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MODEL ______ CONTRACT 951132

ISSUE NO. ______ ISSUED TO: ______

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ABSTRACT

This document is the final report on efforts conducted under JPL Contract 951132, "Fabrication Feasibility Study of a 20 Watt Per Pound Solar Array". The solar arrays under study are to be used to power electric propulsion engines on unmanned spacecraft enroute to a Mars orbit.

The study includes development of Criteria and Requirements, data collection, parametric evaluations and configuration trades of arrays using components and materials and processes that are considered "state-of-the-art" as of January 1966. After establishment of a baseline configuration, both a 10 and a 50 kilowatt prototype preliminary design were developed and small, 1 square foot, sample panels and small scale model arrays were fabricated.

The contract was begun on February 22, 1965, and completed in December, 1965.

KEY WORDS

Criteria and Requirements
Panel Configuration Trades
Array Configuration Trades
Parametric Evaluations
Solar Photovoltaic Cells
Cover Glasses
Dielectric
Busses
Folding Modular Array
Panel Substrate
Rigid Panels
Semi-Rigid Panel
Flexible (Rollup) Panels
Deployment Mechanisms
Release Mechanisms
Interconnectors
Rollup Arrays

CONTENTS

CONTENTS (Cont'd)

FIGURES

FIGURES (Cont'd)

FIGURES (Cont'd)

FIGURES (Cont'd)

TABLES

1.0 INTRODUCTION AND SUMMARY

1.1 INTRODUCTION

This final report summarizes the program activities conducted under JPL Study Contract 951132. The contract was initiated on February 22, and completed in December, 1965. The program evaluated the feasibility of fabricating large deployable, 3 to 50 kilowatt solar photovoltaic arrays. They were to provide electricity at 20 watts per pound at a sun probe distance of 1 Astronomical Unit. The arrays were to be used to drive electric propulsion systems and to provide all other electric power on an unmanned Mars orbiting spacecraft.

Two baseline spacecraft configurations were used. One for launch by an Atlas/Centaur required 10 KW of power at 1 Astronomical Unit. The second by the Saturn IB/Centaur requiring 50 KW of power.

The completed preliminary designs showed predicted results of 24.35 watts per pound for the array launched by the Saturn IB/Centaur and 19.64 watts per pound for the array launched by the Atlas/Centaur.

1.2 SUMMARY

1.2.1 General

A statement of criteria and requirements was formulated and is included in Section 2.0 of this report. Data was extracted from coordination meetings with JPL, Hughes Aircraft Company, Electro-Optical Systems, and from Boeing analyses. Envelope, structural, dynamic, thermal, electrical and mission requirements provided definition of the problems requiring resolution. Figure 1.1-1 summarizes the criteria statement.

Alternate structural, mechanical, electrical, and cell stack systems were configurated. Flexible, rigid, concentrator and semi-rigid panel concepts were examined. Mechanical deployment concepts examined included booms, cable systems, and bourdon tubes.

By evaluation and integration of these concepts, a bonded beryllium structure carrying 8 mil solar cells with 4 mil cover glasses was selected. These were mounted on a stretched fiberglass tape substrate.

When deployed, the array consists of 4 solar panel assemblies in a cruciform pattern. Each solar panel assembly consists of a series of hinged sub-panel assemblies. Each sub-panel assembly consists of a main sub-panel and two hinged auxiliary sub-panels. See Figure 1.2-1.

CRITERIA AND REQUIREMENTS SUMMARY

MISSION PROFILE

- 71 THRU 75 TIME PERIOD
- AMR DIRECT LAUNCH
- 350 DAYS TO ENCOUNTER
- 140 DAYS OPERATION IN MARS ORBIT
- SATURN IB/CENTAUR LAUNCH VEHICLE
- ATLAS/CENTAUR LAUNCH VEHICLE

POWER PROFILE

- 10 OR 50 KW AT A SUN-PROBE DISTANCE OF 1 A.U.
- 0.5 OR 1.1 KW AFTER MARS ENCOUNTER (AT A SUN-PROBE DISTANCE OF 1.67 A.U.)
- POWER CHARACTERISTICS FOR OPTIMIZATION
- 20 WATT PER POUND MINIMUM CONVERSION RATIO

STRUCTURAL AND ENVIRONMENTAL

- NATURAL
- INDUCED

INTERFACES

- PHYSICAL
- FUNCTIONAL

MANUFACTURING AND DEVELOPMENT

- AUTOMATION OF CELL MODULE ASSEMBLIES
- REPAIR CAPABILITIES
- ASSEMBLY AND INSTALLATION TECHNIQUES
- TEST REQUIREMENTS

RELIABILITY

- ELECTRICAL
- MECHANICAL
- STRUCTURAL

CONFIGURATION AND ENVELOPE

- DISCOVER-SERIES LANDER
  - SC-1 FOLDING MODULAR
  - SC-2 FOLDED DISC
  - SC-3 ROLLUP

FIGURE 1.1-1

When the solar array is in a stowed configuration, the panels are folded into four (4) compact stacks. The inner most panel of each stack supports the total weight of each stack during launch. The only attachment to the spacecraft is by the lower hinges of the inner panel. The stowed stacks are connected to each other with a fitting at a point which is 70% of the stack length from the lower hinge point. The total stowed array will then act as a unit.

Array extension is by electric motor and cable retraction. Auxiliary panels are deployed by redundant torsion springs. Separation and unlocking from the stowed position is accomplished by pyrotechnic devices. Redundancy is provided in all critical elements. See Section 6 for full description.

The electrical power subsystem consists of the solar cell modules and power busses to generate the required power and carry the current to the interface at the spacecraft-array joint. Power conditioning is provided on the spacecraft. 28-volt power is provided by sub-panel #1 for telecommunications and for the scientific and engineering experiment package. 100 volt power is provided by the outboard sub-panels to the spacecraft bus for the electrical propulsion engines.

Panel temperature, deployment completion, and separation completion sensors are provided on each panel assembly. Current and voltage sensors for each assembly are provided on the spacecraft power busses.

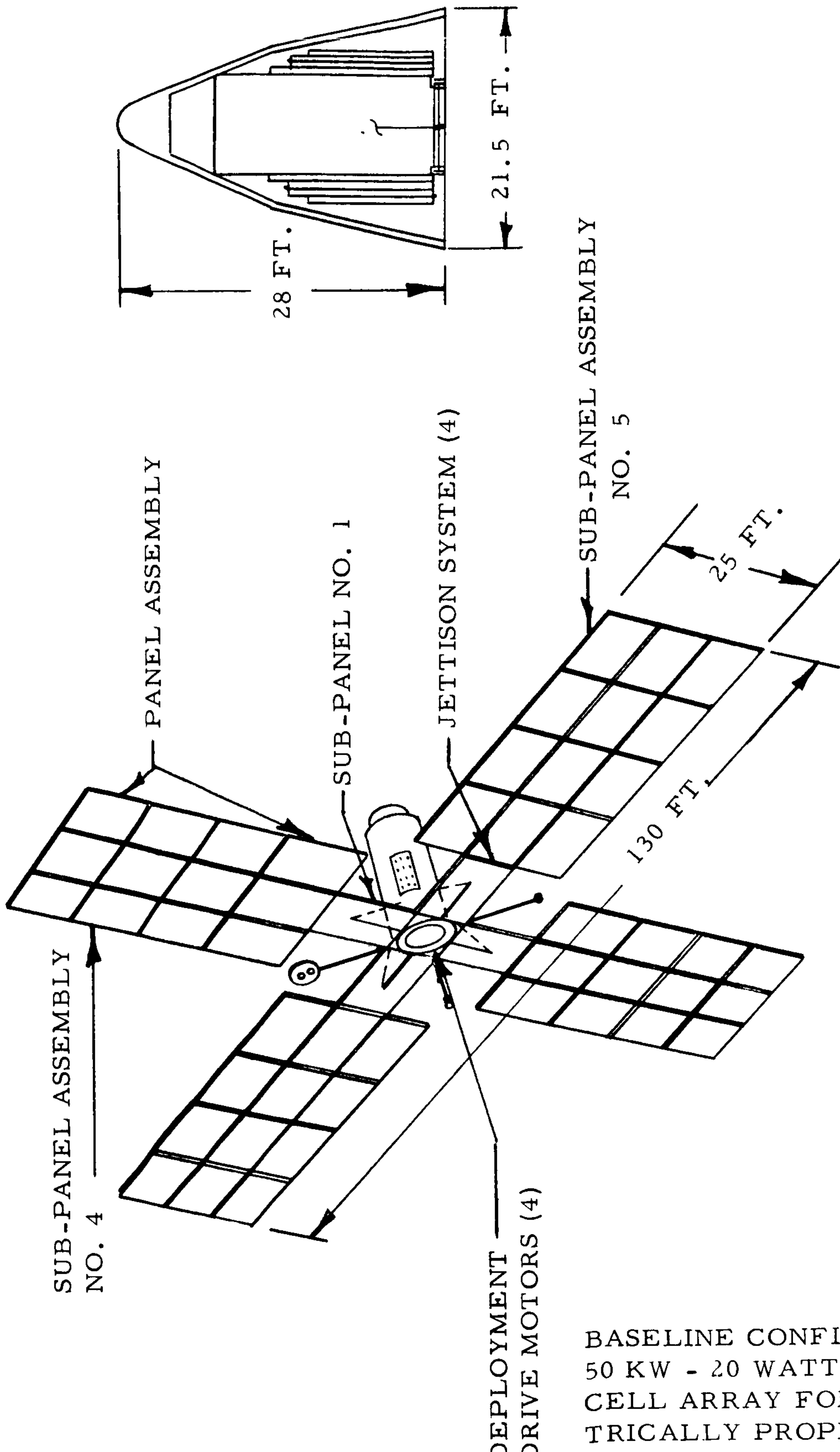
Ground support requirements from fabrication through development, qualification and acceptance testing phases have been developed.

1.2.2 Saturn IB/Centaur Array

The preliminary design of the array for launch by a Saturn IB/Centaur deploys 4944 square feet gross area and 4433 square feet of solar cell module area. Array weight is calculated to be 1959.7 pounds. It yields 47.7 KW of power at 1 Astronomical Unit for 10.8 watts per square foot and 24.35 watts per pound ratios. If a weight contingency of 10% were added, this last factor becomes 21.9 watts per pound. Sufficient stowage space exists under the shroud to add an additional sub-panel assembly to each panel. This would provide approximately 500 square feet of additional deployed area for a 52 kilowatt power output for the array. Preliminary dynamic, thermal, and stress analyses of this design have been completed. One problem area was identified and is discussed in Section 1.6. See Section 6.1 for complete discussion of analysis and array design.

1.2.3 Atlas/Centaur Array

The preliminary design of the Atlas/Centaur array deploys 1004 square feet gross area and 930 square feet of solar cell module area. Array calculated weight is 518.12 lbs. It yields 10.18 KW of power at 1 A.U. for 10.8 watts per



BASELINE CONFIGURATION
50 KW - 20 WATT/POUND SOLAR
CELL ARRAY FOR AN ELEC-
TRICALLY PROPELLED MARS
ORBITER

FIGURE 1.2-1

square foot and 19.64 watts per pound ratio.

Preliminary thermal and static analyses were run. See Section 6.2 for complete array description.

1.2.4 Prototype Program Plan

A program plan for the development of full scale prototype arrays was completed and the results summarized in Section 7. Included are cost and schedule evaluations and a statement of test and ground support equipment and facility requirements. The study indicates that 29 months would be required for the design, development, fabrication and test of a 10 kilowatt Atlas/Centaur launched array and 32 months for a 50 kilowatt Saturn IB/Centaur launched array.

The detailed program plan was included in bi-monthly report number 4, Document D2-23942-4.

1.2.5 Samples and Models

Three one square foot panels were fabricated. They demonstrate the feasibility of the cell mounting and soldering techniques to a hand assembly level. The panels will be solar tested to verify the watts per square foot output used in the preliminary design effort. It is also planned for one panel to be incorporated in an ion engine test to determine the effects of ion impingement on the panel.

A 1/20th scale model of the Saturn IB/Centaur launched array and the Atlas/Centaur launched array were fabricated. These models illustrate the launch packaging and deploying principles of the array designs. A full scale model of one of the overcenter hinge latches used on the array was fabricated to illustrate the deployment latching.

1.2.6 Trade Studies

Concurrently, with the development of the criteria statement, evaluations of the following major elements were conducted.

I. Solar Cell Modules
II. Electrical Bus System
III. Structural Support
IV. Deployment and Jettison Mechanisms
V. Ground Support Requirements

The leading components of each subsystem were integrated and synthesized into several array configurations; two of which are shown on Figure 1.2-2. Each of these configurations were evaluated and compared on the basis of power output potential, weight, reliability, and state-of-the-art.

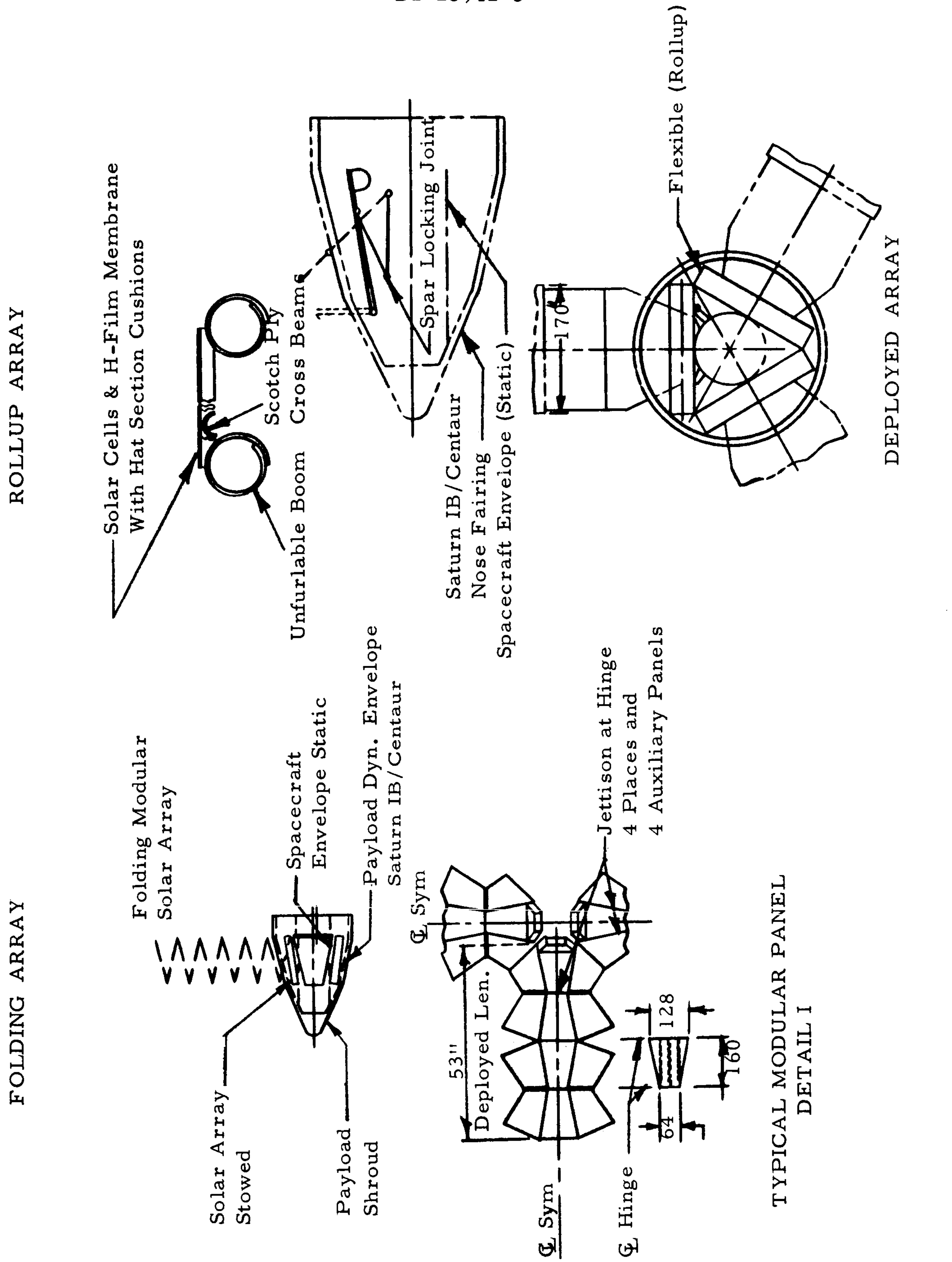
The folding modular array consists of trapezoidal panels joined together by hinges and latches, folded in the stowed position and deployed by one of several actuating systems. The trapezoidal panels best utilize the available conic envelope under the shroud, but create several design and fabrication problems. A typical rectangular semi-rigid panel is illustrated in Figure 1.2-3.

Several panel configurations including aluminum and beryllium flat sheet stringer, aluminum concentrator and semi-rigid panels were analyzed. Only the aluminum concentrator and beryllium semi-rigid panels were capable of meeting the 20 watt per pound conversion ratio requirement. Sufficient area of concentrator panel could not be stowed in the available envelope to meet the 10 and 50 kilowatt power requirements.

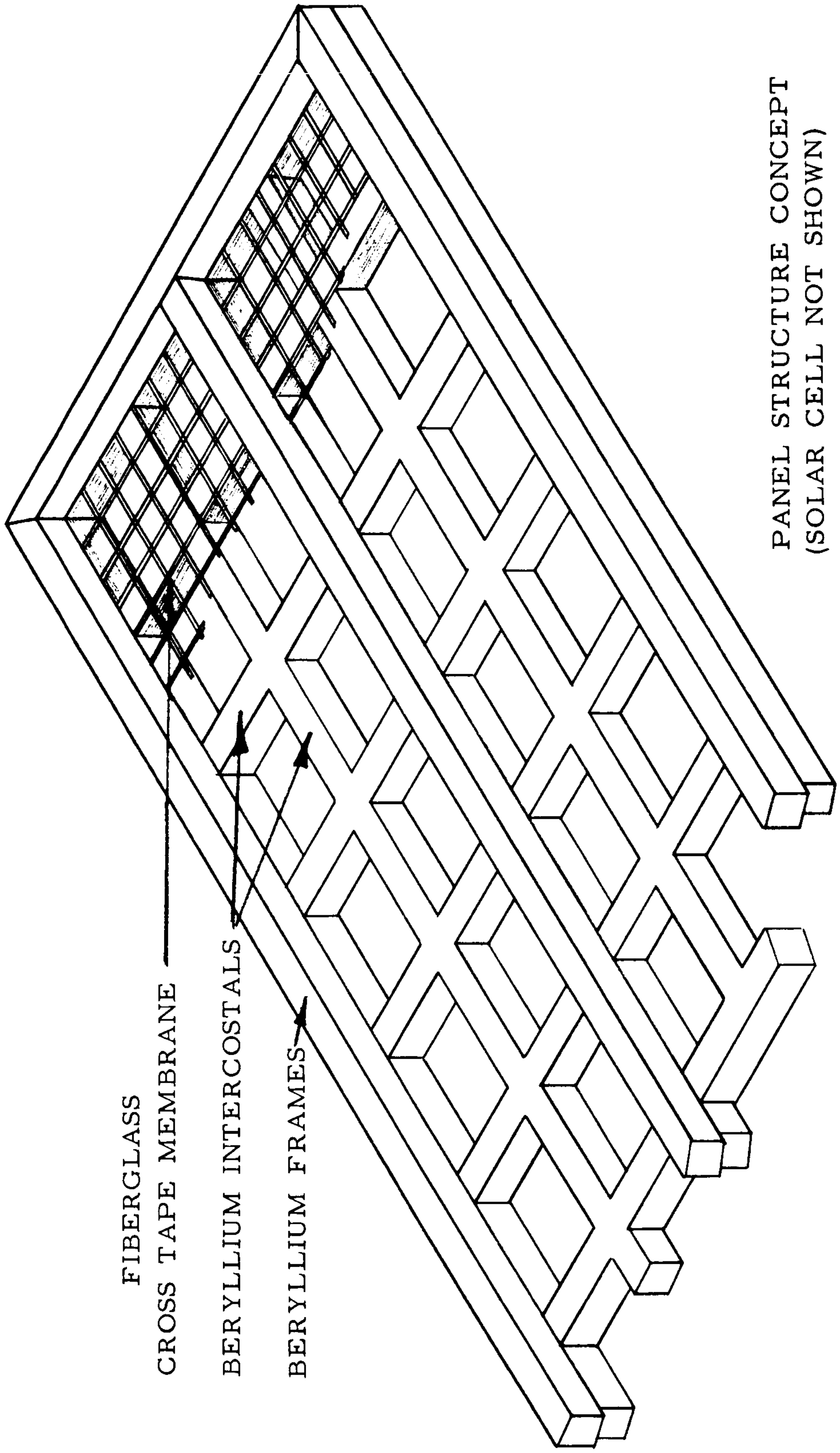
Beryllium and aluminum support structure were compared. Aluminum with a modulus of elasticity of 10×10^6 and a specific density of 0.10 pounds per cubic inch did not meet the weight requirement. Beryllium with a modulus of 43.5×10^6 and a specific density of .067 pounds per cubic inch did meet them. The use of beryllium will result in higher fabrication cost because of special tooling and facility requirements.

For the rollup type array, combinations of rigid panels and curtain panels were considered. The rigid panels would be capable of sustaining the retro loads imposed during injection into the Mars orbit. H-film substrates mounted to collapsible type booms were configurated. Two substrate concepts, shown on Figure 1.2-4, were considered. One used H-film corrugations bonded to a flat sheet of H-film, for solar cell support, and in the other, the cell stack and H-film substrates were protected by foam rubber between the rolled up layers. Beryllium copper DeHavilland STEM type, and collapsible closed section booms, and telescoping aluminum booms were considered for deployment and structural support.

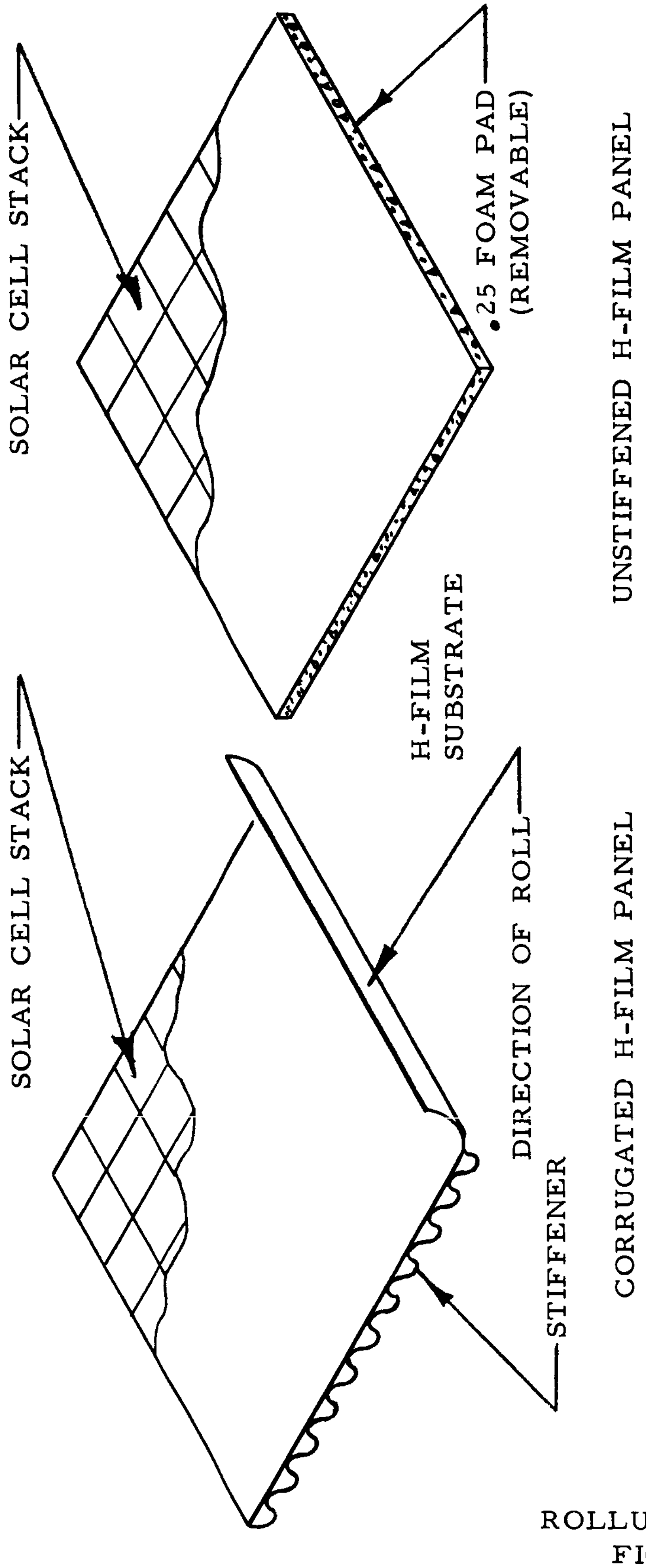
Other new components selected during the trade studies include 8 mil N on P back-connected silicon solar cells and 4 mil microsheet cover glasses. These components provided the best power per pound ratio within the expected state-of-the-art as of January 1, 1966. See Section 4 for full discussion.



ARRAY CONFIGURATIONS
FIGURE 1.2-2



FOLDING MODULAR PANEL
FIGURE 1.2-3



ROLLUP SUBSTRATES
FIGURE 1.2-4

1.3 PROBLEMS

Five (5) major and two (2) minor problems have been either identified or unresolved.

The major problem areas are:

(a) Availability of 2 x 2 cm, 8 mil N on P, back-connected, 11.8% efficient cells at air mass 1 in quantities required.
(b) Single source availability for 4 mil microsheet cover glasses.
(c) Verification of dynamic analyses through test.
(d) Manufacturing development of beryllium frame - fiberglass substrate assembly techniques.
(e) Manufacturing development of solar cell stack bonding and soldering techniques.

Discussions with solar cell vendors indicate differences of opinion as to the availability of sufficient 8 mil solar cells for a 1971 flight article. A summary of these investigations indicates that small quantities, up to approximately 2,000, 11.4% efficient cells at air mass 1 could be available shortly at about $6.50 per cell. If the electrical output for cells to be delivered in quantity in the 1969 to 1971 period are to be increased to the desired AM-1 efficiency of 11.8%, it is estimated that a $15,000 to $20,000 6-month development program would be required. Results cannot be guaranteed. If the program were successful, cells in the quantity required for one complete flight article would cost approximately $4.37 per unit. If the program were not successful, lower efficiency 8 mil cells or 12 mil cells could be used with an approximate 10 percent reduction in the watts per pound ratio.

The above information appears pessimistic to The Boeing Company. Tests of 717 8 mil front-connected cells provided by JPL show an average air mass 1 efficiency of 11.72 percent. The results of these tests are further discussed in Section 6.1.4 of this report.

At the present time, 4 mil cover glasses are available from one vendor only. Additional funding may be required to provide for development of process techniques to ensure adequate supplies of these components for flight article fabrication.

The dynamic analyses have been based on an assumed damping factor for the bonded beryllium structure. Based on past experience, a dynamic magnification ratio of 15 has been estimated. However, arrays of the type and size designed in the program have never been fabricated. Only fabrication and test of a portion of the array structure can provide confidence in the analyses that have been made.

Development of fiberglass substrate assembly techniques on a small scale occurred during the fabrication of the sample panels. Problem evaluation and recommended solutions for full scale production have been made and are reported in Section 6 of this report.

Development of solar cell bonding and soldering techniques to the hand assembly level were accomplished during fabrication of the small sample panels. Discussion of full scale fabrication techniques are included in Section 6 of this report.

Minor or unresolved problems:

(a) Effect of ion engine exhaust impingement on solar cells.
(b) Overstressed members due to dynamic boost loads.

The effect of (a) will be evaluated at a test of one of the square foot panels at an engine manufacturer's facility. Until the test is run, no design effect can be forecast.

In the final dynamic analysis, some members in each sub-panel were found to be severly overstressed during the dynamic boost condition. At least four solutions are known and it is estimated that 80 pounds of structure will be entirely adequate to fix the problem. Additional design work is required to arrive at an optimized design fix. See Section 6 for complete discussion.

1.3.1 Interfaces

Compatible envelopes for the array stacks and the spacecraft structure and subsystems in both the stowed and deployed positions have been determined. Physical and functional interfaces for the 400 cycle deployment motors and locations for the cables were established.

The structural loads imparted at the array-spacecraft attachment have been calculated. See Section 6. The dynamic input from the launch vehicle and spacecraft to the stowed array is specified in the design requirements. The deployed array has been designed to meet stiffness requirements which preclude coupling with the spacecraft guidance and control system.

The electrical interface between the 28 and 100 volt bus systems of the array and the spacecraft has been defined and documented. The array bus system terminates in NAS 1599 connectors.

1.4 CONCLUSIONS

From the results attained by the preliminary designs of the two solar arrays the following is concluded.

1. Twenty watt per pound solar arrays for this mission can be built within a state-of-the-art achievable by the time the panels will be required.

2. Money must be put into several areas to insure this. These are:
   (a) Improvements in the solar cells
   (b) Automation of cell inspection, assembly, and installation
   (c) Prototype or preprototype dynamic and deployment testing

3. Certain generalized conclusions can be drawn regarding the power per pound ratio. These are:
   (a) Increase in size improves the ratio. This is due to a disproportionate increase in mechanism weight per unit area for smaller arrays.
   (b) Available space for structure affects the ratio. One significant difference between the Saturn IB/Centaur's 24.3 watts per pound and the Atlas/Centaur's 19.6 watts per pound is the space available for panel spars. The Saturn, with more available space, is simply more efficient structurally.
   (c) Structural interties between the spacecraft and solar arrays must be held to a minimum. The lowest structural array weight can only be achieved if the arrays are isolated as far as practical from amplified spacecraft structure dynamic loadings.

1.5 STUDY PROGRAM PLAN

The foregoing work was accomplished in accordance with the following plan:

Phase I

(a) Define the problem by determining the criteria and requirements for the array and the array-spacecraft interfaces.

(b) Data collection and evaluation and analysis of state-of-the-art components, and materials and processes that were appropriate to the development of panel and array configurations.

(c) Use of this material to develop alternate panel substrates, cell stacks, and several alternates for support structure, actuators, and mechanisms. The components and subsystems were synthesized into array configurations and compared to the requirements to select the best system configuration.

(d) After establishment of a baseline configuration, a preliminary design for both the 10 and 50 KW arrays was developed.

(e) The prototype designs were evaluated for power to weight ratio. The estimated cost, and the schedule and fabrication development problems to produce a prototype array were determined.

Phase II

(a) Design and fabrication of three identical one square foot sample panels and a small scale hand operated working model of each array.

(b) Fabrication of the sample panels illustrating the cell mounting and interconnection techniques.

Testing of the panels would permit verifying the prototype design and analysis in terms of watts per square foot and for resistance to thermal, thermal shock and acoustic loads. The array models will illustrate the stowage and deployment techniques of the designs and provide visual aids for the final program review.

Phase III

Reporting and documentation of the study results.

2.0 CRITERIA AND REQUIREMENTS

To define the problem, a statement of the general and specific environmental, mission and interface requirements to be met by the solar array power subsystem has been established. The information was used as the baseline for the trade studies and the preliminary designs of two solar photovoltaic arrays which provide primary electrical power for electrically propelled Mars orbiting spacecraft. The preliminary design is a portion of the study to determine the feasibility of fabricating 20 watt per pound arrays for this mission.

One array is mounted to a spacecraft launched by the Atlas/Centaur and provides 3 to 10 kilowatts of electrical power at a sun-probe distance of one astronomical unit. The second array is mounted to a spacecraft launched by a Saturn IB/Centaur and provides 30 to 50 kilowatts of electrical power at a sun-probe distance of one astronomical unit.

Data for this statement was extracted from interface coordination meetings with Jet Propulsion Laboratory, Hughes Aircraft Company, Electro-Optical Systems, and from analyses conducted by The Boeing Company.

2.1 DEFINITIONS - ELECTRICAL

The following terms are defined to establish common understanding.

2.1.1 Electrical Power

Spacecraft prime electrical power is provided by one or more two-wire, ungrounded systems. Power is supplied by solar cells and is available as unregulated direct current at specified voltages.

2.1.2 Load - Electrical

Any component or subsystem utilizing electric power which is normally operated as a unit is considered to be a load.

2.1.3 Nominal System Voltage (Voltages)

This is a steady-state voltage which can be maintained at 100 percent electrical load, when demanded, under specified conditions of environment. It is measured at the main bus terminals.

2.1.4 Voltage Range

This expresses the upper and lower voltage limits which will permit normal operation of load equipment in a mission environment.

2.1.5 Bus Voltage

The bus voltage is measured at the main electrical bus terminals. The measured value will vary with the load, but will always be within the specified voltage range.

2.1.6 Transients - Electrical

An electrical transient is the changing condition of a characteristic which goes beyond the steady-state limits and returns to the steady-state limits within 10 milliseconds.

2.1.7 Electrical Load Profile

A chart or graph showing power demands (watts) at the main electrical bus as a function of time.

2.1.8 Source (Generator) Impedance

The source impedance is a variable which depends on solar intensity, temperature of the solar cells and the amount of electrical load.

2.1.9 Reverse Current

A condition where current flow reverses direction due to a higher generated voltage from the electrical load. Reverse current is defined as having a duration of 10 or more milliseconds and is, therefore, not considered a transient occurrence.

2.1.10 Inrush Current

Current which is greater than normal operating current may be caused by various conditions, but is always associated with switching on an electrical load.

2.1.11 Fault Current

The increment of current which appears at the generator terminals and is caused by a physical short-circuit of the power leads.

2.1.12 Ripple

Ripple is the A.C. variation of voltage about a fixed D.C. voltage during a steady-state D.C. electric system operation.

2.1.13 Solar Array - Configuration

A complete assembly for one vehicle of all structural, mechanical, deployment mechanisms, latches, dampers, and electrical parts and equipment including devices for attaching the array to the spacecraft. Electrical lines to the main electrical bus within the spacecraft are included, but electrical conditioning is not.

2.1.14 Solar Panel

The largest element of the solar array which attaches to the spacecraft.

2.1.15 Sub-Panel

The largest element of a solar panel.

2.1.16 Solar Cell Module

A module is the minimum group of solar cells which will independently produce the electrical subsystem voltage at the output terminals of the cell group. Modules will assume various physical outlines as required by the configuration of the structural support. Electrical power output will vary with the number of solar cells used in the module.

2.1.17 Solar Cell

The smallest electrical producing element of the module. Single crystal silicon elements will be utilized.

2.1.18 Electrical Bus

Metallic conductors which provide electrical continuity between solar cell modules and between each solar panel assembly and the spacecraft.

2.1.19 Bus Crossovers

Flexible or hinged busses which provide electrical continuity across solar panel, sub-panel or solar array hinge points.

2.1.20 Jettison Control Wires

The wires which are used to activate solar panel jettison initiators.

2.2 MISSION ASSUMPTIONS

2.2.1 Spacecraft

Electrically propelled unmanned Mars orbiter.

2.2.2 Duration

Flight time to Mars rendezvous of 350 days (maximum). Mars orbital operating period - 140 days.

2.2.3 Orbital Altitudes

Direct ascent into Mars transfer trajectories.

2.2.3.1 Mars Orbit

Circular - 5,000 kilometers.

2.2.4 Launch Location

Atlantic Missile Range (AMR).

2.2.5 Launch Vehicles

Atlas/Centaur for 3 to 10 kilowatt array. Saturn IB/Centaur for 30 to 50 kilowatt array.

2.2.6 Launch Period

1971 - 1975

2.2.7 Use of 1966 state-of-the-art components, materials and processes.

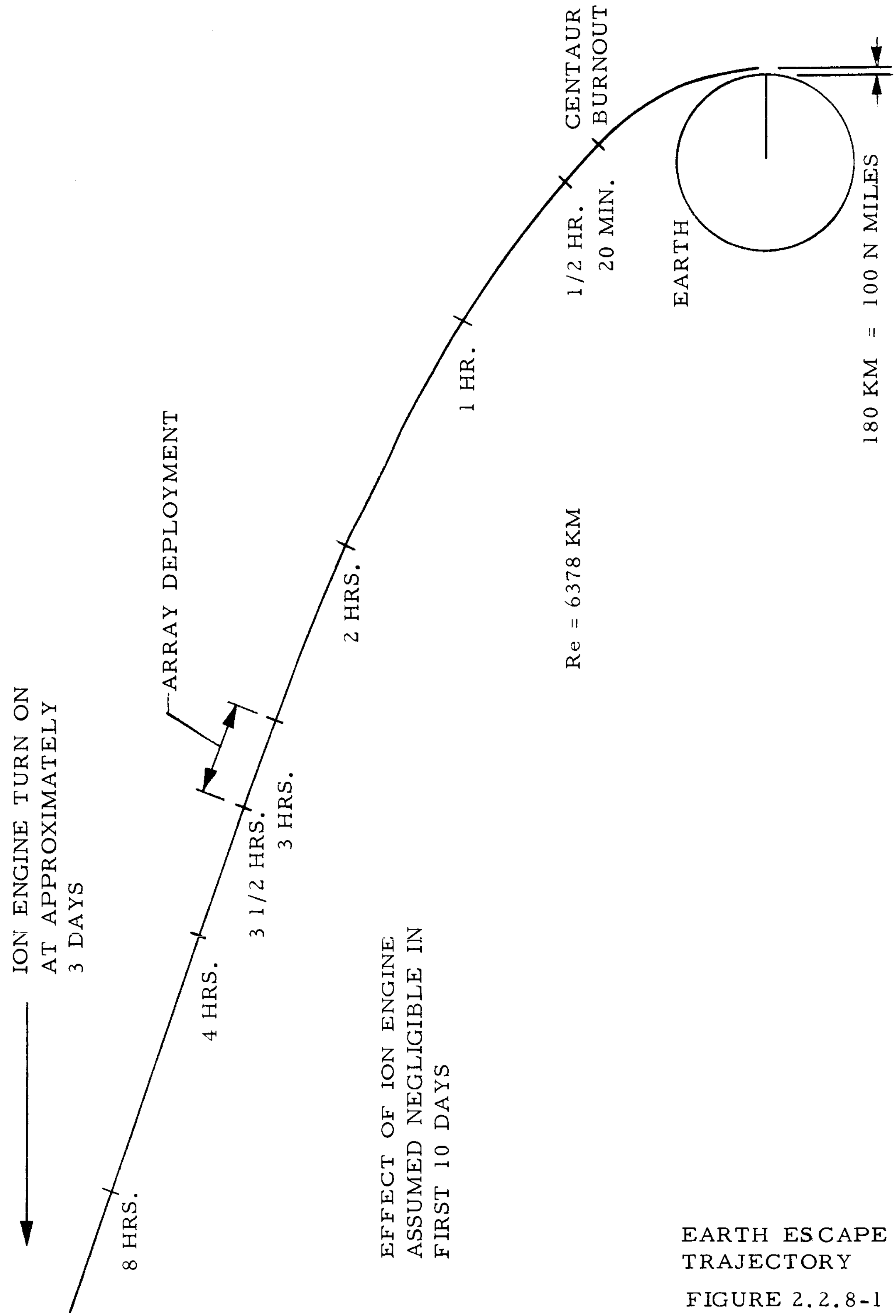
2.2.8 Mission Profile

The mission profile is defined, generally, by Figures 2.2.8-1 thru 2.2.8-4.

Figure 2.2.8-1 defines the earth escape trajectory portion of the mission and shows that array deployment would take place approximately 3 hours after launch.

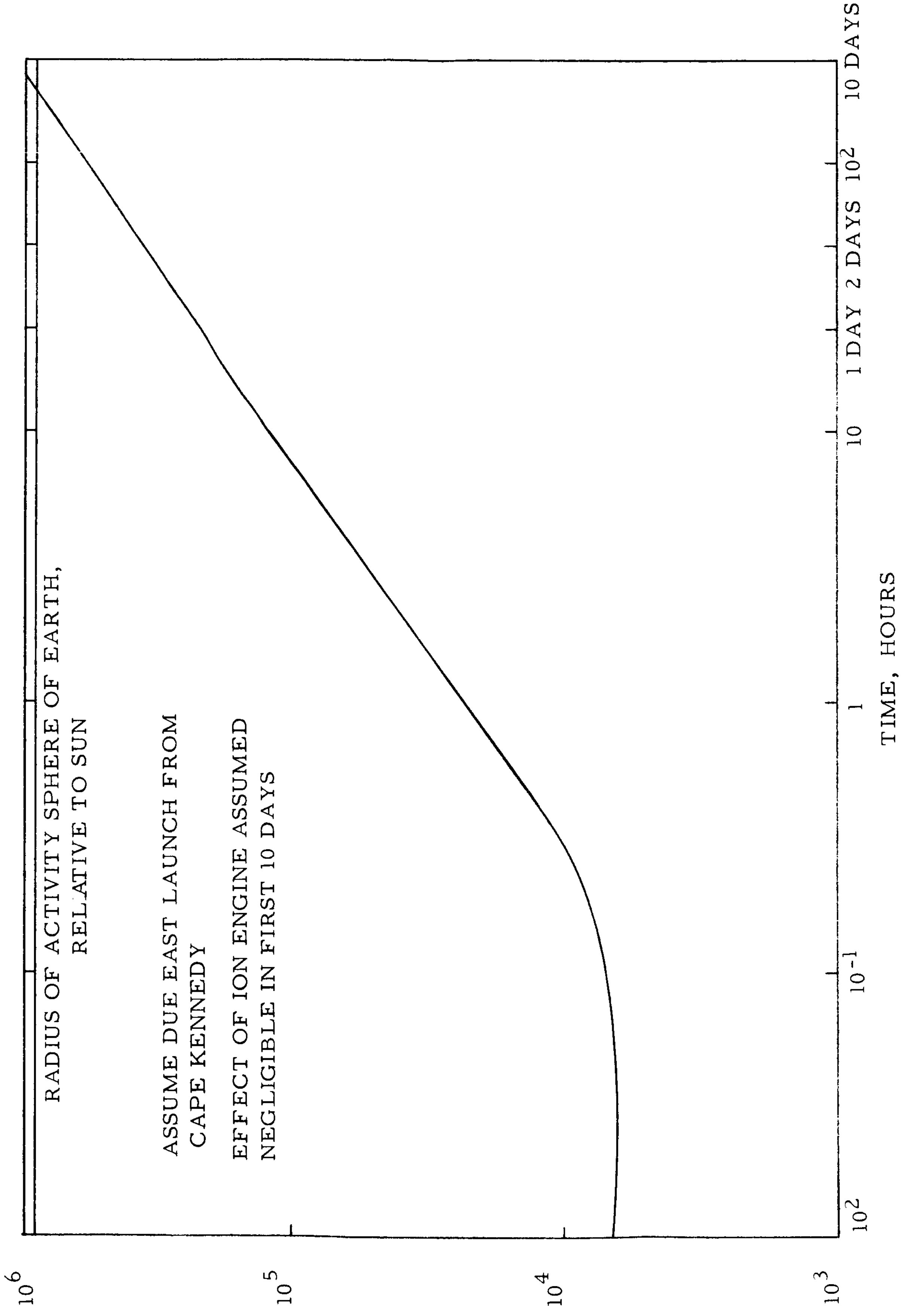
Figure 2.2.8-2 indicates the spacecraft distance from the sun, with respect to time for the first 10 days of the mission.

Figure 2.2.8-3 defines the sun probe distance, in astronomical units, of the spacecraft for the mission reference trajectory.



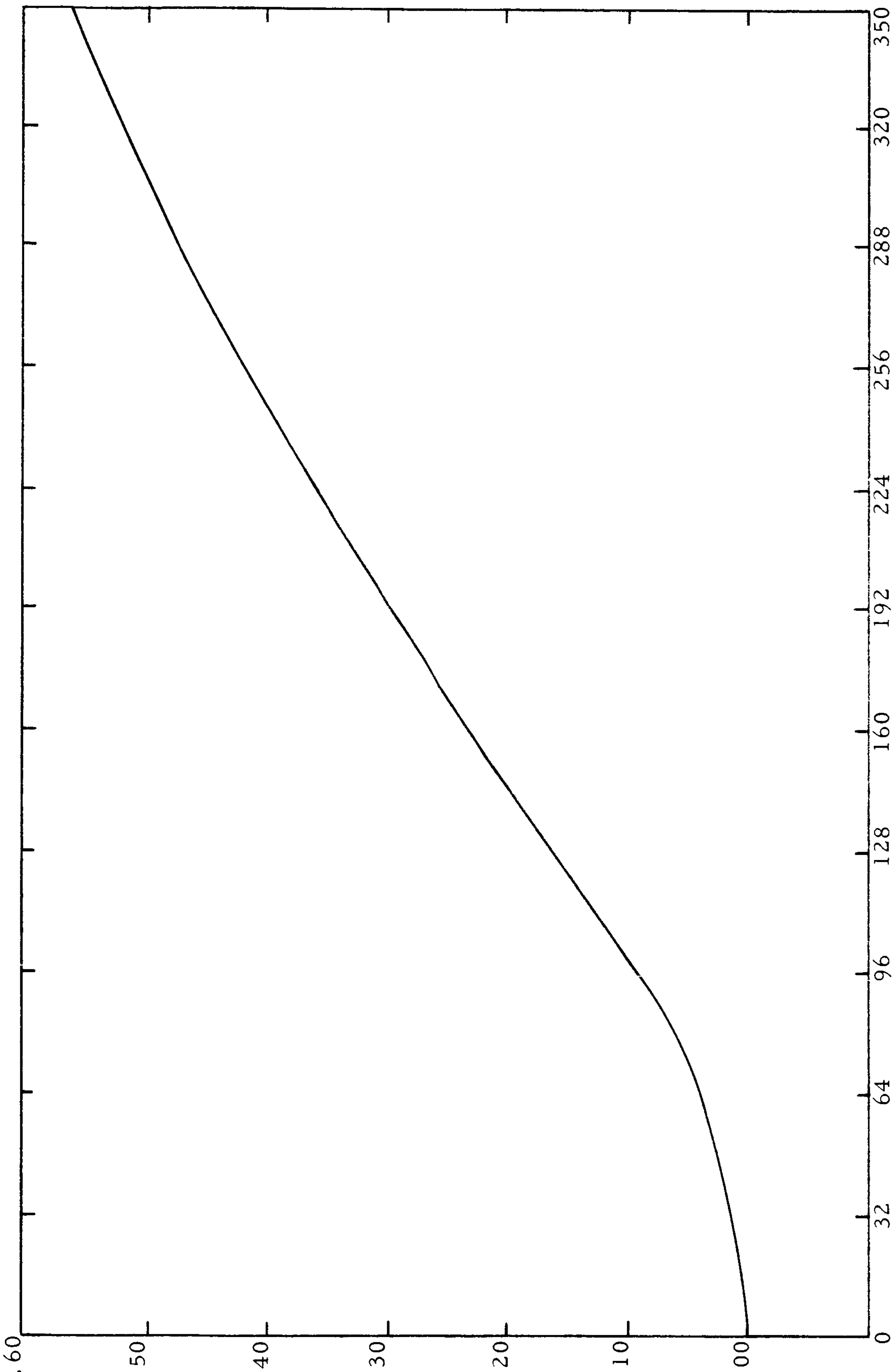
EARTH ESCAPE TRAJECTORY

FIGURE 2.2.8-1



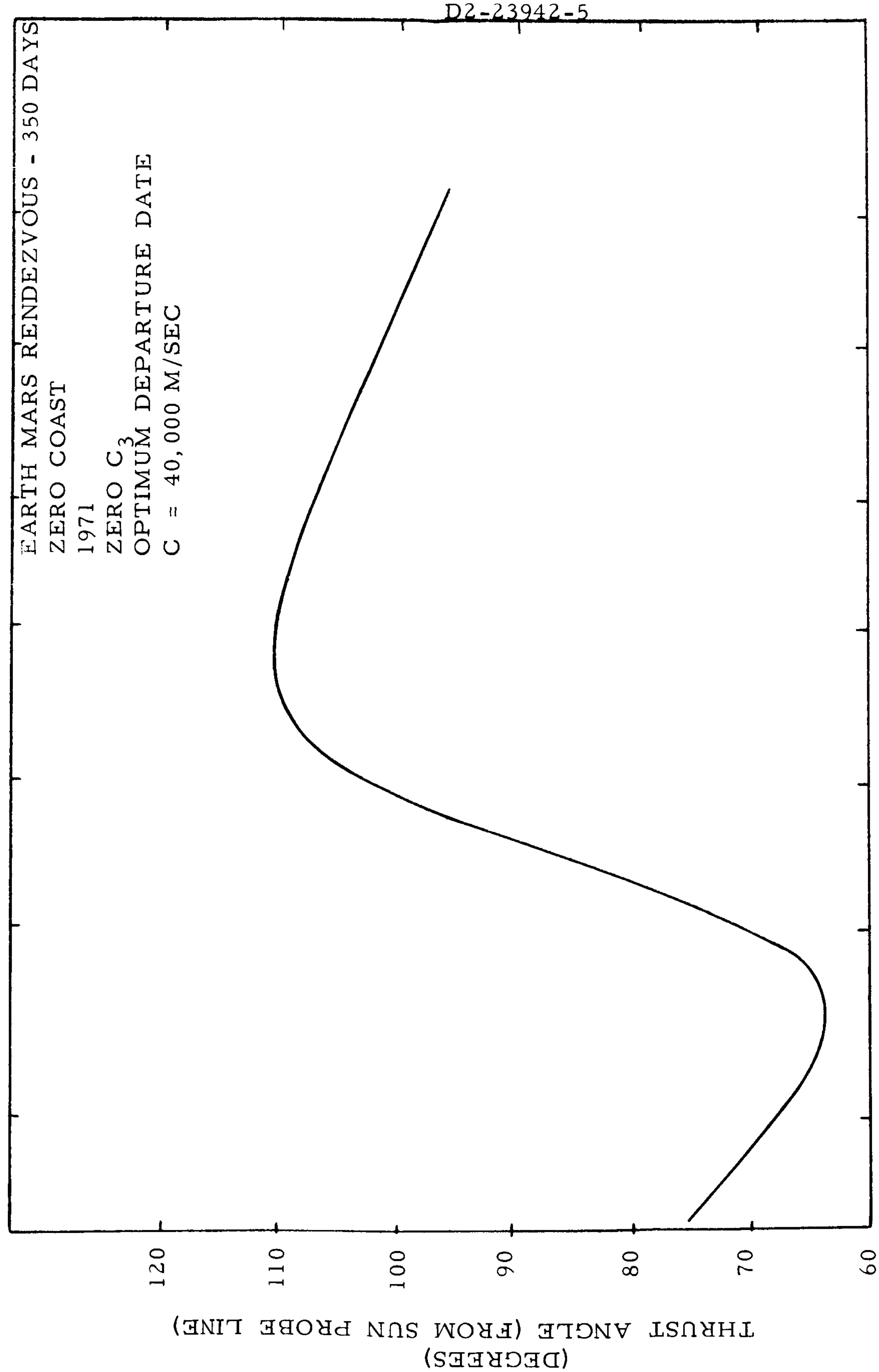
RADIAL DISTANCES VS TIME FOR PARABOLIC ESCAPE
($C_3 = 0$)

FIGURE 2.2.8-2



DISTANCE FROM SUN VERSUS TIME FOR EARTH-MARS RENDEZVOUS REFERENCE TRAJECTORY

FIGURE 2.2.8-3



FLIGHT TIME PROFILE
(DAYS)

FIGURE 2.2.8-4

Figure 2.2.8-4 defines the relationship of the electrical propulsion engine's line of thrust to the sun probe line. The sun probe line is defined as a line through the sun and normal to the solar array.

2.3 SPACECRAFT ENVELOPE REQUIREMENTS

The final physical envelopes and configurations for the spacecraft subsystems, structure, deployable antennas, instruments, and for the solar array were established by coordination with the mission analysis contractors, JPL, and Boeing.

Figures 2.3-1 and 2.3-2 define the gross envelope requirements for the solar array and the spacecraft for the Atlas/Centaur and Saturn IB/Centaur launch vehicles, respectively. Dynamic and static envelopes as well as permissible center of gravity locations are provided.

2.4 SOLAR ARRAY STRUCTURAL AND ENVIRONMENTAL DESIGN CRITERIA

2.4.1 Ground Handling

2.4.1.1 The solar array structural performance shall not be degraded due to cleaning of the surfaces after fabrication. The cleaning operation shall consist of wiping the surfaces with a soft lint-free cloth saturated with a suitable cleaning solvent.

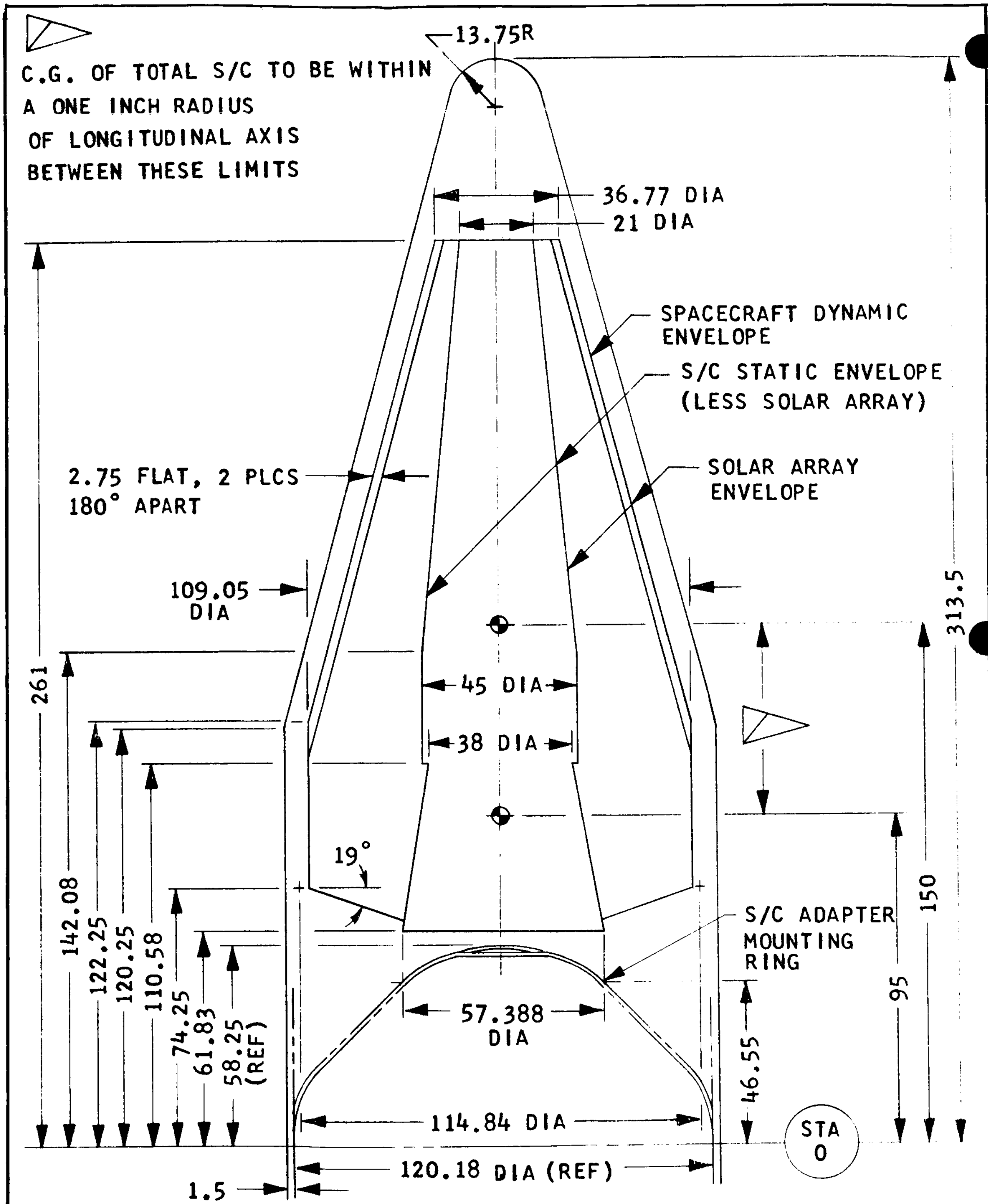
2.4.1.2 The solar array structural and electrical performance shall not be degraded due to transporting a suitably packaged array from the manufacturing area to the spacecraft launch area. The ground handling transportation test specification shall be equivalent to the applicable portions of MIL-STD-810A, dated 23 June 1964, and shall be applied to the outside of the container in its transportable configuration.

2.4.2 Launch Configuration

The solar array in the stowed configuration shall be capable of withstanding without structural, electrical, or mechanical degradation, the following structural load environment. The solar array shall not be required to withstand the dynamic load environment superimposed upon the static load environment.

2.4.2.1 Vibration Environment

2.4.2.1.1 Sinusoidal Sweep at 1.0 Min/Octave - The sinusoidal vibration input shall be swept three times from 2 to 200 cps with the input acceleration levels noted and applied in phase, first normal to the plane of one panel of the stowed array, and then parallel to that plane in a lateral direction. The two vibration inputs shall not be applied simultaneously.



ATLAS/CENTAUR
SPACECRAFT ENVELOPE

FIGURE 2.3-1

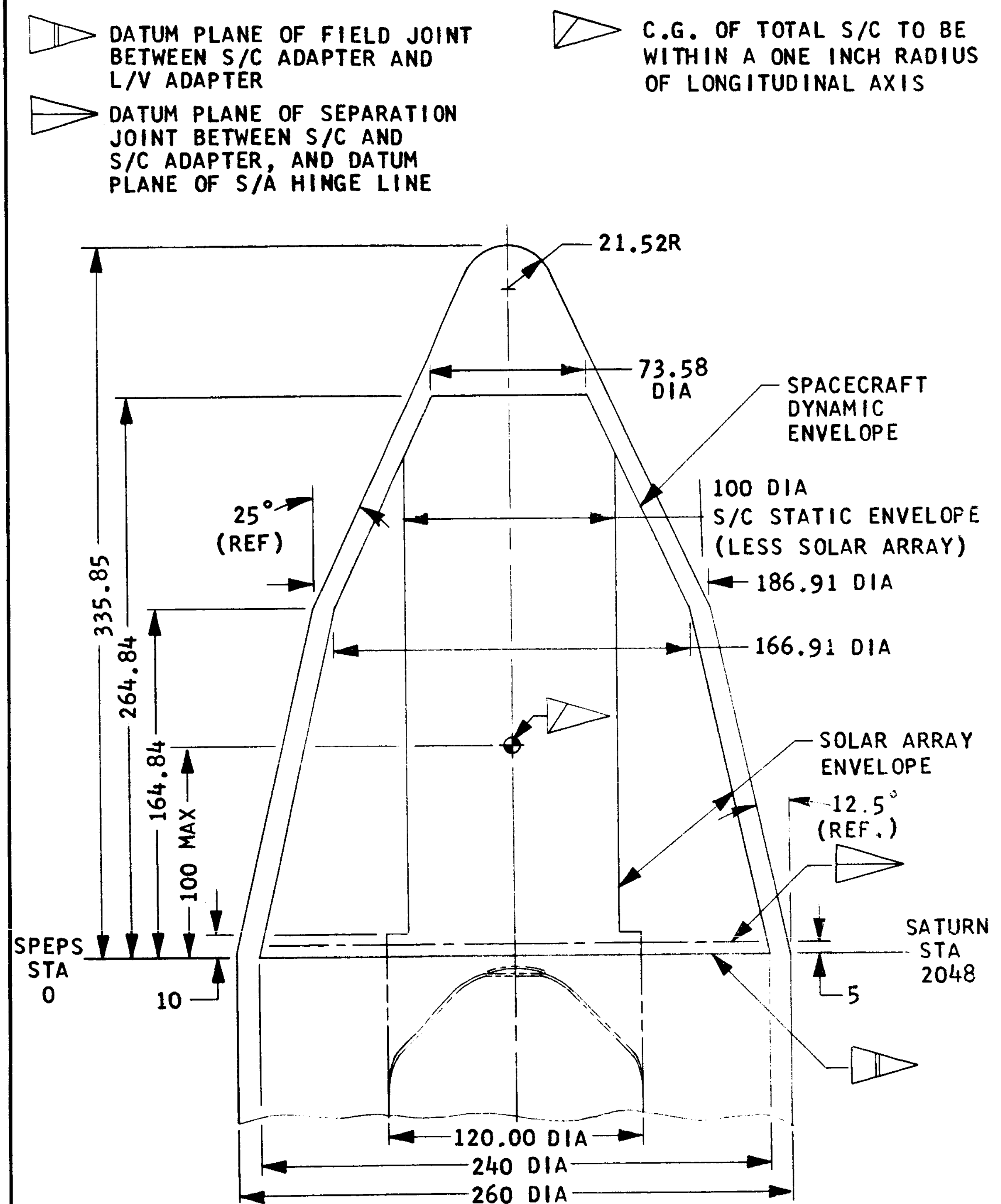


FIGURE 2.3-2 SATURN IB/CENTAUR NOSE FAIRING AND S/C DYNAMIC ENVELOPE

Frequency	Forward Attachment Point	Aft Attachment Point
$2 \leq f < 10$	1.5 "g" rms	1.5 "g" rms
$10 \leq f < 20$	8.5 "g" rms	1.5 "g" rms
$20 \leq f < 50$	5.0 "g" rms	1.5 "g" rms
$50 \leq f < 200$	5.0 "g" rms	2.0 "g" rms

2.4.2.1.2 Random Gaussian Vibration - The random Gaussian vibration shall consist of 3 minutes vibration at 0.2 g^2/cps band limited between 200 and 2000 cps.

2.4.2.2 Static Environment

The static loads shall consist of a steady-state acceleration of 18 "g" directed along the spacecraft longitudinal axis and a 1 "g" steady-state acceleration directed normal to the spacecraft longitudinal axis.

2.4.2.3 Dynamic Characteristics

The first mode resonant frequency of the stowed array shall be less than 10 cycles per second or greater than 25 cycles per second.

2.4.2.4 Acoustic Environment

The stowed arrays shall withstand without degradation the flight acoustic environments specified in Figure 2.4.5-1 during the launch phase.

2.4.2.5 Thermal Environment

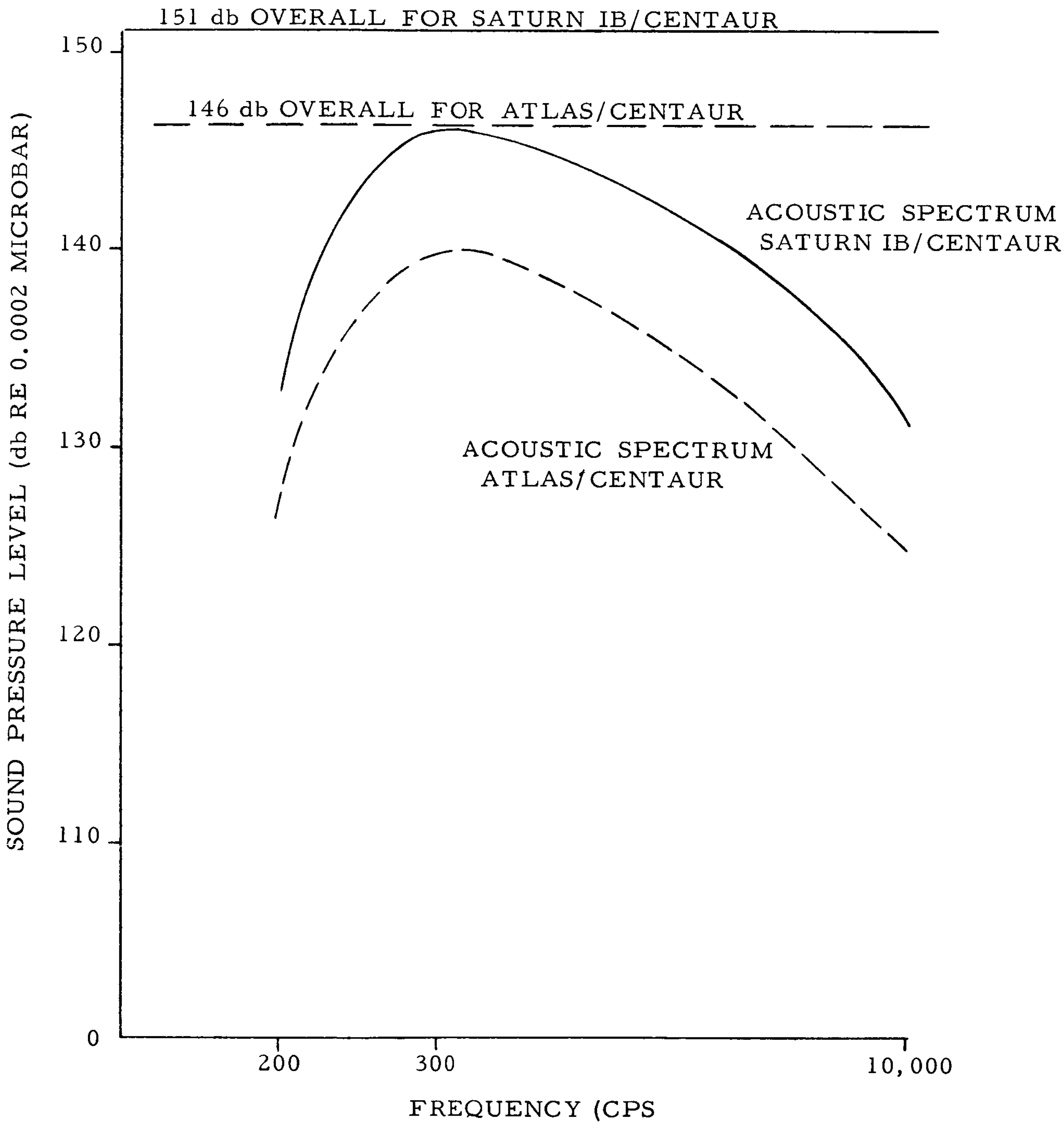
The array shall withstand the transients due to thermal dynamic heating during boost and shroud ejections without structural or electrical degradation.

2.4.3 Deployed Configuration

The solar array shall be capable of full deployment without interference between the array elements and between the array and the spacecraft. The solar array shall be so designed that no structural or electrical degradation will occur due to or during the following conditions.

2.4.3.1 The thermal gradients that will develop between the sun and shade sides of the array due to solar illumination at 140 mw/cm^2 intensity.

2.4.3.2 Transient thermal stresses due to repeated entry into, dwell time within, and emergence from planetary shadow with solar intensity of 50 mw/cm^2, when in a 5000 kilometers circular orbit about Mars.



ATLAS/CENTAUR -
SATURN IB/CENTAUR
ACOUSTIC ENVIRONMENT
FIGURE 2.4.5-1

2.4.3.3 A steady-state acceleration of 3×10^5 "g" directed at 45^o to the plane of the array. This loading simulates that imposed by the cruise engines.

2.4.3.4 A steady-state acceleration of 4 "g" directed normal to the plane and 2 "g" directed in the plane and normal to the span direction (for one panel) of that portion of the array required for secondary power generation in Mars orbit. (Step Functions)

2.4.3.5 Repeated discreet applications to the entire array of a square wave pulse with duration not less than 13 seconds or more than 5 minutes and maximum amplitude of 2×10^{-5} radians/sec^2 pitch angle accelerations.

2.4.3.6 Random Gaussian Vibration band limited between 15 and 1500 cps at 0.002 g^2/cps and 1.5 minutes in duration. This vibration is applied to the portion of the array retained for Mars orbit secondary power generation.

2.4.3.7 Cruise Array Dynamic Characteristics

The first mode resonant frequency of the entire deployed array shall be greater than .04 cycles per second and less than 10 cycles per second.

2.4.3.8 Orbit Injection Array Dynamic Characteristics

The first mode resonant frequency of that portion of the array retained during Mars orbit shall be greater than 5 cycles per second and less than 20 cycles per second.

2.5 MATERIAL SELECTION CRITERIA

The structural, electrical, thermal control, and lubricant materials used in the solar arrays must withstand exposure to the following simulated environments without a loss in any critical design property. These requirements are to be used for material selection purposes only and not for design (see Section 2.4).

2.5.1 Storage at 95% relative humidity at 30^oC for 50 hours.

2.5.2 150 thermal cycles between -20^oC to 60^oC at a rate of change that permits temperature stabilization without excessive thermal shock.

2.5.3 10 thermal cycles between -200^oC and 90^oC at 10^{-7} torr with a one-hour cycle and a temperature stabilization dwell at the extreme temperatures.

2.5.4 The materials must also resist the flight environment without releasing any condensing gases which would decrease the solar cell efficiency or cause electrical shorts, or cause any degradation to spacecraft system operation.

2.5.5 Structural Materials Criteria

The structural components of the various configurations and arrays should have a reliability of 0.999+ with a confidence level of above 90%. To meet this requirement, the material selection and design configuration must consider not only the load conditions of launch and space maneuvers, but also the loads induced by thermal excursions and micrometeoroid impact. The study of the various arrays will consider means of minimizing or isolating thermally induced loads and will attempt to minimize the critical impact area of the structure.

The dosage and energy levels of the particulate radiation encountered during this mission will not produce a significant effect on the metallic structural elements. Polymeric materials will either be shielded or selected to resist a radiation dosage of 10^7 rads without decreasing the critical design properties below the design allowables.

2.5.6 Adhesive Criteria

2.5.6.1 Structural

The structural adhesive must be resistant to all of the mechanical, vibrational and thermal loads induced on the array. When used to bond transparent or partially transparent structural components, the adhesive must also resist particulate radiation of 10^7 rads and ultraviolet radiation equivalent to 245 days of solar radiation at a rate of 2.002 calories/cm^2/minute. A prime requirement of any selected adhesive system is the feasibility of processing and the compatibility of processing procedures with the elements being bonded.

2.5.6.2 Solar Cell Adhesives

Two adhesive system requirements exist in the solar cell area. One is for an adhesive system to attach the solar cells to the structure, the second to bond the solar cell cover glasses to the cells. The requirements for attaching the solar cells to the structure vary with the various arrays under consideration, but in general must have the following properties: high thermal conductivity; low out-gassing in the vacuum environment; a modulus of elasticity compatible with the thermal motion of the cells and structure; and repairability during the fabrication phase.

The adhesive for bonding the cover glasses to the solar cells must be transparent to electromagnetic radiation in the wavelengths from 0.4 to 1.1 microns. The adhesive must be resistant to ultraviolet and particulate radiation to the extent that the transmittance to the solar cells shall not decrease more than 2% during the flight.

2.5.6.3 Thermal Control Coatings

These coatings must be resistant to the ultraviolet and particulate radiation of the flight environment such that no significant change will occur in the design values of absorptance and emittance.

2.5.6.4 Bearings and Lubricants

The bearings and lubricants for use on the various array configurations must function a minimum of 10 times in a 1 "g" earth environment without failure or part replacement. The bearing materials must resist the thermal excursions and particulate radiation of the flight environment without a change in the critical dimensions or the release of any condensing gases.

2.6 ELECTRICAL POWER CRITERIA

Design criteria are presented for solar cells, solar cell modules, cell connections, busses and terminations and for installations. Discreet values are given wherever possible, but in some cases the exact value will be the result of a trade study. Voltage and power requirements for the Mars mission are shown in Figure 2.6-1.

2.6.1 Solar Cell Modules

The solar cell modules will be designed and evaluated in accordance with the following criteria and assumptions.

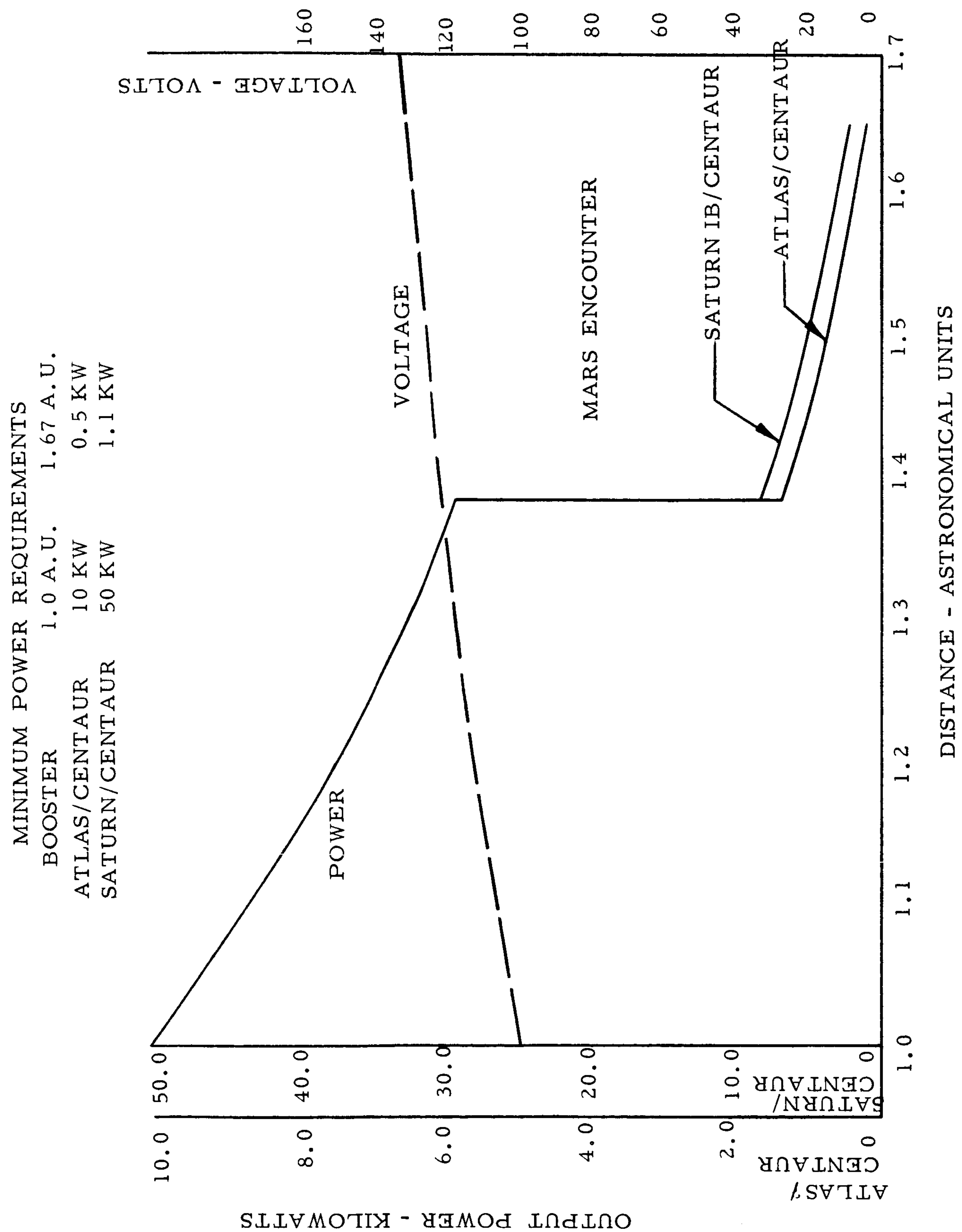
2.6.1.1 Power Output Determination

Power output of the array will be determined by the following equation and assumed design constants:

$$P = (\eta)\,(A)\,(S)\,(P)\,(K_1)\,(K_2)\,(K_3)\,(K_4)\,(K_5)\,(K_6)\,(K_8)\,(K_9)$$

$$1 - (T_C - 28)\,(K_7)$$ Where:

η = efficiency of cell at air mass one under standard test conditions.
A = gross area of panel in square feet.
S = solar intensity falling on cell in watts per square foot.
P = ratio of active cell area to gross panel area
K_1 = factor correcting cell efficiency from air mass 1 to air mass 0 = 0.85
K_2 = factor to account for cover glass losses = 0.98
K_3 = factor to account for process degradation = 0.985
K_4 = factor to account for mismatch losses due to spectral response deviations = 0.98



31 MINIMUM SOLAR ARRAY OUTPUT REQUIREMENTS AT VARIOUS SUN DISTANCES

FIGURE 2.6-1

K_5 = factor to account for solar constant uncertainty and deviation = 0.945

K_6 = factor to account for standard cell calibration and panel test errors = 0.96

T_C = temperature of cell in space in degrees C

K_7 = temperature efficiency coefficient = 0.45%/degree C

K_8 = factor to account for environmental degradation = 0.95

K_9 = factor to account for cosine effect of array misorientation to the sun-probe line. The determination of array nominal electrical power output will be made using zero degree misorientation.

Factors K_3, K_4, K_5, and K_6 are random variables which are unlikely to be all at either the maximum or minimum value. The RMS equivalent of the composite variances of these factors is:

$$K_C = \sqrt{1 - (1 - K_3)^2 + (1 - K_4)^2 + (1 - K_5)^2 + (1 - K_6)^2}$$

$$= \sqrt{1 - (.015)^2 + (.02)^2 + (.055)^2 + (.04)^2} = 1 - .0724 = 0.928$$

2.6.1.2 Cell Efficiency

Power output calculations will be based on N/P 12 mil cells having a median efficiency of 12% tungsten-equivalent air mass 1. Efficiencies of cells having different thicknesses will be based on Heliotek Report No. 1 PO BK 4-312938.

2.6.1.3 Electrical Insulation

The electrical insulation between the solar cells and the metallic substrate or the ground plane will provide a minimum breakdown strength in air at standard temperatures and pressure conditions greater than 3 times the open circuit voltage of the panel. Leakage resistance under the same test conditions will be greater than 10^9 ohms per square centimeter of cell area.

2.6.1.4 Thermal Conductance

The solar cell stack shall be constructed so that the temperature difference between the back side of the cell and the metallic substrate or back side radiating surface will not exceed 5 degrees C with an incident intensity upon the cell of 140 milliwatts per square centimeter.

2.6.1.5 Repairability

The modules will be constructed and materials selected so that defective cells can be replaced in a manufacturing repair area without damage to adjacent cells, electrical insulation, or mounting substrate.

2.6.1.6 Testing

Test terminals shall be provided on each sub-panel to permit ground testing and checkout prior to launch, in a 1 "g" earth field with suitable ground support equipment.

2.6.1.7 Packing Factor

The spacing between adjacent solar cells in a parallel connected group shall not exceed 0.01 inches. Spacing between series connected modules should not exceed 0.030 inches with front connected cells or 0.010 inches with back connected cells. The length and width of the module shall be equal to (NC + 1/2) inches where C = gross length or width of a cell group and N is some interger. The number of 2 x 2 centimeter solar cells per square foot of gross area shall not be less than 217 front connected cells or 221 back connected cells.

2.6.1.8 Compatibility of Materials

The stack shall be designed to use only materials that are compatible with each other and with the space environment and interface requirements.

2.6.1.9 Tolerances

All tolerances will be assumed to be normally distributed about the mean value. Overall tolerances due to variations in the dimensions of the various components comprixing the stack shall be taken as equal to the rms of individual variances. Weight and performance calculations will be based on the sum of the mean values plus the rms of the variances.

2.6.1.10 Interconnections

The cells will be interconnected both in parallel and in series by a metallic conductor. This conductor shall be designed to minimize both thermal and flexual stresses on the cell. The resistance of the interconnection plus solder shall not exceed 2% of the total series resistance of the cell. The joint shall have a strength equal to or greater than the strength of the bond between the silicon and the ohmic contacts. The joining materials shall exhibit stable physical and electrical characteristics in both space and terrestrial environments.

2.6.1.11 Magnetic Field

Solar cell wiring and interconnecting techniques will be designed to minimize the magnetic field produced by the flow of current in the solar array. No magnetic materials will be used in the module stack. The module will be designed to produce a magnetic field (MMF) of less than 1 gamma at a distance of 3 feet from the module.

2.6.1.12 Solar Cell Characteristics

Power outputs, radiation protection, and area requirements will be based on the use of single crystal silicon cells. Efficiency, temperature and temperature relations will be determined by tests conducted upon sample 8 mil cells supplied by JPL.

2.6.1.13 Modules

Two types of module assemblies will be designed. One module assembly will provide auxiliary electrical power to the spacecraft. The output of auxiliary power modules will be nominally 28 volts at maximum power in space at 1 A.U. The other modules will provide power to the propulsion system and will not exceed 100 volts at maximum power at 1 A.U.

2.6.1.14 Cover Glasses

Stack design will be initially based on the use of 6 mil cover glasses. Alternate materials to be considered will include 3 or 4 mil micro sheet, 2 mil glass integrated with the cells, and various transparent plastics.

2.6.1.15 Stack Weight

The lightest and thinnest materials will be used in the stack which are consistent with (1) the functional requirements of the stack, and (2) present and near future fabrication and production technology.

2.6.2 Conductors

The configuration of electrical conductors shall be determined with regard to the following considerations.

(a) Minimum possible weight
(b) Minimum resistivity
(c) Minimum magnetic field
(d) Mechanical strength to endure launch environment
(e) Allowable voltage drops to be determined by trade studies
(f) Exterior finish to be resistant to natural and induced environments
(g) Process adaptability
(h) Redundancy
(i) Thermal coefficient considerations.

2.6.3 Conductor Insulation

Selection of conductor insulating materials shall be made with regard to the following considerations:

(a) Broad spectrum radiation resistance
(b) Heat resistance compatible with manufacturing and assembly processes
(c) Mechanical strength
(d) Notch sensitivity
(e) Flexibility
(f) Dielectric characteristics
(g) Ease of forming or fabricating
(h) Cost
(i) Flight thermal considerations

2.6.4 Electrical Terminals

Terminals shall be used to facilitate maintenance, repair and replacement of electrical components. The following requirements shall be observed.

(a) Voltage drop across any terminal shall not exceed 25 millivolts at rated load.
(b) The terminal shall withstand 50 cycles of manual mating and unmating without replacement of parts.
(c) The terminal shall be accessible for ease of wiring installation and for factory or field checkout.
(d) It shall be rigidly attached to primary or secondary structure.
(e) It shall have minimum possible weight.
(f) Exterior finish of the terminal shall be resistant to both natural and induced environments.

2.6.5 Installation

The installations of wires, terminals, electrical connectors and busses shall conform to the following requirements:

(a) Busses and other wiring shall be installed to minimize magnetic fields.
(b) Installations shall withstand rigors of normal handling and transportation as well as launch and operational maneuvers.
(c) Installation shall be designed to facilitate service and repair activities.

2.6.6 Electroexplosive Devices

The design, installation and test of wires which control the initiation of electroexplosive devices shall meet requirements of AFETRP 80-2, "General Range Safety Plan", Volume 1, paragraphs 3 and 4 of Appendix A.

2.6.7 Electric Connectors

2.6.7.1 All connectors shall have removable crimp type contacts which shall be removed from the back of the connector.

2.6.7.2 All connectors shall have cable strain-relief provisions.

2.6.7.3 All connectors shall have positive moisture seals at the wire entry and at the mating interface.

2.6.7.4 The connector half which contains the socket contacts shall have the following characteristics:

(a) The exposed insert insulator shall be made of hard plastic material.
(b) Socket contact entry shall be protected by the hard plastic insulator.

2.6.7.5 The connector half which contains the pin contacts shall have a rubber insulator at the mating interface.

2.6.7.6 The exterior finish of each connector shall withstand the following conditions.

(a) All common terrestrial environmental conditions listed in NAS 1599, Connectors, General Purpose, Electrical, Environment Resisting.
(b) Exposure to zero mass air density for one year without sublimation.
(c) Fuels used as propellants in launch vehicles.

2.7 MECHANISM RESTRAINTS

2.7.1 The solar array, including release and deployment mechanisms, must fit within the solar array envelope specified for the applicable configuration (Atlas/Centaur nose fairing or Saturn IB/Centaur nose fairing).

2.7.2 In the stowed configuration, the solar array must be protected from any damage which could be caused by shock and vibration during launch, and must be supported in such a way that shock and vibration loads are transferred to spacecraft structure.

2.7.3 Release and deployment mechanisms must withstand the launch environment without damage, and upon command and in proper sequence, must release solar array restraints and extend and lock the array into its deployed position.

2.7.4 In the deployed configuration, under steady-state conditions, the solar array must have sufficient rigidity so that by controlling the attitude of the spacecraft the array can be oriented and maintained in a plane normal to the direction of the sun within $\pm 10^{\circ}$. This tolerance shall include deflections from static spacecraft load inputs and thermal gradients, but shall not include deflections due to dynamic load inputs.

2.7.5 Release, deployment, and locking mechanisms must withstand and be capable of functioning mechanically in the space environment after launch and 4 hours of space flight. They shall function structurally for the duration of the mission.

2.7.6 Release, deployment, and locking of the solar array shall not cause release of loose parts or gases which could damage or impair the function of the solar array or other spacecraft subsystems.

2.7.7 Release, deployment, and locking mechanisms shall be designed so that, with suitable test equipment, their operating function can be checked in a 1 "g" environment. Mechanisms shall be capable of operating a minimum of 10 times in a 1 "g" field in a vacuum of 10^{-5} torr without failure or part replacement.

2.7.8 Since a reduced electrical power output is required in Mars orbit, the array mechanism system shall provide for release and jettison of the major part of the total array at approximately the end of the first 350 days of the mission and prior to Mars orbit injection.

The release and jettison shall accelerate the discarded solar array and any loose parts away from the spacecraft. The design of the jettison system shall minimize perturbing forces to the spacecraft.

2.8 RELIABILITY

2.8.1 Structural Reliability Goal

The probability that the structural members of the solar array will successfully function has been established as .999. This goal is not high for structures in general and is consistent with the design considerations discussed below. The array structure includes all of the array except the solar cells, electrical wiring, and deployment mechanisms, some of which become structural members of the array after deployment.

The failure modes being considered in the design of the array structure are listed below under the various conditions and environments to which the array will be subjected.

Shock loadings and strains associated with packaging, handling, transporting, and installing (array in stowed configuration in 1 "g" earth environment.

- Structural misalignment resulting in deployment difficulty.

- Panel warping, splitting, or loosening causing electrical discontinuities, cell fractures, and panel misorientation.

Extended storage in earth environment:

- Corrosion or deterioration of structural members due to high humidity.

Launch environment:

- Failure of the mounting members supporting the stowed array.
- Premature deployment.
- Structural misalignment that would hamper deployment.
- Thermal dynamic heating.
- Panel bending or buckling affecting array flatness when deployed, and causing cell failures.
- Excessive vibration loads reaching panel causing fatigue damage to cells or interconnections.

Deployment at 0 "g":

- Partial deployment
- Shock load in the structure when the array sections arrive at the limit stops, should motion retarding snubbers fail to function.

Jettison of part of the array prior to retro maneuver:

- Deformation of the panel supporting members affecting panel flatness and cell integrity in the remaining panels because of excessive severing loads or failure of severing devices.

Retro maneuver:

- Deformation of supporting beams or fixtures attaching array to spacecraft resulting from acceleration load.
- Vibration damage.

Planet orbit:

- Excessive strains on panels through thermal expansion cycling of the structure.

2.8.2 Mechanical Reliability Goal

The probability that the mechanisms will successfully deploy 100 percent of the array one time was established as .999+. This goal is based on analysis of a typical set of deployment mechanisms that might be used with a large array.

Part	Failure Data Failures per Cycle	Quantity	Failures for Single Deployment	Source of Data
Actuator (electric winch)	.00004	4	.00016	Lundy Electronic
Release Mechanism (squib pinpuller)	.00020	12	.00240	AVCO Reliability Data Series
Linkage (hinges, cables, pulleys, etc.)	.00005	4 sets	.00020	Engineering Judgement
Latch (Locking)	.000006	48	.00029	FARADA

The figures in the above table give a reliability of .9969 for 100 percent deployment. The squib release mechanism affect the reliability most heavily. Making all 12 release mechanisms redundant gives .9993 for reliability, which is the basis for the goal shown above.

This reliability goal may be conservative. It is recognized that the failure rates used in the above table are the result of wear out (except in the case of the squib release mechanism). Wear out is not a likely mode of failure during operation of the array deployment mechanisms. Furthermore, these mechanisms will have passed the high failure rate period of burn-in during assembly and testing. More probable failure modes of the mechanisms would be malfunctions resulting from stresses imposed during launch (misalignment, electrical discontinuity, etc.), or from cold welding after expsoure to space environment. The failure rates may be considerably lower than those given in the table above. Therefore, the reliability goal of .999+ appears reasonable.

The reliability goal for successful jettison of part of the array prior to a retro maneuver was established as .9999. The basis for this goal is that pyrotechnic devices will perform detachment in twelve places. If the failure rate of a pyrotechnic device is .001 failures per cycle after long exposure to space environment, and if a redundant device is used at each point of detachment, the above reliability goal can be achieved. Without redundance, the expected reliability would be .990.

2.8.3 Electrical Reliability Goal

The reliability goal for the solar cells and electrical interconnections was established by analysis based on typical solar arrays having approximately the circuitry and number of cells as employed on this program. The goals shown below correspond to five percent power loss which was selected as a common basis for the various arrays. The reliability of the 50 KW array does not differ significantly from .98 for power loss as small as 1.0 percent, and for the 10 KW array, there is no significant reliability change down to two percent loss.

Array	Time Days	Maximum Power Loss Per Cent	Reliability
50 KW	350	5.0	.98
1.1 KW	490	5.0	.98
10 KW	350	5.0	.95
0.5 KW	490	5.0	.995

The arrangement of the solar cells in the typical arrays considered in the reliability analysis was seven cells connected in parallel in each row and 200 such rows connected in series. There were as many groups of 200 rows each as required to obtain the rated power.

The following conditions and assumptions were made to perform the analysis:

- Individual cell failure rate is 0.1×10^{-6} failures per hour.
- Failure distribution is exponential.
- Failure occurrence is random with respect to location in the array.
- Primary mode of failure is open circuits. Shorts to ground and cell short circuits will be negligible.
- The above failure rate includes the effects of micrometeoroid damage, differential thermal expansion, and time-dependent defects of the solar cell and its interconnections.

The reliability goal for the electrical busses and connections is .998. The principal mode of failure will be open circuits at bus connections to blocks of solar cells. The failure rate of the conductors, including crossover strips at hinge lines, was considered negligible.

The failure rate for a hand-soldered bus connection in space environment is $.0001 \times 10^{-6}$ failures per hour, derived from data obtained in the Miniteman Program. If there are 2,000 connections in the 50 KW array, the probability of having zero failures for 350 days is:

$$e^{-(2,000)(.0001 \times 10^{-6})(8,400)} = .998$$

This figure is conservative because a small number of failures could occur, resulting in only a small reduction in array power output.

2.9 INTERFACES

Interface coordination is required to establish compatibility between the spacecraft and the array in the following areas.

2.9.1 Spacecraft - Array Structural Interfaces

2.9.1.1 Attachment Points

The solar array design shall be compatible with the solar array to spacecraft attachment points.

2.9.1.2 Materials Compatibility

The solar array structural materials shall be compatible with the spacecraft materials, spacecraft instrumentation, and the mission requirements.

2.9.1.3 Thermal Interaction

The spacecraft - solar array thermal interaction shall be considered on a system basis so as to provide the most efficient combination capable of performing the mission.

2.9.1.3.1 The solar array design shall not be unilaterally restricted by the spacecraft thermal control requirements.

2.9.1.3.2 The deployed array shall experience no shadowing from the earth, moon, other planetary bodies or spacecraft subsystems prior to Mars rendezvous.

2.9.1.4 Dynamic Coupling

The solar array structural design shall be such that dynamic coupling of the solar array with the spacecraft guidance and control equipment is precluded.

2.9.1.5 Mass Center Location

The solar array shall be designed so that displacements of the vehicle mass center due to solar pressure, thermal gradients, and array temperatures are minimized consistent with other array requirements.

2.9.2 Spacecraft Subsystem - Array Interfaces

Design of the array and spacecraft shall take into consideration the following interface problems.

2.9.2.1 Clearance with the exhaust from attitude control jets, retro-rockets and electric thrusters.

2.9.2.2 Satisfactory antenna view angles for communication with earth or a lander vehicle in space or on Mars.

2.9.2.3 View angles for cameras or sensors on the spacecraft.

2.9.2.4 Electrical, magnetic, and radiofrequency interference limits.

2.9.2.5 Space envelope interface, for both stowage and deployment between the arrays, spacecraft, booster, and shroud.

2.9.2.6 The spacecraft attitude control system shall maintain the solar array plane normal to the sun line within $\pm$ 5 degrees.

2.10 MANUFACTURING RESTRAINTS

The manufacturing feasibility of the solar array will be directly proportional to the producibility of the design. The basic restraint that the array "be fabricated with materials and technologies which have been developed, or can be developed to production readiness within one year from the date of issuance of this contract" limits the manufacturing technology to modest extensions of the existing state-of-the-art. Based upon this premise, the proposed designs were reviewed and evaluated to the following criteria of restraints.

2.10.1 Configuration and size of parts must be compatible with normal tooling practices. Very thin, foil gage, parts must be capable of being fabricated with reasonable assurance that damage will not occur and that the part can be handled without damage when reasonable precautions are taken.

2.10.2 Solar panel assembly and solar cell installation normally require the extensive use of bonding materials. The thickness and area of application of these materials must be accurately controlled. These materials, however, are inherently difficult to control in manufacturing. The designs and processes must include control requirements and tolerances that can be maintained in the manufacturing shops.

2.10.3 The control of solar cell processing through the manufacturing shops is dependent upon the comparison of initial testing and grading to subsequent cell testing during the manufacturing sequence. The tolerances set by the design must be adequate to allow a reasonable yield of good assemblies.

2.10.4 The heat required in joining solar cells by soldering can cause degradation in cell performance. The solar cell connecting system must be compatible to low temperature soldering methods and accurate temperature controlling.

2.10.5 The quantity of solar cells to be installed will be of a magnitude that will require some degree of mechanization to reduce cost and flow time and to increase reliability. The extent to which this can be accomplished will depend upon how well the design can be adapted to mechanized techniques for solar cell joining, cover glass assembly, and solar cell testing.

2.10.6 The installation of solar cell assemblies on substrate panels, and the assembly of structural component parts must be accomplished with protective coverings on the operators hands or the handling must be done with suitable mechanical devices. The configuration of these assemblies must be such that the required work can be accomplished while complying with the handling restrictions.

2.10.7 The configuration of the complete array must be such that fixturing for positioning and holding of components and subassemblies can be accomplished to provide support during array assembly.

2.10.8 Manufacturing must be able to repair or replace any component of the solar array at any step during the fabrication sequence. The extent of repairability will be determined by the ease of access to the damaged part plus the possibility of damage to adjacent parts when the repair is made.

2.11 GROUND HANDLING REQUIREMENTS

2.11.1 The ground handling requirements for the solar array will be determined by functional analysis and specific items of standard and special support equipment will be identified. The functional analysis shall cover the following areas.

2.11.2 The program for the development of the solar array structural subsystem.

2.11.3 The program for the development of the flight prototype solar array including electrical testing.

2.11.4 The pre-flight operations with the solar array at Cape Kennedy.

3.0 PANEL TRADE SUMMARIES

The following summarizes the results of the electrical and structural parametric evaluations and analyses. Rigid, semi-rigid and rollup panels utilizing both back and front connected solar cells of varying thicknesses were investigated. The effects of these parameters on the power per pound and per square foot ratios for 10 to 50 KW arrays using several different panel configurations are shown.

Results of studies to determine optimum cell thickness, cover glass thickness, dielectric material, radiation effects, conductor material and conductor sizes are presented. 8 mil, N on P back-connected silicon solar cells and 4 mil micro-sheet cover glasses were selected. H-film was found to be suitable for solar cell insulation. Conductor cables with tape-wrapped H-film offer advantages in regard to weight and heat resistance. Aluminum conductors offer substantial weight saving over copper, but introduce more problems in component design.

The parametric structural analyses, conducted for the rigid solar panel substrates, were based upon an equivalent simply supported beam subjected to sinusoidally varying acceleration at the support points. For such an analysis the dynamic stiffness term (EI/m) required of the structure can be related to the input acceleration leve, the simply supported span length, and the critical stress for the structure.

Parametric structural analyses of membrane solar panel concepts were conducted. Among the parameters investigated were the solar cell installation weight (W_C), the maximum membrane tension load (T), the maximum acceleration in the membrane (Gq), the panel aspect ratio (A/B), and the panel size AxB .

The results of these analyses indicate that, for the panel sizes of interest in this program, the dynamic amplitudes of the membranes will be negligible.

Thermal analyses of four membrane solar panel concepts have been conducted for a range of solar intensities. The results illustrate the dependence of the solar cell temperature upon the solar intensity for deep space conditions.

3.1 SOLAR CELL MODULE EVALUATIONS

This section summarizes the trade studies and parametric evaluations performed on solar cell stack configurations and components. The results of these analyses have been presented in detail in bi-monthly reports D2-23942-1 and D2-23942-2.

In summary, the studies that were performed during the course of this contract include:

(a) P/N cells versus N/P cells
(b) Cell efficiency
(c) Cell thickness
(d) Cover glass thickness
(e) Cover glass ultra violet filter cut-off
(f) Front versus back or wrap around contacts
(g) Magnetic field

3.1.1 Optimum Cell Type

One of the first decisions required was to select whether P/N or N/P cells be employed on the array for this program. Figure 3.1.1-1 presents the results of a part of this study. It shows the relative power density attainable with both types of cells at 1.0 A.U., as a function of temperature. P/N cells exhibit a higher initial efficiency and a lower temperature efficiency coefficient.

Figure 3.1.1-2 summarizes two sets of data. Curve group 1 shows the relation between the degradation experienced by different types of cells during a 350 day mission to Mars as a function of cover glass thickness. These curves indicate that despite the higher initial efficiency of P.N cells, the N/P cells will still deliver more power at Mars. For this reason, N/P cells were selected.

Curve 2 shows the effect of cover glass thickness on the ratio of the power output at Mars to the total array weight. This curve indicates that optimum thickness for this mission would be 2 mils and that it is desirable to use the thinnest glass available down to 2 mils. Two mil cover glasses are not considered state-of-the-art at this time. The minimum thickness commercially available is 4 mils. Therefore, 4 mil thick cover glasses were selected.

3.1.2 Optimum Cell Thickness

Figure 3.1.2-1 summarizes two sets of data which show the effect of solar cell thickness on efficiency and power-to-weight ratio. Curve 1 illustrates the fashion in which cell conversion efficiency varies with cell thickness. Note that cell efficiency decreases rapidly with cell thicknesses less than 10 to 12 mils. Curve 2 shows the manner in which the power-to-weight ratio varies with cell thickness. This analysis includes both the weight of the cells and the estimated change in structure weight caused by the change in cell weight. Note that the maximum power to weight ratio is achieved with 8 mil cells. Therefore, 8 mil cells were chosen for the design.

3.1.3 Optimum Ultra Violet Filter Cut-Off

Figure 3.1.3-1 illustrates the change in power output caused by shifting the cut-off point of the ultra violet reflection filter on the cover glass. This curve takes into account both the change in temperature and light intensity seen by the cell. Note that maximum power is obtained with cut-off wavelengths of 0.35 microns and shorter. Note also that the greatest power output is obtained by using the shortest cut-off point that provides ultra violet protection to the cover glass cell adhesives. Tests have shown the RTV 602 can tolerate wavelengths down to 0.4 microns without degradation. Therefore, the ultra violet filter cut-off selected is 0.4 microns.

It should be noted that blue-red filters were also examined. The analyses indicated that the reduced transmission of these filters to wavelengths between 0.5 to 0.9 microns offset any possible benefit derived from the reflection of infra-red energy. Power output with these filters was about 2% less than that obtainable with standard blue reflecting filters. Therefore, the blue reflecting filter only was selected.

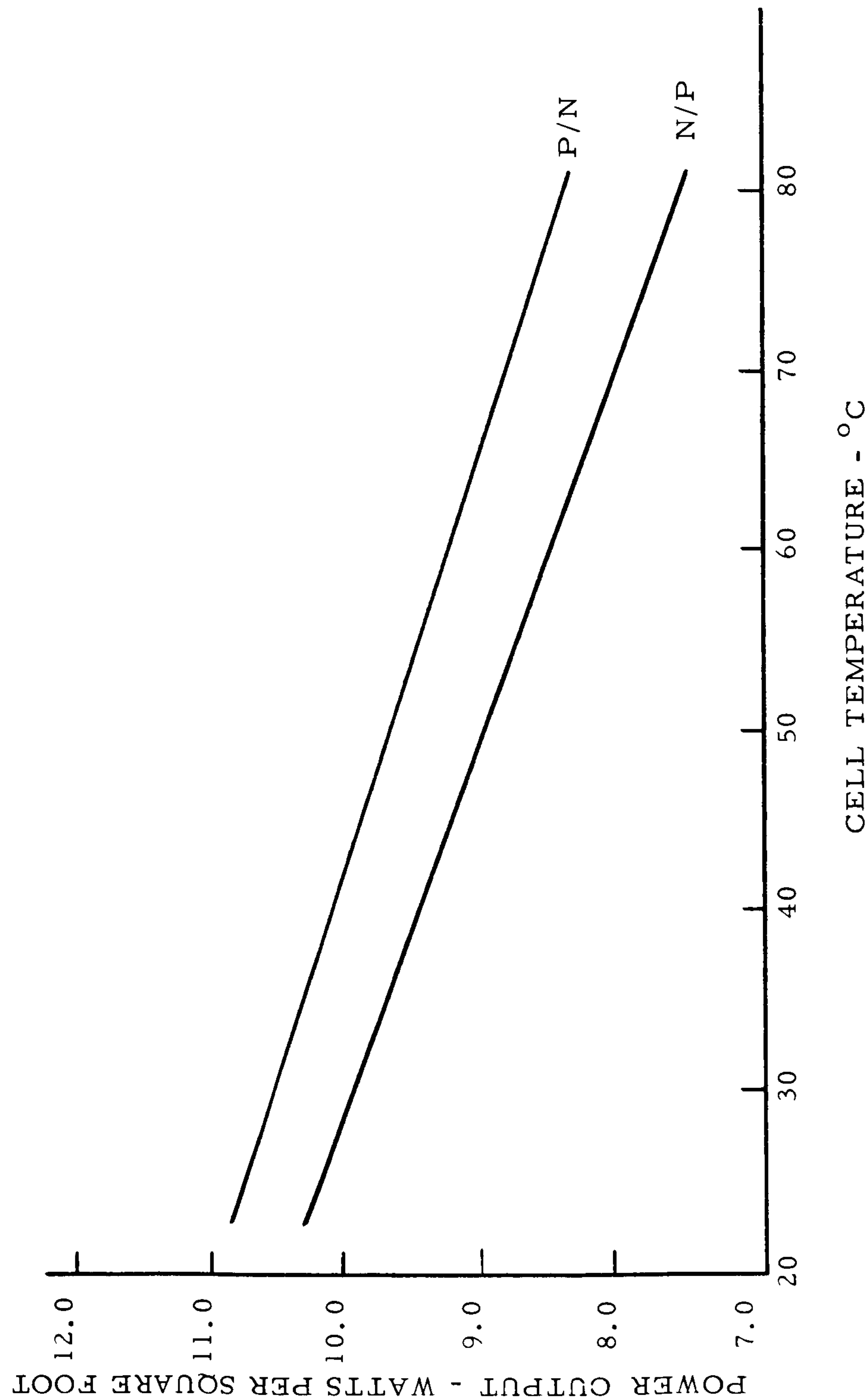
3.1.4 Cell Size and Connection Type

A cell size of 2 x 2 cm was selected for two reasons:

(a) A slightly higher packing density can be achieved by eliminating the gap between two 1 x 2 cm cells.

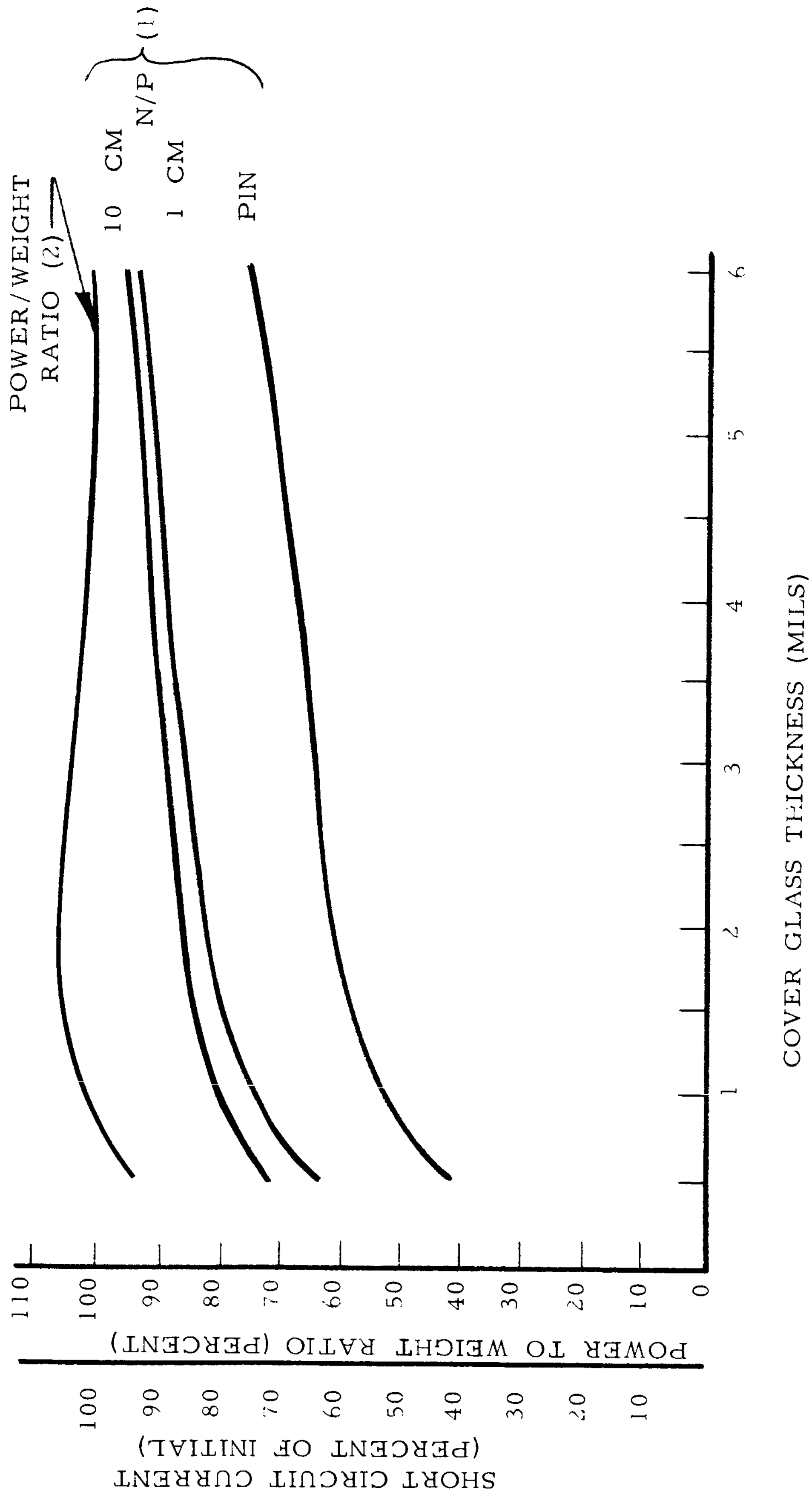
(b) A saving in the purchase and installation cost can be realized. 2 x 2 cm cells cost only 50% more than 1 x 2 cm cells yet produce 100% more power. Installation costs will be lower since only half as many cells must be handled.

Back-connected cells were selected over conventional front-connected cells. Back-connected cells have 5% more active area than front-connected cells. This results in a 4% increase in power.

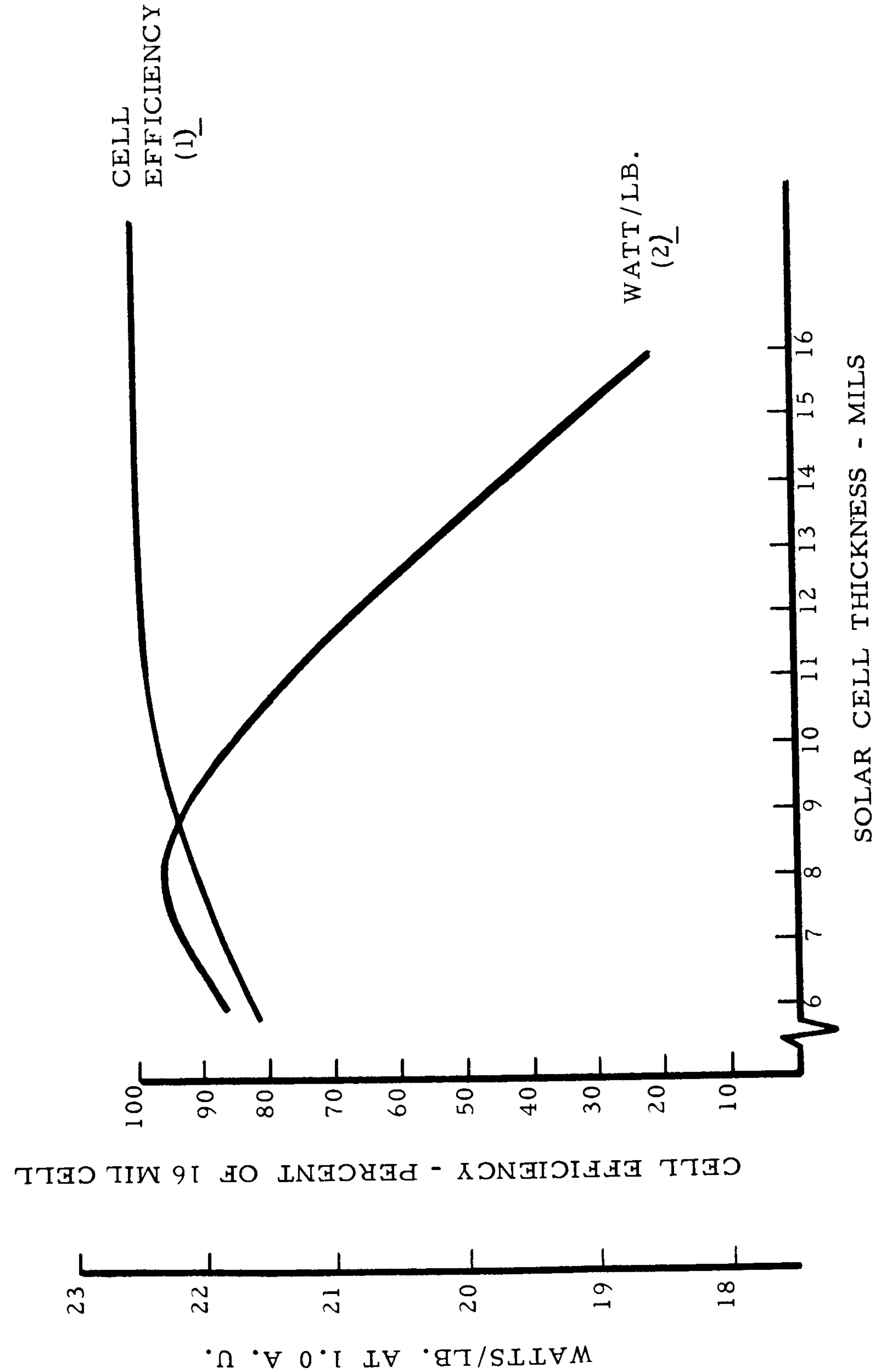


CHARACTERISTICS OF N/P
& P/N SOLAR CELLS

FIGURE 3.1.1-1

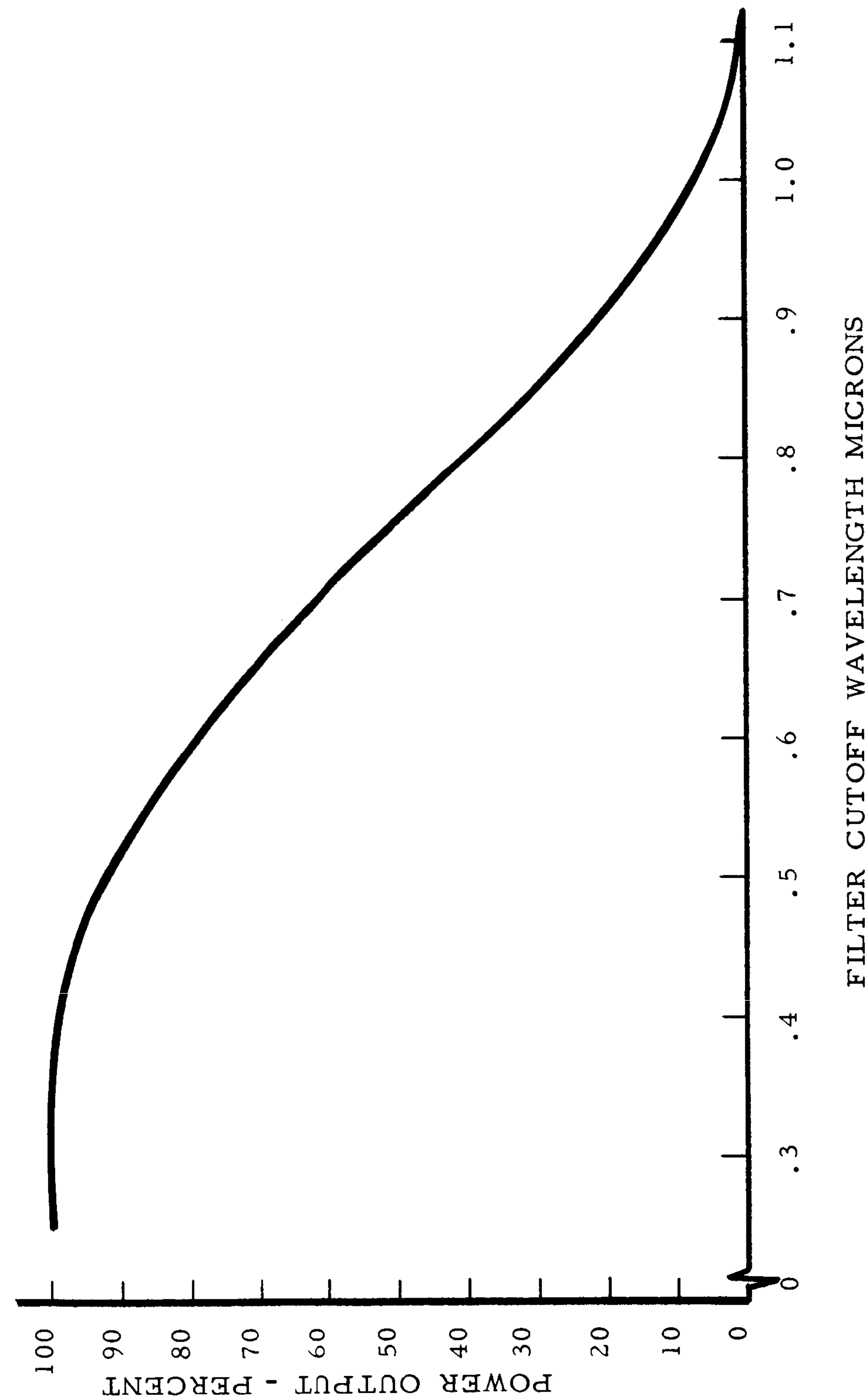


COVER GLASS
THICKNESS STUDY
FIGURE 3.1.1-2



SOLAR CELL THICKNESS STUDY

FIGURE 3.1.2-1



COVER GLASS ULTRA VIOLET FILTER STUDY

FIGURE 3.1.3-1

3.2 ELECTRICAL BUS EVALUATIONS

Trade studies were performed to select the materials and configuration for the electrical busses. Candidate designs were compared for performance capabilities, environmental compatibility, reliability in operation and for cost.

3.2.1 Candidate Metals for Conductors

(a) Copper
(b) Copper Alloy
(c) Silver (for comparison only)
(d) Aluminum
(e) Beryllium

<u>Resistivity</u>

Criteria for resistivity must include weight considerations. Aluminum and beryllium provide equivalent conductivity to copper with one half the weight and this is a strong factor in favor of these two metals.

<u>Tensile Strength</u>

Copper Alloy No. 63* is 98% copper, but the addition of very small amounts of several other metals provides more than double the tensile strength of copper. The tensile strength of beryllium varies over a wide range and the value chosen for Table 3.2-1 is considered to be conservative for thin sheet stock. Tensile strength is not the strongest factor for the choice of a conductor material, but a high tensile strength is desired for small conductors.

3.2.2 Manufactured Forms

(a) Sheet or Strip
All materials listed in Table 3.2-1 can be procured in sheet or strip form to closely controlled dimensions. Insulated metal strip is considered for non-rigid configurations.
(b) Bar, Rod and Tube
All metals can be procured to the required dimensions and tolerances.
(c) Drawn Wire
All materials except beryllium can be drawn at reasonable cost. Drawing beryllium into fine wire for use in stranded cables presents two major problems: (1) the extreme stiffness and lack of ductility complicate manufacturing; and, (2) there is no experience in the application of primary insulating material.

* Trade name of Surprenant Wire Company

3.2.3 Manufacturing Methods

(a) Solderability

The use of solder techniques will be required for connecting solar cells and for attaching some electrical busses to solar cell modules. Conductor materials which are not inherently solderable will require plating or equivalent treatment. Aluminum can be soldered, but beryllium requires solder temperatures which would injure solar cells. Use of either of these two materials would involve plating such as silver or possibly nickel.

(b) Pressure Termination

Terminations which depend upon physical pressure for electrical contact and continuity are vulnerable to the cold flow characteristics of malleable metals. All elements of the terminal which are in stress should have the same thermal coefficient of expansion. As shown in Table 3.2-1, aluminum requires consideration of this characteristic for component design.

(c) Bonding

All metals which have been selected as candidates for conductors can be bonded to the various substrate materials by qualified processes.

(d) Thermal Expansion

A low coefficient of expansion is desired because temperature extremes are large enough to produce high physical stress. Beryllium and copper are the two most satisfactory and aluminum the least satisfactory.

3.2.4 Emissivity

The emissivity of all conductors is too low to be a factor in the choice of a material. Surface treatment can be applied to improve total emissivity. Design objectives require the highest possible emission in the infra-red to keep conductors at the lowest possible temperatures. Anodized aluminum with the surface protected by a vacuum-deposited coating will provide an emission of 0.8 to 0.85 and is highly stable in a space environment. The other metals can be treated to provide oxide coatings, but all such coatings must be removed and the conductor plated at each terminal.

3.2.5 Conclusions

Of all factors mentioned, resistivity per unit weight, manufacturability, and cost are considered important for selection of a conductor material. Aluminum is the lightest and copper is easiest to handle and is the most readily available. Cost for these two materials is approximately the same for standard configuration. In non-standard configurations, aluminum is generally more expensive. Use of beryllium would increase costs materially at all levels of manufacture. Where current carrying capacity is required, such as main feeder busses, aluminum is preferred to obtain all possible weight savings. Where small current carrying capacities are involved, copper would be used if it did not materially increase the weight.

PROPERTIES OF CONDUCTORS*

Conductor Material	Resistivity Micro Ohms per cm,			Density	Res. Per Unit Wt. 20°C	PSI Tensile Strength 20°	Solder-ability	Cold Flow	Coef. of Expansion $\times 10^{-6}$
	-100°C	20°C	55°C						
Copper	1.05	1.67	1.82	8.96	1.00	28,000	Excellent	Good	17.0
Silver	0.97	1.58	1.89	10.6	1.07	18,000	Excellent	Poor	19.1
Aluminum	1.35	2.69	2.84	2.70	0.507	5,000	Fair	Poor	23.5
Alloy #63	1.27+	1.91	2.19	8.94	1.13	60,000	Good	Good	17.0
Beryllium	3.58	4.30	4.51	1.85	0.53	50,000	Poor	Good	13.3

* VALUES DERIVED FROM METALS REFERENCE BOOK, SMITHELLS HANDBOOK OF THERMOPHYSICAL PROPERTIES OF SOLID MATERIALS; GOLDSMITH WATERMAN AND HIRSHHORN.

+ At - 65°C

TABLE 3.2-1

3.2.6 Secondary Cabling Weight

Secondary cabling weight has been estimated for the Saturn/Centaur semi-rigid rectangular array to be .0121 pounds/square foot based on uninsulated, tapered pairs of leads. The weights are shown below for one-fourth of the array.

Total copper on main panels	12.55 pounds
Connectors (15%)	1.88
Adhesive	.07
Shielded pair for panel release	.50
Side panel conductors	1.50
Total	16.50 pounds

3.2.7 Dielectric Insulation - Conductors

Benefits which would result from operating electrical conductors at high temperatures can be observed in Figures 4.3-1 and 4.3-2. Candidates for dielectric insulations which will operate at elevated temperatures are Teflon products TFE and FEP and Dupont H-film. Teflon products are heavier, have less radiation resistance and lower maximum operating temperatures than H-film. Recent developments of Teflon and H-film laminate constructions deserve serious consideration due to the resulting weight savings. Product development and testing are expected to be compatible with the time periods under consideration.

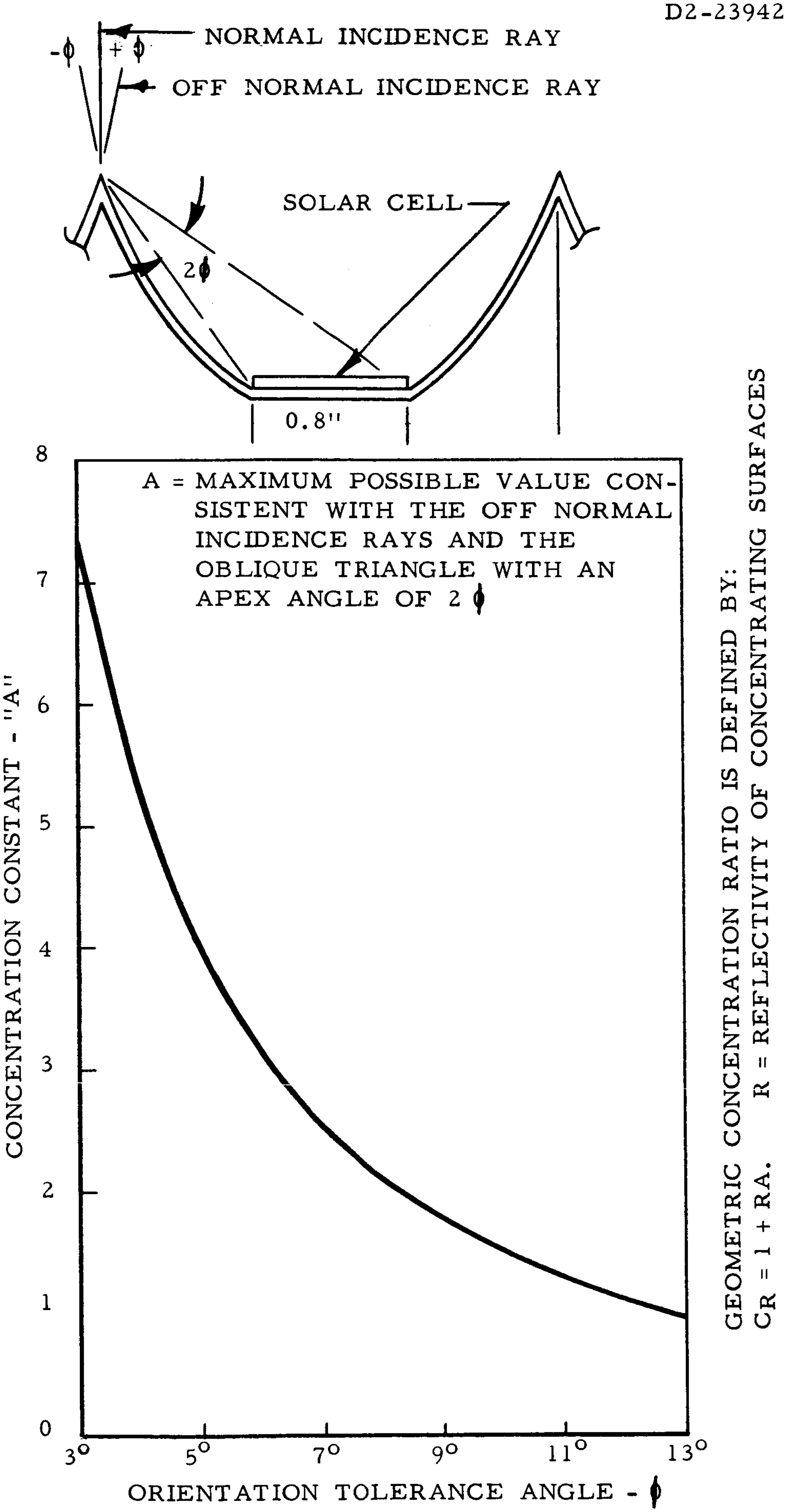
3.3 STRUCTURAL EVALUATIONS

Structural, thermal, dynamic, and stress analyses of many panel concepts were conducted. Among the concepts considered were: (1) concentrators, (2) flat sheet stringer, (3) flexible rollups, and (4) semi-rigid panels.

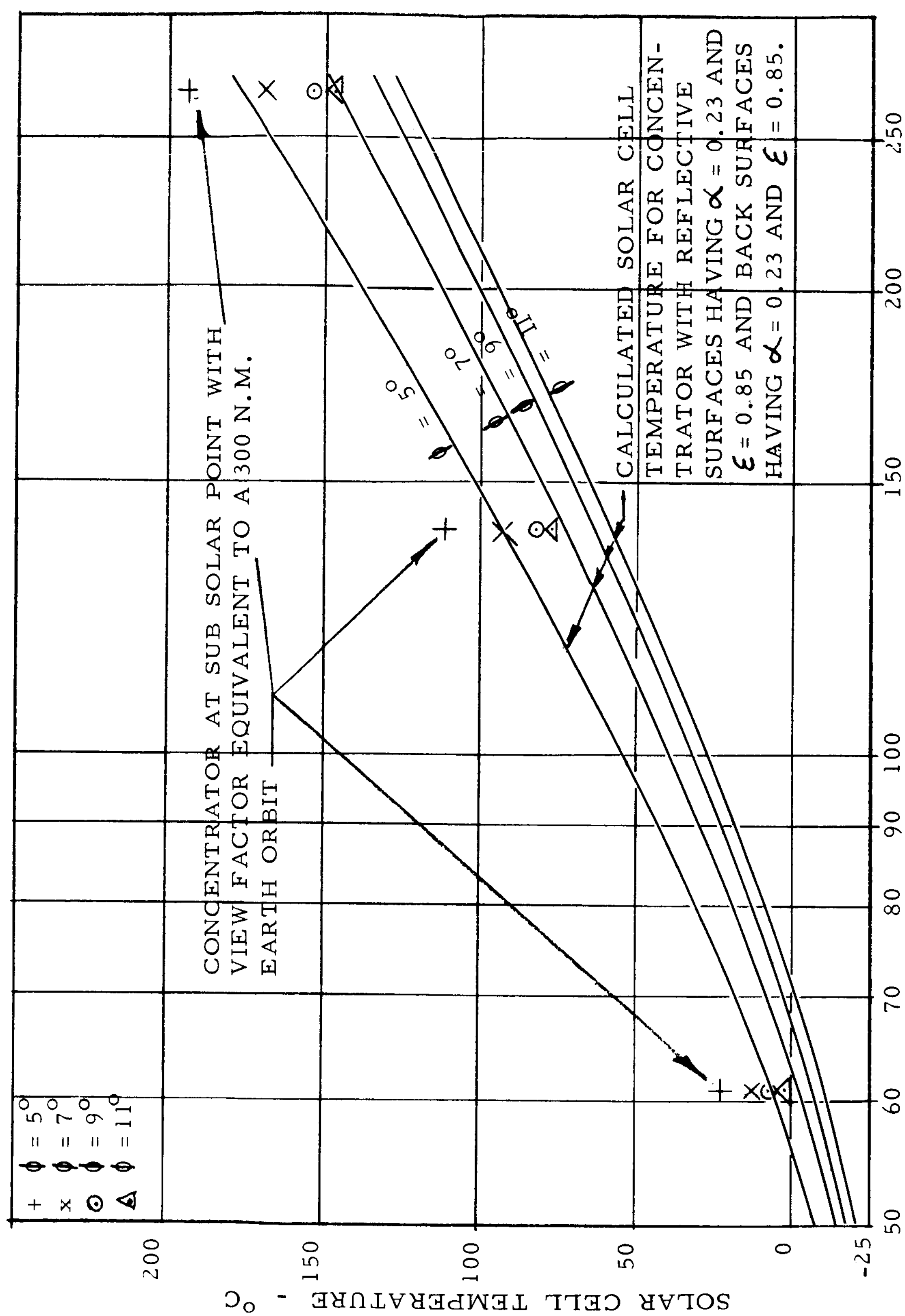
3.3.1 High performance alùminum concentrators mounted on beryllium supporting frames were evaluated. The performance of this concept is predicted to be between 16 and 20 watts per pound. The concentrator concept, with a family of possible orientation angles, is illustrated in Figure 3.3.1-1. Figure 3.3.1-2 summarizes the thermal analysis of the high performance concentrator panel. With the temperatures shown and the reflection losses expected, 8.9 watts per square foot can be expected. Therefore, if a 20 watt per pound array is to be fabricated of aluminum concentrator panels, the maximum unit weight of the assembly cannot exceed 0.45 pounds per square foot. Dynamic, structural, and stress analyses of the concentrator concept is summarized in Figure 3.3.1-3 where the unit weight versus span versus loading curve is shown for solar cell installation weights of $W_C = 0.3$ pounds/square foot of solar cell area. Assuming the mechanisms and supporting frames require as much weight as the panels, the maximum simply supported span for an aluminum concentrator capable of meeting the 20 watt per pound performance is between 20 and 30 inches.

3.3.2 Aluminum and beryllium sheet stringer designs similar to the Mariner 64 configuration were evaluated.

A wide selection of optimized structural designs were considered. The geometries were varied by changing the value for the parameter R. The structural concept shown in Figure 3.3.2-1 illustrates the significance of the parameter R. In these designs the optimum configuration was assumed to be one in which area 1 would just buckle elastically upon full reversal of the moment which buckled area 2. No crippling was considered in the analyses. The results of the analyses are summarized for this critical condition in Figure 3.3.2-2 for aluminum sheet stringer solar panel substrates and Figure 3.3.2-3 for beryllium sheet stringer solar panel substrates. Thermal analysis of these designs indicates that the expected performance would be approximately 10 watts/sq. ft. Consequently the maximum unit weight such a design can have if the performance goal is to be met is 0.50 pounds/sq. ft. Examination of Figure 3.3.2-2 for a Gq loading of 22 indicates aluminum sheet stringer concepts can not deliver 20 watts per pound at 1 A.U. intensity in practical solar array sizes. The beryllium unit weights shown in Figure 3.3.2-2 indicate this concept could deliver 18 watts per pound, and was not considered as being within the 1965 state-of-the-art for fabrication and material availability.



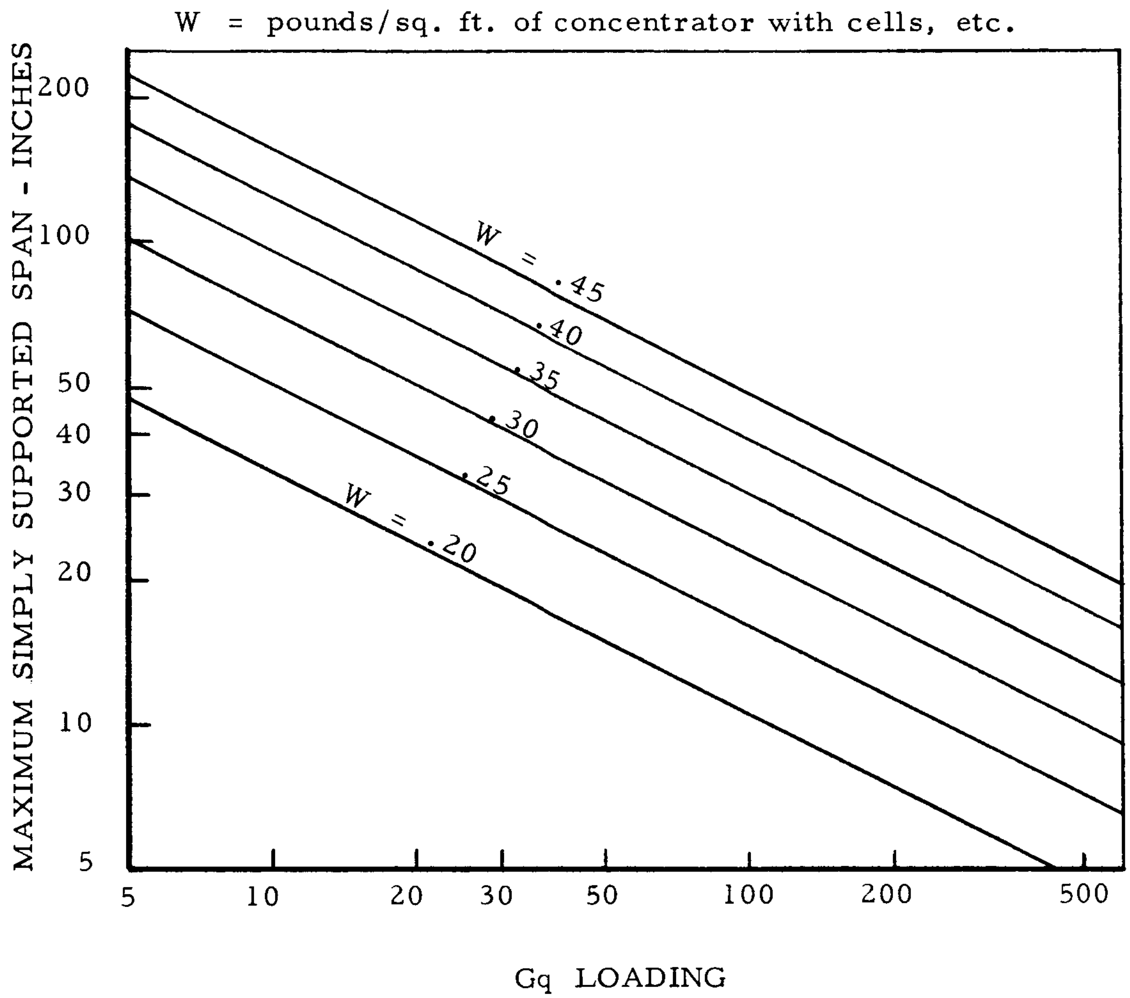
PHOTOVOLTAIC CONCENTRATOR CONFIGURATION
FIGURE 3.3.1-1



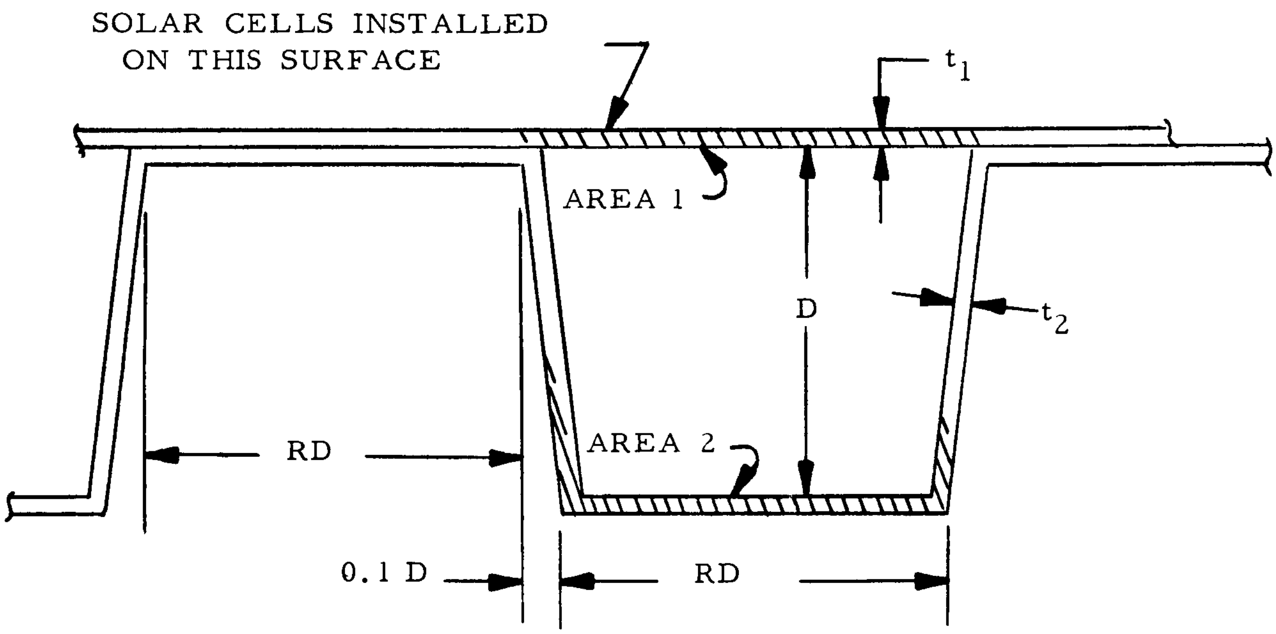
CALCULATED SOLAR CELL TEMPERATURES IN A 6 MIL CONCENTRATOR WITH HIGH EMITTANCE REFLECTIVE SURFACES. CONCENTRATORS ANALYZED ARE ALL UPPER BOUND DESIGNS.

SOLAR CELL TEMPERATURES
FOR CONCENTRATOR PANEL

FIGURE 3.3.1-2



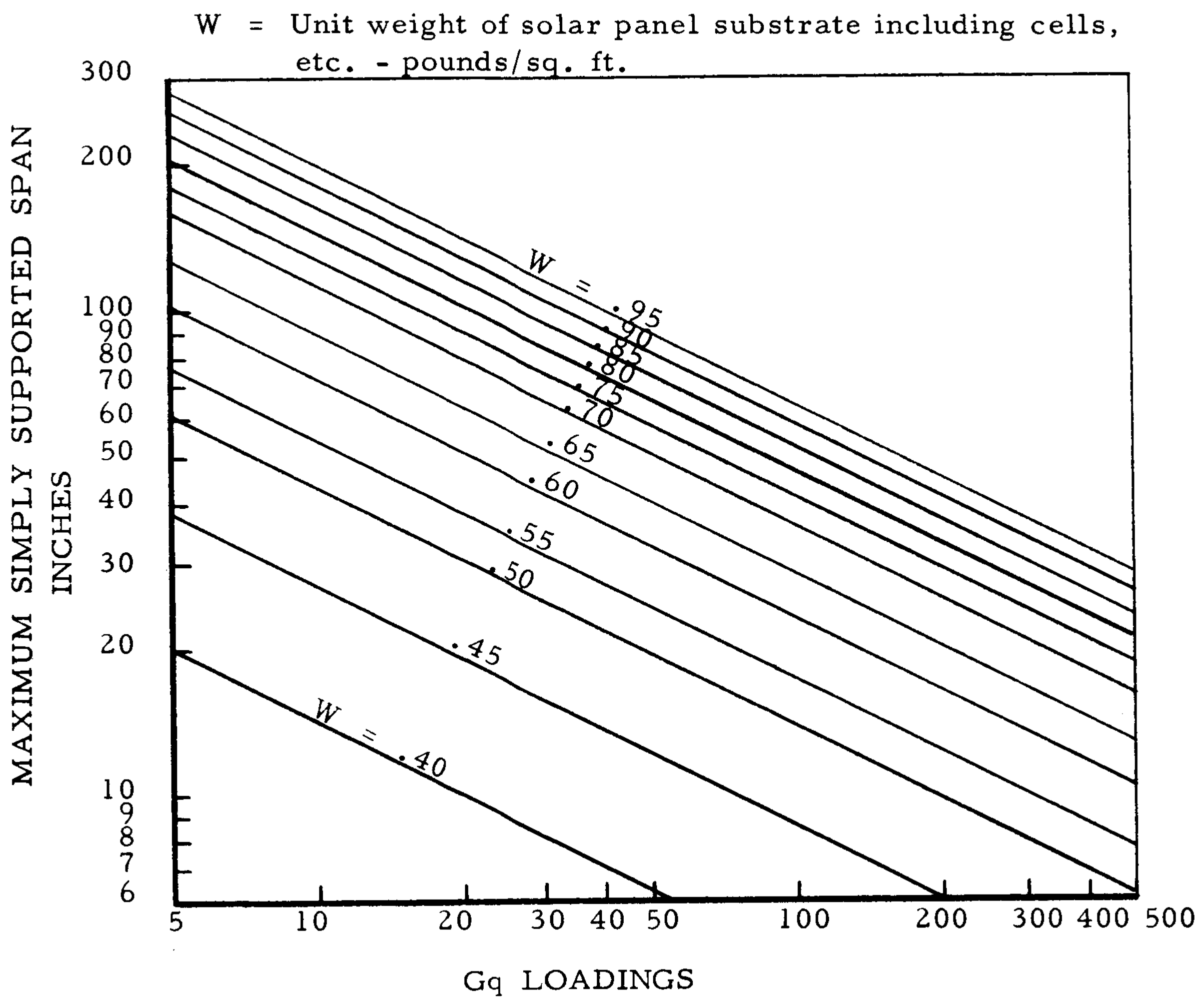
UNIT WEIGHT VS LOADING AND
SPAN UPPER BOUND $\phi = 5^{o}$
CONCENTRATOR LAMINAR X-500
COATING AND $W_c = 0.3$ LBS/SQ. FT.
FIGURE 3.3.1-3



SHEET STRINGER SOLAR PANEL
SUBSTRATE CROSS-SECTION

SHEET-STRINGER PANEL
CONFIGURATION

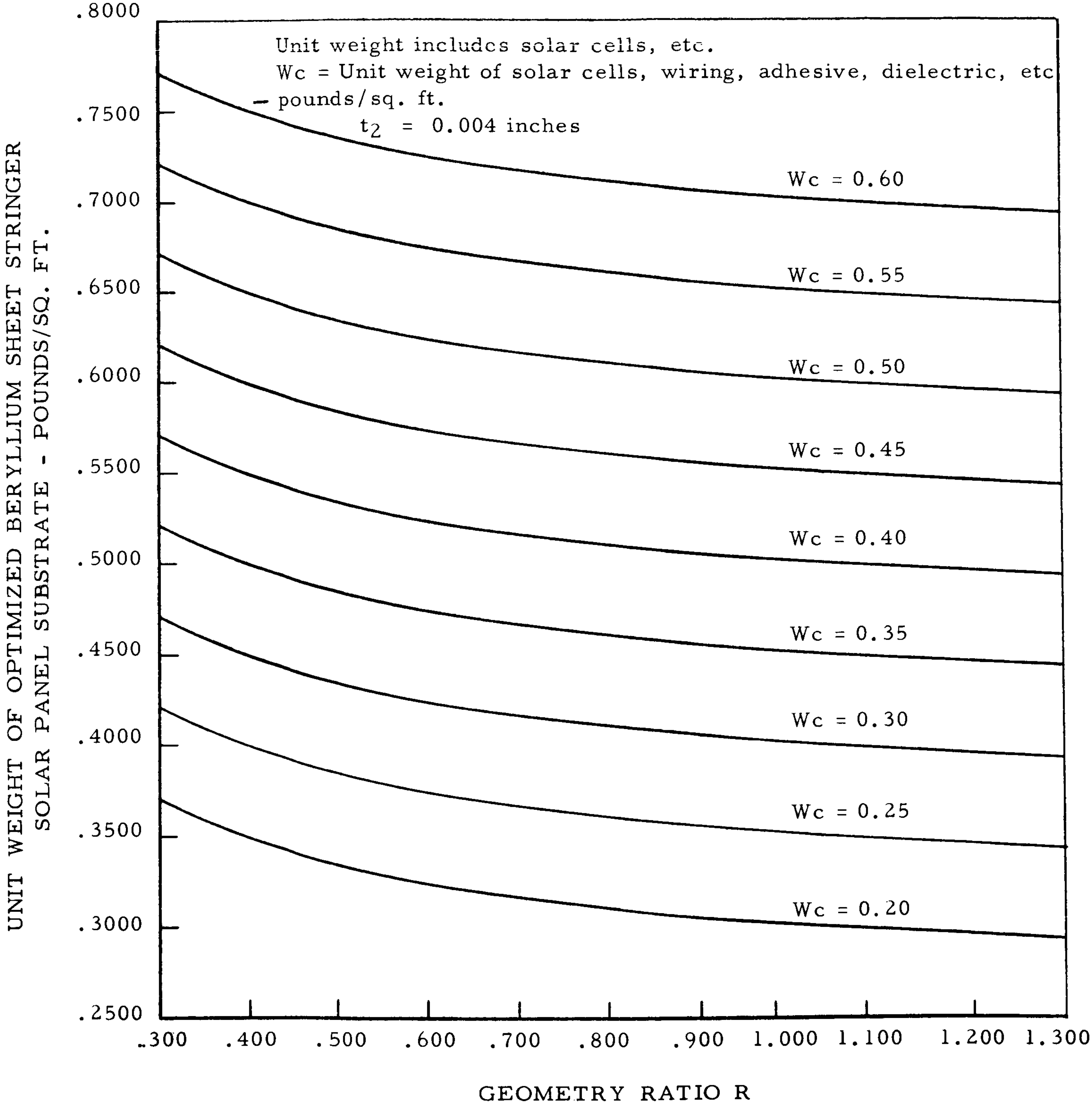
FIGURE 3.3.2-1



Data Suitable for Preliminary Design Purposes Only.

UNIT WEIGHT VS LOADING AND SPAN - ALUMINUM SHEET STRINGER SOLAR PANEL SUBSTRATE
$W_c = 0.3$ POUNDS/FT^2

FIGURE 3.3.2-2



UNIT WEIGHT OF OPTIMUM BERYLLIUM SHEET STRINGER SUBSTRATE - t_2 = 0.004 INCHES

FIGURE 3.3.2-3

3.3.3 Flexible Rollup Arrays

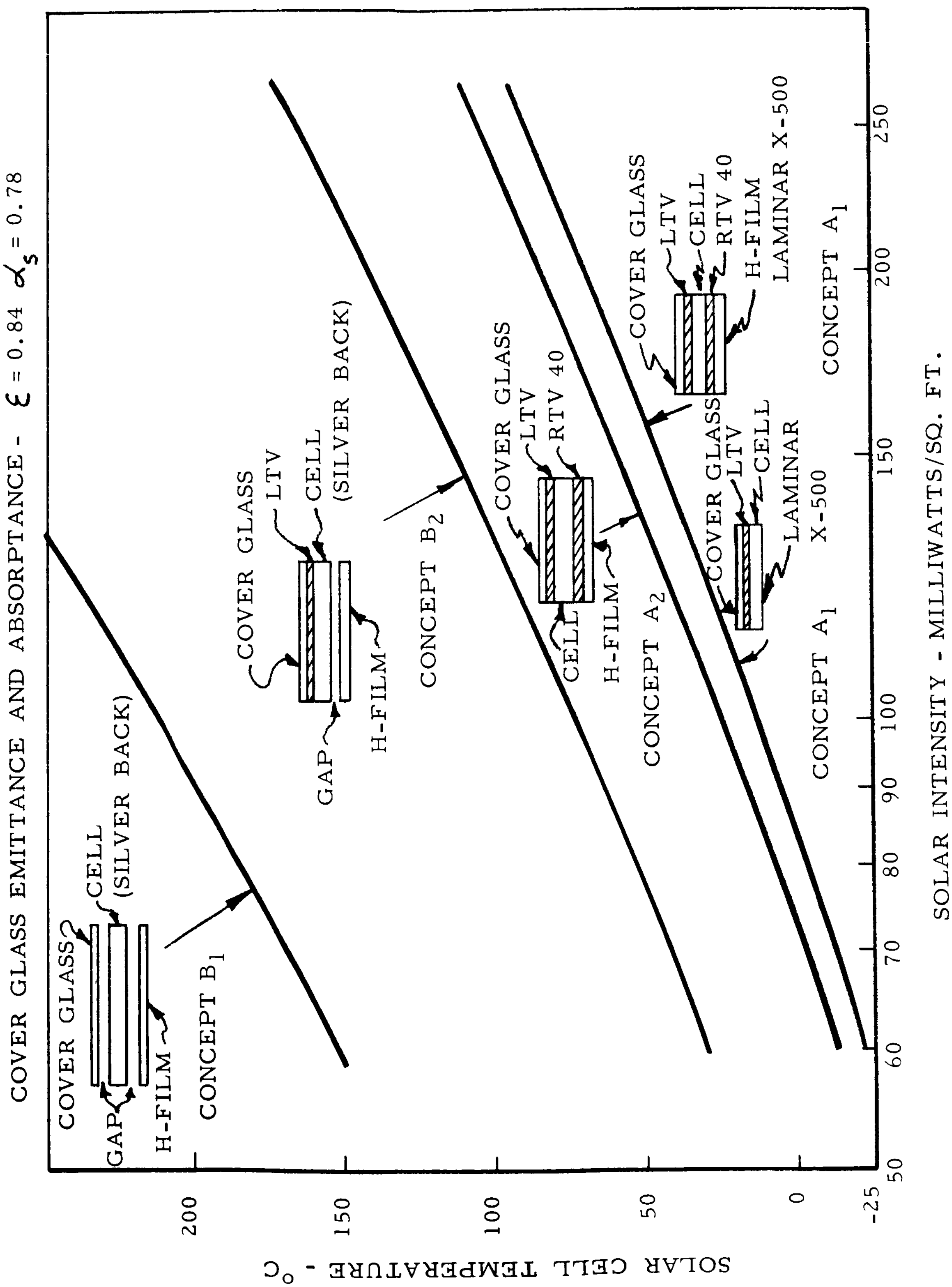
This panel concept appeared to be quite promising from the standpoint of deployed weight. Thermal analysis indicated the concept could deliver 10.5 watts/sq . ft. This concept was not considered to be within the 1965 state-of-the-art definition.

The results of the thermal analysis for both this concept and the semi-rigid concept are shown in Figure 3.3.3-1.

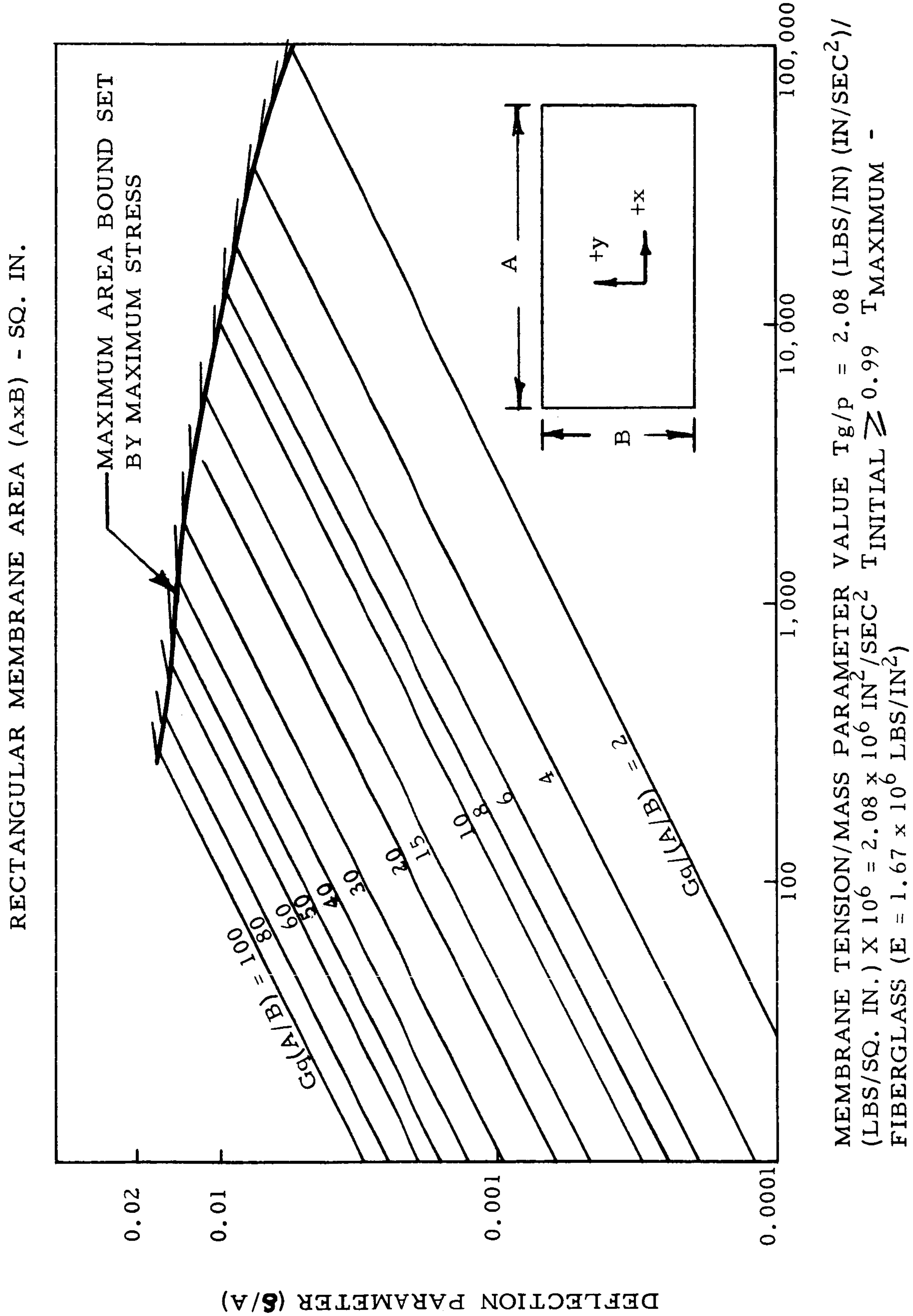
3.3.4 Semi-Rigid Cross Tape Concept

In this concept the solar cells are supported by woven fiberglass tapes in an extremely light weight design. The power performance was indicated to be high and the concept certainly appeared feasible for a 20 watt per pound array. The crucial item in consideration of this crossed tape and semi-rigid stretched membrane is the response of the membrane to vibratory loading. Theoretical analyses of the dynamic characteristics of rectangular membranes is summarized in Figure 3.3.4-1. In this figure, the tension load in the membrane in pounds per inch is the variable T, g = 386 inch/sec./sec., and the weight of the membrane in pounds per square inch is the variable p. The figure summarizes a particular membrane load/mass parameter value whereby the deflection of the membrane can be found from the curves. For a geometry ratio of A/B = 1.25 and Gq = 22 "g", the deflection parameter for a 500 square inch membrane has a value of $\delta/A = 0.006$. Therefore, for A=25 inches, the deflection, $\delta = 0.15$ inches. This deflection is well within the limits set by the frame design. Consequently, this panel concept was selected as a likely candidate for the 20 watt per pound array.

A more complete dynamic analysis of this concept is given in Appendix E of bi-monthly report D2-23942-2.



THERMAL ANALYSIS OF MEMBRANE SOLAR ARRAY CONCEPTS
FIGURE 3.3.3-1



VIBRATING MEMBRANE
ANALYSIS

FIGURE 3.3.4-1

4.0 ARRAY TRADE SUMMARIES

This area includes the configuration of different array concepts for the Saturn IB/Centaur and Atlas/Centaur spacecrafts. Subsystem integration and interface coordination continued throughout this phase of the program. Preliminary analysis of the array power to weight ratio capabilities were conducted and the significant structural and dynamic characteristics of each concept were estimated.

Data collection and evaluation of state-of-the-art components, materials and processes appropriate to the configuration of large solar arrays were conducted. This material was then used to develop several configuration alternatives for cell stack modules, electrical busses, support structure and deployment and jettison mechanisms. These components and systems were synthesized into array configurations which were compared to the requirements to select a near optimum configuration.

Tables 4-1 and 4-2 summarize the results of the weight, specific power, reliability and state-of-the-art trades that have been conducted for several 50 kilowatt folding modular rollup arrays. The analyses were based on the preliminary conceptual layouts discussed previously in this document.

The weights shown for the several panel configurations are based on solar cell stacks utilizing 8 mil N on P, back-connected silicon solar cells and 4 mil microsheet cover glasses. The estimated performances, in watts per pound, were arrived at by dividing the estimated power outputs for each panel type by the predicted weight per square foot of the total array in which the panel is used.

A reliability rating has been assigned to each significant subsystem of the array analyzed. A rating of one (1) indicates a baseline. Each succeeding higher number indicates a prediction of a lower degree of reliability. A lower degree of reliability is predicted when reliability data is not available or when the subsystem exhibits a higher degree of complexity than the baseline.

A state-of-the-art evaluation of the significant array subsystems has also been conducted. The basic guideline for this comparison is a prediction of when the technical know-how for designing and fabricating the components would be available. Availability of the facilities necessary to fabricate and test the large arrays was not included. Those elements rated "A" are considered to be present state-of-the-art. A rating of "B" denotes a prediction that approximately 6 months of development work would be required to bring this item to a state-of-the-art position. A "C" rating denotes a prediction of a 12 to 15 month development period.

Finally, the solar panel area that can be packaged in the available envelope has been developed. Using the results of the power output analyses, a determination of the total kilowatt capability that can be stowed and deployed has been made. The summary indicates that only 45 kilowatts, at 1 Astronomical Unit sun-probe distance, of concentrator panel can be stowed. This is due to the low, 8.95 watts per square foot, power output of this panel configuration. 50 kilowatts of power are achievable using semi-rigid or rollup type panels. The semi-rigid folding modular array was selected for preliminary design. This decision was primarily based on the state-of-the-art comparisons conducted.

FOLDING MODULAR TRADES SUMMARY

ITEM	LBS./ SQ. FT.	RELIABILITY RATING	STATE OF ART RATING	ESTIMATED PERFORMANCE WATTS/POUND	TOTAL KW POSSIBLE ENVELOPE
Concentrator Panel	.263	1	A		45
Semi-Rigid Panel (Cell Stack and Substrate)	.211	2	B		50
Beryllium Support Structure For Concentrator	.108		B		50
Beryllium Support Structure For Semi-Rigid	.208		B		
Motor Drive Deployment	.023	1	A		
Bourdon Tube Deployment	.028	2	B		
Spring and Damper Deployment	.025	1	B		
Miscellaneous	.030				
Busses	.025				
Total Array for Concentrator Concept (Motor Drive)	.449	1	A	19.9	45
Total Array for Semi-Rigid Concept (Motor Drive)	.497	2	B	21.9	50

TABLE 4-1

ROLLUP CONFIGURATION TRADES STUDY

ITEM	LBS./ SQ. FT.	RELIABILITY RATING	STATE OF ART RATING	ESTIMATED PERFORMANCE WATTS/POUND	TOTAL KW POSSIBLE ENVELOPE
Stiffened Panel	.249	1	C		50
Unstiffened - Foam Pad and Storage Drums	.245	2	C		50
As Structure					
STEM Boom	.059	1	A		
Closed Tube Boom	.059	1	C		
Telescoping Boom	.035	2	A		
As Deployment System					
STEM Boom	.079	1	A		
Closed Tube Boom	.079	3	D		
Telescoping Boom	.083	2	A		
Busses	.025				
Total Array using STEM Boom	.412	1	C	25	50
Total Array using Closed Tube Boom	.412	3	C	25	50
Total Array using Telescoping Boom	.392	3	C	26.2	50

TABLE 4-2

4.1 SATURN IB/CENTAUR CONFIGURATION TRADE STUDIES

4.1.1 Folding Modular - Semi-Rigid Array

This configuration consists of four rectangular panel assemblies located symmetrically about the centerline of the spacecraft. Each panel assembly consists of 6 sub-panel units (numbers 1 through 6, Figure 4.1-1). Each sub-panel unit, except #1, has two auxiliary panels. The deployed length of each panel assembly is 68 feet. The sub-panel is composed of a rigid metallic frame and a pretensioned fiberglass tape substrate.

There are three spars in the main panels only. The auxiliary panel when in the folded position transmit their dynamic mass loading through hard point supports to the main spars. Section A-A of Figure 4.1-1 shows a typical cross section. An alternate configuration provides three spars in each sub-panel (main and auxiliary) and is shown in Section A-A, Figure 4.1-2. In this concept the dynamic mass loading of each sub-panel is supported by its own spars. The required weight of substrate structure to arrive at a 20 watt per pound solar array was re-distributed throughout the substrate support structure. This resulted in support structure that was capable of supporting a substrate stress level of approximately 80,000 psi. Using this load and a rectangular panel configuration developed during this iteration, a typical panel (#2 of Figure 4.1-1) was analyzed. An average weight of .4988 pounds/sq. ft. was arrived at for this typical panel. Using this value for the total array resulted in a 20 watt per pound solar array.

4.1.2 Concentrator Solar Array Configuration

The concentrator folding modular solar array consists of four panel assemblies which are folded and stowed within the spacecraft envelope as shown in Figure 4.1-2. Each panel assembly consists of 7 sub-panel units (numbers 1 through 7). Each sub-panel unit has two auxiliaries except #1. Panel #1 does not have auxiliary panels so that spacecraft subsystems such as antennas and sensors may be deployed without interfering with the solar array.

Each panel, main and auxiliary, has three spars. The spars are sized by the mass loading of the cells and substrate during the launch vibration condition with the panels stowed. After the array is deployed and just prior to Mars orbit injection all panels except the #1 main panels (4) are jettisoned. These provide sufficient area to generate 1.1 KW of power during Mars orbit.

The concentrator substrate is nested between the spars to reduce the stowed panel envelope (see Section A-A, Figure 4.1-2). The concentrator substrate has an unsupported span of 50 inches and the ends are fixed to the spars with bonded clips.

The gross area of this configuration is 5517 square feet which produces 45.6 KW and weighs .456 pounds/square foot.

4.1.3 Mechanism Trade Studies

In this trade study phase, two principal systems were considered for primary deploy ent of folding modular arrays in addition to the motor-pulley-cable system selected for the preliminary design. Each competitor was developed in sufficient detail to provide the basis for weight and reliability trades, and state-of-the-art comparisons. The motor-pulley-cable system is described in greater detail in Section 6.1.2 of this report.

4.1.3.1 Torsion Spring Deployment

As shown in Figure 4.1-3, a torsion spring design was established with springs located at each hinge pin. The springs are sized to provide a 14 minute sequence for full deployment.

In the boost configuration, springs are under load at all times. Upon receipt of the deployment signal, ordnance pin pullers release the panel stack and the torsion springs rotate the panels about their hinge lines. A system of panel releases integrated with the joint latches sequence each quarter array. Although the addition of a synchronous deployment rate control would probably reduce spacecraft perturbations, it would complicate the deployment sequence.

4.1.3.2 Bourdon Tube Deployment

A bourdon tube assembly is shown in Figure 4.1-4. Each tube assembly has nine helical turns. There is one tube used per hinge joint. Sequential deployment is controlled by metering the pressurizing gas to the outboard sub-panel assemblies through successively smaller fixed orifices.

The system is made up of a pressure vessel, redundant release valves, filters, and suitable orifices to supply nitrogen gas at 1015 psia. Either 20 or 24 helically formed bourdon tubes are used depending upon the number of solar subpanels required. Each bourdon tube serves as the actuator and damper for each hinge point. Latches are provided at each hinge. Side panel deployment will be accomplished by flat or torsion springs.

Upon earth command, the ordnance pin pullers release the boost tie down structure, and the ordnance valves on each end of the gas supply vessel are operated. Pressurized nitrogen gas is fed in series to each bourdon tube of each panel assembly through metering orifices. As each panel reaches its deployed position, each joint is latched in sequence and the succeeding panel is released for deployment.

4.1.3.3 Motor-Cable System

The system shown in Figure 4.1-5 consists of an electric motor coupled to a cable drum through a gear reduction. The principle of operation follows a standard aircraft control approach. A cable is used to rotate a quadrant to change linear motion to hinge torque. This concept was chosen for both Saturn IB/Centaur and Atlas/Centaur deployment.

The electric motor-harmonic drive unit is similar to that already flown on the Pegasus Meteoroid Detection Satellite and the combination with the pulleys and cables results in a simple system with fewer parts than the spring damper system. Although the bourdon tube concept appears attractive and basically sound, it was not considered to be present state-of-the-art. A full description of the motor-pulley-cable system is given in Section 6 of this report.

4.1.3.4 Auxiliary Panel Deployment

Three basic spring systems for side panel depliyment were considered during the trade studies. These included the carpenter's rule type, a collapsible tube and the torsion spring-damper which was finally selected. The torsion spring design is discussed in the preliminary design Section 6 of this report. Selection of the torsion spring was based on reliability and state-of-the-art considerations.

4.1.3.4.1 Spring Leaf With Arc Section (Carpenter's Rule)

The side auxiliary panels, shown in Figure 4.1-6, once released, will deploy from the energy stored in the spring hinge/damper/lock. From the stowed to the deployed position, the side panel will accelerate at a decreasing rate to the deployed position whereupon the spring will deflect in the opposite direction absorbing the energy of the moving panel. This oscillating motion continues until the initial spring energy of the hinge/damper/lock is absorbed by internal friction and dissipated as heat. Any subsequent changes in spacecraft attitude can cause the side panels to oscillate momentarily, but the panels will always come to rest when there is zero stress in the hinge/damper/lock. Analysis of a flat spring system is shown in D2-23942-2, Appendix G.

4.1.3.4.2 Collapsible Spring Tube

The side auxiliary panels shown in Figure 4.1-6, once released, will deploy from the energy stored in the collapsible spring tube. Deployment action and reaction to spacecraft attitude change will be the same as for the spring leaf system.

4.1.3.5 Hinge Joint Actuating Quadrants

Use of the motor-pulley-cable deployment system requires the use of cable quadrants to supply the operating torque for individual joints. The action of a cable about a sheave as the hinge of a joint is opened requires that the sheave always provide a moment arm about the hinge pin. A means must be provided to keep the cable in the sheave track without interference to the cable as it moves through the joint. The method of Figure 4.1-7 features a sun and planetary gear set for each side of the sheave. Ratios are selected to assure all three guards are nexted away from the cable as deployment continues.

The use of cable guard shear pins was the selected method for the preliminary design and is discussed in Section 6.

4.1.3.6 Mars Panel Jettison

Array separation as the spacecraft is prepared for retro into the Mars orbit is accomplished by a single ordnance actuator. The deployment cable and control and monitor wires are cut and the structure separated at prepared joints. Figure 4.1-8 shows one method which was studied. Upon receipt of the jettison signal, dual ordnance squibs are fired by dual bridge wires from redundant wiring circuits. Expanding gas drives two cable cutters against an anvil, cutting the deployment cable and the signal and monitor wires. As the gas continues to expand, pistons apply a force to retract carefully fitted structural plugs by actuating a rocker arm which is connected to a tension cable of fiberglass.

The use of a mechanical structural clamp was also studied. This was the selected concept and is described in Section 6.1.2 of this report.

4.1.4 Three Wing Folding Modular Solar Array (Figure 4.1-9)

A three-wing folding modular solar array was investigated to determine what deployed area could be achieved with rectangular and trapezoidal shaped panels. The object was to ascertain what the trades were for supplying improved view angles for spacecraft subsystems.

Using a typical panel thickness of 2.50 inches, the gross stowable area for the rectangular panels is 4113 square feet and for the trapezoidal panels, it is 4323 square feet. Since the area achieved is not sufficient to produce 50 KW, an extension of the Saturn shroud of 42.00 inches is required. This would result in a deployed panel length of 81.5 feet from the spacecraft centerline. The trapezoidal panel array investigated resulted in a decrease in the available area for the ion engine view angle.

4.1.5 Saturn/Centaur Modular Array Configuration Weights

Three modular configurations were weight analyzed. These are the trapezoidal semi-rigid array using a tape substrate, the rectangular semi-rigid array with tape substrate, and the rectangular concentrator array. Weights for secondary cabling and mechanisms were based on constant factors. The weights of the rectangular panels are based on the weight of the second most inboard panel. Although the weight of the inboard panel in pounds/square foot is greater than this panel, the other panels more than compensate for this. The weights in pounds per square foot of these panels are shown below. The cell stack weights are based on the use of 8 mil silicon solar cells and 6 mil microsheet cover glasses.

SEMI-RIGID	TRAPEZOIDAL	RECTANGULAR
Cell Stack plus Substrate	.2123	.2123
Secondary Cabling	.0250	.0250
Panel Frame and Spars	.2796	.2080
Mechanisms (12%)	.0620	.0535
	.5789	.4988

CONCENTRATOR	RECTANGULAR
Cell Stack	.0905
Substrate	.1500
Stiffeners	.0316
Secondary Cabling	.0250
Spars	.0780
Miscellaneous	.0300
Mechanisms (12%)	.0460
	.4511

4.1.6 Solar Array Rollout Extendible Boom

The concept shown in Figures 4.1-10 and 4.1-11 provides a solar array system that makes use of two types of solar array panel deployment systems, namely, fixed or rigid and rollout.

Basically there are three panel assemblies spaced equally within the envelope provided.

The fixed or rigid sub-panel is rotated 90° during deployment and is capable of withstanding the inertia forces generated during insertion into Mars orbit. The rollout portion of the array panels are jettisoned prior to Mars orbit.

The fixed array structure provides for the support of the rigid sub-panels on a truss member. This truss is supported directly off both ends of the flexible array containers. Two nylon cables crisscross the truss to rigidize the frame. Two additional nylon cables span between the truss and the center envelope to fix the array truss for aft loads when the array is in the deployed position. The array container has the fixed truss secured on its end and is rotated 90° during deployment by a torsion spring system. The container preloads and locks the truss in the down position. This preload condition resists all forward loads directed on the truss.

The extendible boom provides for the deployment and support of the flexible array. The system secured to each container and truss frame is a seven section gas-operated telescoping boom. The first or base section of the boom will contain a pressure bottle that will supply the gas required for erecting the boom.

Each section of the goom will employ teflon seals that will provide the sliding surfaces during boom deployment as well as be the seals for retaining the gas system.

The flexible panel consists of silicon solar cells bonded to a film member and secured to a multiple buildup of various materials. This curtain of material is then rolled up on an eight-inch mandrel inside a thirty-inch fiberglass array container.

4.1.7 Saturn/Centaur Extendible Boom Weight Analysis

A weight analysis was made for the extendible boom configuration of Figures 4.1-10 and 4.1-11. The results shown here may be compared with weight of the two alternate rollup designs in paragraph 4.1.11.

Cell Stack + Substrate	.2332
Secondary Cabling	.0250
Cushion	.0408
Crossbeams	.0098
Guides	.0020
Cable and Attachments	.0010
Fixed Array Frame	.0006
Mandrels	.0217
Drums	.0417
U-Joints, Drive, Attachment, etc.	.0136
Gas, Bottles, and Mounts	.0055
Miscellaneous	.0100
Unit Weight	.4163

4.1.8 Flexible Rollup Solar Array

The concept that is shown in Figure 4.1-12 provides alternate techniques for deployment of the flexible rollup solar array. This concept also provides a rigid portion of fixed array capable of withstanding the inertia forces generated during insertion into Mars orbit. The flexible rolled-out portion of the array is jettisoned prior to Mars orbit insertion.

Figure 4.1-12 shows a 3-arm array and a 6-arm array. The 3-arm array requires deployment of the flexible rolled-out portion to a length of 161 feet. The 6-arm array employs a second tier of 3 drums and fixed array trusses forward and centered between the aft tier drums. Because of reduced storage space, the forward arms are narrower than the aft arms. The 6-arm array requires deployment of the flexible rolled-out portion to a length of 110 feet.

The fixed array is hinged for compact stowage and is mounted to structure at its base using a motor-driven cylindrical shaft. A cable and pulley arrangement is used to rotate the outboard section about the hinge while the inboard section rotates about the shaft centerline. Latches are provided so that, when fully deployed, the fixed array locks in the extended position. A storage drum for the flexible rollup solar array is mounted to the outboard end of each section of rigid array.

The flexible solar array consists of silicon solar cells bonded to a 3 mil H-film membrane supported longitudinally along each side by a collapsible deployment boom, and laterally by fiberglass crossbeams. In the stowed condition, each section of flexible array is rolled up on an 8-inch diameter mandrel in a storage drum mounted outboard of the corresponding section of fixed array. The end of each deployment boom is permanently attached to and supported by the rigid array. The storage drum is attached to the rigid array through a pyrotechnic device. After the rigid array is fully deployed, a squib is ignited to release the storage drum from the section of rigid array. The flexible array unwinds from the mandrel with energy for mandrel rotation derived from that stored in the collapsed deployment booms. As the array unwinds the storage drum moves outward. When the flexible array completely unwinds from the mandrel, the storage drum is jettisoned by simply allowing it to continue its travel outward away from the spacecraft. Two configurations of deployment boom are shown in Figure 4.1-12, the STEM (Storable Tubular Extendible Member), and the Collapsible Tube.

4.1.9 STEM Configuration

The STEM booms are formed out of beryllium copper sheet spring material heat treated into a circular section such that the edges of the material overlap by approximately 180°. The boom elements are stored in a strained, flattened condition when wound on the mandrel. Bending strength and buckling resistance of the STEM boom may be increased for a given formed diameter by nesting several elements together. This design is shown in Section D-D.

4.1.10 Collapsible Tube Configuration

The collapsible tubular boom shown in Section A-A provides an alternate to the STEM boom as a means for deploying the flexible rollup solar array. Instead of an open tube which rolls up on the mandrel as a single sheet, this device is a closed tube which rolls up on the mandrel as a double sheet. The tube would be fabricated in mating halves of thin gage beryllium copper. The tube halves are then joined by seam welding along the mating flanges followed by heat treating to a temper suitable for sheet spring material. The tube halves are formed on a tapered forming die to produce a tapering boom for increasing strength to correspond with increasing bending moment toward the inboard end of the boom. Since the collapsible tubular booms rolls up as a double sheet, it occupies less width on the mandrel than a STEM boom of the same tube diameter.

4.1.11 Cushion for Flexible Array

In order to protect the silicon solar cells and cover glasses from damage when stowed, a cushion is rolled up on the mandrel so as to separate adjacent layers of the flexible solar array. Two configurations for such a cushion are shown in Figure 4.1-12. One configuration shown in Section C-C utilizes a separate flexible polymer foam cushion, and includes a separate spring-wound auxiliary mandrel. The function of the auxiliary mandrel is to wind up the cushion as the flexible array unwinds from the main mandrel. The cushion is then jettisoned along with the storage drum.

In the other configuration shown in Section G-G, the cushion is integrated with, and will remain attached to, the flexible array. This cushion will consist of a corrugated layer of H-film bonded between corrugations to the back of the flexible array. When rolled up on the mandrel, the corrugations will depress and provide the required cushioning effect against the solar cells. In the deployed condition, the corrugated H-film layer provides structural support for the flexible array. Corrugations might be arranged in a chevron pattern to supplement both the longitudinal stiffness provided by the deployment booms and the lateral stiffness provided by the crossbeams. The degree of panel stiffness afforded by the corrugated H-film layer will also facilitate handling of the array during fabrication and assembly prior to attachment of deployment booms and crossbeams.

4.1.12 Saturn/Centaur Rollup Array Configurations

A weight analysis was made for the collapsible tube and the STEM boom rollup arrays. The closed boom and STEM boom arrays were analyzed for configurations using six 110 foot arms or three 161 foot arms. Although the rigid sections of these arrays are heavier than the rollup sections, the area of the rigid section is a small enough percentage of the total area that for preliminary analysis the effect is nominal. Therefore, analysis has been based primarily on the rollup sections. The weights of the collapsible tube and STEM boom are shown with both a separate polymer cushion and an attached H-film cushion substrate.

	110 Ft. Arm		161 Ft. Arm	
Collapsible Tube	Separate Cushion	H-Film Cushion	Separate Cushion	H-Film Cushion
Cell Stack + Substrate	.2332	.2332	.2332	.2332
Secondary Cabling	.0250	.0250	.0250	.0250
Cushion	.0281	.0408	.0281	.0408
Boom	.0484	.0484	.0662	.0662
Crossbeams	.0098	.0098	.0098	.0098
Storage Drums	.0555	.0555	.0507	.0507
Array Mandrel	.0322	.0322	.0208	.0208
Cushion Mandrel	.0096	--	.0063	--
Rollers, Drive, Etc.	.0200	.0150	.0100	.0075
	.4618	.4599	.4501	.4540
STEM Boom				
Same as above except:				
Subtract Closed Boom	-.0484	-.0484	-.0662	-.0662
Add STEM Boom	.0448	.0448	.0877	.0877
	.4582	.4563	.4716	.4755

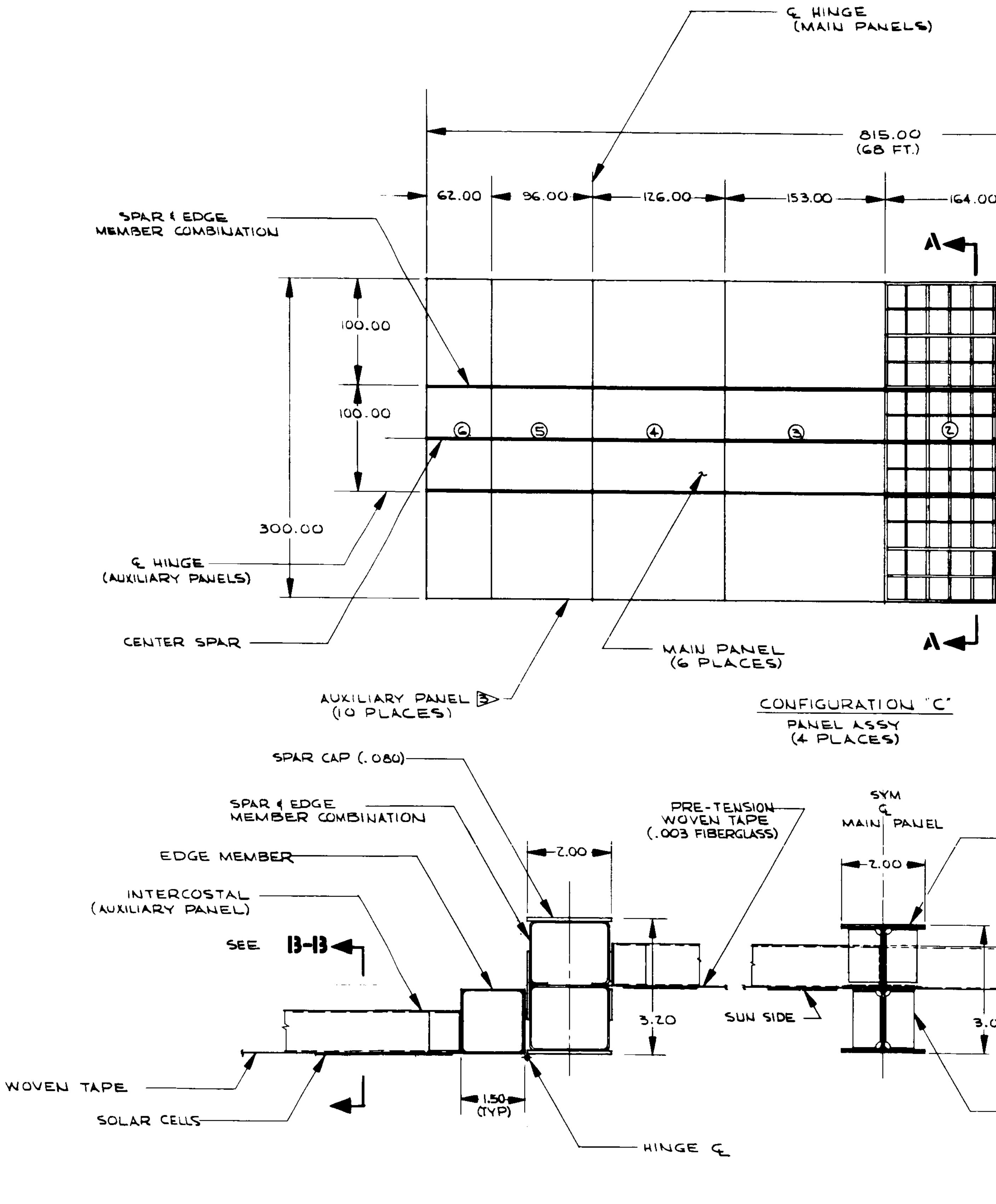
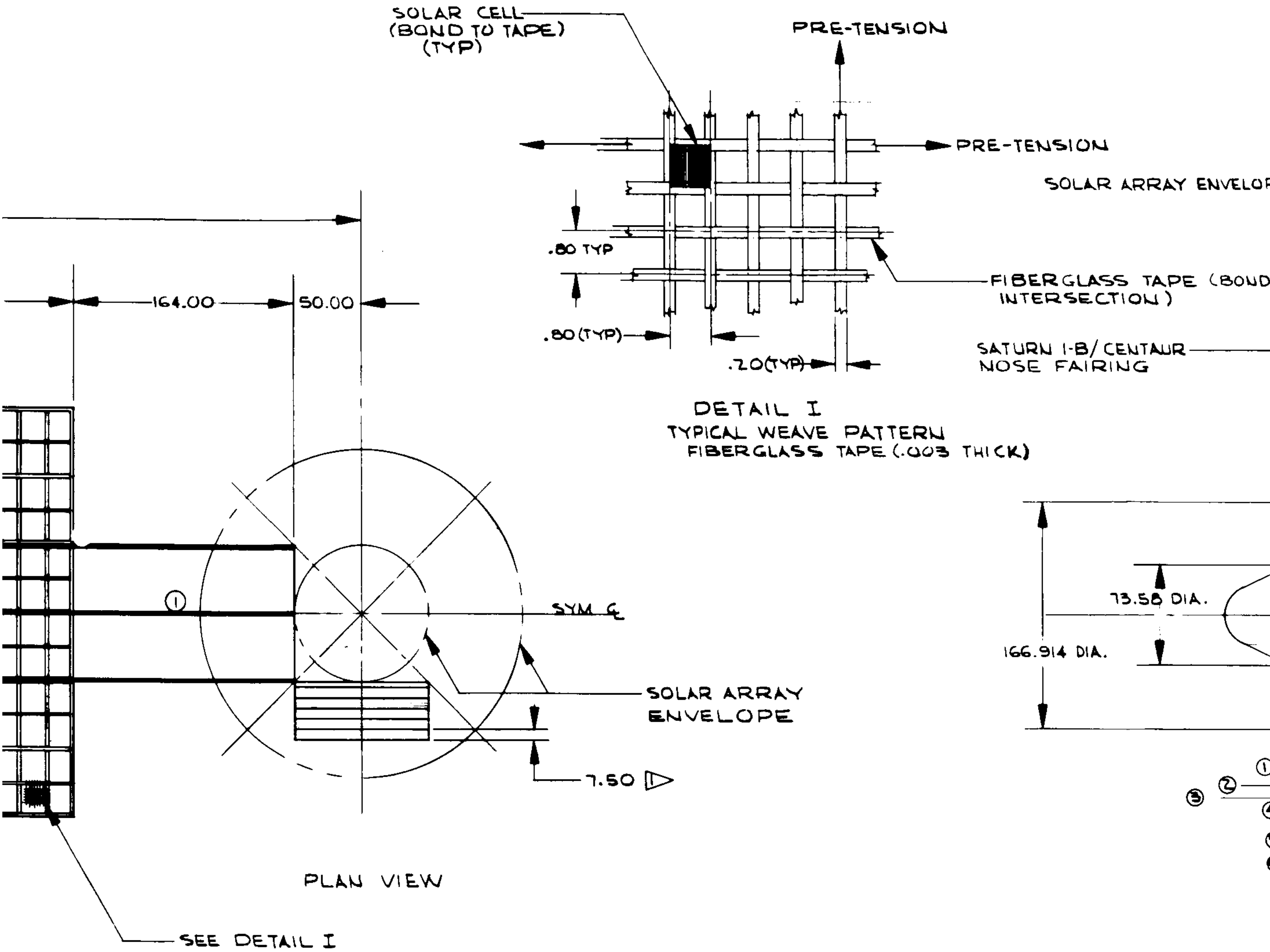
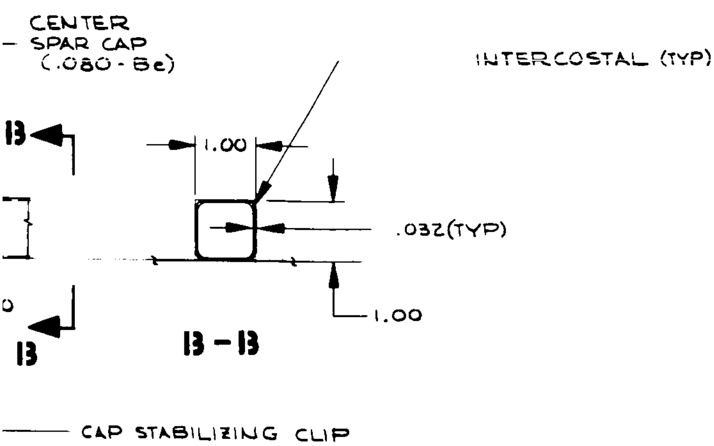

℄ HINGE (MAIN PANELS)
815.00 (68 FT.)
62.00
96.00
126.00
153.00
164.00
SPAR & EDGE MEMBER COMBINATION
A
100.00
100.00
300.00
℄ HINGE (AUXILIARY PANELS)
CENTER SPAR
AUXILIARY PANEL (10 PLACES)
MAIN PANEL (6 PLACES)
CONFIGURATION "C"
PANEL ASSY (4 PLACES)
SPAR CAP (.080)
SPAR & EDGE MEMBER COMBINATION
EDGE MEMBER
INTERCOSTAL (AUXILIARY PANEL)
SEE B-B
2.00
PRE-TENSION WOVEN TAPE (.003 FIBERGLASS)
SYM ℄ MAIN PANEL
3.20
SUN SIDE
WOVEN TAPE
SOLAR CELLS
1.50 (TYP)
HINGE ℄
A-A

Fig 4.1-1

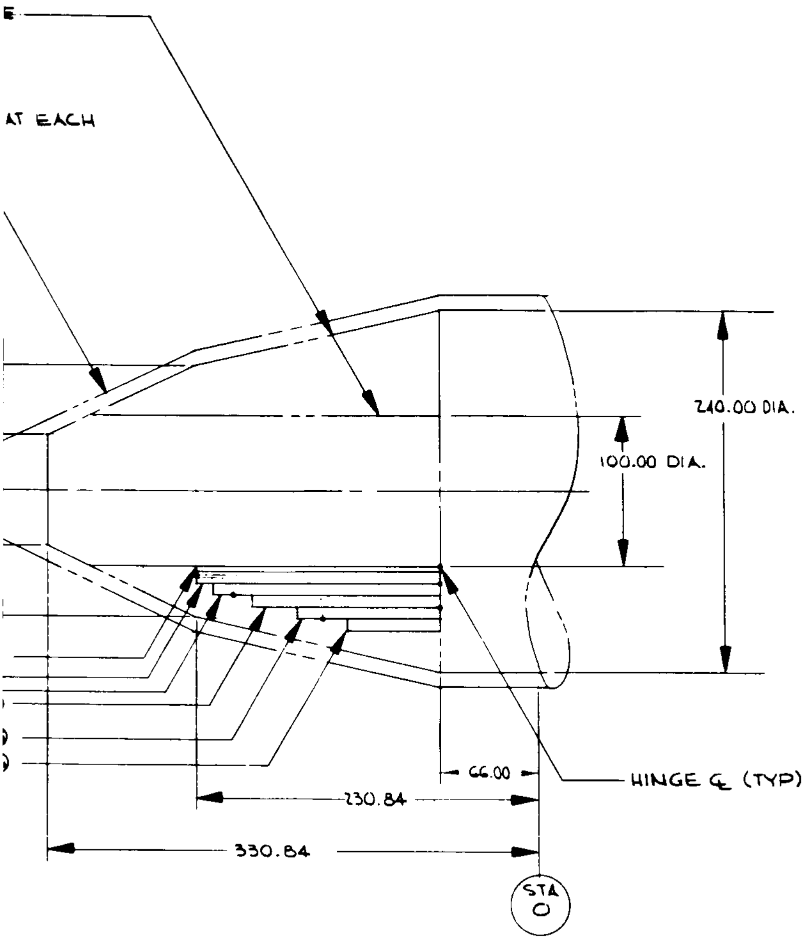
①





	PANEL ▷	CONFIG 'C' AREA SQ. FT.	WT LBS.
	①	114	
	②	342	
	③	319	
	④	262	
	⑤	200	
	⑥	129	
1 PANEL ASSY		1366	
1 SOLAR ARRAY		5464	

Fig 4.1-1

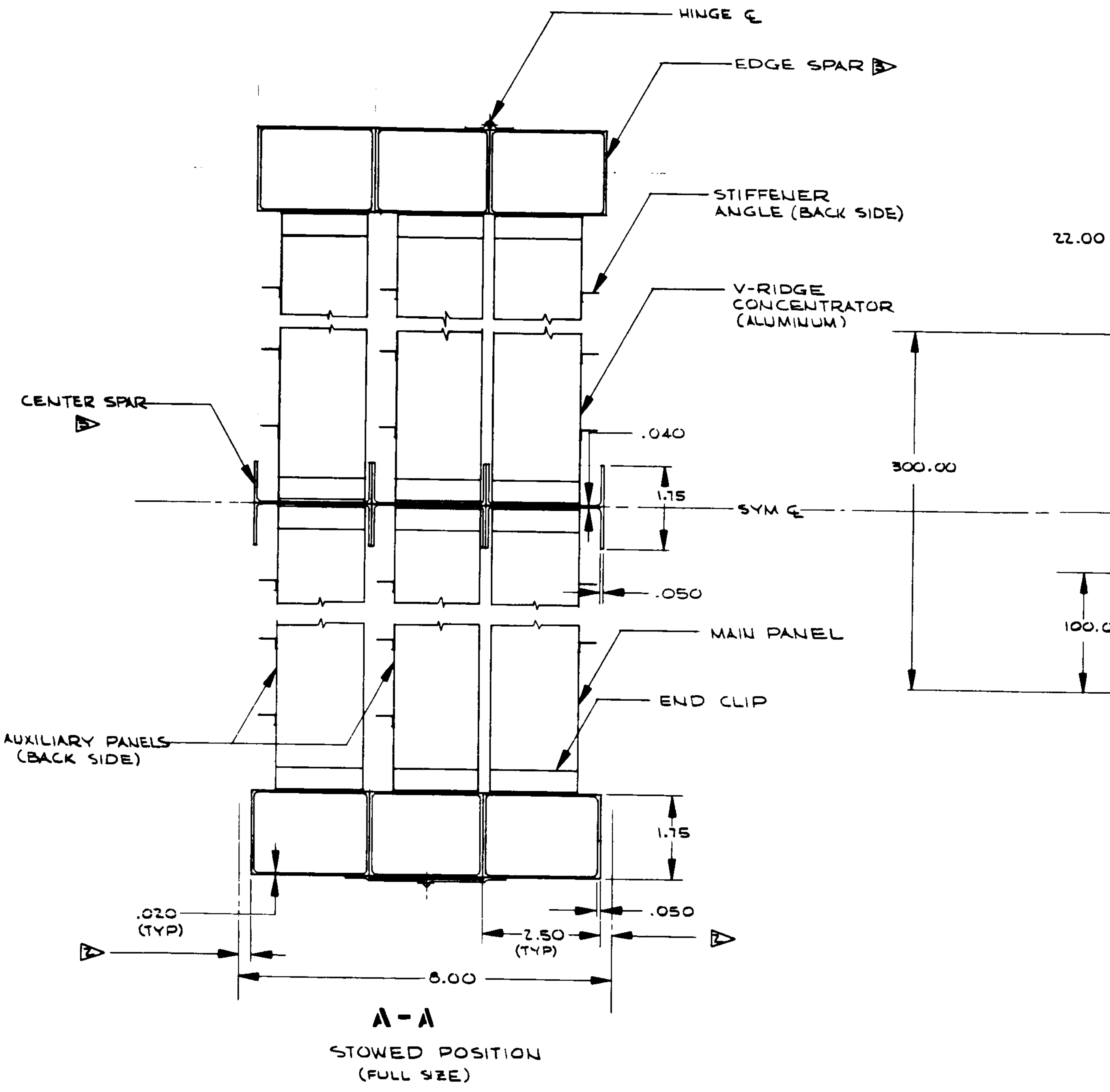


3 THE AUXILIARY PANELS IN THE STOWED POSITION LOAD THE MAIN PANEL SPARS THROUGH THE USE OF VISCOUS DAMPERS.

1 CONSISTS OF 1 MAIN PANEL & 2 AUXILIARY PANELS. (EXCEPT FOR 1.)

FIGURE 4.1-1 MODULAR SOLAR ARRAY
SEMI-RIGID
SATURN IB/CENTAUR

3



HINGE ℄
EDGE SPAR
STIFFENER ANGLE (BACK SIDE)
22.00
V-RIDGE CONCENTRATOR (ALUMINUM)
CENTER SPAR
.040
300.00
1.75
SYM ℄
.050
100.0
MAIN PANEL
END CLIP
AUXILIARY PANELS (BACK SIDE)
1.75
.020 (TYP)
.050
2.50 (TYP)
8.00
A-A
STOWED POSITION (FULL SIZE)

Fig 4.1-2

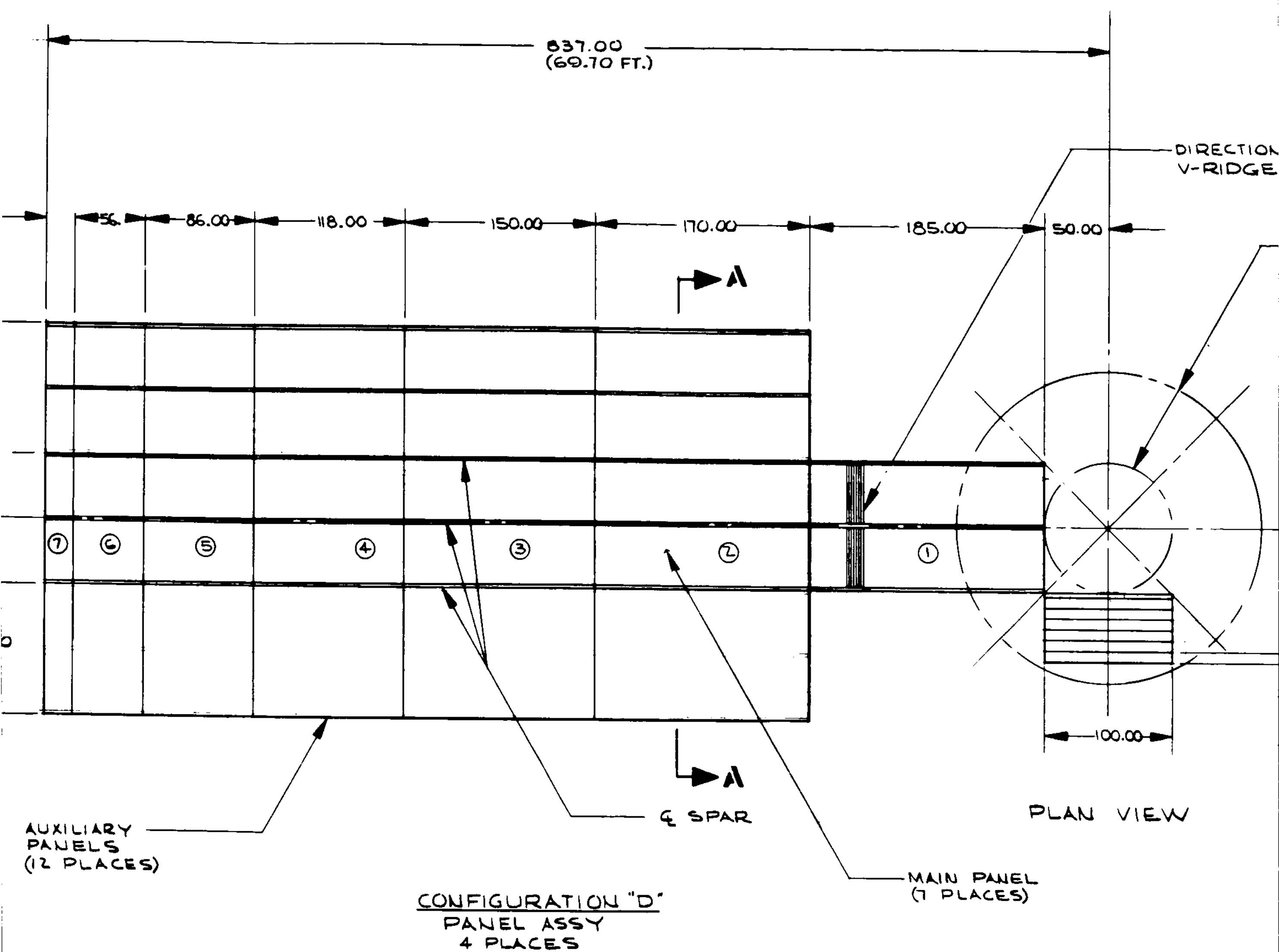
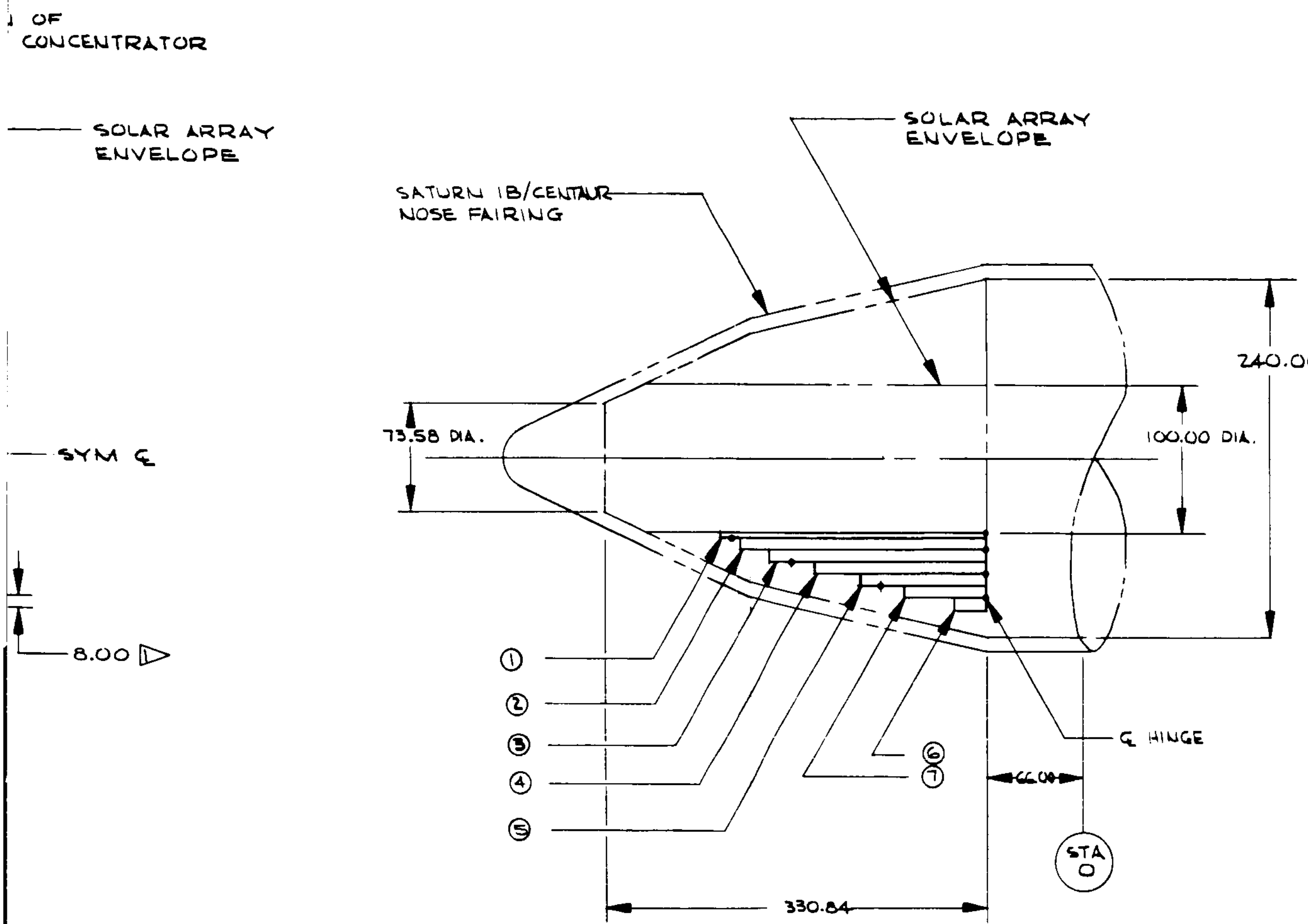

837.00
(69.70 FT.)
DIRECTION
V-RIDGE
56.
86.00
118.00
150.00
170.00
185.00
50.00
A
A
7
6
5
4
3
2
1
100.00
AUXILIARY
PANELS
(12 PLACES)
℄ SPAR
MAIN PANEL
(7 PLACES)
PLAN VIEW
CONFIGURATION "D"
PANEL ASSY
4 PLACES
1 PA
1 S

Fig 4.1-2



PANEL	CONFIG. 'D'	
1	AREA SQ. FT.	WT. LBS.
1	128	
2	354	
3	312	
4	246	
5	177	
6	117	
7	45	
NEL ASSY	1379	
LAR ARRAY	5516	

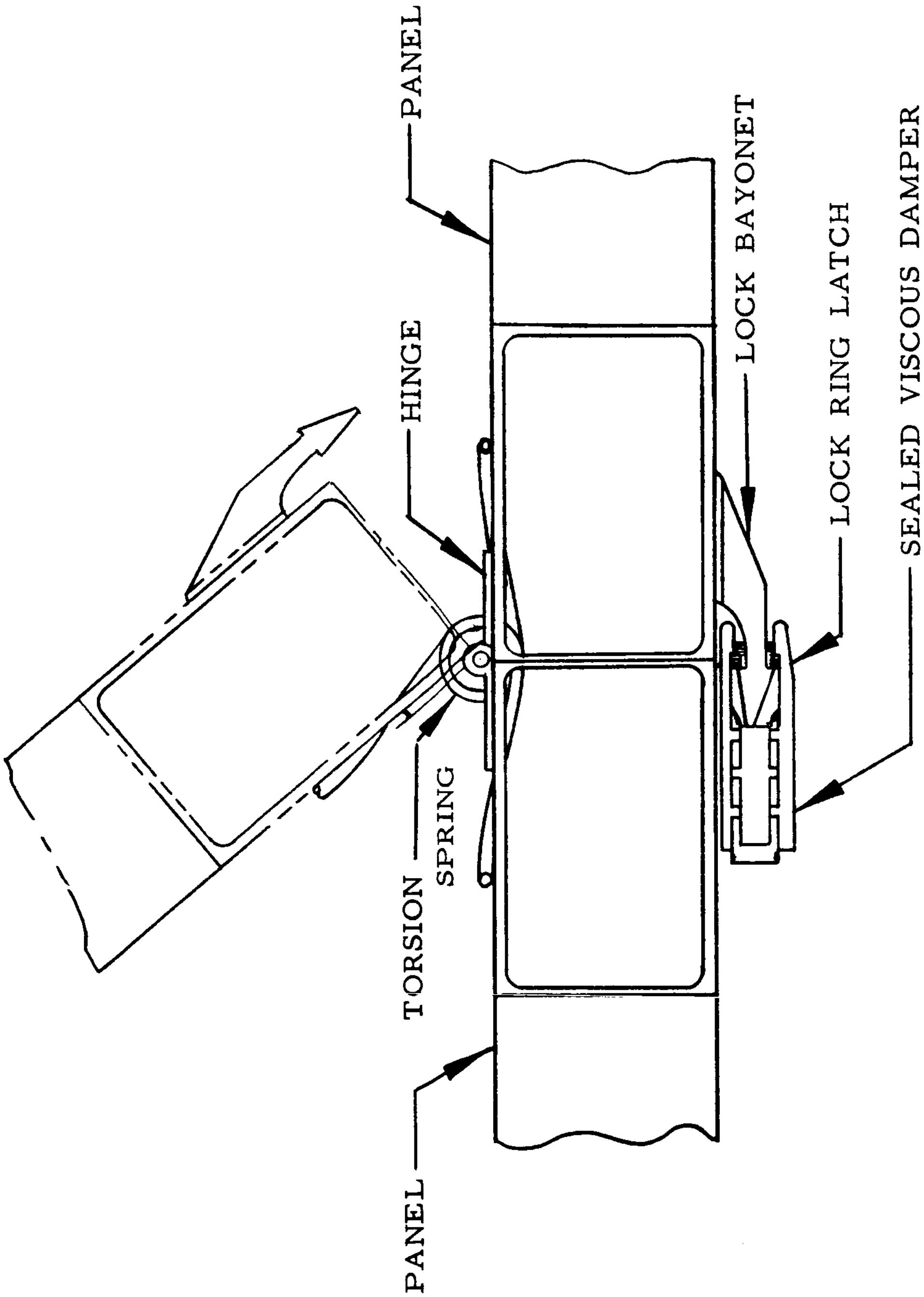
3 Beryllium Spars to be Built up Sections by Bonding, Brazing or Riveting.

2 .250 Allowable Space to Provide Clearance Between Spars as Required.

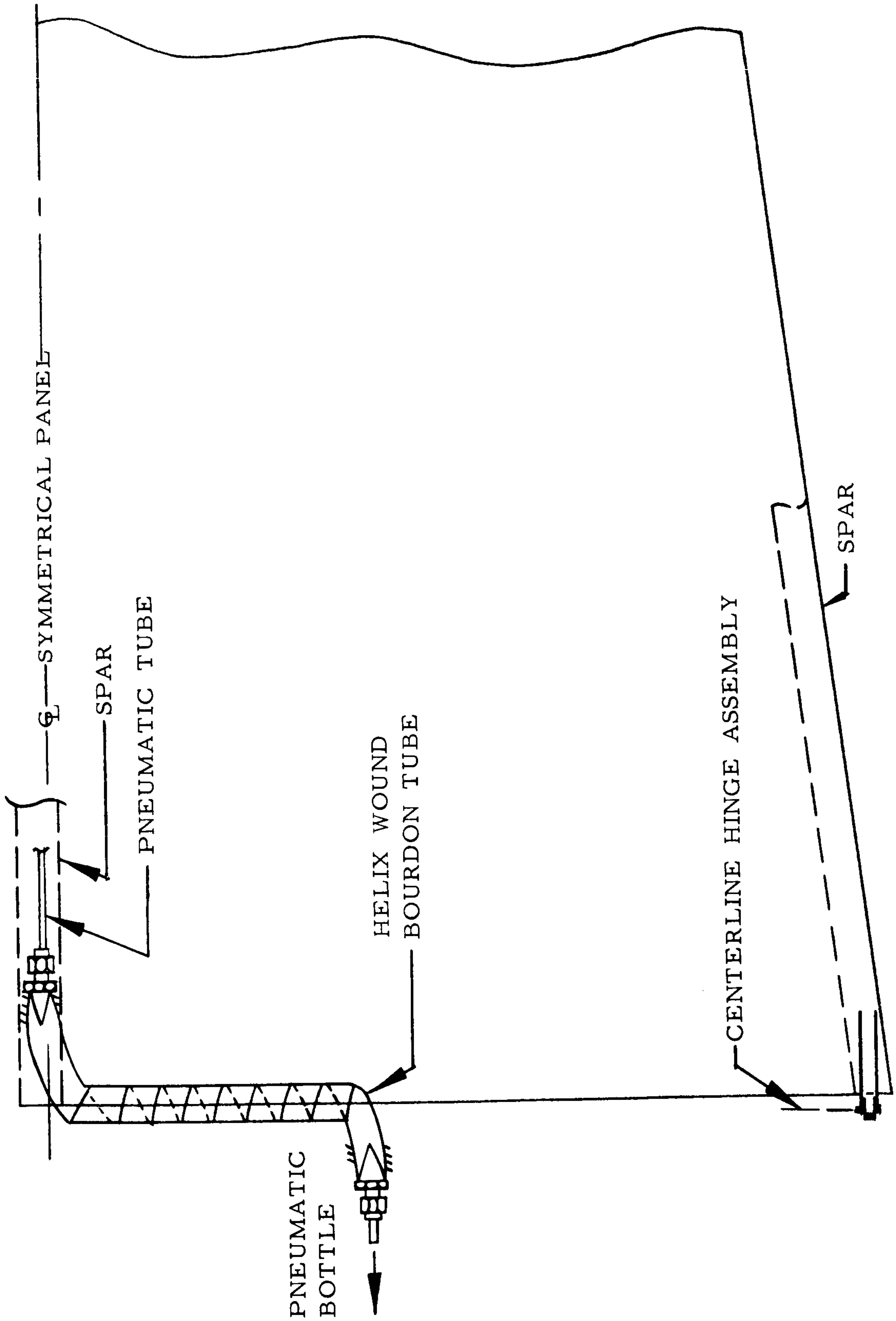
1 Consists of One Main Panel And Two Auxiliaries Except for 1

FIGURE 4.1-2 MODULAR SOLAR ARRAY CONCENTRATOR SATURN IB/CENTAUR

3

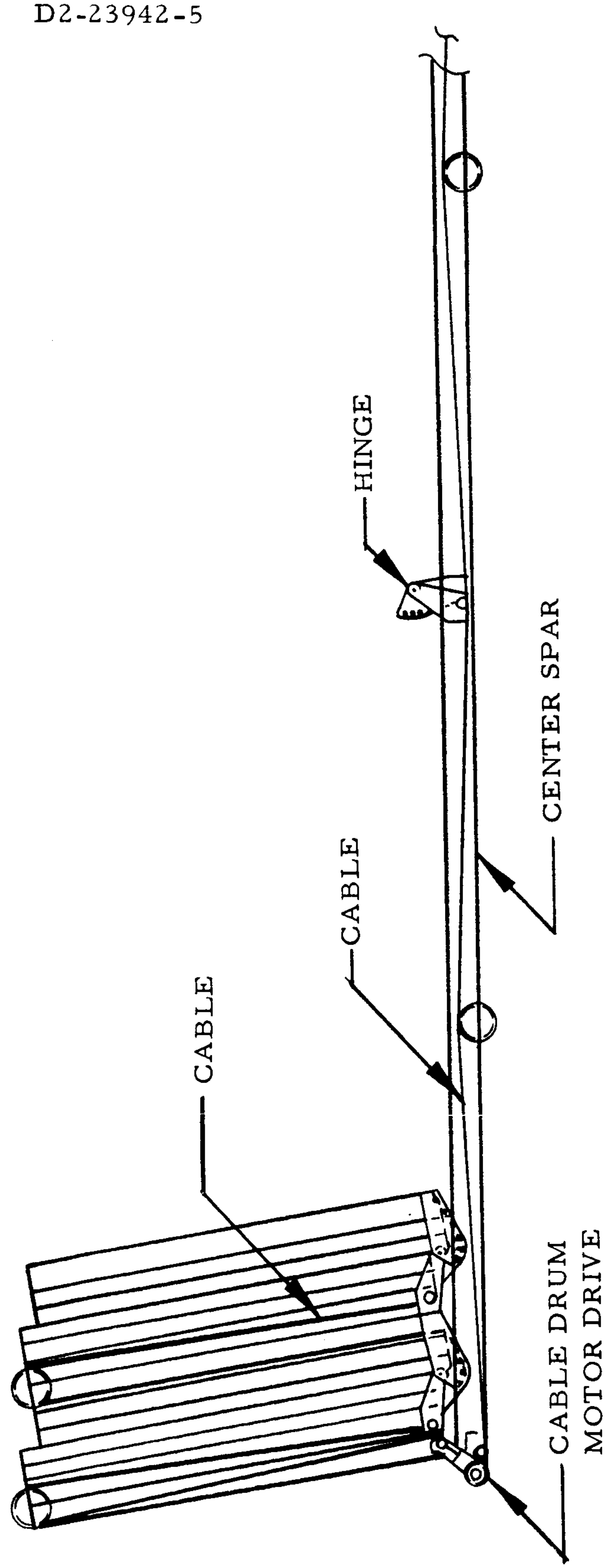


SPRING DAMPER
FIGURE 4.1-3



BOURDON TUBE
FIGURE 4.1-4

MOTOR - PULLEY - CABLE



MOTOR-PULLEY-CABLE
FIGURE 4.1-5

AUXILIARY PANEL DEPLOYMENT

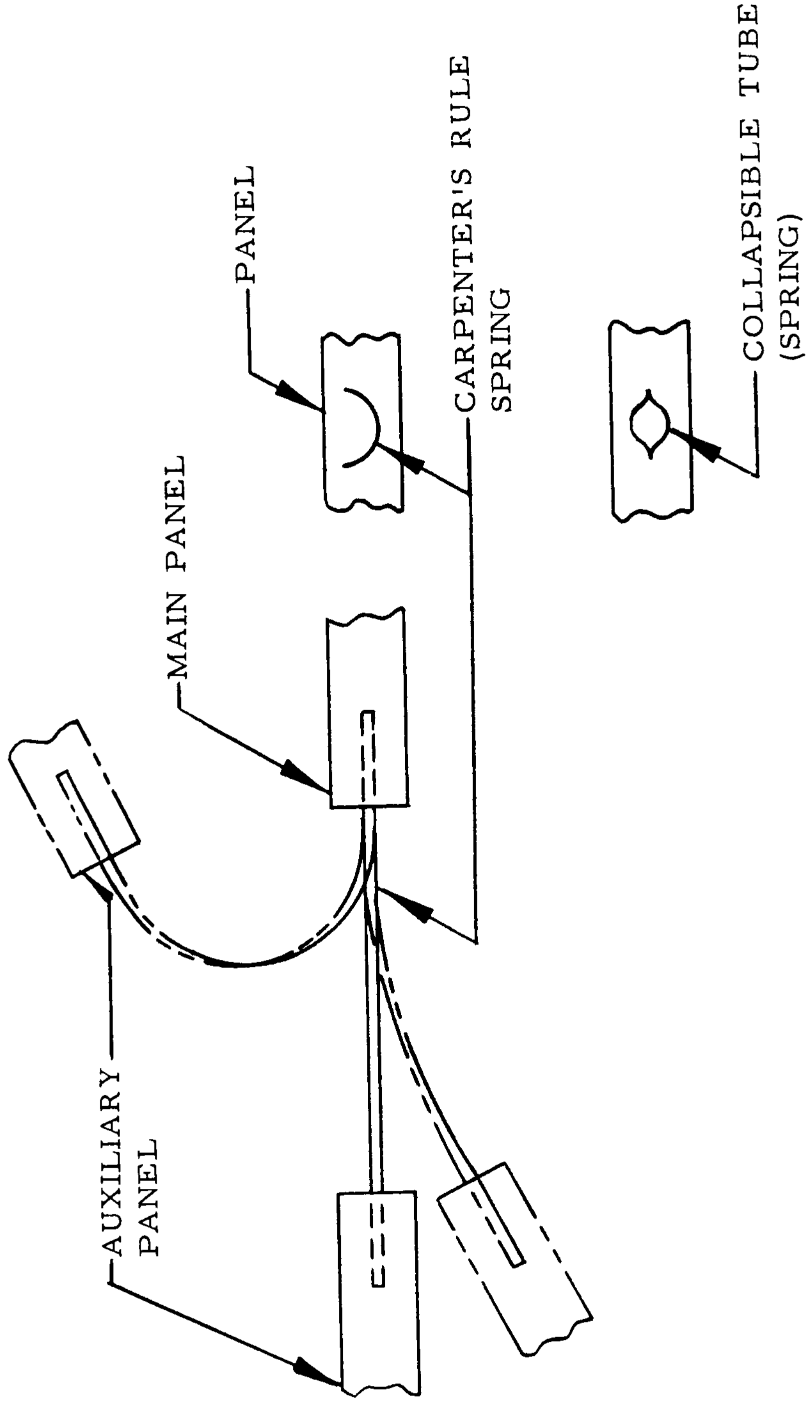
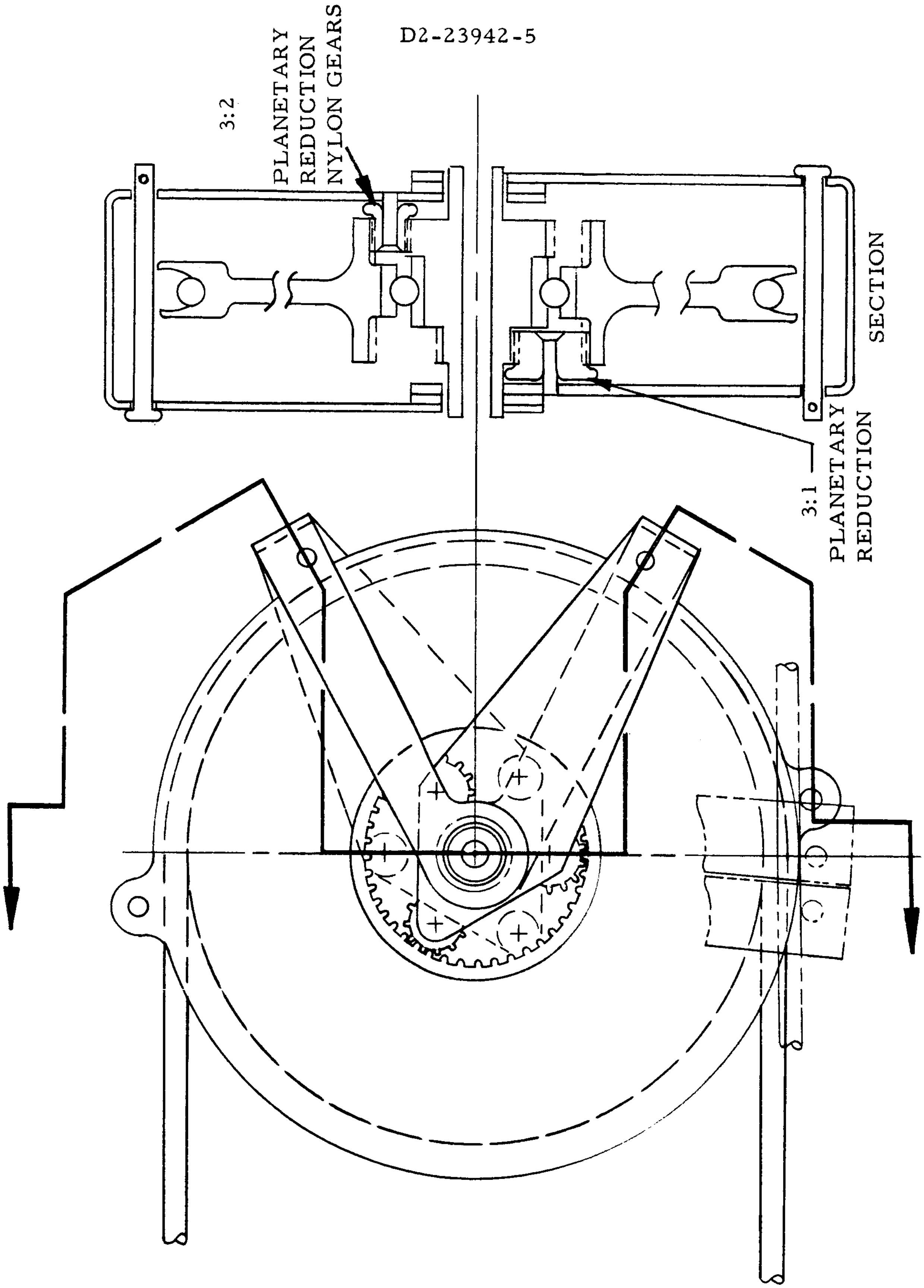
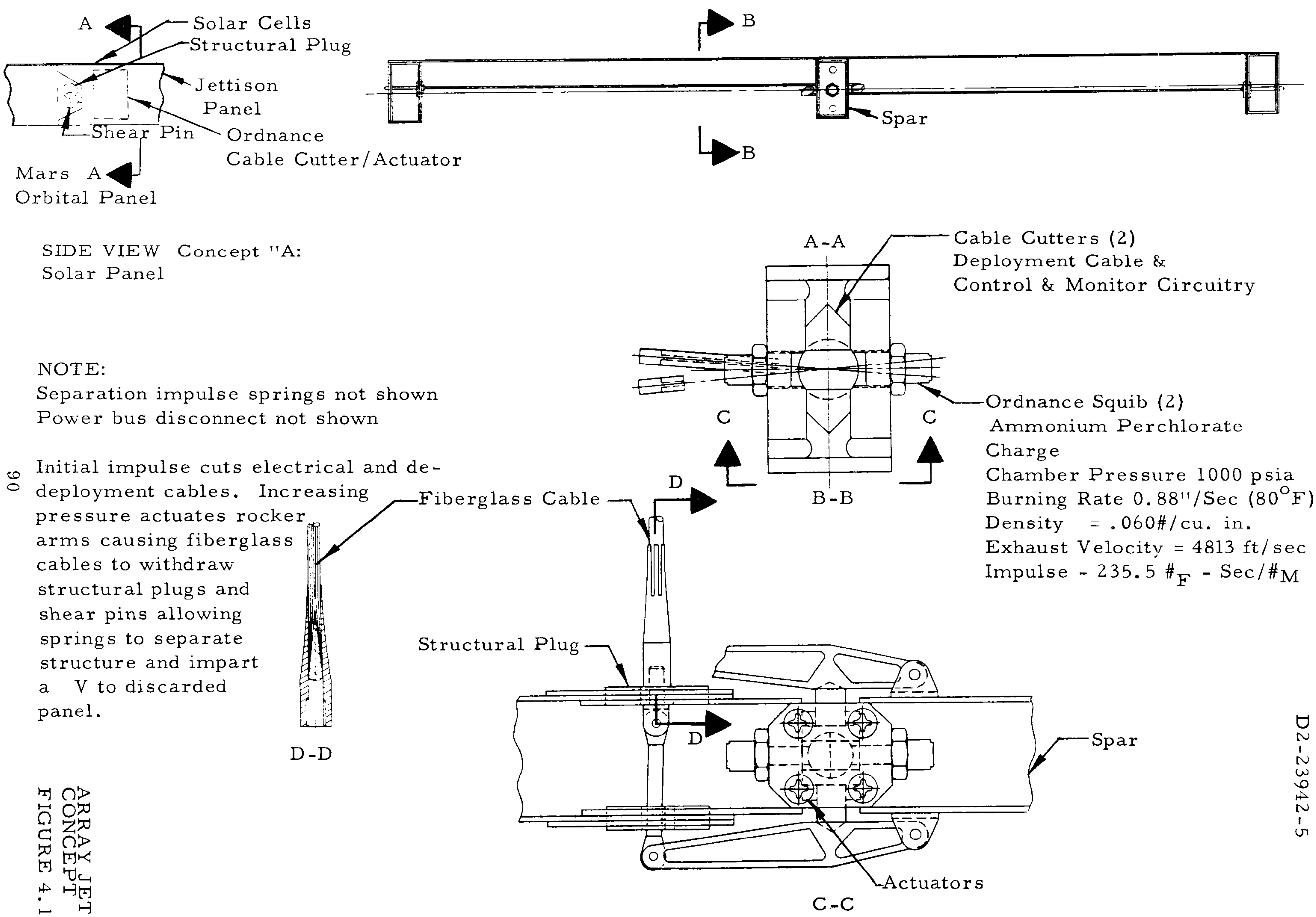


FIGURE 4.1-6

AUXILIARY PANEL
DEPLOYMENT
FIGURE 4.1-6



SHEAVE & GUARD
SOLAR PANEL HINGE POINT
FIGURE 4.1-7



NOTE:
Separation impulse springs not shown
Power bus disconnect not shown

Initial impulse cuts electrical and de-deployment cables. Increasing pressure actuates rocker arms causing fiberglass cables to withdraw structural plugs and shear pins allowing springs to separate structure and impart a V to discarded panel.

ARRAY JETTISON CONCEPT
FIGURE 4.1-8

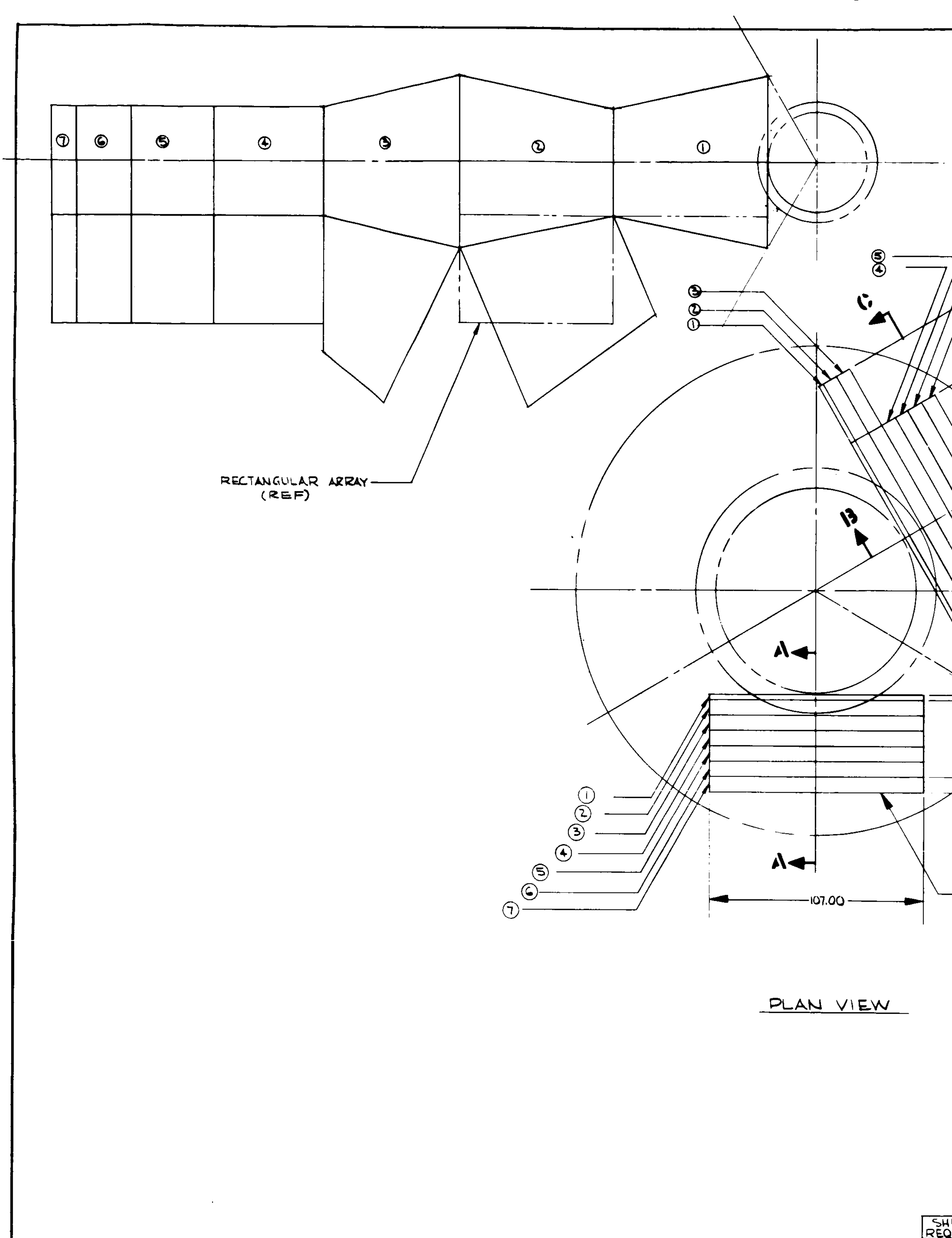
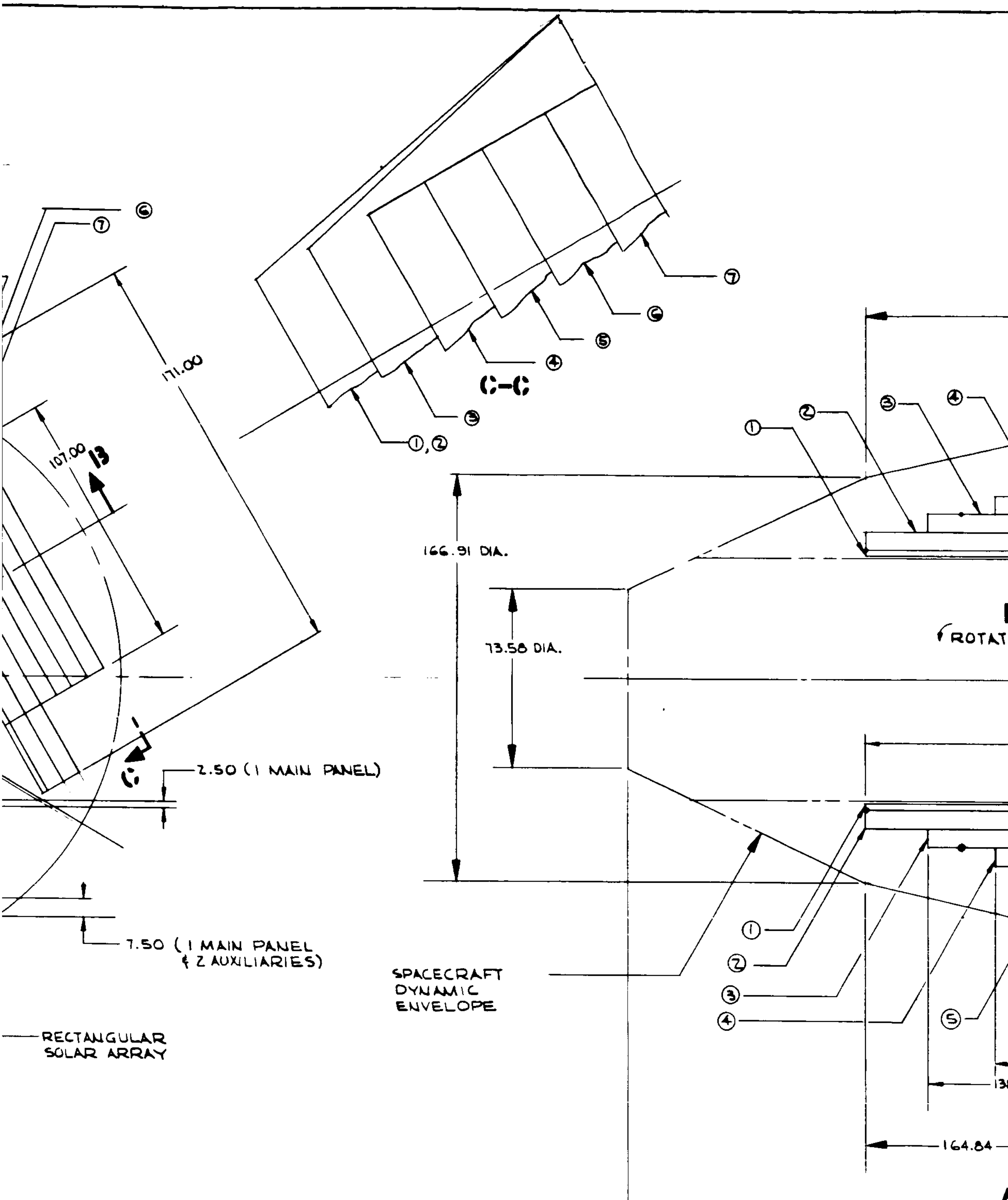


Fig 4.1-9
①



PANEL NO.	AREA (SQ. FT.)	
	RECTANGULAR	TRAPEZOIDAL
1 2 3 4 5 6 7	114 342 308 245 185 122 55	131 393 310 245 185 122 55
1 ASSY 1 ARRAY	1371 4113	1441 4323
ROUD EXTENSION D TO MEET 5500 SQ. FT.	42 IN.	

Fig 4.1-9 (2)

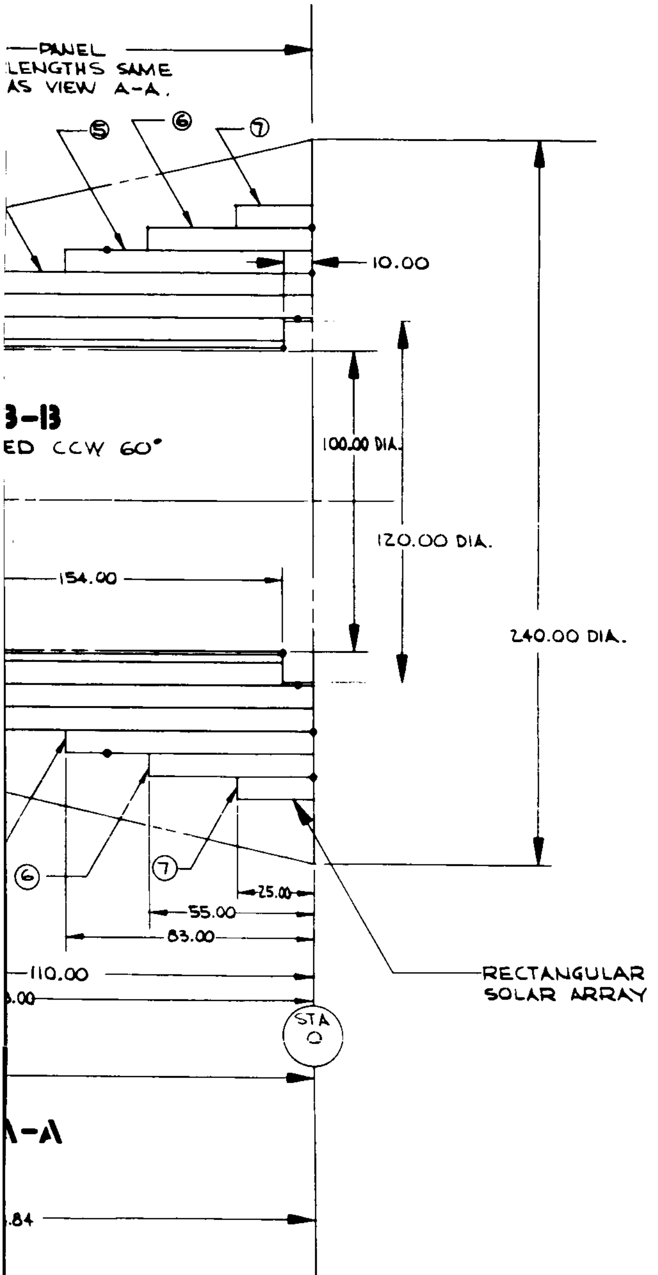
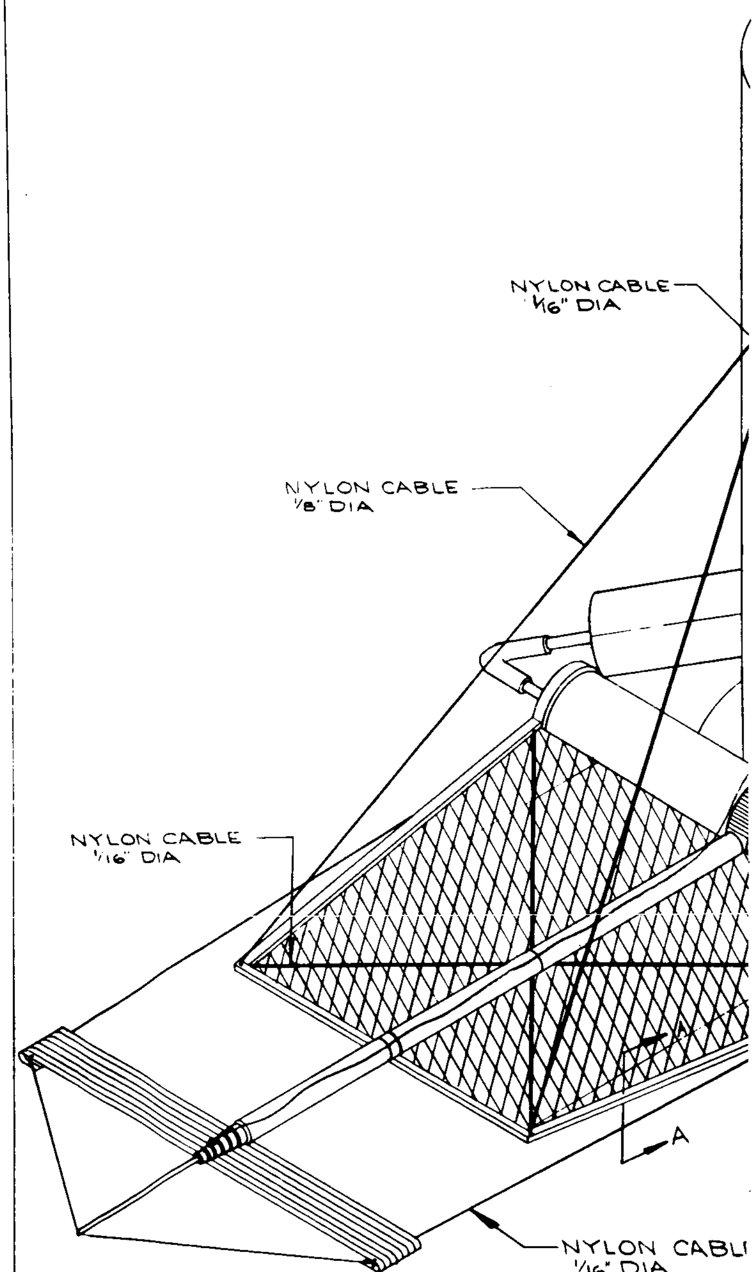


FIGURE 4.1-9 FOLDING MODULAR SOLAR ARRAY 3 WING-SATURN IB/CENTAUR

3



ISOMETRIC
(OUT/LOCKED

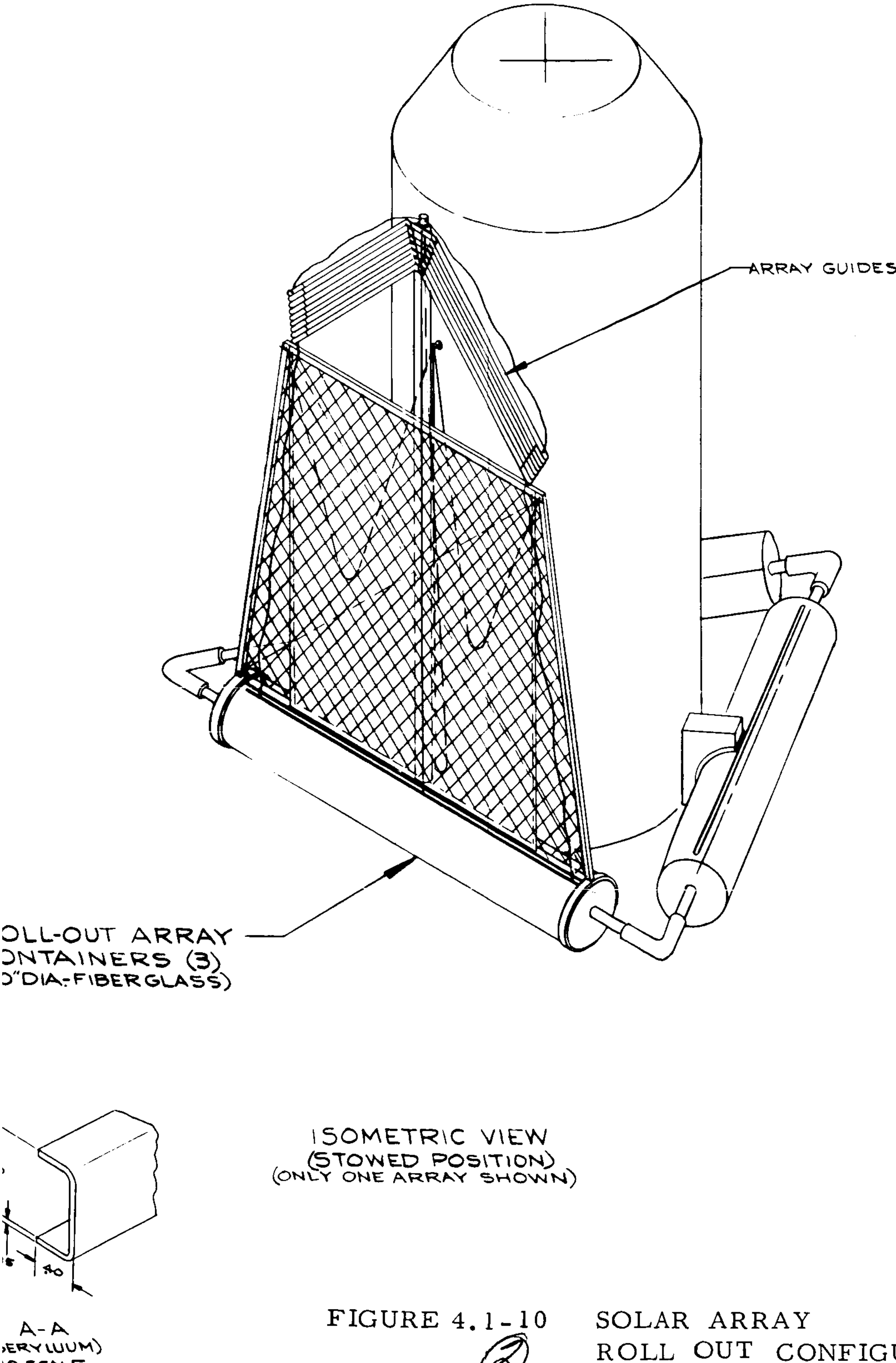
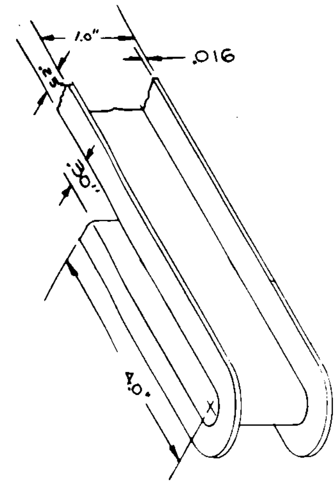
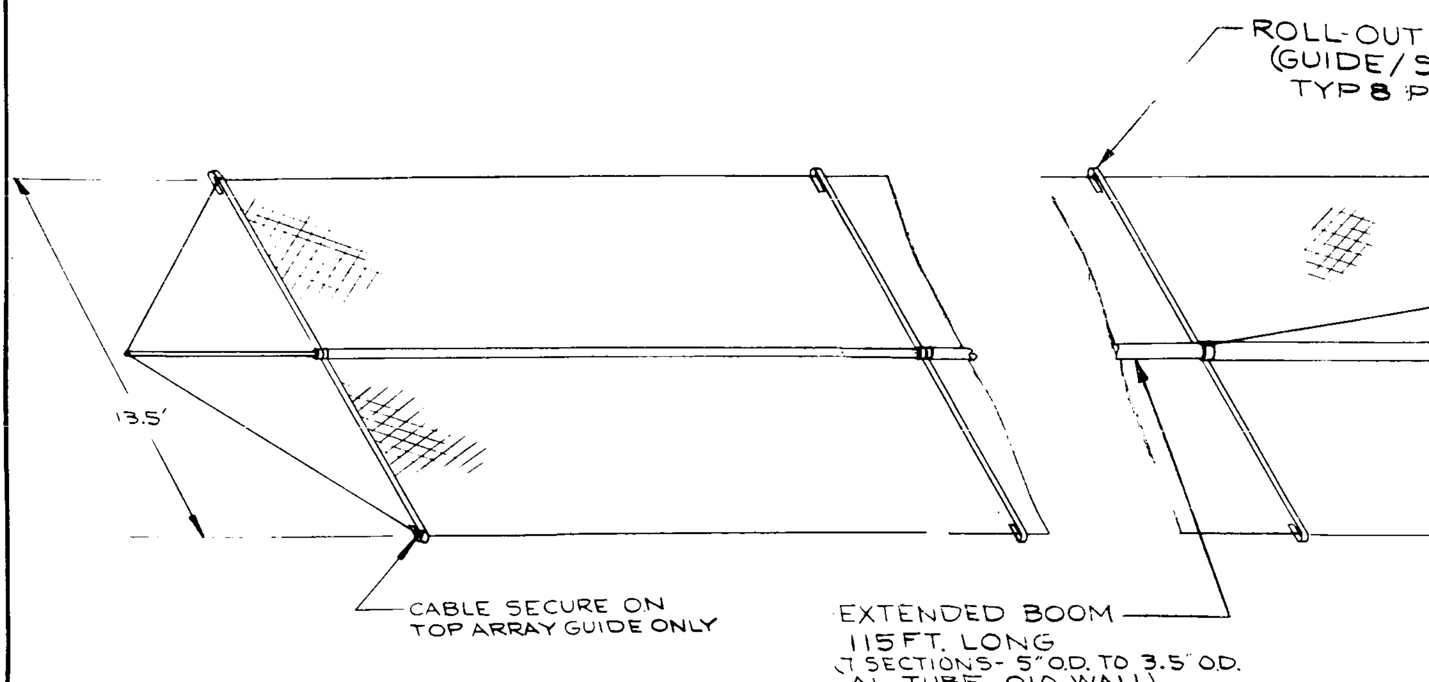


FIGURE 4.1-10 SOLAR ARRAY
ROLL OUT CONFIGURATION
EXTENDING BOOM

2



DETAIL I_{A7}

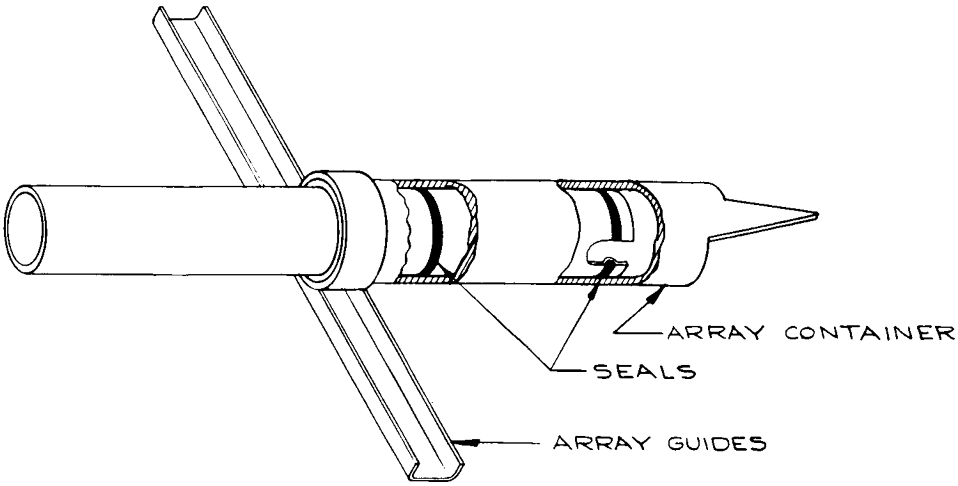


ISOME
BOOM/ARRA
(TYP 3 PLA

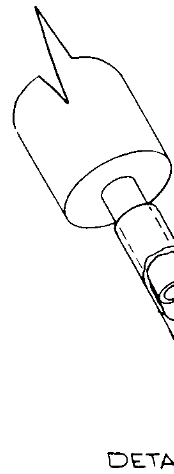
Fig 4.1-11

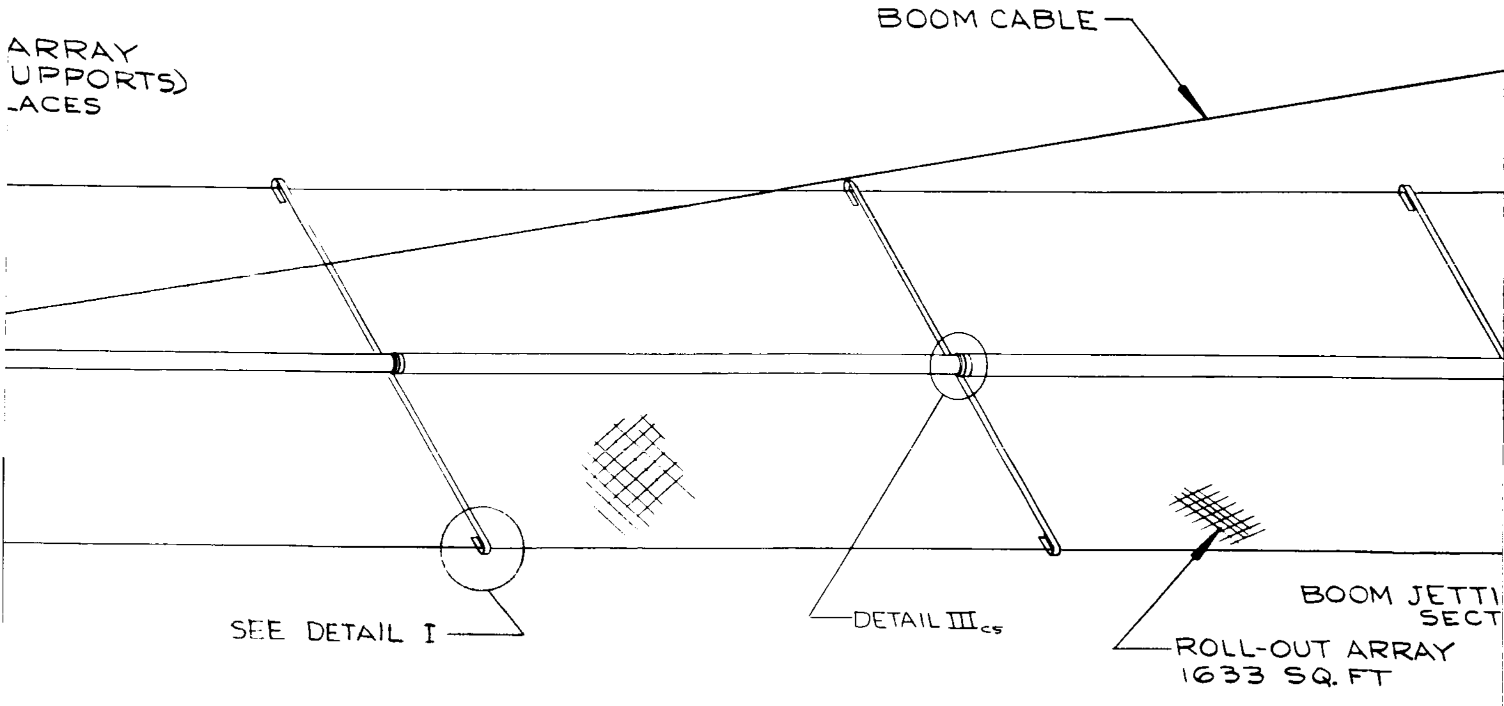
6

0



DETAIL III_{B4}





TRIC VIEW
Y EXTENDED
.CES)

Fig 4.1-11

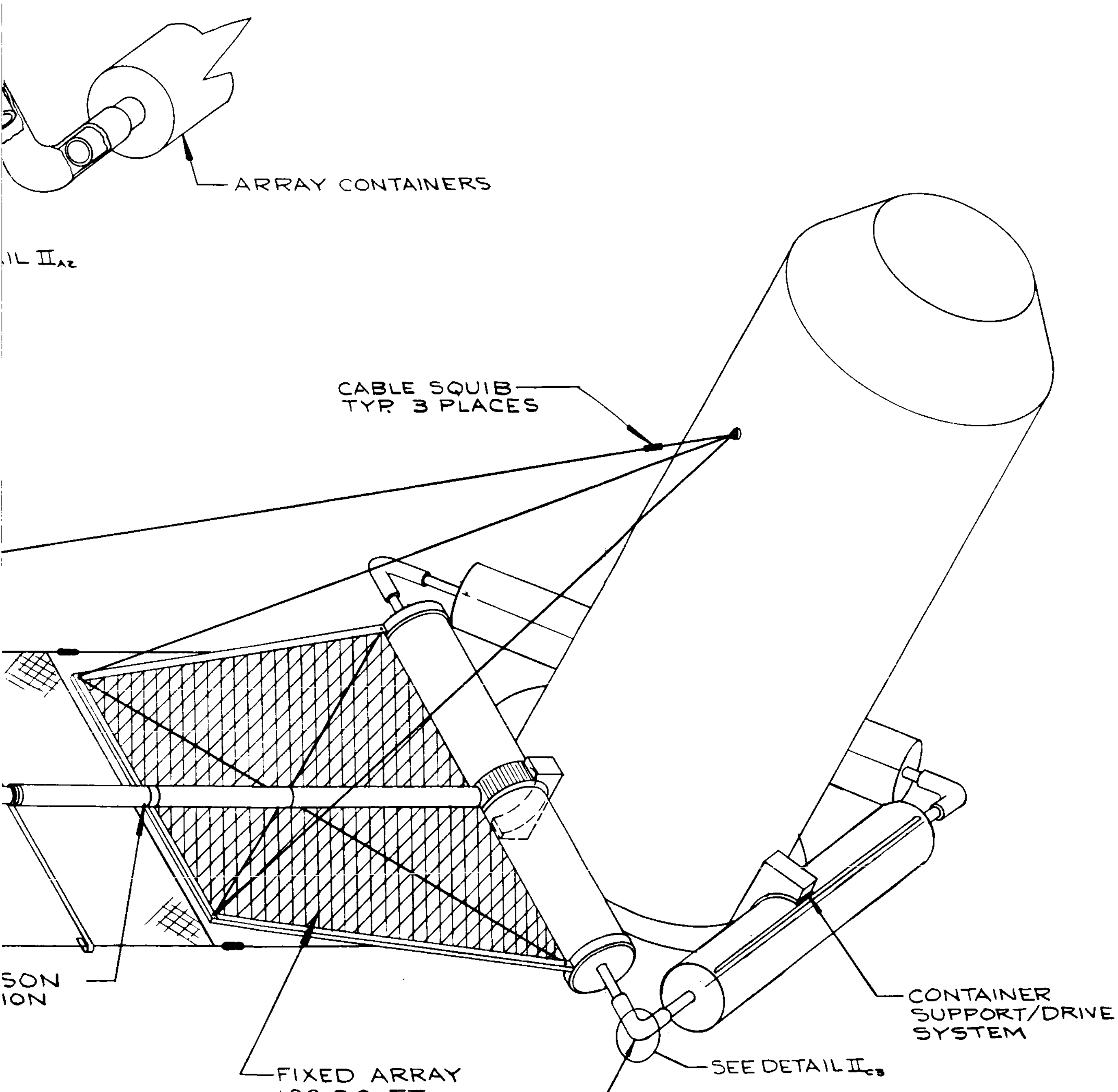
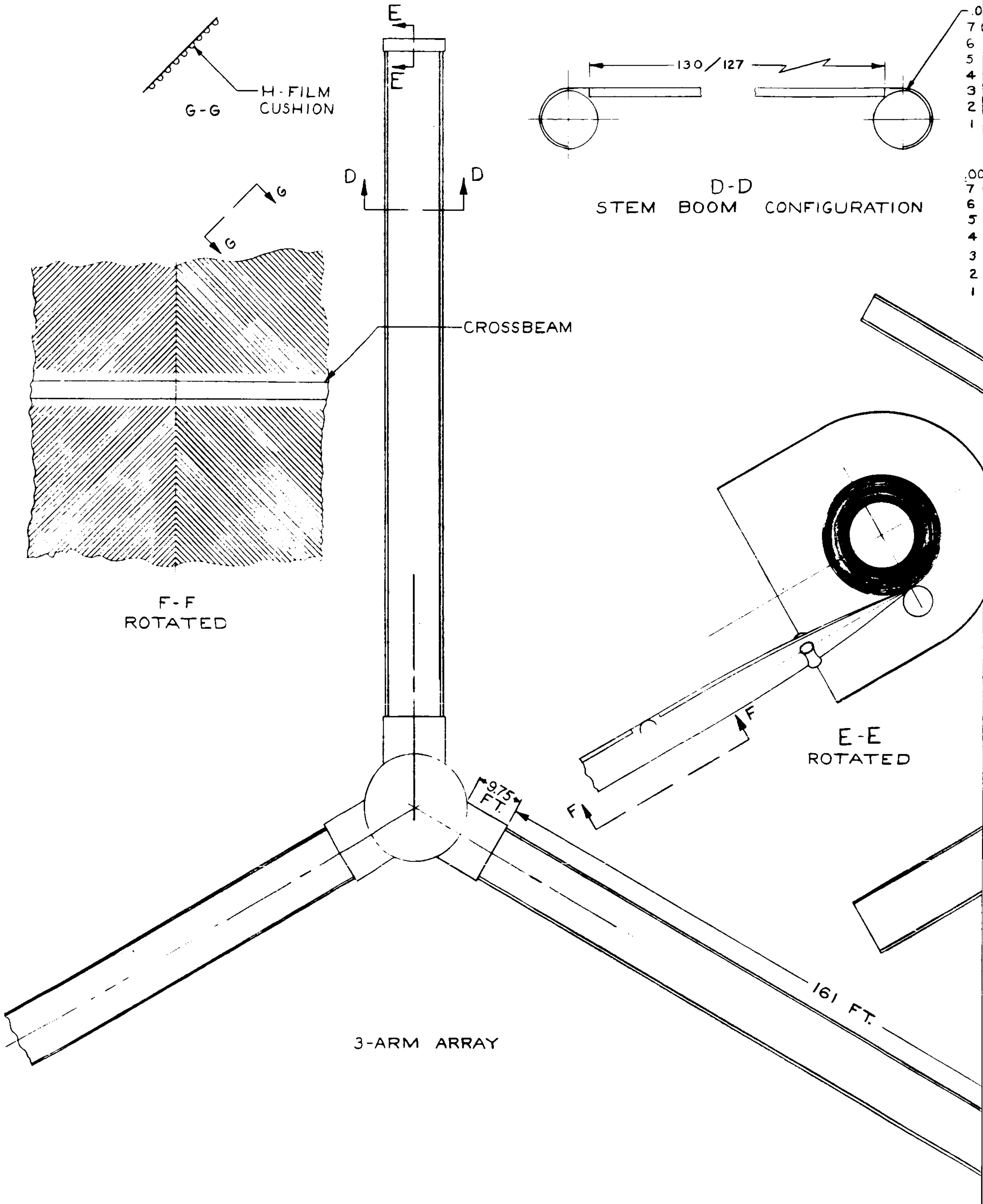


FIGURE 4.1-11 SOLAR ARRAY -
ROLL OUT - CONFIGURATION
EXTI NDING BOOM

(3)



E
E
H-FILM
CUSHION
G-G
130/127
D-D
STEM BOOM CONFIGURATION
D
D
G
G
CROSSBEAM
F-F
ROTATED
E-E
ROTATED
F
F
9.75 FT.
161 FT.
3-ARM ARRAY

Fig 4.1-12

(7)

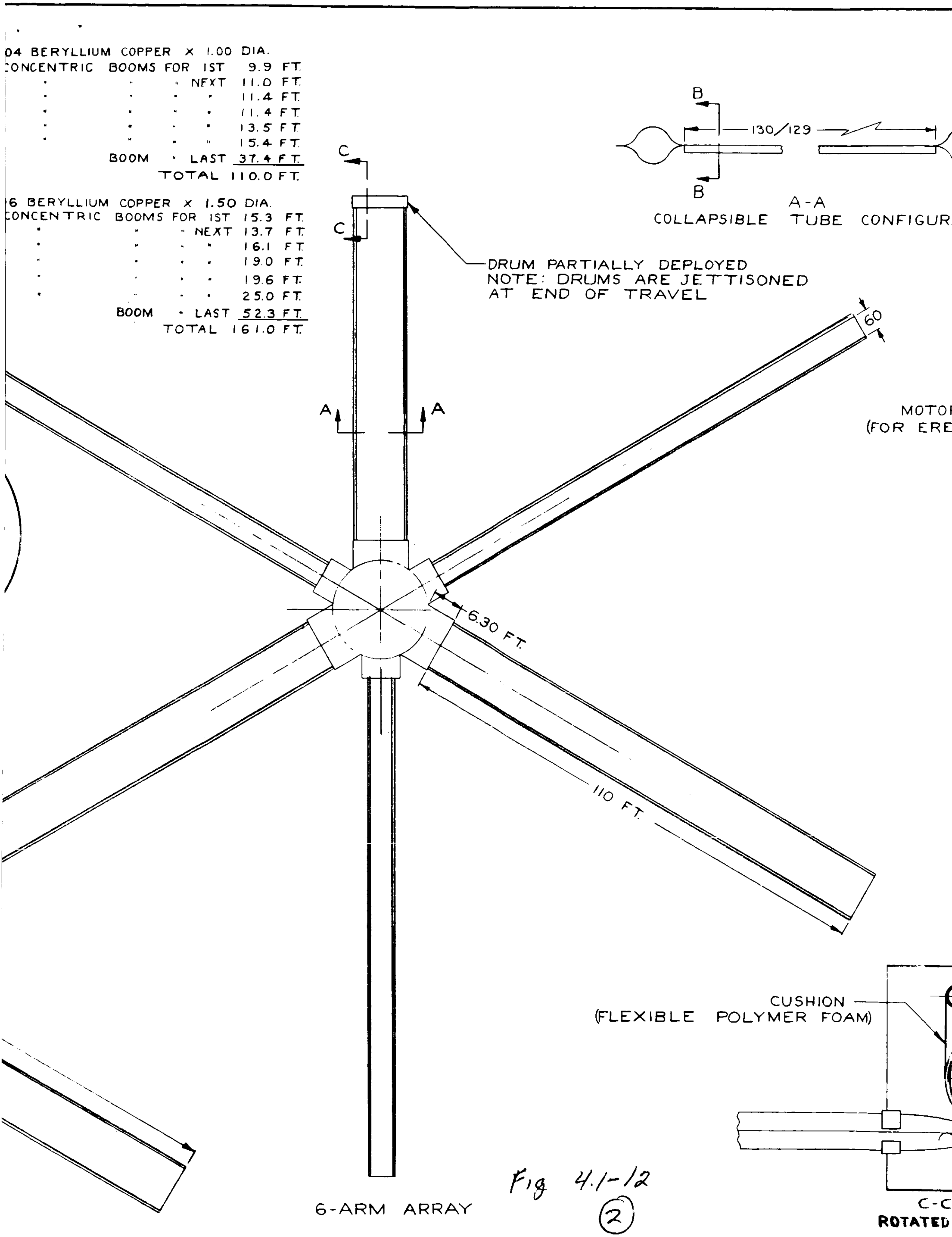

04 BERYLLIUM COPPER x 1.00 DIA.
CONCENTRIC BOOMS FOR 1ST 9.9 FT.
" NEXT 11.0 FT.
11.4 FT.
11.4 FT.
13.5 FT.
15.4 FT.
BOOM " LAST 37.4 FT.
TOTAL 110.0 FT.
16 BERYLLIUM COPPER x 1.50 DIA.
CONCENTRIC BOOMS FOR 1ST 15.3 FT.
" NEXT 13.7 FT.
16.1 FT.
19.0 FT.
19.6 FT.
25.0 FT.
BOOM " LAST 52.3 FT.
TOTAL 161.0 FT.
C
C
A
A
DRUM PARTIALLY DEPLOYED
NOTE: DRUMS ARE JETTISONED AT END OF TRAVEL
B
130/129
B
A-A
COLLAPSIBLE TUBE CONFIGUR
60
MOTO
(FOR ERE
6.30 FT.
110 FT.
CUSHION
(FLEXIBLE POLYMER FOAM)
6-ARM ARRAY
C-C
ROTATED

Fig 4.1-12

(2)

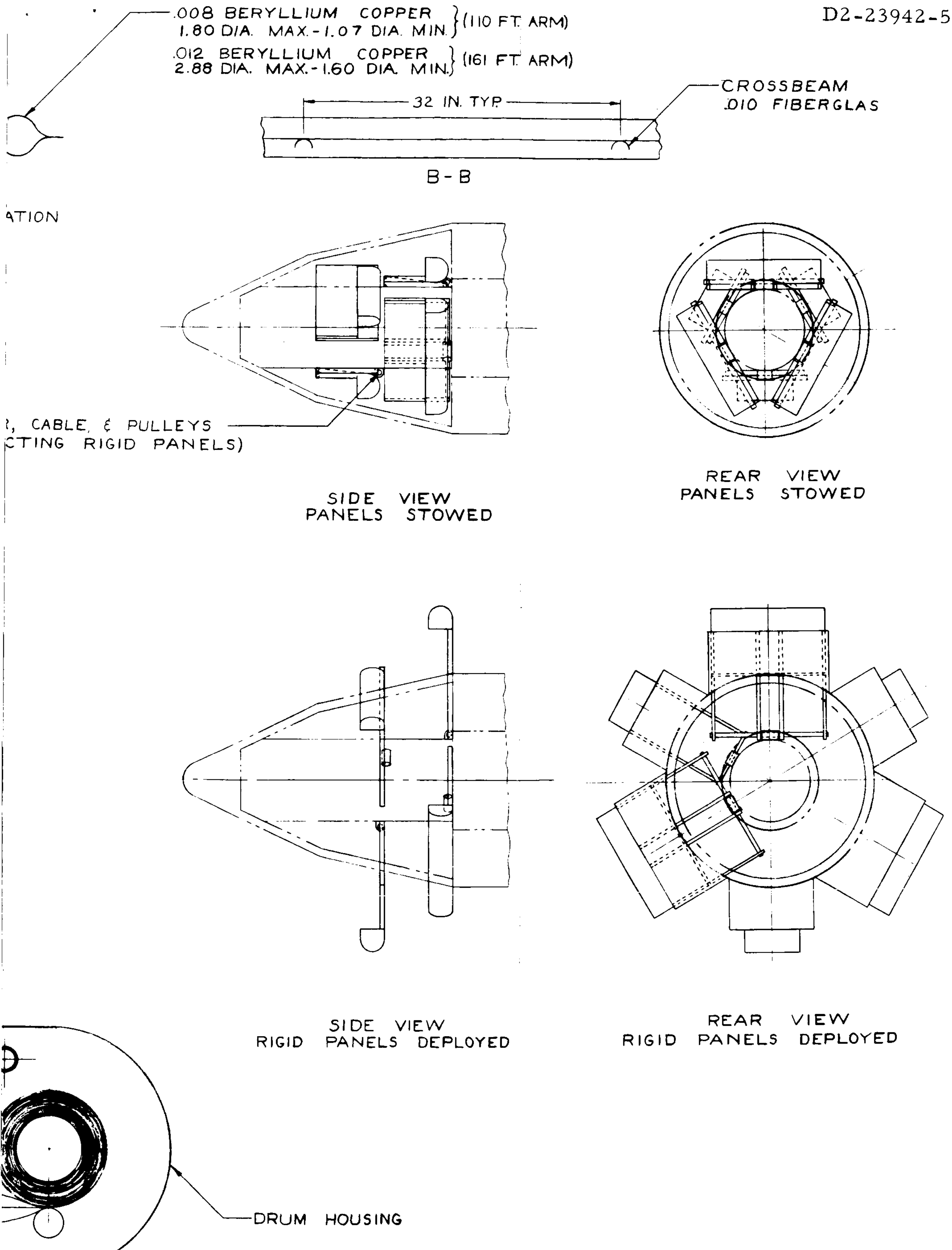


FIGURE 4.1-12 EXTENDIBLE BOOM ROLLUP ARRAY

3

4.2 ATLAS/CENTAUR CONFIGURATION STUDIES

4.2.1 Semi-Rigid Solar Array Configurations

4.2.1.1 Six Panel Array (Figure 4.2-1)

This is a folding modular solar array stowed within the envelope of the Atlas/Centaur Launch Vehicle. The concept consists of two arrays, configuration #1 (3 panel assemblies), and configuration #2 (3 panel assemblies). Configuration #1 is hinged to the base of the spacecraft envelope and configuration #2 is hinged at the top of the envelope. Each panel assembly consists of two main panels and four auxiliary panels. An optional solar vane is also shown. When the arrays are deployed, the lower array (configuration #1) casts a shadow on the upper array (configuration #2) by an amount determined by the spacecraft orientation tolerance angle. This shadow reduces the effective area of solar cells.

An alternate array concept is to hinge six panels at the base (configuration #1) or at the top (configuration #2).

Each panel consists of three spars, semi-rigid substrate and substrate support structure.

The main disadvantage of this array is the lack of adequate voids in the deployed array to accommodate spacecraft subsystems.

4.2.1.2 Four Panel Array (Figure 4.2-2)

This folding modular solar array consists of four panel assemblies hinged at the top and stowed within the Atlas/Centaur envelope. The deployed length of each panel assembly is 29.5 feet about the spacecraft centerline. In the deployed position there is more space for deployment of spacecraft subsystem equipment as compared to the six panel configuration. The individual panels are basically the same as for the six panel array above.

4.2.2 Atlas/Centaur Modular Array Configuration Weight Analysis

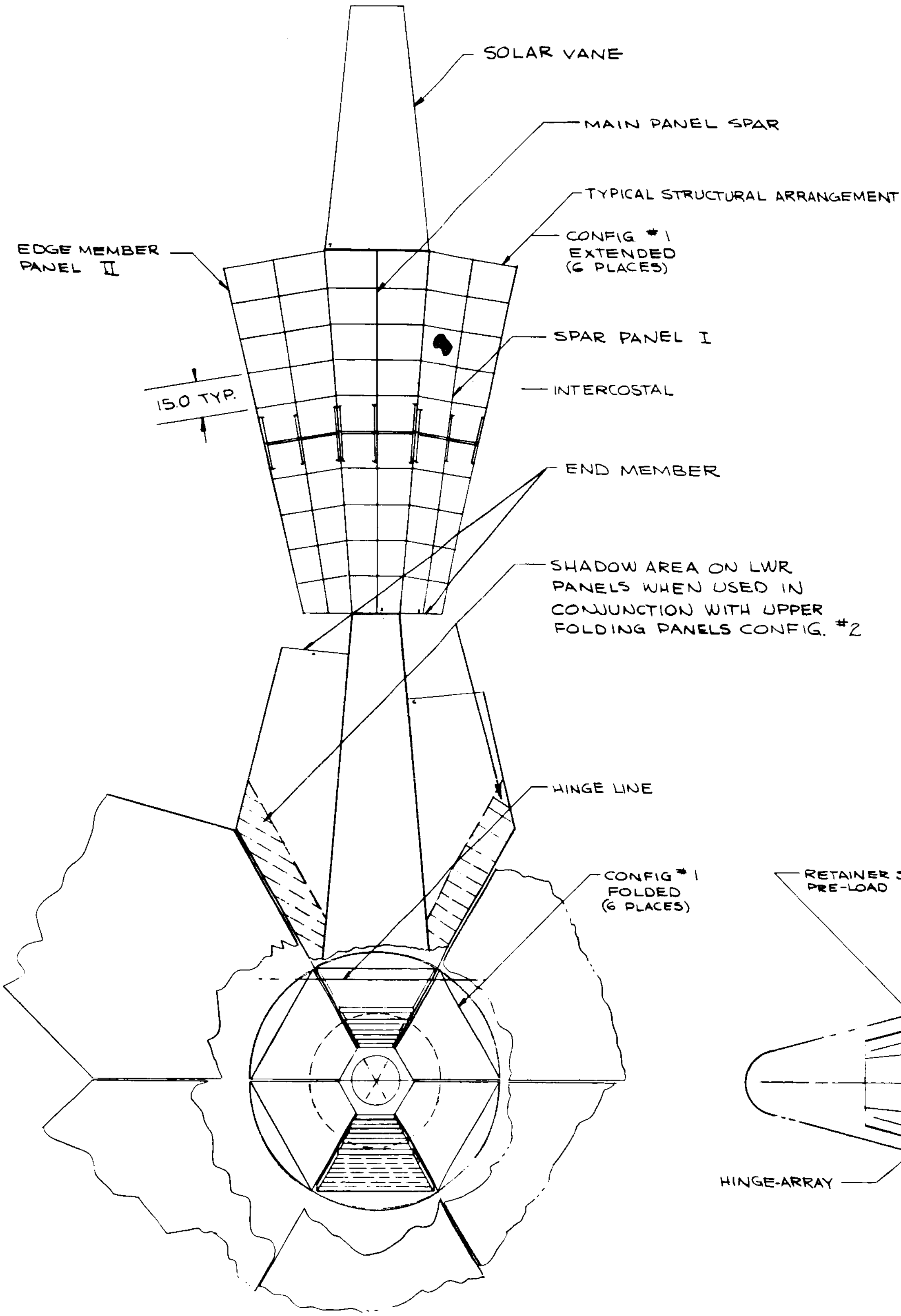
The array has six panel assemblies which may all be attached to the upper or lower end of the payload vehicle. The sub-panels in this configuration are semi-rigid trapezoids using tape substrate. The weights for lower and upper attachment are shown below. The primary difference is in hinging the sub-arrays to the vehicle.

	Lower	Upper
Cell Stack + Substrate	.2123	.2123
Secondary Cabling	.0250	.0250
Support Structure	.1488	.1402
Mechanisms	.0718	.0582
	.4579	.4357

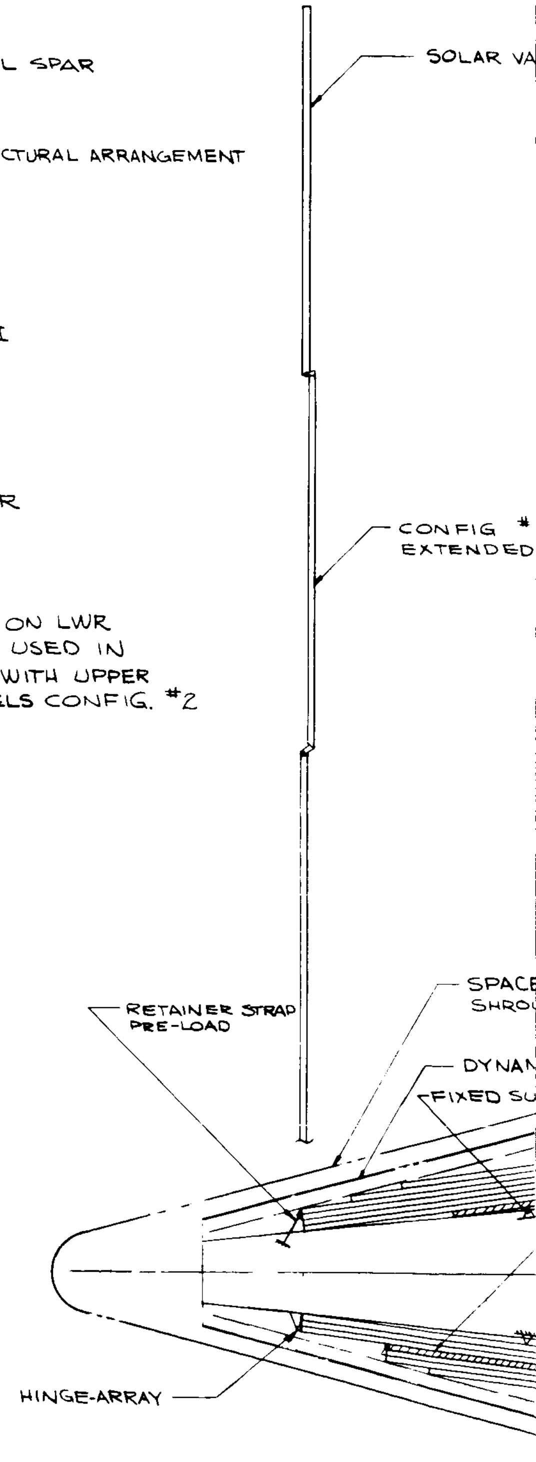
4.2.3 Rollup Solar Array Configuration

STEM Boom - Rollup Solar Array (Figure 4.2-3) - The follup array is stowed within the shroud of the Atlas/Centaur. The solar array consists of four solar panel assemblies, each of which has a semi-rigid panel and a flexible rollup solar panel. The flexible rollup panel is jettisoned prior to the Mars terminal maneuver and the semi-rigid panel is retained for secondary power generation.

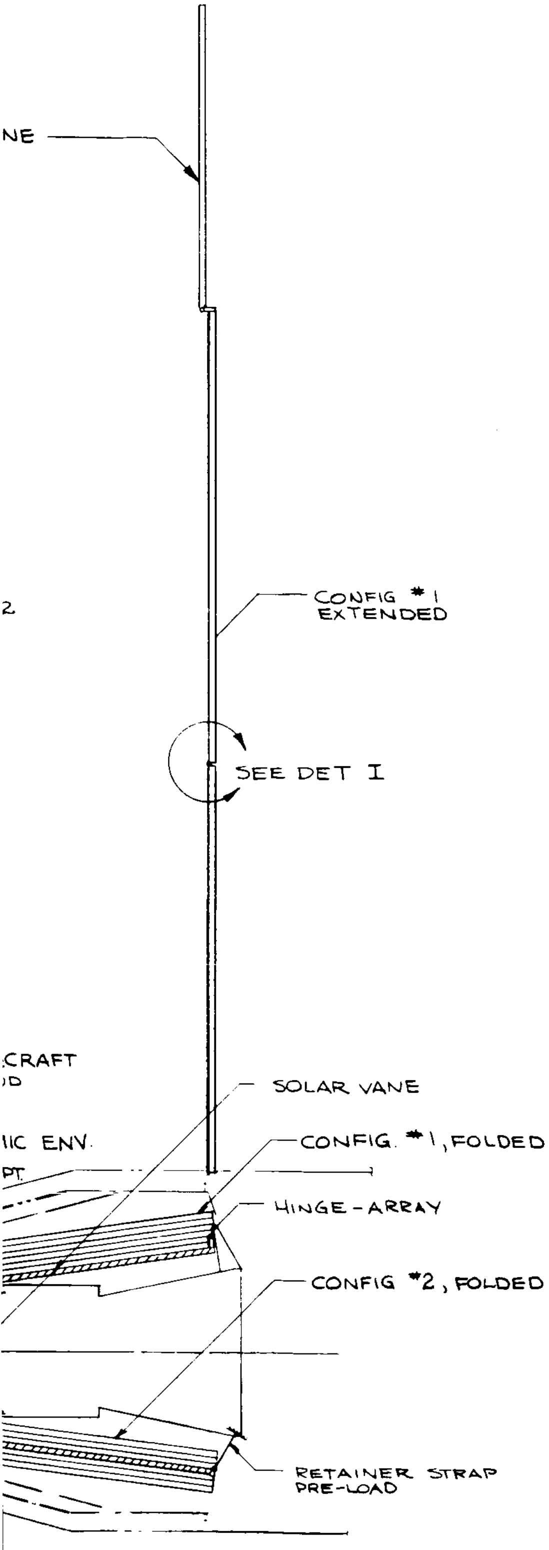
The semi-rigid panel has a pretensioned fiberglass tape substrate supported by a beryllium frame. At the outboard end of the deployed semi-rigid panel, the 20.0 diameter case for the flexible rollup substrate is attached. When the squib is fired, the rollup case is propelled outboard by the stored energy in the stowed "STEM" (Storable Tubular Extendible Member) boom. The boom is a .50 diameter 3 mil beryllium copper tube. The rollup substrate is a 3 mil H-film with a corrugated cushion to protect the cells during stowage. Lateral cross beams are used to maintain the distance between the STEM boom spars.



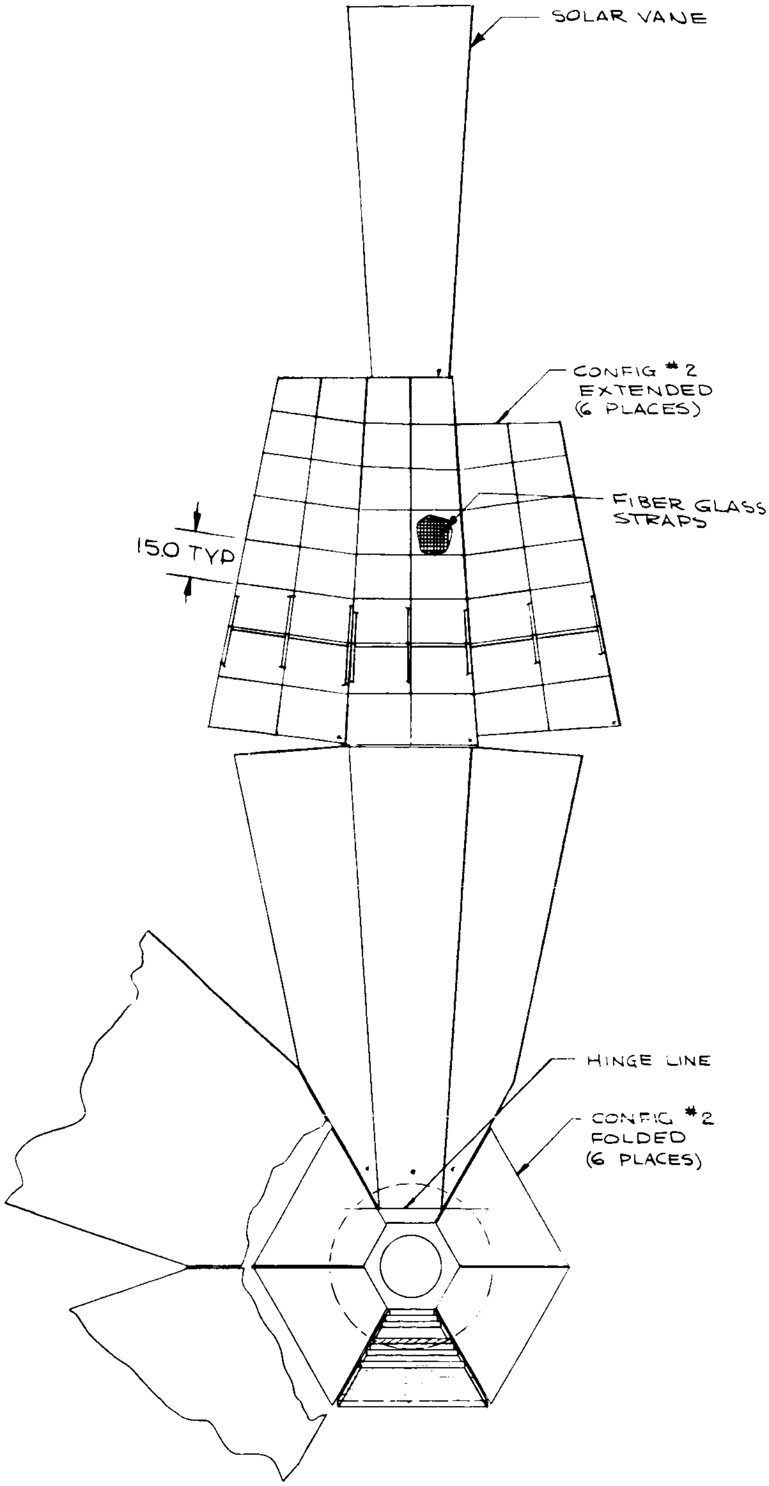
CONFIGURATION #1 - LOWER FOLDING
PLAN VIEW



CONFIGURATIO
SIDE VIE



N - *1 & *2
W- COMPOSITE



CONFIGURATION #2 - UPPER FOLDING
PLAN VIEW

OPTIONA
DE

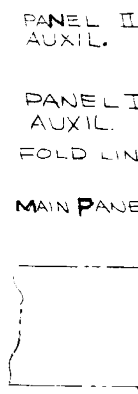
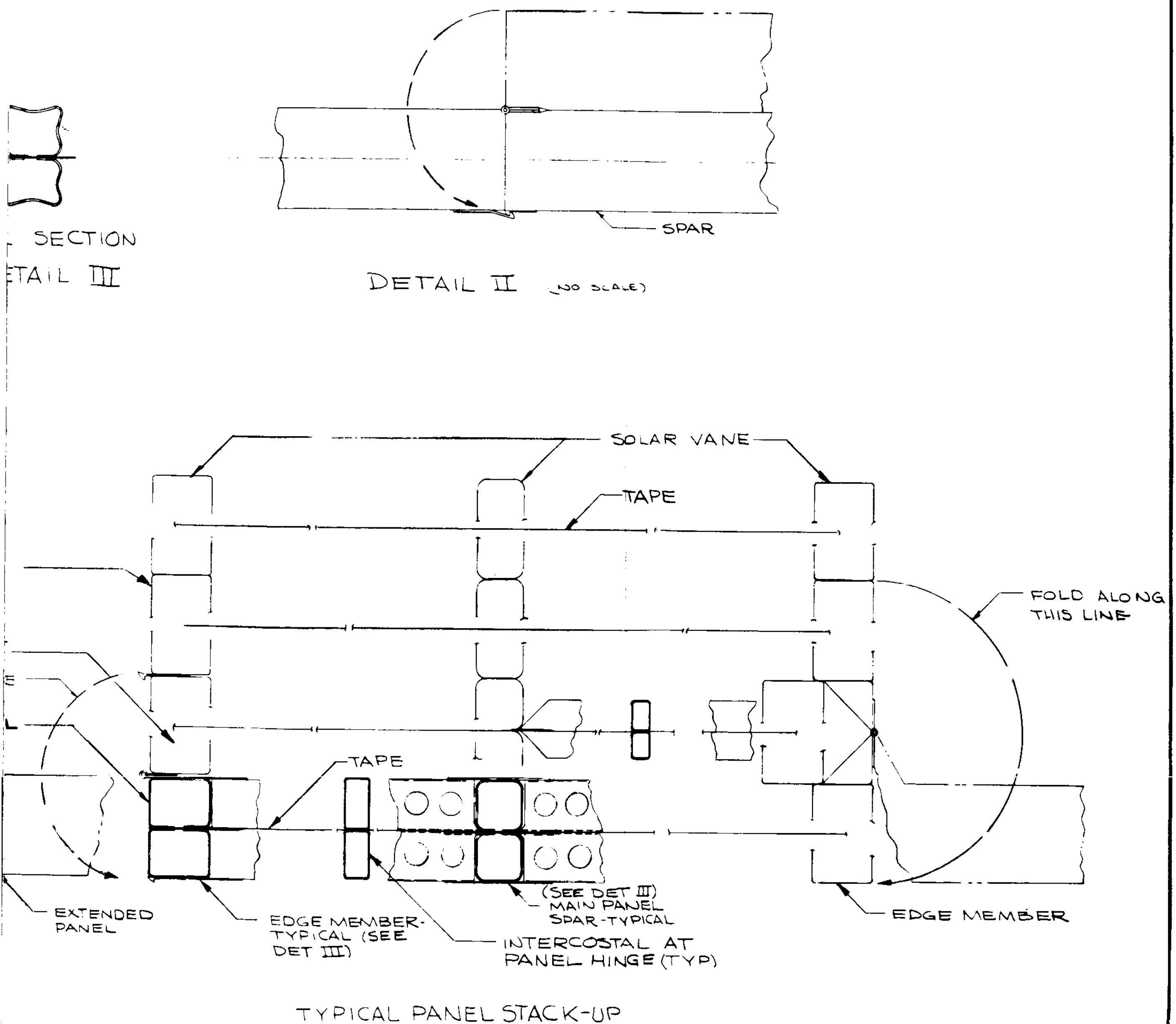


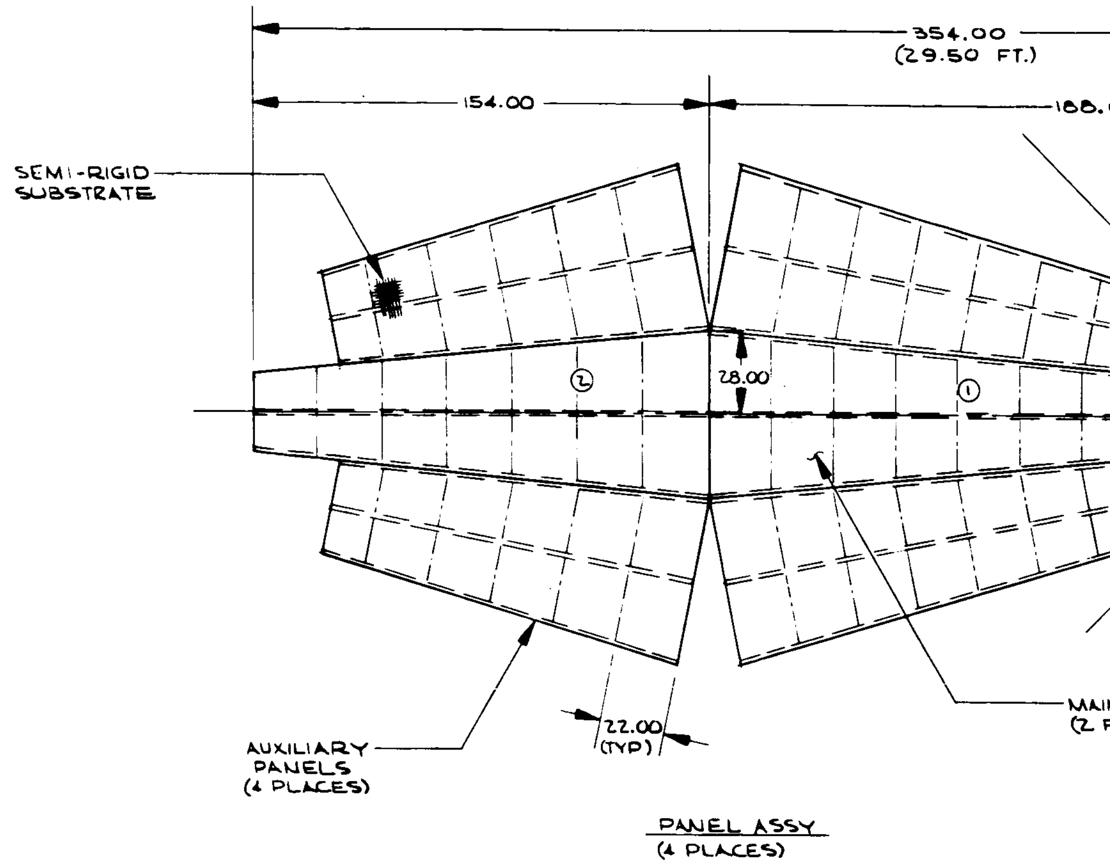
TABLE A	
COMBINATIONS	
NO.	CONFI
3	#1
3	#2
6	#1
6	#2
SOLAR VANE	
6	#1
6	#2



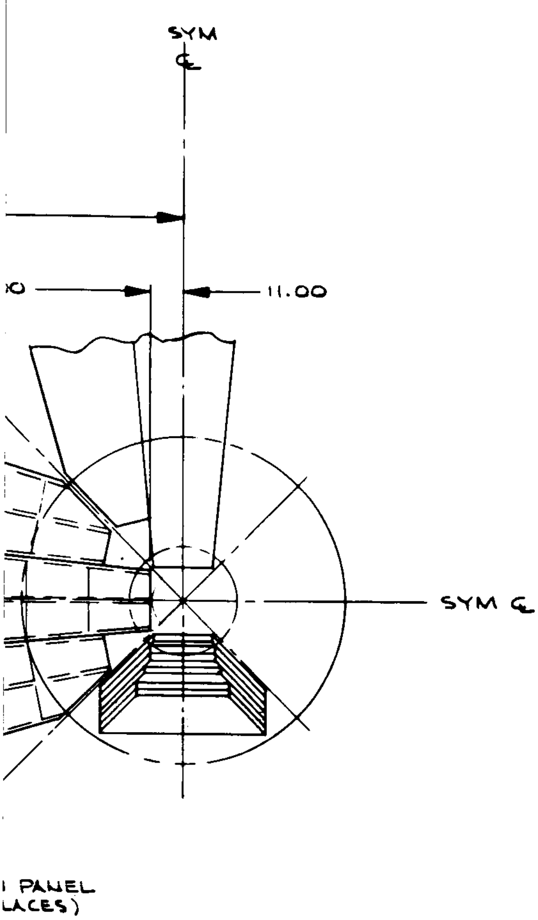
REA vs PANEL COMBINATION
AREA - FT²
DOES NOT INCLUDE SOLAR VANE
523 (CORRECTED FOR SHADOW)
577
1100 FT² TOTAL
1122 FT² TOTAL
1134 FT² TOTAL
AREA FT²
140 FT² TOTAL
180 FT² TOTAL

FIGURE 4.2-1 SOLAR ARRAY - SIX PANEL - SEMI-RIGID - ATLAS/CENTAUR

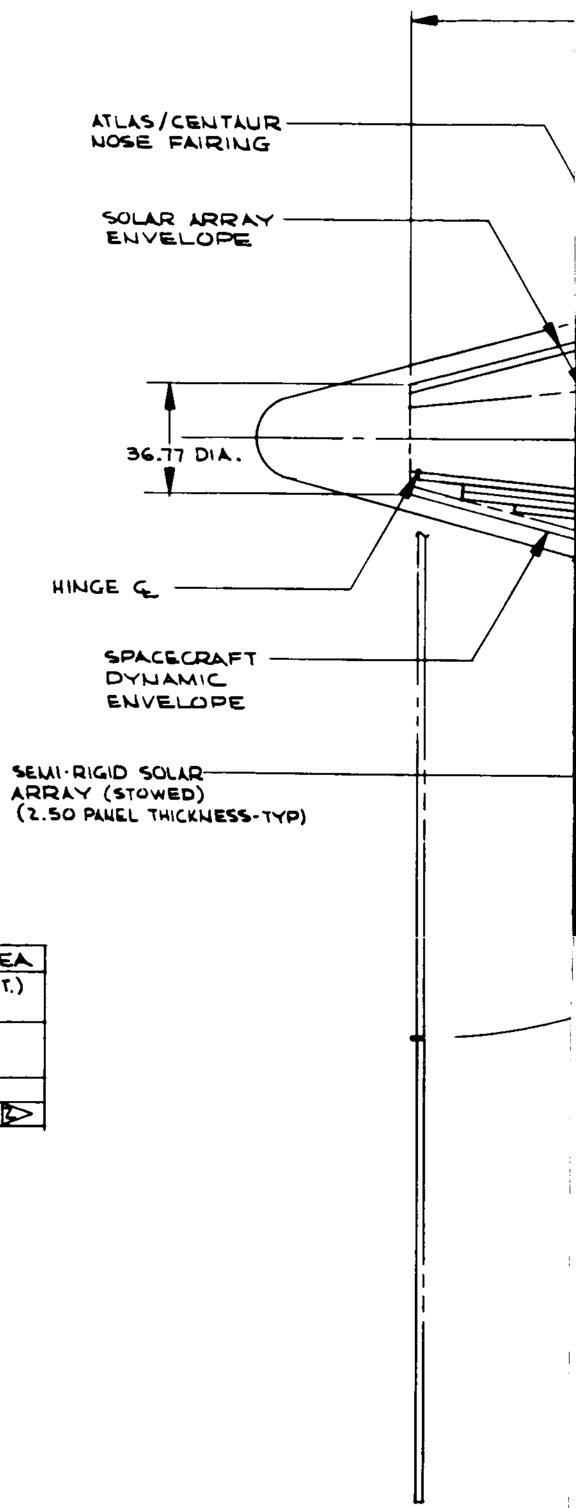
(3)


354.00
(29.50 FT.)
154.00
188.
SEMI-RIGID
SUBSTRATE
28.00
AUXILIARY
PANELS
(4 PLACES)
22.00
(TYP)
PANEL ASSY
(4 PLACES)

4.2-2



PLAN VIEW
(SCALE: 1/20TH SIZE)



SOLAR ARRAY AREA	
PANEL ▷1	AREA (SQ. FT.)
①	139 SQ. FT.
②	118 SQ. FT.
PANEL ASSY	257 SQ. FT.
SOLAR ARRAY	1028 SQ. FT. ▷2

FIG 4.2-2

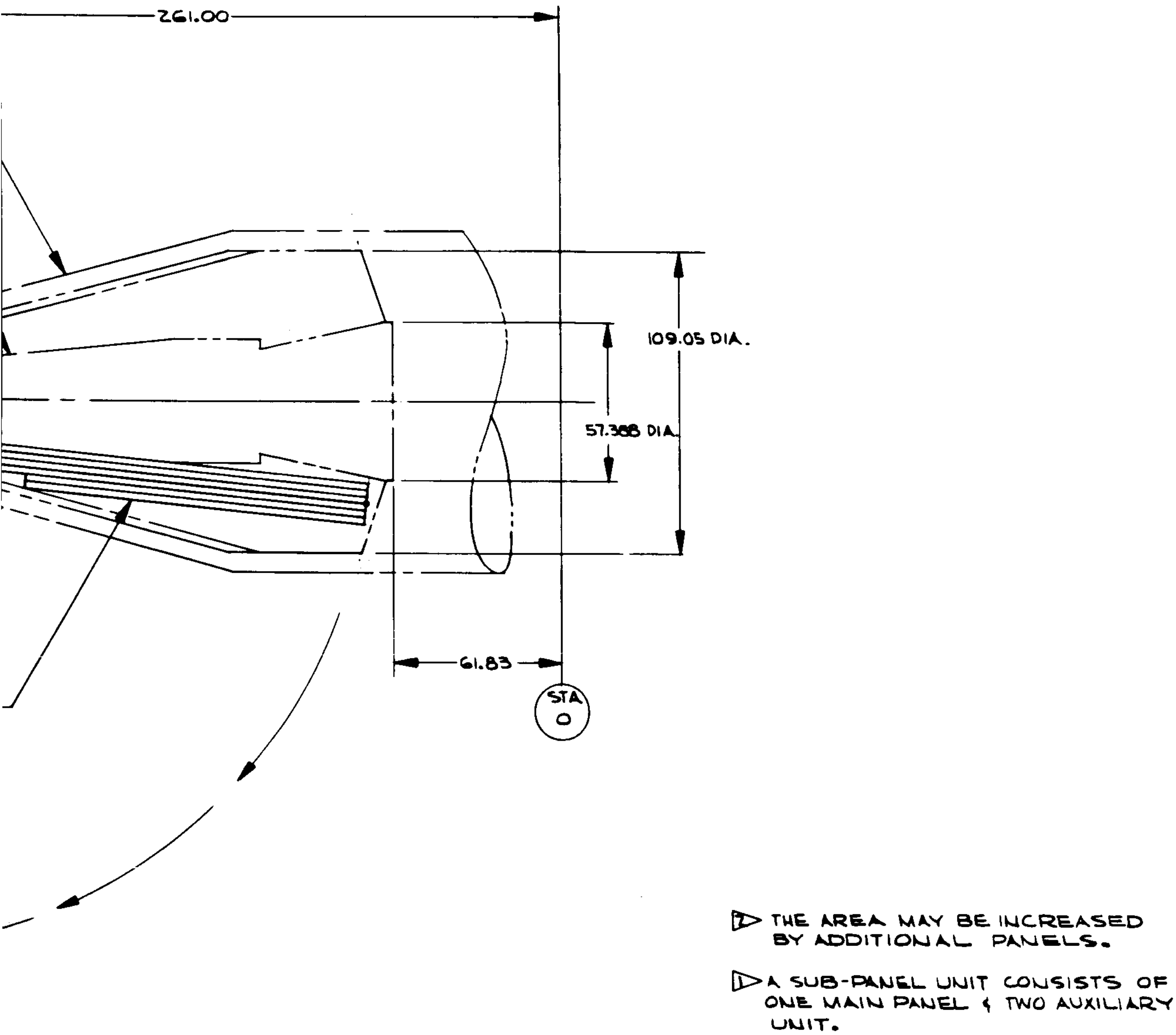


FIGURE 4.2-2 SOLAR ARRAY - FOUR PANEL ATLAS/CENTAUR

3

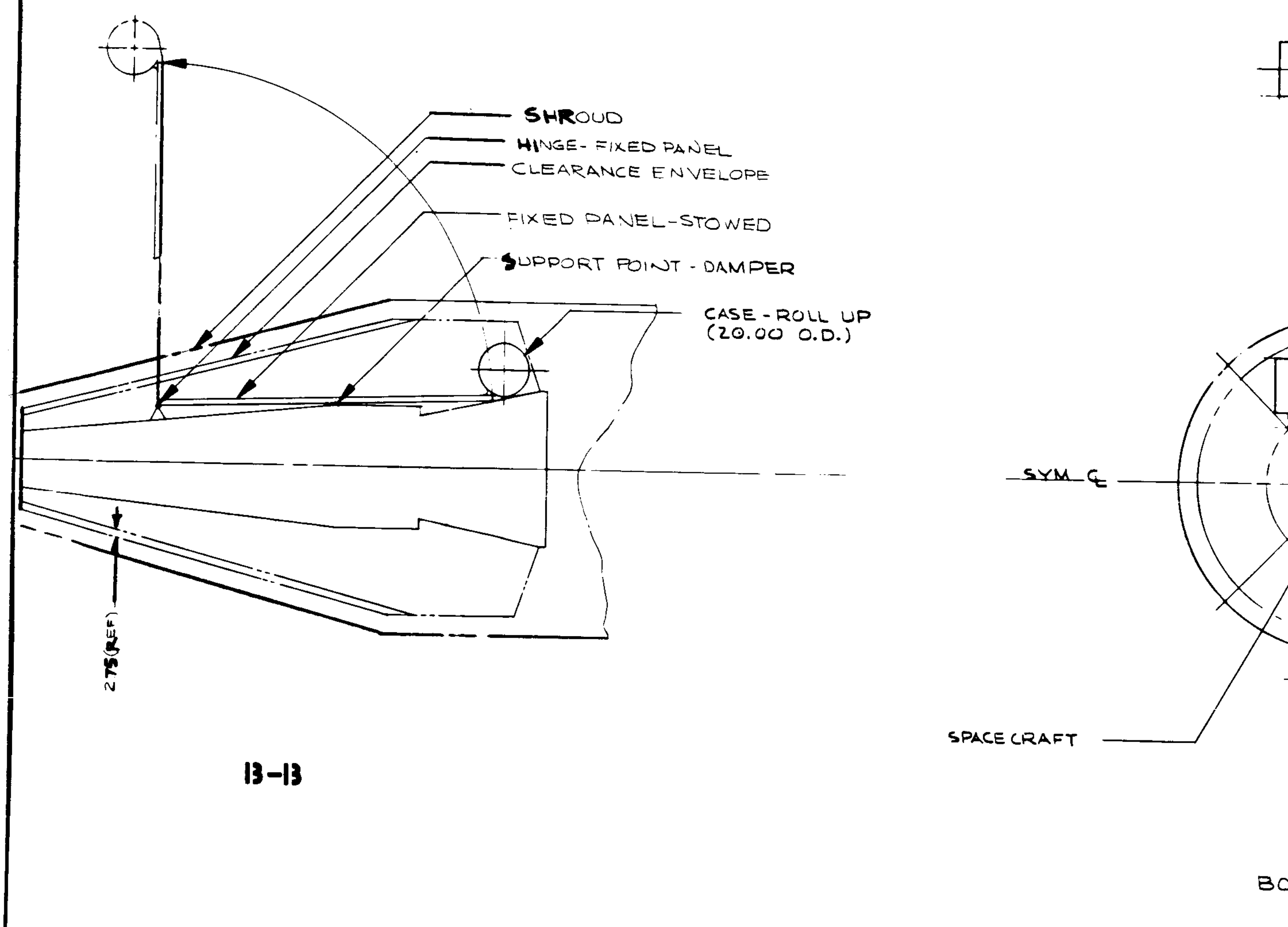

SHROUD
HINGE- FIXED PANEL
CLEARANCE ENVELOPE
FIXED PANEL-STOWED
SUPPORT POINT - DAMPER
CASE - ROLL UP
(20.00 O.D.)
2.75(REF)
B-B
SYM ℄
SPACE CRAFT
BO

Fig 4.2-3 (1)

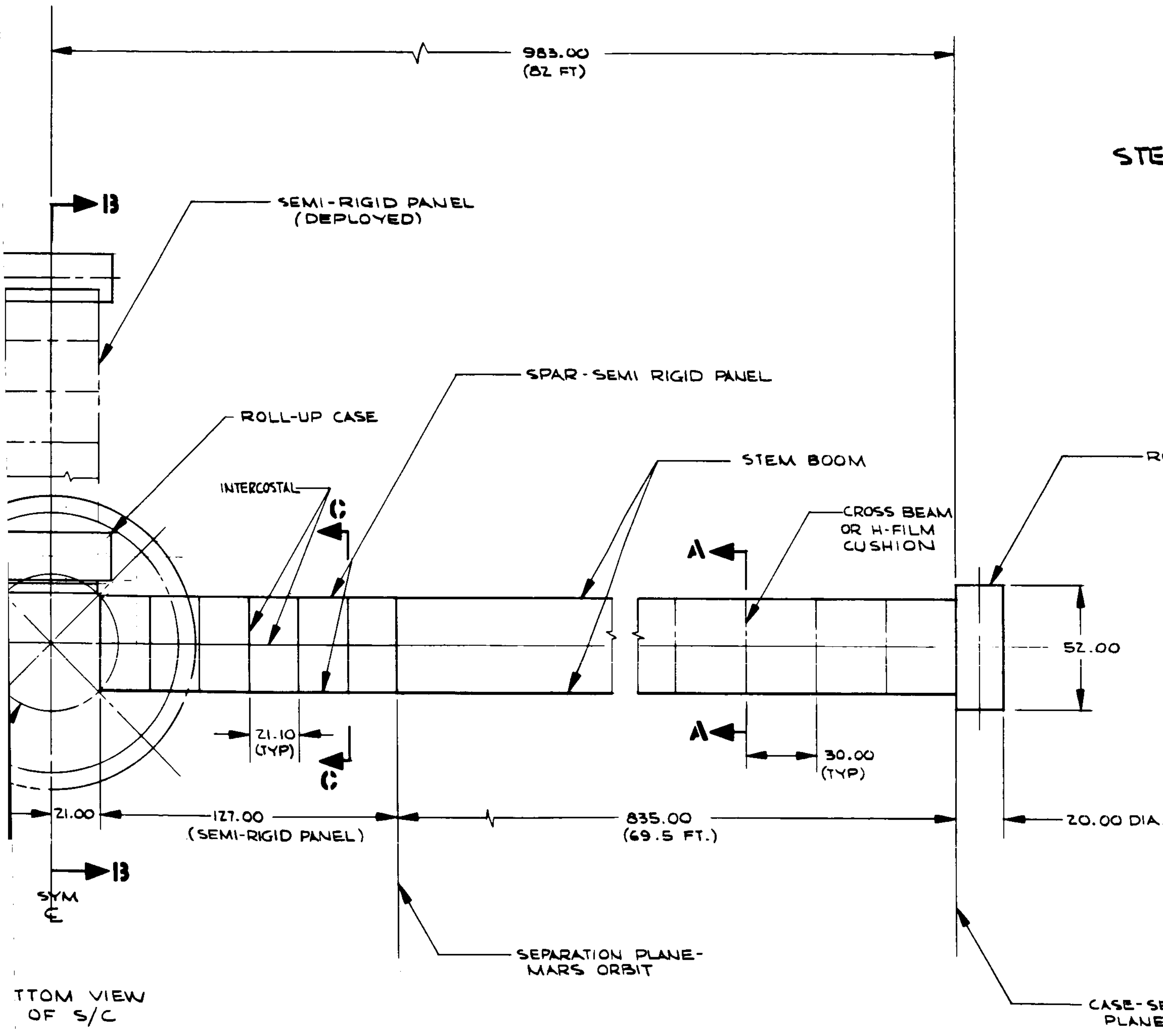
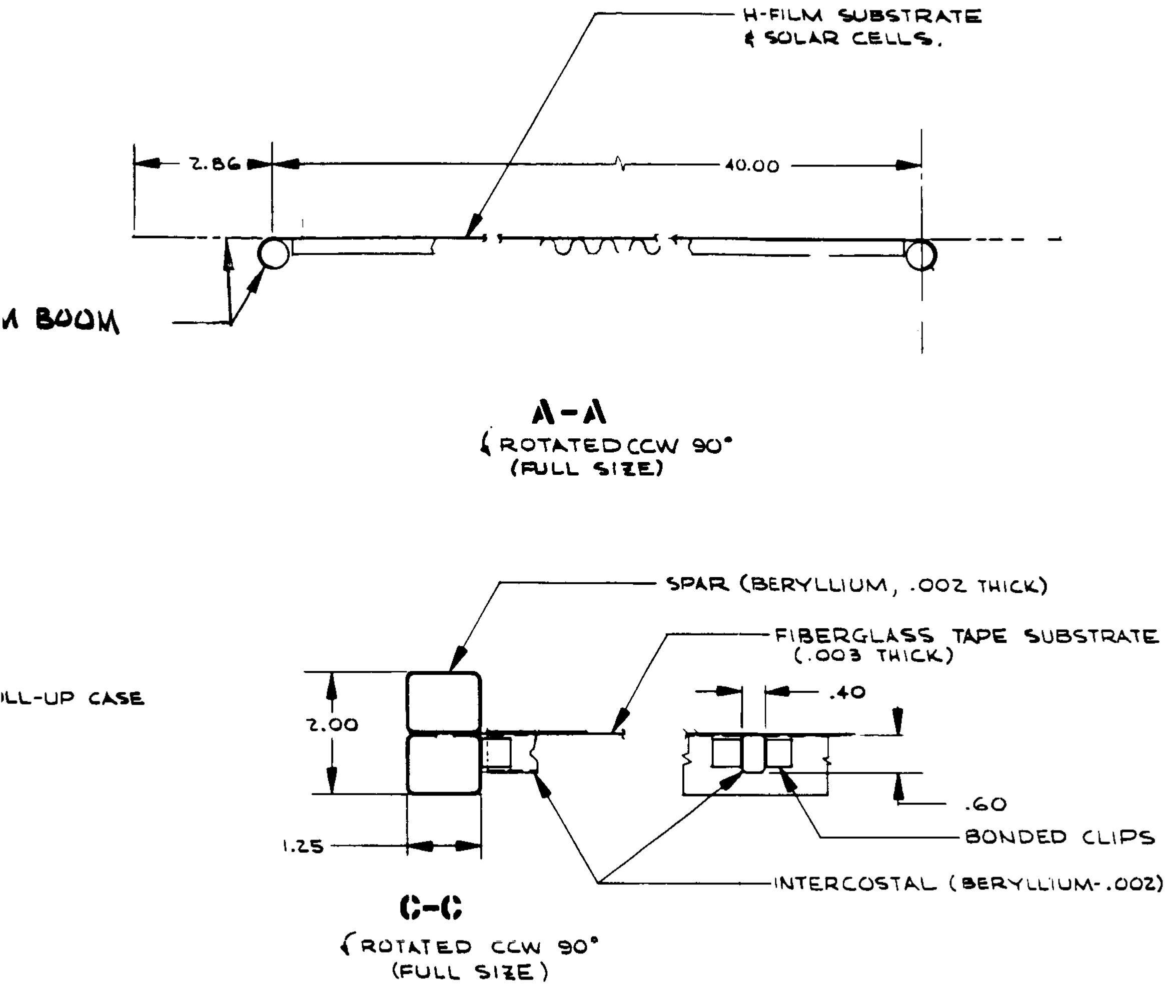

983.00
(82 FT)
STE
SEMI-RIGID PANEL
(DEPLOYED)
B
SPAR-SEMI RIGID PANEL
ROLL-UP CASE
STEM BOOM
INTERCOSTAL
C
CROSS BEAM
OR H-FILM
CUSHION
A
52.00
21.10
(TYP)
A
30.00
(TYP)
C
21.00
127.00
(SEMI-RIGID PANEL)
835.00
(69.5 FT.)
20.00 DIA.
B
SYM
℄
SEPARATION PLANE-
MARS ORBIT
TTOM VIEW
OF S/C
CASE-SE
PLANE

Fig 4.2-3



	AREA
SEMI-RIGID	141 SQ. FT.
ROLL-UP	926 SQ. FT.
GROSS	1067 SQ. FT.

FIGURE 4.2-3 ROLL-UP SOLAR ARRAY
STEM BOOM
ATLAS/CENTAUR

3

4.3 ELECTRICAL BUS TRADES

Conductor thermal studies have been made. They show that heat generated by IR losses results in severe limitation for weight reduction considerations. The increase in radiation area for aluminum favors that material to a small extent; but modest changes in radiating area have only a small effect on the equilibrium temperature. Aluminum was chosen for wire sizes larger than AWG 16 principally because of the favorable resistivity/density ratio. This study indicates that conduction, the one remaining possibility for greater heat rejection should be considered for future design.

4.3.1 Electrical Conductor Analyses

Studies were made to determine the effects of current densities in copper and aluminum conductors. The following assumptions were made to eliminate complications.

(a) Multistranded, twisted wire was considered to be thermally homogeneous
(b) Emissivity of copper and aluminum conductors was assigned a value of 0.4.
(c) Temperature of space T_s was assumed to be $40^{o}R$
(d) Radiation was assumed for only one-half of the wire surface area due to the position of nearby structure.

Conductor current carrying capability versus conductor temperature is shown in Figures 4.3-1 and 4.3-2. The following formula is used to obtain stabilized conductor temperatures.

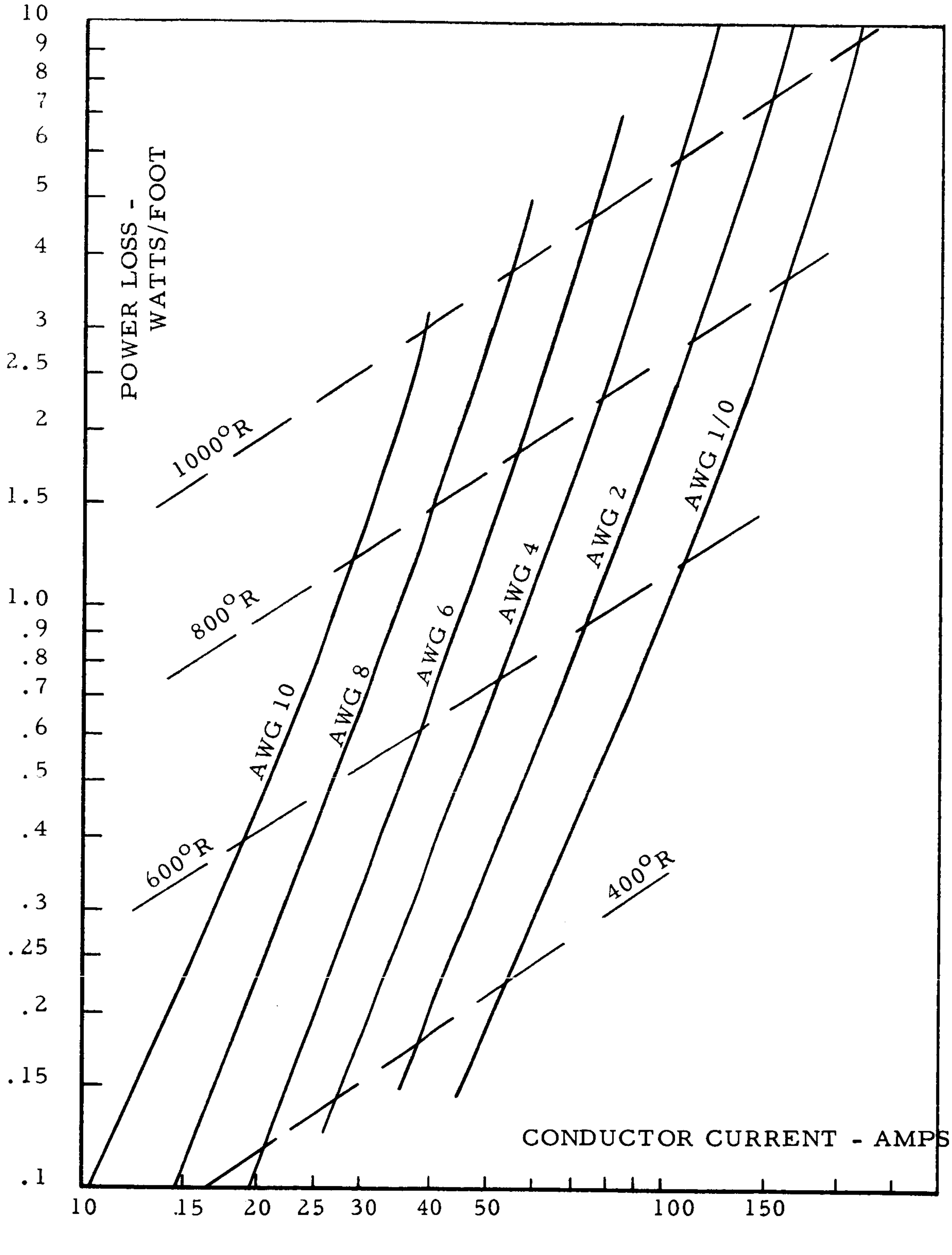
$$T_E = \sqrt[4]{\frac{3412\,P}{\sigma A \varepsilon} + T_S^4}$$

Where:

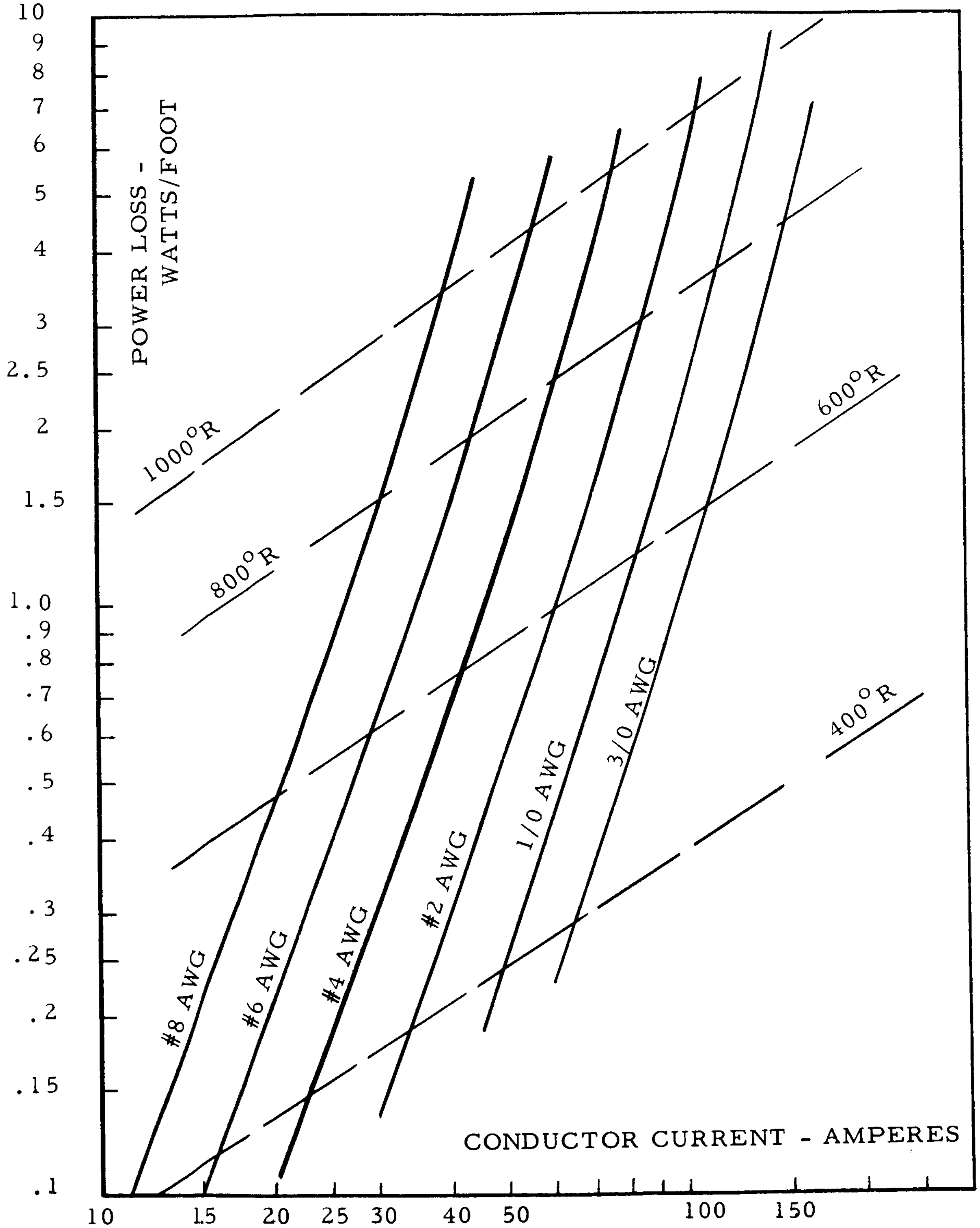
T_E = Equilibrium temperature
T_S = Temperature of space
P = Power loss (KW)
σ = Stephan-Boltzmann Constant
A = Radiating Area
ε = Emissivity

The resistivities for all conductors for various operating temperatures were derived from "Handbook of Thermophysical Properties of Solid Materials", Goldsmith, Waterman and Hirshhorn. Table 4.3-3 shows weights for various sizes of conductors. The weights shown are for the conductor material only and do not include insulation.

Bus operation at 800°F is feasible for either copper or aluminum and that is the approximate upper limit for the molded nylon wire support. A comparison of current carrying capacities shown here and recommended values for commercial wire in free air reveals a substantial derating for space. Air conduction and convection account for approximately 20 to 30 percent of the heat transfer under terrestrial conditions. Possible solutions to a decrease in conductor weight would be the development of conductors with large radiating areas or the use of structure as an electrical conductor.



CONDUCTOR CHARACTERISTICS - COPPER
FIGURE 4.3-1



CONDUCTOR CHARACTERISTICS - ALUMINUM
FIGURE 4.3-2

4.4 MATERIAL EVALUATIONS

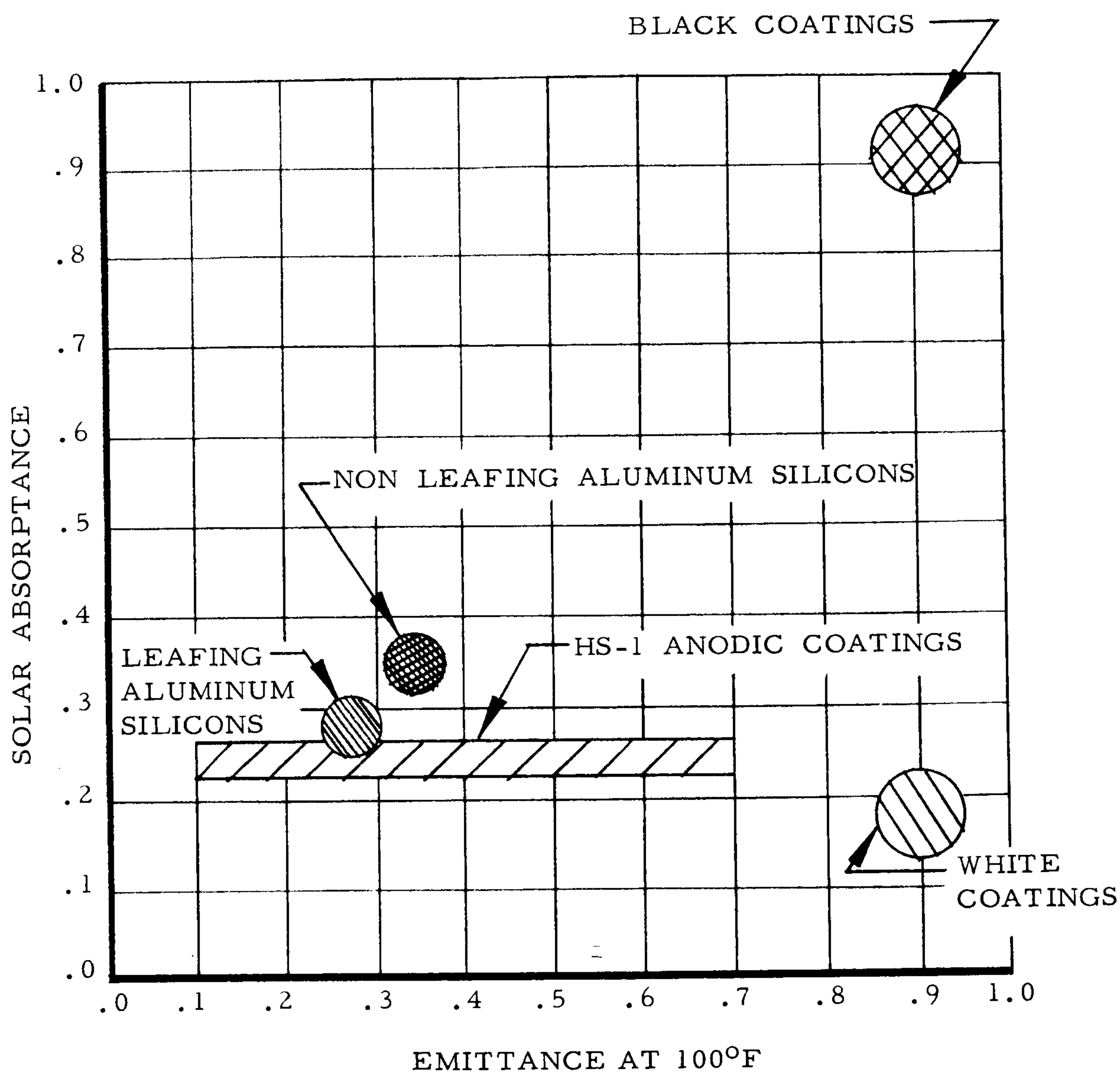
Materials selected for use on the solar array are space qualified and have been used in previous space missions.

4.4.1 The epoxy-polyamide adhesives have been used on the IMP spacecraft (FM-1000) and on the Mariner '64 solar panels. Their performance in space has apparently been satisfactory. The structural loads developed in the array during launch are direct functions of the structural damping provided by the adhesive system used in fabrication of the bonded beryllium frame members. Several polymeric systems were considered. Most exhibited little energy absorbing capability or had not been space qualified. The epoxy-polyamide adhesive "BMS 5-29" exhibits desirable structural damping characteristics and was, therefore, selected as the structural bonding agent.

4.4.2 The materials such as aluminum, beryllium, copper beryllium, silver and titanium have all been successfully used on previous space missions. The polymeric materials used in the mechanism system (Delrin and Dacron) have been selected for their unique properties of strength, and lubrication capabilities.

Thermal control coatings were investigated and the black high emittance coatings were selected. Experimental data for the optical properties of these coatings are shown in Figure 4.4-1. The emittance values obtained from the black coatings indicated a dependence of emittance value upon the coating thickness.

A more complete and specific detailed material evaluation is given in Section 5.4 of bi-monthly report D2-23942-1.



OPTICAL CHARACTERISTICS
OF SELECTED THERMAL
CONTROL COATINGS
FIGURE 4.4-1

4.5 RELIABILITY EVALUATIONS

During the configuration stage of the program, reliability trade studies of the several deployment system concepts were conducted. The following failure data were assembled for the various components used in the deployment mechanisms.

Component	Failures Per Cycle	Source of Data
Squib Actuator	.0001	Conax; Holex; McCormick-Selph; HiShear
Latch (Locking)	.000006	FARADA
Mechanical Release	.00001	FARADA
Motor Drive	.00004	Lundy Electronic
Viscous Damper	.000004	FARADA
Spring (Torsion) (2 Redundant)	Negligible	FARADA
Bourdon Helix	.0000045	Glassco
Hinge	.00001	FARADA
Switch	.000007	FARADA
Orifice	.0000013	FARADA
Gas Supply	.000007	AVCO Reliability Data Series
Flexible Array Roll-Out	Not Available	
Extendible Boom (Rollup)	Not Available	
Extendible Boom (Telescoping)	Not Available	

Except in the case of the squib actuator, the failure rates for the items in the above table include wearout and fatigue failures. The use of these rates has a conservative effect on the predicted deployment mechanism reliability because wearout and fatigue are not expected modes of failure.

Thus far, it appears the squib device is the controlling item in the magnitude of the reliability predictions.

Table 4.5-1 shows the reliability estimates used to support trade studies among the candidate deployment mechanisms for the folding modular array configuration.

Reliability estimates were made for the Roll-Out Array deployment mechanisms under consideration. The following preliminary ranking was established.

Configuration	Rank	Remarks
DeHavilland Boom	1	About 100 used in space.
Pneumatic Telescoping Boom	2	Feasible. Subject to gas leakage.
Closed Tube Boom	3	Unproven.

DEPLOYMENT METHOD	COMPONENT	QUANTITY	FAILURES/CYCLE
Motor-Cable Drive	Squib	8	.0008
	Latch	64	.00038
	Mech. Release	36	.00036
	Motor Drive	4	.00016
	Dashpot	40	.00016
	Spring	40	
	Hinge	64	.00064
	Switch	4	.000028
			.002534
			R = .9975
Bourdon Tube	Squib	12	.0012
	Latch	64	.00038
	Mech. Release	36	.00036
	Dashpot	40	.00016
	Bourdon Tube	24	.00011
	Spring	40	
	Hinge	64	.00064
	Switch	4	.000028
	Orifice	24	.00003
	Gas Supply	4	.000028
			.002936
			R = .9971
Spring and Damper	Squib	8	.0008
	Latch	64	.00038
	Mech. Release	36	.00036
	Dashpot	64	.00026
	Spring	64	
	Hinge	64	.00064
	Switch	4	.000028
			.002468
			R = .9975

RELIABILITY ESTIMATES

TABLE 4.5-1

4.6 MANUFACTURING FEASIBILITY EVALUATION

The solar array and solar panel configurations were evaluated by considering the state-of-the-art and manufacturing capabilities of the various technologies required to fabricate the arrays. The evaluations are based upon the engineering information developed during the trade studies and does not include tolerances or process limits.

A state-of-the-art rating code was applied to the evaluations. This code considers technical capability only and does not include manufacturing readiness. It is assumed that development programs would start no later than January 1, 1966. The following rating code and definitions are used:

A = Presently State-of-the-Art. Development not required.
B = Predicted to be State-of-the-Art by May 1, 1966.
C = Predicted to be State-of-the-Art by February 1, 1967.
D = Possible State-of-the-Art after February 1, 1967.

4.6.1 Beryllium Structure Fabrication

The approach that was used to design the beryllium structural members is compatible with existing fabricating techniques. The state-of-the-art for beryllium fabrication has not progressed to the point where development is not required. The size of the parts (length of spars, etc.) will present tooling and handling problems. State-of-the-art rating Code B is assigned.

4.6.2 Collapsible Tube Fabrication (Closed Tube Boom)

Three areas of technology are involved - forming, assembly, and heat treating. The techniques to be applied are not new, but the length of the parts to be fabricated will require considerable development of tools and methods. In addition to the actual fabrication, testing and handling problems will have to be resolved before production readiness is attained. State-of-the-art rating Code D is assigned.

4.6.3 H-Film Bonding

The application of using H-film as a substrate for solar cell installations would involve bonding to structure and to solar cells. To date there has been little experience with this material. All of the information that is available would indicate that major development would not be required. The size and the hat section stiffener configuration would require tool development before production readiness is attained. State-of-the-art rating Code C is assigned.

4.6.4 Fiberglass Bonding

The use of fiberglass tape as a substrate for solar cell installation would involve existing techniques for bonding. It is not anticipated that additional technique development would be required. This is particularly true if BMS 5-29 adhesive is specified. The main concerns will be the tooling to accommodate size and pretensioning, and the availability of 3 mil thick tape. State-of-the-art rating Code B is assigned.

4.6.5 State-of-the-Art Ratings for Array Configurations

A summary of state-of-the-art ratings for specified configurations is presented in Figure 4.6-1. The ratings were determined from the manufacturing feasibility evaluations of the technologies required for fabrication. The limiting factor which determined the rating is listed for each rating except "A", Present State-of-the-Art.

ARRAY	SUBSYSTEM	TYPE	STATE-OF-THE-ART RATING	LIMITING PARAMETER
Folding Modular	Panel	Concentrator	A	
		Fiberglass Tape Semi-rigid	B	Pretension tape and solar cell installation
	Support Structure	Beryllium Spar	B	Beryllium fabrication
	Actuating Mechanism	Motor Drive	A	
		Bourdon Tube	B	Tube Fabrication
		Spring & Damper	B	Damper Fabrication
	Total Array	Concentrator	A	
		Semi-Rigid	B	
Rollup	Panel	Stiffened Panel	C	Bonding to H-Film
		Unstiffened Panel	C	Bonding to H-Film
	Support Structure	STEM Boom	A	
		Closed Tube Boom	D	Assembly
		Telescoping Boom	A	
	Actuating Mechanism	STEM Boom	A	
		Closed Tube Boom	D	Assembly
		Telescoping Boom	A	
	Total Array	STEM Boom	C	H-Film Bonding
		Closed Tube Boom	D	Closed Tube
		Telescoping Boom	C	H-Film Bonding

ARRAY CONFIGURATION TRADES SUMMARY

TABLE 4.6-1

5.0 PRELIMINARY DESIGN SYSTEM DEFINITION

During the first four months of the program the design objectives to be met by the array were established. In addition, analyses and key trade studies were conducted. This effort resulted in the following baseline configuration and subsystem definitions for the preliminary designs shown in Section 6 of this bi-monthly report.

5.1 BASELINE CONFIGURATION

Based upon the established criteria and requirements and the analysis conducted, the following baseline configuration, shown in Figure 5.1-1, has been established.

5.1.1 Four folding modular panels will be symmetrically mounted to the spacecraft. Each panel consists of a sub-panel number 1 and sub-panel assemblies 2 through 5. Each sub-panel assembly consists of a main sub-panel and two auxiliary sub-panels.

5.1.2 The semi-rigid sub-panel shall consist of a fiberglass substrate with beryllium structural support.

5.1.3 Pulley-cable systems will be used for main panel deployment and spring and damper actuation for auxiliary panel deployment.

5.1.4 The major portion of the arrays will be jettisoned prior to injection of the spacecraft into Mars orbit. The jettison system shall be actuated by sealed pyrotechnique devices. Separation shall be completed using spring devices.

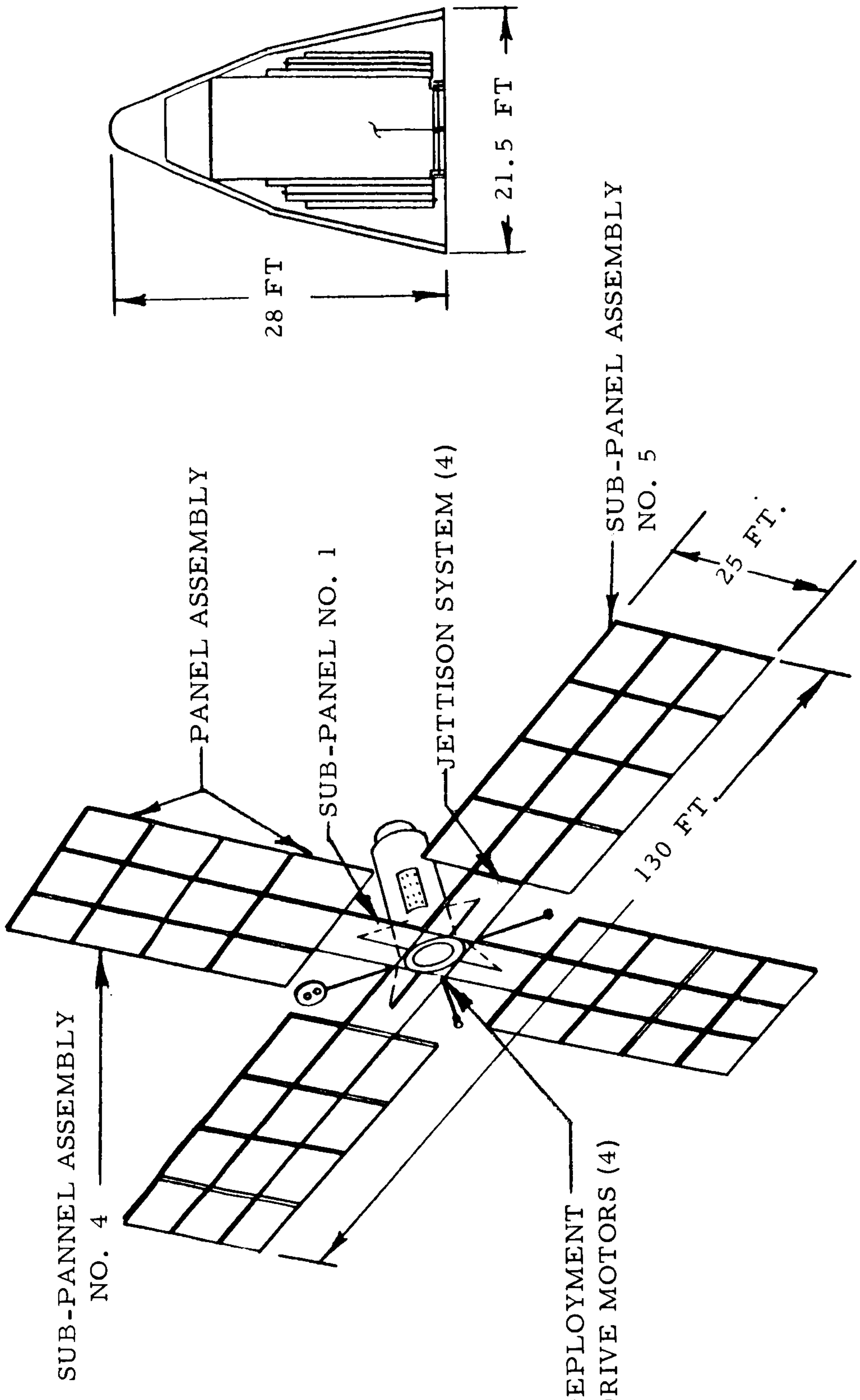
5.1.5 The cell modules shall include use of 2 x 2 cm, 8 mil, back-connected, N on P silicon photovoltaic cells and 4 mil microsheet cover glasses.

5.2 MAJOR SUBSYSTEMS

The solar array power system will consist of the following major subsystems.

5.2.1 The electrical power subsystem shall include the necessary solar cells, cover glasses, interconnectors and electrical busses to generate the required power and carry the current to the interface connectors at the spacecraft-array joint. Power conditioning is not included in this system. The bussing system shall be capable of bridging the several hinge points and shall be compatible with the array jettison system. Major design studies include:

(a) Selection of optimum interconnector material.



BASELINE CONFIGURATION
50 KW - 20 WATT/POUND SOLAR
CELL ARRAY FOR AN ELEC-
TRICALLY PROPELLED MARS
ORBITER

FIGURE 5.1-1

(b) Optimum number of cells in the series-parallel arrangement to provide the required power and redundancy to meet reliability goals.
(c) Installation procedures and processes for module and bus elements.

5.2.2 The structural portion of the array shall consist of the sub-panel substrate, substrate support members, panel support spars and panel attachments to the spacecraft. Necessary elements are provided to carry the static and dynamic loads imposed by the launch vehicles and spacecraft propulsion and attitude control subsystems.

Major design studies include:

(a) Comparison of woven fiberglass tape substrate with one of perforated fiberglass sheet.
(b) Determination of structural and substrate support spacing and configuration.
(c) Spar configuration.
(d) Optimization of sub-panel shapes and sizes from a manufacturing standpoint within the constraint of required panel solar cell area.
(e) Shaping of the panel configurations to provide compatibility with spacecraft subsystems in the deployed mode.
(f) A dynamic analysis to define the array characteristics.
(g) A thermal analysis to define the array characteristics and check gradient and other thermal problems.
(h) Loads analysis to determine array-spacecraft tie-down and need for boost damper.

5.2.3 The array deployment subsystem consists of release mechanisms, actuators, hinges, latches, and dampers necessary to effect synchronous extension of the panels to a fixed position. Extension will be accomplished with a minimum disturbance to the spacecraft. The system will include the wiring for sequencing of the various releases and actuators. Electrical power for the release mechanisms and actuators will be provided by spacecraft secondary power supply.

Major design studies include:

(a) Development of a deployment sequence.
(b) Selection of optimum electric motor-gear drive.
(c) Assembly of hinges and latches for proper fit up.

5.2.4 The jettison subsystem separates the major portion of the array from the spacecraft prior to injection of the vehicle into a Mars orbit. The system includes pyrotechnique devices, springs, and the necessary wiring from the pyrotechnics to the spacecraft command and control console and secondary power system. Suitable devices are provided to assure proper separation of the electrical busses on the array.

5.2.5 Sensors are mounted on the array to make the following engineering measurements during the mission.

5.2.5.1 Array Deployment
5.2.5.2 Panel Temperatures
5.2.5.3 Jettison Completion

5.2.6 A ground support equipment subsystem will be defined through the use of functional analyses.

5.3 MAJOR ARRAY-SPACECRAFT INTERFACES

The following major interfaces are defined in the Preliminary Design.

5.3.1 Forward and aft array-spacecraft structural and mechanical attachments.

5.3.2 Array-spacecraft spatial relationships for the array stowed and deployed.

5.3.3 Array electric motor-drive system envelope requirements in the spacecraft.

5.3.4 Array deployment cable routing and envelope requirements.

5.3.5 Array electric bus interface with spacecraft bus.

5.3.6 Interfaces for the signal and power cables required on the array to initiate release and deployment mechanisms.

5.3.7 Array sensor wire interfaces.

6.0 PRELIMINARY DESIGN AND ANALYSIS SUMMARY

This section summarizes the preliminary designs prepared during the program. The major elements of a 10 and 50 kilowatt folding modular solar photovoltaic array were defined to a considerable level of detail. These details have been documented in bi-monthly reports D2-23942-3 and D2-23942-4.

Power output and weight evaluations of the two array designs developed demonstrate that 20 watts per pound at 1 Astronomical Unit is a feasible goal for the mission investigated. The ratios calculated were 24.35 watts per pound for the Saturn IB/Centaur launched array, and 19.64 watts per pound for the Atlas/Centaur launched array. Redundant elements to meet the reliability goals established were included in the designs and weight analyses. Although the designs are of a preliminary nature, sufficient details were developed to verify the results. Our analyses demonstrate that the basic concept has considerable flexibility. Dynamic response of the array can be varied by simple changes to the panel assembly support points. Excessive stress concentrations can be eliminated by tailoring the constant section frame structure to the critical loads without large weight penalties. This can be achieved by varying either the material gages or tapering the frame dimensions. Although 8 mil solar cells and 4 mil cover glasses were used as the basis of the module designs, 12 mil cells and/or 2 mil cover glasses could be utilized. Deployment of the main sub-panels and auxiliary sub-panels are to some extent conducted independently. That is a failure of one or more auxiliary sub-panels to deploy will not prevent deployment of the remainder of the panel assembly.

Significant interface coordination with Jet Propulsion Laboratory and the spacecraft and mission contractors was conducted throughout the program. Subsystem envelope, and dynamic and electrical interfacing functions that were defined are summarized in the appropriate paragraphs of this section.

6.1 SATURN IB/CENTAUR ARRAY

6.1.1 Launch Configuration (Figure 6.1-1)

The folding modular array is stowed symmetrically (4 places) about the spacecraft centerline and is enclosed by the shroud of the Saturn IB/Centaur vehicle. All points on the solar array clear the spacecraft (shroud) dynamic envelope by 2 inches. Each stowed solar panel assembly is supported at STA 8.70 by the spacecraft and they are attached to each other at STA 114.00. At STA 10.0 and 114.00, tension tie-rods are used to clamp and pre-stress the panel spars so that the total stowed assembly acts as one unit under load.

All spars are separated by a .25 gap prior to pretensioning the tie-rods. Silicon rubber spacer pads (5 places) and aluminum pads (2 places) (see Section C-C) are bonded to the spars. The silicon rubber pads act only as elastic supports on the spars and the aluminum pads act as compression and shear load transfer members. The tension tie-rods carry all of the tension loads. The aluminum pads are either "ball-cone" or "ball-slot" combinations. Shear in the Z-axis direction is carried by a "ball-cone" at STA 10 and a "ball-slot" at STA 114. Putting the shear capability between spars at only one point allows the spars to slide longitudinally with respect to each other (the ball-slots will be dry-lubricated) under a bending load. As a result the resonant frequency of the stowed panel assembly will be approximately 10 cps.

Pad thicknesses are varied so that pretensioning the STA 114 tie-rods prestresses the spars in bending, and applies a compression load at all of the pad locations. The pre-stressed spars force the stowed panel assembly to act as a unit. Each of the 4 stowed panel assemblies are attached to the spacecraft at the 3 main hinges at STA 8.70 and the diagonal tension tie at STA 58. The outboard main hinges of each panel assembly have shear capability in the Z-axis and in a plane normal to the panels. The center main hinge has shear capability in all 3 axes.

6.1.2 Deployed Configuration (Figures 6.1-2 and 6.1-2A)

The array is deployed to a gross area of 4,944 square feet. It is deployed and locked in a rigid configuration capable of resisting cruise mode load conditions. Just prior to the terminal maneuver, all the sub-panels that provide ion engine power outboard of sub-panel #1 are jettisoned. This leaves the four 28 volt power generation sub-panels attached to the spacecraft. These sub-panels and attachments will resist Mars orbit injection maneuver loads.

The solar array has four panel assemblies and each assembly is made up of sub-panel #1, and sub-panel units #2 through #5. A sub-panel unit has two auxiliary panels and a main panel. Sub-panel #1 does not have auxiliary panels in order to provide area for deployment of subsystem components.

Each sub-panel has 3 longitudinal spars, lateral and longitudinal intercostals, edge members, and solar cells bonded on a fiberglass substrate. The edge spars of the main panels are locked at the hinge centerlines for the total span of the array. This provides the main structural path for the deployed configuration. The structural members are formed from beryllium sheet and bonded with BMS 5-29 adhesive. Bonding of the structure improves the dynamic load capability by lowering the dynamic magnification factor.

The fiberglass tape substrate (3 mil thick x .20 wide at .80 spacing) is a unidirectional tape and is procured in an uncured state. It is formed into a mesh pattern and cured under pressure to provide a homogeneous fiberglass matrix. Bi-directional fiberglass sheet was considered as a substrate, but was rejected when it was determined that the straightness of the filaments could not be assured. If these filaments were not straight, they would be cut when the "waffle pattern" holes were cut out of the fabric. This reduces the structural integrity of the substrate. Selection of uni-directional tape thus results in a more reliable structural substrate. Once the substrate is formed, it is pre-stressed to 10 pounds per inch and bonded to the panel support structure. With the panel module sizes used on this design the dynamic amplitudes of the substrate will be less than ± 0.40 inches. The 0.75 inch half spar depth prevents contact of adjacent cell stacks in the stowed condition. For thermal control, Laminar X-500 is applied to the dark side of the substrate, solar cells, and structure.

Originally, the deployed solar panel assemblies were supported by a diagonal tension tie for cruise or retro load in the +Z direction and a hinge lock at the spacecraft hinge centerline reacted the loads in the -Z direction. However, the diagonal tension tie requirement was changed to a diagonal compression member because of an expected load increase in the -Z direction.

The 3 main hinges, per solar panel assembly, are the primary structural attachment points to the spacecraft. These fittings are made from 6-6-2 Titanium Alloy to achieve the highest strength to weight ratio.

6.1.3 Saturn IB/Centaur Mechanisms

The cable drum and pulley system shown in Figure 6.1-3, sheets 1 thru 3, was chosen as the deployment method for the main panels. The preloaded torsion spring deployment with overcenter linkage lock shown in Figure 6.1-4 was chosen for deploying the auxiliary panels. This concept demonstrates lighter weight and reliability equal to the bourdon tube or spring damper systems and is composed of essentially state-of-the-art components. The deployment sequence is summarized in Figure 6.1-5 and is outlined in detail in paragraph 4.2.1 of Document D2-23942-3, Bi-Monthly Report Number 3.

As a result of this study, it has been concluded that panels of the size discussed can be successfully deployed in space and that mechanism weights do not increase proportionally to an increase in deployed area. More specifically, mechanism weight performance is improved with increasing areas to be deployed.

Deployment of the panels could have been speeded up, but at the cost of a greater power requirement, active in lieu of passive damping and the risk of introducing perturbating forces on the spacecraft. Even at the present slow rate, certain panels have a tip speed of 3 inches per second.

Deployment of the panels could have been slowed down further, but even in the designed system, hardware sizes are at a minimum for standard components. A further step to micro-mineature equipment would not have met the program state-of-the-art goals.

6.1.3.1 Separation for Mars Orbit

Prior to Mars orbit injection, all but 1.1 kilowatt must jettison. The ordnance device shown in Figure 6.1-3, Sheet 3, cuts the deployment cable, the monitor signal wires and the ordnance signal wires. The charge also withdraws a torque tube locking pin. This allows a double cam at each spar to release a structural clamp within the spar. Upon structural release, two preloaded compression springs extend to separate the main power bus connectors, and to eject the discarded solar panels from the spacecraft. Although the impulse force is not great, the separation direction is at 90° to the spacecraft centerline; hence, the panels move outward from the spacecraft.

6.1.3.2 Hinges

Sub-panel #1 hinges are the only ones required to withstand launch boost and retro loads. All other hinges are required to withstand deployment torques and attitude control moments. Panel hinges feature self-centerline bearings using a fabroid material on the sliding surfaces which prevents vacuum welding and provides lubricity. Deployment centrifugal forces and hinge friction are negligible for the angular velocities encountered. A resistance spring is provided to prevent excessive deployment velocities from any external forces.

The sub-panel #1 to the spacecraft center hinge incorporates a quadrant with which deployment torque is applied to the hinge. A negator spring is used as a cable guard to prevent the cable from being displaced for any reason. The auxiliary panel hinges use a pin bushing of fabroid plastic material to prevent vacuum welding and to reduce friction.

6.1.3.3 Latches

The main latches of sub-panel units #2, #3, #4, and #5 are bayonet type. Two latching dogs with locking pins are spring driven into a slotted striker plate. The latch is mounted on the spar opposite the hinge fittings at each main joint. One latch is provided at each outboard spar. Latching action begins as the bayonet enters the slotted plate and the protective sheath is driven back to expose the latching dogs. A piston is spring driven against the latching dog driving and locking pins. The pivot connection of the latching dogs force them outward and against the slotted plate. As the piston approaches the end of its travel, the driving and locking pins enter the slot of the piston and provide a positive lock (see Section E-E, Figure 6.1-3, Sheet 1).

A shear pin on the piston of the latch for joints 3 and 5 is retracted into the bayonet housing as the latching dogs are locked. As the pin retracts, one or more retainer cables are released allowing the next sub-panel unit and an adjacent auxiliary panel to deploy. These retainer cables are required only during the deployment sequence and are not loaded during the boost phase.

The overcenter link latches of the #1 joint and all auxiliary panels are of simple pin joint design. Drilling of one link pivot hole at final assembly assures perfect angular control of the panel during deployment. Springs provide the overcenter driving energy as well as holding force. Split washers and Belleville springs lock the pivot joints (see Figure 6.1-4).

6.1.3.4 Sheaves

Sheaves are of Delrin material which provides nearly friction free rotation, possesses good wearing characteristics, and is suitable for use in a space environment. Cable guard pins are of aluminum and are located in accord with good aircraft design techniques.

6.1.3.5 Motor, Reducer, and Cable Drum (Figure 6.1-6)

This system is a compact unit housed within the physical envelope of the cable drum. The motor used is a size 15 Kearfott synchronous motor T-170-001 with a speed of 8000 rpm. The cable drum speed is .375 rpm resulding in a cable speed of 3.0 inches per minute. Speed reduction is accomplished in a two stage harmonic drive system with an overall drive ratio of 21,333:1. The first stage of the harmonic drive will be 120:1 ratio and the second stage 178:1 ratio.

Use of the harmonic drive allows the motor and first stage reduction to be hermetically sealed and allows the second stage to be sealed by a rotating seal. The low speed and sealed compartment features will reduce lubrication problems.

Four drive units weighing 2.5 pounds each are used. Each unit is the primary drive for one panel set and the redundant drive for the panel set opposite. Power for the drives is provided from spacecraft systems. Mounting space and cable clearances are also provided within the spacecraft.

6.1.3.6 Deployment Cable

The cable is 1/8 inch diameter Dacron with a breaking strength of 240 pounds. The synchronous motor deploys the panels at a constant angular velocity. The cable will stretch up to 7 percent at each joint against the increasing load of the hinge resistance spring and latch sheath. This feature negates the need for main panel dampers.

6.1.3.7 Diagonal Brace

The diagonal brace assembly consists of two links with a detent latch. It provides a resistance load on the deployment cable and a tension/compression structural tie between the panels and the spacecraft for Mars orbit injection loads. A tension tie only is shown on Figure 6.1-3, Sheet 2. However, the concept has been changed to include compression capability.

6.1.3.8 Panel Hold Down Release

The panel hold down system for the boost phase is fastened on the edge of the first panel by a dual shear pin system. This is extracted by an ordnance device when deployment is initiated. The system is shown in Detail I, Figure 6.1-1, and in Detail III and Section X-X of Figure 6.1-3, Sheet 3. Reliability will be maintained at a high level by redundant firing circuitry, dual bridge wires in the squibs, and dual squibs in each pin puller assembly. Each squib can be fired by one bridge wire and is adequate to withdraw both shear pins in one corner.

6.1.3.9 Panel Release at Mars Orbit

A single ordnance device is provided at the #2 joint of each solar panel. This actuates mechanical release of the panel structure and severing of the deployment cable with two guillotine cutters. The system is shown in Section T-T of Figure 6.1-3, Sheet 3, and is made up of the two cutters and a pin puller. The system features dual squibs, dual bridge wires and dual power circuitry. Either bridge wire will fire the squib and either squib will complete the separation by actuating all functions.

6.1.3.10 Spring Design

Five types of springs are used in this preliminary design.

(a) Compression springs for Mars orbit panel separation, electrical disconnect and clamp jaw opening, main panel latches, and damper springs.
(b) Torsion springs for main hinge resistance force, auxiliary panel drive and latch, panel separation torque tube drive, and panel release springs.
(c) Negator spring as cable guard for hinge joint number 1 control quadrant.
(d) Disc (Belleville) springs for the auxiliary panel latch locks.
(e) Extension spring for joint number 1 sub-panel release and Mars entry diagonal brace.

To avoid use of magnetic materials, only beryllium copper has been considered for springs.

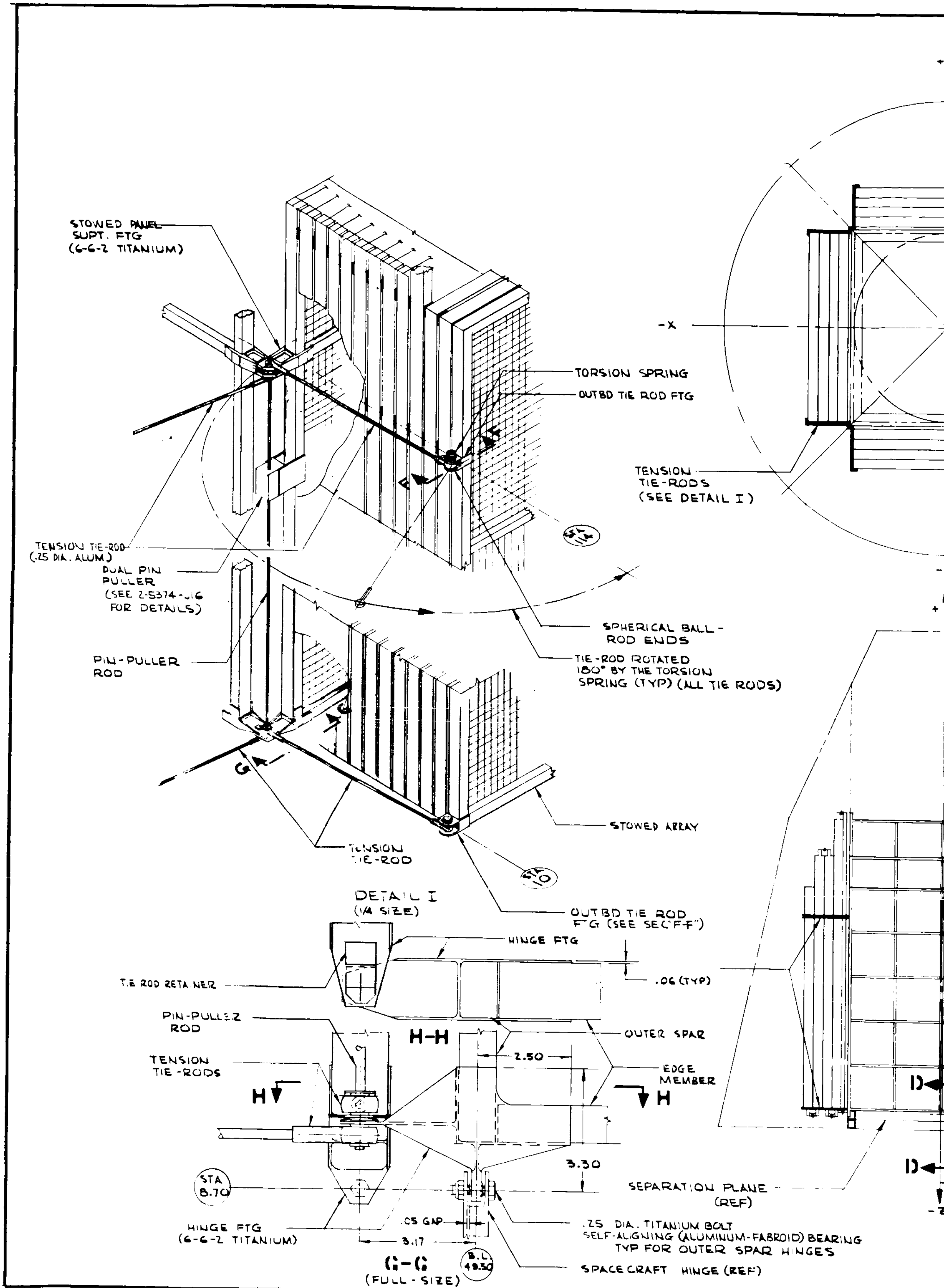

STOWED PANEL SUPT. FTG (6-6-2 TITANIUM)
TORSION SPRING
OUTBD TIE ROD FTG
-X
TENSION TIE-RODS (SEE DETAIL I)
TENSION TIE-ROD (.25 DIA. ALUM.)
DUAL PIN PULLER (SEE 2-5374-J16 FOR DETAILS)
PIN-PULLER ROD
SPHERICAL BALL-ROD ENDS
TIE-ROD ROTATED 180° BY THE TORSION SPRING (TYP) (ALL TIE RODS)
STOWED ARRAY
TENSION TIE-ROD
OUTBD TIE ROD FTG (SEE SEC "F-F")
DETAIL I (1/4 SIZE)
HINGE FTG
TIE ROD RETAINER
.06 (TYP)
PIN-PULLER ROD
H-H
OUTER SPAR
TENSION TIE-RODS
2.50
EDGE MEMBER
3.30
STA 8.70
SEPARATION PLANE (REF)
HINGE FTG (6-6-2 TITANIUM)
.05 GAP
3.17
.25 DIA. TITANIUM BOLT
SELF-ALIGNING (ALUMINUM-FABROID) BEARING
TYP FOR OUTER SPAR HINGES
G-G (FULL-SIZE)
B.L 49.50
SPACECRAFT HINGE (REF)

Fig 6.1-1

Ⓕ

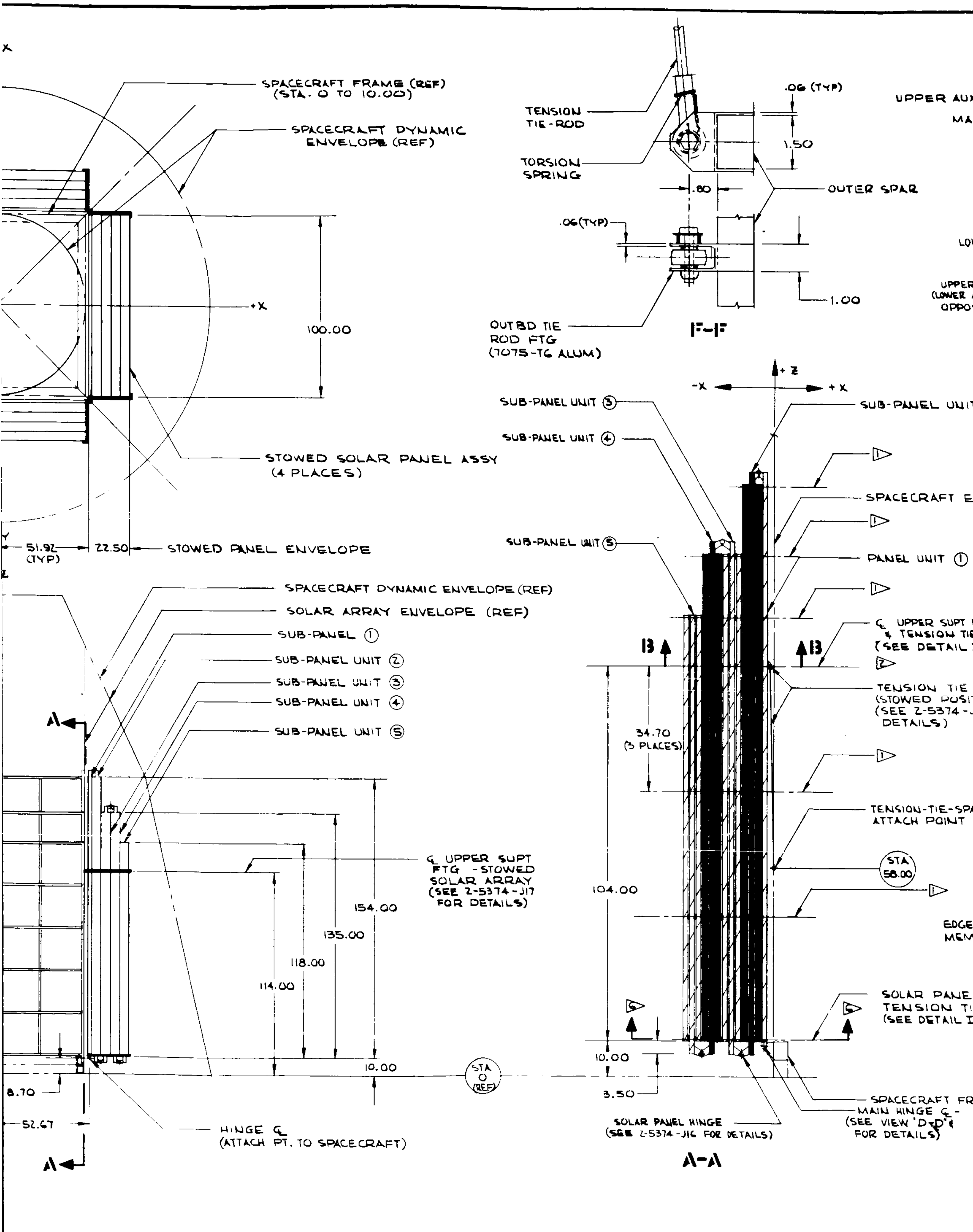
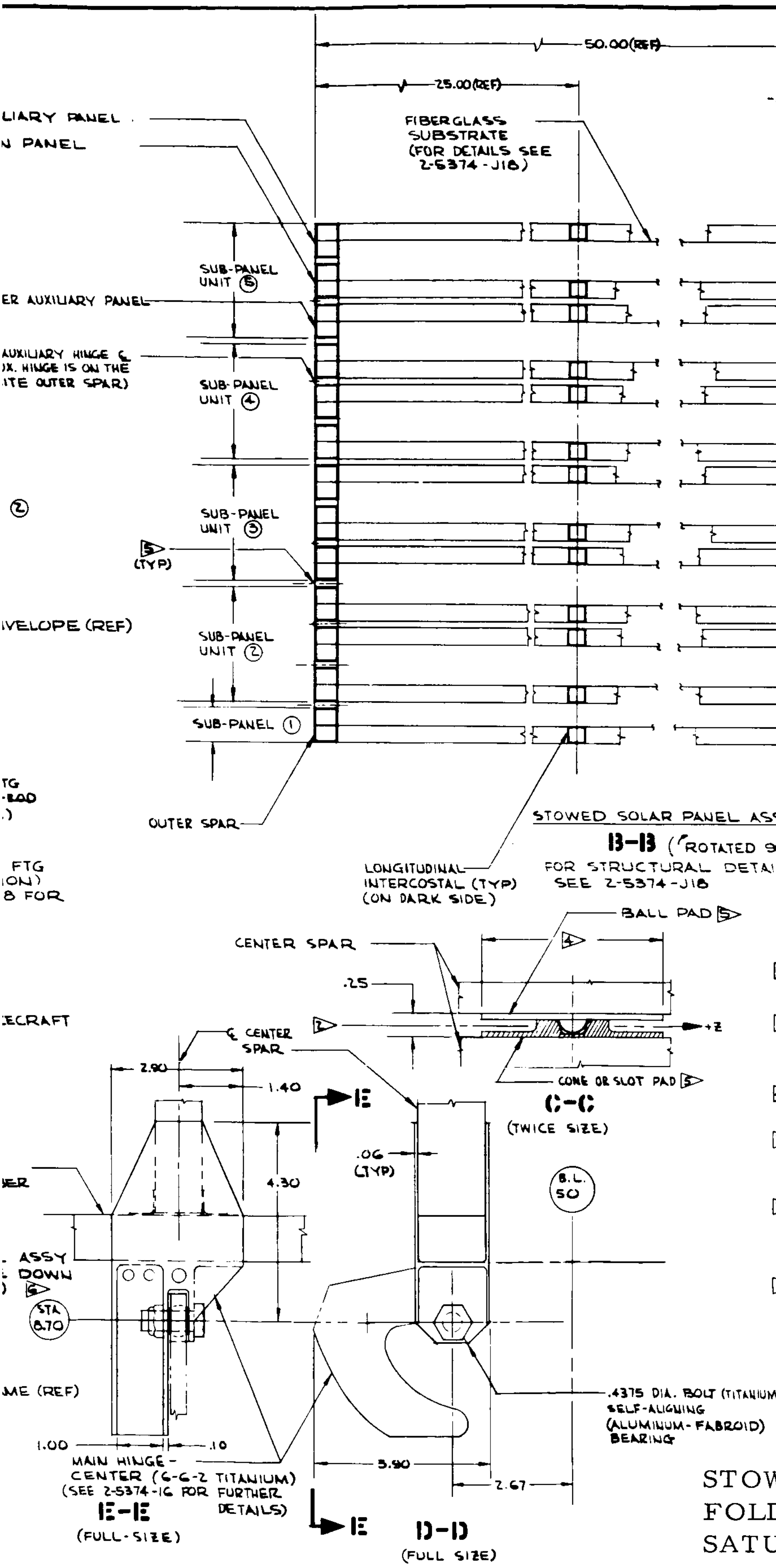

SPACECRAFT FRAME (REF)
(STA. 0 TO 10.00)
SPACECRAFT DYNAMIC ENVELOPE (REF)
+X
100.00
STOWED SOLAR PANEL ASSY
(4 PLACES)
51.92
(TYP)
22.50
STOWED PANEL ENVELOPE
SPACECRAFT DYNAMIC ENVELOPE (REF)
SOLAR ARRAY ENVELOPE (REF)
SUB-PANEL ①
SUB-PANEL UNIT ②
SUB-PANEL UNIT ③
SUB-PANEL UNIT ④
SUB-PANEL UNIT ⑤
℄ UPPER SUPT FTG - STOWED SOLAR ARRAY (SEE 2-5374-J17 FOR DETAILS)
154.00
135.00
118.00
114.00
10.00
STA 0 (REF)
8.70
52.67
HINGE ℄
(ATTACH PT. TO SPACECRAFT)
TENSION TIE-ROD
TORSION SPRING
.06 (TYP)
1.50
.80
OUTER SPAR
.06(TYP)
1.00
OUTBD TIE ROD FTG
(7075-T6 ALUM)
F-F
UPPER AUX
MA
LOW
UPPER
(LOWER
OPPO
+Z
-X
+X
SUB-PANEL UNIT ③
SUB-PANEL UNIT ④
SUB-PANEL UNIT ⑤
SUB-PANEL UNIT
SPACECRAFT E
PANEL UNIT ①
℄ UPPER SUPT
& TENSION TIE
(SEE DETAIL
B
B
TENSION TIE
(STOWED POSI
(SEE 2-5374-
DETAILS)
34.70
(3 PLACES)
TENSION-TIE-SPA
ATTACH POINT
STA 58.00
104.00
EDGE
MEM
SOLAR PANE
TENSION TI
(SEE DETAIL
10.00
3.50
SOLAR PANEL HINGE
(SEE 2-5374-J16 FOR DETAILS)
SPACECRAFT FR
MAIN HINGE ℄
(SEE VIEW 'D-D'
FOR DETAILS)
A-A
A
A

Fig 6.1-1



6> SAME AS SEC. 'B-B' EXCEPT THAT 2> & 3> ARE REVERSED.

5> PAD MATER AL-7075-T6 BONDED WITH BMS 5-25. ALL CONTACTING SURFACES LUBRICATED WITH MOLY-DI-SULFIDE. CONES ARE FOR 'X' & 'Z' SHEARS AND SLOTS ARE FOR 'X' OR 'Z' SHEARS.

4> LENGTH OF PAD VARIES WITH THE APPLIED SHEAR.

3> AT STA 114.00, ALONG THIS SHEAR PLANE, THE CENTER SPAR WILL HAVE ZERO SHEAR CAPABILITY (THEY WILL BE SILICONE RUBBER PADS) & THE OUTER SPARS WILL HAVE 'X'-AXIS SHEAR CAPABILITY ONLY.

2> AT STA. 114.00, ALONG THIS SHEAR PLANE, THE CENTER SPAR WILL HAVE SHEAR CAPABITY (SEE DETAIL 'B-B') IN THE 'Z'-AXIS & THE OUTER SPAR IN THE 'X' & 'Z'-AXIS (SEE DETAIL 'B-B').

1> ℄ OF SILICONE RUBBER SPACER PADS. THE NOMINAL THICKNESS IS .25. THE THICKNESSES WILL VARY IN ORDER THAT THE PROPER PRE-LOAD WILL BE APPLIED AT THE PADS TO INSURE THAT THE STOWED PANEL ASSYS WILL ACT AS ONE UNIT UNDER LOAD.

FIGURE 6.1-1 3

STOWED CONFIGURATION
FOLDING MODULAR ARRAY
SATURN IB/CENTAUR

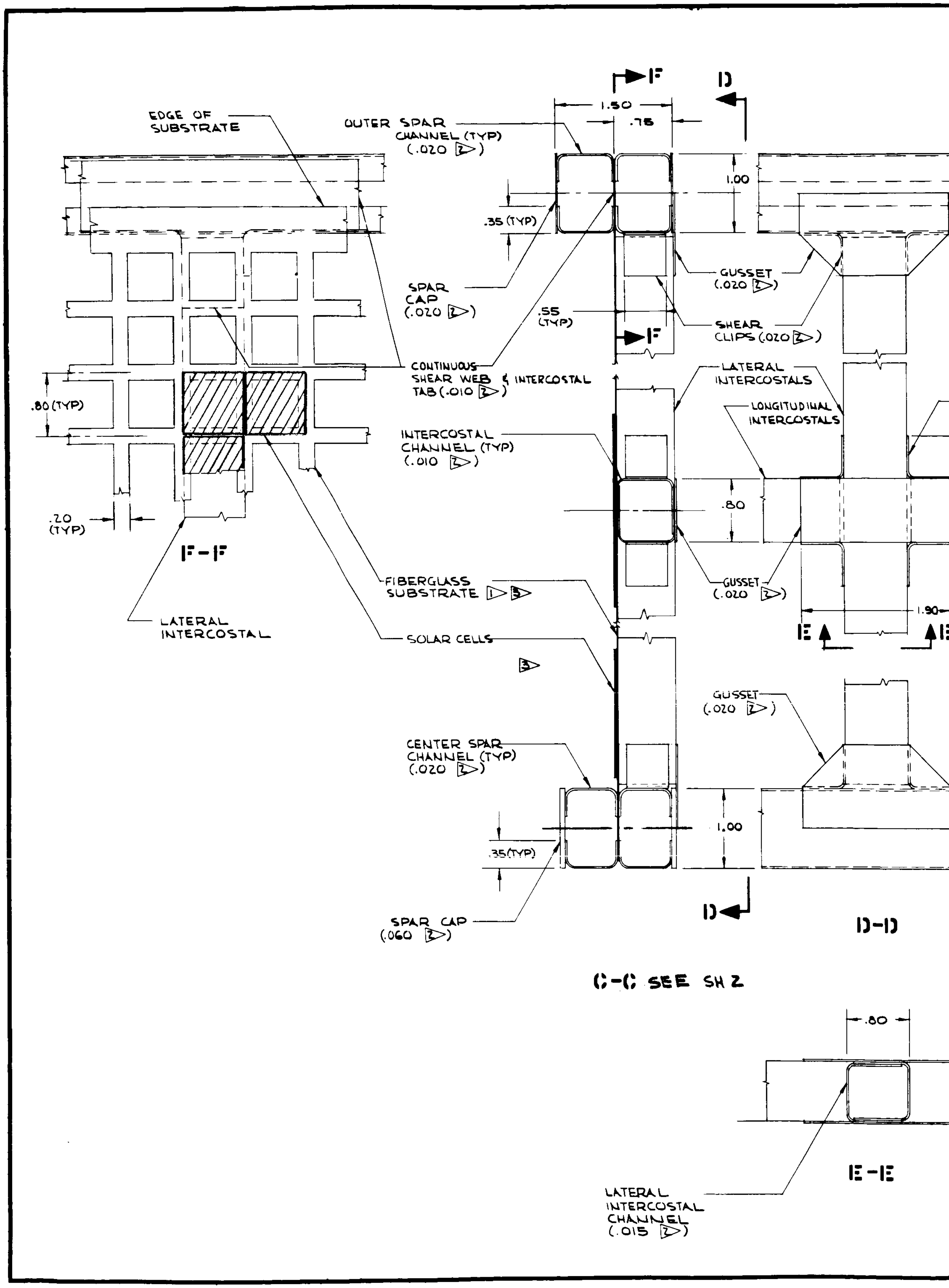
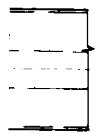
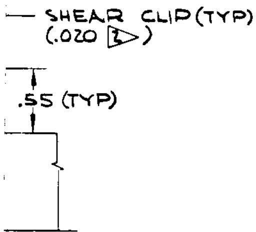
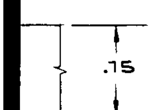
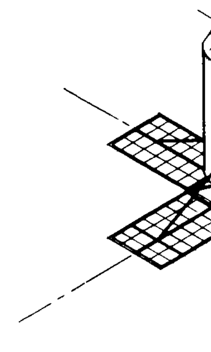

EDGE OF SUBSTRATE
OUTER SPAR CHANNEL (TYP) (.020 2)
1.50
.75
1.00
.35 (TYP)
SPAR CAP (.020 2)
.55 (TYP)
GUSSET (.020 2)
SHEAR CLIPS (.020 2)
CONTINUOUS SHEAR WEB & INTERCOSTAL TAB (.010 2)
LATERAL INTERCOSTALS
LONGITUDINAL INTERCOSTALS
.80 (TYP)
INTERCOSTAL CHANNEL (TYP) (.010 2)
.80
.20 (TYP)
F-F
GUSSET (.020 2)
FIBERGLASS SUBSTRATE 1 3
1.90
LATERAL INTERCOSTAL
SOLAR CELLS
3
GUSSET (.020 2)
CENTER SPAR CHANNEL (TYP) (.020 2)
1.00
.35(TYP)
SPAR CAP (.060 2)
D-D
C-C SEE SH 2
.80
E-E
LATERAL INTERCOSTAL CHANNEL (.015 2)

Fig 6.1-2 ①

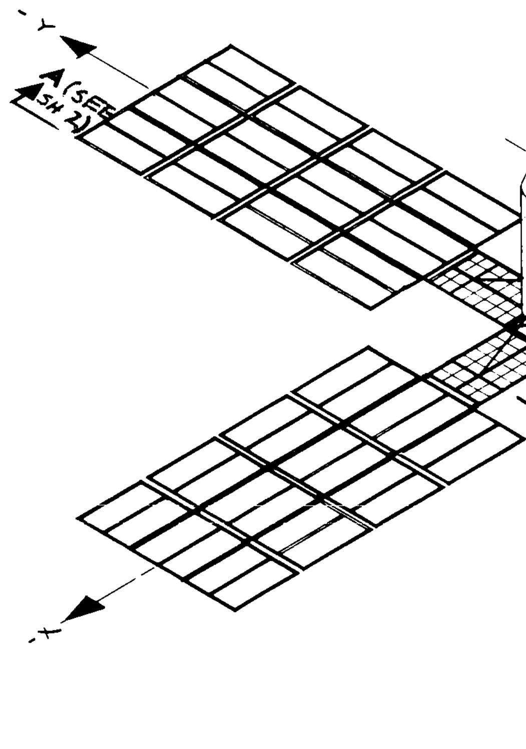








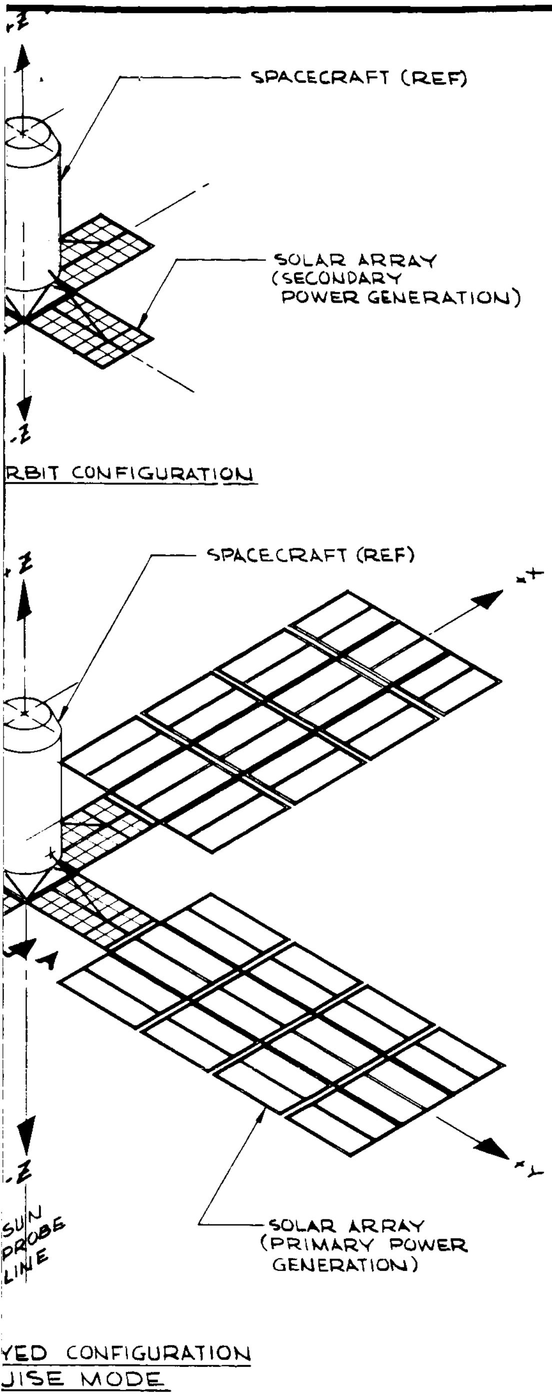
MARS O



DEPLO
CR

SUB-PANEL OR SUB-PANEL UNIT	AREA (SQ. FT)
SUB-PANEL (1)	107
SUB-PANEL UNIT (2)	321
SUB-PANEL UNIT (3)	281
SUB-PANEL UNIT (4)	281
SUB-PANEL UNIT (5)	246
1 SOLAR PANEL ASSY	1236
SOLAR ARRAY	4944
SPARS & EDGE MEMB	-278
NET SOLAR CELL AREA	4666

Fig 6.1-2
(2)



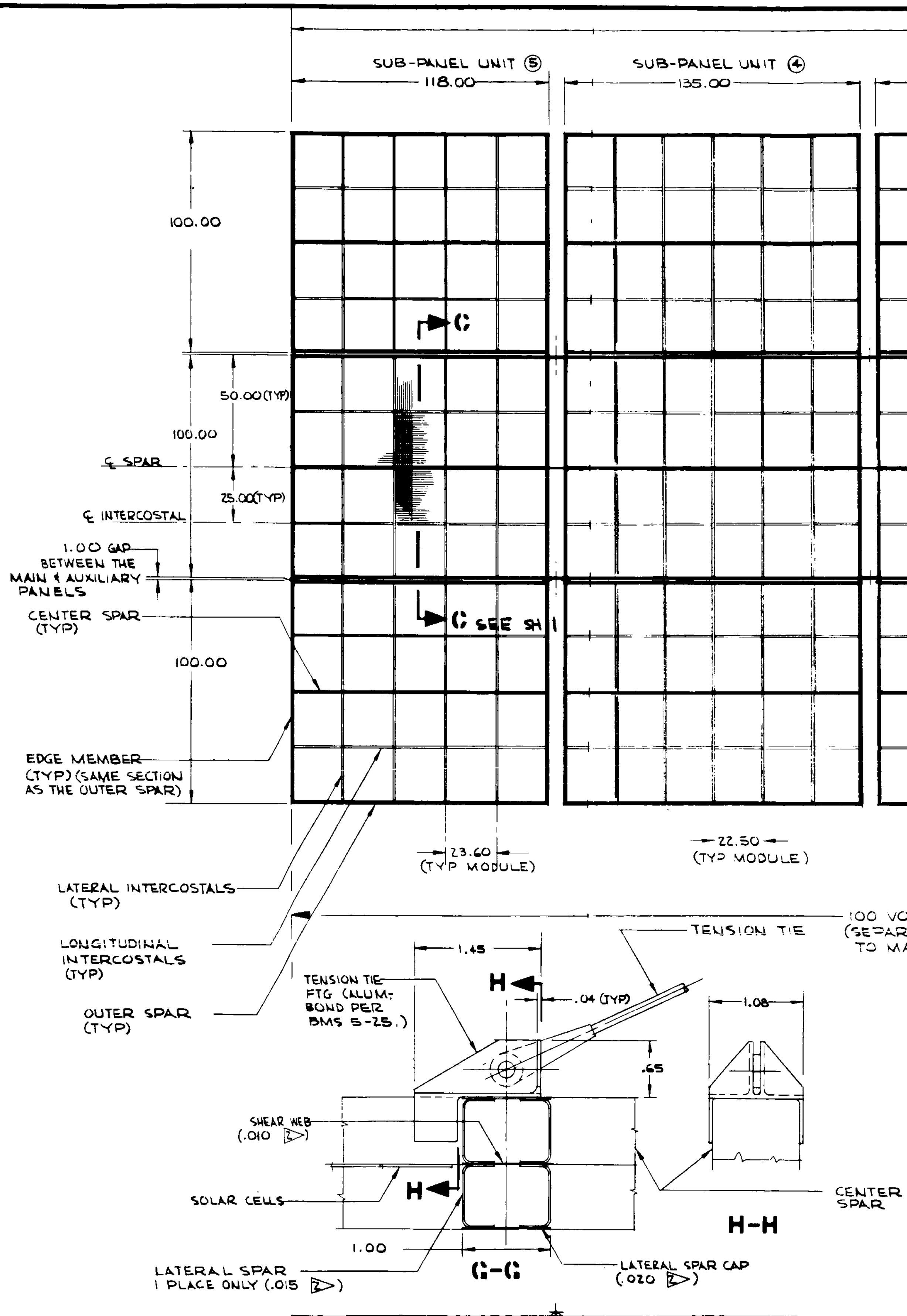
3 APPLY LAMINAR X-500 (3 MILS THICK) BLACK POLYURETHANE THERMAL CONTROL COATING TO THE DARK SIDE OF THE SUBSTRATE AND SOLAR CELLS ONLY.

2 HOT CROSS-ROLLED BERYLLIUM SHT. PER AMS 7902. BOND ALL JOINTS WITH BMS 5-29 EXCEPT AS NOTED.

1 THE SUBSTRATE IS A 3MIL UNIDIRECTIONAL FIBERGLASS PRE-PREG (EPOXY RESIN) TAPE WHICH IS WOVEN AS SHOWN & CURED. THEN THE TAPE IS PRE-TENSIONED TO 10 LBS./INCH & BONDED TO THE PANEL STRUCTURE.

FIGURE 6.1-2 DEPLOYED CONFIGURATION
FOLDING MODULAR
SOLAR ARRAY
SATURN IB/CENTAUR

5



SUB-PANEL UNIT ⑤
118.00
SUB-PANEL UNIT ④
135.00
100.00
50.00(TYP)
100.00
℄ SPAR
25.00(TYP)
℄ INTERCOSTAL
1.00 GAP BETWEEN THE MAIN & AUXILIARY PANELS
CENTER SPAR (TYP)
100.00
EDGE MEMBER (TYP) (SAME SECTION AS THE OUTER SPAR)
C
C SEE SH 1
23.60 (TYP MODULE)
22.50 (TYP MODULE)
LATERAL INTERCOSTALS (TYP)
LONGITUDINAL INTERCOSTALS (TYP)
OUTER SPAR (TYP)
TENSION TIE
1.45
TENSION TIE FTG (ALUM-BOND PER BMS 5-25.)
H
.04 (TYP)
.65
1.08
SHEAR WEB (.010 2)
SOLAR CELLS
H
1.00
LATERAL SPAR 1 PLACE ONLY (.015 2)
C-C
LATERAL SPAR CAP (.020 2)
H-H
CENTER SPAR

Fig 6.1(1)-2A

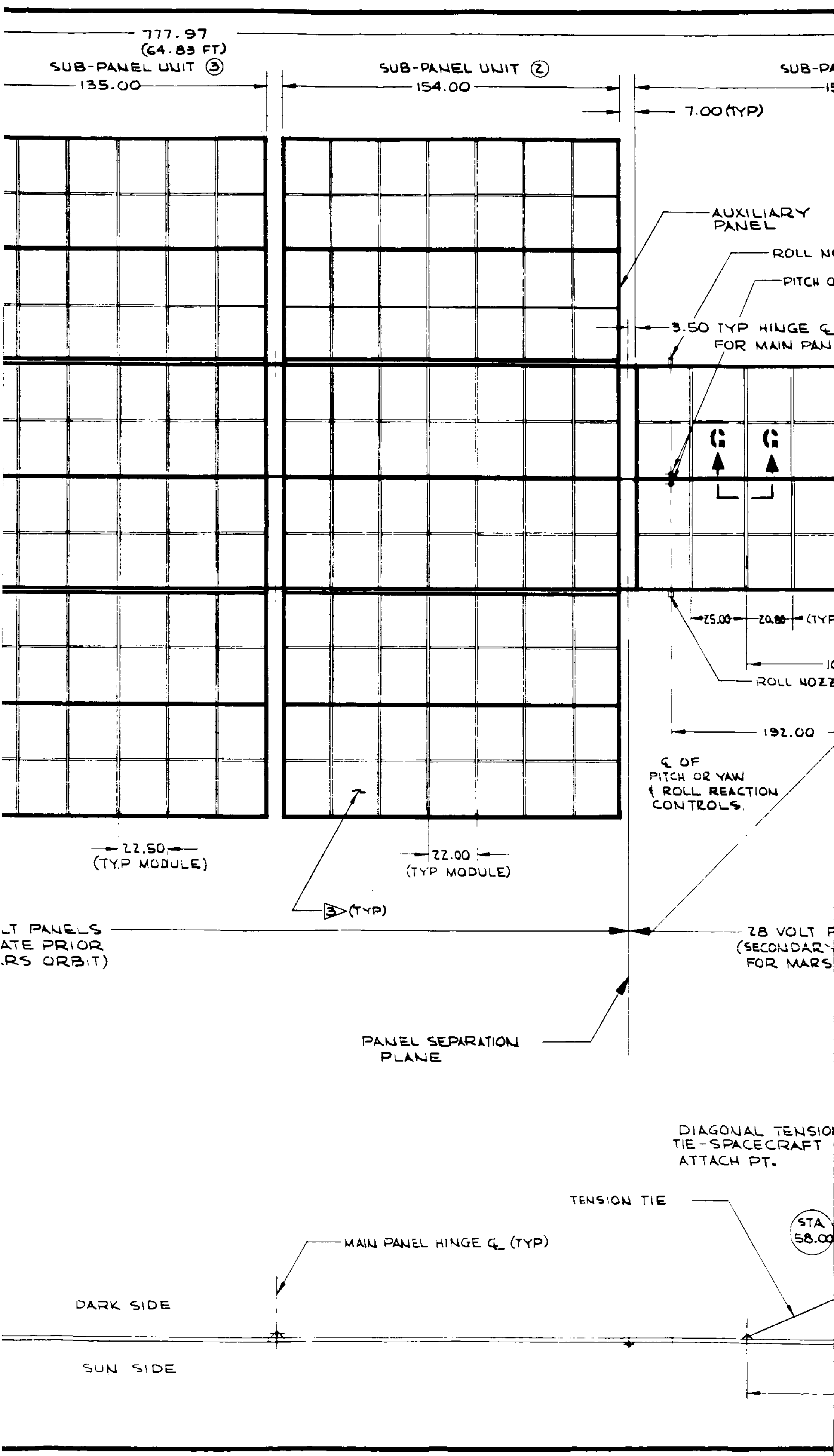
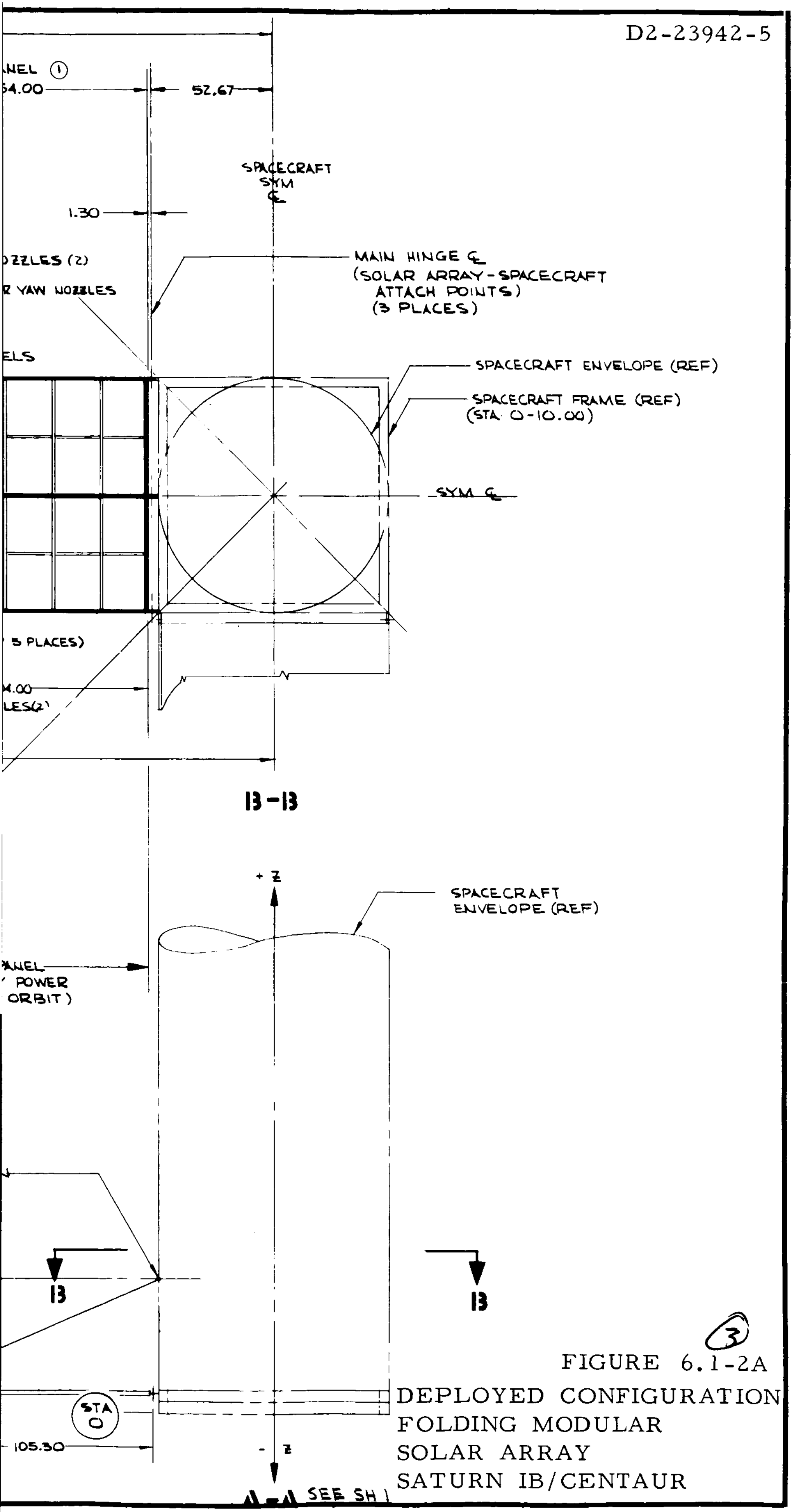
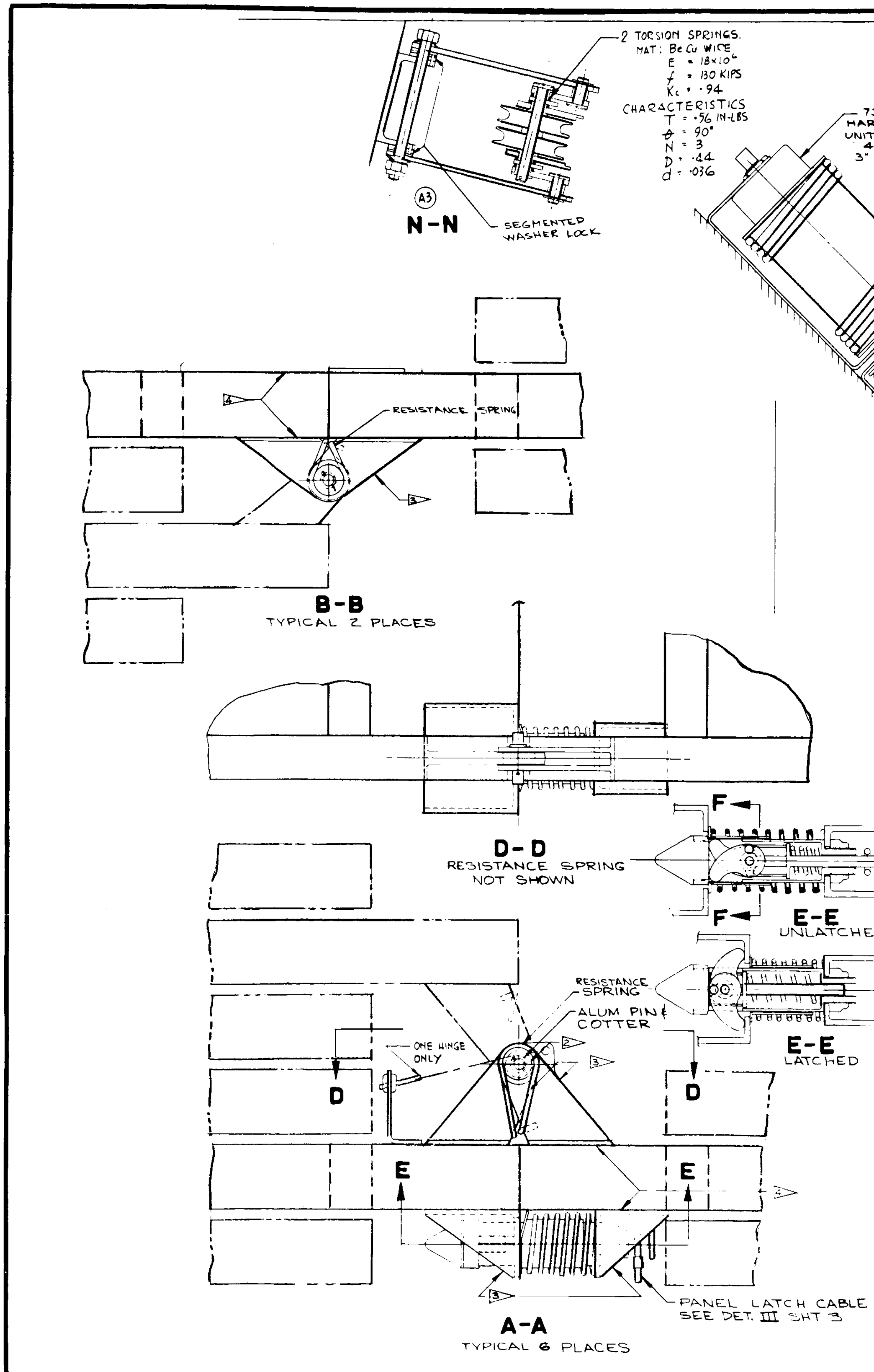

777.97
(64.83 FT)
SUB-PANEL UNIT ③
135.00
SUB-PANEL UNIT ②
154.00
SUB-PA
7.00 (TYP)
AUXILIARY
PANEL
ROLL N
PITCH O
3.50 TYP HINGE ℄
FOR MAIN PAN
25.00
20.80
(TYP
ROLL NOZZ
192.00
℄ OF
PITCH OR YAW
& ROLL REACTION
CONTROLS.
22.50
(TYP MODULE)
22.00
(TYP MODULE)
3 (TYP)
LT PANELS
ATE PRIOR
RS ORBIT)
28 VOLT F
(SECONDARY
FOR MARS
PANEL SEPARATION
PLANE
DIAGONAL TENSION
TIE-SPACECRAFT
ATTACH PT.
TENSION TIE
STA
58.00
MAIN PANEL HINGE ℄ (TYP)
DARK SIDE
SUN SIDE

Fig 6.1-2A (2)



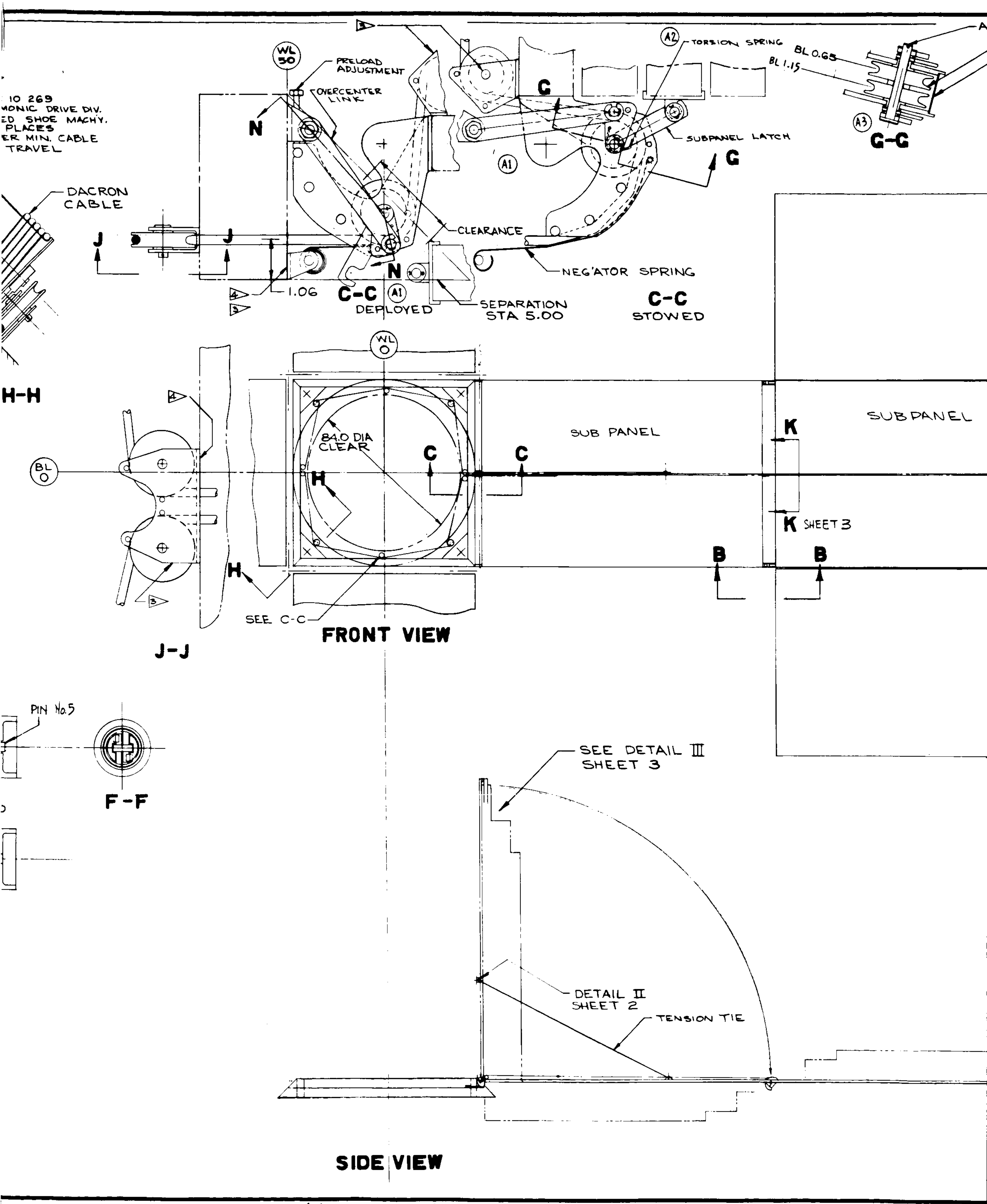
③

FIGURE 6.1-2A
DEPLOYED CONFIGURATION
FOLDING MODULAR
SOLAR ARRAY
SATURN IB/CENTAUR



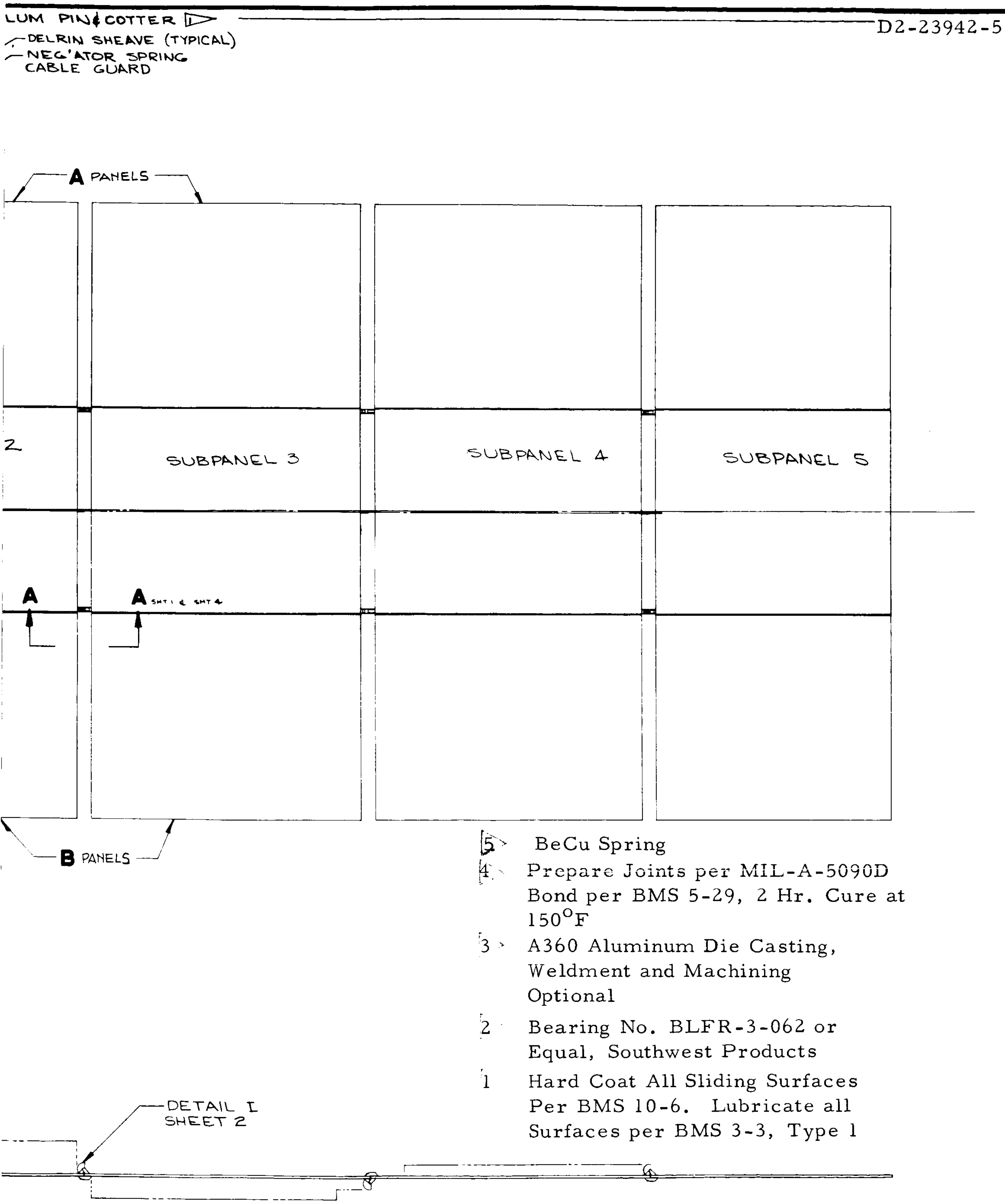
2 TORSION SPRINGS.
MAT: Be Cu WIRE
E = 18×10⁶
f = 130 KIPS
Kc = ·94
CHARACTERISTICS
T = ·56 IN-LBS
θ = 90°
N = 3
D = ·44
d = ·036
A3
N-N
SEGMENTED WASHER LOCK
RESISTANCE SPRING
B-B
TYPICAL 2 PLACES
D-D
RESISTANCE SPRING NOT SHOWN
F
E-E
UNLATCHE
E-E
LATCHED
RESISTANCE SPRING
ALUM PIN & COTTER
ONE HINGE ONLY
D
E
PANEL LATCH CABLE
SEE DET. III SHT 3
A-A
TYPICAL 6 PLACES

Fig 6.1-3



10 269
MONIC DRIVE DIV.
ED SHOE MACHY.
PLACES
ER MIN. CABLE
TRAVEL
WL 50
PRELOAD ADJUSTMENT
OVERCENTER LINK
N
DACRON CABLE
J
J
CLEARANCE
1.06
C-C
DEPLOYED
N
A1
SEPARATION STA 5.00
NEG'ATOR SPRING
C-C
STOWED
C
A1
A2
TORSION SPRING
SUBPANEL LATCH
G
BL 0.65
BL 1.15
A3
G-G
H-H
WL O
BL O
84.0 DIA CLEAR
H
C
C
SUB PANEL
SUBPANEL
K
K SHEET 3
B
B
H
SEE C-C
FRONT VIEW
J-J
PIN No.5
F-F
SEE DETAIL III SHEET 3
DETAIL II SHEET 2
TENSION TIE
SIDE VIEW

Fig 6.1-3
(2)



5 BeCu Spring

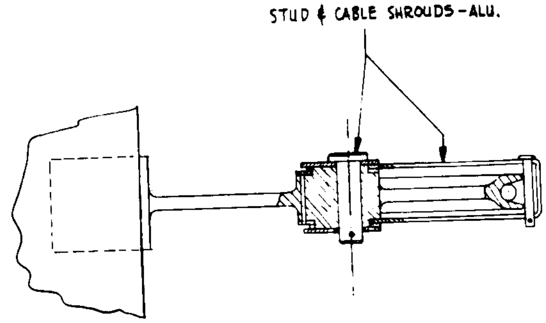
4 Prepare Joints per MIL-A-5090D Bond per BMS 5-29, 2 Hr. Cure at 150°F

3 A360 Aluminum Die Casting, Weldment and Machining Optional

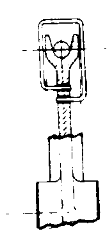
2 Bearing No. BLFR-3-062 or Equal, Southwest Products

1 Hard Coat All Sliding Surfaces Per BMS 10-6. Lubricate all Surfaces per BMS 3-3, Type 1

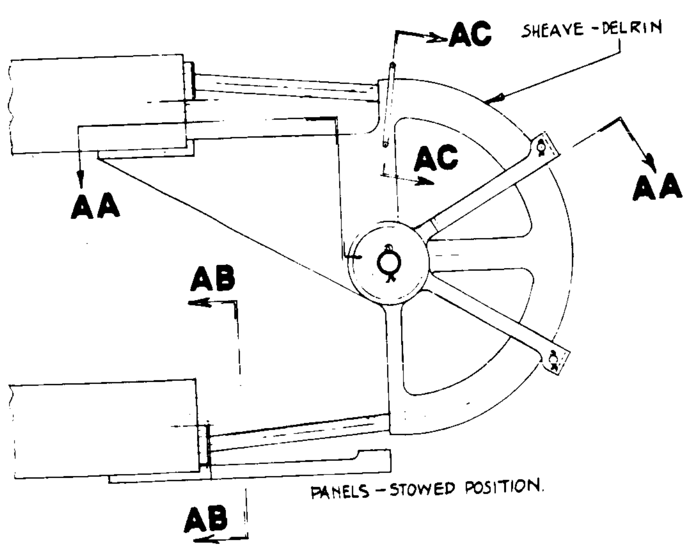
FIGURE 6.1-3 MECHANISM INSTALLATION
SHEET 1 OF 3 50 KW SOLAR ARRAY
SATURN IB/CENTAUR



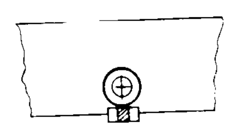
AA-AA



AC-AC

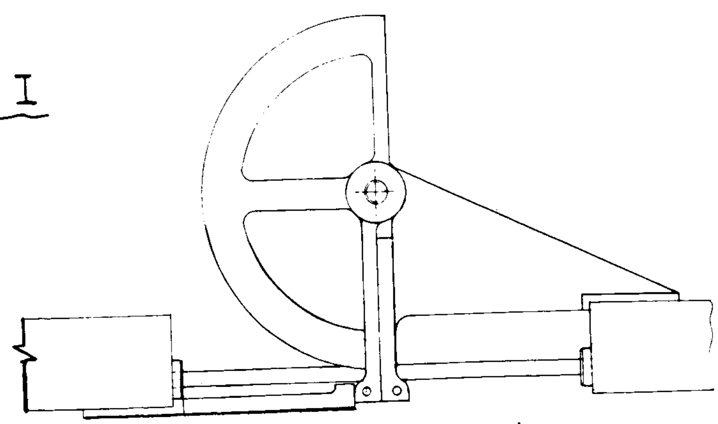


PANELS-STOWED POSITION.

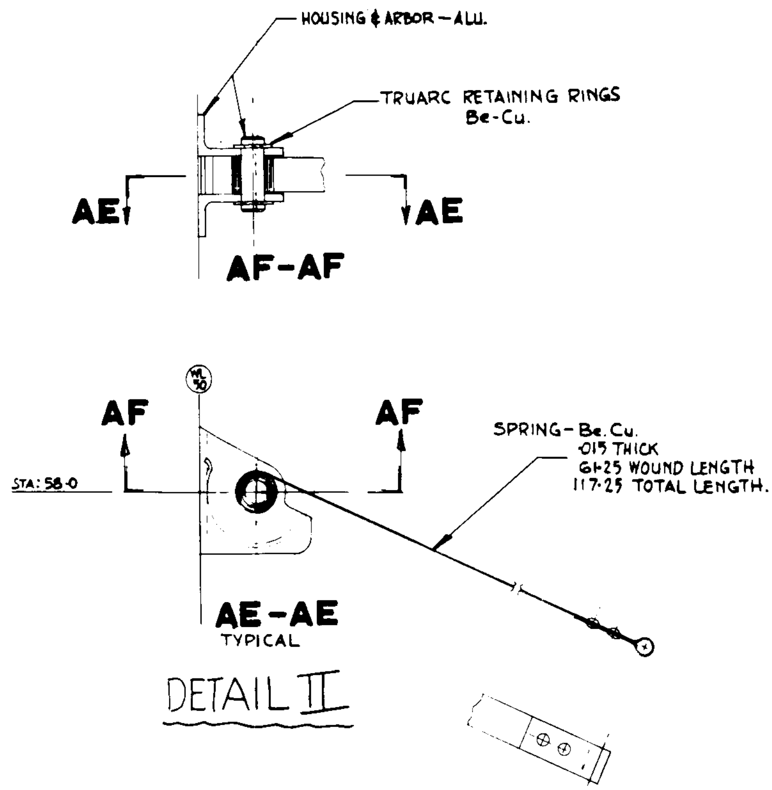


AB-AB

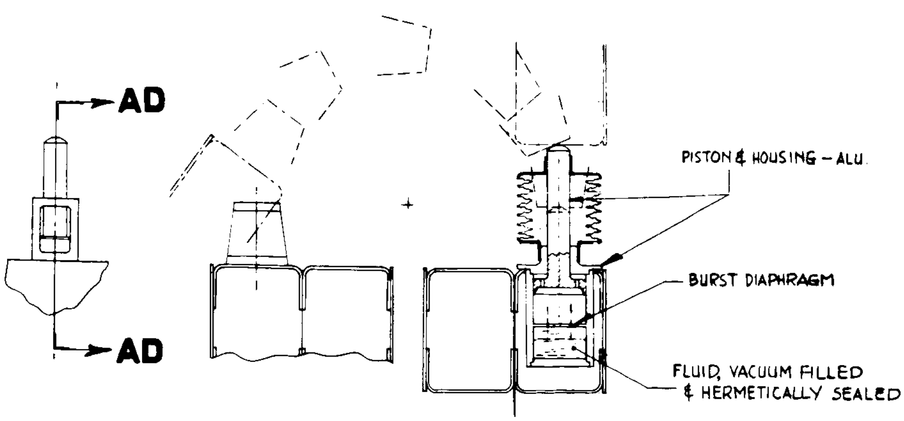
DETIAL I



PANELS-DEPLOYED POSITION



DETAIL II



AD-AD

DAMPER DETAIL
TWICE SIZE

FIGURE 6.1-3 MECHANISM INSTALLATION
SHEET 2 OF 3 50 KW SOLAR ARRAY
SATURN IB/CENTAUR

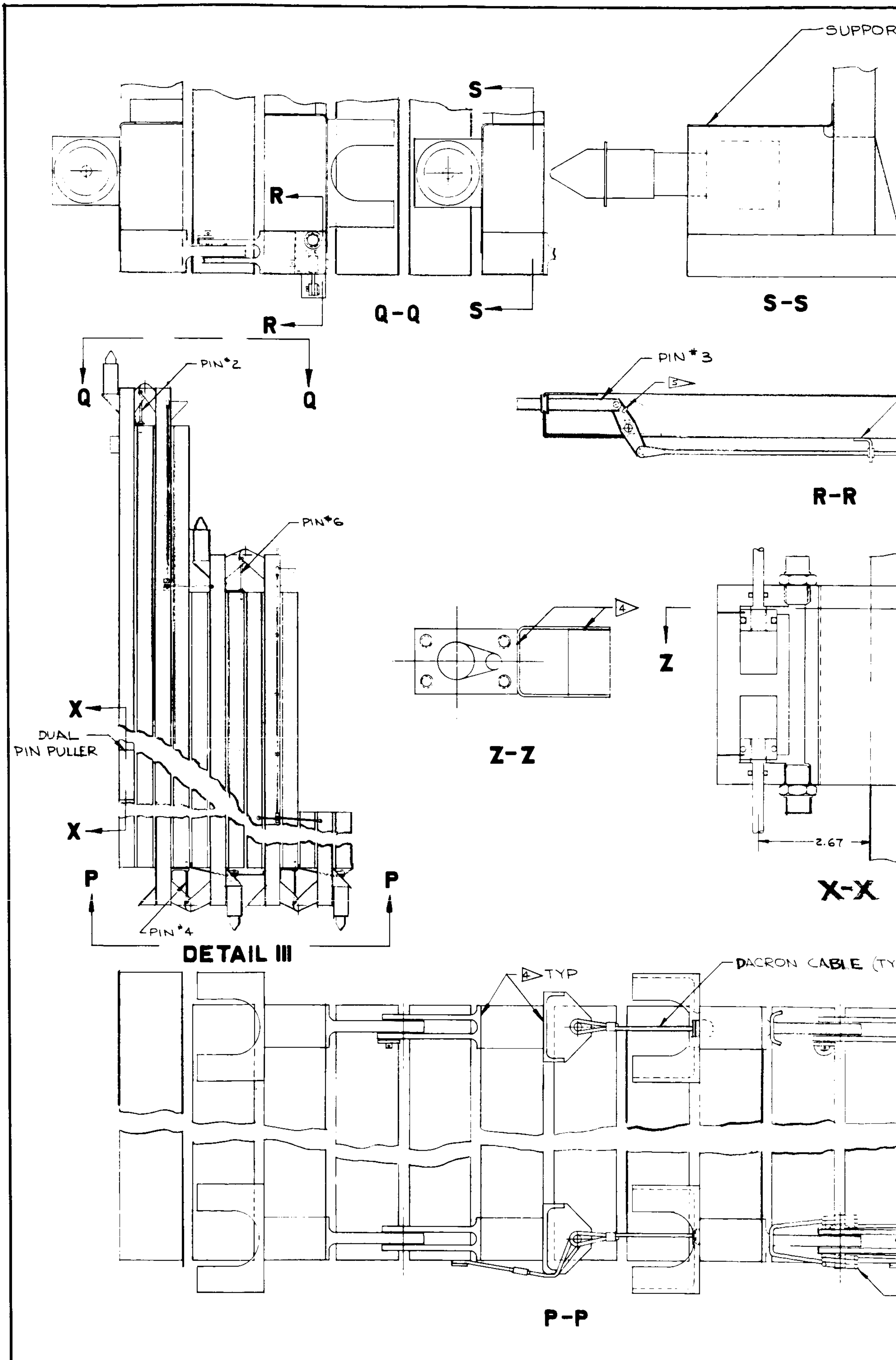
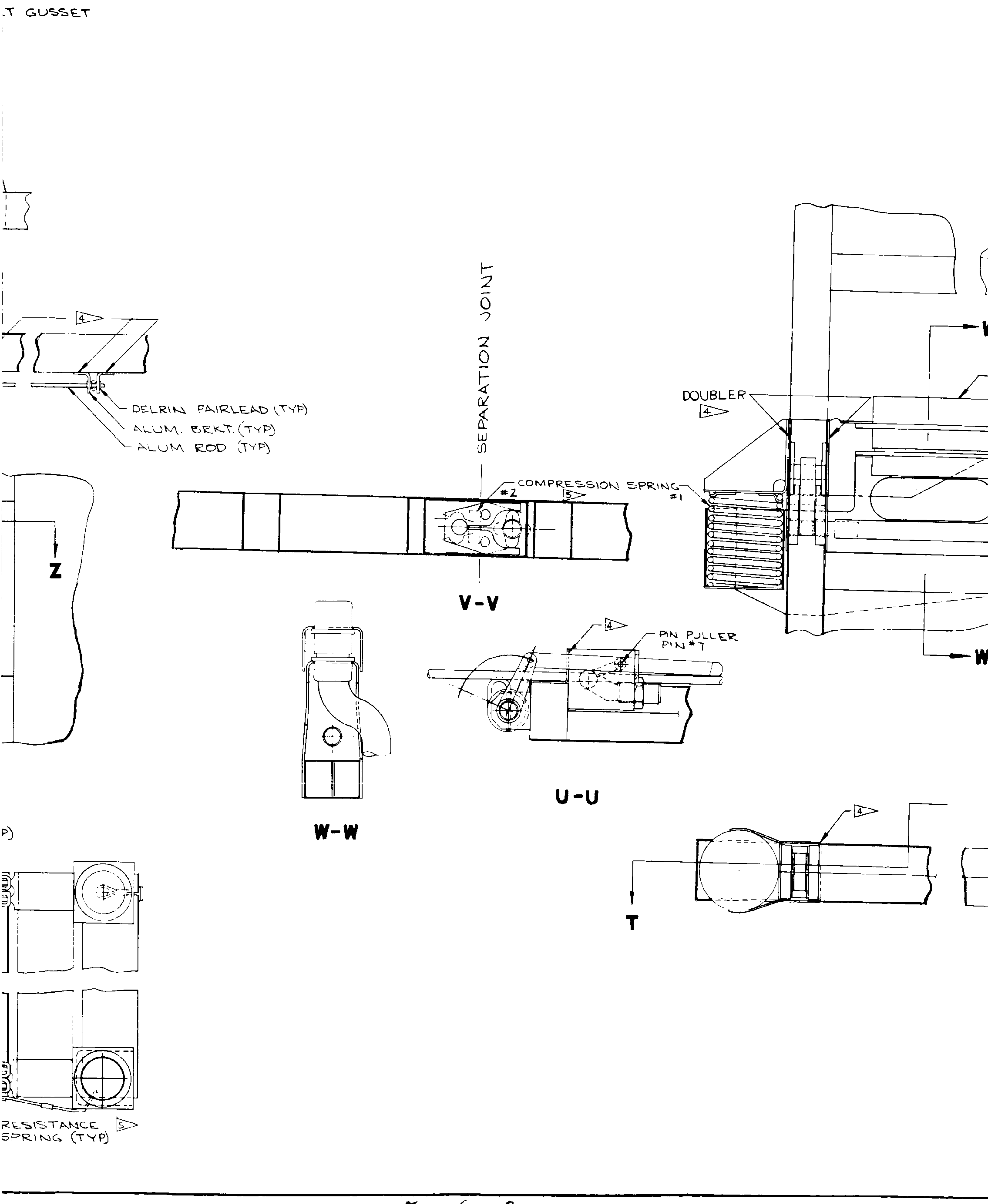

SUPPOR
S
R
Q-Q
S
R
S-S
PIN #2
Q
Q
PIN #3
3
R-R
PIN #6
4
Z
Z-Z
X
DUAL
PIN PULLER
X
2.67
X-X
P
P
PIN #4
DETAIL III
4 TYP
DACRON CABLE (TY
P-P

Fig 6.1-3



T GUSSET
4
DELRIN FAIRLEAD (TYP)
ALUM. BRKT. (TYP)
ALUM ROD (TYP)
SEPARATION JOINT
Z
COMPRESSION SPRING
#2
5
#1
DOUBLER
4
W
V-V
4
PIN PULLER
PIN #7
W-W
U-U
4
T
P)
RESISTANCE
SPRING (TYP)
5

Fig 6.1-3
(2)

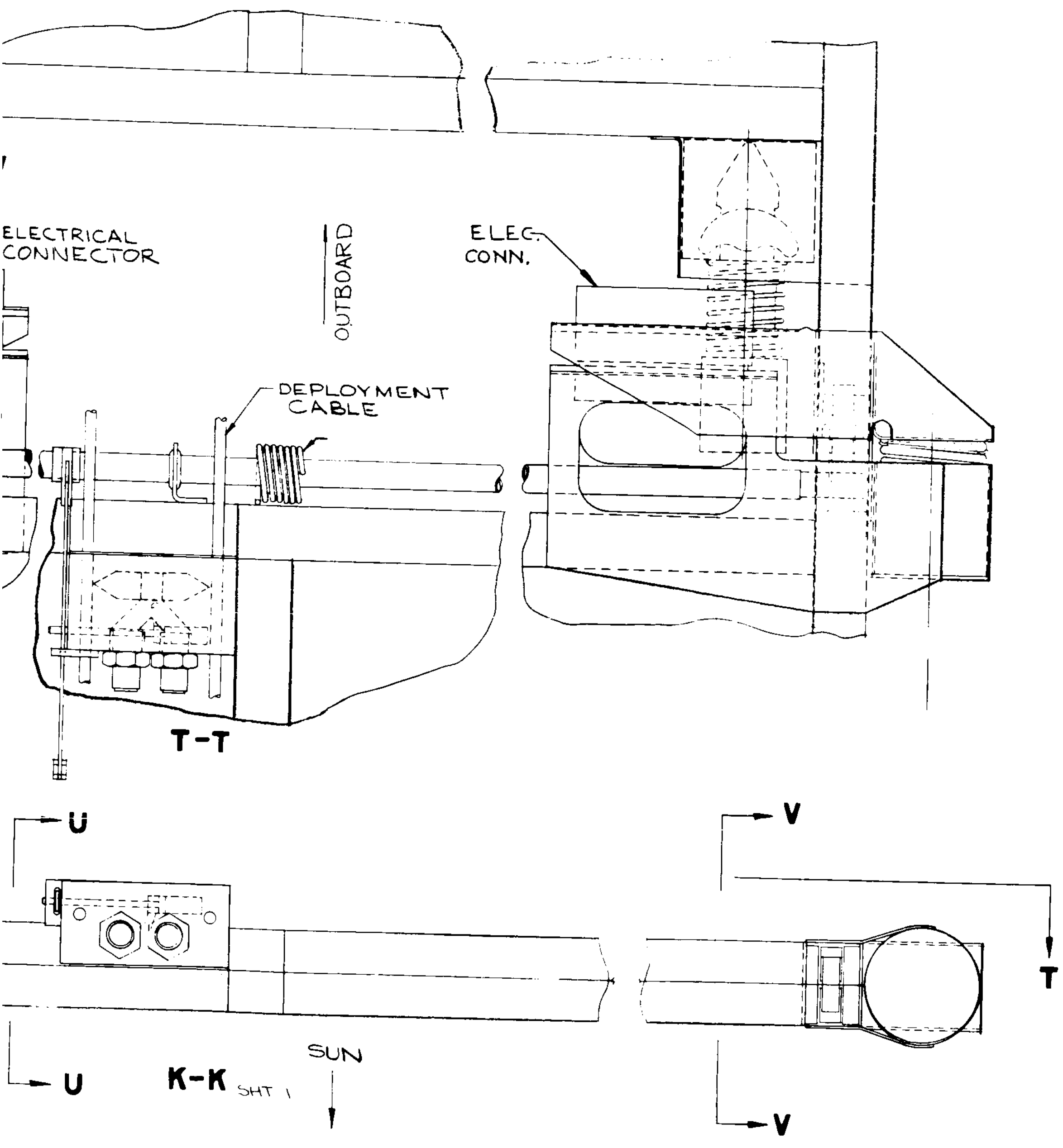


FIGURE 6.1-3 MECHANISM INSTALLATION
SHEET 3 OF 3 50 KW SOLAR ARRAY
SATURN IB/CENTAUR

FIG 6.1-3
3

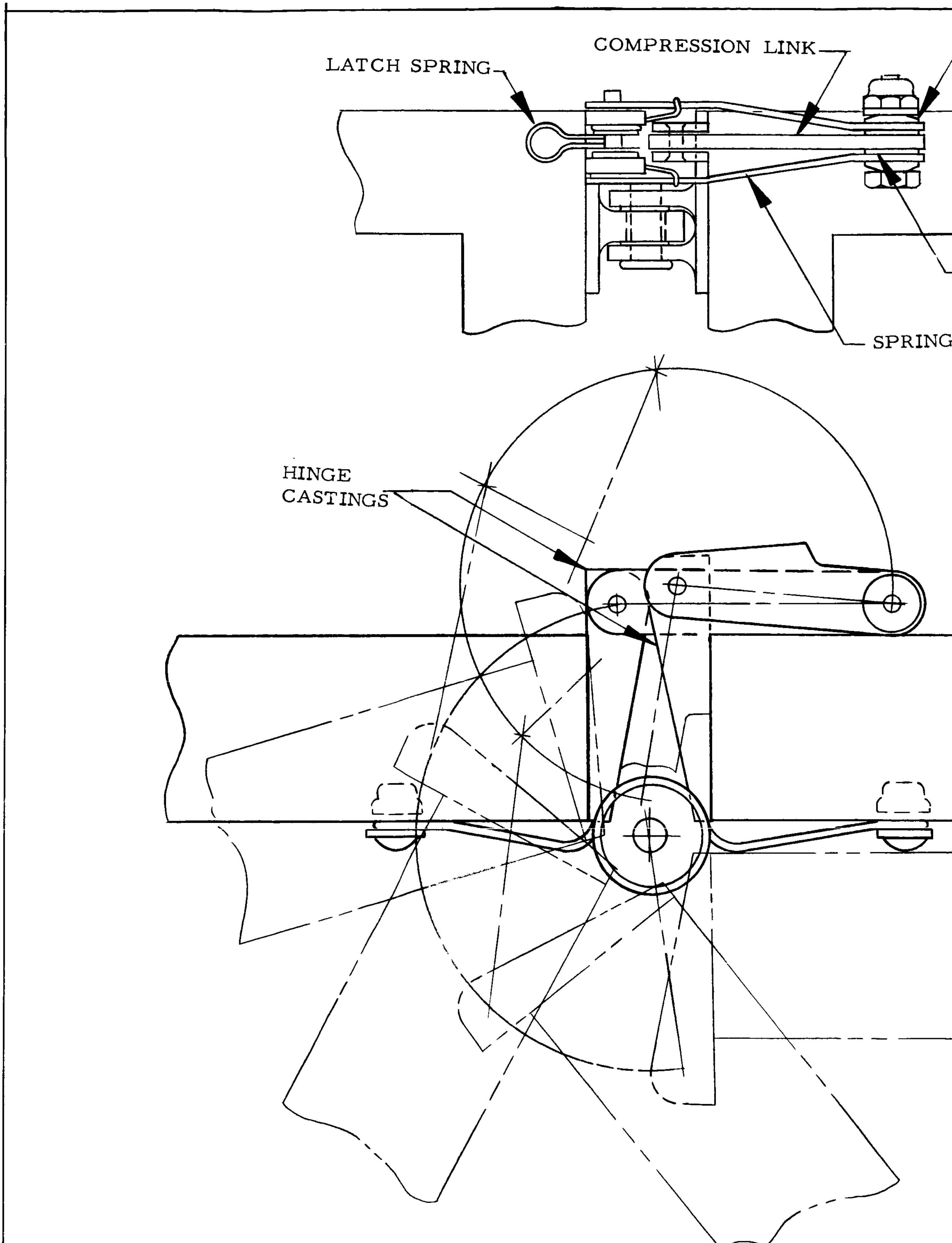


Fig 6.1-4

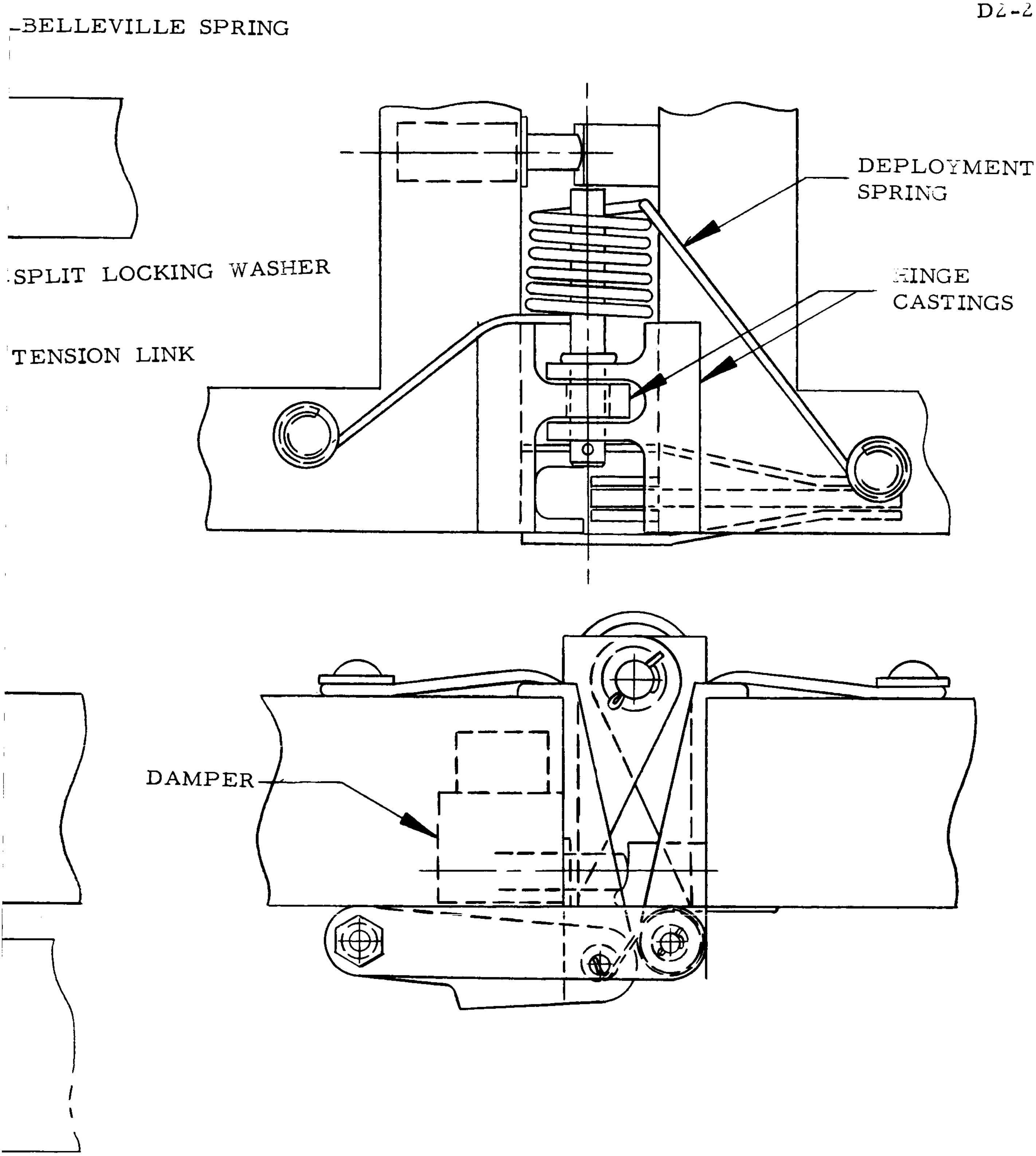
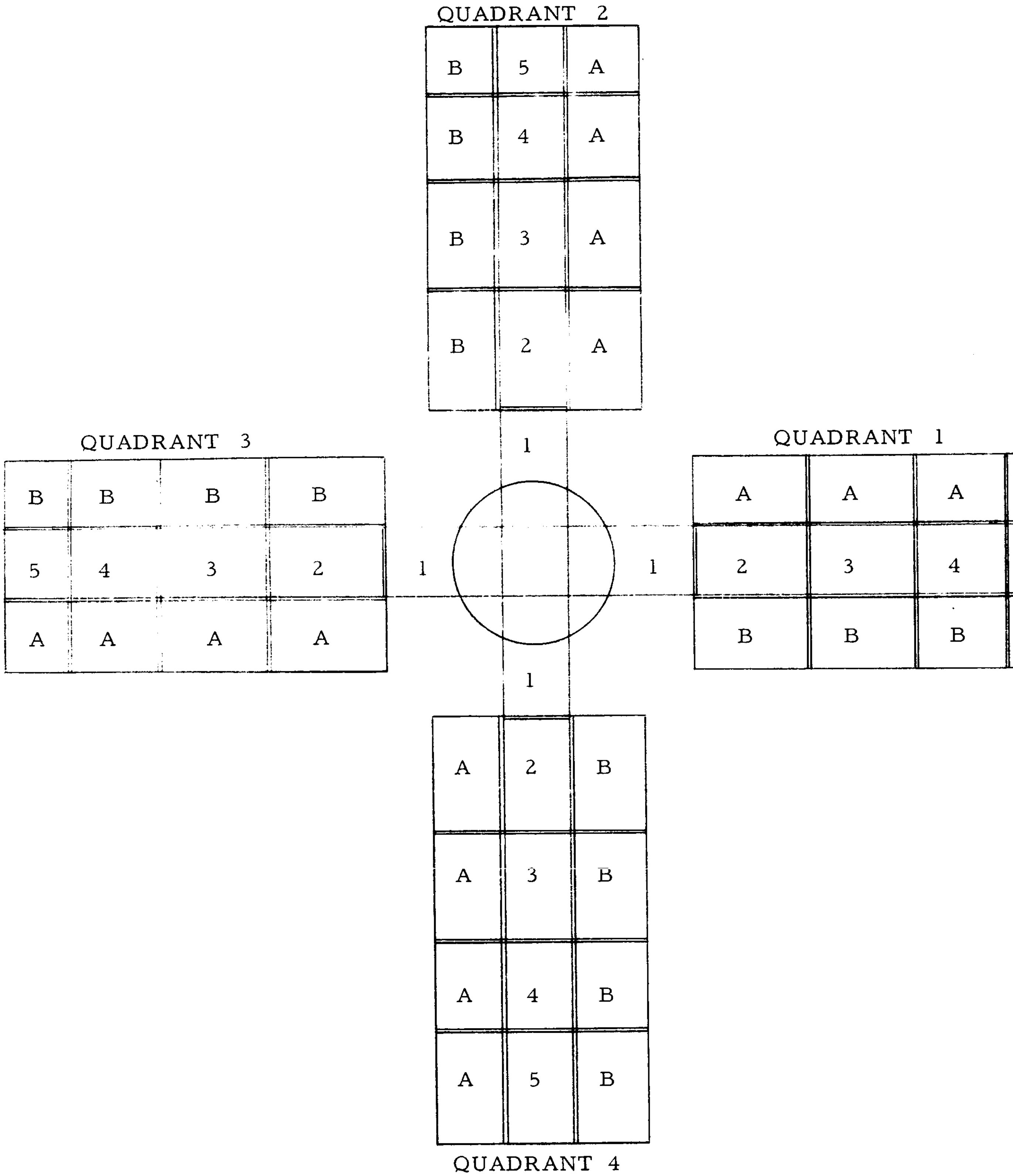


FIGURE 6.1-4 OVERCENTER HINGE - LATCH

(2)



FOLDING MODULAR SOLAR ARRAY
(FULLY DEPLOYED)

Fig 6.1-5
0

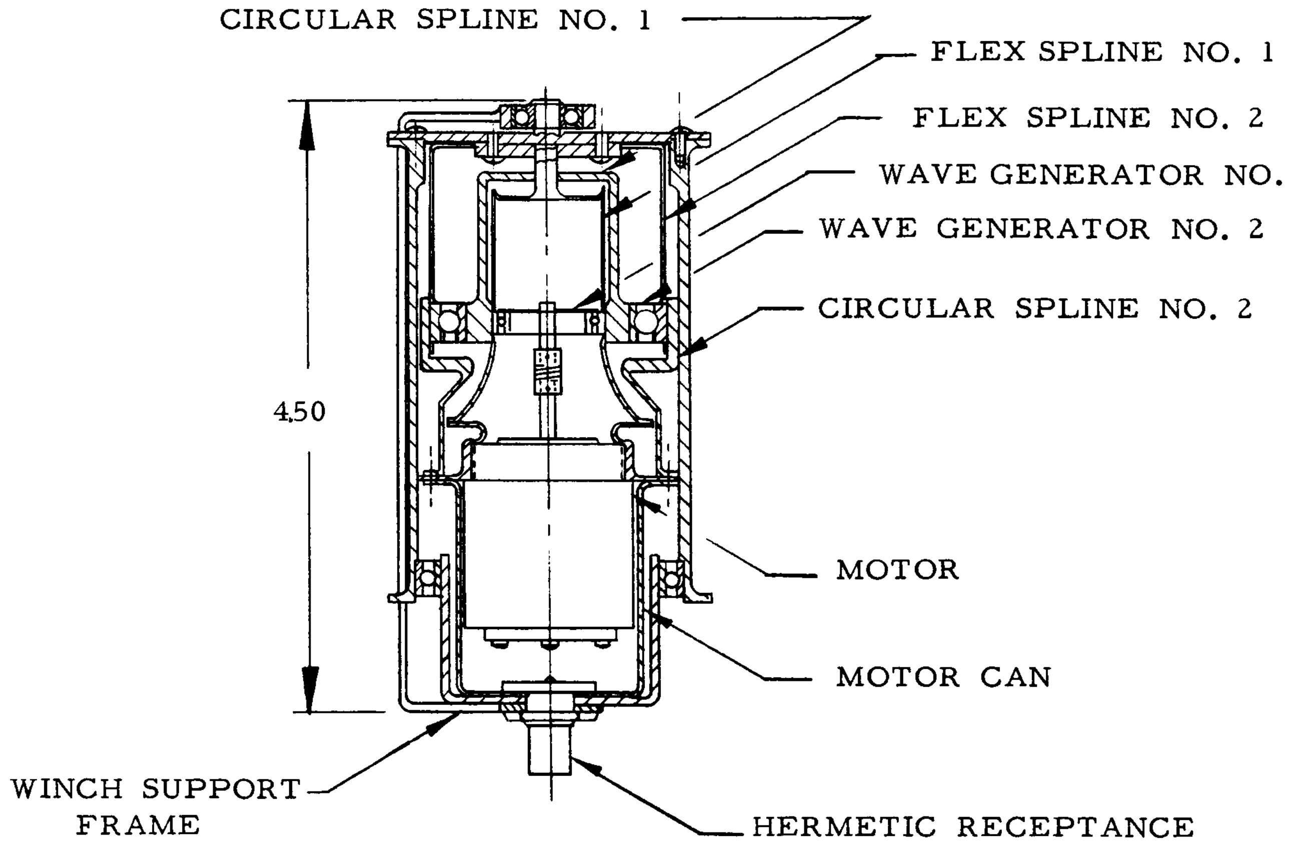
DEPLOYMENT SEQUENCE
MOTOR PULLEY AND CABLE SYSTEM

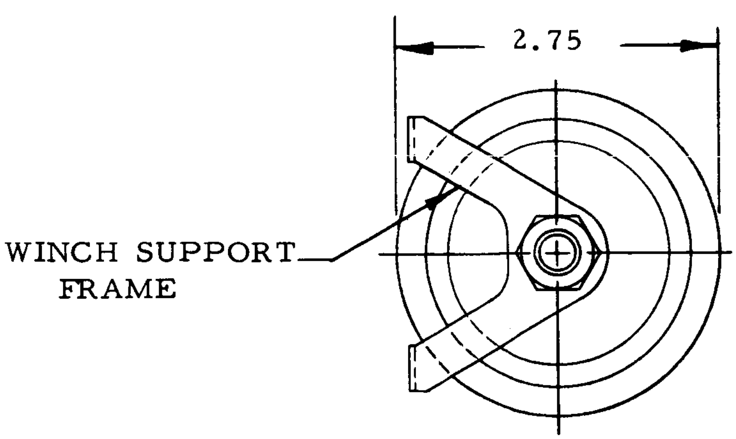
TIME	
0	Apply voltage to 4 dual pin pullers and 4 motor drive units simultaneously to unlatch the array and begin deployment. (Detail I of Figure 6.1-1, and Section X-X of Figure 6.1-3). Rotate all folded arrays 90° about their main hinge line at the base of the spacecraft and latch joint #1 by use of an overcenter link.
79 sec	Release sub-panels 2, 3, 4, and 5.
81 sec	Rotate sub-panels 2, 3, 4, and 5 180° about hinge line #2.
195 sec	Release sub-panel 2A. Rotate sub-panel 2A 180° and latch.
244 sec	Release sub-panels 3, 4, and 5.
246 sec	Latch joint #2 and rotate sub-panels 3, 4, and 5 about hinge line #3.
360 sec	Release sub-panel 2B. Rotate sub-panel 2B 180° and latch.
411 sec	Latch joint #3 and release sub-panel 3B, 4, and 5. Rotate sub-panels 4 and 5 about hinge line #4.
525 sec	Release sub-panel 3A. Rotate sub-panel 3A 180° and latch.
574 sec	Release sub-panels 4A and 5. Rotate sub-panel 4A 180° and latch.
576 sec	Latch joint #4. Rotate sub-panel 5 about hinge line #5.
690 sec	Release sub-panel 4B. Rotate sub-panel 4B 180° and latch.
741 sec	Latch joint #5. Release sub-panels 5A and 5B, rotate 180° and latch. De-energize deployment motors.
840 sec	Deployment complete, signal received.

840 sec/60 = 14 minutes

SOLAR ARRAY
DEPLOYMENT SEQUENCE

FIGURE 6.1-5 (2)





ELECTRIC MOTOR-HARMONIC DRIVE

FIGURE 6.1-6

6.1.4 Solar Cell Modules

6.1.4.1 Cell Efficiency

The average efficiency of the cells proposed for the design was established at 11.8% air mass 1. This decision was reached after conferring with three solar cell vendors (Heliotek, Texas Instruments, and RCA). Two of the vendors felt that 8 mil back-connected cells having an average efficiency of 11.8% could be produced in quantity. These vendors acknowledged that 8 mil cells produced in the experimental quantities in 1965 would not meet this requirement, but improvement in subsequent production runs would raise efficiency to the desired value. One vendor, RCA, felt that a development program would be necessary to attain this efficiency.

To obtain a clear picture of the efficiency of 8 mil cells produced under JPL contract in 1965, Boeing tested all SiO coated 8 mil cells supplied to Boeing for use on the model panels. These tests were conducted under a tungsten lamp set to an equivalent intensity of 100 mw/cm^2 by means of a Heliotek standard cell 130. Color temperature of the lamp was adjusted to $2800^{\circ}K$ using a Heliotek color temperature meter. Cell temperature was maintained at $28^{\circ}C + 1^{\circ}C$ and cell current was observed at a cell voltage of 0.460 volts. The distribution of these cells is shown in Figure 6.1.4-1. Note that the average efficiency of these cells was 11.72%. This data confirms the reasonability of the 11.8% efficiency established for the design.

6.1.4.2 Interconnector Design

A detailed study of the solar cell interconnector was performed. Figure 6.1.4-2 shows the relation between cell conversion efficiency and the resistance of the connection between cells. A connector design was evolved using 2 mil expanded silver metal having the configuration shown in the inset. Resistance of the connection between cells considering the resistance in the "p" layer of the cell, the solder joints and connector was calculated to be 0.02 ohms. With this value of resistance, the reduction in cell efficiency will be less than 1%.

6.1.4.3 Stack Design

The solar cell stack is shown in Figure 6.1.4-3. The stack consists of the following items.

(a) A 0.004 inch thick cover glass equipped with a 0.4 micron ultra violet reflecting filter.
(b) A 0.001 inch thick layer of RTV 602 bonding the cover glass to the cell.
(c) A 0.008 inch thick 2 x 2 cm solar cell equipped with back contacts and having an average efficiency of 11.8%, air mass 1.

(d) A layer of RTV 40 bonding the cell to the substrate. The thickness varies from 0.001 inches beneath the connector, to 0.004 inches beneath the cell.
(e) A 0.002 inch thick expanded silver mesh connector.

As described previously, the stack will be supported by 0.003 inch thick fiberglass webs. A 0.003 inch thick coating of Laminar X-500 will cover the entire back side surface of the array and serve to increase the emittance of the panel. Nominal spacing between adjacent cells will be 0.012 inches.

6.1.4.4 Module Layout

The solar cell layout for the Saturn IB/Centaur array is shown in Figure 6.1.4-4. As described previously, the array consists of 4 panels with each panel in turn consisting of 5 sub-panel assemblies. Sub-panel #1, nearest to the vehicle, carries the 28 volt power source plus 4 - 100 volt modules. Sub-panel assembly #2 carries 13 - 100 volt modules, sub-panel assembly #3 and #4 each carry 12 - 100 volt modules. Sub-panel assembly #5 carries 10 100 volt modules.

A seven cell parallel connected group is the basic building block for both the 28 and 100 volt modules. The 28 volt modules consist of 54 series-connected 7 cell groups arranged in a 6 row rectangular layout. The 100 volt modules consist of 236 series connected 7 cell groups arranged in a 4 row rectangular layout.

6.1.4.5 Array Power Analysis

The power output of both the Saturn IB and Atlas arrays were calculated using the equations presented in the criteria statement. The Saturn IB/Centaur array will develop 47.7 KW at 1.0 A.U. The 28 volt power source will develop 1.18 KW at 1.67 A.U. The Atlas/Centaur array will develop a total power output of 10.18 KW at 1.0 A.U. Its 28 volt power source will develop 0.51 KW at 1.67 A.U.

The power voltage and current voltage characteristics of the arrays are shown in Figure 6.1.4-5.

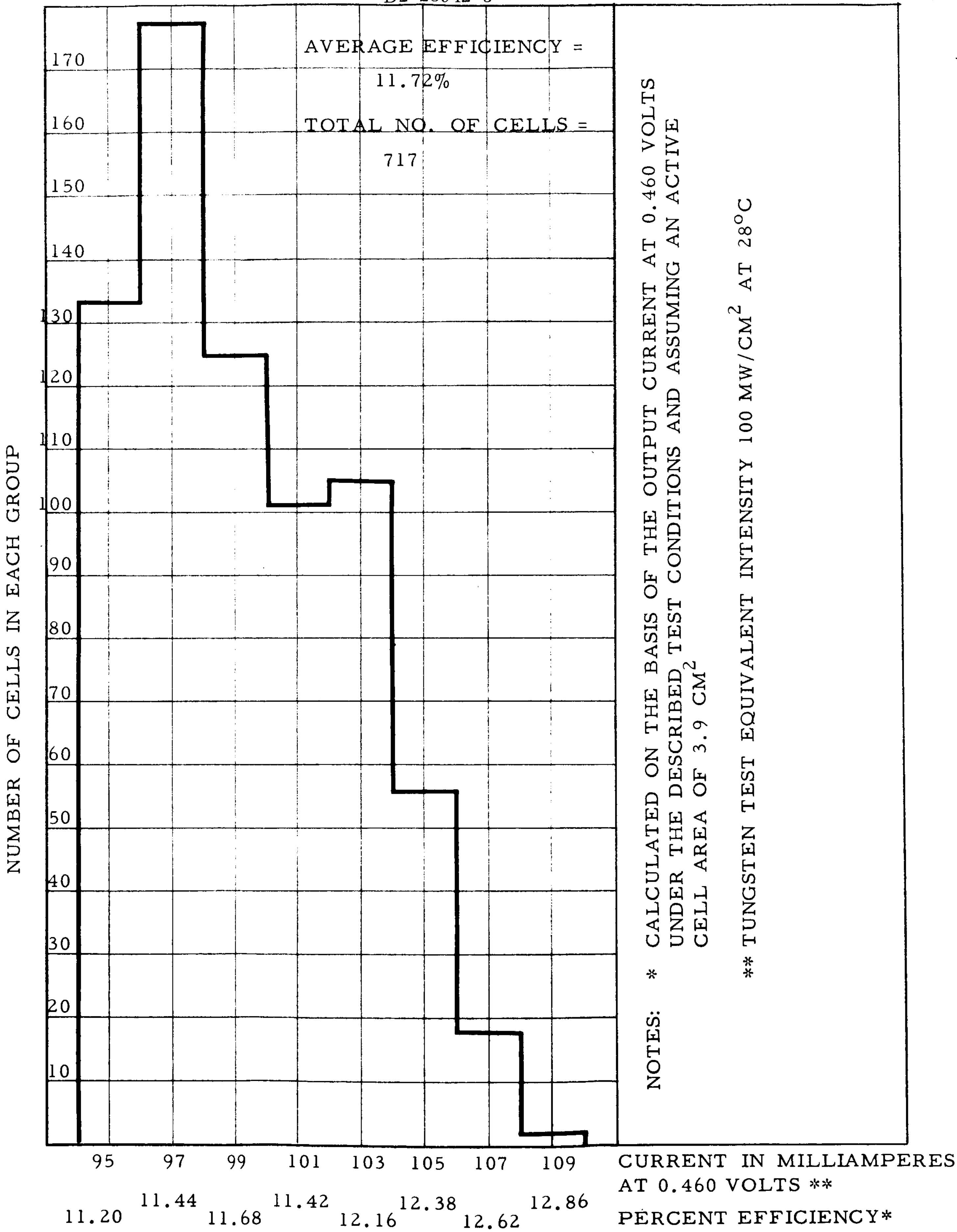
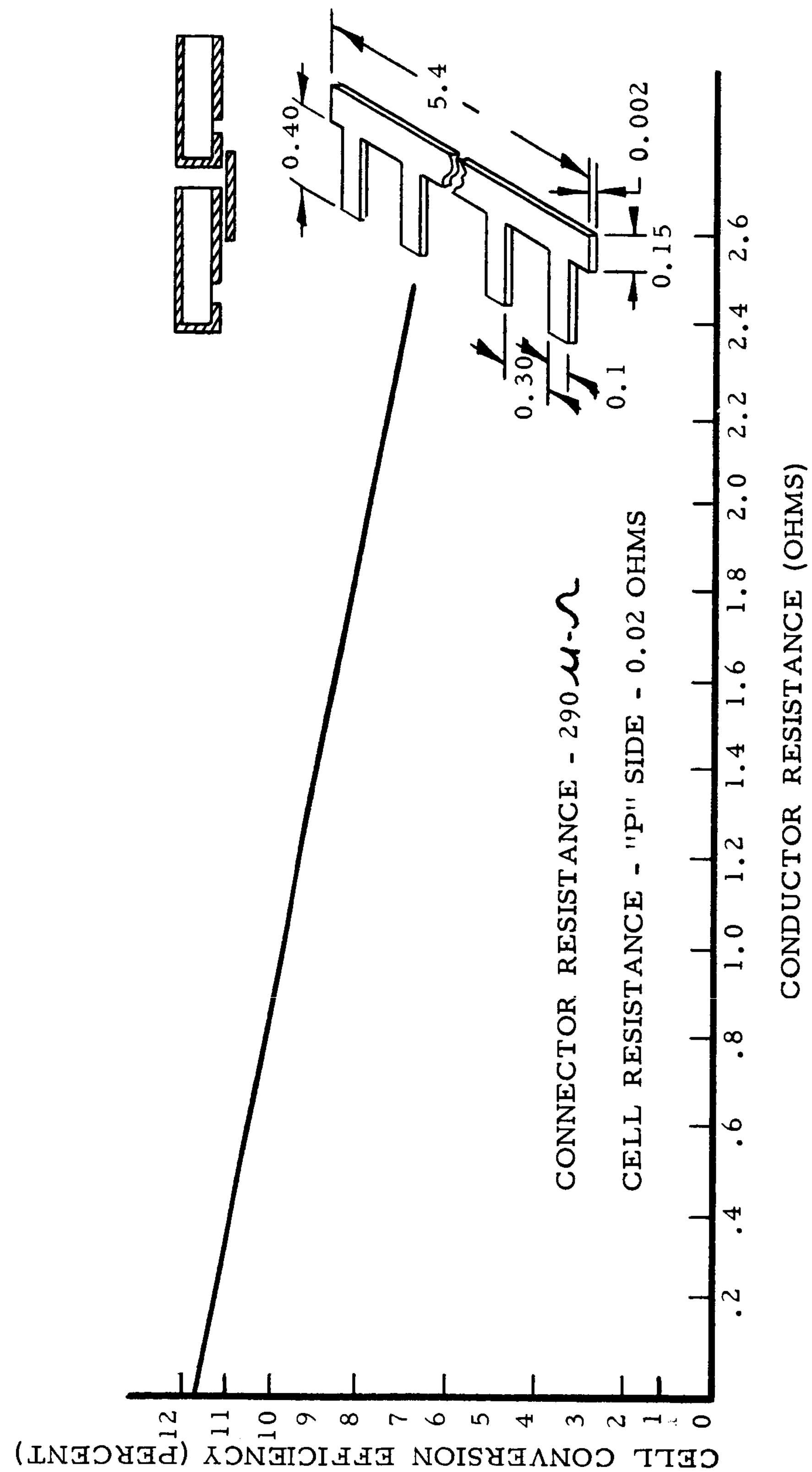
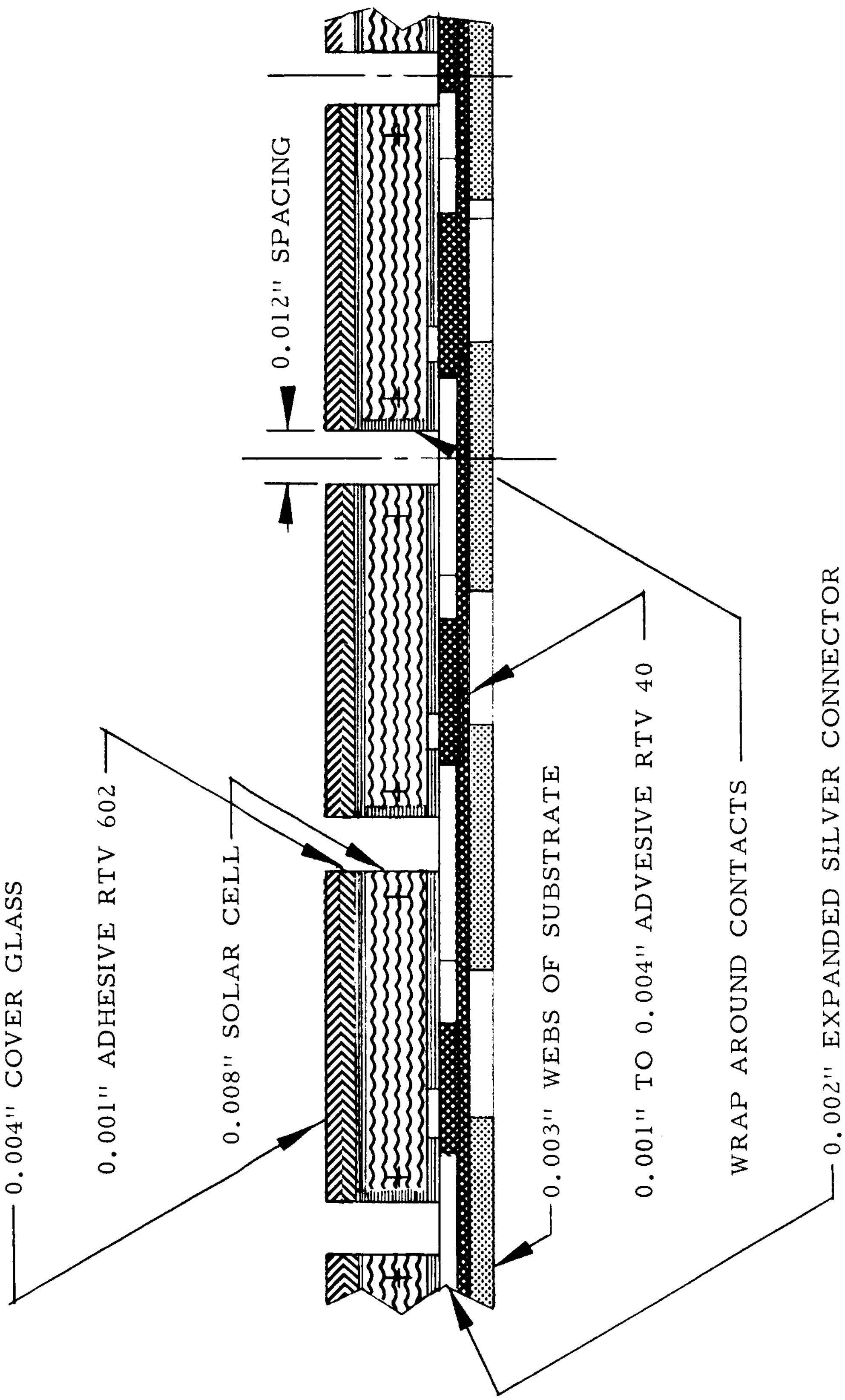


FIGURE 6.1.4-1
SOLAR CELL EFFICIENCY - SAMPLE PANELS

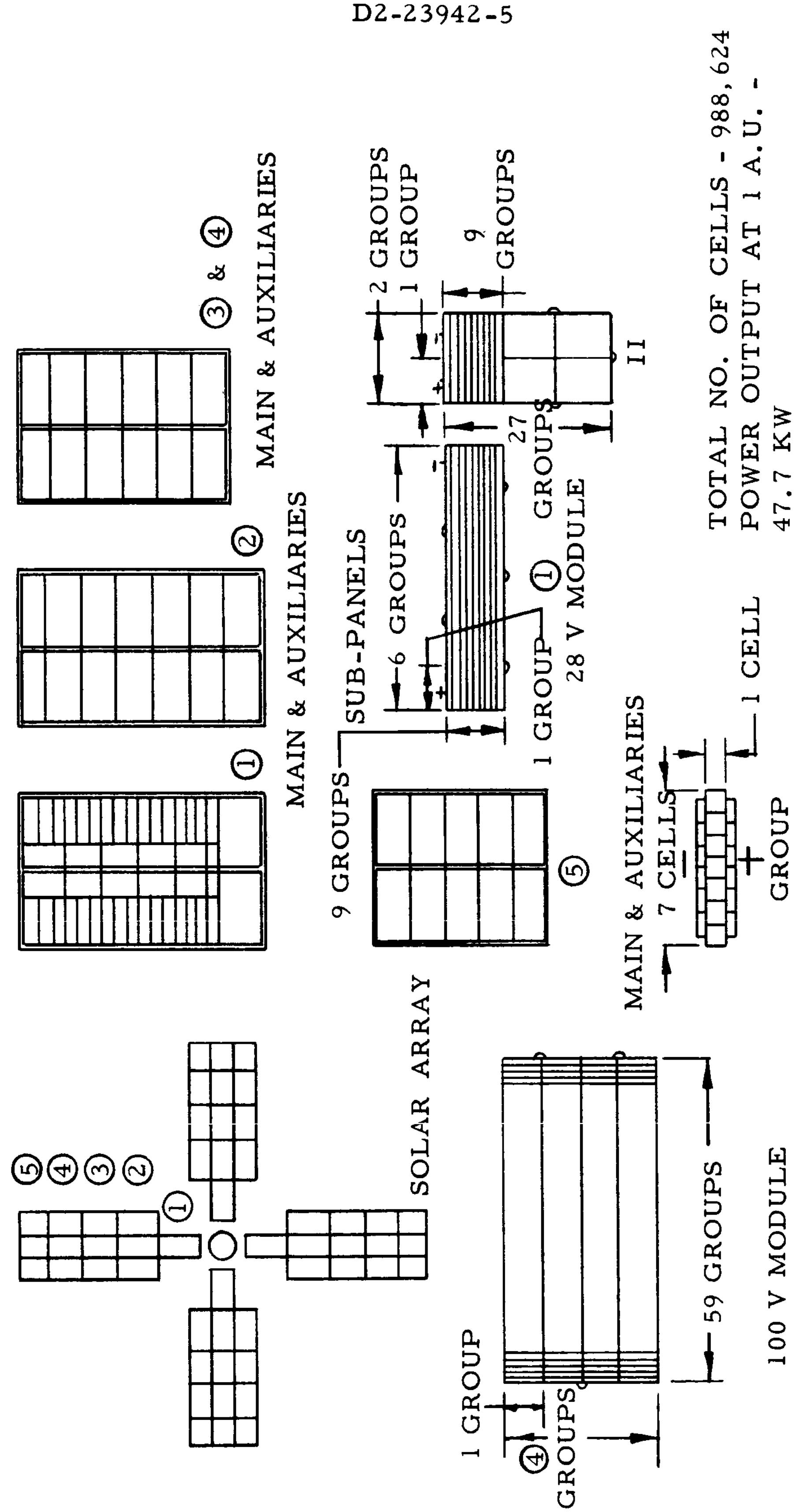


CELL CONVERSION EFFICIENCY
VS CONNECTOR RESISTANCE
FIGURE 6.1.4-2



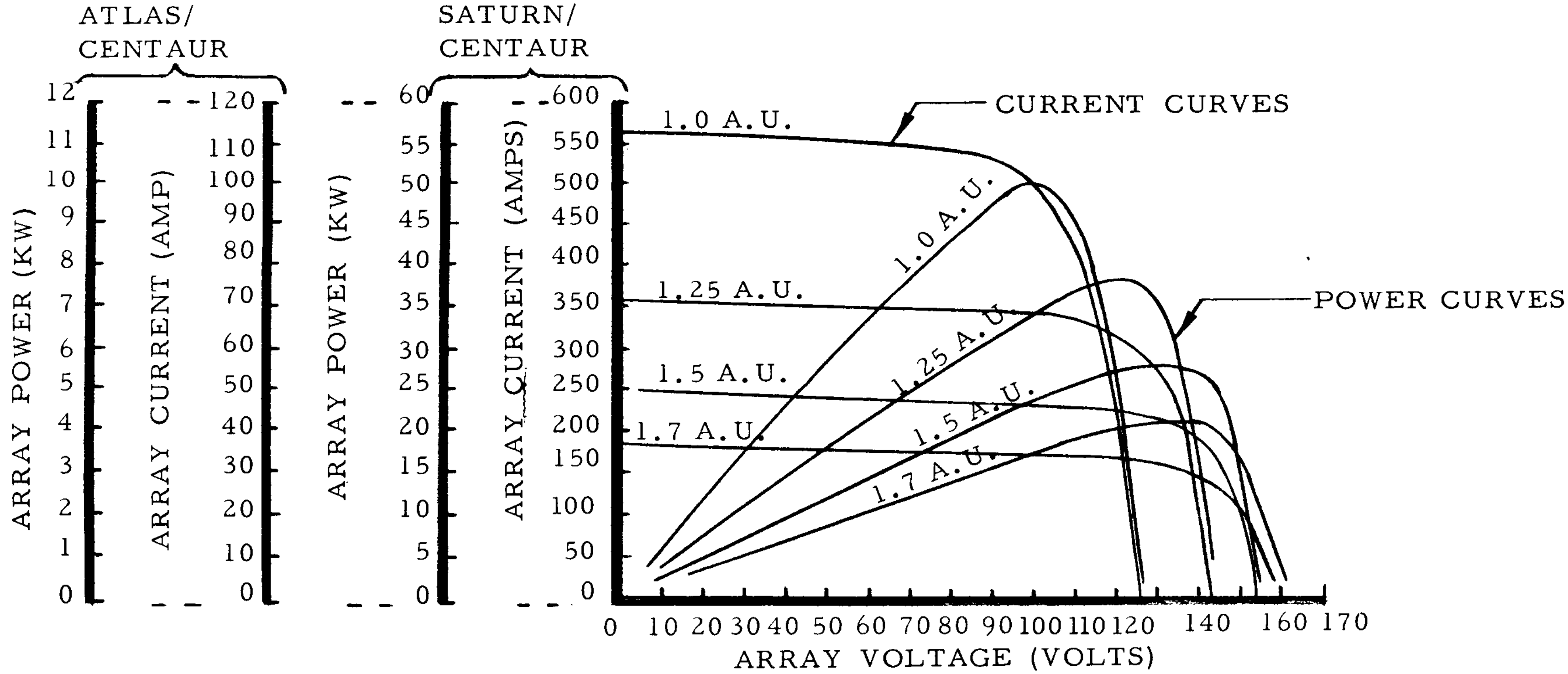
SOLAR CELL STACK

FIGURE 6.1.4-3



SOLAR CELL LAYOUT
SATURN IB/CENTAUR
ARRAY

FIGURE 6.1.4-4



$P = (\eta)(N)(A)(S)(K_1)(K_2)(K_C)(K_8)(K_9)(K_{10}) \quad 1-(T_C - 28)(K_7)$

η = 11.8% Air Mass 1

N = 988624 Cells - Saturn Centaur
209880 Cells - Atlas Centaur

A = 4.0 CM^2

K_1 = Air Mass 1 - Air Mass 0 = 0.85

K_2 = Cover Glass Loss 0.98

K_C = Process degradation mismatch - solar constant uncertainty calibration errors = 0.93

K_7 = Temperature coefficient = -0.45%/C^o

K_8 = Environmental Degradation = 1.0 - 1.0 A.U.
0.95 - 1.7 A.U.

K_9 = Misorientation = 1.0

K_{10} = Low Light Intensity 1.0 - 1.0 A.U.
0.95 - 1.7 A.U.

P = 47.7 KW Saturn/Centaur   P = 10.2 KW Atlas/Centaur

ARRAY ELECTRICAL
CHARACTERISTICS
FIGURE 6.1.4-5

6.1.5 Electrical Conductors

The conductors and interconnectors for the power, jettison, and command and monitor systems and their interfaces are defined in this section.

6.1.5.1 System Description - Power Busses (Figure 6.1.5-1)

6.1.5.2 General Configuration - 100 Volt System

The electrical wiring for each panel is symmetrical on the centerline of structural symmetry; that is, power busses proceed from outboard to inboard without crossing the centerline of the central sub-panels. This provides eight independent power sources up to the spacecraft central power panel. This concept evolved from structural constraints. However, it also offers a type of redundancy which improves mission reliability when considering possible electrical bus failures.

6.1.5.3 General Configuration - 28 Volt System

The 28 volt system is the same as the 100 volt system except for size, part of the single inboard sub-panel is used for a power source.

6.1.5.4 System Protection

The system is designed to minimize total loss of power from recognized failure modes. The following design characteristics have been used:

(a) Seven solar cells are connected in parallel.
(b) Each module (28 or 100 volt) feeds into the power system through two diodes, one in the positive lead and one in the negative lead.
(c) Dual wiring is provided between each diode and the sub-panel terminal.

6.1.5.5 Wiring Provisions

<u>Monitor - Panel Deployed and Latched</u> - Latching of the outboard sub-panel closes a switch circuit to give positive indication of panel deployment. Two hermetically sealed switches are mounted on the outboard support member of the central #4 sub-panel and are actuated by a striker mounted on the outboard sub-panel. Wiring consists of two #22 AWG which connect the switches to the spacecraft. Wire breaks are provided at the junctions of sub-panels #2 and #3, and at #3 and #4. The wire break between sub-panels #1 and #2 is at the jettison connector.

<u>Monitor - Jettison Accomplished</u> - Contacts number 44 and 45 of the outboard jettison connector are connected with a short wire loop. When the connector separates this circuit opens and the signal can be relayed to the ground monitor.

Sensors - Temperature sensors are shown on the inboard sub-panel and #2 central sub-panel. Wires for the sensors are routed through the jettison connectors to the #2 sub-panel.

Squibs - Squib wiring is not routed through the jettison connector. Wires are cut by guillotine action on electrical activation of the squib initiators.

Wires - All wire is to be in accordance with the forthcoming military specification for H-film insulated wire. All monitor wires are unshielded, twisted pair, size 22 AWG. Squib wires are shielded and jacketed to meet AFETR requirements. Size 18 AGW wire is selected for squib wiring.

6.1.5.6 Bus Crossovers

Bus crossovers are required to carry power across the hinge line of each sub-panel. The cross section and configuration of each part is determined by its physical location. Beryllium copper alloy was chosen because each crossover must have spring qualities as well as conductive capability. Conductivity of the alloy chosen is about 22 percent of copper, but the length is comparatively short and the cross section is chosen accordingly.

Insulation is not used on the bus crossovers for two reasons:

(a) The configuration makes it difficult, and
(b) Any added material would adversely affect the spring qualtities. Each crossover is firmly positioned by the mounting hole with two flats so that a small amount of H-film on adjacent structure will give adequate protection from shorts with minimum weight.

Bus crossover configurations are shown in Figures 6.1.5-2 and 6.1.5-3.

6.1.5.7 Wire Busses

All wire of size 16 and smaller will be copper conductor and larger sizes will be aluminum.

Each circuit member (plus or minus voltage) will be made of two wires. This accomplishes two purposes:

(a) Improves heat radiation, and
(b) Reduces electromagnetic field.

Electromagnetic fields are minimized by means of the H. A. Milloit patent (Perfection Mica Company - Licensee). The essence of this patent is a wire-weave which provides a field reduction of 12 to 20 db over twisted pair for low frequencies. There is no available information regarding the reduction for d-c fields, but it can be assumed to be approximately the same as low frequency alternating fields.

6.1.5.8 Copper Busses - Flat

Copper strips, .002 to .006 inch thick will connect each module to the nearest screw terminal. The strips are attached to the substrate on the side opposite the solar cells. All junctions and attachments are brazed, except for the junction with the solar cell which is soldered per Figure 6.1.5-4. Brazing is accomplished with a non-cadmium bearing, silver solder foil. The material selected is Eutectic Alloy Corporation No. 1800 alloy. Brazing is done with a resistance-weld hand tool. Low temperature solder is used at the solar cell junction to facilitate replacement of cells and to reduce temperature at cell contact surface.

0.002 inch tape is used for short runs and thicker tapes are used for long runs or for connecting two or more modules in parallel enroute to the terminal assembly.

The copper tapes are cemented to the glass fiber substrate. In the event that a negative and positive lead must cross each other, they will be separated by a 3 mil barrier of H-film. The positive and negative conductor tapes are routed on adjacent glass fiber tapes and, consequently, are 0.80 inch apart.

Wires are supported by molded nylon blocks which are cemented to the spars as shown in Figure 6.1.5-5. Self-binding nylon ties pass through holes in the support block and are cemented to prevent the possibility of a loose tie. The ties are expendable if rework of wire runs is required. Supports are mounted 10 to 14 inches apart.

6.1.5.9 Diode Installation - Sub-Panels

The solar cell connector used to connect the terminal groups of cells, plus and minus 28 volts, plus and minus 100 volts, has a 90° tab which projects into the dark side of the sub-panel where inter-module wiring is located. (See Figure 6.1.5-6) Diodes are soldered to the solar cell connector tab and to two copper tapes. The diode will be cemented to the glass fiber substrate and the exposed terminals will be completely covered with General Electric RTV-40 compound. The other termination of the dual copper tape is the terminal assembly shown in Figures 6.1.5-2 and 6.1.5-3.

6.1.5.10 Electrical Interface - Panel Jettison System

All panel jettison squib wiring terminates in an NAS 1599 type electrical connector as shown in Figure 6.1.5-7. Squib characteristics will be in agreement with safety provisions which are presently in effect at the Air Force Eastern Test Range (Cape Kennedy). The nominal current requirement for each bridge wire is five amperes. The maximum current is determined by the series resistance of each squib circuit and by the limiting characteristics of the squib firing circuits. The maximum current provided by the Boeing squib amplifier used on the Lunar Orbiting Spacecraft is 13 amperes and this value is proposed for the 20 watt per pound array. Squib power is required for 50 milliseconds, but the power peak should not exceed 5 milliseconds. Maximum power requirements for the maneuver to jettison all panels simultaneously would then be the product of the number of squibs per spacecraft (8), times the number of bridges per squib (2), times the peak current (13A), which equals 208 amperes for 5 milliseconds. The voltage for the squib system is assumed to be 28 volts d-c, nominal.

6.1.5.11 Electrical Interface - Command and Monitor (Figure 6.1.5-7)

A. Central Computer and Sequencer
   1. Stored Commands
      a. Deploy solar cell panels
      b. Jettison outer panels
   2. Ground Command
      Same as Stored Commands

B. Telecommunications - Monitor the following functions:
   1. Panel No. 1 Voltage and Current, 28 volts
   2. Same for Panels 2, 3, and 4.
   3. Panel No. 1, Voltage and Current 100 volts
   4. Same for Panels 2, 3, and 4
   5. Panel No. 1, Panel deployed and latched (Circuits A and B)
   6. Same for Panels 2, 3, and 4
   7. Panel No. 1 Jettison accomplished
   8. Same for Panels 2, 3, and 4
   9. Temperature Sensors - 6 per panel

   Total Number of Channels - 56

6.1.5.12 Electrical Interface - Motor Actuator

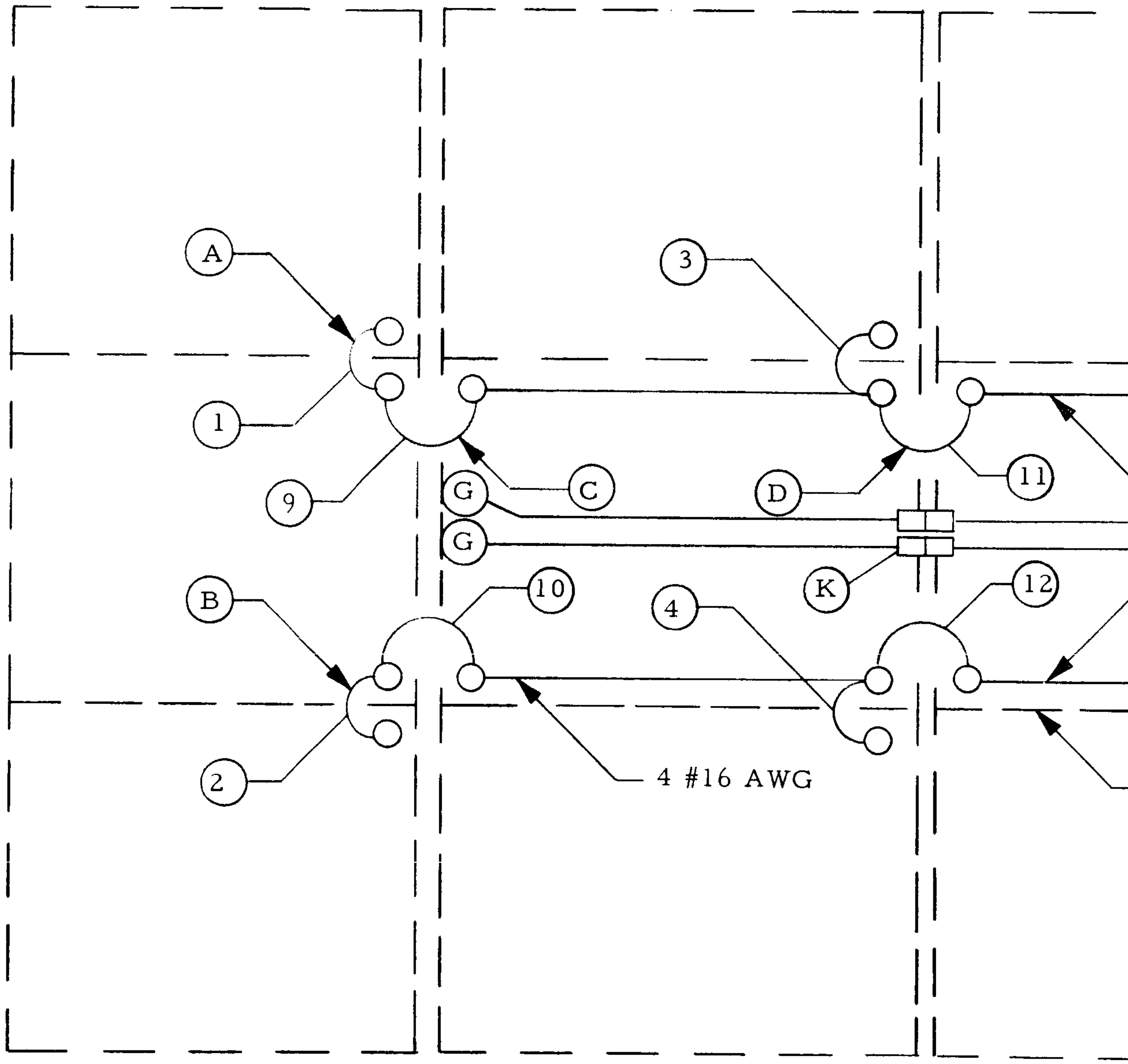
Each motor power input cable will terminate in an NAS 1599 type connector as shown on Figure 6.1.5-7. A six-contact arrangement has been selected to provide for design contingencies. The maximum power is estimated to be 12 watts per panel for a maximum of 15 minutes.

6.1.5.13 Electrical Interface - Power Conductors

Very little information is available for connector derating in a space environment. The four central contacts, see Figure 6.1.5-7, are not used in order to minimize heat transfer from the center of the insert to the outer shell. Two connectors with six contacts for the 100 volt bus and two contacts for the 28 volt bus will average 20 amperes per contact which gives a derating factor of 50 percent. This figure is believed to be conservative, however, choice of the next smaller size connector does not allow sufficient derating for a space environment.

6.1.5.14 Electrical Interface - Spacecraft Busses

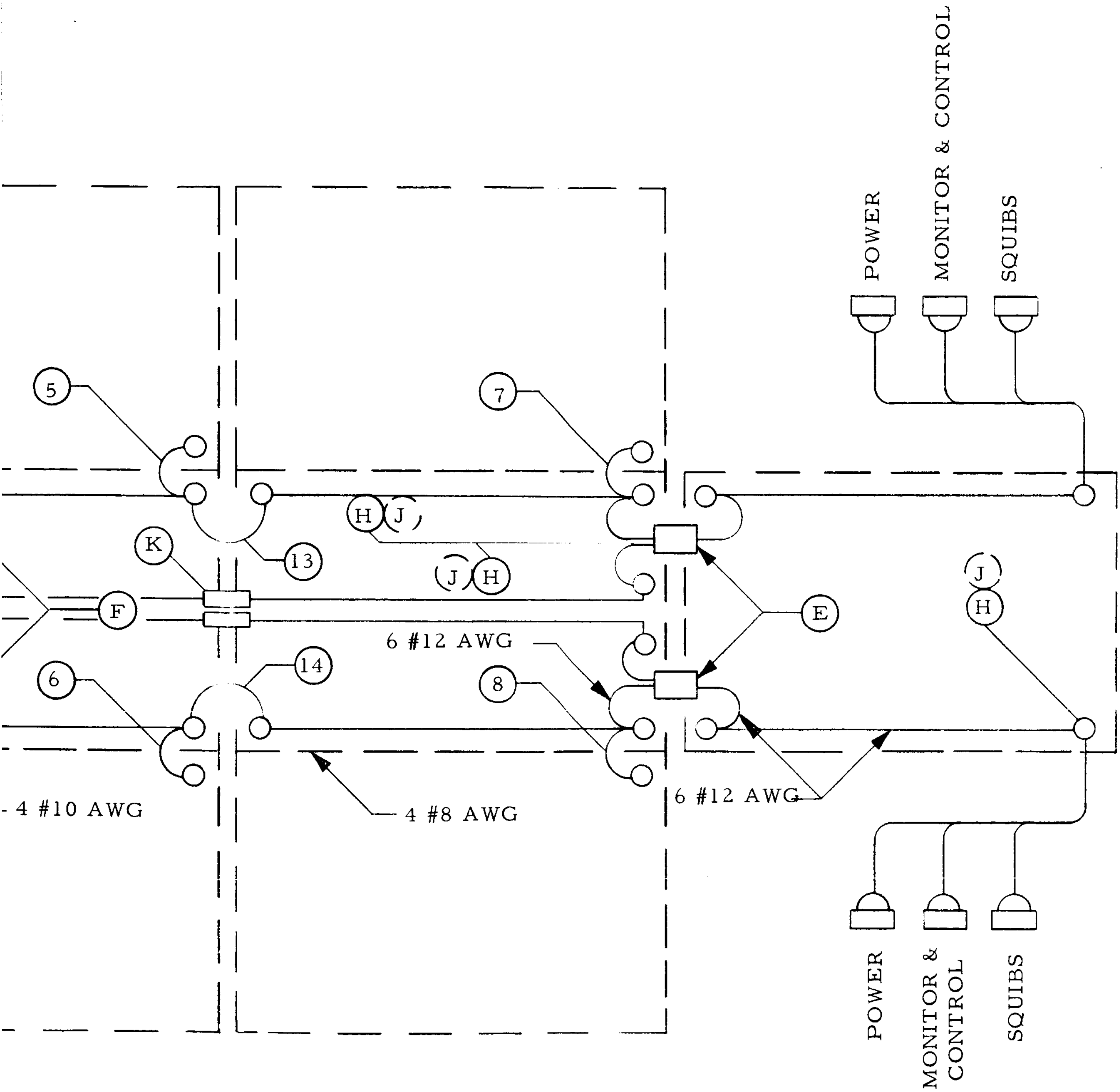
The essential elements of the array interface with spacecraft busses are shown in Figure 6.1.5-8. Shunts are shown in the input leads for each 28 volt and 100 volt panel. Monitoring for voltage and current at each shunt will provide a general indication of success for panel deployment and, in addition, will provide useful engineering data throughout the mission.



LEGEND

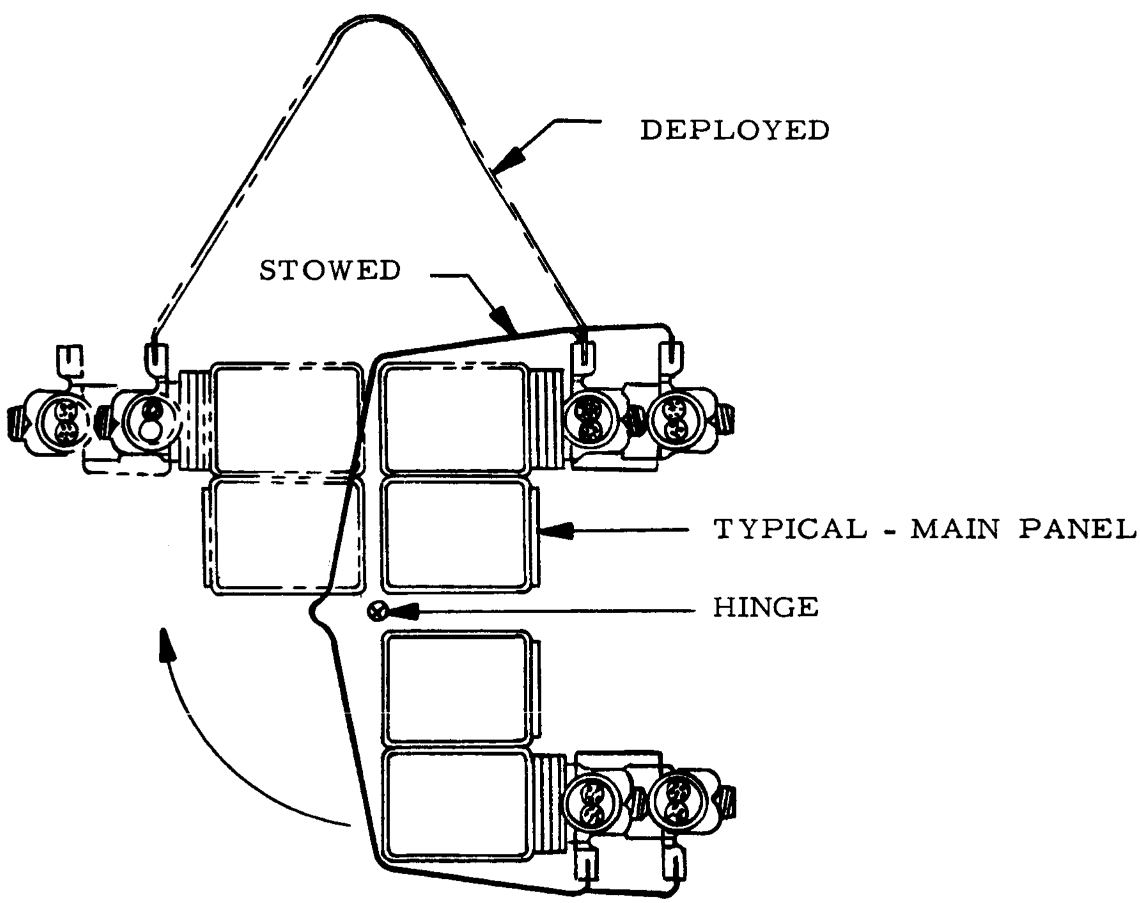
A	Bus Crossover - Inside Hinge Configuration (Figure 6.1.5-3)
B	Bus Crossover - Outside Hinge Configuration (Figure 6.1.5-2)
C	Bus Crossover - Main Panel to Main Panel
D	Bus Crossover - Main Panel to Main Panel
E	Jettison Connector
F	Power Bus (Cables
G	Panel Deployed and Latched
H	Temperature Sensor - Upper
J	Temperature Sensor - Lower
K	Connector - Panel Deployed and Latched

158



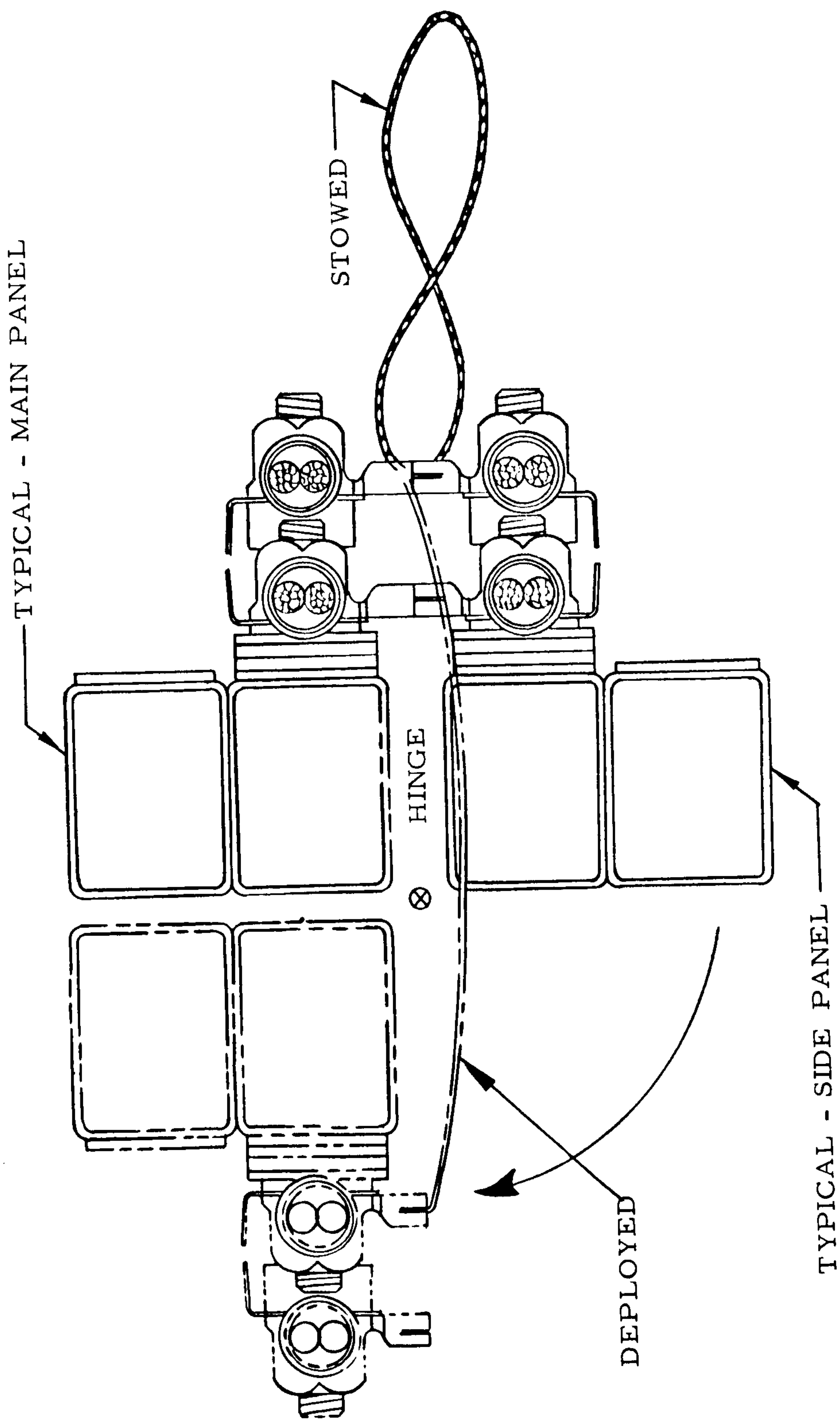
BUSSES AND CONDUCTORS PHYSICAL LOCATION AND IDENTIFICATION

FIGURE 6.1.5-1



BUS CROSSOVER - DETAIL
OUTSIDE HINGE

FIGURE 6.1.5-2



BUS CROSSOVER - DETAIL
INSIDE HINGE
FIGURE 6.1.5-3

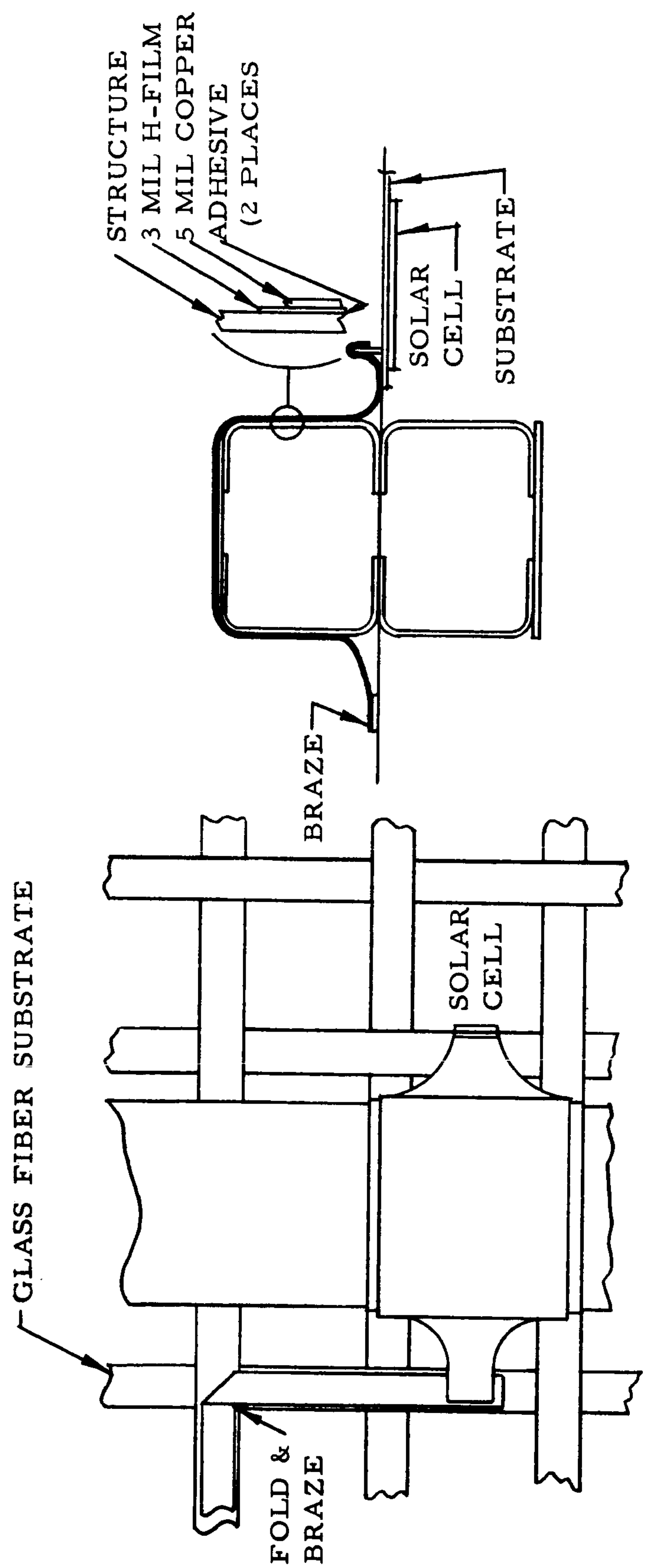
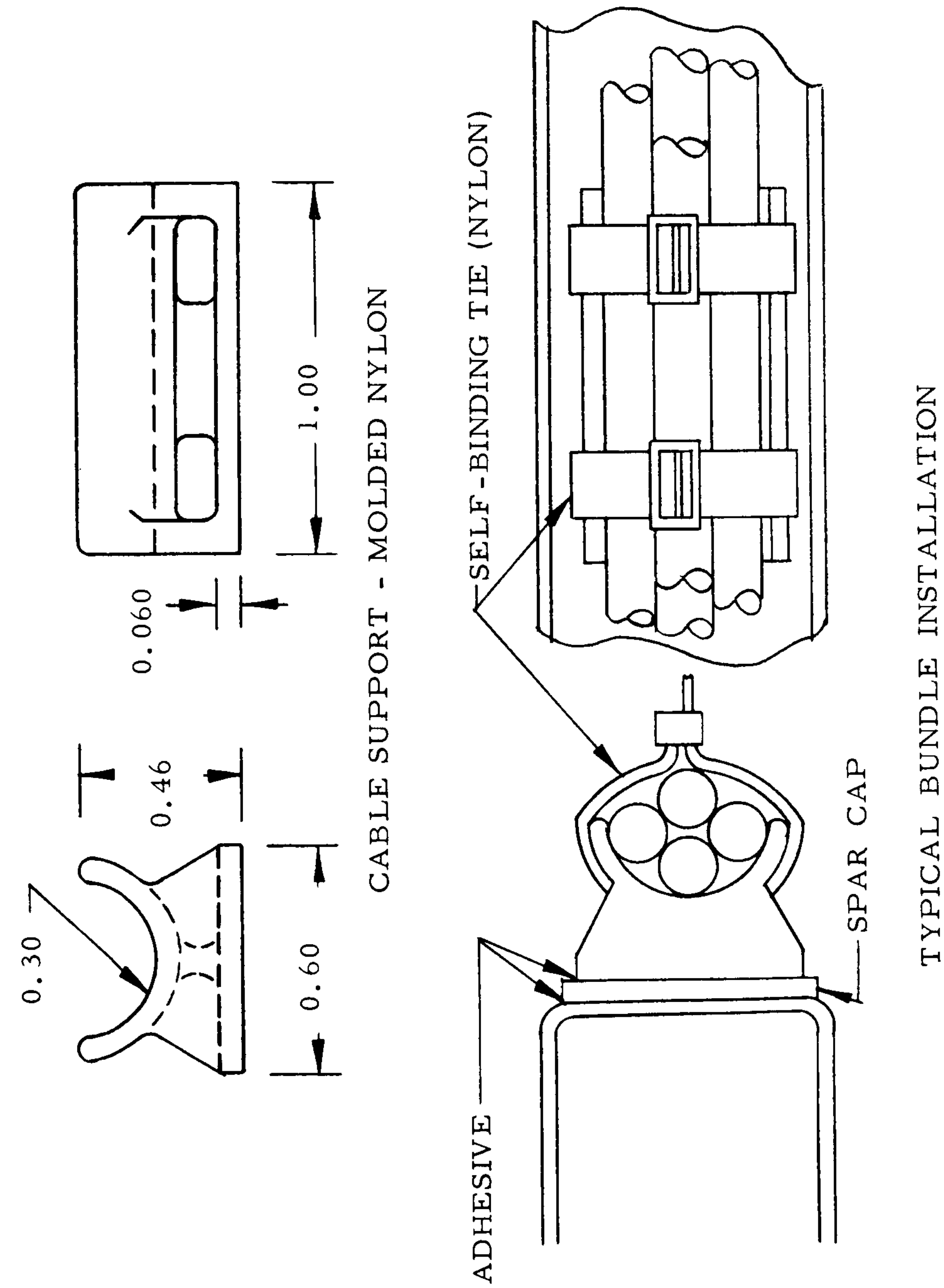
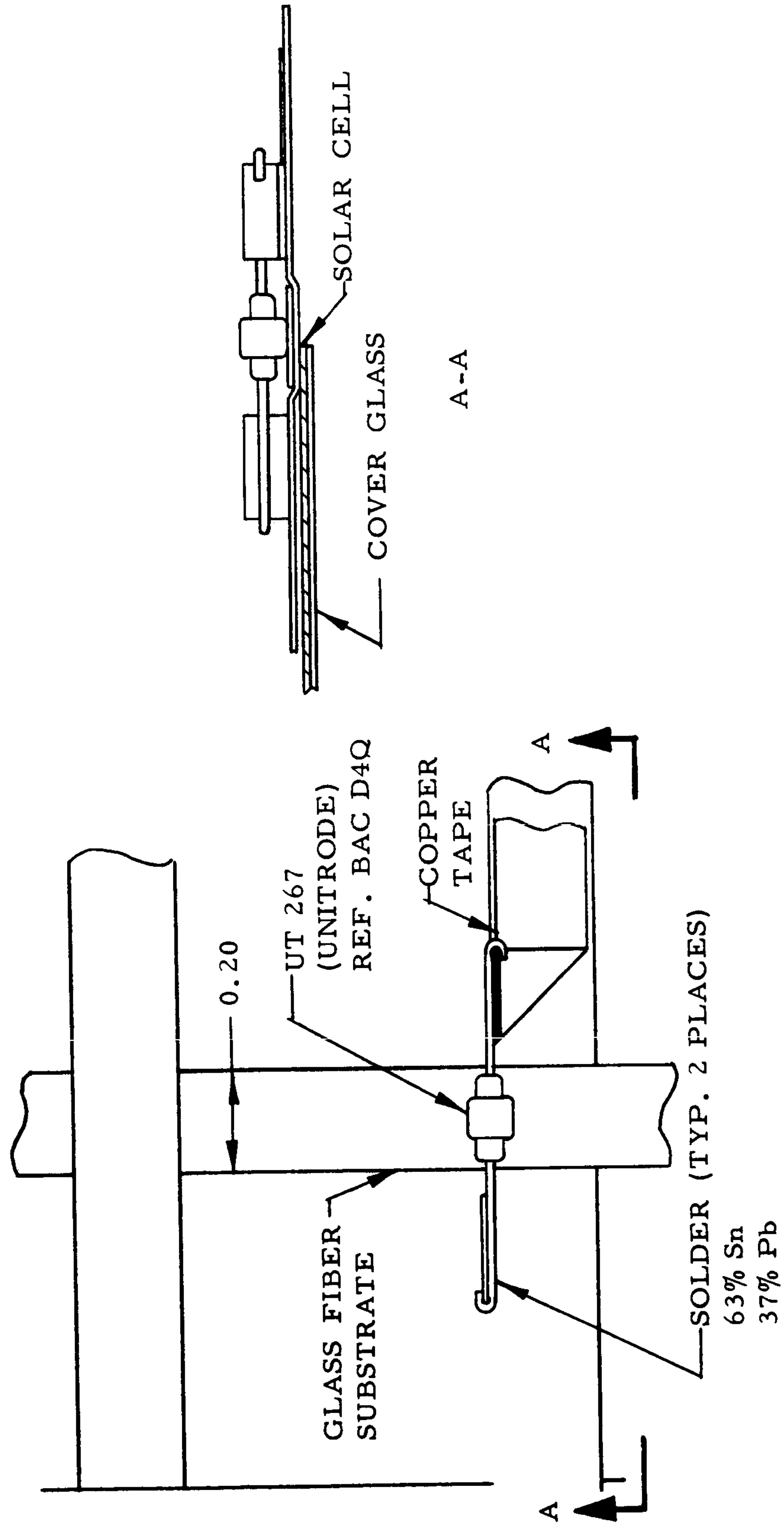


FIGURE 6.1.5-4
ELECTRICAL CROSSOVER AT
LATERAL SPAR - TYPICAL
INSTALLATION



WIRE BUNDLE SUPPORT

FIGURE 6.1.5-5



DIODE INSTALLATION -
SUB-PANEL SUBSTRATE

FIGURE 6.1.5-6

	CONTACT IDENTIFICATION
SQUIB NO. 1, CIRCUIT A NEG	A
SQUIB NO. 1, CIRCUIT A POS	B
SQUIB NO. 1, CIRCUIT B NEG	C
SQUIB NO. 1, CIRCUIT B POS	D
SQUIB NO. 2, CIRCUIT A NEG	E
SQUIB NO. 2, CIRCUIT A POS	F
SQUIB NO. 2, CIRCUIT B NEG	G
SQUIB NO. 2, CIRCUIT B POS	H
SHIELD - SQUIB NO. 1	J
SHIELD - SQUIB NO. 2	K

INTERFACE

CANNON ELECTRIC COMPANY

PV6R12B - 10 PNS
ELECTRIC PLUG

CONTROL SQUIB FIRING

	CONTACT IDENTIFICATION
SIGNAL - PANEL DEPLOYED AND LATCHED (CIRCUIT A)	A D
SIGNAL - PANEL DEPLOYED AND LATCHED (CIRCUIT B)	E H
JETTISON ACCOMPLISHED (CIRCUIT A)	J K
JETTISON ACCOMPLISHED (CIRCUIT B)	L M
SPARE	N S
SPARE	T W
SPARE	X PP

INTERFACE

CANNON ELECTRIC COMPANY

PV6R24B - 61 PNS

MONITOR AND CONTROL FUNCTION

THRU

THRU

THRU

THRU

THRU

	CONTACT IDENTIFICATION
MOTOR LEADS	A
	B
	C
	D
	E
MOTOR LEADS	F

INTERFACE

CANNON ELECTRIC CO.
PV6R10B - 6 PNS

POWER CONNECTOR -
ARRAY DEPLOY MOTOR

	CONTACT NUMBER
NOT USED	1
NOT USED	2
NOT USED	3
100 - VOLT BUS - NEG	4
100 - VOLT BUS - POS	5
100 - VOLT BUS - NEG	6
100 - VOLT BUS - POS	7
100 - VOLT BUS - NEG	8
100 - VOLT BUS - POS	9
28 - VOLT BUS - NEG	10
28 - VOLT BUS - POS	11
NOT USED	12

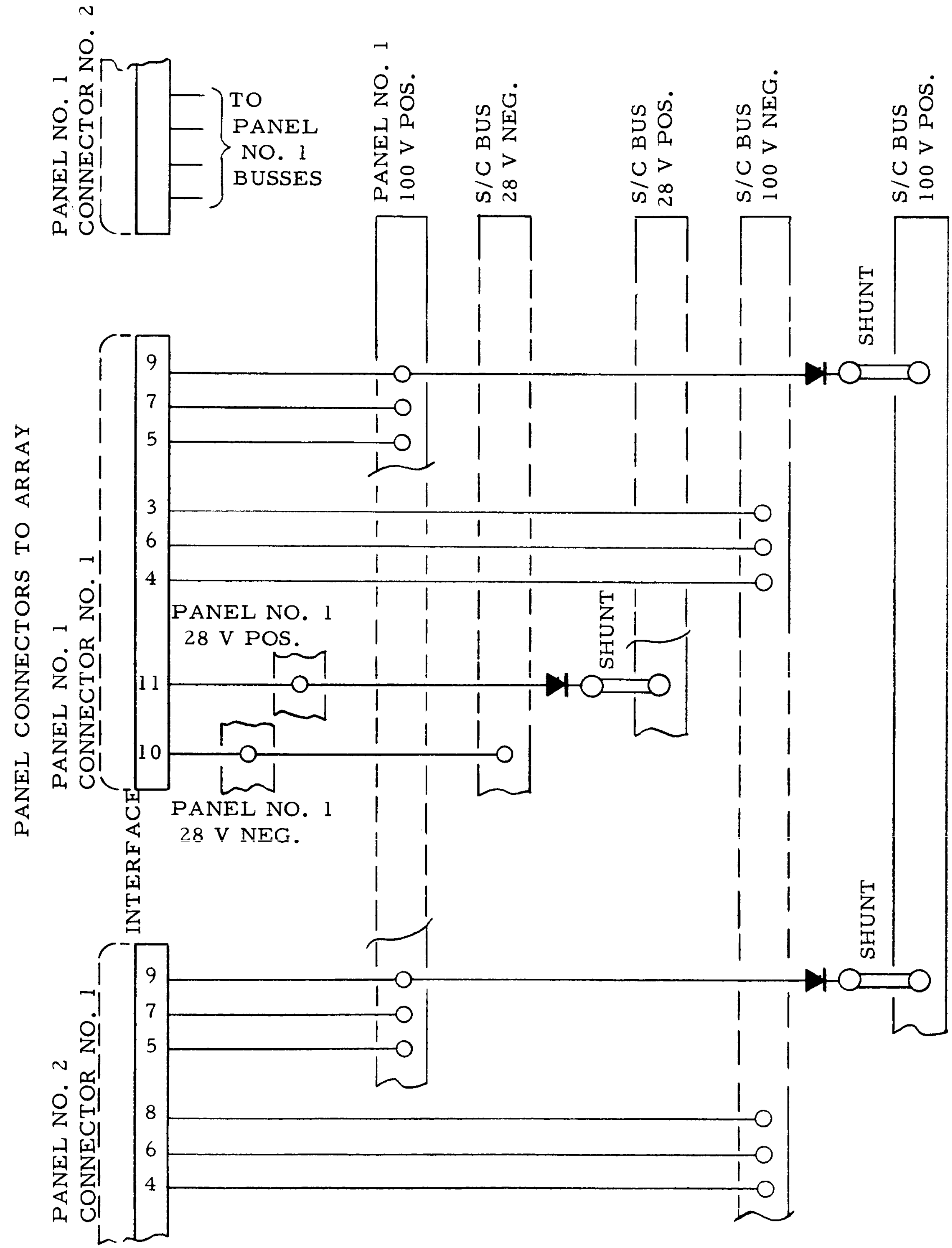
INTERFACE

CANNON ELECTRIC CO.
KV6R22T - 12 PNS
ELECTRIC PLUG

ELECTRICAL POWER

ELECTRICAL INTERFACE PLUG
SCHEMATIC & IDENTIFICATION

FIGURE 6.1.5-7



SUGGESTED ARRANGEMENT - ARRAY POWER TO S/C BUSSES

FIGURE 6.1.5-8

6.1.6 Dynamic Analysis

The Saturn IB/Centaur solar array has been analyzed to determine its response to the dynamic environment. The first mode resonant frequency of the stowed array was calculated to be 10.4 cycles per second. The maximum amplitudes of motion for each node are given in Table 6.1.6-2.

A single quadrant of the array was chosen for analysis. The stowed quadrant composed of 13 sub-panels in a stack was synthesized to an analogous frame by summing the member stiffnesses and the nodal masses. The response of this analogous frame to a sinusoidal vibration input at nodes 1, 3, 5, 26, and 30 was determined by matrix methods using stiffness matrices, mode shapes as generalized coordinates, and LaGranges equations of motion. Throughout the analysis a distributed damping coefficient of 3% of critical damping was assumed. This value is consistent with state-of-the-art structural bonding techniques and materials.

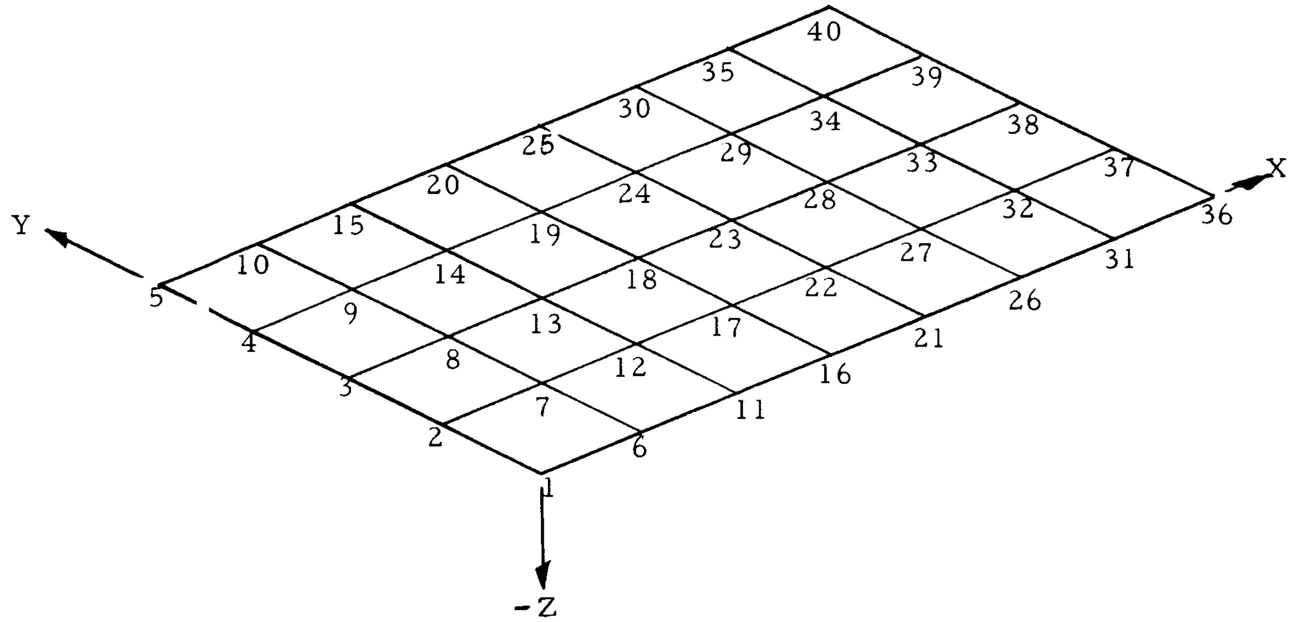
Results of this analysis are summarized as follows:

1. Acoustic Loads on Substrate and Solar Cells
   a. Acceleration -- 0.14 ft/sec^2, rms.
   b. Displacement -- 2.64×10^{-5} in.
2. Cruise Array, Bending
   a. First Mode -- between .064 and .088 cps
   b. Second Mode -- between .256 and 3.52 cps
3. Mars Orbit Array
   a. First Mode Bending -- 2.2 cps
   b. Second Mode bending -- 9 cps
4. Hinge Load During Deployment Sequence
   a. Maximum load on spacecraft hinge -- 1257 ft/lb/sec.
   b. Maximum load on panel hinge parallel to spacecraft hinge -- 380 ft/lb/sec.
   c. Maximum load on side panel hinge -- 21 ft/lb/sec.

   These loads assume:
   a. Rigid body
   b. Latch time of 0.1 second
   c. Force is normal to supports
5. Mass moment of inertia about spacecraft
   $I_h^{tot} = 23,705$ slug - ft^2

The responses to random loads have not been computed because these inputs have been band limited by the criteria to 200 cps to 2000 cps for the stowed array and 15 cps to 1500 cps for the deployed array. The resonant frequencies for these two arrays are sufficiently below the band limits so as to preclude significant responses.

The analogous frame is shown in Figure 6.1.6-1 and the response of the analogous frame to a 1.5 "g" rms sinusoidal vibration normal to the plane is given in Table 6.1.6-2. The δ_i^n matrix is given. The i symbol identifies the node and n identifies the input frequency. The response δ_i^n is the dynamic amplitude in inches and defines the dynamic envelope of the array quadrant relative to its rigid body motion. The complete dynamic analysis is given in bi-monthly report D2-23942-4, Appendix C.



IDEALIZED QUADRANT STRUCTURE

NODE	LUMPED WEIGHT - POUNDS
1, 6, 11, 16, 21, 26, 5, 10, 15, 20, 25, 30	12.252
31, 35	10.498
36, 40	7.630
37, 39	6.924
38	8.008
2, 3, 4	11.618
7, 9, 12, 14, 17, 19, 22, 24, 27, 29	19.933
32, 34	17.761
8, 13, 18, 23, 28	23.429
33	19.435

ANALOGOUS FRAME

FIGURE 6.1.6-1

NODE	f_1 10.4 cps	f_2 15.8 cps	f_3 22.5 cps	f_4 30.4 cps	f_5 35.8 cps	f_6 38.2 cps
SPACECRAFT RIGID BODY MOTION	.1925	.0836	.0410	.0224	.0162	.0142
2	.3581	.3674	.3706	.3698	.3708	.3717
4	.3709	.3748	.3727	.3716	.3706	.3689
6	.2791	.3436	.3604	.3704	.3737	.3765
7	.2468	.3485	.3660	.3718	.3819	.3806
8	.3033	.3805	.3788	.3755	.3787	.3722
9	.3207	.3926	.3755	.3779	.3800	.3679
10	.3412	.3832	.3637	.3686	.3714	.3710
11	.2384	.3266	.3546	.3705	.3722	.3783
12	.1900	.3440	.3704	.3736	.3817	.3817
13	.2606	.3946	.3918	.3790	.3813	.3720
14	.2940	.4081	.3801	.3769	.3785	.3678
15	.3426	.3944	.3578	.3639	.3681	.3717
16	.2757	.3353	.3588	.3719	.3683	.3742
17	.2092	.3599	.3837	.3744	.3714	.3750
18	.2487	.4044	.4026	.3787	.3750	.3700
19	.2958	.4154	.3876	.3708	.3682	.3681
20	.3686	.3980	.3575	.3599	.3641	.3727
21	.3513	.3595	.3674	.3730	.3670	.3690
22	.2637	.3805	.3953	.3741	.3625	.3680
23	.2494	.4015	.4054	.3758	.3681	.3682
24	.3046	.4086	.3939	.3676	.3606	.3682
25	.3935	.3902	.3627	.3615	.3646	.3728
27	.2717	.3768	.3895	.3614	.3661	.3698
28	.2295	.3828	.3995	.3729	.3646	.3686
29	.2786	.3808	.3908	.3725	.3662	.3684
31	.2871	.3537	.3652	.3631	.3750	.3791
32	.1751	.3481	.3791	.3627	.3683	.3794
33	.1656	.3539	.3912	.3722	.3671	.3711
34	.1927	.3524	.3928	.3812	.3694	.3638
35	.2822	.3437	.3777	.3823	.3769	.3656
36	.1722	.3266	.3580	.3568	.3802	.3885
37	.1022	.3239	.3731	.3641	.3755	.3822
38	.7799	.3221	.3835	.3732	.3736	.3746
39	.1068	.3197	.3865	.3835	.3771	.3675
40	.1735	.3145	.3819	.3924	.3834	.3623

Dynamic response (δ) in inches. (Total spatial motion would be equal to the dynamic response plus rigid body motion.)

TABLE 6.1.6-2

6.1.7 Stress Analysis

6.1.7.1 Summary

The watt per pound performance of the solar array is directly dependent upon the structural weight which in turn depends upon the stress levels in the structural design. Since the watt per pound performance of the array was a critical item in the feasibility study, the structural design and analyses were much more detailed than those normally required for feasibility studies.

The stress analysis is based upon the results of the detailed dynamic analysis previously reported in the fourth bi-monthly progress report and summarized in Section 6.1.6 of this report.

Results of the stress analysis indicate some areas of the quadrant in which bending stresses are excessive due to vibration normal to the quadrant plane. The high stresses in these areas can be eliminated by one of several suggested alternate methods without weight increase. The stress analysis also indicates the quadrant is seriously under strength for the in plane shear load condition. The requisite modification to alleviate these high stresses can be accomplished by several suggested methods with at most a weight increase to the total array of 80 pounds.

The most satisfactory solution to the high stress problem for in plane shear loading could be determined by a trade study conducted during any subsequent contract culminating in a Type Approval solar array for the Saturn IB/Centaur.

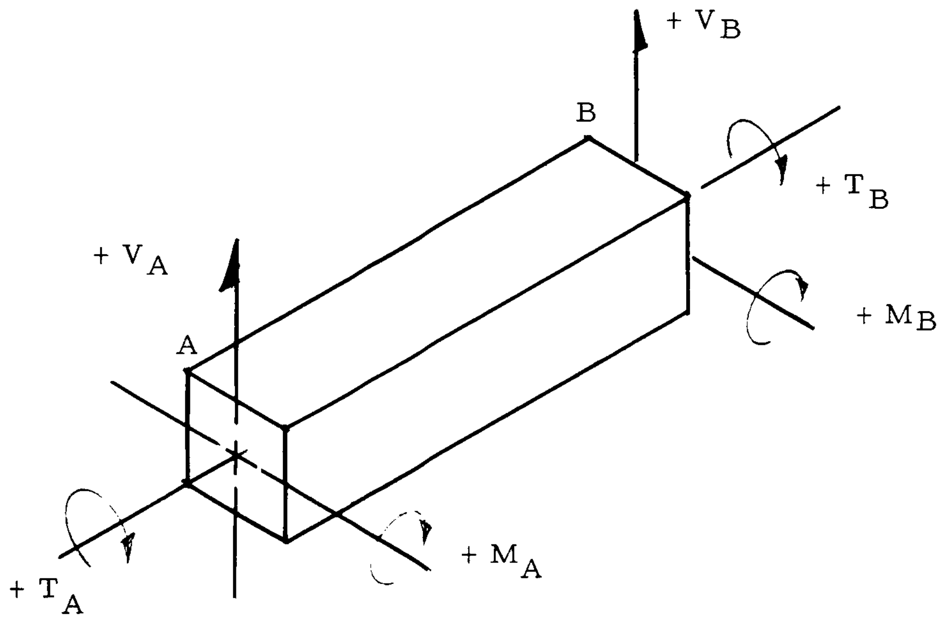
6.1.7.2 Analysis Assumptions

The analysis is predicated upon the following assumptions:

(a) The distributed structural damping coefficient is 3% of critical.
(b) The quadrant is pin supported at the forward attachment points by the adjacent quadrants.
(c) The critical loading condition is vibration at the first mode resonant frequency normal to the plane of the quadrant.
(d) The summation of stiffnesses and masses for the members can be used to form an analogous frame for analysis purposes.
(e) Coupling between normal and in plane vibration modes does not occur.

6.1.7.3 Analysis

Based upon the above assumptions, the first mode resonant frequency of the stowed array was calculated to be 10.4 cps. The maximum amplitudes of motion for each node are given in Table 6.1.6-2. The bending moments, torsions and shear forces in the analogous frame were obtained through a matrix algebra technique using the dynamic response amplitudes and the stiffness matrix to obtain a fictitious constraint load matrix which was subsequently used in a multiple beam analysis program to obtain the loads for each member of the analogous frame. The sign convention for moments, torsions, and shears in shown in the sketch.



6.1.7.4 Normal Loading

The bending and shear stress have been calculated for each member in the analogous frame for the loads induced by a 1.5 "g" rms sinusoidal vibration normal to the plane of the frame and oscillating at 10.4 cps. The input vibration was imposed at the three main hinges at the aft end of the frame and also at the two forward attachment points. Although dynamic loads were computed for the first three mode frequencies, stresses and margins of safety (M.S.) were computed only for the fundamental frequency. (See Tables 6.1.7-1 and -2)

Twenty-four of the sixty-seven members of the frame have negative margins of safety under this load condition. Of these twenty-four over stressed members, twenty are grossly over stressed and must be redesigned. There are thirty-two grossly understressed members in the frame. Reduction of gage in the under stressed members coupled with increase in gage of the over stressed members can result in safe margins for the redesigned frame without weight increase.

The present design of this array provides bumper pads to allow one sub-panel in the total stack to slide, in the longitudinal direction, relative to the adjacent sub-panels. The dynamic and stress analysis conducted was predicated upon the capability of the shear slippage between these elements. Providing a shear transfer system between adjacent sub-panels in the stack will result in more efficient utilization of the structural material and increase the fundamental frequency from 10.4 cps to 105 cps and will reduce the amplitude from 0.39 inches to 0.004 inches. The outermost elements in the stack would be the principal bending material and the inner elements would be primarily shear material.

As a result of this modification to the shear transfer characteristic between the elements of the stacks, the total weight of the array quadrant will decrease. A new critical load condition will develop if the inner elements in the stack are reduced to minimal sections; that load condition is buckling for the sub-panel frames and intercostals due to the substrate membrane tension load.

Another structural modification which will directly affect the dynamic response of the quadrant is multiplicity of support ties between adjacent quadrants instead of the single tie locations used in the analysis.

Incorporation of all three modifications (member resizing, shear transfer, multiple supports) would reduce the total array weight significantly thus increasing the specific performance in terms of watts per pound.

In summary, although some members in the sub-panel structural assemblies are seriously under strength, the structural capability of the array can be readily increased without increasing the total array weight.

6.1.7.5 In Plane Loading

For in plane loading, the critical condition is assumed to be a static load of 1.121 "g" with the dynamic reaction of the adjacent quadrants imposed at the forward attachment points. The analogous frame and member I's are shown in Figure 6.1.7-3.

The resultant 23,580 pound load at the forward quadrant attachments is the sum of the static load on the frame plus the dynamic reaction of the adjacent frames vibrating at 10.4 cps.

The analogous frame is converted to an analogous bent by summing the I/L values story by story, for one half the frame. The analogous bent is then:

STRUCTURE SY

$F_S = K_s\,(t/b)^2 \leq 20$ KSI $\qquad F_b = K_b\,E\,(t/b)^2 \leq 38$ KSI $\qquad f_{s\,\tau} = \quad \tau\, d/$

$K_s = 8.11$/WEBS $\quad K_s = 4.84$/CAPS $\quad K_b = 21.6$/WEBS $\quad K_b = 3.61$/CAPS

MEMBER		d/J $(IN)^{-3}$	d/J $(IN)^{-3}$	C/I $(IN)^{-3}$	Q/Ib $(IN)^{-2}$	τ A #/IN	M IN#	M IN#	V #	V #
A	B	WEBS	CAPS	CAPS	WEBS	A&B	A	B	A	
1	6	.14	.21	1.26	.63	16,900	-957	+27,750	-1,218	+1
6	11	.14	.21	1.26	.63	9,974	-27,470	-8,910	+1,654	-1
11	16	.14	.21	1.26	.63	4,213	+8,898	-16,827	+360	
16	21	.14	.21	1.26	.63	1,061	+17,103	+64,464	-3,708	+3
21	26	.14	.21	1.26	.63	6,754	-65,090	-105,223	+7,742	-7
26	31	.088	.13	.63	.95	18,761	+105,552	+92,643	-9,010	+9
31	36	.088	.13	.63	.95	14,411	-92,103	-6,213	+4,469	-4
3	8	.088	.13	.63	.95	1,600	+675	+61,100	-2,810	+2
8	13	.088	.13	.63	.95	771	-60,590	-20,776	+3,698	-3
13	18	.088	.13	.63	.95	494	+20,715	+2,630	-1,061	+1
18	23	.088	.13	.63	.95	1,429	-3,114	+1,122	+90	
23	28	.088	.13	.63	.95	1,677	-366	+7,131	-307	
28	33	.088	.13	.63	.95	170	-7,471	-1,344	+401	
33	38	.088	.13	.63	.95	462	+757	+11,920	-576	
2	7	5.30	4.98	20.58	82.0	119	+734	-500	+56	
7	12	5.30	4.98	20.58	82.0	45	-99	-39	+6	
12	17	5.30	4.98	20.58	82.0	55	+66	-39	-1	
17	22	5.30	4.98	20.58	82.0	91	+49	+173	-10	
22	27	5.30	4.98	20.58	82.0	490	+136	+67	-9	
27	32	5.30	4.98	20.58	82.0	456	-210	-210	+19	
32	37	5.30	4.98	20.58	82.0	91	-39	+87	-2	
1	2	.14	.21	1.26	.63	957	+16,900	+957	-3,427	+3
2	3	.14	.21	1.26	.63	1,691	-68,900	+1,691	+5,986	-5
6	7	3.34	3.14	10.60	4.24	280	-6,926	-2,830	+390	
7	8	3.34	3.14	10.60	4.24	319	+2,900	+1,922	-193	
11	12	3.34	3.14	10.60	4.24	11	-5,761	-2,782	+342	
12	13	3.34	3.14	10.60	4.24	16	+2,882	+2,051	-197	
16	17	3.34	3.14	10.60	4.24	275	-5,274	-2,440	+309	
17	18	3.34	3.14	10.60	4.24	285	+2,476	+1,432	-156	
21	22	3.34	3.14	10.60	4.24	627	-5,692	-2,359	+322	
22	23	3.34	3.14	10.60	4.24	317	+1,777	+415	-88	

MMETRIC ABOUT MEMBER 3--8--13--18--23--28--33--38

J $f_{sv} = \frac{VQ}{Ib}$ $f_b = \frac{Mc}{I}$ $\Sigma f_s = f_{s\tau} + f_{sv}$ $R_s = \Sigma f_s / F_s$ $R_b = f_b / F_b$ $M.S. = \frac{1}{\sqrt{R_s^2 + R_b^2}} - 1$

	$f_{s\tau}$		f_{sv}	Σf_s		f_b		F_s		F_b	
	PSI	PSI	PSI	PSI	PSI	PSI	PSI	PSI	PSI	PSI	PSI
	WEBS	CAPS	WEBS	WEBS	CAPS	A	B	WEBS	CAPS	WEBS	CAPS
,218	2,366	3,549	767	3,133	3,549	1,206	34,965	20,000	20,000	38,000	38,000
,654	1,396	2,094	1,042	2,438	2,094	34,612	11,227	20,000	20,000	38,000	38,000
-360	590	885	227	817	885	11,211	21,202	20,000	20,000	38,000	38,000
,708	149	223	2,336	2,485	223	21,550	81,225	20,000	20,000	38,000	38,000
,742	945	1,418	4,877	5,822	1,418	82,013	132,581	20,000	20,000	38,000	38,000
,010	1,651	2,439	8,560	10,211	2,439	66,498	58,365	20,000	20,000	38,000	38,000
,469	1,268	1,873	4,246	5,514	1,873	58,025	3,914	20,000	20,000	38,000	38,000
,810	141	208	2,670	2,811	208	425	38,493	20,000	20,000	38,000	38,000
,698	68	100	3,513	3,581	100	38,172	13,089	20,000	20,000	38,000	38,000
,061	43	64	1,008	1,051	64	13,050	1,657	20,000	20,000	38,000	38,000
-90	126	186	86	212	186	1,962	707	20,000	20,000	38,000	38,000
+307	148	218	292	440	218	231	4,493	20,000	20,000	38,000	38,000
-401	15	22	381	396	22	4,707	847	20,000	20,000	38,000	38,000
+576	41	60	547	588	60	477	7,510	20,000	20,000	38,000	38,000
-56	631	593	4,592	5,223	593	15,106	10,290	20,000	20,000	38,000	38,000
-6	239	224	492	731	224	2,037	803	20,000	20,000	38,000	38,000
+1	292	274	82	374	274	1,358	803	20,000	20,000	38,000	38,000
+10	482	453	820	1,302	453	1,008	3,560	20,000	20,000	38,000	38,000
+9	2,597	2,440	738	3,335	2,440	2,799	1,379	20,000	20,000	38,000	38,000
-19	2,417	2,271	1,558	3,975	2,271	4,322	4,322	20,000	20,000	38,000	38,000
+2	482	453	164	646	453	803	1,790	20,000	20,000	38,000	38,000
,427	134	201	2,159	2,293	201	21,294	1,206	20,000	20,000	38,000	38,000
,986	237	355	3,771	4,008	355	86,814	2,131	20,000	20,000	38,000	38,000
-390	935	879	1,654	2,589	879	73,416	29,998	20,000	20,000	38,000	38,000
-193	1,065	1,002	818	1,883	1,002	30,740	20,373	20,000	20,000	38,000	38,000
-342	37	35	1,450	1,487	35	61,067	29,489	20,000	20,000	38,000	38,000
+197	53	50	835	888	50	30,549	21,741	20,000	20,000	38,000	38,000
-309	919	864	1,310	2,229	864	55,904	25,864	20,000	20,000	38,000	38,000
-156	952	895	661	1,613	895	26,246	15,179	20,000	20,000	38,000	38,000
-322	2,094	1,969	1,365	3,459	1,969	60,335	25,005	20,000	20,000	38,000	38,000
+88	1,059	995	373	1,432	995	18,836	4,399	20,000	20,000	38,000	38,000

STRESS ANALYSIS OF
SATURN CENTAUR
SOLAR ARRAY QUADRANT

TABLE 6.1.7-1

MEMBER		d/J		C/I	Q/I, b	A	M		V	
A	B	WEBS	CAPS	CAPS	WEBS	A&B	A	B	A	B
26	27	.088	.13	.63	.95	329	+25,515	+55,767	-3,251	+3,25
27	28	.088	.13	.63	.95	186	-54,821	-39.813	+3,785	-3,78
31	32	3.34	3.14	10.60	4.24	540	-4,350	-2,273	+265	-26
32	33	3.34	3.14	10.60	4.24	291	+1,726	+622	-94	+9
36	37	.088	.13	.63	.95	6,213	-14,411	-8,578	+920	-92
37	38	.088	.13	.63	.95	6,126	+8,670	-2,673	-240	+24

MEMBER		R_s WEBS	R_s CAPS	R_b WEBS	R_b CAPS	M.S. WEBS	M.S. CAPS
1	6	:16	.18	.92	.92	+.071	+.067
6	11	.12	.10	.91	.91	+.089	+.092
11	16	.04	.04	.56	.56	+.781	+.781
16	21	.12	.01	2.14	2.14	-.534	-.532
21	26	.29	.07	3.49	3.49	-.714	-.713
26	31	.51	.12	1.75	1.75	-.451	-.430
31	36	.28	.09	1.53	1.53	-.358	-.348
3	8	.14	.01	1.01	1.01	-.019	-.010
8	13	.18	-	1.00	1.00	-.016	0
13	18	.05	-	.34	.34	+1.910	+1.941
18	23	.01	.01	.05	.05	+18.612	+18.612
23	28	.02	.01	.12	.12	+7.217	+7.217
28	33	.02	-	.12	.12	+7.217	+7.333
33	38	.03	-	.20	.20	+3.946	+4.000
2	7	.26	.03	.40	.40	+1.096	+1.493
7	12	.04	.01	.05	.05	+14.625	+18.608
12	17	.02	.01	.04	.04	+21.371	+23.272
17	22	.07	.02	.09	.09	+7.772	+8.220
22	27	.17	.12	.07	.07	+4.464	+6.199
27	32	.20	.11	.11	.11	+3.380	+5.427
32	37	.03	.02	.05	.05	+16.150	+17.570
1	2	.11	.01	.56	.56	+.752	+.786
2	3	.20	.02	2.28	2.28	-.563	-.561
6	7	.13	.04	1.93	1.93	-.483	-.482

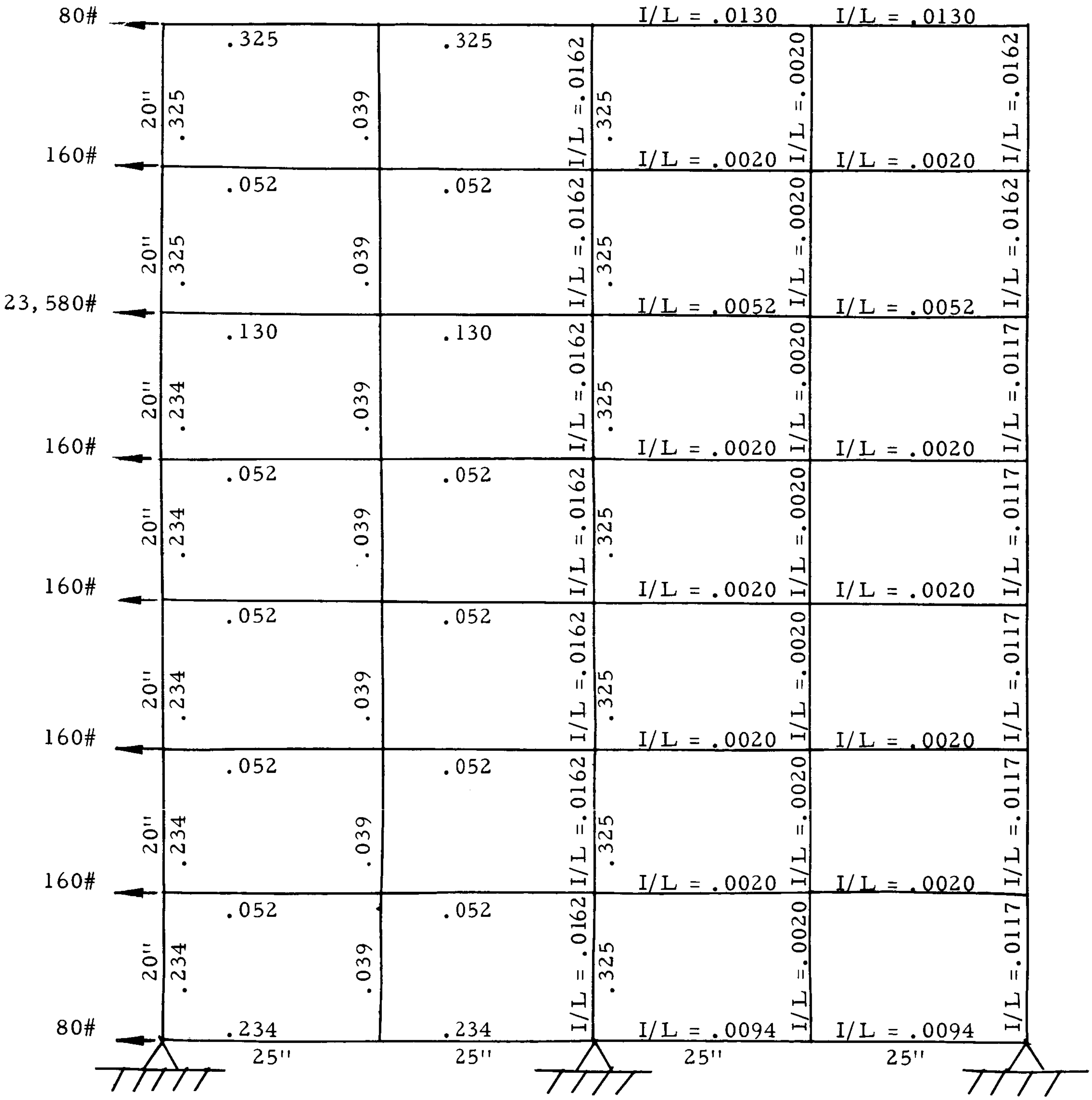
MEMBER		R_s WEBS	R_s CAPS
7	8	.09	.05
11	12	.07	-
12	13	.04	-
16	17	.11	.04
17	18	.08	.04
21	22	.17	.10
22	23	.07	.05
26	27	.16	-
27	28	.18	-
31	32	.15	.08
32	33	.07	.05
36	37	.07	.04
37	38	.04	.04

	f_s		f_{sv}	f_s		f_b		F_s		F_b	
	WEBS	CAPS	WEBS	WEBS	CAPS	A	B	WEBS	CAPS	WEBS	CAPS
1	29	43	3,088	3,117	43	16,074	35,133	20,000	20,000	38,000	38,000
5	16	24	3,596	3,612	24	34,537	25,082	20,000	20,000	38,000	38,000
5	1,804	1,696	1,124	2,928	1,696	46,110	24,094	20,000	20,000	38,000	38,000
4	972	914	399	1,371	914	18,296	6,593	20,000	20,000	38,000	38,000
0	547	808	874	1,421	808	9,079	5,404	20,000	20,000	38,000	38,000
0	539	796	228	767	796	5,462	1,684	20,000	20,000	38,000	38,000

	R_b		M.S.	
	WEBS	CAPS	WEBS	CAPS
	.81	.81	+.227	+.232
	1.61	1.61	-.379	-.379
	.80	.80	+.248	+.250
	1.47	1.47	-.322	-.320
	.69	.69	+.440	+.447
	1.58	1.58	-.371	-.368
	.50	.50	+.981	+.990
	.92	.92	+.071	+.087
	.91	.91	+.078	+.099
	1.21	1.21	-.180	-.175
	.48	.48	+1.061	+1.072
	.24	.24	+3.000	+3.110
	.14	.14	+5.868	+5.868

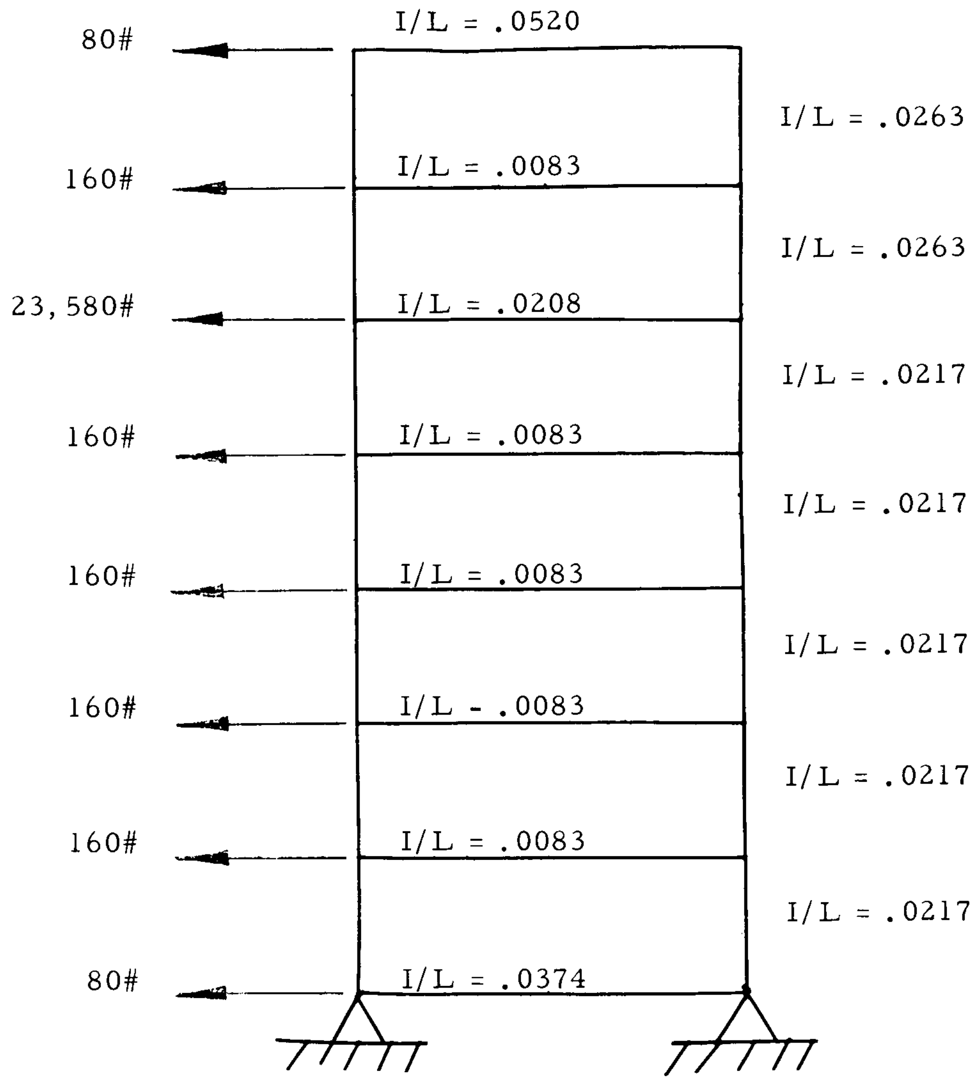
TABLE 6.1.7-2

STRESS ANALYSIS OF
SATURN CENTAUR
SOLAR ARRAY QUADRANT



ANALOGOUS FRAME
IN-PLANE BENDING

FIGURE 6.1.7-3



The stiffness factors are then modified and used to compute distribution factors. The bent is separated along the line of contra flexure points to simplify the sideway distribution procedure. For the horizontal beams the modified stiffness becomes: $K_M = 3/4\ K_o \times 2/Lo = 3/2\ K_o = 1.5\ Ko$

The distribution factors are calculated from the formula: $DF_i = \frac{K_i}{\Sigma K_i}$ at the joint.

As shown in Figure 6.1.7-4, the beam carry over factor for each story is -1 from top to bottom and 0 from bottom to top on the left. Along the line of contra flexure the story carry over factor is 0 from top to bottom and +K above/K below from bottom to top. Along the contra flexure line the moment carried up is the fixed end moment + the distributed moment summed at the story below and multiplied by the story carryover factor.

The values for the factors and the moments are shown in the figures. The final moments and shears on the analogous bent are shown in Figure 6.1.7.3. These moments and shears are then proportioned to the analogous frame by the ratio:

$$\frac{(I/L)}{\Sigma(I/L)}\ \text{Beam} \qquad \times \qquad \frac{(I/L)}{\Sigma(I/L)}\ \text{Column}$$

for each story and bay and are given in Figure 6.1.7-6.

For example, the proportioning factor for the bay within nodes 21 31, 23 33 is calculated by:

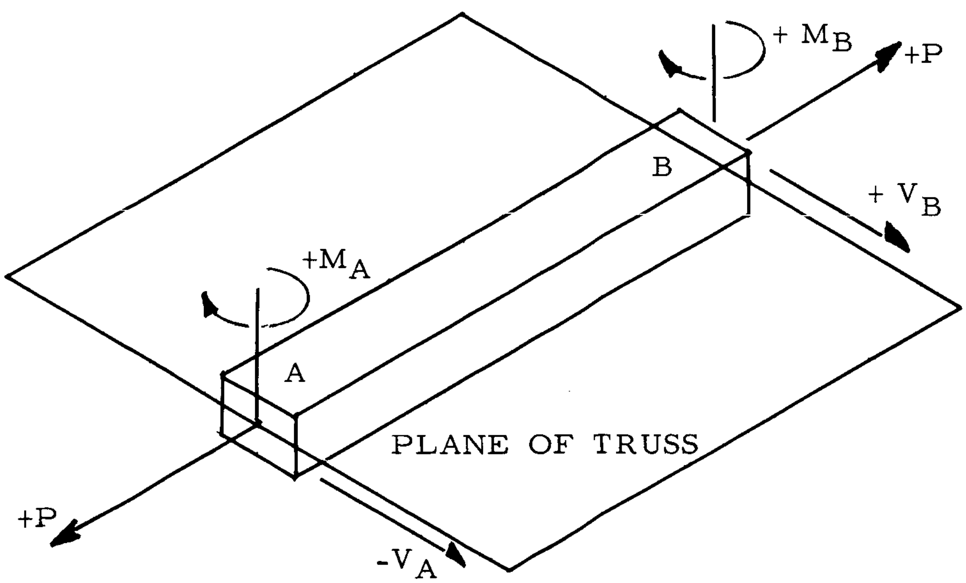
$$P.F._{\text{beams at 27}} = \left(\frac{.0104}{.0208}\right)\left(\frac{.0040}{.0481}\right) = .0415$$

The column factor at node 27 $= 2 \times P.F._{\text{beam 27}} = .083$

What moment is not carried by node 27 is carried equally by nodes 26 and 28.

Following the sign convention previously given, the stresses and margins of safety have been calculated and are given in Table 6.1.7-7 for one half of the frame. Due to the truss action, however, there are axial loads induced in the members. Since the truss is under in plane shear loads, there are no out of plane bending moments induced nor are any torsional moments induced in the members.

Compression loads are designated minus and tension loads plus.



CONTINUE FROM PT. B

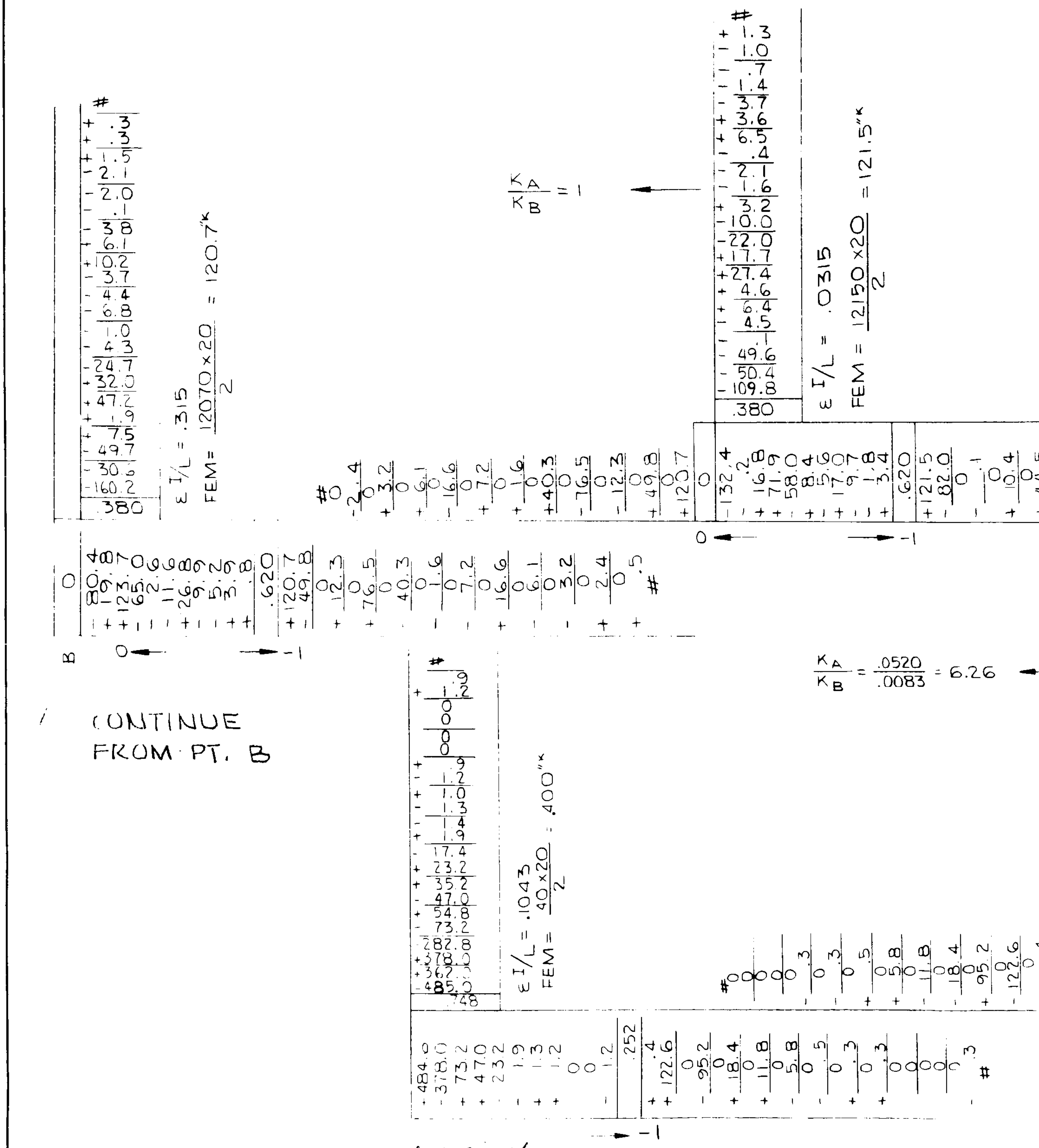
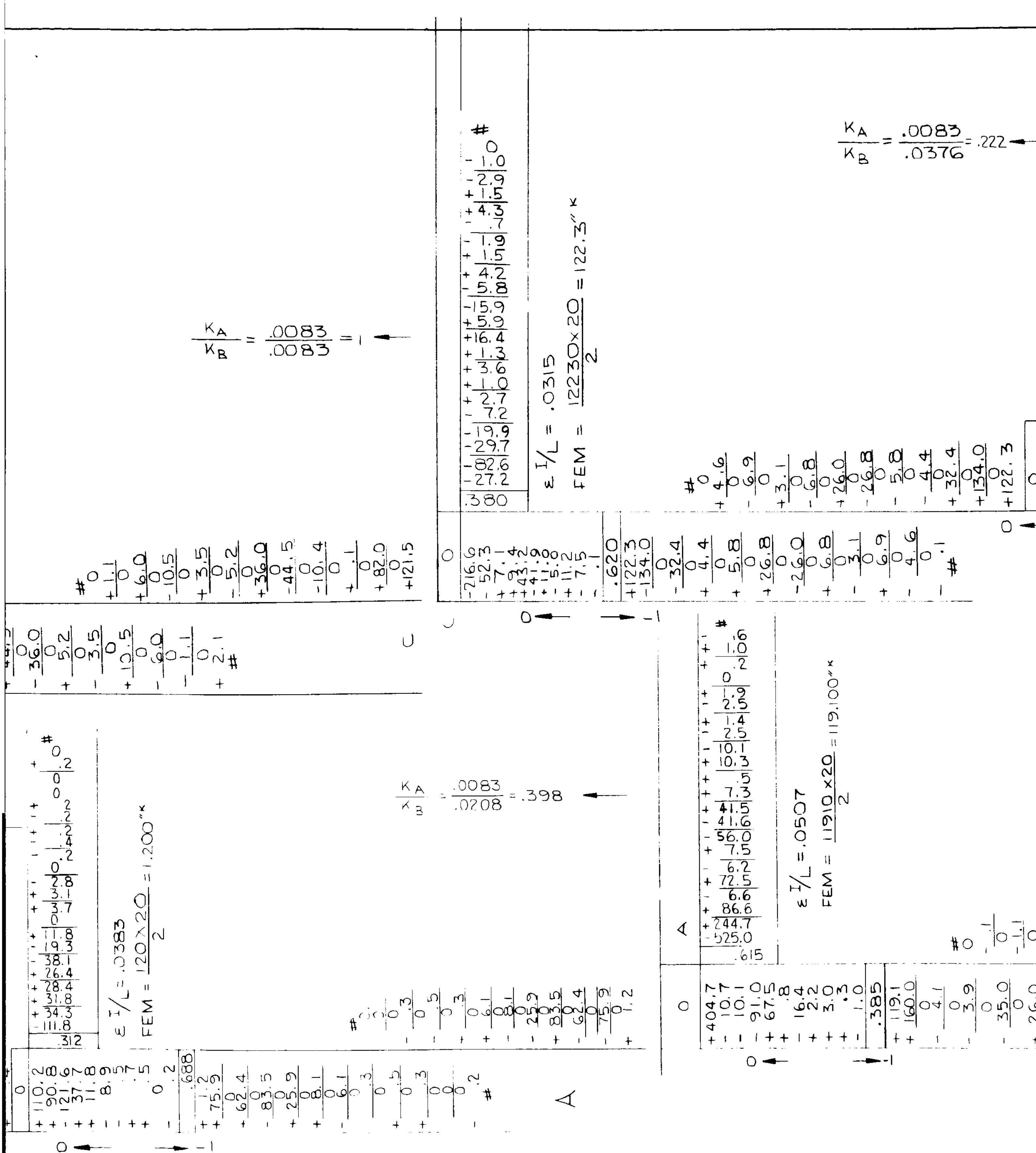


Fig 6.17-4
(7)



KA/KB = .0083/.0376 = .222
KA/KB = .0083/.0083 = 1
KA/KB = .0083/.0208 = .398
ε I/L = .0315
FEM = 12230×20/2 = 122.3"K
ε I/L = .0507
FEM = 11910×20/2 = 119,100"K
ε I/L = .0383
FEM = 120×20/2 = 1,200"K

FIG 6.17-4 (2)

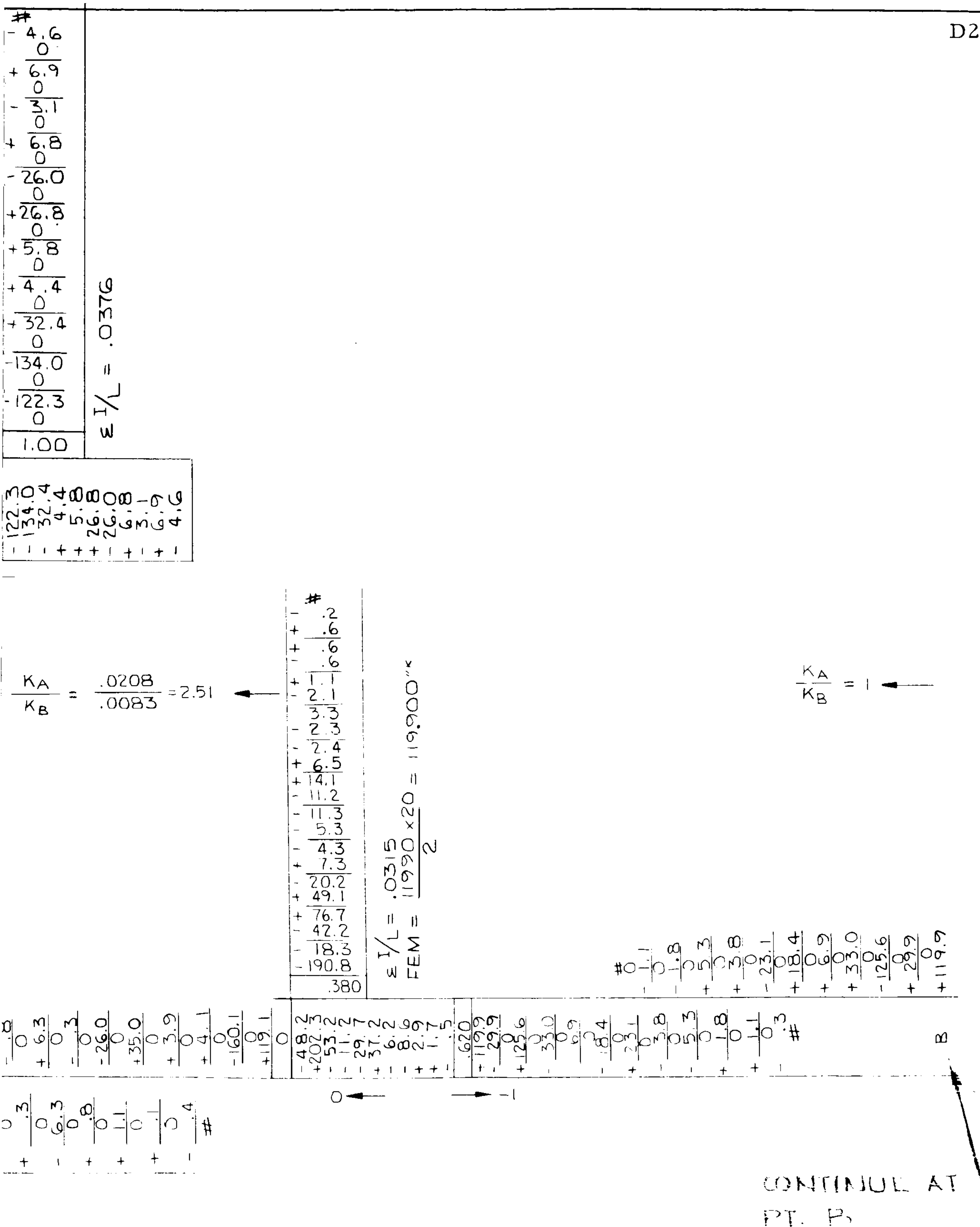
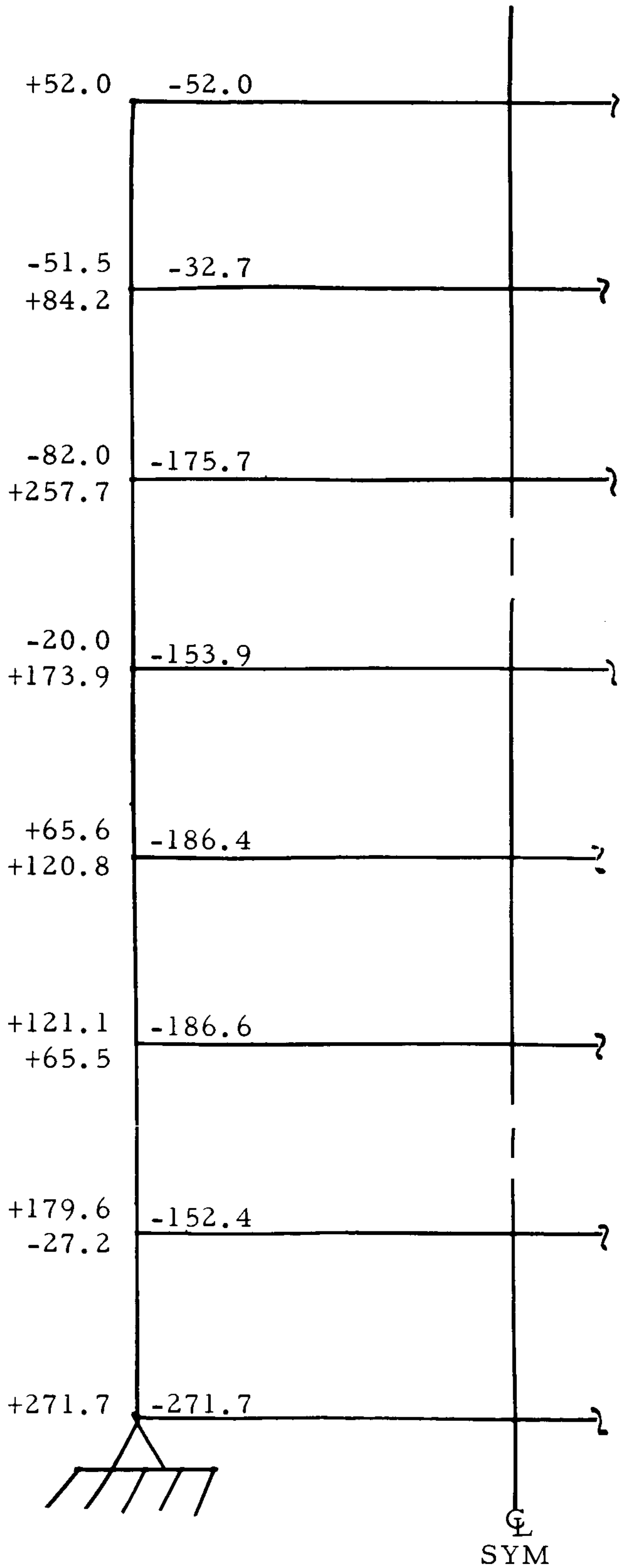


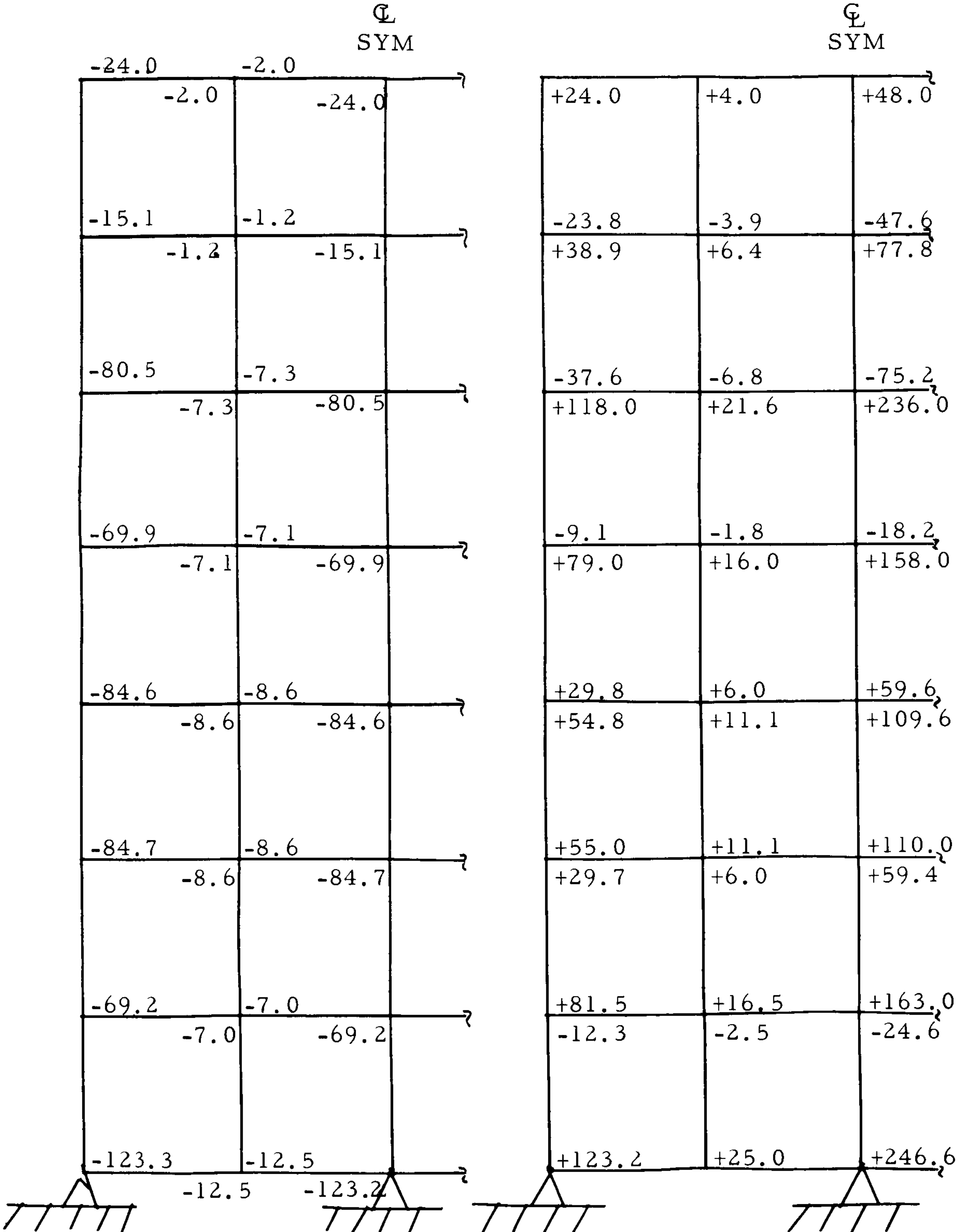
FIGURE 6.1.7-4 STRESS ANALYSIS
SOLAR ARRAY QUADRANT
SATURN IB/CENTAUR

3



ALL MOMENTS IN THOUSAND INCH POUND UNITS
(INCH - KIPS)

BENDING MOMENTS IN
ANALOGOUS BENT
FIGURE 6.1.7-5



BEAM MOMENTS IN ANALOGOUS FRAME

COLUMN MOMENTS IN ANALOGOUS FRAME

ALL MOMENTS IN THOUSAND INCH POUND UNITS (INCH-KIPS)

BEAM AND COLUMN MOMENTS
FIGURE 6.1.7-6

STRUCTURE IS SYMMETRIC IN MOMENT AND SHE

$F_b = F_d = 38$ KSI $F_s = 20$ KSI $R_C = f_c / F_d$ $R_b = f_b / F_b$ $R_S = f_s / F_s$ M.

MEMBER		AREA	C/I	$Q/_{Ib}$	M ~ INk		P~ k	V ~ k	f_c	f_b KSI	
A	B	$IN.^2$	$IN.^{-3}$	$IN.^{-2}$	A	B			KSI	WEBS	CA
1	6	2.86	2.14	.535	+123.3	-12.3	-18.79	5.55	-6.56	263.9	26
6	11	2.86	2.14	.535	+81.5	+29.7	-15.74	5.56	-5.49	174.4	17
11	16	2.86	2.14	.535	+55.0	+54.8	-12.01	5.49	-4.19	117.7	11
16	21	2.86	2.14	.535	+29.8	+79.0	-8.28	5.44	-2.89	169.1	16
21	26	2.86	2.14	.535	-9.1	+118.0	-5.20	5.45	-1.81	252.5	25
26	31	4.94	1.53	.231	-37.6	+38.9	-1.69	.01	-.34	59.5	5
31	36	4.94	1.53	.231	-23.8	+24.0	-1.04	.01	-.21	36.7	3
3	8	4.94	1.53	.231	+246.6	-24.6	0	11.10	0	377.3	377
8	13	4.94	1.53	.231	+163.0	+59.4	0	11.12	0	249.4	24
13	18	4.94	1.53	.231	+110.0	+109.6	0	10.98	0	168.3	168
18	23	4.94	1.53	.231	+59.6	+158.0	0	10.88	0	241.7	241
23	28	4.94	1.53	.231	-18.2	+236.0	0	10.90	0	361.1	361
28	33	4.94	1.53	.231	-75.2	+77.8	0	.13	0	119.0	11
33	38	4.94	1.53	.231	-47.6	+48.0	0	.02	0	73.4	7
2	7	.61	10.25	1.92	+25.0	-2.5	0	1.12	0	256.3	256
7	12	.61	10.25	1.92	+16.5	+6.0	0	1.12	0	169.1	16
12	17	.61	10.25	1.92	+11.1	+11.1	0	1.11	0	113.8	11
17	22	.61	10.25	1.92	+6.0	+16.0	0	1.10	0	164.0	164
22	27	.61	10.25	1.92	-1.8	+21.6	0	.99	0	221.4	221
27	32	.61	10.25	1.92	-6.8	+6.4	0	.02	0	69.7	69
32	37	.61	10.25	1.92	-3.4	+4.0	0	.03	0	41.0	41
1	2	2.86	2.14	.535	-123.3	-12.5	-2.63	5.44	-.92	263.9	263
2	3	2.86	2.14	.535	-12.5	-123.3	-1.51	5.44	-.53	263.9	263
6	7	.91	7.69	1.44	-69.2	-7.0	+.03	3.05	0	532.1	532
7	8	.91	7.69	1.44	-7.0	-69.2	+.07	3.05	+.01	532.1	532
11	12	.91	7.69	1.44	-84.7	-8.6	-.05	3.73	0	651.3	651
12	13	.91	7.69	1.44	-8.6	-84.7	-.02	3.73	0	651.3	651
16	17	.91	7.69	1.44	-84.6	-8.6	-.03	3.73	0	651.3	651
17	18	.91	7.69	1.44	-8.6	-84.6	0	3.73	0	651.3	651
21	22	.91	7.69	1.44	-69.9	-7.1	+.03	3.08	0	537.5	537
22	23	.91	7.69	1.44	-7.1	-69.9	-.08	3.08	+.01	537.5	537
26	27	2.37	3.08	.173	-80.5	-7.3	+6.29	3.51	+2.6	247.9	247
27	28	2.37	3.08	.173	-7.3	-80.5	+5.32	3.51	+2.2	247.9	247
31	32	.91	7.69	1.44	-15.1	-1.2	0	.65	0	116.1	116
32	33	.91	7.69	1.44	-1.2	-15.1	0	.65	0	116.1	116
36	37	4.94	1.53	.231	-24.0	-2.0	0	1.04	0	36.7	36
37	38	4.94	1.53	.231	-2.0	-24.0	0	1.04	0	36.7	36

AR ONLY, ABOUT MEMBER 3-8-13-18-23-28-33-38

$$\text{S.} = \sqrt{\frac{1}{R_s^2 + (R_b + R_c)^2}} - 1$$

	f_s	$f_c + f_b \sim$ KSI		$R_b + R_c$		R_s	M.S.	
PS	CAPS	WEBS	CAPS	WEBS	CAPS	CAPS	WEBS & CAPS	
.9	2.97	270.5	270.5	7.118	7.118	.148	-.8595	
.4	2.98	179.9	179.9	4.734	4.734	.149	-.789	
.7	2.93	121.9	121.9	3.208	3.208	.146	-.688	
.1	2.91	172.0	172.0	4.526	4.526	.145	-.779	
.5	2.92	254.3	254.3	6.692	6.692	.146	-.851	
.5	0	59.8	59.8	1.574	1.574	0	-.365	
.7	0	36.9	36.9	.971	.971	0	+.030	
.3	2.56	377.3	377.3	9.929	9.929	.128	-.899	
.4	2.57	249.4	249.4	6.563	6.563	.128	-.848	
.3	2.54	168.3	168.3	4.429	4.429	.127	-.774	
.7	2.51	241.7	241.7	6.360	6.360	.125	-.843	
.1	2.52	361.1	361.1	9.503	9.503	.126	-.895	
.0	.03	119.0	119.0	3.132	3.132	.0015	-.681	
.4	0	73.4	73.4	1.932	1.932	0	-.482	
.3	2.15	256.3	256.3	6.745	6.745	.107	-.852	
.1	2.15	169.1	169.1	4.450	4.450	.107	-.775	
.8	2.13	113.8	113.8	2.995	2.995	.106	-.666	
.0	2.11	164.0	164.0	4.316	4.316	.105	-.768	
.4	1.90	221.4	221.4	5.826	5.826	.095	-.828	
.7	.04	69.7	69.7	1.834	1.834	.002	-.455	
.0	.06	41.0	41.0	1.079	1.079	.003	-.073	
.9	2.91	264.8	264.8	6.968	6.968	.145	-.856	
.9	2.91	264.4	264.4	6.958	6.958	.145	-.856	
.1	4.39	532.1	532.1	14.003	14.003	.219	-.928	
.1	4.39	532.1	532.1	14.003	14.003	.219	-.928	
.3	5.37	651.3	651.3	17.140	17.140	.268	-.942	
.3	5.37	651.3	651.3	17.140	17.140	.268	-.942	
.3	5.37	651.3	651.3	17.140	17.140	.268	-.942	
.3	5.37	651.3	651.3	17.140	17.140	.268	-.942	
.5	4.44	537.5	537.5	14.145	14.145	.222	-.929	
.5	4.44	537.5	537.5	14.145	14.145	.222	-.929	
.9	.61	250.5	250.5	6.592	6.592	.030	-.848	
.9	.61	250.1	250.1	6.582	6.582	.039	-.848	
.1	.94	116.1	116.1	3.055	3.055	.047	-.673	
.1	.94	116.1	116.1	3.055	3.055	.047	-.673	
.7	.24	36.7	36.7	.966	.966	.012	+.035	
.7	.24	36.7	36.7	.966	.966	.012	+.035	

TABLE 6.1.7-7

STRESS ANALYSIS OF
SATURN CENTAUR
SOLAR ARRAY QUADRANT

The prevalence of negative margins throughout the structure indicates that some measure of relief must be provided. At the present time four candidate modifications have been conceived. These four with their respective advantages and disadvantages are:

1. Provision of firmer shear tie across the faying surfaces of the panel stack. This concept alleviates the normal bending stress in the quadrant because of the increase in effective I. Likewise the resonant frequency increases and the reaction load at nodes 26 and 30 decreases. As a result of the decrease in reaction load, the loading of the adjacent supporting quadrants is reduced. This concept by itself is unlikely to relieve the stresses in the supporting quadrants sufficiently.

2. Installation of an external diagonal strut between nodes 1 and 30 and between nodes 5 and 27 to provide a direct load path for the support reaction load. This concept will develop very high compressive loads in the members between nodes 1 — 26, 26 — 30, 1 — 5, and 5 — 30; unless external framework is provided. This concept will result in an increase in structural weight. The extent of the increase is not known at this time.

3. Installation of additional support ties between adjacent quadrants. This concept directly alters the mode shapes and frequencies for normal vibration of a quadrant and will result in lower reaction loads spread over a longer length of the adjacent supporting quadrants. The normal bending stresses and the side shear bending stresses will be reduced. However, increasing the number of latches and tie points may present reliability problems with some increase in array weight.

4. Construction of main sub-panel #1 as a shear beam with a continuous shear web of thin beryllium sheet in place of the crossed fiberglass tapes on this sub-panel only. With this concept the reaction load is carried down to the main hinges in the deep shear beam made up of the beryllium web, intercostals, frames, and some additional T or L stiffeners on main sub-panel #1. A small penalty on electrical performance of the solar cells on this sub-panel will ensue. Furthermore, there will be a weight penalty of approximately 80 pounds to the total array if this concept is used. This concept would require the least redesign of the array structure and mechanisms with the least effect on the system reliability analysis.

In summary, although stress levels are excessive under certain loading conditions, the remedies proposed pose no undue penalties to the system performance. From the stress analysis standpoint, a 50 KW, 20 watt per pound solar array of this type is certainly feasible.

6.1.8 Thermal Analysis

6.1.8.1 Summary

Thermal analysis of the Saturn IB/Centaur launched solar array was reported in detail in bi-monthly reports D2-23942-2 and D2-23942-4. The analyses are summarized in this section of the final report.

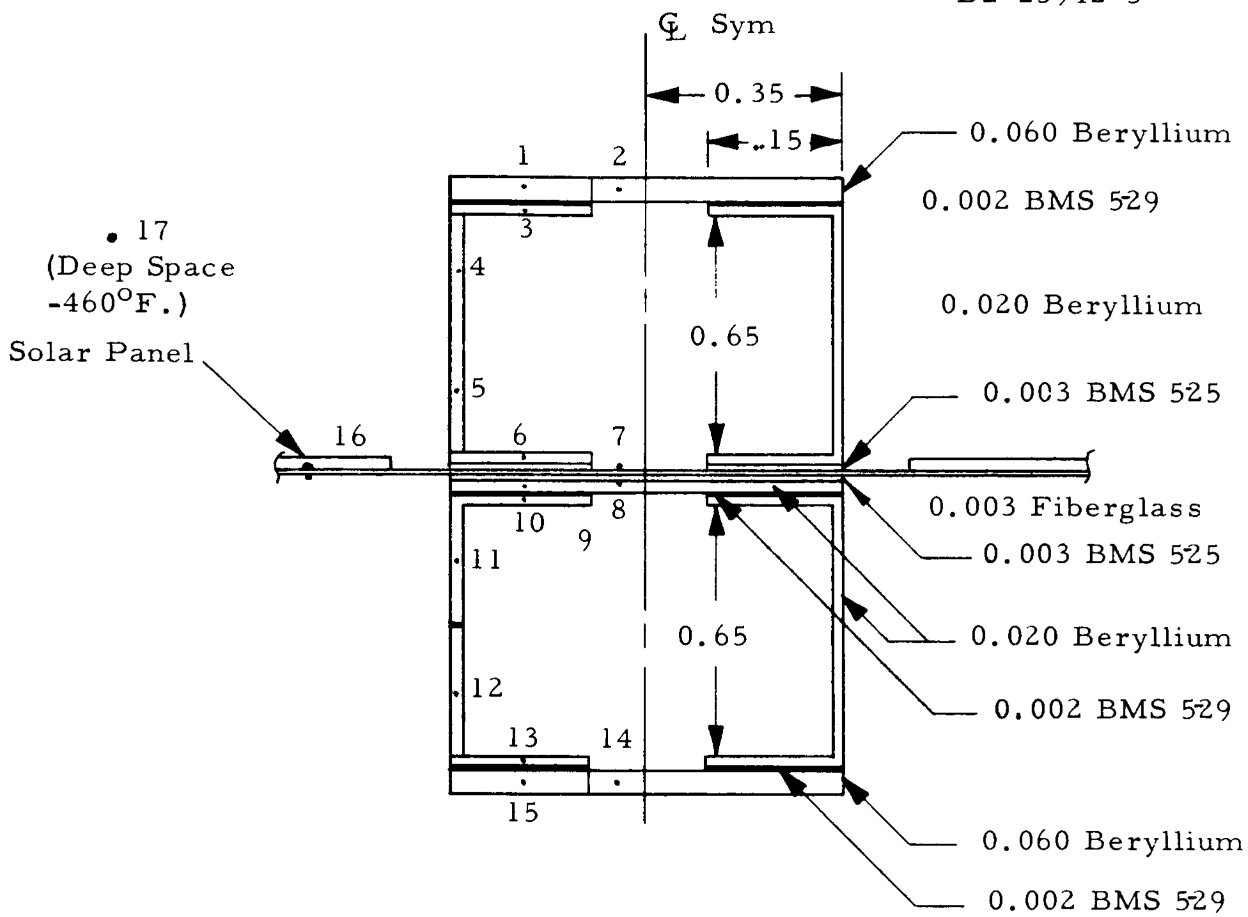
A 3 mil thick, Laminar X-500 thermal control coating is used on the back of the solar cells, substrate and beryllium frames in this design. Experimental data indicates that the emittance of this paint varies with thickness for thickness less than 5 mils. Disagreement exists as to the emittance value, 0.85 to 0.95, that should be applied to the thermal analysis of the array. This results in some variation for predicted temperatures for solar cells and array structure. The results obtained in this analysis were based on an assumed emittance value of 0.95. If the emittance value is reduced to 0.85, the 24.35 watts per pound performance ratio would be reduced approximately 10 percent to 22.91 watts per pound.

The analyses indicate that thermal distortion of the deployed array is minimal and presents no major problems for design of large area arrays of this type.

6.1.8.2 Analysis

The steady state thermal analysis of the large area solar arrays is based on the conductivities, emittances, and absorptances shown on the thermal model Figure 6.1.8-1. Results of the analysis shown in Table 6.1.8-2 indicate no appreciable thermal gradient exists in the structure if a high emittance thermal control coating is applied to the shaded side of the structure.

In the analysis, it was assumed that the spar absorbs energy from the solar cells by radiation, but the temperature of the solar cells remains constant at the values noted. These solar cell temperatures have been calculated for thermal equilibrium conditions. Because the steady state temperatures of the structure differ very little from the solar cell temperatures, this assumption as to solar cell - structure energy interchange is considered valid.



THERMAL DATA USED IN ANALYSIS

MATERIAL	THERMAL CONDUCTIVITY
Fiberglass	0.02 BTU-In/In^2-Hr-°F
Beryllium	8 BTU-In/In^2-Hr-°F
BMS 529	0.00833 BTU-In/In^2-Hr-°F
BMS 525	0.0125 BTU-In/In^2-Hr-°F

FOR BERYLLIUM:

Solar Absorptance = 0.48
Solar Reflectance = 0.22
Emittance - 0.09, Ext. and Int. Surfaces
High-Emittance Paint (ϵ = 0.95) is considered to be used on external surfaces for lower half section of spar.

SOLAR HEAT INPUTS:

LOCATION	SOLAR CONSTANT
Venus	267 Milliwatts/sq. cm.
1/2 way Venus-Earth	189 Milliwatts/sq. cm.
Earth	140 Milliwatts/sq. cm.
1/2 way Earth-Mars	88 Milliwatts/sq. cm.
Mars	60 Milliwatts/sq. cm.

THERMAL MODEL
50 KW ARRAY
FIGURE 6.1.8-1

NODE	Emittance = 0.09 for Nodes 1-15 Both Surfaces Except: Emittance = 0.95 for external surfaces at Nodes 11, 12, 14, and 15 — LOCATION					Emittance = 0.09 for Nodes 1-15, Both Surfaces (External and Internal) — LOCATION				
	Venus °F	1/2 Way Venus - Earth °F	Earth °F	1/2 Way Earth Mars °F	Mars °F	Venus °F	1/2 Way Venus - Earth °F	Earth	1/2 Way Earth-Mars	Mars °F
1	196.1	140.3	96.3	34.7	-8.71	515.3	436.7	373.6	286.5	222.9
2	196.3	140.5	96.4	34.8	-8.66	515.4	436.8	373.7	286.6	223.0
3	195.2	139.7	95.8	34.4	-8.92	514.7	436.3	373.3	286.3	222.8
4	192.4	137.6	94.3	33.4	-9.58	512.7	434.9	372.3	285.6	222.3
5	189.6	135.6	92.8	32.5	-10.24	511.2	433.8	371.5	285.1	222.0
6	186.5	133.4	91.1	31.4	-10.99	509.9	432.9	370.8	284.7	221.7
7	183.9	131.5	89.7	30.5	-11.60	508.8	432.1	370.2	284.3	221.5
8	183.8	131.5	89.7	30.5	-11.60	508.8	432.1	370.2	284.3	221.5
9	183.8	131.5	89.7	30.5	-11.61	508.8	432.1	370.2	284.3	221.5
10	182.8	130.8	89.2	30.2	-11.85	508.4	431.8	370.0	284.2	221.4
11	179.7	128.5	87.5	29.1	-12.61	507.1	430.9	369.3	283.8	221.1
12	177.3	126.8	86.2	28.3	-13.19	506.3	430.3	368.9	283.5	220.9
13	175.3	125.4	85.1	27.6	-13.65	505.9	430.0	368.7	283.4	220.8
14	174.6	124.8	84.7	27.3	-13.83	505.7	429.9	368.6	283.3	220.8
15	174.7	124.9	84.8	27.4	-13.80	505.7	429.9	368.6	283.3	220.8
16	206	150	104	40	-7	206	150	104	40	-7
(Driven Node - Solar Panel										

TEMPERATURE DISTRIBUTION
SATURN IB/CENTAUR
SOLAR ARRAY CENTER SPAR
TABLE 6.1.8-2

6.1.9 Reliability Evaluation

Included in this section is the reliability summary for the mechanical and electrical portions of the 50 kilowatt solar array to be installed on a Saturn IB/Centaur launched spacecraft. In general, the reliability goals discussed in Section 2 have been verified analytically for the preliminary design of the array. In all cases historical failure data for representative components were applied in the analysis. In certain areas where reliability deficiencies were anticipated, as indicated in the following paragraphs, it was necessary to employ redundancy to achieve reliability goals. These redundant elements have been included in the design and weight analyses.

6.1.9.1 Deployment Mechanism Reliability

For 100 percent array deployment, the reliability of the deployment mechanism meets the goal of .999. The failure rates used for all components, except the ordnance devices, are 25 percent of the values obtained from failure data sources. This adjustment of failure rates reflects the light type of service expected of the mechanisms relative to the more severe service to which the available failure data apply. If this adjustment is not made, the calculated reliability is .997.

Redundancy incorporated in the main panel drive system ensures that this relatively complex mechanism has a very small probability of failure. Without redundancy a single failure in the drive system would limit maximum deployment to 75 percent of the array. The torsion springs which deploy the side panels are redundant because a significant increase in reliability is obtained for a very small addition in weight.

6.1.9.2 Jettison Reliability

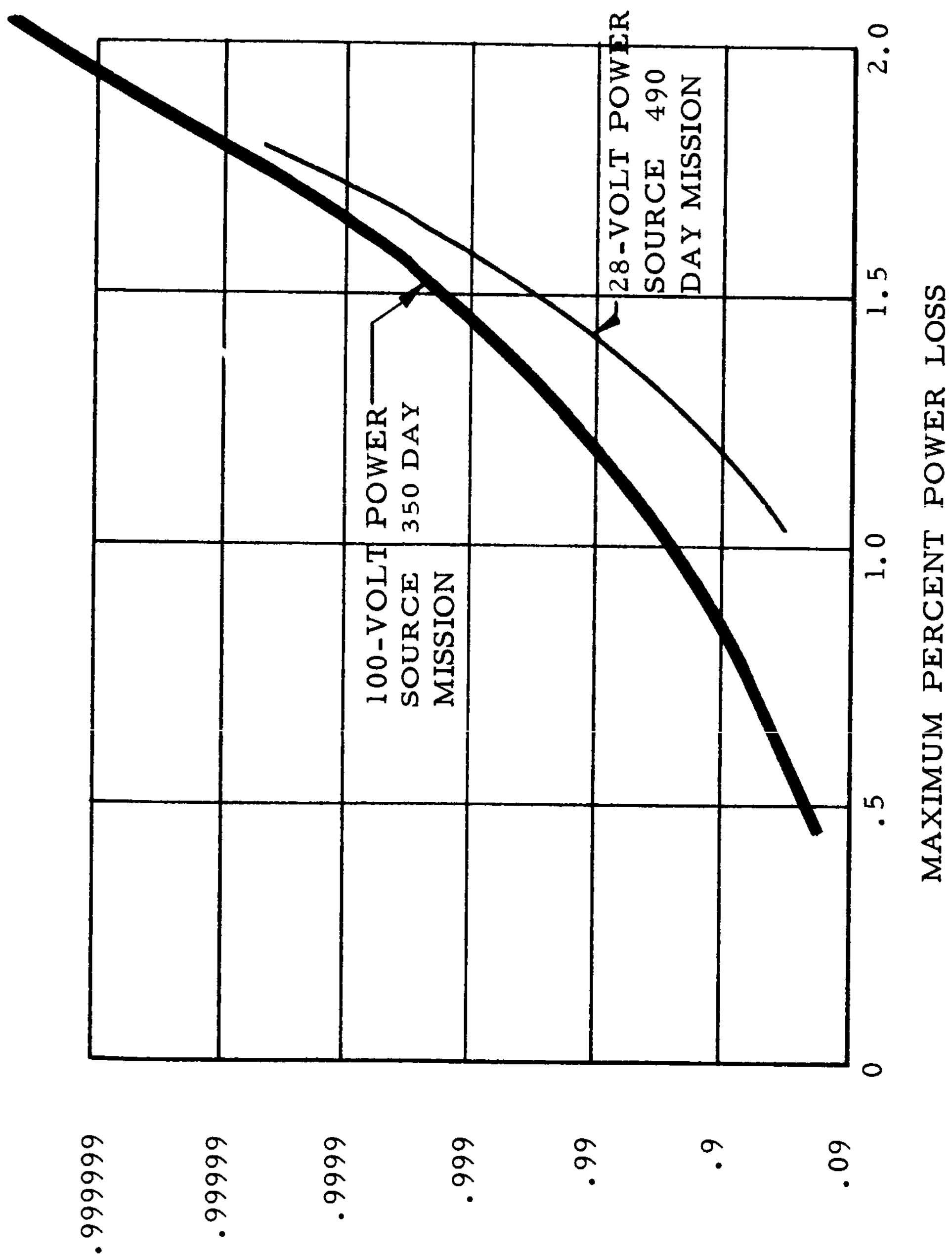
The probability of successful jettison of the 100 volt portion of the array prior to Mars encounter is .999996. The reliability goal established was .9999. Some uncertainty exists about the effect of 350 days exposure to deep space environment on the jettison components. Therefore, redundant squibs, bridgewires, electrical circuits, and mechanical parts were incorporated. To provide additional conservatism in the analysis, a failure rate of .001 failures per firing, instead of the normal .0001, was assumed for the individual ordnance devices.

6.1.9.3 Electrical Busses and Bus Connection Reliability

The probability of failure in the electrical bus system is negligibly small. The connections between the busses and the blocking diodes in the solar cell modules were made redundant to meet the reliability goal of .998. The failure rate used for the soldered connections is $.0001 \times 10^{-6}$ failures per hour.

6.1.9.4 Solar Cell Array Reliability

The reliability of the solar cells in the array was found to be the predominant factor in determining the electrical system reliability. Figure 6.1.9-1 shows the electrical system reliability versus maximum expected loss of electrical power output for the 100 volt and 28 volt power sources at the end of 350 days and 490 days respectively. Redundancy is an essential feature of solar arrays and is built in by the series-parallel arrangement of the cells. The cells in the modules were electrically arranged so that the open circuit failure of a single cell causes the loss of output of only one seven cell string. The open circuit failure rate used for an individual cell is 0.1×10^{-6} failures per hour. The occurrence of short circuit failures in the array was assumed negligible. The curve shows that there is a reliability of .999 that no more than 1.5 percent of rated power will be inactive for the 100 volt system at Mars encounter.



ELECTRICAL SYSTEM
RELIABILITY

FIGURE 6.1.9-1

6.1.10 Weight Analysis - Saturn IB/Centaur

A summary weight statement of the Saturn IB/Centaur configuration is shown below in terms of both total array weight and pounds per square foot. The major difference between the preliminary design and the trade study concept is a reduction in frame weight. There is an overall decrease from .4988 to .3959 pounds per square foot. A more detailed analysis of the final design is included in bi-monthly report number 3, Document D2-23942-3.

Cover Glasses	218.16 Pounds	.0441 Pounds/sq. ft.
Adhesives	78.48	.0159
Cells	464.40	.0939
Connectors	79.28	.0160
Substrate	76.08	.0154
Thermal Coating	136.40	.0276
Frame	531.32	.1070
Mechanisms	201.24	.0407
Electrical Connections	174.36	.0353
	1959.72 Pounds	.3959 Pounds/sq. ft.

6.1.11 Manufacturing Feasibility Evaluation

The successful fabrication of the semi-rigid folding modular solar array will depend largely upon the ability of manufacturing shops to maintain close control of fabrication tolerances affecting the gross weight of the completed arrays. Manufacturing methods and techniques must be developed so that consistent and reliable results are obtained. The process materials that are used throughout the array (adhesives, silicone rubber, solder, paint, etc.) have stringent requirements concerning thickness and quantity. Assembly techniques and tooling innovations must be developed to control the process material application and compensate for the large size and fragile nature of the array.

6.1.11.1 Beryllium Structure Fabrication

Past experience with forming beryllium sheet has shown that the hot creep forming technique consistently produces good parts with the least amount of risk. Accordingly, creep forming was logically selected to form spar and intercostal components. The forming, trimming, and machining is followed by an etch to remove oxidation and micro cracks and prepare the surfaces for bonding.

Beryllium forming requires development of tools and methods for each new configuration. The length of the parts for this array will present problems in forming to maintain straightness. Restraining features in the dies will have to be worked out and the proper relief of stresses within the parts will be important.

The assembly of beryllium parts into spars, intercostals and sub-panel frames utilizes conventional adhesive bonding techniques. Bonding assembly jigs are required to locate and hold the parts throughout the process. The beryllium is expected to bond satisfactorily when proper precautions are taken to maintain clean parts and prevent oxidation after etching. The closed box beam construction using formed channels will provide a straighter assembly, but the bond lines may not be consistent in thickness due to the inability to apply pressure on the inside of the spar.

The assembly of the solar array requires certain components to be matched and fastened on assembly. The ball and cone devices used to take the shear forces while the array is stowed must be assembled in this manner. It is not feasible to use mechanical fastening devices that require drilling holes in the beryllium. Match on assembly items will be installed by the metal bonding process.

6.1.11.2 Fiberglass Tape Substrate Assembly

There are several methods that can be used to assemble the fiberglass tape into the grid pattern required for the solar array. A method of wrapping the tape on a mandrel, in a manner similar to filament tape winding (Figure 6.1.11-1) has been selected for economy of assembly and the ability to make two assemblies on one mandrel. The tape must be cured after assembly. This would be accomplished by placing the assembly between pressure plates and into an autoclave under controlled pressure, temperature and time. The resulting assembly is fused together at the tape intersections and the surface is in a flat plane suitable for solar cell mounting.

6.1.11.3 Fiberglass Tape Substrate Assembly to Panel Frames

The cured fiberglass substrate is bonded to the beryllium frames with an epoxy polyamide adhesive and cured at room temperature. A tension of 8 pounds per tape must be maintained during the bonding operation. A bonding fixture would be required to hold the frame and substrate in relative position. A weight or constant tension device would be clamped to each tape. Adhesive is applied and all components are clamped for adhesive cure.

6.1.11.4 Cover Glass Assembly

The use of back-connected solar cells will allow cover glasses to be assembled before solar cells are soldered. In order to maintain controlled and consistent assembly, a mechanized progressive assembly is recommended. A method and the required equipment must be developed to obtain the desired results.

6.1.11.5 Solar Cell Soldering

The soldering of solar cells into parallel groups and series strings connected with two mil expanded silver connectors presents a challenge to develop tooling and techniques that obtain reliable results. The 8 mil thick cells with their 4 mil thick cover glass require careful handling, accurate positioning and controlled heat, pressure and time for making solder joints.

The positioning and handling can be realized in a fixture (Figure 6.1.11-2) which locates the cells and connectors for a complete string to be soldered and installed on the substrate without removing the cells from the fixture. The fixture would consist of a flexible molded material such as Poly Flex, Eponyl or silicone rubber mounted on an aluminum back up plate to provide rigidity. The solar cells are placed in the cavities face down and the connectors placed in the locating notches and over the cells. The assembly could then be passed under a soldering device for single or multiple solder joint completion. Solder is applied as preforms at the desired locations.

To obtain reliable solder joints between the silver connectors and the solar cells, a controlled sequence of pressure, heat and cooling of short duration which will not cause cell degradation is required. The pulse soldering method has the necessary ingredients to fill these requirements. It is conceivable that pulse soldering in combination with the proposed fixture could be mechanized to permit a more economical assembly of the large number of solar cells involved. Preliminary soldering tests have been made using the pulse soldering method. The solder joints were satisfactory and the cells were not degraded.

6.1.11.6 Solar Cell Installation on Substrate

The strings of solar cells are retained in their soldering fixtures for installation on the fiberglass substrate. A one tenth inch wide strip of primer is applied to the edges of the cells. After proper drying, a quantity of cells and their fixtures for a complete sub-panel, are placed in a master locating fixture which positions all cell strings for a sub-panel. The sub-panel structure, containing the substrate with RTV 40 applied, is positioned over the prelocated cell strings. The soldering fixtures are not removed. The entire sub-panel assembly is now sealed in plastic sheet and a vacuum is applied. The cells are now under uniform pressure to reduce bond line thickness while the vacuum removes any remaining air from the RTV. After a pre-cure, excess RTV is removed and the assembly is left to cure for 48 hours. The sub-panel is completed by making all remaining electrical connections and spraying the back side with Laminar X-500.

6.1.11.7 Bond Line Thickness

The thickness of the adhesive bonding material between two faying surfaces after curing is the bond line thickness. This parameter of the array assembly requires very close control in order to maintain the power to weight ratio required. The techniques for applying BMS 5-29 adhesive to structure parts, RTV 602 for cover glass assembly and RTV 40 for solar cell installation will be unique for this design.

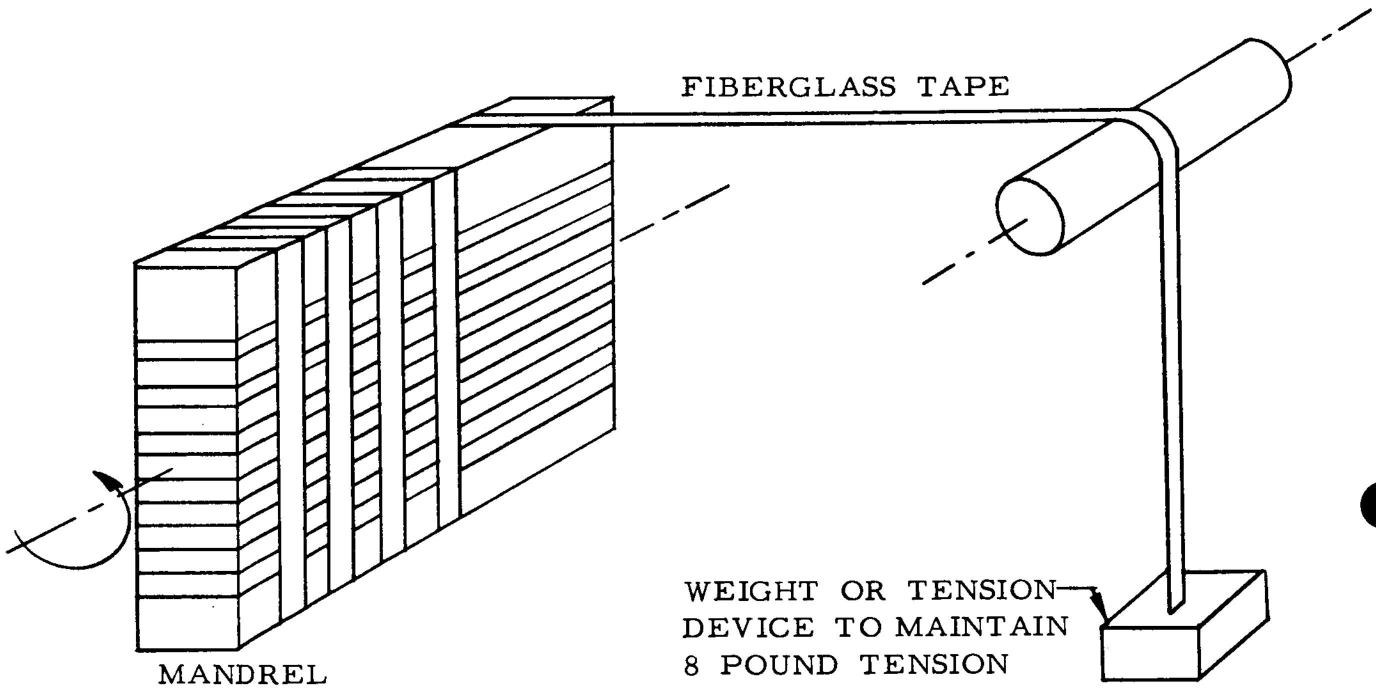
Structural parts will require careful fit-up on assembly to reduce bond line thickness. Equipment for dispensing measured amounts of RTV will have to be developed to insure consistent results. Spray methods of application should be considered although a change in adhesive material may be required to implement this method. The final techniques used can only be determined by experimenting with the various materials and methods.

6.1.11.8 Repair Techniques

The ability to repair structural components of the array is going to be limited to the replacement of complete subassemblies. The bonded construction of the sub-panels will not permit disassembly of the damaged part without causing damage to the adjacent part. The bond cannot be dissolved with solvents, a mechanical means of separation must be used. This type of repair should not be attempted, especially after solar cells are installed. A sub-panel containing the damaged parts would be removed at the hinge connections and replaced by a new sub-panel.

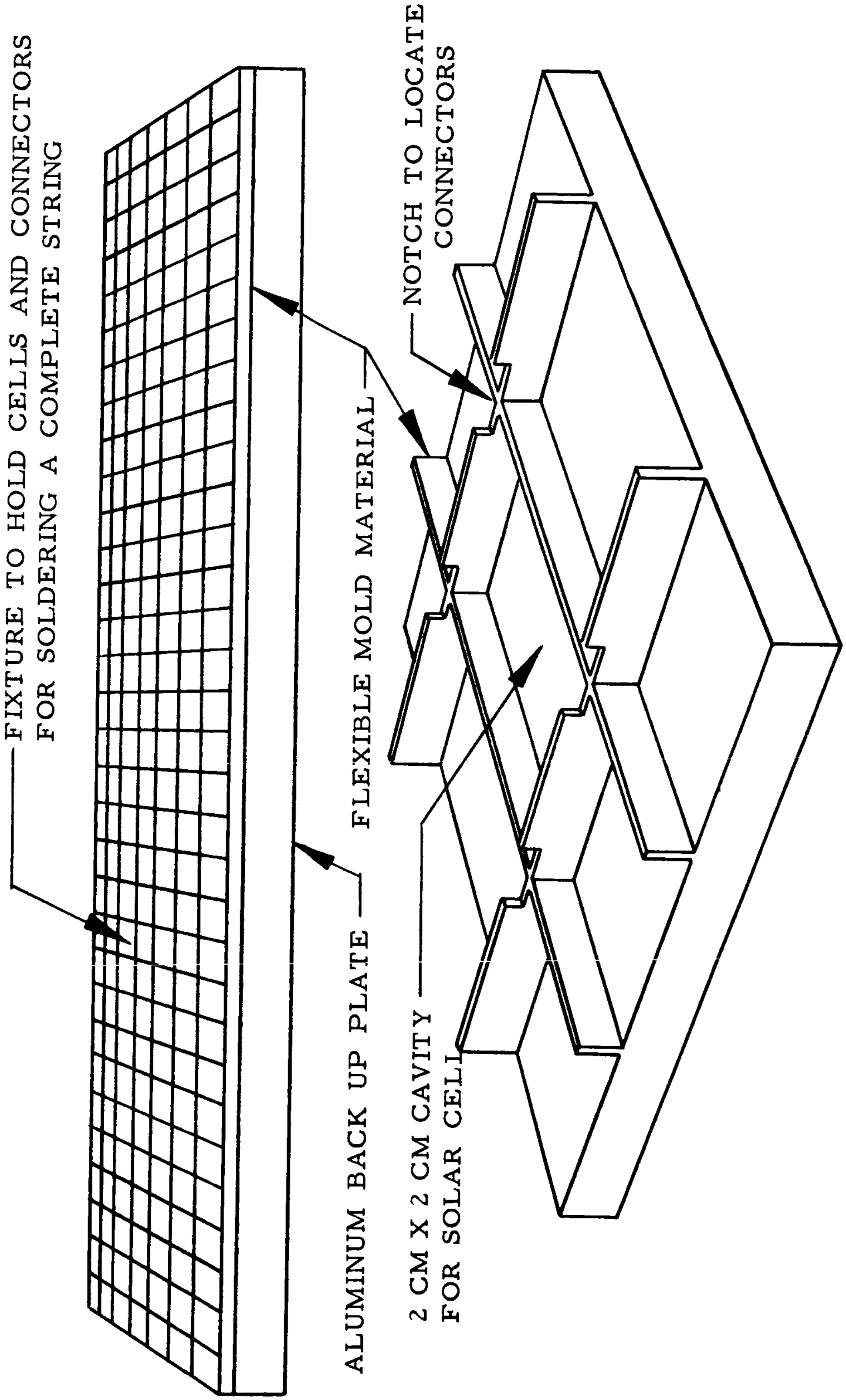
The replacement of individual solar cells can be accomplished with varying degrees of difficulty as the assembly progresses. Before solar cell installation on the substrate, a cell can be removed by desoldering and inserting a new cell with the same current rating.

Soldering is done in the same manner as for regular production. To replace a cell after installation requires careful removal by prying it loose from the connectors and substrate with a thin blade from the back side of the cell. All of the paint and RTV is thoroughly removed from the substrate and connectors. The connectors are then formed up (away from the substrate). A new cell is prepared by soldering connector material to the back side at both ends. After priming the cell and applying new RTV to the substrate the cell is positioned on the substrate with the connectors mating with the existing connectors. The soldered connection is completed with a hand held tweezer type pulse soldering tool. The sub-panel would be in a vertical position to facilitate this type of repair.



FIBERGLASS TAPE
SUBSTRATE ASSEMBLY

FIGURE 6.1.11-1



SOLAR CELL
SOLDERING

FIGURE 6.1.11-2

6.2 ATLAS/CENTAUR ARRAY

The preliminary design for a 10 kilowatt solar photovoltaci array launched by an Atlas/Centaur vehicle is summarized in this section. Envelope, configuration, structural, mechanical, and electrical designs are discussed and illustrated. The necessary analyses that were conducted during the program are also summarized.

6.2.1 Structural Configuration

The folding modular solar array, Figures 6.2-1 and 6.2-2 is stowed within the shroud of the Atlas/Centaur launch vehicle. The panel assemblies are rectangular and have a gross deployed area of 1004 square feet. The stowed array is designed to withstand the static and dynamic load conditions experienced during the launch environment. At approximately three hours after launch, squibs are fired that release the tension tie-down rods permitting the solar panel assemblies to deploy. The panel assemblies then deploy to the cruise mode configuration and supply primary 100 and 28 volt power to the spacecraft. Just prior to the Mars orbit retro maneuver, -3 and -4 panel assemblies, and the outboard sub-panel units of -1 and -2 panel assemblies are jettisoned. This leaves sub-panel unit #1 to provide 28 volt power for the spacecraft subsystems.

The folding modular solar array consists of -1, -2, -3, and -4 panel assemblies. Each panel assembly has three sub-panel units. Each sub-panel unit is made up of two auxiliary panels and a main panel. Each panel (main and auxiliary) has two spars, two edge members, and later intercostals. These structural members are made of bonded beryllium using a BMS 5-29 epoxy Polyamide as the adhesive. The spars and lateral members are formed from beryllium sheet. To facilitate fabrication, beryllium extrusions should be considered as a substitute for the sheet. The substrate is a pre-impregnated fiberglass tape that is formed into a matrix sheet and cured under temperature and pressure to provide a one piece substrate to which the solar cells are applied. The dark side of the solar panels are painted with a Laminar X-500 thermal control coating. The substrate tapes are pretensioned to 8 pounds per tape to minimize the amplitude produced by launch conditions.

During the launch environment each panel assembly is attached to the spacecraft with two hinges at Station 8.00, and each panel stack is attached to the adjacent panel stack at Station 107 with a pin-connected fitting. Tension tie-down rods at Stations 8.00 and 107 pre-load the panel stack spars so that they will be better able to withstand the vibration environment. A system of shear balls in cones and slots is installed in the gap between the adjacent spars so that each spar will transmit shear, but allow movement in the longitudinal direction.

A diagonal tension-compression brace is provided on two #1 sub-panels to withstand the high Mars retro "g's". Since the other two panel assemblies are jettisoned prior to this maneuver, a lock at the spacecraft hinge is sufficient to take the light loads encountered during the cruise mode.

6.2.2 Mechanism Design

A cable drum and pulley system similar to the Saturn/Centaur configuration described in paragraph 6.1.2, has been chosen for the Atlas/Centaur. Auxiliary panels are again deployed by torsion springs. The preliminary design is shown in Figures 6.2-3 and 6.2-4. The deployment sequence is summarized in Figure 6.2-5. Individual hardware elements are similar to those described for Saturn IB/Centaur and are not repeated here.

6.2.3 Solar Cell Modules

The solar cell layout for the Atlas/Centaur is shown in Figure 6.2.3-1. This array consists of two pairs of dissimilar panels. Each panel in turn consists of 3 sub-panel assemblies. One pair of panels carries both 28 volt and 100 volt modules. The other panel pair carries only 100 volt modules.

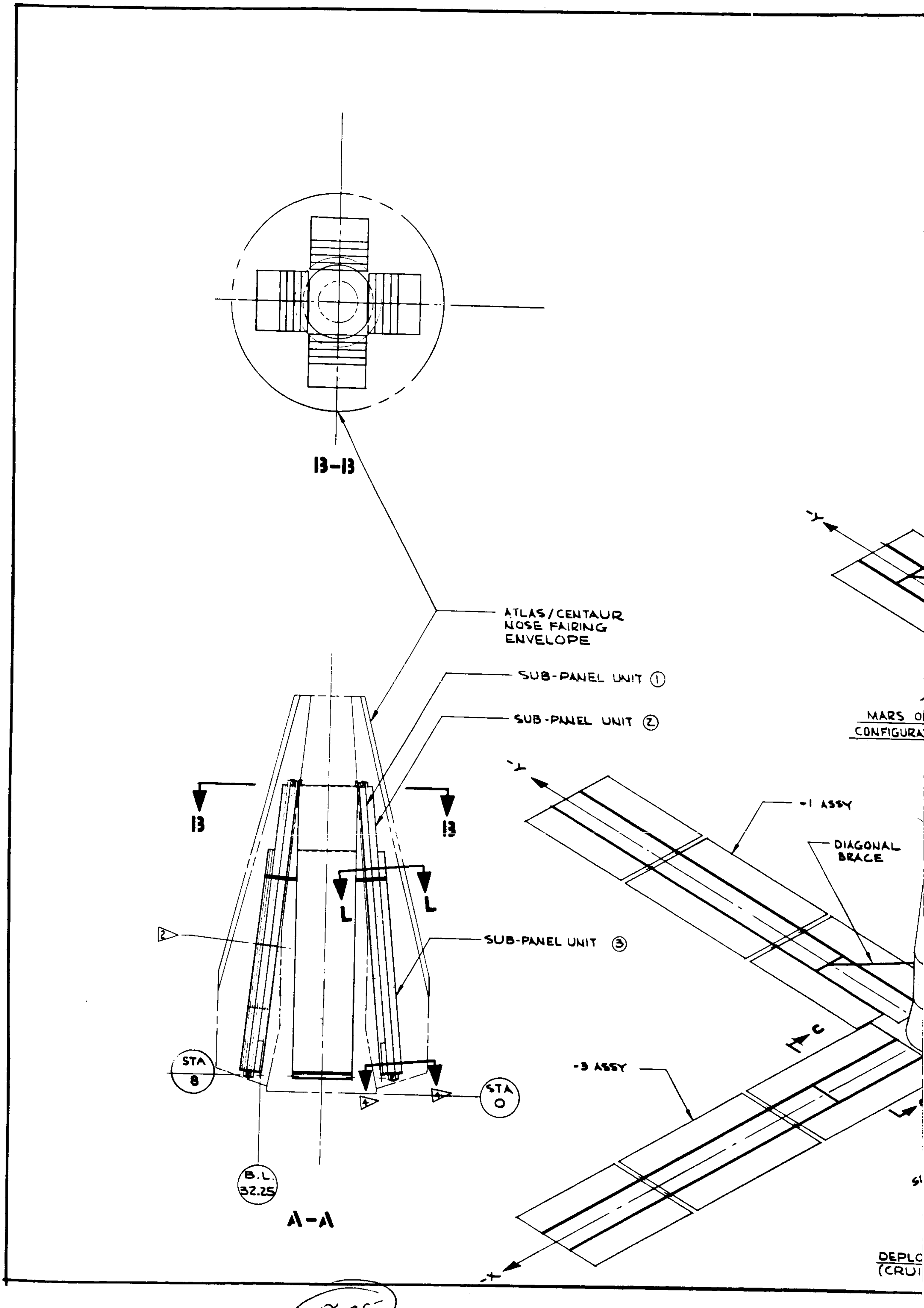
Sub-panel assembly #1 carries 28 28-volt modules and 2 1/2 100-volt modules. Sub-panel assembly #2 carries 9 100-volt modules and sub-panel assembly #3 carries 8 1/2 100-volt modules.

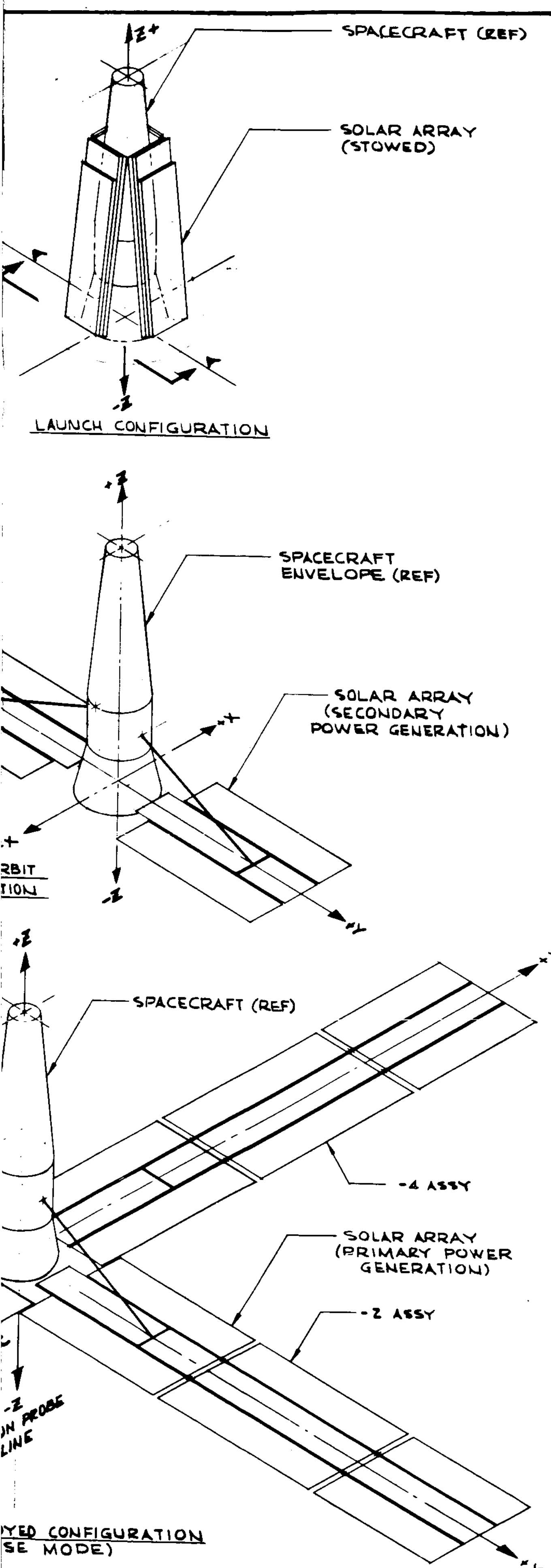
A nine cell parallel connected group is the basic building unit for both the 28 and 100 volt modules. The 28 volt modules consist of 54 series connected nine cell groups arranged in a single string. The 100 volt modules consist of 236 series connected nine cell groups arranged in a four row rectangular layout.

The array carries a total of 209,880 cells and has a power capability at 1.0 A.U. of 10.18 KW.

6.2.3.1 Magnetic Field Analysis

An analysis was conducted to determine if the proposed solar cell layout would yield magnetic field strengths acceptable to the vehicle. In this plan the adjacent solar cell strings are laid so that the current flowing in one string is in an opposite direction to that of its neighbors. The results of this analysis are shown in Figure 6.2.3-2 and indicate that the magnitude of the magnetic field strength decays rapidly with distance from the panel. In view of the large size of the array, this relation suggests that instruments sensitive to magnetic fields can be located far enough away from the panels so that field strength will be within acceptable limits. The proposed interconnection plan was thus deemed acceptable for use in the design.


B-B
ATLAS/CENTAUR
NOSE FAIRING
ENVELOPE
SUB-PANEL UNIT 1
SUB-PANEL UNIT 2
B
B
L
L
2
SUB-PANEL UNIT 3
STA
8
4
4
STA
0
B.L.
32.25
A-A
-Y
-Y
-1 ASSY
DIAGONAL
BRACE
C
-3 ASSY
-X



4. THIS TENSION TIE IS SIMILAR TO THAT SHOWN IN SECTION L-L EXCEPT THAT INBOARD FITTINGS ARE NOT CONNECTED TO EACH OTHER BETWEEN PANELS & REQUIRE SEPARATE PIN PULLERS

3. SHEAR BALL & CONES AT POINTS MARKED "B". SHEAR SLOTS BETWEEN ALL OTHER PANELS. CONES & SLOTS OPPOSITE AT STA. 98.94.

2. ℄ OF SILICONE RUBBER SPACER PADS. NOMINAL THICKNESS IS .25 INCH BUT THICKNESS WILL VARY IN ORDER THAT THE PROPER PRE-LOAD WILL BE APPLIED AT THE PADS TO INSURE THAT THE STOWED PANEL ASSYS WILL ACT AS ONE UNIT UNDER LOAD.

1. HOT CROSS ROLLED BERYLLIUM SHT PER AMS 7902. BOND ALL JOINTS WITH BMS 5-29.

FIGURE 6.2-1 FOLDING MODULAR SOLAR ARRAY ATLAS/CENTAUR

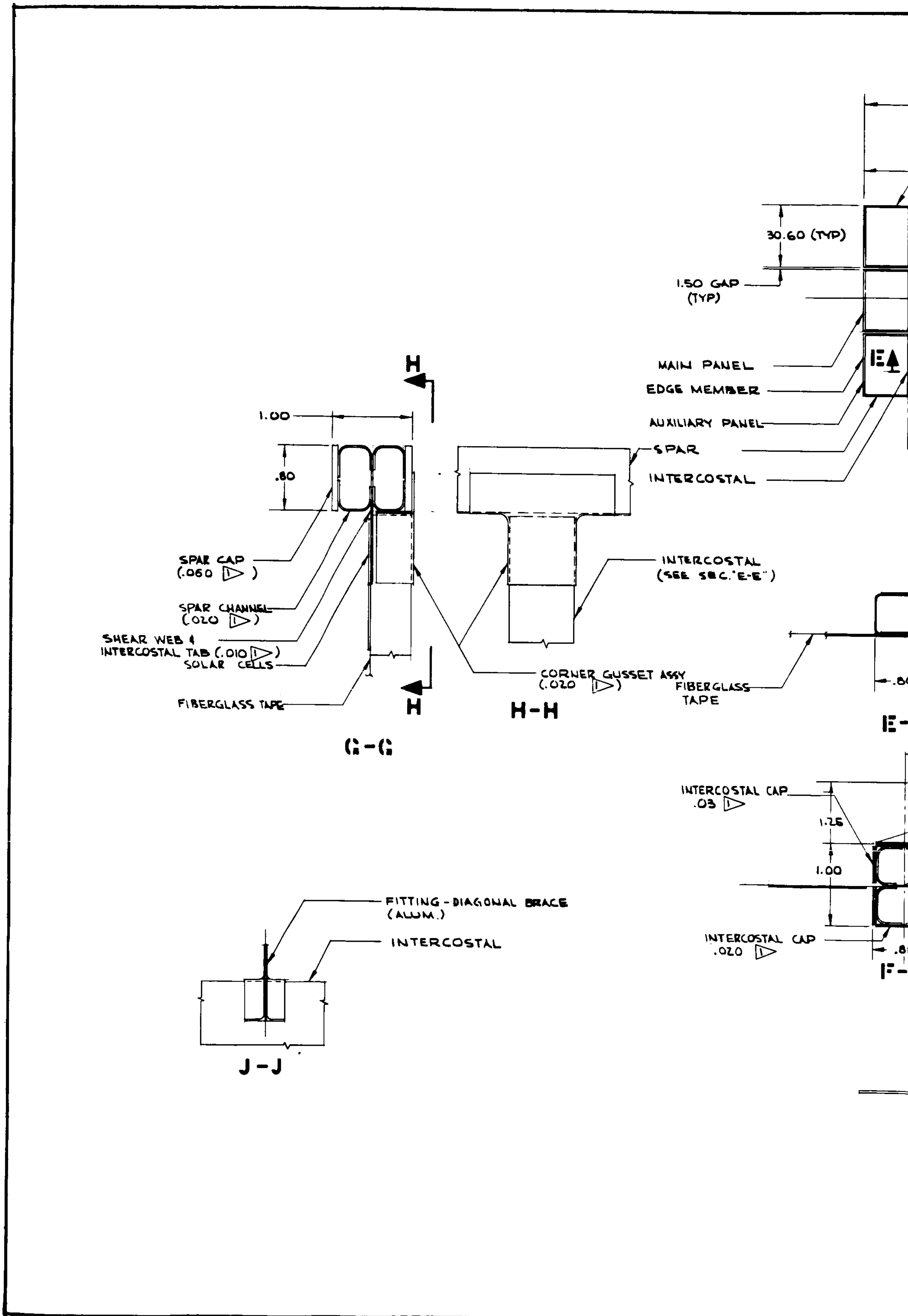

30.60 (TYP)
1.50 GAP (TYP)
MAIN PANEL
EDGE MEMBER
AUXILIARY PANEL
SPAR
INTERCOSTAL
H
1.00
.80
SPAR CAP (.060 1)
SPAR CHANNEL (.020 1)
SHEAR WEB & INTERCOSTAL TAB (.010 1)
SOLAR CELLS
FIBERGLASS TAPE
H
G-G
INTERCOSTAL (SEE SEC."E-E")
CORNER GUSSET ASSY (.020 1)
H-H
FIBERGLASS TAPE
INTERCOSTAL CAP .03 1
1.25
1.00
INTERCOSTAL CAP .020 1
FITTING-DIAGONAL BRACE (ALUM.)
INTERCOSTAL
J-J

Fig 6.2.2 ①

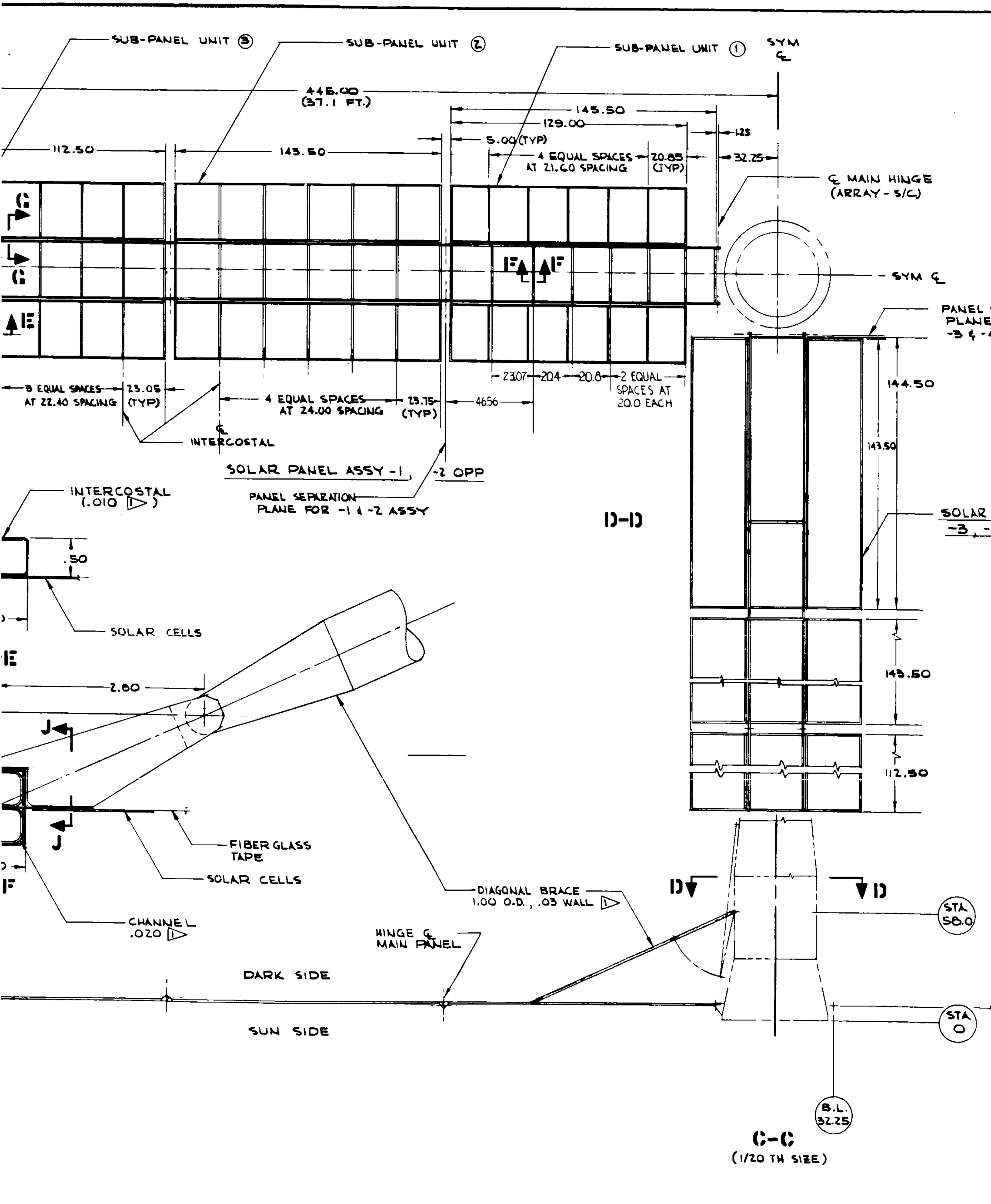

SUB-PANEL UNIT 3
SUB-PANEL UNIT 2
SUB-PANEL UNIT 1
SYM ℄
445.00 (37.1 FT.)
145.50
129.00
112.50
143.50
5.00 (TYP)
.125
4 EQUAL SPACES AT 21.60 SPACING
20.85 (TYP)
32.25
℄ MAIN HINGE (ARRAY - S/C)
SYM ℄
PANEL PLANE -3 & -
3 EQUAL SPACES AT 22.40 SPACING
23.05 (TYP)
4 EQUAL SPACES AT 24.00 SPACING
23.75 (TYP)
46.56
23.07
20.4
20.8
2 EQUAL SPACES AT 20.0 EACH
144.50
℄ INTERCOSTAL
SOLAR PANEL ASSY -1, -2 OPP
PANEL SEPARATION PLANE FOR -1 & -2 ASSY
143.50
D-D
SOLAR -3, -
INTERCOSTAL (.010 1)
.50
SOLAR CELLS
E
2.80
J
J
F
143.50
112.50
FIBER GLASS TAPE
SOLAR CELLS
DIAGONAL BRACE 1.00 O.D., .03 WALL 1
D
D
STA 58.0
CHANNEL .020 1
HINGE ℄ MAIN PANEL
DARK SIDE
SUN SIDE
STA 0
B.L. 32.25
C-C
(1/20 TH SIZE)

Fig 6.2-2
(2)

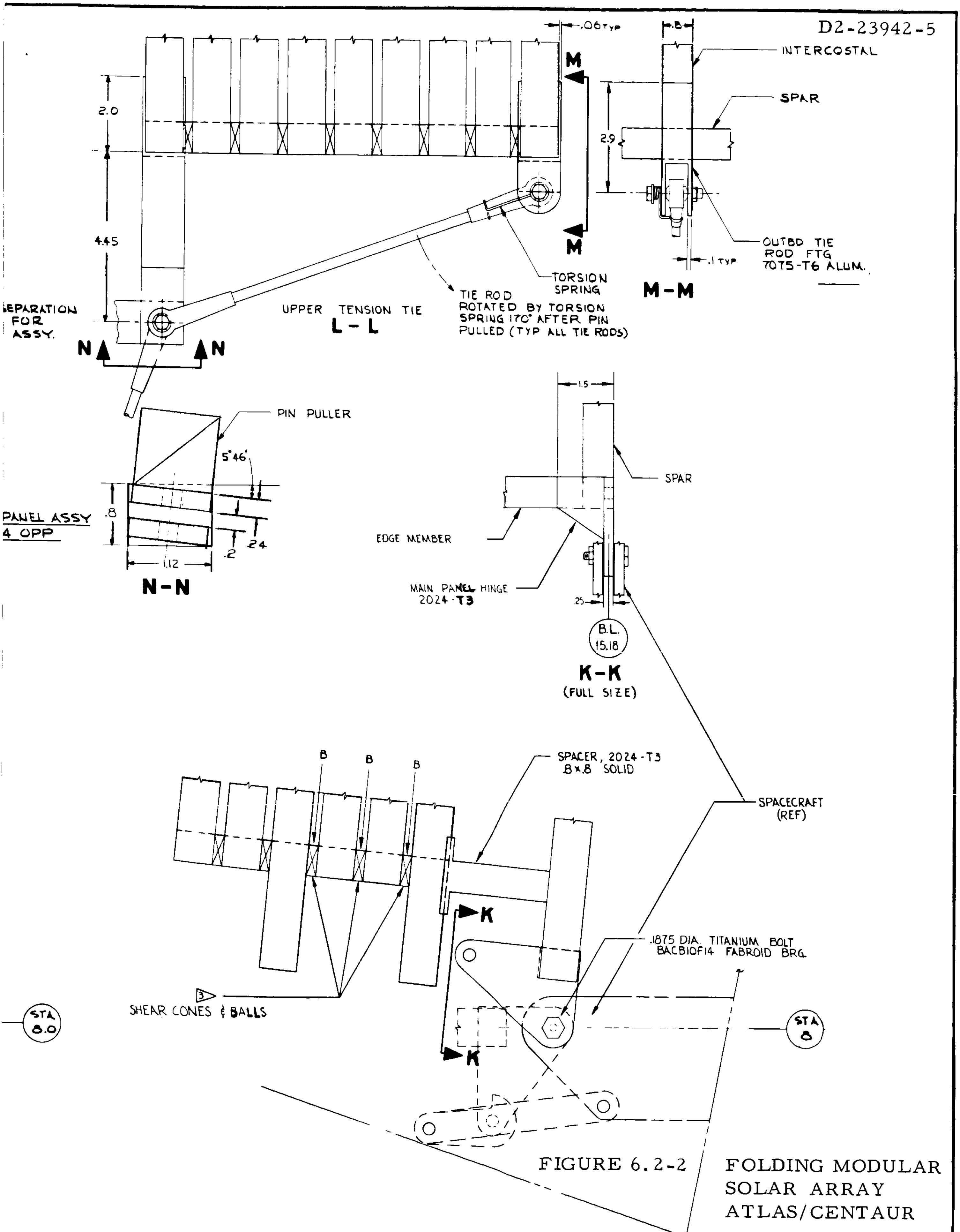


FIGURE 6.2-2

FOLDING MODULAR SOLAR ARRAY ATLAS/CENTAUR

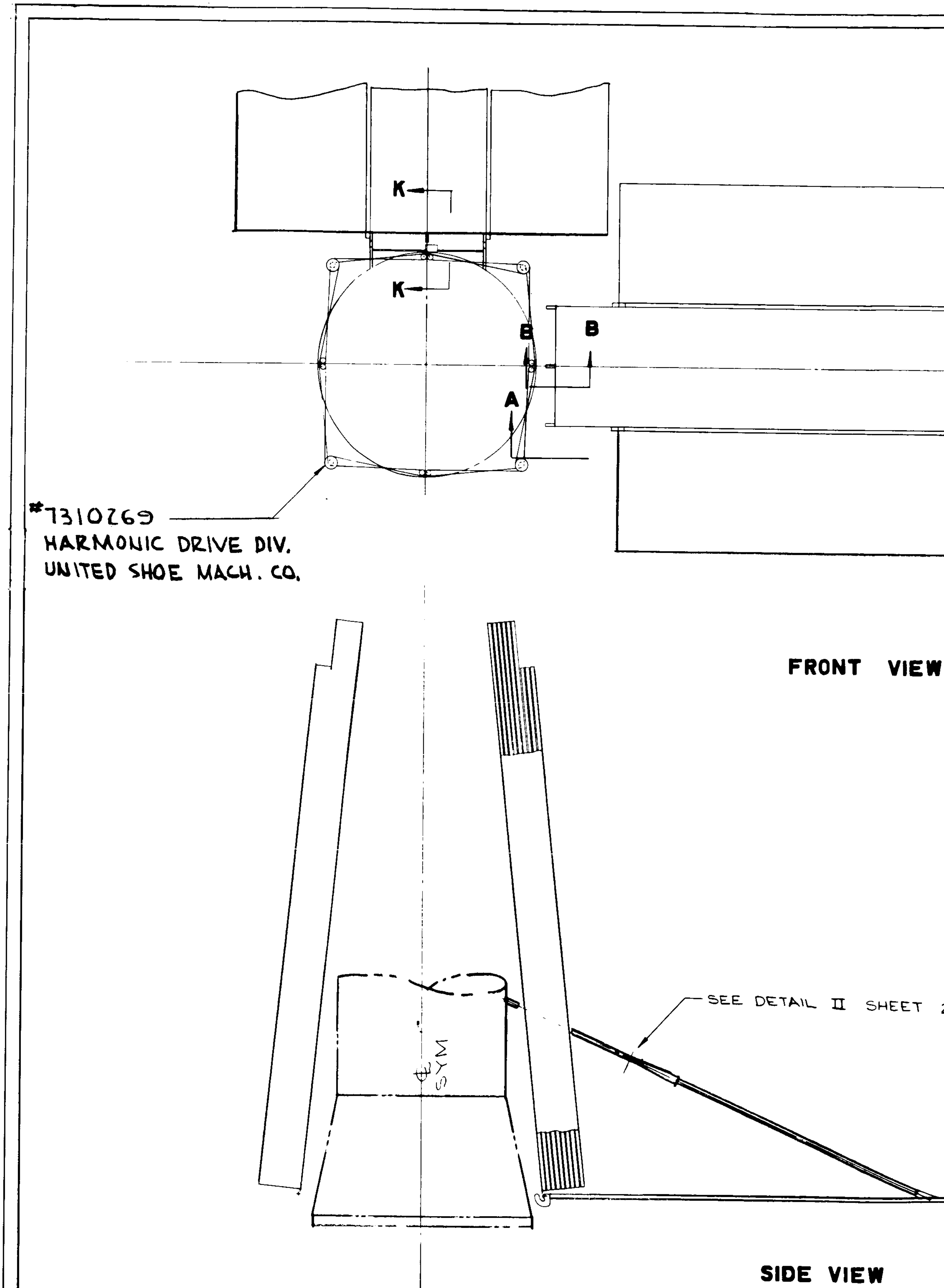

K
K
B
B
A
*7310269
HARMONIC DRIVE DIV.
UNITED SHOE MACH. CO.
FRONT VIEW
SEE DETAIL II SHEET
SYM
SIDE VIEW

Fig 6.2-3 (1)

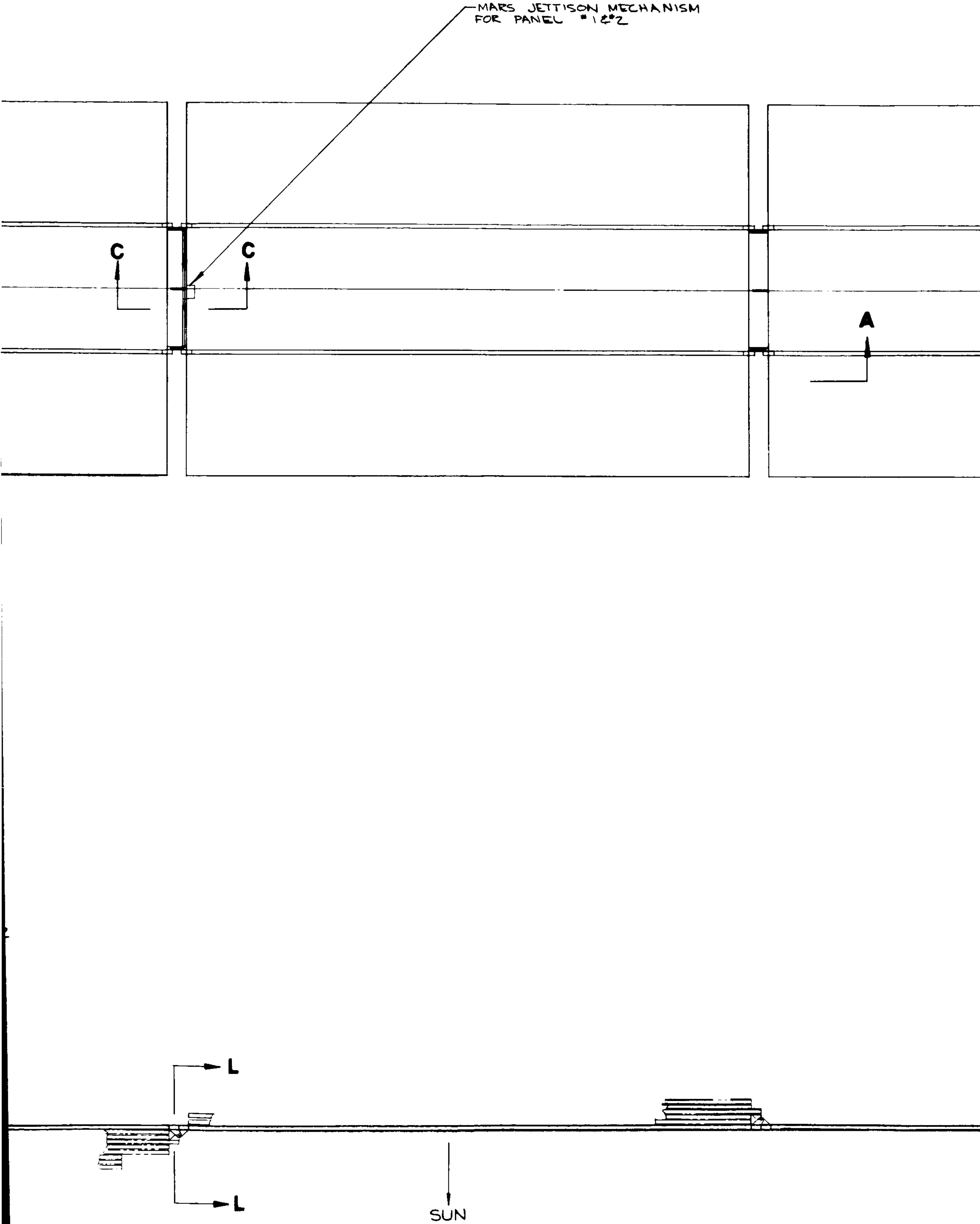
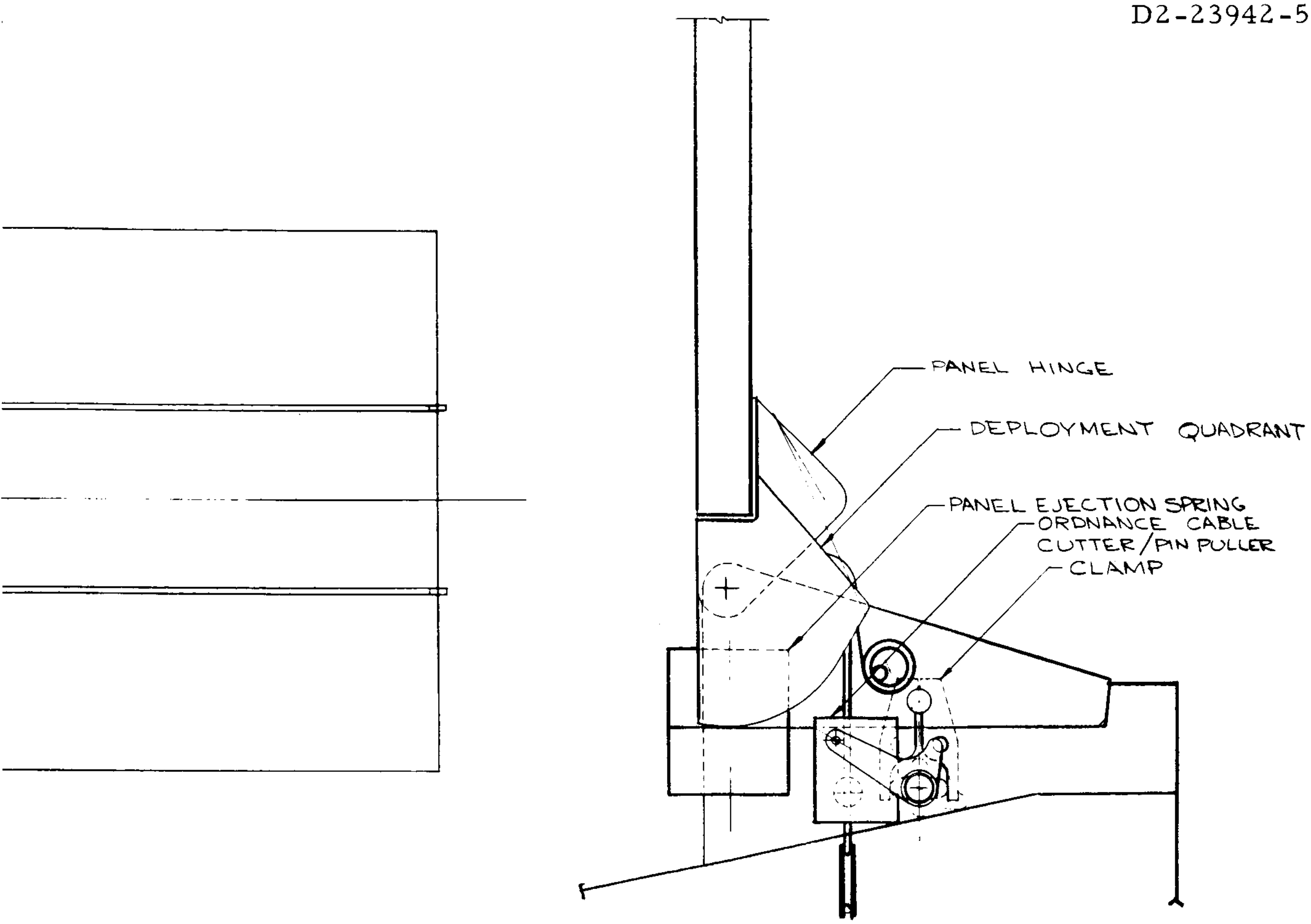


Fig 6.2-3 (2)



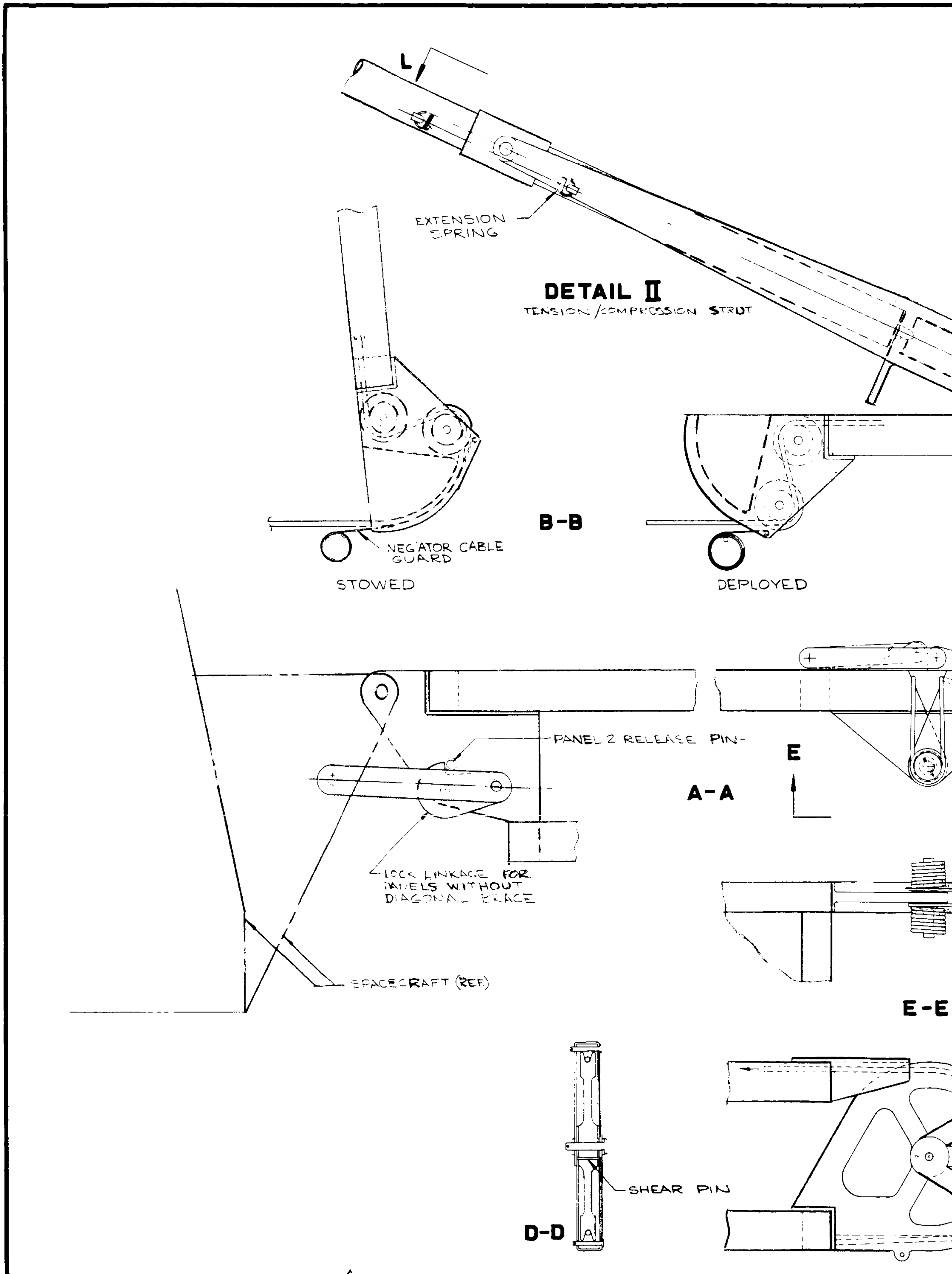
K - K

MARS JETTISON MECHANISM FOR PANELS #3 & #4. TORQUE TUBE DRIVE SPRINGS NOT SHOWN, SYSTEM IS SIMILAR TO SATURN/CENTAUR

SEE FIG. 6.2-4 FOR SECTION VIEWS NOT SHOWN.

FIGURE 6.2-3 MECHANISM INSTALLATION
10 KW SOLAR ARRAY
ATLAS/CENTAUR

3



L
EXTENSION SPRING
DETAIL II
TENSION/COMPRESSION STRUT
B-B
NEGATOR CABLE GUARD
STOWED
DEPLOYED
PANEL 2 RELEASE PIN
E
A-A
LOCK LINKAGE FOR PANELS WITHOUT DIAGONAL BRACE
SPACECRAFT (REF)
E-E
SHEAR PIN
D-D

Fig 6.2-4
(1)

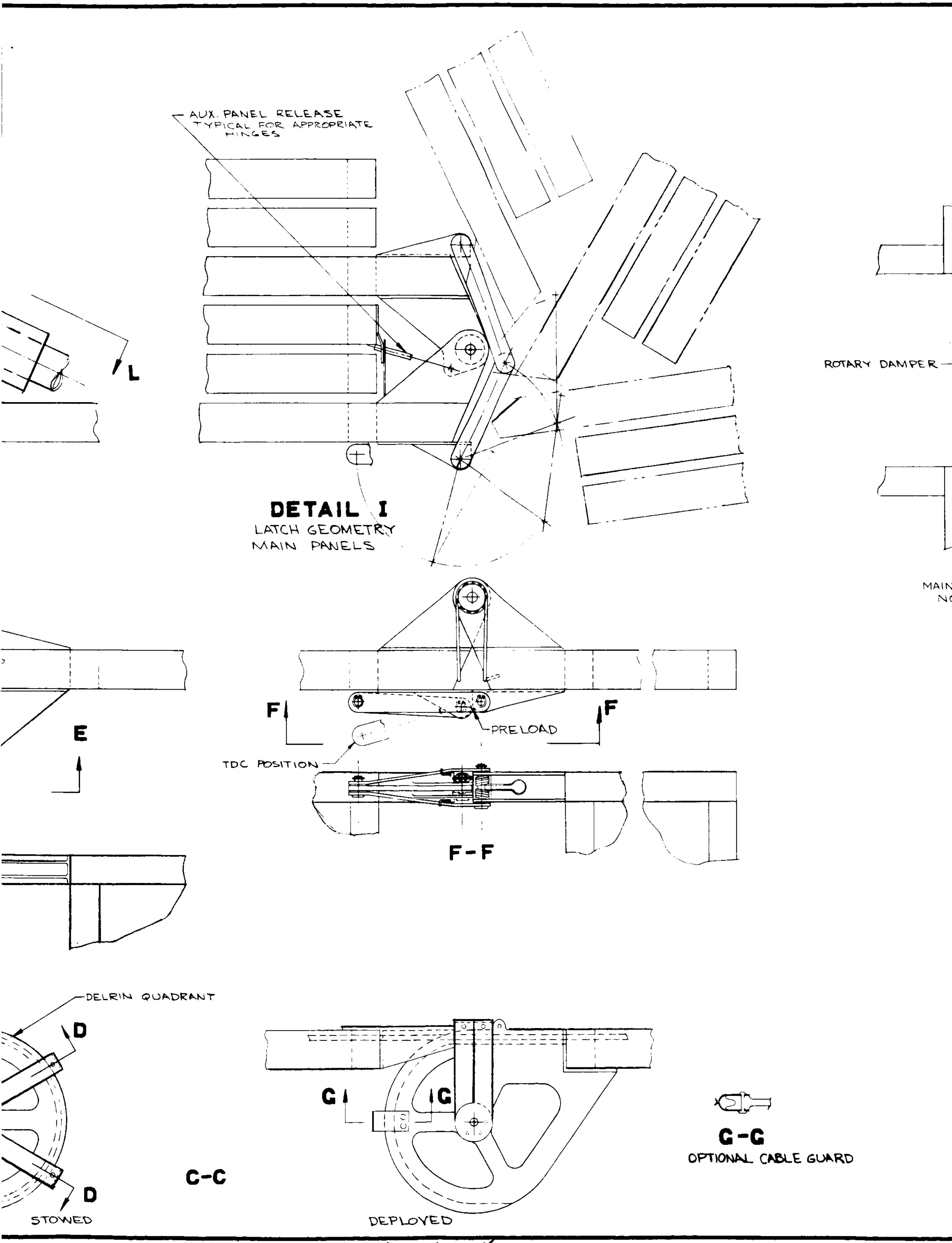
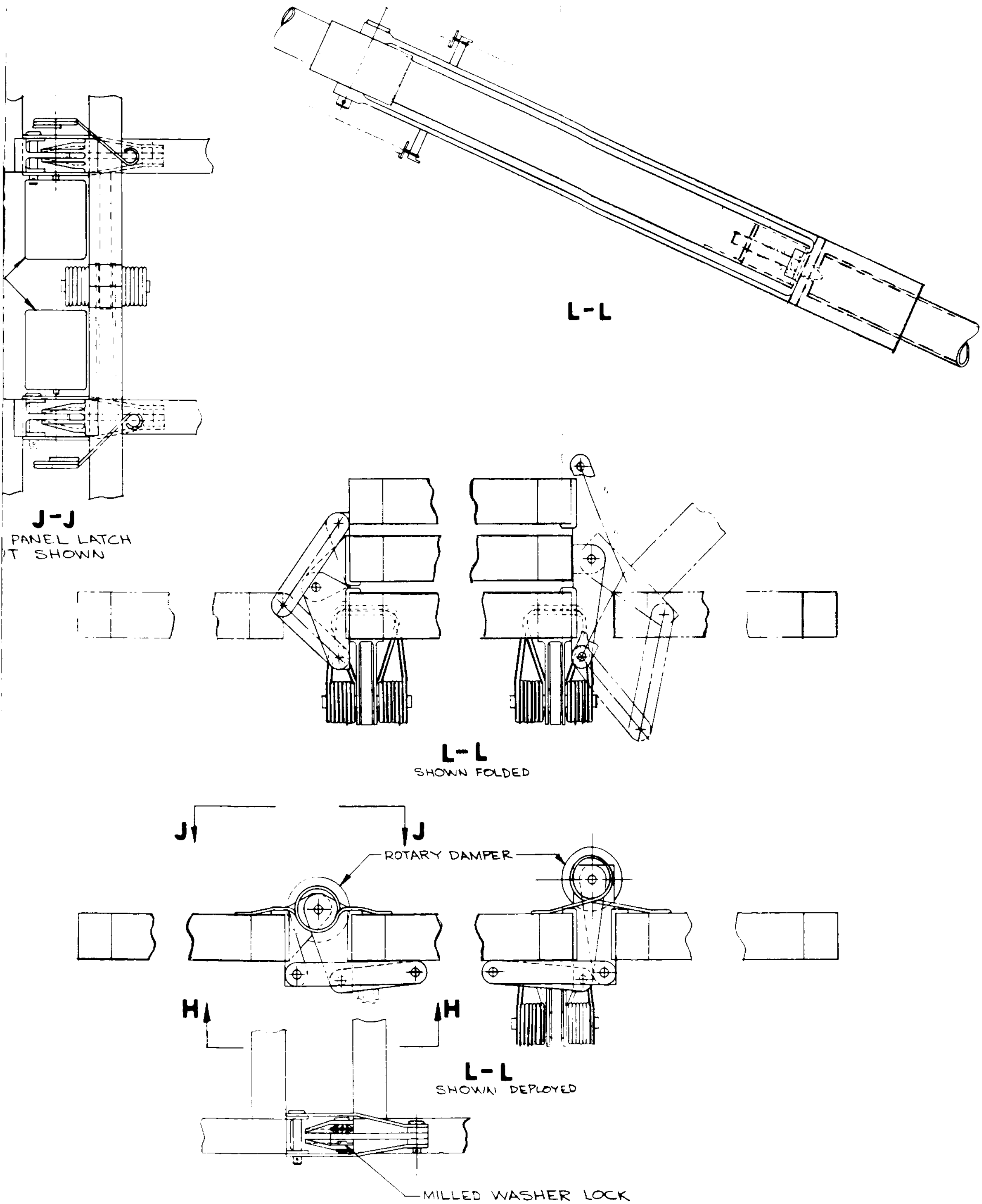

AUX. PANEL RELEASE
TYPICAL FOR APPROPRIATE HINGES
L
DETAIL I
LATCH GEOMETRY
MAIN PANELS
ROTARY DAMPER
E
F
F
PRELOAD
TDC POSITION
F-F
DELRIN QUADRANT
D
D
C-C
STOWED
G
G
DEPLOYED
G-G
OPTIONAL CABLE GUARD

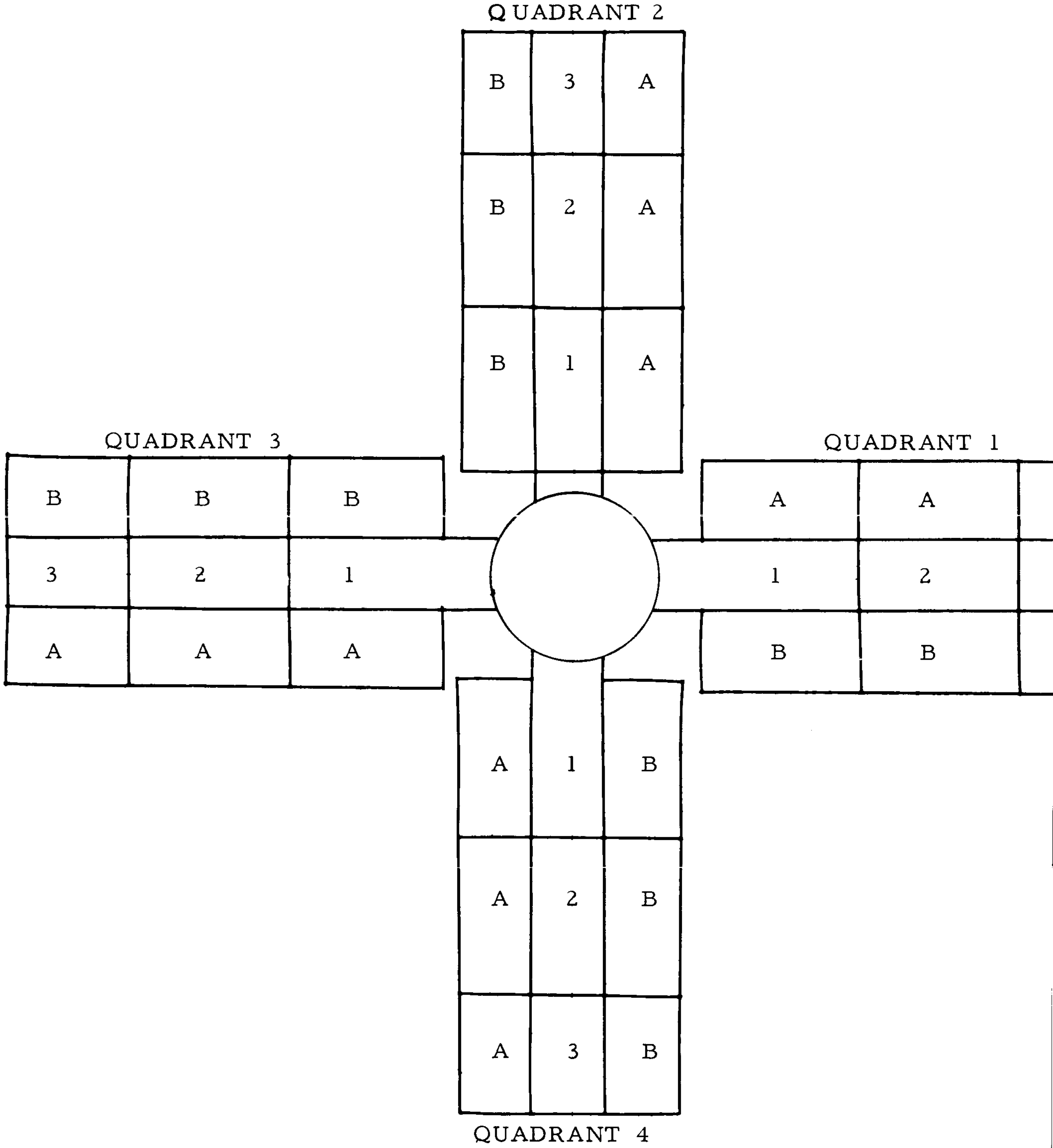
Fig 6.2-4 (2)



H-H
PRELOAD ON ATLAS AUX. PANELS NOT REQD.
LATCH SPRING NOT SHOWN

FIGURE 6.2-4 MECHANISM INSTALLATION
10 KW SOLAR ARRAY
ATLAS/CENTAUR

3

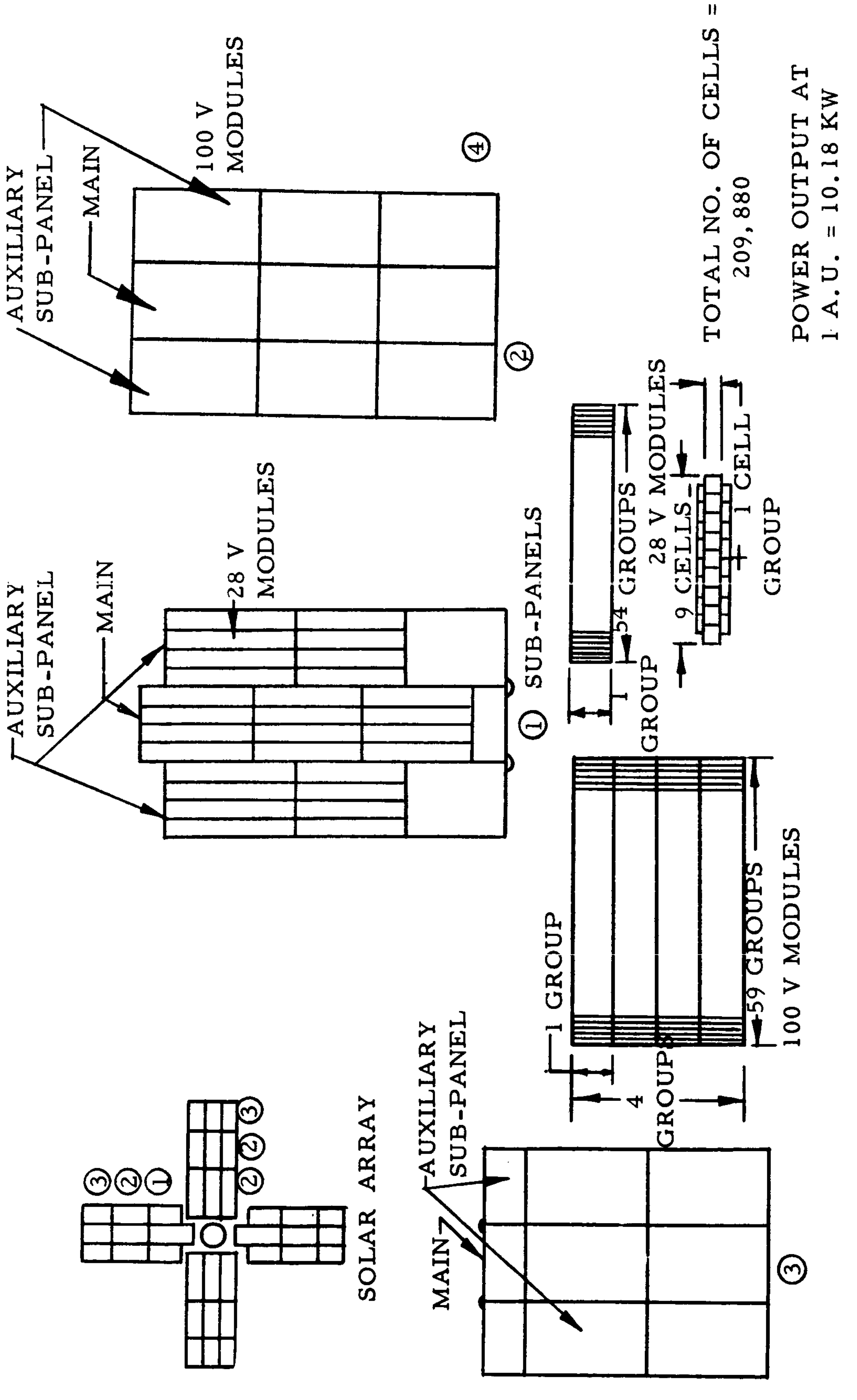

QUADRANT 2
B
3
A
B
2
A
B
1
A
QUADRANT 3
B
B
B
3
2
1
A
A
A
QUADRANT 1
A
A
1
2
B
B
A
1
B
A
2
B
A
3
B
QUADRANT 4

TIME	
0	Apply voltage to pin pullers and drive units. Rotate all folded panel assemblies 90°
70 sec	Latch joint #1, release panels #2 and #3 as an assembly
165 sec	Resease sub-panel 1A
207 sec	Latch joint #2, release sub-panel 3 and sub-panel 2A
240 sec	Release sub-panel 1B
282 sec	Latch sub-panel 2A, release sub-panel 3A
315 sec	Latch sub-panel 1B
344 sec	Latch panel #3, release sub-panel 3A
357 sec	Latch sub-panel 2B
419 sec	Latch sub-panel 3A, release sub-panel 3B
494 sec	Latch sub-panel 3B, deployment complete, Signal received.

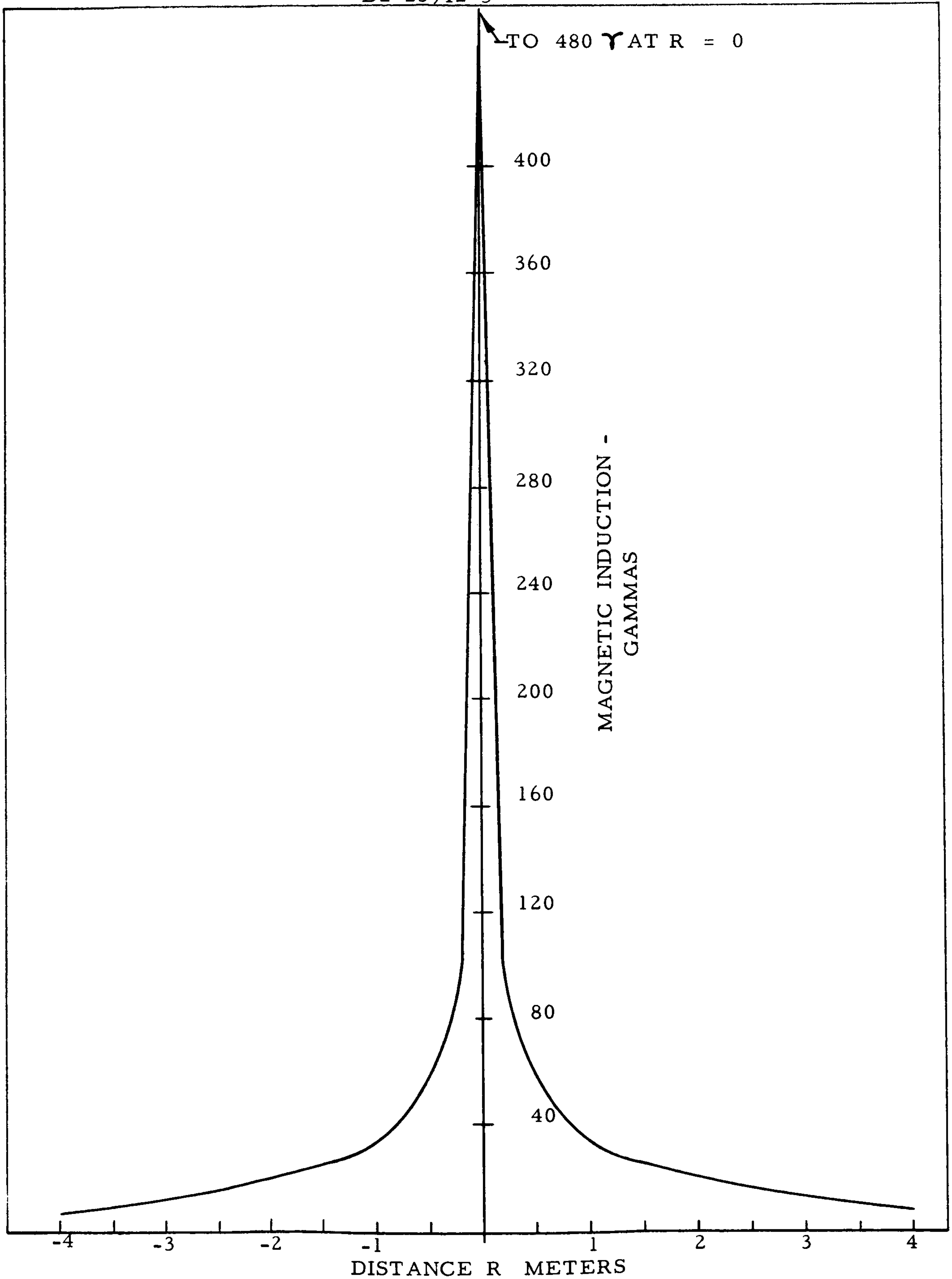
A
3
B

ATLAS/CENTAUR SOLAR ARRAY DEPLOYMENT SEQUENCE

FIGURE 6.2-5



SOLAR CELL LAYOUT
ATLAS/CENTAUR ARRAY
FIGURE 6.2.3-1



MAGNITUDE OF RESULTANT MAGNETIC INDUCTANCE VECTOR OF LONGITUDINAL & TRANSVERSE CURRENTS VS DISTANCE R

FIGURE 6.2.3-2

6.2.4 Electrical Conductors

The conductors and interconnectors for the power, jettison and the command and monitor system and their interfaces are defined in this section.

6.2.4.1 System Description - Power Busses (Figure 6.2.4-1)

6.2.4.2 General Configuration - 100 Volt and 28 Volt Systems

The power bus system for the Atlas/Centaur is identical to the Saturn/Centaur except for scale. Wire diameters, crossover busses and power connectors have been specified according to electrical power requirements.

6.2.4.3 Jettison Connector

The Atlas/Centaur configuration requires one jettison connector per sub-panel. This connector is identical to the Saturn part but the Atlas requires one half the number of parts. Figure 6.2.4-1 shows the location for jettison connectors as installed on the -1 and -2 solar cell panel assemblies. On the -3 and -4 panel assemblies the jettison connectors are located at the panel/spacecraft separation plane.

6.2.4.4 Wiring Provisions Command and Monitor

Wiring provisions for command and monitor are similar to the Saturn configuration.

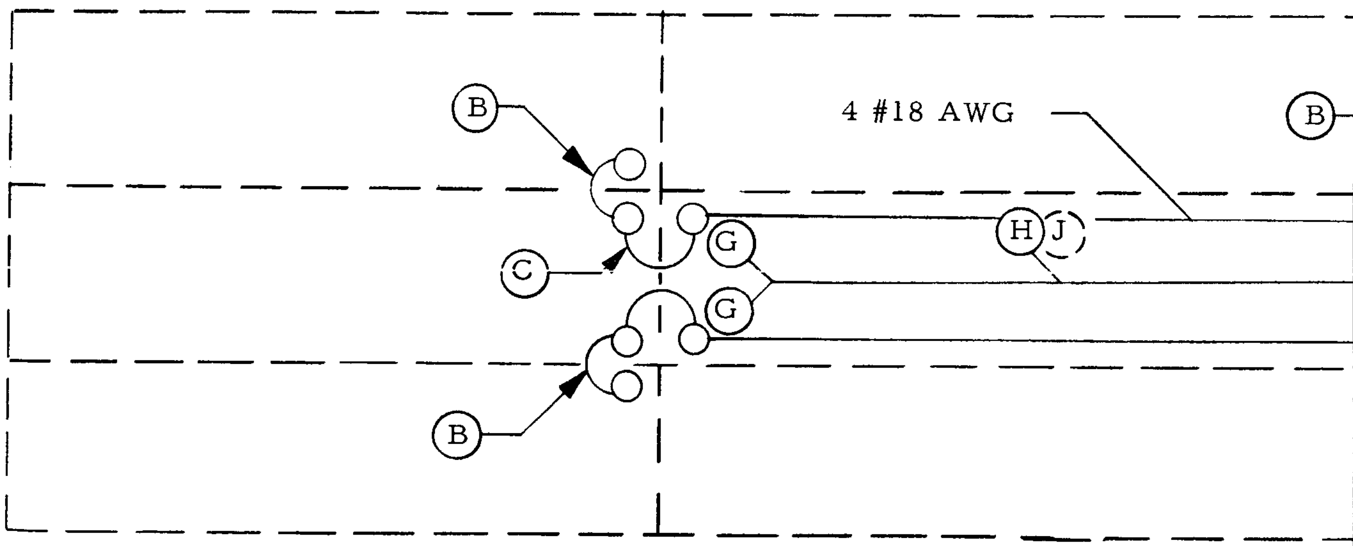
6.2.4.5 Electrical Interfaces (Figure 6.2.4-2)

Smaller electrical connectors have been chosen for the Atlas/Centaur configuration for monitor and control functions and for electrical power. The connectors specified for Saturn are used for the Atlas control - squib firing and array deployment motor. The electrical interfaces for Atlas/Centaur are the same as Saturn/Centaur except as noted above.

6.2.5 Dynamic Analysis

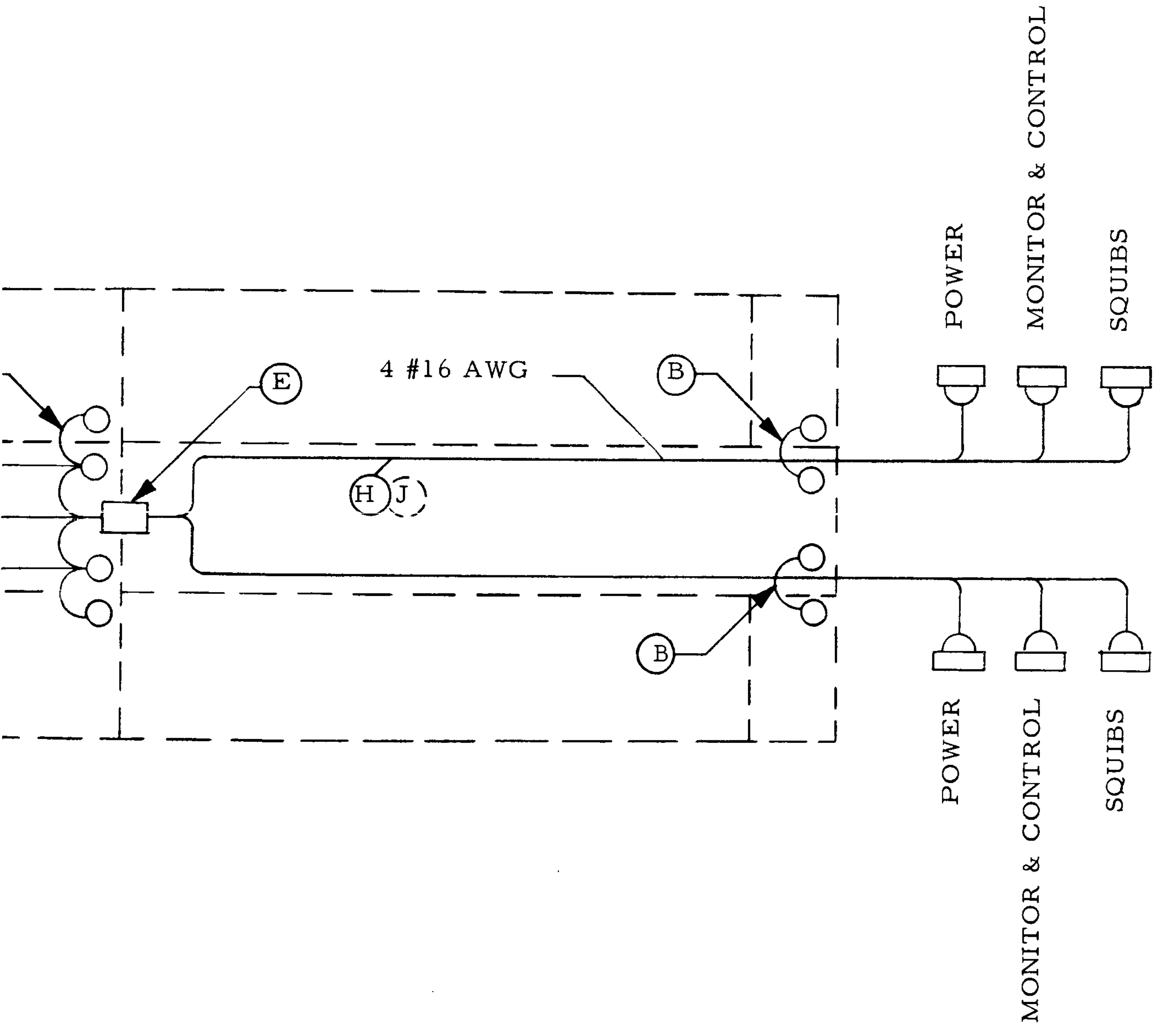
The Atlas/Centaur solar array was analyzed for its dynamic characteristics and for the loads imposed upon the mounting hinges during imposition of a launch environment.

The first two resonant frequencies of the stowed array in normal vibration are: $f_1 = 12.7$ cps and $f_2 = 50.8$ cps. For the loading normal to the plane the loading was assumed to be quasi static where the 1.5 "g" rms input was magnified by Q = 15 for a mean quasi static acceleration of 22 "g". For such a loading the hinge loads at the base are 1310 pounds/hinge. For an in plane side shear loading the load between quadrants is 1860 pounds.



LEGEND

B	Bus Crossover - Outside Hinge Configuration
C	Buss Crossover - Main Panel to Main Panel
E	Jettison Connector
G	Panel Deployed and Latched
H	Temperature Sensor - Upper
J	Temperature Sensor - Lower



BUSSES AND CONDUCTORS
PHYSICAL LOCATION & IDENTIFICATION
ATLAS/CENTAUR

FIGURE 6.2.4-1

	CONTACT IDENTIFICATION
SQUIB NO. 1, CIRCUIT A NEG	A
SQUIB NO. 1, CIRCUIT A POS	B
SQUIB NO. 1, CIRCUIT B NEG	C
SQUIB NO. 1, CIRCUIT B POS	D
SQUIB NO. 2, CIRCUIT A NEG	E
SQUIB NO. 2, CIRCUIT A POS	F
SQUIB NO. 2, CIRCUIT B NEG	G
SQUIB NO. 2, CIRCUIT B POS	H
SHIELD - SQUIB NO. 1	J
SHIELD - SQUIB NO. 2	K

CANNON ELECTRIC CO.

PV6R12B - 10 PNS
ELECTRIC PLUG

CONTROL - SQUIB FIRING

INTERFACE

	CONTACT
SIGNAL - PANEL DEPLOYED AND LATCHED (CIRCUIT A)	
SIGNAL - PANEL DEPLOYED AND LATCHED (CIRCUIT B)	
JETTISON ACCOMPLISHED (CIRCUIT A)	
JETTISON ACCOMPLISHED (CIRCUIT B)	
SPARE	

CANNON ELECTRIC COMPANY

PV6R16B-26 PNS

MONITOR AND CONTROL FUNCTIONS

IDENTIFICATION

THRU

THRU

THRU

INTERFACE

CONTACT IDENTIFICATION	
A	MOTOR LEADS
B	
C	
D	
E	
F	MOTOR LEADS

INTERFACE

CANNON ELECTRIC
PV6R10B - 6 PNS

<u>POWER CONNECTOR -</u>
<u>ARRAY DEPLOY MOTOR</u>

	CONTACT NUMBER
NOT USED	1
NOT USED	2
NOT USED	3
NOT USED	4
100 - VOLT BUS - NEG	5
100 - VOLT BUS - POS	6
100 - VOLT BUS - NEG	7
100 - VOLT BUS - POS	8
100 - VOLT BUS - NEG	9
100 - VOLT BUS - POS	10
28 - VOLT BUS - NEG	11
28 - VOLT BUS - POS	12
NOT USED	13
NOT USED	14

INTERFACE

CANNON ELECTRIC CO.

KV6R18 - 14 PNS
ELECTRIC PLUG

<u>ELECTRICAL POWER</u>

ATLAS/CENTAUR
ELECTRICAL INTERFACE
PLUG SCHEMATIC & IDENTIFICATION

FIGURE 6.2.4-2

In the deployed configuration the array will exhibit fundamental bending frequencies of: $f_1 = 0.25$ cps and $f_2 = 1.5$ cps.

The mass moment of inertia of the array about the spacecraft centerline is 1934 slug ft^2.

The hinge loads during deployment have been calculated to be 170 ft/lb/sec on the spacecraft to array hinge. The portion of the array retained for secondary power generation about Mars will have fundamental resonant frequencies of: $f_1 = 0.6$ cps and $f_2 = 2.4$ cps.

The response to acoustic loading is expected to be 0.14 ft/sec^2 rms acceleration and 2.64×10^{-5} inches displacement. The response of the array to random loads have not been computed because of the band limited inputs and the frequency mismatch between array and input loading. Random load response is expected to be negligible.

6.2.6 Stress Analysis

The structural design analysis of the Atlas/Centaur solar array quadrant given in the fourth bi-monthly progress report (D2-23942-4, Appendix A) was reviewed for purposes of stress analysis. Since the primary concern of the program has been the Saturn IB/Centaur solar array, little additional stress analysis was justified for the Atlas/Centaur array.

The geometry of the Atlas/Centaur array and its support configuration are such that the in plane shear load condition is even more critical for this array than for the Saturn IB/Centaur array. Consequently, the conclusions reached in the Saturn IB/Centaur stress analysis (6.1.7) are applicable to this design as well.

The high stresses extant in the array due to in plane shear loading can be readily alleviated by several different means. The most satisfactory method of relieving these stresses could be determined by a trade study conducted during any subsequent contract.

6.2.7 Thermal Analysis

Thermal analysis of the Atlas/Centaur launched solar array was reported in detail in bi-monthly reports D2-23942-2 and D2-23942-4. The analyses are summarized in this section of the final report.

At a solar intensity of 1 Astronomical Unit the expected temperature gradient of the spars is expected to be 7^oF. This will result in less than a 2 degree tip deflection of the deployed array. The stabilized cell temperature is expected to be 104^oF. This is based on the use of a 3 mil layer of

Laminar X-500 applied to the dark side of the array. An emittance value of 0.95 was used in the analysis for this coating.

Figure 6.2.7-1 illustrates the thermal model used. Table 6.2.7-2 summarizes the results for the 29 nodes investigated at several locations in space.

6.2.8 Reliability Evaluation

Included in this section is the reliability summary for the mechanical and electrical portions of the 10 kilowatt solar array to be installed on an Atlas/Centaur launched spacecraft. To meet the reliability goals given in Section 2, redundant elements were incorporated in the same areas of the design as in the Saturn IB/Centaur array.

6.2.8.1 Deployment Mechanism Reliability

The deployment mechanism reliability is estimated to be .9992 for 100 percent array deployment compared to the goal of .999. The approach to the analysis was the same as that for the Saturn IB/Centaur system.

6.2.8.2 Jettison Reliability

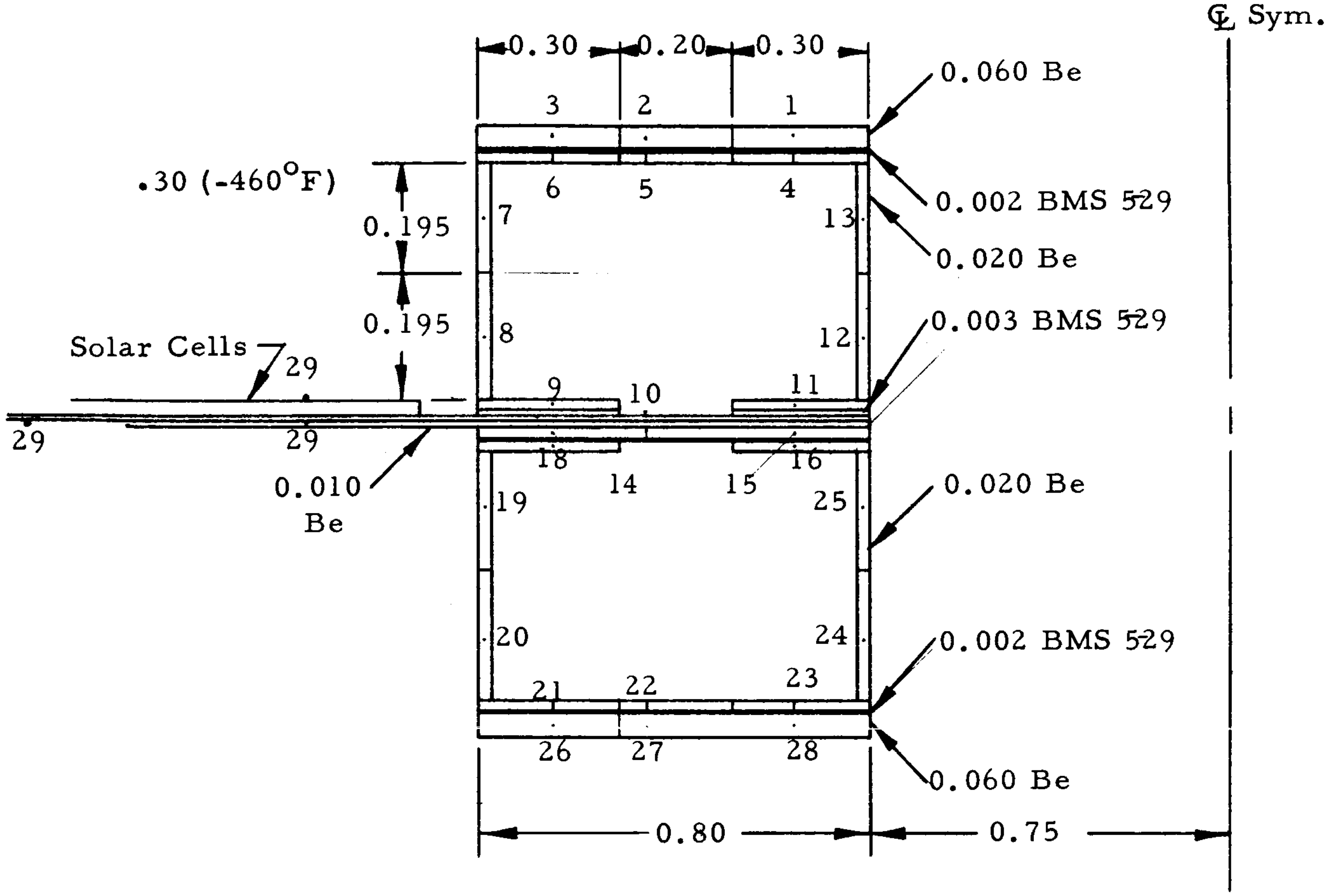
The probability of successful jettison of the 100 volt portion of the array is .999996 which exceeds the goal of .9999. A single analysis served both the Atlas/Centaur and Saturn IB/Centaur arrays.

6.2.8.3 Electrical Busses and Bus Connection Reliability

As in the case of the Saturn IB/Centaur array the electrical bus system reliability far exceeds the goal of .998. The bus systems for the two arrays are the same except in size.

6.2.8.4 Solar Cell Array Reliability

Figure 6.2.8-1 shows the electrical system reliability versus maximum expected loss of electrical power output for both the 100 volt and 28 volt power sources. As with the Saturn IB/Centaur array, the reliability of the solar cells in the array is the predominant factor determining the electrical system reliability. The same type of analysis was carried out for both Atlas/Centaur and the Saturn IB/Centary arrays. In addition to size, the principal difference in the arrays is that the Atlas/Centaur array has nine cells in parallel per group while the Saturn IB/Centaur array has seven.



THERMAL DATA USED IN ANALYSIS

MATERIAL	THERMAL CONDUCTIVITY
Fiberglass	0.02 BTU-In/In^2-Hr-°F
Beryllium	8 BTU-In/In^2-Hr-°F
BMS 529	0.00833 BTU-In/In^2-Hr-°F
BMS 525	0.0125 BTU-In/In^2-Hr-°F

FOR BERYLLIUM:

Solar Absorptance = 0.48
Solar Reflectance = 0.82
Emittance = 0.09, Exterior and Interior Surfaces
High-Emittance Paint (ϵ = 0.95) is considered to be used on external surfaces for lower half section of spar.

SOLAR HEAT INPUTS:

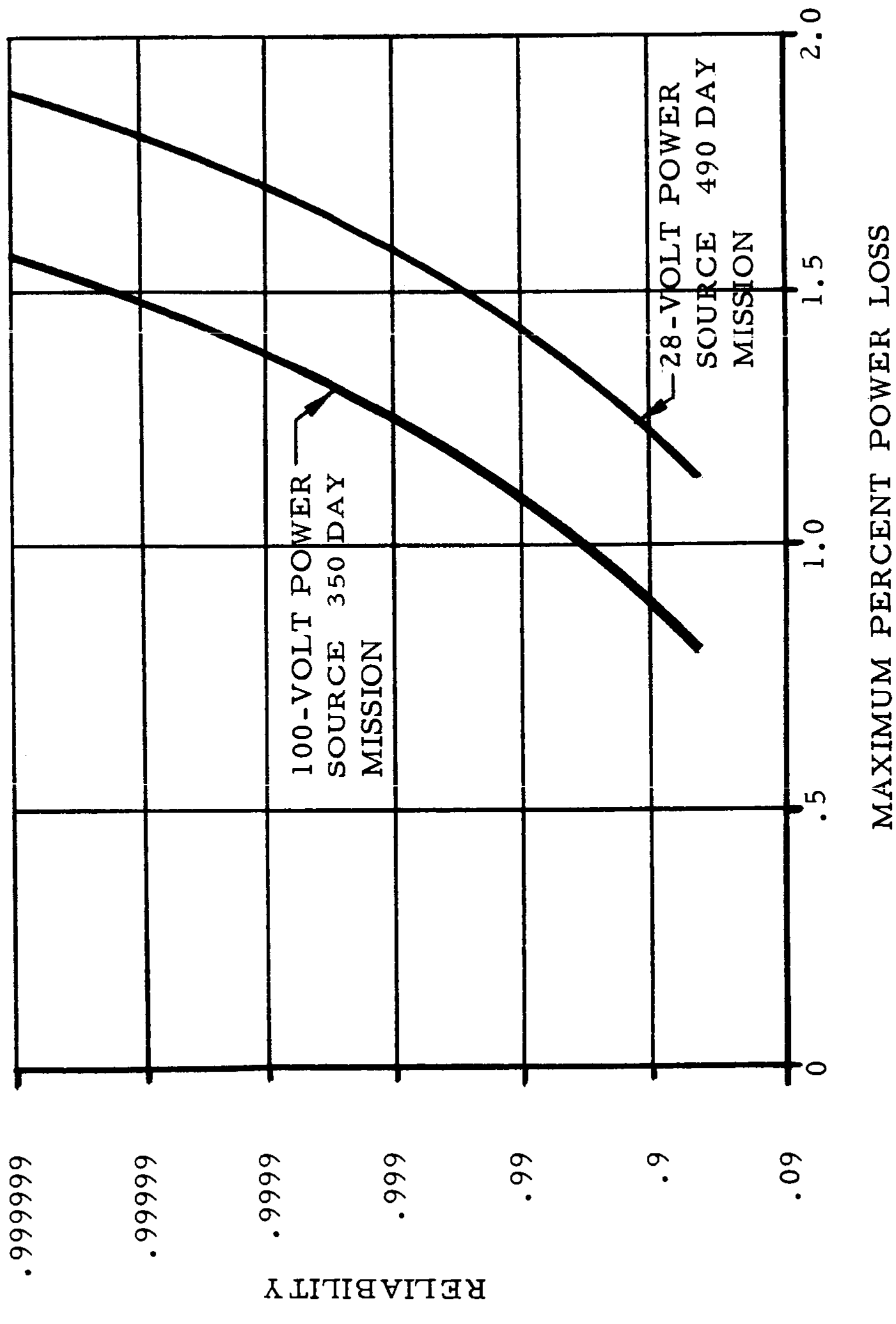
LOCATION	SOLAR CONSTANT
Venus	267 Milliwatts/sq. cm.
1/2 Way Venus-Earth	189 Milliwatts/sq. cm.
Earth	140 Milliwatts/sq. cm.
1/2 Way Earth-Mars	88 Milliwatts/sq. cm.
Mars	60 Milliwatts/sq. cm.

THERMAL MODEL
ATLAS/CENTAUR ARRAY
FIGURE 6.2.7-1

NODE	Venus °F	1/2 Way Venus - Earth °F	Earth °F	1/2 Way Earth - Mars °F	Mars °F
1	173.62	127.14	90.95	37.25	-1.35
2	173.68	127.19	90.98	37.27	-1.32
3	173.66	127.18	90.98	37.27	-1.31
4	172.89	126.62	90.56	37.00	-1.50
5	173.24	126.87	90.75	37.13	-1.41
6	173.01	126.71	90.63	37.06	-1.45
7	171.46	125.61	89.83	36.59	-1.76
8	170.22	124.73	89.17	36.19	-2.03
9	168.50	123.49	88.25	35.63	-2.41
10	166.19	121.81	87.98	34.82	-2.99
11	168.41	123.39	88.14	35.51	-2.53
12	170.15	124.64	89.08	36.09	-2.14
13	171.50	125.62	89.81	36.54	-1.82
14	166.24	121.87	87.04	34.89	-2.92
15	166.12	121.75	86.91	34.75	-3.05
16	165.22	121.09	86.41	34.43	-3.28
17	166.18	121.81	86.98	34.82	-2.98
18	165.41	121.29	88.61	34.64	-3.07
19	163.75	120.11	85.75	34.14	-3.38
20	162.67	119.32	85.14	33.75	-3.66
21	161.50	118.41	88.41	33.22	-4.10
22	161.27	118.23	84.25	33.10	-4.21
23	161.41	118.31	84.31	33.12	-4.20
24	162.36	119.00	84.82	33.42	-3.99
25	163.41	119.76	85.40	33.79	-3.74
26	161.03	118.05	84.12	33.01	-4.26
27	160.99	118.02	84.08	32.98	-4.29
28	161.00	118.01	84.08	32.97	-4.30
29 Solar Cell	206	150	104	40	-7

For Nodes 19-20 and 24-28, ϵ = 0.95 at exterior surface, ϵ = 0.09 at interior surfaces. For all other nodes on spar, ϵ = 0.09 at both exterior and interior surfaces. (Results given to 1/100 °F merely to show the small temperature difference existing between adjacent nodes.)

TEMPERATURE DISTRIBUTION
FOLDING MODULAR ARRAY SPAR
ATLAS/CENTAUR
TABLE 6.2.7-2



ELECTRICAL SYSTEM
RELIABILITY - 10 KW ARRAY

FIGURE 6.2.8-1

6.2.9 Weight Analysis - Atlas/Centaur

A summary of the weights for the Atlas/Centaur configuration are shown below in pounds per square foot and total array weight. The increases in cover glass, adhesive, cell, and connector weight per square foot over that of the Saturn/Centaur configuration is due to a more full use of available substrate. Frame, mechanisms, and electrical connections increased in terms of pounds per square foot due, in part, to envelope and minimum gauge limitations and partly because the weights of some items are not primarily area dependent. A more detailed breakdown of the Atlas/Centaur configuration weights is included in bi-monthly report number 4, Document D2-23942-4.

Cover Glasses	46.33 Pounds	.0460 Pounds/sq. ft.
Adhesives	16.67	.0165
Cells	98.64	.0979
Connectors	16.84	.0168
Substrate	15.51	.0154
Thermal Coating	27.79	.0276
Frame	165.49	.1641
Mechanisms	82.36	.0817
Electrical Connections	48.49	.0481
	518.12 Pounds	.5141 Pounds/sq. ft.

7.0 PROGRAM PLAN

The development of a Program Plan for a full scale prototype solar cell array of the 20 Watt Per Pound class, and adaptable for the Saturn IB and/or Atlas/Centaur launched spacecraft, is based on the present state-of-the-art. It is likely that methods for beryllium and titanium forming and handling, and various bonding techniques for solar cell utilization will be more complete by the time this program would be implemented. However, this planning is predicated on our present knowledge in this field. Also, it is assumed that a contractor selected for this program will have, as a part of his facilities, the necessary equipment and capital assets available to accomplish the task. Therefore, this study has not included General Facilities as an overall part of the cost. Major items that would be peculiar to this program, however, have been discussed and costed separately.

This study has also included the size and type of a complete manufacturing and testing facility that will be necessary to handle a solar array program of this magnitude. This information and the approximate cost is shown in Section 7.6.

7.1 MASTER PHASING SCHEDULES

Master Phasing Schedules are shown for both the Saturn IB/Centaur solar cell array, Figure 7.1-1, and the Atlas/Centaur solar cell array, Figure 7.1-2. Both programs are very similar and differ only in the length of time required for each and the size of the program and cost.

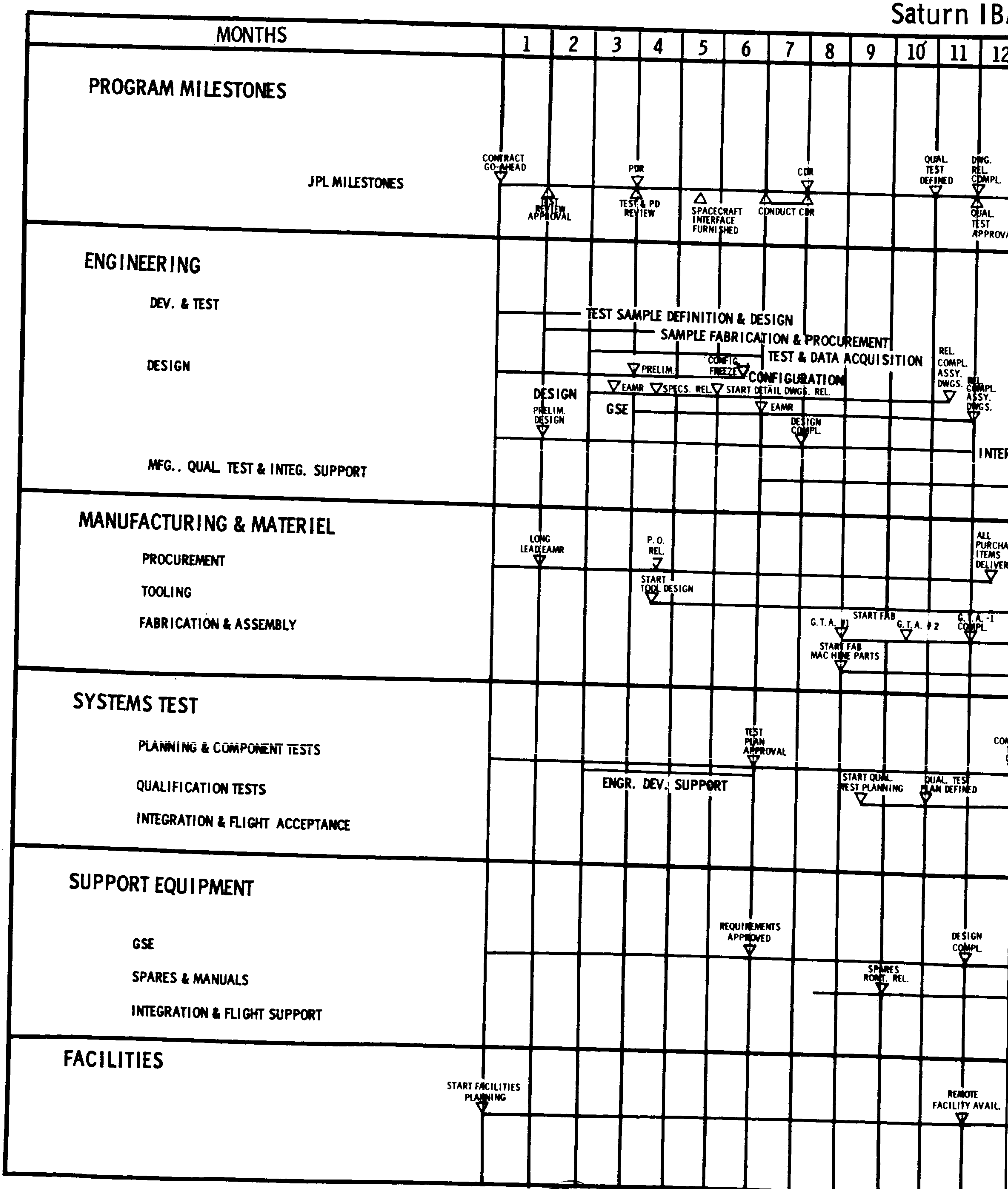
The Engineering effort would be essentially the same for either program, and, therefore, is shown the same on each phasing chart. From a program go-ahead until a flight ready status, a program for the Saturn IB/Centaur shows 32 months. The Atlas/Centaur program would take 29 months.

The Manufacturing effort and the Materiel requirements for this type of a program is by far the most costly portion. As shown on the Master Phasing Schedules, the length of time necessary to complete the manufacturing is the pacing item for the program. The manufacturing plan is described in Section 7.5.

Close coordination between the Manufacturing and the Ground Support Equipment Groups reveals the fact that much of the support equipment and the tooling can be the same. This has been considered in our study. The support equipment list is shown in Section 7.4.

The Systems Test Schedule and activity is shown on Master Phasing Schedules and is also described in the test plan detailed in Section 7.3.

MASTER PHAS
FULL SCALE SC
Saturn IB



NG SCHEDULE

AR CELL ARRAY

entaur

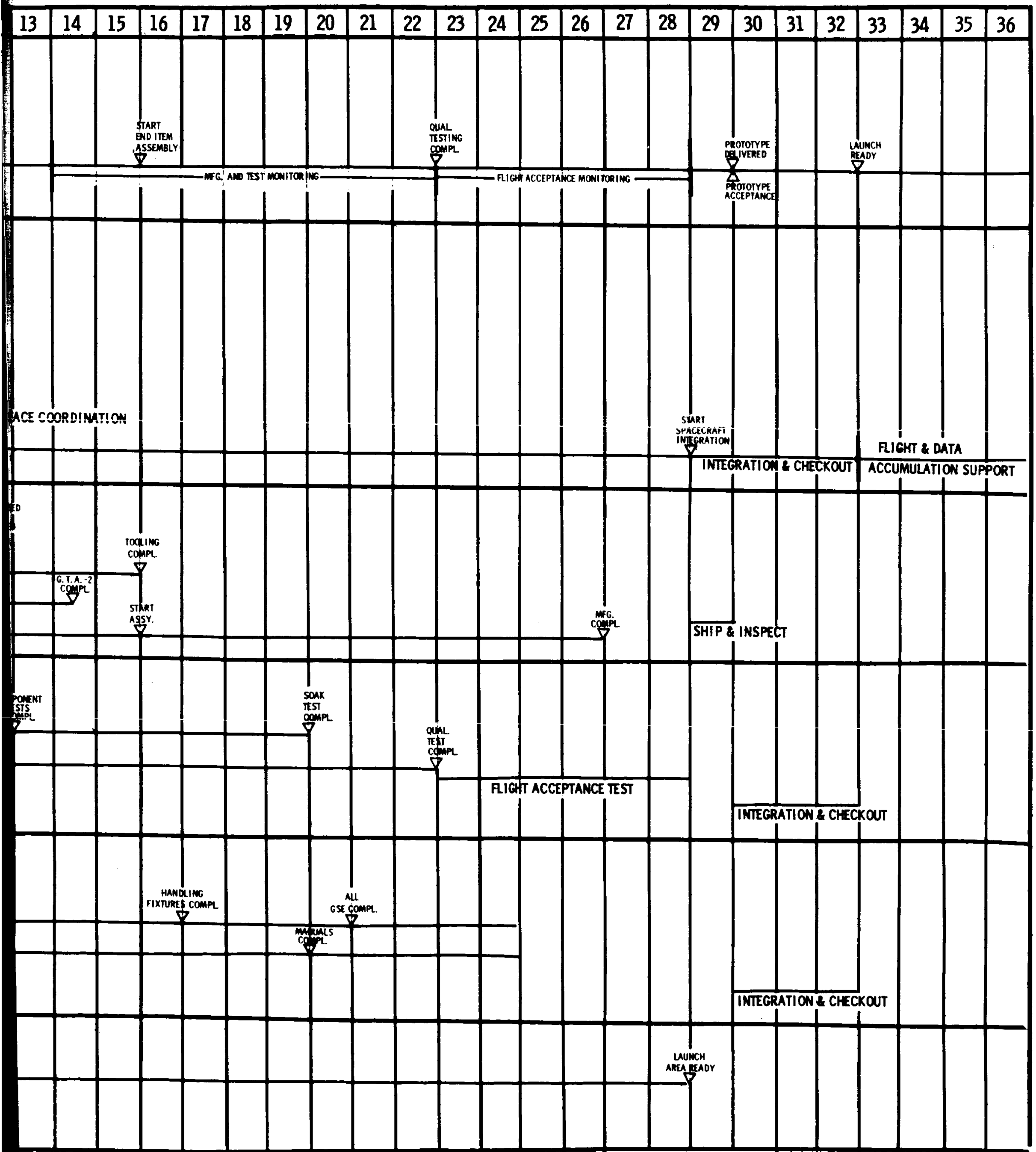
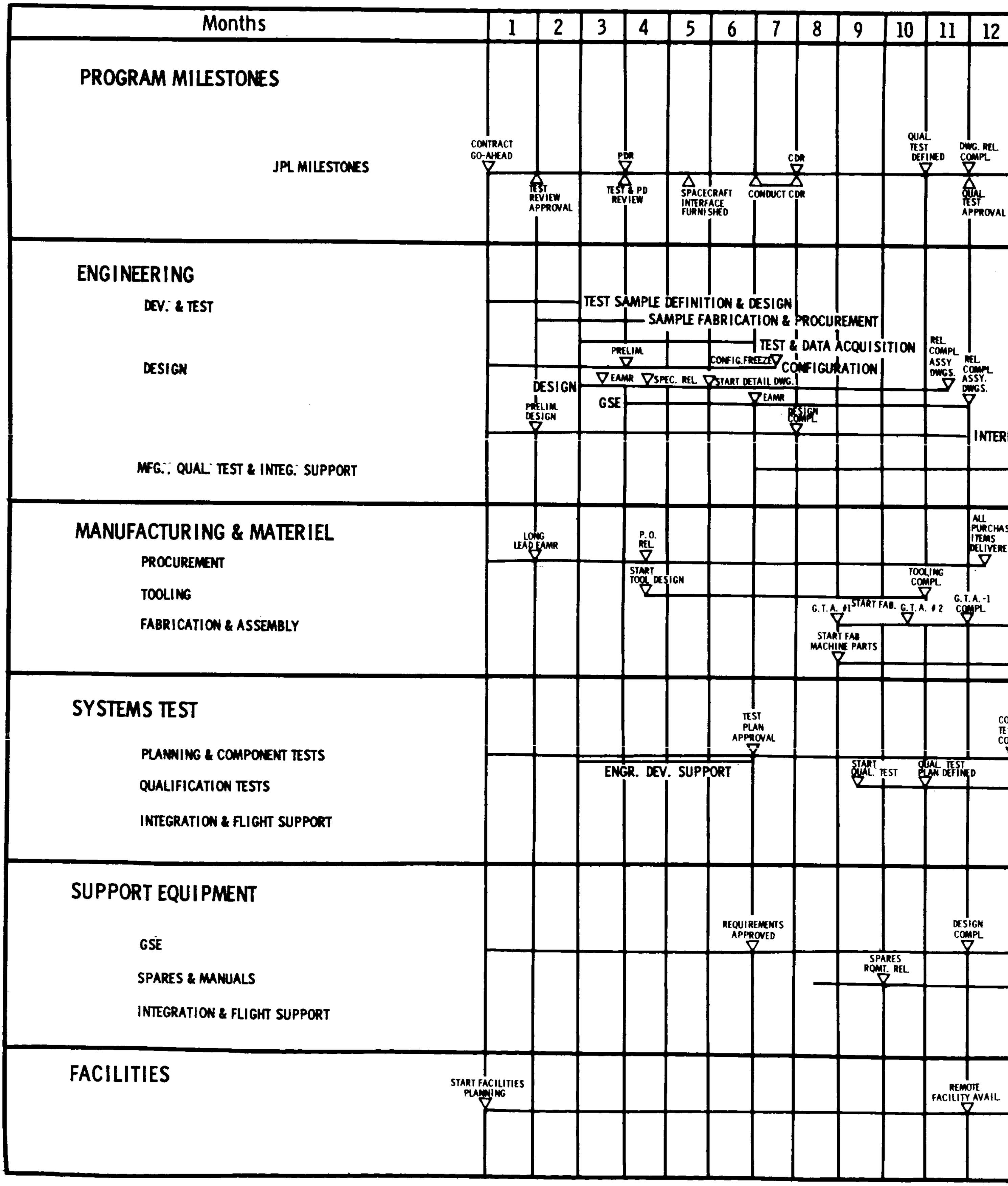


Figure 7.1-1

MASTER PHA
FULL SCALE S
Atlas/C



SING SCHEDULE
LAR CELL ARRAY
entaur

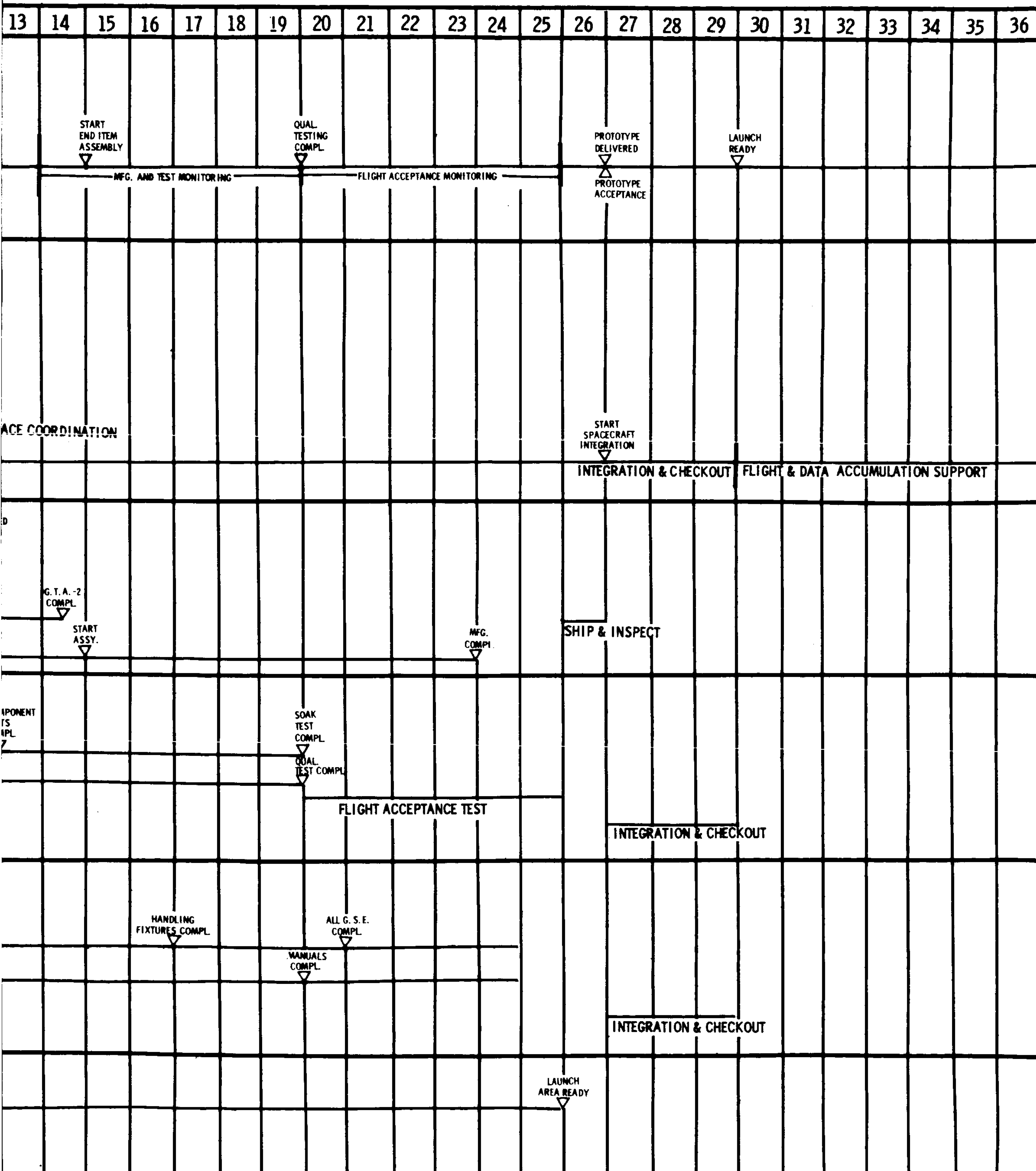


Figure ~~7.1-2~~

7.2 ENGINEERING PLAN

An Engineering Plan for either of these programs, Figure 7.2-1, is shown in four phases.

The Engineering Development Phase will start with the definition and design of the various types and quantity of test samples required. The development testing, conducted in the Engineering Laboratories, is necessary to develop the design criteria and specifications. Quality Assurance is not shown here as a separate item. It is our belief that a sound engineering organization automatically includes as a standard practice the necessary value engineering, maintainability, reliability, configuration control, and quality control that will furnish quality assurance. Preliminary design should be completed by the fourth month. However, the development phase will continue until the seventh month of the program.

The Project Design Phase will include interface coordination with the spacecraft contractor, the establishment of a final configuration, and the formal design for the structural, mechanical, and electrical systems. During this phase the ground support equipment design and the qualification test definition are planned to be completed.

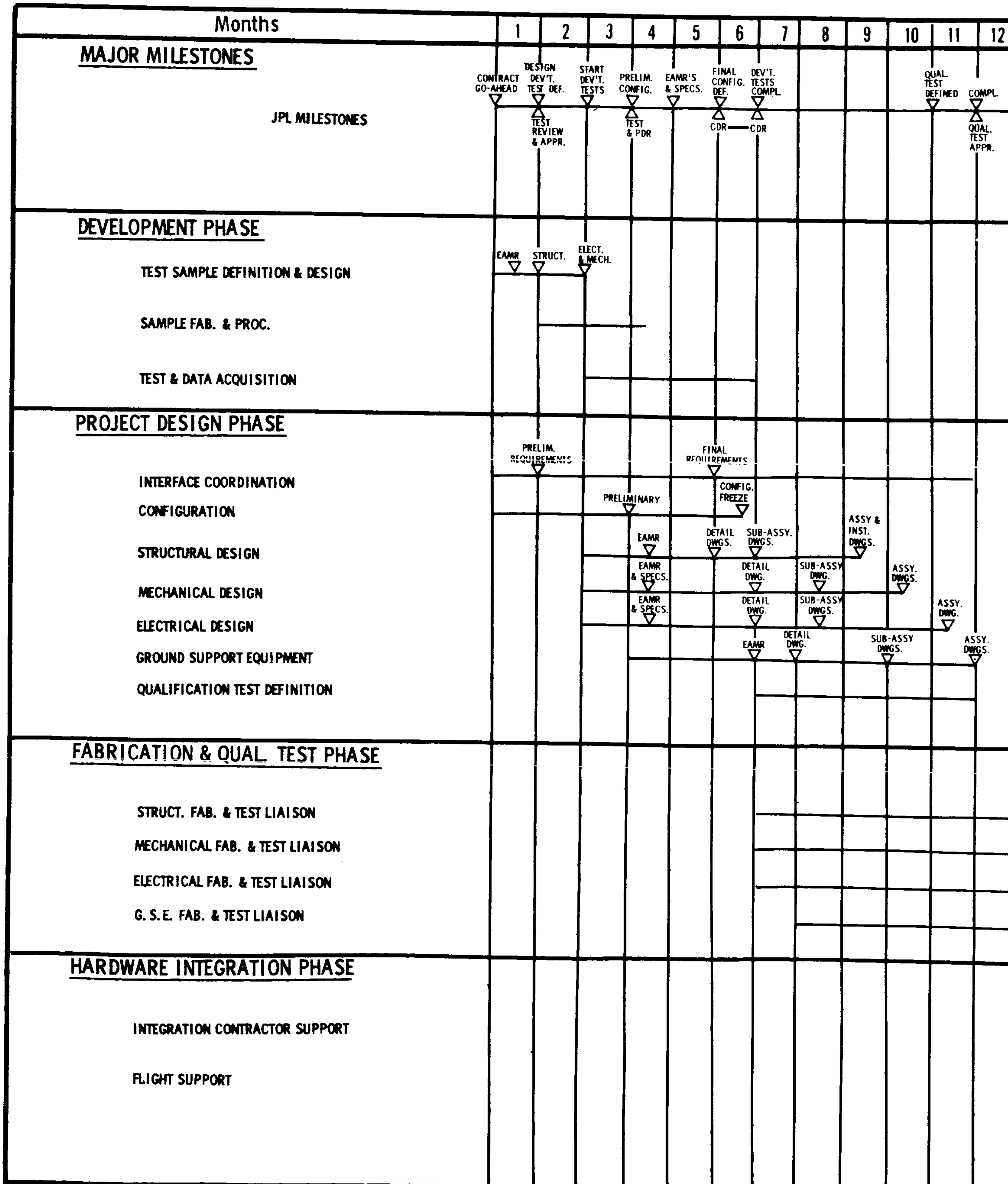
Critical Design Review can be accomplished during the seventh month of the contract. All engineering drawings would be released by the eleventh month. The total engineering effort for the Saturn IB/Centaur prototype array and the Atlas/Centaur prototype array will be basically the same; therefore, are shown alike on both Master Phasing Schedules.

The Manufacturing and Qualification Phase will consist largely of engineering support to the manufacturing effort and the systems test organization.

The Hardware Integration Phase involves engineering support to the integration contractor from the time the end item is delivered to the customer until after the flight program is concluded.

FULL SCALE SOLAR CELL

Saturn IB/Centaur



ARRAY ENGINEERING SCHEDULE
or Atlas/Centaur

D2-23942-5

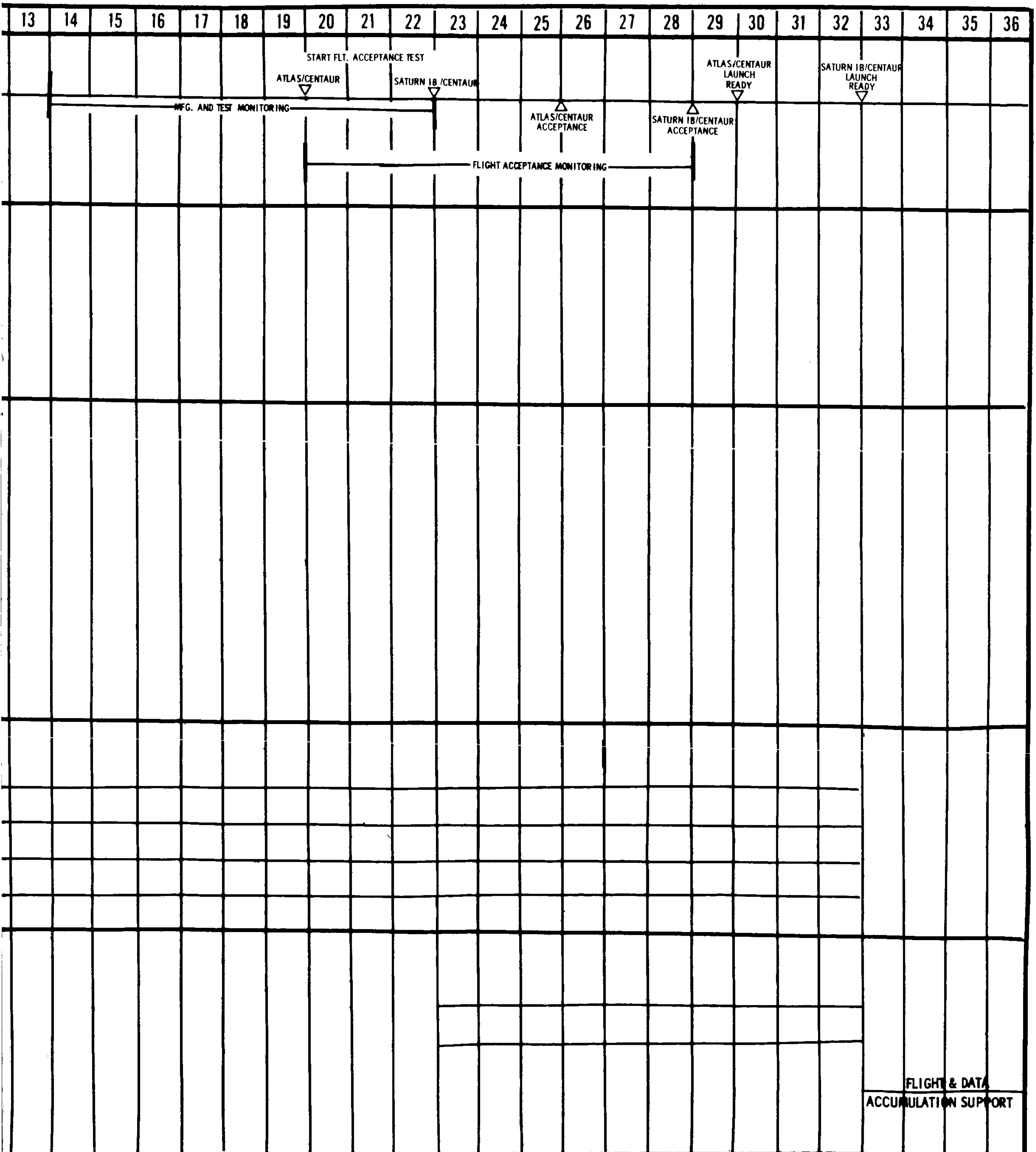


Figure 7.2-1

7.3 TEST PLAN

General - The solar array test program applies to either the solar array planned for launch on the Atlas/Centaur vehicle or the solar array planned for launch on the Saturn IB/Centaur vehicle. The test and checkout activity as regards development and qualification of the solar array shall essentially be finished upon completion of solar array installations on the spacecraft at the launch site. However, program personnel will probably be needed during subsequent launch site and mission activities. The brief resume of the test plan in this final report is described in detail in the fourth bi-monthly report, Document D2-23942-4.

7.3.1 Test Program Sequencing

The solar array test program essentially involves two groups of test articles. The first group contains small component hardware and their applicable tests. The second group contains the major test articles and their applicable tests. The major test articles range in size from the sub-panel unit to the total solar array. Included in the group of major test articles is the operational end item or solar array mission article.

The test program would be conducted primarily at the factory site and the remote site or high altitude location. The remote site would evaluate solar cell performance in response to stimulation by the sun's rays.

The normal event sequences for the testing are:

Small Component Testing	Major Article Testing
1. Receive from Manufacturing	1. Receive from Manufacturing
2. Inspection	2. Inspection
3. Ground Ambient Tests	3. Factory Stimulation of Solar Cells
4. Test Evaluation	4. Remote Site Stimulation of Solar Cells
5. Vacuum Chamber Tests	5. Ground Ambient Tests
6. Test Evaluation	6. Test Evaluation
	7. Vacuum Chamber Tests
	8. Test Evaluation
	9. Remote Site Stimulation of Solar Cells

7.3.2 Test Categories

The tests involved in the program are classified under one of four groups. The four groups include:

(a) Design Criteria Tests - To establish specific design characteristics to be included in the component and subsystems design.

(b) Design Development Tests - To functionally prove the feasibility of proposed material, processes and designs.
(c) Qualification Tests - To demonstrate satisfactory functional capability under stringent mission operating conditions.
(d) Flight Acceptance Tests - To demonstrate satisfactory functional capability under simulated mission conditions; also, to demonstrate that the mission article has been fabricated to and functions per the design requirements.

7.3.3 Solar Array

Subdivision - Basic to a clear understanding of the various test program activities is knowledge of end item nomenclature. The principal items composing the solar array are illustrated as follows:

(a) The solar array consists of four separate arms or panel assemblies.
(b) A panel assembly consists of four sub-panel units plus one sub-panel. A panel assembly thus contains 13 sub-panels.
(c) A sub-panel unit consists of three sub-panels, a main sub-panel, plus the two auxiliary sub-panels.
(d) The sub-panel is the basic item in the array structure and is operationally and structural integral in itself.

Description - Because of power output requirements, the solar array total area is relatively large. The area of the 1/4 part or panel assembly is approximately 1200 square feet, and the weight is approximately 500 pounds. Since the array has been designed to deploy in a zero "g" environment, the overall solar array structure is relatively weak in the earth environment in order to conserve weight. Thus, the factors of size, weight and structural weakness plus the major requirement of maintaining a specific electrical energy output pose special problems when devising a test program. The approach taken for testing the solar array is given in the following section.

Test Approach - Establishment of the approach to testing the solar array required division into two basic programs; the small component testing and the activities involving the major test articles. Although it was required that members of both groups demonstrate satisfactory functional capability in the anticipated space environment, the sheer size differences resulted in the following test approaches for most of the small components and all of the major test articles:

(a) Obtaining weight and balance
    1. Stowed Position
    2. Deployed Position
(b) Assembly of the sub-panels into the panel assembly.
(c) Imposing ascent and launch environment on the stowed array.
(d) Deployment
(e) Production of electrical energy

(f) Space environment during flight to Mars.
(g) Jettison.
(h) Retrorocket loading.
(i) Mars orbital environment.
(j) Solar cell maintenance.
(k) Magnetic field mapping.

7.3.4 Small Components

7.3.4.1 Design Criteria Tests

The small component part of the test program involves testing on all articles other than the major test articles. The major test articles include Ground Test Article No. 1 (GTA-1), Ground Test Article No. 2 (GTA-2), and Mission Article No. 1 (MA-1). MA-1 is the operational end item. Therefore, the small component hardware includes springs, hinges, motors, etc.

The small component test program involves both design criteria tests and design development tests.

The required design criteria tests in the solar array test program have been listed under subsystem headings. The tests shall be conducted per applicable engineering specifications.

A. Deployment
   - Springs

B. Electrical
   - Solar cell interconnections

7.3.4.2 Design Development Tests

The required design development tests in the test program have been listed under subsystem headings. The tests should be conducted per prescribed engineering specifications.

A. Deployment
   1. Damper Assembly Sub-Panels A and B
      -Test Requirements:
      a. Ground ambient environment
      b. Vacuum environment (10^{-5} Torr)
   2. Motor Reducer Cable Drum
      -Test Requirements:
      a. Ground ambient environment
      b. Vacuum environment
   3. Springs
      -Test Requirements:

  - a. Ground ambient environment
  - b. Vacuum environment
- 4. Latches
  - a. Articles required:
    - (1) Main sub-panel latches
    - (2) Auxiliary sub-panel latches
  - b. Test requirement: (NOTE: Each specific test environment and subsequent test article operation should have latches assigned solely for that test.)
    - (1) Ground ambient environment
    - (2) Vacuum environment
- 5. Dual Pin Pullers
  - a. Articles required:
    - (1) Two complete dual pin puller assemblies
    - (2) Structural shear pins
    - (3) Piston retainer shear pins
    - (4) Five sets of dual pin puller assembly propellant
  - b. Test requirement: (NOTE: Each specific test environment and subsequent test article operation should have hardware assigned solely for that specific test.)
    - (1) Ground ambient environment
    - (2) Vacuum environment

B. Jettison
- 1. Ordnance Cable Cutter/Pin Puller
  - a. Articles required:
    - (1) Two ordnance cable cutter/pin puller systems
    - (2) Guillotine retainer shear pins
    - (3) Pin Puller retainer shear pins
    - (4) Five sets of propellant
    - (5) An explosion container with the capability of measuring the force of the contained explosion
  - b. Test requirements: (NOTE: Each specific test environment and subsequent test article operation should have hardware assigned solely for that specific test.)
    - (1) Ground ambient environment
    - (2) Vacuum environment
- 2. Mars Structure Release Clamp and Cam
  - a. Articles required:
    - One set of the Mars structure release clamp and cam
  - b. Test requirements:
    - Vacuum environment

C. Structures
- 1. 3 Mil Fiberglass Tape Membrane
- 2. Beryllium Frames
- 3. Hinges
- 4. Shear Cones
- 5. Tape to frame joint and intercostal to frame joint

6. Intercostal to Frame Joint

D. Electrical

1. Solar Cell - Cover Glass Adhesive
   - Test Requirements:
   a. Ultra-violet radiation
   b. Particulate radiation
2. Thermal Control Coatings
   - Test Requirements:
   Particulate radiation - Check integrity of the test articles; accomplish particulate radiation tests, using 5 test articles for each test.
3. Electrical Breakdown
4. Maintainability
5. Magnetic Field
6. Solar Cells
   a. Articles required:
      (1) Various types and makes of solar cells. This should include 3 lots of 300 cells to the lot, selected from standard production runs. The 3 lots should each be from a different manufacturer.
      (2) Solar cell test console
      (3) Monochrometer
      (4) Light source
      (5) Radiation sources (1 mev electrons)
      (6) Vacuum chamber
   b. Test requirements:
      (1) Ground ambient environment, with the temperature controlled
      (2) Accomplish a physical inspection of the cells
      (3) Using a white light source, check the electrical characteristics of the cells
      (4) Using the monochrometer, check spectral response of the solar cells.
      (5) Mechanical strength tests
      (6) Solderability tests
      (7) Particulate radiation resistance tests
7. Thermal Cycling
   - Test Requirements:
   Electrical continuity check of all test articles

7.3.5 Major Test Articles

There are essentially three major test articles in the solar array test program. These include:

Ground Test Article No. 1 (GTA-1)
Ground Test Article No. 2 (GTA-2)
Mission Article No. 1 (MA-1)

MA-1 is the operational end item and so will be installed on the mission spacecraft. MA-1 is considered as one of the test articles in that it is required to undergo a specific test sequence in the program.

The qualification tests in the program consist of those tests required to be accomplished by the two ground test articles (GTA-1 and GTA-2). The flight acceptance tests in the program consist of those tests required to be accomplished by the mission article (MA-1).

The specific test sequences and the detailed test requirements for the above articles are given in the following sections.

7.3.5.1 Ground Test Article No. 1 (GTA-1) - Description

This test article basically consists of one complete prototype sub-panel plus the prototype structure of a panel assembly. The main features of this test article include:

1. The panel assembly structure is prototype and so shall be capable of being stowed and deployed. The structure includes fiberglass substrate.
2. Sub-panel No. 1 shall be prototype including installation of solar cells.
3. Simulated panels and/or weighted strips shall be placed in the other individual sub-panels so that resulting mass characteristics of this panel assembly will be the same as for the prototype.
4. Prototype hardware including connectors, busses, etc., plus sufficient wiring to enable electrical continuity checks, shall be installed on the panel assembly.
5. Groups of forty-nine cells (rows of seven) shall be installed in the following locations.
    a. In the center of the rectangles formed by:
        (1) the intercostals
        (2) the intercostals and spars
        (3) the intercostals, spars and edge members
    b. Over the intersections of the intercostals
    c. In the middle of the intercostal spans; i.e., at intercostal midpoints between any two support points.

<u>GTA-1 Requirement</u>

GTA-1 shall meet the requirements for a structural and operational demonstration test article in the test program. The detailed requirements which GTA-1 shall satisfy are listed as follows:

1. Stowing the panel assembly.
2. Withstanding launch and ascent flight environment in the stowed position.
3. Deploying the panel assembly in a zero "g" environment.
4. Accomplishing jettison operation.

5. Withstanding retrorocket loads on sub-panel No. 1.
6. Sub-panel No. 1 withstanding space environment from deployment on through orbit about Mars.

7.3.5.2 Ground Test Article No. 2 (GTA-2) - Description

This test article consists of four sets of jettison test articles. The articles include main sub-panel No. 1 plus a dummy jettison panel. The main features of this test article include:

1. The main sub-panels No. 1 should have the normal structural members, except there is no requirement for fiberglass substrate. The structure may be fabricated from aluminum or some other substitute material. Simulation of prototype sub-panel mass characteristics is not required.
2. The dummy panels-jettison demonstration must have capability of vectorially measuring the jettison force.
3. Prototype electrical connectors, busses, etc., plus sufficient wiring to enable electrical continuity checks, shall be installed on the No. 1 sub-panels.
4. The installation shall be such that jettison may be initiated in a manner similar to that on the actual mission.

GTA-2 Requirement

GTA-2 meets the requirement for a test article which is prototype in all aspects of the jettison installation and operation. The detailed requirements which GTA-2 shall satisfy are listed as follows:

1. Accomplish simultaneous jettison of the four No. 2 sub-panels.
2. Accomplish simultaneous jettison of the four No. 2 sub-panels in the Mars temperature environment.
3. Determination of satisfactory jettison force simultaneously applied to the four No. 2 sub-panels.

7.3.5.3 Mission Article No. 1 (MA-1) - Description

MA-1 is the operational end item and thus consists of four prototype panel assemblies. The four panel assemblies shall be maintained integrally throughout the flight acceptance testing. Any changes in the sub-panel makeup of the panel assemblies shall be indicated by flight acceptance testing.

7.3.6 GTA-1 Test Sequence

1. Receive sub-panels from manufacturing and inspection.
2. Weigh and balance the individual sub-panels.
3. Assemble the individual sub-panel units.
4. Electrical continuity check of the individual sub-panel units.

5. Zero "g" deployment of the individual sub-panel units. Factory solar simulation check after deployment.
6. Assemble and panel assembly in the stowed configuration.
7. Weight and balance.
8. Electrical continuity check.
9. Zero "g" deployment of the panel assembly. Factory solar simulation check after deployment.
10. With the panel assembly in the deployed configuration, make attachments for vibration testing. Conduct torsional vibration and lateral bending vibration tests.
11. Factory solar simulation.
12. Place panel assembly in space chamber in the stowed configuration.
13. Simulation of ejection of protective cover around spacecraft and solar array just prior to solar array deployment. Evaluate thermal reaction of stowed panel assembly.
14. Assemble main sub-panel No. 1 with the jettison mechanism and the dummy panel-jettison demonstration.
15. Place main sub-panel No. 1 in space chamber. During the chamber testing, periodic electrical continuity checks must be made. Control vacuum and temperature to accomplish the following:
    a. Thermal shock, to simulate environment upon deployment.
    b. Thermal soak, to simulate environment during flight to Mars.
    c. Jettison demonstration. Measure ejection force vector.
    d. Thermal shock, to simulate environment while orbiting Mars.
16. Vibration test of main sub-panel No. 1. Environment after jettison.
17. Factory solar simulation.
18. Retro load test on main sub-panel No. 1.
19. Factory solar simulation.
20. Assemble main sub-panel No. 1 with the rest of the panel assembly in the stowed configuration.
21. Electrical continuity check.
22. Acoustic tests.
23. Electrical continuity check.
24. Vibration tests, to simulate launch and ascent flight vibrations.
25. Vibration tests, to establish dynamic response of the stowed panel assembly.
26. Electrical continuity check.
27. Vibration tests of a mass simulated complete array.

7.3.7 GTA-2 Test Sequence

1. Receive sub-panels and other hardware from manufacturing and inspect.
2. Assemble dummy panels with the No. 1 sub-panels.
3. Acoustic tests.
4. Electrical continuity check.
5. Vibration tests, simulating launch and flight vibrations.
6. Electrical continuity check.

7. Assemble the four sets of sub-panels in the array deployed configuration.
8. Electrical continuity check.
9. Jettison in Mars temperature environment. Electrical continuity check after jettison.

7.3.8 MA-1 Test Sequence

1. Receive disassembled sub-panels from manufacturing and inspect.
2. Magnetic field mapping. To be accomplished only on one main sub-panel No. 1. Use similar procedure as that used in the small component magnetic field testing. Progressively sum the results to obtain the magnetic field of the deployed solar array.
3. Weight and balance the individual sub-panels.
4. Assemble the individual sub-panel units.
5. Electrical continuity check of the individual sub-panel units.
6. Zero "g" deployment of the individual sub-panel units. Factory solar simulation check after deployment.
7. Assemble the panel assemblies in the stowed configuration.
8. Weight and balance.
9. Acoustic tests.
10. Electrical continuity checks.
11. Assemble the four panel assemblies into the vibration shaker. Accomplish tests in the following sequence:
    a. Simulation of launch and flight vibrations.
    b. Electrical continuity checks.
    c. Vibration test to verify frequencies, amplitudes and mode shapes.
    d. Electrical continuity checks.
12. Zero "g" deployment of the panel assemblies. Factory solar simulation check after deployment.
13. Remote site solar stimulation of each sub-panel.
14. Store solar array, as required.

Following the remote site solar stimulation tests, the MA-1 panel assemblies shall be placed in storage until shipment. Upon shipment, the following activities will take place in order:

1. Shipment
2. Receiving inspection at Cape Kennedy
3. Accomplish tests on individual sub-panels per following procedure:
    a. Stimulate each sub-panel in turn with fluorescent lamp bank, while maintaining ambient temperature on the sub-panel with fans or other cooling source.
    b. Obtain V-I characteristics. There should be no drift in the open circuit voltage.
    c. Remove all light source stimulation of the sub-panel.

d. Pass current through the dark panel in forward direction. Use 700 ma (milli-amps) per module.

e. Obtain IR image using Philco thermal scanner.

f. Examine data for:

   1. Open cells
   2. Shorted cells
   3. Bad solder joints
   4. Storage as required
   5. Assemble in stowed position on the spacecraft.

7.4 GROUND SUPPORT EQUIPMENT REQUIREMENTS

7.4.1 Introduction

The following analysis of GSE functional requirements is based on the Saturn IB/Centaur solar array configuration. Functionally, the GSE requirements for the Saturn and Atlas arrays are identical; physically, they differ only in size. Where significant differences exist, they are noted.

GSE functional requirements have been derived from manufacturing and test flow diagrams, shown in Figures 7.4-1, 7.4-2, and 7.4-3. Cape Kennedy operations have been detailed in these diagrams up to and including installation of the solar array on the spacecraft. The requirements for GSE to support integrated spacecraft test and launch activities of the solar array installations are not included in this study. Requirements for spares and manuals are not specifically identified.

7.4.2 Flow Diagrams

GSE functional requirements have been derived from manufacturing and test flow diagrams for Ground Test Articles (GTA) -1 and -2, and Mission Article (MA) -1. These flows in turn are based on the manufacturing and test plans as presented in Section 3.3 and 3.5 of the fourth bi-monthly report, Document D2-23942-4.

The flow diagrams are arranged to represent the progress of the hardware through sequential fabrication and test operations. Ground support equipment requirements are indicated directly below each operation. (Figures 7.4-1, 7.4-2, and 7.4-3)

Individual GSE items are assigned an identifying item number and descriptive title such as (B-14), Zero G, Deployment Fixture. When an equipment item is first identified on a flow diagram, the full nomenclature is used; only the item number is shown for repetitive requirements.

The large size and delicate nature of the solar sub-panels and panel assembly requires specially designed transportation and handling equipment. For the individual sub-panels, a Sub-Panel Handling Fixture (A-1) has been identified. The sub-panel remains in its handling fixture at all times, except when undergoing testing. Handling fixtures can be inter-locked and placed on a framework (A-7), Deployed Panel Assembly Support Fixture, to support a panel assembly in the deployed configuration. A cover plate (A-2), shipping container (A-3), and heat seal bag (A-4) are provided for shipping sub-panels as individual subpanels rather than as four stowed panel assemblies.

For handling of the panel assembly in the stowed condition, a support fixture (A-8) is provided. This fixture remains with the panel assembly at all times when in the stowed position and interfaces with the zero "g" deployment fixture (B-14), vibration fixture (B-15), weight and balance fixture (B-16), and transporter (A-9). The transporter (A-9) is used for inplant transportation of the panel assembly from one area to another.

Solar testing is confined to sub-panels because of the extreme size and delicate construction of the panel assembly. Sub-panels first undergo preliminary testing in the factory using the solar panel excitation equipment (B-9) to detect any gross defects in operation. After successful completion of these tests, the sub-panels are shipped to a high altitude solar test site. Here the sub-panels undergo standard solar testing. An equatorial mount (B-1) is provided to maintain the sub-panel normal to the sun vector. Sub-panel operating temperature is controlled by means of the cooling air plenum (B-24) which directs conditioned air onto the dark side of the sub-panel. A portable air supply (B-25) provides cooling air to the plenum.

A portable shelter (B-3) provides protection during test setup. Panel performance is monitored and recorded by the solar panel test console (B-8). Standard solar irradiation instrumentation such as pyroheliometer (B-4) and standard solar cells (B-5) is provided.

Panel assembly deployment while in a 1 "g" environment is demonstrated with the zero "g" deployment fixture (B-14). Panel deployment takes place in a vertical plane. Auxiliary panel deployment against 1 "g" forces is accomplished with the aid of booster springs. Dampers are provided to slow deployment of auxiliary panels aided by gravity. Proper functioning of auxiliary panel flight deployment springs is demonstrated by testing sub-panel units separately prior to installation into the panel assembly.

Immediately following deployment demonstration, the deployed panel is subjected to torsional and lateral vibration. The deployed panel assembly structural test fixture (B-26) is provided for these tests. Zero "g" deployment fixture (B-14) removal and structural test fixture (B-26) installation takes place simultaneously so that the deployed panel is disturbed as little as possible. It may be necessary to re-install portions of the zero "g" fixture (B-14) to facilitate returning the panel assembly to the stowed position at the conclusion of vibration testing.

Thermal and vibration fixtures are provided for individual sub-panels, and the stowed panel assembly as required by the test plan. Sub-panel and stowed panel assembly weight and center-of-gravity is determined with a weight and balance fixture (B-16). A jettison fixture (B-21) and dummy panel (B-27) are used in conjunction with GTA-2 to demonstrate operation of the jettison mechanisms and ordnance.

A detailed description of each of the following items of equipment is recorded in Section 3.4.5 of the fourth bi-monthly report, Document D2-23942-4.

7.4.3 Transportation and Handling Equipment

Item No.	Nomenclature
A-1	Sub-Panel Handling Fixture (Figure 7.4-4)
A-2	Sub-Panel Cover Plate (Figure 7.4-4)
A-3	Sub-Panel Shipping Container (Figure 7.4-4)
A-4	Heat Seal Bag
A-5	Sub-Panel Dolly (Figure 7.4-4)
A-6	Plastic Dust Cover
A-7	Deployed Panel Assembly Support Fixture (Fig. 7.4-5)
A-8	Stowed Panel Assembly Support Fixture
A-9	Stowed Panel Assembly Transporter

7.4.4 Monitoring and Checkout Equipment

Item No.	Nomenclature
B-1	Equatorial Mount (Figure 7.4-6)
B-2	Shade Tube (Figure 7.4-10)
B-3	Portable Shelter (Figure 7.4-9)
B-4	Pyrheliometer
B-5	Standard Solar Cell and Water-Colled Cell Block
B-6	JPL Reference Cells
B-7	Skylight Sensor
B-8	Solar Panel Excitation Equipment
B-13	Zero "G" Deployment Fixture (Fig. 7.4-8 and 7.4-8A)
B-14	Zero "G" Deployment Fixture (Suspension) (Fig. 7.4-7)
B-15	Stowed Panel Assembly Vibration Fixture
B-16	Weight and Balance Fixture
B-18	Solar Panel Deployment Monitor
B-19	Sub-Panel Thermal Test Fixture
B-20	Stowed Panel Assembly Thermal Test Fixture
B-21	Jettison Demonstration Fixture
B-22	Ordnance Circuit Test Set
B-23	Sunlight Attenuator (Figure 7.4-10)
B-24	Cooling Air Plenum (Figure 7.4-10)
B-25	Portable Air Supply
B-26	Deployed Panel Assembly Structural Test Fixture
B-27	Dummy Panel - Jettison Demonstration
B-28	Retro Load Test Fixture

7.4.5 Maintenance and Service Equipment

Item No.	Nomenclature
C-1	Solar Cell Repair Kit
C-2	Structure Repair Kit
C-3	Connector and Wire Bundle Repair Kit

7.4.6 Simulators

Item No.	Nomenclature
D-1	Dummy Solar Panels (Mechanical)
D-2	Solar Panel Simulator (Electrical)

7.4.7 Zero "G" Deployment Fixture

Two alternate fixture designs are presented. In both, panel deployment takes place in a vertical plane. Auxiliary panel deployment against one "g" forces is accomplished with the aid of booster springs. Dampers slow deployment of auxiliary panels aided by gravity. Monitors are provided to measure and record hinge angle and motor current as a function of time.

Alternate #1 - Suspension Arm System (See Figure 7.4-7)

This configuration consists of an overhead, jointed truss of aluminum or steel tubing and adjustable steel cable guywires. The panel assembly is supported at the center of the main sub-panels hinges by counter-weighted arms which allow vertical movement. A floor mounted vertical steel tube simulates the spacecraft structure and provides hinge points for main sub-panel #1. Fixture counterweights are located in wooden mock-ups which simulate the other three stowed panel assemblies. Deployment forces are provided by the panel assembly cable drum drive motor with the overhead suspension arm system passively following panel deployment. Auxiliary panel booster springs and dampers are required to negate the effect of gravity.

Alternate #2 - Air Bearing Fixture (See Figures 7.4-8 and 7.4-8A)

The air bearing zero "g" deployment fixture is composed of four main elements: a table, the air bearings and extensions, a dummy spacecraft structure, and the booster springs and dampers. Each element is discussed below.

Table - The table consists of heavy aluminum castings which have a flat upper surface and a deep waffle pattern below. The castings are coated with 1/4 inch epoxy which is machined to provide a flat, smooth surface. Cracks between castings are filled with beeswax. Table leveling is accomplished by means of jacks. The table will measure approximately 30 feet by 70 feet.

Air Bearings - Five air bearings are required for the deployment of a panel assembly. They are attached to the panel assembly at the main panel hinge points by means of extension rods. Air is supplied under pressure through pneumatic lines and flexible hoses. The bearings are isolated from the support rods by ball bearings to provide maximum freedom of motion. Three smaller air bearings attached to the sub-panel edge members are required to demonstrate deployment of sub-panel units.

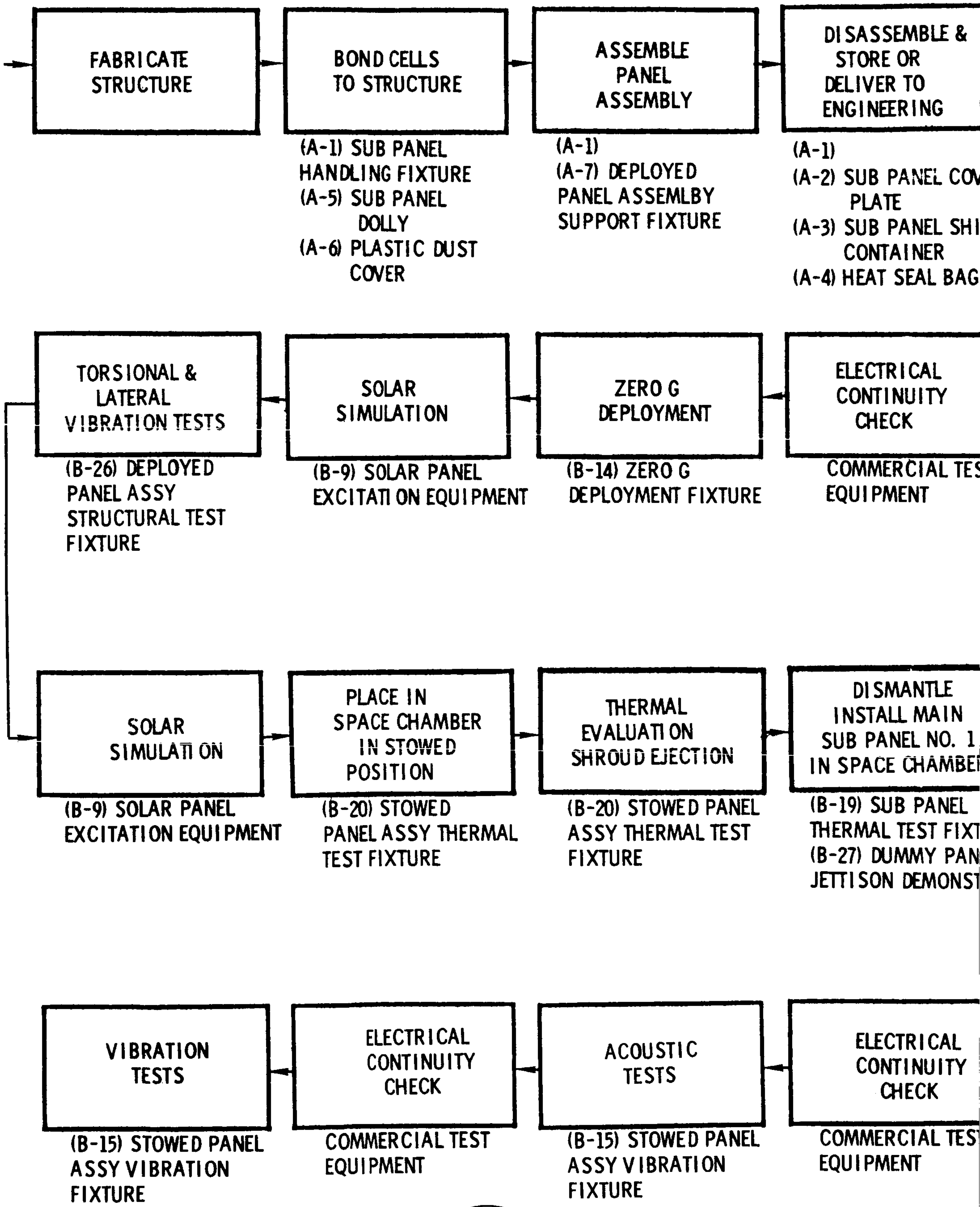
Dummy Spacecraft Structure - The dummy spacecraft structure provides a mounting surface for the stowed panel assembly support fixture and houses the cable drum drive motors.

Booster Springs and Dampers - These items are required to negate the effect of gravity on auxiliary panel deployment. They are installed on the panel assembly prior to the start of testing.

7.4.7.1 Recommendation

The Alternate #1 "Suspension Arm System" is selected on the basis of preliminary estimates of cost, availability, and confidence level. Alternate #1 was used in costing the program.

FAB
FUNCTIONAL
GROUN



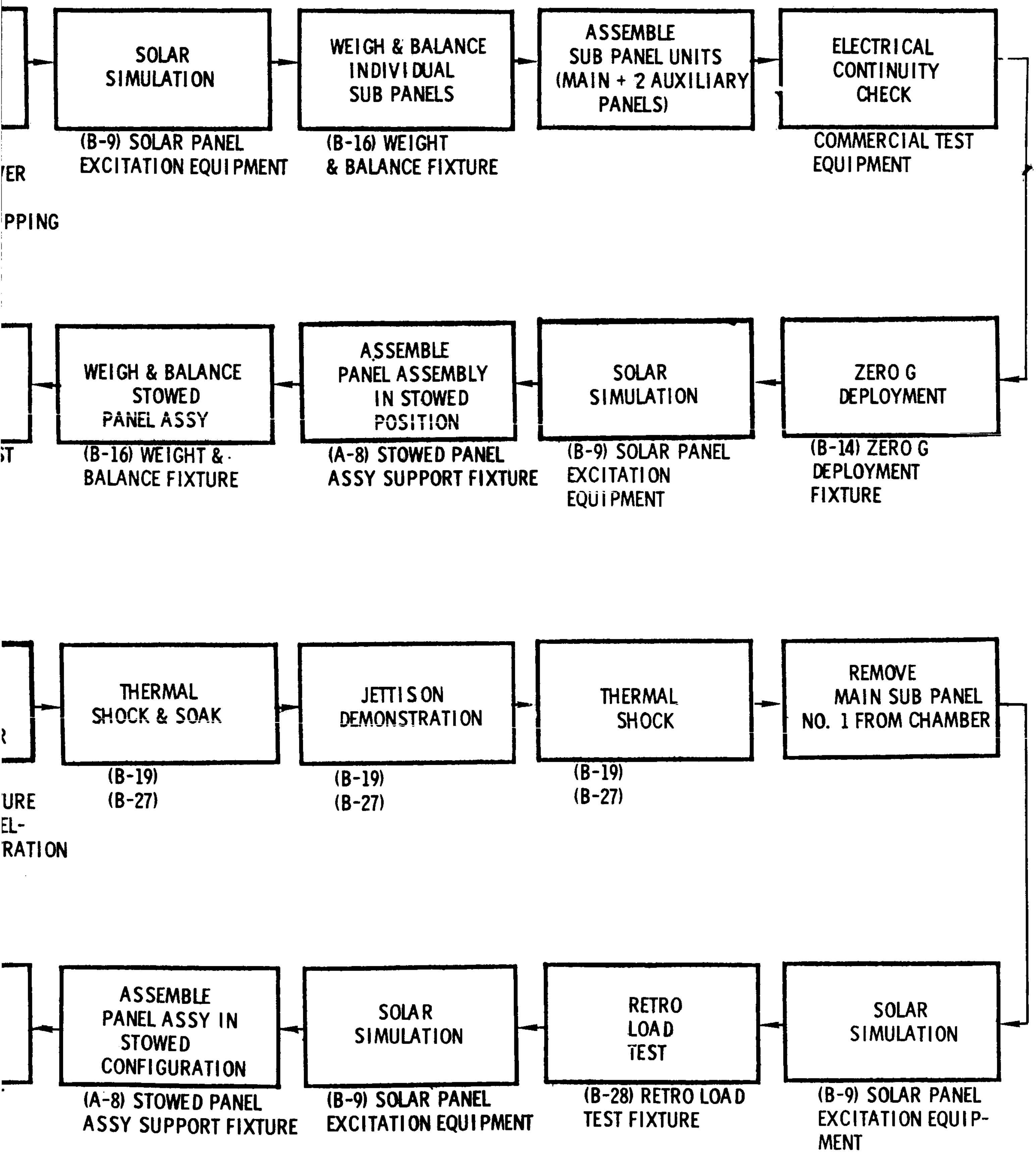

SOLAR SIMULATION
(B-9) SOLAR PANEL EXCITATION EQUIPMENT
WEIGH & BALANCE INDIVIDUAL SUB PANELS
(B-16) WEIGHT & BALANCE FIXTURE
ASSEMBLE SUB PANEL UNITS (MAIN + 2 AUXILIARY PANELS)
ELECTRICAL CONTINUITY CHECK
COMMERCIAL TEST EQUIPMENT
ER
PPING
WEIGH & BALANCE STOWED PANEL ASSY
(B-16) WEIGHT & BALANCE FIXTURE
ASSEMBLE PANEL ASSEMBLY IN STOWED POSITION
(A-8) STOWED PANEL ASSY SUPPORT FIXTURE
SOLAR SIMULATION
(B-9) SOLAR PANEL EXCITATION EQUIPMENT
ZERO G DEPLOYMENT
(B-14) ZERO G DEPLOYMENT FIXTURE
THERMAL SHOCK & SOAK
(B-19)
(B-27)
JETTISON DEMONSTRATION
(B-19)
(B-27)
THERMAL SHOCK
(B-19)
(B-27)
REMOVE MAIN SUB PANEL NO. 1 FROM CHAMBER
URE
EL-
RATION
ASSEMBLE PANEL ASSY IN STOWED CONFIGURATION
(A-8) STOWED PANEL ASSY SUPPORT FIXTURE
SOLAR SIMULATION
(B-9) SOLAR PANEL EXCITATION EQUIPMENT
RETRO LOAD TEST
(B-28) RETRO LOAD TEST FIXTURE
SOLAR SIMULATION
(B-9) SOLAR PANEL EXCITATION EQUIPMENT

FIGURE 7.4-1

FABRICATION AND TEST
FUNCTIONAL FLOW & GSE REQUIREMENTS
GROUND TEST ARTICLE NO. 2

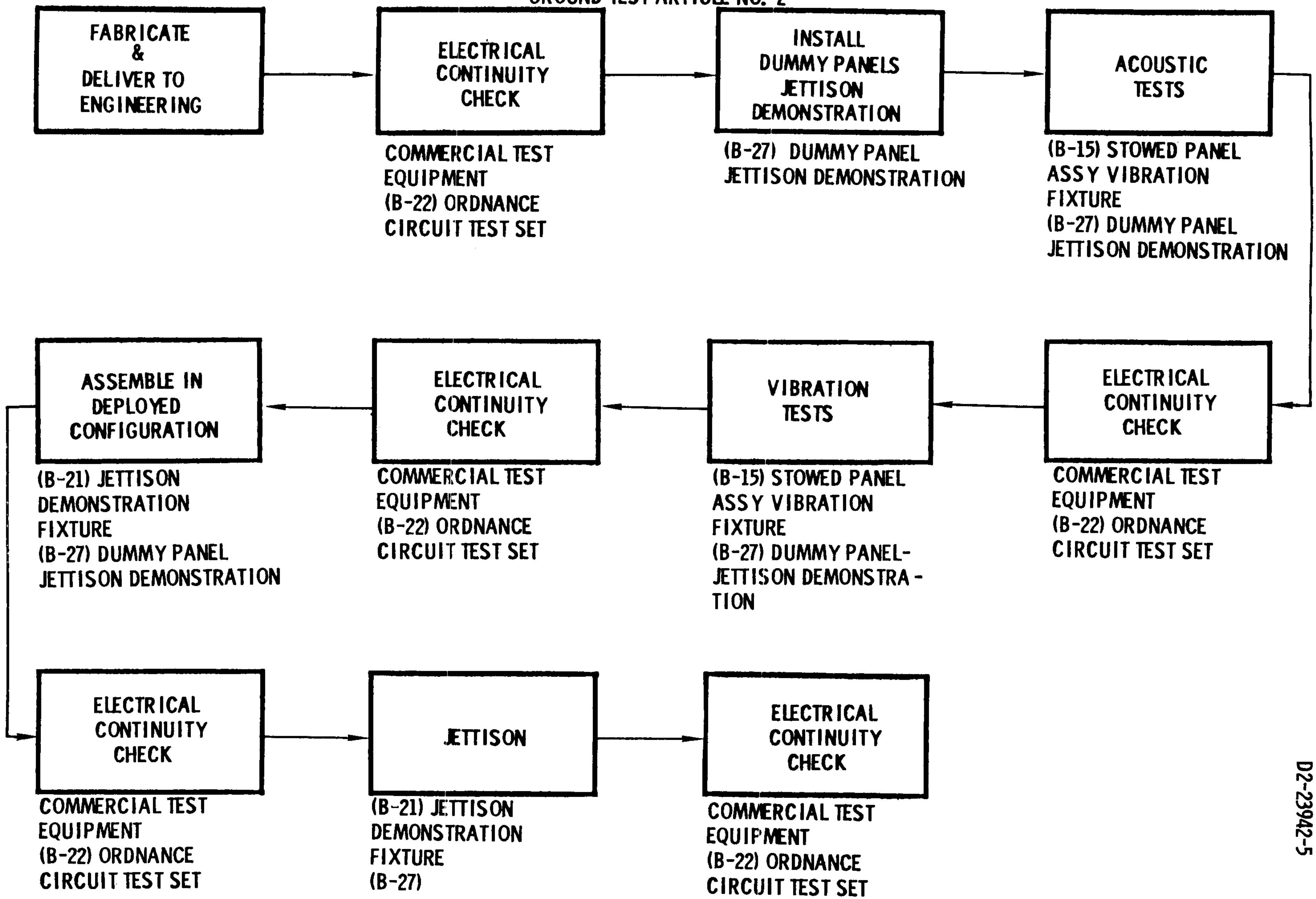
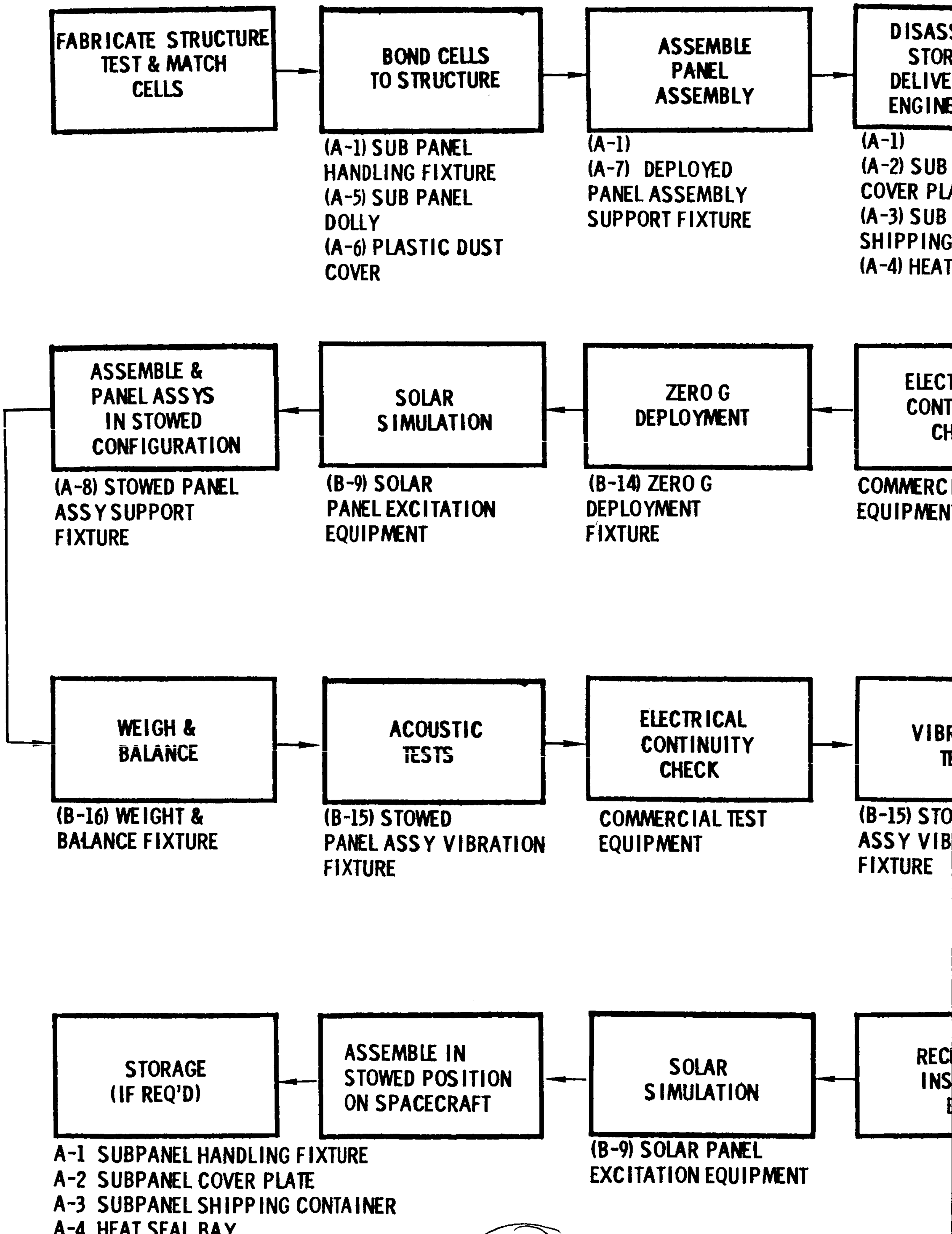


FIGURE 7.4-2

FABRICAT
FUNCTIONAL FLO
MISSION



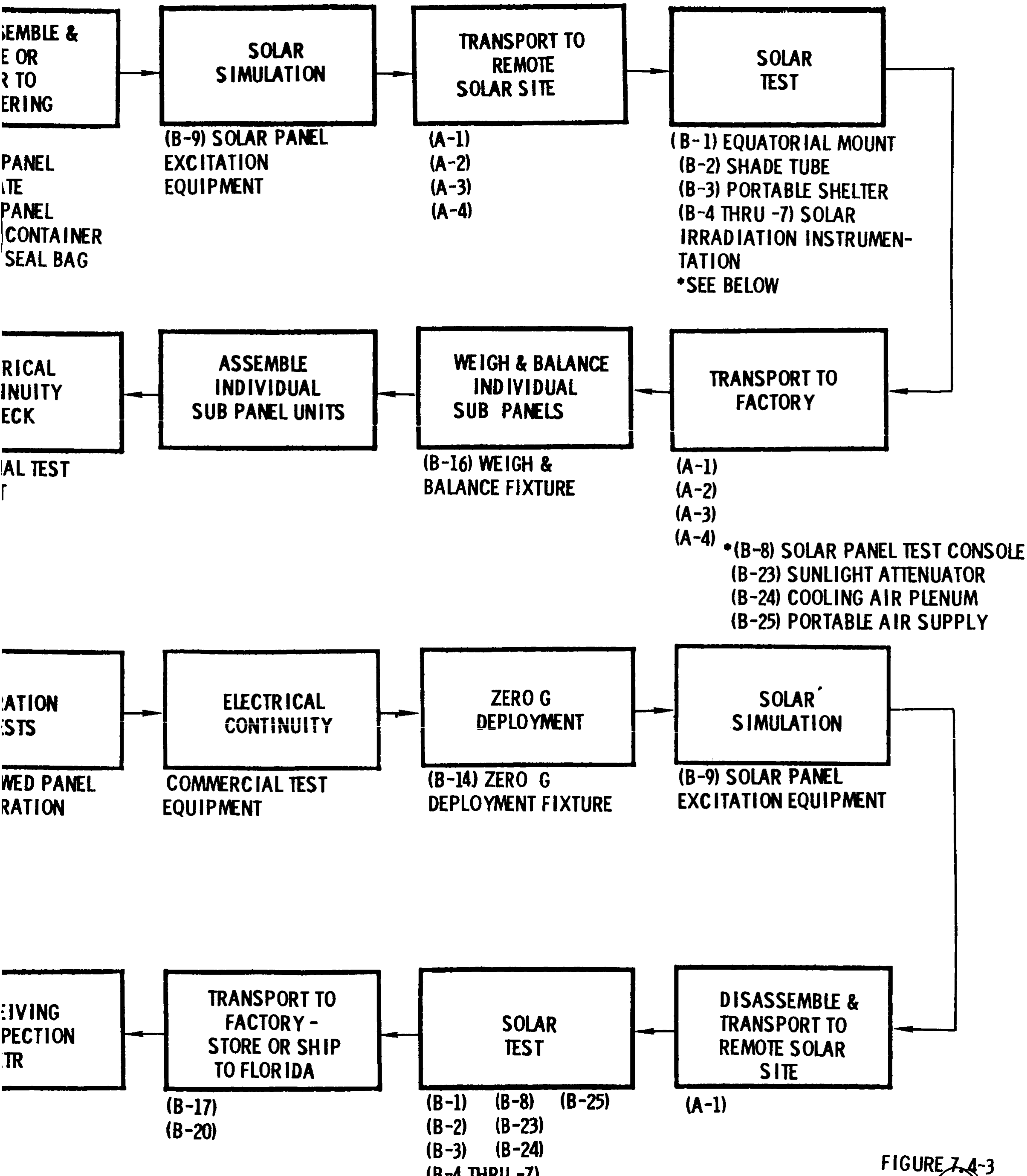
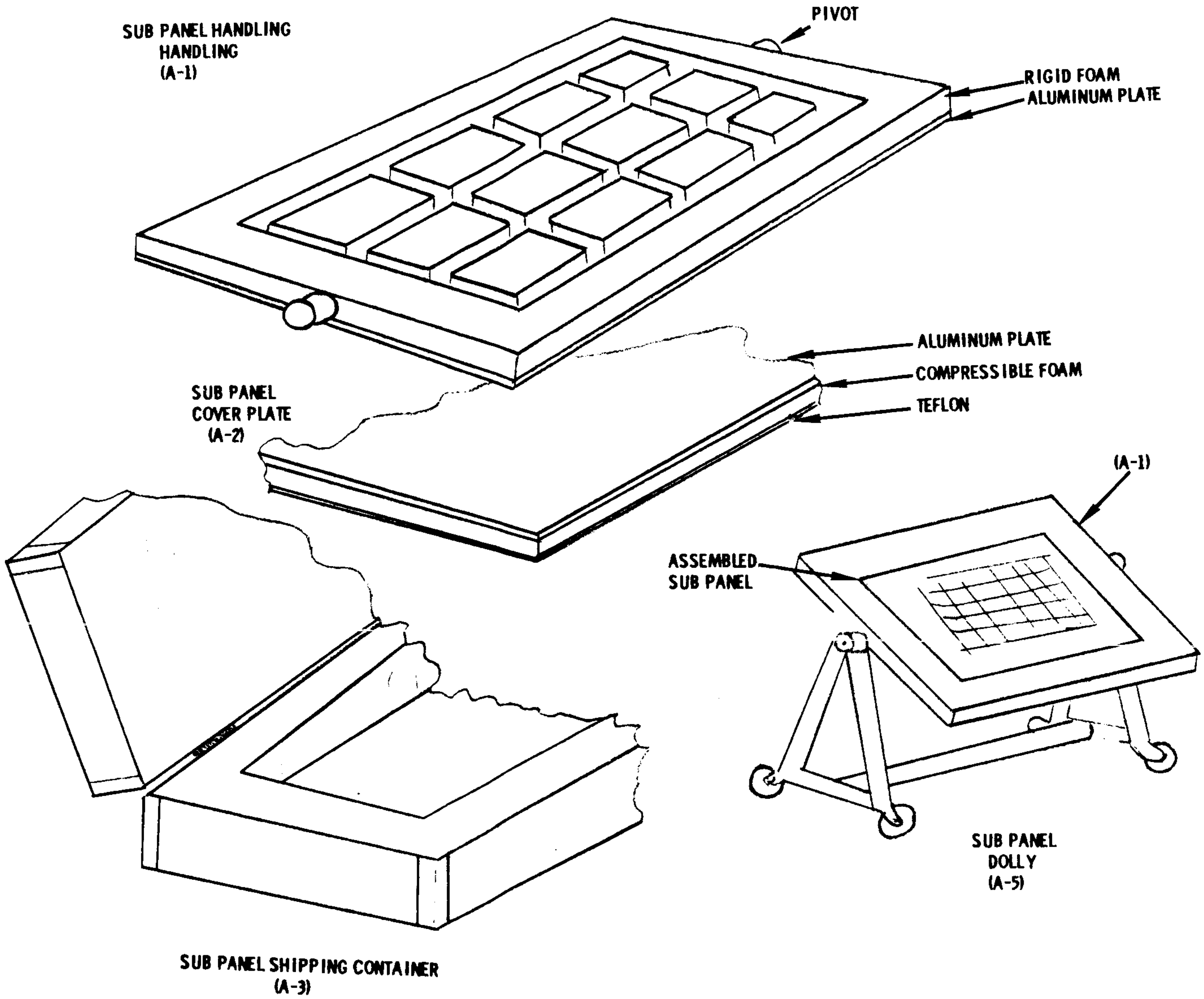
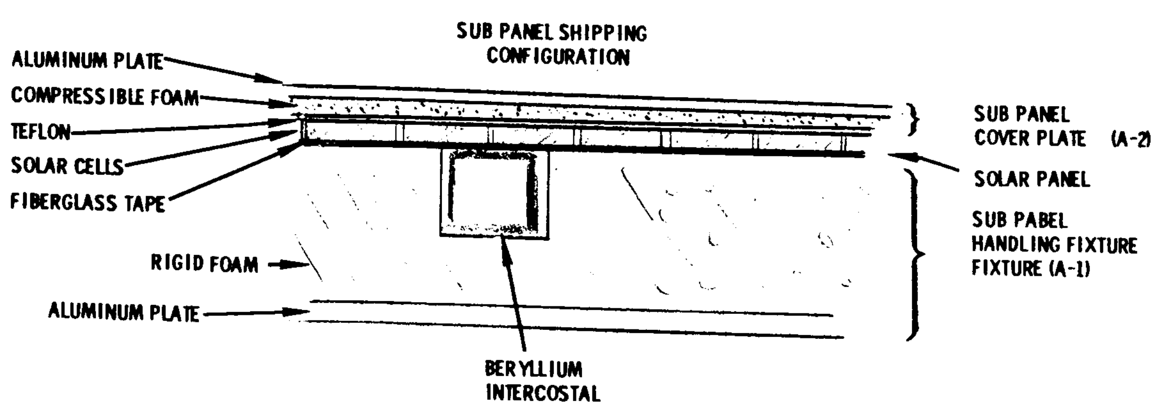

EMBLE &
E OR
R TO
ERING
PANEL
TE
PANEL
CONTAINER
SEAL BAG
SOLAR SIMULATION
(B-9) SOLAR PANEL EXCITATION EQUIPMENT
TRANSPORT TO REMOTE SOLAR SITE
(A-1)
(A-2)
(A-3)
(A-4)
SOLAR TEST
(B-1) EQUATORIAL MOUNT
(B-2) SHADE TUBE
(B-3) PORTABLE SHELTER
(B-4 THRU -7) SOLAR IRRADIATION INSTRUMENTATION
*SEE BELOW
RICAL
INUITY
ECK
AL TEST
ASSEMBLE INDIVIDUAL SUB PANEL UNITS
WEIGH & BALANCE INDIVIDUAL SUB PANELS
(B-16) WEIGH & BALANCE FIXTURE
TRANSPORT TO FACTORY
(A-1)
(A-2)
(A-3)
(A-4)
*(B-8) SOLAR PANEL TEST CONSOLE
(B-23) SUNLIGHT ATTENUATOR
(B-24) COOLING AIR PLENUM
(B-25) PORTABLE AIR SUPPLY
ATION
STS
WED PANEL
RATION
ELECTRICAL CONTINUITY
COMMERCIAL TEST EQUIPMENT
ZERO G DEPLOYMENT
(B-14) ZERO G DEPLOYMENT FIXTURE
SOLAR SIMULATION
(B-9) SOLAR PANEL EXCITATION EQUIPMENT
EIVING
PECTION
TR
TRANSPORT TO FACTORY - STORE OR SHIP TO FLORIDA
(B-17)
(B-20)
SOLAR TEST
(B-1) (B-8) (B-25)
(B-2) (B-23)
(B-3) (B-24)
(B-4 THRU -7)
DISASSEMBLE & TRANSPORT TO REMOTE SOLAR SITE
(A-1)

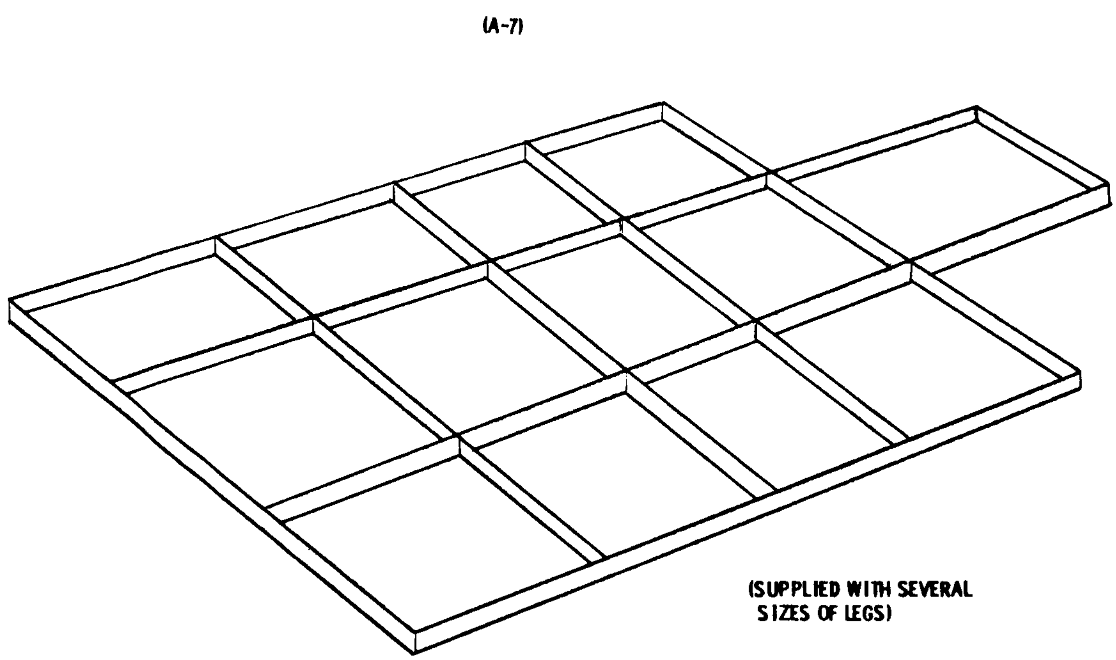
FIGURE 7.4-3

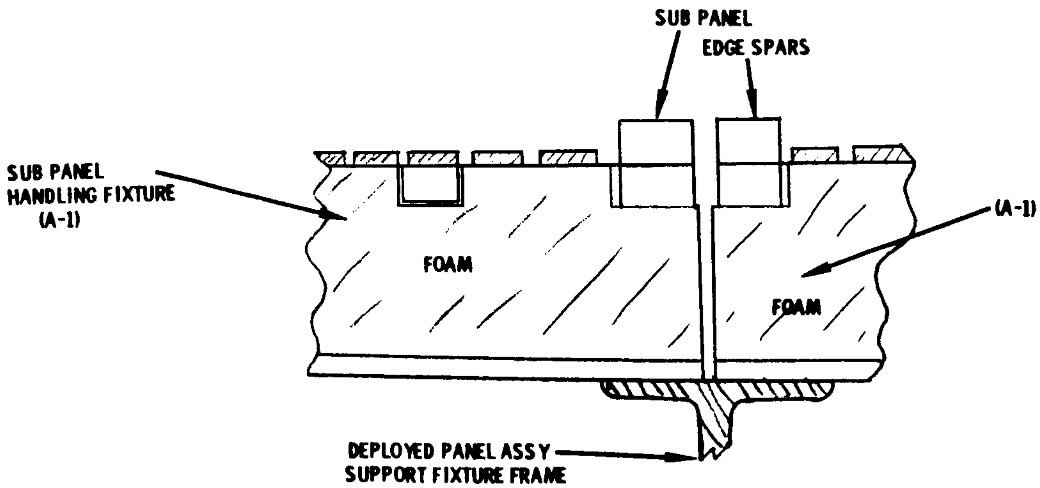




SUB PANEL SHIPPING & HANDLING

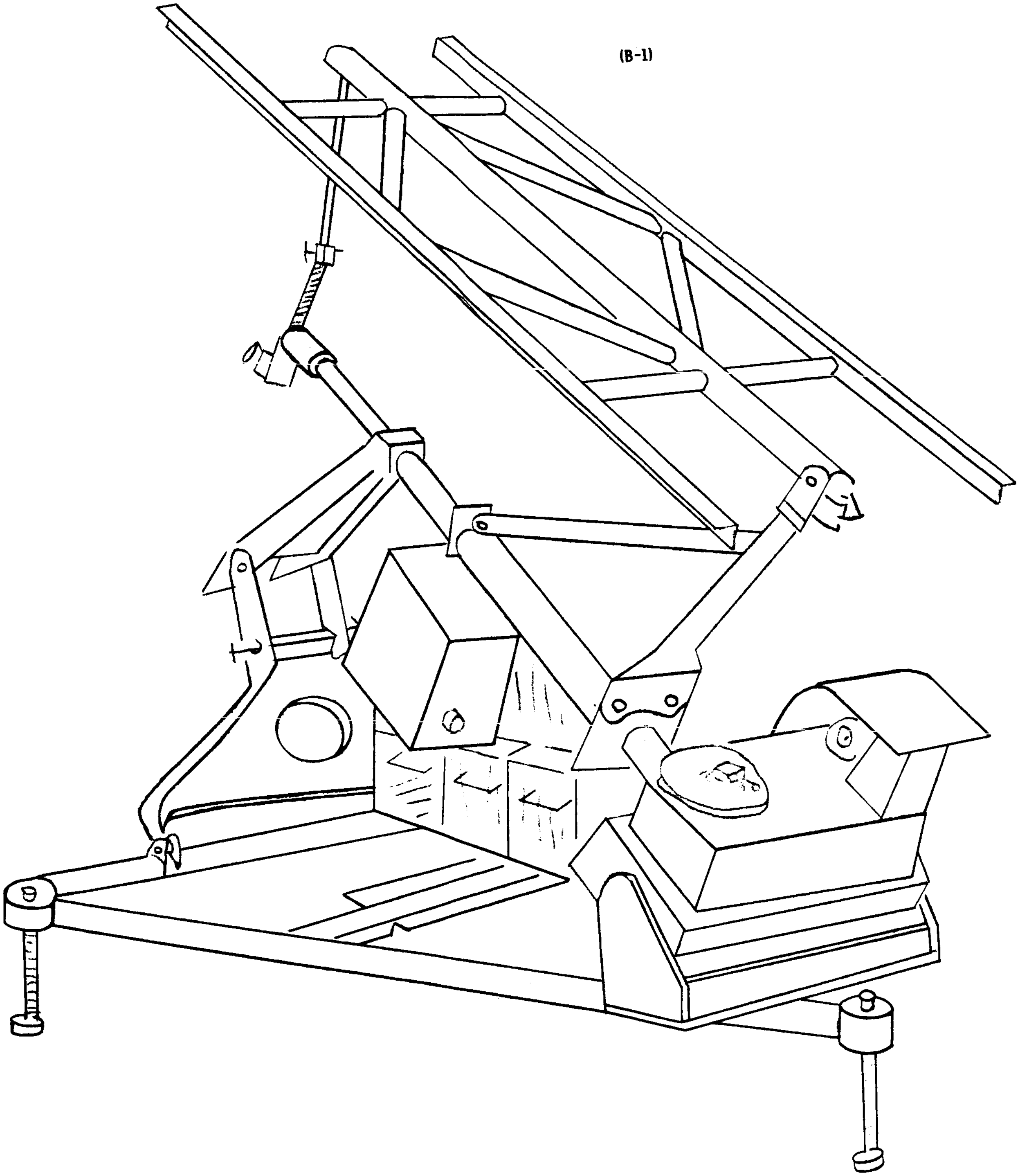
FIGURE 7.4-4





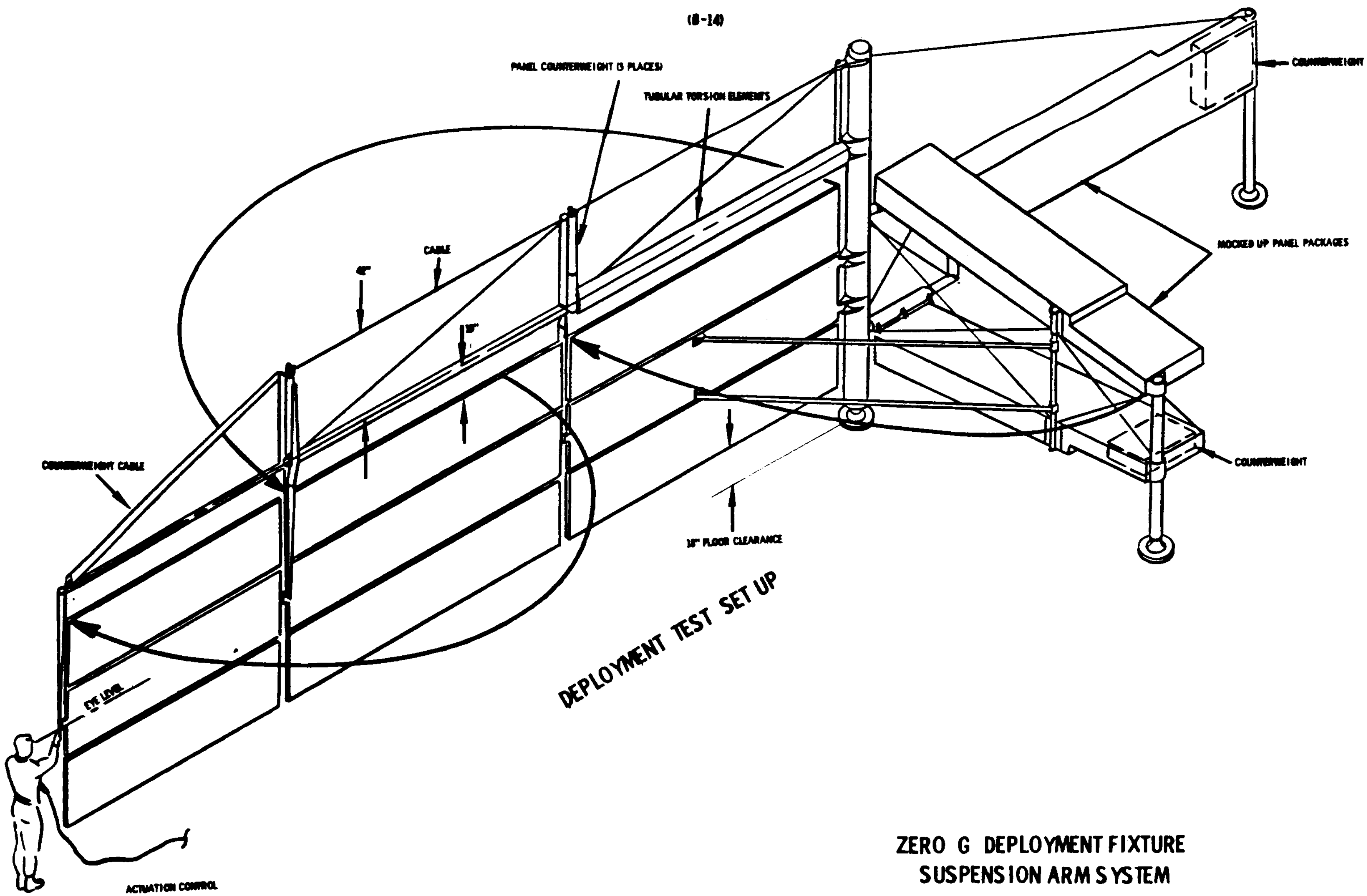
DEPLOYED PANEL ASSEMBLY
SUPPORT FIXTURE

FIGURE 7.4-5



EQUATORIAL MOUNT

FIGURE 7.4-6



ZERO G DEPLOYMENT FIXTURE
SUSPENSION ARM SYSTEM

FIGURE 7.4-7

ZERO G DEPLOYMENT FIXTURE
(AIR BEARING CONFIGURATION)
ALTERNATE # 2

(B-13)

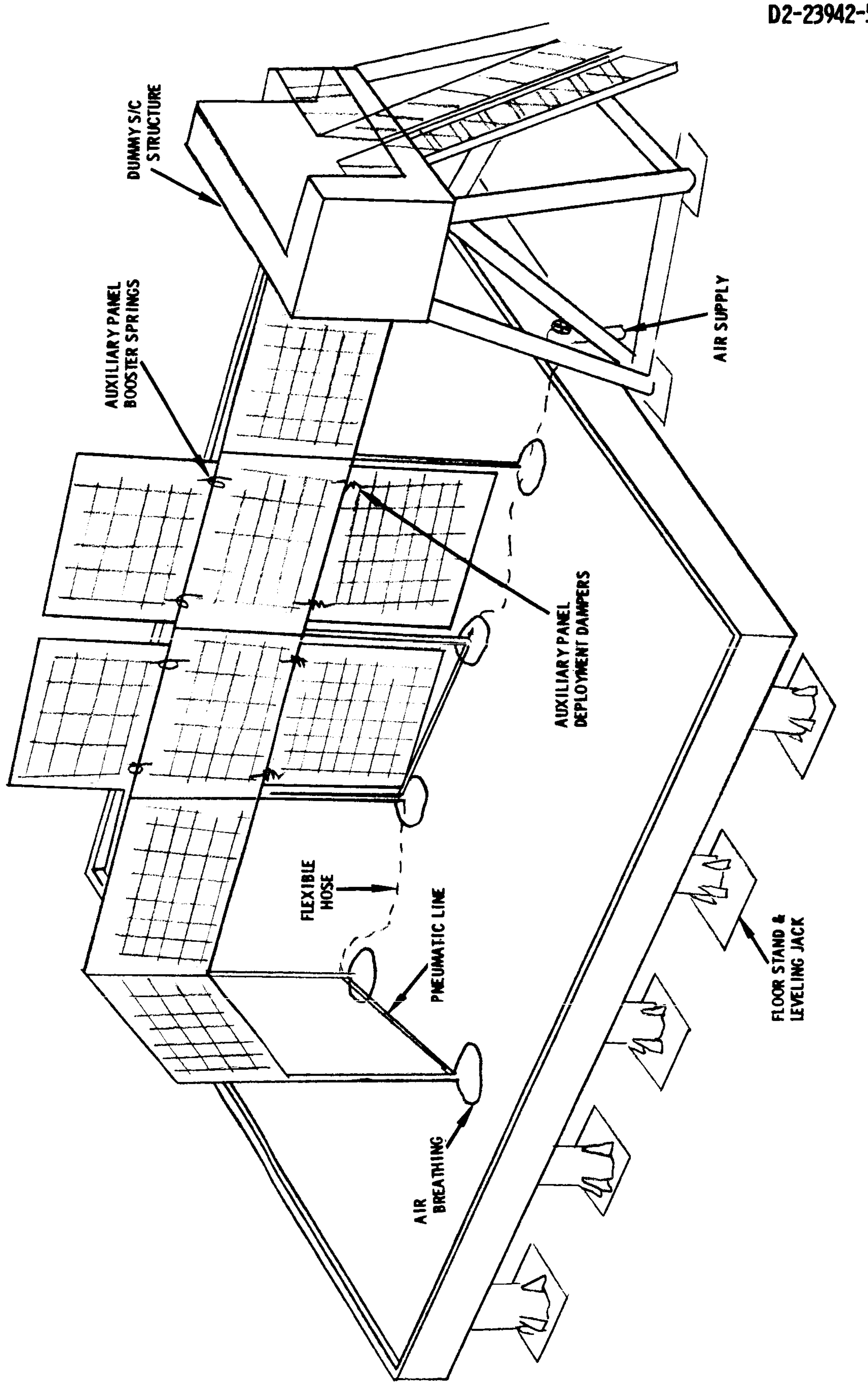
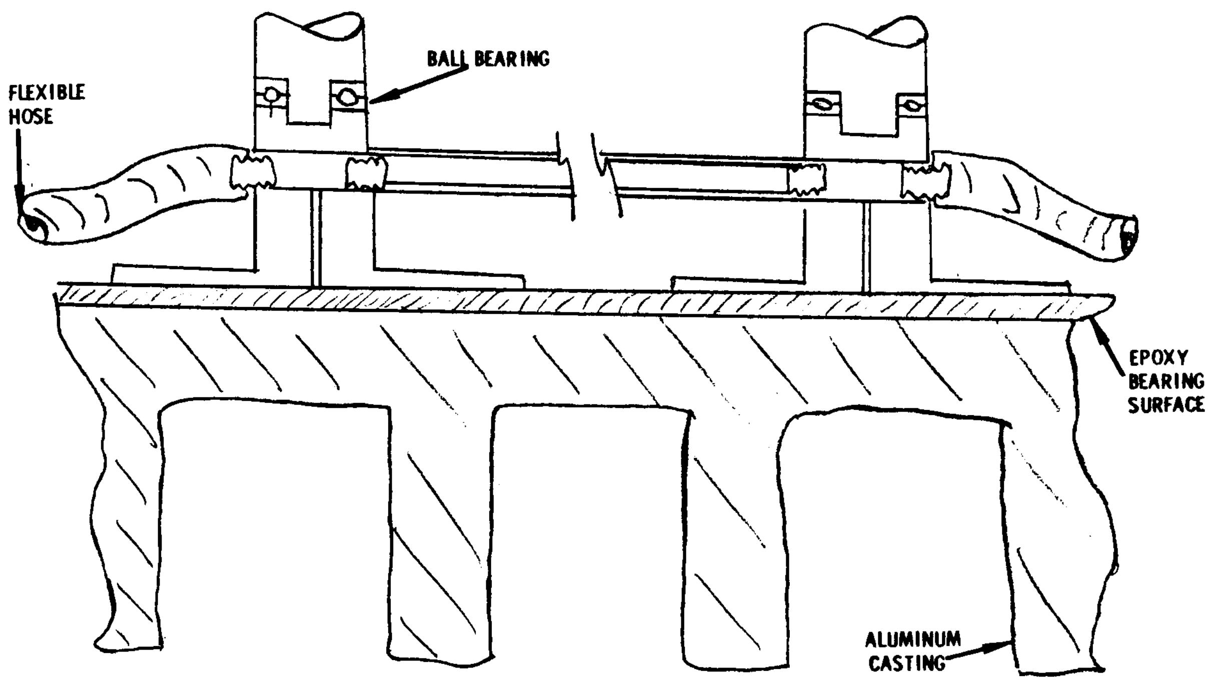


FIGURE 7.4-8

AIR BEARING DETAIL



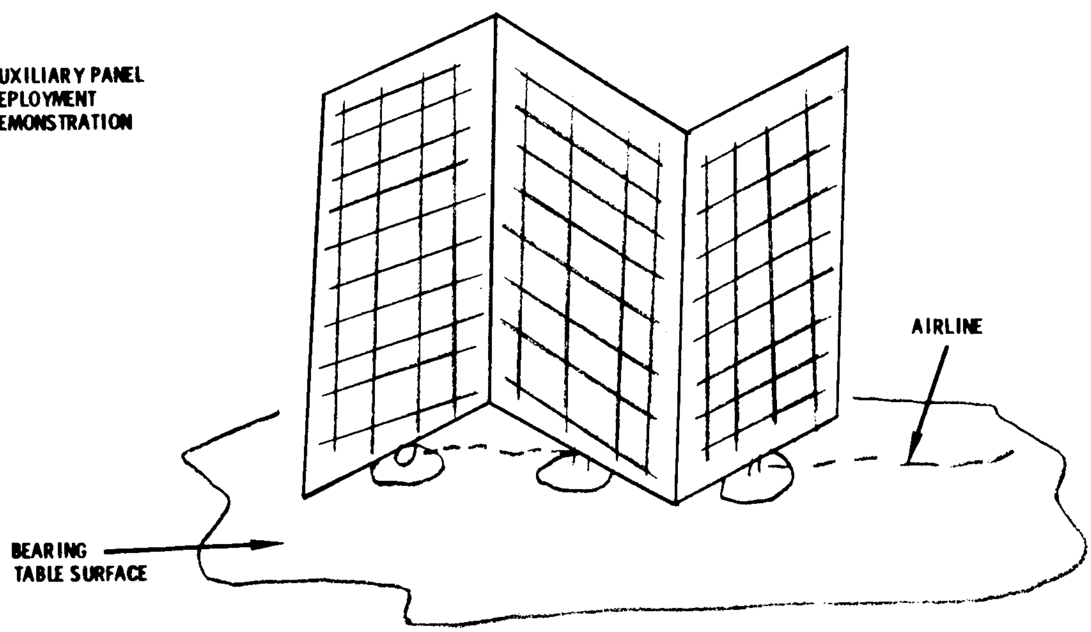


FIGURE 7.4-8A

PORTABLE SHELTER

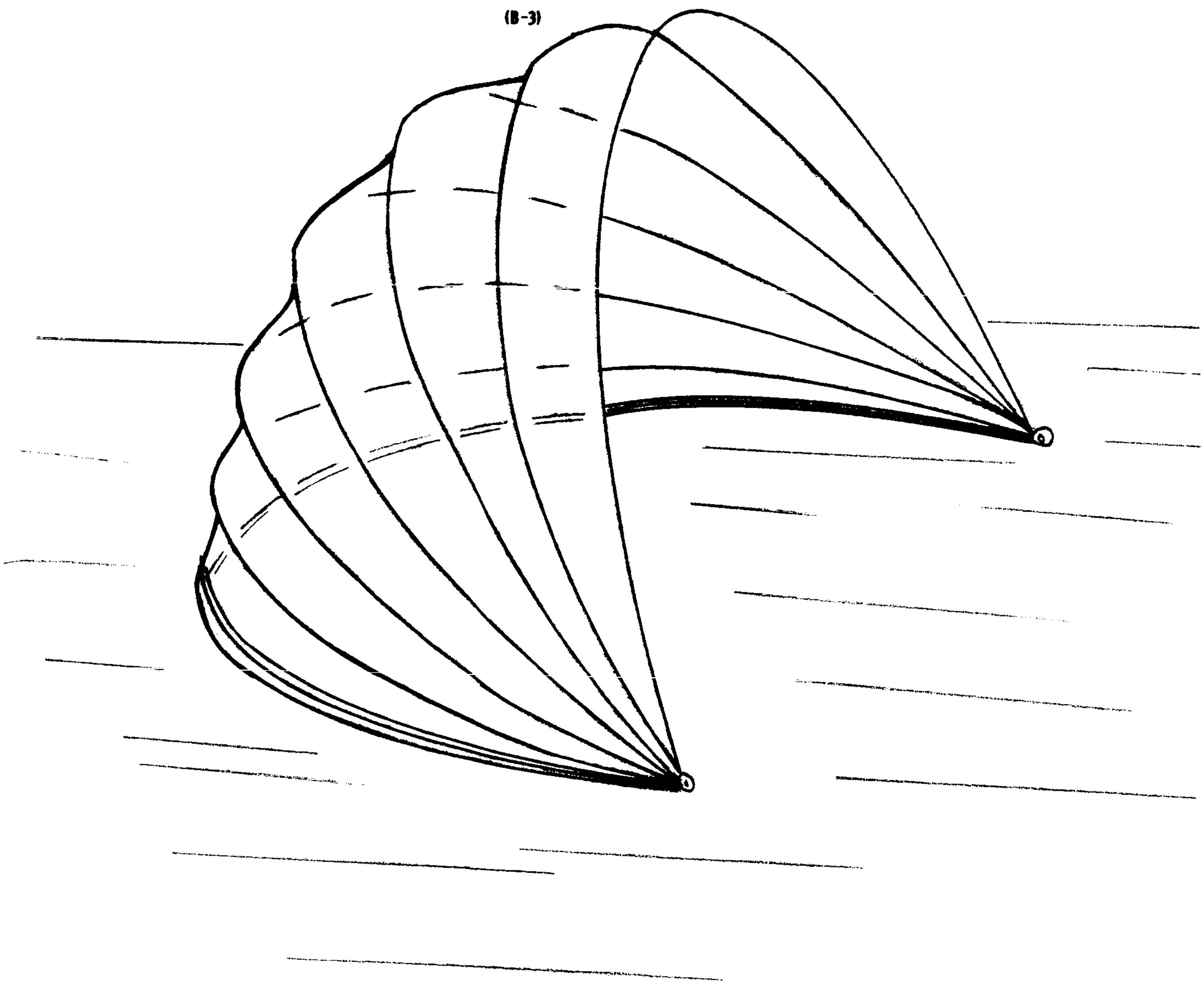


FIGURE 7.4-9

SOLAR SUB PANEL TEST SET UP

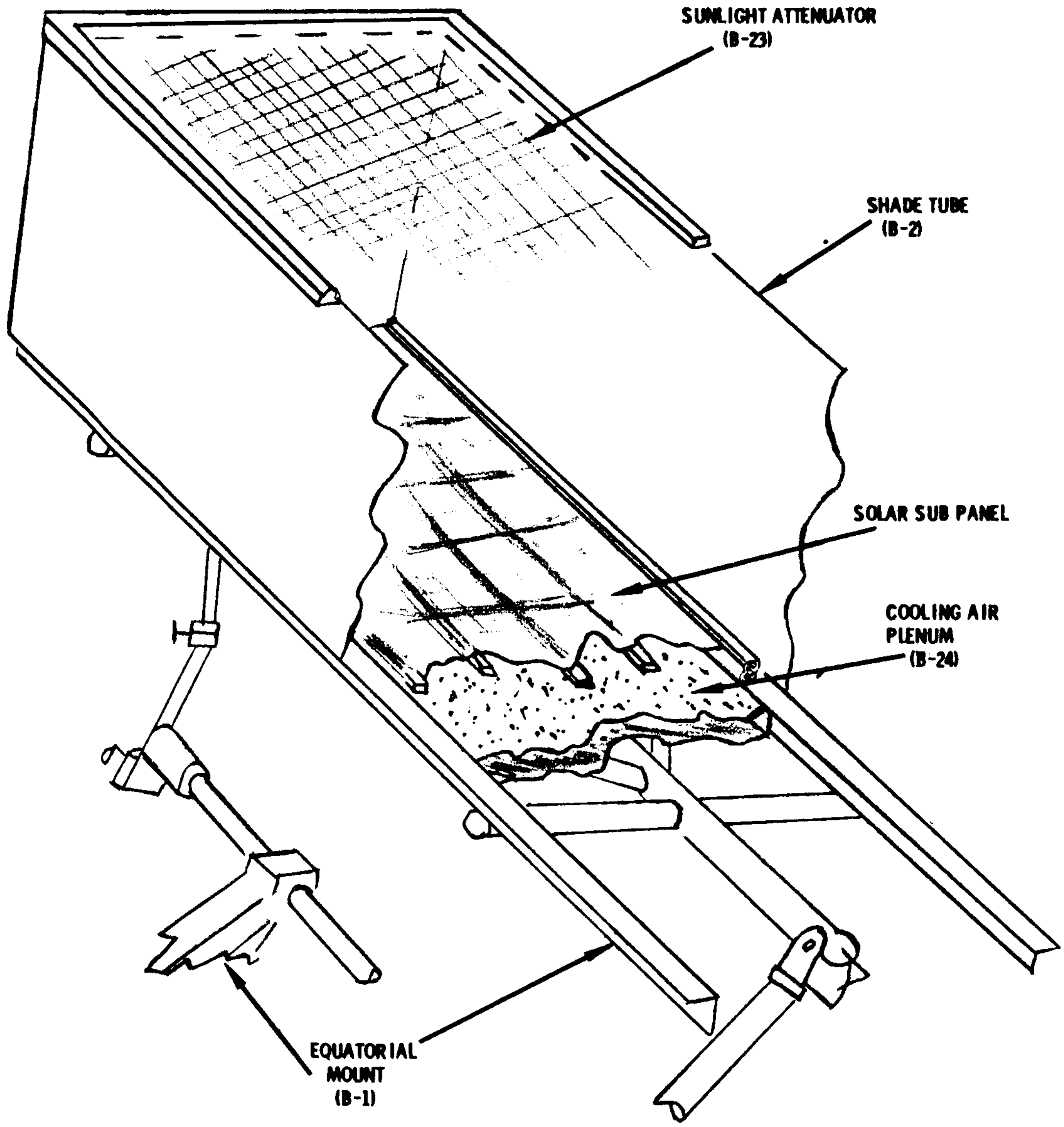


FIGURE 7.4-10

7.5 MANUFACTURING PLAN

The plan to fabricate, assemble, and test large, light-weight solar arrays starts with support from a manufacturing development organization. The work centers around developing techniques and tooling criteria so that details may be fabricated and assembled, and solar cells may be installed on a volume basis. With large panel sizes, certain factors must be extended and qualified to provide a satisfactory confidence level. Prototype tools, parts, and adhesive-bonded assemblies should be made and tested to establish production performance to engineering specifications. Automatic or semi-automatic concepts must be developed.

7.5.1 Fabrication

The cutting, forming and handling of beryllium is of major concern due to the toxic health hazards involved and the exacting long cycle forming operations. The beryllium parts used for the structural framework originate from thin-gaged sheets, generally .015 and .020 inch thickness. They can only be effectively cut to net size by abrasive sawing methods. The U-type channels used for spars and intercostals must be creep-formed between electrically heated ceramic dies at about 1300°F and with a bake-time of about 3 hours. The overall processing will remain complex until rectangular extruded beryllium tubing is economically obtained. The fabrication of titanium fittings and aluminum parts can be accomplished by conventional methods.

7.5.2 Assembly Bonding

The assembly of details to form the solar panels begins with the bonding of formed beryllium channels to produce spars, end pieces, and intercostals. The straightness of the long channels is critical in obtaining consistent adhesive bond-lines for the 13 foot long rectangular spars. The rectangular tubing is cut to exact lengths and loaded into sub-panel assembly jigs. To minimize array weight, all materials, including the bonding adhesives, should be accurately measured and applied. Progressive assembly by bonding using heat for curing, must be monitored so that no degradation occurs in previously completed steps.

The "basket weave" substrate made of bonded fiberglass tape is accomplished in sub-panel sizes and bonded to the beryllium framework. Hinge fittings, deployment mechanisms and other structural members will also be bonded to the beryllium framework. Where possible, all hinge fittings and deployment mechanisms should be drilled and reamed prior to solar cell and electrical installation work.

Throughout the assembly operations a high standard of cleanliness must be imposed in the handling and storage of parts. Any contaminants may be seriously detrimental to the quality of bonding achieved.

7.5.3 Solar Cell and Electrical Installations

Sub-panel frames are moved to a class 100,000 clean room, as defined by Federal Specification 209, for installation of electrical components, solar cells and mechanical details. The panels are then ready for final operational and functional testing and inspection.

No state-of-the-art problems are anticipated in solar cell and electrical operations. However, an extensive training program will be required for the installing personnel. Studies should be made to develop mechanized methods for soldering solar cells to produce a reliable and economical assembly. Prototype assemblies are required early in the program to provide for methods analysis and testing procedure development. Methods for replacement of solar cells and cover glasses and other field repairs must be developed and proven.

7.5.4 Quality Control Test

Receiving inspection is required on all materials and equipment to verify conformity with engineering specifications.

Sub-panel testing for required power and voltage output under controlled stimulation, and for continuity of deployment sensor cables, jettison cables, and power busses is also a requirement.

Full panel level tests of manual deployment and folding into the stowed position to verify mechanical freedom, function of mechanical devices, and integrity of electrical circuits are necessary.

7.5.5 Packaging and Shipping

Due to the fragile nature of the solar panels, packaging and shipping containers must be designed and fabricated for in-plant handling of panel frames. These closed containers, lined with cushion materials will provide security against shock damage, contamination and other detrimental conditions when the panels are in storage or between shop moves. Upon completion of tests and acceptance by Quality Assurance, the solar array must be dis-assembled and individual panels packed in their shipping containers.

7.6 FACILITY REQUIREMENTS

7.6.1 General

The facilities identified for use on the 20 Watt Per Pound Solar Cell Array program should be similar to those located at the Boeing-owned Space Center located near Kent, Washington, which is used as a model in this analysis.

The tailoring of facilities adequate to handle a program of this complexity requires knowledgeable planning for the fabrication and test support needs. In addition to the class 100,000 clean area and support shops for machining and bonding beryllium structures, the availability of a space chamber must be considered. These facilities should include equipment and support services for fabrication, tooling, and other support functions such as development and environmental testing. Office area for management, control, engineering, and operations functions should be located in close proximity to the development, manufacturing and test area to insure rapid reaction to the overall requirements of the program. Basically the program should be a centrally controlled function with fabrication and testing located at a central site, excluding of course, the necessary high altitude testing.

The major support area required within the complex to support this type program is 30,000 square feet for the Saturn/Centaur configuration, and 20,000 square feet for the Atlas/Centaur configuration. Incorporated in this area will be both class 100,000 and class "S" clean areas and beryllium fabrication facilities that are controlled in accord with all safety health hygiene standards.

7.6.2 Production Facilities

An integrated facility for fabrication, cleaning, bonding, and assembly and test should be established in the complex. This should contain a complete beryllium fabrication facility incorporating all the necessary autoclaves, deoxidizing lines and brazing facilities to supplement the necessary machine tooling to create an autonomous beryllium production facility. Figure 7.6-1 indicates the layout, sizing, and flow pattern of the assembly and test complexes required on both the Saturn IB/Centaur, and the Atlas/Centaur programs. Paragraph 7.6.4 indicates the manufacturing equipment required to support the two programs.

7.6.3 Stores

Material and individual cells can first be received into the stores area which is approximately 2,400 square feet for Saturn IB/Centaur and 1,600 square feet for Atlas/Centaur. Storage in both class 100,000 clean areas and the normal storage area is provided to support both the cell group assembly and the beryllium fabrication facility.

7.6.4 Manufacturing Capital Facilities

Saturn IB/Centaur

Bonding Area

1 - Autoclave
10' dia. x 15' long
100 psi - 500°F
1 - Oven, Electric
vented 10 x 15' long x 6' high
300°F
1 - Cleaning and De-oxidizing
Line for beryllium
Tank sizes 3'W x 4'D x 15'L
Complete with 1000# O.H.
1 - Refrigerator 0°F
600 cubic feet

Cell Area

1 - Soldering Oven
Conveyorized for cell group soldering
Elec. 10" wide x 150" long
30 - Electronic Type Benches
13 - Induction Brazing Set - Portable (connector braze)
1 - Photovoltaic Automatic Cell Tester - 1000 cells/hour

Beryllium Area

1 - Stone Saw
18" wheel, Bed Size 36" x 20"
1 - Radius Grinder
6" wheel - 20' travel
1 - Vapor Degreaser Facility
spray immersion tank 18" x 18" x 15" long with vent system
1 - Hydroblast Machine
2' dia. x 4' long
1 - Saw, Band - Do All 36"
1 - Mill, Knee #2

Atlas/Centaur

Bonding Area

1 - Autoclave
8' x 15' long
100 psi - 500°F
1 - Oven, Electric
vented 8 x 15' long x 6' high
300°F
1 - Clean and De-oxidizing
Line for beryllium
Tank sizes 3'W x 4'D x 12'L
Complete with 1000# O.H.
1 - Refrigerator 0°F
600 cubic feet

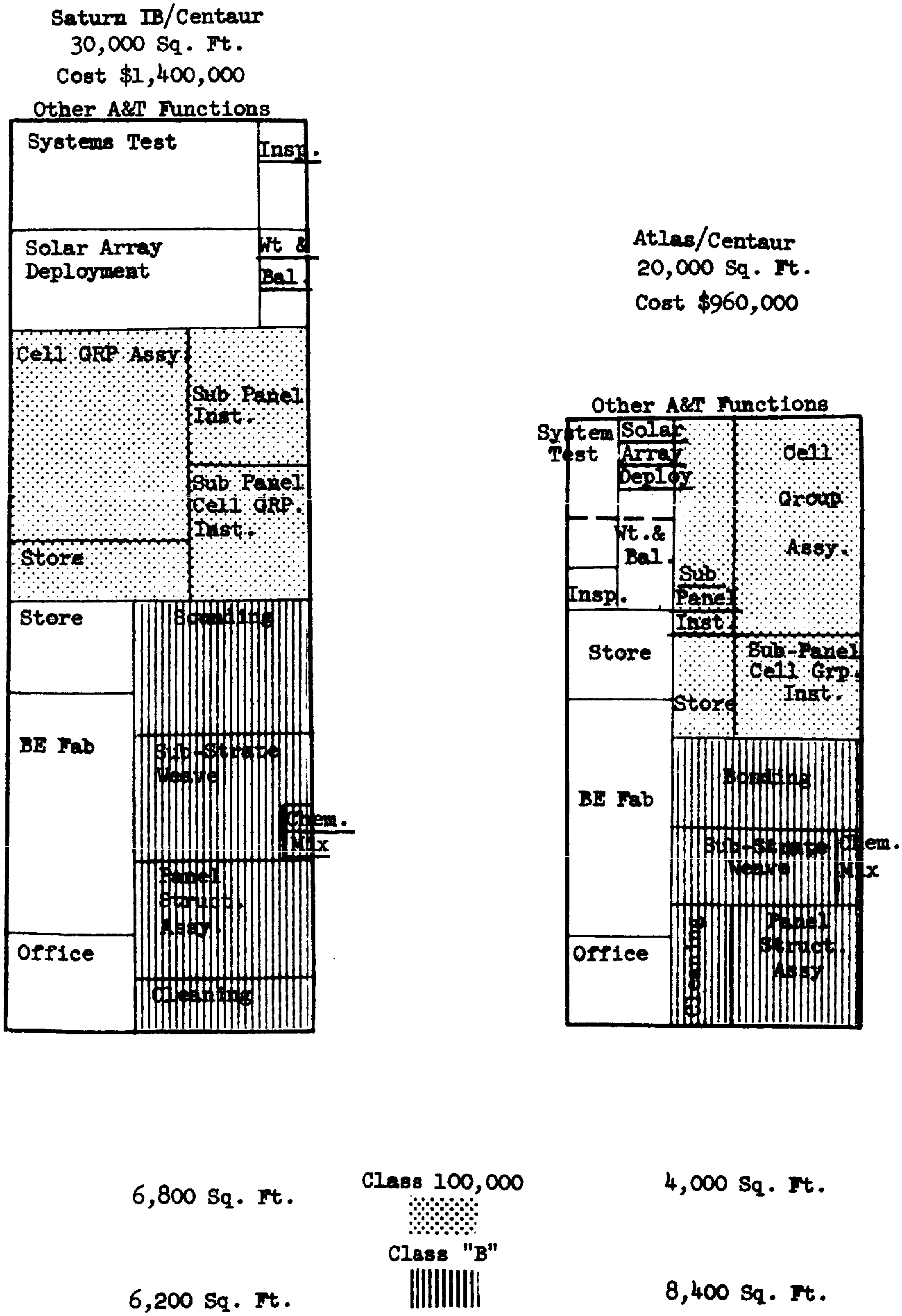
Cell Area

1 - Soldering Oven
Conveyorized for cell group soldering
Elec. 10" wide x 150" long
30 - Electronic Type Benches
13 - Induction Brazing Set - Portable (connector braze)
1 - Photovoltaic Automatic Cell Tester - 1000 cells/hour

Beryllium Area

1 - Stone Saw
18" wheel, Bed Size 36" x 20"
Radius Grinder
6" wheel - 12' travel
Vapor Degreaser Facility
spray immersion tank 18" x 18" x 12' long with vent system
1 - Hydroblast Machine
2' dia. x 4' long
1 - Saw, Band - Do All 36"
1 - Mill, Knee #2

SOLAR PANEL ARRAY
ASSEMBLY AND TEST



D2-23942-5

FIGURE 7.6-1

7.7 COSTS

7.7.1 General Costing Assumptions

A cost estimate is presented here for the development, test, and fabrication of prototype solar arrays for the Atlas/Centaur and the Saturn IB/Centaur concepts. Basis of the estimate was a comprehensive evaluation of the manpower required to perform these major tasks. Since solar cells and beryllium material are critical items from a fabrication and material cost standpoint, supplier quotations were obtained.

The estimate is intended for budget planning purposes. Labor rates are those projected for calendar year 1968. Materials and fabrication techniques are considered to be within the state-of-the-art capability of this time period. The following specific costs have been included.

- Direct labor and overhead.
- Major items of material, with a material burden added but no utilization factor.
- An administrative burden cost.
- Other direct costs such as taxes, freight, and manuals.
- Travel for a representative number of trips and destinations for program coordination.
- Travel and subsistence for technical personnel at the launch site during integration and checkout.
- Hardware consisting of two ground test articles and one prototype mission article.
- Engineering design and development costs of professional engineering effort and the support of engineering laboratories and shops.
- Test includes the same.
- Manufacturing includes all the functions of planning, tooling, inspection and production.

Specifically excluded from consideration in this estimate were:

- Fee.

- Facilities - The value of required facilities was determined, but the assumption is made that these would be available as capital space and equipment by the contractor.

7.7.2 Cost Evaluation

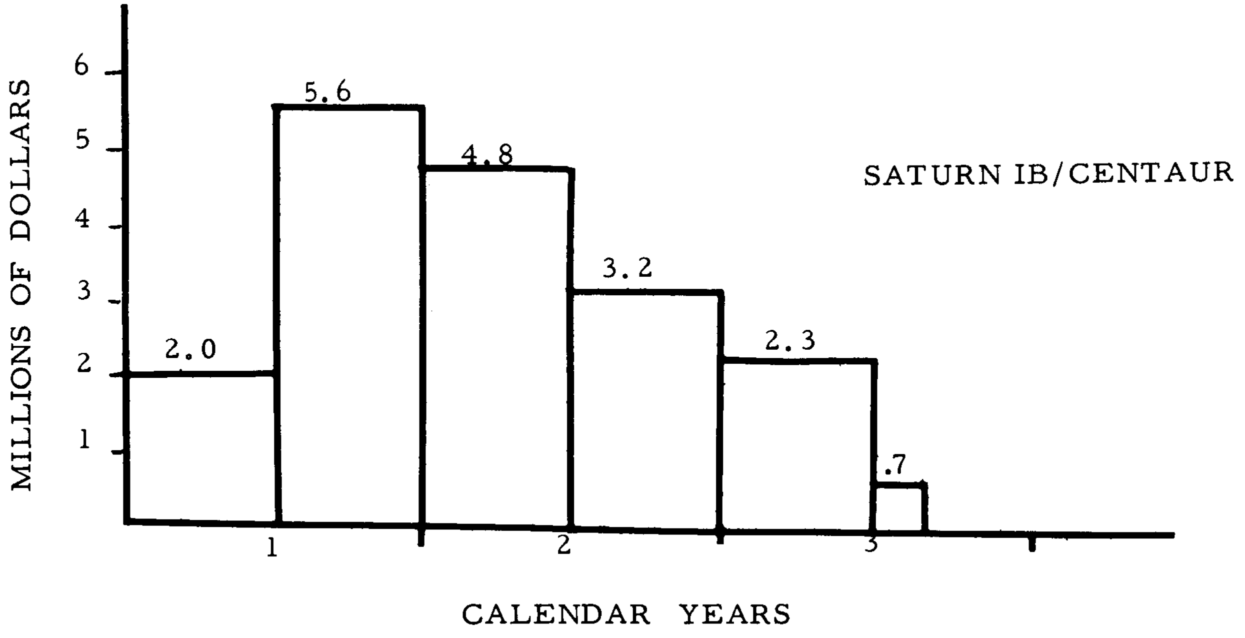
Material costs represent fifty-nine percent of the total cost of the solar array program for the Saturn IB, and thirty-eight percent of the total cost for the Atlas/Centaur. Since solar cells represent half of the total material costs, the conclusion is readily apparent that improvements in solar cell manufacture or efficiency would have a major impact on program costs.

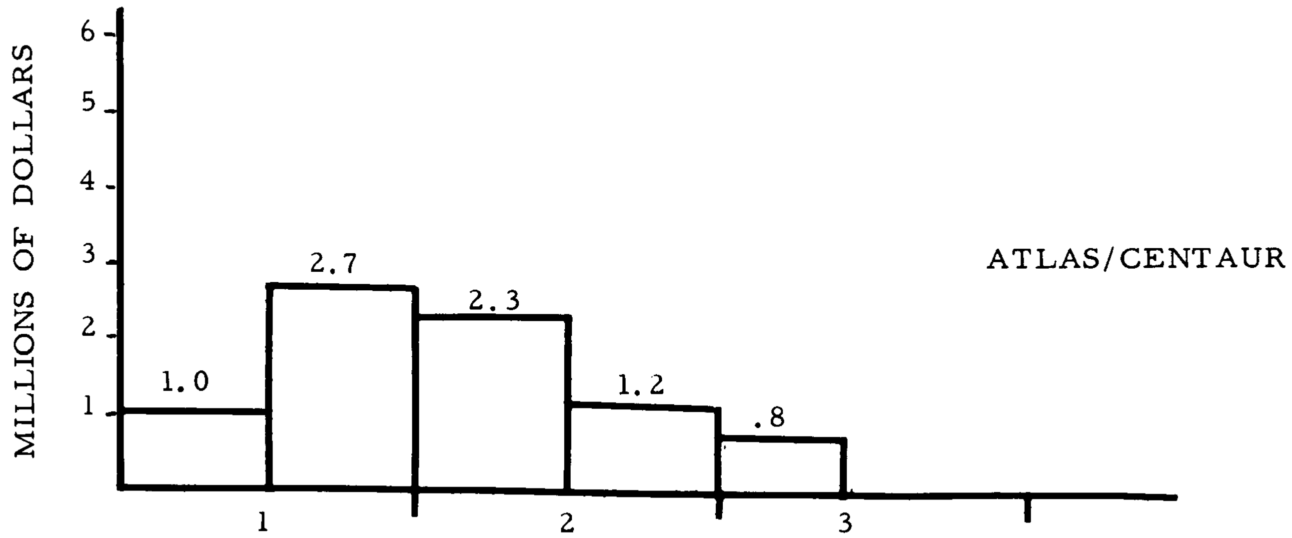
Another area of potential reduction for a solar array program is manufacturing. Eighteen percent of total cost is in manufacturing. The percentage figure is difficult to compare with other programs because of the distortion in cost relationships occasioned by high material costs. However, the real dollar cost is relatively high, and is so because of the difficult and hazardous nature of beryllium fabrication. A substantial reduction in manufacturing costs would be realized by a feasible substitution of structural material.

7.7.3 Comparative Cost Estimates (Dollars in Thousands)

Cost Element	Atlas/Centaur	Saturn IB/Centaur
Engineering Design & Development	$1,091	$1,129
Systems Test Program	705	705
Integration and Checkout	147	173
Manufacturing	1,434	3,223
Material	3,079	11,040
Ground Support Equipment		
Engineering	119	123
Manufacturing	1,099	1,259
Material	98	149
Travel	83	91
Spares	226	713
TOTAL PROGRAM COST	$8,081	$18,605

FUNDING REQUIREMENTS





Funding is based on the master schedules. (Figure 7.1-1 and Figure 7.1-2) It is by calendar year in six month increments for ease in matching to any go-ahead date.

FIGURE 7.7-1

8.0 SAMPLE PANELS AND MODELS

This section summarizes the work accomplished under Phase 2 of the program. Three identical one square foot sample panel sections of the array were designed and fabricated. These samples illustrate the basic cell mounting and interconnection techniques and can be used for certain solar, thermal, thermal shock and acoustic tests. The panel design utilizes aluminum frames, 10 mil thick fiberglass tape, and the 8 mil thick front connected 2 x 2 centimeter silicon solar cells and 6 mil microsheet cover glasses supplied by Jet Propulsion Laboratory. Although the preliminary design of the array is based on beryllium structure, 3 mil fiberglass tapes, back connected cells and 4 mil cover glasses, the basic bonding, soldering and interconnection techniques are closely enough matched to verify the design and fabrication techniques involved. Test results can be successfully interpolated by analysis.

A 1/20th scale hand operated working model of the Saturn IB/Centaur and the Atlas/Centaur solar array have been designed and fabricated. Each model illustrates the stowed relationship of the four array panel assemblies and the spacecraft envelope. On each model two of the panel assemblies are capable of being extended in a 1 "g" field and will illustrate the launch packaging and deploying techniques of the array. In addition, a full scale model of one of the overcenter latching hinges has been fabricated. When operated by hand, this model will illustrate the latch technique used by most of the sub-panels.

8.1 SAMPLE PANELS

8.1.1 Structural Design

A sample panel of one square foot frame area was designed as part of the Phase II program for the feasibility study. As shown on Figure 8.1-1, the frame of the panel is a channel section of 2024-T3 aluminum, 1.2 inches wide and bonded in the corners with BMS 5-29. The width of the channel permits mounting of cells in a manner similar to the cells mounted on the intercostals of the array preliminary designs.

The back of the panel is coated with Laminar X-500 thermal control coating as is the array preliminary design.

8.1.2 Electrical Design

The electrical design for the solar panel models is shown in Figure 8.1-2. Each model carries 196 solar cells. The grouping consists of 28 series-connected sets of 7 parallelled cells. Front connected 8 mil thick solar cells and 6 mil thick cover glasses supplied by JPL were installed.

The cells were interconnected by connectors similar to that proposed for use on the Atlas/Centaur and Saturn IB/Centaur array preliminary designs. The connector was modified to permit its use with front connected cells. The solar cell string was terminated into press-fit teflon-insulated, stud-type terminals mounted on the sidewall of the structure.

The power output of each panel is expected to be 8.1 watts under standard test conditions of 100 mw/cm^2 and a cell temperature of 28°C.

8.1.3 Sample Panel Fabrication

Three sample panels were fabricated to demonstrate the solar cell installation on the fiberglass tape substrate. Bonded aluminum was used for the frame. Pretensioned fiberglass tape was cured in the grid pattery configuration to provide a substrate that is representative of that to be used on the full array. After curing, the fiberglass was bonded to the aluminum frames while maintaining an eight pound pull on each tape.

Solar cells, supplied by JPL, were tested, classified and matched into groups of seven such that the average current of each group is equal to the average current of the lot. A two mil thick expanded silver connector was soldered to the contact strip on the top surface of each cell in the seven cell groups. Cover glasses were bonded to the active face of the cells. The seven-cell groups were placed face down into a master locating fixture which positioned all the cells for one panel. The connectors were then soldered to the back side of the cells.

The installation of solar cells on the panel was accomplished by priming the cells and the fiberglass substrate and applying RTV 40 to the substrate. The frame and substrate was positioned on the prepositioned and soldered solar cells. Weights were placed on the assembly to reduce bond line thickness. The assembly was cured at room temperature and excess RTV removed. The panels were completed by spraying Laminar X-500 on the back of the cells and frame.

During fabrication of the sample panels, various assembly problems were encountered. Many of the problems are identified with the sample panel only and would not apply to the full scale array. The following is a brief outline of the problems.

1. The fiberglass tape could not be reduced in thickness with any consistancy. The full thickness of 0.010 inch was used. 3 mil figerglass tape should be procured for full scale production.

2. The pretension on the tapes increased as the curing fixture expanded at the elevated temperature. This did not present a significant problem on the small panels but would be a major consideration on a full size array.

3. Bonding the fiberglass around the radius corner and to the edge of the frame was a problem on the curved configuration. Excess adhesive was required to secure an adequate bond.

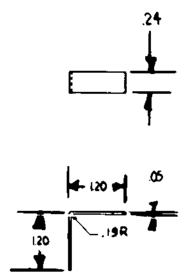
4. It was a problem to form the 0.010 inch offset in the 0.002 inch thick expanded silver connector material for front connected cells. A positive location in the forming tool was difficult to maintain because of the fragile nature of the material. This problem would not occur with the back connected cells specified on the full scale array.

5. The sheared and formed edges of the connectors consisted of the open ends of the mesh material which acted like small hooks. These hooks catch on tools, storage trays, clothing, etc., and sometimes result in distorted connectors or mislocated parts for assembly.

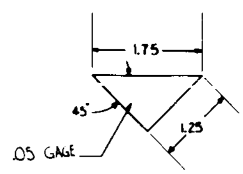
6. The connector configuration, the thin cells, and the front to back connections on the cells all contributed to a problem of maintaining alignment of cell groups and strings. It was necessary to work under a microscope during soldering to maintain proper spacing between the cells.

7. The silver contact surfaces of the solar cells separated from the cell in many cases. This condition was found before and after assembly and usually occurred on the negative contact strip or at the edges of the cell back.

8. The silver surfaces of the cells would tarnish rapidly, but this did not appear to affect the soldering. It was mainly an appearance problem.

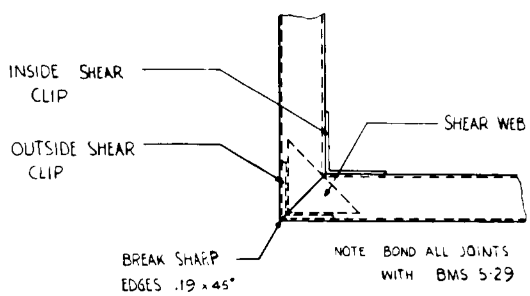


CLIP, INNER
AND OUTER



SHEAR WEB

NOTE: CLIPS AND WEB TO BE MADE OF
SAME MATERIAL AS FRAME



NOTE BOND ALL JOINTS
WITH BMS 5-29

DETAIL I

Fig 8.1-1

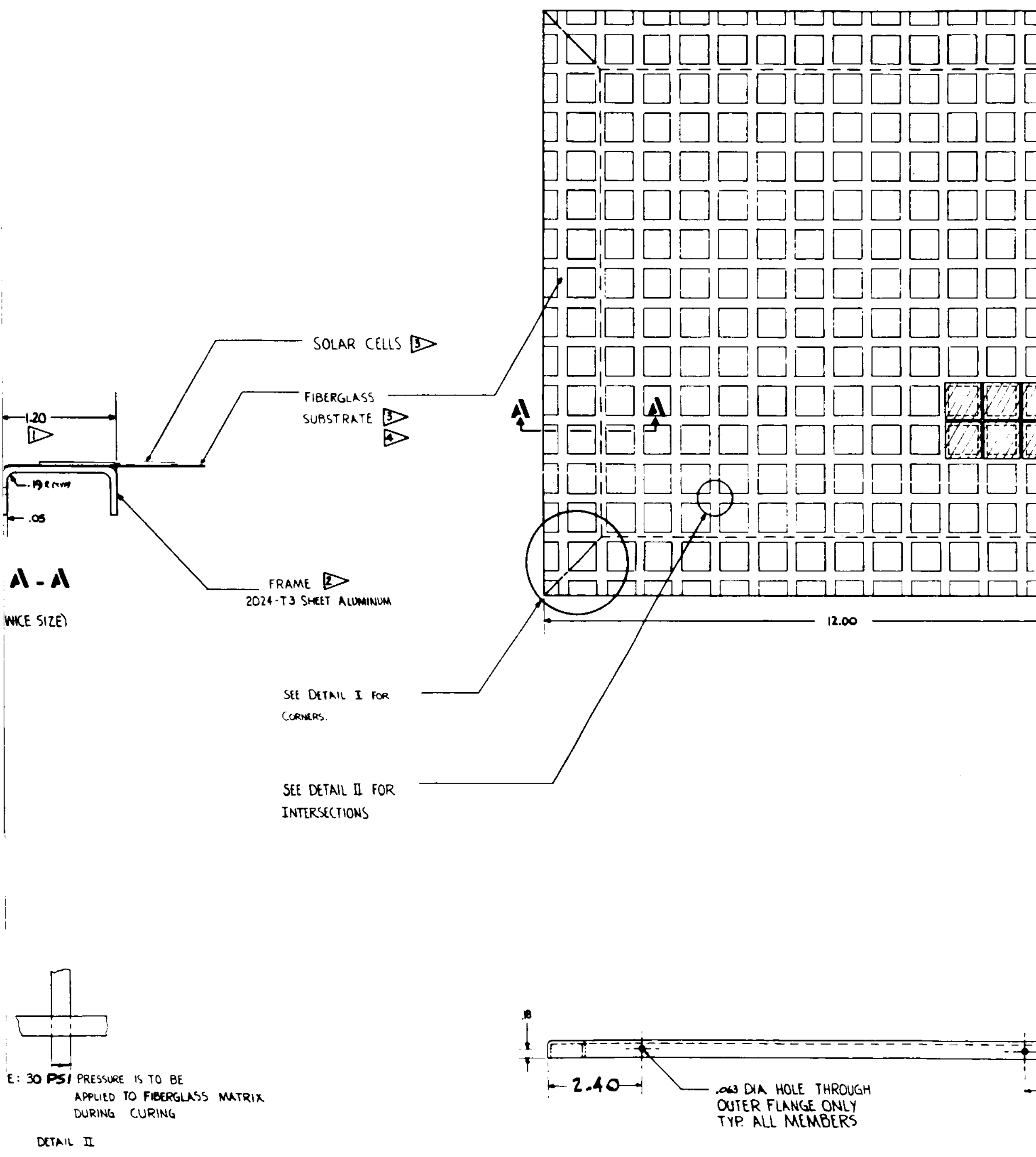
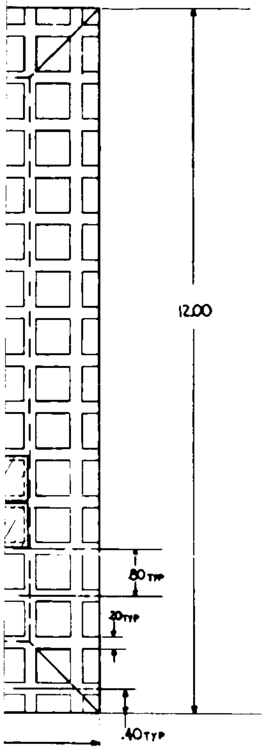
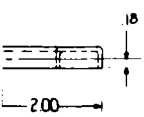

SOLAR CELLS 3
FIBERGLASS SUBSTRATE 3 4
1.20 1
.19 R (TYP)
.05
A - A
(TWICE SIZE)
FRAME 2
2024-T3 SHEET ALUMINUM
A
A
12.00
SEE DETAIL I FOR CORNERS.
SEE DETAIL II FOR INTERSECTIONS
.18
2.40
.063 DIA HOLE THROUGH OUTER FLANGE ONLY TYP. ALL MEMBERS
E: 30 PSI PRESSURE IS TO BE APPLIED TO FIBERGLASS MATRIX DURING CURING
DETAIL II

Fig 8.1-1





4. THE SUBSTRATE IS XP 251S UNCURED SCOTCHPLY PRE PREG TAPE WHICH IS PRE-TENSIONED TO 3 LBS/TAPE AND BONDED TO FRAME PER SPEC BAC 5010 TYPE 38 VENDOR RECOMMENDED CURE: 30 MINUTES AT 300°F. SEE NOTE ON DETAIL II.

3. BACK OF CELLS AND TAPES ARE TO BE COATED WITH LAMINAR X-500

2. TOLERANCES ON FRAME NOT CRITICAL BUT CROSS SECTIONS MUST MATCH FOR BONDING

1. ALL TOLERANCES ± .03 UNLESS OTHERWISE NOTED

FIGURE 8.1-1 SAMPLE PANEL FOLDING MODULAR SOLAR ARRAY

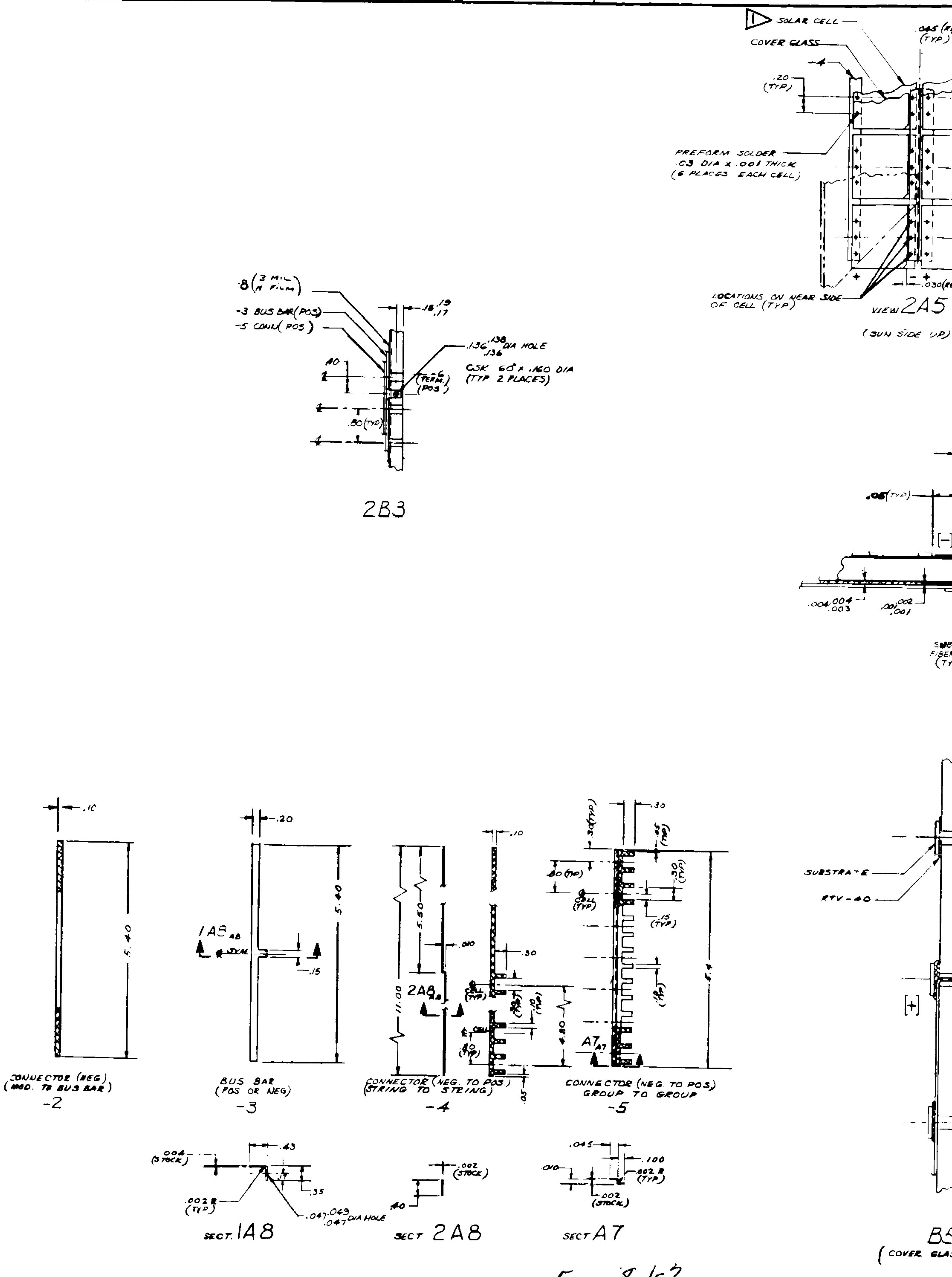
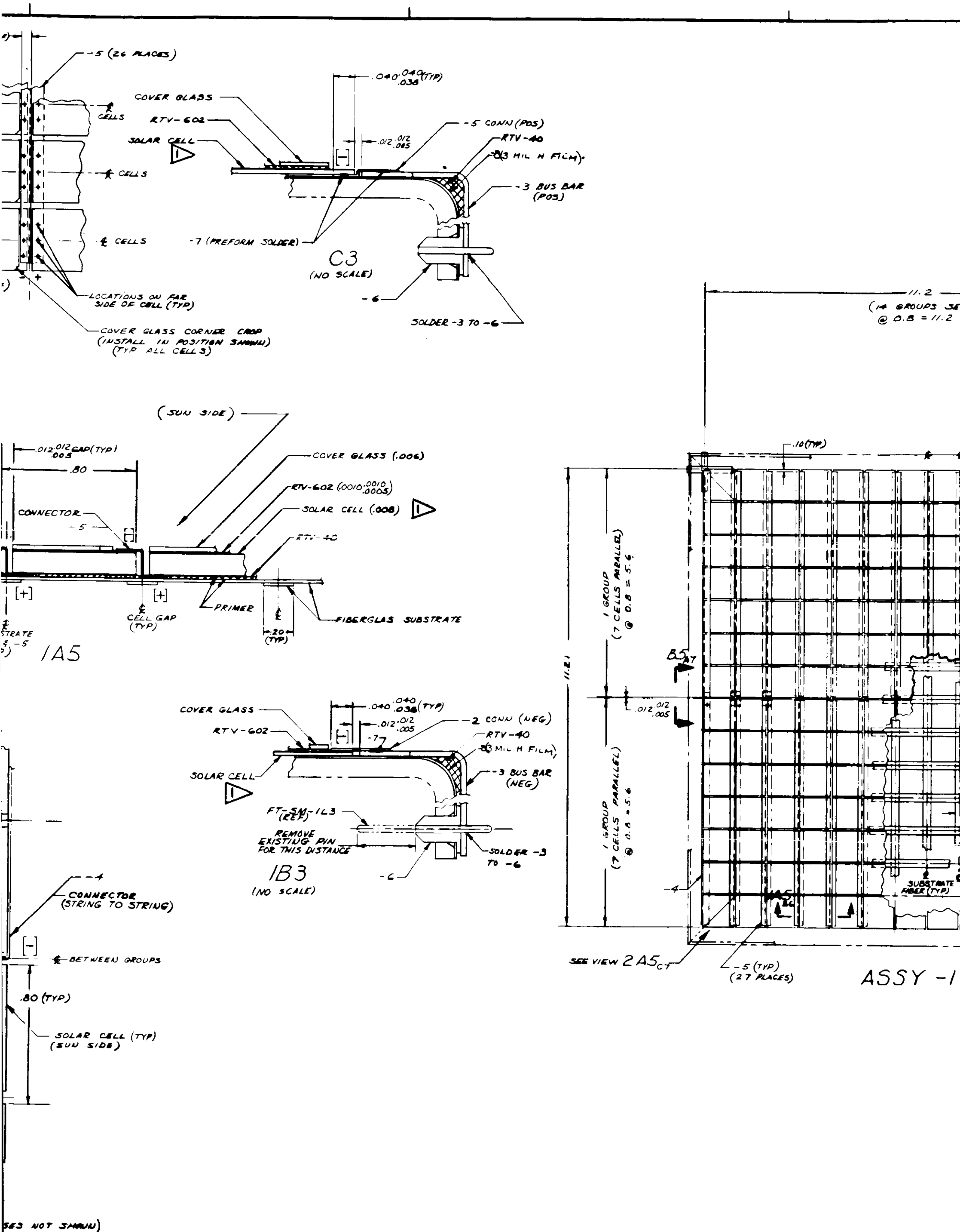

SOLAR CELL
COVER GLASS
PREFORM SOLDER
.03 DIA x .001 THICK
(6 PLACES EACH CELL)
LOCATIONS ON NEAR SIDE OF CELL (TYP)
VIEW 2A5
(SUN SIDE UP)
-8 (3 MIL H FILM)
-3 BUS BAR (POS)
-5 CONN (POS)
CSK 60° x .160 DIA
(TYP 2 PLACES)
2B3
SUBSTRATE
RTV-40
CONNECTOR (NEG)
(MOD. TO BUS BAR)
-2
BUS BAR
(POS OR NEG)
-3
CONNECTOR (NEG. TO POS.)
(STRING TO STRING)
-4
CONNECTOR (NEG TO POS)
GROUP TO GROUP
-5
SECT. 1A8
SECT 2A8
SECT A7

Fig 8.1-2

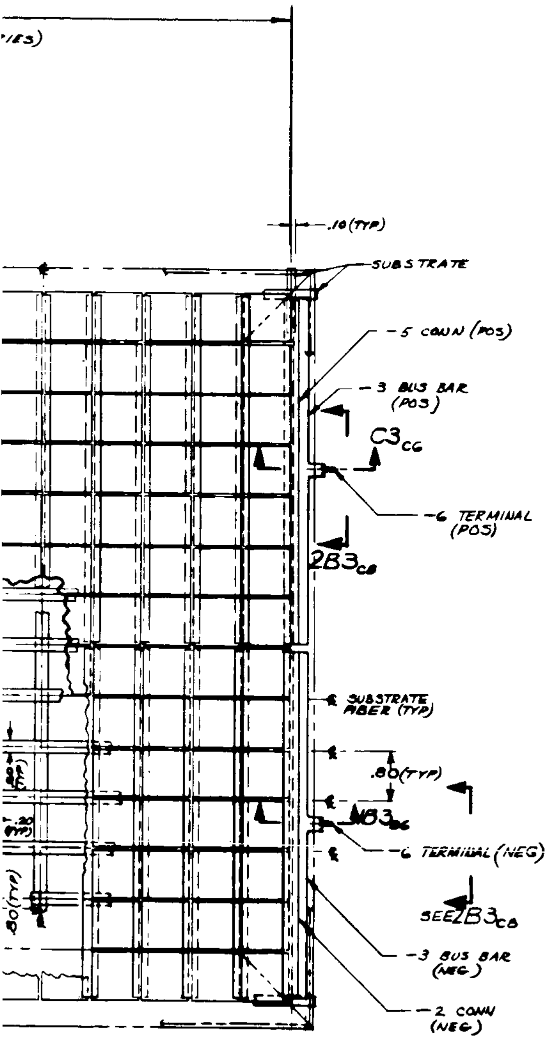
①



-5 (26 PLACES)
COVER GLASS
RTV-602
SOLAR CELL
CELLS
.040 .040/.038 (TYP)
.012 .012/.005
-5 CONN (POS)
RTV-40
-8 (3 MIL H FILM)
-3 BUS BAR (POS)
-7 (PREFORM SOLDER)
C3
(NO SCALE)
-6
SOLDER -3 TO -6
LOCATIONS ON FAR SIDE OF CELL (TYP)
COVER GLASS CORNER CROP (INSTALL IN POSITION SHOWN) (TYP ALL CELLS)
(SUN SIDE)
.012 .012/.005 GAP (TYP)
.80
COVER GLASS (.006)
RTV-602 (.0010 .0010/.0005)
SOLAR CELL (.008)
CONNECTOR -5
RTV-40
PRIMER
CELL GAP (TYP)
FIBERGLAS SUBSTRATE
.20 (TYP)
1A5
COVER GLASS
RTV-602
SOLAR CELL
.040 .040/.038 (TYP)
.012 .012/.005
-7
-2 CONN (NEG)
RTV-40
-8 (3 MIL H FILM)
-3 BUS BAR (NEG)
FT-SM-1L3 (REF)
REMOVE EXISTING PIN FOR THIS DISTANCE
SOLDER -3 TO -6
1B3
(NO SCALE)
-6
-4
CONNECTOR (STRING TO STRING)
BETWEEN GROUPS
.80 (TYP)
SOLAR CELL (TYP) (SUN SIDE)
11.2
(14 GROUPS SE... @ 0.8 = 11.2
.10 (TYP)
1 GROUP (7 CELLS PARALLEL) @ 0.8 = 5.6
11.21
.012 .012/.005
SUBSTRATE FIBER (TYP)
SEE VIEW 2A5 C7
-5 (TYP) (27 PLACES)
ASSY -1
...SES NOT SHOWN)

Fig 8.1-2

-8	INSULATOR	
-7	SOLDER PREFORM	
-6	TERMINAL-PIN	
-5	CONNECTOR - GROUP TO GROUP	2
-4	CONNECTOR - STRG TO STRG	2
-3	BUS BAR	
-2	CONNECTOR-MOD TO BUS BAR	2
-1	SOLAR PANEL ASSY	



Cells to be installed per solar cell installation specification 2-5415-2-38

2 Make from .002 gage expanded metal sheet P.N 2Ag5-6/0 Exmet Corp., Bridgeport, Conn.

1 RCA or Heliotek silicon solar cells. (Provided by Engineering)

FIGURE 8.1-2 SOLAR CELL INSTALLATION

3

8.2 ARRAY MODELS

8.2.1 Hinge Assembly Model - Solar Array (Figure 8.2-1)

The hinge assembly model demonstrates the typical hinge-lock configuration. utilized on the folding modular array preliminary design.

In a stowed position, the hinge halves are folded 180 degrees. During deployment and prior to locking a simulated damper is depressed to absorb the kinetic energy of the panels. The hinge is locked by the overcenter links which are forced past dead center by a torsion spring on the link. On the preliminary design hinges there is a slotted washer at the link pivot joint that prevents the link from unlocking. However, the model has omitted this lock to facilitate repeating of the hinge action.

8.2.2 Solar Array - 1/20th Size Model - Saturn IB/Centaur (Figures 8.2-2 and 8.2-3)

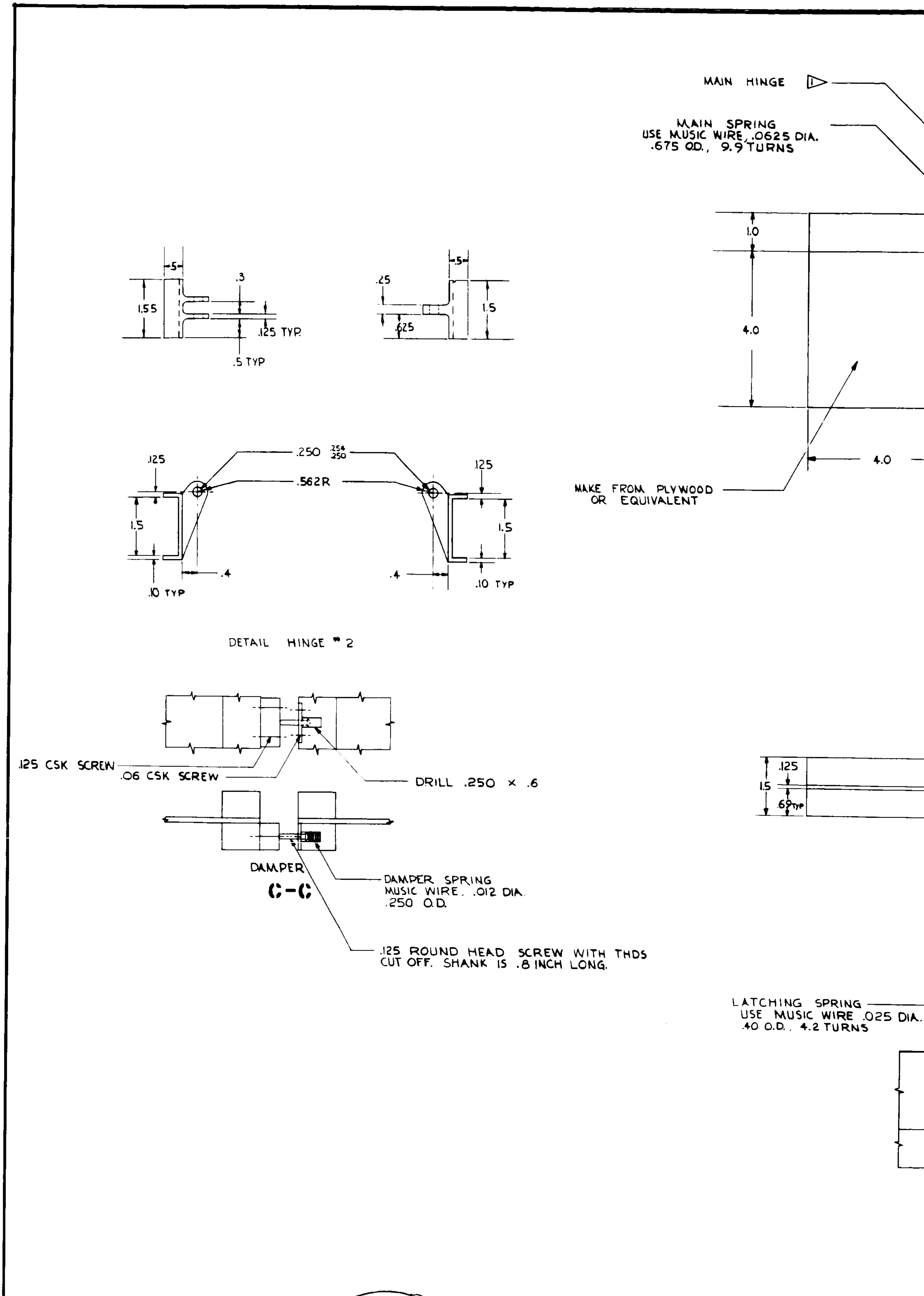
The 1/20 size solar array working model demonstrates approximately the stowed and deployed positions of the array. Only two of the four solar panel assemblies are operated in order to minimize the effects of the one "g" field. The model is also placed on its side to further reduce this effect.

The model is operated manually by a handle. The first motion produced releases the tension-tie down rods which are rotated to a clear position by a torsion spring. This releases the panels for deployment. Further rotation of the handle rotates a drum on which a nylon tape is wound. This nylon tape provides the deployment force to extend the panel assemblies. As the panels are rotated about a hinge centerline, a torsion spring provides a reaction to the tape deployment torque to control the rate of deployment of the sub-panel assemblies. As each main panel is rotated to its deployed position, a flat spring lock engages a notch and simultaneously releases the next sub-panel set for deployment. The auxiliary panels are released in an ordered sequence by a cam operated link-lock at each main panel hinge. The auxiliary panels are deployed by torsion springs and are positioned properly by a mechanical stop.

The solar panels and the spacecraft envelope are made from aluminum. The panels are blue on the sun side and black on the dark side to simulate the solar cells and thermal control coating respectively. The spacecraft is painted black to simulate a thermal control coating.

8.2.3 Solar Array - 1/20th Size Model - Atlas/Centaur (Figures 8.2-4 and 8.2-5)

The Atlas/Centaur working model was made to the same scale (1/20th size) as the Saturn IB/Centaur model in order that there would be a direct size relationship for comparison purposes. The hinges and mechanism details were simplified because of the reduction in area of this model as compared to the Saturn IB/Centaur model. Two panel assemblies (opposites) are deployable while the other two are fixed in the stowed position. The deployment sequence is the same as the Saturn IB/Centaur model. A diagonal brace was added on this model to demonstrate the action of this tension-compression member during deployment.


MAIN HINGE
MAIN SPRING
USE MUSIC WIRE, .0625 DIA.
.675 OD., 9.9 TURNS
1.0
4.0
4.0
MAKE FROM PLYWOOD
OR EQUIVALENT
.5
.3
1.55
.125 TYP.
.5 TYP
.25
.625
1.5
.250 .254/.250
.562R
.125
1.5
.10 TYP
.4
DETAIL HINGE # 2
.125 CSK SCREW
.06 CSK SCREW
DRILL .250 × .6
.125
1.5
.69 TYP
DAMPER
C-C
DAMPER SPRING
MUSIC WIRE, .012 DIA.
.250 O.D.
.125 ROUND HEAD SCREW WITH THDS
CUT OFF. SHANK IS .8 INCH LONG.
LATCHING SPRING
USE MUSIC WIRE .025 DIA.
.40 O.D., 4.2 TURNS

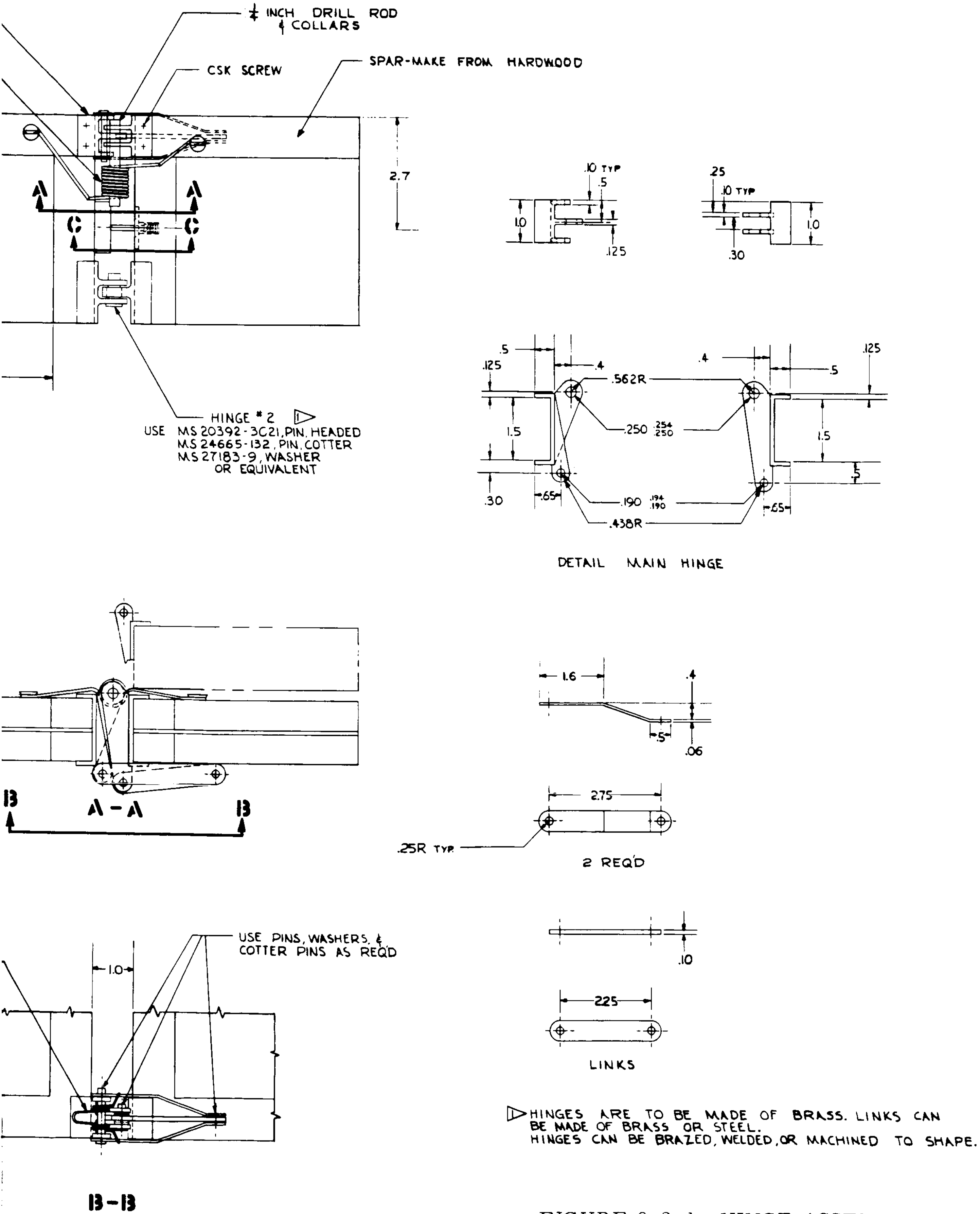
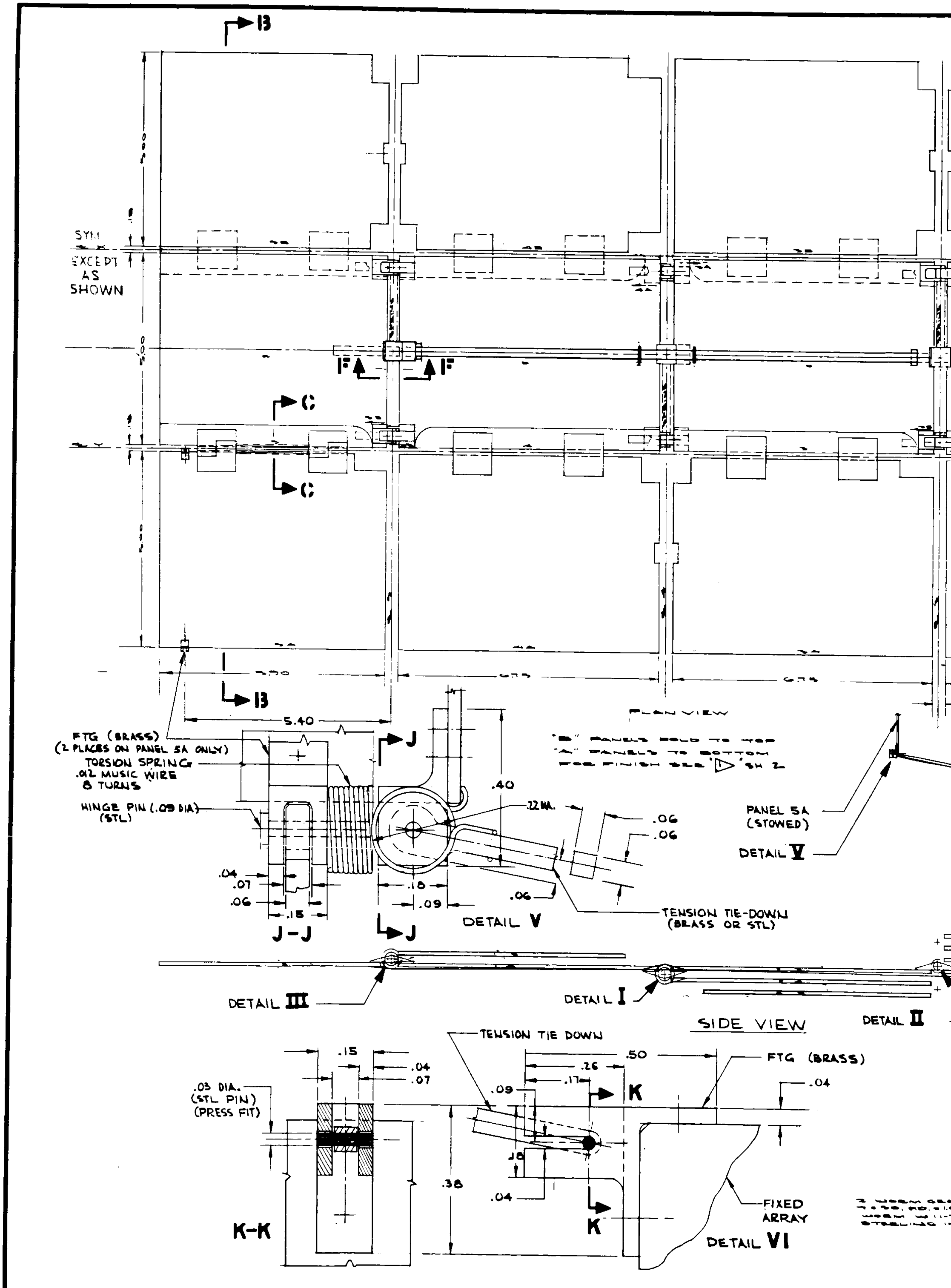


FIGURE 8.2-1 HINGE ASSEMBLY MODEL SOLAR ARRAY


SYM EXCEPT AS SHOWN
B
F F
C C
5.40
FTG (BRASS) (2 PLACES ON PANEL 5A ONLY)
TORSION SPRING .012 MUSIC WIRE 8 TURNS
HINGE PIN (.09 DIA) (STL)
.40
.22 DIA.
.06
.06
.04
.07
.06
.15
.18
.09
.06
J-J
J J
DETAIL V
TENSION TIE-DOWN (BRASS OR STL)
PANEL 5A (STOWED)
DETAIL V
DETAIL III
DETAIL I
SIDE VIEW
DETAIL II
TENSION TIE DOWN
.15
.04
.07
.03 DIA. (STL PIN) (PRESS FIT)
.50
.26
.17
.09
K K
FTG (BRASS)
.04
.38
.18
.04
FIXED ARRAY
K-K
DETAIL VI

289

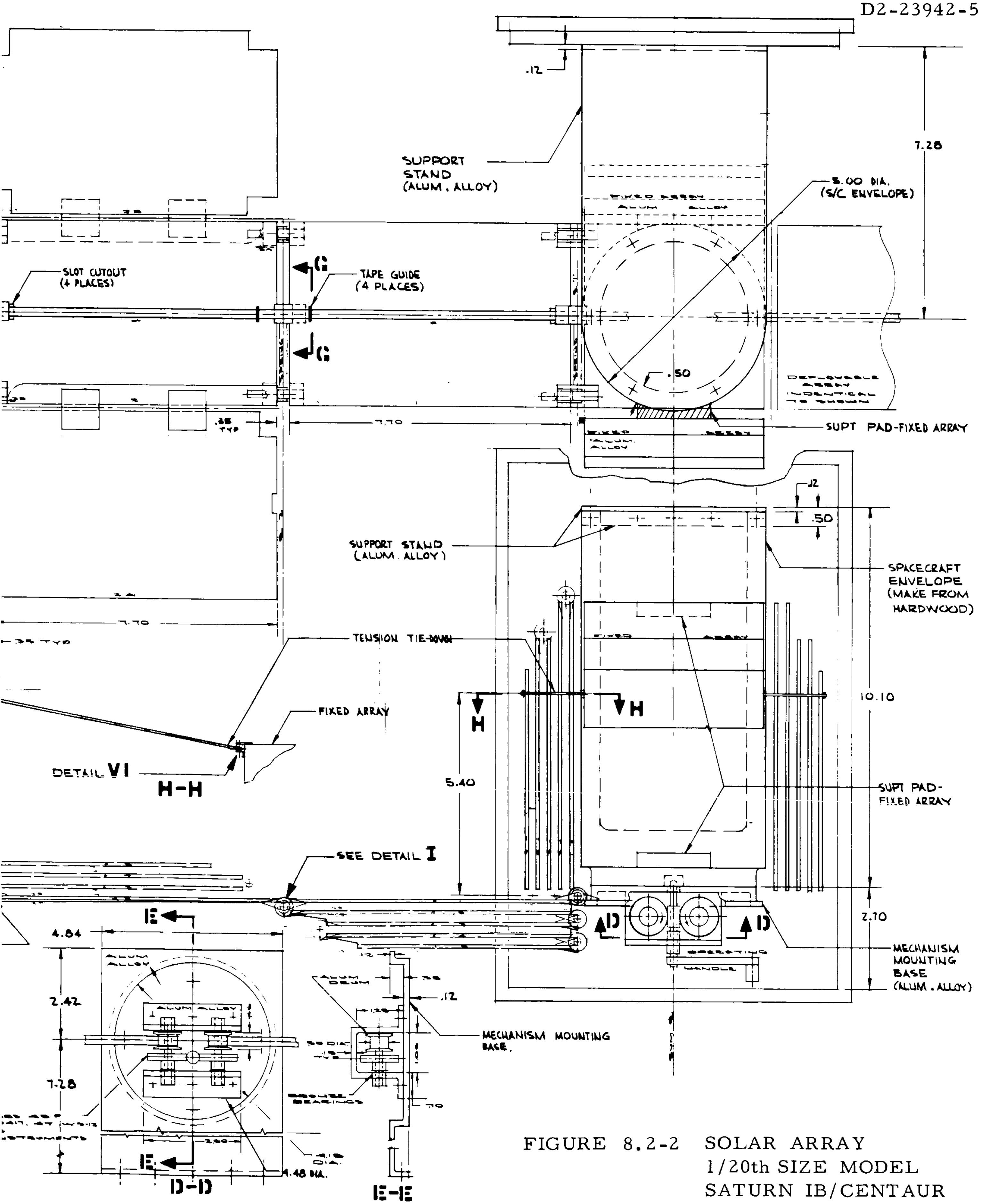
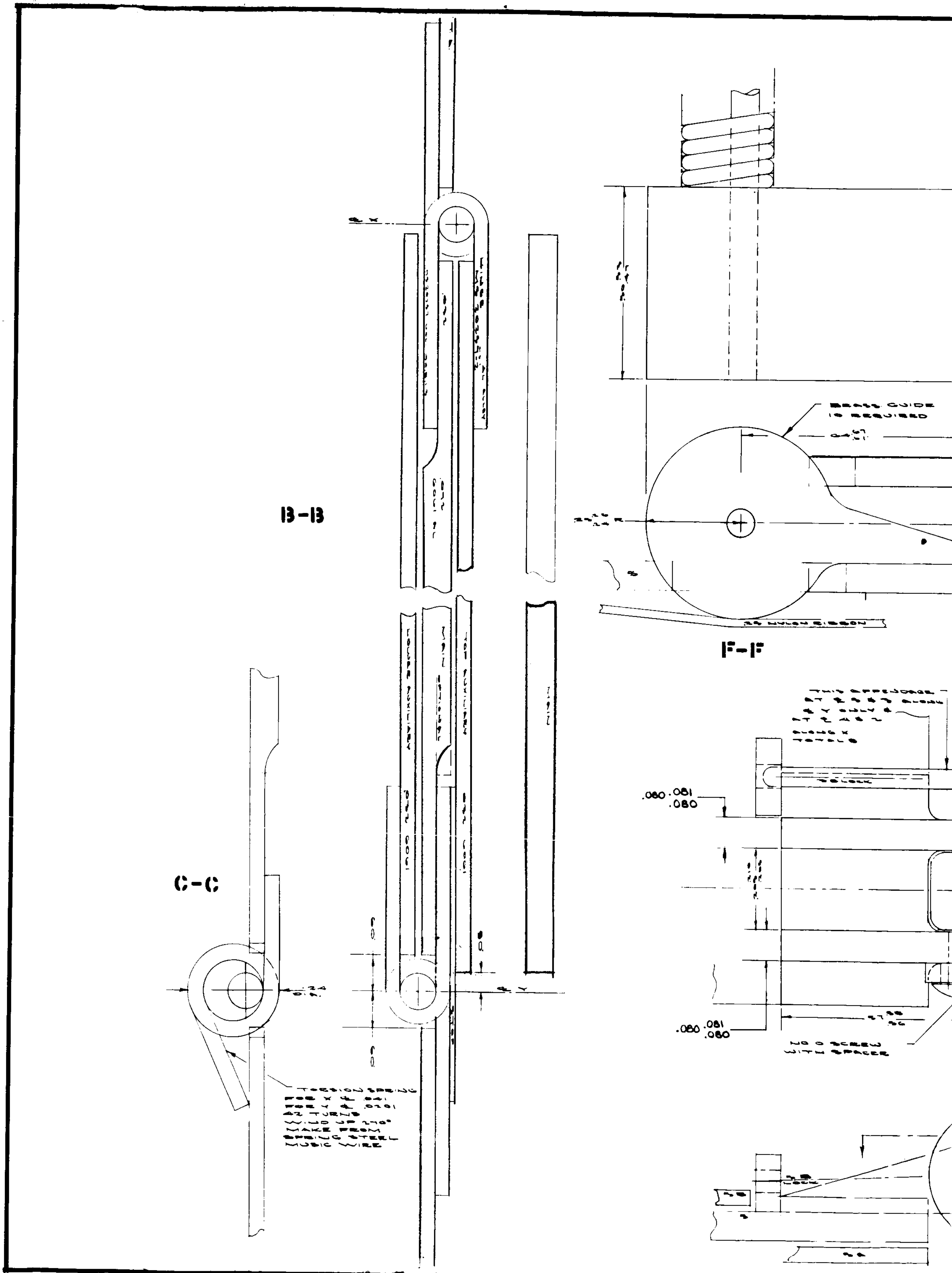


FIGURE 8.2-2 SOLAR ARRAY 1/20th SIZE MODEL SATURN IB/CENTAUR


B-B
C-C
F-F
.080 .081 .080
.080 .081 .080
TORSION SPRING
42 TURNS
MAKE FROM
SPRING STEEL
MUSIC WIRE
BRASS GUIDE
NYLON RIBBON
NO 0 SCREW
WITH SPACER

F1G 8.2-3
①

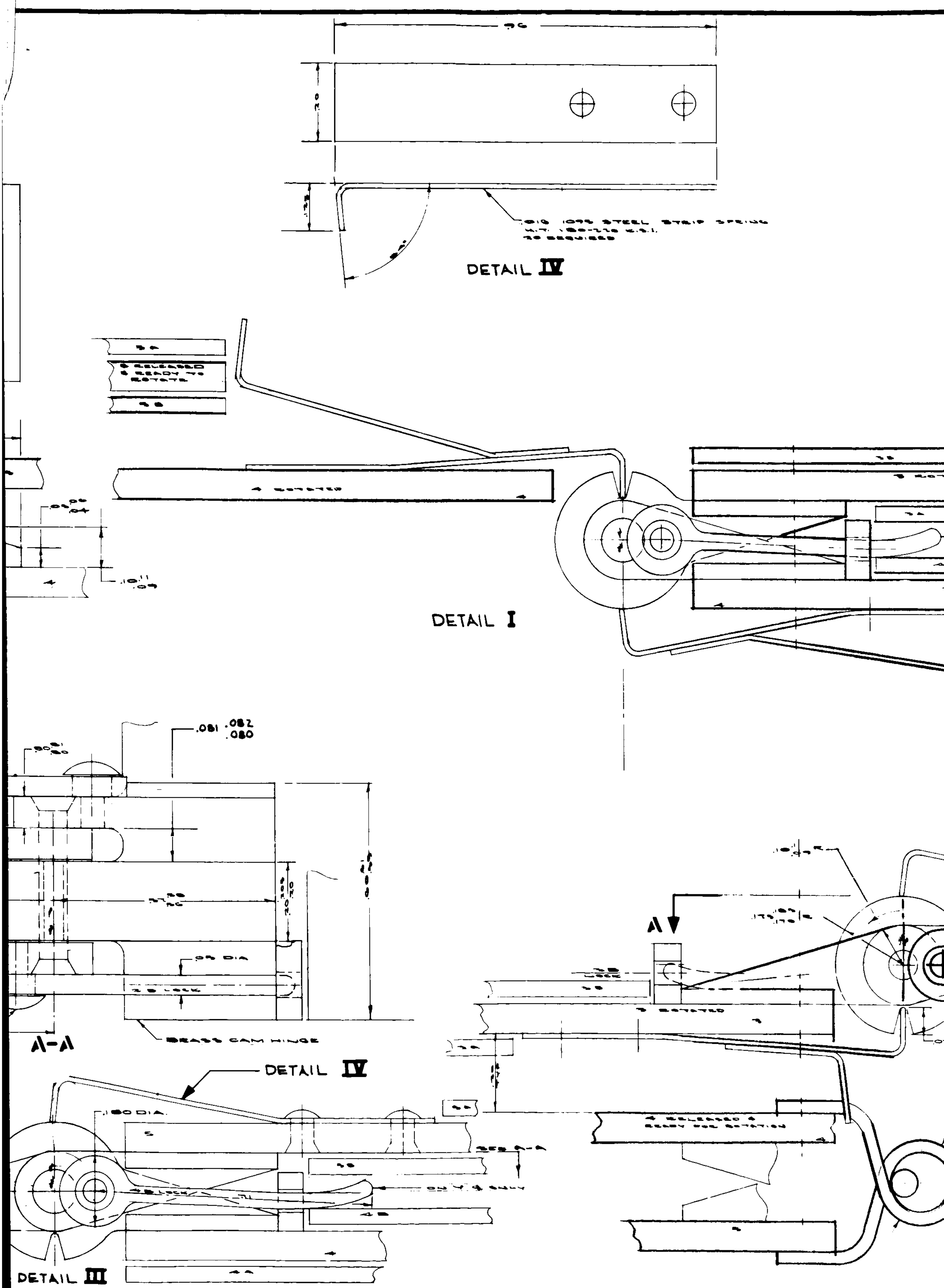

DETAIL IV
DETAIL I
.081 .082
.080
A
A-A
DETAIL IV
.180 DIA.
DETAIL III

Fig 8.5-3
(2)

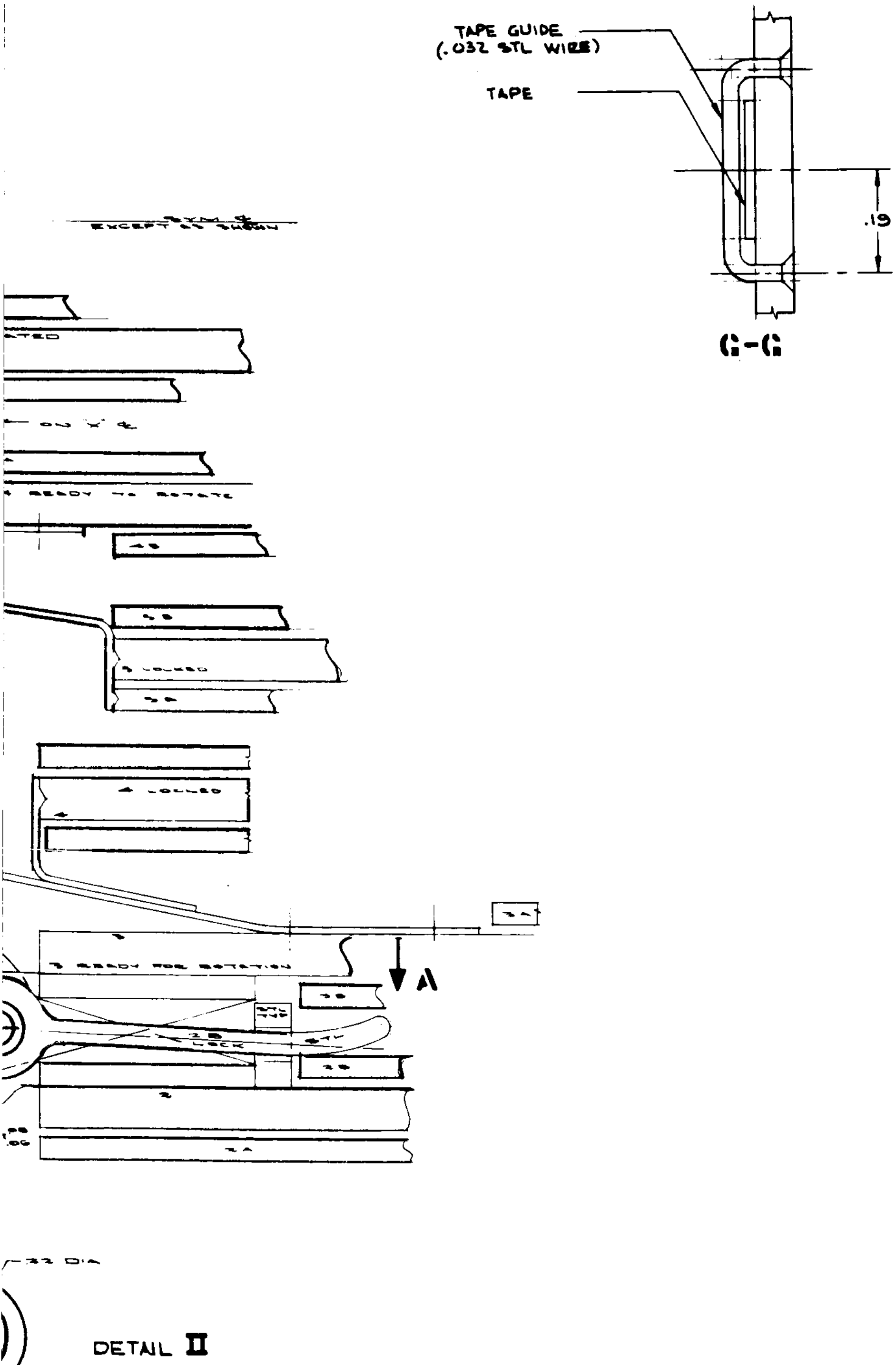


FIGURE 8.2-3 SOLAR ARRAY
1/20th SIZE MODEL
SATURN IB/CENTAUR

3

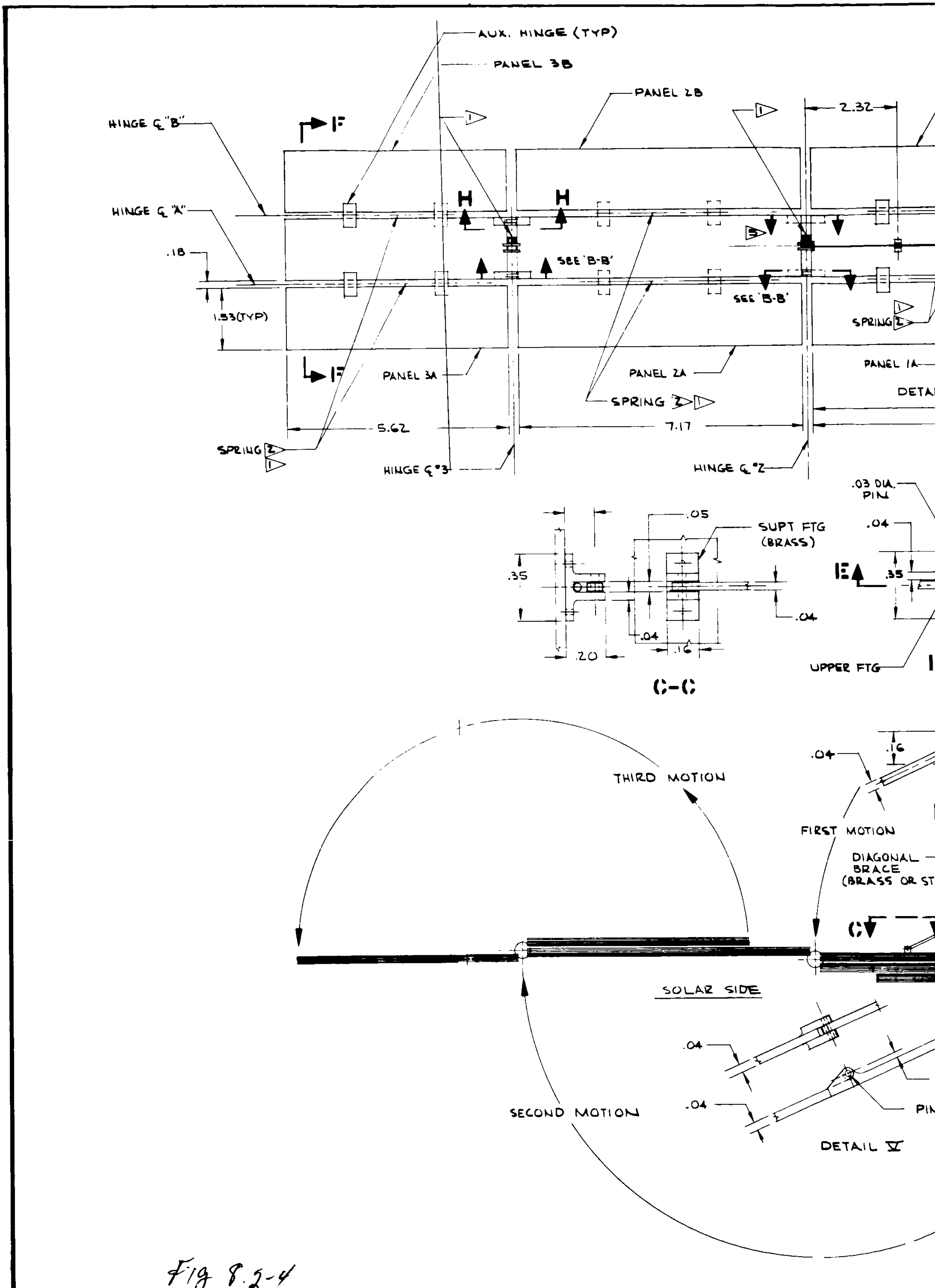

AUX. HINGE (TYP)
PANEL 3B
PANEL 2B
HINGE ℄ "B"
HINGE ℄ "A"
2.32
F
H
H
SEE 'B-B'
SEE 'B-B'
.18
1.53(TYP)
SPRING 2
PANEL 3A
PANEL 2A
PANEL 1A
SPRING 2 1
5.62
7.17
SPRING 2 1
HINGE ℄ "3"
HINGE ℄ "2"
DETAI
.03 DIA. PIN
.05
SUPT FTG (BRASS)
.04
.35
.35
.04
.04
.16
.20
E
UPPER FTG
C-C
.16
.04
THIRD MOTION
FIRST MOTION
DIAGONAL BRACE (BRASS OR ST
C
SOLAR SIDE
.04
.04
PIN
SECOND MOTION
DETAIL V

Fig 8.2-4

①

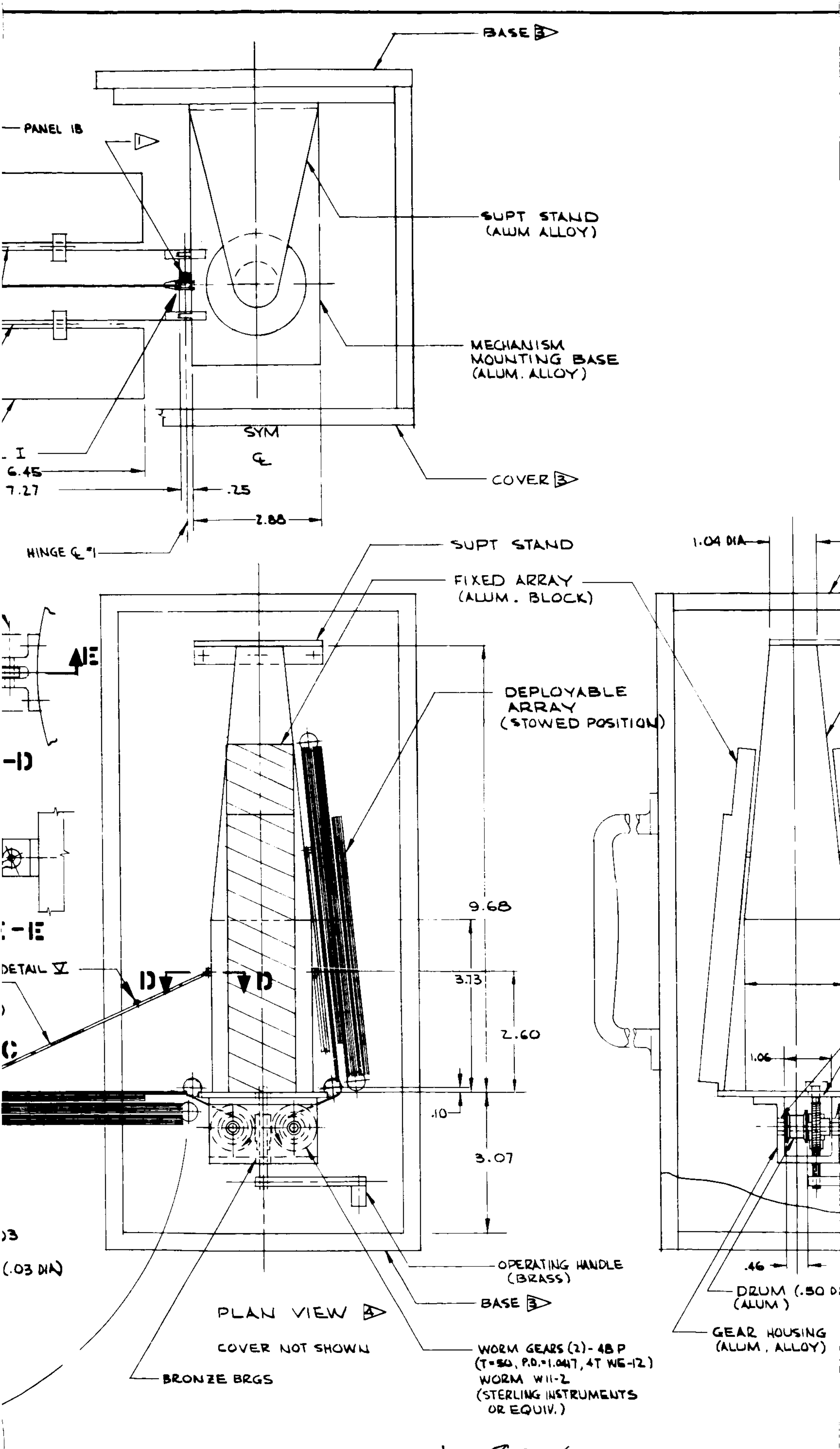
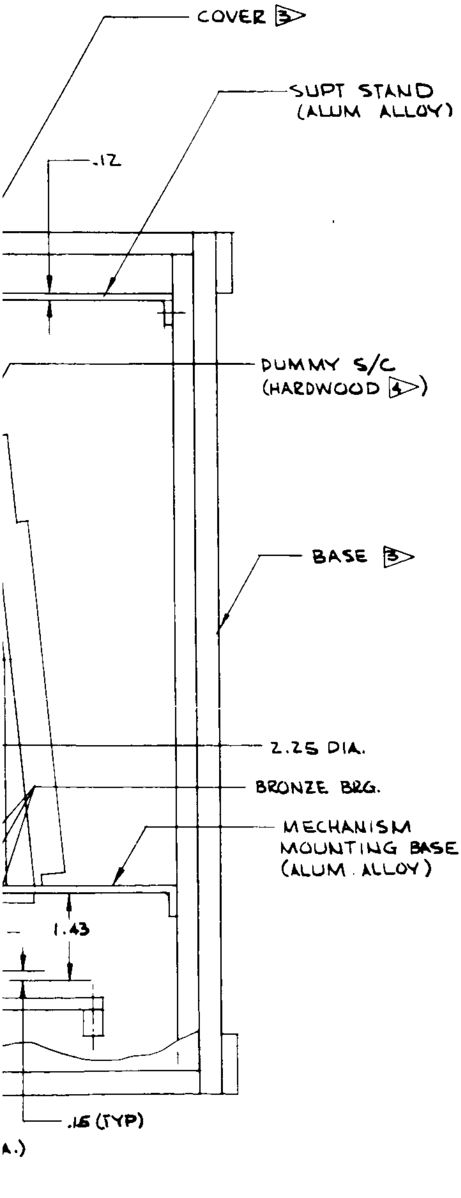

BASE 3
PANEL 1B
1
SUPT STAND (ALUM ALLOY)
MECHANISM MOUNTING BASE (ALUM. ALLOY)
SYM ℄
6.45
7.27
.25
2.88
COVER 3
HINGE ℄ #1
SUPT STAND
FIXED ARRAY (ALUM. BLOCK)
DEPLOYABLE ARRAY (STOWED POSITION)
1.04 DIA
E
-D
-E
DETAIL V
D
D
C
9.68
3.73
2.60
.10
3.07
1.06
.03
(.03 DIA)
OPERATING HANDLE (BRASS)
BASE 3
PLAN VIEW 4
COVER NOT SHOWN
BRONZE BRGS
WORM GEARS (2) - 48 P (T=50, P.D.=1.0417, 4T WG-12)
WORM W11-2
(STERLING INSTRUMENTS OR EQUIV.)
.46
DRUM (.50 D
(ALUM)
GEAR HOUSING (ALUM. ALLOY)

Fig 8.2-4



5 SAME AS "H-H" EXCEPT HOOK-LOCK IS OMITTED FROM FLAT SPRING

4 FINISH NOTE: ALL ALUM. SURFACES WILL BE BLUE ANODIZED EXCEPT THE BACK SIDE WILL BE BLACK ANODIZED. THE SPACECRAFT DUMMY WILL BE PAINTED BLACK.
THE WORKING SURFACES OF THE MECHANISM WILL REMAIN UNFINISHED.

3 THE BASE & COVER WILL BE MADE FROM BIRCH OR MAPLE. STAIN LIGHT TAN & APPLY SATIN FINISH VARNISH. USE A BAC-H10E-6 HANDLE ON THE COVER & 4 CORBIN CABINET LOCKS (BAC-C12D) OR EQUIV. TO ATTACH THE BASE TO THE COVER.

2 IF THE TORSION SPRINGS SELECTED IN 1 DEPLOY THE AUX. PANELS 'A' IN LESS THAN 2 SECS. THE SPRING TORQUES WILL BE ADJUSTED UNTIL THE DEPLOYMENT TIME IS GREATER THAN 2 SECS.

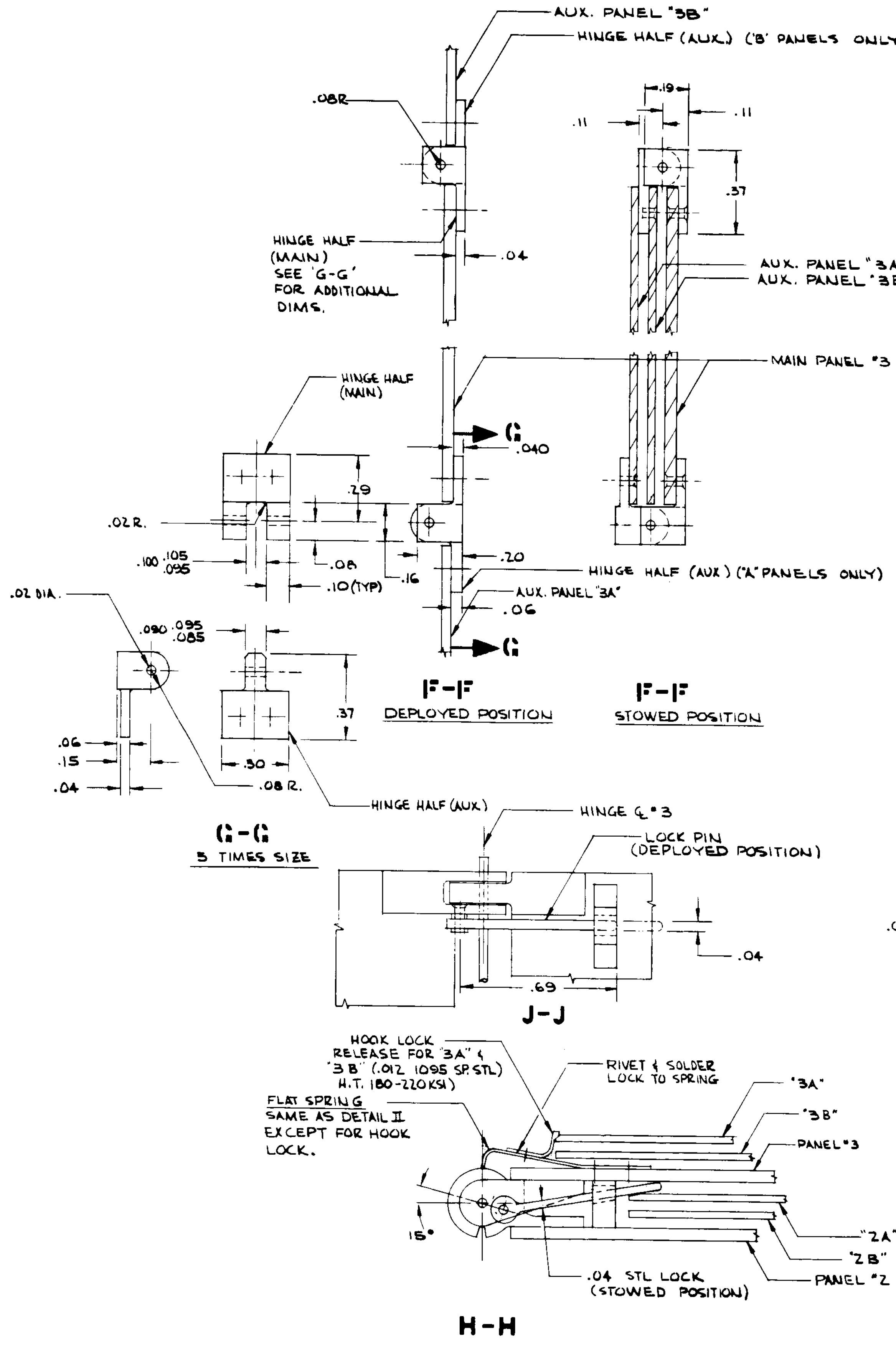
1 SPRING STL MUSIC WIRE TORSION SPRINGS WILL BE USED AS FOLLOWS:

LOCATION	MEAN DIA.	WIRE DIA.	WINDUP	NO. OF COILS
HINGE ℄ #1	.208	.0128	180°	10.4
" ℄ #2	.208	.0162	90°	12.6
" ℄ #3	.208	.0204	90°	16.0
" ℄ "A"	.208	.0204	270°	42
" ℄ "B"	.208	.0128	270°	42

THE ABOVE SPRINGS MAY BE ADJUSTED AS REQ'D UPON COORDINATION WITH ENGRG.

FIGURE 8.2-4 SOLAR ARRAY
1/20th SIZE MODEL
ATLAS/CENTAUR

3


AUX. PANEL "3B"
HINGE HALF (AUX.) ('B' PANELS ONLY
.08R
.19
.11
.11
.37
HINGE HALF (MAIN) SEE 'G-G' FOR ADDITIONAL DIMS.
.04
AUX. PANEL "3A
AUX. PANEL "3B
MAIN PANEL "3
HINGE HALF (MAIN)
G
.040
.29
.02 R.
.100 .105 .095
.08
.16
.10 (TYP)
.20
HINGE HALF (AUX) ("A" PANELS ONLY)
AUX. PANEL "3A"
.06
G
.02 DIA.
.090 .095 .085
.37
F-F
DEPLOYED POSITION
F-F
STOWED POSITION
.06
.15
.04
.30
.08 R.
HINGE HALF (AUX.)
HINGE ℄ "3
LOCK PIN (DEPLOYED POSITION)
G-G
5 TIMES SIZE
.04
.69
J-J
HOOK LOCK RELEASE FOR "3A" & "3B" (.012 1095 SP. STL) H.T. 180-220 KSI)
RIVET & SOLDER LOCK TO SPRING
FLAT SPRING SAME AS DETAIL II EXCEPT FOR HOOK LOCK.
"3A"
"3B"
PANEL "3
15°
"2A"
"2B"
PANEL "2
.04 STL LOCK (STOWED POSITION)
H-H

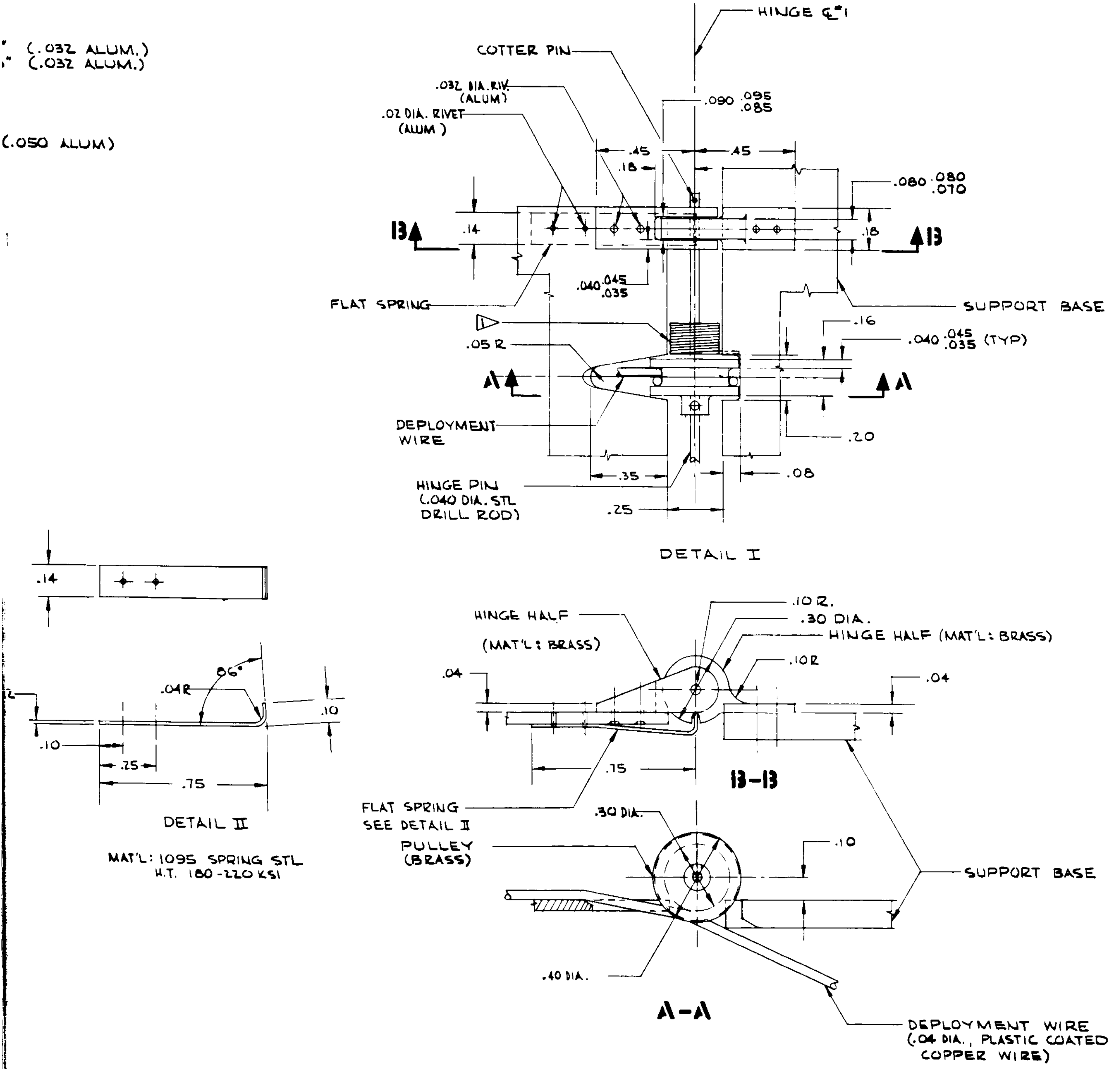


FIGURE 8.2-5 SOLAR ARRAY 1/20 SIZE MODEL ATLAS/CENTAUR