

FINAL REPORT

SERT II SOLAR ARRAY

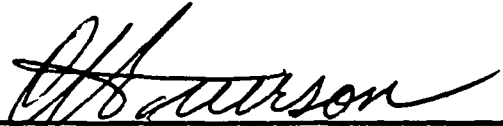
CONTRACT NAS 3-11512

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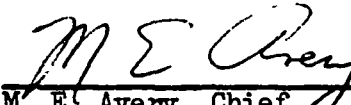
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F O R E W O R D

This document constitutes the Lockheed Missiles and Space Company final report on the SERT II Solar Array Contract. It is submitted to NASA, Lewis Research Center, in compliance with Paragraph 8.5, Exhibit A, Contract NAS 3-11512, Statement of Work, dated 30 November 1968.

SUMMARY

The SERT II Solar Array design was derived from a basic solar array module design which was successfully flown on several military missions. The basic design was modified only as necessary to satisfy the requirements of the SERT II Mission.

Modifications were made to the deployment mechanism and the release mechanism system, 2 ohm-cm solar cells were used instead of the 10 ohm-cm cells used for the military program. In addition, changes were made in the solar array instrumentation and in electrical harnesses which provide power from the solar collector assemblies to the Spacecraft Support Unit.

Changes to the deployment mechanism design were required because the basic design provided for the arrays to deploy in a drooped and canted position. The SERT II program requires the array to deploy in a plane perpendicular to the vehicle Z-axis. A stronger actuator mechanism spring was also added.

The release mechanism was redesigned to provide assurance that the arrays would deploy under failure conditions where either pin puller was actuated with a single squib. Redundancy in the squib firing switch circuitry was also provided.

The 10 ohm-cm solar cells used for the basic design were replaced with 2 ohm-cm cells to provide for the higher operating voltage for the SERT II Mission. The solar collector assemblies were wired to provide two power outputs. Seventy-six panels were wired to furnish 1550 watts minimum at 68 VDC to the SSU Thrustor section, and 14 panels were wired to provide 286 watts minimum power at 34 VDC for the SSU Housekeeping section.

The module level test program was limited to performance of only those tests required to demonstrate that the flight worthiness of the original

solar array module design had not been invalidated by the modifications required by the SERT II Program, and that the SERT II solar arrays would withstand the vibration levels of a THORAD launch. The following tests were conducted:

- Deployment Mechanism Engineering Evaluation Tests
- Release Mechanism Evaluation Tests
- Vibration Confidence Tests
- Flight Acceptance Tests

A thorough review of the designs, hardware, test programs, test data, engineering analysis, and product assurance assessments of the SERT II Solar Array has been completed. LMSC concludes that the solar arrays are flight worthy and meet the requirements of the Statement of Work, Contract NAS 3-11512.

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SECTION 1.0

INTRODUCTION

1.0 PURPOSE

The final report presents the mission requirements, design and test philosophies, concepts considered in arriving at the final design, description of the final design, and the testing accomplished to prove flight worthiness of the SERT II Solar Array modules.

1.1 SCOPE

This report will include discussion of the following:

- Design requirements specified by the Work Statement to Contract NAS 3-11512.
- Design philosophy.
- Mechanical and electrical design concepts considered in arriving at the final design.
- Final design. This section will describe the mechanical design of the deployment mechanism, the release mechanism, the mounting bed assembly, and the solar array wing assembly. The description of the electrical design will include solar cell selection, array wiring, pyrotechnic release mechanism circuitry and instrumentation. Final design of the thermal controls will also be described in this section.
- Test program. This section will present the test philosophy and describe the tests conducted to demonstrate flight worthiness of the SERT II solar array final design.
- Conclusions.

SECTION 2.0

DESIGN REQUIREMENTS

The following requirements dictated the final design of the SERT II Solar Array system:

2.1 FLIGHT PROGRAM DESCRIPTION

The SERT II flight program will be a minimum 6-month continuously operated life test of a mercury bombardment ion thruster system. The spacecraft will be launched on a Thorad/Agena D from WTR into a 540 nautical mile circular orbit. To provide sun synchronism with continuous sunlight for a minimum of six (6) months, the spacecraft is scheduled for launch in March 1969, into an orbit inclined at 99.10 degrees.

After injection, the Agena orients itself nose down and then becomes an orbiting platform which furnishes power from the solar array to the spacecraft which remains attached to the Agena vehicle. Attitude control in three axes is furnished by gravity gradient forces in conjunction with control moment gyro torques. The Agena mounted solar array furnishes power for all housekeeping and experiment requirements.

The component of ion engine thrust tangent to the orbit causes the orbit altitude to increase or decrease by approximately 63 miles over the minimum 6-month duration, depending on which of two thrusters is operating. This altitude change and other orbit parameters are determined by the STADAN network. The STADAN network also provides command and data acquisition support.

The spacecraft structure contains the two ion thruster systems, a low-g accelerometer, an RF (radio frequency) noise experiment, a contamination experiment, two ion beam plasma probes, a spacecraft potential probe and a back-up acquisition cold gas control system.

The spacecraft is mounted to the Spacecraft Support Unit (SSU) structure which contains the telemetry system, command system, data storage system, attitude control system and the power system.

The SSU-Agena interface occurs at the forward end of the Standard Agena Clamshell Shroud (SACS) adapter ring.

All systems hardware for the solar arrays was developed to meet the SERT II mission objectives outlined below:

- Orbit

The SERT II orbit is to be a sun synchronous circular orbit inclined at 99.10 degrees. The orbit will vary approximately from 540 to 603 or from 540 to 478 nautical miles dependent on which thruster is operating during a minimum 6-month period due to a tangential component of thrust produced by one of the ion thrusters.

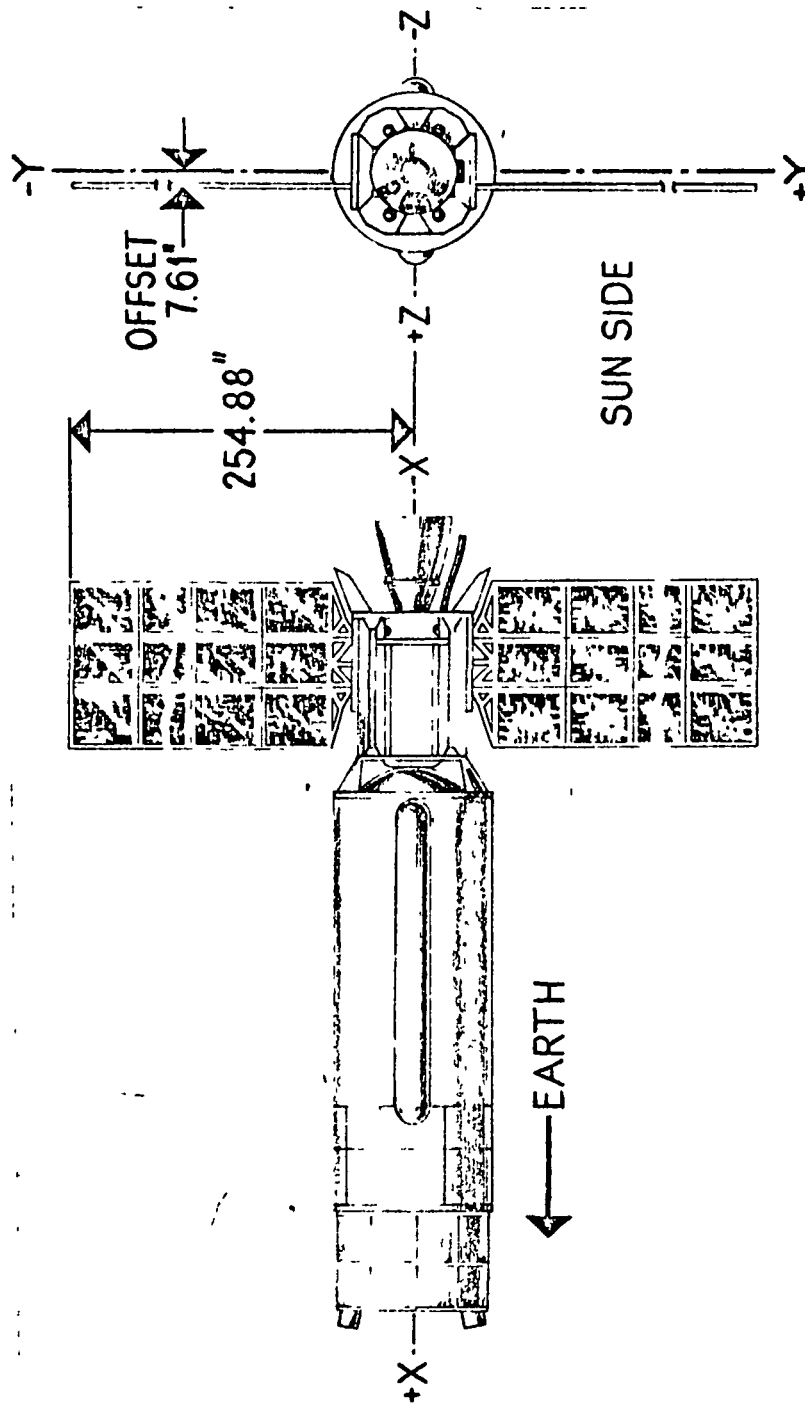
The orbit perpendicular precesses at a rate such that the orbit is in constant sunlight for a minimum of six months. During this period, the angle between the orbit perpendicular and the earth-sunline varies between 0 and 16 degrees.

Following mission completion, the angular deviation of the orbit perpendicular from sun alignment increases steadily until each orbit is partially eclipsed during a 2-month period. Operation of the spacecraft and its experiments will be continued for as long as possible. During the 2-month eclipse interval certain housekeeping functions will remain active and other subsystems will be activated for brief periods to check spacecraft conditions.

Following the period of partial solar eclipse, the spacecraft with its experiments will be reactivated and ion thruster operation will be resumed for as long as possible.

- Satellite Orientation

The on-orbit orientation of the satellite is with the spacecraft end of the Agena pointed earthward, with the deployable solar array extended and oriented in the plane of the orbit and the Agena +Z axis facing the sun. See Figure 2-1.



SERT II VEHICLE ORIENTATION

FIGURE 2-1

2.2 STRUCTURE

The flight solar array shall consist of two deployable solar platforms, associated latching and deployment mechanisms, solar panels, solar cell modules and sub-modules with solar cells and support structure, complete in all mechanical, structural and electrical aspects.

2.3 LOCATION AND MOUNTING

The solar array module shall be constructed so that in the stowed position it is located within the rectangular truss work aft of the Agena tank section and within the envelope of the Agena Booster adapter structure, and symmetrical about the vehicle longitudinal axis.

2.4 RELEASE MECHANISM

The release mechanism shall be designed such that the mechanism will function as designed when either pin puller is actuated by a single squib. Pin puller squibs shall be actuated by redundant timer switch circuits.

2.5 DEPLOYMENT

The solar array shall be designed to deploy in such a manner that the plane formed by the extended array is parallel to the vehicle Y/X plane. The long axis of each wing is normal to the longitudinal axis of the vehicle. Each wing shall lie in the same plane. The wing planes need not pass through the longitudinal axis although this is preferred, but in any case the plane of the array shall be within 8.00 inches of the Y/X plane. The solar array shall be capable of simulated zero-G deployment in a one G field when properly supported by an extension and testing jig.

2.6 ORIENTATION

Array orientation shall be non-heliotropic, but so oriented as to maximize the solar input in the presence of acceptable deviations induced by the gravity gradient orientation of the spacecraft.

2.7 WEIGHT

A detailed weight record of the solar array shall be effected and maintained current throughout the program. The weight of the flight solar arrays complete with latching and deployment mechanisms shall not exceed 445 pounds.

2.8 POWER OUTPUT

The solar array shall be functionally separated into two sections which provide power to the SSU and spacecraft in the following manner:

- Thrustor section - The thrustor section of the array shall provide power to the power conditioners in the spacecraft for ion engine operation. At air mass zero and 25°C the thrustor section shall deliver 1550 watts minimum at 68 VDC, at maximum power (plus 34 VDC and minus 34 VDC referenced to array center tap). The output wiring shall be a three line system comprised of twisted, shielded cable sized not to exceed 0.60 VDC drop from the Agena to the SSU interface at maximum power point. The shields shall be insulated and not grounded but carried through the interface.
- Housekeeping section - The housekeeping section of the array shall supply power to the housekeeping equipment in the SSU and spacecraft. The housekeeping section shall deliver 286 Watts minimum at 34 VDC at maximum power. The output wiring shall be a two line system comprised of twisted, shielded cable sized not to exceed 0.30 VDC drop from the Agena to the SSU interface at maximum power point. The shield shall be insulated and not grounded, but carried through the interface.
- Open circuit voltage - Under conditions of normal operation, with the spacecraft in a constant sunlight orbit and with a sunline incidence angle of less than sixteen degrees from normal, the open circuit voltage (Voc) shall not exceed 75.0 Volts for the thrustor section and 35.0 Volts for the housekeeping section. These maximum voltages shall be inherent

characteristics of the array and shall not be achieved by limiters or regulators.

2.9 INSTRUMENTATION

Instrumentation for the solar array system consists of two basic groups, one group for instrumenting the events associated with deployment and the second group for instrumenting the temperature and power characteristics of the array assembly.

- Group I

A deployment monitoring switch shall be installed in such a manner as to indicate the solar array arm has deployed in excess of 95% of the full open position.

- Group II

The following instrumentation shall be provided to monitor the temperature and power characteristics of the solar array:

- (1) Temperature Sensors

IMSC shall provide and install a total of six (6) thermistor sensors to monitor representative temperatures on the solar array.

- (2) Control Cell - Short Circuit Current

IMSC shall furnish one separate solar cell to monitor cell short circuit current. The short circuit control cell shall be shorted through a one ohm, $\pm 1\%$ precision resistor, located as close to the control cell as possible.

- (3) Pitch Rate Cell -

IMSC shall furnish one separate solar cell which shall be wired in parallel with the short circuit current control cell prior to temperature calibration of the control cell test panel. The pitch rate cell shall be mounted in an unobstructed space on the back surface of the Control Cell Test Panel, but shall not be mounted directly opposite the control cells.

(4) Control Cell - Open Circuit Voltage

IMSC shall furnish one separate solar cell connected to a 2.5K ohm, $\pm 1\%$ precision resistor to monitor solar cell open circuit voltage.

(5) Control Cell - Maximum Power

IMSC shall furnish one separate solar cell connected to a precision resistor sized for maximum power at AMO with the sunline perpendicular to the active face.

(6) Control Cell Mounting and Calibration -

The control cells and pitch rate cell described above shall be mounted on a single control cell test panel. One of the 6 thermistor sensors specified above shall be mounted on the back surface of the control cell test panel. The panel shall then be calibrated for voltage open circuit, current short circuit, and current maximum power versus 40°C, 50°C, 60°C and 70°C temperature conditions.

2.10 MAGNETIC DISTURBANCE TORQUES

The solar array shall be designed, constructed, and wired in a manner so as to minimize the torque resulting from any current loops or magnetic fields produced from electrical loading and operation for any position in orbit.

2.11 THERMAL CONTROL

The satellite orbit precesses to provide the spacecraft with approximately six months of continuous sunlight followed by periods of darkness of 18 minutes or less per orbit for a period of two months. The thermal design will recognize these constraints and shall be accomplished by passive means.

NOTE: LMSC has been advised that the launch of the SERT II mission has been rescheduled to September 1969. The effort required to adapt the solar array system to accommodate a fall launch is not included in this report. The work required to comply with Change Order Number 12 to Contract NAS 3-11512 will be described in a subsequent analysis report.

SECTION 3.0

DESIGN PHILOSOPHY

From the initial stages of proposal preparation the design philosophy has been to use, as much as possible, the existing flight-qualified Solar Array design. This was to preserve the flight qualification status of the array module thereby greatly reducing the required testing effort, modification to existing tooling and AGE equipment.

To the maximum extent possible, existing hardware from previous flight programs was to be utilized. Bearings, extension mechanisms, dampers, mounting brackets, clamps, and other supporting hardware were to be manufactured and assembled such that the solar array is suitably supported on the vehicle, to prevent damage which might impair the flight worthiness of the array, during ground transportation, launch and injection.

The existing flight qualified solar array module from which the SERT II solar array module design was derived consists of three module components: (1) an arm assembly capable of mounting existing designed solar collector panels (approximately 14 by 18 inches per panel); (2) an erection or deployment mechanism for extending and positioning the arm assembly at a controlled rate; and (3) a support structure for supporting and retaining the folded arm assembly and erection mechanism during ascent and for supporting the extended arm assembly while in orbit.

The existing solar array module was designed and qualified for an Atlas boost launch environment and an on-orbit deployed position with a 35-degree droop toward the center of gravity of the vehicle and a 40-degree twist with respect to the array longitudinal axis. See Figure 3-1.

For this design, a simple slider crank device was used which rotated under the action of a single liquid-damped spring actuator, thus extending and angularly positioning the array in a single, continuous cycle (Figure 3-2). This mechanism consists of a frame, an adjustable liquid-damped spring actuator, a slider, and a support assembly. The inboard leaf is rotated about a fixed pivot on the support assembly with

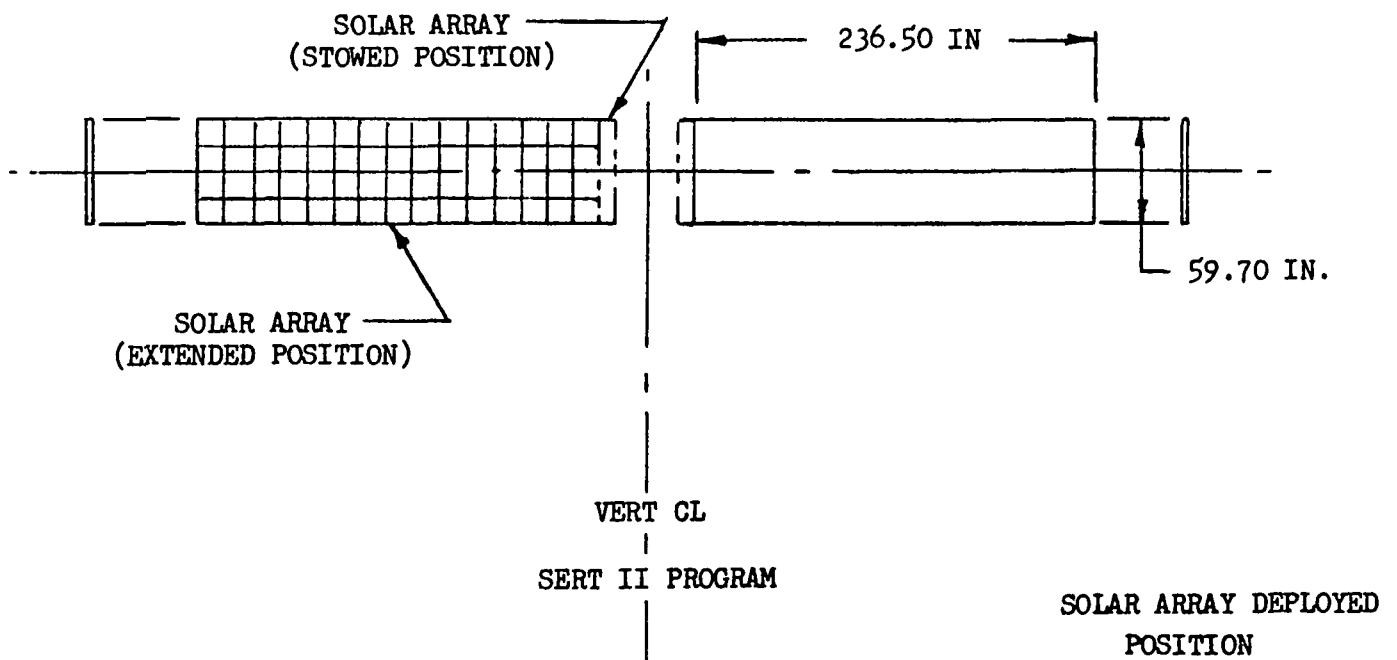
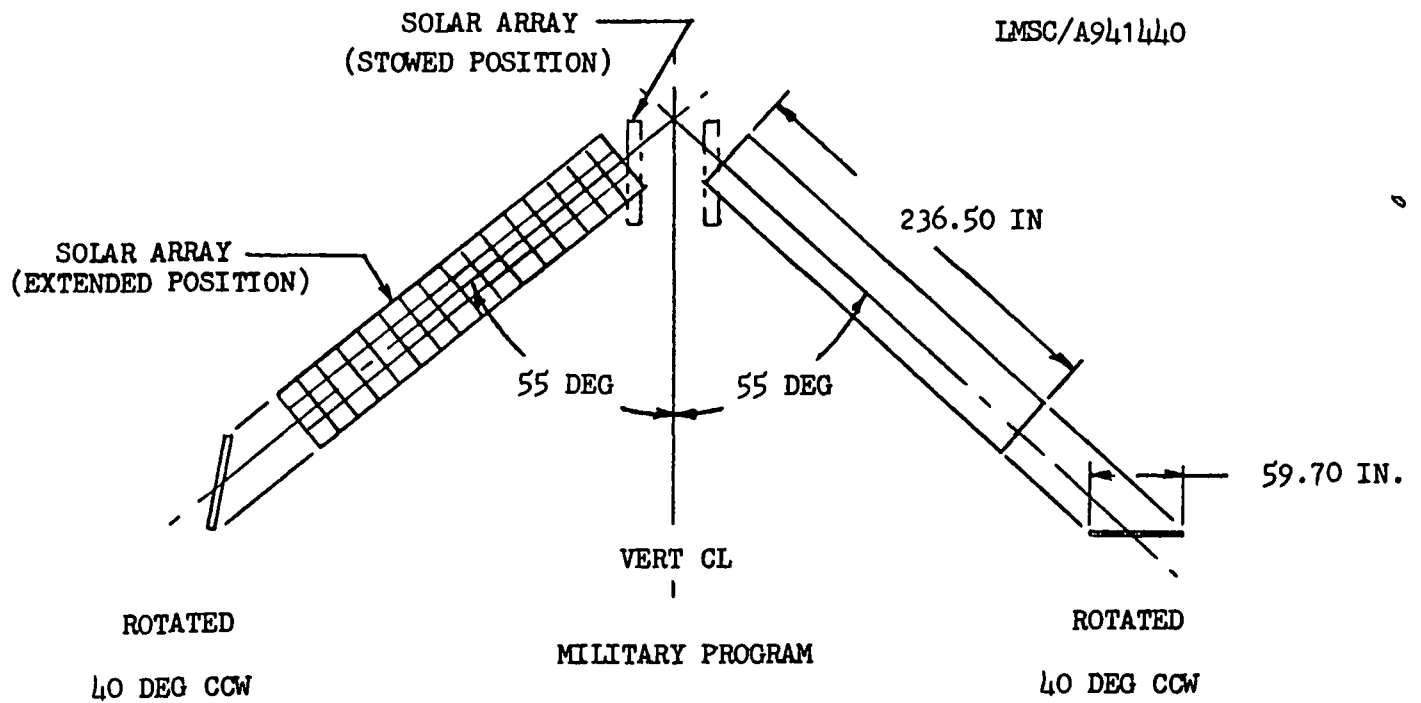
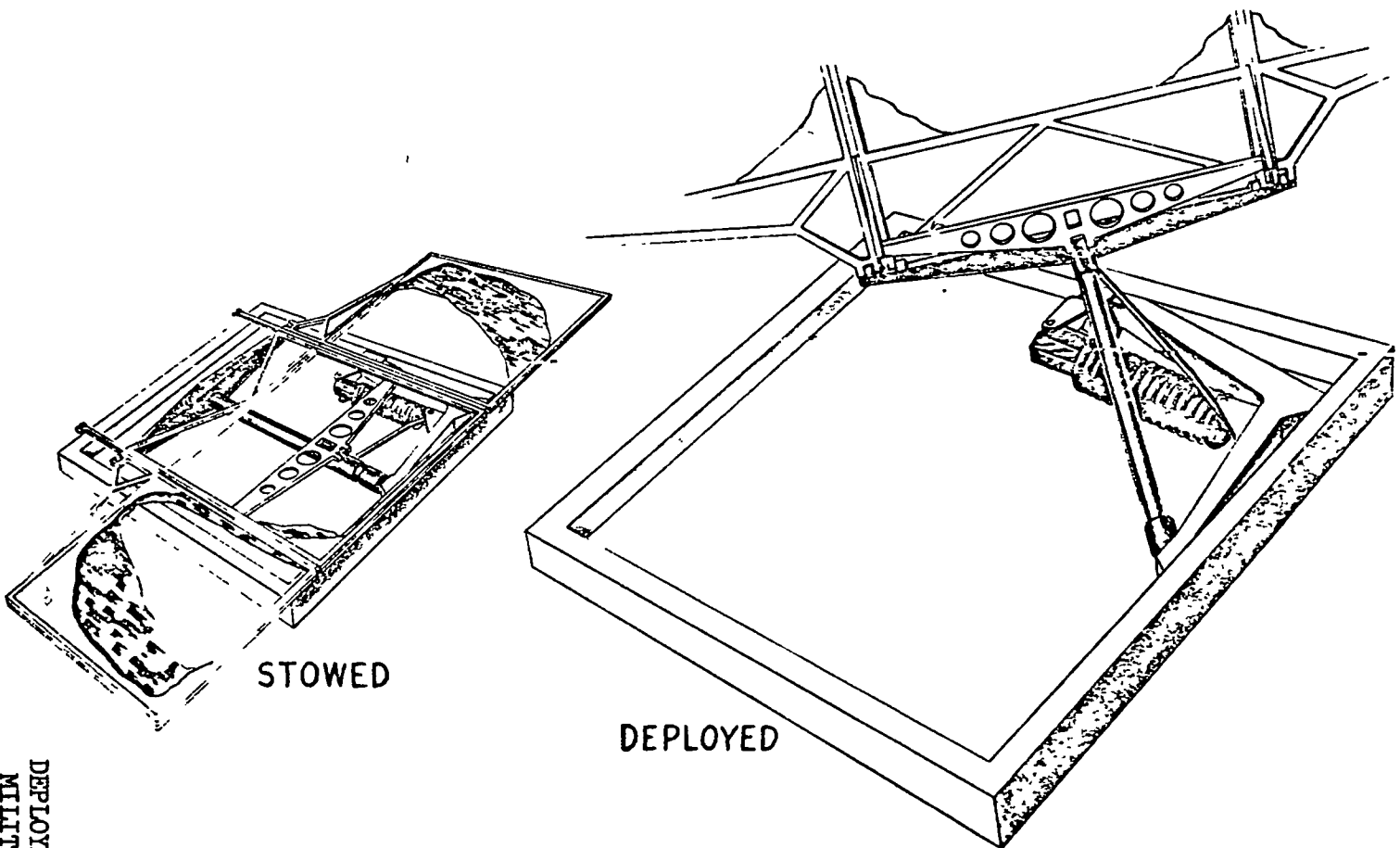


FIGURE 3-1



DEPLOYMENT MECHANISM
MILITARY PROGRAM

FIGURE 3-2

the inriggers attached through a single fitting to the slider. During an extension, the actuator rotates the bellcrank which moves the slider along the support assembly tube until the full-extension position is attained. The actuator is used to control the extension rate and to complement the array arm forces when they are approaching the minimum output.

While both the launch environment and the on-orbit deployed position of the solar arrays for the SERT II Program differ from that of the Military Program, by analysis much of the design techniques, fabrication methods, flight and test results are directly applicable to this program.

SECTION 4.0

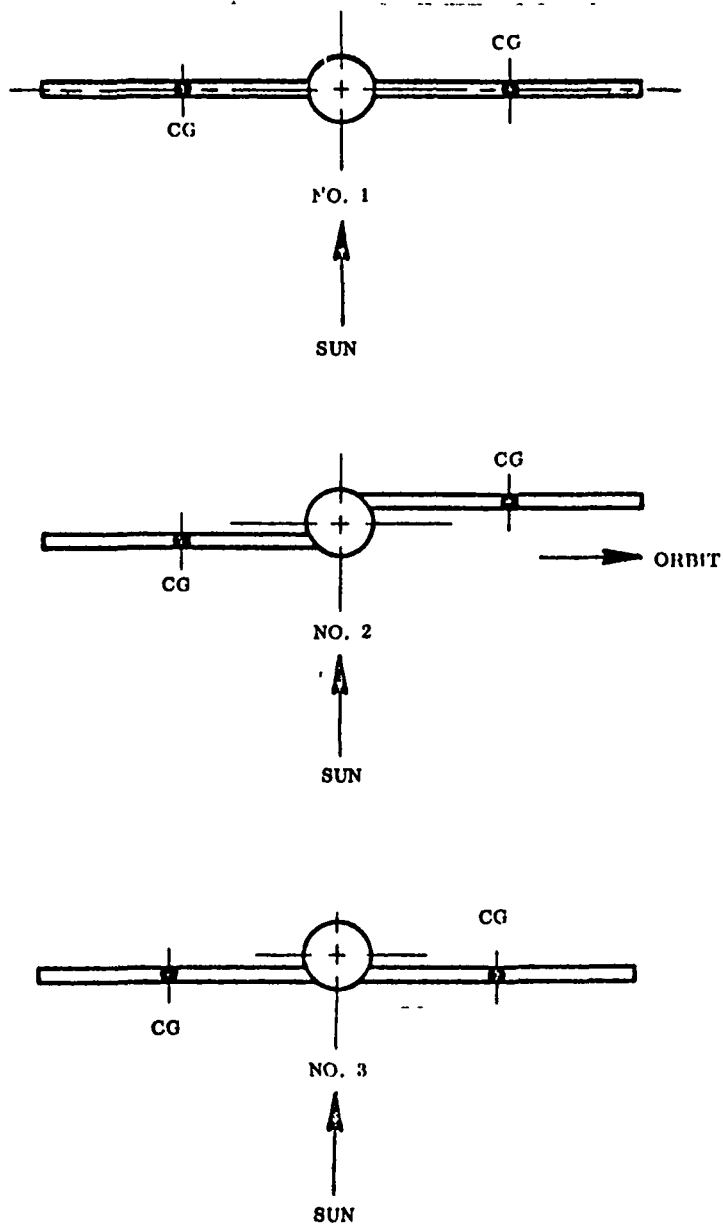
DESIGN CONCEPTS

4.1 MECHANICAL

4.1.1 Deployment Mechanism - The primary modification to the existing solar array module design flown on other Agena vehicles is a redesign of the deployment mechanism to extend the arm assembly straight out instead of the drooped and canted configuration shown in Figure 3-1. By extending the array arm straight out from the vehicle the on-orbit orientation requirements of the SERT II mission will be met. Figure 4-1 shows the three deployed solar array position options considered for this modification. Position Number 3 has been chosen for the SERT II mission based upon mission requirements and modification complexities.

Positions number 1 and 2 reflect array deployment configurations that effectively place the center of gravity of each arm on the vehicle centerline to enhance vehicle stability. To achieve position 1, however, requires considerable modification to the existing solar array deployment mechanism in that the actuator would be required to pivot in two planes and stabilization rods to be added to the inboard portion of the inriggers. Figure 4-2 illustrates this set up while Figure 4-3 describes the deployment geometry associated with this configuration. To achieve position number 2 would require less modification to the deployment mechanism, however, it would require a right and left hand array arm in order to have the collector assemblies facing in the same direction. This in turn would require a redesign of the array arm structure in addition to the redesign of the deployment mechanism.

Investigations indicate that orbit perturbations resulting from the center of gravity offset associated with position number 3 do not warrant the effort required to eliminate it. The deployment mechanism



SOLAR ARRAY POSITION

FIGURE 4-1

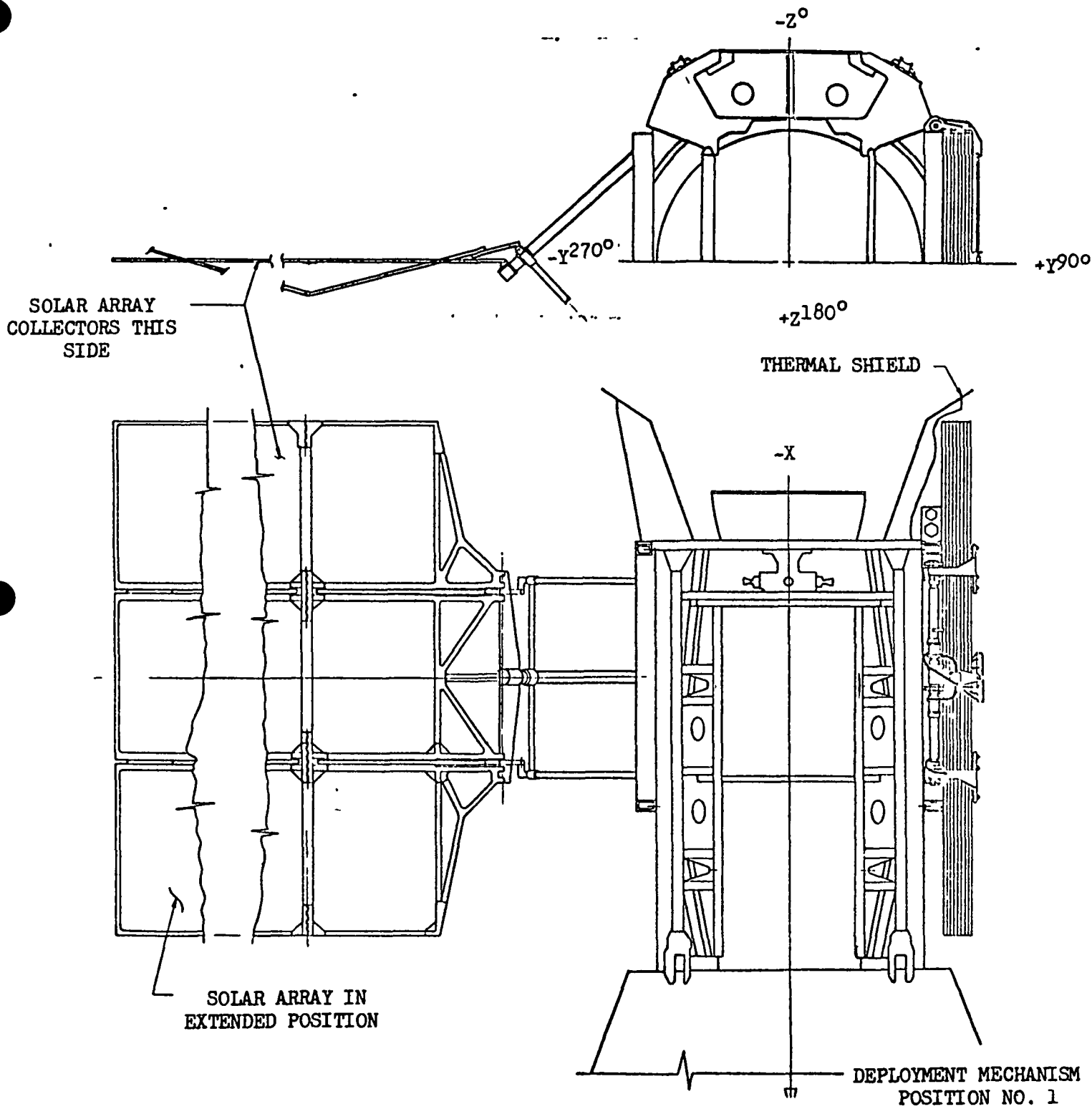
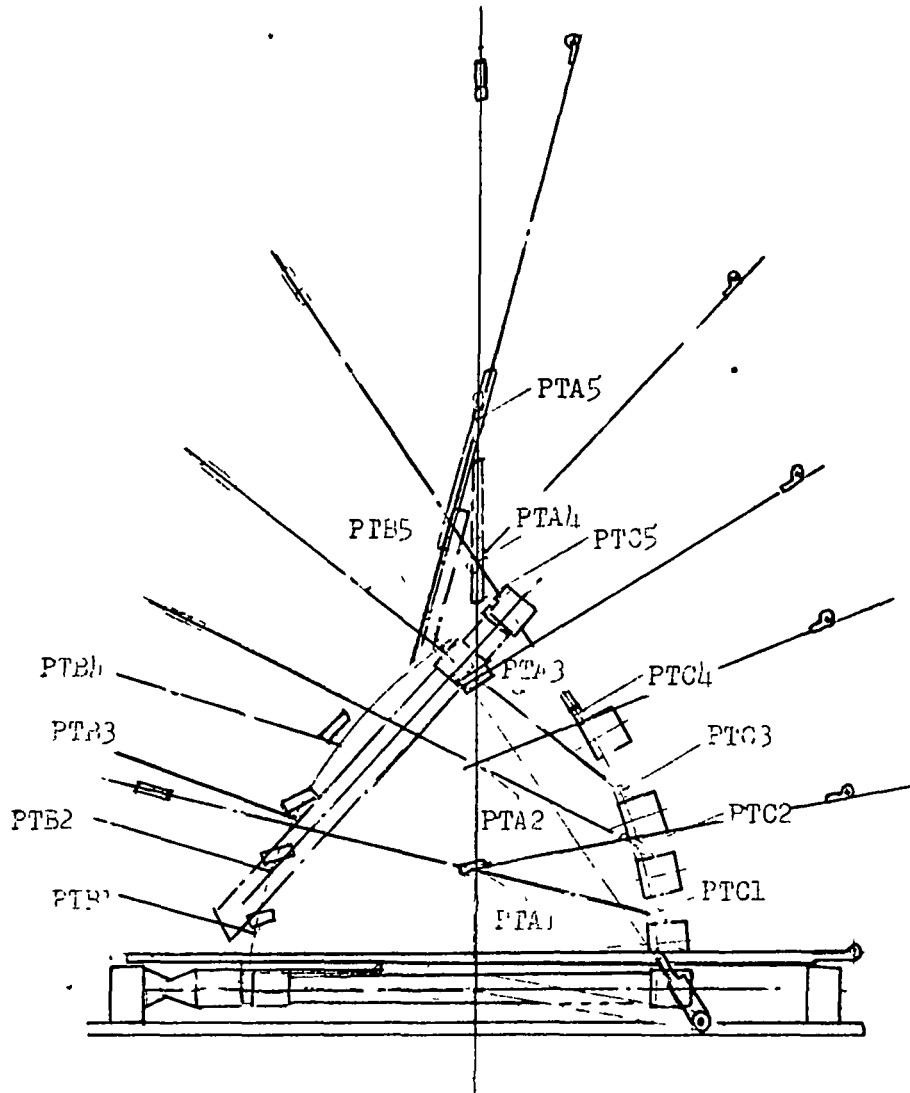


FIGURE 4-2



DEPLOYMENT GEOMETRY
POSITION NO. 1

FIGURE 4-3

design considered for achieving position number 3, the chosen design, is illustrated in Figure 4-4 and its deployment geometry in Figure 4-5. As it was desirable to have the same mechanical advantage in the actuator linkage geometry as that in the Military Program, a total of 16 separate mechanical advantage variations were examined. Additionally, three different methods of achieving the advantages were analyzed, i.e., long idler arm, short idler arm and slotted cradle. The slotted cradle concept was chosen as presenting the most economical means of achieving the desired conditions. This mechanism is further discussed in Paragraph 5.1.1.

4.1.2 Release Mechanism - LMSC initially proposed that the release mechanism basic design successfully flown on several military program vehicles be retained for the SERT II Program. During the SERT II Solar Array Conceptual Design Review, Mr. Conger of LeRC expressed concern over the reliability of the solar array release mechanism. This concern stemmed from the fact that 2 pin pullers were employed to effect release of the array and that failure of either pin puller to function would prevent the array from deploying.

LMSC was directed to initiate a design study to develop a release system which would allow the array to deploy normally with the actuation of 2 pin pullers, with the added feature that malfunction of either of these pin pullers would not prevent the array from deploying.

Although a conscientious attempt was made to preserve as much of the existing design as practicable, it was decided that this should not be imposed as a constraint which would limit freedom of thought in developing new concepts.

A description of the existing design and its potential failure mode is presented below. Also, preliminary design concepts and the recommended final design are discussed.

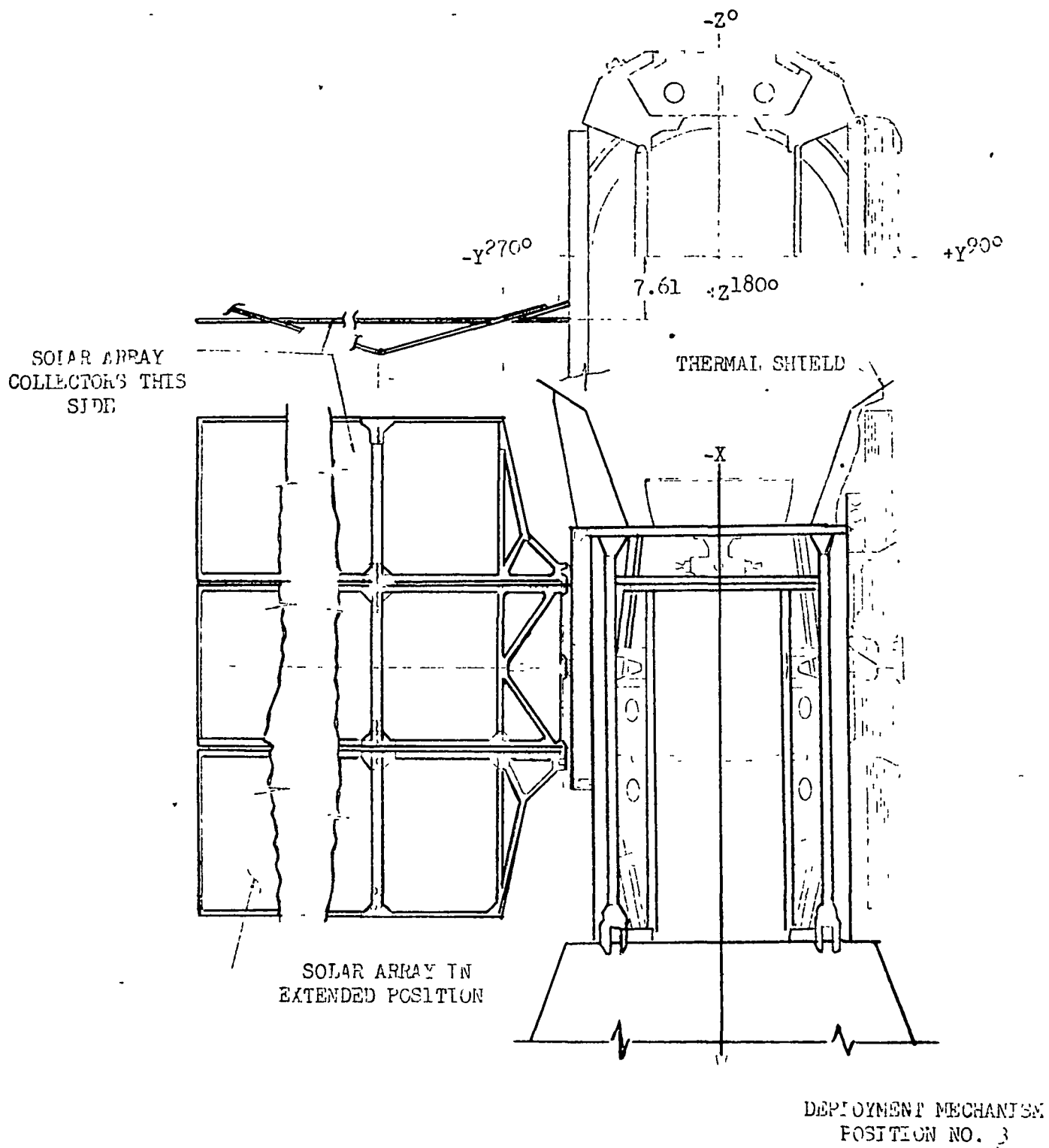
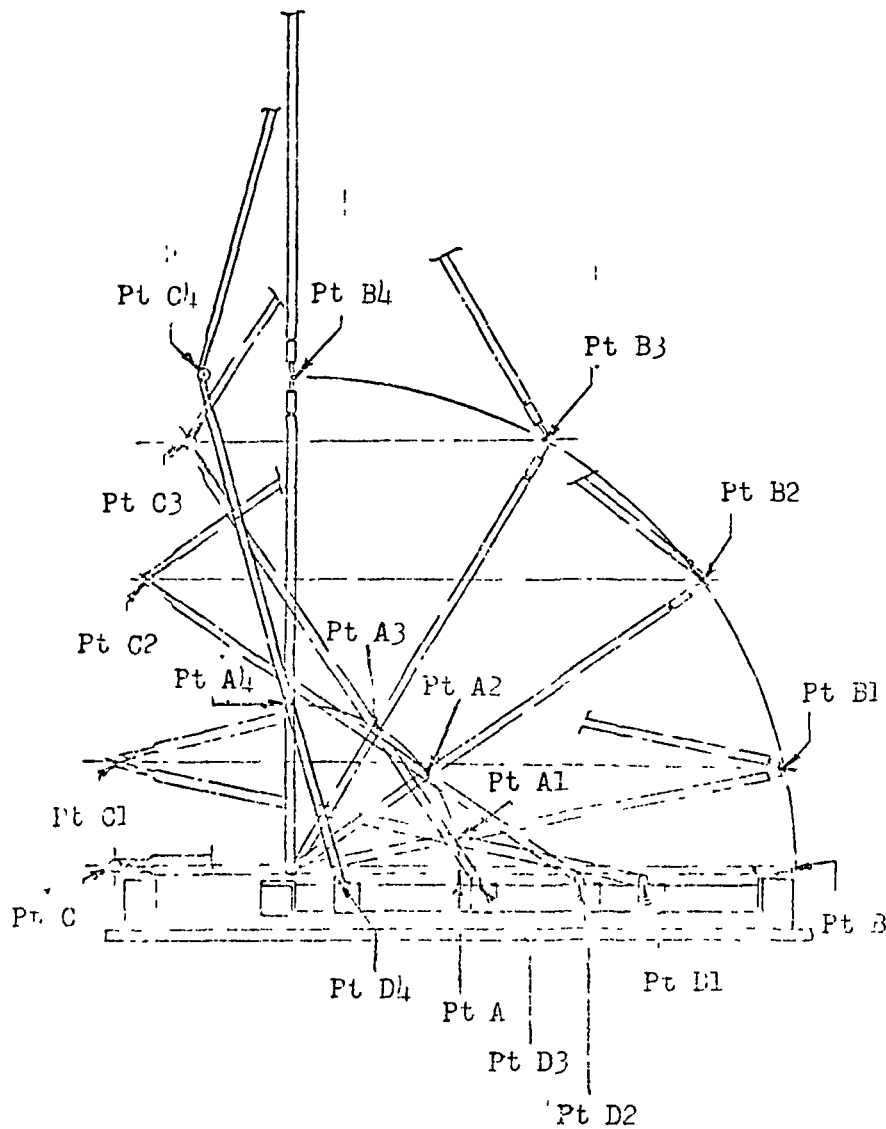


FIGURE 4-4



DEPLOYMENT GEOMETRY
POSITION NO. 3

FIGURE 4-5

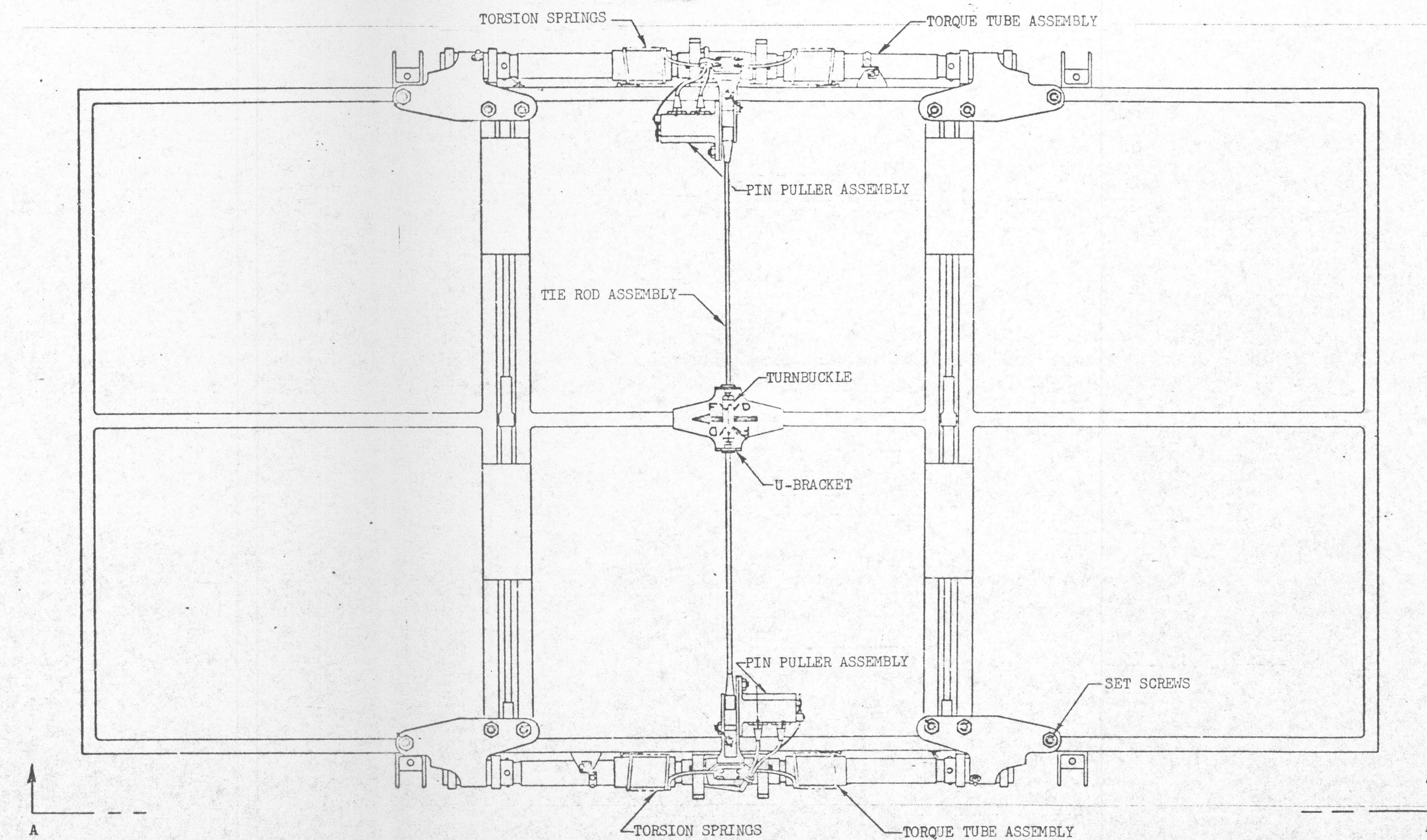
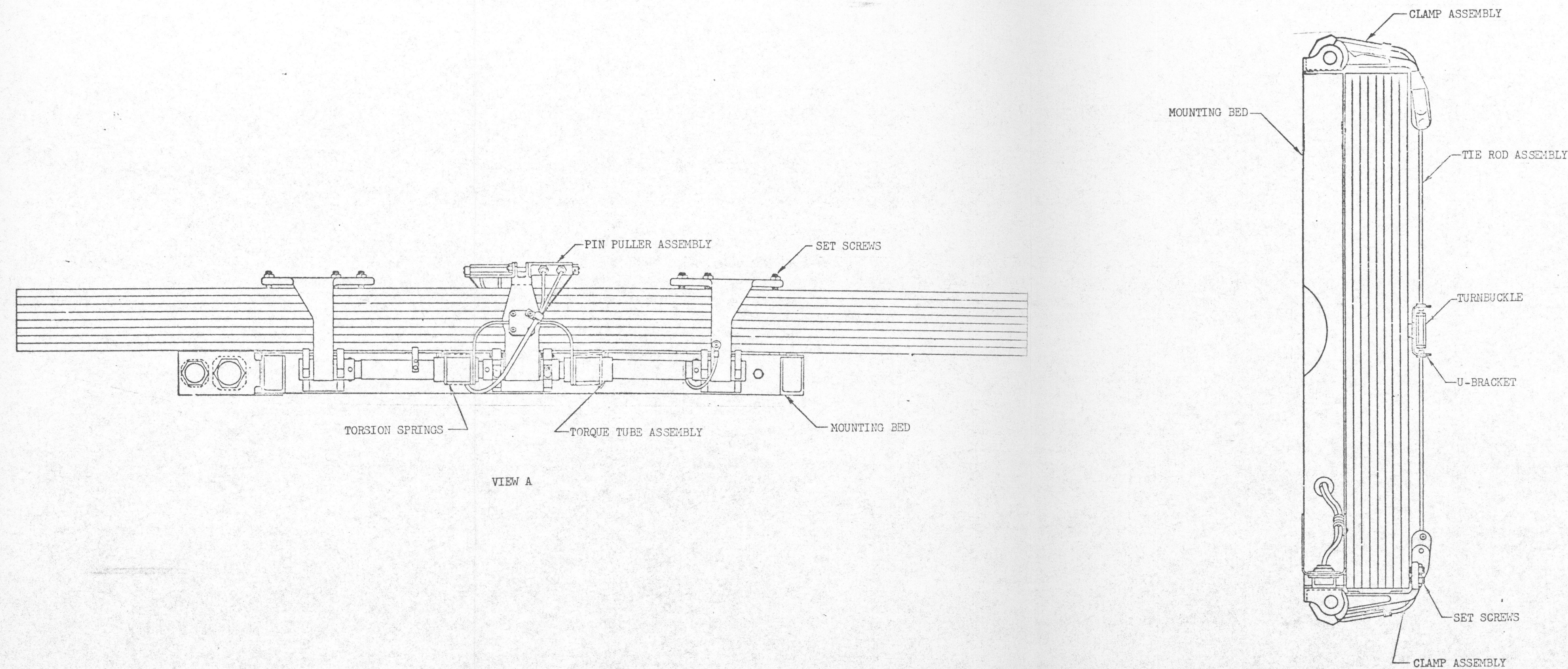
4.1.2.1 Existing Release Mechanism Design - The existing release system consists of 2 torque tube assemblies, 1 tie rod assembly, 2 pin puller assemblies and 2 torsion springs. Each torque tube assembly is comprised of three clamp assemblies which are joined into an integral unit with tube sections. The torque tube assemblies are mounted within journals on either side of the mounting bed and are free to rotate. These assemblies are pre-loaded to the released position by torsion springs.

The center clamp of each torque tube assembly is essentially a clevis with mounting provisions for 1 pin puller. The tie rod end fits within the clevis and is held by the extended piston of the pin puller. The opposing torque tube assemblies are drawn together by the tie rod assembly which employs a turnbuckle at its midpoint.

Design analysis and strain gage test data indicate that the tie rod (limit load) tension will be in the order of 450 to 500 lbs. This load is transmitted to the pin puller pistons in double shear. The tie rod passes through a U-shaped bracket which is fastened to the outer solar array panel and is retained by the bracket to prevent fly-away.

A total of 14 set screws contained in the clamp assemblies provide the force which restrains the folded array assembly against the mounting bed. Pin puller actuation releases the ends of the tie rod and the torsion springs cause the torque tubes to rotate away from the array, leaving the system free to deploy. In the event that either pin puller malfunctions, the associated torque tube cannot rotate to free the array due to the entrapment of the tie rod between the "U" bracket and the pin puller piston.

The existing release mechanism design from which the SERT II design was derived is shown in Figure 4-6.



RELEASE MECHANISM
EXISTING DESIGN

FIGURE 4-6
Page 4-9

4.1.2.2 Preliminary Design Concepts - The following discussion describes the various release mechanism design concepts which were considered:

- Concept "A"

Since the prime failure mode of the existing design involves entrapment of the tie rod within the "U" bracket, it would logically appear that deletion or modification of the bracket to allow release of the tie rod is the most expeditious solution. The shortcomings of this approach are:

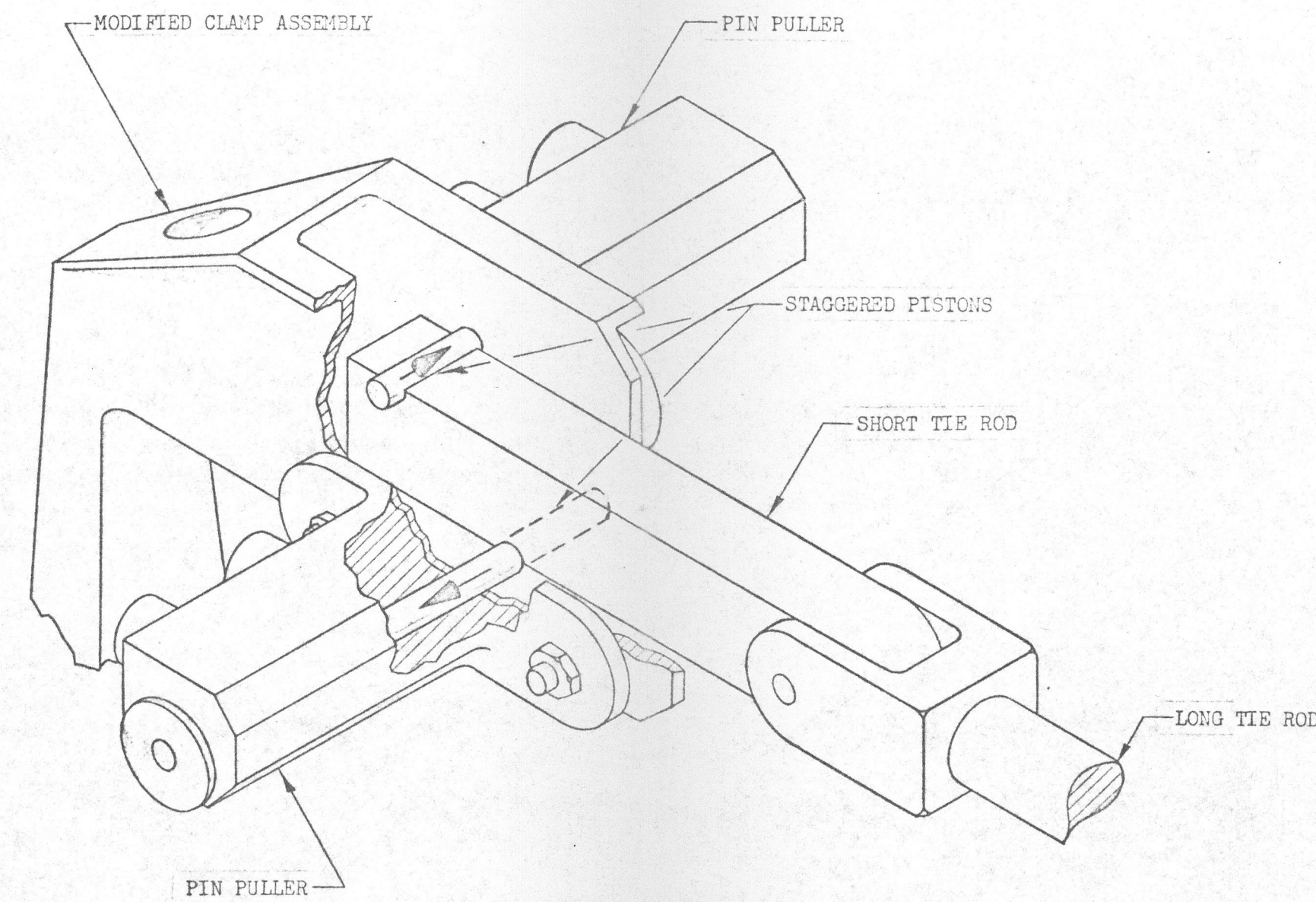
1. Normal functioning of both pin pullers would allow the tie rod to leave the array as a free orbiting body, offering the possibility that it could impact the vehicle or arrays at any time.
2. Failure of either pin puller would cause the rod to remain attached to the clamp assembly where it could swivel freely, increasing the probability that the rod may strike the array.

Various approaches were considered which would provide restraint of the tie rod in the failure mode. Each of these, however, tended to impose restraint on the tie rod during normal separation with a subsequent reduction of reliability.

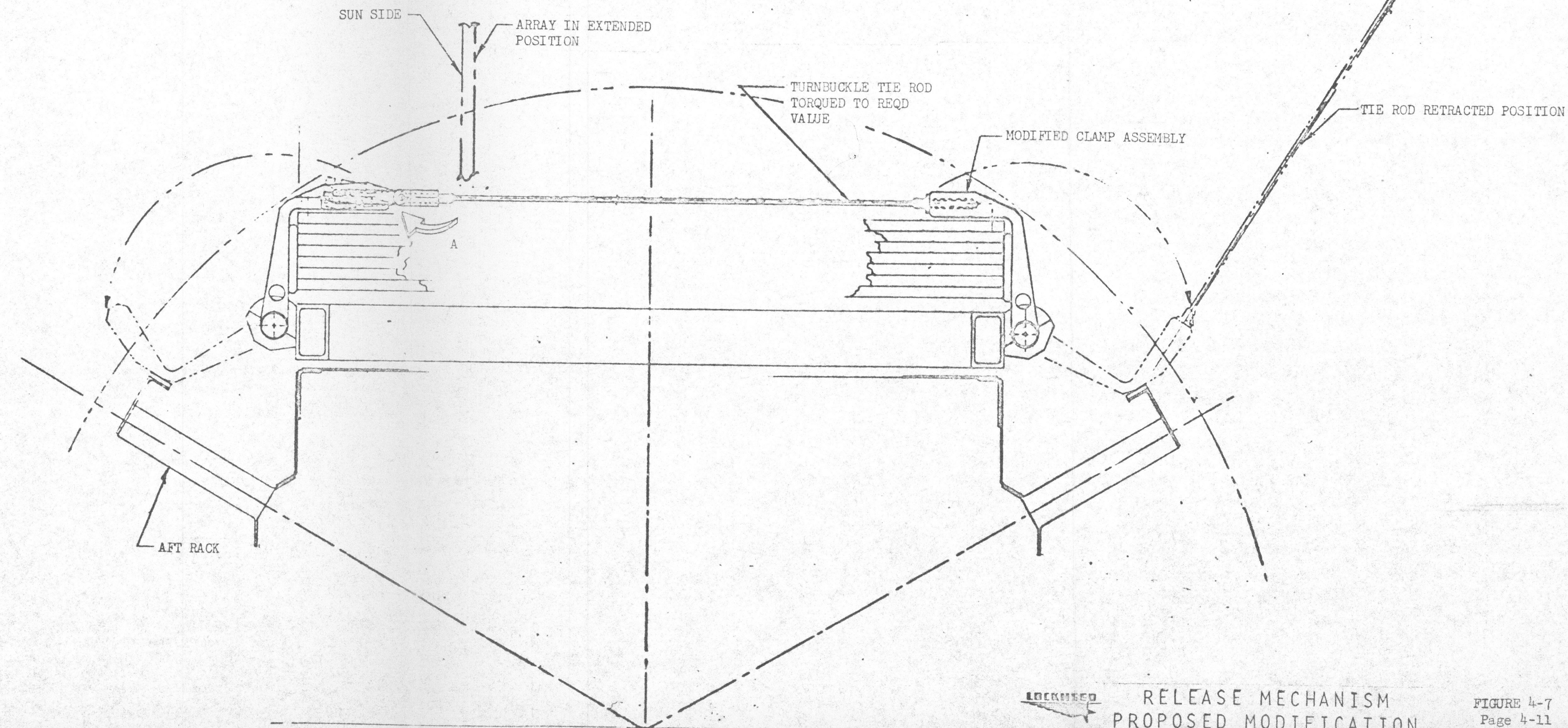
A variation of this concept was considered wherein the tie rod and pin pullers were replaced by a cable and pyrotechnic cable cutters, respectively. This system exhibited the same deficiencies previously noted.

- Concept "B"

As depicted in Figure 4-7, the tie rod is rigidly fixed in one clamp assembly and both pin pullers are installed on the opposite clamp in a staggered fashion. The tie rod terminates in a fitting which is free to swivel at the tie rod connection. This fitting includes two cutouts which hook over and under the pin puller pistons. The orientation of these hooks in conjunction with tie rod tension will maintain proper engagement of the



ARTIST'S CONCEPT
VIEW A



LOCKED

RELEASE MECHANISM
PROPOSED MODIFICATION
CONCEPT "B"

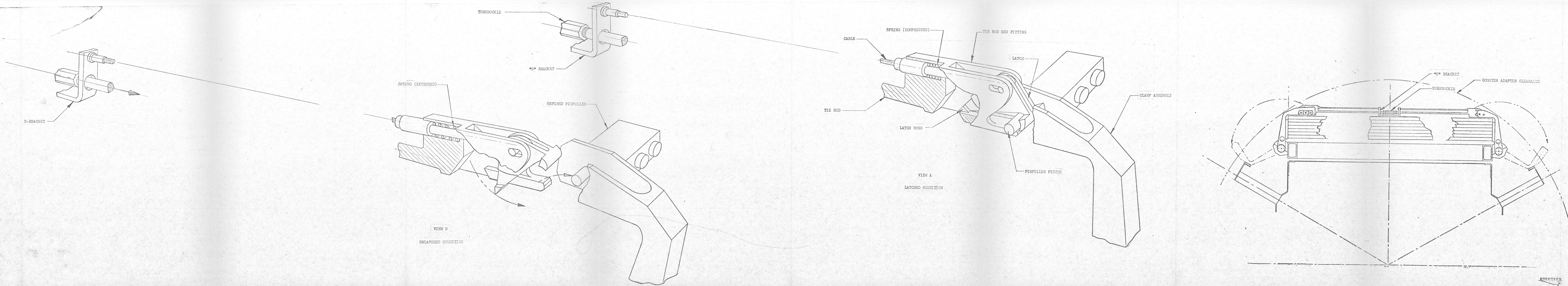
fitting with the pistons (Figure 4-7, View A). The actuation of both pin pullers releases the fitting and the torque tube swings the rod away, providing more than ample clearance for the array to deploy. In the event that either pin puller malfunctions, the end fitting will swivel to disengage itself from the piston.

- Concept "C"

This concept was developed with the intent of retaining all of the existing design with the exception of the tie rod. As shown in Figure 4-8, a sear-type latch is incorporated at both ends of the tie rod. Minor redesign of the "U" bracket is required but its configuration and function remain essentially unchanged. The retaining hook is spring loaded into the unlatched condition and requires tie rod tension to maintain engagement. Small bosses on this latch prevent it from rotating (Figure 4-8, View A). In the failure mode, the spring causes the tie rod to move toward the unfired pin puller while the latch hook is held in its original position by a cable attached to the "U" bracket. A slot in the tie rod end fitting permits the relative motion. A relief slot in the end fitting provides clearance for the latch bosses after approximately one-quarter inch of relative motion and torque tube rotation causes the latch to open (Figure 4-8, View B). Although only one design is presented here, a multitude of variations of this same basic concept were considered. The major drawback of this design is the intricacy of its detail parts including the close machining tolerances required to ensure proper functioning.

- Concept "D"

This concept departs most radically from the existing mechanism and deletes the rod completely. The function of torque tube



clamping action is achieved by incorporating over-center hinged links between the lower end of the clamp assemblies and brackets cantilevered from the mounting bed as shown in Figure 4-9. Pin pullers are connected to the center pivot of the over-center links through a cable and pulley system and actuation of either or both pin pullers will break the link into a position where the torque tube torsion springs will release the clamps. The primary advantages of this system are:

1. Deletion of the tie rod and bracket.
2. Through the incorporation of additional cables and pulleys, both arrays can be released by the same pin pullers.

Disadvantages include:

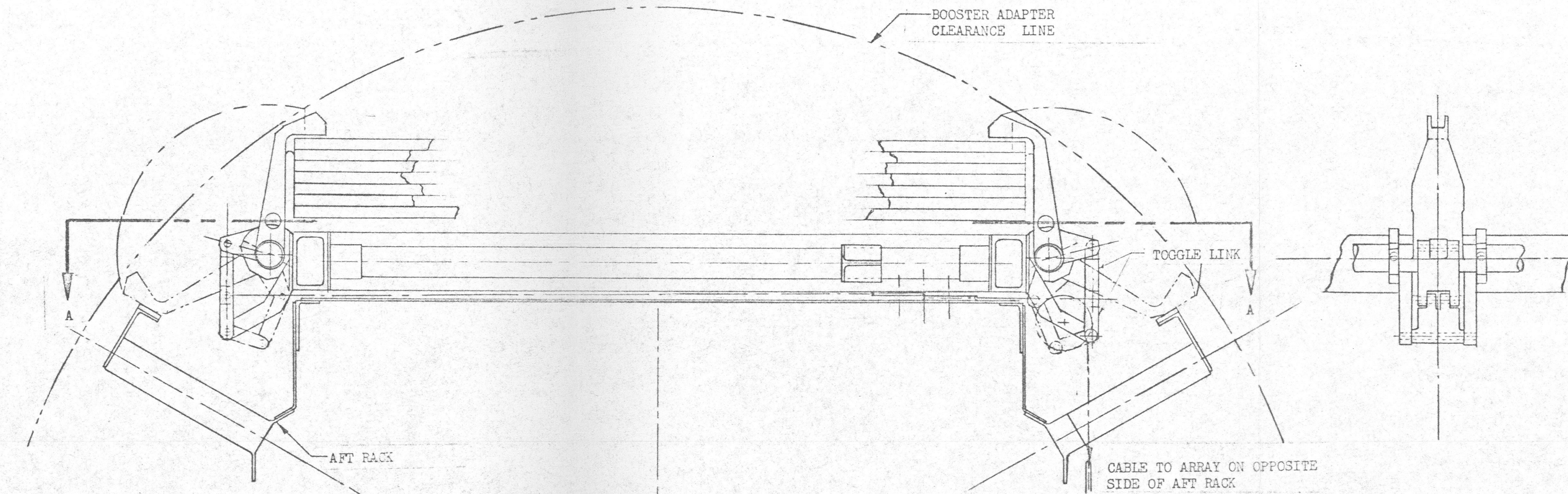
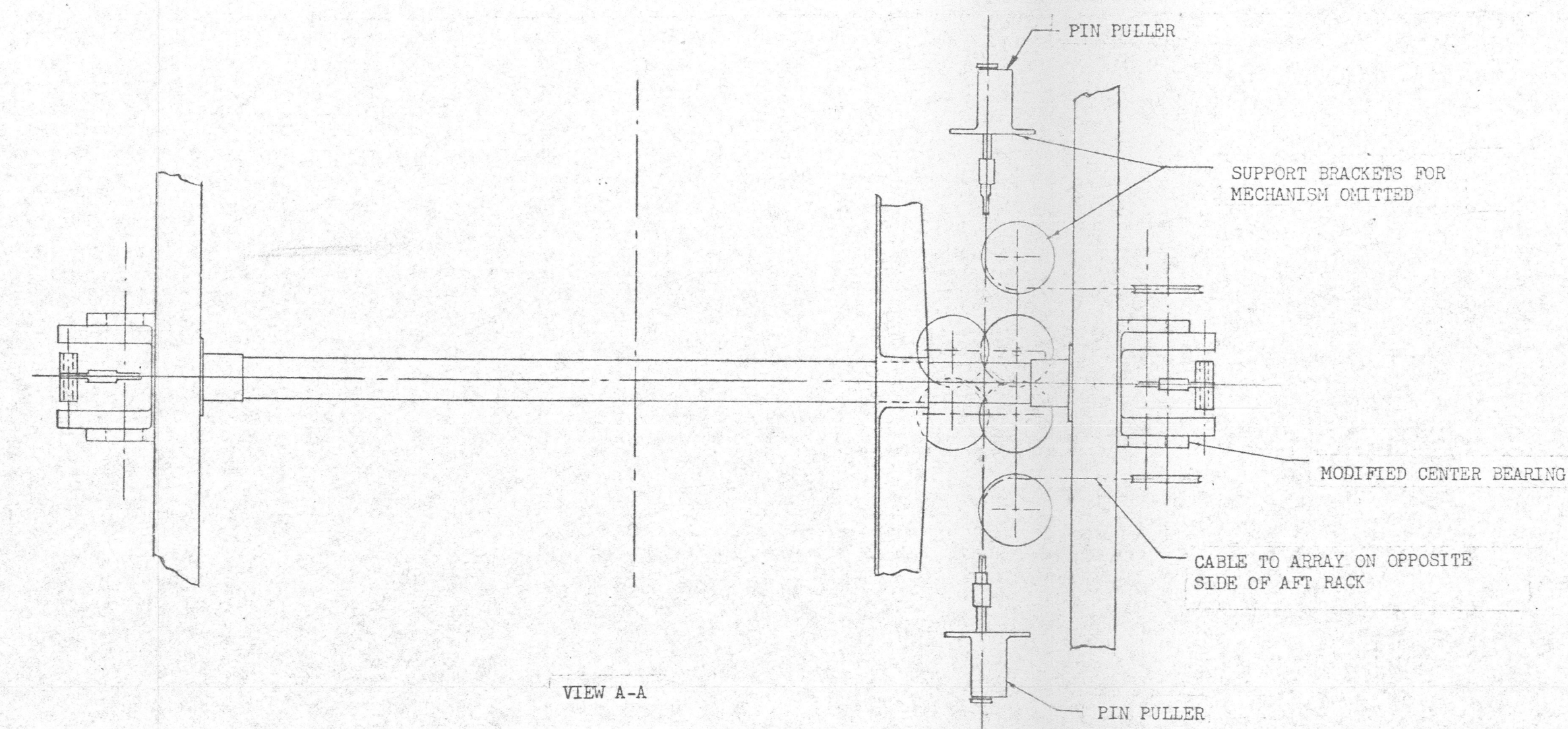
1. Prohibitive impact on cost and schedule due to redesign and fabrication of major system components.
2. Difficulties involved in achieving proper rigging installation, preload, etc., including associated development testing.

- Concept "E"

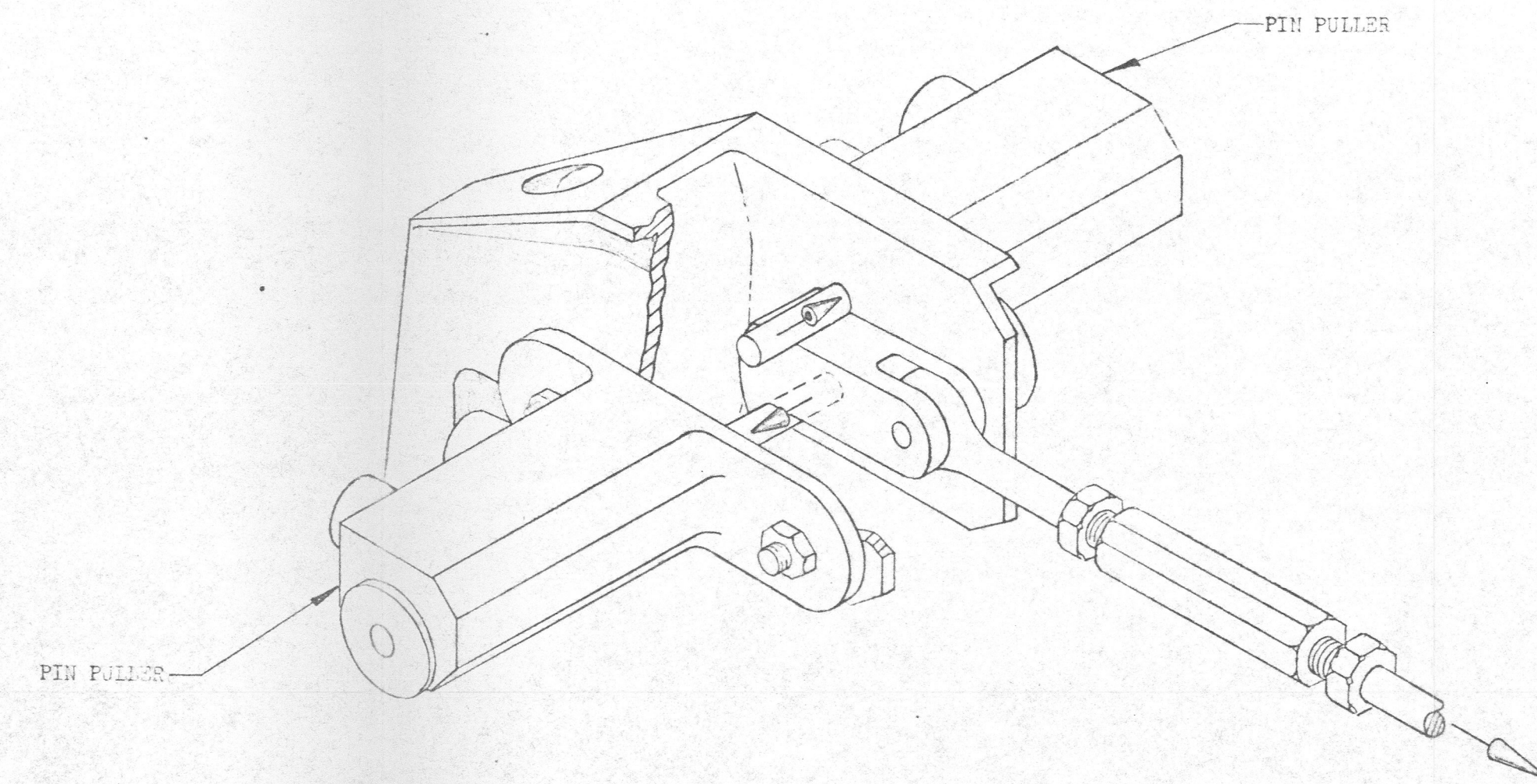
This design is an outgrowth of Concept "B". As shown in Figure 4-10, View A, the only departure is in the orientation of the pin pullers with a resulting change in the tie rod end fitting. The significant benefit derived from this reorientation is that the tie rod tension load is distributed to both pin pullers, essentially reducing the individual piston shear load by a factor of two. It is highly improbable that this load distribution could be achieved, as depicted in Concept "B", due to the manufacturing tolerances which must be considered.

- Preliminary Concepts - Summary

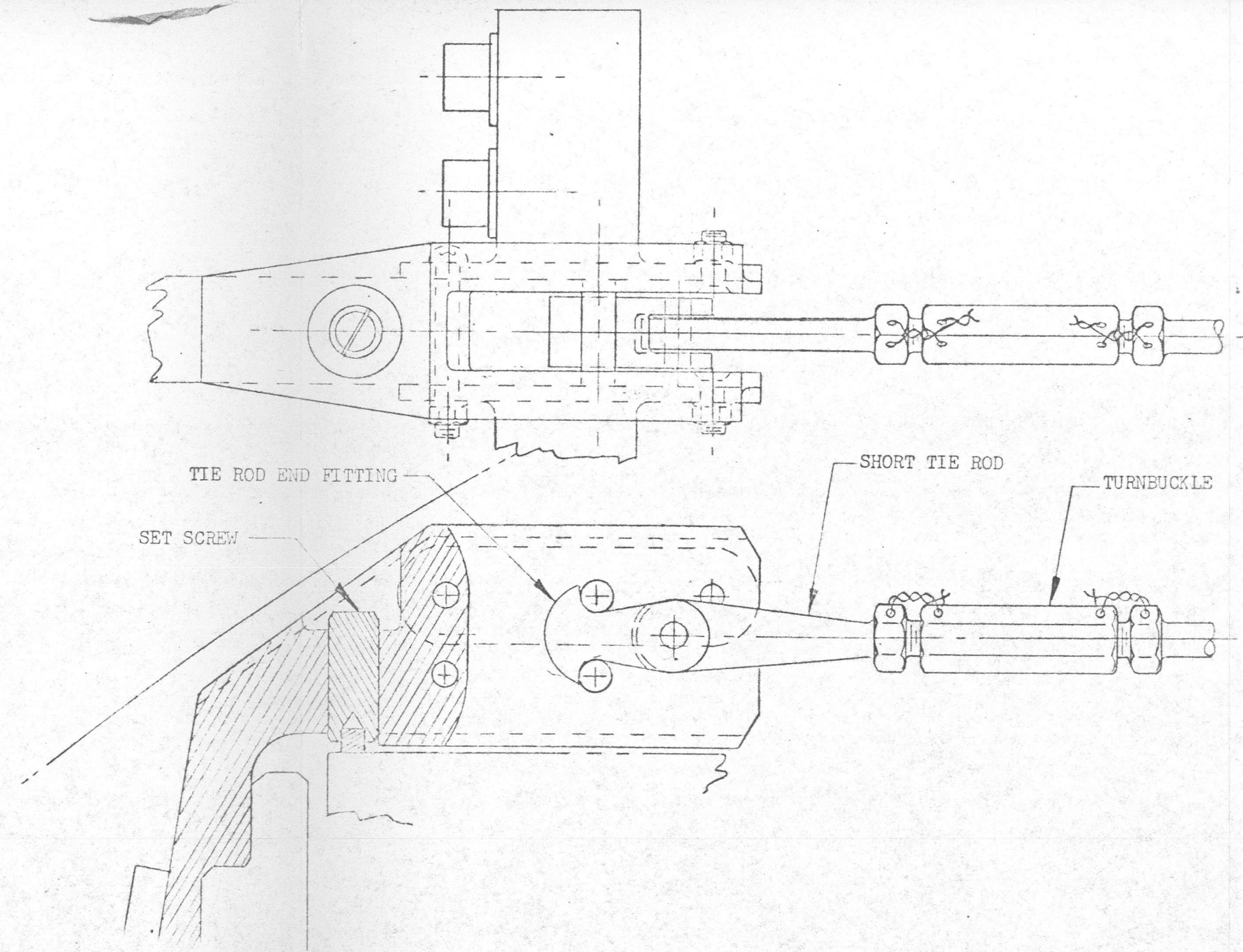
The preceding concepts were presented at the Solar Array Final Design Review on 22 May 1968. The consensus among LeRC and LMSC attendees was that Concept "E" offered the most desirable



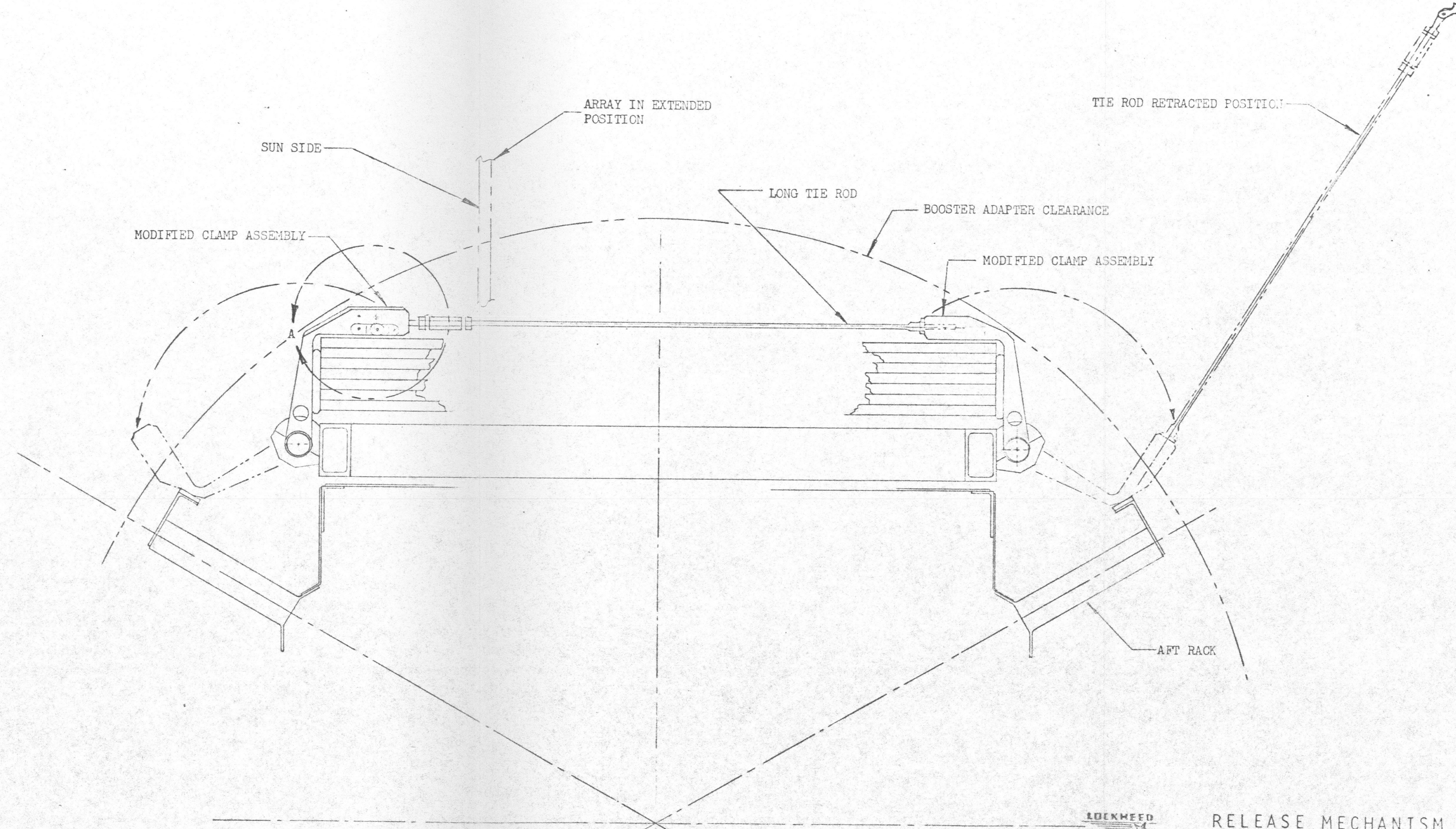
RELEASE MECHANISM
PROPOSED MODIFICATION
CONCEPT "D"



ARTIST'S CONCEPT



VIEW A



RELEASE MECHANISM
PROPOSED MODIFICATION
CONCEPT "E"

advantages because of the following considerations:

1. Simplicity of component parts.
2. Functional simplicity.
3. Maximum utilization of existing hardware.
4. Reduced pin puller shear loads enhance the possibility that one squib in either pin puller will effect release of the array.

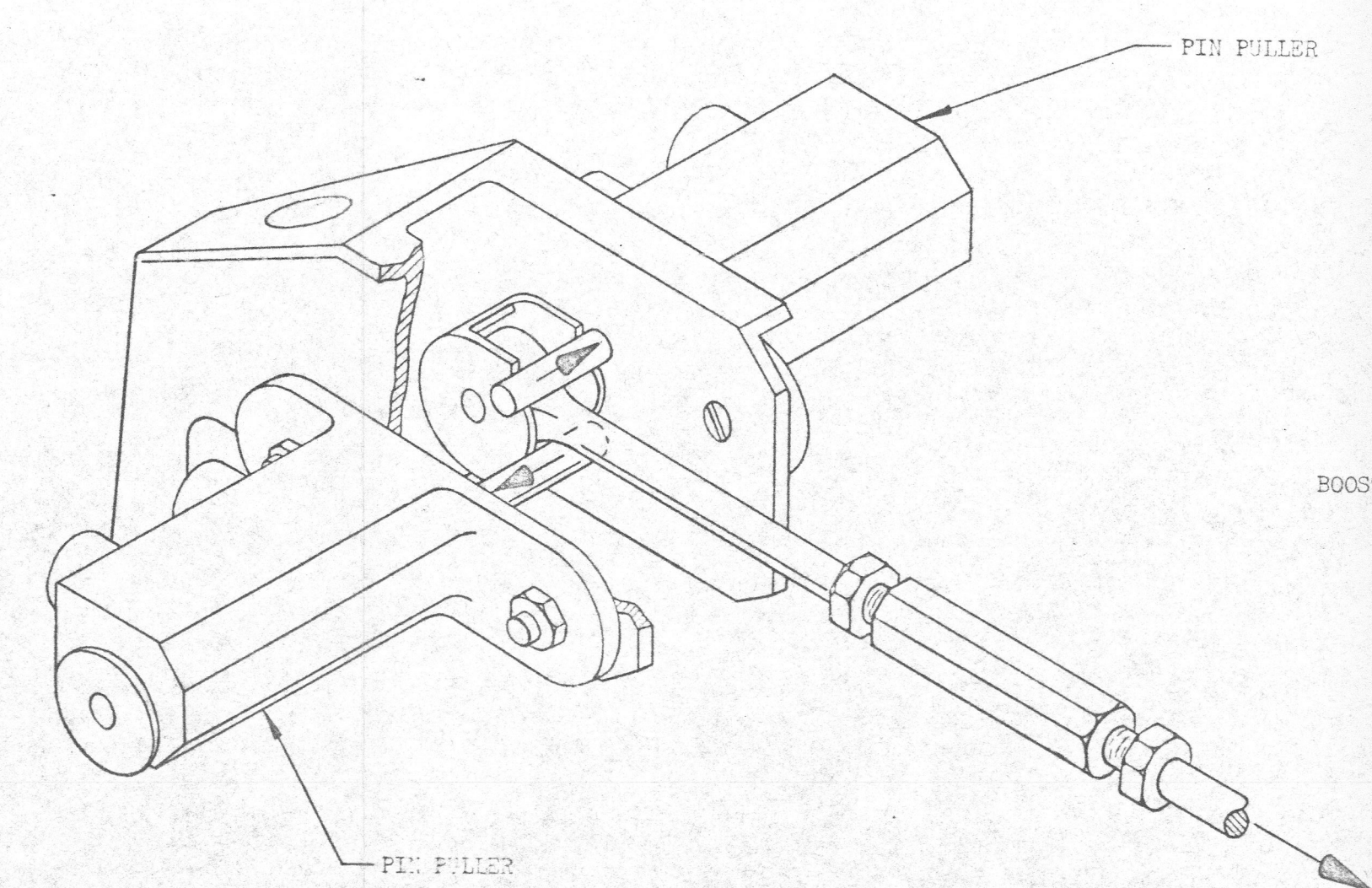
Subsequent to the Design Review Meeting, additional study effort was concentrated on this design to assess any inherent deficiencies which may not have been readily apparent. Deficiencies were found which are identified and resolved in the following discussion of the recommended design.

- Recommended Design

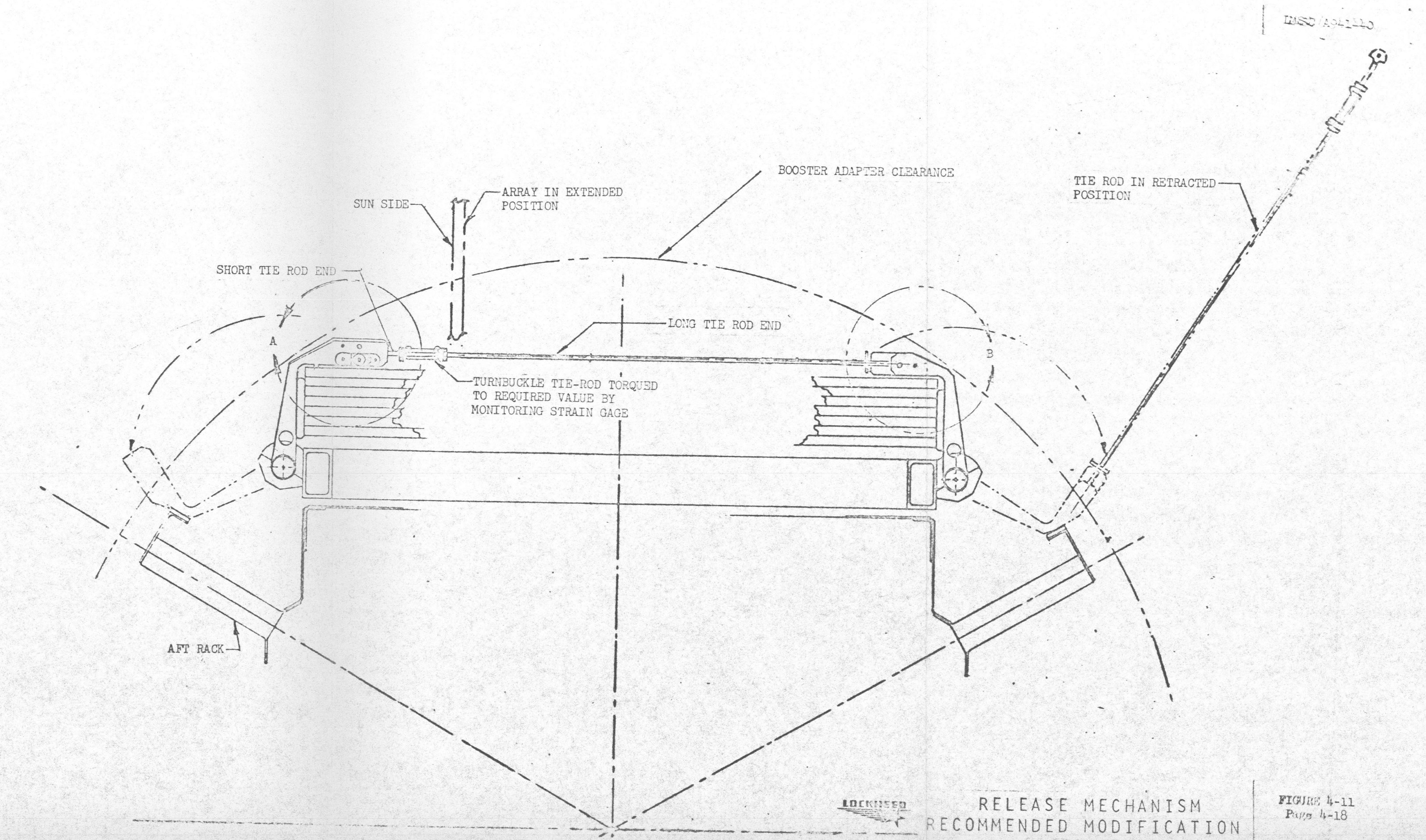
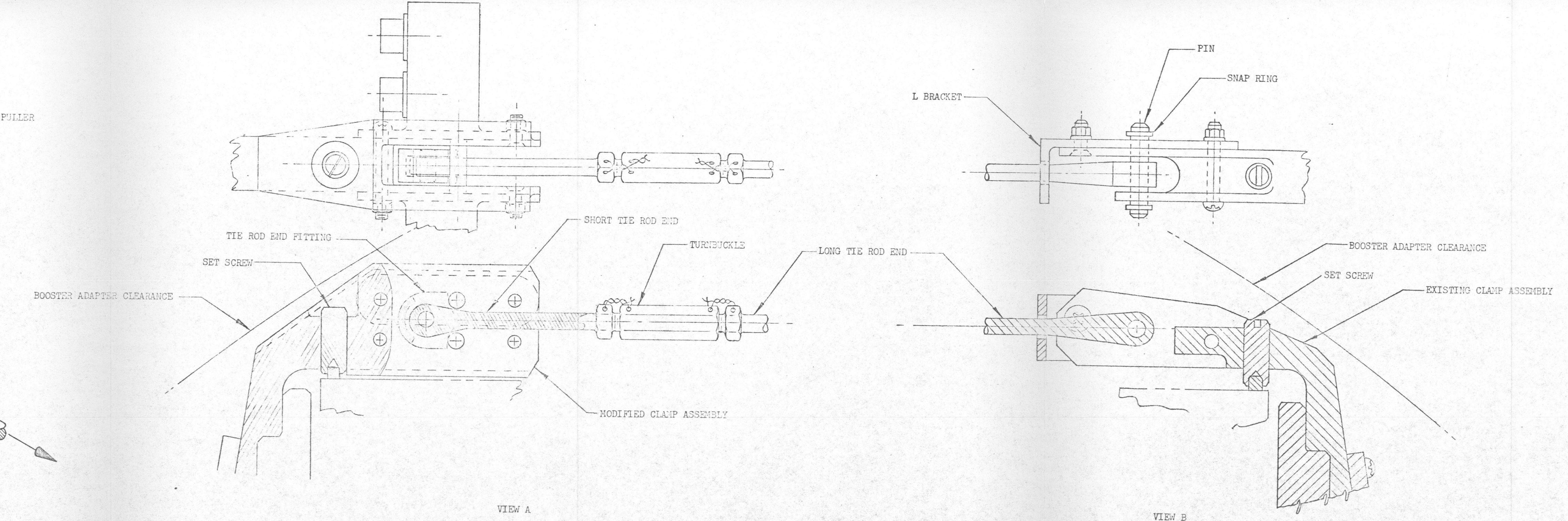
The design concept depicted in Figure 4-11 represents the culmination of the foregoing study effort and is, in the opinion of IMSC, the most suitable means of achieving the release system redundancy desired by LeRC.

This design employs the same principle as Concept "E". Concept "E", however, exhibited two major deficiencies:

1. Due to fabrication and/or assembly tolerances it is virtually impossible to achieve the desired relationship of pin pullers to the tie rod. (A line intersecting the piston centers should be normal to the tie rod center line). The tie rod fitting would tend to bear against both pistons by rotating at its attachment to the tie rod, but tie rod tension would counteract this tendency. As a result, load distribution to the pin pullers would not be equal.
2. The same tolerances mentioned above could cause bending of the tie rod due to its rigid attachment within the clamp assembly at the fixed end.



ARTIST'S CONCEPT



As shown in Figure 4-11, View A, the problem of unequal load distribution on the pin pullers has been solved by relocation of the pivot point of the tie rod end fitting. In this configuration the fitting will align itself on the pin puller pistons without being adversely influenced by the tie rod tension. This should result in equal loading at the pin pullers.

At the fixed end of the rod the proposed design shown in Figure 4-11, View B, utilizes the existing clamp assembly and solves the bending problem by providing a pin joint at the tie rod end. The rod passes through an "L" bracket which provides rotational restriction for the rod in the released condition. Impact of the clamp assembly against the stop pad will cause a "whipping" action of the tie rod. Preliminary investigation indicated that the forces involved would probably not cause the tie rod to take a permanent set in bending. Subsequent test data analysis confirmed this preliminary investigation and will be discussed in a later section of this report.

The width of the clevis opening for the tie rod and end fitting is a function of pin puller piston extended and retracted length. Maximum use of this space is realized by press fitting a pin in the short tie rod end while providing slip fit clearance between the pin and the swivel fitting.

In the recommended design the clamp assembly, long tie rod section, and turnbuckle was to have been fabricated from the same materials as the existing design (7075-T6 Aluminum Plate, 7075-T651 aluminum Alloy Rod, and 303SE CRES, respectively). Subsequent analysis performed during detail design indicated that the use of Titanium for the entire tie rod, toggle assembly, and turnbuckle was more desirable due to its high strength and low coefficient of expansion.

4.2 ELECTRICAL

To meet the requirements of the SERT II program several improvements to the existing design were necessary. These improvements are as follows:

- NASA requirements for more power at a higher maximum power voltage resulted in a change from 10 ohm-cm solar cells to 2 ohm-cm cells.
- Minor changes to the solar cell filter covers to incorporate recent vendor design improvements.
- Gold plating thickness was changed from 100 micro inches to 10-30 micro inches to prevent potential fracture on the Kovar bus tabs due to excessive intermetallic compounds.
- Widening of Kovar bus tabs to provide a greater conductive surface area and a stronger joint between submodules. Holes were provided to permit quality control inspection of solder fillets.
- Solar Array harnesses were rewired to permit full power capability without adding another Peculiar J-Box on the Agena.
- Deletion of most of the Solar Array instrumentation. These were deleted as not being required for the SERT II mission.
- Addition of solar calibration cells. These cells are intended to provide NASA with measurements of solar cell maximum power, open circuit voltage, short circuit current, and pitch rate.

The initially proposed electrical design concept and the final flight hardware design are essentially the same. The preliminary design was presented during the SERT II Solar Array Conceptual Design Review on 19 December 1967. The final flight hardware design will be described in Section 5.0 of this report. The principal design changes involved redesign of the pyrotechnic circuitry of the solar array release mechanism and the addition of a pitch rate monitor cell to the 3 cell module.

SECTION 5.0

FINAL DESIGN

5.1 MECHANICAL DESIGN

Each solar array is designed in the form of a jack-in-the-box, with a mechanical tiedown and an electrical interface. Each box contains an extensible arm assembly powered entirely by springs and controlled by liquid damping. The lid on the box, also spring loaded, is released by the action of two squib-operated pin pullers. When the lid is released, the array arm automatically extends to a prealigned position.

Three features are typical of each module component. First, normally used tolerances for all rotating joints are relaxed to provide increased clearance and preclude the possibility of hangup due to foreign particle intrusion or vehicle ascent and orbit heating. Second, all rotating or sliding joints have either the shaft or the journal coated with a baked on, solid-film lubricant to minimize friction and prevent electrochemical action. Third, all exposed surfaces of the structure are finished to specific thermal control requirements for collector-panel temperature control.

The mounting-bed and extensible arm assembly design and fabrication techniques used on the SERT II Program are the same as those previously used by LMSC on an Air Force satellite program.

The deployment mechanism, while using major components and techniques previously developed, does differ from the Air Force system in the motion imparted to the array assembly. Additionally, it does not become an assembly, as such, until its component parts are installed into the mounting-bed.

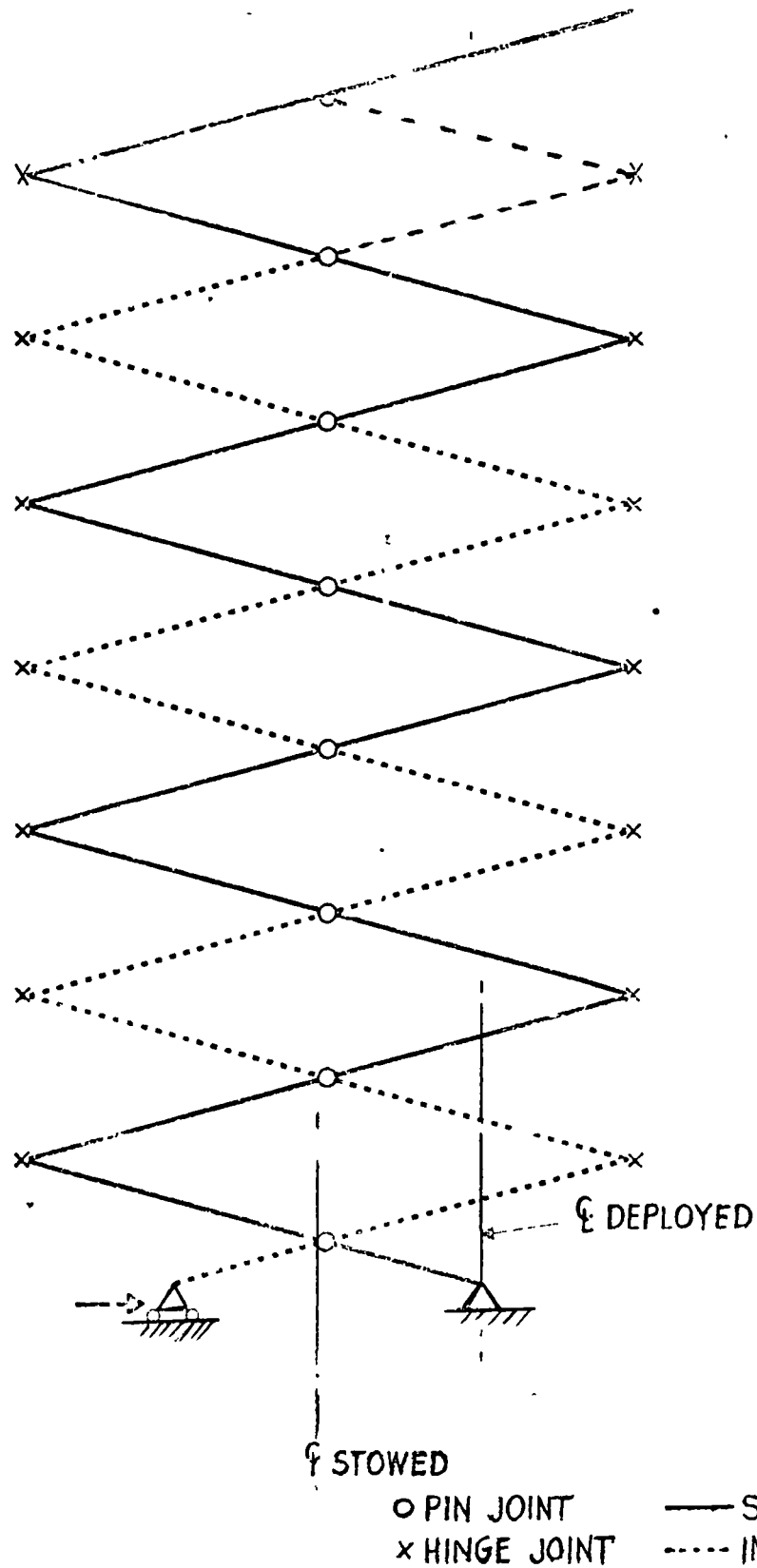
5.1.1 Deployment Mechanism - The concept of using a sliding sleeve on a fixed tube to erect and position the "lazy tongs" of the solar array assembly has been incorporated in the design of the SERT II Solar Array Module.

The inboard leaf of the array, is rotated about a fixed pivot on the support (cross bar) assembly with the inriggers attached through a single fitting to the slider. Figure 5-1 illustrates the array geometry. During extension, the actuator - through appropriate linkage - moves the slider along the support tube until the full extension position is attained. See Figure 5-2. The actuator is used to control the extension rate and complement the array arm forces when they are approaching their minimum output. A fluorosilicone fluid is used as the damping medium because of its relatively stable temperature characteristics. Rate control is adjustable by throttling the fluid flow with a needle valve within the piston rod. A compression spring surrounding the actuator, aided by the mechanical advantage of the mechanism, complements the forces built into the arm assembly.

Full arm extension and position is accomplished by adjusting the erecting mechanism when it is installed in the mounting bed. Additionally, an angular adjustment of the array assembly with respect to the mounting-bed/deployment mechanism assembly has been incorporated in the inrigger assembly. This adjustment is of limited capability and is intended to aid in coping with manufacturing tolerances.

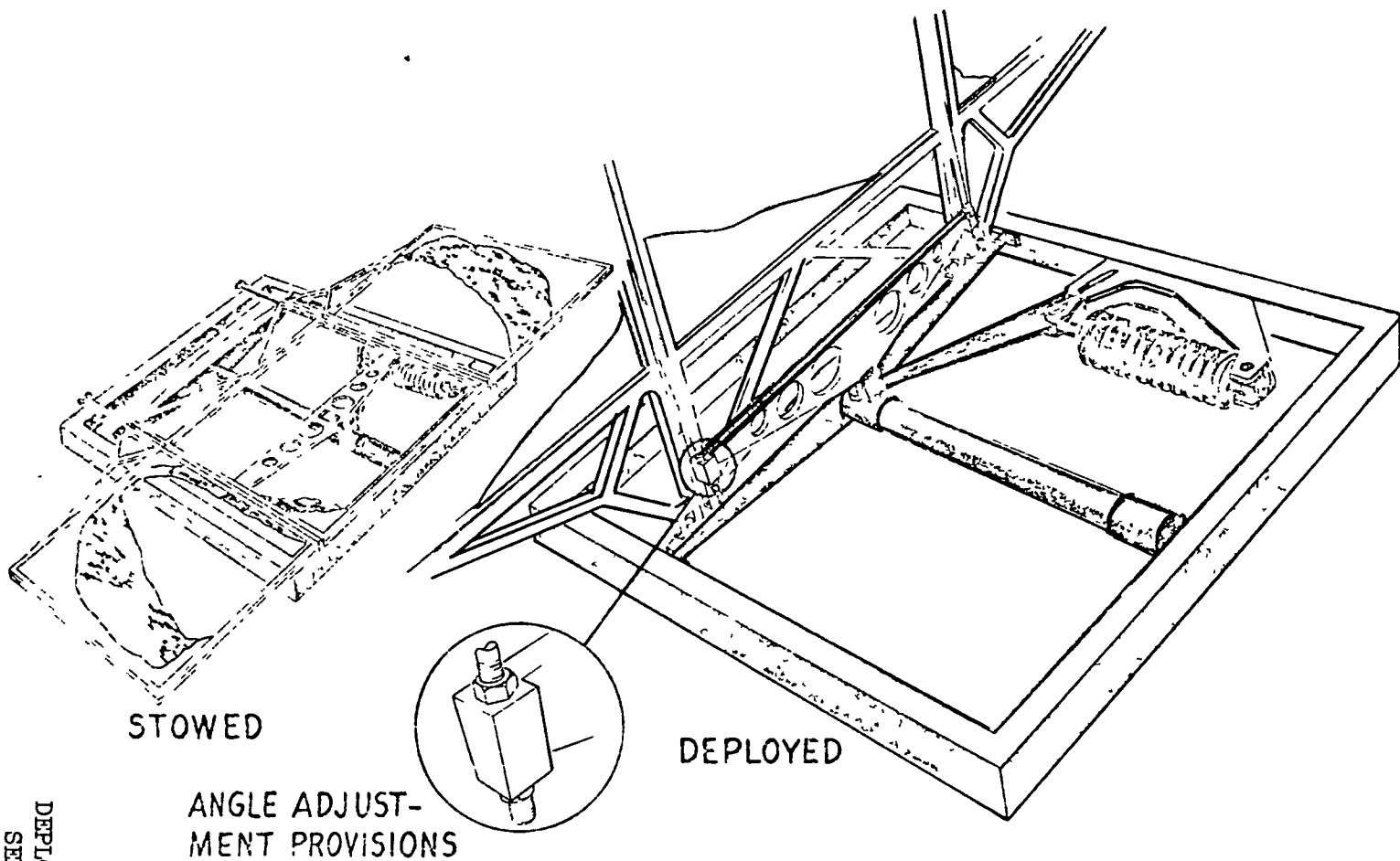
The tube of the support assembly is machined from 1.250-IN. diameter 4130, condition N, Type 1 chrome-moly steel. The tube is first ground to an outside diameter of $1.2448 + 0.0000 - 0.0020$ inches. It is then chrome plated to $1.2525 + 0.0020 - 0.0000$ diameter and ground to the required diameter of $1.2468 + 0.0017 - 0.0000$ inches. The surface finish of the tube is 32 rms. concentricity of 0.0005 TIR and an overall waviness along the 27.60 in. tube length is 0.001 TIR. The final operation in the manufacture of the tube is to hand burnish the outside surface with molybdenum disulfide per MIL-M-7866 until a smooth, glossy surface has been produced.

The slider is fabricated from 6061-T-651 aluminum bar. The inside diameter of the slider is machined to a diameter of $1.2530 + 0.0008, -0.0000$ in. with



SOLAR ARRAY GEOMETRY

FIGURE 5-1



DEPLOYMENT MECHANISM
SERT II PROGRAM

FIGURE 5-2

a surface finish of 16 rms. Chamfers of 0.05 by 45 deg are machined into both ends of the part. The entire inside diameter of the slide is coated with baked-on dry film lubricant to a finished diameter of 1.2500 +0.0008 -0.0000 in. Incorporated within the slider, in addition to previously mentioned features, are provisions for the mechanism's stop.

The stop performs a dual function, (a) it stops the action of the deployment at the predetermined extension position and (b) it trips the micro-switch used to signal the completion of the extension cycle.

5.1.2 Release Mechanism - The recommended design of the solar array release mechanism, described in paragraph 4.1.2.2 and illustrated in Figure 4-11, was selected as the final design.

As a result of analysis conducted subsequent to the adoption of the recommended release mechanism design, a positive torque tube stop, P/N 1385645, was added to the design. The purpose of the stop is to prevent the clamp and tie rod assembly from rebounding and impacting the array.

During the release mechanism evaluation tests described in Section 6.3, a test anomaly involving clamp-set screw misalignment occurred. The design was modified to correct for the misalignment. The modification consisted of installing a shim under the key in the center clamp of each torque tube assembly to provide vertical alignment of the clamp, parallelism between key and key-way, and centering of the clamp set screws over their pads. In addition, set screw pads were bonded on top of the existing pads at the 4 outboard frame locations. The pads have a lip on the inner edge, toward the center of the array, as an added provision to prevent the set screw from moving beyond the edge of the pad during vibration environment.

Deployment testing, conducted subsequent to the vibration confidence tests, demonstrated that the modified hardware would function as designed. Engineering drawings were released to accomplish the above modifications on the flight hardware.

5.1.3 Mounting Bed - The mounting bed supplies the ascent protection for the arm assembly and deployment mechanism. It mounts on the Agena vehicle aft equipment rack prealigned to the geometric axes. It can be considered as the container for the array arm and deployment mechanism and consists of a frame and two retaining clamps.

The mounting bed frame is an integral structure with journals for the retaining clamps along each side. The retaining clamps are integral spring-loaded structures which rotate in the frame journals. A total of 14 set screws in the retaining clamps are tightened to specific requirements and in a prescribed sequence for retaining the folded arm assembly.

The center fittings on each retaining clamp mate with the grooves on the arm assembly to provide major acceleration support for the arm. The groove and the clamp fitting are both tapered to ensure free separation. A tie rod links the two retaining clamps. Two squib-operated pin pullers mounted on one center clamp joins the tie rod and the retaining clamps. When the arm and erection mechanism are assembled into the mounting bed, the tie rod is preloaded with an adjustment nut. The retaining clamp set screws are then torqued to preload the arm assembly. Once the pin-pullers are fired, the retaining clamps are released, and the array arm is free to extend. (Reference Figure 4-11).

Material used in the fabrication of the basic framework is AZ31B-F extruded rectangular magnesium tubing, welded to form a box framework. To this framework are added the necessary journals, pads, tie-down bracketry, etc., required to support the array assembly, deployment mechanism, and for attachment to the launch vehicle.

Dimensional control of the bed assembly is achieved by performing all machine operations after welding and through the use of controlled tooling. Figure 5-3 depicts the basic framework and critical dimensional control.

5.1.4 Solar Array Assembly - The array assembly is a "lazy-tong" type mechanism, consisting of a number of connecting links. One-half of the links (leaves) are hinged together in one continuous assembly, providing support for the collector panels. The other half of the links (inriggers)

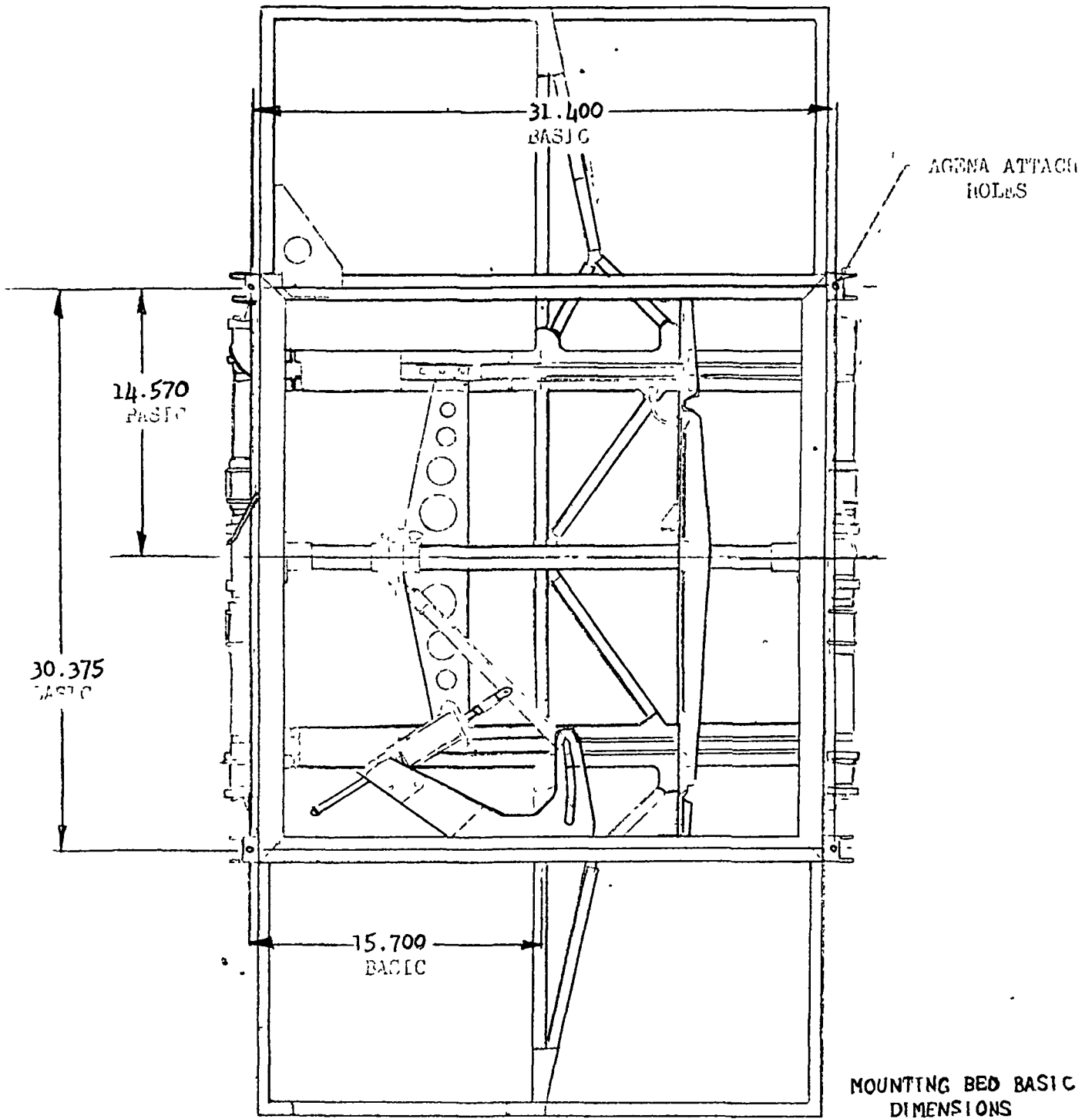


FIGURE 5-3

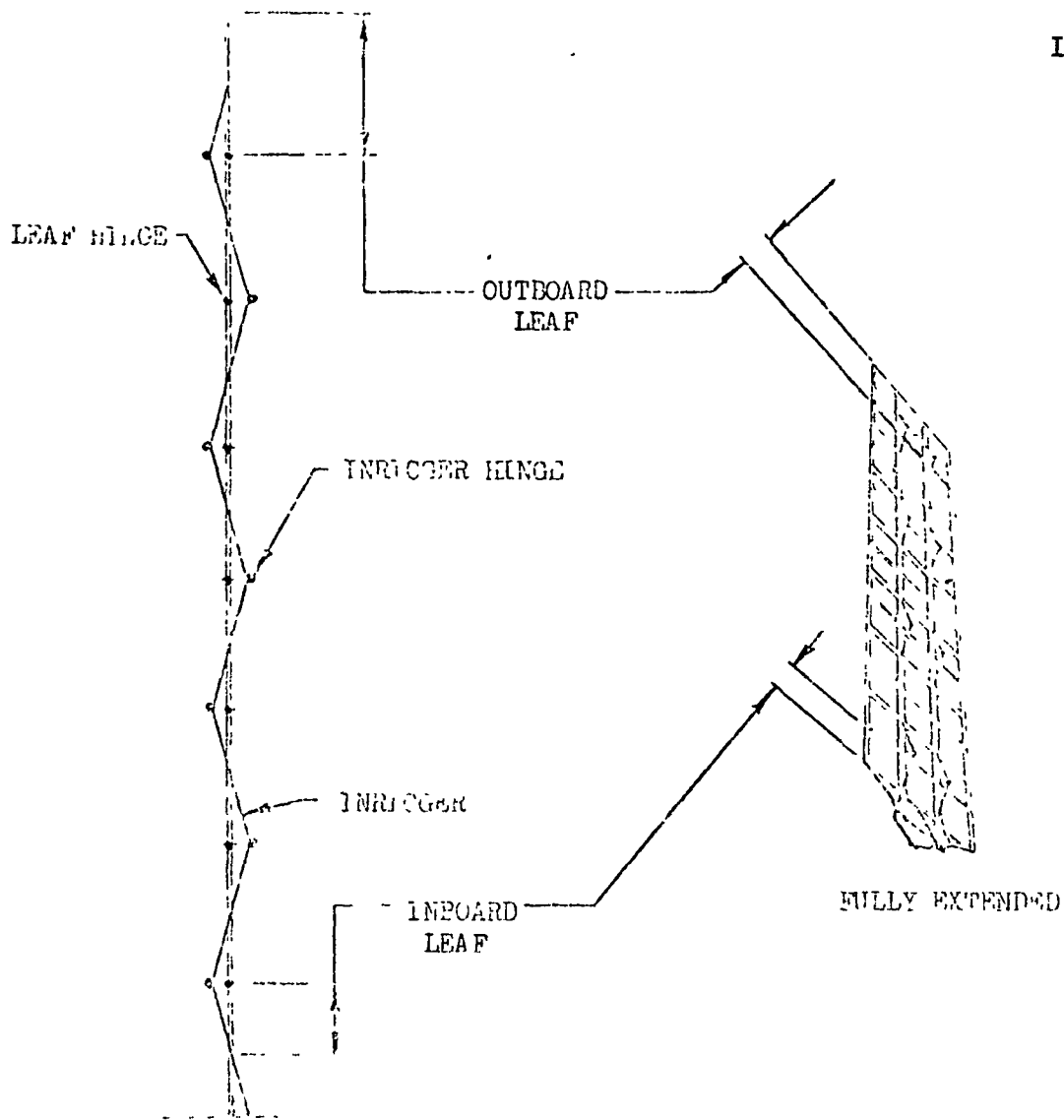
are hinged together and pivot about pins fixed within each leaf assembly. See Figure 5-4. When the retaining torque bar/side clamps are released by the action of two squib-operated pin pullers, the arm automatically extends to a prealigned position. The solar array assembly is powered entirely by springs and controlled by liquid damping. The power for extension of the arm is distributed over the entire arm assembly. Small torsion springs replace two or three lobes at each hinge section. These are complemented by larger torsion springs installed at each inrigger joint. The load is uniformly distributed in this manner because of the size of the array and the very thin sections from which it is fabricated.

When fully extended, the leaf sections are in line, forming one planar surface; the inriggers remain at an angle of 150 degrees with each other (15 degrees with the leaves). With this linkage arrangement, the assembly is less flexible and the leaves are held in a flatter position than would be possible with the inriggers in the leaf plane.

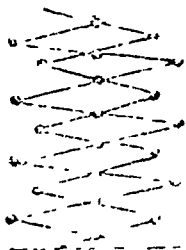
In a folded position, the array arm envelope is made as thin as possible by nesting the inriggers into their respective leaves. Tapered grooves are cut normal to the hinge lines on each side of the folded assembly to accommodate the mounting bed center fittings.

The sides of the frames are held perpendicular and parallel with respect to a base side within ± 0.015 in; flatness along the side is held to 0.020-in., and the maximum mismatch between the mounting surface of the collector panel and the centerline of the frame is held to 0.010 in. In addition, controlled tooling is employed for drilling the 14 mounting holes required for installation of each solar collector panel into the frames.

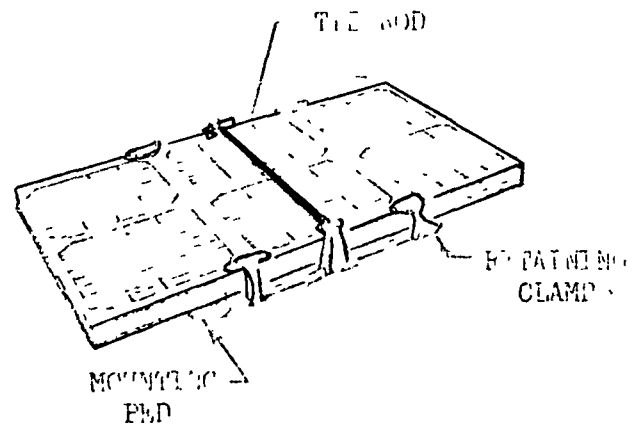
Dimensional control of the inrigger assemblies is again achieved through the use of controlled tooling. Using the centerline and the side surface of the pivot point as the reference point, the end fittings are held parallel with respect to the reference surface while the distance from the pivot point centerline to the holes in the end fittings is held to ± 0.005 in. in the longitudinal direction and ± 0.002 in. in the vertical direction.



FULLY EXTENDED



PARTIALLY
EXTENDED



SOLAR ARRAY ASSEMBLY

FIGURE 5-4

Material used in the fabrication of the panel frames is AZ31B-F extruded rectangular magnesium tubing, with the tubes being cut to length and the flanges trimmed prior to assembly. To achieve greater stiffness in the last or out-board leaf, two of the tubes are bonded together using MIL-A-5090 Airframe Structural Adhesive. Nondestructive testing of the bonded assemblies is accomplished by use of a simple beam flexure method.

For the SERT II Program, the corner and center fittings to be used in the assembly of the panel frames are machined from magnesium bar rather than from the magnesium sand castings used on the military program. This change, based on the recommendations of design and production engineering, requires less lead time for procurement and contributes to the ease of manufacture. These fittings, together with the rectangular tubing, are riveted together using controlled tooling for holding the close dimensional tolerances previously discussed.

The inrigger assemblies on all but the inboard, second, and third leaves are fabricated from 7075-T651 aluminum alloy and 0.312 in. diameter steel rod. To minimize the power output loss due to shadowing from the inriggers, the cross-sectional area of the steel rods has been minimized, consistent with good design practices.

Since the bending loads on the inriggers increase as they go inboard, it was necessary to strengthen the three inboard members. A design using titanium inriggers (6AL-4V per AMS 4911) of maximum cross-sectional area is required in this area. The titanium alloy was selected over steel or beryllium as the best material giving the combination of high strength-weight ratio with minimum cost.

In the assembly of the frames and inriggers, a 0.187-inch diameter pin, passing through the inriggers into the frame assemblies, provides the rotational joint.

MS-2001 aluminum extruded hinges are mounted on the rectangular framework using controlled tooling for accurate location. These hinges provide structural attachment between panel frames and provide adequate strength for stowed and orbital extension of the assembly. Additionally,

to provide the capability for handling the framework assembly in a 1-g environment, 0.050 CRES 301 $\frac{1}{2}$ hard splice plates are added between frames.

Mechanical assembly of the SERT Solar Array assembly consists of mating the inboard leaf, six center leaves, and one outboard leaf. This assembly, with the exception of the final chining operation, is accomplished in a large assembly jig. As part of the mechanical assembly, torsion springs and hinge pins are employed at the leaf hinge lines and inrigger joints. These springs and pins provide for permanent attachment of the subassemblies and are the primary source of energy for array extension.

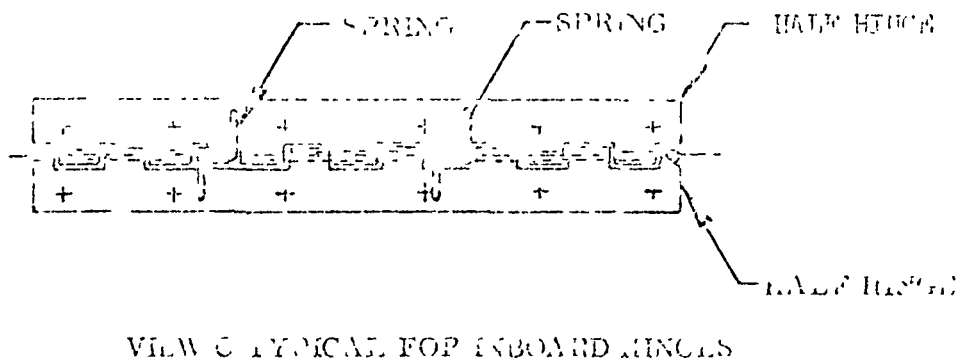
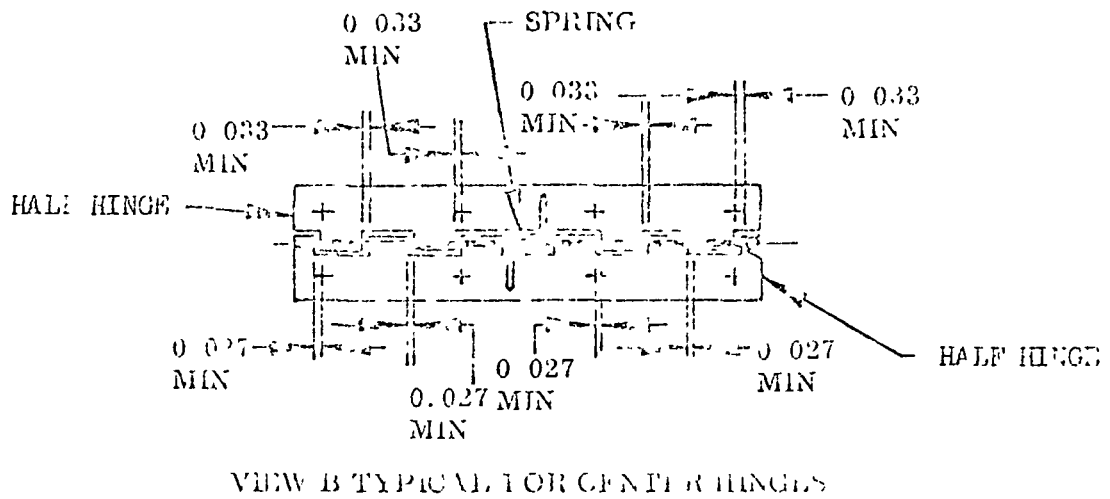
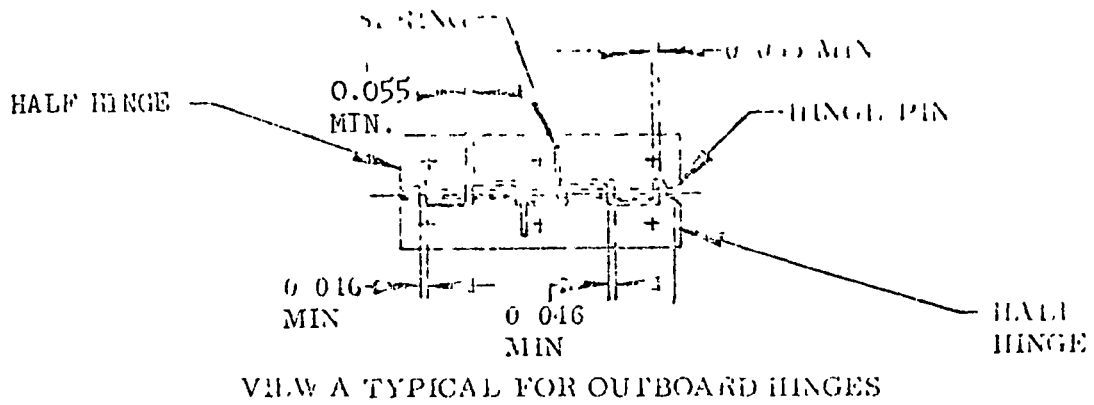
Each hinge line contains six torsion springs and each spring has a nominal torque of 1.2 in.-lb. in the retracted position and a torque of 0.4 in.-lb. at full extension. In the completed assembly there are a total of 42 of these torsion springs providing a torque of 50.4 in.-lb. and 16.8 in.-lb. in the retracted and extended positions, respectively.

Torsion springs, rated at 7.02 in.-lb. retracted and 2.00 in.-lb. extended, are installed at each of the 14 inrigger joints. These springs provide an opening torque of 98.28 in.-lb., with 28.0 in.-lb. remaining at full extension.

The total opening force then available from the hinge line and inrigger torsion springs is 148.68 in.-lb. with 44.8 in.-lb. remaining at full extension.

The possibility of binding between adjacent hinge lobes due to manufacturing tolerance build-up combined with thermal expansion of the leaves has been investigated. As a result, clearance between the adjacent lobes on the outer hinges was increased to that shown on Views A and B of Figure 5-5.

Because of their location in relation to the center line of the array, the inner hinges did not require clearance modifications. Thermal analysis indicated that the effects of thermal expansion would not exceed the accumulated manufacturing and assembly tolerance build-up.



HINGE CLEARANCE

FIGURE 5-5

Corrosion resistant steel pins per MS 20253-3 are used in the assembly of the leaves. A machined pin of 7075-T6 aluminum is used at each of the inrigger joints. All of the pins are coated with baked-on molydisulfide dry film lubricant.

To hold alignment and to prevent the extended solar array assembly from being forced over center by the spring force remaining in the hinge line, the inrigger, and the erection mechanism actuator, stop pads are added to the leaf assemblies at each inrigger hinge line -- a total of 14 places.

The final operation for completion of the mechanical assembly of the solar array arm is the machining of a vee-groove in each side of the assembled and folded array arm. This groove mates with the solar array mounting bed and allows the longitudinal ascent loads from the folded solar array to be transmitted to the mounting bed and into the basic satellite structure.

The electrical assembly of the SERT Solar Array assembly consists of the installation and wiring of the 45 solar collector panels into the mechanical assembly previously discussed, and the electrical acceptance testing of the completed assembly.

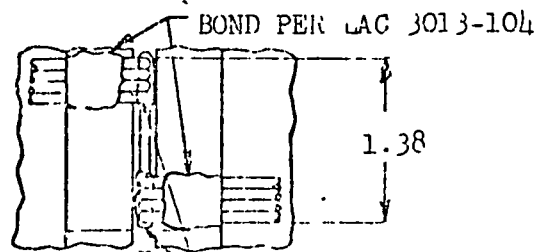
Each collector panel is installed into the array frame using 14 10-32 flathead screws. Torque value on each screw is limited to 12 to 15 in.-lb to prevent stripping of the threaded holes in the magnesium collector panels. The head of each screw is sealed in place with an epoxy-base adhesive per MIL-A-8623 to prevent loosening during transit and vehicle ascent.

Mounting holes in the collector panel corners are threaded through the panel and are tapped to accept RH threaded fasteners. The remaining 10 holes in the collector panels are drilled and tapped to accept a 10-32 LH thread. This method of installation has been used on all IMSC programs using similar collector panels and eliminates the possibility of installing fasteners of insufficient length in the four corners. It also prevents the bottoming out of fasteners in the remaining 10 holes.

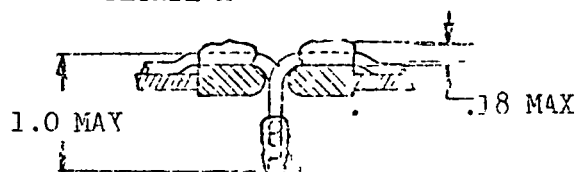
The lazy-tong design lends itself well to the solution of the problems inherent in wiring an extensible structure. The wires, installed along the back of the leaves opposite the cell side where hinge lines occur, pass either through or around the hinge in such a way that a loop is formed providing for free deployment. This loop is bonded in place with enough flexibility remaining to allow repeated extension and retraction of the arm without resulting in failure of the wire.

The electrical harness routing between two adjacent leaf assemblies is illustrated in Figure 5-6.

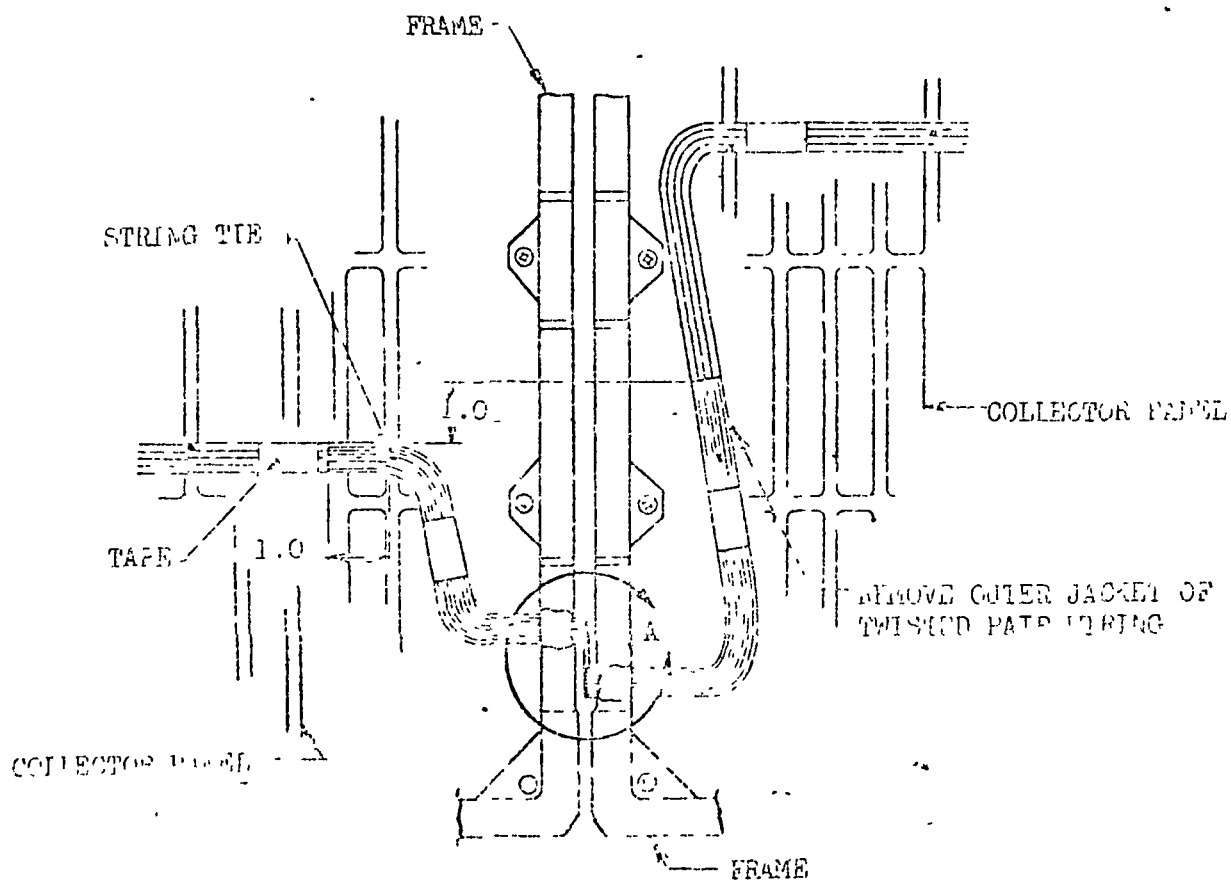
A silicone rubber pad is cemented to the center of the active side of each collector panel. These pads provide cushioning between the adjacent collector panels for prevention of damage to the panels during shipment and during vehicle launch. Additionally, adhesive-backed aluminum foil is added to each pad to preclude the pads from adhering to each other after being under compression in the stowed position for extended periods.



DETAIL A



COAT WIRES AT FORMED BENDS
AS SHOWN PER LAC 3013-104



ELECTRICAL HARNESS ROUTING

FIGURE 5-6

5.1.5 Strength Analysis of Mechanical Design - Table 5-1 presents a summary of the results obtained from a structural analysis performed to substantiate the strength of the SERT II program peculiar components of the solar arrays designed for and used in the Military Program. As the load levels for the SERT II mission are within the qualification levels for the Military Program components, only those components that require modification or those components that are loaded in a different manner as a result of the structural modifications have been reviewed.

The primary structural components of the solar array module are the solar panel frames, inriggers, mounting bed and erection mechanism. The structure is designed to resist orbit loads due to attitude changes by the inriggers and solar panel frames acting together to form a single cantilever beam. During ascent the arrays are folded and clamped to the mounting bed. The mounting bed is attached to the Agena aft rack by four stand-offs. The arrays are deployed by initiating a pyrotechnic device which causes the stowage clamps to relax thereby releasing the arrays. The solar panel hinges are spring loaded and cause an immediate surge when the stowage clamps are relaxed. The deployment rate is controlled by a damped spring-type actuator mechanism which restricts the deployment rate during the early stages and sustains the deployment rate during the final part of the erection period. The time required to erect the arrays is on the order of one-half minute.

Results of a joint investigation with orbit thermodynamics personnel of the release mechanism tie rod tension has indicated that the predicted thermal environment does not create excessive tie rod tension. The maximum increase in tie rod tension due to temperature gradients between the mounting bed and tie rod which occurs at approximately 2100 seconds after liftoff is 100 lb. The maximum increase at time of deployment is 50 lb. When added to an initial preload of 400 lb the combined loads are within the design limit load of 500 lb (see Case III b below).

ITEM	PART NO.	LOAD	M. S.
Solar Array Mechanical Assembly			
Fitting, Pivot, Inrigger	1367090	I	High
Fitting, Main Inboard Frame	1367071	I	High
Pin, Pivot, Solar Array	1367092	I	High
Link, Adjusting, Main Inrigger	1386818	I	High
Fitting, Main Inrigger	1386782	I	High
Nut, Adjusting, Main Inrigger	1386819	I	High
Pin, Crosshead	1381956	I	High
Pin, Connecting	LS8876-3-34	I	High
Pin, Connecting	LS8876-3-22	I	High
Erection Mechanism Assembly			
Cradle Assembly	1386723	IIIa	High
Pin, Retainer	1386820	IIIa	High
Link	1386726	IIIa	+0.74
Eyebolt	LS8945-1-B	IIIa	+0.05
Crosshead	1386733	IIIa	High
Spar	1386720	I	High
Clip, Spar	1386727	I	High
Tube, Support	1386537	I	High
Support, Tube	1386729	I	High
Mounting Bed Assembly			
Clamp, Center	1388601	IIc	High
Toggle	1388597	IIb	High
Connecting Pin	1388656	IIIb	+0.72
Tie Rod End Fitting	1388598	IIIb	+0.70
Turnbuckle	1388664	IIIb	High
Tie Rod	1388599	IIIc	High
Support, Tie Rod	1388600	IIIc	+0.66
Pin	1381956-3	IIIb	High
Clamp, Center	1386803	IIIc	+0.56
Clamp, Triple	1386802	IIIb	+0.57
Fitting, End	1386801	IIIb	+0.87
Stop	1385645	IIIc	+0.36
Spacer	1385643	IIIc	High
Bolt, Mount	LS8762-4-14	IIa	+0.30
Stop	1371034	IIIc	+0.07

Note: All margins of safety greater than 1.00 are considered as "High".

SUMMARY OF MINIMUM MARGINS OF SAFETY

TABLE 5-1

Analysis indicates that the critical structural member is the link and eyebolt assembly which connects the actuator to the crosshead and resists the surge load on the erection mechanism during the early part of the erection period. This component has survived the dynamic erection tests without visible degradation. The basic arrays as used for the Military Program, were tested for both ascent and orbital loads to load factors equal or greater than those specified for the SERT II mission (Table 5-2).

The following Limit Loads and Ultimate Factors of Safety were utilized for this analysis:

- Case I - Orbit Loads = 0.032 G normal to plane of extended array
- Case II - Ascent Loads
 - a) $N_x = 9.5 \text{ G}$
 $N_y = \pm 2.5 \text{ G}$
 - b) $N_x = 9.5 \text{ G}$
 $N_z = \pm 2.5 \text{ G}$
 - c) Tie rod tension = 400 lb
- Case III - Erection Loads
 - a) Peak tension load on actuator - 405 lb
 - b) Peak tie rod tension = 500 lb
 - c) Peak kinetic energy of stowage clamp assembly = 60 in - lb

NOTES:

<u>Load Case</u>	<u>Duration</u>	<u>U.F.S.*</u>
I	Short	1.25
II(a)	Short	1.25
II(b)	Short	1.25
II(c)	Long	1.50
III(a)	Short	1.25
III(b)	Short	1.50
III(c)	Short	1.50

* Ultimate Factor of Safety = 1.50 in cases where personnel safety is in jeopardy.

LOAD CONDITION	ULT. DESIGN LOAD*	TEST LEVELS
Case I	0.04 G	0.04 G
Case IIa	$N_x = 11.9 \text{ G}$ $N_y = \pm 3.1 \text{ G}$	$N_x = 11.9 \text{ G}$ $N_y = \pm 3.8 \text{ G}$
Case IIIb	$N_x = 11.9 \text{ G}$ $N_z = \pm 3.1 \text{ G}$	$N_x = 11.9 \text{ G}$ $N_z = \pm 3.8 \text{ G}$
Case IIIa	506 lb.	**

* Ultimate design loads are 125% of the design limit loads.

** The deployment mechanism has survived repeated dynamic erection tests without visible degradation.

STRUCTURAL TEST SUMMARY
MILITARY PROGRAM SOLAR ARRAY

TABLE 5-2

5.2 ELECTRICAL DESIGN

5.2.1 Array Power Wiring - The SERT II Solar Array power subsystem is based upon the use of silicon Type N (N-on-P) 2 ohm-cm photovoltaic solar cells. The change from solar cells having base resistivity of 10 ohm-cm to the 2 ohm-cm type was required to insure that the assembled array would provide the increased power and higher voltage specified for the SERT II mission. The N-on-P 2 ohm-cm solar cells were designed and tested by the vendor in accordance with LMSC Specification 1420134. Fused silica solar cell filter covers with vacuum deposited film coatings are used in conjunction with the solar cells to provide thermal characteristics and radiation protection to the solar cell panels for space application. The filter covers were designed and tested in accordance with LMSC Specification 1420144. The solar cells and filter covers are assembled into solar collector assemblies.

The solar collector assemblies were designed, constructed and tested in accordance with LMSC Specification 1419896. The solar collector assemblies were designed to provide an electrical output of 19.8 watts minimum, with a minimum power average of the delivered lot of 20.4 watts, at an output voltage of 34.0 VDC. Power output profiles of the individual solar collector assemblies were furnished to LeRC by LMSC Technical Memorandum titled Final Extrapolated I-V Curves for SERT II Solar Array Subsystem, dated 26 July 1968.

The solar collector assemblies are installed in the +Y and -Y solar array assemblies as described in Paragraph 5.1.4 of this report. A total of 90 solar collector assemblies are utilized. Seventy-six collector panel assemblies are wired to provide 1550 watts at 68 VDC for spacecraft thruster power. Fourteen solar collector assemblies are wired to provide 286 watts minimum power at 34 VDC for the spacecraft support unit housekeeping section.

The power wiring is terminated as described in Paragraph 2.8 and as shown in Figure 5-7. Wiring details are specified by LMSC Drawings 1386793 and 1386794. The power wiring from terminal board TB6 to the SSU interface

SERT SOLAR ARRAY SYSTEM

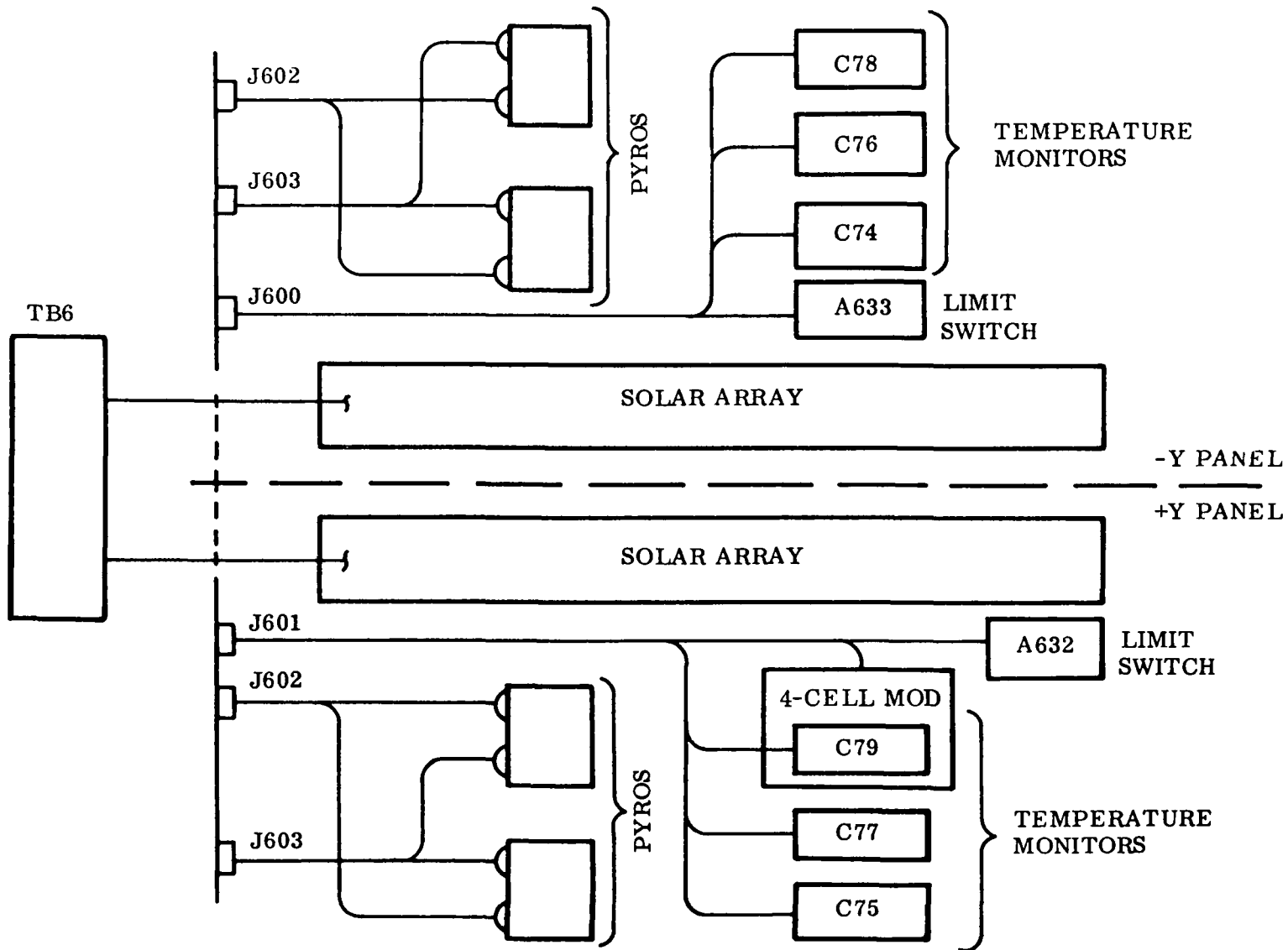


FIGURE 5-7

IMSC/A9411410

is number 8 gauge and meets the voltage drop requirements described in Paragraph 2.8 of this report.

5.2.2 Release Mechanism Pyrotechnic Wiring - The pyrotechnic wiring as initially proposed and as adopted for final design is shown by Figure 5-8. The pyrotechnic circuits were redesigned to provide higher functional reliability of the release mechanism.

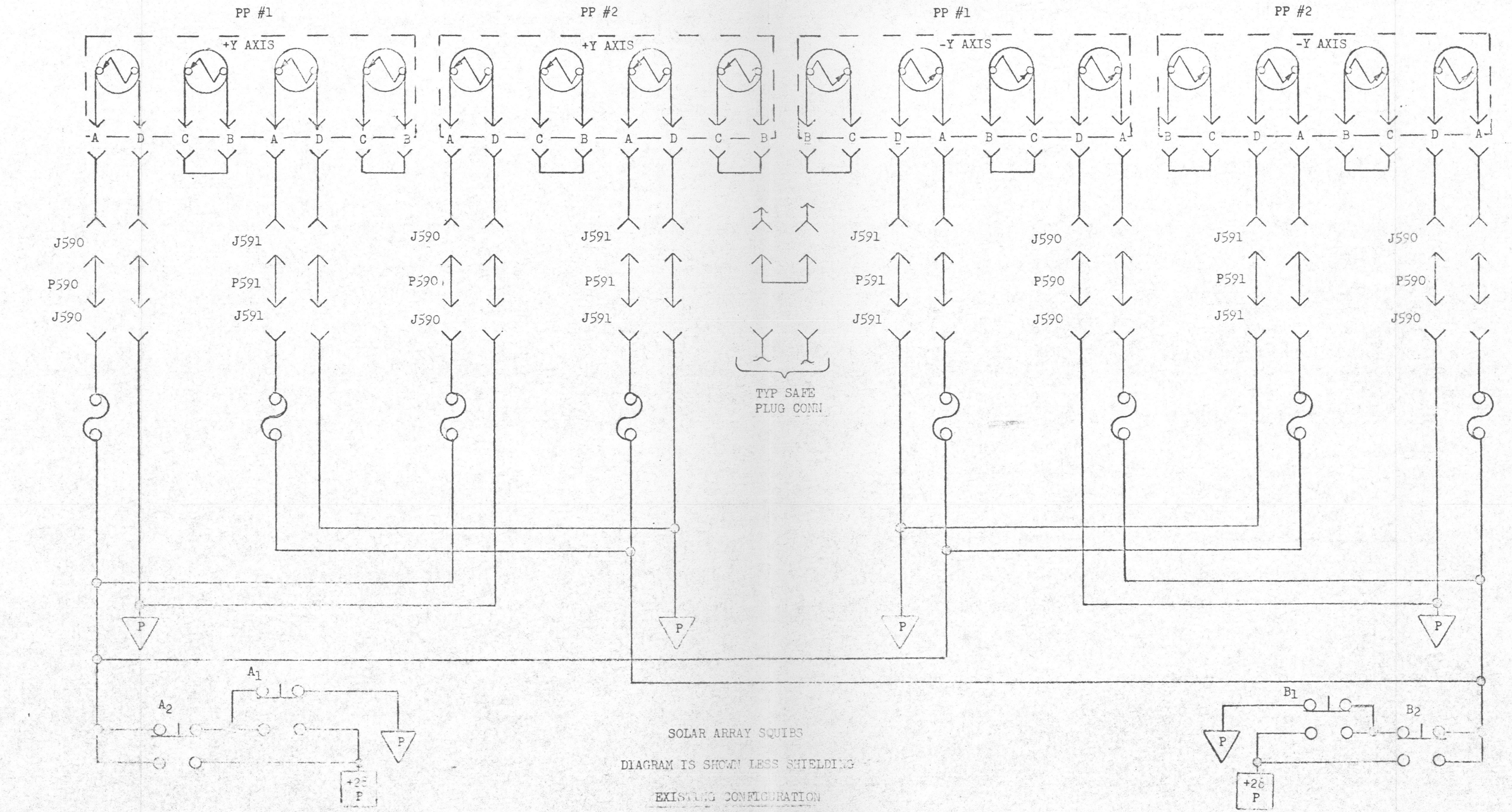
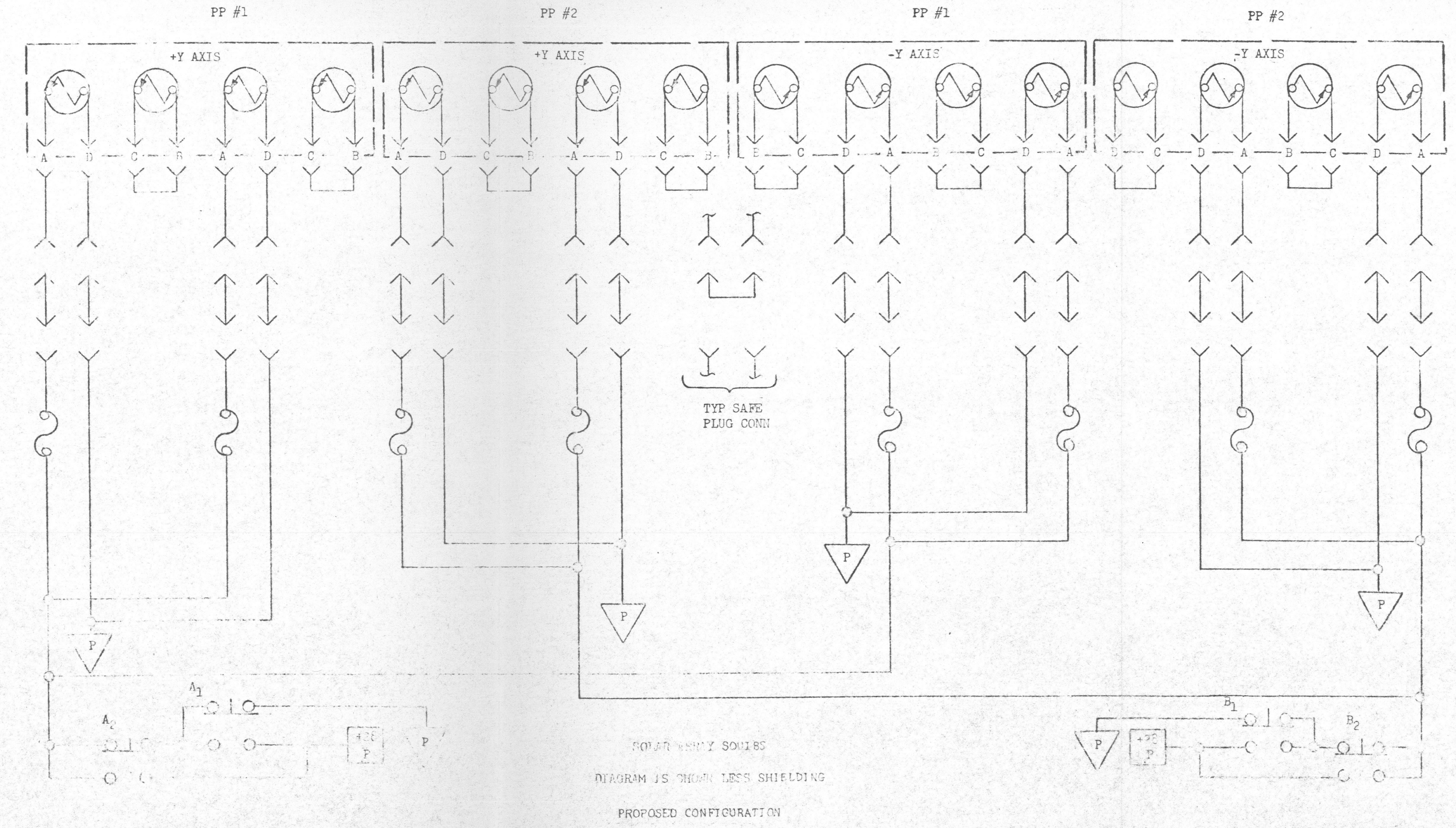
5.2.3 Instrumentation - Table 5-3 lists the instrumentation installed on the solar array modules. Figure 5-7 is a diagram of the solar array electrical system. The four-cell module assembly is shown in an artist's sketch as Figure 5-9. The unconditioned signals from the temperature monitors and the calibrated four-cell module assembly are transmitted through the SSU/Agena interface connector to the SSU telemetry subsystem. The signals from the solar array extension limit switches are transmitted through the Solar Array/Agena interface connectors to the Agena flight control J-Box and thence to the Agena telemetry subsystem.

5.3 THERMAL CONTROL DESIGN AND ANALYSIS

Thermal control of the SERT II solar array modules is achieved by passive means through the selective application of White Kemacryl lacquer (LAC 37-4033 Class I) and Mystic 7402 foil (LAC 24-4199). The application of these surfaces is specified by LMSC Drawing 1386328, Thermal Control Surfaces.

5.3.1 Ascent Analysis - Detailed analysis of the ascent and transfer orbit phase, based on the above design results in the following predictions:

- The maximum and minimum temperatures of the outer +Y solar panel are 69°F and 55°F respectively. The -Y solar array is in the shadow of the aft rack during the transfer orbit. It receives only earthshine fluxes in addition to exchanging energy with the array system and space. As a result, the temperature of the outer most -Y solar panel drops from 69°F at Agena separation to -47°F at Deployment. The maximum and minimum of each of the solar panels and the array actuator assemblies are given in Table 5-4.



SERT SOLAR ARRAY INSTRUMENTATION

Nomenclature	Module Location	Meas No.	Transducer	Range
Limit Switch Nos.				
1	+Y	AG32	MS24547-1	ON-OFF
2	-Y	AG33	MS24547-1	ON-OFF
Thermistor Nos.				
1	+Y	C75	} 1618702-5	
2	+Y	C77		
3	+Y	C79		
4	-Y	C74		
5	-Y	C76		
6	-Y	C78		
Control Cell Nos.				
1	+Y	C50	} 1464843 Cover Glass & 1462389-1 Solar Cells	567 MV *
2	+Y	C51		132 MA *
3 & 4	+Y	C52		128 MA *

* AT 25° C AMO

TABLE 5-3

IMSC/A9212210

FOUR-CELL MODULE ASSEMBLY

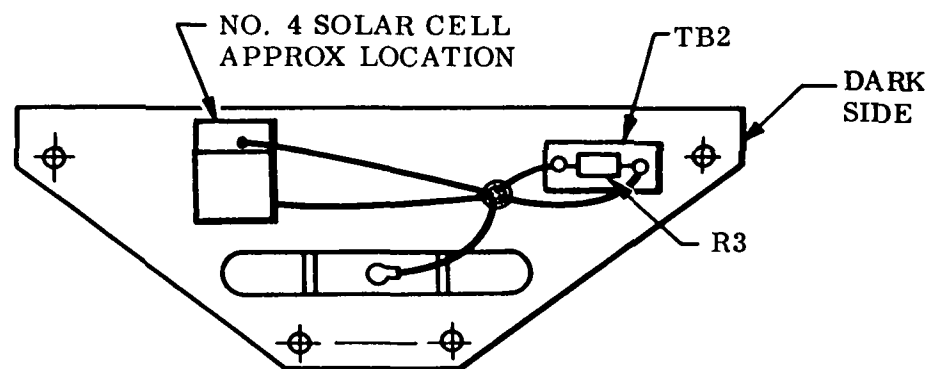
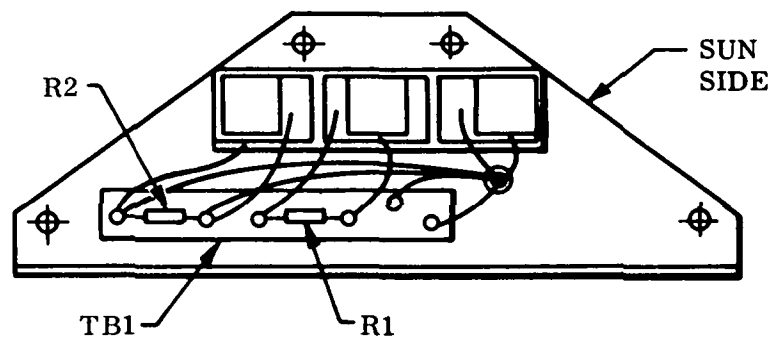


FIGURE 5-9

Component	T max °F	T min °F
+Y Solar Panel 1 (outside)	69.	56.
+Y Solar Panel 2	63.	60.
+Y Solar Panel 3	61.	60.
+Y Solar Panel 4	61.	60.
+Y Solar Panel 5	61.	60.
+Y Solar Panel 6	62.	60.
+Y Solar Panel 7	64.	60.
+Y Solar Panel 8 (Inside Half Panel)	66.	60.
-Y Solar Panel 1 (Outside)	69.	-47.
-Y Solar Panel 2	60.	15.
-Y Solar Panel 3	60.	39.
-Y Solar Panel 4	60.	52.
-Y Solar Panel 5	60.	58.
-Y Solar Panel 6	61.	60.
-Y Solar Panel 7	64.	60.
-Y Solar Panel 8 (Inside Half Panel)	66.	60.
+Y Actuator	75.	45.
-Y Actuator	75.	45.

TABLE 5-4

SERT II SOLAR ARRAY ASCENT ANALYSIS

MAXIMUM AND MINIMUM COMPONENT TEMPERATURES

5.3.2 Orbit Analysis - Temperature predictions were obtained by means of a thermal mathematical model of the solar arrays and the aft section of the Agena, from the engine nozzle extension to the propellant tanks. The solar arrays were broken up into a series of nodes based on their view factors to the Agena. Because of the magnitude of the view factors, it was necessary to have smaller nodes for sections of the solar arrays closest to the vehicle. The SERT solar array mathematical model node layout is shown in Figure 5-10.

For sunline incidence angles of zero and 16 degrees temperature predictions were obtained for a range of the value of the parameters that affect the solar arrays. The major conditions considered in the analysis were:

1. Altitude
2. Solar Constant
3. Efficiency of Solar Cells
4. Distance from Agena (Temperature gradient within the array)
5. Heat Transfer Normal to Solar Cell (Temperature gradient through cell)
6. Solar Array Frame (Temperature gradient within module)

The following summarizes the results obtained for the specified orbital conditions:

First in orbit, sunline incidence angle of zero.

Since it is expected that the vehicle will be launched in March, a maximum solar flux of 460 BTU/Hr-ft² may be incident on the solar array. This is based on the nominal value for the solar constant and the possible deviation in this solar constant. The temperatures for this condition are given below:

<u>Temperature Predictions (°F)</u>	<u>Solar Array Efficiency</u>
146°F	= 0
132°F	= 0.1

SERT II SOLAR ARRAY THERMAL NODE LAYOUT

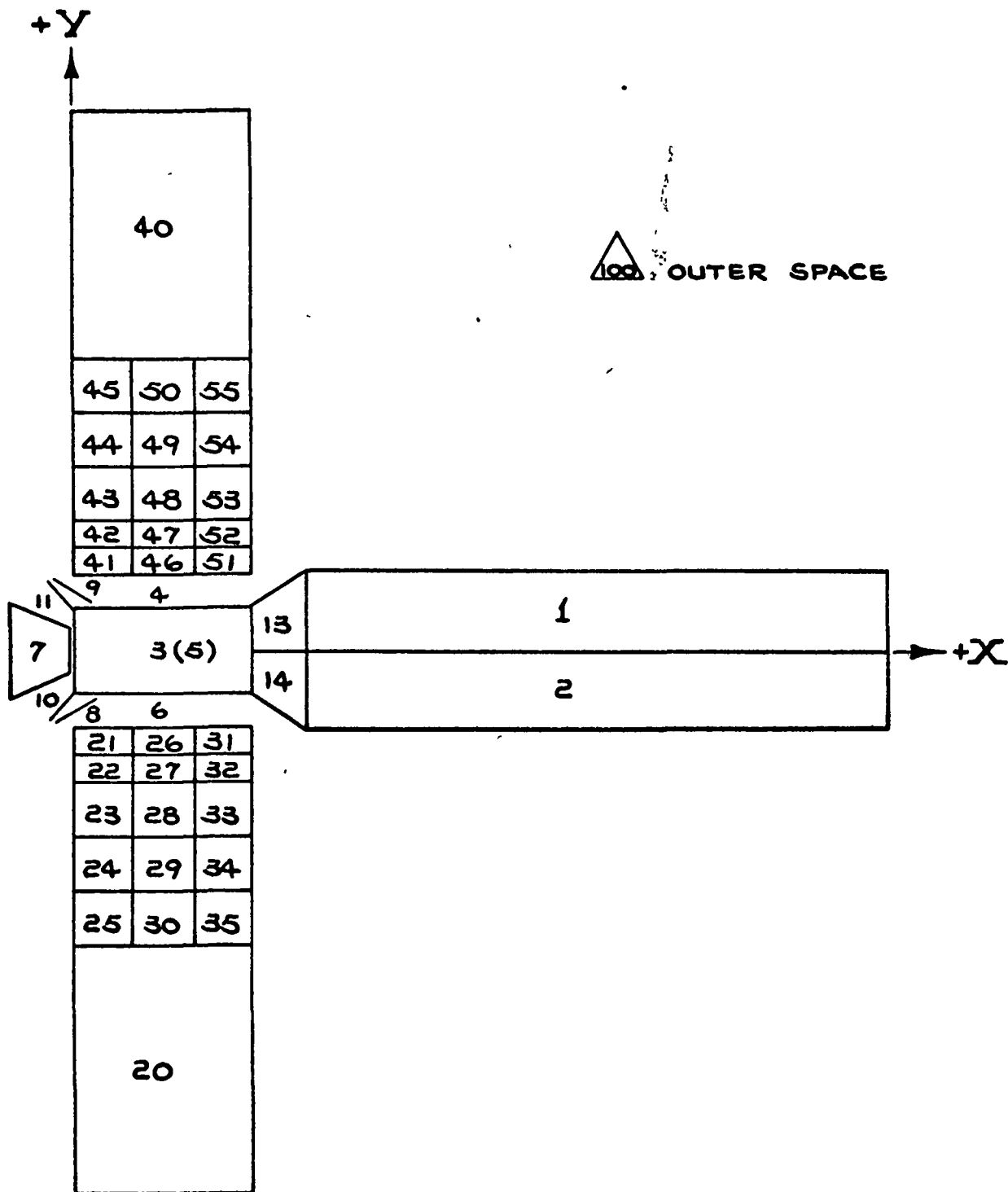


FIGURE 5-10

The values are based on the optical properties of the solar cells determined from the analysis of data from previous military program flights. This analysis indicated optical properties of $\alpha_s = 0.8$, $\epsilon = 0.8$ for the solar cells. These predictions are the maximum estimated temperatures for the solar array and include uncertainties. It is estimated that due to uncertainties there may be a maximum deviation of $\pm 12^\circ\text{F}$ from the nominal prediction. If the prediction for the $\eta = 0$ case is expressed on a nominal basis, it is $134^\circ\text{F} \pm 12^\circ\text{F}$.

Six months have elapsed sunline incidence angle of 16° .

The temperature predictions for this condition were based on a solar constant of $443 \frac{\text{BTU}}{\text{Hr} \cdot \text{ft}^2}$. This is based on the maximum solar constant for the month of September. The temperature predictions for a solar array efficiency of .10 and 0 are shown in Figure 5-11. Even though the solar vector remains constant on the arrays (since the solar array is in the plane of the orbit), there is a variation in the temperature predictions due to the change in albedo as a function orbit position angle.

These are maximum temperatures and include uncertainties. The minimum temperatures can be obtained by deducting 24°F from any point on either the $\eta = .10$ or $\eta = 0$ curve.

Six months have elapsed, 18 minute eclipse, sunline incidence angle of zero.

The temperature predictions for this condition was based on a solar constant $\dot{q}_s = 443 \frac{\text{BTU}}{\text{Hr} \cdot \text{ft}^2}$ as in the immediately preceding analysis. The time-temperature curves for efficiencies of zero and .10 are shown on Figure 5-12. The orbit position angle at which the vehicle is eclipsed for 18 minutes is arbitrary since no condition physically exists in which the vehicle will be eclipsed at a sunline incidence angle of zero.

As previously stated in this report, the present thermal control design of the SERT II solar array modules is based on analysis of an assumed Spring launch. IMSC is in receipt of Change Order 12 to Contract NAS 3-11512 which specifies a Fall launch date. Analysis of the new ascent and orbit conditions will be required, and, redesign effort is anticipated to insure optimum thermal control design.

SOLAR ARRAY TEMPERATURE PREDICTIONS

Sunline Incidence Angle of 16°
 $= 74^\circ$, $h = 621$ miles, $\dot{q}_s = 443$ BTU/ht-ft²

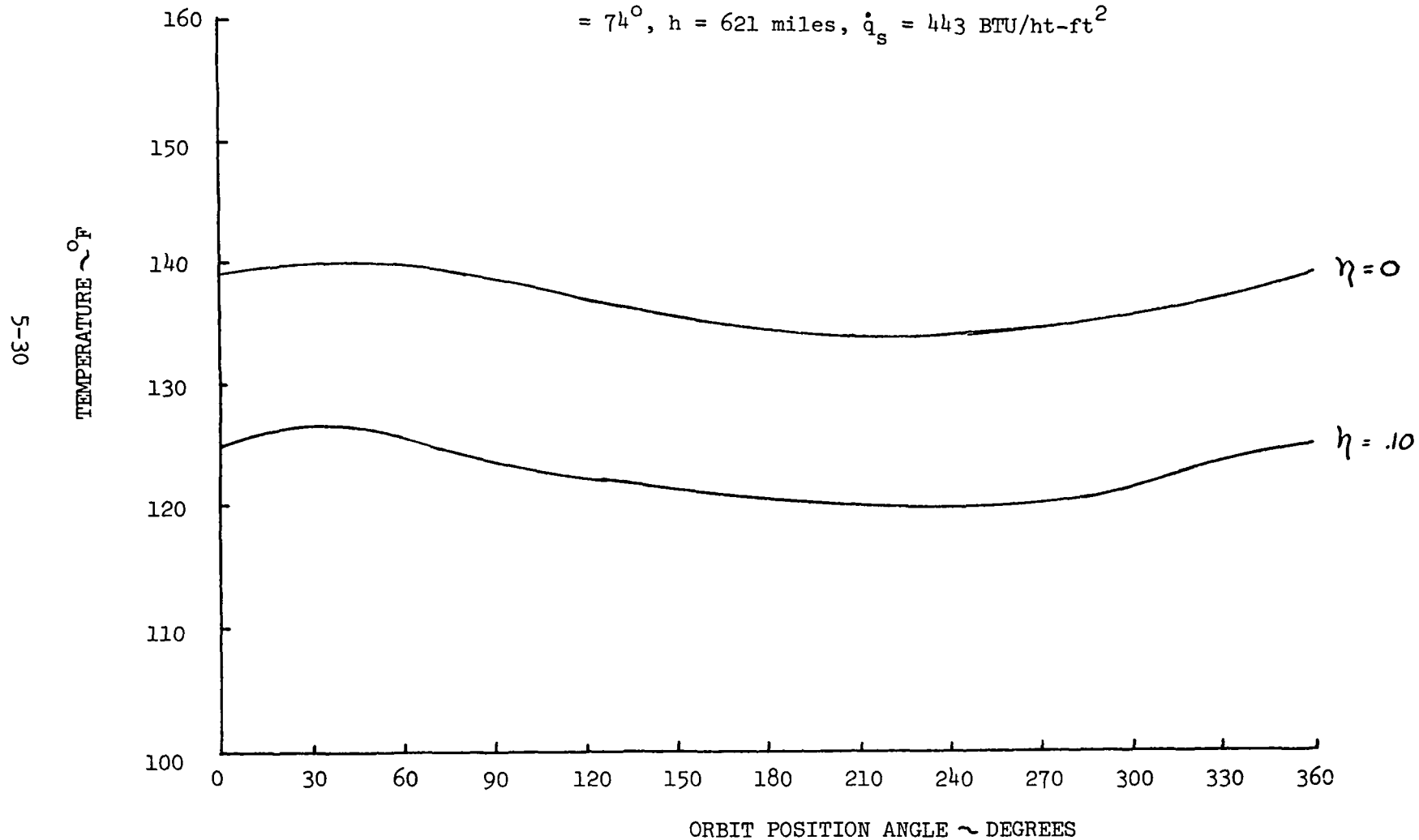


FIGURE 5-11

LMSC/A941440

ANALYSIS RESULTS

18-minute eclipse
Sunline incidence angle of zero
 $h = 621$. statute miles
 $\dot{q}_s = 443$. BTU/hr-ft²

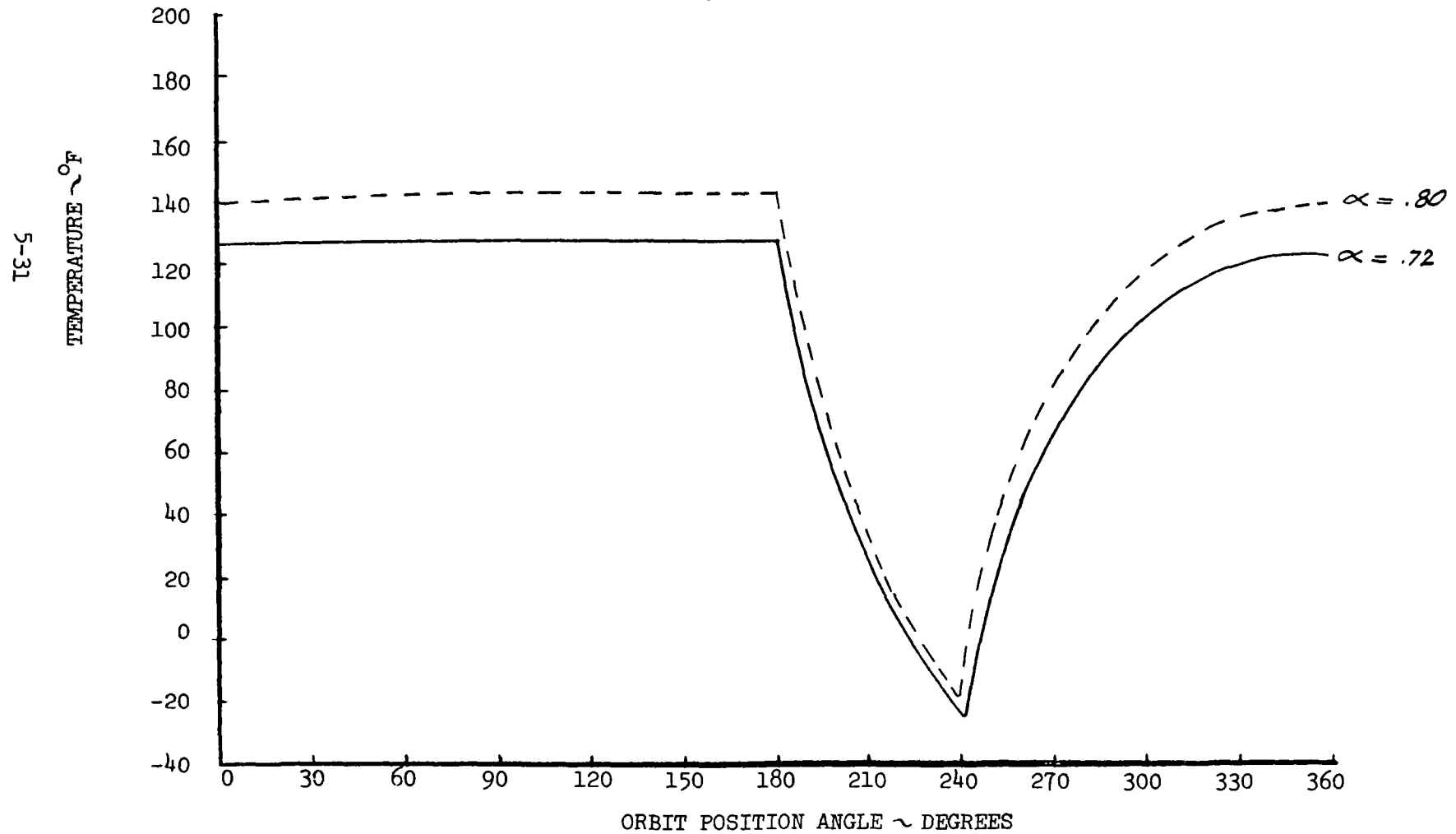


FIGURE 5-12

5.3.3 Solar Array Release Mechanism Analysis - In addition to the above analysis, a separate thermal analysis of the revised SERT II solar array release mechanism was performed to verify the adequacy of the thermal design. The detailed thermal analysis has demonstrated that the thermal design is effective in minimizing temperature excursion in the tie rods, pin pullers and clamps. The increase in tie rod tension due to temperature differentials, as described in Paragraph 5.1.5, was minimized by the use of thermal reflective tape. The tape was applied to the solar array module which is on the shade side during ascent.

SECTION 6.0

TEST PROGRAM

6.1 TEST PHILOSOPHY

The SERT II Solar Array test program was based upon the premise that the military program solar array from which the SERT II solar array design was derived was to be provided as a flight-proven "off-the-shelf" array to be modified as necessary to meet the SERT II mission requirements. The Air Force Program which developed this array subjected it to rigorous qualification tests and certified it as qualified. Further, the basic "off-the-shelf" solar array has been proven by a series of successful flights to be flight worthy.

It was agreed that the SERT II solar array test program would be restricted to those tests required to assure that the flight worthiness of the basic design had not been invalidated by modification of the design to meet SERT II mission requirements.

The SERT II Solar Array Module test program consisted of the following tests:

- Engineering Evaluation Tests
- Release Mechanism Evaluation Tests
- Vibration Confidence Tests
- Flight Acceptance Tests

6.2 ENGINEERING EVALUATION TESTS

The objective of this test program was to verify that the SERT II Solar Array mechanical deployment assembly would perform the deployment as designed. These tests were not intended to be "qualification level tests" to prove the original design integrity. They were only intended to provide confidence that the modifications to the qualified equipment had not invalidated the flight worthiness of the solar array system.

This test program was limited to performance of ten solar array deployments.

6.2.1 Test Specimen Design and Construction - The design and construction of the test specimen was in accordance with IMSC Drawing 1386798, SERT II Solar Array Evaluation and Confidence Test Module.

6.2.2 Test Set Up - The deployment tests were conducted, under ambient test laboratory temperature and pressure conditions, using a test fixture which simulated deployment of the solar array assembly in a zero gravity environment. The array arm was supported at the center of each leaf by low friction roller assemblies that rode in an overhead track.

Each roller assembly was adjusted to support only the weight of the leaf to which it was attached. The leaves were allowed to both rotate and translate along the overhead track. The mounting bed was supported by a pedestal and counterbalanced about the support point to minimize the loads imposed on the array assembly. While the array arm was extending, the mounting bed was allowed to orient itself freely to its prealigned position. After assembly of the solar array module in the test fixture, the mounting bed assembly was aligned perpendicular to the extended solar array arm. Extension of the array was initiated by remotely operating the pin pullers with a pneumatic pin puller system.

6.2.3 Instrumentation - The actuator assembly, P/N 1368196-501, was instrumented with strain gages which provided continuous readout of the load being applied to the deployment mechanism arm assembly, P/N 1386726-501, throughout the entire deployment cycle. Strip chart recordings were made of this instrumentation readout.

6.2.4 Photographic Coverage - High speed motion picture coverage was provided during the entire deployment cycle to permit analysis of displacements, velocities, and accelerations of the panel tip motion and the angular motion of the inboard solar array panel.

Still camera coverage was provided to document the overall test setup and to record possible test anomalies.

6.2.5 Solar Array Flatness Measurements - Panel flatness measurements were obtained at the completion of each array extension. Angularity measurements were taken at the hinge joints between panels 7 and 8, 5 and 6, and panels 1 and 2. The array panels are numbered outward from the mounting bed and erection mechanism assembly interface.

6.2.6 Summary of Test Results - A summary of the results obtained during the ten deployments is presented in Table 6-1. Excellent repeatability of deployment times and panel flatness measurements was demonstrated for each actuator linkage configuration tested. The actuator linkage configurations listed in Table 6-1 are described in Figure 6-1. Deployments 1 through 5 were conducted with the actuator linkage configuration A-C. Average deployment time for this configuration was 53.6 seconds. Tests 6 through 8 utilized actuator linkage configuration A-D. Deployment time averaged 50.58 seconds. The last two deployments were conducted with actuator linkage A-E configuration. Average deployment time for this configuration was 47.85 seconds. Actuator linkage configuration A-C was selected as the optimum flight hardware design.

The maximum panel flatness misalignment recorded was one degree. This measurement was recorded after the first deployment. The misalignment was less than one degree on all subsequent deployments.

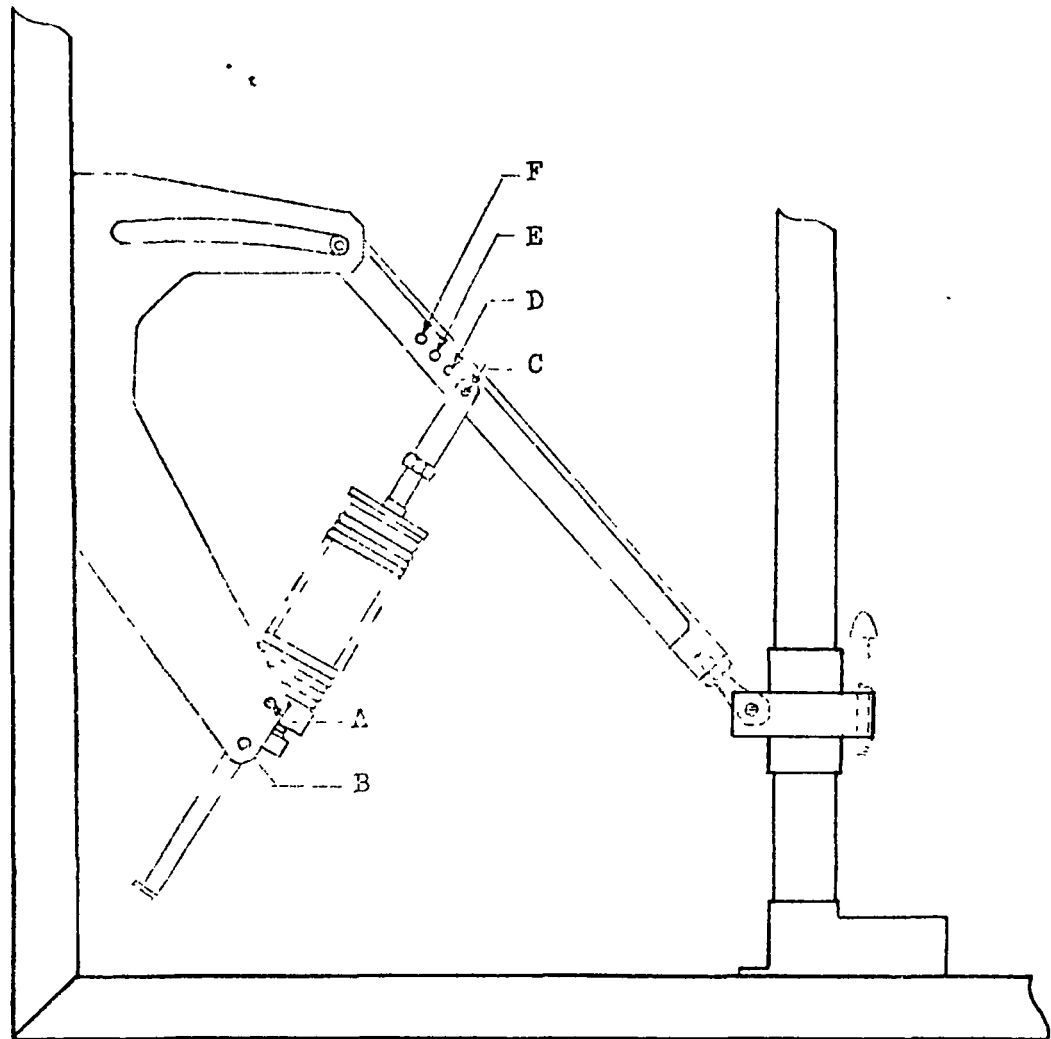
A detailed analysis of the film data was presented in LMSC/A925371, Test Report, SERT II Solar Array Engineering Evaluation Test, dated 19 April 1968.

Based upon the test data analysis it was considered that the deployment system as tested was adequate to perform the desired array deployment function. However, analysis indicated that, under worst-on-worst conditions involving weakest allowable spring forces and maximum array hinge and wire bundle resistance, the array may fail to fully open the last 5 degrees of travel.

TEST NO.	DEPLOYMENT TIME IN SECONDS	PANEL FLATNESS MEASUREMENTS	ACTUATOR LINKAGE CONFIGURATION
1	54	7 & 8 179.0° 5 & 6 180.0° 1 & 2 179.0°	A to C
2	54	7 & 8 180.0° 5 & 6 179.7° 1 & 2 179.8°	A to C
3	54	7 & 8 179.7° 5 & 6 179.8° 1 & 2 179.2°	A to C
4	53.5	7 & 8 180.0° 5 & 6 179.7° 1 & 2 179.4°	A to C
5	53	7 & 8 180.0° 5 & 6 179.7° 1 & 2 179.4°	A to C
6	52	7 & 8 180.0° 5 & 6 179.8° 1 & 2 179.4°	A to D
7	50	7 & 8 180.0° 5 & 6 180.0° 1 & 2 180.0°	A to D
8	49.75	7 & 8 180.0° 5 & 6 180.0° 1 & 2 179.8°	A to D
9	48.9	7 & 8 180.0° 5 & 6 179.3° 1 & 2 179.3°	A to E
10	46.8	7 & 8 180.0° 5 & 6 179.3° 1 & 2 179.4°	A to E

SOLAR ARRAY DEPLOYMENT TEST SUMMARY

TABLE 6-1



ACTUATOR IN PLACE ADJUSTED TO

FIGURE 6-1

IMSC was directed to replace the basic actuator spring with a spring having a greater force constant. The change was based upon the results of an analysis performed by the IMSC Dynamics Group which indicated that the larger spring would provide added confidence of full deployment without requiring redesign of other hardware.

The deployments performed as a part of the confidence tests, along with the dynamic analyses performed after the tests, as well as the deployments performed as a part of the acceptance testing of the flight hardware, proves that the new spring in the SERT II Solar Array application is satisfactory.

6.3 SOLAR ARRAY RELEASE MECHANISM ENGINEERING EVALUATION TEST

The primary objective of this test program was to demonstrate that the redesigned release mechanism would function properly under simulated failure conditions which would require the array to deploy with either pin puller being actuated individually with only a single squib firing. Additional objectives were to demonstrate that the release mechanism components would not physically interfere with the array deployment, and, that the pyrotechnic shock caused by the new pin puller configuration would not significantly degrade the power output capability of the solar collector assemblies.

This test program was limited to 1 complete pneumatically activated solar array deployment, electrical continuity, resistance, and power output tests, and to pyrotechnically initiated tests of the redesigned pin puller-tie rod release mechanism assembly.

6.3.1 Test Set Up - Three separate and distinct test set ups were required for this test series. They were as follows:

- Deployment and Power Output Tests - The deployment test was conducted under ambient test laboratory conditions using the same deployment test fixture described in Paragraph 6.2 above. The tie rod assembly was instrumented with strain gages to

facilitate monitoring tie rod strain during the tightening of the clamp set screws during stowing operations. High speed motion picture coverage was provided during the entire deployment cycle to permit analysis of displacements, velocities, and accelerations of the panel tip motion and the angular motion of the inboard solar array panel.

Power output tests of solar collector assembly under 1685 were conducted using a light source while the test array was fully extended in the deployment fixture.

- Release Mechanism Evaluation Tests - The test specimen was mounted on an Agena aft rack structure. The aft rack was in turn mounted on a heavy load plate. A cable/bungee suspension system was used to simulate a zero gravity condition on the panels. The zero G simulation system was attached to a beam approximately 20 feet above the floor.

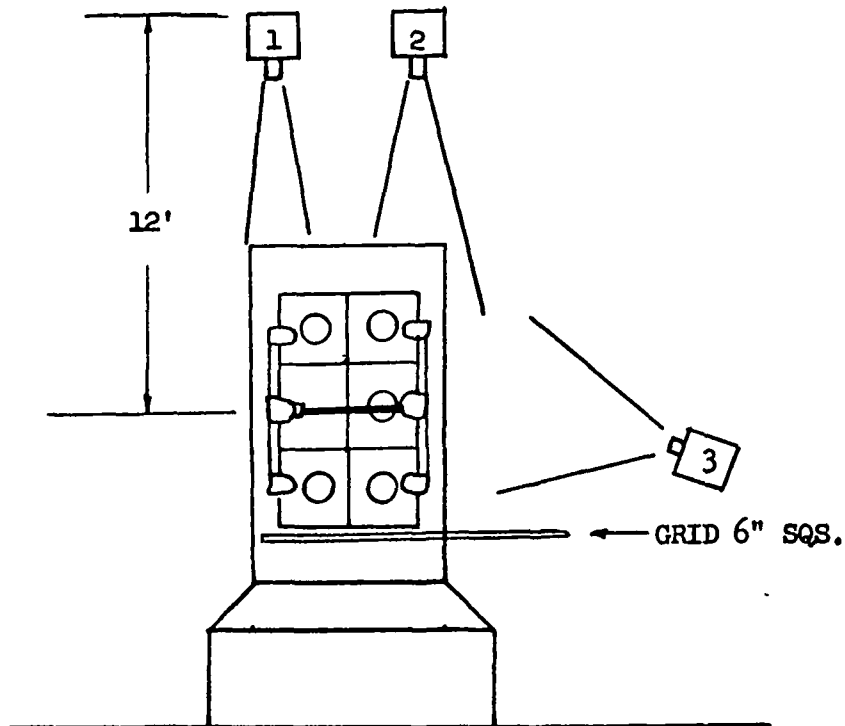
The array was restrained during each release mechanism test to restrict movement of the outboard panel to a maximum of 24 inches.

The release mechanism was actuated by firing live squibs in flight type pin pullers.

Primary instrumentation consisted of three high speed motion picture cameras, two ENDEVCO 2225 crystal accelerometers, and two calibrated strain gages. Secondary instrumentation consisted of five current shunt bars for measuring squib current, and a voltage monitor to record pyro power supply voltage.

Camera location and camera data is furnished in Figure 6-2.

The two ENDEVCO accelerometers were used to measure the magnitude of the pyrotechnic shock near the center of a functional collector panel, and at a point on the supporting collector assembly frame near the pin puller location.



CAMERA DATA

500 FPS-MITCHELL-HS-16

120° SHUTTER

PLUS X NEGATIVE FILM

#1 - 32 MM LENS - VIEW LATCH

#2 - 25 MM LENS - VIEW TIE ROD

#3 - 32 MM LENS - OVERALL VIEW

IRIG TIMING

FILM LOG NO. OS3790

FIGURE 6-2

CAMERA POSITION FOR DATA RELEASE MECHANISM TEST

6.3.2 Summary of Test Results - The SERT II Solar Array Release Mechanism Engineering Evaluation Test Program was successfully completed on 11 September 1968. The test program consisted of the following tests:

- Electrical continuity, resistance, and high potential tests, performed before and subsequent to the release mechanism tests.
- Six pyrotechnically actuated release mechanism tests with the solar array restrained to limit maximum deployment to 24 inches.
- Array circuit continuity and resistance tests prior to and after each release mechanism test.
- Squib circuit and fusistor resistance measurements before and after each release mechanism test.
- One full deployment test using pneumatically actuated pin pullers.
- Solar Collector Assembly Power Output Tests accomplished before and after the Release Mechanism Tests.
- Three solar collector assembly power output tests conducted at the JPL Table Mountain Sunlight Test Facility.

Analysis indicates that all electrical test data obtained during the continuity, resistance, high potential, and power output tests were within acceptable limits.

Evaluation of accelerometer measurements made in the course of operation of the solar array release mechanism in various modes indicates that the use of two pin pullers instead of one does not cause any significant increase in shock level on the solar array structure or collector panels.

Six modes of operation of the release mechanism were successfully tested. These were:

- Test 1 - Two pin pullers, four squibs. Array restrained so that extension was prevented.

- Test 2 - One pin puller, two squibs, followed 100 milliseconds later by the remaining pin puller with two squibs.
- Test 3 - Number 1 pin puller with two squibs. Number 2 pin puller with dummy squibs and not actuated.
- Test 4 - Number 2 pin puller with two squibs. Number 1 pin puller with dummy squibs and not activated.
- Test 5 - Number 1 pin puller only with one squib active.
- Test 6 - Number 2 pin puller only with one squib active.

The data films from the SERT II release mechanism tests have been reduced to yield time histories of the motion of the torque tube clamps and solar array panels. A deflection curve for the tie rod at the time its clamp contacted the stop pad was also determined.

Although strain energy stored in the solar array by the torquing procedure caused sizable initial angular rates in the torque tube clamps, analysis indicates that no interference problems exist between the rebounding clamps and deploying solar panels even if no friction is present in the clamp assemblies. Additionally, the predicted bending moment at the tie rod base for a frictionless clamp assembly is lower than the bending moment required to yield the tie rod.

Only a small portion of the energy input to the solar array during the torquing procedure appears as kinetic energy of the solar panels. Consequently, no significant decrease in predicted deployment times will result.

A more detailed analysis of the results of this test program was presented in IMSC/A937479, SERT II Release Mechanism Engineering Evaluation Test, dated 11 October 1968.

6.3.3 Conclusions - The primary objective of this test program was to demonstrate that the redesigned solar array release mechanism would function as designed. Emphasis was placed upon testing of various squib-pin puller failure modes. Specifically, LeRC direction required demonstration of satisfactory single pin puller operation with either pin puller being actuated by a single squib.

The second test objective was to demonstrate that the pyrotechnic shock and pin puller operation would not significantly degrade the power output capability of the solar collector assemblies.

The third objective of this test program was to demonstrate that there would be no mechanical interference between the components of the release mechanism system and the deploying solar array assembly.

All of the above objectives were satisfactorily met. LMSC considers the solar array release mechanism system, as presently designed, qualified to support LeRC specified SERT II mission requirements.

6.4 SOLAR ARRAY MODULE CONFIDENCE TEST

The objective of this test program was to verify that the solar array assembly will withstand the higher dynamic loading anticipated from the Thorad launch vehicle.

This test program was limited to the following:

- Vibration tests of the solar array module in the X, Y, and Z axes.
- Electrical continuity and harness resistance measurements.
- Deployment tests of the solar array module performed before and after vibration tests.
- Power output tests of Solar Collector Assembly, Serial 1657, conducted before and after vibration tests.
- Sunlight testing of Solar Collector Assembly, Serial 1657, conducted at the JPL Table Mountain Test Facility, Wrightwood, California.

The design and construction of the test specimen was in accordance with LMSC Drawing 1386798, SERT II Solar Array Evaluation and Confidence Test Module, except that pneumatically actuated pin pullers were used to initiate deployment of the array.

6.4.1 Test Set Up -

- Deployment Tests - The deployment test was conducted, under ambient test laboratory temperature and pressure conditions, using a test fixture which simulated deployment of the solar array assembly in a zero gravity environment as described in Paragraph 6.2 above.

The tie rod assembly was instrumented with strain gages to facilitate torquing the clamp assemblies to specified values during solar array stowing operations. The output of the two calibrated strain gages was continuously monitored with a BIH SR4, Model 120, Strain Indicator and a BIH PSBA-20, Model 3, Balance Unit during the torquing operation.

Electrical continuity and insulation resistance measurements were obtained using a Solar Array Test Console, Part Number 1412610-17H, and appropriate interconnect adapter cables and test leads.

Power output tests were conducted with the array fully deployed in the extension fixture and the Light Source Assembly, Part Number 1365446-17H, positioned 5 ± 0.25 inches from the active collector panel.

Solar array deployment times were derived from analysis of film data.

High speed motion picture coverage was provided during the entire deployment cycle to permit analysis of displacements, velocities, and accelerations of the panel tip motion, and the angular motion of the inboard solar array panel.

- Vibration Tests - A 15 thousand pound force shaker was used for all vibration tests. For the X-axis and Z-axis vibration tests the test specimen was mounted on a heavy steel table which rested in a horizontal position on 7 Team tables. The armature of the shaker was rotated to the horizontal.

For the Y-axis vibration test the specimen was mounted on an aluminum table which in turn was attached to the shaker armature. The armature was rotated to the vertical position for this portion of the test.

The solar array test console described above was moved from the Solar Array Test Facility and used to measure insulation resistance subsequent to vibration testing in the X and Y axes.

Twenty-three ENDEVCO accelerometers were mounted on the solar array structure. Three control accelerometers, Numbers 1, 2, and 3, were mounted to the attachment fixture.

Two calibrated strain gages were bonded to the small end of the tie rod, 180° apart, to facilitate torquing the clamp assemblies.

Still camera coverage was provided to document the overall test setup and to record possible test anomalies.

The setup for the Table Mountain panel power output tests is shown in Figure 6.3.

6.4.2 Summary of Test Results - The results of the electrical continuity, resistance, and power output tests indicate that all parameters measured were within acceptable limits.

The data films from the pre- and post-vibrational Solar Array Deployment tests have been reduced to yield time histories of the motion of the solar array tip. Analytical time histories of array tip motion for the minimum and maximum time deployments expected under test conditions have also been determined. The test and analytical time histories are in close agreement throughout extension of the solar array. Both the pre- and post-vibrational deployment times were within the range of times predicted by analysis.

The results of the pre-vibration and post-vibration solar collector assembly panel power output tests are summarized in Table 6-2. Due to inclement weather at the Table Mountain test site over an extended period of time, it was necessary to conduct the post-vibration test using a calibrated Standard Sunlight Simulator.

Comparison of the pre-vibration and post-vibration test results indicates that no significant decrease in panel power output capability resulted from the vibration testing.

TEST	PANEL NUMBER	Voc	Isc	OUTPUT	
				VOLTS	MA
Pre-Vibration (Sun Simulator)	1657	43.1	680	34.0	621
Pre-Vibration Sunlight	1657	43.2	683	34.0	623
Post-Vibration Sun Simulator	1657	43.3	667	34.0	596

TABLE 6-2
POWER OUTPUT TEST RESULTS

The test data did not reveal any unusual activity of the structure during vibration test and the solar array structure and the active panel showed no signs of functional or structural degradation subsequent to the confidence tests.

The solar array has, therefore, demonstrated capability to withstand vibration environment in excess of the expected flight environment.

A more detailed presentation of the Solar Array Module Confidence Test set up description and subsequent test data analysis was presented in LMSC/A938941, SERT II Solar Array Module Confidence Test, dated 25 October 1968. A summary of the results of the Solar Collector power output tests has been included in this report since the test was not completed in time to be included in the test report cited above.

6.5 FLIGHT ACCEPTANCE TESTS

The SERT II flight solar array modules were tested in accordance with LMSC Specification 1420657. The acceptance test program consisted of the following:

- Inspection of all hardware for satisfactory compliance with engineering drawings.
- Harness resistance and electrical continuity measurements.
- Extension limit switch electrical and mechanical function tests.
- Thermistor probe continuity tests.
- Solar collector panel GO-NO-GO power output tests.
- Mechanical push-pull force measurements of the solar array assembly.
- Solar array assembly to mounting bed assembly alignment verification.

- Hold down clamp and tie rod release mechanism operation under pin puller simulated failure conditions.
- Array deployment tests under simulated zero gravity conditions.

The test set up for this test program was essentially the same as described in Paragraph 6.2.

6.5.1 Summary of Acceptance Test Results Data Recording - The test data covering the solar array module electrical acceptance tests are recorded on the data sheets accompanying LMSC Test Procedure MTP 30548. The data covering the mechanical alignment and deployment tests are recorded on the data sheets accompanying LMSC Test Procedure MTP 30549. Additional data for solar array deployment analysis are recorded on film number OS 3799. All data indicated above are included with the DD-250 Data Package.

6.5.2 Hardware Review - A review of all hardware and wiring installation indicated satisfactory compliance with the engineering drawings except that application of thermal reflective tape had not been accomplished. The launch date slip, under Contract Change Order Number 12 to NAS 3-11512 contract will alter this requirement. Therefore, completion of this item is being held in abeyance pending engineering documentation revision.

6.5.3 Electrical Tests and Alignments - The test data recorded in MTP-30548 and MTP 30549 have been reviewed. All acceptance criteria have been met.

6.5.4 Solar Array Deployment Tests - The data films from the deployment tests on the 1365451-505 and 1365451-507 SERT II solar array modules were reduced to yield time histories of the motion of the solar array tip. Analytical time histories of array tip motion for the minimum- and maximum-time deployments have also been determined. The deployment times for the flight solar array modules, as determined by testing, are all within the range of deployment times predicted by analysis.

LMSC/A941440

The deployment times recorded during testing of the -507 solar array module were 33.3, 33.1, and 30.0 seconds. The deployment times recorded during testing of the -505 solar array module were 33.0, 31.6, and 32.4 seconds. The deployment times are all within the range of 29.6 to 40.3 seconds as predicted to occur by analytical methods.

SECTION 7.0

CONCLUSION

A thorough review of the designs, hardware, test program, test data, engineering analysis, and Product Assurance assessments of the SERT II Solar Array has been completed. The review concludes that the SERT II Solar Array meets or exceeds all requirements of the Statement of Work, Contract NAS 3-11512 with the exception of the predicted on orbit open circuit voltage maximums.

The present predicted maximum open circuit voltages for a March launch are 83.1 volts and 41.5 volts respectively for the Thrustor and House-keeping sections. The present maximum permissible open circuit voltages are 75.0 volts and 35.0 volts.

LMSC has submitted a letter to LeRC requesting that the maximum permissible open circuit voltages be increased to 85.0 volts for the Thrustor Section and 43.0 volts for the Housekeeping section. Action is pending on this request at the time of publication of this report.

SECTION 8.0

ADDENDUM

The summary reports on the following pages 8.2, 8.7 and 8.9 cover the results of solar array module modifications and testing activities occurring subsequent to the scheduled issue date of the final report.

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LMSC/A941440

In reply refer to:
LMSC/1958407
69-20/538

10 September 1969

Mr. C. Siegert, Project Manager (6 cys)
SENT II Solar Array - Mail Stop 54-3
NASA Lewis Research Center
21000 Brookpark Road
Cleveland, Ohio 44135

Subject: Summary Test Report on Array Module Modification, Shade
Shield Installation, and Thermal Taping, SENT II Solar
Array Contract NAS 3-11512

Reference: (a) Contract Amendment No. 14 to NAS 3-11512, 23 May 1969
(b) Contract Amendment No. 12 to NAS 3-11512, 8 Nov. 1968

Dear Mr. Siegert:

The purpose of this summary report is to describe modifications of the SENT II Solar Array Assembly accomplished in compliance with References (a) and (b), and to summarize the results of the post modification tests in accordance with paragraph 3.3, Exhibit A of subject contract. Accordingly, each module was removed from the shipping container and the array assembly removed from the mounting bed. The solar array assemblies were modified by shorting out six submodules in each of the thirty-eight solar collector panels which comprise the thruster section of each assembly.

A sun shade-shield for the short circuit current test cell on the four cell module assembly was designed, fabricated, and installed.

Thermal tape was installed on the 1365451-507 module in lieu of the -505 module as a result of the launch date change from a spring to a fall launch.

Engineering drawings and the solar array module assembly specification were changed to specify modification and retest requirements. New dash numbers were created for the Solar Array Module Assembly Drawing. 1365451-505 was changed to -509 and 1365451-507 was changed to -511.

Following modification, electrical and mechanical tests were successfully accomplished to verify conformance with the revised specification and drawing requirements.

Each module was reassembled, stowed with clamp assemblies torqued to flight values, electrically tested for circuit resistance and continuity, returned to their respective shipping containers, and placed in temporary storage.

Description of Solar Collector Assembly Modification and Tests

After removal from the mounting bed, each solar array assembly was installed in the vertical holding fixture and the specified submodules prepared for soldering by removing thermal paint and cleaning areas where jumper wires were to be soldered. The silicon thermal paint was removed by etching with Lonco Number 504 Cream Wire Stripper. After paint removal, the areas to be soldered were thoroughly cleaned with Isopropyl alcohol. Shorting was accomplished by soldering two short jumper wires to each submodule. Continuity tests were then performed to verify shorting. The solder joints and adjacent areas were inspected, cleaned with Isopropyl alcohol, and the reworked areas repainted with thermal silicon paint.

Electrical insulation resistance, continuity, and illumination tests were performed on each solar array assembly after completion of modification of submodules. Insulation resistance and continuity measurements, in all cases, met specification requirements of 50 megohms minimum for insulation resistance and 0.85 ohms maximum for continuity.

Illumination tests were conducted to determine short circuit current, output load, and open circuit voltage. All short circuit current measurements met or exceeded the minimum test requirement of 450 mA. All output load measurements met the minimum test limit of 375 mA at 30 vdc. Open circuit voltage measurements are summarized in Tables 1 and 2. The test results show an average reduction of approximately 3 Volts per collector assembly as a result of the submodule modification.

Insulation resistance and continuity tests were repeated for each module after they had been reassembled, stowed, and torqued to flight values. All resistance and continuity measurements conformed to specification requirements.

Sun Shade-Shield Installation

The design drawing for the Standard Four Cell Module Assembly, LMSC Drawing 1390722, was revised to define the design, installation and alignment requirements for a shade-shield for the short-circuit current test cell. Alignment requirements were as follows:

The upper edge of the shade-shield shall have a minimum vertical height, above the active surface of the short circuit current control cell, of .280 inches and a horizontal distance, from the edge of the active surface, of .000 inches $\pm$.005 inch.

Post installation measurements confirmed that the shade-shield alignment requirements were met.

Thermal Tape Installation

Thermal tape was installed on the 1365451-507 (redesignated -511) module only, in compliance with Reference (b). Application of the thermal tape, to support a full

Table 1

SOLAR COLLECTOR PANEL ILLUMINATION TEST RESULTS
(-Y Module)

Panel Number	Open Circuit Voltage		Panel Number	Open Circuit Voltage	
	** -505	-509		** -505	-509
45	43.0	40.0	26	42.0	40.0
42	41.0	40.0	29	42.0	40.0
39	43.0	40.0	32	42.0	40.0
36	42.0	40.0	35	42.0	39.0
33	42.0	40.0	38	42.0	40.0
30	42.0	40.0	41	42.0	40.0
27	42.0	40.0	44	42.0	40.0
24	41.0	40.0	43	42.0	40.0
21	42.0	39.0	40	42.0	40.0
18	42.0	40.0	37	42.0	40.0
15	42.0	40.0	34	42.0	40.0
12	42.0	40.0	31	42.0	40.0
9	41.0	40.0	28	42.0	40.0
6	42.0	40.5	25	42.0	40.0
3	42.0	40.0	22	42.0	40.0
* 2	42.0	44.0	19	42.0	40.0
* 5	42.0	43.0	16	42.0	40.0
* 8	42.0	41.0	13	42.0	40.0
* 11	41.0	43.0	10	42.0	40.0
* 14	42.0	44.0	7	42.0	40.0
* 17	42.0	44.0	4	42.0	40.0
* 20	42.0	43.5	1	42.0	40.0
23	42.0	40.0			

* These panels were not modified

** Column under -505 lists pre-modification V_{oc} test results

-509 column shows post-modification test results

Table 2

SOLAR COLLECTOR PANEL ILLUMINATION TEST RESULTS
(+Y Module)

Panel Number	Open Circuit Voltage		Panel Number	Open Circuit Voltage	
	** -507	-511		** -507	-511
45	42.0	38.0	26	42.0	40.0
42	44.0	40.0	29	42.0	40.0
39	43.0	40.0	32	42.0	39.5
36	43.0	40.0	35	42.0	40.0
33	42.0	40.0	38	42.0	40.0
30	42.0	40.0	41	42.0	40.0
27	43.0	40.0	44	43.0	40.0
24	43.0	39.0	43	42.0	40.0
21	43.0	39.5	40	42.0	40.0
18	43.0	40.5	37	42.0	40.0
15	44.0	40.0	34	42.0	40.0
12	43.0	40.0	31	43.0	40.0
9	43.0	40.0	28	42.0	40.0
6	41.0	38.5	25	42.0	40.0
3	43.0	38.5	22	43.0	40.0
* 2	43.0	43.0	19	42.0	40.0
* 5	42.0	43.0	16	43.0	40.0
* 8	43.0	42.0	13	42.0	40.0
* 11	43.0	43.5	10	42.0	40.0
* 14	42.0	43.0	7	42.0	40.0
* 17	43.0	44.0	4	42.0	40.0
* 20	43.0	44.0	1	43.0	40.0
23	42.0	40.0			

* These panels were not modified

** Column under -507 lists pre-modification V_{oc} test results

-511 column shows post-modification test results

SUMMARY TEST REPORT
CONTRACT NAS 3-11512

LMC/A958407
-5-

launch, had been delayed until this more convenient time when the Solar Array Assembly would be removed from the mounting bed for the panel modifications.

X-Ray of Pyrotechnic Harness Connectors

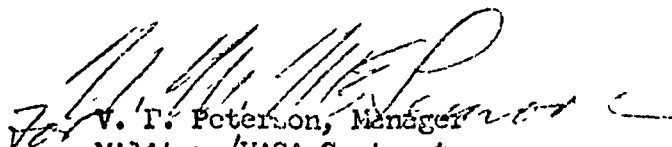
X-ray tests were made on all pyrotechnic harness connectors because of a suspected thin wall condition. Two connectors were found to be defective and were replaced and successfully retested for insulation resistance and circuit continuity.

Conclusions and Recommendations

Analysis of pre- and post-modification test data shows that the modification of the solar collector assemblies effectively reduced the open circuit voltage by approximately 3 Volts per panel. Also, results of resistance and continuity tests, conducted with each array extended and in the stowed position, shows that previous acceptance tests have not been invalidated during the modification activity. However, it should be noted that post-modification array deployment tests were not conducted in conjunction with this modification program.

It is LSC's recommendation that the SERT II Solar Array Modules are ready to support the SERT II Mission fall launch period.

LOCKHEED MISSILES & SPACE COMPANY
SPACE SYSTEMS DIVISION


V. T. Peterson, Manager
Military/NASA Contracts

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In reply refer to:
LSC/A963523/69-20/553

Lockheed
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12 December 1969

Mr. Cliff Siegert, Project Manager (6 copies)
SERT II Solar Array - Mail Stop 54-3
NASA Lewis Research Center
21000 Brookpark Road
Cleveland, Ohio 44135

SUBJECT: SUMMARY TEST REPORT ON ARRAY MODULE THERMAL TAPING CHANGE AND
DEPLOYMENT TESTS, SERT II SOLAR ARRAY CONTRACT NAS 3-11512

REFERENCE: (a) Change Orders 21 and 22 to Contract NAS 3-11512

Dear Mr. Siegert:

This summary report is submitted in accordance with Paragraph 6.3, Exhibit A, of subject contract to describe the thermal taping change accomplished and to summarize the results of array module deployment tests conducted. The thermal taping change and deployment tests were accomplished by authority of Reference (a) change orders.

The solar array modules 1363451-509 and -511 were removed from their storage containers. Thermal taping was removed from the -511 module (Fall Launch Configuration) and proper taping was applied to the -509 module (Spring Launch Configuration). As a result of this change the -511 module reverted to the -507 and the -509 to the -505 dash number identifications.

An abbreviated deployment test was performed successfully on each module. The extension time was 35 seconds for the -505 module and 37 seconds for the -507 module. Both extension times were within the specified values of 20 to 45 seconds. A slight adjustment to the power harness service loop had to be made during the extension test preparations to allow the array to fully open. This service loop had been formed in accordance with harness installation drawings at the time the array wing was reassembled to the mounting bed subsequent to the thruster panel modifications. At that time no deployment tests were conducted to confirm the adequacy of the service loop.

Subsequent to the thermal taping change and during the deployment test phase, the array modules were subjected to a degaussing process with magnetic field intensity measurements taken. The results of this activity are covered in a separate letter report.

Mr. H. W. Flohn
Page 2

In reply refer to:
LMSC/A263928/67-20/538

SUBJECT: SUMMARY TEST REPORT ON ARRAY MODULE THERMAL TAPING CHANGE AND
DEPLOYMENT TESTS, SENT II SOLAR ARRAY CONTRACT NAS 3-11512

Following completion of the deployment tests, the array modules were stowed with clamp assemblies torqued to flight values, electrically tested for circuit continuity and insulation resistance, and placed in their storage/shipping containers. Insulation resistance and continuity measurements met specification requirements of 50 megohms minimum for insulation resistance, and 0.85 ohms maximum for continuity.

The array modules are now configured for a spring launch period and are ready to support the SENT II mission.

LOCKHEED MISSILES & SPACE COMPANY
SPACE SYSTEMS DIVISION

VTP/CRM/ep

V. F. Peterson, Manager
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IMSC/A941440

In reply refer to:
IMSC/A953982
69-20/536

17 December 1959

Mr. C. Siegert, Project Manager (w/6 copies)
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Subject: Summary Letter Report on Degaussing and Magnetic Field Intensity Measurements of SERF II Solar Array Modules Contract NAS 3-11512

Reference: (A) IMSC TTX (A969706) to LORC, Subject: SERF II Solar Array Contract NAS 3-11512, Proposal for Degaussing Solar Array Modules and Measuring Field Intensities, dated 12 November 1959

Dear Mr. Siegert:

The purpose of this summary report is to describe the degaussing of the SERF II Solar Array Modules and to summarize the results of the residual magnetic field intensities measured on the deployed array modules. This effort was accomplished under Change Orders 21 and 22 as proposed in Reference (A).

A test fixture was built which consisted of 300 turns of equivalent 12 gage wire, spaced 1/2 inch apart on a wooden structure rectangular in shape and of dimensions capable of enclosing the folded array in either of the two longer axis, (Approx. 3 ft high by 7 ft wide by 7 ft long). Maximum current was 8 1/4 amperes at 60 Hz which was estimated to provide a 10 gauss field within the coil.

All field strength measurements were made with a Heliflux, Magnetic Aspect Sensor (Type RAM-3, Serial Number 1154) manufactured by Schenck Instrument Co., Silver Spring, Maryland. This instrument (hereafter called the magnetometer) provides a voltage output proportional to field strength (the scale factor is 0.25 gauss/volt).

Each solar array module was degaussed in the folded configuration with final residual magnetic field strength measurements performed in the extended position. To measure the effectiveness of the degaussing coil, one magnetized solar collector assembly (panel) was measured and degaussed. Magnetometer measurements taken before and after the degaussing process indicated that the panel was effectively demagnetized. Using one panel as a basic unit, the array module was considered as a group of magnetic dipoles contributing to composite magnetic generators. The folded array module was considered as three generators at the center of each group of folded panels along the longer axis and two generators at the center of each group of panels along the shorter axis. Magnetometer measurements were taken at these points as a guide in demagnetizing

To: Mr. Cliff Sargent, NASA JSC

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Subj: Summary Letter Report on Degaussing and Magnetic Field
Intensity Measurements of SERT II Solar Array Modules
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the effectiveness of the degaussing process. Each open array module was considered to consist of 15 generators in the plane of the array along the center of each half-leaf and 3 generators in the plane of the array along the center of each of the three rows of 15 panels. Magnetometer measurements were taken at these points to assess the magnitude of residual magnetism remaining after the degaussing process of the folded module.

Multiturn magnetometers (gradimeters) which would provide automatic null of the earth magnetic field were not available. Therefore, the following procedure was adopted for the measurements of the modules in both deployed and folded configurations. First the array was physically removed a sufficient distance to preclude affecting the fixed magnetometer head. Next the instrument was aligned such that the instrument gave a zero output for all folded array measurements or a constant reference reading for measurements in the open array position, thus effectively nulling out effects of the earth's field strength. Finally, the folded array was moved to a close proximity of the magnetometer head (20 to 50 inches) and the voltage change and distance was recorded.

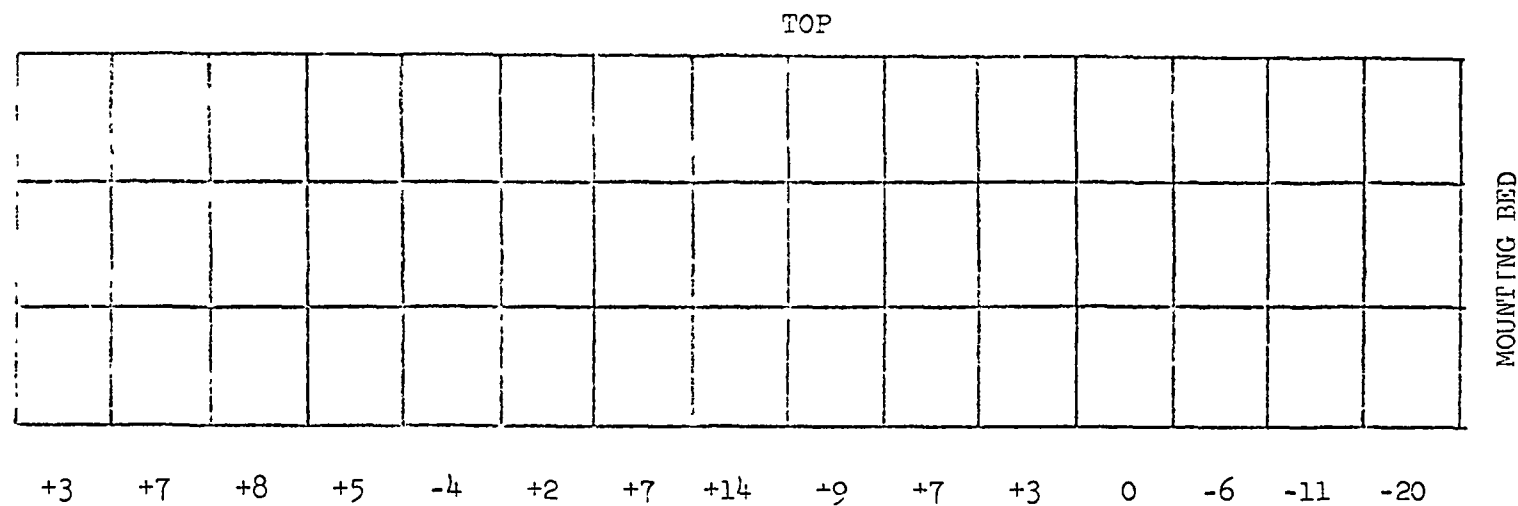
The degauss procedure was to insert the module into the coil, slowly increase the current, subject the unit to the maximum field for a short period of time, then slowly decrease the field to zero. A typical sequence was 2 minutes rise time, 2 minutes soak, and a 2-minute decay time. After the degaussing, measurements of field strength at varying distances were recorded. This procedure was repeated many times with various module orientations within the coil. When the magnetometer readings showed no further reduction of field strength the procedure was terminated. The procedure reduced the high initial readings to 1/2 or less but had a lesser affect in reducing values which were initially low.

After completion of the degaussing operation the arrays were given a deployment test. Subsequent to deployment a sequence of magnetometer readings was taken while the array was in the extender in an open position. Figure 1 summarizes the -505 module magnetometer readings (in millivolts) while Figure 2 provides similar data for the -507 module. Each of the readings was taken at a constant distance of 30 inches from the edge of each of the designated magnetic generator.

For the assessment of the residual magnetism remaining in the array modules, the total scaled value was determined. This determination was made by the summation of the magnetometer values in millivolts obtained at a 30 inch distance along the bottom of the array from each half-leaf. From Figure 1, the total for the -505 module is 24 millivolts, and from Figure 2, the total for the -507 module is 26 millivolts.

Assuming each magnetic generator to be at a distance r along the extension of the longitudinal axis of the magnet and given the magnetic field intensity H (as derived from magnetometer reading taken at distance r) the magnetic moment was calculated to be:

DIFFERENTIAL MAGNETOMETER READINGS IN MILLIVOLTS
 AT A DISTANCE OF 30 INCHES
 FOR -505 SOLAR ARRAY MODULE



Delta Millivolts = (Earth + Array) - (Earth Alone)

Side Shown is Non-Active Surface

Figure 1

DIFFERENTIAL MAGNETOMETER READINGS IN MILLIVOLTS

AT A DISTANCE OF 30 INCHES

FOR -507 SOLAR ARRAY MODULE

TOP

MOUNTING BED

-17 -4 +2 +6 +5 +5 +4 +4 +3 +3 +3 +4 +3 +2 +5

Delta Millivolts = (Earth + Array) - (Earth Alone)

Side Shown is Non-Active Surface

Figure 2

To: Mr. Cliff Siegert, NASA LeRC

IMSC/A963982
Page 3

Subj: Summary Letter Report on Degaussing and Magnetic Field
Intensity Measurements of SERT II Solar Array Modules
Contract NAS 3-11512

For -505 Module:

$$\begin{aligned} M &= \frac{Hr^3}{2} = (1/2) \frac{.25 \text{ gauss}}{\text{volt}} \frac{2.54^3 \text{ cm}^3}{\text{in}^3} \\ &= \frac{2.05 \text{ gauss-cm}^3}{\text{volts in}^3} \\ &= (2.05)(.024)(30)^3 \\ &= (.205)(24 \times 10^{-3})(27 \times 10^3) \\ &= 1328 \text{ gauss-centimeter}^3 \end{aligned}$$

For -507 Module:

$$\begin{aligned} M &= \frac{2.05 \text{ gauss cm}^3}{\text{volts in}^3} \\ &= (2.05)(.028)(30)^3 \\ &= (2.05)(28 \times 10^{-3})(27 \times 10^3) \\ &= 1550 \text{ gauss-centimeter}^3 \end{aligned}$$

Therefore, the total magnetic moment of both modules is approximately 2900 gauss-centimeters³.

It was shown that the solar array modules were effectively degaussed to a level which results in residual magnetic moment on the order of 3000 gauss-cm³. IMSC cannot make an exact evaluation of the vector direction of the residual magnetization due to the limitations of the instrumentation that was available and used. The magnitude as estimated by linear summation is less than 3000 gauss-cm³ for both modules. IMSC experience indicates this is a practical lower limit with further reduction requiring highly specialized and sophisticated equipment and facilities. A magnetic moment value of 3000 gauss-cm³ is well within the allowance made in the Agena spacecraft magnetic moment analysis for orbit stability and therefore, IMSC recommends that no further action be taken.

LOCKHEED MISSILES & SPACE COMPANY
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