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SPACE CONSTRUCTION SYSTEM ANALYSIS

PART 2 FINAL REPORT
PLATFORM DEFINITION

APRIL 1980

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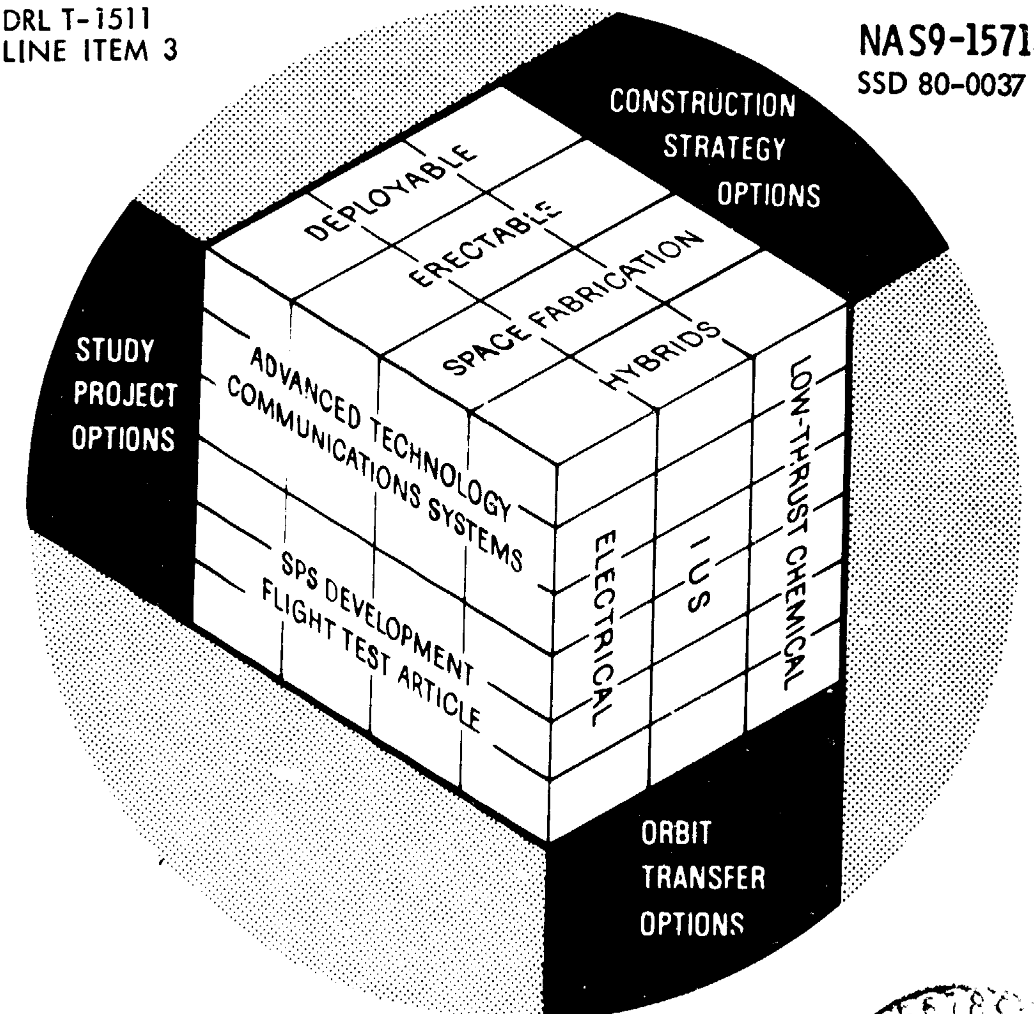

Rockwell International

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Space Systems Group
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SPACE CONSTRUCTION SYSTEM ANALYSIS
PART 2, FINAL REPORT

Platform Definition

MARCH 1980

NAS9-15718

Principal Authors: R. J. Hart
H. L. Myers

Satellite Systems Division
Space Systems Group
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FOREWORD

This report summarizes the mission and system requirements and accompanying design definition for the Engineering Technology Verification Platform (ETVP). The ETVP was selected at the end of Part 1 of the study for use as a model system to study space construction concepts and processes. The system requirements, platform design definition, and related rationale in this document are the study products from Tasks 6 and 7 of Contract NAS9-15718, Space Construction System Analysis Study. This contract effort was conducted by the Space Operations and Satellite Systems Division, Space Systems Group, of Rockwell International Corporation for the National Aeronautics and Space Administration, Johnson Space Center. The work was administered under the technical direction of the Contracting Officer's Representative (COR), Mr. Sam Nassiff, Spacecraft Systems Office, Spacecraft Design Division, Johnson Space Center.

The study was performed under the direction of Ellis Katz, Study Manager. The following persons made significant contributions toward the completion of the analyses reported herein.

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Major documents resulting from Part II of the contract effort are listed below:

Space Construction System Analysis, Part 2, Platform Definition, Final Report	SSD 80-0037
Space Construction System Analysis, Part 2, Construction Analysis, Final Report	SSD 80-0038
Space Construction System Analysis, Part 2, Cost and Programmatics, Final Report	SSD 80-0039
Space Construction System Analysis, Final Report, Space Construction Experiments Concepts	SSD 80-0040
Space Construction System Analysis, Part 2, Executive Summary, Final Report	SSD 80-0041

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1.0 INTRODUCTION

1.1 PURPOSE AND SCOPE

1.1.1 Purpose

This document summarizes the top level system requirements and presents the accompanying conceptual design for an Engineering and Technology Verification Platform (ETVP) system. The ETVP is intended to be a versatile tool or facility for use in developing both large area space systems technologies and then in developing "user technologies" which require large area space systems.

The ETVP concept was selected at the end of Part I of the study to be the reference configuration for the analysis of space construction processes and technologies. Thus, the purpose of this document is to present the system scenario/requirements and rationale along with a preliminary conceptual design for the ETVP system.

Sufficient platform design definition is required to satisfy several key needs, (1) to enable design interactions with the construction processes to be understood, (2) to verify the feasibility of construction out of the orbiter, and (3) to establish trends and determine sensitivities to mission resource requirements. Although design completeness is required to satisfy these needs, emphasis was on the rapid development of a representative and reasonable design with traceable requirements rather than highly rigorous approaches required to produce an optimized design. Where insufficient data were available and/or where extensive trades would be required to determine optimum characteristics, sensible judgments were applied based on appropriate qualitative factors to establish needed system requirements and definitions. In this way, a suitable platform design was developed while conserving resources for the main study objective of construction analysis. The resulting platform definition serves as a model for the space construction analyses.

1.1.2 User Survey

The requirements development included a "user" survey in which key contacts were made with members of private companies and with government agencies to determine the general acceptance, fundamental needs and/or desirability for an ETVP type system. The consensus viewpoint indicated the ETVP could serve an important role in many future space projects. Candidate payload areas were identified and usage scenarios prepared as a rapid means for defining system requirements. The advanced communications technology mission as a rapidly growing world need was selected as the reference payload for use in quantifying specific payload interface requirements.

1.2 ORGANIZATION

This document is organized into two main sections, System Requirements and Platform Design which are followed by a series of supporting appendixes.

The System Requirements section begins with an all encompassing statement of system objectives which drive the system requirements. This is followed by paragraphs on the major mission and subsystem requirements which are derived from the mission objectives with emphasis on the advanced communications technology mission/payload.

The Platform Design section defines the platform configuration which evolved from the mission and subsystem requirements.

The appendicies contain various trade studies and supporting rationale for selection of specific elements or features of the platform configuration.

2.0 ENGINEERING AND TECHNOLOGY VERIFICATION PLATFORM SYSTEM REQUIREMENTS

This section summarizes the top level system requirements for the Engineering and Technology Verification Platform (ETVP). These requirements form the basis for the preliminary design activity necessary to define and size the platform hardware. The resulting preliminary design is used as a reference configuration for end-to-end space construction analyses. The main purpose of the construction analyses is to define preferred construction methods and processes, identify the important interactions between the platform design and the construction system design and operation, and to outline the important technology development efforts required to support the design and space construction of the ETVP.

2.1 ETVP SYSTEM OBJECTIVES

2.1.1 Introduction

The Engineering and Technology Verification Platform (ETVP) is intended to be a versatile development tool or facility able to support a variety of test payloads. It is envisioned to have an IOC in the late 1980's time period. In addition it is intended to serve as a demonstration project for large area systems requiring construction in space. In meeting these objectives the ETVP must also provide high legacy values to even larger future space constructed systems such as the Satellite Power System. The implications of these three objectives are discussed in the following paragraphs.

2.1.2 Technology Development Facility

In the role of a developmental tool the ETVP is envisioned to be a shuttle tended free-flying facility with LEO and GEO capabilities for supporting large user payloads requiring any of the following features: more than one shuttle flight for delivery/assembly, more electrical power or longer mission duration than the shuttle can economically provide. Candidate users include communications development, SPS technology advancement payloads, space based radar, high resolution radiometry and large optical systems and their applications.

To focus the scope of the ETVP design effort the system requirements to be specified herein will be based on the advanced communications technology needs. This avoids the need for developing rigorous definitions of user payloads for all categories. It is intended, however, that the basic ETVP design concept be suitable for all user categories, but, to save study resources the configuration to be used for construction analyses will be sized to support the communications development mission. It is felt that the structure could be lengthened, if necessary, to meet other payload needs and that detailed subsystem elements could be resized to reflect the specific needs of other payloads. None of these potential resizing characteristics is expected to significantly affect the construction analysis, and thus, while the ETVP design used as a reference configuration in the study is to be based on the communication payloads, it and the construction processes derived from it will fundamentally have the capability of supporting all payload categories.

It is believed that an ETVP concept designed for reasonable levels of versatility, once it is placed in operation, will be utilized by the entire user community. Payloads will be sized and tailored to it much the same as payloads are now being designed to fit in the Orbiter cargo bay. Thus, inasmuch as the orbiter will become the main element in the space transportation system, the ETVP can become a basic development facility or tool for furthering various technology areas.

2.1.3 Demonstration Project

As mentioned earlier, in addition to being a versatile development tool, the ETVP is intended to serve as a demonstration project for large area space systems. In this capacity it can provide experience and learning in the design and construction of lightweight space structures and can actually serve as a test-bed to explore the behavior of space structures in the zero "g" gravity gradient and thermal environments. It could also serve as a meaningful experiment and demonstration in the area of control dynamics interactions with structural modes.

It could be outfitted with appropriate instrumentation and during the initial mission phases following construction, special test operations could be performed to produce the desired dynamics data. Certain structural behavior data could be developed for the structure alone (no modules or subsystems installed) if they were needed for basic technology reasons. This would require a construction strategy calling for completion of the structure before the installation of systems and components. This may not be the optimum strategy, but it does represent some of the versatility and options available to maximize the overall ETVP project effectiveness and its value to the national space program.

Similarly, the platform could further be used to investigate system interactions with orbit transfer propulsion in terms of TVC dynamics and loads and vibration environments imposed on subsystems and payloads. It would naturally serve as a useful demonstration covering facets of space construction such as beam joining/welding, module/component installations, structural alignment, etc.

Another very important technology objective could be in the demonstration of on-orbit servicing techniques and concepts, possibly including both manned servicing modes and remote/teleoperator modes suitable to early GEO applications (before manned missions to GEO are possible). Additionally, experience will be gained in high power (and possibly high voltage) electrical power generation, distribution and switching systems and components. Thus, in addition to being a development tool the platform can be considered as a set of experiments in itself, particularly during construction and initial mission phases.

2.1.4 Legacy

To achieve the final project objective, high legacy value, a configuration concept with a high potential for automation and high construction productivity is required. A tri-beam configuration made up of space fabricated beams with cable bracing to rigidize the structure will meet this objective. It offers the automation and packaging efficiency of space fabricated beams and related construction processes and will give experience in the erection of three-dimensional structures that will be needed for future large area space systems.

The philosophy reflected in these three objectives will be used in establishing system/subsystem requirements for the ETVP. It will basically be designed and sized for the advanced communications mission for construction analysis purposes, but it will have the inherent capability for supporting a variety of user areas and to also serve as a test-bed, demonstration project covering many large space system issues.

2.2 CANDIDATE PAYLOADS

2.2.1 Introduction

Beginning in the fall of 1978, the need for better understanding of future space requirements was keenly felt as a strong driver on platform construction design. As the first step, an initial industry survey was made in the fall of 1978 which investigated the utility of a large space platform for the common carrier industry. Most carriers (WU, ASC, SBS, COMSAT) are in favor of large platforms, especially if replacement and repair of communications systems is feasible in geosynchronous orbit. AT&T was the least enthusiastic and pointed out problems of sharing space on a platform by a multitude of users (ownership, cost, interference, reduction in competition, the alternative approach of large capacity increase of ground facilities at possibly lower cost).

In February, Collins Transmission Systems Division of Rockwell was asked to perform a communications platform study to include:

- Capability of current satellite constellation
- Projected future demand
- Capability of future satellites
- Projection of possible saturation of orbital arc and frequency spectrum.
- The use of a multiple beam communications platform to relieve congestion by frequency reuse.
- The evolution of the communications platform concept.

The study was completed and published in the Part I Final Report, Document SSD 79-0126, June 1979. This report concluded that due to the extraordinary growth of satellite communications, the limited bandwidth, and the crowded occupancy of the orbital arc, saturation will occur in the 4/6 GHz band in 1989-1992 and in the 12/14 GHz band in 1993-1996. This saturation is due to the single-beam per satellite system in present use. By utilizing multiple-beam satellites, the frequency reuse factor can be greatly increased and the saturation alleviated so that much more traffic can be handled. The study went on to recommend an Engineering and Technology Verification Platform to verify certain multiple-beam antenna characteristics in time for commercial platform applications in the 1990 to 1994 time frame. At the same time this antenna work is being carried out, the report recommended that other users share in the use of the Engineering and Technology Verification Platform in order to reduce cost and to benefit from redirected use of the multiple-beam antenna. Suggested user experiments included: propagation measurements, RFI, low-cost TV, electronic mail pilot program, data relay, and emergency aircraft beacon locating.

2.2.2 User Survey

A user survey for candidate payloads for the Engineering and Technology Verification Platform (ETVP) was made during the summer of 1979. The survey was conducted with three major thoughts in mind: (1) continue common-carrier survey to review ETVP systems concept with system specialists in the communications industry, (2) review selected antenna multiple-beam pilot-test concepts with antenna specialists, and (3) identify experimental payloads (other than the multiple-beam antennas) for possible testing on the platform.

This survey began by reviewing satellite communications system studies recently conducted by Aerospace, Ford, Hughes, Western Union, Comsat, ITT, and Rockwell. Three pilot antenna systems suitable for testing on the platform were selected. Fourteen questionaires detailing these concepts were mailed out for user response. Each potential user was then telephoned for response (most people did not reply to the mailed questionaire) and those found to be interested were visited (see Figure 2.2.2-1).

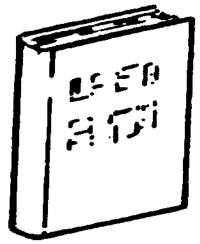
The briefing used on the user survey trip and the full results are given in Appendix G. A summary of results is given here in Table 2.2.2-1. Other payloads which have been suggested for the ETVP are:

- SPS Development Test Payloads
- Soil Radiometer, Antenna, 30-m (NASA Langley/GE)
- Infrared Radiometer, Antenna, 2.5-m (SAMSO/Rockwell)
- Land-Mobile Radio, Antenna, 50-m (NASA Headquarters/Aerospace)
- Interferometer, Ships, Aircraft Location (NASA Goddard/Hughes)
- Pilot Testing of Space-Based Radar Antenna Concepts, Antenna, 40-60-m (RADC)

2.2.3 Payload Definition Summary

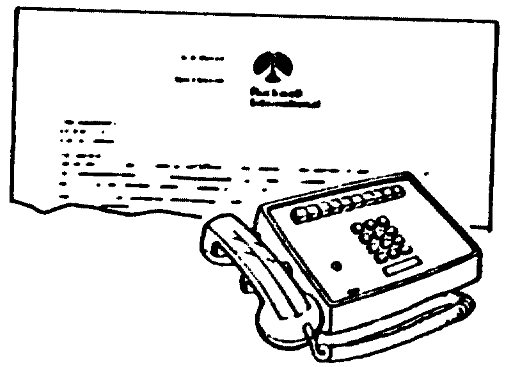
Four antenna concepts were selected for pilot testing on the ETVP, three from reference studies and one from the user survey. The four concepts are:

LITERATURE RESEARCH



- COM SYSTEMS
- RADIOMETERS
- SURVEILLANCE
- SPS

CONTACTS



- FAIRCHILD INDUSTRIES
- NASA HQ
- DOD/AFRDS
- NASA LANGLEY
- NASA GODDARD
- NASA LEWIS
- COMSAT
- WESTERN UNION
- NASA/MSFC
- JOINT COUNCIL, ED TV
- AT&T LONG LINES
- AEROSPACE CORP.

VISITS

- WL MORGAN - COMSAT
- DON NOWAKOSKI - WESTERN UNION
- JOHN McELROY - NASA HQ
- FRANK NORWOOD - JOINT COUNCIL ED. TV
- PETER FOLDES - GENERAL ELECTRIC
- AL YEH - BELL LABS
- BOB HARRIS - AT&T LONG LINES
- IDO NAKAMURA - AEROSPACE CORP.

Figure 2.2.2-1. User Survey

Table 2.2.2-1. User Survey Results

FAVORS ETVP	PROBLEMS
Western Union	• Switching, timing and buffering • Cost • Non-continguous beams favored
Comsat	• NASA funding • Repair, replacement in space
NASA Headquarters	• Test switching between 20/30 & 12/14 GHz • 20/30 GHz capacity may preclude large platforms • Favors NASA/commercial joint venture
Joint Council Ed. TV	• Ed. TV needs to be self-sustained • School timing problems
General Electric	• Single point failure • Antenna isolation • 20/30 & 12/14 GHz capacity may preclude large platforms • BFN best solution
Bell Labs	• Sidelobe problem • Scanning best solution • Lab demo progressing
Rome Air Dev. Center	• Interested in phased array radar test on ETVP
DOES NOT FAVOR ETVP	
AT&T Long Lines	• Not feasible • Too costly • Projected demand too high • Plenty of ground capacity

1. A scanning phased array which uses movable spot antenna beams to communicate to dispersed ground stations in a Time Domain Multiple Access (TDMA) mode. As many as 20 fixed beams and 10 pairs of movable beams are envisioned for a full commercial system. For the pilot test only one pair of movable beams and four fixed beams need be tested to checkout this concept.

2. A number of fixed beams (100 to 250 for the full future system) of which a significant fraction needs to be contiguous and all need good sidelobes. One solution to this dual problem is to use multimode corrugated feedhorns (for their sidelobe performance) and to interleave the spots from three antennas to obtain contiguous coverage. For the pilot test only two interleaving antennas need be used with 10 multimode horns each.

3. Another fixed beam concept is one in which good sidelobes and contiguous coverage is given by beam-forming networks. This system has an added potential advantage of reconfigurability in which imperfections in the parabolic reflector can be partially compensated whether caused by attitude error or thermally induced defocusing. The interaction of variable phase shifters, hybrid networks, and the interleaving and extraction of various frequency bands requires extensive testing to verify this complex method. A pilot test with 10 beams should be sufficient to checkout this concept.

4. The fourth concept selected for test on the ETVP was suggested by more than half those people interviewed on the user survey. The testing of the 20/30 GHz band by actual space/ground communications links was very appealing. A second part of this test is switching from 20/30 GHz during rainstorms to a backup 12/14 GHz.

2.2.4 Other Possible Payloads

As mentioned in section 2.2.1 User Survey, there are 6 additional payloads which are tentative candidates for testing on the ETVP. The reasons these are not included as actual payloads at this time range from a primitive state of development of concept to a reduced level of urgency as compared to the specific communications technology payloads chosen. Nevertheless as time goes on one or more of these payloads may indeed become a prime choice. Three of these candidate payloads of which more detailed characteristics are available are discussed in this section.

2.2.4.1 SPS Development Test Payloads

The SPS technology advancement program is an on-going activity aimed at developing a comprehensive understanding of the technical requirements, the economic practicality and the social and environmental acceptability of the solar power satellite for meeting growing energy needs. Specific SPS development objectives include the following four primary areas and encompass both LEO and GEO test situations. The four primary areas are: (1) microwave power transmission system, (2) photovoltaic power generation system, (3) space construction processes and evaluation, and (4) space structures.

Power transmission test issues include both klystron and amplitron RF power generator concepts, evaluation of the retro-directive phase control system and factors affecting power transmission efficiency.

Photovoltaic power generation test issues include high voltage (up to 40,000 volts) power distribution and switching, arcing phenomena and high voltage power loss to surrounding plasma.

Space construction test issues include automated fabrication processes, large element assembly and alignment, large structure deployment and component installation concepts.

orbit satellites could sweep over target areas at relatively close range. This eases the antenna size, but increases pointing and tracking difficulties. Better choices seem to be a 6 hour orbit, 12 hour orbit, or even geosynchronous orbit. It is desired to have something like a 50 km beam footprint. Consequently, the antenna sizes are 90, 140, and 225 meters, respectively. Before the full scale radar is deployed, system tests will be conducted with an antenna in the 40 to 70 meter range. A 60 meter version is shown attached to the ETVP in Figure 2.2.4-2.

The rate stability of the platform should be at least 1/10 of antenna beamwidth over approximately 10 scan periods of the radar. This works out to be .028/10 deg/sec for the 60 meter radar. The pointing accuracy can be about 0.25 degree since the radar images known landmarks. A data rate of 4 Mbits/sec should be adequate for a radar which processes most of the data on-board. However, more data may be needed for a test system. In any case, 50 Mbits/sec will certainly suffice. Gross sizing assumptions are:

	Mass (kg)	Power (Watts)
(2500 W Avg.) Transmitter	1,100	10,000
Processor, Receiver	400	1,200
Antenna, 60 m (1 ply)	1,800	
Active Lens, 15 m (3 ply)	400	
	3,700 kg	11,200 Watts

2.2.5 Payload Requirements

The payloads to be installed on the ETVP shall conform to the following requirements.

2.2.5.1 Payload Location

The payload shall interface with the ETVP at one or more of the attach ports. There are eight attach ports located at the ends of the four long cross-beams. Although the three attach ports located on the aft thrust structure are intended primarily for the orbital transfer propulsion modules, they may be used for payload installation when otherwise unoccupied (Figure 2.2.5-1).

2.2.5.2 Payload Installation

The payload shall be capable of installation on the ETVP in the following modes:

1. In LEO, transportation and installation by the Shuttle Orbiter. The construction fixture will be available attached to the ETVP.

2. In GEO, transportation and installation by Teleoperator. Teleoperator docking ports are provided on the ETVP, located close to the payload attach ports on the long cross beams. There are no Teleoperator docking ports on the aft thrust structure.

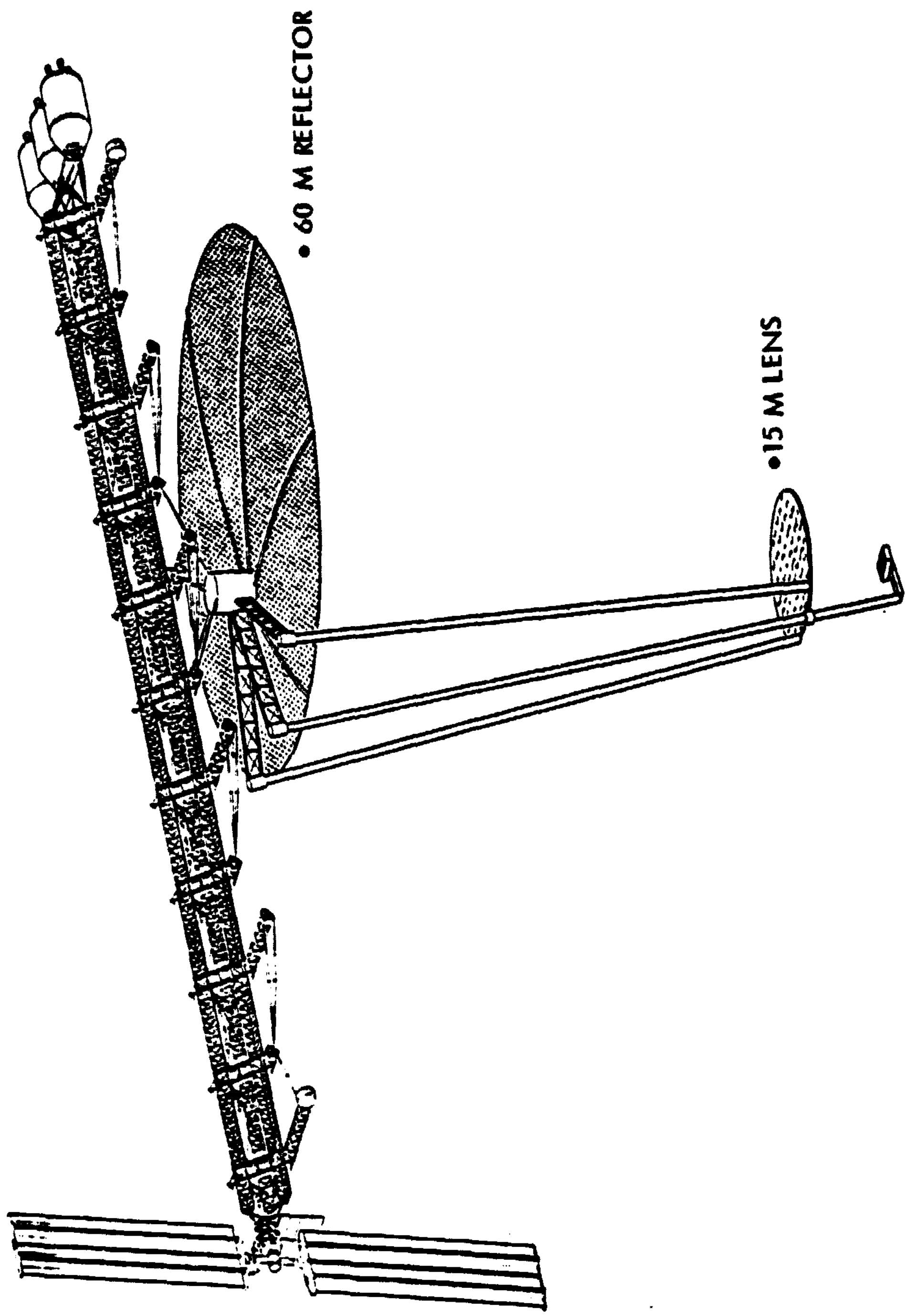


Figure 2.2.4-2. ETVP—Space Base Radar Test

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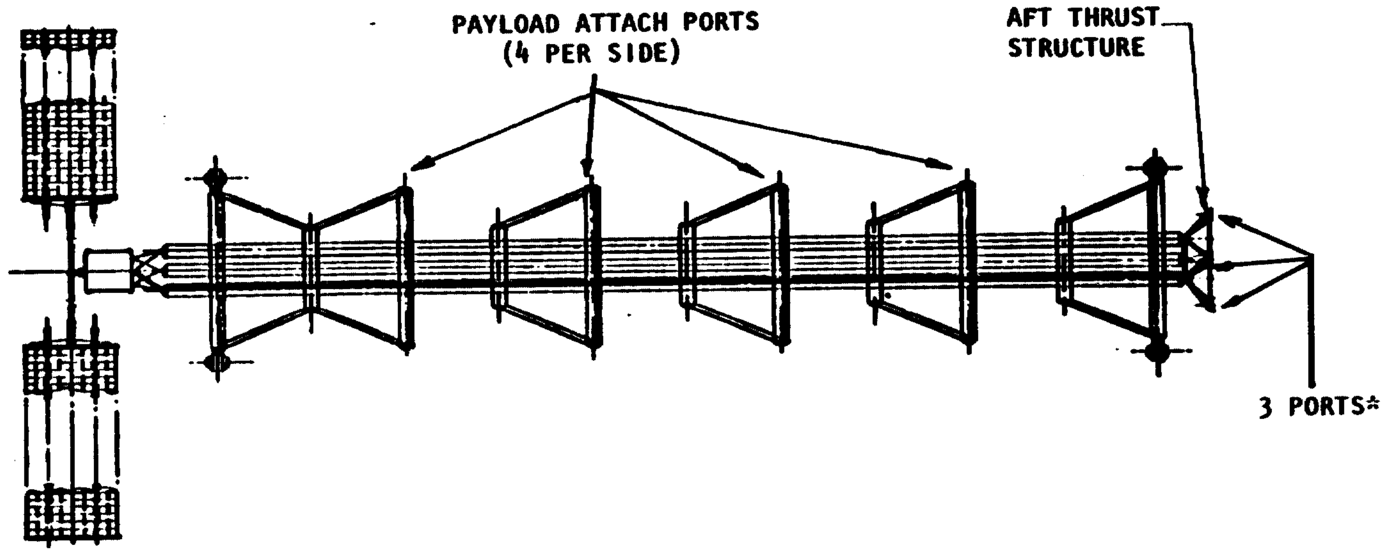


*Primarily for orbit transfer propulsion modules, but may be used for payloads.

Figure 2.2.5-1. Location of Payload Attach Ports

2.2.5.3 Payload Interface Configuration

The interface between the payload and the ETVP shall conform to the following:

1. Attach port configuration (Section 3.1.2, Figure 3.1.2-2).

2. Interface electrical connector (Figure 2.2.5-2).

3. Electro-mechanical latches (Figure 2.2.5-3).

2.2.5.4 Payload Power and Data Requirements

The electrical connector across the attach port interface shall supply the following connections to the payload:

1. Two pairs of power leads. Nominal maximum power shall be 30 kW dc.

   Note 1: It is possible to join together the power for the two power leads and obtain power up to 60 kW dc.

   Note 2: The power available for the three attach ports on the aft thrust structure is 2 kW each.

2. Four coax cables (Payload Data).

   Note: Not supplied to the aft thrust structure attach ports.

3. Four twisted shielded pairs (Housekeeping).

4. Five twisted shielded pairs to the crossbeam attach ports only (Hardline Backup).

5. Two twisted shielded pairs to the aft structure attach ports only (Hardline Backup).

2.2.5.5 Payload Pointing and Target Aquisition

Payloads shall contain the necessary equipment (servos, drive motors, sensors, etc.) for target aquisition.

The ETVP shall maintain the following pointing accuracy in LEO and GEO.

Attitude determination	0.050°
Control dynamics	0.100°
Structural thermal deformation	0.080°

The manufacturing and assembly inaccuracies between each ETVP attach port and the Attitude Reference System in the SCM shall be measured during the ETVP construction phase, and shall be recorded. The final installation accuracy shall be within 0.21 deg.

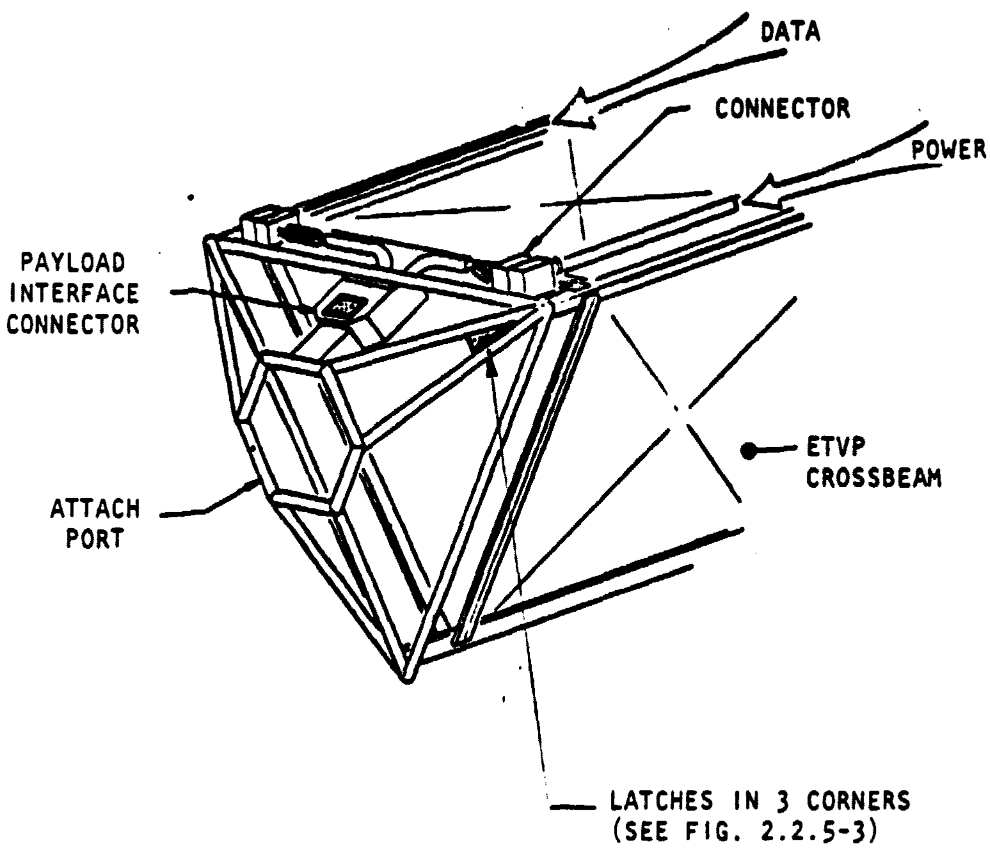


Figure 2.2.5-2. Payload Interface Connector

ELECTRO-MECHANICAL LATCHES

3 FOR EACH PAYLOAD ATTACH PORT

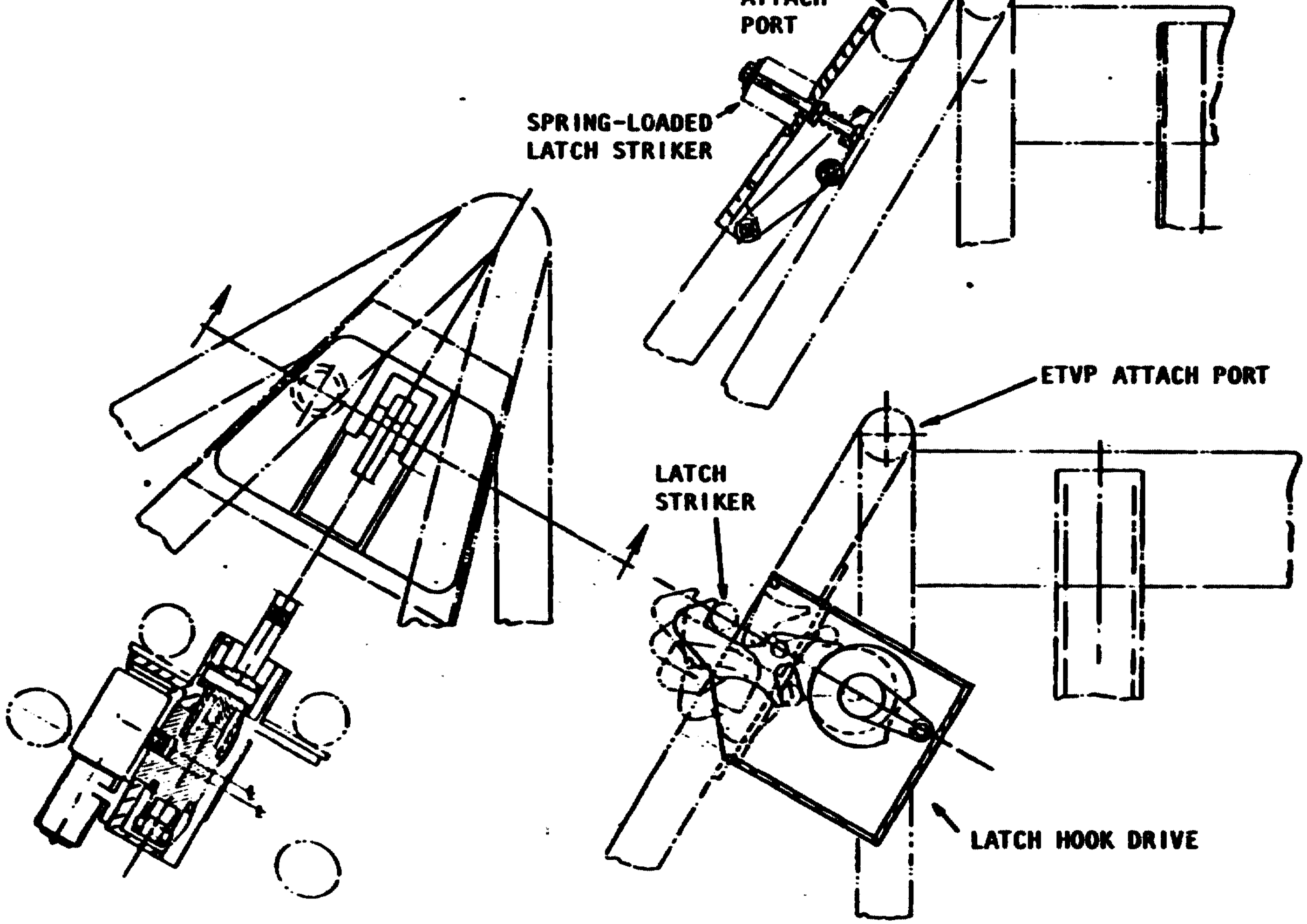


Figure 2.2.5-3. Payload Interface Requirements

2.2.5.6 Orbit Transfer

Payloads installed in LEO and intended for operation in GEO shall survive the orbit transfer maneuvers and accelerations ($T/W_{max} = 0.2$) without significant degradation. Payloads may be designed to fold or retract in LEO and then to deploy to operational configuration in GEO.

2.2.5.7 Thermal Control Requirements

The payload shall provide its own thermal control capability.

2.3 REFERENCE MISSION DEFINITION

2.3.1 Introduction

The ETVP reference mission contains 8 major phases in which any or all of the following elements may differ; the character of the operations, the configuration of the flight system or the flight environment and/or mission geometry. The 8 mission phases are shown pictorially in the mission profile of Figure 2.3.1-1 and are listed as follows:

- Platform Activation
- LEO Tended Operations
- LEO Free-Flight Operations
- Orbit Transfer Propulsion Build-up
- LEO-GEO Orbit Transfer
- GEO Activation
- GEO Technology Mission Operations
- GEO Servicing Operations

These mission phases begin after the construction of the ETVP is completed. Thus, the various operating situations reflected within these mission phases will serve as a basis for establishing system/subsystem requirements through the active life of the vehicle. Each mission phase is described below in terms of the types of flight modes involved and the mission operations to be performed.

2.3.2 Platform Activation

During this phase of the mission all subsystems will be powered up, initialized and configured into an appropriate LEO operating mode. Although certain continuity checks and other confirmation procedures will likely be performed during the construction process, the activation phase will be the first time all subsystems are fully powered and their operating performance verified. It is envisioned that this mission phase will be accomplished with the shuttle orbiter attached to the platform via the construction fixture. At the end of this mission phase it is presumed the ETVP will be declared "operational" and capable of achieving its LEO technology mission objectives. Prior to this event, however, the shuttle may separate, perform a final fly-around inspection sequence and any other final checkouts which cannot be performed with the shuttle attached.

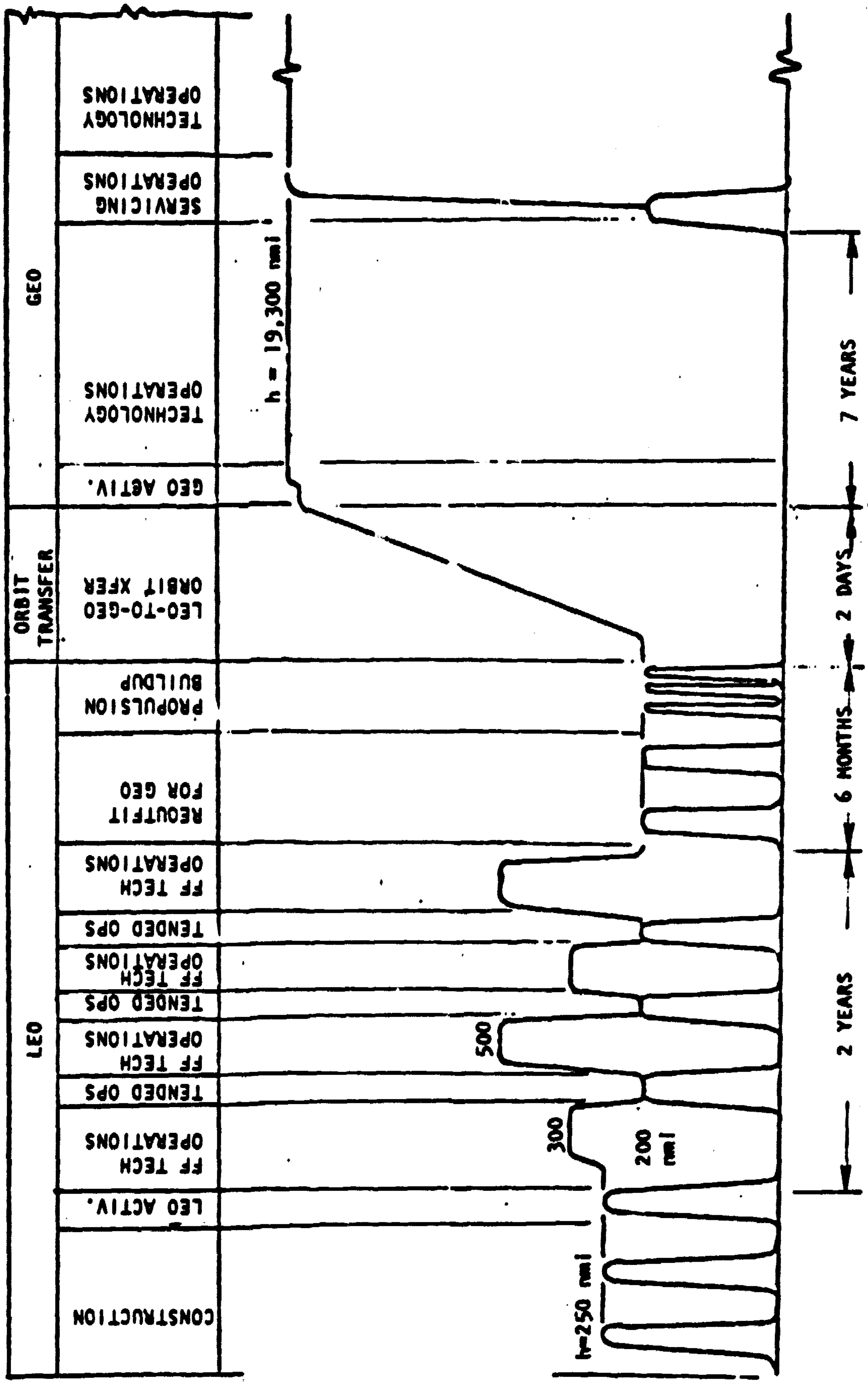


Figure 2.3.1-1. ETVP Mission Profile

The construction fixture is presumed to remain attached to the platform to aid in later shuttle revisits in LEO. This initial activation process could also include checkout and activation of the initial payload systems installed during the construction process.

The activation phase will occur at the construction orbit, assumed to be 250 nmi at 28.5 degrees inclination.

2.3.3 LEO Tended Operations

As part of the LEO operations scenario it is envisioned that shuttle revisits will be required to add or exchange payload equipment and/or for servicing either platform subsystems or payload gear. LEO tended operations, then, include those mission periods when the shuttle is attached, but after activation. Since it is likely the platform will operate at higher orbit altitudes than suitable for space construction, the platform must perform a fly-down maneuver prior to each shuttle revisit (assuming shuttle payload delivery performance is a factor in the revisit mission objectives). As a gendral guide it is presumed that revisit missions will be scheduled at 6-month intervals during LEO operations. Shuttle attachment to the ETVP will be via the construction fixture, which through its translation capability will provide access to needed payload/subsystem locations on the platform. The platform is assumed to be powered down for shuttle tended operation, but because of fixture translation requirements will be required to provide its own power. Cooperative rendezvous aids are assumed to be provided by the platform to permit efficient shuttle rendezvous operations.

2.3.4 LEO Free-Flight Operations

This mission phase is the main LEO operational period in which the technology payloads are exercised and operated to meet their respective development/test objectives. This phase includes the platform fly-up and fly-down operations to the payload operating orbits and back down to the shuttle revisit orbit. Payload operating orbits are expected to fall in the 300 to 500 nmi altitude range, but could conceivably be higher. Shuttle revisit orbit altitude is assumed to be 200 nmi.

Various flight modes are envisioned to be required in support of payload operations. These include LVH, IH and possibly LOS tracking of ground targets. Body pointing to the sun or body pointing to a TDRS satellite may also be required.

During payload operations the platform will provide electrical power, pointing/stability and command and control functions to the payloads.

It is envisioned that any given flight mode need be sustained, disturbance free, no longer than one complete orbit period since most line-of-sight conditions in LEO are subject to interruptions, occultation, etc., within this time period.

2.3.5 Orbit Transfer Propulsion Build-up

In addition to the LEO operations many technology payloads require development testing in GEO. This mission phase, then, encompasses the activities and operations involved in the delivery and installation of the propulsion modules required to perform the LEO to GEO orbit transfer. The platform is envisioned to be flown down to the shuttle revisit orbit (or as low as possible above the revisit orbit) and then operated in a minimum drag configuration during the propulsion build-up phase. The propulsion modules are assumed to be installed from the shuttle with the shuttle berthed to the construction fixture. Propulsion checkout and readiness verification is also assumed to be accomplished with the shuttle attached. After overall system readiness for GEO transfer is confirmed the platform will be separated from the construction fixture. The fixture will be returned to earth with the orbiter. The platform payloads are assumed to be installed and configured to their orbit transfer condition prior to this phase.

2.3.6 LEO-GEO Orbit Transfer

This phase begins after the shuttle leaves following delivery and installation of the final propulsion module. The platform and payload systems are presumed to have been configured for orbit transfer in the preceding mission phase. This phase could include a period of LEO orbit coast to accurately determine the orbit ephemeris and to achieve the proper nodal position for the perigee burn. The appropriate guidance data must be loaded into the platform G&N subsystem and the platform must be oriented to the required thrusting attitude. The LEO to GEO transfer may follow a direct ascent profile (typically 3 burns) or may be comprised of a series of perigee and apogee burns. Monitor and command and control functions must be maintained during portions (if not all) of this flight phase. This phase ends with the verification of satisfactory completion of the final GEO insertion burn.

2.3.7 GEO Activation

This phase begins after satisfactory arrival in GEO. However, some position (longitude) adjustments may be necessary to achieve the placement precision desired for the platform and are considered part of this mission phase. After safe arrival the platform subsystems must be configured to their GEO operating condition. This includes deploying solar arrays, switching from battery power to the solar arrays, and orienting to an "earth looking" LVH flight mode with the solar arrays sun tracking. The phase ends with the platform at its GEO orbit position and with all subsystems verified as operational, including EW and NS stationkeeping functions. At this point the platform is declared ready to support payload operations. Payload activation is not considered a part of this mission phase.

2.3.8 GEO Technology Mission Operations

Once the platform arrives at its assigned geosynchronous location, is checked out and declared operational, this phase begins. The activation and operation of all technology payloads will be performed during this mission phase. The platform is presumed to be operating in an earth looking LVH flight mode with periodic stationkeeping maneuvers to control both East-West and North-South drift. As a developmental vehicle, interruptions in payload operations will be permitted for platform housekeeping functions (if required). This is in contrast to the need for uninterrupted service in the case of operational platforms providing commercial services. Platform relocation maneuvers will also be included in this mission phase if they are required for payload operations.

2.4 CONFIGURATION CONCEPT

The configuration of the ETVP is the result of implementing the following objectives.

- The platform will have the versatility to accept multiple antenna payloads or single large area payloads.
- The platform concept will have the capability to be configured as an SPS test article.
- The stiffness (.005 Hz) of the platform will be compatible with the antenna requirements.
- The platform concept will minimize the construction effort and the construction equipment complexities.
- The concept will be compatible with STS capabilities and services.
- The platform will be fabricated and assembled from the Shuttle orbiter.
- The concept will permit LEO servicing with/without EVA activity and provide for remote servicing in GEO.
- The solar arrays will have the capability of 2 DOF and be retractable for boost to GEO.
- The platform will have the capability to be boosted to GEO utilizing low-thrust chemical propulsion engines.
- Attachments to the platform, such as the RCS modules and antenna payloads, will utilize a common berthing/attach port concept.
- Modules will be capable of being handled and berthed/attached to the platform utilizing the orbiter RMS.

The platform as illustrated in Figure 2.4.1-1 utilizes a linear, tri-beam, structural arrangement that meets the structural and operational objectives. The construction operations are discussed in Report SSD 080-0038, Construction Analysis and are considered to represent a relatively minimum complexity construction and assembly process while utilizing the Shuttle orbiter capabilities.

This configuration imposes certain requirements on the replaceable modules such as the RCS, payloads, and GEO transfer engines. The modules must contain orbiter RMS grappling provisions, have compatible attach port physical mating provisions, and have compatible utility interface connection provisions.

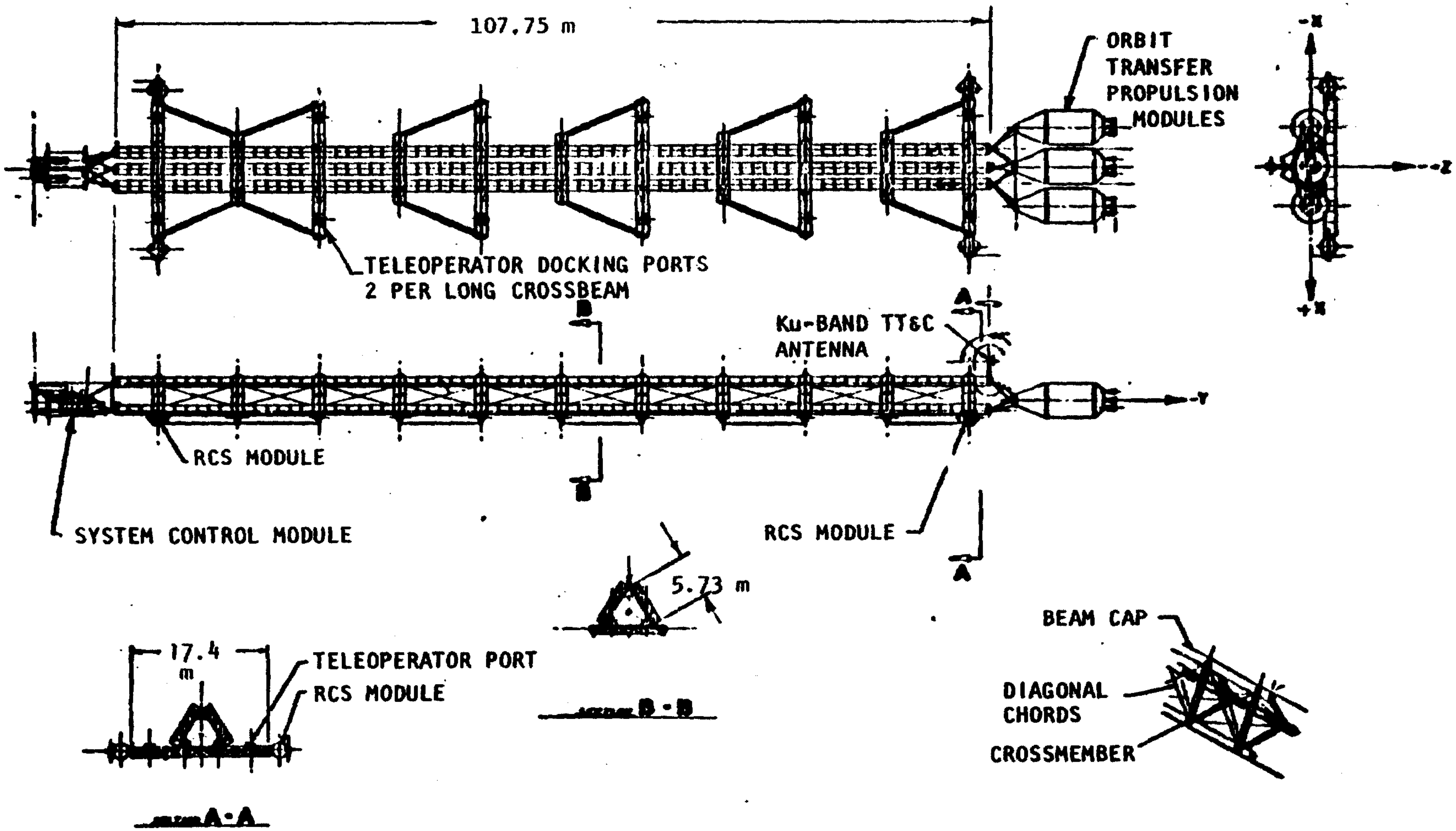


Figure 2.4.1-1. ETVP Configuration

The platform systems are all clustered in one location—the system control module (SCM). The individual subsystem installations on the SCM require the capabilities for remote servicing for GEO operations.

A detailed description of the platform, including a discussion of the features on the replaceable modules and of the SCM, is presented in Section 3.0.

2.5 SUBSYSTEM REQUIREMENTS

This section outlines the main functional and performance requirements for each of the ETVP subsystems. As indicated in Section 2.1, ETVP System Objectives, the requirements are mainly matched to the advanced communications technology development mission needs, but are also conceived to not preclude their adaptation to other technology missions. Further, they are based on the reference mission profile and ETVP configuration concept presented in Sections 2.3 and 2.4 respectively.

2.5.1 Structural Requirements

2.5.1.1 General Guidelines

Purpose

The ETVP structure shall provide the support and mounting for the ETVP subsystems and for the payloads. The subsystems shall include RCS, orbit transfer propulsion modules, the system control module, solar arrays and other equipment. The specific complement of advanced communication payloads shall be as discussed in Section 3.3.

The structure shall be designed to operate in LEO and in GEO and to transport the subsystems and payloads without detriment from one orbit to the other.

Fabrication

The structure shall be capable of being fabricated and assembled using the Shuttle Orbiter as a base in LEO. The structure components and material shall be stowable to efficiently utilize the weight and volume capabilities of the Shuttle Orbiter and to minimize the number of missions required for platform construction.

Automation

The ETVP structure shall consist mainly of elements (beams) fabricated automatically by the General Dynamics SCAFEDS beam builder. Other elements shall be used as required to implement the functional and structural requirements.

The structure shall be designed to facilitate automatic assembly and joining of elements and components, and for ease of installation of the ETVP subsystems and payloads.

Materials

The materials for the structure shall be selected to minimize thermal distortion.

2.5.1.2 Platform Structure

The structural requirements "significant" to the engineering technology verification platform structure configuration and the particular supports for subsystems equipment are described herein. The term "significant" refers to those requirements that directly affect the structure configuration, member arrangement, and sizing in a manner important to the construction fixture design and the construction operations.

Tri-Beam Structural Requirements

The structural requirements delineated herein apply to the basic platform tri-beam structure shown on Figure 2.5.1-1, consisting of the basic machine-made beam elements, the X-bracing system complete with intersection fittings, the thrust structure, support strut assembly, outrigger stabilization struts, and the subsystem attachment ports.

Construction Operations

The loads imposed on the machine-made beams during beam translation, beam joining, X-bracing installation and tensioning shall not induce any permanent detrimental deformation. The maximum imposed tension in any cord is 2360 N (530 lb) ± 150 N (34 lb).

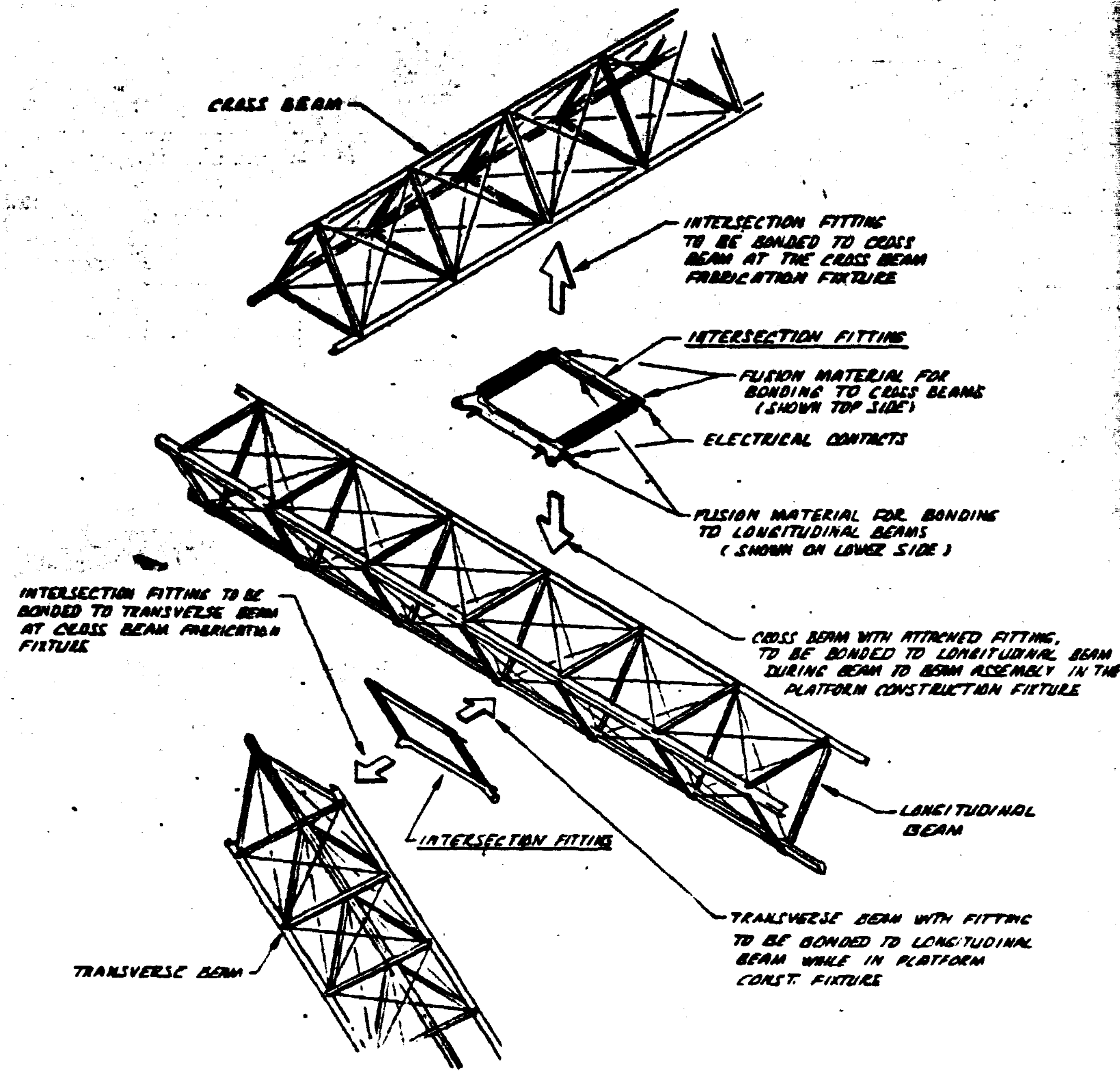
The tri-beam structural configuration shall be fabricated to the following dimensional tolerances:

- The departure from straightness along the length shall not exceed .005 m/m.
- The section angle of twist shall not exceed .01 degree per meter of length.

Attachment of subsystems equipment shall utilize "soft-docking techniques".

Orbit Transfer Maneuver

The structure shall sustain the loads induced during transfer from low earth orbit to geosynchronous orbit, in conjunction with both the associated pretension and thermal gradient induced loads. The thrust loading results from the particular platform and equipment mass distribution (Section 3.2.1) exposed to a T/W = .20. To preclude any significant load amplification, sequential startup of the thrusters shall be utilized. The thermal gradients used are the worst case of a 55°C difference between the individual caps in any machine-made beam, between the average of the three caps in each of the three machine-made beams, and between the average of the caps of each machine-made beam and the X-bracing cords. The machine-made beam cord pretension is



BEAM TO BEAM ATTACHMENT

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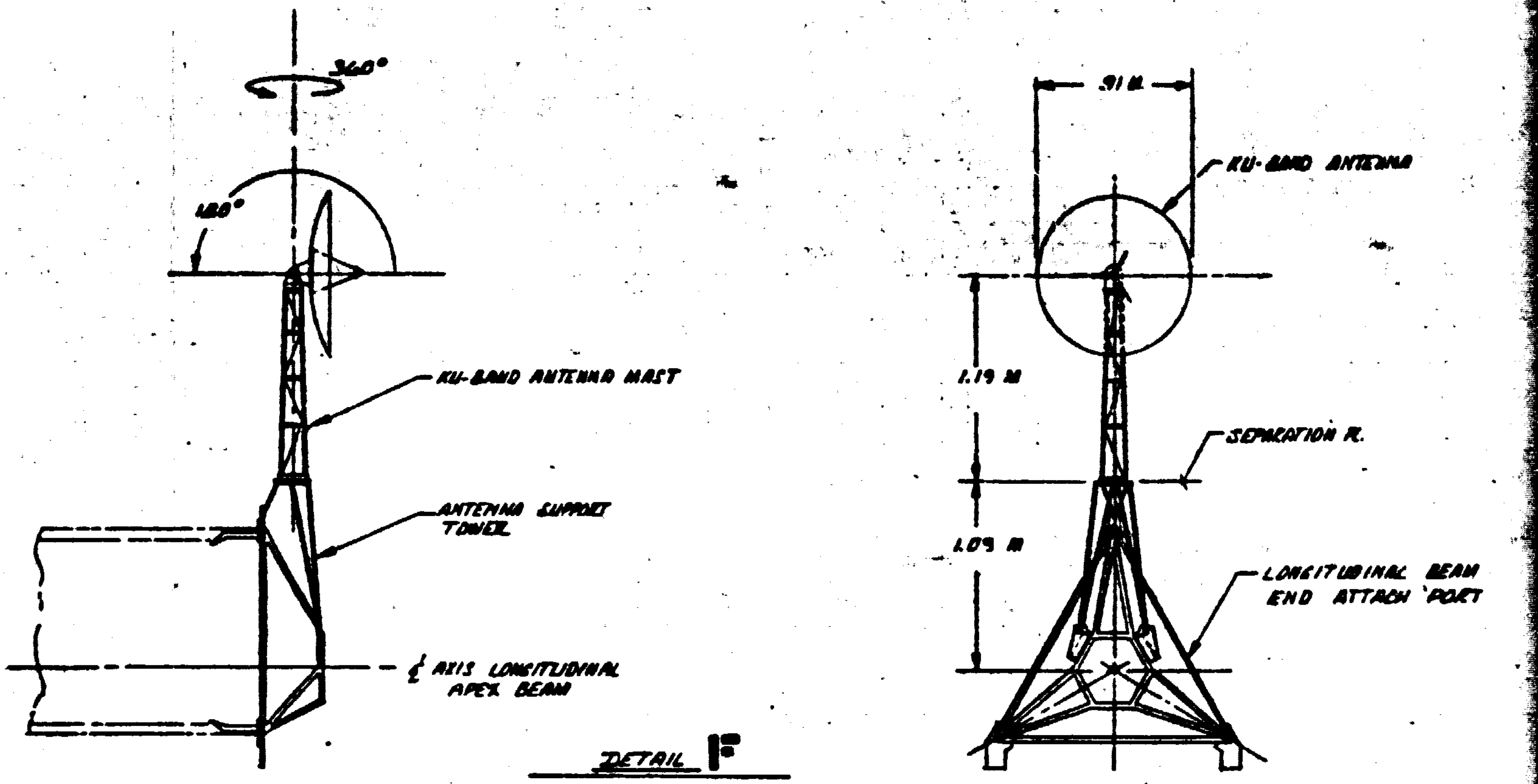
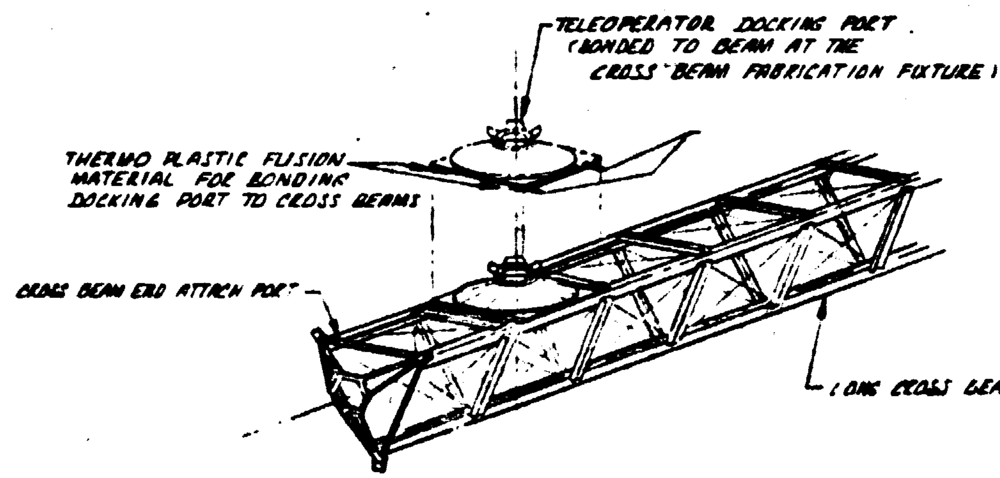


DETAIL F



DETAIL L

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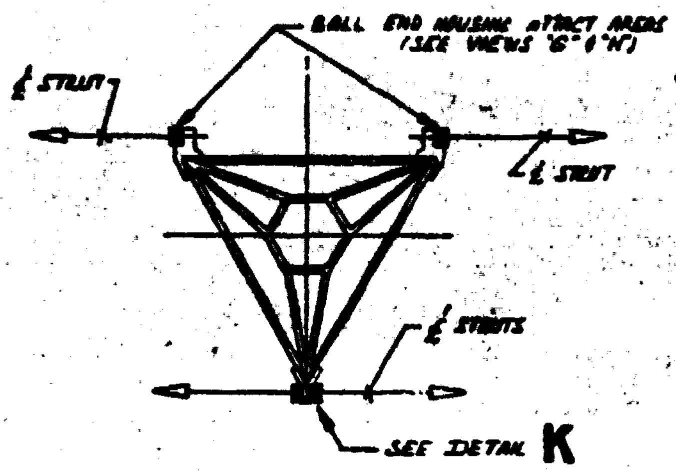
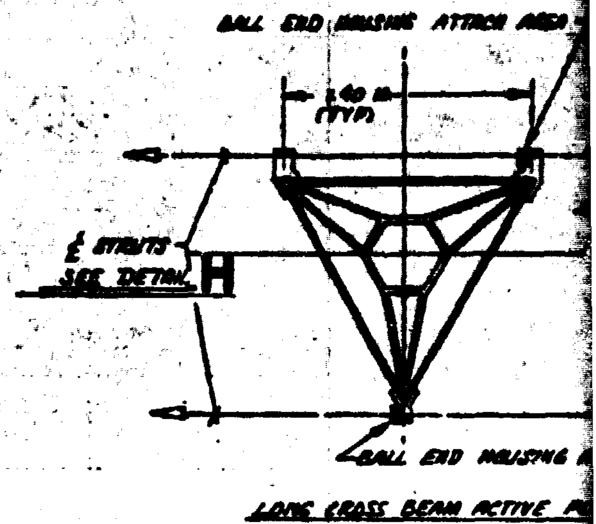
ANTENNA

...ON R.

...TUDINAL BEAM
ATTACH PORT



VIEW J - J



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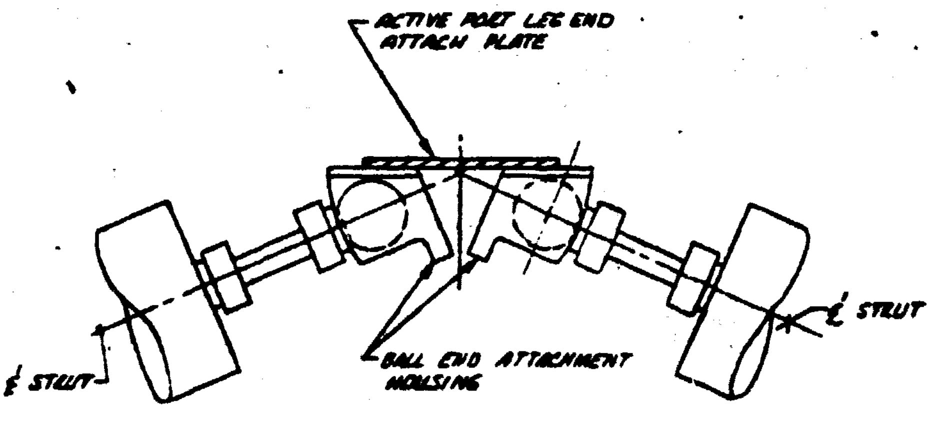

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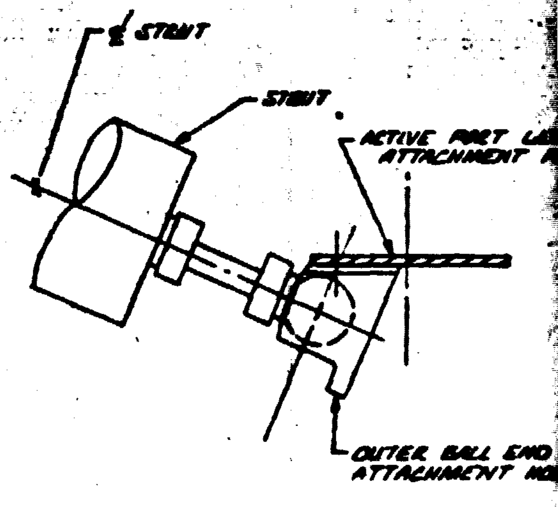

DETAIL H

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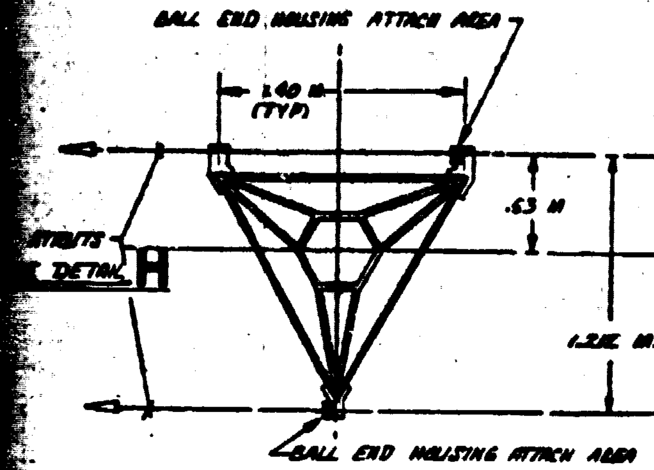
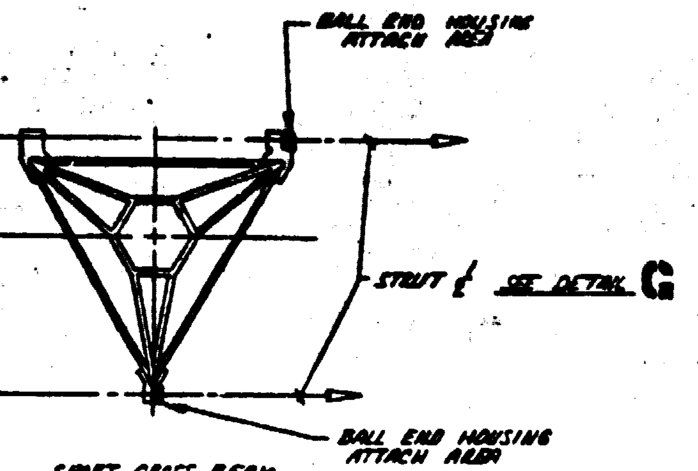


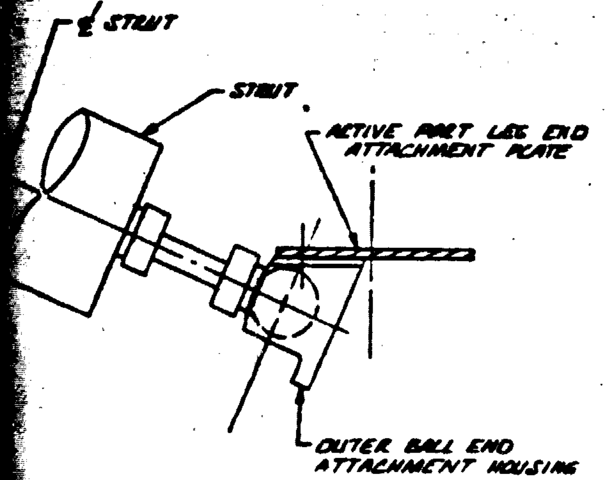
LONG CROSS BEAM ACTIVE PORT

VIEW D - D



SHORT CROSS BEAM ACTIVE PORT

VIEW E - E



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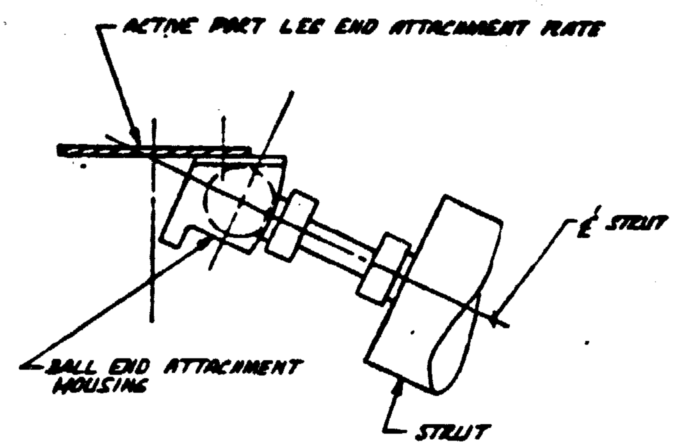

DETAIL G

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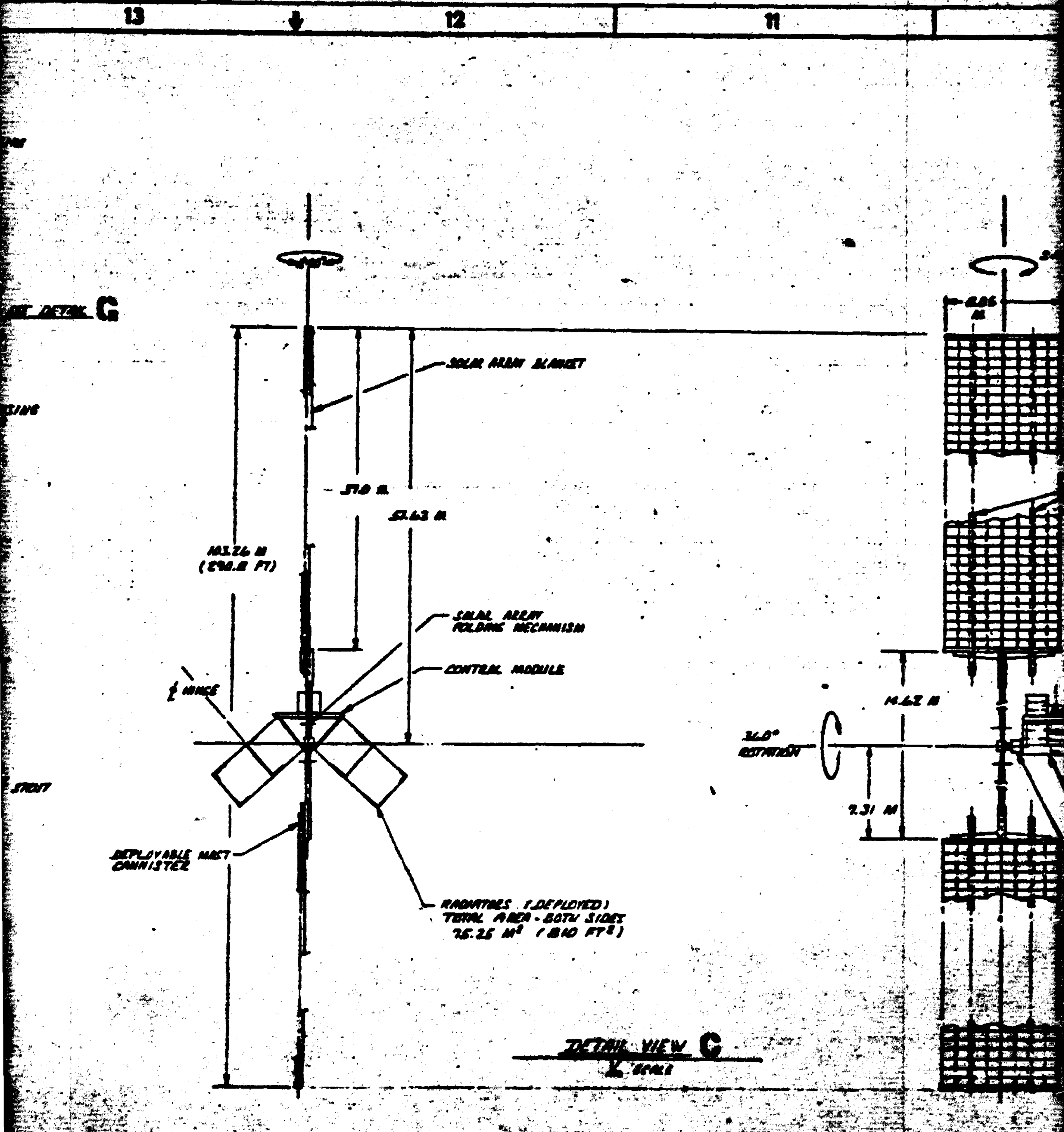

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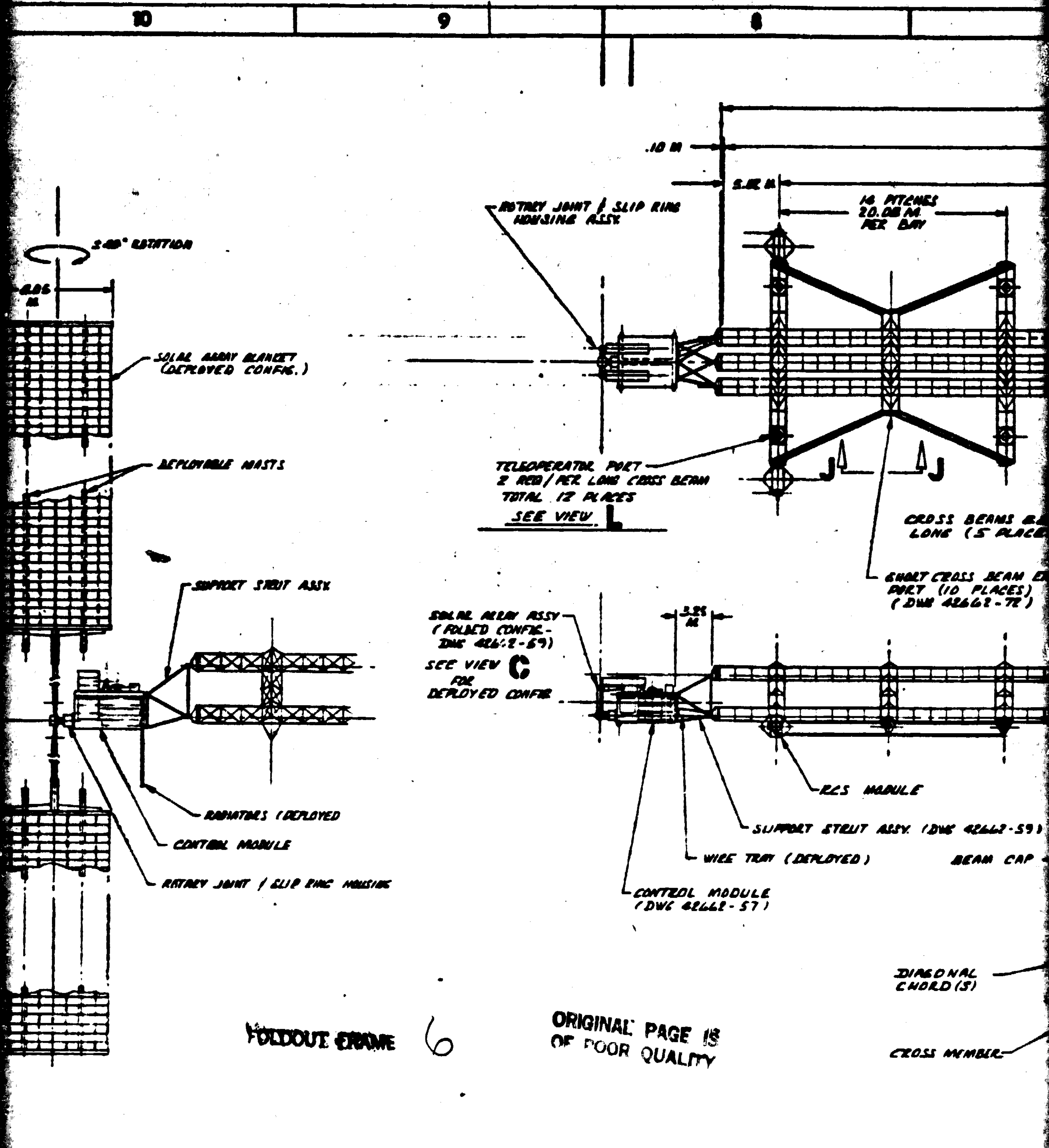

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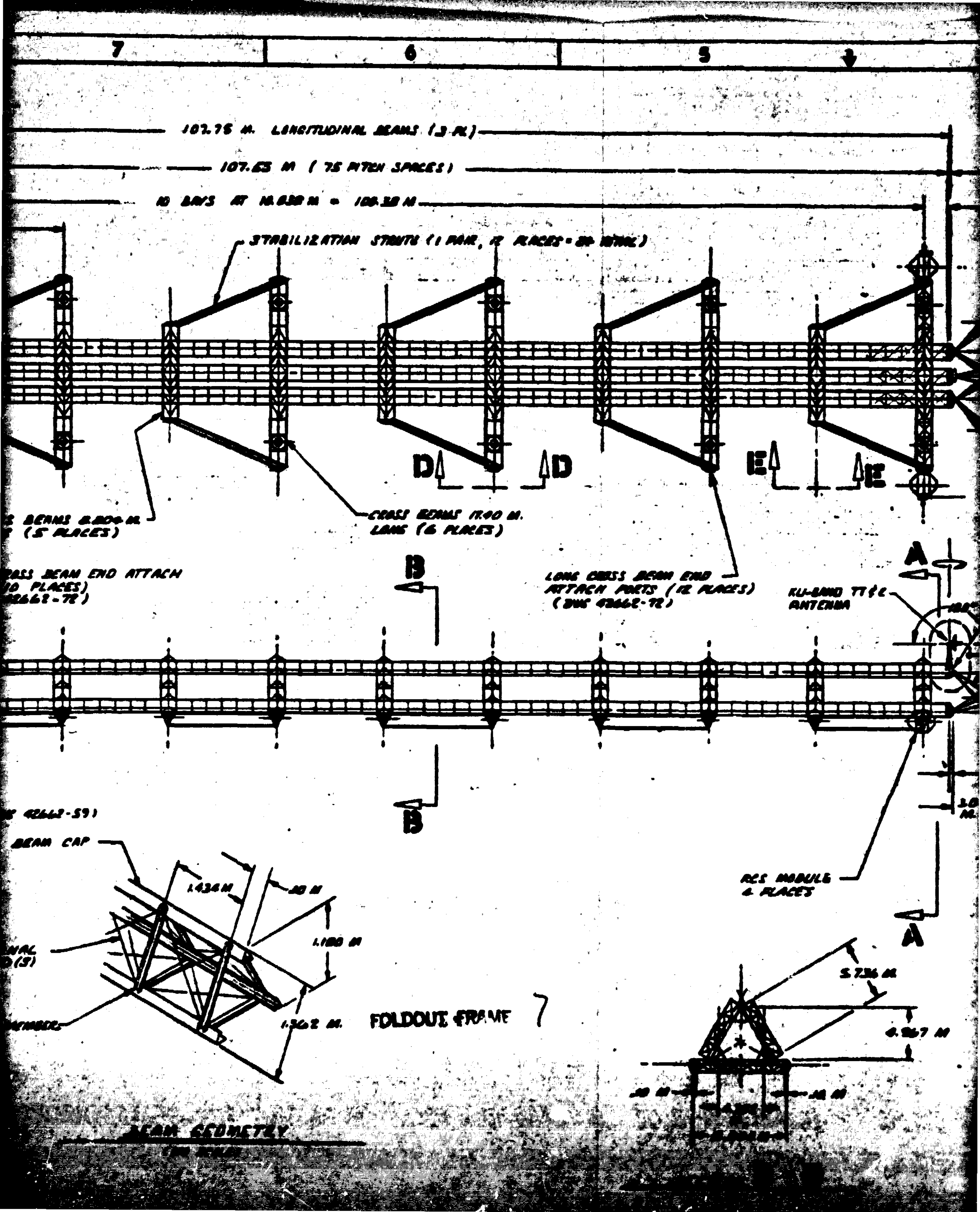
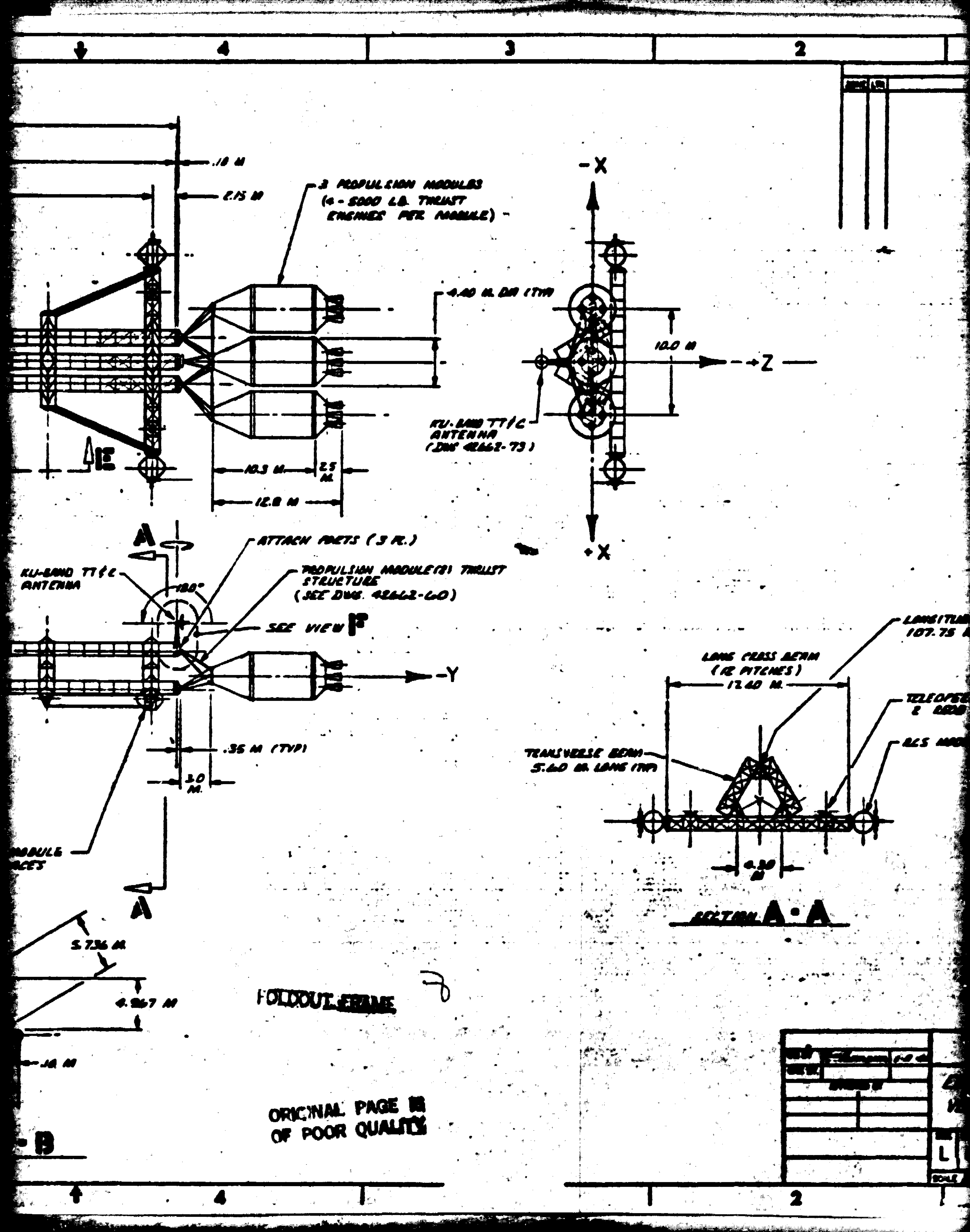
10 9 8


7
6
5
107.75 M. LONGITUDINAL BEAMS (3 PL)
107.65 M (75 PITCH SPACES)
10 BAYS AT 10.838 M = 108.38 M
STABILIZATION STRUTS (1 PAIR, 12 PLACES = 24 TOTAL)
D
D
E
E
CROSS BEAMS 17.40 M. LONG (6 PLACES)
(5 PLACES)
B
B
A
A
LONG CROSS BEAM END ATTACH PORTS (12 PLACES)
KU-BAND TT&C ANTENNA
RCS MODULE 4 PLACES
BEAM CAP
1.434 M
1.180 M
1.362 M.
5.736 M
4.967 M
BEAM GEOMETRY
FOLDOUT FRAME 7



.18 M
2.15 M
3 PROPULSION MODULES
(4 - 5000 LB. THRUST
ENGINES PER MODULE)
4.40 M. DIA (TYP)
-X
10.0 M
-Z
KU-BAND TT/C
ANTENNA
(DWG 42662-73)
10.3 M
2.5 M.
12.8 M
+X
A
ATTACH PARTS (3 PL.)
KU-BAND TT/C
ANTENNA
PROPULSION MODULE(3) THRUST
STRUCTURE
(SEE DWG. 42662-60)
180°
SEE VIEW F
-Y
.35 M (TYP)
3.0 M.
LONG CROSS BEAM
(12 PITCHES)
12.40 M.
TRANSVERSE BEAM
5.60 M. LONG (TYP)
4.30 M
SECTION A-A
5.736 M
4.967 M
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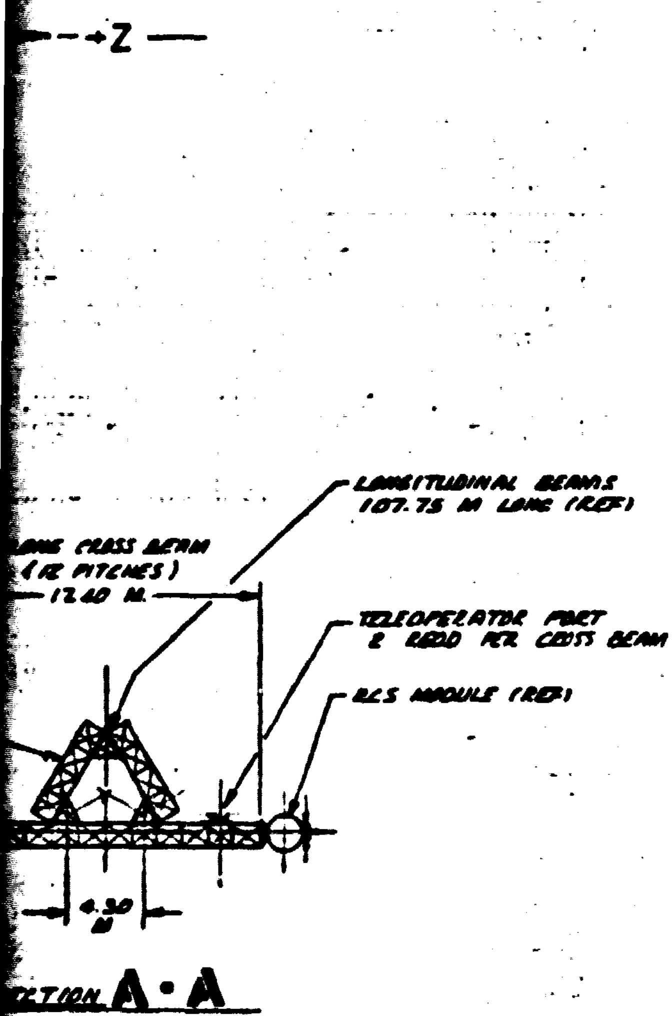

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Figure 2.3.1-1.

REVISIONS				
ZONE	LTR	DESCRIPTION	DATE	APPROVED

Rockwell International Corporation
Space Division
12214 Lakewood Boulevard • Downey, California 90241

ENGINEERING & TECHNOLOGY VERIFICATION PLATFORM

SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	42662-45A

SCALE 1/20 SHEET

42662-45A

180 N (40 lb). The tri-beam X-bracing peak tension is described above. The structure stiffness shall be compatible with the guidance and control system to minimize delta V losses to acceptable levels.

Operational Configuration

To permit the guidance and control system to maintain the antenna "short term" pointing accuracy within .03 degree, the overall structural configuration minimum modal frequency shall be greater than .005 Hz.

During operation of the communication antennas, the structural deformation of the platform structure shall be limited to preclude rotation between the communication antenna electrical axis and CMG reference axis to a value no more than six minutes of arc when combined with the antenna deformation. This value shall be maintained during the worst combination of the following conditions:

- Thermal gradients of the magnitude stated above.
- N-S stationkeeping maneuvers utilizing 44.5 N (10 lb) thrusters, E-W stationkeeping maneuvers utilizing 4.45 N (1 lb) thrusters.
- Attitude control maneuvers utilizing 4.45 N (1 lb) thrusters.

Both stationkeeping and attitude control thrusters shall be initiated in such a manner that any cumulative structural deflections do not exceed the six minute requirement specified above.

<u>Subsystem Structural Requirements</u>

The structural requirements delineated herein apply to the major supported subsystems components, i.e., the communication antennas, solar array panel and conductor runs, RCS propulsion components, rotary joints, and systems control module.

Construction Operations

The loads imposed on the subsystems equipment during all the construction operations starting with removal from the cargo bay to installation on the platform structure shall not induce any detrimental deformation in the structure subsystem.

The installation of electrical conductor lines shall accommodate the differential expansion and/or contraction of the lines relative to the supporting structural elements without detrimental deformation to either the structure or conductor.

Prior to the orbit transfer operation, the solar panels shall be stowed and secured to the tri-beam structure as shown on Figure 2.5.1-1. The communication antenna feed columns shall be maintained in the stowed position as shown on Figure 2.5.1-1 until completion of the orbit transfer maneuver.

Orbit Transfer Maneuver

All the subsystems defined above shall survive the orbit transfer thrust buildup, steady-state thrusting operations, and thrust termination segments of the orbit transfer maneuver without any detrimental deformation.

Solar Panel and Antenna Feed Column Deployment

Subsequent to orbit transfer thrust culmination and platform stabilization, the solar panels and communication antenna feed columns shall be deployed, to the configuration shown on Figure 2.5.1-1, without any detrimental deformation.

Operational Configuration

The communication antenna structural elements comprising the feed column and reflector structures in combination with platform deformation shall be designed to limit deflections of the antenna RF axis to six arc minutes during operation. The six minutes are to be maintained during the worst combination of 55°C thermal gradients and the stationkeeping and attitude control forces delineated above.

2.5.2 Electric Power Requirements

2.5.2.1 Introduction

The Electric Power System (EPS) shall provide all of the electric power for the operation of the ETVP in LEO, orbit transfer and GEO. Electric power during the construction phase and for ETVP systems checkout shall be provided by the Orbiter. Electric power for untended operations between Orbiter construction missions shall be provided by the construction fixture which remains attached to the ETVP until orbit transfer.

2.5.2.2 EPS Provisions

The EPS shall have the following provisions:

Solar Arrays

The primary source of power shall be solar arrays. The solar arrays shall be provided with two degrees of freedom:

1. A continuous 360° rotation about the longitudinal axis of the platform.

2. A ±40° "nodding" motion normal to the 360° rotation.

The solar arrays shall have a power output at beginning of life (BOL) of 60 kW at 258 Volts.

The solar arrays shall be capable of retracting and extending upon command, They shall be retracted for stowing and transporting in the Shuttle Orbiter, and for orbit transfer from LEO to GEO. The solar arrays shall be extended in LEO and in GEO.

Batteries

Batteries shall be provided to power the ETVP during eclipse periods. The batteries shall have the capability of being recharged by the solar arrays via charge/discharge control equipment.

Distribution System

The distribution system shall distribute power to the ETVP subsystems (including the RCS and the orbit transfer propulsion modules) and to the eight payload interfaces at the ends of the long crossbeams. The distribution system shall be capable of delivery 30 kW at each of the eight interfaces; additionally it shall be capable of combining two 30-kW power sources to provide a total 60-kW load capacity.

Payload Interfaces

The payloads shall interface with the ETVP via the attach ports at the ends of the long crossbeams. Each attach port shall contain the electromechanical and the electric equipment required to attach and operate a payload. The electrical equipment shall include relays for switching of power sources and converter/regulators to convert from bus dc voltage to the required payload voltage.

Control and Data Lines

Control and "Housekeeping Data" lines shall be provided to the ETVP subsystems and to the payload interfaces.

Wiring and Cabling Installation

Where possible, electrical wiring shall be pre-installed on the ground (e.g., in the SCM, the solar array and rotary joint assembly). The cabling along the platform longitudinal and the crossbeams shall be designed to facilitate automatic installation during the ETVP construction period. The long cable runs along the platform shall provide for differential thermal expansion.

Redundancy

Where feasible, redundancy shall be provided throughout the EPS. The power buses from the solar arrays and batteries to the ETVP subsystems and to the payload interfaces shall be redundant. Spare control and data lines shall be provided.

Checkout

Provisions shall be made for automatic checkout of continuity and function of all the ETVP subsystems and payload interfaces. The checkout shall be conducted from the Orbiter via a "drag cable". The SCM shall be provided with a PIDA type connector to interface with the "drag cable" and to provide the necessary checkout wiring.

2.5.3 Guidance, Navigation and Control Requirements

2.5.3.1 Introduction

The Guidance, Navigation and Control (GN&C) subsystem is divided into the following five functions:

- Attitude Reference—sensors that determine the attitude of the platform with respect to the local vertical or inertial space.
- Attitude Control—momentum exchange and mass expulsion systems that are used to orient the platform.
- Translation Control—mass expulsion systems used to provide thrust for velocity changes.
- Flight Control Computer—computer used to perform the computations to solve the GN&C system equations.
- Rendezvous and Docking—transponder and terminal ranging aid for GEO rendezvous and final precision closure/berthing control.

The purpose and makeup of the above functions are given in the following sections.

2.5.3.2 Attitude Reference

The attitude reference subsystems shall determine the attitude of the ETVP with respect to various reference frames. The sensor subsystem shall determine the attitude of the platform reference to an accuracy of 0.05 degree. The attitude reference hardware shall consist of:

1. Three inertial quality floated rate integrating gyros in a strapped down configuration.
2. Three strapped down star trackers with an accuracy of 18 $\widehat{\text{sec}}$. The three star trackers shall give the required accuracy of platform attitude information in both LEO and GEO from which the platform orientation to the local vertical and solar panel orientation can be derived. The strap-down gyros shall be used during star acquisition periods and during thrusting periods.

2.5.3.3 Attitude Control

The Attitude Control subsystem shall supply the torques that are used to control the attitude of the ETVP. There are two methods of applying torques to the ETVP.

- Momentum Control by CMG
- Mass Expulsion by RCS

The momentum storage shall be provided by 3 two-degrees-of-freedom CMGs of 10,500 nms capacity each. The CMGs shall be sized so that any two can absorb the cyclic and secular momentum of 21,000 nms accumulated by the platform during an orbit without disturbing the platform orientation.

The RCS, as used in the attitude control system, performs the maneuvers described in Table 2.5.3-1 and dumps the momentum absorbed by the CMG's during the orbital period. Momentum storage shall be sized to permit combined CMG desaturation with orbital stationkeeping maneuvers.

Table 2.5.3-1. RCS Attitude Control Requirements

LEO Requirements

	Stabilization	Attitude Error	Time
2 Yr	LVH, Y-POP	0.1 deg 5.0 deg	75% 25%
0.5 Yr	IH, Y-POP	0.1 deg 5.0 deg	75% 25%
0.25 Yr	LVH, Streamlined	5.0 deg	100%

Attitude Changes

3 - Single Axis/day	50% @ 0.03 deg/sec, 50% @ 0.1 deg/sec
2 - 3-Axis/day	50% @ 0.03 deg/sec, 50% @ 0.1 deg/sec

Orbit Transfer Requirements

8 - Single Axis 180° rotations @ 0.1 deg/sec
8 - ½ LEO period 3-axis attitude hold @ 5.0 deg
16 - 3-axis attitude rotations for target acquisition @ 0.1 deg/sec

GEO Requirements

Translation Maneuvers

E-W Stationkeeping and eccentricity control
N-S Stationkeeping
Initial positioning
Station repositioning (4)

Attitude Orientation and Control Maneuvers

4 - 3-axis maneuvers/yr @ 0.05°/sec
5 - 3-axis maneuvers @ 0.1°/sec
CMG momentum dump
30 day RCS backup

2.5.3.4 Translation Control

Translation control of small ΔV maneuvers shall be provided by the RCS. Large ΔV maneuvers for LEO to GEO shall be provided by the orbit transfer propulsion system.

The propulsion modules shall be capable of 3-axis attitude control during the thrusting maneuvers. The TVC system shall accept commands from the Flight Control Computer to steer the propulsion module thrust to give the required thrust direction changes. The TVC system shall contain all the electronic and electro/hydro/mechanical systems that are required to give satisfactory response and damping through the frequency range. The angular sweep of the TVC shall be ±5 degrees which shall encompass the limits of the travel of the platform center of mass.

The RCS shall be utilized to make small orbit correction maneuvers to counter orbit perturbation disturbances such as:

- Non-spherical earth
- Drag forces - Aero at LEO
  - Solar Pressure at GEO
- Uncertainty in orbit velocity corrections

The RCS shall be sized to perform the preliminary ΔV budget given in Table 2.5.3-2 along with the attitude maneuvers and stabilization schedule in Table 2.5.3-1.

2.5.3.5 Flight Control Computer

The Flight Control Computer shall consist of two basic sections, the flight system and the ground system. The ground based system contains the computing capability required for state vector computations, the star map for the star tracker, the computation of the ΔV requirements, and the dynamic model of the ETVP for comparison with the actual platform dynamics for verification of system performance. The on-board flight computer shall have the capacity of converting the ground based commands to the correct analog or digital format required by the sensors and actuators of the GN&CS. The on-board system shall have the capability of providing antonomous GN&C in case of temporary loss of ground based commands.

2.5.3.6 Servicing

The GN&C equipment shall be capable of being serviced in LEO and GEO. The primary mode of servicing in LEO shall be by the Orbiter. The primary mode of servicing in GEO shall be by Teleoperator or similar vehicle. Servicing shall include repair, replacement refurbishing and other tasks as required.

Table 2.5.3-2. RCS Translational Requirements

ORBIT MANEUVER nmi	km	ΔV REQUIRED m/sec	ft/sec
250-300		51.45	169
300-200		102.94	338
200-500		308.83	1014
500-200		308.83	1014
200-300		102.94	338
300-200		102.94	338
200-500		308.83	1014
500-200		308.83	1014
Sum		1698.58	5577
20% Contingency		339.59	1115
LEO TOTAL		2038.17	6692
GEO N-S Stationkeeping (7 yr)		349	1145
GEO E-W Stationkeeping (7 yr)		46	151
			1296
2 Station Changes plus return		23	75
GEO TOTAL		418	1371

2.5.4 Thermal Control Requirements

2.5.4.1 Introduction

The Thermal Control Subsystem (TCS) shall provide for thermal control of the ETVP to maintain structures and subsystems within acceptable temperature limits. The TCS shall include insulation, heat transport, heat rejection equipment and thermal coatings as required.

2.5.4.2 Payloads

No platform thermal control shall be provided for payloads. Payloads shall provide their own.

2.5.4.3 Thermal Control During Construction

In some cases, subsystem modules and/or components will be installed many days or weeks before the EPS system is activated. Platform power will therefore not be available for thermal control. Sensitive subsystem equipment shall be capable of 90 days of "cold soak" in the LEO space environment. If passive techniques are not adequate, provisions shall be made for interfacing an auxiliary power/thermal control package to maintain the required temperature during the "cold soak" period. This auxiliary power/thermal control package may be integrally designed into the subsystem unit or may be considered as special construction support equipment.

2.5.4.4 Thermal Control During Mission Operations

The thermal control subsystem shall provide adequate heat rejection capability to maintain subsystem temperatures during LEO, orbit transfer and GEO mission phases.

Structural temperature gradients shall be controlled to limit cyclic structural flexure and resulting misalignments (payload mount to platform attitude reference) to less than .05 degree.

2.5.5 Tracking Telemetry and Control (TT&C) Subsystem Requirements

2.5.5.1 S-Band

The TT&C shall provide S-band links for tracking and control in LEO, during orbit transfer and in GEO. The command data rate shall be a maximum of 72 Kbits/sec. The transmission capability (return link) for telemetry and data shall be 192 Kbits/sec. The TT&C shall have the capability to communicate simultaneously with the Shuttle Orbiter, (directly in LEO, indirectly in GEO) with GPS or with other spacecraft, see Figures 2.5.5-1 and 2.5.5-2.

2.5.5.2 Ku-Band

The TT&C shall provide Ku-band links in LEO to the ground through TDRS with approximately 95% availability in time. Forward commands shall have a maximum rate of 216 Kbits/sec and the return data rate shall be 50 Mbits/sec, see Tables 2.5.5-1 and 2.5.5-2.

2.5.5.3 Test Capability

The TT&C shall have built-in test capability for performing tests to aid in failure diagnosis for the ETVP subsystems. This shall be available for use in the checkout performed subsequent to the ETVP construction, and during LEO and GEO operations.

2.5.6 Reaction Control System (RCS) Requirements

2.5.6.1 Purpose

The purpose of the Reaction Control System (RCS) shall be to perform stationkeeping and attitude control maneuvers during LEO and GEO and to perform attitude control maneuvers during orbit transfer.

2.5.6.2 RCS Sizing

In the Clarke orbit (geosynchronous equatorial orbit) the RCS shall provide translation maneuvers to maintain a specified longitudinal position over the earth within 0.05 degree and for attitude control in conjunction with the translation burns.

The tankage shall be sized on the basis of a 7 year resupply interval. The estimated requirements are shown in Table 2.5.6-1.

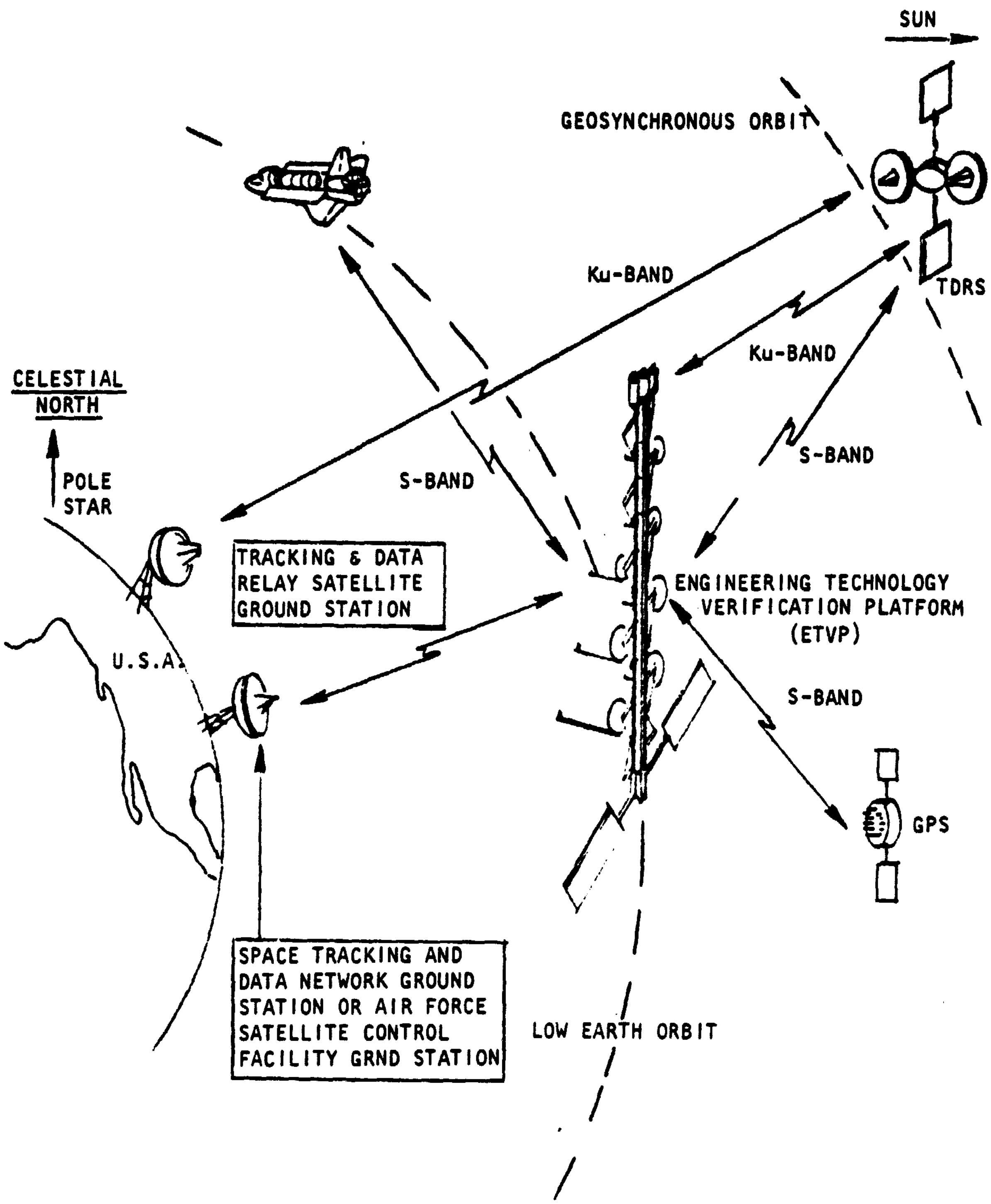

Figure 2.5.5-1. Communication Links for ETVP in Low-Earth Orbit

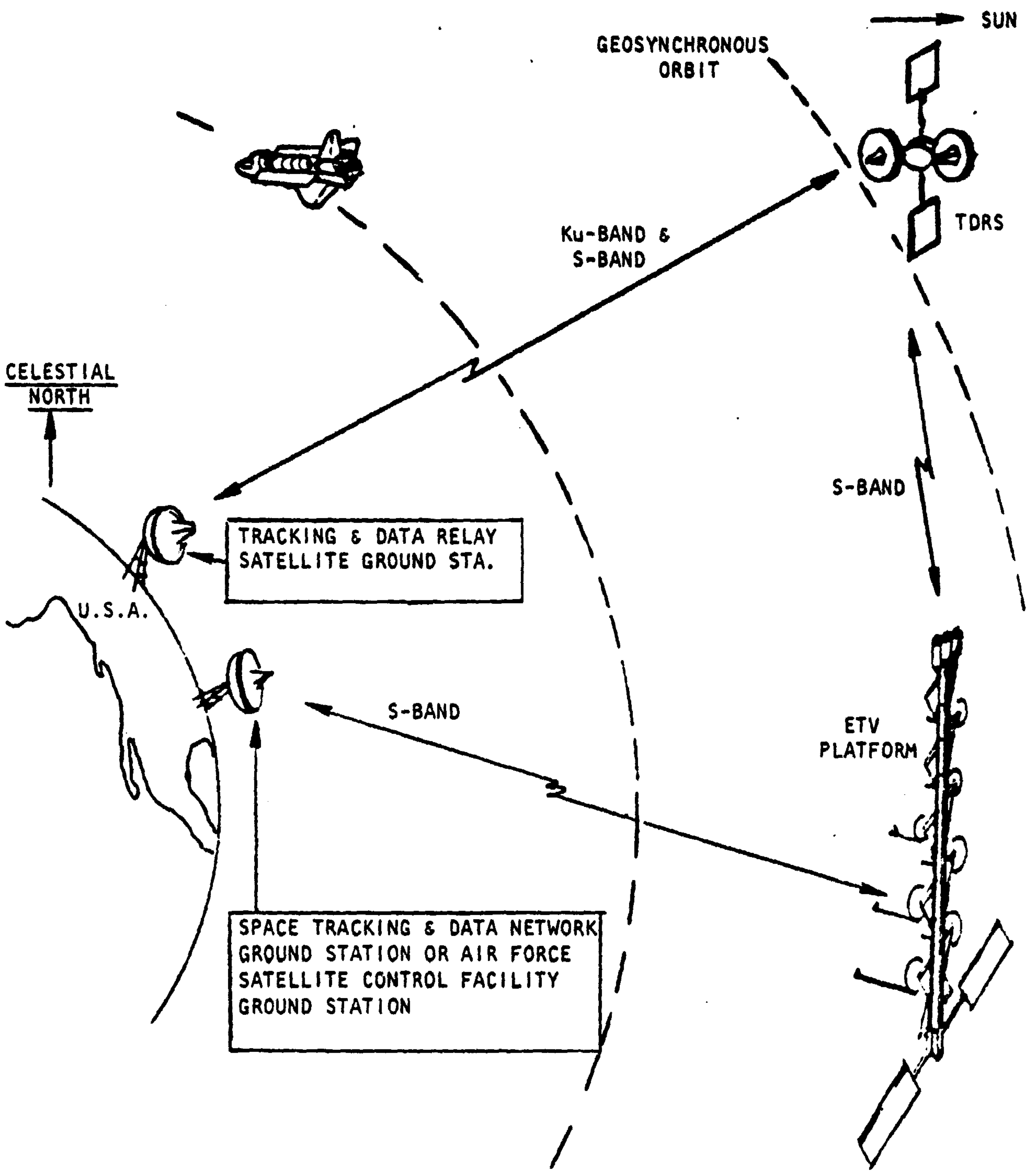

Figure 2.5.5-2. Communication Links for ETVP in Geosynchronous Orbit

Table 2.5.5-1. Link Capacity for Engineering Technology Verification Platform at LEO

LINK	ONE WAY OR TWO WAY?	FREQUENCY	DATA RATE	NOTES
S-LINK PM TO STDN OR SCF (OR ORBITER)	TWO	RETURN 2200-2300 MHz FORWARD 2025-2120 MHz	192 Kb/s 72 Kb/s	THIS LINK IS RELAYED TO STDN, SCF OR TDRS. SCF HAS 256 Kb/s (RETURN) RF POWER = 6 W ANTENNA = 4 HELICES
FM TO GROUND (STDN OR SCF)	ONE	RETURN 2250 MHz (10 MHz BANDWIDTH)	4 Mb/5	RF POWER = 10 WATTS ANTENNA = 0.27m HORN
PM TO TDRS (THROUGH TDRSS)	TWO	RETURN 2200-2300 MHz FORWARD 2025-2120 MHz	192 Kb/s 72 Kb/s	RF POWER = 100 WATTS ANTENNA = 4 HELICES
Ku-LINK TO ORBITER OR TO TDRSS	TWO	RETURN 14.85-15.15 MHz	50 Mb/x DIGITAL OR 4.5 MHz ANALOG TV 2 Mb/s PAYLOAD 192 Kb/s TELEMETRY	RF POWER = 50 WATTS ANTENNA = PARABOLIC (0.9 M DIAMETER) TIME COVERAGE: ≥95%
		FORWARD 13.75-13.80 GHz	216 Kb/s COMMANDS	HAS DATA & PN RANGE

Table 2.5.5-2. S-Band Link Capacity for Engineering Technology Verification Platform at GEO

LINK	ONE WAY OR TWO WAY?	FREQUENCY	DATA RATE	NOTES
S-BAND PM TO STDN, SCF OR TDRSS	TWO	RETURN 2200-2300 MHz FORWARD 2025-2120 MHz	192 Kb/s 72 Kb/s	RF POWER = 100 W ANTENNA = 4 HELICES
FM TO GROUND (STDN OR SCF)	ONE	RETURN 2250 MHz (10 MHz BANDWIDTH)	4/Mb/s	RF POWER = 10 W ANTENNA = 0.27m HORN

Table 2.5.6-1. RCS Propellant Requirements (7 Year Mission)

Translation maneuvers	m/s	(fps)	kg	(lb)
• East-west stationkeeping	14.3	(46.9)		
• North-south stationkeeping	349.0	(1145.2)		
• Eccentricity control	31.6	(103.6)		
• Initial positioning on station	6.0	(19.6)		
• Station repositioning (4)	22.6	(74.4)		
Total	423.5	(1389.7)	5303	(11666)
Attitude orientation and control maneuvers			kg	(lb)
• 4 - 3 axis maneuvers per year at 0.05 deg/sec			63	(138.6)
• CMG momentum dump			365.6	(804.3)
• 30 day RCS backup			4.5	(10.0)
• Transfer orbit attitude control 5-3 axis maneuvers at 0.1°/s			60.9	(134.0)
Total			494.0	(1086.9)
Grand total propellant carried			5796.8	(12752.9)
Total RCS weight			8073.6	(17761.9)
RCS weight per quadrant			2018.4	(4440.5)

2.5.6.2 RCS Location

The RCS shall be located to provide the optimum control moment and to minimize exhaust impingement on payloads and platform systems.

2.5.6.3 RCS Installation

The RCS units shall be designed to interface with the ETVP in the same manner as the payloads at the ends of the long crossbeams (Figures 2.2.5-1, 2.2.5-2, and 2.2.5-3).

2.5.6.4 Installation and Servicing

The RCS units shall be capable of being installed and serviced in LEO by the Shuttle Orbiter and in GEO by a Teleoperator.

Servicing may include repair or replacement of the RCS thruster units or components thereof. It shall include replacement of complete quad modules based on a seven-year resupply interval.

2.5.6.5 Choice of Propellants

The propellants selected for the ETVP RCS shall be compatible with the seven year resupply interval.

2.5.7 Orbit Transfer Propulsion System (OTPS) Requirements

2.5.7.1 Purpose

The Orbit Transfer Propulsion System (OTPS) shall be used to transfer the assembled ETVP with payloads from LEO to GEO.

2.5.7.2 Location and Mounting Interface

The OTPS shall consist of three modules mounted on the aft thrust structure. The interface between the OTPS modules and the ETVP shall be similar to the ETVP/payload interface, reference figures 2.2.5-1, 2.2.5-2, and 2.2.5-3. Power and data lines shall be supplied from the ETVP to the OTPS through each attach port as follows:

1. 2 pairs of power lines, 2 kW dc

2. 4 twisted shielded pairs (Housekeeping)

3. 2 twisted shielded pairs

2.5.7.3 Transportation and Installation

Each OTPS module shall be capable of transportation to LEO in the Shuttle Orbiter and of being installed on the ETVP using the RMS and the construction fixture.

2.5.7.4 Propellant Storage Life

The quantity of propellant shall allow for boil off over the total storage life which includes the shuttle turn around time between missions for installing the three modules.

2.5.7.5 Thrust Vector Control (TVC)

Thrust Vector Control shall be obtained in pitch, yaw and roll by gimballing the engines in the OTPS modules.

The modules shall be sized to the largest single module which can be delivered to orbit by the Shuttle (28,860 kg). Three such modules can deliver up to 41,100 kg from LEO to GEO. Propellants can be off-loaded for platform weights below this value.

2.6 ETVP SERVICING REQUIREMENTS

2.6.1 Design Life

The ETVP shall be designed for an overall life of 20 years with a span of 7 years between normal resupply missions.

2.6.2 Scope of Servicing

ETVP servicing shall include the following:

1. Normal resupply of expendables (RCS fuel, batteries).

2. Removal/replacement of payloads. Payloads may be replaced because of updated technology, new technology, and breakdown.

3. Removal/replacement of subsystems or components.

4. Repair of subsystems.

5. Repair of payloads.

6. Switching of redundant elements built into the platform.

2.6.3 Resupply, Replacement and Repair

The ETVP shall be designed for ease of resupply, replacement and repair by use of the Shuttle Orbiter in LEO and by the Teleoperator in GEO. Docking facilities shall be provided for access to all areas of the ETVP. The primary mode of docking in LEO shall be by means of the construction fixture which shall have the capability of translating back and forth along the platform. Provisions shall be made to enable the docked Shuttle Orbiter to rotate a minimum of ±360° about the axis of the docking port. Because the construction fixture will not be available in GEO, docking facilities shall be provided on the platform for the Teleoperator. Such docking facilities shall be passive, i.e., they shall not be equipped with active latches or with electrical interconnects. The Teleoperator shall be the active vehicle for GEO servicing.

Components which are included in the remove/replace category shall be compatible with the Shuttle Orbiter RMS and with a similar manipulator system on the Teleoperator. Suitable grappling fittings and targets shall be provided to enable the RMS/teleoperator to locate and grasp the component. The use of a PIDA type mounting for components shall be considered where appropriate. Where electrical connectors require make and break actions the design shall ensure the correct alignment of the pins before mating.

2.6.4 Failure Diagnosis

The ETVP subsystems shall have built-in failure diagnosis capability. The failure diagnosis system shall:

1. Switch redundant components where such capability exists.

2. Advise ground control by TT&C of failures and/or corrective action taken.

2.6.5 Rendezvous

The ETVP shall be provided with suitable equipment to enable the Shuttle Orbiter and the Teleoperator to locate and rendezvous with the platform.

2.6.6 Redundancy

It is realized that initial concept definition studies cannot rigorously deal with the many detailed issues involved in the massive three-way trade of project cost/funding, component life/redundancy and various on-orbit servicing approaches. However, the baseline system definition must include the identification of the basic elements of these issues in order to be complete. It is necessary as part of a practical ETVP design to establish first order estimates of the total subsystems hardware complement, including redundancy, which must be delivered and installed as part of the space construction process. Thus, a set of initial guidelines to be applied to the ETVP system was formulated. The system requirements related to these issues can thus be identified and while the assumptions applied here will be the subjects of much future analysis during Phase B and C studies these preliminary requirements will meet the desired design traceability objective set for the study.

The principal guidelines are summarized in paragraph 2.6.6.1 along with the key supportive rationale. The application of these guidelines to the individual subsystems are presented in paragraph 2.6.6.2.

2.6.6.1 Principal Guidelines and Rationale

Servicing Guidelines

1. The ETVP design concept shall be compatible with unmanned remote servicing approaches.

   Rationale: This is a basic "given" requirement grossly specified in the contract SOW and later clarified in discussions with the customer. It recognizes the eventual need and benefits to be gained with on-orbit servicing and was input to the study to generate an initial understanding of its impact on platform design and related space construction processes. Manned servicing will likely be employed during the LEO phases of platform operations, but the unmanned remote servicing approach was specified as a design requirement in recognition of the high costs and potential early unavailability of manned servicing to GEO.

2. Special operating modes or conditions shall be permitted during servicing operations.

   Rationale: The ETVP platform is a developmental tool designed to carry a variety of experimental payloads. The value of the continuation of experiment activity during servicing operations was not felt to warrant the probable cost. Brief interruptions in experiment activity are not expected to have serious impacts on the national space program or even the project utilizing the platform. This is in contrast to high capacity commercial services such as the communications platform in Part I of the study in which interruptions could affect thousands of users. The main requirement affecting the design of the servicing concept is that all essential functions be maintained and/or the ability to recover from an outage of the function is provided.

3. For current study purposes the design servicing interval shall be based on RCS consumables as limited by GEO delivery capability of three orbit transfer propulsion modules.

   Rationale: The main issues affecting servicing interval are beyond the scope of the study. They include technology growth rate and related update requirements and intervals (mainly payloads, but possibly subsystems also), redundancy and component MTBF factors, and costs of increasing subsystems life versus costs of logistics and servicing operations. Since these issues cannot be fully treated here the arbitrary consideration of sizing to the lowest feasible number of propulsion modules was selected as a cost driven basis for establishing the servicing interval. With this approach the maximum RCS propellant load which can be delivered on the platform to GEO with three orbit transfer propulsion modules will be used. Based on preliminary platform weights and RCS consumption estimates this is expected to yield a 5 to 7 year consumables life at GEO. This is generally consistant with current estimates of desired technology update intervals for communications and other developmental payloads. Thus, this approach is reasonably compatible with payload update requirements and assures a low cost platform within the gross definition practice of initial concept studies.

Redundancy Guidelines

1. The ETVP platform system shall be designed with one tier of redundancy.

   Rationale: This requirement is driven by program cost implications, particularly "front end" costs, and also by the fact that the platform is to be designed for on-orbit servicing. High levels of redundancy and long life components lead to high program costs. Since the platform is designed to be serviced in space we can capitalize on this feature by designing to only one tier of redundancy, thereby reducing program costs.

2. The ETVP platform system shall be designed to a fail safe criteria. Options shall be permitted at the subsystems level to incorporate fail operate capability if it is inherent in the subsystems design or can be included at virtually no added cost.

Rationale: Again, this guideline is driven by program cost considerations, but it also recognizes that experimental programs are inherently more tolerant of interruptions for servicing than systems providing high capacity commercial services. The cost balance between costs to assure non-outage versus costs due to an outage will invariably shift toward the outage tolerant side for experimental programs. This is mainly due to the lower cost impacts of outages with these types of programs. No commercial revenues are being lost.

Note: There is one exception to this single tier, fail safe design guideline. This is in the electrical power supply to the TT&C subsystem. To assure at least diagnostic contact with the platform can be maintained in the event of a serious primary power failure a second tier of redundancy is required. This will provide emergency backup power to the TT&C subsystem so ground based diagnostic teams can analyze platform status and determine a workaround solution to the failure situation.

Packaging Guidelines

The following general rules shall be considered in establishing packaging arrangements for platform subsystems (particularly those in the control module). They are in part based on or derived from the redundancy/servicing issues discussed above. They would also be affected by the design of the servicing system which is undefined for this study. However, replaceable unit commonality and access provisions inherently acknowledge potential servicing system influences on the platform design. The main packaging considerations to be applied are:

1. Isolate redundant elements from primary units to allow independent servicing of primary and backup units.

2. Consider grouping of backup elements into single replaceable units in conjunction with commonality of replaceable modules.

3. Provide reasonable access to each replaceable unit considering unamnned remote servicing modes.

4. Consider commonality of replaceable units to reduce design complexity of the servicing system. Commonality can include such features as size/shape, mechanical latching mechanisms, electrical connectors and actuation concepts, installation alignment aids and module retention interfaces with the servicing system.

2.6.6.2 Guideline Applications to Individual Subsystems

This section translates the general platform system guidelines defined above into specific requirements for each of the subsystems. The main elements of each subsystem are identified along with specific requirements for redundancy and/or replaceability.

Guidance, Navigation and Control Subsystem

1. Precision attitude reference package (includes star trackers, gyros and accelerometers).

   Requirement: Design as a single (non-redundant) replaceable unit. Fail safe redundancy to be provided by a backup gyro pack plus sun sensor.

2. Backup attitude reference package.

   Requirement: Design as a single replaceable unit. This is the fail safe redundancy identified above.

3. Coarse attitude reference unit (includes sun sensors and magnetometer).

   Requirement: Design as a single replaceable unit. This unit aids initial setup of precision pointing modes, but is non-essential in that other techniques using other measured parameters (solar array sun sensors, antenna gimbal angles, etc.) can be used in conjunction with ground based analysis to perform this function.

4. Control Moment Gyros (CMG's).

   Requirement: For baseline design use three two-axis CMG's sized for any two to do the mission. This gives fail ops capability. All units are to be space replaceable. An option exists to utilize a larger number of existing CMG's sized to smaller missions. This could save program costs. Similar redundancy (50%) should be applied with this option.

5. Guidance and Control Computer.

   Requirement: Design for dual replaceable units. This yields fail ops capability and will assure a stable target for servicing revisit operations (in event of primary computer failure) and will allow continuation of vital control functions during servicing operations.

   Note: Rapid advances in solid state technology including adaptive self-healing computers may make the GN&C computer a candidate for exclusion from the redundancy/replaceability requirement. Twenty year MTBF computers would not require replacement.

Electrical Power Subsystem

1. Solar Array Wings.

   Requirement: Design for the solar arrays to be integral with the rotary joint (non-space replaceable). The system is comprised of four PEP wings (two PEP systems) thereby having inherent redundancy. The expected life in the combined LEO/GEO platform environment is sufficient to perform the mission without space replacement.

2. Rotary Joint.

Requirement: Design as an integral unit with the solar arrays as above. The combined assembly is to be designed for space installation and hence could be replaced in space. However, sufficient internal redundancy in both conductor paths and motor driven units will be incorporated to preclude the need for planned on-orbit replacement.

3. Main Power Buses.

Requirement: Design for dual main power buses. This will provide full redundancy. Space replacement of electrical/signal lines is not planned. A second level of redundancy is required in the form of an emergency backup power supply for the TT&C subsystem. This will allow diagnostic contact to be maintained with the platform in the event of a major power failure as discussed earlier.

4. Energy Storage.

Requirement: The baseline design utilizes seven batteries to provide maximum continuous "day-night" power in GEO operations. This inherently provides at least fail-safe capability in that the payloads can time-share the remaining available power in the event of a battery failure. If the payloads actually flown do not require the full power of the platform, complete redundancy may exist and time-sharing may not be required. Battery charger units are integral with the battery module and thus provide the same level of redundancy. Battery modules are to be designed for space replaceability.

5. Power Conditioning.

Requirement: The baseline design requires payloads to provide their own converter/regulator functions. For platform subsystems (housekeeping functions) dual converter/regulators shall be provided in concert with the dual main power buses discussed above. Each C/R unit will be space replaceable. A third C/R unit will be required for the emergency backup power system supplying the TT&C subsystem. This unit maybe integral with the TT&C subsystem and would be replaceable with it.

6. Load Isolation Switches.

Requirement: In the baseline design these are integral with the payload electrical interface unit and are pre-installed on special payload attach ports. They shall be designed with internal redundancy so that space replacement is not required.

Tracking, Telemetry and Command Subsystem

1. S-Band Communication Unit.

Requirement: Design for dual replaceable units. This system provides the vital data and command link for ground control of the platform. It would provide essential housekeeping data for ground workaround analyses in event of major platform failures. It also provides the vital command link which may be required to establish a stable "safe" rendezvous target for servicing revisits.

2. K-Band Communication Unit.

Requirement: Design as a single (non-redundant) replaceable unit. This system provides the wide-band data stream in support of payload operations. Some redundancy for this data stream is inherent in the payloads themselves (such as the communications technology payloads, etc.). Even if some payload data is lost due to a failure in this system it would not affect the health of the platform and could be restored with a servicing mission. Thus, in the interest of lower program costs only a single replaceable unit is specified.

3. Central Processor.

Requirement: Design for dual replaceable units. The central processor is vital to the command and control of the platform, similar to the S-band unit above.

3. Recorder.

Requirement: Design for dual replaceable units. Redundancy is required to assure essential diagnostic data will be available.

Thermal Control Subsystem

1. Internal Freon Loops.

Requirement: Design for sufficient system redundancy to meet platform design life. This system is internal within the control module and thus highly protected from the meteorite environments. To avoid the complexity/cost associated with designing for pump replacement (with fluid couplings) built in redundancy is specified.

2. Heat Pipe Radiation.

Requirement: Design for sufficient initial surface area to meet heat rejection requirements over platform life span allowing for expected heat pipe damage from meteorites. Heat pipe radiators are very damage tolerant thereby imposing only a small increase in surface area to yield a long service life. Thus, the complexity/cost factors associated with space replaceability are also eliminated for the radiator system.

3. RCS Heaters.

Requirement: Design for sufficient redundancy to meet RCS quad service life. The system would be replaced with the installation of new quads after propellant depletion. It is integral with the quad design.

Reaction Control Subsystem

1. Basic Quad Unit.

Requirement: Design for space replaceability. The 4-quad system utilized on the platform is inherently redundant, i.e., any three quads operative can perform the required stationkeeping and CMG desaturation functions. However, the propellant remaining in a failed quad at the time of the failure would be lost from the mission requiring earlier than planned RCS replacement servicing.

2. RCS Thrusters.

Requirement: Design for redundant thruster nozzles if required to meet thruster life/burn time limits. The thrusters are an integral part of the quad unit and will be replaced with it.

2.7 GROWTH REQUIREMENTS

There are two types of growth considerations which must be applied to the ETVP: (1) built-in extra initial capacity, and (2) design adaptability to future sizing.

The built-in extra initial capacity refers to oversizing the system capacity to support additional payloads and/or higher than estimated support needs for individual planned payloads. In the current ETVP concept this includes extra payload attach ports with their electrical and signal interface services. Also, electrical power service at each payload attach location includes two 30 kW power buses which can be combined to provide a total of 60 kW at each location. This allows the flexibility for installing a high power payload at any location. Other forms of built-in extra capacity could be the capability for additional flight modes such as special inertial hold orientations and/or more precise platform pointing capability.

The second growth consideration, design adaptability, addresses the ability of the basic design concept to accept resizing without greatly affecting the construction technology. For example, the length of the tri-beam structure could be increased to 200 meters and beyond with no impact on the basic construction processes. The same construction fixture and subsystems installation techniques could be used. Similarly, electrical lines, J-boxes and other elements could be resized (within reasonable limits) and still utilize the same installation techniques and their related construction support equipment.

On the other hand, changing the depth/spacing of the tri-beam members would have a significant impact on the construction technology. The detailed mechanism for welding/joining space fabricated beams would likely be the same, but the construction fixture design would require resizing along with tension cable installers, etc.

Thus, early attention must be given to growth considerations in the design of the ETVP to assure adequate concept adaptability to future mission objectives and to provide sufficient initial capacity for a range of expected payloads.

3.0 PLATFORM DESIGN

3.1 OVERALL CONFIGURATION

This section presents the design definition of the ETVP system which meets the requirements specified in Section 2.0.

3.1.1 General Description

The engineering test and verification platform (ETVP), as illustrated in Figure 3.1.1-1 is configured to accept various communication antennas for development testing. With appropriate modifications, the ETVP can also be configured as an SPS test article utilizing the structure to support solar array panels as illustrated in Figure 3.1.1-2.

The principal portion of the platform is 107.7 m long. Recent end-to-end construction analysis results suggested significant construction system and operations benefits would result from this slightly downsized platform. This change eases critical remote manipulator reach constraints. Thus, the final configuration is the downsized version which is compatible with the construction system definition and timelines presented in SSD 80-0038.

Although much of the subsystem data presented elsewhere in this report are based on a 136 m platform, no significant impacts on subsystem sizing and/or concept feasibility are expected. Electrical wire runs will be shorter and certain attitude control elements might be slightly reduced in size, but not sufficiently to alter orbiter bay packaging considerations or the handling of these elements within the construction process. Thus, in the interests of focusing study resources on the main study objective of space construction, the design interactions refining all subsystems to this new dimension were not performed. However, the basis for each subsystem design is clearly presented and future refinements and requirements can easily be made.

Eight attaching ports provide for the attachment of antennas, or provide the attachment of structural support elements for very large antennas. The ports are located on the ends of the crossbeams which are 17.4 m long and are spaced 20 m apart. This arrangement will accommodate a group of antennas up to 20 m in diameter. Appropriate power, data, and signal interfaces are also provided at each port.

Electrical power is provided by four solar array panels generating approximately 60 kW. The solar array panels are located adjacent to the system control module (SCM). A trade study was performed which investigated symmetrical versus asymmetrical solar array arrangements. The asymmetrical arrangement as illustrated in Figure 3.1.1-2 reflects the results of the trade. Appendix A documents this trade study. Power conditioning and power storage batteries are mounted on the SCM. CMG's, sensors, and their appropriate controls, and TT&C equipment are also mounted on the SCM.

Four RCS engine modules are located at the extremities of the platform; these units provide both attitude control and translation thrusting capability.

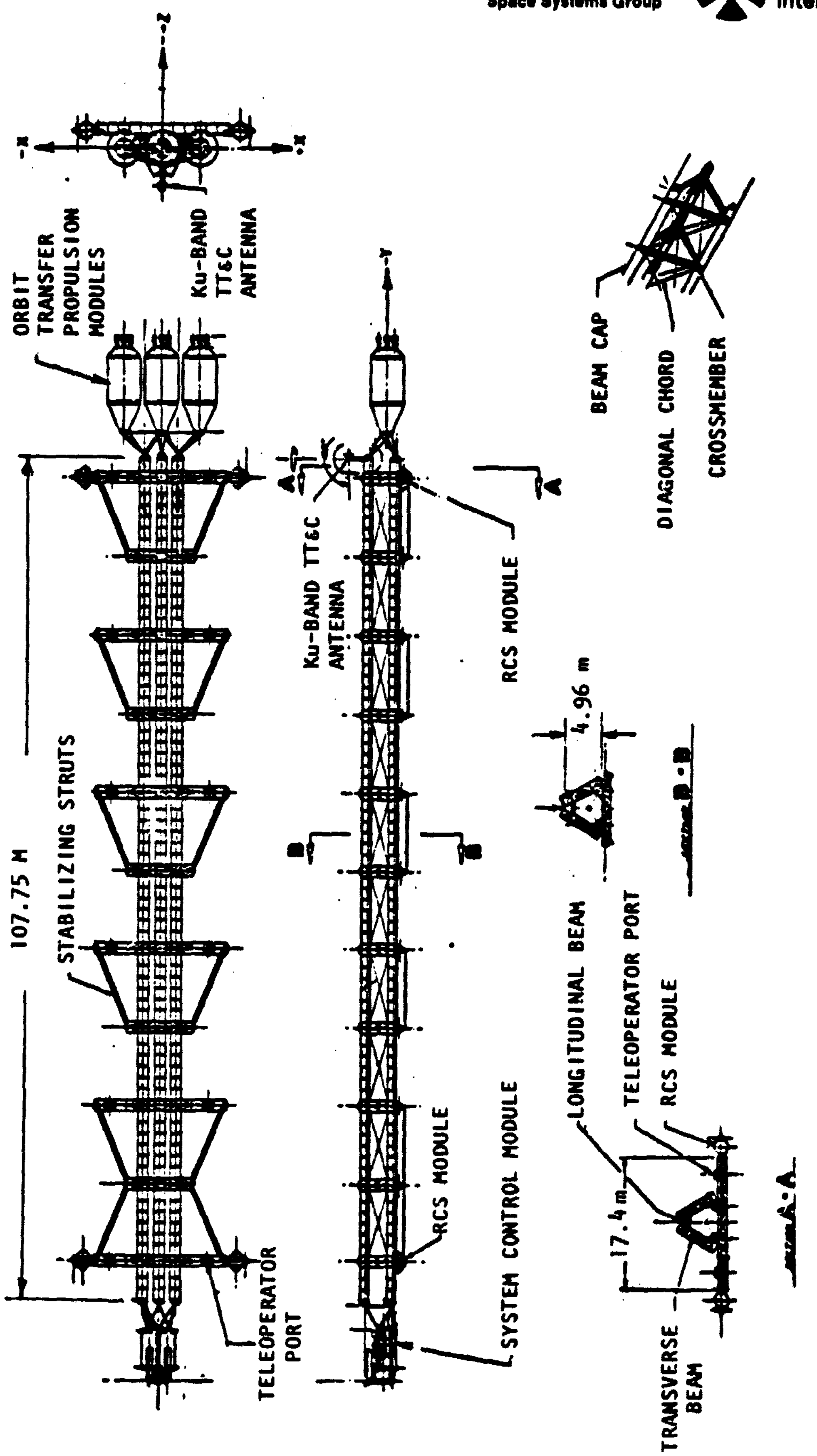

Figure 3.1.1-1. Engineering Technology and Verification Platform

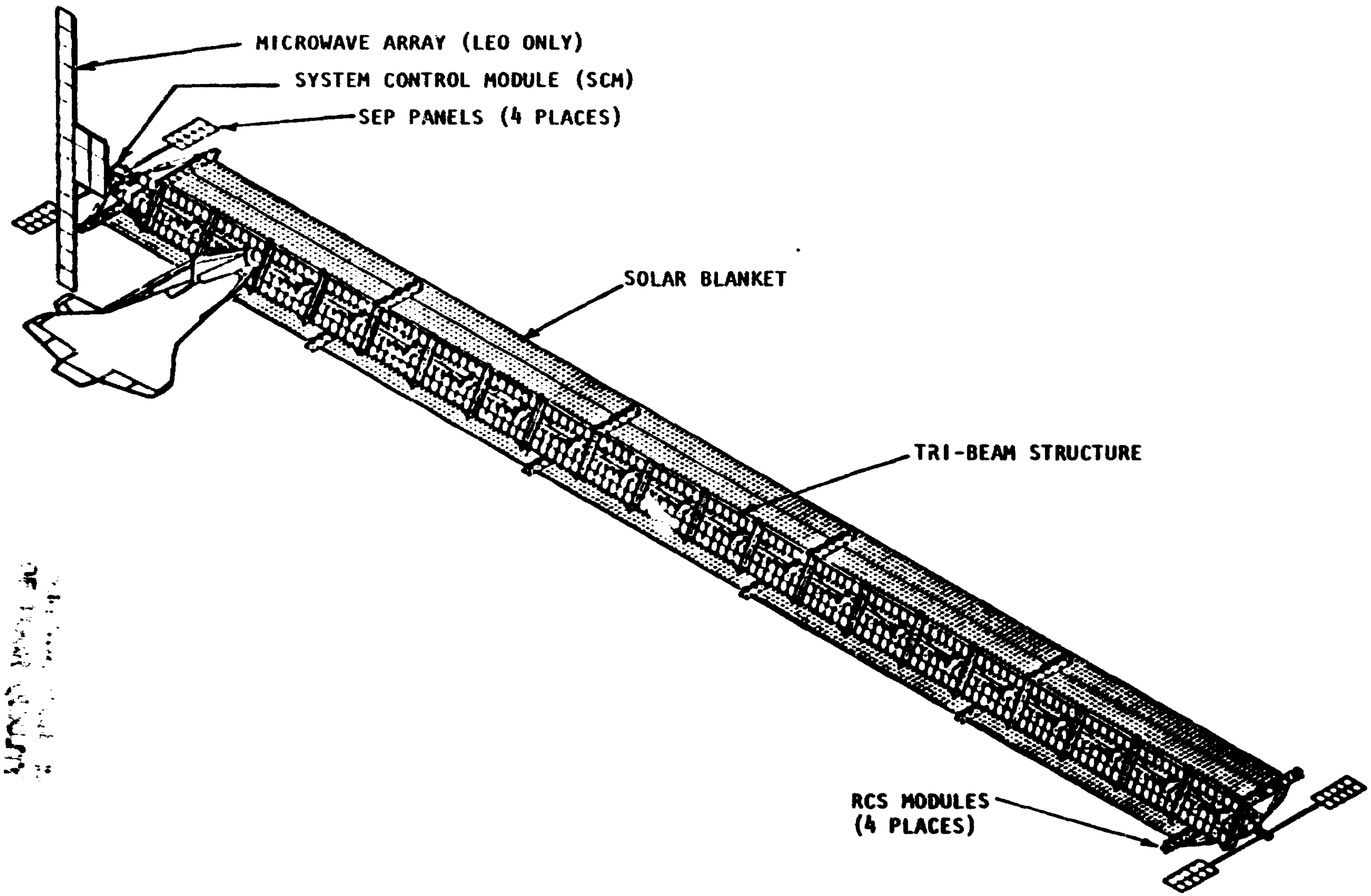

Figure 3.1.1-2 ETVP SPS (LEO) Test Configuration

The ETVP is capable of operation at GEO as well as at LEO, and is transported to GEO from LEO with low-thrust chemical propulsion modules.

3.1.2 Structural Arrangement

The primary structure of the ETVP consists of automated space-fabricated beams arranged in a triangular section 4.96 m deep with in-plane mounted cross-members (Figure 3.1.1-1). Appendix B contains the trade study data that were developed for the selection of the in-plane cross-members.

The depth of the tri-beam, in conjunction with the 17-m cross-members, permits the RMS to reach all extremities of the platform structure. This size tri-beam also permits the construction fixture to be folded and stowed within the orbiter cargo bay for transport to the construction altitude.

The 17.4 m long crossbeam member, in conjunction with the 10-m bay spacing, permits the installation of a pair of stabilization struts by the RMS without ETVP translation, thus simplifing the construction process. Figure 3.1.2-1 illustrates this operation.

The beam cap and diagonal cords are sized for the orbit transfer operation and to limit torsional deflections during the attitude control RCS firing to acceptable levels. The selection of the diagonal cord concept versus a vierendeel arrangement is documented in Appendix C. Dual stabilization struts at each payload attach port provide torsional stability of these transverse members. Dual stabilization struts are also utilized at each RCS module to react the orbit transfer bending loads induced in the transverse beams. The tri-beam structure is divided into ten bays, each approximately 10 m long. This spacing provides adequate column stability during the orbit transfer mode.

Strut arrangements provide the support of the system control module and of the orbit transfer propulsion modules.

The orbit transfer thrust structure consists of strut members providing a structural bridge between the ends of the three longitudinal members of the ETVP and the three propulsion module locations. Attach ports are located on the ends of the struts which define the propulsion module interfaces. Power, data, and control interfaces are also provided at each of the attach ports. Ball and socket interface attachments are provided for the struts at the ETVP longitudinal members. Appendix D contains the trade data generated for the selection of the thrust structure.

The SCM support struts are arranged as a foldable package with ball and socket end attachments that mate with both the platform structure and the SCM structure. This arrangement was selected to minimize the construction time and complexity even though the foldable struts package less efficiently in the orbiter payload bay than the individual "dixie cup" type struts. Appendix E documents the trade performed which selected this configuration.

All subsystem modules and payloads are secured to the structure with attaching ports; these ports provide the capability to attach the modules

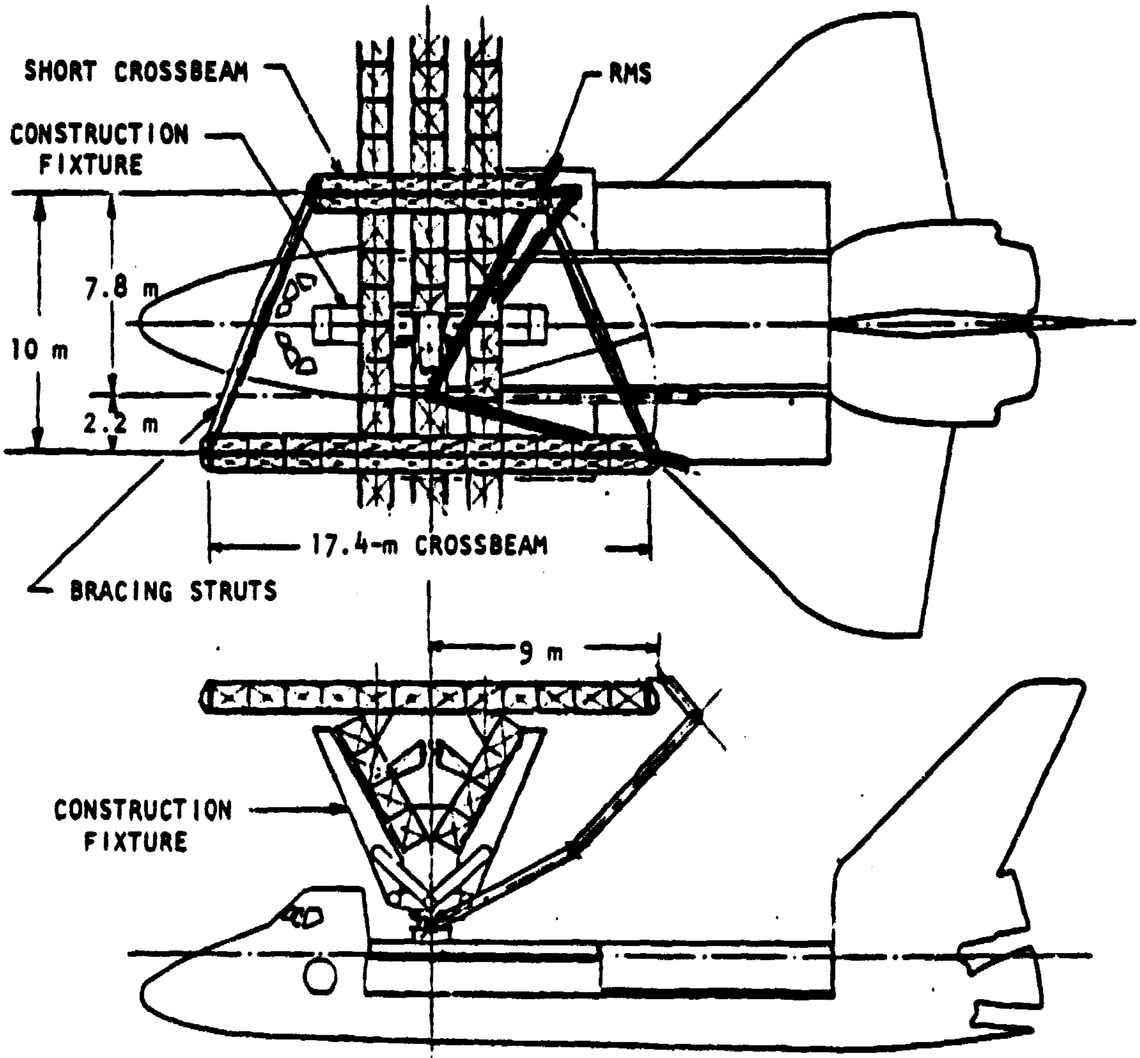

Figure 3.1.2-1. Stabilizing Strut Installation Concept

with a single mating motion. The ports also provide the capability to remove the modules, as required, for resupply or for unscheduled servicing. The attach ports also contain the utility interfaces required for the operation and control of each module. The configuration of the attach ports is the result of a trade study and is documented in Appendix F.

Three configurations of attach ports are utilized on the ETVP. The three configurations are illustrated in Figure 3.1.2-2 and Figure 3.1.2-3. Twelve Configuration A ports are utilized. These ports contain not only the physical and utility interfaces, but also provide the stabilization strut connection interface.

Configuration B port is identical to the Configuration A port, except that Configuration B is used only to provide attachment for the stabilizing struts. Therefore, no physical latching is provided nor any utility interfaces.

Configuration C ports (Figure 3.1.2-3) are utilized on each end of the longitudinal members. These attach ports provide the attachment of the strut supports for the thrust structure and for the CSM support struts.

3.1.3 RCS Module Arrangement

Four individual modules provide the attitude and the orbit makeup thrust. Each module contains its own propellant, engines, valves, and controls. The modules are sized for 7-year GEO operations, both in the quantities of propellant and nozzle lifetimes. Resupply consists of the removal of the spent module and the installation of a new module assembly. This arrangement requires only the necessity for making electrical-type connections for power and data control; no fluid connections are required. Appropriate environmental protection and thermal control are provided on each module. The capability for maintenance of individual components could also be provided.

Figure 3.1.3-1 illustrates the RCS mdoule configuration, and indicates the berthing port and the utilities interface at the port. The concept also indicates a grappling fitting which will be used for RMS handling and installation.

3.1.4 Electrical Wire Routing

Figure 3.1.4-1 is a schematic illustration of the electrical power, data, and communication lines routing. These lines are installed along only one of the base longitudinal members or the tri-beam configuration. This arrangement was selected in order to minimize construction time and equipment required. At each of the intersections of the long cross-members, an electrical connection is made which distributes power and data lines to the attach port/payload interface. (Figure 3.1.4-2)

A discussion of the line installation procedure for both the crossbeams and the longitudinal member is presented in SD 80-0038, Construction Analysis.

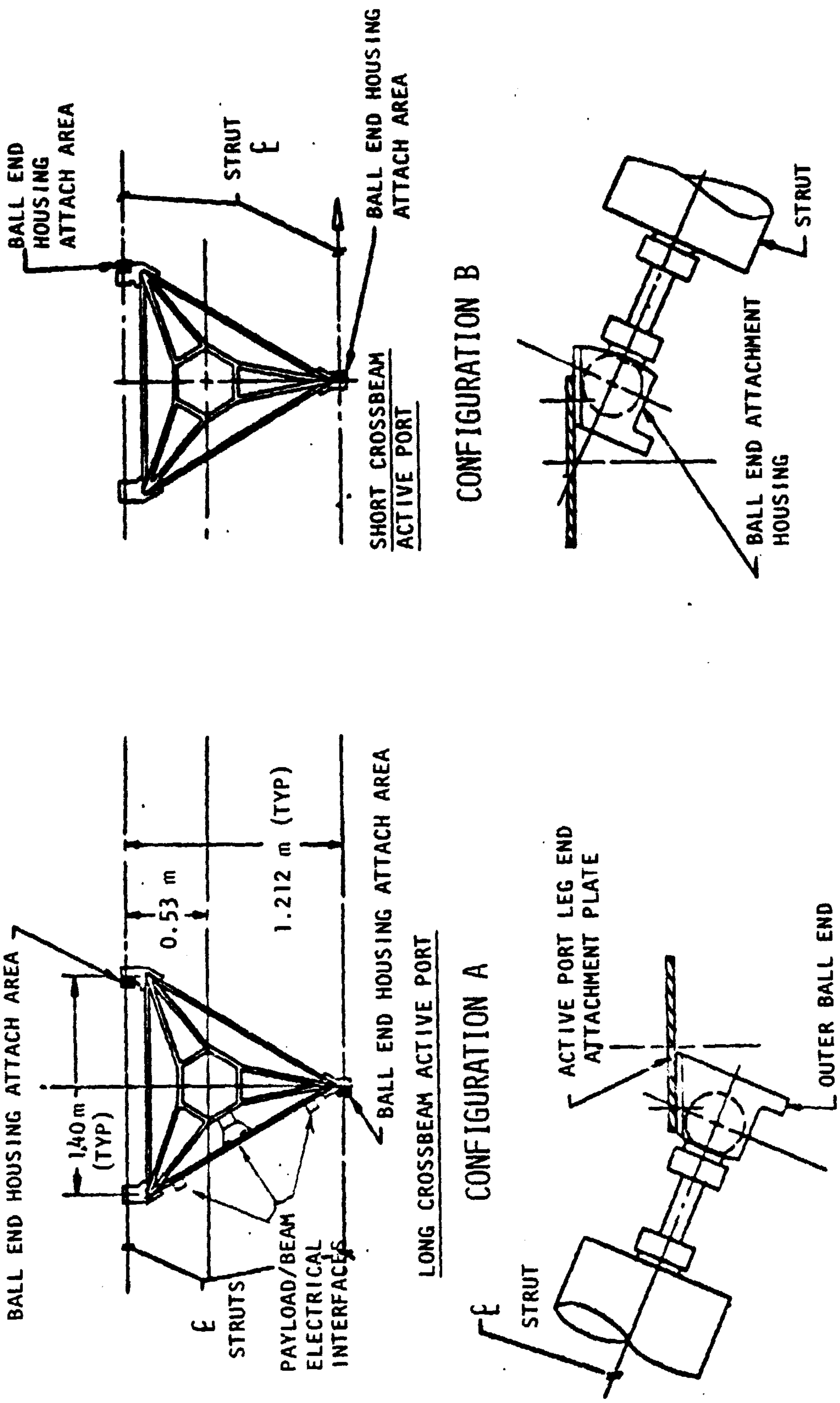

Figure 3.1.2-2. ETVP Attach Port Concept Configurations

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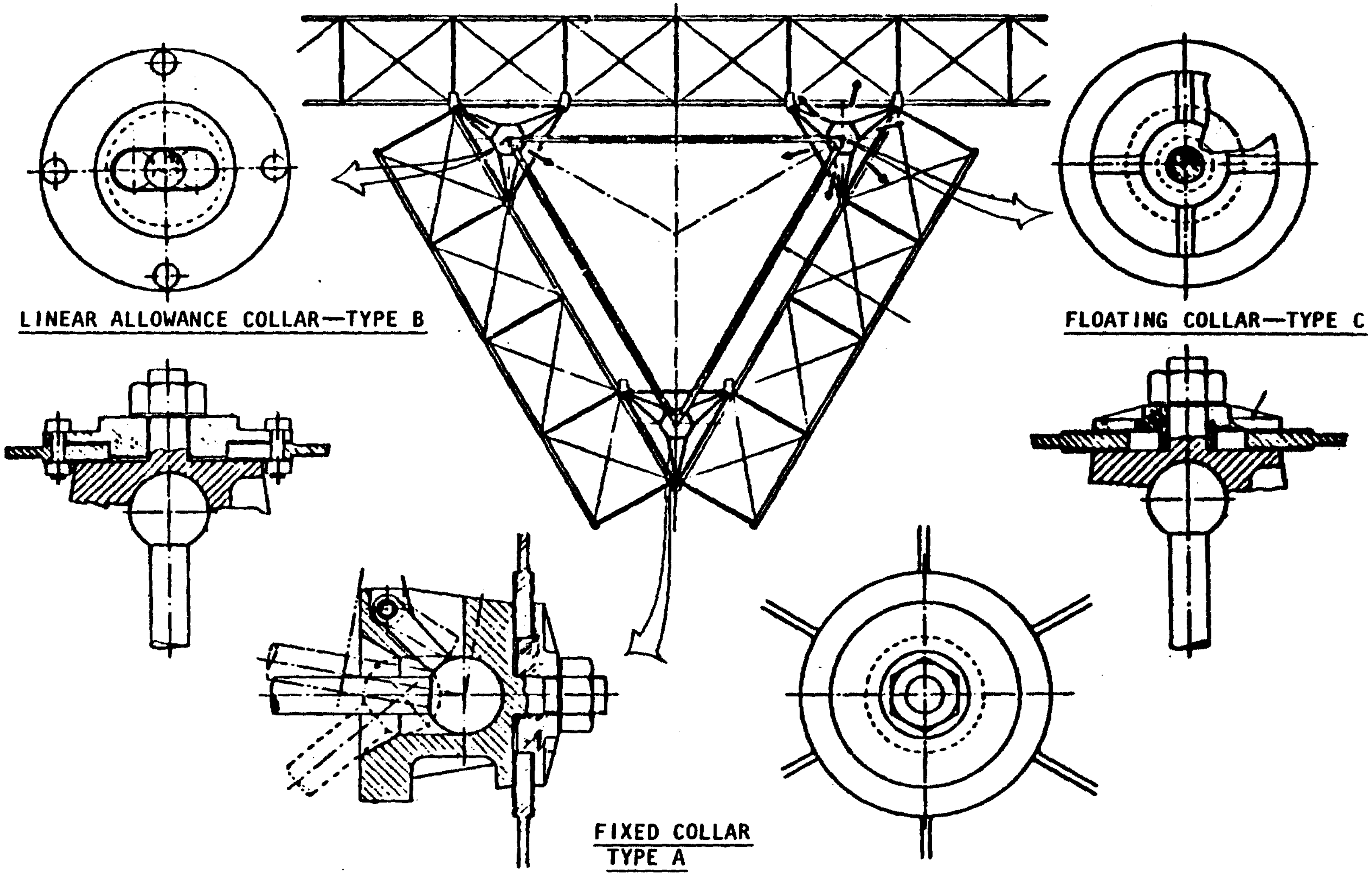

Figure 3.1.2-3. ETVP Longitudinal Beam Attach Port

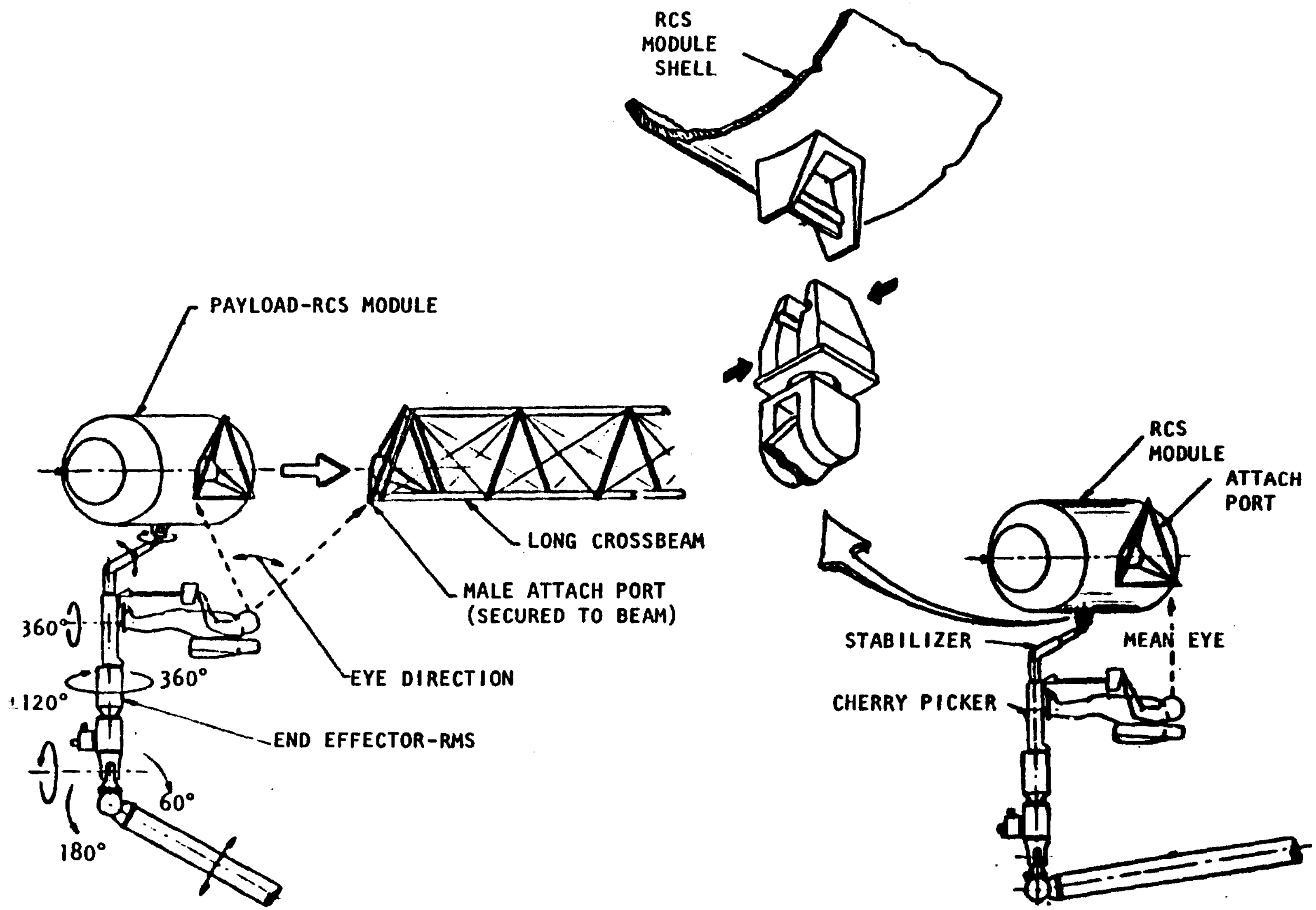

Figure 3.1.3-1. ETVP RCS Module Installation Concept

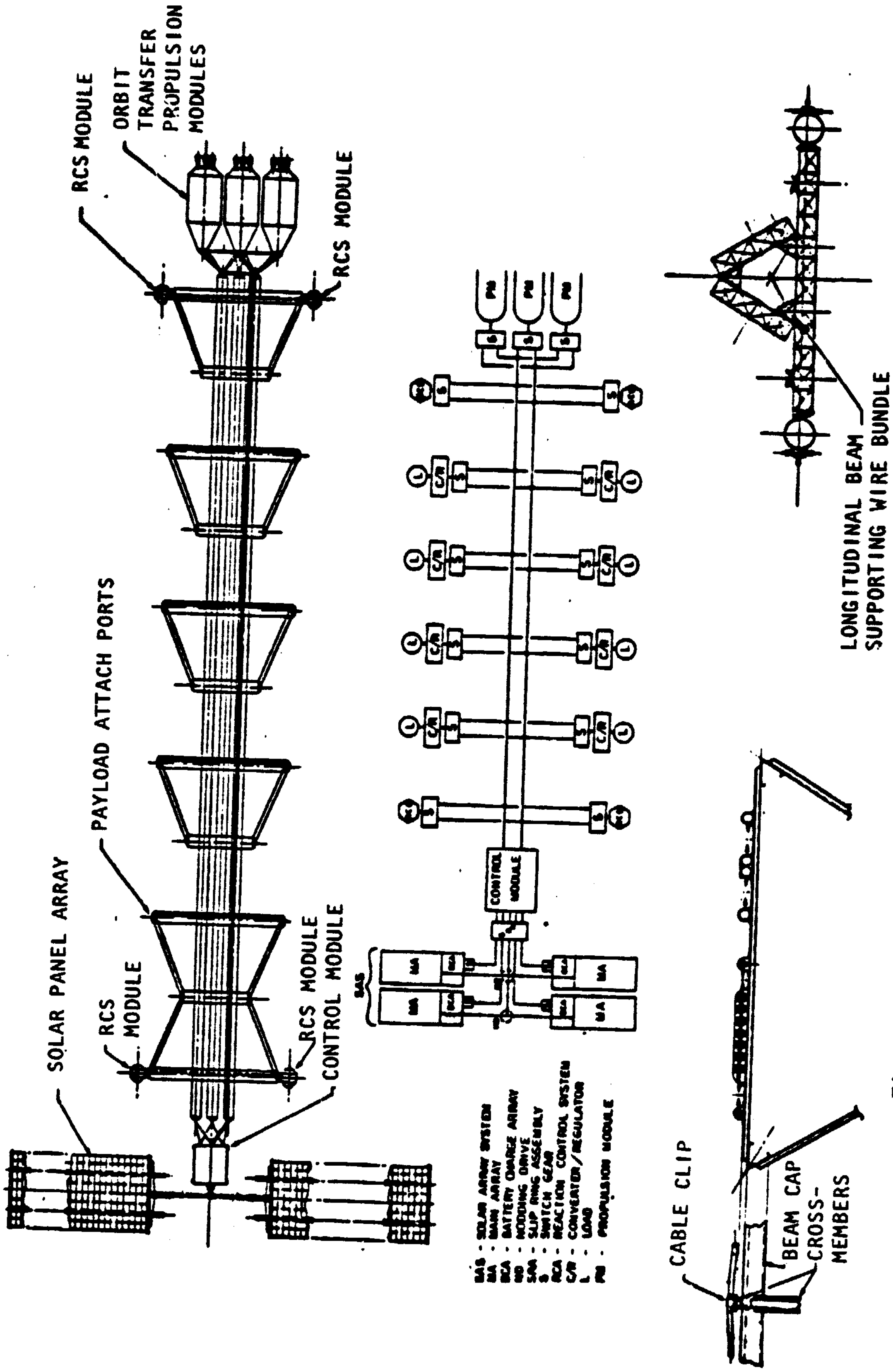

Figure 3.1.4-1. ETVP Electrical Wire Routing Schematic

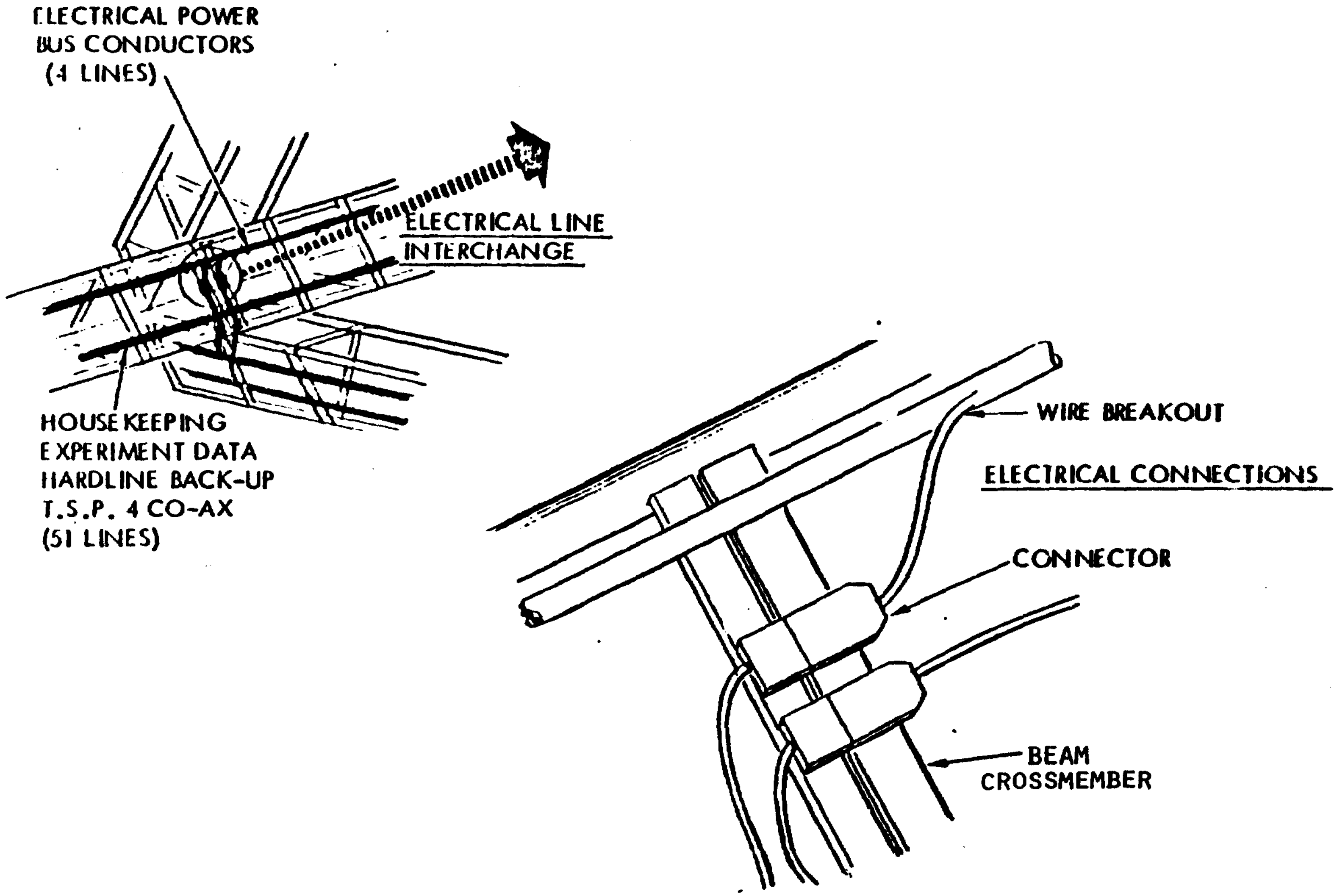

Figure 3.1.4-2. Electrical Power and Data Distribution Concept

3.1.5 System Control Module (CSM)

The module is a structural system which integrates all of the normal control subsystems for the Engineering and Technology Verification Platform into a single unit which can be accommodated in the Shuttle orbiter as a payload on the second flight.

Preliminary trade studies were conducted to examine alternative configurational approaches. The major considerations included: structural arrangement, accommodation of the wide variety of subsystem modules in an arrangement which would be amenable to on-orbit servicing (replacement) by a remotely controlled teleoperator, structural interface compatibility with the tri-beam platform on one end and the solar array power system on the other, and compatibility with the orbiter standard payload retention systems. The concept evaluation indicated the choice of a hexagonal-shaped body as the preferred approach. However, subsequent more definitive analysis of configurations and stowage concepts for the thermal control radiator panels resulted in a change of the selected concept from the hexagonal-shape to the "T" cross sectional-shape. The general arrangement of the structure and accommodated subsystems for the T-concept are shown on Figure 3.1.5-1.

The control module has been configured to maximize the effective use of cargo bay length and cross section. The top of the "T" is in plane with the orbiter payload retention interface at orbiter station 414. Four trunnions provide direct attachment to standard payload retention latches on the orbiter. The vertical web of the T-structure extends to the bottom of the cargo bay and incorporates the keel fitting. The structural configuration is shown in Detail B on Figure 3.1.5-1.

The control module has an overall length of 5.1 meters and features subsystems mounted on the upper surface of the "T" and both sides of the vertical "T". These three surfaces incorporate coldplates which are part of the active thermal control system.

The operational philosophy is to be able to service the ETVP remotely by use of a teleoperator. Therefore, the subsystem components have been grouped into modules to facilitate this servicing objective, rather than handling each individual component element. Where possible, associated subsystem components have been grouped. In those cases where there was redundancy, the redundant components were grouped separately.

For representative purposes in this configuration, the module retention and servicing concept shown is similar to that currently being developed for the NASA GSFC Multi-Mission Modular Spacecraft (MMS). Each module is attached by two latch screws which, in the case of the MMS, are driven by a special-purpose end effector "tool" grasped by the RMS standard end effector and rotated by the RMS wrist drive. In removing a module, one latch screw is disengaged, then the RMS moves to the other latch position, unscrews the latch screw and, while still grasping the released module, translates the module to a replacement position. A new module is installed in a reverse operation. In the Engineering Technology Verification Platform concept, the servicing teleoperator would be equipped with a mechanized arm, similar to the orbiter RMS which has the required rotating end effector. Detail A on Figure 3.1.5-1 illustrates

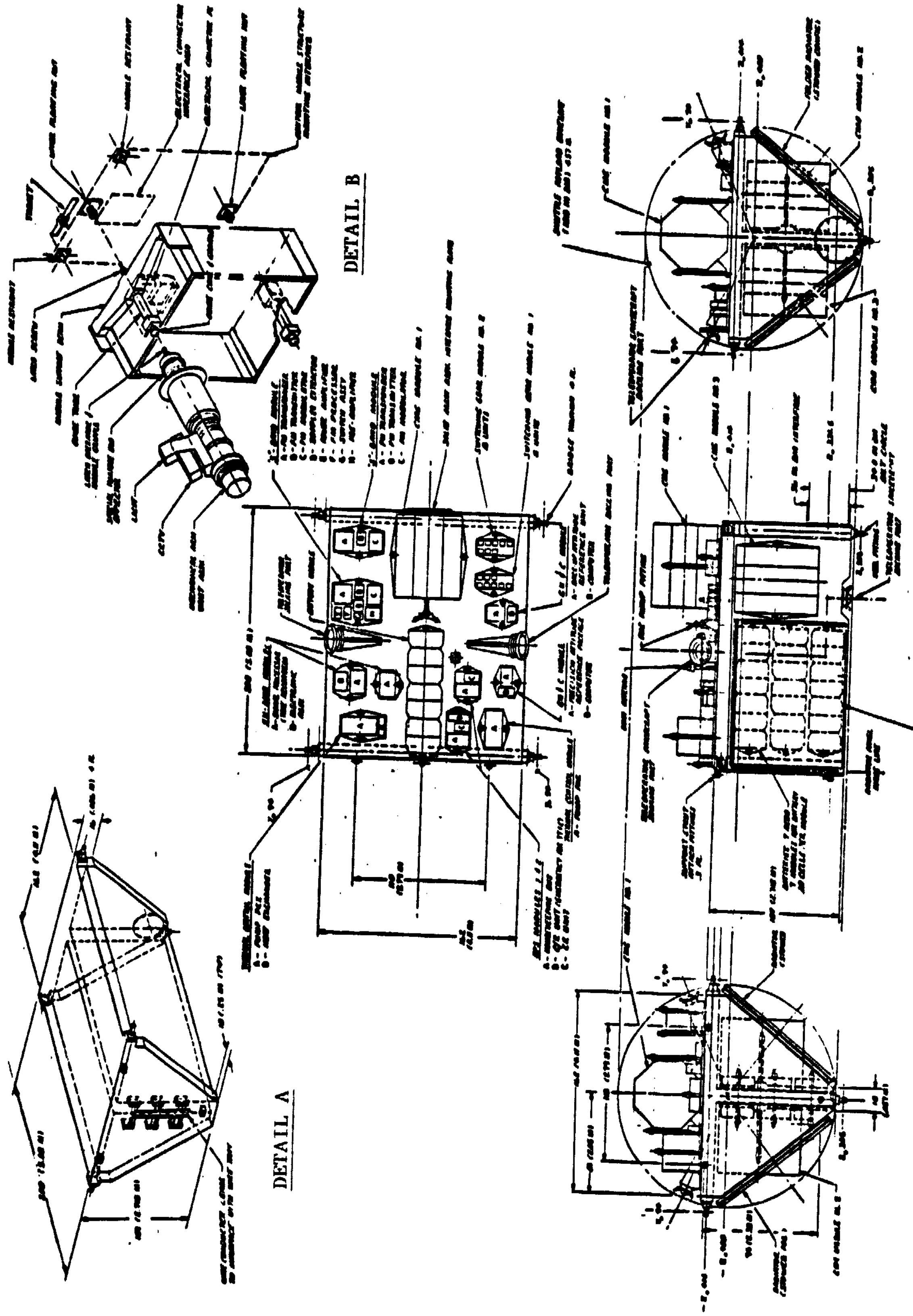

Figure 3.1.5-1. System Control Module

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the major features of the module exchange and retention system. The electrical connector interface is located in the mating surface and the mating is automatically accomplished where the module latch screws are driven.

The largest modules accommodated on the control module are the three CMG's. The physical size of these units was a major driver in the configuration development of the control module. One unit is mounted on the upper surface of the T-structure and one each on the sides of the vertical web of the T, all at the forward end of the control module (in ETVP flight configuration).

The electrical power system nickel hydrogen battery cells have been consolidated into seven large units to reduce the number of items being handled in the servicing mode. Three battery units are attached to each side of the vertical web of the T-structure adjacent to the CMG's, and the seventh unit is located on top.

All the subsystem modules are located on the top surface. The modules include: EPS 1 and 2, thermal control 1 and 2, GN&C 1 and 2, switching gear 1 and 2, Ku-band 1 and 2, and S-band 1 and 2.

Three interfaces have been provided for docking a teleoperator to the control module to accomplish module exchange. The passive docking port is an 18-inch-diameter three-petal unit. Two ports are located on the top surface, one on each side, and a third is on the bottom of the structural "T" vertical web. The three locations will provide the necessary access to all subsystem mounting surfaces by the teleoperator.

The forward end bulkhead on the control module incorporates a 36-inch-diameter bolt circle structural interface for attachment of the solar array system.

The thermal control system utilizes two deployable radiators with a total single-sided flat panel area of 37 m^2. Each panel consists of two folded half-panels which are mounted on the lower triangular edges of the aft structural bulkhead of the control module. The panels are stowed lengthwise along the control module and lay over the vertical web mounted battery modules. Remotely-controlled latches release the panels and dampened spring-powered hinges with positive locks, swing the panels to the deployed position.

The wiring system necessary to interconnect the subsystems modules and the solar array system on one end and the major interface with the platform on the other end is integrated into the control module basic structure. As previously mentioned, each subsystem module mounting foot print on the control module structure provides the appropriate electrical connector half which mates with the other half on the module side. This mating is automatically accomplished when the modules are attached to the structure. Major electrical connectors for power and data are provided at the forward bulkhead to interface with the solar array system. At the aft bulkhead, a series of connectors mate with one end of the wiring system in the deployable wire tray. The wire tray is a hinged, deployable structural system containing the wires which extend across the 355 cm space between the control module and the tri-beam platform.

3.1.6 Conclusion

Space construction had a major influence on the final configuration of the ETVP. The sizing of members, the spacing of members, attaching arrangements for modules and structures are some of the parameters that were affected by the space construction operation. Construction from the Shuttle orbiter also had a significant influence on the general arrangement of the ETVP. Table 3.1.6-1 lists some of the elements of the platforms configuration and the construction item that incluenced it.

Table 3.1.6-1. Space Construction Influence on the ETVP Configuration

PLATFORM ELEMENTS	CONSTRUCTION ITEM	REMARKS
LINEAR CONFIGURATION	BEAM BUILDER	LONG CONTINUOUS MEMBERS MOST EFFICIENT USE OF BEAM BUILDER
5-M TRI-BEAM	RMS CONSTRUCTION FIXTURE	REACH ENVELOPE FROM ORBITER SIZE AND MOUNT LOCATION IN ORBITER
17-M CROSS-MEMBERS	RMS	REACH ENVELOPE FROM ORBITER FOR MODULE ATTACH.
10-M BAY SPACING	RMS	REACH OF BOTH ENDS OF STABILIZATION STRUTS FROM ONE ORBITER POSITION FOR INSTALLATION
IN-PLANE CROSS-MEMBERS	CONSTRUCTION FIXTURE	LESS COMPLEX FIXTURE, LESS CONSTRUCTION TIME
ATTACH PORTS	INSTALLATION OPERATION	SINGLE MOTION FOR ATTACH/REMOVAL OF MODULES
CANTILEVERED SECTION OF TRI-BEAM AT SCM	CONSTRUCTION FIXTURE	FIXTURE CONFIGURATION; LOCATION OF FIXTURE FOR SEPARATION FROM PLATFORM
THRUST STRUCTURE	ORBITER TRANSPORT	ORBITER PAYLOAD BAY PACKAGING EFFICIENCY
SCM SUPPORT TRUSS	INSTALLATION OPERATIONS	ORBITER PAYLOAD BAY PACKAGING; MINIMIZE INSTALLATION TIME
SCM	MAINTENANCE OPERATIONS	REMOTE MAINTENANCE IN GEO
ELECTRIC WIRE ROUTING AND CONNECTIONS	CONSTRUCTION SUPPORT EQUIPMENT	CONNECTION OPERATIONS (EVA); REDUCE TIME FOR WIRE LAYING

3.2 SUBSYSTEMS

3.2.1 Structural Subsystem

Objectives

The objectives of the structural design analyses conducted in this space construction study of an engineering technology verification platform were:

- o To ensure construction system study realism by representing structural configurations that are indeed suitable for the total spectrum of mission requirements.
- o To ensure identification and understanding of those particular requirements for structural integrity which significantly impact construction.
- o To support the total systems weight analysis through definition of component structural sizes.

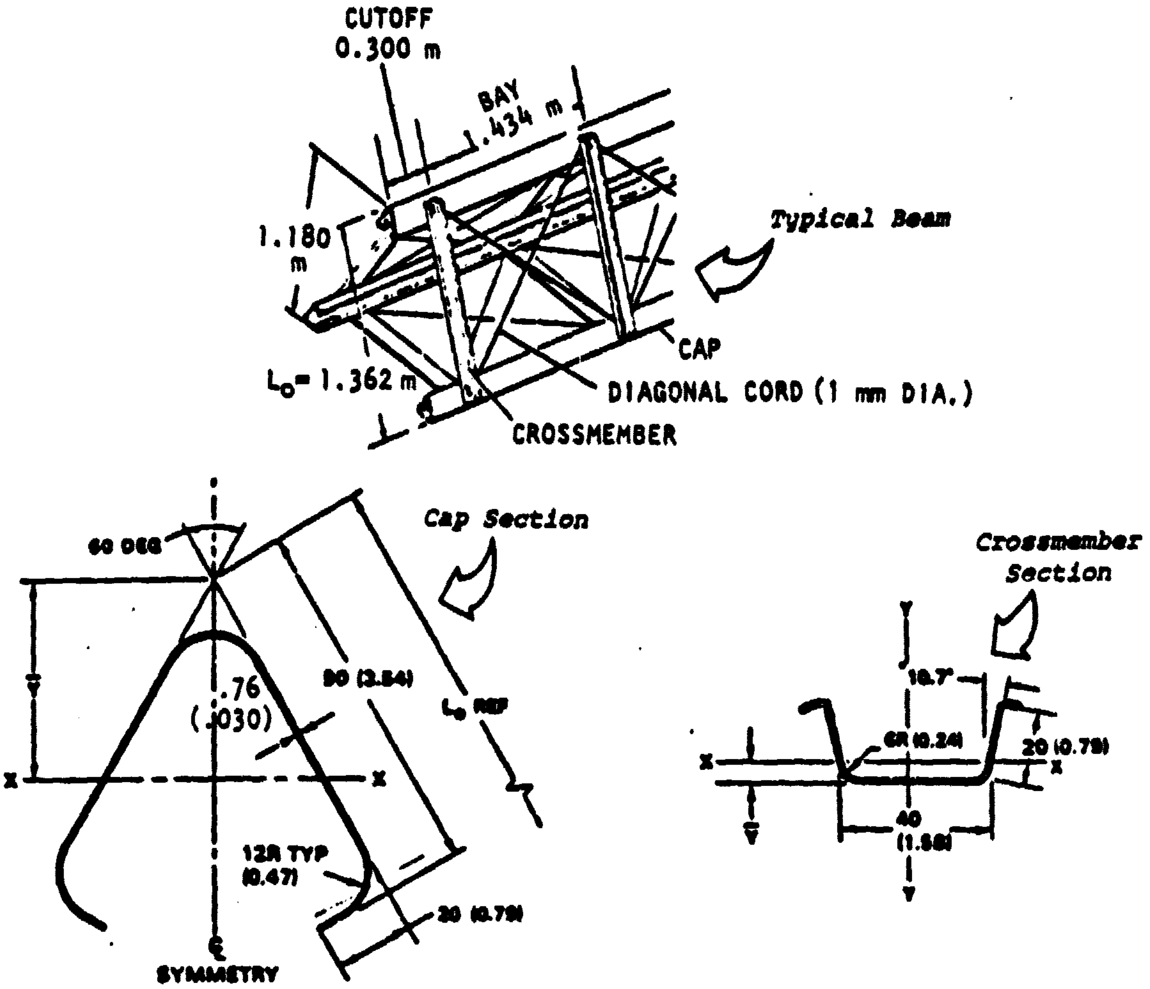
Platform Configuration Description

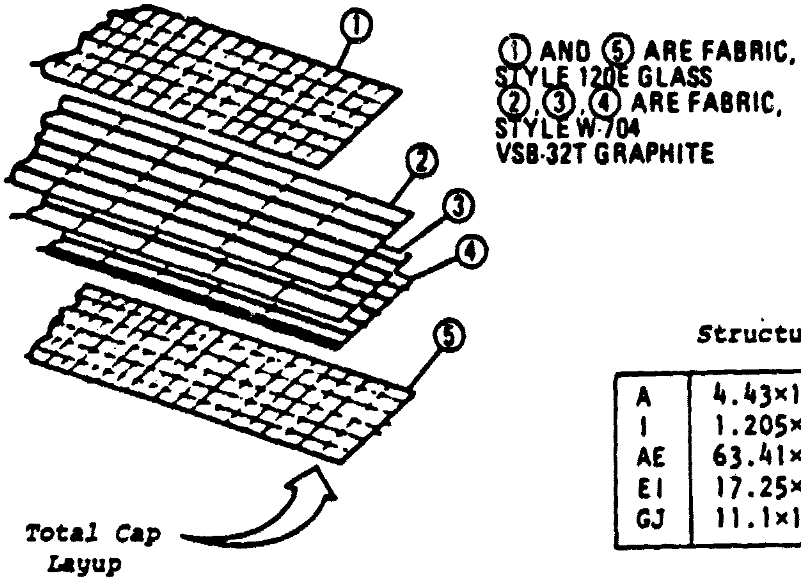
The structural configuration of this space-fabricated engineering technology verification platform, pictorially described on Figure 2.5.1-1, Section 2.5.1, is that of a tri-beam with outriggers. The tri-beam utilizes the machine-made beam element (Figure 3.2.1-1), currently being developed by General Dynamics under Contract NAS9-15310, as the basic structural member from which the tri-beam is fabricated. However, its basic cap thickness, diagonal cord diameter and pretension and hence its basic structural performance characteristics (Figure 3.2.1-1) have been increased, as shown in Figure 3.2.1-2. These increases are within the permissible envelope of changes (per conversation with the General Dynamics study manager). The tri-beam cross section has a side dimension of 4.2 m (center-to-center of beam element) with truss behavior provided by the "X" system of diagonal tension cables (4.7 mm diameter graphite composite) shown in all bays but the first short bay adjacent to the systems control module.

The tri-beam is the basic strongback to which the orbit transfer propulsion modules are mounted through provision of a thrust structure (Drawing 42662-60)*and to which the solar array and control module are mounted through provision of a support strut assembly (Drawing 42662-59).* Both structures are deployable space trusses. The four RCS thruster module packages are mounted to the end outriggers and will be placed so that their center of mass (associated with orbit transfer) will be nominally on the neutral axis of the outriggers. Mounting provision for eight antennas is provided at the extremeties of the remaining four outriggers.

An array of 12 pairs of 163 mm (6.4 in.) diameter stabilization struts having a wall thickness of 1 mm (.04 in.) is provided as shown on Drawing 42662-45A. The four pairs of struts at the extremities of the configuration are provided to preclude excessive bending moments being applied to the outriggers during orbit transfer. The eight inner pairs of struts are provided to torsionally stiffen and limit antenna rotation compatible with the required pointing errors during N-S stationkeeping maneuvers.

*See Appendix A, SSD 80-0038





Structural Characteristics

A	$4.43\times10^{-4} m^2$ (0.687 in.^{2})
I	$1.205\times10^{-4} m^4$ (289.7 in.^{4})
AE	63.41×10^6 N (14.26×10^6 lb)
EI	17.25×10^6 Nm^2 (6.014×10^9 lb-in^2)
GJ	11.1×10^3 Nm^2 (3.87×10^6 lb-in^2)

Figure 3.2.1-1. GD Baseline Machine-Made Beam

MACHINE-MADE BEAM MODIFICATIONS

- CAP GAUGE INCREASED TO 1.25 mm (0.050 in.)
- DIAGONAL CHORD DIAMETER INCREASED TO 2 mm (0.080 in.)
- DIAGONAL CHORD PRE-TENSION INCREASED FROM 10 TO 40 LB

REVISED STRUCTURAL CHARACTERISTICS

A	7.38×10^{-4} m^2 (1.15 in.^{2})
I	2.0×10^{-4} m^4 (483 in.^{4})
AE	106×10^{6} N (23.8 10^6 lb)
EI	28.7×10^6 N-m^2 (10×10^9 lb-in.^{2})
GJ	44.4×10^3 N-m^2 (15.5×10^6 lb-in.^{2})
KAG	88.8×10^3 N (20.0×10^3 lb.)

Figure 3.2.1-2. Modifications to Machine-Made Beam and Revised Structural Characteristics

During transfer to geosynchronous orbit, the antenna feed column masts are stowed to preclude a prohibitive penalizing of the antenna feed column design, large thrust vector inclinations due to c.g. travel, and prohibitive joint moments. The solar panels are stowed to avoid penalizing their structural design. The antenna reflector, however, is deployed to eliminate the risk of deployment at geosynchronous orbit and because antennas of this size are designed for 1-g ground deployment with the electrical axis oriented horizontally (according to Reference 1). The reflector weights and moment arms are small compared to that of the feeds.

The structural sizes of the individual members of the thrust structure and control module support strut assembly are delineated in Figure 3.2.1-3. The sizes shown, while derived to satisfy load strength/stability requirements are expected to be acceptable for stiffness requirements. These two structural components have each been designed to be deployable structural arrangements packageable into the Shuttle cargo bay and contain the appropriate clevises and locking devices shown on the referenced drawings.

The structural attachment of the antennas, RCS modules, thrust structure, support strut assembly, and stabilizing struts to the ends of the machine-made beams are accomplished by male/female attach port structures. The male attach port (Drawing 42662-72) is provided with legs (Detail "A") that are directly welded to the interior of the beam caps. The female attach points are ground-attached to the antennas, RCS module, etc.

The tri-beam construction utilizes the lap joint attachment concept and intersection fitting presented in zones 23 and 24 of Drawing 42662-45A. The design concept proposes joining the fitting to each of the machine-made beams through fusion bonding (Table 3.2.1-5). Each fitting contains a pair of clevises to receive the "X"-bracing tension cable end fittings. The tension cables will be pretensioned to 2360 N (530 Lbs) + 150 N (34 lbs) by means of the construction techniques described in SSD 80-0038. The structural suitability of that technique is substantiated in Section 5.3.1.

Platform Structural Configuration Options-Selections-Rationale

The studied structural configuration options, actual selections, and associated rationale that resulted in the final platform configuration are summarized, for review convenience, in Table 3.2.1-1, with further information regarding each design option provided in the referenced documentation.

Structural Analysis Model Description

The NASTRAN finite element structural analysis model used in the strength, stability and modal analyses is described herein.

Constructed during the earlier stages of the study, the basic X-braced tri-beam model including outriggers and stabilizing struts is that of the 136.8 meter long platform shown on Drawing 42662-45. The thrust structure, support strut assembly and stick model of the control module and solar array are in accord with the latest configuration shown on Drawing 42662-45A. Structural sizes are in accordance with the data shown in Figures 3.2.1-1, -2, and -3.

	MEMBER	CRITICAL COMPRESSION LOAD (N)	CRITICAL MEMBER LENGTH (M)	TUBE INSIDE DIA (MM)	WALL THICKNESS (MM)
THRUST STRUCTURE	604 THROUGH 609	24,132	3.73	117	1.0
	610 THROUGH 615	111,840	2.96	163	1.0
	616, 637, 638, 621, 639, 640	46,490	4.2	163	1.0
	651, 652, 653	109,025	4.2	163	1.45
SUPPORT STRUT ASSY	507 THROUGH 518	11,192	4.0	117	1.0

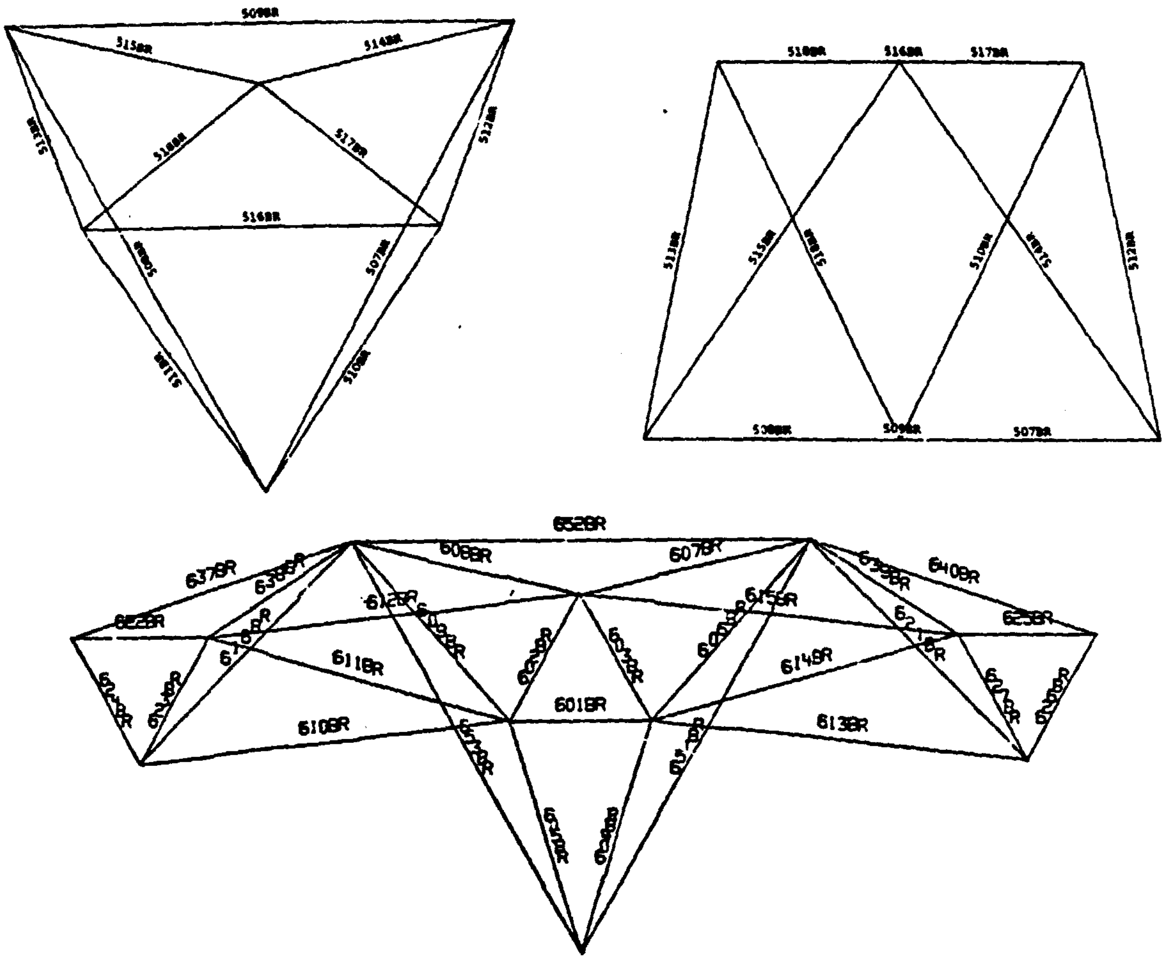


Figure 3.2.1-3. Thrust Structure/Support Strut Assembly Configuration and Sizes

Table 3.2.1-1. Configuration Options-Selections-Rationale

DESIGN OPTION	SELECTION	REFERENCE DRAWING	RATIONALE	DOCUMENTATION
Tri-beam depth	4.967M	Dwg 42662-45A Section B-B	• Maximum depth compatible with RMS reach to ends of outriggers. • Compatible with structural requirements. • Minimum fixture size	3.1.1
Outrigger Spacing	20.08M	Dwg 42662-45A	• Compatible with clearance requirements for installation of 20 m diameter antennas.	3.1.1
Tri-beam bay spacing	10.04M		• Compromise of desired minimum number of cross-members and X-bracing cables with required torsional stiffness and desired minimum cable pretension load.	3.1.1 and 3.2.1
Conventional vs. Staggered joints	Conventional		• Simplicity of fixture	Appendix B
Vierendeel vs. X-bracing	X-bracing		• Legacy to SPS • One less propulsion stage	Appendix C
Thrust structure concept	Deployable truss	Dwg 42662-60	• Best compromise for structural simplicity and contingency for engine failure.	Appendix D
Outrigger bracing	Stabilizing struts to antennas	Dwg 42662-45A	• Pairs of struts required to torsionally stiffen outrigger to maintain platform pointing accuracy.	3.2.1
	Stabilizing struts to RCS pods		• Single strut precludes excessive bending moment in outrigger. • Pair of struts for design commonality.	

The NASTRAN model used for determination of the orbit transfer induced internal loads is described by the CRT plots of Figure 3.2.1-4. The same model was used for the platform column stability analysis. Appropriate stick model representations of the deployed antennas (Figure 3.2.1-5 and Table 3.2.1-2), deployed solar array, center propulsion stage at burnout, and the remaining miscellaneous equipment mass applied to the platform structure model resulted in the NASTRAN model for the modal analysis of the operational configuration. Minor modifications of this model to represent stowage of the antenna feed columns and solar array and the presence of the three fully-loaded propulsion stages (28,409 kg ea.), defined the orbit transfer configuration model. Finally, the magnitude and distribution of the masses used in the internal loads, column stability, and modal analysis are those described in Table 3.2.1-3.

Relative to the final slightly downsized configuration of Drawing 42662-45A, the analysis model provides slightly conservative internal loads (10 to 15%) and significantly conservative modal frequency and column stability values (50-100%).

Structural Requirements

The space-fabricated tri-beam structural requirements are delineated in Section 2.5.1.

Structural Analysis

The structural analyses performed to support the design definition and verify the suitability of the structural configuration to satisfy the foregoing requirements are delineated herein. These analyses utilize a safety factor of 1.5 applied to limit load.

Total Platform Structure Column Suitability

A NASTRAN stability analysis was performed to verify the column suitability of the total platform structure. The NASTRAN model (described in the foregoing articles) was loaded by a .20 g acceleration of the mass distribution (Table 3.2.1-3). The Eigenvalue obtained is 3.2, with a $M.S. = 1 - \frac{1.5}{3.2} = .53$. This margin has sufficient allowance for the secondary effects of fabrication non-straightness and thermal gradient-induced bending.

It is pertinent to note, that the compression loads due to pretension of the diagonal cord and X-bracing of the tri-beam are included in the local cap stability considerations but not in the foregoing Euler stability analyses. The platform column stability is not influenced by the pretension loads.

Machine-Made Element Strength Review

Of primary concern is the column stability of the individual open-section cap of the machine-made beam element.

The most critical machine-made beam loads extracted from the NASTRAN internal loads model are tabulated in Table 3.2.1-4.

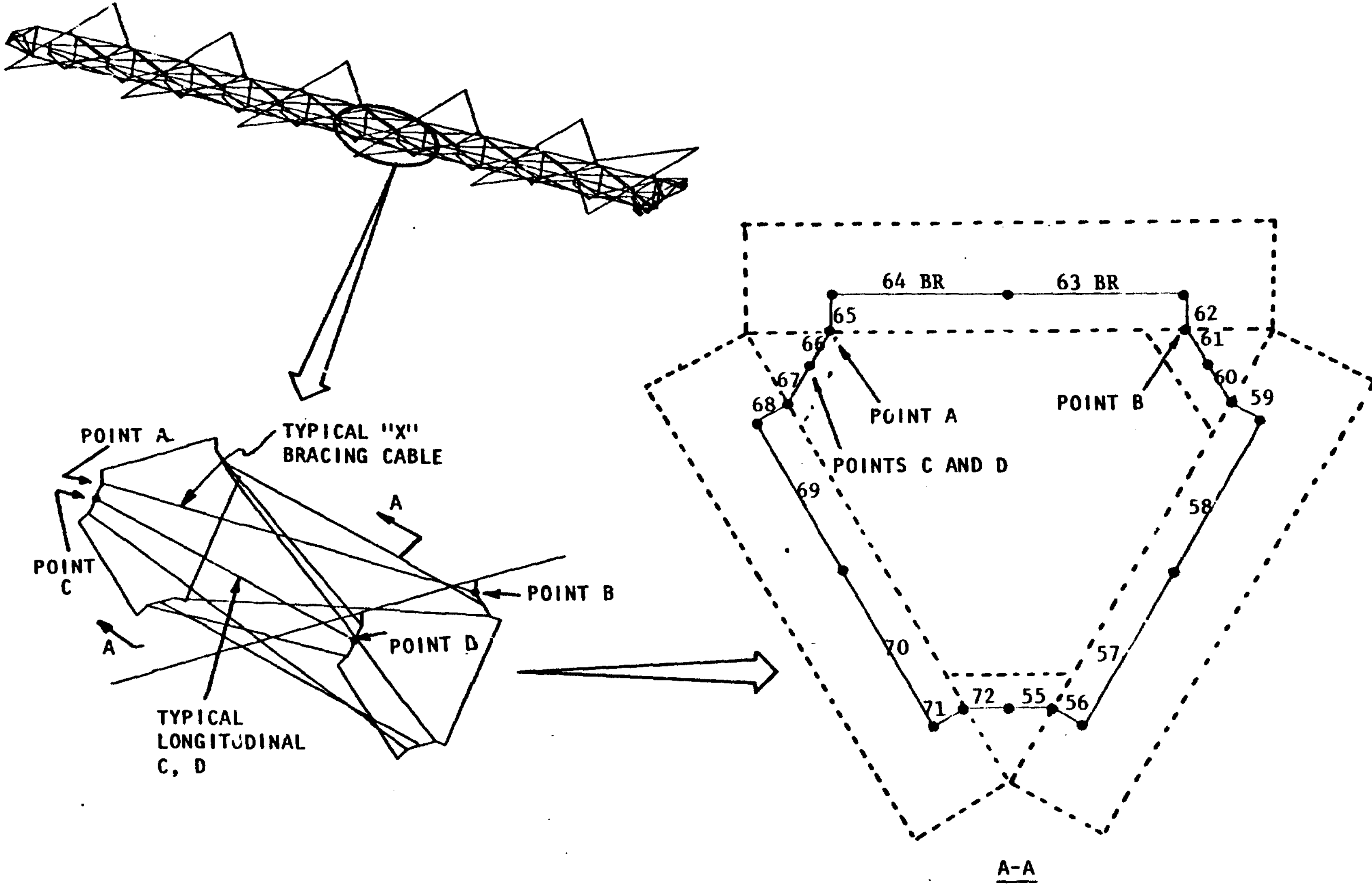
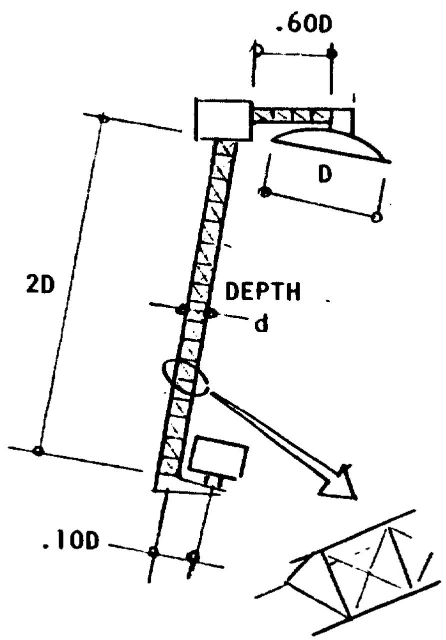


Figure 3.2.1-4. CRT Plots of Platform Structure Nastran Model

Table 3.2.1-2. Antenna Feed Column Structural Characteristics



D (m)	FEED MASS (kg)	d (m)	EI ($N/m^2 \times 10^{-10}$)	GJ ($N/m^2 \times 10^{-10}$)	FEED COLUMN THERMAL ROTATION (AT FEED) (MIN)
6.0	134	0.228	28	0.69	1.8
7.5	1932	0.381	299	1.93	1.3
13.8	140	0.456	113	2.76	2.1
20.5	195	0.761	522	7.67	1.8

ANTENNA BOOM HAS SAME DESIGN AS FEED SUPPORT COLUMN.

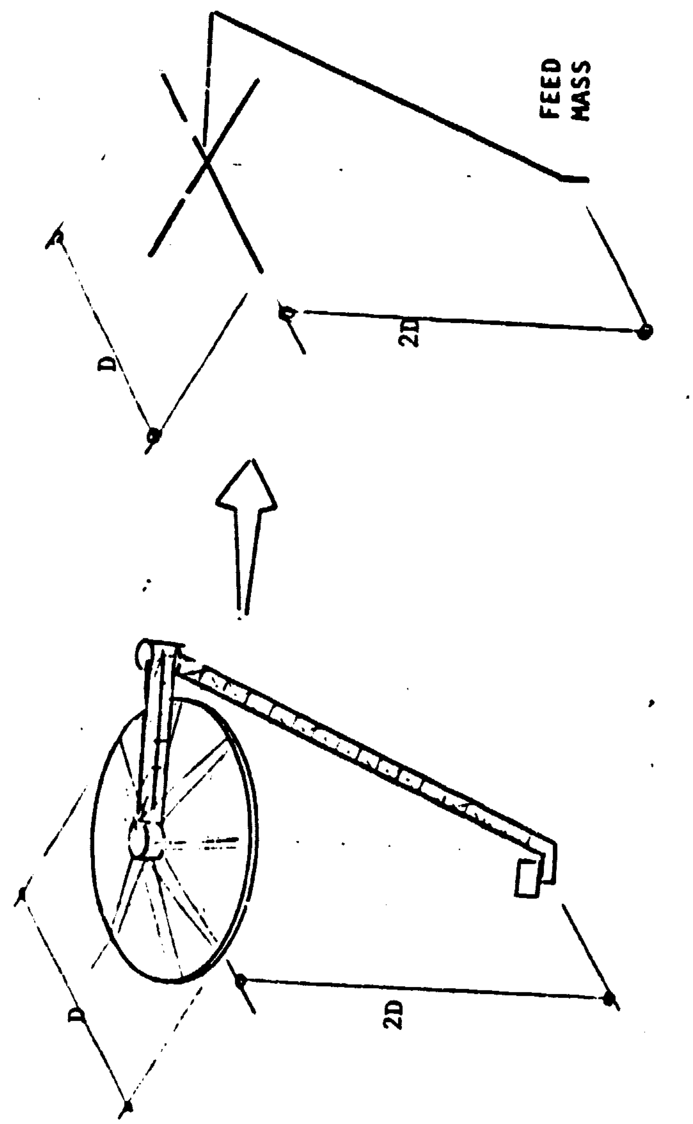


Figure 3.2.1-5. Antenna Structure Stick Model

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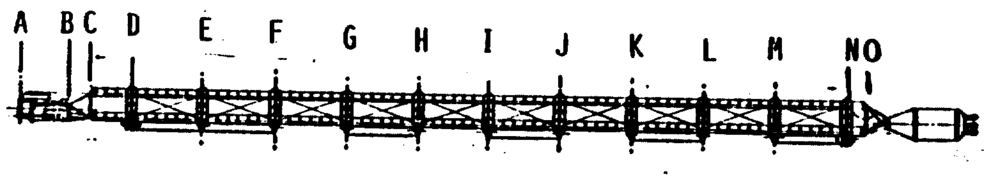
Table 3.2.1-3. Mass Distribution Platform Structure



ITEM	DISTRIBUTED MASS (kg/m)	DISTRIBUTED REGION	TOTAL DISTRIBUTED MASS (kg/m)	CONCENTRATED MASS (kg)	LOCATION	TOTAL MASS (kg)
SOLAR ARRAY	-	-		920	AT A	920
CONTROL MODULE, ROTARY JOINT, BATTERIES, CMG	1455	A TO B	10760			10760
RF BOXES				700	AT G & H	700
PLATFORM STRUCTURE, DOCKING PORTS, AND ANTENNA HARNESS	70	B TO M	9530			9530
2-6.0 METER ANTENNAS	-	-	-	1140	AT F	1140
2-7.5 METER ANTENNAS	-	-	-	4820	AT H	4820
2-13.8 METER ANTENNAS	-	-	-	1466	AT J	1466
2-20 METER ANTENNAS	-	-	-	1956	AT L	1956
2 RCS PODS	-	-	-	3757	AT D	3757
2 RCS PODS	-	-	-	3757	AT N	3757
					TOTAL	38806

Table 3.2.1-4. Orbit Transfer Induced Ultimate Loads - Machine-Made Beam

Element No.	Location*	Type Member	Axial Compression (N)	Bending Moment (NM) M_1	Bending Moment (NM) M_2	Shear (N) V_1	Shear (N) V_2
267	STA I	Longitudinal	31,292	3228	1072	196	71
279	STA M	Longitudinal	39,453	2729	1104	205	94
279	STA O to N	Longitudinal	38,200	580	Neg.	415	Neg.
195	STA N	Cross-Beam	59	337	488	98	67
98	STA I	Cross-Beam	105	2406	525	9158	801
173	STA M	Cross-Beam	Neg.	-2008	71	-8495	846

*See Table 3.2.1-3.

Torsion loads are negligible.

The most critical load on an individual cap is 15875 N (3570 lbs). Since the baseline beam being developed by General Dynamics is used (except for the modifications discussed previously), the analysis presumes, pending a static test of the prototype beam, that the individual cap ultimate load capability is 6583 N (1480 lbs), i.e., the value quoted in Reference 1.

The baseline design is not limited by Euler buckling, but by local buckling and torsional buckling criteria. It is expected that use of the .050 gauge, increases the 6583 N strength value as described below.

In metals, the total axial load capability governed by local buckling criteria would be increased by the ratio of the thickness cubed. The torsional stiffness has the same ratio of increase. This ratio $(1.67)^3$ is a factor of 4.65. For composites, this value may be somewhat less. Conservatively using the ratio squared, the allowable load is 18,360 N (4125 lbs). In view of the foregoing, although the baseline design exhibited a reduced axial stiffness above 2540 N because of local buckling, significant reduction is not expected in this design.

Maintenance of pretension of all six diagonal cords of every bay in the machine-made beams is highly desirable, but not essential. It is essential that one diagonal cord in each face of the three faces per bay is maintained in tension. The analysis review of the entire internal loads model indicated the following:

Referring to the figure in Table 3.2.1-3, the worst case in the tri-beam longitudinals between STA C to N is demonstrated from the data in Table 3.2.1-4 and Figure 3.2.1-6.

- The .05 gage cap shortening (39,453 N compression) = 0.53 mm
- 2 mm cord thermal-induced (55° C) length change along X-axis = .29 mm
- 2 mm cord shear-induced length change along X-axis = <u>2.1</u> mm

  Peak relative length change = 2.92 mm
- Initial diagonal cord elongation along X-axis = 3.10 mm

  Remaining elongation = .18 mm at ultimate load

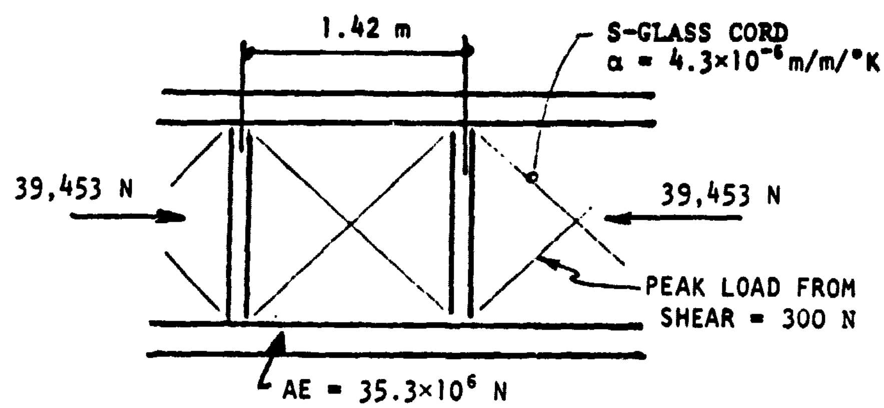


Figure 3.2.1-6. Machine-Made Beam Elevation View

Creep is not expected to be significant for the stress level in the glass which is approximately 3% of ultimate (but remains to be demonstrated by development tests). Hence, all cables are maintained in tension.

Again, referring to Table 3.2.1-3, the same analysis conducted for the single bay of the tri-beam longitudinal between STA N and O indicates the applied shear of 415 N (Table 3.2.1-4) will result in one cable in each of the faces being slack. This is not expected to be any problem, since the remaining tension cable is structurally adequate. Even so, this condition can be eliminated by shifting the control module so that the total platform mass center of gravity is on the neutral axis of the tri-beam.

The internal loads of Table 3.2.1-4 (elements 98 and 173) indicate ultimate shear loads up to 9158 N (2060 lbs) are imposed in the crossbeams at the lap joints. Variation of joint stiffness by two orders of magnitude of stiffness has not significantly changed these loads. The region of high shear is shown in Figure 3.2.1-7. These shears could result in the cord

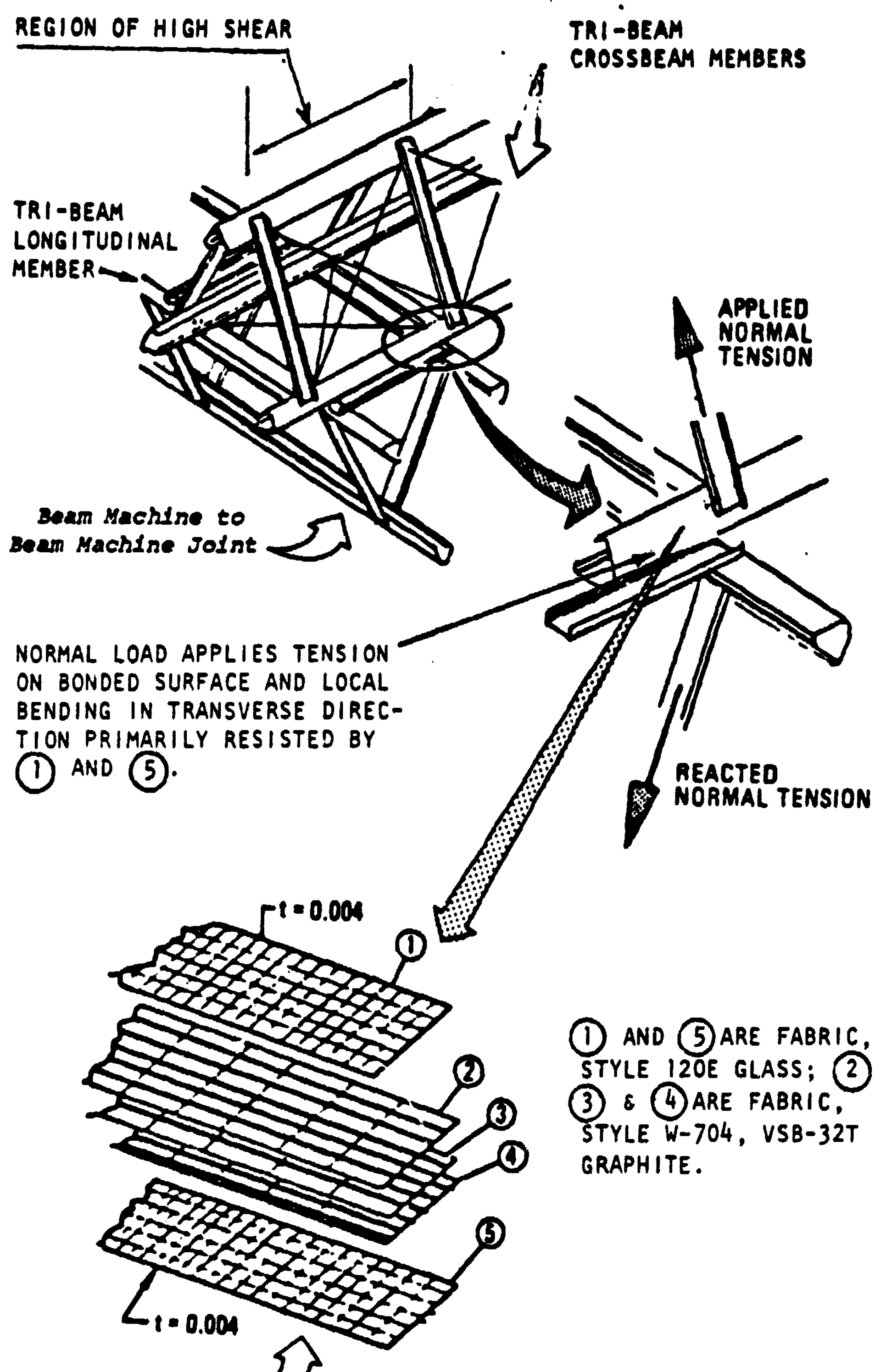


Figure 3.2.1-7.
Beam-to-Beam Load Considerations

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being slack in each of the three faces in the region of high shear. The loss of stiffness associated with the two cords being slack is expected to have a negligible affect on tri-beam stiffness. However, any future study should investigate the column suitability of the channel section cross-members of the machine-made beam and a small increase in cord cable diameter.

Lap Joint Strength Review

Of primary concern to this strength review is the machine made beam-to-beam joint capability. Figure 3.2.1-7 illustrates the general nature of the local tension loading resulting from joint moment. The concern is the transverse bending induced in the laminate.

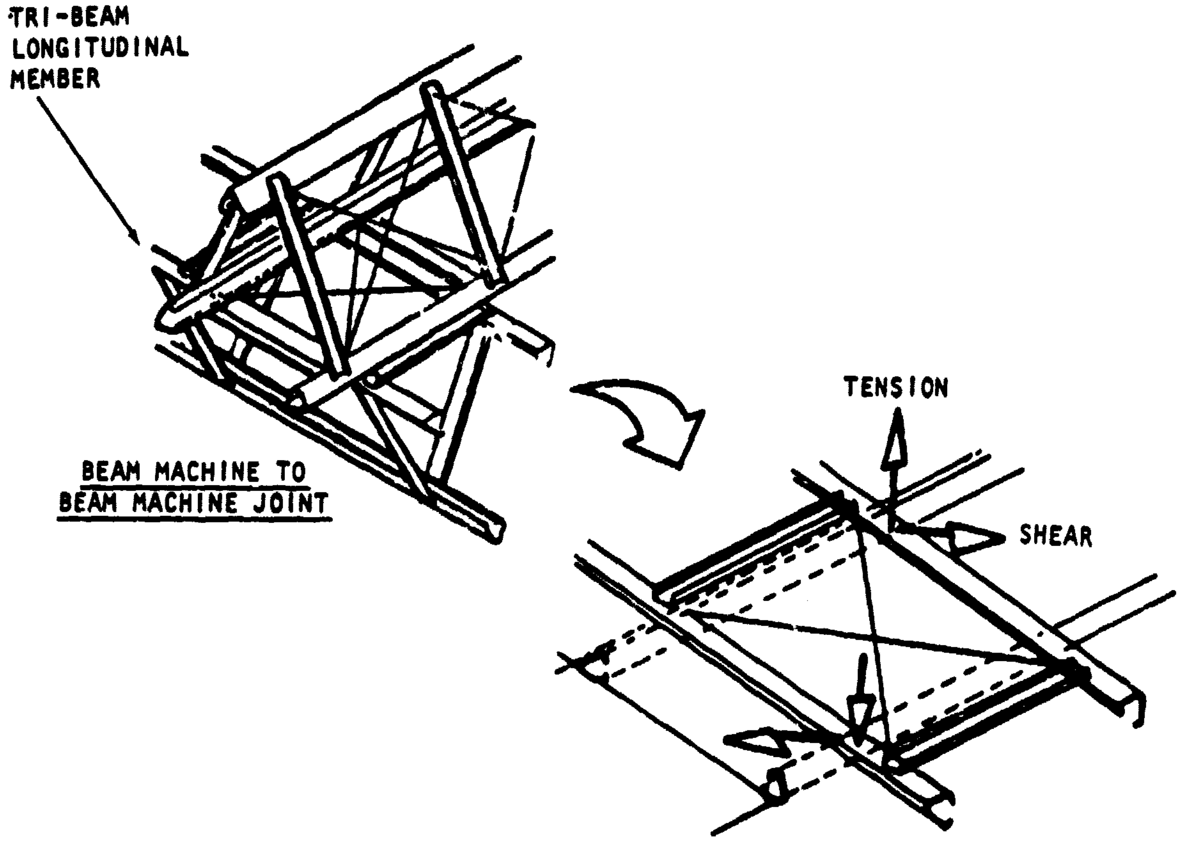
The joint loads have been determined from the NASTRAN model grid point loads output. The peak local tension and shear loads imposed at the cap-to-cap interface reinforced by the intersection fitting material (not shown in Figure 3.2.1-7) is tabulated in Figure 3.2.1-8. The loads were determined from a computer program that converted the X, Y, Z axis forces and moments (T1, T2, T3, R1, R2, R3) to the appropriate local shear and tension forces shown. The figure illustrates the nature of the peak ultimate local tension and shears which are 1170 N (263 lb) and 2750 N (618 lb). The shears appear within the design capability, but a development test similar to that illustrated in Figure 3.2.1-9 is required to verify the tension load suitability. This test configuration will satisfy the stated concern without the fabrication expense of two total beam sections.

It is pertinent to note that the above described peak tension loads are due to the center of mass of the RCS pods and antennas being offset from the attachment interface. Placement of the center of mass at the interface plane will result in the most critical tension being reduced to 312 N (70 lb). The port designs to achieve this would be unique.

Intersection Fitting Materials Investigation

As previously stated, the lap joint attachment concept utilizes the intersection fitting presented in Zones 23 and 24 of Drawing 42662-45A (page 2-25). The results of the materials investigation, conducted to determine the concept feasibility from a materials point of view, are presented herein.

The surfaces of the intersection fitting are coated with a thermo-plastic material which is heated by means of resistance wires built into the fitting. The intersection fittings are joined to the crossbeam at the No. 2 construction station by applying pressure while the contact surfaces of the fittings are heated. The crossbeam and fittings are then positioned on the longitudinal by the beam positioning device. Joining to the longitudinal is again achieved by pressure and resistance heating. The desirable characteristics of the thermo-plastic material are listed in Table 3.2.1-5. The first joining method conceived used laser heating, and a series of tests were conducted. Although the laser concept was supplanted by the idea of resistance heating (mainly for ease of construction), the results of the laser have some bearing on the problem of joining the intersection fittings and, consequently, they are included.



GRID POINT	T_1 (N)	T_2 (N)	T_3 (N)	R_1 (NM)	R_2 (NM)	R_3 (NM)	TENSION (N)	SHEAR (N)
2	-1856	-31	-27	160	-41	21	62	558
26	5665	979	-65	-466	1485	-22	777	1726
98	9158	801	-107	-525	2406	-69	1170	2750
170	8495	-276	NEG.	-106	1486	-84	625	2532

Figure 3.2.1-8. Maximum Ultimate Joint Loads

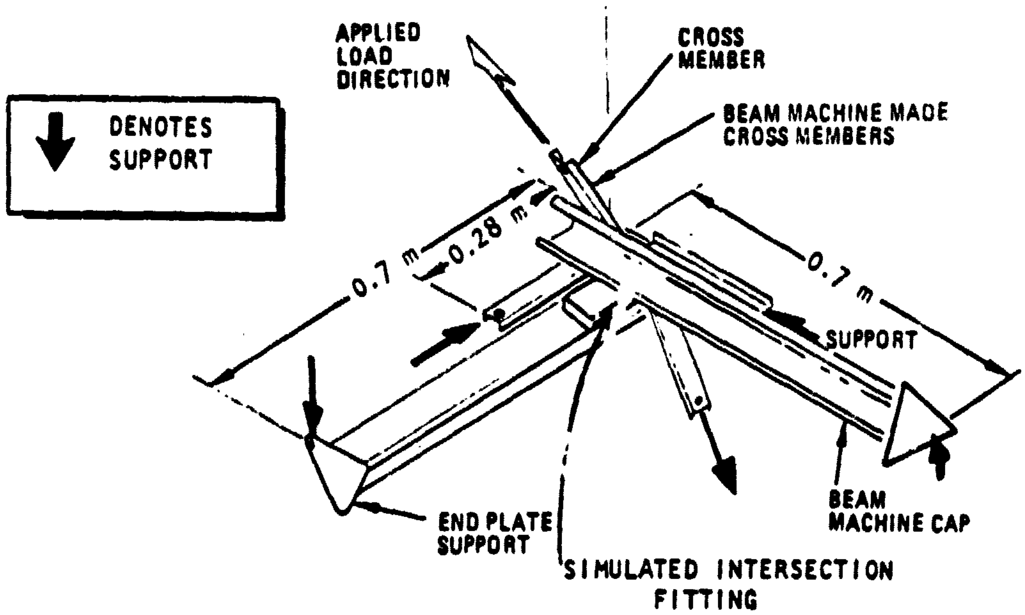


Figure 3.2.1-9. Development Test Article—Lap Joint Capability

Table 3.2.1-5. Intersection Fitting Desirable Characteristics

• Thermo-plastic	• Low substrate contact angle
• 350°F minimum softening temperature	• No degradation on remelt
• Narrow softening range	• Compliant
• No volatiles, low vapor pressure	• 30-year life in vacuum
• Substrate adherent	
• High surface tension	

Initial tests were conducted by laser-bonding single-lap shear specimens of a graphite-epoxy composite (GY70/E793) using a thermoplastic adhesive (Henkel Corp. Versalon 1200). The adhesive was premelted in vacuum to remove residual volatiles from the manufacturing process and applied to one of the graphite-epoxy coupons. A coated and an uncoated coupon were placed in contact under a few grams force pressure. The area of the joint was then irradiated on one side with the expanded beam of a CO_2 laser at a target irradiance of 1.5 W/m^2. Adhesive melting took place and capillary forces caused flowing through the entire contact area. A full bond with fillet resulted.

While a thermoplastic adhesive is the first choice of a bonding agent, since repair or separation is readily accomplished by remelting, thermosetting adhesives may also be used. A series of lap shear specimens were prepared by a similar technique using a thermosetting adhesive with scrim (3M AF143). Specimens were delivered to the test laboratory for tensile test machine evaluation. The results are not yet available.

The objective of the preliminary test was achieved—the demonstration that laser heating of a precoated contact zone could produce adhesive melting or cure without the local application of tooling to the contact zone.

Tri-Beam X-Bracing Pretension

A pretension of 2360 N (530 lb) ± 150 N (34 lb) will be used in each of the tri-beam X-bracing cords that are 4.7 mm (0.188 in.) in diameter. This pretension is sufficient to maintain pretension in the cords despite the peak machine-made beam shortening, transverse shear, and a 55°C increase in cord temperature relative to the caps. This is demonstrated as follows for the 12.9-m bay length between STA. M and N.

• Peak machine-made beam shortening due to 39,453 N compression in each cap	4.9 mm
• 4.7-mm cord thermal (55°C) induced length change along X-axis ($\alpha = 0.36\times10^{-6}$ m/m°C)	0.26 mm
• 4.7-mm cord shear induced length change along X-axis	5.52 mm
Peak relative change	10.68 mm
• Initial diagonal cord elongation along X-axis	11.2 mm
Remaining elongation at ultimate load	0.52 mm

Thrust Structure, Support Strut Assembly, and Outrigger Stabilizing Struts

For the individual elements of these major components, Table 3.2.1-3 describes the critical axial compression loads, member lengths, and resulting cylindrical tube inner diameter and wall thickness dimensions. All of these elements were sized as pin-ended elements; hence, using the Euler and local stability requirements

$$P = \frac{\pi^2 EI}{\alpha \ell^2} \text{ and } P = .40\pi E t^2$$

where α is a factor = 1.25 to preclude magnification of thermal gradient imposed deflection and other secondary effects (Section 5.3.1).

All of the above elements are sized by orbit transfer thrust. Referring to Figure 3.2.1-3, the inner thrust structure (elements 604 to 609) is sized by second-stage boost, while the remainder of the thrust structure is sized by first-stage boost (outboard engines firing).

Orbit Transfer Configuration Modal Analysis

The analysis performed to estimate the minimum natural frequency of this platform in the orbit transfer mode is discussed herein. As stated previously, the antenna feed columns and solar panel are stowed. Presently, a minimum requirement has not been established. The minimum modal values are, therefore, presented for future guidance and control evaluations.

The analysis was performed on NASTRAN for the start of orbit transfer with all propulsion modules in full. The following data were obtained:

Start of Orbit Transfer

First mode (torsion) frequency = .033 Hz
Second mode (bending) frequency = .099 Hz

It is noted that these values are expected to be 50 to 100% higher for the actual platform shown on Drawing 42662-45A. Also, it is pertinent to note the torsional frequency, if necessary, can be increased by a factor greater than three by a combination of reduced bay spacing to 5.04 m, increased tri-beam depth to 5.4 m and doubling of the X-bracing cord area. The first item would represent the most significant increase in construction time. Use of the latter two options alone would represent an increase factor of approximately 1.5. Increased X-bracing cord area represents increased pre-tension loads.

Operational Configuration Modal Analysis

To determine the minimum natural frequency of this platform, it was necessary to estimate the structural characteristics of the 20.5, 13.8, 7.5, and 6-meter-diameter antenna feed columns, reflector structure, and reflector structure support boom. The EI and GJ data shown in Table 3.2.1-2 were derived from analysis of the derived stick model shown in Figure 3.2.1-5. The GJ and EI data shown were determined to provide a natural frequency at least = 0.10 Hz for the antenna mounted to an infinitely rigid base.

The antenna structure data shown were incorporated into the previously described model of the tri-beam. Figure 3.2.1-10 presents the first modal frequency obtained from the described NASTRAN model. The frequency determined is .007 Hz (0.005 required) and is a torsional mode. This value is 50 to 100% conservative for the actual platform.

The same comments pertaining to increase of the minimum modal frequency (if required) as stated for the orbit transfer modal analyses are applicable.

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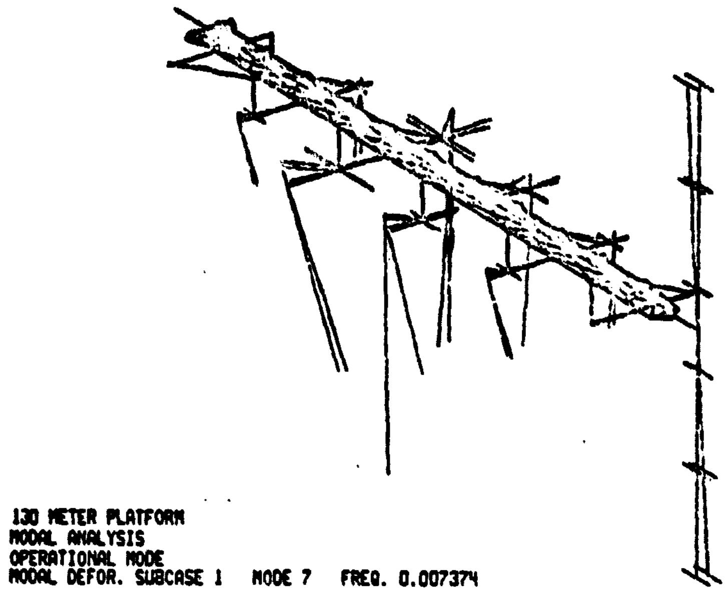


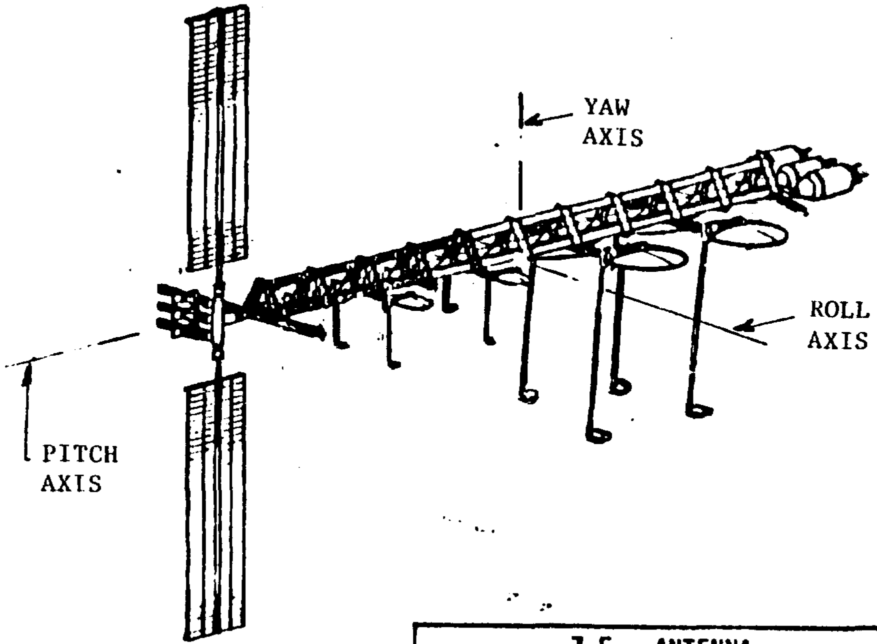
Figure 3.2.1-10. CRT Plot Operational Configuration Minimum Modal Frequency

Platform Dimensional Stability Analysis

A summary of the antenna and platform structural deformation errors relative to the CMG reference is shown in Figure 3.2.1-11. The data shown were estimated from similar data derived for the 230-meter platform synthesized in Part I of the study. The peak error will be less than six arc-min., providing pitch attitude control and N-S stationkeeping maneuvers are not concurrent. Otherwise, a settling time approaching one hour may be required.

REFERENCES

1. Space Construction Automated Fabrication Experiment Definition Study (SCAFEDS), Convair Division, General Dynamics, CASD-ASP77-017 (26 May 1978).



- ATTITUDE CONTROL—1 LB THRUSTERS
- E-W STATIONKEEPING—1 LB THRUSTERS
- N-S STATIONKEEPING—10 LB THRUSTERS
- ANTENNA DESIGN—0.10 Hz
- OUTRIGGERS—TORSION BRACED

ERROR SOURCE	AXIS	7.5-M ANTENNA				20.5-M ANTENNA			
		FEED COLUMN	REFLECTOR	TRI-BEAM		FEED COLUMN	REFLECTOR	TRI-BEAM	
				LOCAL	BODY			LOCAL	BODY
THERMAL GRADIENTS	Noted	1.3 ROLL PITCH	.40 ROLL YAW	NEGL.	.9 ROLL YAW	1.8 ROLL PITCH	.5 ROLL YAW	.30 ROLL	1.5 ROLL YAW
PITCH ATTITUDE CONTROL	PITCH	<.10	NEGL.	NEGL.	1.8	<.10	NEGL.	NEGL.	2.6
NORTH-SOUTH STATION-KEEPING	ROLL	1.5	NEGL.	.5	NEGL.	.6	<.10	.1	NEGL.

Figure 3.2.1-11. Structural Deformation Contribution to Pointing Error (Minutes)
(100-m-Long Platform)

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3.2.2 Electrical Power

3.2.2.1 Summary

The electrical power subsystem (EPS) provides 24.2 kW electrical load (BOL) continuous power capability in LEO operations and 44.5 kW in GEO to satisfy ETVP subsystem and platform payload requirements. A sketch showing the component layouts is presented in Figure 3.2.2-1. Key features of the EPS subsystem include:

- Asymmetrical solar array configuration
- 600-m^2 solar array area (60-kW)
- 30 kW capability to any one of 8 payload interfaces
- Seven 50 Ah Nickel Hydrogen batteries (66.8 kWh)
- Dual buses to insure power redundancy
- Redundant bus switching
- Power lines separated from control and data lines
- Deadfacing through switchgear provided on power side of payload

The solar array was sized to satisfy the load requirements given in Table 3.2.2-1 utilizing SEP solar array technology. The driving requirements are to deliver power at a specified voltage (250 V dc) and power level (24.2-44.5 kW) on a continuous basis for a duration of approximately 20 years. Periodic maintenance and repair requirements are to be considered.

A summary of the EPS assemblies is given in Figure 3.2.2-2. The components of the EPS were selected on the basis of space construction considerations (ease of installation), weight and technology status. Nickel hydrogen batteries provide a longer life and lower weight system. Potentially, NiH_2 offers lower cost. SEPS solar array technology was ground ruled since this technology readiness has been established and NASA plans to develop the hardware for other programs (e.g., SEPS, Power System).

Major configuration issues that were evaluated during the study are given in Table 3.2.2-2. Further discussion of these trade issues is presented in following paragraphs.

3.2.2.2 Subsystem and Hardware Characteristics

The overall configuration of ETVP is shown in Figure 3.2.2-1. Four solar array wings are located at the forward end. Power from the array is transmitted through a rotary joint to the system control module (SCM). From the SCM power is distributed to subsystems and payload attach ports (eight). The Electrical Power functional schematic is shown in Figure 3.2.2-3. Battery charging utilizes dedicated solar array areas. Provisions are made on the array for switching to excess battery charging array area for supporting other loads, e.g., during GEO operation where battery charging demand is small. Note: Other battery charging concepts are available (e.g., load bus voltage for charge and boost regulator for discharge) but were not traded off. Details of EPS system control are

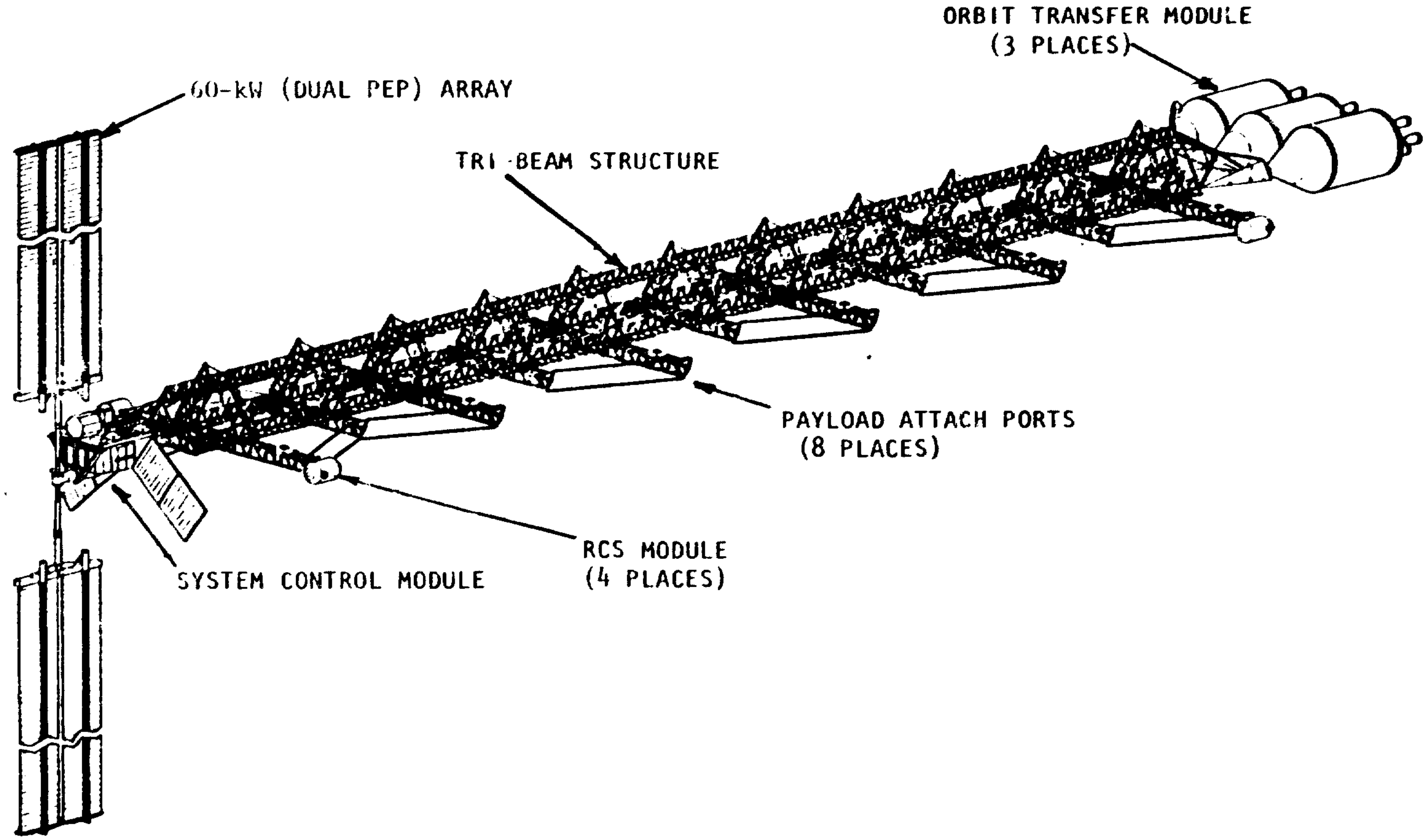


Figure 3.2.2-1. Engineering and Technology Verification Platform (ETVP)

Table 3.2.2-1. Power Requirements

•SUNLIGHT OPERATION: LEO, 200 NMI					GEO				LEO-TO-GEO ORBIT TRANSFER	
	GENERATED AVAILABLE POWER				GENERATED AVAILABLE POWER					
	POWER (kW)	ENERGY (kWh)	AFTER LOSSES (kW)	USEFUL POWER (kW)	POWER (kW)	ENERGY (kWh)	AFTER LOSSES (kW)	USEFUL POWER (kW)	GEN. POWER (kW)	USEFUL POWER (kW)
SOLAR ARRAY TOTAL	60.0*				60.0*				60.0*	
S/A TO MAIN BUS	30.3		24.2		55.7		44.5		30.3	
S/A TO BATTERY	29.7	27.5(a)	(b)		4.3	98.2	(d)			
SCM				5.0		(c)		5.0		5.0
RCS				2.0				2.0		
PROPULSION MODULE										6.0
PAYLOAD				17.2				37.5		
SUBTOTAL				24.2				44.5		11.0
• ECLIPSE OPERATION										
BATTERY TO BUS	30.3	18.3	24.2		55.7	65.2	44.5		30.3	
SCM		(e)		5.0		(f)		5.0		5.0
RCS				2.0				2.0		
PROPULSION MODULE										6.0
PAYLOAD				17.2				37.5		
SUBTOTAL				24.2				44.5		11.0

*BOL POWER

(a) 0.926×29.7 = 27.5 kWh
(b) 0.99×0.90×0.996×0.90×30.3 = 24.2 kW
(c) (24-1.17)×4.3 = 98.2 kWh
(d) 0.99×0.90×0.996×0.90×55.7 = 44.5 kW
(e) 0.604×30.3 = 18.3 kWh
(f) 1.17×55.7 = 65.2 kWh

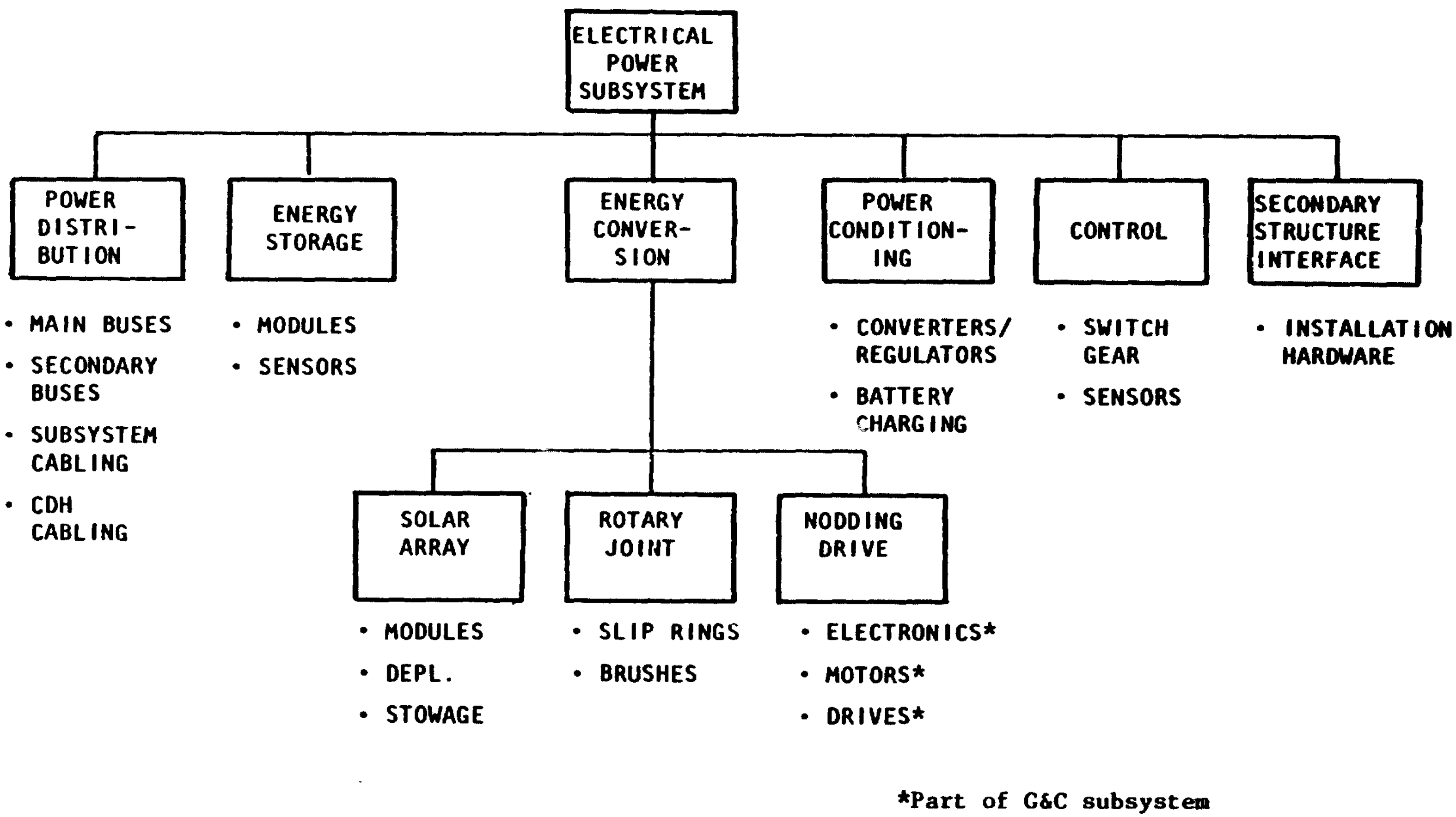


Figure 3.2.2-2. Electrical Power Subsystem Assembly Tree

Table 3.2.2-2. EPS Major Configuration Issues

ELEMENTS	SELECTION	RATIONALE	OPTIONS	COMMENTS
SOLAR ARRAY	LOCATED AT FWD END OF PLATFORM	OVERALL CONFIGURATION CONSIDERATIONS (LESS COMPLEX PM INTERFACE & EASIER CONSTRUCTION TASK)	SYMMETRICAL (ONE ON EACH END)	SMALL EFFECT ON POWER DISTRIBUTION
	SEPS TECHNOLOGY BASE	AVAILABLE AND COMMONALITY WITH OTHER NASA PROGRAMS	ADV. TECHNOLOGY (E.G., 2-MIL CELLS)	
POWER DISTRIBUTION	SINGLE-POINT GRND IN CONTR. MODULE	REDUCES POSSIBILITY OF GROUND LOOPS		
	DUAL BUSES FOR PWR DISTRIBUTION	REDUNDANCY IN POWER TRANSFER		
	TSP & COAX FOR C&DH	ISOLATES FROM POWER BUS TO MINIMIZE NOISE INTERFERENCE		
	FLEXIBLE CONTINUOUS CABLING	EASE OF SPACE CONSTRUCTION (HANDLING & INSTALLATION)	SECTIONALIZED BUSES, RIGID CABLING	
	HIGH VOLTAGE (250 VDC)	REDUCE WEIGHT, ENHANCES FLEXIBILITY & EASE OF INSTALLATION		250 VDC UPPER LIMIT DUE TO CONVERTER/REGULATOR CONSTRAINTS
ENERGY STORAGE	NiH_2 BATTERIES	WEIGHT ADVANTAGE, COST EFFECTIVE, LONGER LIFE	NiCd	
SWITCH GEAR	VACUUM INTERRUPTERS	HIGH SPEED, HIGH VOLTAGE/CURRENT STATE OF ART, NO CONTACT MAINTENANCE (SEALED—NO ARCING), LONG LIFE, NOT SENSITIVE TO MOUNTING ORIENTATION	KNIFE SWITCHES	CONTACTS ARE OVER-DESIGNED TO ACCEPT SURGE CURRENTS
ROTARY JOINT	SLIP RINGS AND BRUSHES	MAXIMUM FLEXIBILITY, WITHIN STATE OF ART	FLEXIBLE CABLES	
CONVERTERS/ REGULATORS	SW. CONV./REG. (HIGH FREQ. SW.)	WEIGHT AND EFFICIENCY		

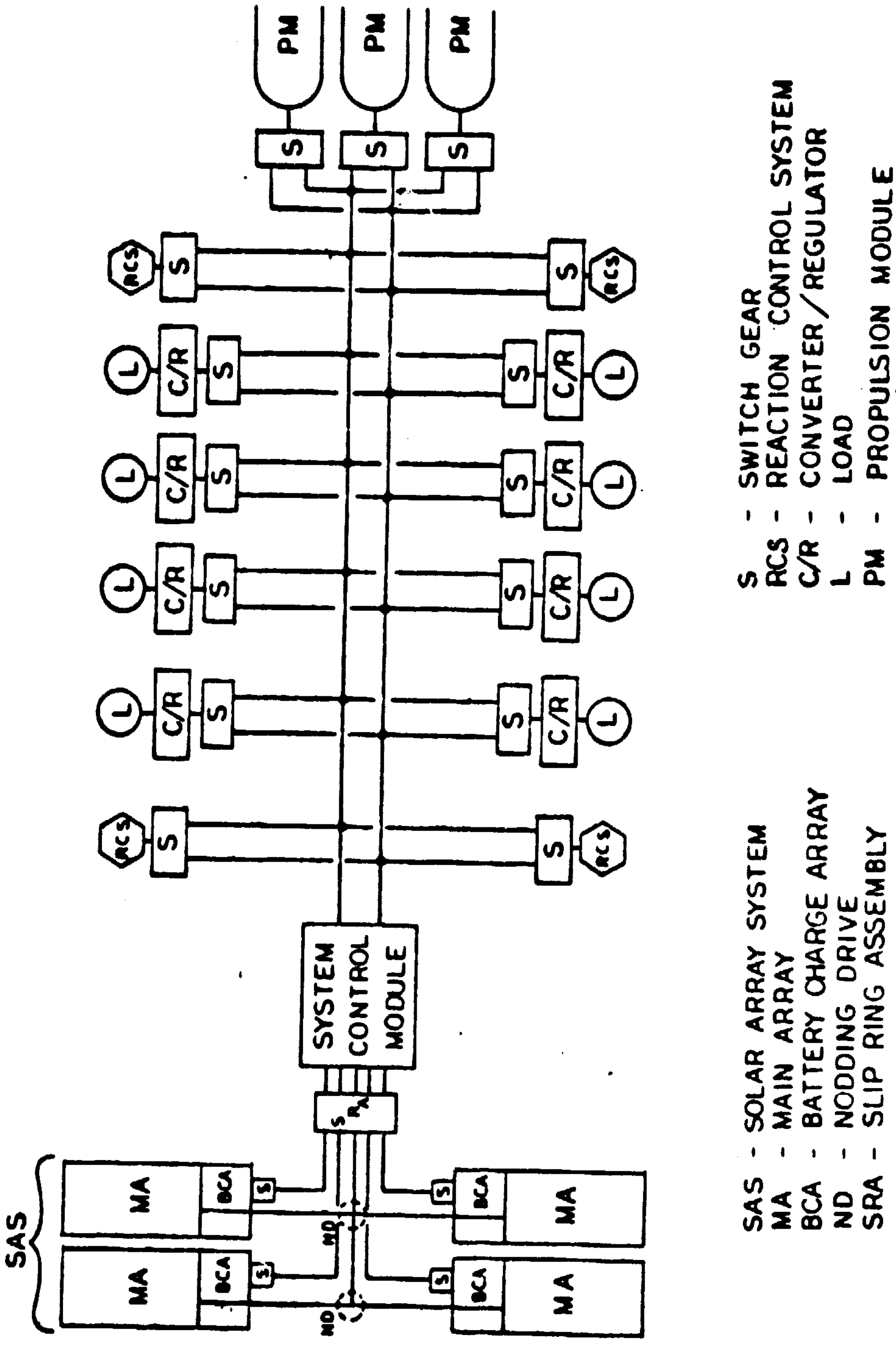


Figure 3.2.2-3. Electrical Power Functional Schematic

shown in Figure 3.2.2-4. Four power lines from the arrays supply seven battery charge units (seven batteries). Power from the seven batteries (during eclipse) is fed into the main power bus network for distribution.

The solar arrays (4 wings) will output 60 kW (BOL). Low Earth Orbit operations are relatively short in duration (in the order of two years) and very little (negligible) degradation due to the space environment will occur. GEO and orbit transfer solar array degradation were not evaluated. The voltage level is 250 V dc (unregulated) at the EPS/subsystem interface. Any modification required by the loads is undefined and further investigation is required. Batteries are provided to supply full operational capability during eclipse. Solar array sun tracking is provided by G&C subsystems.

The EPS has the capability to provide up to 30 kW at any one of 8 payload interfaces. Redundant distribution buses are included in the baseline concept. Physical characteristics of each major component is shown in Table 3.2.2-3.

3.2.2.3 Available Power and Battery Capacity

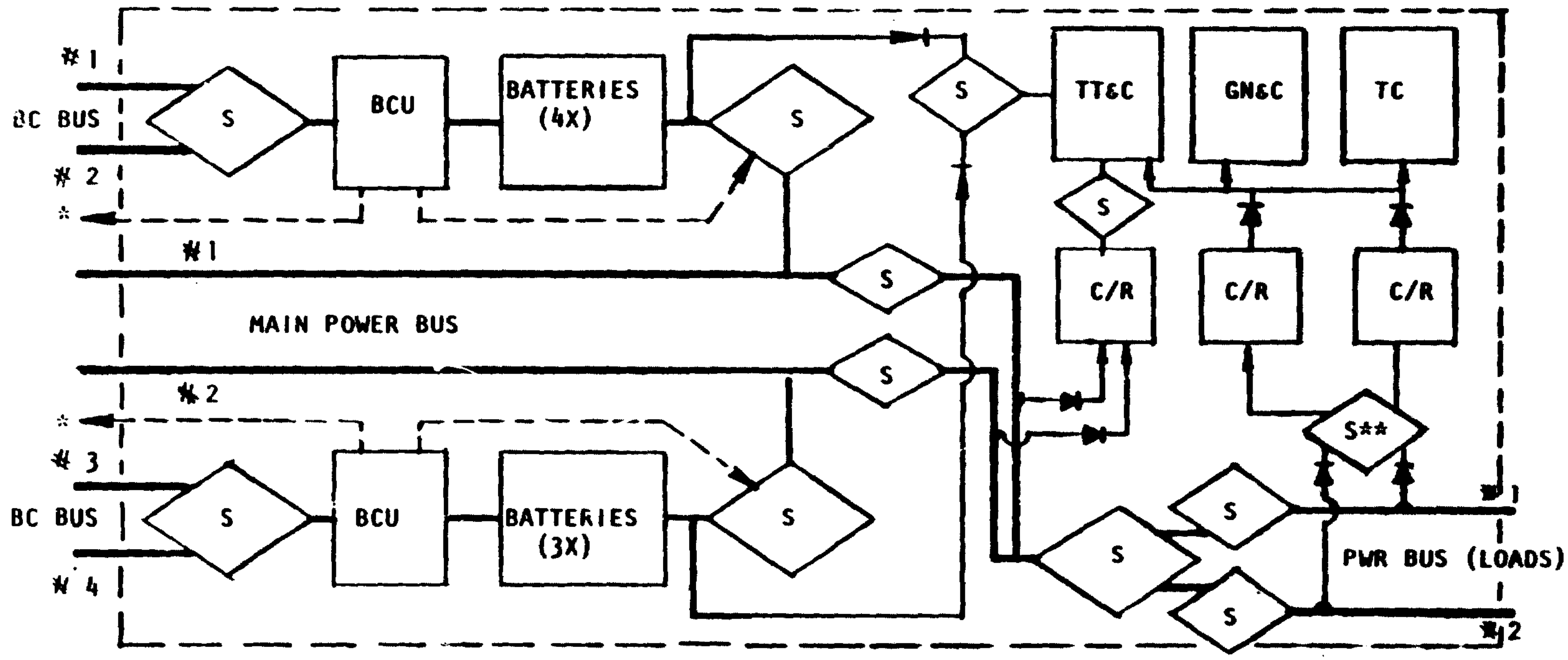
Criteria to determine available power and required battery capacity are based on the mission profile (Figure 2.3.1-1) assuming worst case eclipse periods for each orbit, and a BOL total solar array capability of 60 kW. Furthermore, a switching arrangement will be used where 3 solar cell modules are connected in series for battery charging versus the usual 2 modules for bus voltage. For each orbit under consideration it can then be determined what portion of the array is needed for battery charging, and also the available array power for the payload as illustrated in Table 3.2.2-4.

A block diagram of the EPS showing system efficiency is presented in Figure 3.2.2-5. An energy balance will be accomplished through proper switching of dedicated solar array modules into the main bus through a microcomputer when they are no longer needed. The microcomputer will sense the battery voltage level and when properly charged will disconnect portions of the solar array modules to achieve an energy balance condition.

3.2.2.4 Power Generation

The primary power generation system consists of the four solar array wings. The major components of the solar array are: two extendable/retractable 39 m (1536 in) masts, two solar cell blankets 37.5 m (1476 in) by 3.99 m (157 in), preloadable covers, ascent support containers, blanket tensioning guide cable systems, canister deployers and support structures. A sketch of one PEP designed wing is shown in Figure 3.2.2-6. The masts extend and retract the foldable blankets. The panels are mechanically hinged to each other along their long dimension to permit storing of the panels. The solar cells in the electrical module are electrically configured 5 cells in parallel by 306 cells in series.

Each module is configured to produce a nominal voltage of 129 V. Two modules are connected together in series to provide a nominal voltage of 258 V. The power produced by each panel is 316 W (BOL) for a total of 15.8 kW per wing.



LEGEND

BC BUS—BATTERY CHARGE BUS
BCU —BATTERY CHARGE UNIT, CONTAINS CHARGE/DISCHARGE CONTROLLER
C/R —CONVERTER/REGULATOR
S —SWITCH GEAR
* —TO SWITCH GEAR ON SOLAR ARRAY SYSTEM (SAS)
** —SWITCH GEAR CAN CONNECT EITHER C/R TO BUS NO. 1 OR NO. 2

TT&C—TRACKING, TELEMETRY & COMMAND SUBSYSTEM
GN&C—GUIDANCE, NAVIGATION & CONTROL SUBSYSTEM
TC —THERMAL CONTROL SUBSYSTEM

Figure 3.2.2-4. EPS—System Control Module

Table 3.2.2-3. Electrical Power Distribution and Control Characteristics

COMPONENT	QTY	UNIT WT. (kg)	DIMENSION OR VOLUME (cm^3)	AVERAGE PWR REQ. (WATTS)	DUTY CYCLE†	COMMENTS
SOLAR ARRAY WING	4	612	4 m 37.5 m (EACH)	TBD	S	OUTPUT PWR—60 kW (EOL) NORMAL TO SUN VECTOR
ELEC. POWER DISTRIB.						
MAIN BUSES	TOTAL SUBSYSTEM WEIGHT	140	25 SQUARE MILLIMETERS		C	TRANSMISSION η = 90%
SECONDARY FEEDERS		80.5	25 SQUARE MILLIMETERS		C	
INTERTIE BARS		5.0			C	(60 kW RATING ON DISTRIBUTION)
PWR MOD SUM BUS		2.0	AWG NO. 9		C	
PWR MOD BUS		1.5	AWG NO. 14		C	
INTERCONNECT WIRING		5.0			C	
C&DH: COAX		5.5	COAX		C	
TSP		13.8	TSP		C	
SUN ORIENT. MEMBERS						
MAIN		13.5	25 SQUARE MILLIMETERS		S	
SECONDARY FD.		17.8	25 SQUARE MILLIMETERS		S	
INSTALLATION ~ 10%		28.5			S	
ENERGY STORAGE						
CELLS + CONTAINER	49	67	CELL: DIA = 8.9 cm;	-	E(C)	7 BATTERIES WITH
BATTERY CHARGER (C/D,			HEIGHT = 22.4 cm;			210 CELLS/BATTERY
FLEX CABLES,			30 CELLS PER MODULE	-		(50 AH CELLS)
SENSORS, HEATERS	7 ASSY	28.29			C	
INSTALLATION (10%)		2.86				
SWITCH GEARS						
TRIPLE SW. GEAR	8	11.25		14	C	SIZED FOR 30 kW @ 30 V
SINGLE SW. GEAR	23	3.94		4	C	
INSTALLATION (10%)		18.06				
ROTARY JOINT					C	
DRIVES: α-AXIS	1	36		TBD	S	
β-AXIS	2	17		TBD		
FLEX LINES	1	7				
CONVERTER/REGULATOR	2	7.2	20.32×20.32×30.48 cm	500	C	15 kW RATING
	1	0.2	10.16×10.16×15.48 cm	100	C	MIN. RATING (APPROX. FEW HUNDRED WATTS)

†C = CONTINUOUS, S = SUNLIGHT, E(C) = ECLIPSE (CONTINUOUS)

Table 3.2.2-4. Available Power Calculations

ORBIT ALTITUDE (NMI)	ORBIT PERIOD (HR)	ECLIPSE [E] DURATION (HR)	SUNLIGHT OPERATION* BATTERY CHARGE (Wh) [1.5×E×P]	BATTERY CHARGE (Wh)/SUNLIGHT (h)
200	1.53	0.604	0.906 P	0.906 P/(1.53-0.604) = 0.978 P
300	1.60	0.593	0.890 P	0.890 P/(1.60-0.593) = 0.884 P
500	1.73	0.583	0.875 P	0.875 P/(1.73-0.583) = 0.763 P
19,300	24	1.17	1.755 P	1.755 P/(24-1.17) = 0.0769 P

THE FOLLOWING SHOWS SOLVING FOR P (I.E., AVAILABLE PAYLOAD POWER)

ORBIT ALTITUDE (NMI)	TOTAL S/A CAPABILITY (W) [C]	C = 60 kW ▸ P (kW) [P]	BATTERY CAPACITY (Ah)** [(P×10³×E/250)/0.80]	QTY OF BATTERIES (50 Ah EACH)
200	1.978 P	30.33 kW	91.6 Ah	2
300	1.884 P	31.85	94.4	2
500	1.763 P	34.03	99.2	2
19,300	1.0769 P	55.72	326.0	/

*P IS AVAILABLE PAYLOAD POWER LOSSES

**DEPTH OF DISCHARGE (DOD) = 0.80

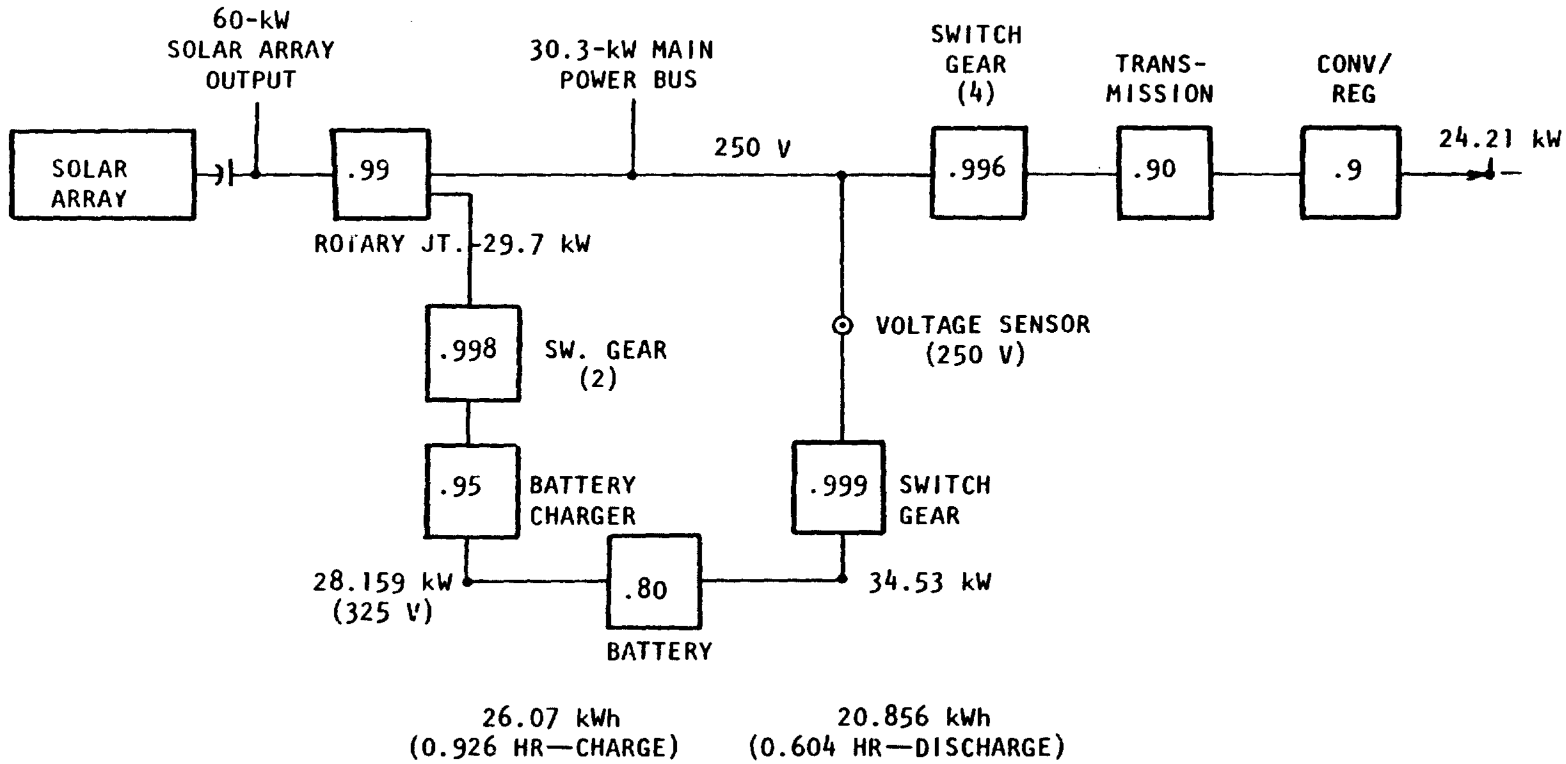
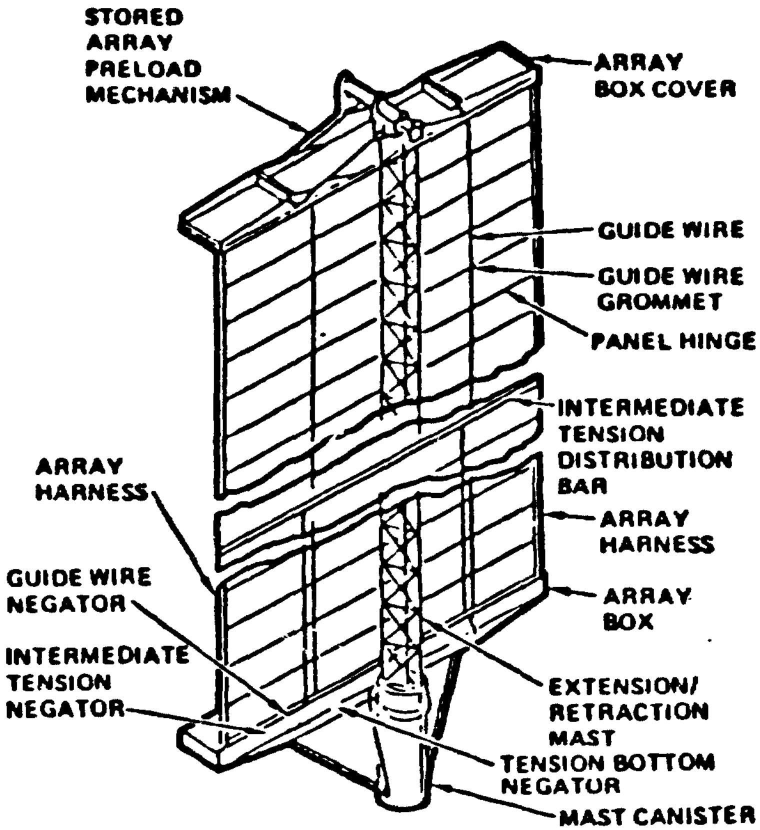


Figure 3.2.2-5. Block Diagram of EPS with Efficiency



	PEP
SI CELL THICKNESS (MIL)	8
CELL SIZE (CM)	2×4
CELL EFFICIENCY (%)	12.8
TYPE OF CELL	2×4 CM
COVER THICKNESS (MIL)	6
KAPTON SUBSTRATE THICKNESS	1/2 MIL
BLANKET HARNESS TYPE	A
INTERCONNECTS	WELDED
PANEL SIZE (M)	0.75×4
WING SIZE (M)	4×37.5
NO. PANELS/WING	50
AREA/WING (M^2)	150
OUTPUT OF WING (KW)	15.8
MASS	612 KG

Figure 3.2.2-6. PEP Wing for ETVP

A study was performed to trade the pros and cons of symmetrical versus asymmetrical location of the solar arrays. Appendix A details this study and explains why the asymmetrical configuration was selected.

Solar Array Harness

The array interconnect system is designed to have a single location for crossing a fold line. The connections are made at the outboard edge of the electrical modules, two per module. The array harness is a flat cable conductor (FCC) assembly mounted on the back at the two long edges of the blanket.

3.2.2.5 Power Distribution (PD)

The PD transfers required power from either solar arrays or batteries to the various loads. A number of trades were conducted to determine the most suitable PD concept:

Grounding

A single point ground scheme is preferred over multiple grounding to eliminate possibilities of ground current loops. The subsystem control module (SCM) was selected for the single ground point (reference point).

Bus System/Voltage Level

A dual bus system was selected to provide redundant power utilizing proper switching. Voltage levels of 250 V dc provided minimum weight witbin the constraints of existing converter/regulator hardware.

Cables

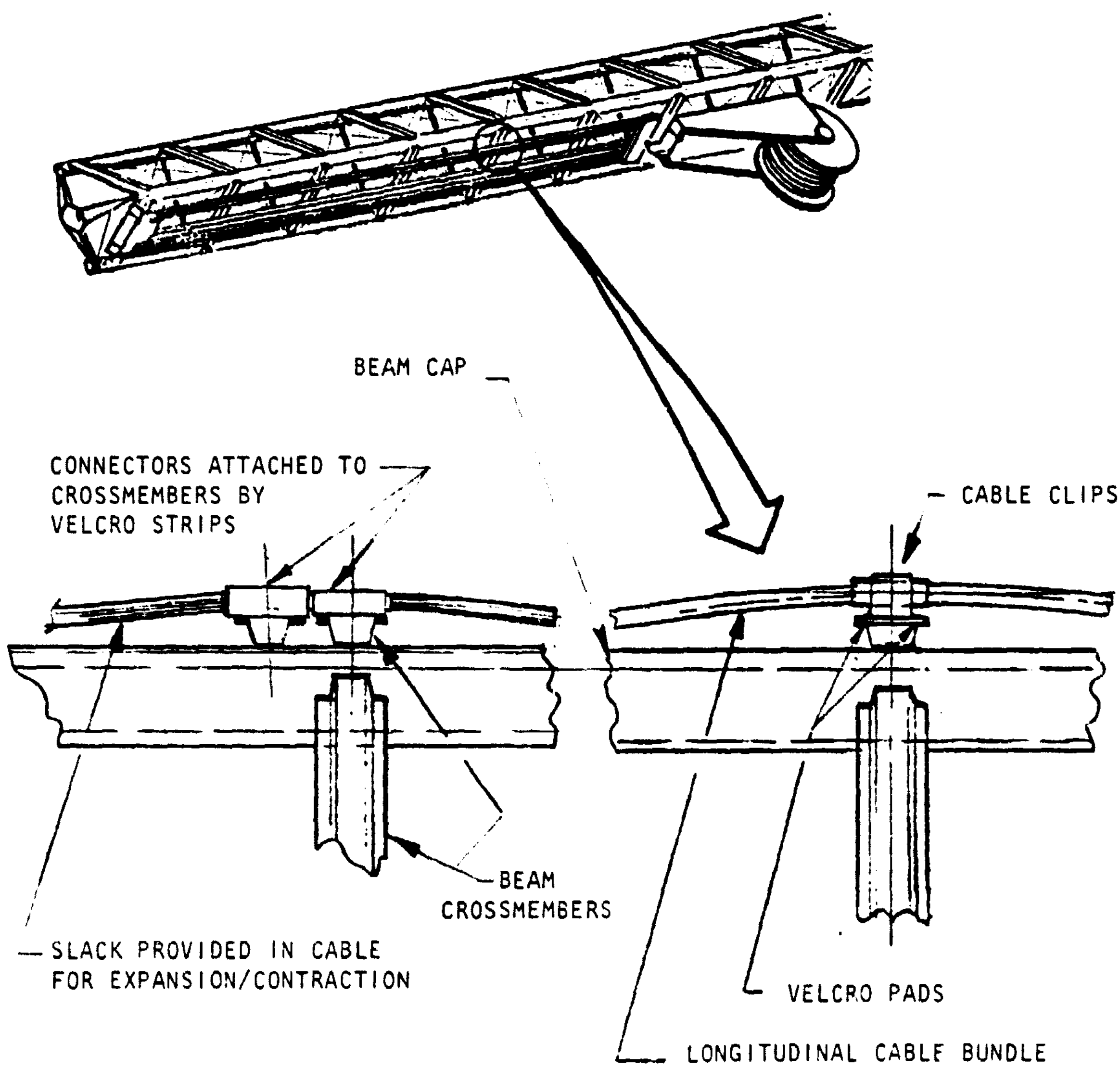
Consideration was given to segmented cables, rigid cables, and flex cables. Flex cabling proved to be most desirable due to construction and installation considerations. The installation of power cabling and command, data and housekeeping (CD&H) cabling is presented in Figure 3.2.2-7. It was decided that the CD&H lines should be separated from the power lines.

The main buses were sized to provide a maximum of 30 kW to any of the load points on the platform. The secondary buses to the RCS quads were sized for a load of 2 kW, and to the orbit transfer modules for a load of 6 kW. Deadfacing was provided to each of the payload interfaces through switchgear on the power system side. The various methods by which energy is transferred from the main buses to the secondary buses are presented in Figures 3.2.2-8 and 3.2.2-9.

The various EPS power distribution and CD&H cable sizes and the general layout for constructing the various harnesses are presented in Figure 3.2.2-10.

3.2.2.6 Energy Storage

The energy storage requirement is ~18.1 kWh in LEO and ~66.8 kWh in GEO. There are 7 batteries required each having 210 (50 Ah) cells. The cells were grouped into 49 replaceable modules and are mounted on the SCM. The packaging



- INSTALLED AT WORK STA. 1
- INTEGRAL CABLE RUN
  - SINGLE REEL
  - FLEX RAILROAD TRACK CONCEPT
  - INTEGRAL CROSSBEAM CONNECTORS
- AUTOMATED END CONNECTION
- AUTOMATED VELCRO ATTACHMENT

Figure 3.2.2-7. Wire Installation—Longitudinal Beam

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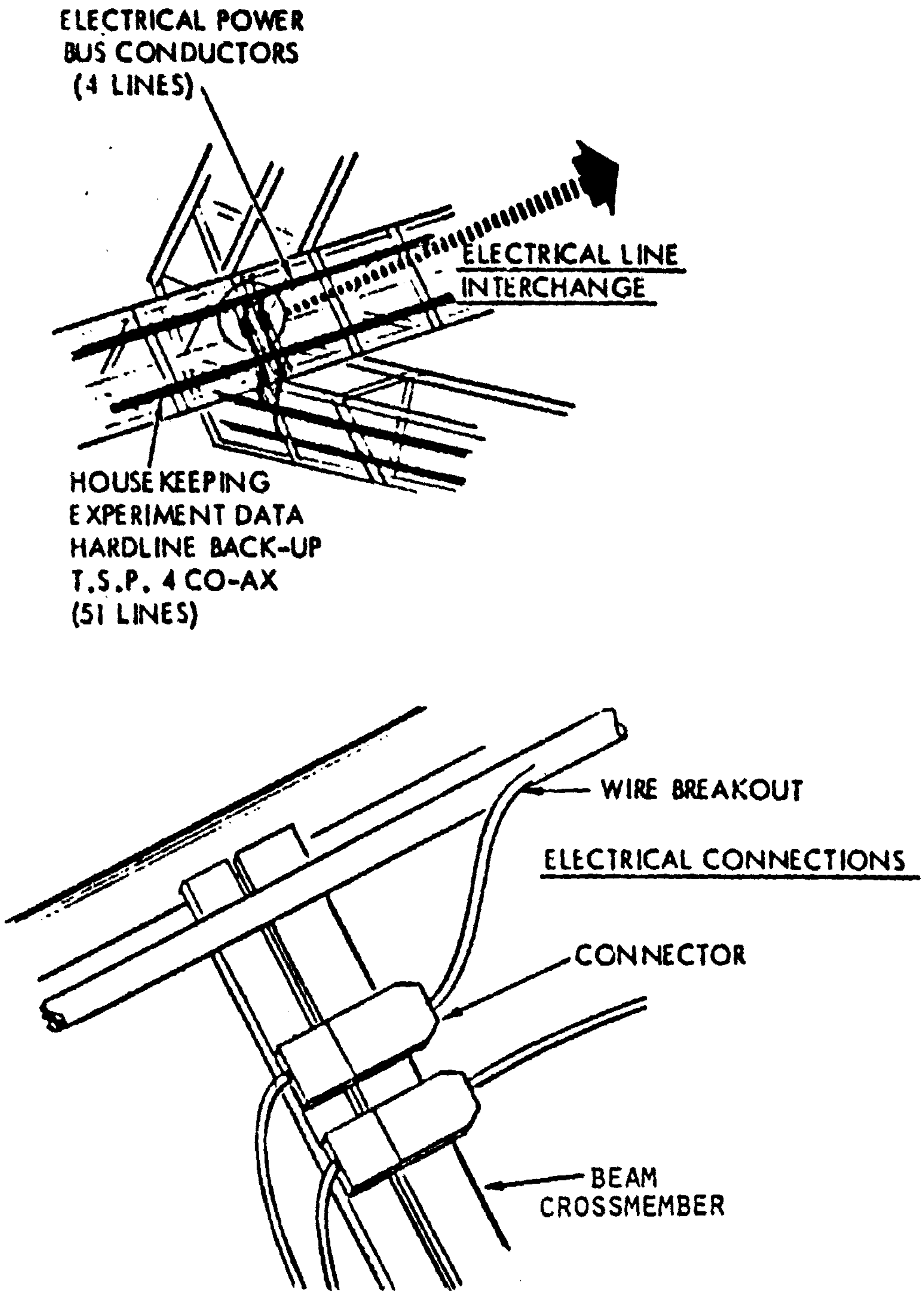

Figure 3.2.2-8. Electrical Power and Data Distribution Concept

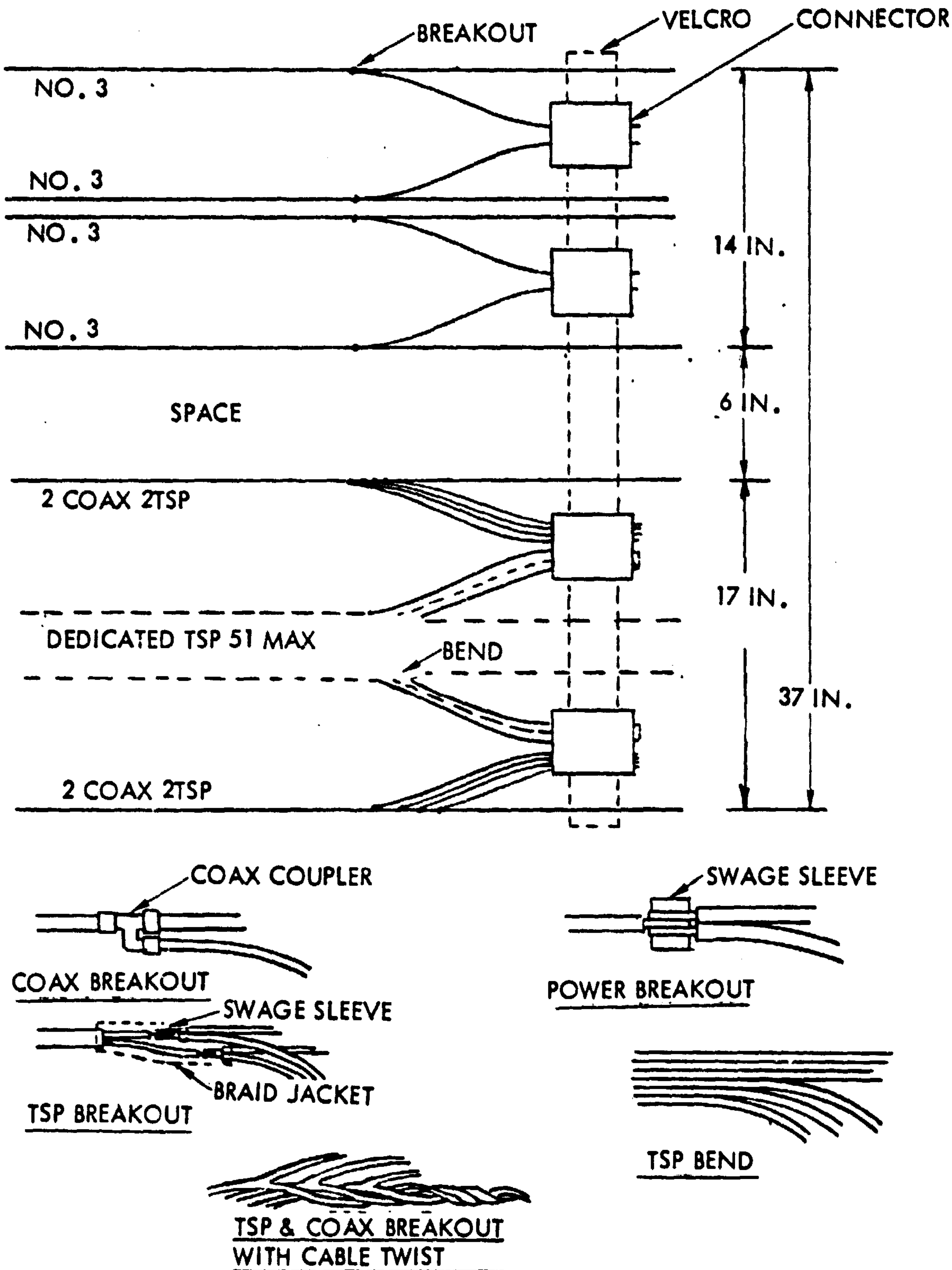

Figure 3.2.2-9. Longitudinal Cable—Side View

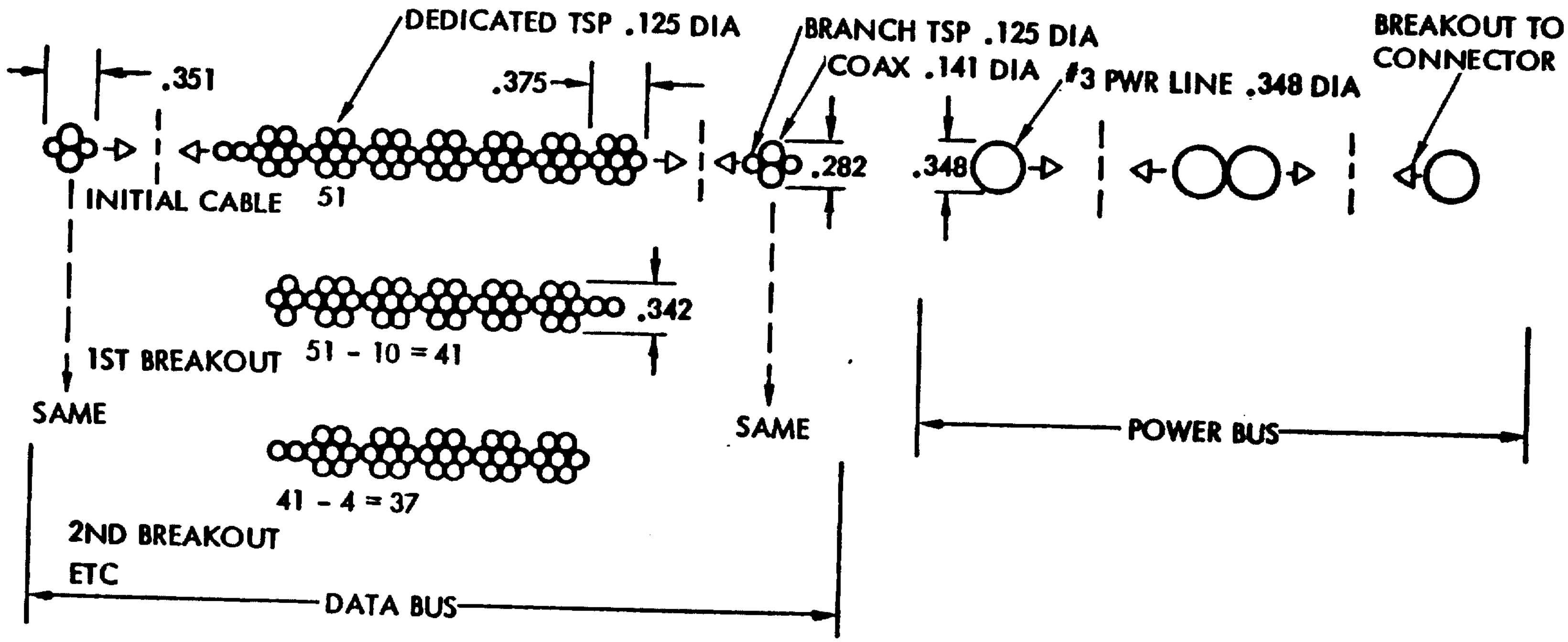

Figure 3.2.2-10. Longitudinal Cable—End View

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arrangement of the cells is presented in Figure 3.2.2-11. NiH_2 batteries potentially solve many of the problems associated with more conventional nickel-cadmium (NiCd) batteries. Specifically NiH_2 batteries do not require periodic reconditioning, are not damaged by extended periods of open-circuit standby, and tolerate both overcharge and over-discharge better than NiCd batteries. Each battery is provided with its own charger. NiH_2 batteries are being tested successfully at 80% depth of discharge. The NiH_2 battery cells are AFAPL-developed cells. The batteries are charged through the BCU which receives the power from the solar array. For the LEO configuration 64 solar array modules in parallel have been dedicated for charging the batteries and 10 modules are utilized in GEO. Knife switches are utilized to place the excess modules back into the PDS during GEO configuration. The switching scheme utilized for the energy balance system is presented in Figure 3.2.2-12.

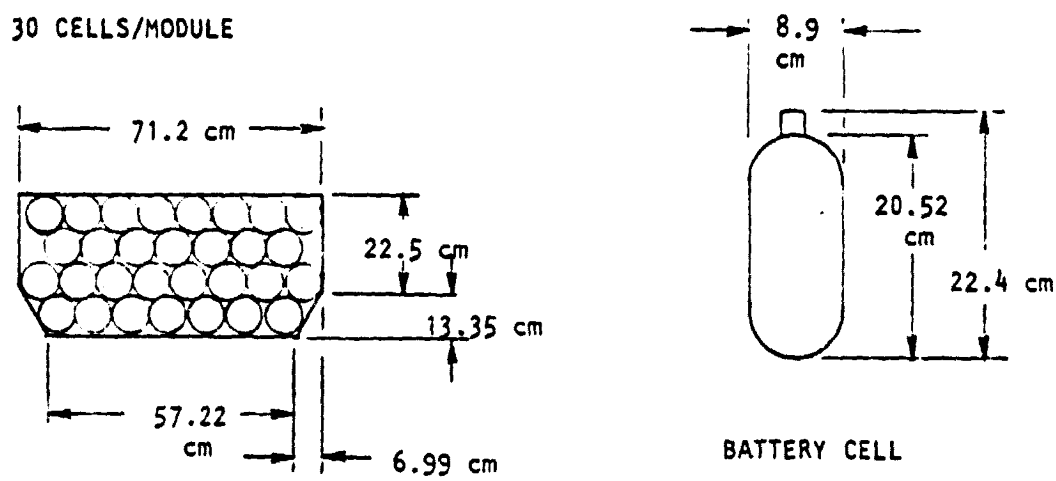


Figure 3.2.2-11. Battery Cell Arrangement

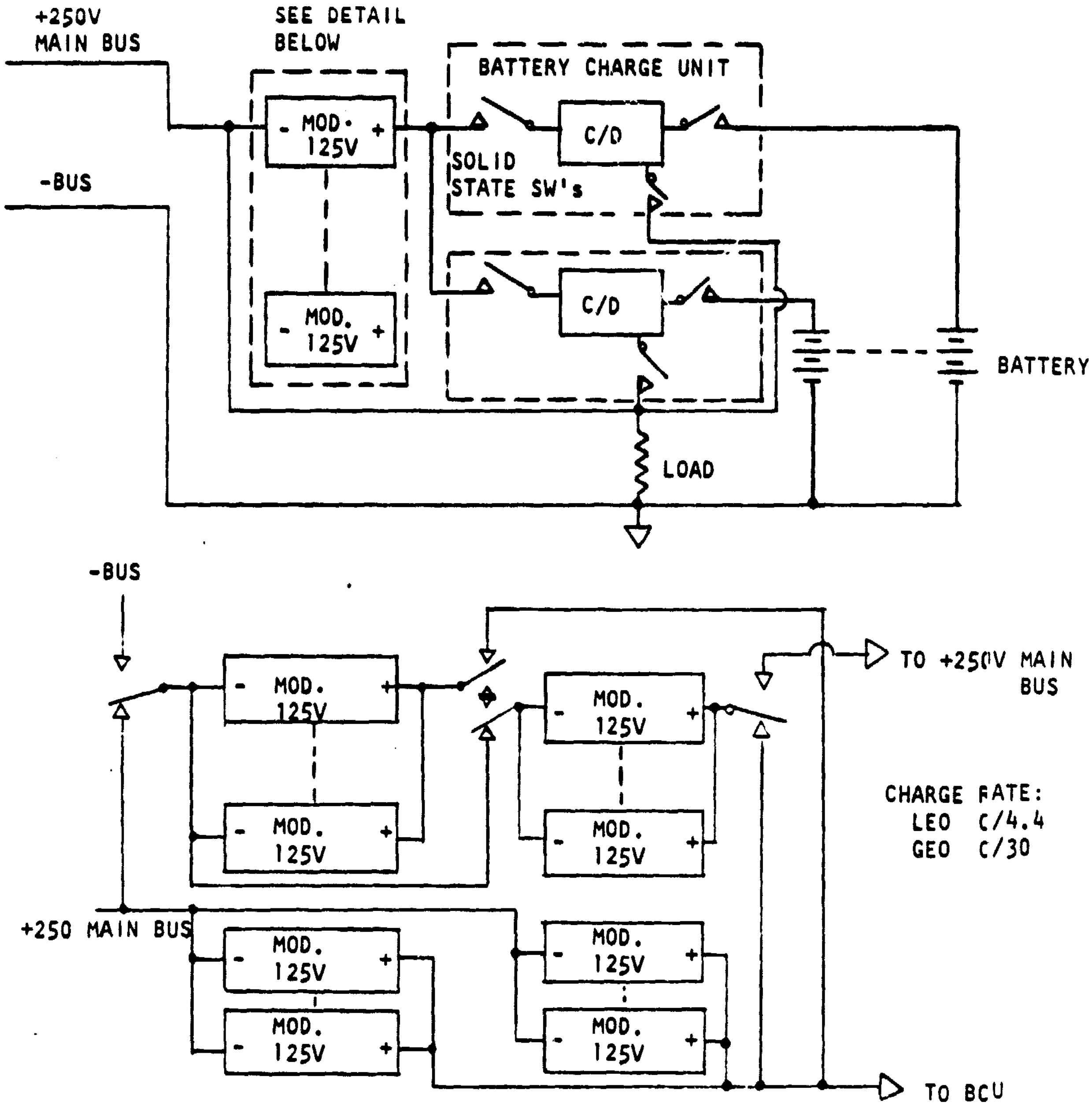
3.2.2.7 Switch Gear

Switch gear of the vacuum interrupter type was selected for interrupting power at the various locations shown in Figures 3.2.2-3 and 3.2.2-4. Power isolation is required for maintenance and repair and failure detection. Reasons for selecting the vacuum interrupter switch gears include:

1. Vacuum interrupters can be mounted in any orientation and configuration providing that the minimum required axial center line spacing between adjacent interrupters is maintained for electrical considerations.

2. These switches have low contact resistance which remains low and stable for the life of the contactor.

3. There is no contact maintenance required, because generally the contacts are hermetically sealed. Therefore, no arcing occurs during interruption.



FOR LEO 64 MODULES IN PARALLEL ARE REQUIRED FOR CHARGING
FOR GEO 10 MODULES IN PARALLEL ARE REQUIRED FOR CHARGING
REMAINING 54 MODULES ARE SWITCHED BACK TO MAIN BUS

Figure 3.2.2-12. Switching Arrangement for Energy Balance

4. The contacts generally can withstand currents many times greater than their ratings.

5. They exhibit longer life than other types of swtich gear.

In general two types of switch gear will be utilized: (1) single-pole/single-throw (SPST), and (2) double-pole/double-throw (DPDT). The SPST switches will be utilized where only power interruption is required. The DPDT switches will generally be used where power transfer or deadfacing is being considered. To minimize complexity and weight and reduce power losses associated with switch gear, knife switches are being considered for the array switching of the 54 modules in GEO. This switching is a one-time task and, therefore, the knife switching technique appears practical for this type of function. The switching of these modules is shown in Figure 3.2.2-12. The switch gear at the payload points is an integral part of the electrical connectors (see Figure 3.2.2-13) and consists of 3-DPDT switch gears providing power redundancy as well as deadface interfacing with the payloads.

3.2.2.8 Rotary Joint

A rotary joint is utilized to transfer energy from the solar array wings to the platform to meet the power requirements of the subsystems and payloads while allowing continuous rotation. The energy is passed through slip rings and brushes. The design criteria and the basic requirements are presented in Figure 3.2.2-14. To provide full orientation capability of each solar array at all times in GEO and LEO, the solar arrays are equipped with nodding drives, to allow rotation around the longitudinal axis of each array over angles of ±40 degrees. This provides full power capability throughout the maximum sun beta angle range (0 to 52 degrees) experienced by the platform.

3.2.2.9 Converter/Regulator

Converter/regulators provide the necessary voltage/power levels and regulation to support TT&C, GN&C, TC, orientation electronics, switch gear and sensors. Switching regulators control the output voltages providing high efficiency. The efficiencies of these converter/regulators varies between 85 to 92 percent (depending on the ratio of input to output conversion voltages). The converter/regulators used during normal operations have a power dissipation of approximately 500 W. The converter/regulator for emergency standby power is sized for the minimal power requirements associated with back-up communications. This assures the availability of diagnostic data at ground mission control in the event of major system failures. The power dissipation of the emergency converter/regulator is approximately 20 W. The electronics of the operational C/R was estimated to fit within a unit package of 20.31×20.32×48 cm, and the emergency C/R within a unit package of 10.16×10.16×15.24 cm.

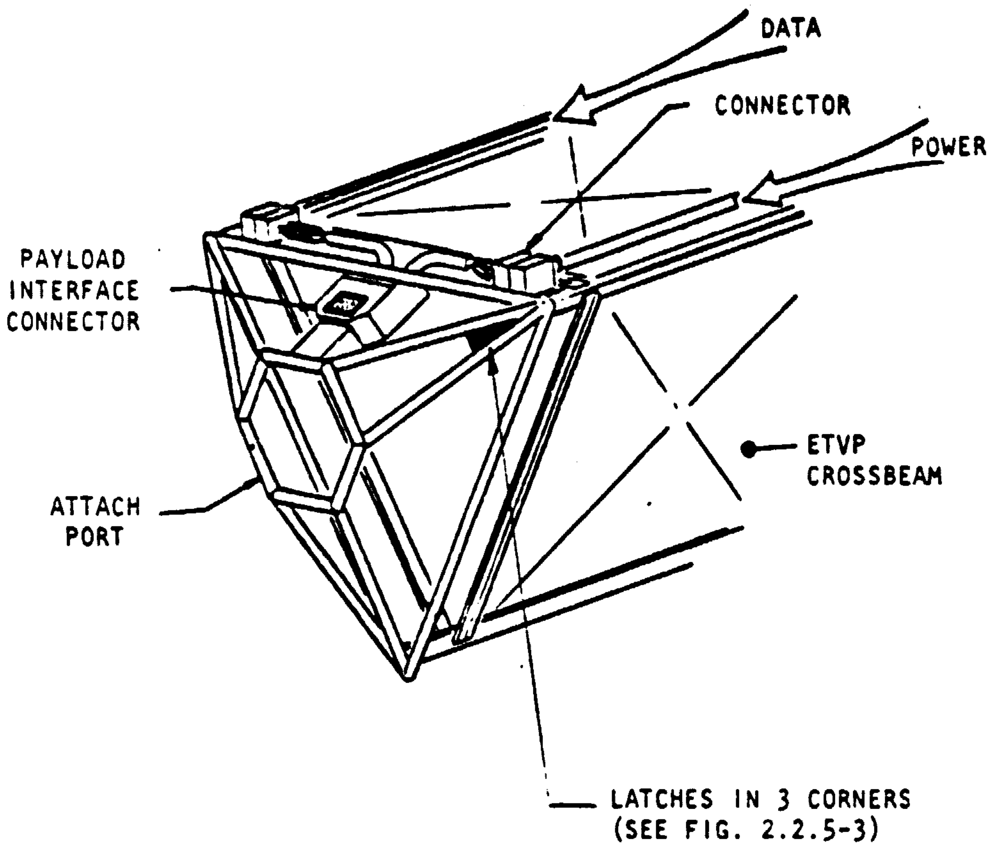


Figure 3.2.2-13. Payload Interface Connector

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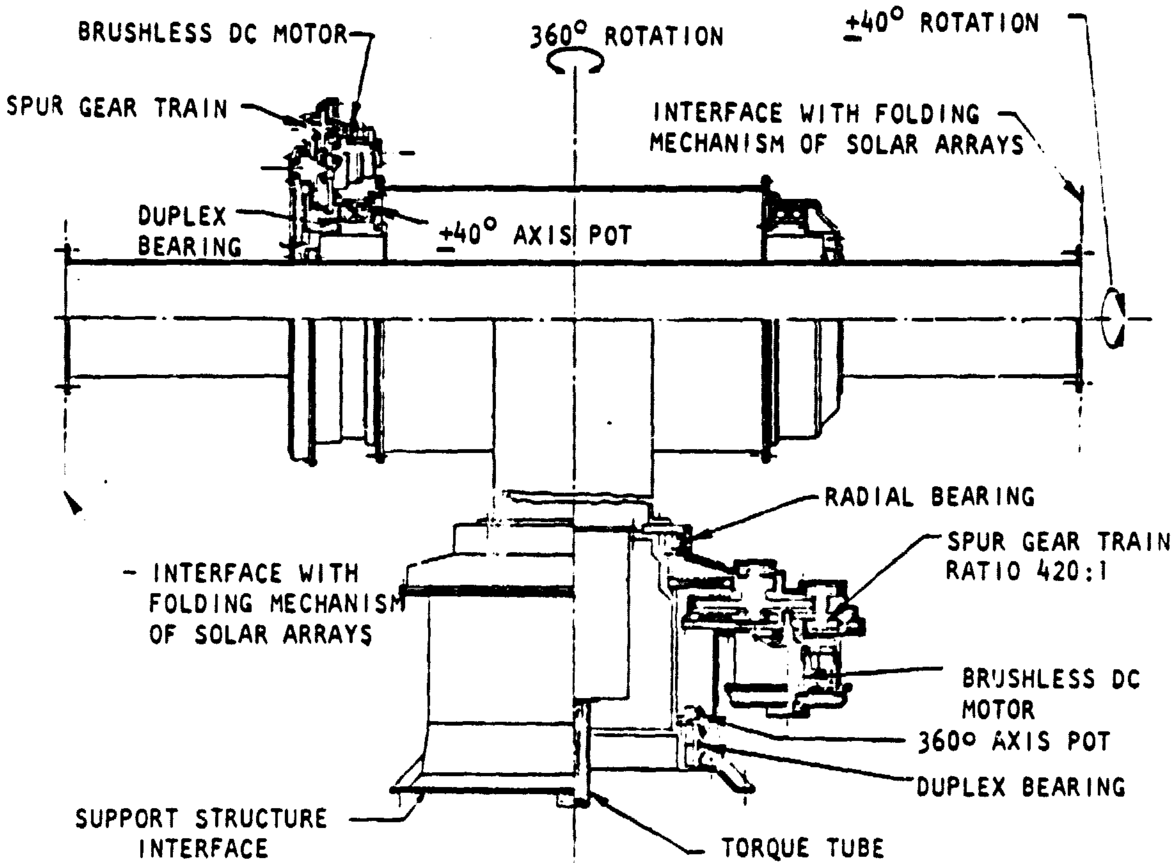


BASIC REQUIREMENTS

- CONDUCTOR PATHS:
  - MAIN POWER
  - BATTERY CHARGING POWER
  - BATTERY CHARGE SWITCHING
  - ARRAY DEPLOYMENT DRIVE
  - NOD ANGLE DRIVE
  - TEMPERATURE SENSOR
  - SUN SENSOR
  - AT LEAST 48 CONDUCTOR PATHS
- DRIVE RATES (DEG/SEC)
  - MAX = 0.1
  - LEO MODE = 0.065
  - GEO MODE = 0.0042
  - LOCK FOR ORBIT XFER
- DESIGN FOR BOOST LOADS (T/W = 0.2)
- ACCURACY, ±0.5 DEG
- ON-ORBIT C/O & SERVICING

DESIGN CRITERIA

- POWER/SIGNAL SEPARATION
- SEPARATE OPPOSITE POLARITY
- REDUNDANT FUNCTIONAL PATHS
- ACCESS FOR INSPECTION/SERVICE

Figure 3.2.2-14. Rotary Joint

3.2.3 Guidance, Navigation & Control (GN&C)

The main elements of the GN&C subsystem are summarized in Table 3.2.3-1. The system is designed to provide post-construction autonomous GN&C with supervisory commands being received from ground stations. The GN&C controls the platform attitude, points the solar arrays, and provides orbit station-keeping during both LEO and GEO operations. Additionally, the GN&C provides attitude control and thrust vector control during the orbit transfer maneuvers.

3.2.3.1 GN&C Components

The components that make up the GN&C subsystem, except for the control moment gyros (CMG's) are present-day flight hardware. The system is based on the hardware proposed for the Multi-mission Modular Spacecraft. As such, the inertial reference unit, star trackers, and linear accelerometers are precisely mounted to a rigid base which is then mounted in the system control module. This assembly, "Precision Attitude Reference," is mounted in the system control module so that the star trackers field of view is pointing away from earth and sweeps out a broad sector of the celestial sphere on each orbit.

Inertial Reference Unit

The inertial reference unit selected is the NASA "Standard High Performance Inertial Reference Unit (DRIRU II)" developed under the responsibility of the NASA Jet Propulsion Laboratory. This unit weighs 16.9 kg (37.2 lbs) maximum, occupies a volume of 16147.25 cm^3 (985.37 in^3), with dimensions of 31.2x22.9x22.6 cm (12.28x9.02x8.90 in). The unit requires 7.5 watts per channel at a nominal 28 V dc. One prime and one backup unit is required.

Spacecraft Computer

The spacecraft computer selected in the NASA "Standard Spacecraft Computer-II (NSSC-II)" developed under the responsibility of the Marshall Space Flight Center. This unit weighs 13.2 kg (29.0 lbs), occupies a volume of 10,970 cm^3 (669.72 in^3), with dimensions of 28.6x30x14 cm (11.26x11.81x 5.51 in). The nominal unit uses 170 watts, but the high reliability unit with a Fault Tolerant Memory selected in the ETVP application here uses 240 watts at a nominal 28 V dc.

Star Tracker

The star tracker selected is the NASA "Standard Mod II Fixed-Head Star Tracker (Mod II - FHST)" developed under the responsibility of the Goddard Space Flight Center. This unit weighs 9.06 kg (19.96 lbs), occupies a volume of 14,060.10 cm^3 (858 in^3) with dimensions of 17.78x19.05x45.72 cm (7.0x7.5x 18.0 in). The unit normally uses 18.0 watts at a nominal 28 V dc with the shutter requiring an additional 3 watts.

Table 3.2.3-1. Elements of the GN&C Subsystem

ITEM	NO. REQ'D	WEIGHT EA (KG)	DIMENSIONS EA (CM)	VOLUME EA (CM^3)	TOTAL WT (KG)	TOTAL VOL (CM^3)	POWER (W)
PRECISION ATTITUDE REFERENCE							
DRIRU	1	16.90	31.2x22.9x22.6	16,147	16.90	16,147	1x7.5 = 7.5
STAR TRACKER	2	9.06	17.8x19x45.7	14,060	18.12	28,120	2x21 = 42.0
LINEAR ACCELEROMETER	3	0.07	2.6 dia x 4.5	24	0.14	72	3x1 = 3.0
COURSE ALIGNMENT							
SUN SENSOR	1	0.11	5.1 dia x 5.1	103	0.11	103	None
MAGNETOMETER (Range ±500 MG)	1	0.45	5.1x10.2x7.6	393	0.45	393	1x0.3 = 0.3
BACK-UP ATTITUDE REFERENCE							
DRIRU	1	16.90	31.2x22.9x22.6	16,147	16.90	16,147	1x7.5 = 7.5
COMPUTER							
NASA STD	2	13.20	28.6x30x14	10,970	26.40	21,940	2x240 = 480
RENDEZVOUS AND DOCKING AIDS							
TELEMETRY							
LIGHTS							
REFLECTOR (LASER)							
TORQUE PRODUCERS							
CMG	3	352	1.41m dia x [illegible]				
RES	4 QUADS						
THRUST VECTOR CONTROL SYS.							

Linear Accelerometer

The linear accelerometer selected has characteristics similar to the Bendix LB PID 50 Accelerometer. This is an inertial grade Hi-g, Miniature Linear Accelerometer with hydrostatic gimbal suspension. The unit weighs 68 grams (0.15 lb), has a volume of 23.89 cm^3 (1.46 in^3) with dimensions of 2.62 cm dia x 4.45 cm long (1.03 in dia x 1.75 in long). The power requirements are less than 1 watt.

Control Moment Gyro (CMG)

The CMG's sized for the ETVP are approximately four times the size of the Skylab ATM CMG's. No technical problems are expected in the development of this size CMG. The ETVP CMG's are 2-degree-of-freedom units. Based on published guidelines, these units each weigh 352 kg (776 lb) and occupy a volume of 2.69 m^3 (94.84 ft^3) with dimensions of 1.41 m dia x 1.72 m long (4.63 ft dia x 5.64 ft long). The nominal power required is 219 watts.

Sun Sensor

The sun sensor is part of the coarse alignment system, which is used to reorient the platform in LEO in case of a platform tumble after construction is completed and the primary system assumes command. A typical sensor is the Ball Bros. two-axis C105 instrument. This unit weighs less than 0.11 kg (0.25 lb) occupies a volume of 103 cm^3 (6.28 in^3) with dimensions of 5.08 cm dia x 5.08 cm high (2 in dia x 2 in high). This unit does not consume any power.

Magnetometer

The magnetometer is part of the coarse alignment system. The unit has capabilities similar to a Develco, Inc. instrument (Model No. 104600). This unit weighs 0.45 kg (1 lb), occupies a volume of 393 cm^3 (24 in^3) with dimensions of 5.08x10.16x7.62 cm (2x4x3 in). The nominal power requirement is 0.3 watt.

3.2.3.2 Flight Modes

During a span of approximately 2-1/2 years after construction, the ETVP will be operated in LEO at altitudes of 370 km (200 nmi), 556 km (300 nmi) and 926 km (500 nmi). Refurbishment, test, and operation of subsystems and operational checkout of experiments will be conducted at these altitudes.

The operational checkout will be conducted in the same attitude as the experiment will experience in GEO; that is, the Y-axis or long axis of the platform perpendicular to the orbit plane (Y-POP). The base of the triangular truss +Z axis will be pointed toward the earth, the X-axis will complete the orthogonal system.

The transfer between the various LEO altitudes and GEO will be conducted in a orbit transfer attitude. The attitude for this operational mode is Y-axis in the direction of the applied thrust.

In GEO, the operational attitude is Y-POP and Z toward the earth. Since the principal axis of inertia is close to the geometric axis, the platform will be oriented so that the Y principal axis will be POP.

3.2.3.3 Pointing Accuracy

The accuracy at which the experiments can be aligned with their targets on the ground is a function of the accuracy of each of the components in the pointing loop. The accuracy breakdown is given in Table 3.2.3-2. The components chosen for the system have errors less than those allowed in this table.

Table 3.2.3-2. Antenna Pointing Accuracy Error Budget

	Budget (deg)
Attitude determination	0.050
Control dynamics	0.100
Thermal deformation of structure and feed horn boom	0.08.
Thermal deformation of antenna reflector	0.06
Manufacturing and assembly tolerance	0.210
RSS total (one beam width)	0.258

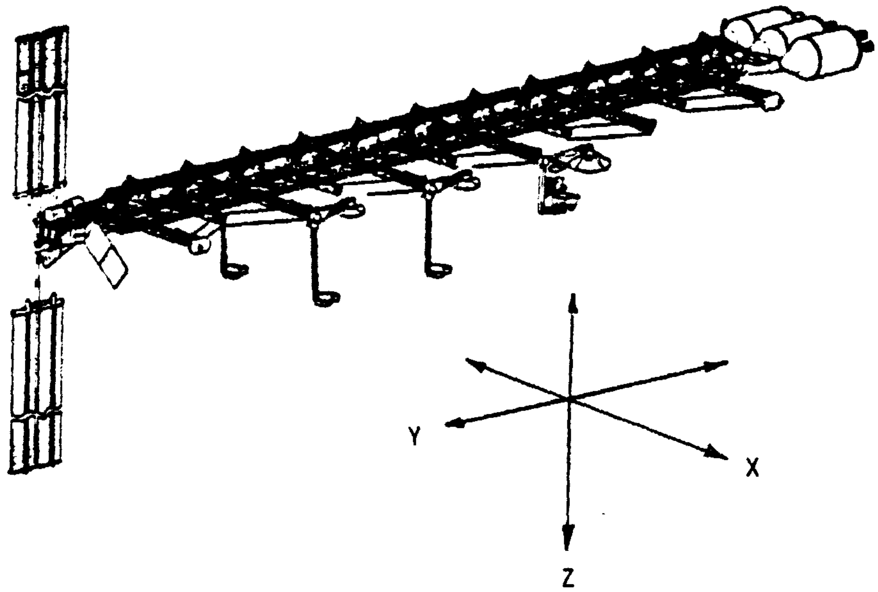
3.2.3.4 Disturbance Torques & CMG Sizing

In order to determine the size and number of CMG's required for the ETVP, an analysis of the major disturbance torques acting on the platform was made. The three major torques acting on a satellite in earth orbit are gravity gradient, aerodynamics, and solar pressure. The aerodynamic torque is significant only in LEO, while gravity gradient and solar pressure (differing in relative magnitudes) are significant in both LEO and GEO.

A digital simulation program, Momentum Accumulation and Dumping (MAD), was utilized to determine the torques and the resultant momentum acting on the platform. The mass properties of the platform modeled in the simulation are given in Table 3.2.3-3. These mass properties were derived at the onset of the study and used throughout to eliminate the need for iterative solutions of the same problem. The same mass properties were used in the operational configuration in both LEO and GEO.

Table 3.2.3-4 gives the orientation of the platform that was simulated to determine the size of the control system actuators. Two runs were made with the principal axes of the platform aligned to the orbiting local vertical reference system.

Table 3.2.3-3 ETVP Operational Configuration Mass Properties



PARAMETER	VALUE
MASS	39,210 KG
I_{xx}	108.951E6 KG-M^2
I_{yy}	3.963E6 KG-M^2
I_{zz}	109.595E6 KG-M^2
I_{xy}	0.0
I_{xz}	0.008722 KG-M^2
I_{yz} m	-3.861E6 KG-M^2

Table 3.2.3-4. ETVP Momentum Requirements Per Orbit

ORBIT PARAMETERS		MOMENTUM (NMS)					
		SECULAR (PER ORBIT)			CYCLIC		
ALT (KM)	ATTITUDE	X-AXIS	Y-AXIS	Z-AXIS	X-AXIS	Y-AXIS	Z-AXIS
35863	Y-POP/LV PA	22603	550	22603	0	325	0
370	Y-POP/LV PA	866	7125	866	19136	1000	20786
556	Y-POP/LV PA	902	550	902	6000	500	6000
35863	Y-POP/LV 10°	21000	7667	22500	0	0	1200
370	X-POP/LV PA	112154	114	1573	0	100	125
370	X-POP/LV 0°	96120	0	0	0	6	6
556	X-POP/IH 0°	1572	0	0	87409	0	0
556	Y-POP/IH 0°	44473	0	0	5000	1291	3720

PA PRINCIPAL AXIS OF PLATFORM ALIGNED WITH LOCAL VERTICAL
0° 10° DIFFERENCE BETWEEN PLATFORM AXIS AND LOCAL VERTICAL
IH INERTIAL HOLD

Figure 3.2.3-1 shows the total torques, solar and gravity gradient, acting on the ETVP in GEO over a 4 day period (4 orbits). The platform itself is maintaining an earth pointing reference, while the solar panels are rotating relative to the platform to maintain a solar inertial orientation.

The integration of the body axis torques relative to an inertial set of axes is presented in Figure 3.2.3-2. The secular X-axis torque is due to solar pressure acting on the asymmetric solar panel configuration.

The inertial momentum is transformed relative to the platform axis system in Figure 3.2.3-3. The secular inertial X-axis momentum transforms into the divergent oscillatory momentum about the X and Z body axes. These two axes are 90 degrees out of phase so that the actual momentum to be stored is determined by one axis alone (i.e., the combined momentum is never larger than the peak value for either axis).

When the ETVP is operational in LEO, the torques acting upon it are gravity gradient, aerodynamics and solar pressure. For the conventional satellite in LEO the solar pressure torque is normally neglected, but due to the large asymmetric solar panel area, 592 m^2 (6372 ft^2), and a lever arm of 63.29 m (207.64 ft), the magnitude of the solar pressure torque, 0.25 nmi (0.18 ft-lb_f), is not negligible relative to the gravity gradient or aerodynamic torques and cannot be ignored.

The momentum accumulated for the various attitudes of the ETVP considered is given in Table 3.2.3-4. In GEO the momentum accumulated is mainly secular. This momentum must be countered by the expenditure of RCS propellant. The use of CMG's for this mode of operation is to provide a stable earth-pointing base for the experiments.

The CMG's are sized to maintain control for one orbit in 370 km LEO. For this configuration, the CMG's must swing their momentum vectors approximately $\pm$21,000 Nms in each orbit to absorb the cyclic momentum and the small secular momentum. In GEO, by effectively positioning the CMG momentum vectors at one end of their travel, two orbits can be controlled before desaturating with the RCS. There is no difference in RCS propellant consumption, only in the length of time of continuous operation.

3.2.3.5 CMG Desaturation

The CMG's must be desaturated when they cannot absorb the predicted momentum to be accumulated during the next experiment period. In 370 km LEO the secular momentum is small, but it does drive the CMG momentum vector unidirectionally, so that during the cyclic swing of the next orbit, the limit of momentum absorption will be reached and control will be lost.

To prevent loss of control, CMG desaturation in LEO is planned to be every orbit at the ascending or descending mode. Since orbit stationkeeping takes place at this point, some saving in RCS propellant will occur by the combination of these two maneuvers.

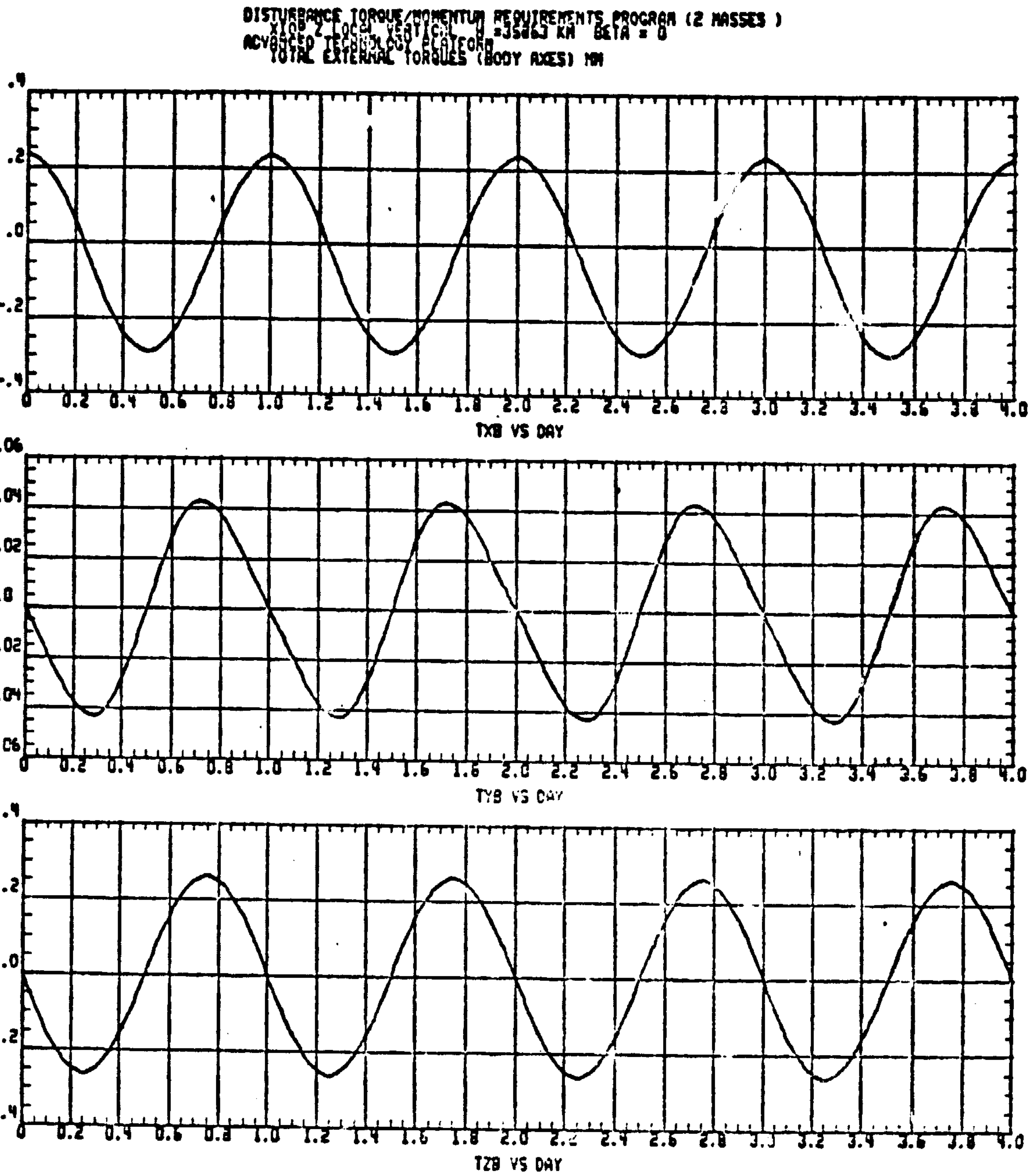


Figure 3.2.3-1. GEO Total Torque - Body Axis

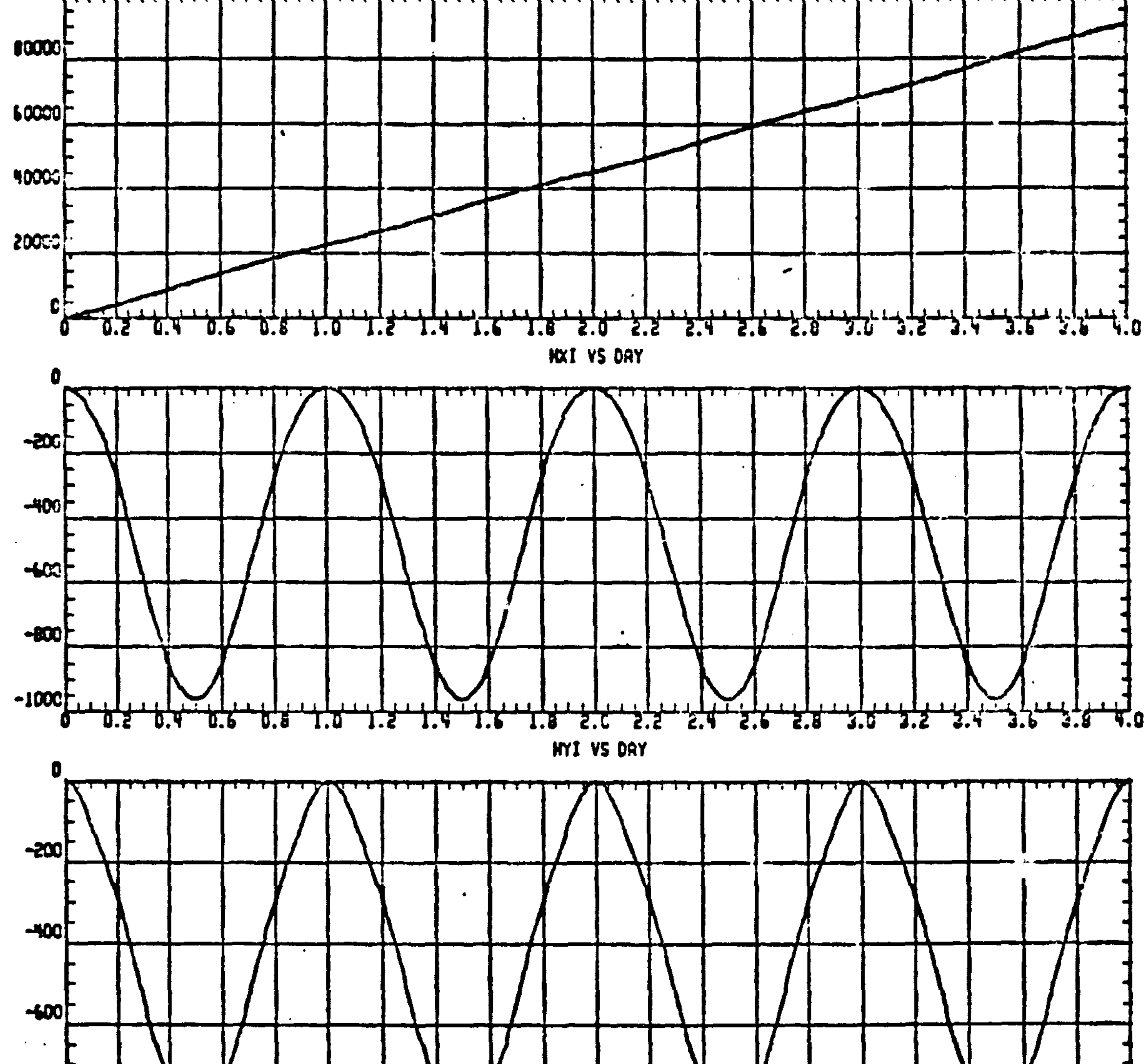


Figure 3.2.3-2. GEO Accumulated Momentum, Inertial Axis

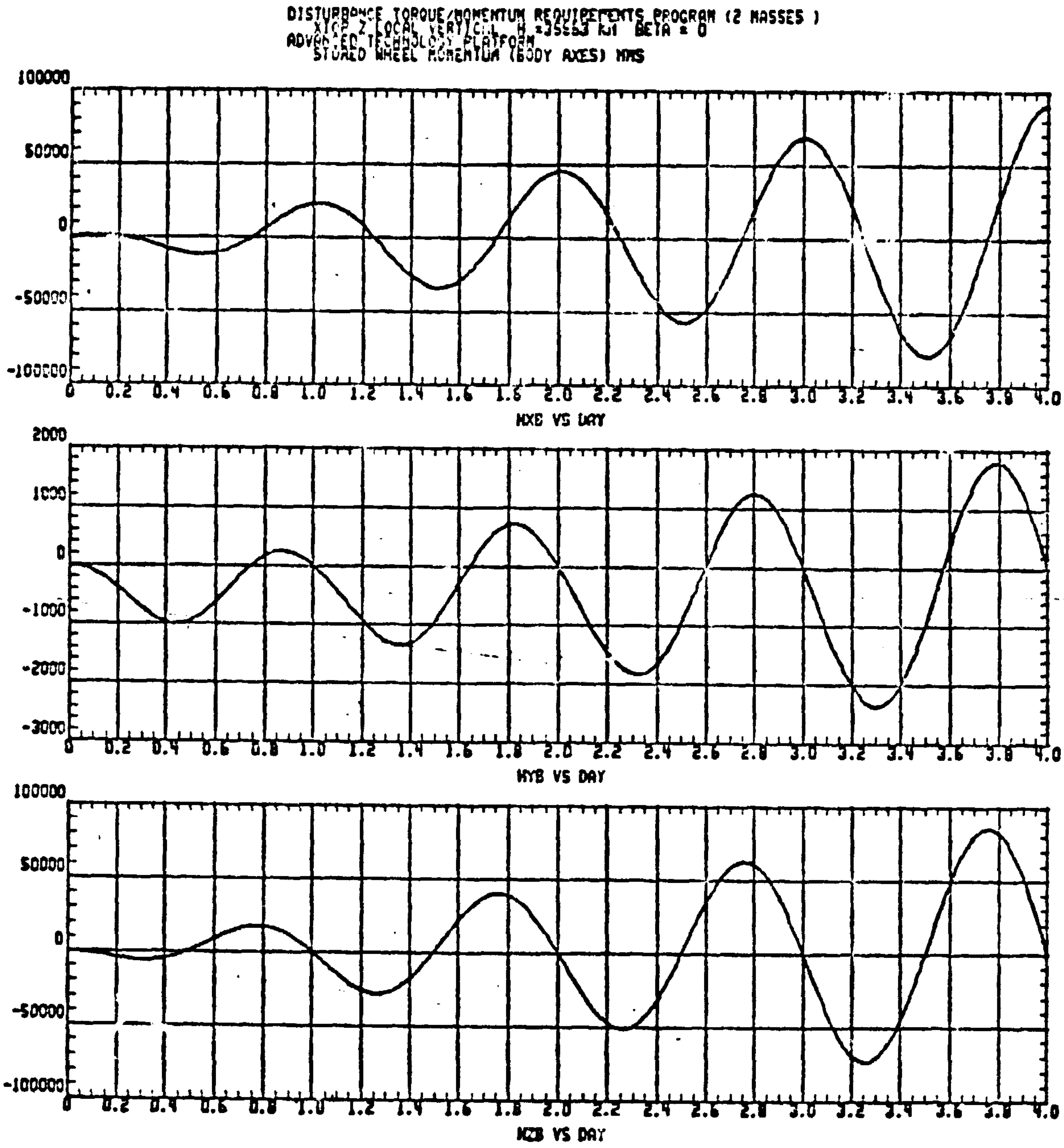


Figure 3.2.3-3. GEO Accumulated Momentum - Body Axis

The same procedure of CMG desaturation and stationkeeping is anticipated in GEO. In GEO, CMG desaturation can occur every other orbit by allowing the momentum vector to go unidirectionally from stop to stop. Whenever stationkeeping occurs, CMG desaturation should take place also in order to take advantage of the propellant savings.

3.2.3.6 Attitude Maneuvers

Attitude maneuvers can be made with either the RCS or the CMG. Use of the RCS is preferable since CMG maneuvers are relatively slow 0.00866 deg/sec. The number of maneuvers to be made is probably small and will not add greatly to the propellant budget. Table 3.2.3-5 shows the propellant required to maneuver the ETVP operational configuration at various rates.

3.2.3.7 Rendezvous Aids

In LEO, the rendezvous and berthing aids are on the construction fixture and consist of a transponder for rendezvous and lights for target illumination during the final 305 meter (1000 ft) of closure and grappling. In GEO the rendezvous and docking for maintenance and resupply is done by teleoperator command of a pilotless vehicle.

The aids in GEO for rendezvous are the same as in LEO except that the construction fixture is not attached and docking has to take place at several locations. At each of the locations that the maintenance vehicle must dock, illumination must be provided either by the platform or the maintenance vehicle. Additionally, a reflective target for ladar/radar range/range rate data must be provided for aid in docking. Visual aids are required to assist in maintaining the correct angular orientation. As in both the Apollo and Shuttle programs, man-in-the-loop real time simulations will determine the best type of visual aid to be used.

3.2.3.8 Terminal Closure

An investigation of the terminal closure phase of an orbiter mission during the construction scenario was conducted in order to complete the overall mission timelines. The terminal trajectory to be used has the complication that the RCS plume impingement on the target vehicle must be minimized. Plume impingement on the platform can cause contamination from the products of combustion, localized thermal stresses on the structure, and rotational motion.

The terminal closure, from 305 to 915 meters (1000 to 3000 ft) to stationkeeping distance, of the baseline orbiter with a plume impingement sensitive payload was investigated by the Crew Training and Procedures Division at NASA/JSC. The results are given in NASA Report JSC-12776, "PDRS-III Shuttle Engineering Simulator Post-Simulation Report," dated 7 November 1977.

The simulation was conducted with a specific payload defined, but the conclusions state that no problems are anticipated except in extreme mass/geometry cases. Whether the ETVP falls in this category cannot be stated in certainty at this time.

Table 3.2.3-5. Propellant Required to Operate at Various Rates

MANEUVER	KG OF PROPELLANT, I_{SP} = 280 SEC, TO GIVE A RATE CHANGE OF:		
	0.1°/SEC	0.05°/SEC	0.03°/SEC
$\dot{\theta}$	0.227	0.114	0.068
$\dot{\phi}$	1.077	0.539	0.323
$\dot{\psi}$	1.268	0.634	0.382
3-AXIS	2.573	1.286	0.773
2-AXIS	1.495	0.750	0.455

The simulation study investigated the ability of the astronaut to maneuver the baseline orbiter to within RMS reach distance of a gravity-gradient stabilized payload and to stationkeep. A combination of four approach paths and three braking techniques was evaluated. Plume impingement or overpressure, propellant consumed, and time were variables considered.

The four approach paths were:

1. Direct - a continuation of a normal intercept trajectory from below and infront of target, Figure 3.2.3-4.

2. $\overline{V}$ - along the velocity vector, Figure 3.2.3-5.

3. $\overline{R}$ - along the radius vector, Figures 3.2.3-6 and 3.2.3-7.

4. $\overline{H}$ - along the momentum vector, out-of-plane approach.

The three braking techniques were:

1. +Z - PRCS
2. ±X - PRCS
3. ∓X - PRCS (Tail first)

Of the above, the combination of the $\overline{R}$ approach path and the ±X - PRCS was determined to be superior to all other approach techniques from the standpoint of minimal plume impingement and operational simplicity.

The $\overline{H}$/±X had the minimum plume effect on the payload but had the largest propellant usage. More important, however, the $\overline{H}$ approach technique was determined to be only conditionally feasible because of its relative complexity and significant probability of orbiter/payload collision.

A typical $\overline{R}$ approach from 305 m (1000 ft) below the target is reproduced in Figure 3.2.3-8. The trajectory goes from 305 meters (1000 ft) to about 11 meters (35 ft) in 30 minutes. Figures 3.2.3-9, -10, and -11 show 5 minutes of stationkeeping at a range of approximately 12 m (40 ft).

For the final two minutes, the relative velocities between the orbiter and payload are less than 0.009 m/s (0.03 ft/sec) along the X or velocity vector, 0.01 m/s (0.04 ft/sec) in the Y body (out of plane) direction, and 0.009 m/s (0.03 ft/sec) along the radius vector or Z body direction. The RSS of these is 0.02 m/s (0.06 ft/sec), which is within the ±0.03 m/s (0.1 ft/sec) limit set forth in Part I of the study.

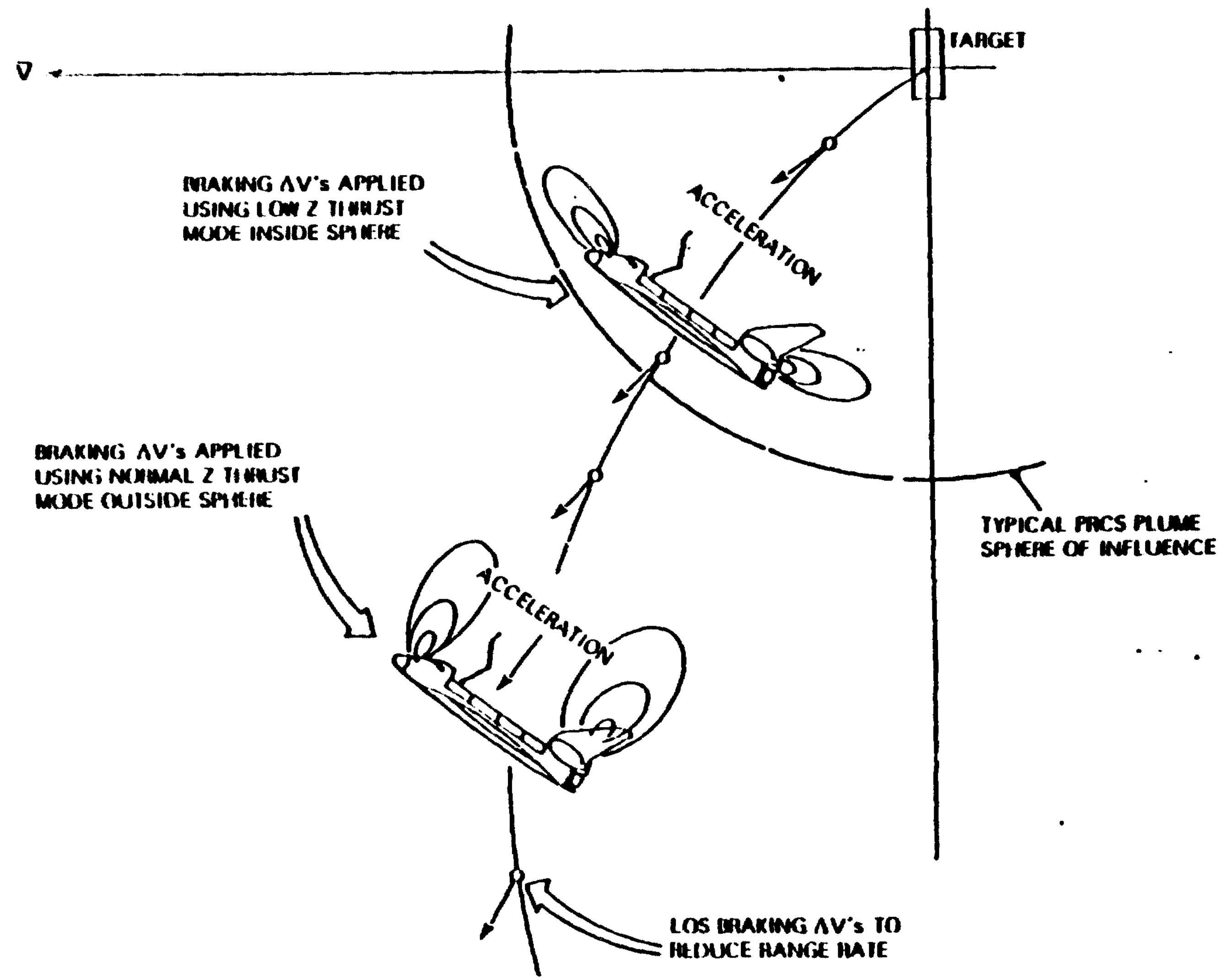


Figure 3.2.3-4. Direct Approach to Close-In Stationkeeping

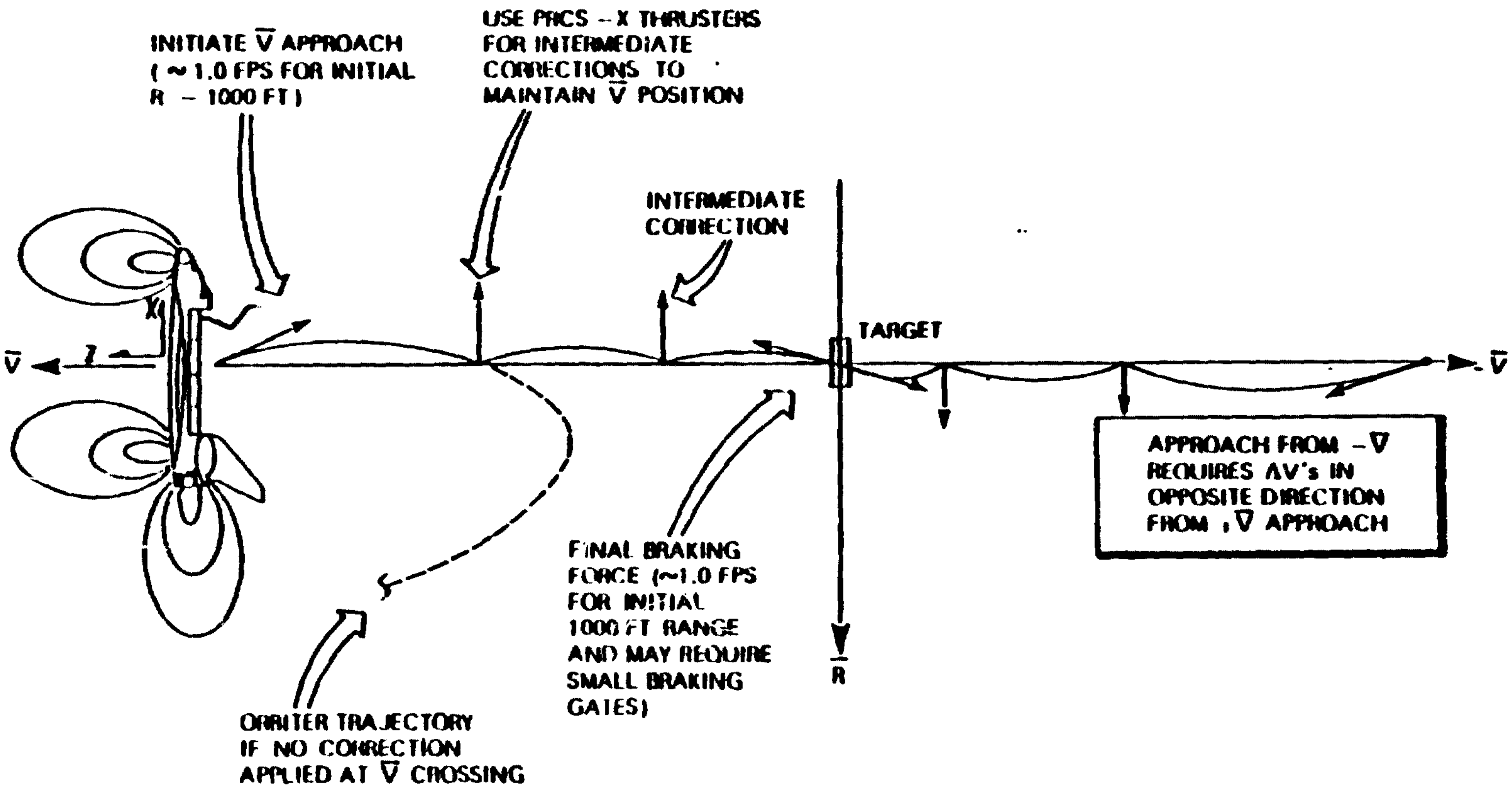


Figure 3.2.3-5. $\overline{V}$ Approach Technique

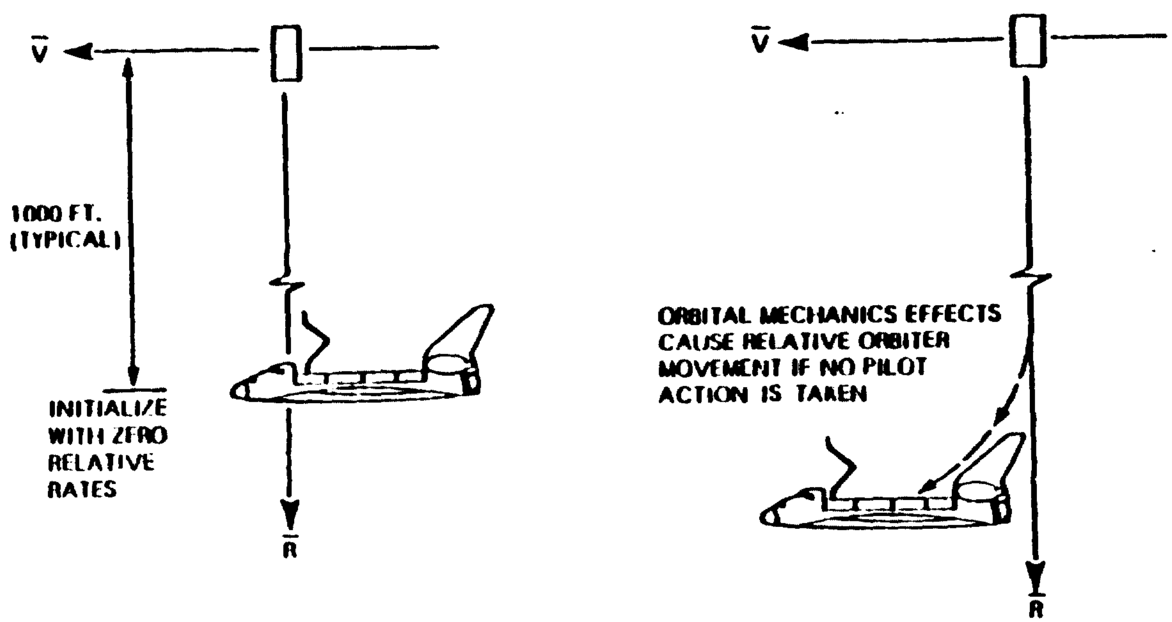


Figure 3.2.3-6. $\bar{R}$ Approach Uses Orbital Mechanics Forces for Braking

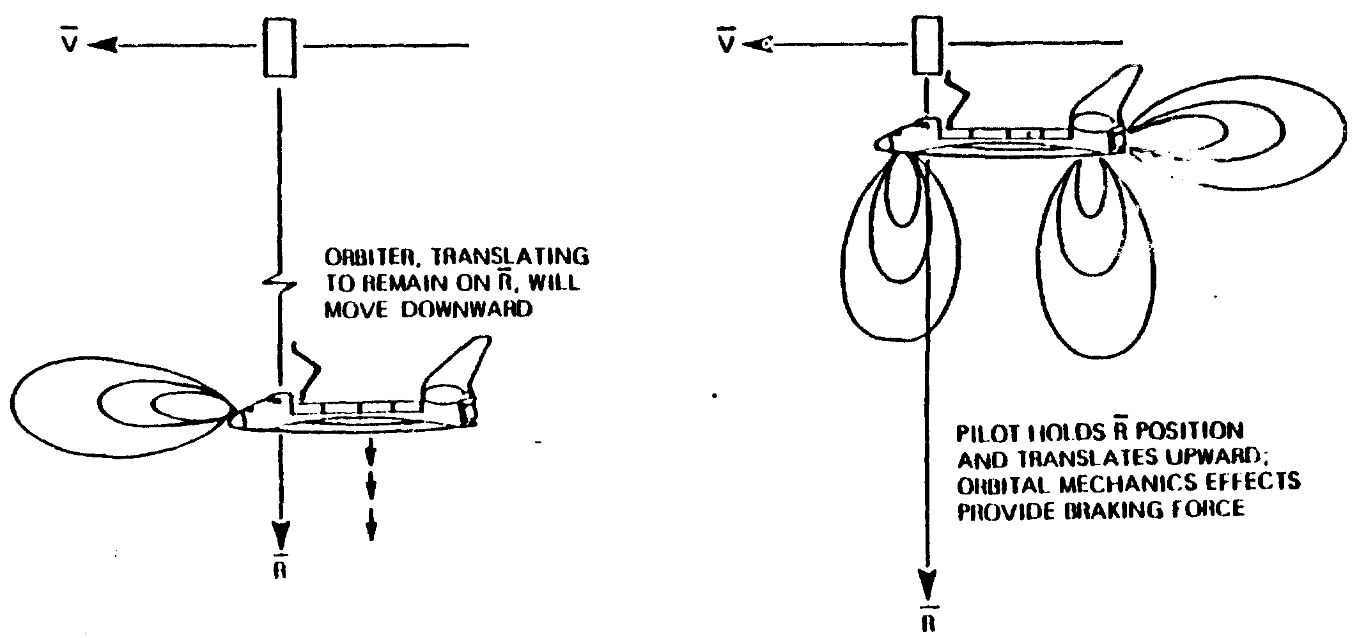


Figure 3.2.3-7. $\bar{R}$ Approach Uses Orbital Mechanics Forces for Braking

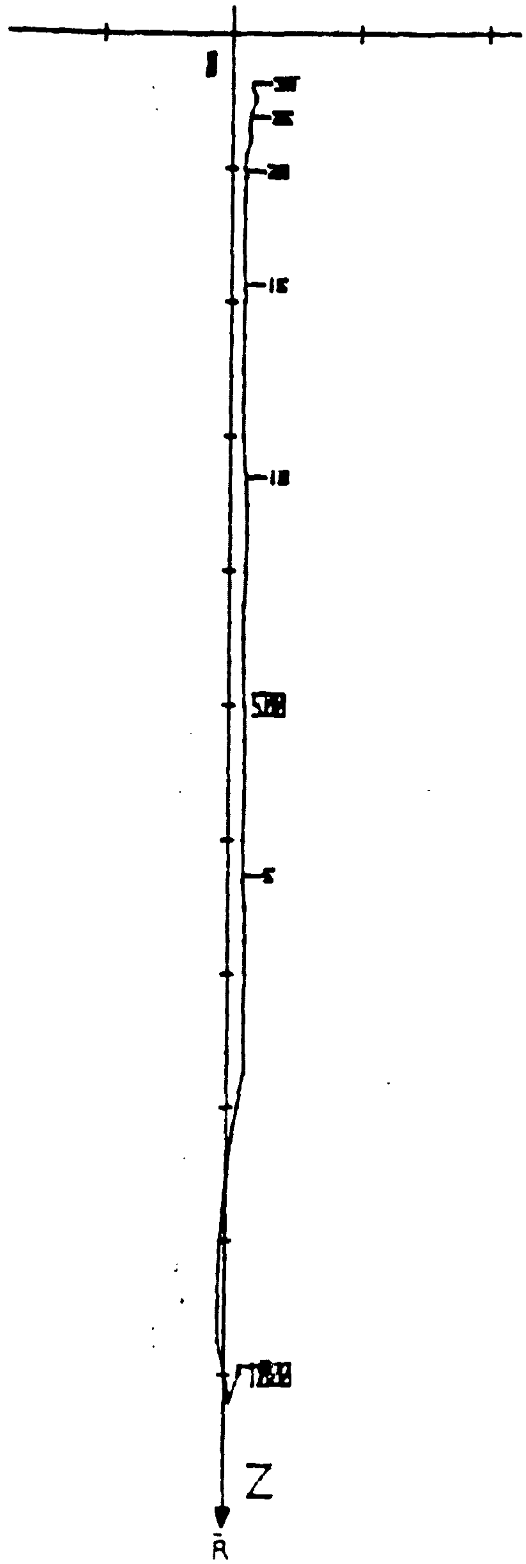


Figure 3.2.3-8. $\overline{R}$ Approach From 1000 Ft. Below Target

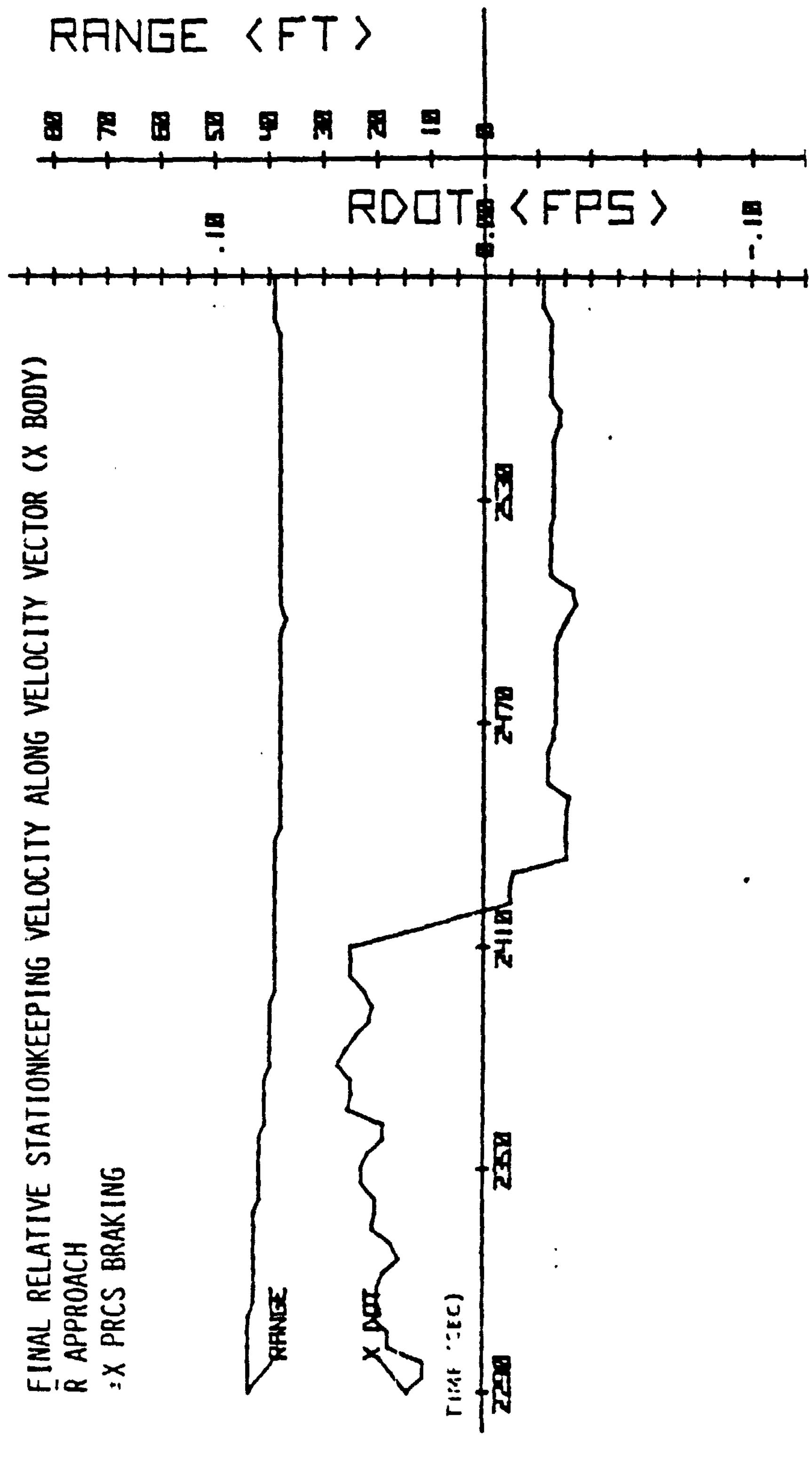
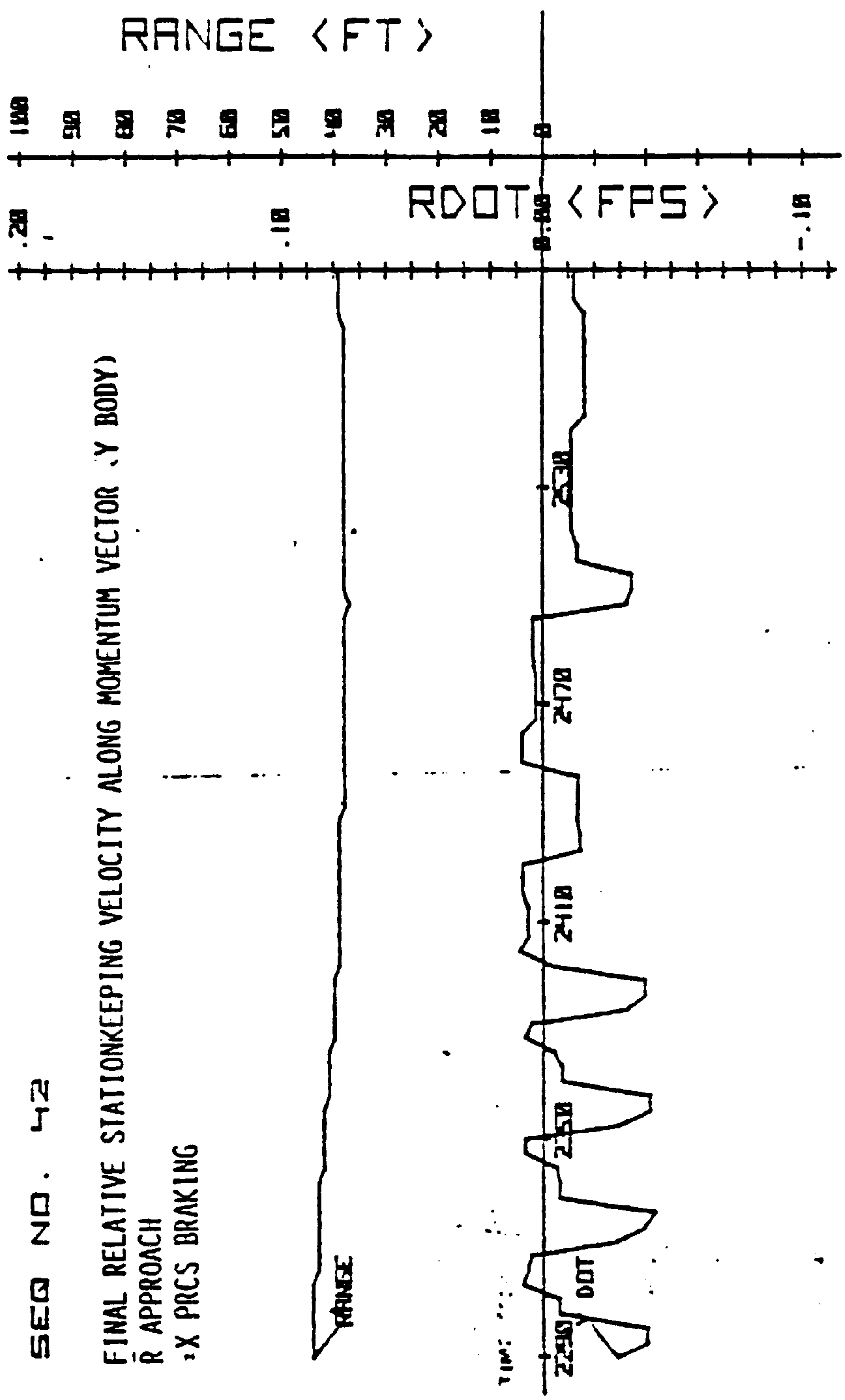


Figure 3.2.3-9. R̄ Approach, Final Relative Velocity (X-Body)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80
70
60
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40
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RDOT <FPS>
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SEQ NO. 42
FINAL RELATIVE STATIONKEEPING VELOCITY ALONG MOMENTUM VECTOR (Y BODY)
R APPROACH
+X PRCS BRAKING
RANGE
RDOT
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2290
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2470
2530

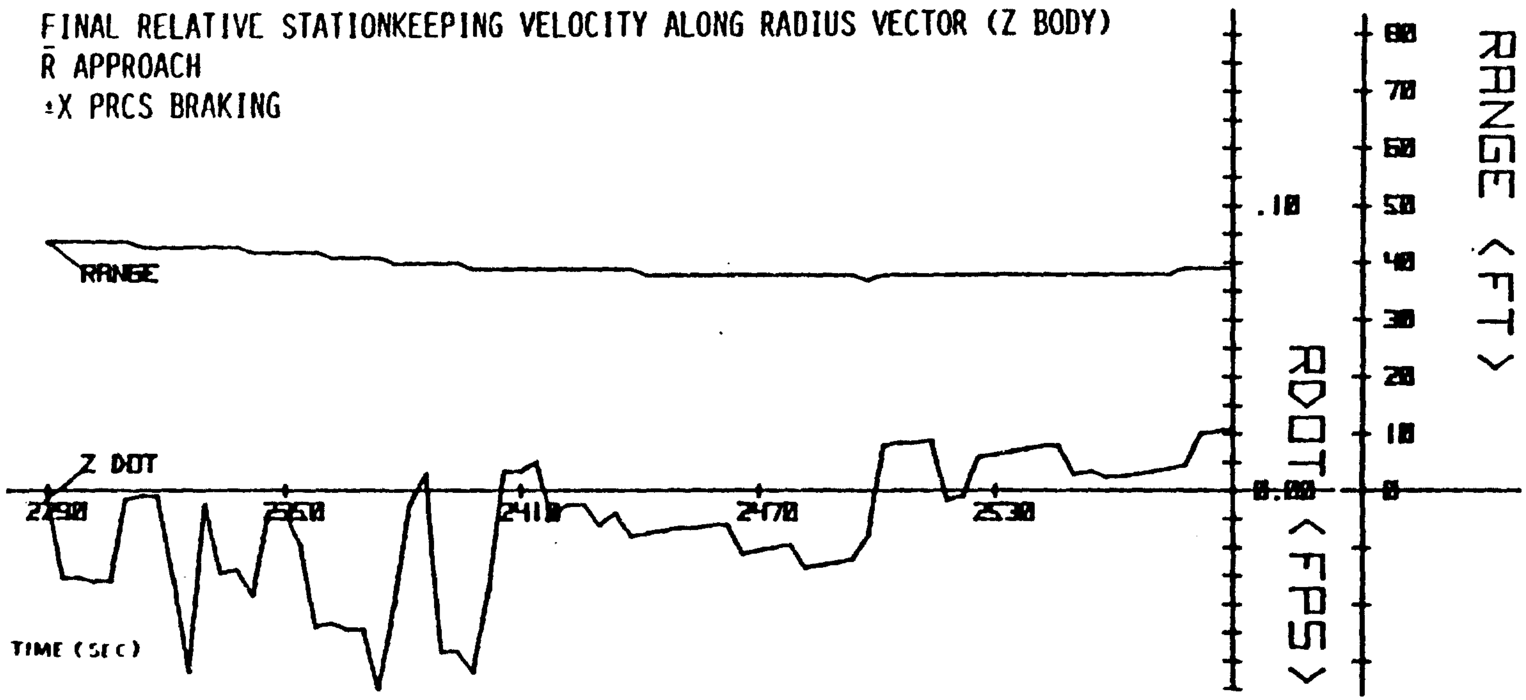


Figure 3.2.3-11. R̄ Approach, Final Relative Velocity (Z-Body)

3.2.4 Thermal Control

3.2.4.1 Summary

The main elements of the thermal control subsystem (TCS) are summarized in Table 3.2.4-1. Temperatures of all components are maintained within allowable levels during LEO, orbit transfer, and GEO operations. Control is provided by an active, pumped fluid loop system which utilizes Freon-21 as the coolant. Waste heat is rejected by a heat pipe radiator selected because of lifetime considerations, reliability, and meteoroid damage tolerance. The required planform area of the radiator (half the radiating area) is 37.2 m^2. Trades and sizing analyses for the individual subsystem elements are discussed below.

3.2.4.2 Radiator Requirements

Radiator requirements were derived from a consideration of heat loads, design temperature extremes and cost/weight/reliability/lifetime considerations. The heat loads include both component dissipation levels and environmental heating. The latter contribution to radiator sink temperature is shown in Figure 3.2.4-1 for LEO operation. Due to the platform attitude and radiator orientation during the sunlit portion of the orbit, solar loading is equal to S cos $(90-\beta)$, where β is the sun beta angle (angle between sun line and the orbit plane) and S is the solar constant. For an orbital inclination of 28.5°, the maximum beta angle (equivalent to maximum solar loading) is 52°. Values of the figure were based on a solar constant, S, of 1393 W/m^2, albedo = 0.35, and earth emission magnitude of 220 W/m^2. At geosynchronous altitude, earth emission and albedo contributions are negligible (<1 W/m^2) and solar heating is again governed by the equation S cos $(90-\beta)$. For an equatorial orbit β_{max} is 23.5° and the maximum sun loading is 555 W/m^2.

Environmental loads are minimal for this fixed radiator configuration because the panels are parallel to the orbit plane for the baseline Y-POP, Z LV platform attitude. If future missions dictate other orientations radiator area would be substantially increased. Subsystem component dissipation loads used to determine radiator requirements are shown in Table 3.2.4-2.

Although power requirements exceed those listed, some components, e.g., CMG's, radiate directly to the environment. Also, there will be some battery load during the sunlit portion of the orbit but this was neglected in the current analysis.

Initial evaluation of component heat rejection requirements indicated the undesirability of individual, localized, passive component radiators due to design complexity and the difficulty of obtaining favorable orientations for all the radiator panels. Centralized radiators could use either a pumped fluid-loop system or heat pipes to transport waste heat to selected radiator panels. The heat pipe option was discarded because of design complexity, cost, and uncertainty; required long length (~3 m), variable conductance, diode heat pipes (possibly with multiple 90° bends) are beyond current or anticipated near-term state of the art. Transport to the radiator panels can be accomplished using the fluid loop directly or by heat pipes. The heat pipe system was selected because of the large weight penalty associated with protecting a fluid loop system from micrometeoroid damage

Table 3.2.4-1. Thermal Control Subsystem Component Listing

COMPONENT	QTY	TOTAL WEIGHT kg	DIMENSIONS cm	POWER W
(Radiator)	(2)	(294)	304.8x609.6	-
Heat pipes	156	105.5	1.27 OD×182.88	-
Skin	-	51.9	609.6×304.8×.061	-
Coldplate HX	-	29.8	25.4×304.8	-
Core	-	43.1	-	-
Silver Teflon coating	-	29.8	0.0127 thick	-
Flex lines, fluid, adhesive	-	19.9	-	-
Flow bypass controller	1	14.0	53.34×76.2×15.24	20 (nom.)
(Fluid Loop)		(234.5)		
Pump package	4	82	72.26×32.77×36.07	250 (each)
Heat exchanger	1	6.1	34.29×10.85×18.75	-
Accumulator fluid	-	136.4	-	-
Piping, fluid, valves	-	10	1.905 line OD	
(Coldplates)	(76)	(196.3)		
Batteries	49	154.8	35.56×71.12	
Cont./Reg./Dist.	8	9.5	22.86×40.64	
	1	1.0	20.32×40.64	
	1	1.3	25.4×40.64	
	2	4.2	25.4×66.04	
GN&C	1	2.6	45.72×45.72	
	2	6.2	40.64×60.96	
TT&C	1	2.3	38.10×48.26	
	2	3.5	33.02×40.64	
	2	3.6	33.02×43.18	
	2	1.4	17.78×27.94	
	1	1.2	27.94×33.02	
	2	2.5	27.94×35.56	
	1	1.2	22.86×40.64	
	1	1.0	22.86×33.02	
TOTAL TCS WEIGHT		(724.8)		

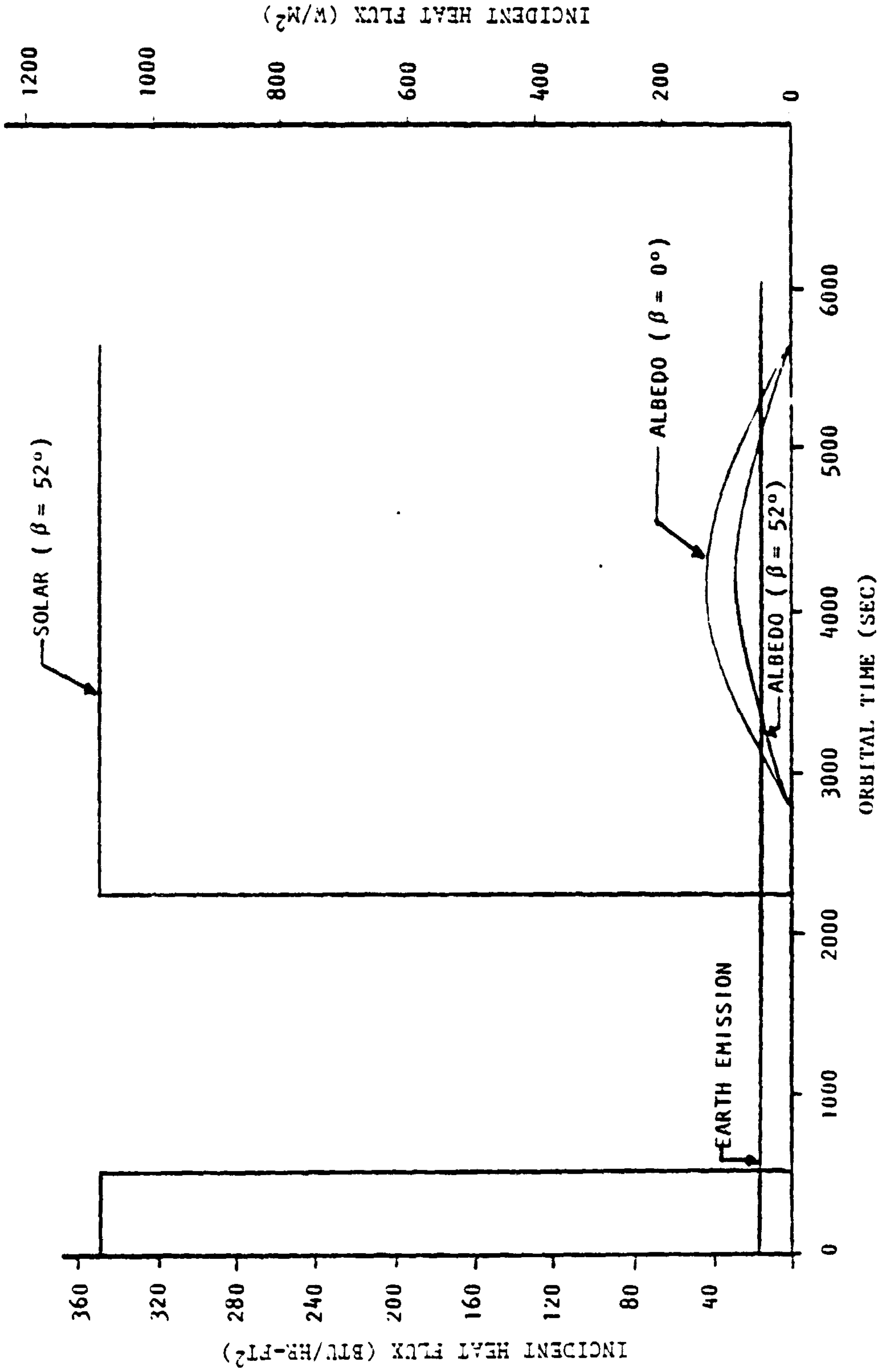


Figure 3.2.4-1. LEO Radiator Heating Rates

during the radiator operational lifetime. The selected hybrid system (pumped fluid loop/heat pipe radiator) is significantly lighter than the active system; for design reliability of 0.9, weight saving is about 50% for a 20-year lifetime.

Table 3.2.4-2. Subsystem Dissipation Requirements

Subsystem	Power Level (watts)
GN&C	300
EPS	
Batteries	14,000 (GEO eclipse) 7,500 (LEO eclipse)
Cont./Reg./Dist.	950
TT&C	250
TCS	500

3.2.4.3 Micrometeoroid Protection

Heat pipe redundancy required to account for loss due to micrometeoroid was found to be about 33%. The micrometeoroid analysis was based on a flux model proposed by NASA (Ref. 1). The incident meteoroid flux distribution varies with particle size according to the relation

$$\log N_t = -14.37 - 1.213 \log m + \text{Log Ge} \qquad (1)$$

where

N_t is the average meteoroid flux (particles/m^2/sec) of mass m or greater

m is the particle mass (g)

r is the distance from the earth's center in earth radii

The term Ge is the defocusing factor to account for focusing of the particle flux due to the earth's gravitation and is given as:

$$Ge = 0.568 + \frac{0.432}{r} \qquad (2)$$

Equation (1) applies to particles in the range 10^{-6} to 1 g. These are considered the most likely to damage the radiator since smaller particles do not have sufficient kinetic energy to penetrate an appreciable wall thickness and larger particles are assumed to not be present in enough quantity to have an appreciable encounter probability; however, for the long mission lifetimes (20 years) associated with advanced platforms, improved flux models for larger particles could be required.

Empirical relationships have been developed by Rockwell which determine material wall thickness required to prevent puncture by a meteoroid particle of mass m (Ref. 2). The required wall thickness to prevent dimpling, spalling, and perforation is:

$$t_w = \frac{1.75\ f_f \rho_p^{0.133} V_p^{2/3} m^{0.367}}{H_t^{1/4} \rho_t^{1/6}} \tag{3}$$

where

t_w is the required wall thickness (cm)

f_f is the spall factor (>1)

ρ_p is the particle density (g/cm^2)

V_p is the particle velocity (km/sec)

m is the particle mass (g)

H_t is the Brinell hardness of the tube wall

ρ_t is the density of the wall (g/cm^3)

The probability of at least one impact by a particle of size m is given by the Poisson distribution,

$$P = 1 - e^{-N_t A \theta} \tag{4}$$

where

P is the impact probability of an individual element

N_t is the average meteoroid flux

A is the projected exposed area (m^2)

θ is the total exposure time (sec)

Equations (1), (2), (3), and (4) were used to define the required heat pipe redundancy and component shielding.

3.2.4.4 Sizing

The hybrid heat pipe radiator system was sized to reject system heat loads and other platform heat sources. The primary external thermal contributor is the solar array. In low earth orbit the solar array can reach 70°C. For a view factor of 0.05 this is equivalent to a thermal input to the radiator of

22 W/m^2. In geosynchronous orbit the solar array will be significantly cooler, maximum operating temperature will be about 42°C, and thermal input to the radiator will be under 16 W/m^2.

Radiator sizing for different orbital environments is shown in Figure 3.2.4-2. Property degradation estimates were based upon published experimental property measurements (Ref. 3) and Global Positioning Satellite (GPS) on-orbit flight data. Sizing is governed by the dissipation extremes during the eclipse portion of geosynchronous orbit operation. The required planform area of the radiator (half the radiating area) is 37.2 m^2 (400 ft^2). Although not shown, orbit transfer would present a lesser requirement. This conclusion holds even if the radiator experiences direct solar loading.

The heat pipe radiator schematic is shown in Figure 3.2.4-3. The heat pipes are located on 1.21-cm (3.08-in.) centers. A total of 156 heat pipes are required. The container is 6061-T6 aluminum, and the basic design utilizes a trapezoidal wick and ammonia working fluid.

3.2.4.5 Deployment

The radiator system utilizes a spring load deployment mechanism. The folded radiator panels are operated by a compressed nitrogen gas system with redundant components. Individual segments of the radiator are 3.048×3.048 m (10×10 ft).

3.2.4.6 Fluid Loop

The fluid loop schematic is shown in Figure 3.2.4-4. The basic plumbing/piping distribution for the platform consists of two Freon-21 loops. Two pump packages (four pumps) and accumulators are included in each loop with all connections in parallel so that only one pump operates at any time. The basic pump package configuration is currently used on orbiter (Specification SVHS6426). Operating lifetimes are estimated to be at least five years for these orbiter components, although this capability has not yet been demonstrated. Improvements in seal and lubricant technology will, hopefully, extend these limits. Waste heat from the pump motors is dissipated into the circulating fluid and eventually rejected by the radiators.

Battery heat dissipation drives loop flow rate and radiator outlet temperature. Radiator outlet temperature is controlled by the orbiter flow proportioning valve which permits a dual position setting. During eclipse periods, the radiator outlet is set at -1.1°C (30°F) to accommodate the high battery loads and in sunlight the temperature setting is 4.4°C (40°F). Fluid-loop flow rate is 2454.5 kg/hr (5400 lb/hr) to maintain the outlet temperature of the battery coldplates under 20°C. This requires a Freon-21 flow rate of 1227.3 kg/hr (2700 lb/hr) in each loop, which is well within the capacity of the current pump design. Consequently, the proposed concept would allow three pumps (and one accumulator) to fail in each loop over the operational lifetime. If orbiter experience indicates that this allotment is inadequate, then an additional pump package in each loop may be required. The design assumes that pump system lubricants would survive over the system lifetime and this requirement might require advanced lubricant concepts.

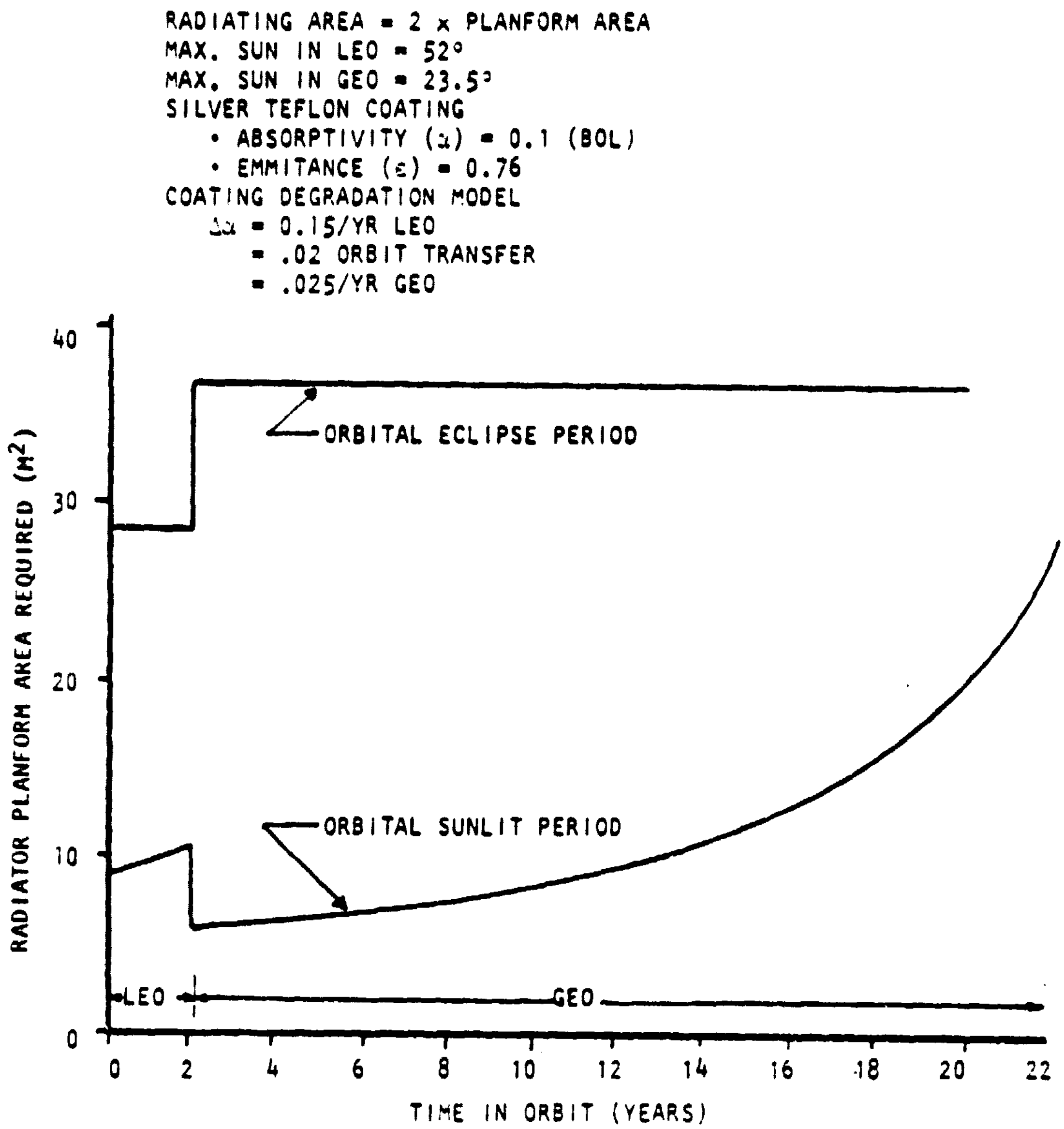


Figure 3.2.4-2. Control Module Radiator Requirements

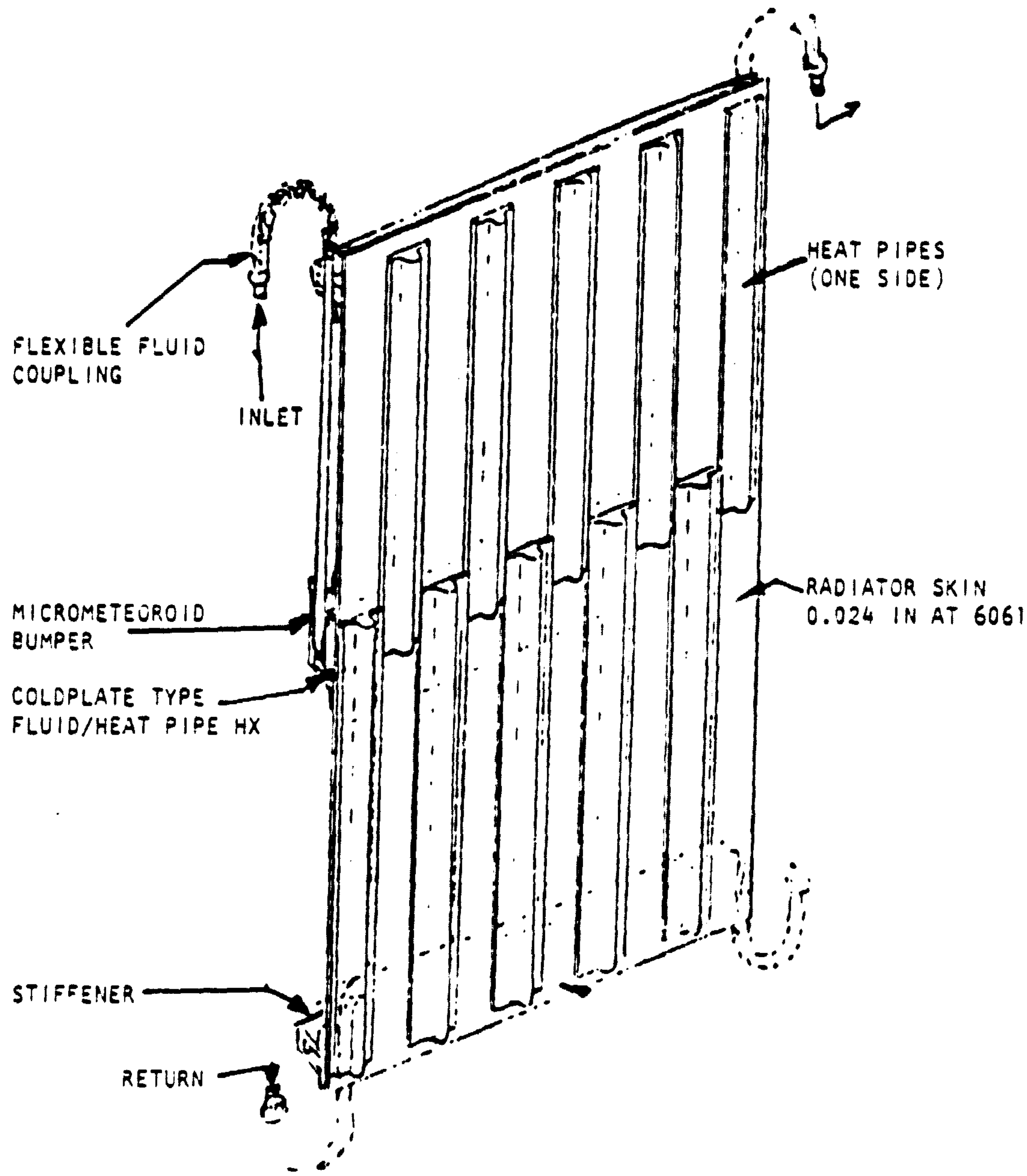
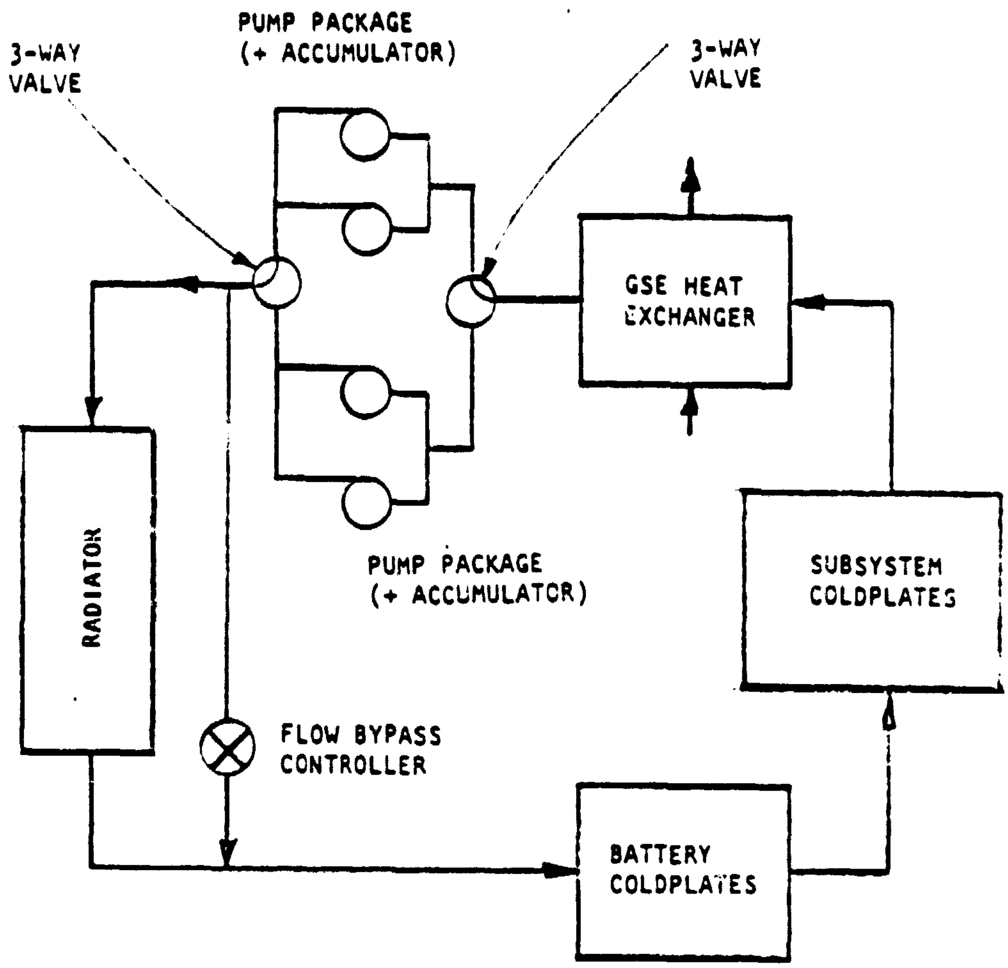


Figure 3.2.4-3. Heat Pipe Radiator Schematic



NOTE: SCHEMATIC ILLUSTRATES ONLY ONE OF DUAL COOLANT LOOPS WHICH OPERATE SIMULTANEOUSLY. RADIATOR, COLDPLATES, AND HEAT EXCHANGER ARE COMMON TO BOTH LOOPS

Figure 3.2.4-4. Fluid Loop Schematic

Heat exchanger (orbiter GSE heat exchanger, Specification SVSH 6424) has been included in the fluid loop, although the heat rejection system is not required to accommodate any payload thermal loads. The exchanger is included to support thermal control during orbiter ground and launch operations and to support potential on-orbit maintenance operations which might require radiator shutdown.

3.2.4.7 Coldplates

For the baseline system, aluminum was selected because of its light weight and high heat transfer rates. Placement of the component coldplates in the loop was dictated by their allowable operating temperatures. The current design assumes that allowable battery temperatures are between -5°C and 20°C with all other components capable of operating between 0°C and 40°C.

Individual coldplates were sized to allow enough area for component attachment and separation. Redundant components were located on different coldplates to guarantee system integrity during changeout and/or servicing operations.

3.2.4.8 Other TCS Elements

Due to the open structure configuration, components will require insulation and shielding to provide meteoroid protection and isolation from the thermal environment. Impact calculations for 0.99 component reliability (≈0.9 system reliability) yield required typical armor thickness of 0.20 cm (0.080 in.) for component packages. The protective layer would be reduced to about 0.05 cm (0.020 in.) if it is installed as a bumper instead of armor, e.g., not integral to the package. For the baseline design, however, the armor approach was implemented to facilitate potential servicing operations. Thermal isolation is also required for space-facing surfaces on coldplate-mounted components, and this can be accomplished by applying MLI blankets; 20 layers are required to provide insulation and allow for some effectiveness loss due to meteoroid impact.

Non-coldplate-mounted elements, like CMG's and RCS modules, require heaters to maintain allowable temperature levels. For components, such as CMG's, temperature-sensitive electronics are effectively isolated from external walls and supplemental heaters are intrinsic to the baseline design. RCS module heater requirements for the nitrogen tetroxide-monomethyl hydrazine system assumed an NTO/MMH mass ratio of 1.6 to 1 with four modules containing 1450 kg (3190 lb) of propellant in each module. For a black-paint coating, each module requires approximately 250 watts of power. This requirement derives from geosynchronous orbit environmental loading and assumes that LEO orbit transfer loads are more benign. Requirements for coatings, insulation, shielding, and heaters have been incorporated into individual component mass definitions.

3.2.4.9 Summary

The active thermal control system presented herein, combined with appropriate coatings, insulation, and heaters as required, is a long-life system capable of maintaining platform component temperatures within allowable levels during operational low earth, transfer and geosynchronous orbits.

3.2.4.10 References

1. Weidner, D. K., "*Space Environment Criteria Guidelines for Use in Space Vehicle Development (1969 Revision)*," NASA TM X-53957, October 17, 1969.

2. Richardson, A. J., "*Meteoroid Environment and Protection*," Appendix B of "*Technological Requirements Common to Manned Planetary Missions*," Report SD 67-621-3, Rockwell International, pp. 1-20, January 1968.

3. Kurland, Richard M., et al., "*Properties of Metallized Flexible Materials in the Space Environment*," SAMSO TR 78-31, January 1978.

3.2.5 <u>Tracking, Telemetry and Control (TT&C)</u>

3.2.5.1 Summary

The TT&C system uses S-band and Ku-band. The S-band has two subsystems:

1. Phase-modulated (PM) links provide tracking and two-way communication with the ground or through the TDRSS.

2. Frequency-modulated (FM) links provide direct one-way data transmission to ground.

The Ku-band is a two-way communication system which transmits data through TDRS. The TT&C components with sizes and weights are summarized in Table 3.2.5-1. A functional schematic of the TT&C is shown in Figure 3.2.5-1. The locations of the antennas is shown in Figure 3.2.5-2.

3.2.5.2 S-Band

The ETV platform's S-band TT&C system is comprised of two independent subsystems. The phase-modulated (PM) links provide tracking and two-way communications direct to ground or through the TDRSS. The frequency-modulated (FM) link provides for the transmission of data direct to ground (one-way). See Figure 3.2.5-2. Reliability is enhanced by using redundant units or else by providing internally redundant circuits. The single exception to this is the S-band antenna switch assembly.

The PM system consists of four helical antennas located on the ETV platform to communicate in varied directions. These antennas serve the Phase Modulation links to the USAF Satellite Control Facility (SCF) ground stations, both direct and TDRS-relayed, as well as through the NASA Spaceflight Tracking and Data Network (STDN) ground stations. This link is compatible with the Shuttle orbiter and in LEO the orbiter can act as a relay for communications, if desired. Four helicals are needed to insure 40% coverage even if the ETV platform tumbles in orbit.

Table 3.2.5-1. TT&C Summary of Size and Weight

	WEIGHT EACH (KG)	VOLUME EACH (M^3)	SIZE LxWxH (CM)	TOTAL VOLUME (M^3)	TOTAL AREA (M^2)	TOTAL WEIGHT (KG)
S-BAND						
PM TRANSPONDER (2 UNITS)	7.9	.0075	30×25×10	.0148	.150	15.8
PM PROCESSOR (2 UNITS)	4.1	.0037	15×25×10	.0074	.075	8.2
DOPPLER EXTRACTOR (INTERNAL RED.)	7.4	.0074	25×30×10	.0074	.075	7.4
POWER AMPLIFIER (100W) (INT. RED.)	14.4	.0148	20×37×20	.0158	.074	14.4
PRE-AMPLIFIER (INT. RED.)	11.6	.0120	20×30×20	.0121	.060	11.6
FM TRANSMITTER & ANTENNA (2 UNITS)	3.0	.0037	25×15×10	.0074	.075	6.0
FM PROCESSOR (INTERNAL RED.)	5.2	.0056	25×15×15	.0112	.037	5.2
SWITCH ASSEMBLY (SINGLE UNIT)	3.0	.0037	25×15×10	.0037	.037	3.0
KU-BAND						
.91 M DIA. ANTENNA DIA. (SINGLE)	13.0	.1620	91 D × 25	.1620	.650	13.0
ELECTRONIC ASSY. (INTERNAL RED.)	44.0	.3940	45×35×25	.3940	.157	44.0
SIGNAL PROCESSOR (TAPE RECORDER) (2 UNITS)	22.5	.3192	38×30×28	.6384	.228	45.0
					TOTAL	173.6

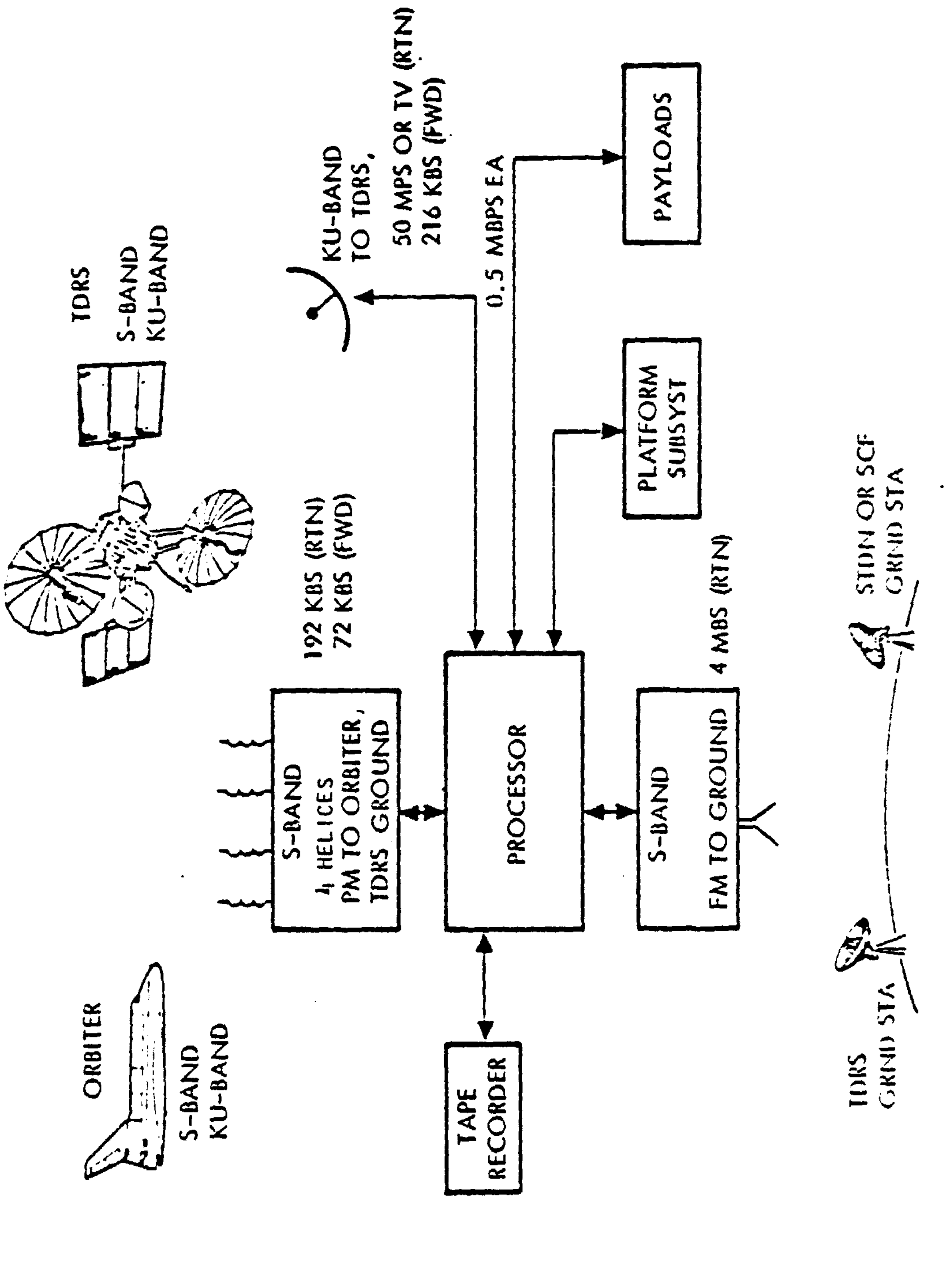


Figure 3.2.5-1. TT&C Functional Schematic

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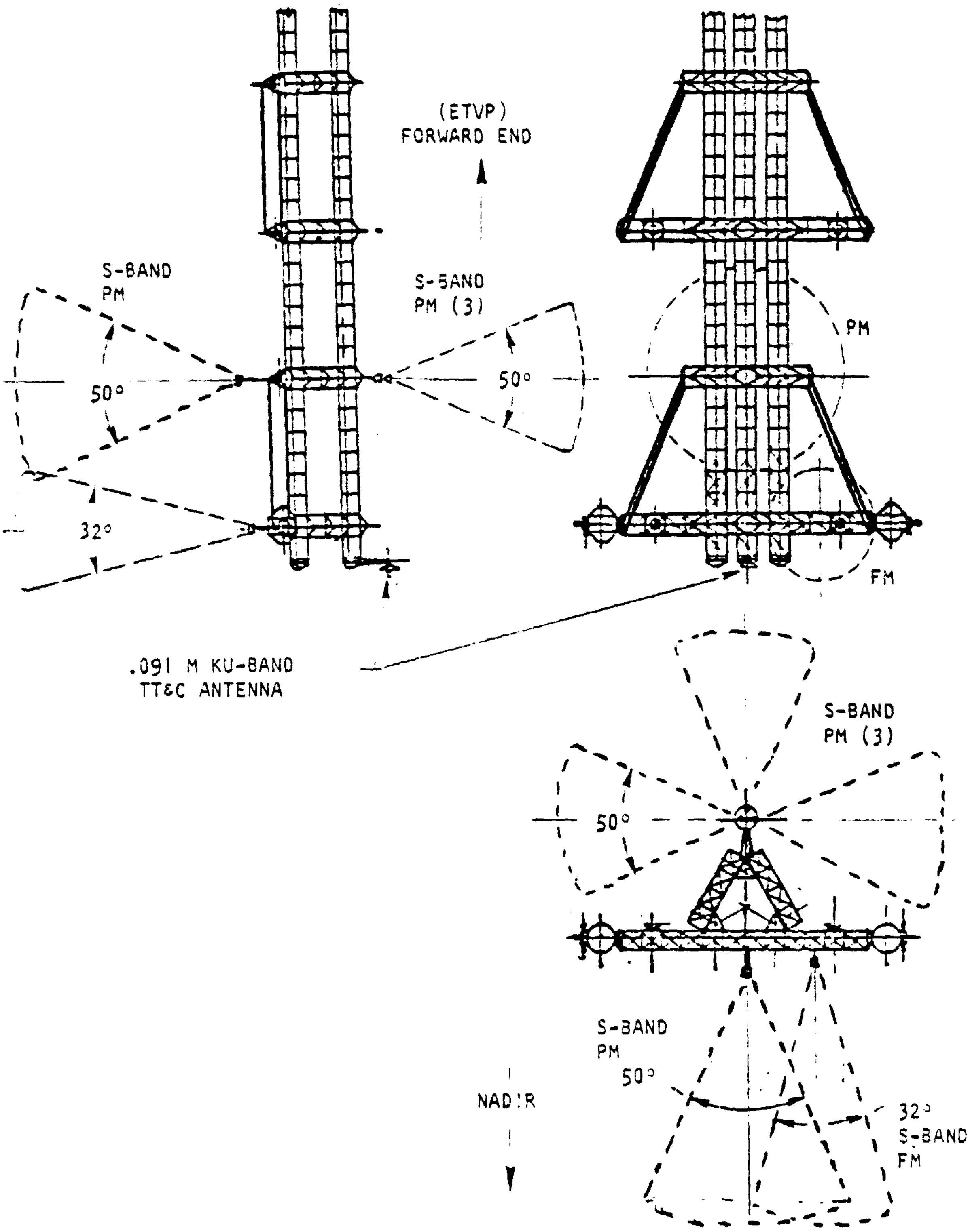

Figure 3.2.5-2. S-Band and Ku-Band Antenna Locations

In the TDRS PM link, the power amplifier generates 100 watts to help overcome the large space loss between LEO and GEO. Convolutional coding is also used in order to provide a better system margin. When communicating directly to ground, only 10 watts power is needed (even without coding) since the ground uses a 26-m aperture.

The FM link consists of one wide coverage horn with 16 dB gain. This link provides additional downlink capability from the ETV platform to ground stations (not TDRSS). The FM signal may be modulated by television, engine data, analog, digital, recorder playback, or real-time payload data (4 Mbits/sec maximum). The wide-coverage horn (32 degree beamwidth) is located on the bottom side of the platform which faces the earth stations in LEO and in GEO.

The FM link provides a high-capacity link direct to ground as a back-up to TDRS in LEO and at the primary link in GEO. The FM transmitter operates at 2250 MHz with an output power of 10 watts. Both baseband and RF filtering are provided to reduce out-of-channel interference to the PM and payload receivers. Less power is needed for this link than for the PM link, since no attempt to work with the 3.5-m TDRS antenna is made and a 16-dB horn is used rather than a 10-dB helix.

3.2.5.3 Ku-Band

The Ku-band parabolic antenna is mounted on the aft apex of the ETVP to allow viewing of the TDRS while the platform is in LEO. The Ku-band TT&C is a two-way communication system which transmits up to 50 Mbits/sec data (14.85-15.15 GHz) on the return link to the 3.5-m antenna on TDRS and receives up to 216 Kbits/sec (13.75-13.80 GHz).

A 0.91-m parabolic monopulse tracking antenna with a small acquisition horn is the only antenna used. Acquisition of the TDRS is aided via designation by the signal processor of an angle around which a spiral search is conducted. Acquisition thereafter is automatic. Ground control of antenna angle is also possible using the S-band link.

Mounted with the antenna are the antenna drive mechanism, drive electronics, traveling-wave tube (TWT) transmitter, and receiver front end.

Mounted remotely from the antenna is the signal processor package which includes tape recorders and the data management circuitry. Location for this package is in the control module located near the solar array.

3.2.6 Reaction Control System (RCS)

3.2.6.1 Summary

The RCS is sized to provide for the control of the antenna platform at the operational altitude in the Clarke (geosynchronous equatorial) orbit (GEO). The RCS provides control for both translation maneuvers (station-keeping) to maintain position in orbit over the earth reception target area to within 0.05 degree, and for attitude orientation control in conjunction with the attitude and velocity control subsystem (AVCS). In addition, during the low earth orbit (LEO) phases of the mission the RCS provides the necessary impulse for all the LEO orbit transfer maneuvers.

3.2.6.2 Features

The key features of the RCS are summarized in Table 3.2.6-1 and the RCS module is illustrated in Figure 3.2.6-1.

Table 3.2.6-1. RCS Summary

Propellants	N_2O_4/MMH
Pressurization gas	Helium
Total Impulse	16.7×10^6 N-sec (3.8×10^6 lb-sec)
Number of modules	4
Number of thrusters	16
Thrust, Each	
12 thrusters	4.4 N (1 lb_f)
4 thrusters	44 N (10 lb_f)
Total weight	8074 kg (17762 lb)

3.2.6.3 Configuration

A 16-thruster configuration, grouped in four modules with propellants and located at the four corners of the rectangular shaped platform, provides an RCS that meets the mission and functional requirements. The corner locations were selected to provide maximum-length moment arms and to avoid thruster exhaust impingement on the vehicle structure and components.

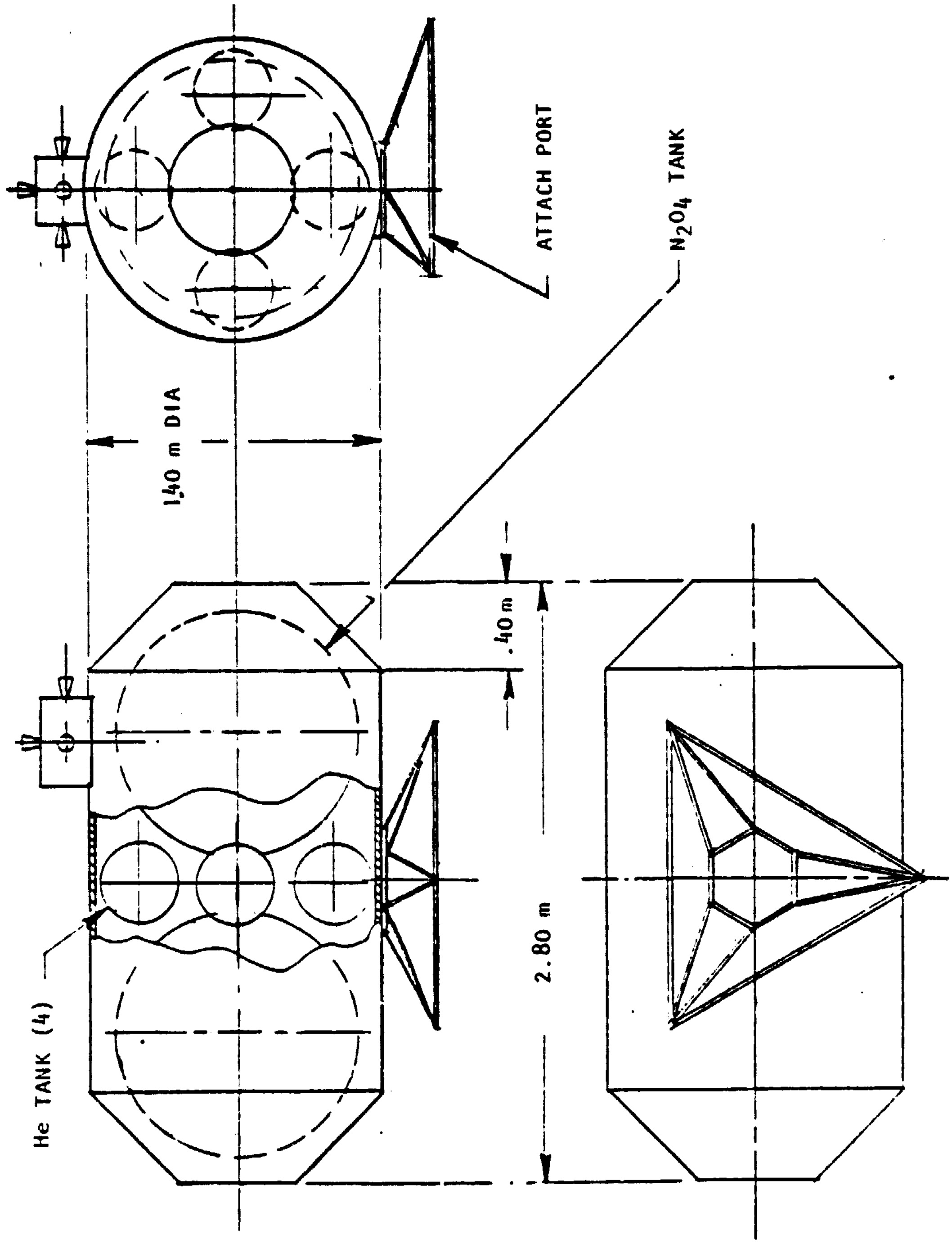

Figure 3.2.6-1. RCS Module

Each RCS module contains an oxidizer tank, a fuel tank, and helium pressurization tanks located within a structural shell that acts as a micro-meteoroid shield and for thermal control. On one side of the module, an assembly of four thrusters is located, with an attach port on the opposite side for mating and attachment to the platform structure.

Each assembly of four thrusters (per module) consists of one 44-N (10-lb_f) thruster (N-S stationkeeping) and three 4.4-N (1-lb_f) thrusters oriented for CMG momentum dumping about the pitch, roll and yaw axes, and for E-W station-keeping. Operation of the four-module RCS includes firing two 44-N (10-lb_f) thrusters at a time for N-S stationkeeping, and alternately two or four 4.4-N (1-lb_f) thrusters at a time for three-axis CMG momentum dump and E-W stationkeeping.

3.2.6.4 Propellants

Storable propellants (N_2O_4/MMH) were selected for the RCS to be compatible with long-duration propellant storage for the seven year resupply interval and still provide reasonable performance. Bi-propellant RCS thrusters, such as those being developed by Aerojet (2-N (0.5-lb_f), 22-N (5-lb_f), and 445-N (100-lb) thrusters) provide a steady-state specific impulse of 2750 N-sec/kg (280 sec) to 2890 N-sec/kg (295 sec). Structural mass fraction efficiency of 0.718 was assumed. These performance values were used in sizing propellant quantities since relatively long pulse durations are required. (For N-S stationkeeping, the 44-N (10-lb_f) thrusters fire for 33 seconds duration, and the shortest pulse for the 4.4-N (1-lb_f) thruster is 13 seconds duration for pitch attitude CMG dumping.) RCS propellant requirements were based on a platform weight of 31136 kg (68500 lb) without RCS, and with RCS a platform weight of 39210 kg (86262 lb). Each RCS module contains 1449 kg (3188 lb) propellant and has an initial gross weight of 1268.5 kg (4440.5 lb) which does not include the docking port. The total gross weight of the four RCS modules is 8074 kg (17762 lb).

Control of the antenna platform in GEO is required for the anticipated life of the vehicle which is assumed to be 20 years' duration. For purposes of sizing the RCS propellant quantities, a resupply interval of seven years in GEO is assumed. The life of storable propellant tankage, feed and propellant management devices, and other components is estimated to be 7 to 10 years.

3.2.6.5 Manuevers

The functional requirements for the RCS in GEO consist of:

1. Translational maneuvers for stationkeeping and platform positioning on station.

2. Attitude orientation and control maneuvers.

3.2.6.6 Stationkeeping

The translation impulses for stationkeeping consist of maneuvers to counteract the perturbations caused by the earth's tesseral harmonics (east-west stationkeeping), lunar-solar gravity (north-south stationkeeping), and solar pressure (eccentricity maintenance).

East-west stationkeeping velocity requirements and limit cycle duty times for synchronous satellites are shown in Figures 3.2.6-2 and 3.2.6-3. These velocity requirements are limited to correcting the inplane perturbations of the orbit due to the earth's nonspherical shape. Due principally to the ellipticity of the earth's equatorial surface, the semi-major axis of the orbit will be changed causing the satellite to drift toward the nearest stable node (76° East or 140° West). Yearly velocity requirements to maintain a satellite on station for the equatorial region 60-140° W are shown in Figure 3.2.6-2. The maximum velocity required is 2.03 meters per second per year however, occurs for a satellite located near 120 degree east longitude.

Limit cycle duty times are also presented in Figure 3.2.6-3. This cycle time is the time between the delta velocity maneuvers required to maintain the satellite within a region of the desired longitude (deadband). When the satellite has drifted to one edge of the deadband, velocity is added to the orbit such that it will reverse its drift. The satellite will then drift to the other deadband edge before the geopotential effects slow the drift rate to zero and again cause it to drift toward the nearest stable node. The minimum time between required stationkeeping maneuvers again occurs at 120 degree east longitude. For a deadband of 0.05 degree the minimum limit cycle time is 14.3 days.

For sizing the RCS, the maximum velocity was assumed as the necessary design impulse for the east-west stationkeeping requirement. The maneuver itself consists of two nearly identical burns approximately 12 hours apart. The magnitude and the frequency of the correction maneuver depends on the longitude of the equatorial station of the platform.

The combined effect of the lunar-solar gravitational acceleration produces a very long period (53 years) oscillation of the Clarke orbit inclination. The out of plane motion induced by the moon is of the order of 2.5 times greater than the sun. For first order approximation the two effects can be combined; the magnitude of the regression being determined from the geometry of the orbit planes initial inclination and the right ascension of the ascending node, and the inclination of the lunar orbit with respect to the earth's equator. The lunar orbit plane regresses around the ecliptic in a period of 18.6 years. The lunar orbit plane inclination to the ecliptic remains constant as the nodes regress which results in an oscillation of the lunar orbit plane inclination to the earth's equator. The lunar orbit plane inclination varies between 18.4 and 28.6 degrees during the 18.6 year cycle. The position within this cycle determines the magnitude of the correction velocity and the frequency of impulse application to overcome the perturbation (Figure 3.2.6-4). The latter is also a function of the maximum permissible excursion caused by the gravitational disturbance.

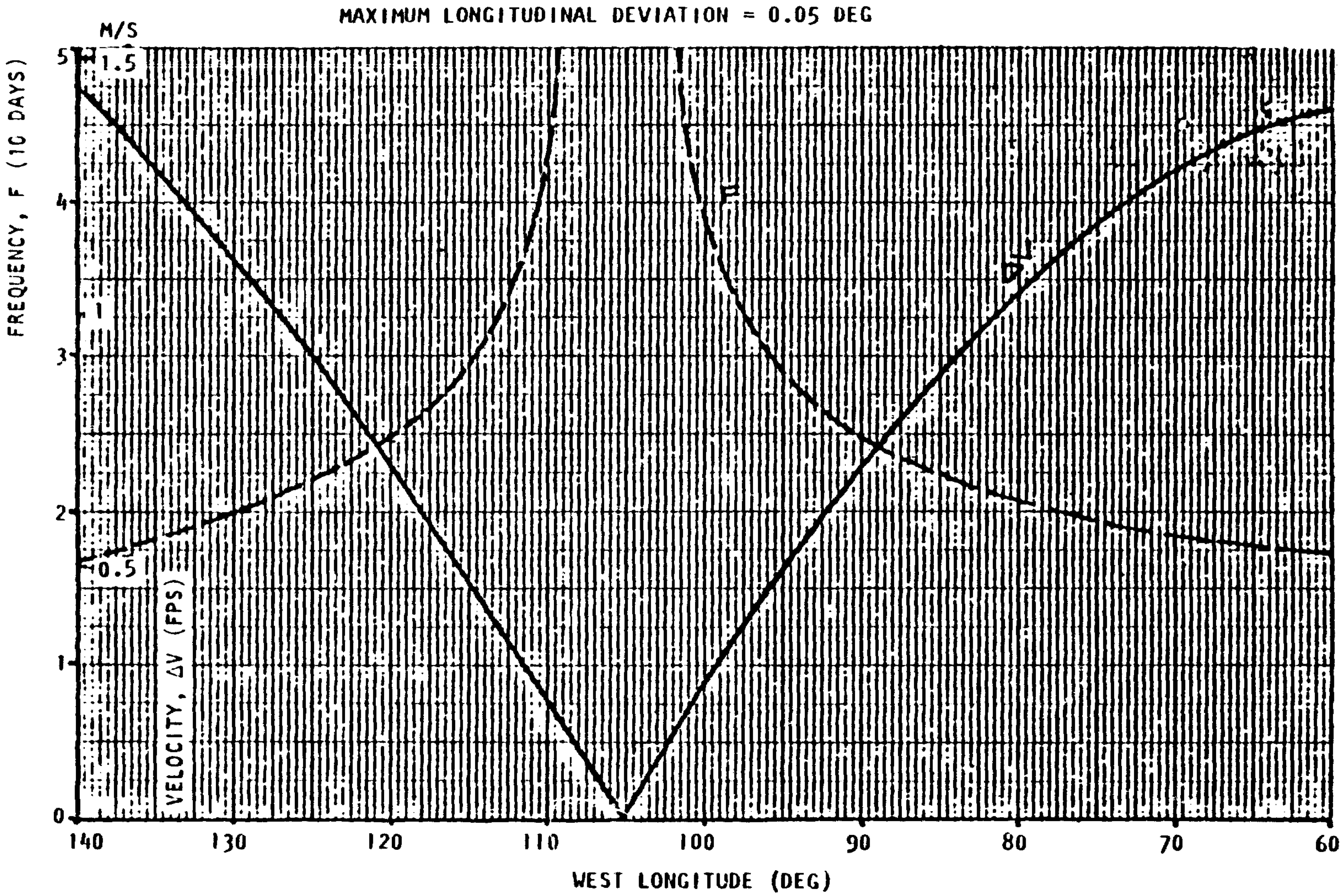

Figure 3.2.6-2. Clarke Orbit East-West Stationkeeping Requirements (One Year)

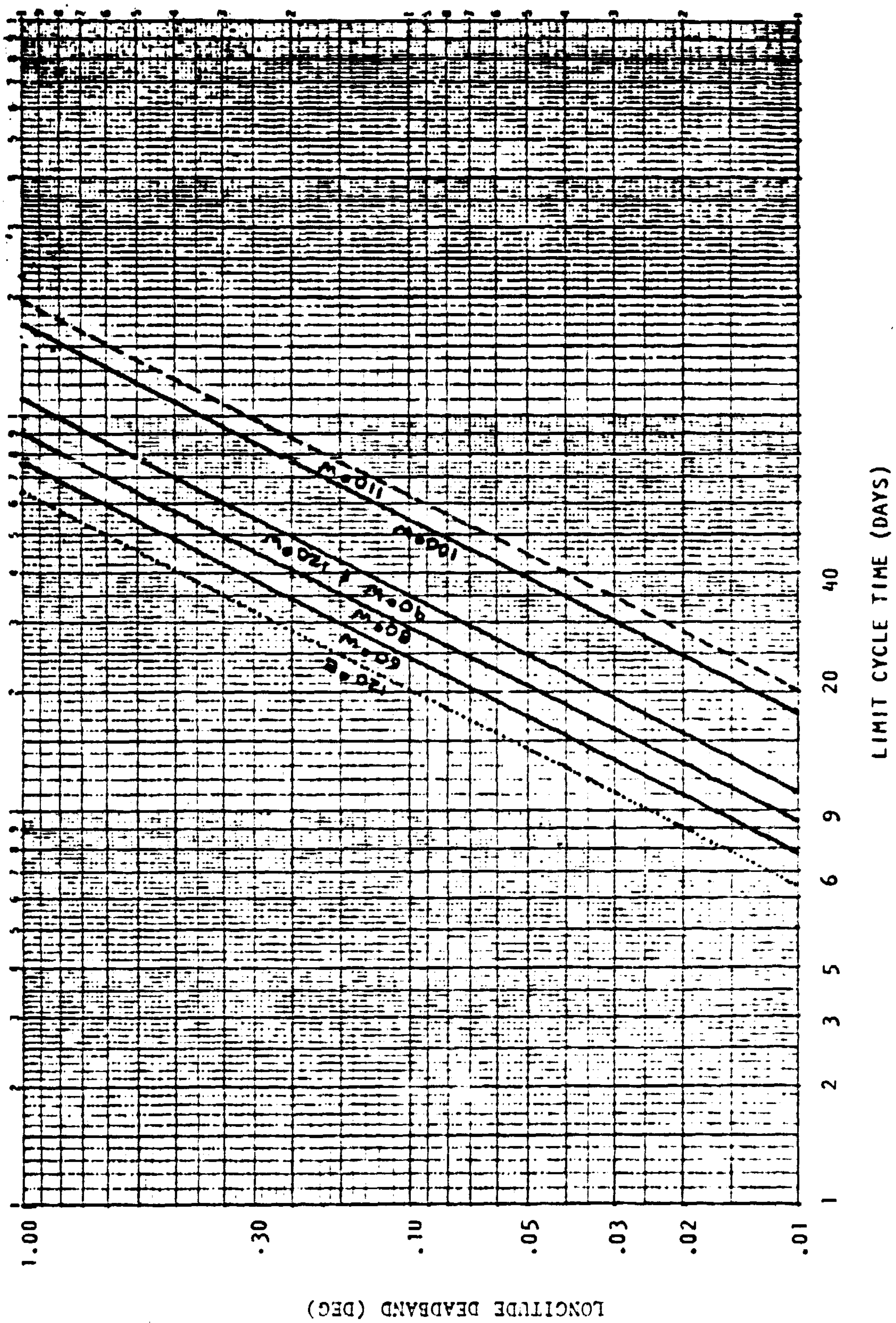

Figure 3.2.6-3. Tesseral Harmonic Perturbations - Clarke Orbit Limit Cycle Time for Stationkeeping

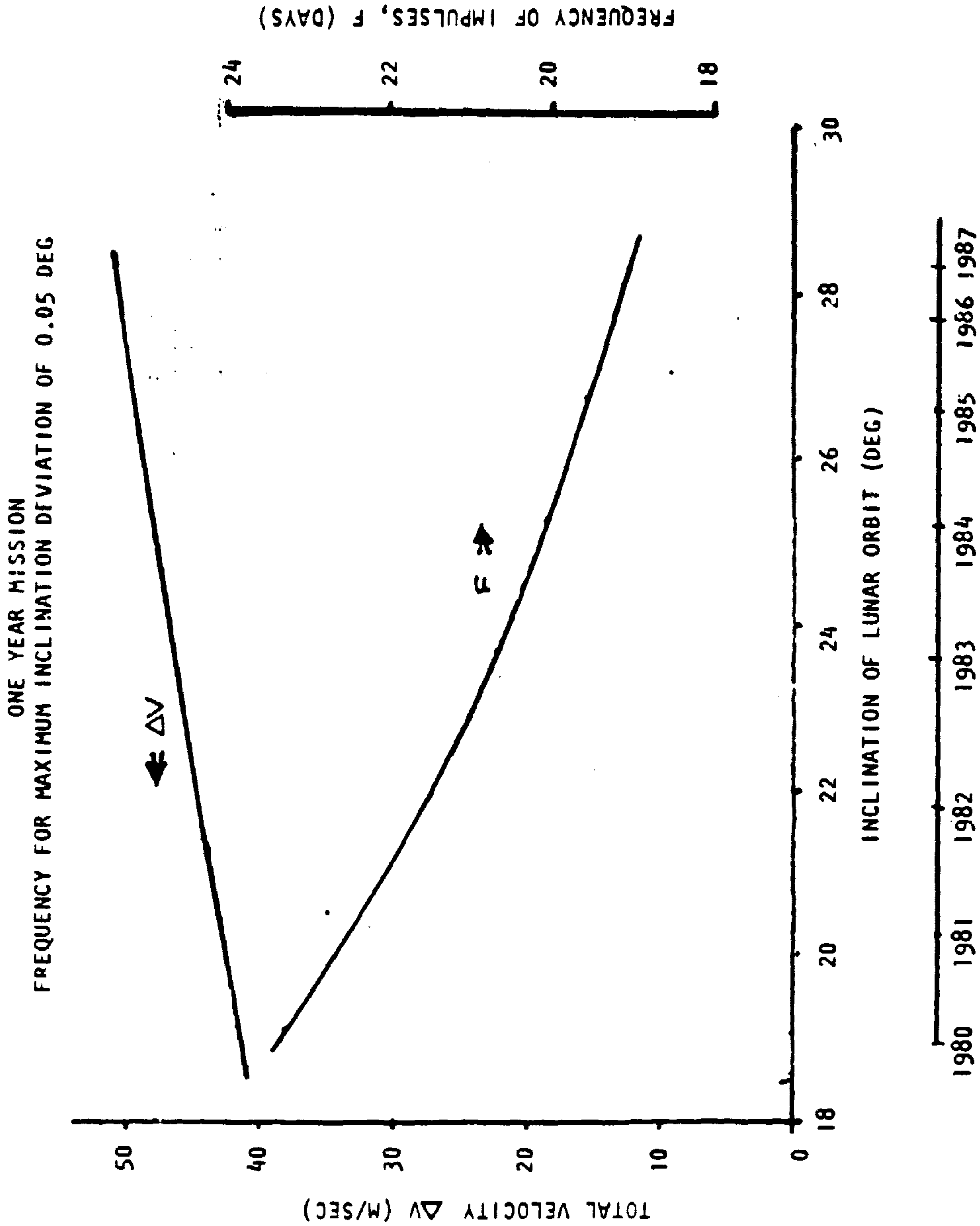

Figure 3.2.6-4. North-South Stationkeeping Requirements
Clarke Orbit (Nominal Injection)

The 18.6 year cycle average inclination is 23.5 degrees, while for the 9.3 year peak in the cycle the average inclination is 26.9 degrees. The average Clarke orbit perturbations resulting from the lunar-solar influence is presented in Figure 3.2.6-5.

The major effects of solar radiation pressure on a geosynchronous orbit is a yearly oscillation in eccentricity and a rotation of the line of apsides. For an initially circular geosynchronous orbit, solar pressure will cause the eccentricity to increase to some maximum value after six months and return to circular after one year. The maximum value for the eccentricity is a function of the satellite effective area-to-weight ratio. For the platform in the Clarke orbit this ratio is approximately (0.02 m^2/kg (0.1 ft^2/lb) . For this area-to-weight ratio the maximum eccentricity perturbation and corresponding daily longitudinal librations are illustrated in Figure 3.2.6-6. The correction maneuver is usually performed by applying two impulses nearly 12 hours apart (one at perigee and the other at apogee). The frequency and the yearly ΔV requirements for these maneuvers are shown in Figure 3.2.6-7.

The GEO perturbation sources, effects, and velocity requirements to overcome them are summarized in Figure 3.2.6-8. The total ΔV requirement per year is 56.4 m/sec or 394.9 m/sec for the entire seven year mission.

The spacecraft positioning maneuver ΔV requirement is entirely dependent on the repositioning drift rate desired during the maneuver and, of course, the number of such maneuvers desired during the entire seven year mission. Figure 3.2.6-9 illustrates the relationship between the position drift rate and the corresponding velocity increment required to initiate or stop the drift. Twice the 2.85 meters per sec/degrees per day value is required to complete the entire maneuver; that is to initiate and terminate the spacecraft drift.

For the seven-year mission it is assumed that besides the one initial positioning maneuver four additional repositioning maneuvers will be required. All these maneuvers are to be performed with a longitudinal drift rate of one degree per day. The ΔV requirement for these five maneuvers is 28.5 m/sec.

3.2.6.7 Attitude Orientation & Control

The RCS attitude orientation and control maneuvers consist of periodic momentum dump of the control moment gyros (CMG's) and some discrete preplanned attitude maneuvers during the LEO to GEO transfer and the seven year GEO mission. The weight and the moment-of-inertia characteristics used for the computation of the propellant demand for this control phase of the mission are shown in the mass properties summary in Table 3.2.6-2.

The RCS thrusters are fired every 12 hours (twice a day) for CMG desaturation expending propellant at a rate of 0.0715 kg/maneuver (0.143 kg/day). It is possible that some of the orbit-keeping maneuvers may be combined with the CMG desaturation maneuvers. The resulting

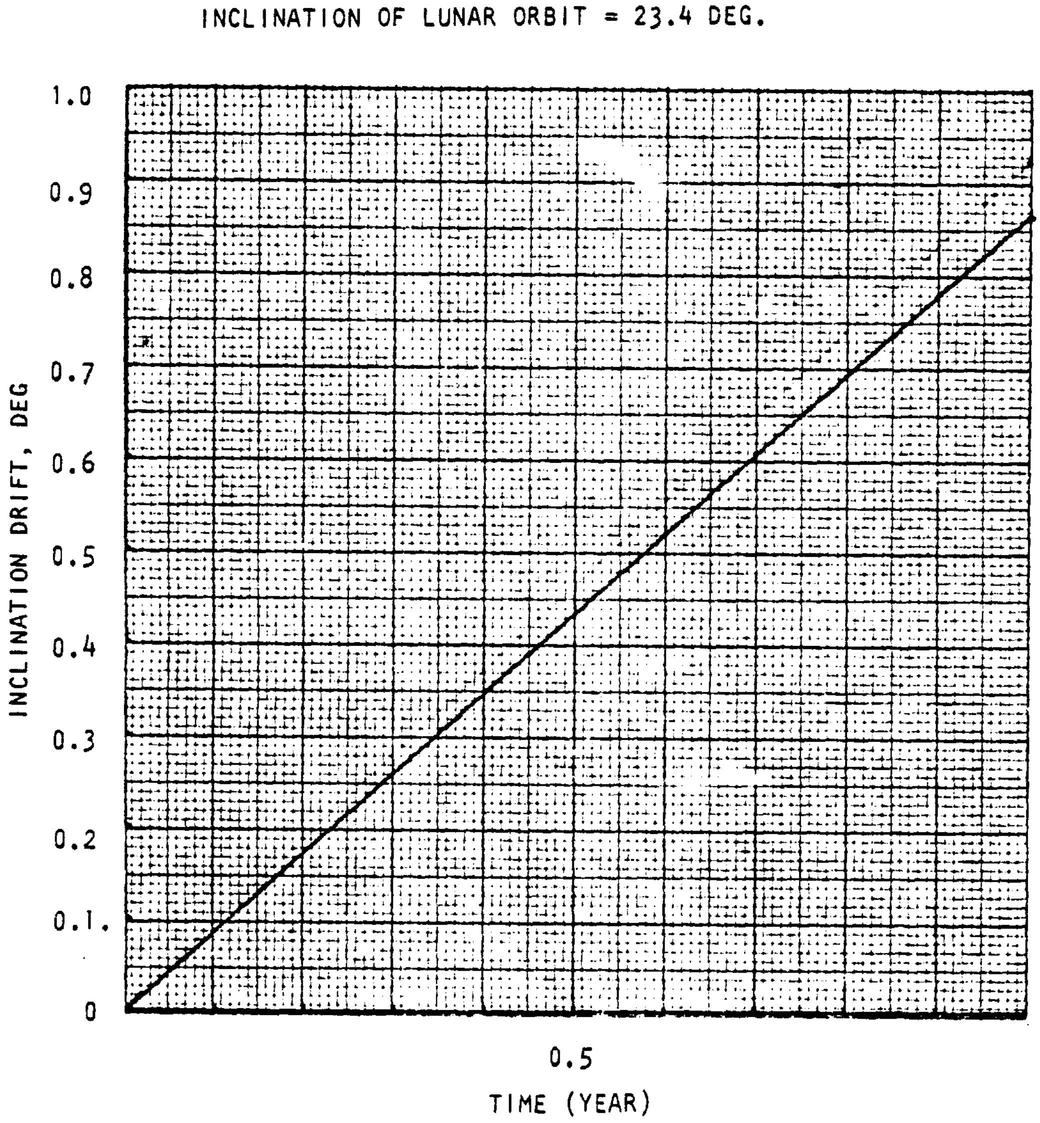

Figure 3.2.6-5. Lunar Solar Perturbations—Clarke Orbit

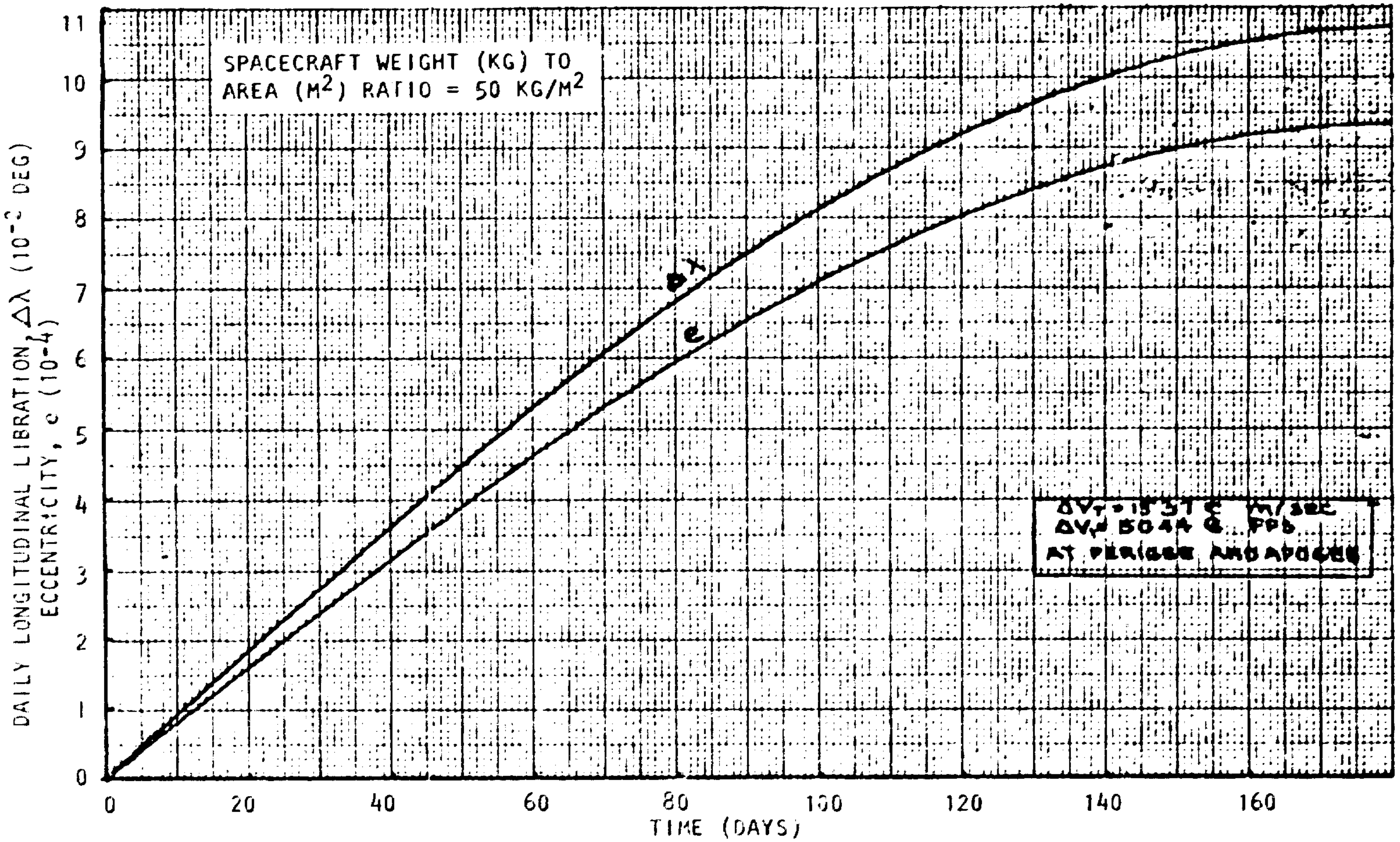

Figure 3.2.6-6. Solar Pressure Perturbations - Clark Orbit

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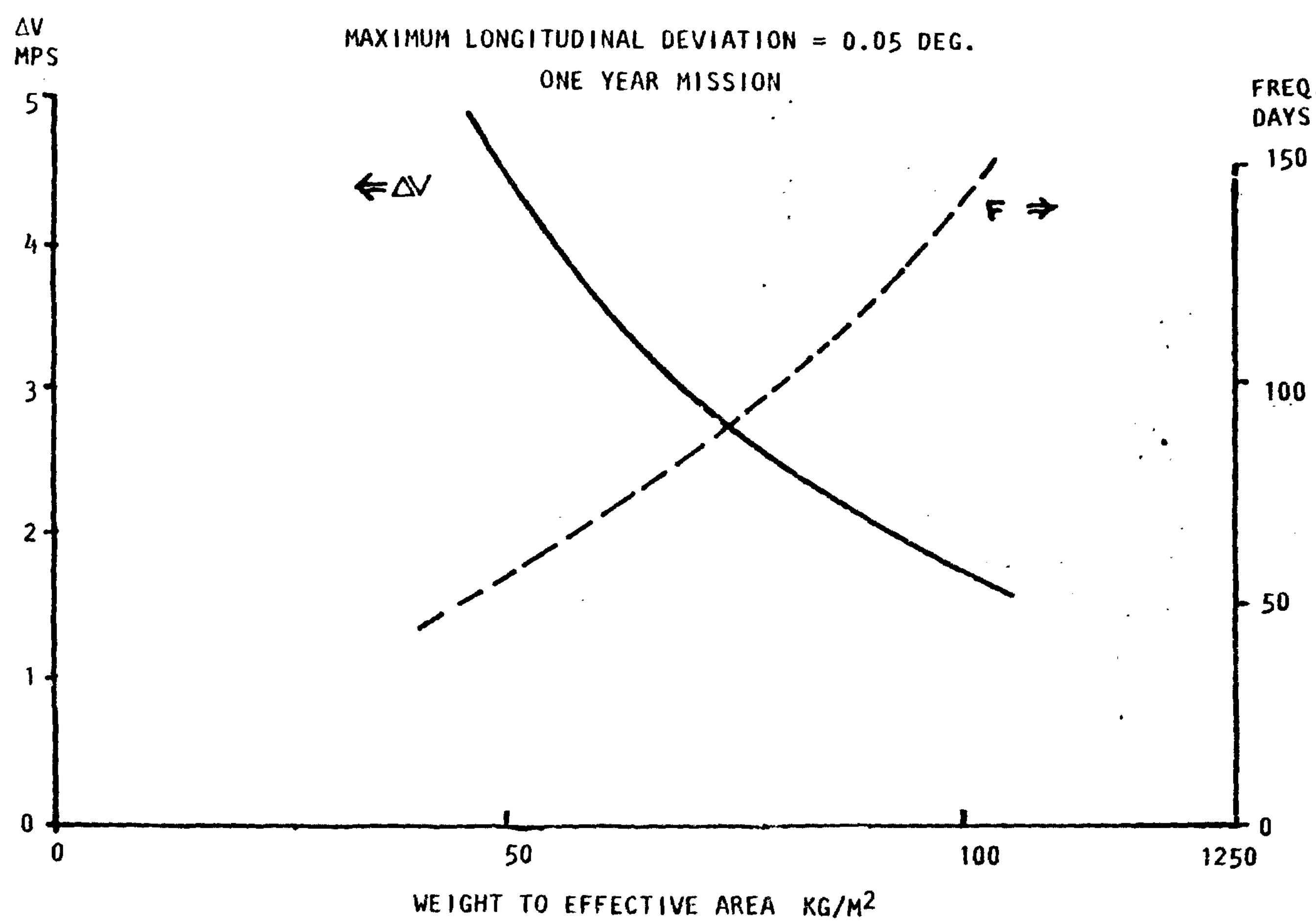


Figure 3.2.6-7. Eccentricity Control Requirements

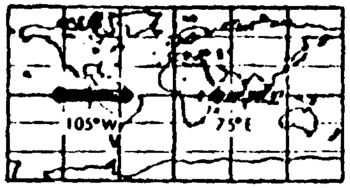
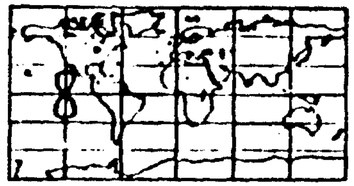
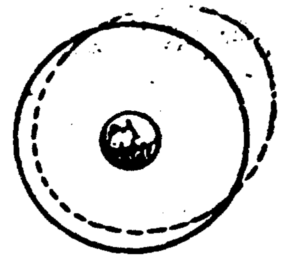
PERTURBATION SOURCE	TESSERAL HARMONIC	LUNAR-SOLAR GRAVITY	SOLAR PRESSURE
PERTURBATION EFFECT	 APPARENT LONGITUDINAL DRIFT	 INCLINATION PERTURBATION	 ORBIT ECCENTRICITY PERTURBATION
MANEUVER FREQUENCY ΔV PER MANEUVER ΔV PER YEAR	14.3-44.5- DAYS $\Delta V_1 \approx \Delta V_2$=0.04-0.014-0 M/S 2.0 M/S/YR	Δi = 0.0025 DEG/DAY ΔV = 6150 sin $\Delta i/2$ M/S f = 19.7 DAYS ΔV = 2.68 M/S 49.9 M/S/YR	$\Delta e = 8.04\times10^{-6}$ DEG/DAY ΔV = 1537 Δe M/S f = 56.5 DAYS $\Delta V_1 = \Delta V_2$=0.342 M/S 4.5 M/S/YR
SEVEN YEAR TOTAL	14.3 M/S (46.9 FPS)	349 M/S (1145.2 FPS)	31.6 M/S (103.6 FPS)

Figure 3.2.6-8. Clarke Orbit Perturbation Sources and Effects

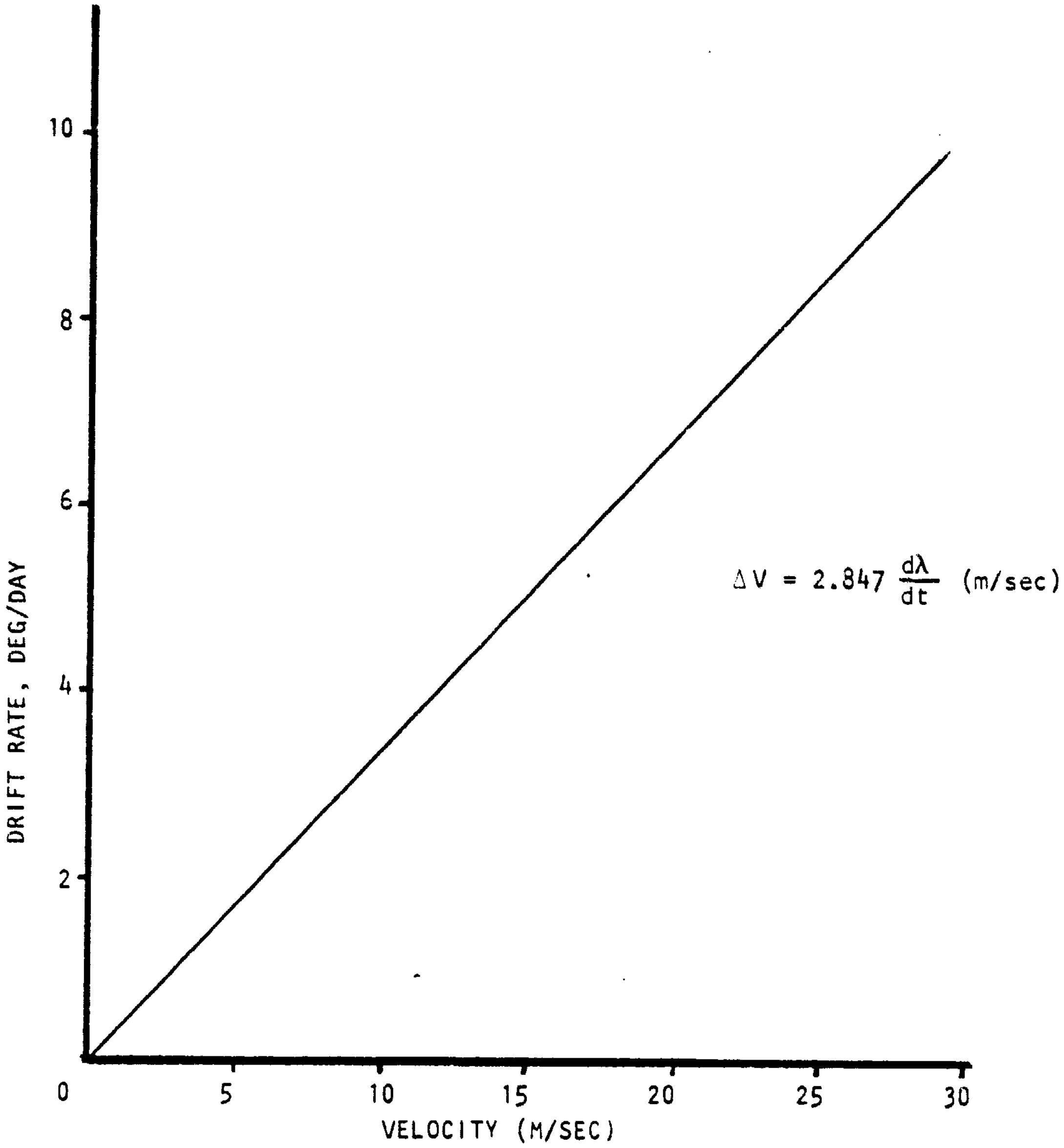
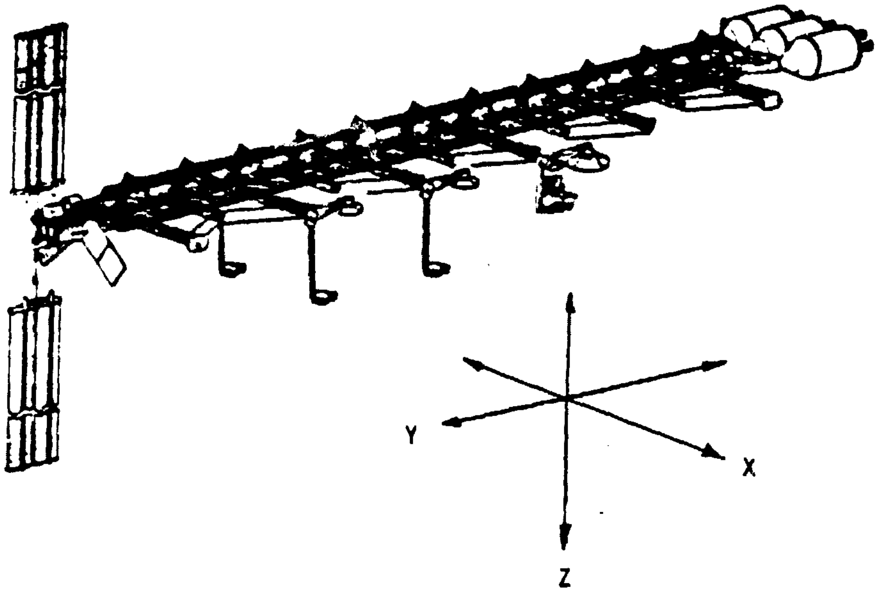
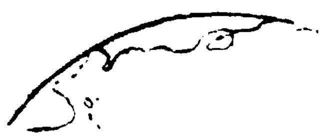


Figure 3.2.6-9. Velocity Required to Establish or to Stop a Drift Rate of a Spacecraft in Clarke Orbit

Table 3.2.6-2. ETVP Operational Configuration Mass Properties



PARAMETER	VALUE
MASS	39,210 KG
I_{xx}	108.951E6 KG-M^2
I_{yy}	3.963E6 KG-M^2
I_{zz}	109.595E6 KG-M^2
I_{xy}	0.0
I_{xz}	0.008722 KG-M^2
I_{yz} m	-3.861E6 KG-M^2



propellant savings from such a combination of maneuvers could be translated into extended mission durations (interval between servicing visits).

Each year in geosynchronous orbit, four discrete three-axis attitude change maneuvers will be performed. The attitude rate during these maneuvers will be 0.05 deg/sec. Each attitude change maneuver requires two sets of impulses, first to start the rotation of the platform and the second set to terminate the rotational rate at the new attitude. For the RCS propellant requirement estimate it is assumed that the conservative method of performing three independent single-axis maneuvers for one three-axis maneuver will be employed.

For the above type of attitude change 2.25 kg of propellant per maneuver will be required.

In addition to the basic mission attitude change maneuvers already discussed five three-axis maneuvers at 0.1 deg/sec are allocated during the LEO to GEO transfer phase. The total amount of RCS propellant for this phase of the mission is 60.9 kg.

3.2.6.8 GEO Propellant Requirements

The total RCS requirements for the seven (7) year mission are summarized in Table 3.2.6-3. The total RCS weight is 8073.6 kg or 2018.4 kg per quad.

Table 3.2.6-3. RCS Propellant Requirements (7-Year Mission)

Translation maneuvers	m/s	(fps)	kg	(lb)
o East-west stationkeeping	14.3	(46.9)		
o North-south stationkeeping	349.0	(1145.2)		
o Eccentricity control	31.6	(103.6)		
o Initial positioning on station	6.0	(19.6)		
o Station repositioning (4)	22.6	(74.4)		
Total	423.5	(1389.7)	5303	(11666)
Attitude orientation and control maneuvers			kg	(lb)
o 4 - 3-axis maneuvers per year at 0.05%			63	(138.6)
o CMG momentum dump			365.6	(804.3)
o 30 day RCS backup			4.5	(10.0)
o Transfer orbit attitude control 5 - 3-axis maneuvers at 0.1°/s			60.9	(134.0)
Total			494.0	(1086.9)
Grand Total propellant carried			5796.8	(12752.9)
Total RCS weight			8073.6	(17761.9)
RCS weight per quadrant			2018.4	(4440.5)

For mission durations of 5 to 10 years the total RCS weight increment per year is approximately 1250 kg. This sensitivity of the total RCS requirement to the GEO servicing interval is illustrated in Figure 3.2.6-10.

3.2.6-9 LEO Propellant Requirements

The unserviced mission duration in low earth orbit is assumed to be six months. The functional requirements for the RCS in this operational mode consists of:

1. Translation maneuvers for orbit transfer from the Shuttle rendezvous compatible orbit altitude to some higher mission operations or test orbit altitude and back.

2. Attitude orientation and control maneuvers required in the mission operations or test orbit.

The maximum RCS propellant loading for this phase of the mission is based on the GEO mission requirements. Thus, the capability to perform translational and rotational maneuvers in LEO will be derived for a system capacity sized to the GEO mission.

The translation velocity requirements can be expressed as a function of orbit altitudes

$$\Delta V_T = \Delta V_p + \Delta V_a$$

$$\Delta V_T = \sqrt{\frac{\mu}{r_o}}\left\{\sqrt{\frac{2r_f}{r_o + r_f}}\left(1-\frac{r_o}{r_f}\right) + \sqrt{\frac{r_o}{r_f}} - 1\right\}$$

or for the altitudes of interest (370 - 1000 km)

$$\frac{\Delta V_T}{\Delta V_H} \quad 0.55 \text{ m/sec/km}$$

Each six month mission phase will require two such translation maneuvers, one to go from the 370 km (200 nmi) Shuttle orbit to the mission operations orbit, and the second of the two sets of impulses to return to the orbiter rendezvous orbit.

The propellant that remains may be used for the attitude orientation and control maneuvers. These attitude orientation and control maneuvers consist of CMG desaturation maneuvers requiring approximately 0.3 kg of propellant per day. This value is reasonable for orbital altitudes in the 500 km range.

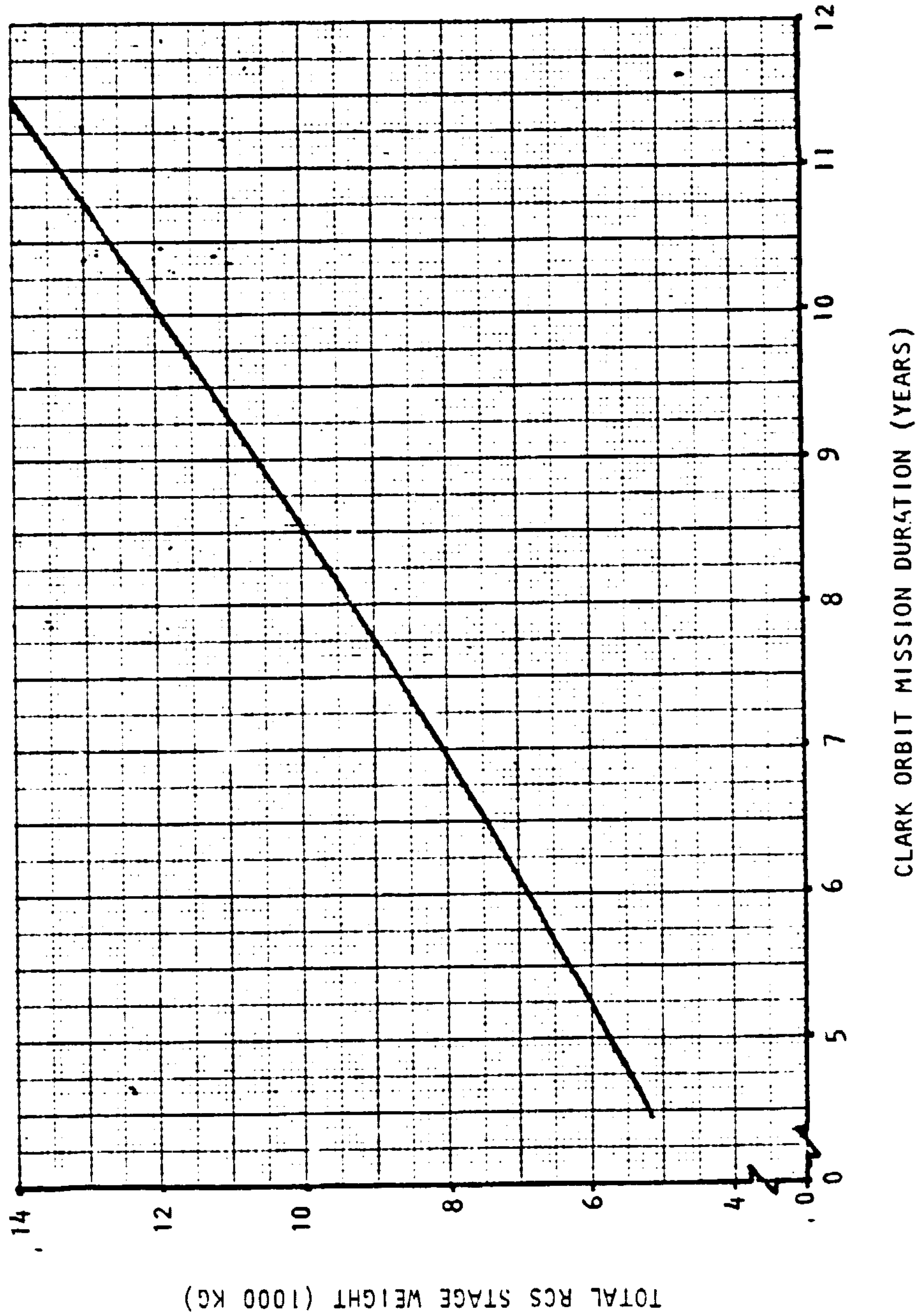


Figure 3.2.6-10. RCS Module Weight

The attitude change maneuvers were assumed to have the following distribution:

20%	3-axis maneuvers	@	0.1 deg/sec.
20%	3-axis maneuvers	@	0.03 deg/sec.
30%	single axis maneuvers	@	0.1 deg/sec.
30%	single axis maneuvers	@	0.03 deg/sec.

The above distribution results in the usage of 2.07 kg of propellant per "average" attitude maneuver.

Based on these ground rules and the RCS propellant requirement for geosynchronous orbit maintenance (mission duration of 5, 7 and 10 years) the relationship between the number of permissible average attitude maneuvers per day can be expressed as a function of LEO operations orbit altitude (Figure 3.2.6-11). For the seven year baseline mission this operation altitude will have to be less than 800 km (430 nmi).

Assuming LEO operations will require two attitude maneuvers per day the permissible LEO operations altitude now can be expressed as a function of the geosynchronous orbit mission duration (Figure 3.2.6-12).

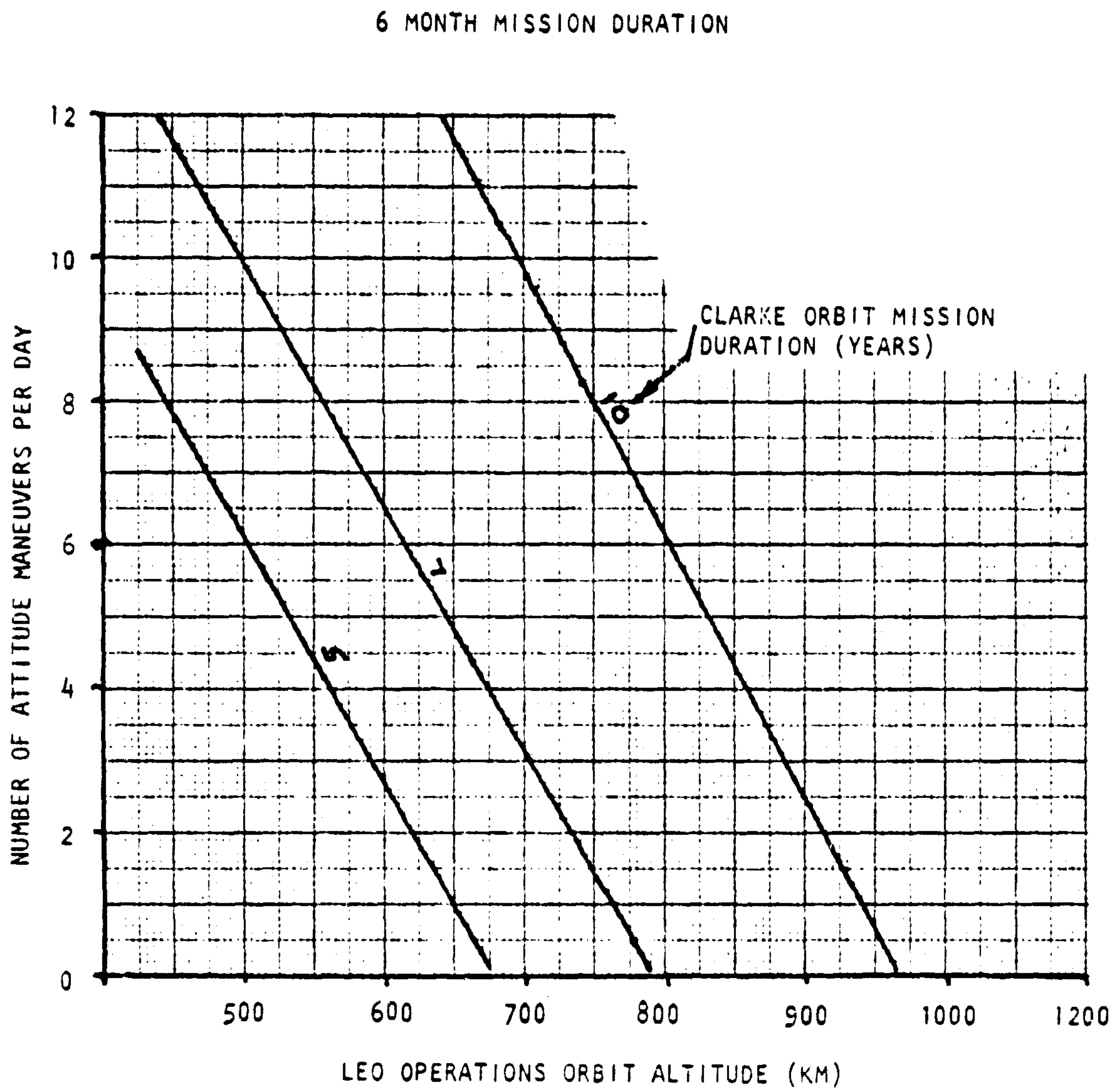


Figure 3.2.6-11. Frequency of Attitude Maneuvers

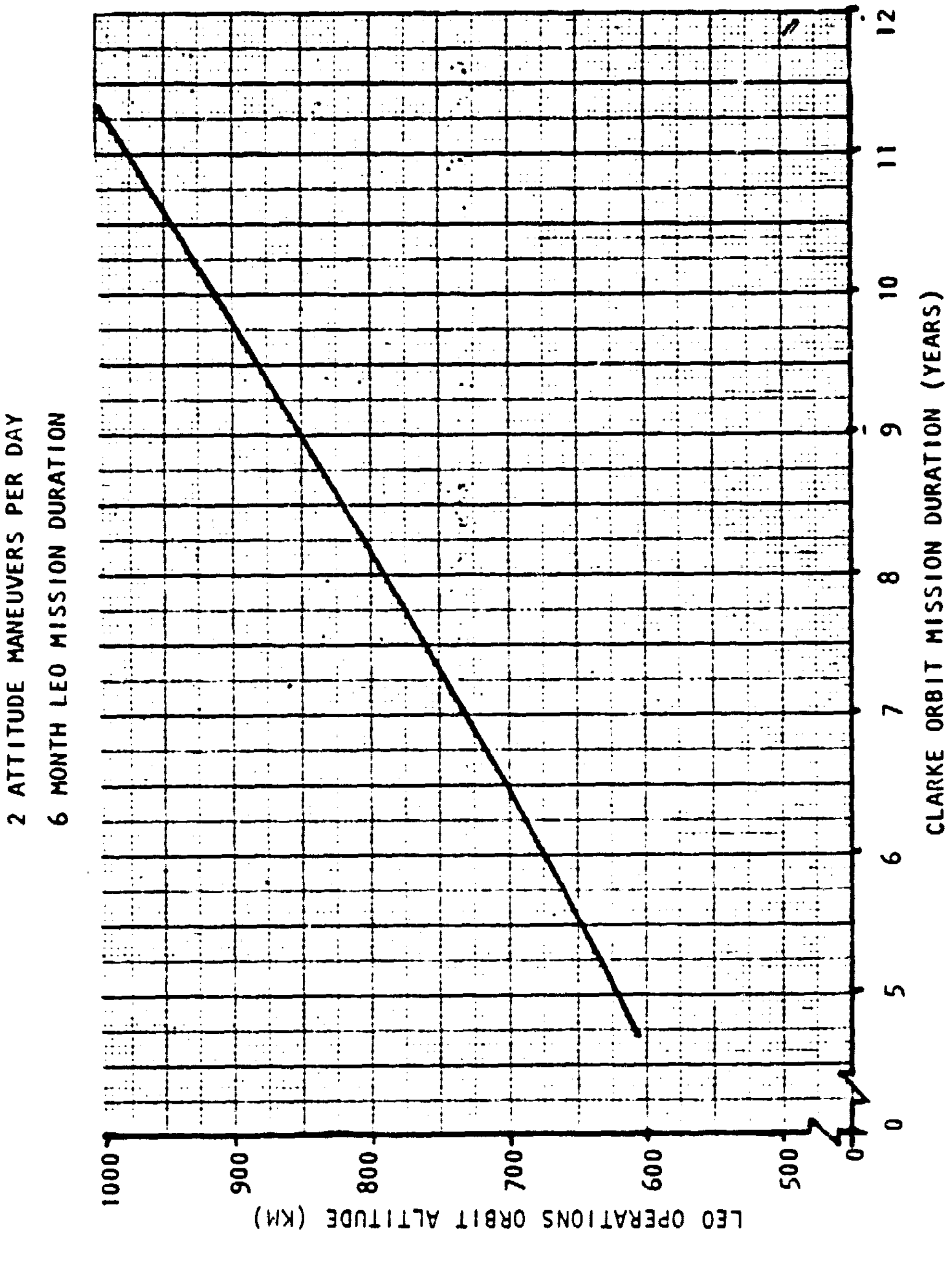


Figure 3.2.6-12. LEO Operations Orbit Altitude

3.2.7 Orbit Transfer Propulsion System

3.2.7.1 Summary

A cluster of three low-thrust propulsion modules is provided for transporting the antenna platform from the construction altitude in LEO up to the operational altitude at GEO.

The significant requirements for the orbit transfer propulsion module include the following: thrust-to-weight (T/W) ratio, velocity increment, maximum size and number of modules, propellant storability, and thrust vector control (TVC).

A maximum T/W of 1.96 N/kg (0.2 lb_f/lb_m) is imposed on the propulsion module design by the structural limitations of the space-fabricated structure.

3.2.7.2 Description

The key features of the orbit transfer propulsion system are summarized in Table 3.2.7-1. The low-thrust propulsion (LTP) module is illustrated in Figure 3.2.7-1.

Table 3.2.7-1. Orbit Transfer Propulsion Summary (7 Year GEO Mission)

PROPELLANTS	LO_2/LH_2
Total impulse	112.1×10^6 N-sec (25.2×10^6 lb-sec)
Number of modules	3, parallel
Firing/staging sequence	2/1 modules
Number of engines	12
Thrust, each	22,240 N (5000 lb_f)
T/W, max.	1.96 N/kg (0.2 lb_f/lb_m)
Ignition weight: each	26,774 kg (58,904 lb)
3 modules*	80,322 kg (176,712 lb)
Boiloff	3%
Loaded weight: each	27,475 kg (60,446 lb)
3 modules	82,425 kg (181,338 lb)
*For payload weight of 39,210 kg (86,262 lb).	

The overall dimensions of the module are compatible with orbiter payload bay size and the overall length is within the 10.7 m (35 ft) length target for OTV design. This is accomplished in part by the use of multiple 22,240-N (5,000-lb) thrust engines which are short, and eliminate the need for nozzle retraction mechanisms.

A single oxidizer tank, fuel tank, and helium pressurization gas tanks are located within a structural shell that acts as a micrometeoroid shield. The design features the use of non-integral propellant tanks with multi-layer insulation (MLI) for control of boiloff. Based on prior studies, an allowance for one-inch thich MLI would control boiloff of LO_2/LH_2 propellants to 0.7%

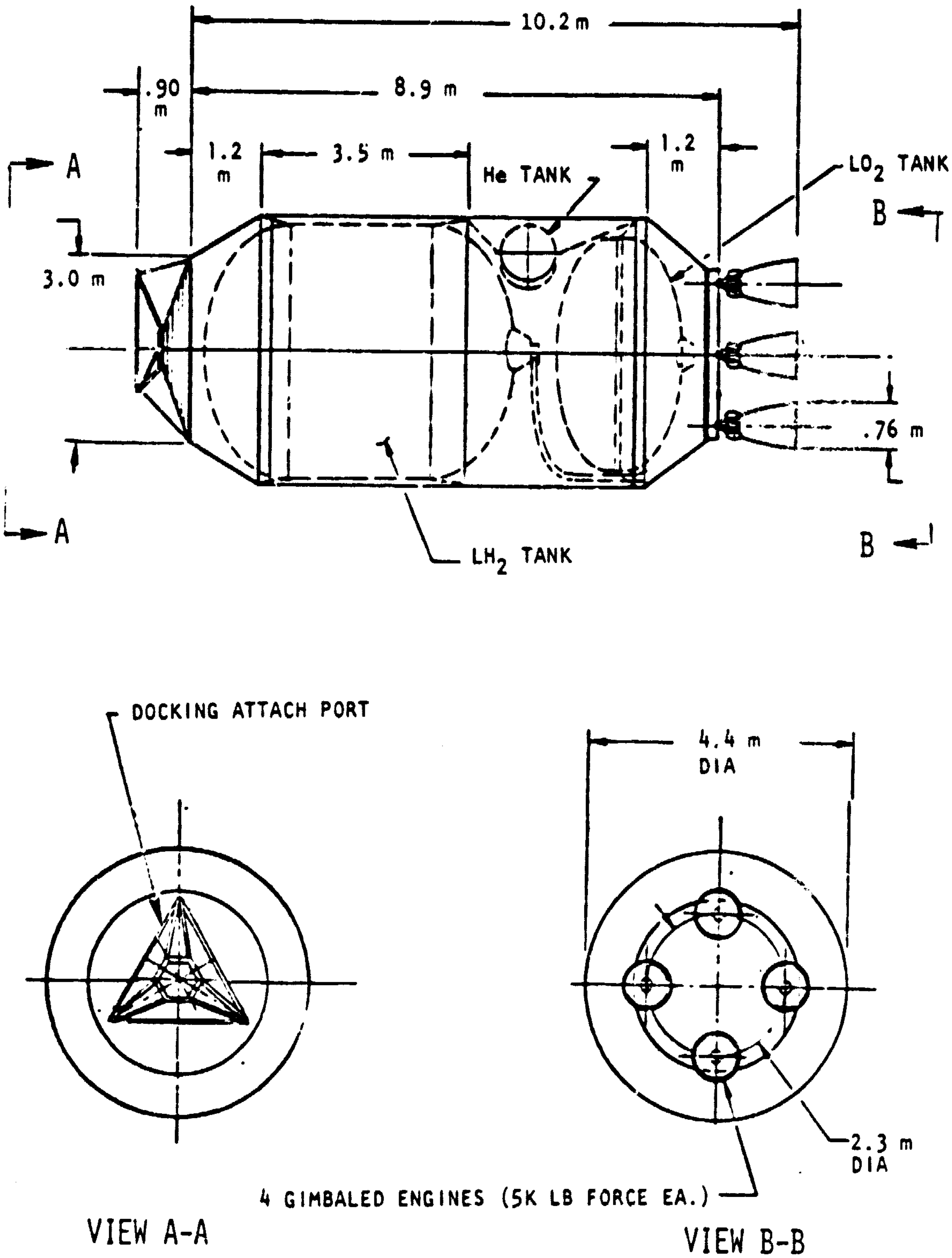


Figure 3.2.7-1. Orbit Transfer Propulsion Module

per week of on-orbit holding time. An allowance of 3% boiloff was assumed for a four-week period to transport all three modules to LEO.

3.2.7.3 Weight

The propulsion module design weight summary with maximum propellant loading is shown in Table 3.2.7-2. The inert weight includes allowances for subsystems such as structure, thermal control, avionics, propulsion, residual fluids and contingencies, based on prior studies of NASA Tug and USAF Orbit-to-Orbit Shuttle (OOS). With maximum propellant loading, the three LTP modules are capable of transporting a maximum of 41,136 kg (90,500 lb). This would include enough RCS propellant to operate 8.55 years in the Clarke orbit without servicing. The sensitivity of LTP module weight to geosynchronous orbit mission duration is illustrated in Figure 3.2.7-2.

Table 3.2.7-2. LTP Maximum Propellant Load Conditions

	kg	(lb)
Maximum gross weight	28,864	(63,500)
Maximum propellant load	25,284	(55,626)
Inert weight	3,579	(7,874)
Stage mass fraction	0.876	
3% propellant boiloff	758	(1,669)
Usable propellant (after boiloff)	24,526	(53,957)

3.2.7.4 Engine Performance

Each of the four engines include provisions for two-axis gimbaling for TVC. The engine is a staged combustion design based on the technology development of the Advanced Space Engine (ASE). The performance and size of the 22,240 N (5000 lbf) thrust engine is summarized in Table 3.2.7-3.

3.2.7.5 Propellant

Propellant storability is a requirement for the entire elapsed time from propellant tanking to burnout. The use of cryogenic propellants requires adequate insulation for tanks to minimize boiloff propellant losses. Transit times to LEO and, subsequently, to GEO are relatively short (measured in minutes and hours), so that the elapsed time that impacts boiloff the greatest is the time required in LEO to accumulate the necessary number of propulsion modules. This elapsed time may be on the order of four weeks, based on the following simplified scenario:

- A single Space Shuttle orbiter is dedicated to the construction of the platform and OTV delivery.
- The orbiter requires a two-week turnaround period between flights.
- A total of three propulsion modules is required, and this determines the orbiter flights required to transport them to LEO.

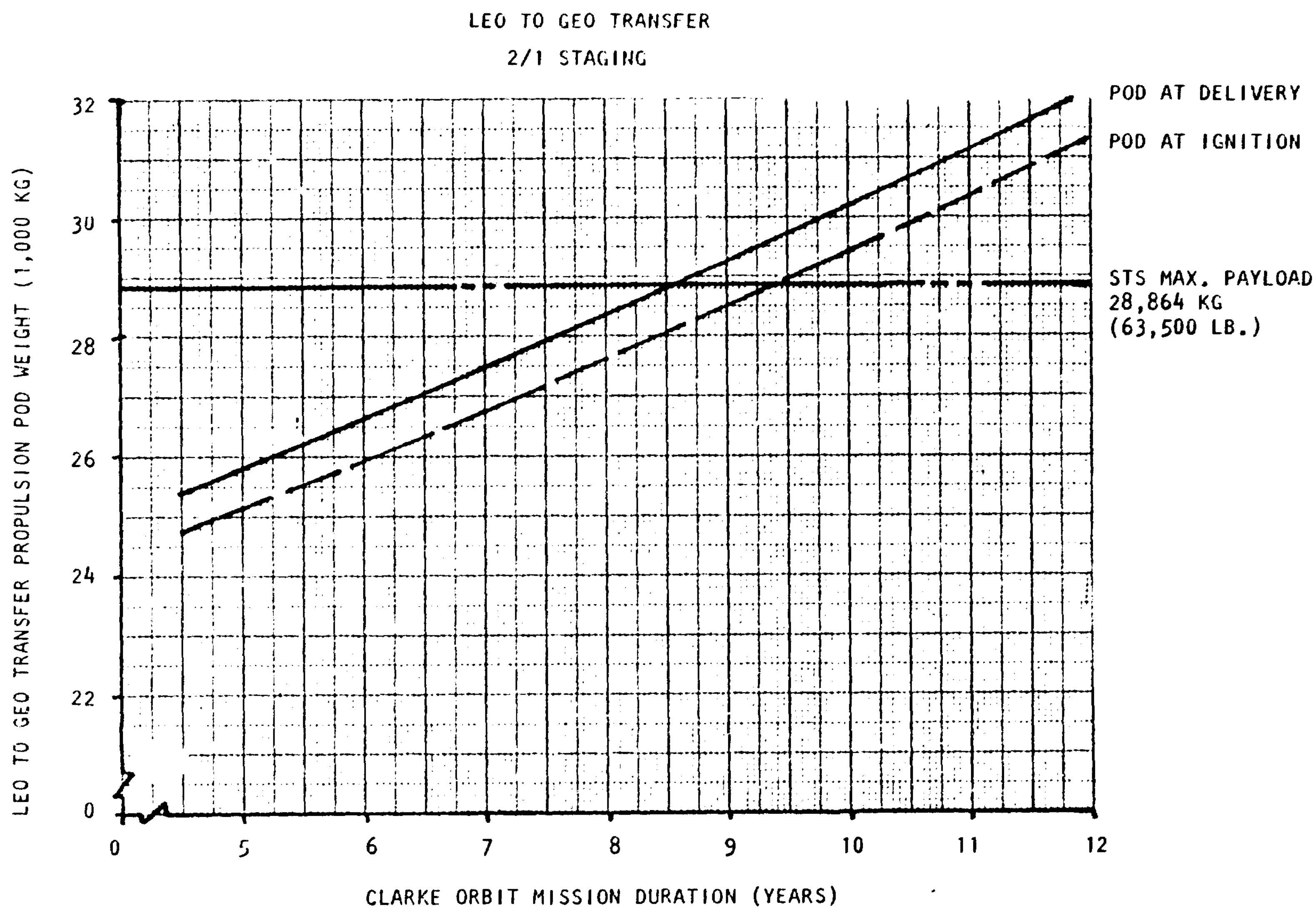


Figure 3.2.7-2. Single Propulsion POD Weight

Table 3.2.7-3. Engine Performance Summary

Thrust	22,240 N (5000 lb)
Chamber pressure	10,342 kPa (1500 psia)
Nozzle expansion area	400:1
Propellants	LO_2/LH_2
Mixture ratio, O/F	6:1
Specific impulse	4,580 N-sec/kg (467 sec)
Overall length	1.32 m (52 in.)
Nozzle exit diameter	0.76 m (30 in.)
Weight	49.9 kg (110 lb)

From this example it can be seen that the third module arrives in LEO four weeks after the first module. Propellant boiloff will be held to approximately 0.7% per week.

3.2.7.6 Staging

The required velocity increments as a function of T/W are shown in Figure 3.2.7-3. This curve shows the velocity requirements for orbit transfer from a 28.5-degree inclined LEO of approximately 370 km (200 nmi) altitude to geosynchronous equatorial orbit (GEO). For a T/W range of 0.96 to 1.96 N/kg (0.1 to 0.2 lb_f/lb_m), a ΔV of 4273 m/sec (14,020 ft/sec) is required, based on a two-impulse burn and Hohmann transfer. For the 2/1 staging sequence the first two propulsion modules will increase the spacecraft velocity by 2296 m/sec (7445 fps). The third and last propulsion module will complete the transfer orbit insertion maneuver and also deliver the ΔV required for apogee injection in the geosynchronous orbit. The ΔV for this stage is 2004 m/sec (6775 fps). It should be noted from the curve that ΔV requirements at T/W values less than 0.96 N/kg (0.1 lb_f/lb_m) increase greatly, which is due to larger gravity losses occurring with the longer burn times associated with less acceleration. However for these conditions multiple perigee burn transfer techniques may be employed to reduce the transfer velocity requirement at the expense of a modest increase in trip time. This relacionship is illustrated in Figure 3.2.7-4.

The three propulsion modules are operated in a 2-1 firing/staging sequence. Operating in this mode, the three module cluster is capable of transporting a maximum 39,210 kg (86,262 lb) of payload from LEO to GEO (after allowing 3% propellant boiloff from each module). The total firing time of each module is approximately 20 minutes at full four-engine thrust per module. In actual practice, durations slightly longer will result when paired engines are shut down to control T/W, and when sequential startup and shutdown by engine pairs are done in ten-second intervals in order to reduce the dynamic amplification of the platform structure during these thrust load transients.

The initial two modules require a single firing for the perigee burn, then are staged off at burnout, and the remaining module is fired to achieve the remaining perigee burn ΔV. A second start for the final module is then required for the apogee burn to circularize the orbit at GEO.

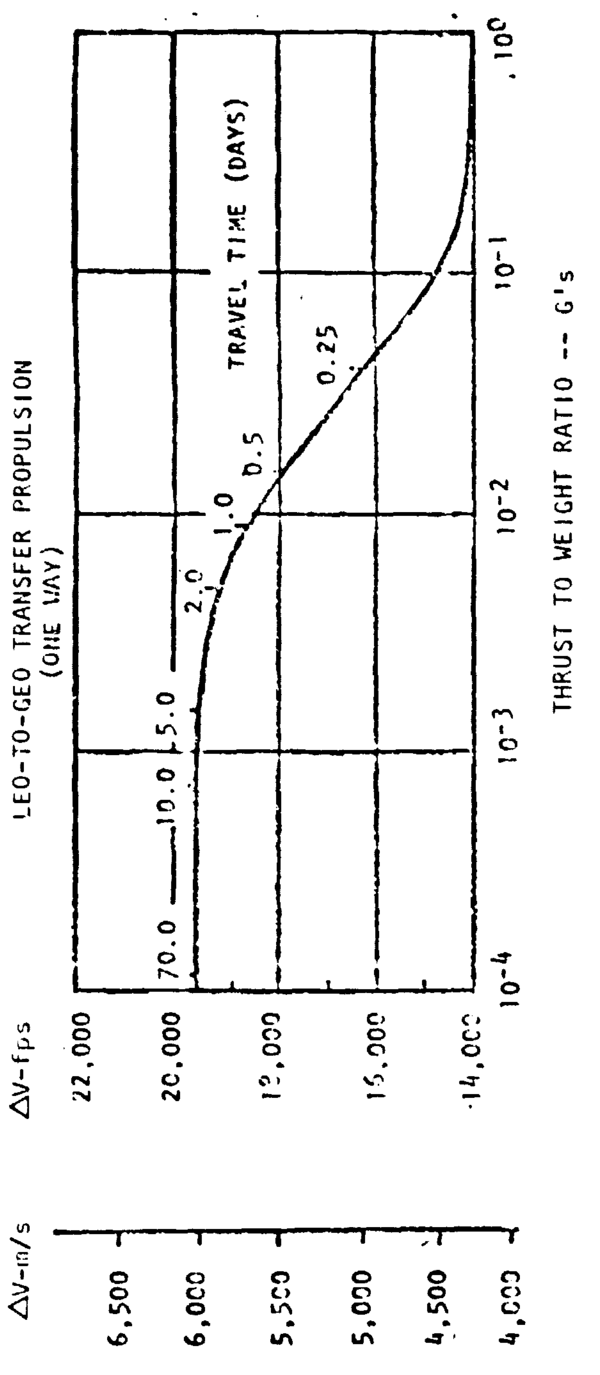


Figure 3.2.7-3. Delta-V Requirements Vs. T/W

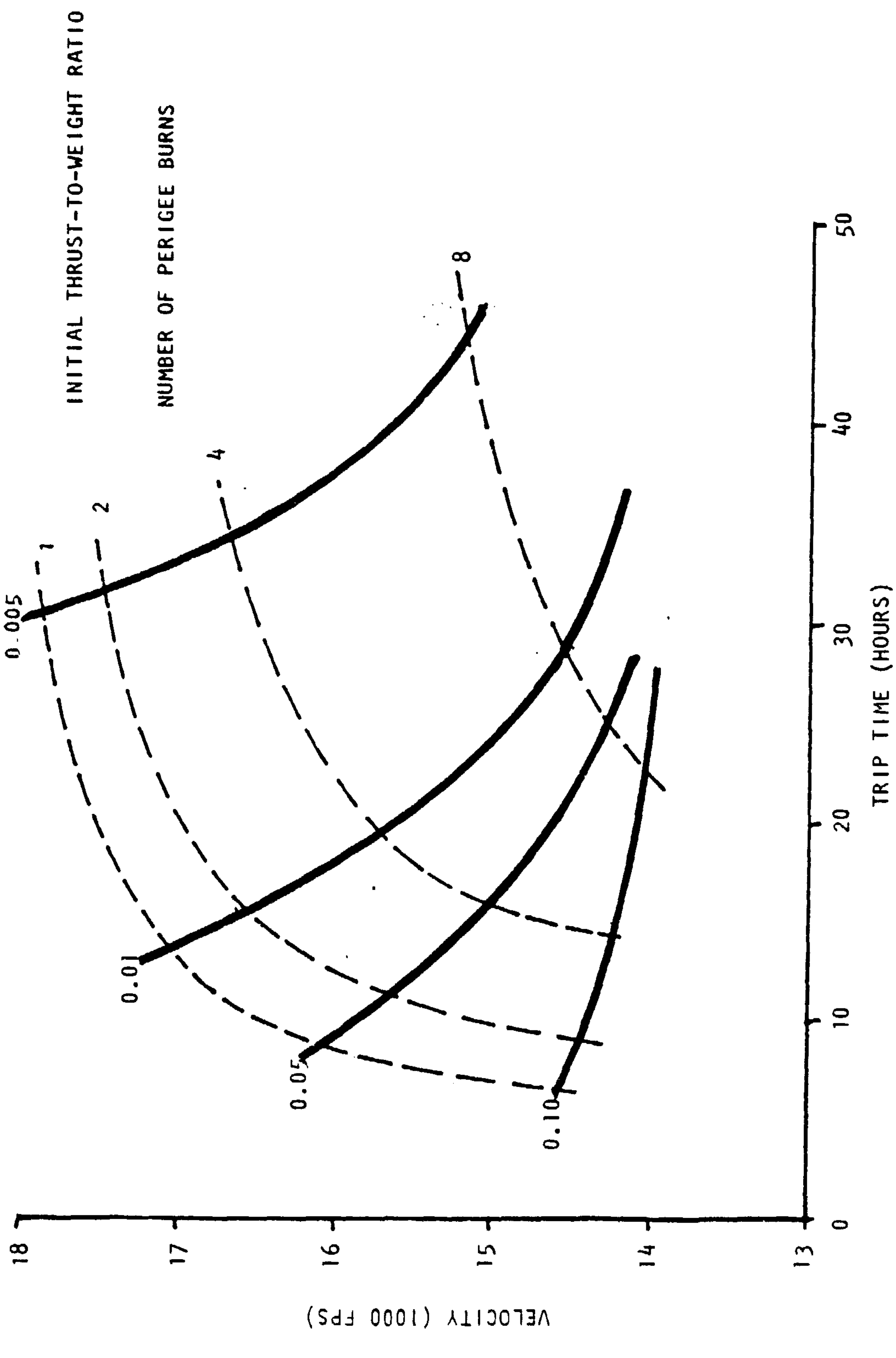


Figure 3.2.7-4. Estimated Velocity Requirements for Multi-Perigee Burn Transfer to Clarke Orbit

T/W Control

With multiple engines per module, the T/W is controlled to remain below the 1.96 N/kg (0.2 lb_f/lb_m) structural limit (space-fabricated tri-beam) by sequential shutdown of engines in pairs. For example, during the two-module burn, the initial T/W is 1.49 N/kg (0.152 lb_f/lb_m) with all engines firing (266,890 N or 40,000 lb_f thrust total) and it remains below 1.96 N/kg (0.2 lb_f/lb_m) until just prior to burnout; then, if two engines are shut down on each module the T/W would be reduced to 1.22 N/kg (0.124 lb_f/lb_m) at burnout. For the remaining single module burns made subsequently, the initial T/W is 1.35 N/kg (0.138 lb_f/lb_m) with all engines firing (88,965 N or 20,000 lb_f thrust) and it remains below 1.06 N/kg (0.2 lb_f/lb_m) for the initial half of the burn; then, two engines are shut down to bring the burnout T/W to 1.05 N/kg (0.107 lb_f/lb_m). During the orbit transfer phase, the antennae feed horn assemblies and the solar arrays are retracted.

Thrust Vector Control

Thrust vector control is provided by the gimbaled engines. During orbit transfer steering, pitch is provided about the X-axis, yaw about the Z-axis, and roll about the Y-axis. The gimbaled engines are ganged in pitch and yaw and differentially gimbaled with the outer modules for roll control. This mode of TVC with the multiple engine configuration provides control under all conditions of 2-1 module staging and paired engine operation for T/W control or structural dynamic deamplification. Multiple engines provide the necessary flexibility for meeting these varied conditions.

3.3 PAYLOAD DEFINITION

This section describes those payloads selected in Section 2.2.2 for use on the Engineering and Technology Verification Platform (ETVP). These payloads have been used to size the platform itself and to definitize various support requirements including power, mass, data rate for TT&C, pointing, alignment, and necessary stability for the platform. The four antenna types chosen are:

1. Interleave - contiguous fixed beam
2. Beam-forming matrix - contiguous fixed beam
3. Phased array - fast scanning beam
4. Noncontinguous fixed beam

Payload parameters are summarized in Figure 3.3.0-1 and Table 3.3.0-1.

3.3.1 Beam Interleaving

The interleave concept is shown in Figure 3.3.1-1. To obtain -35 to -40 dB sidelobes, a highly tapered aperture illumination of the parabolic reflector is required. The best way to obtain this illumination pattern for multiple-beam antennas is to use multi-mode horns in a densely arranged fashion. Even so, the adjacent beam spots on earth will have poor crossovers (only -13.5 dB). By interleaving patterns from three separate antennas, good crossovers (-4.5 dB) can be obtained with excellent sidelobes.

To test this concept on the ETV platform it is sufficient to use two interleaved apertures. The number of feedhorns per aperture can be reduced from 73 to a pilot test 10. By moving the horns inside the feedhorn assembly, the full 73 positions can be checked and performance verified. The feedhorn assembly is the same size as required for 73 horns.

The spot size of 0.26 degrees corresponds to 219 total beams maximum for the entire continental United States. The transmitting antennas are sized from

$$D = \frac{1.35\ \lambda}{\Delta\theta} = \frac{(1.35)\ 0.025}{0.045} = 7.5 \text{ meters}$$

where

1.35 = factor to account for large aperture weighting

λ = 0.025 meter for f = 12 GHz

$\Delta\theta$ = 0.26 degree or 4.5 mR

In like manner, the diameter for the 14-GHz receiving antennas is 6.4 meters.

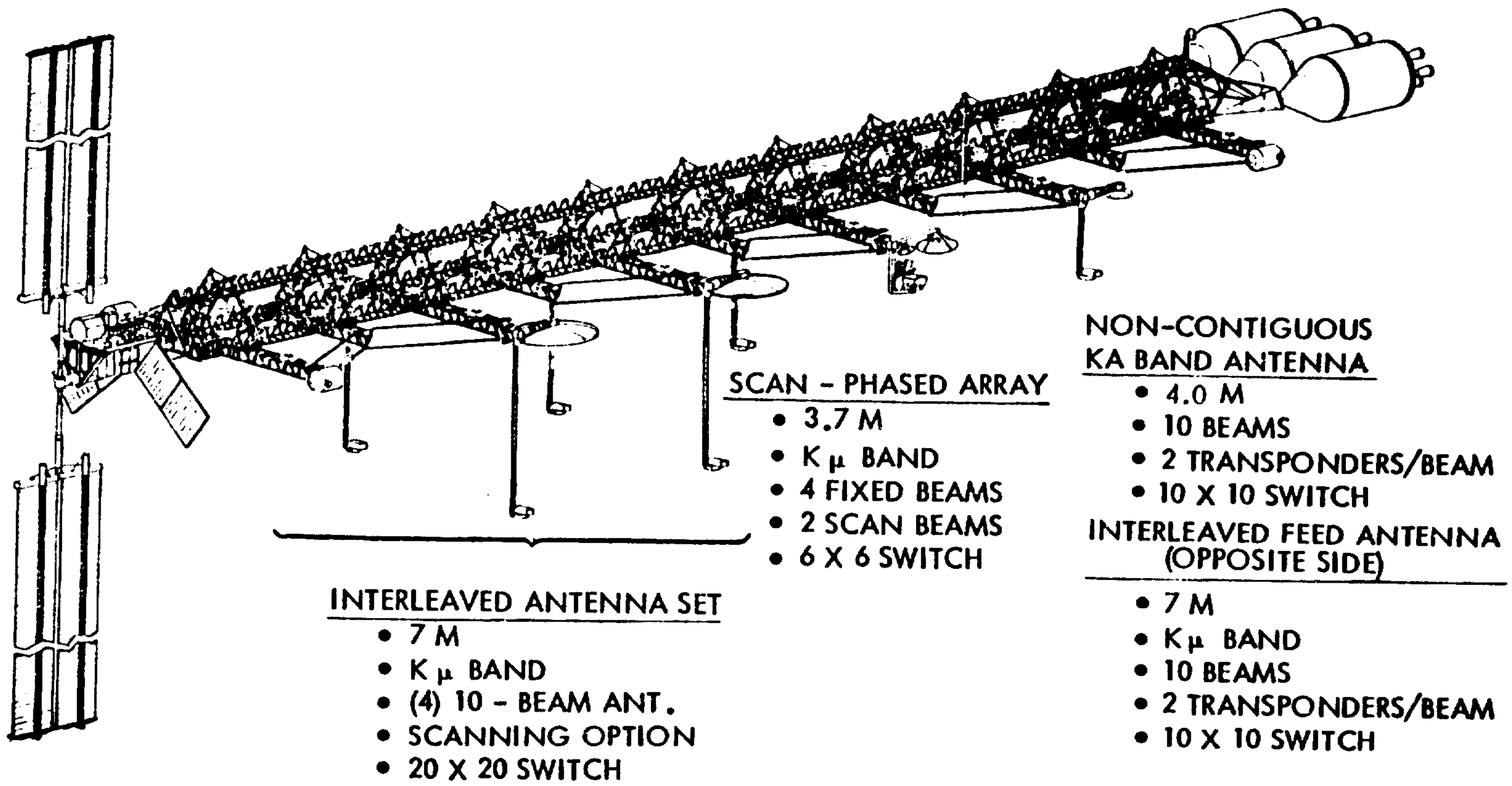


Figure 3.3.0-1. ETVP - COM (GEO) Test Version

Table 3.3.0-1. Recommended Antenna Payloads for the ETV Platform

12/14 GHz	Interleave	Scanning Phased Array	Beam-Forming Network	Non-Contiguous Beams (20/30 GHz)
Antenna	Two at 7.5 m Two at 6.4 m	One at 3.7 m	7.5 m	4 m
Feedhorns	2×10 2×10	2 scan + 4 fixed	10	10
Spot size	0.26° 0.26°	0.5°	0.26°	0.26°
Tx	2×20 (30 W)	1 (165 W) 4 (30 W)	20 (30 W)	20 (12 W)
Other				
Payload mass Payload power	2460 kg 5950 W	800 kg 1340 W	1100 kg 2980 W	1360 kg 2000 W
Total	5720 kg and 12.3 kW			

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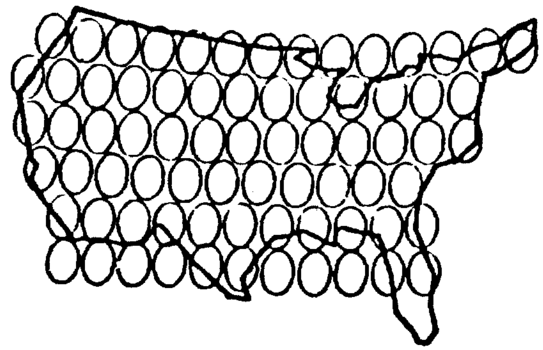
ADVANTAGES

CONTIGUOUS BEAMS
HIGH FREQUENCY REUSE
MODERATE COMPLEXITY

DISADVANTAGES

REQUIRES 3 APERTURES



REQUIREMENTS

- -35 TO -40 dB SIDELOBES
- -35 dB CROSS POLARIZATION ISOLATION

STEP 1

HIGHLY TAPERED
APERTURE ILLUM (-15 dB)

POOR CROSSOVER WITH
73 BEAMS OVER USA
(-13.5 dB)

STEP 2

INTERLEAVE BEAMS

3 IDENTICAL ANTENNAS
3 X 73 = 219 BEAMS
PRODUCES -4.5 dB CROSSOVER

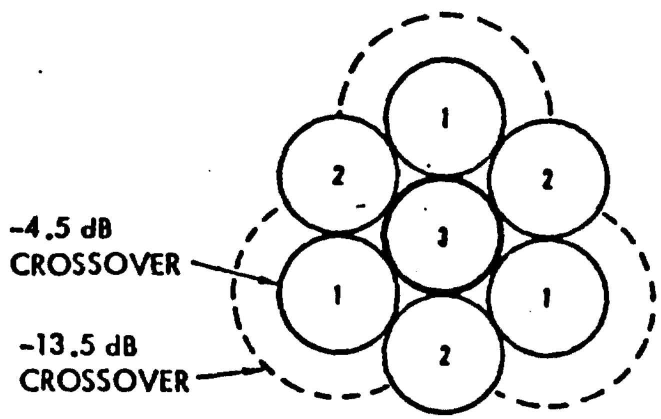


Figure 3.3.1-1. Interleave Concept

Separate transmitting and receiving antennas are used here for several reasons. First, the same spot sizes and crossover levels can be obtained with separate antennas. Also, the filtering problem is greatly simplified. It may be that the transmitters and receivers will be joined onto the same dish - only a more extensive study can make this determination.

A power calculation shows that 3 watts per carrier is quite adequate for 36 Mb/sec at 10^{-5} bit-error rate with a 2.6 m ground antenna and a 48° mask angle through 4 km (horizontal layer) of 16-mm/hour heavy rain at a range of 37,000 km at 12 GHz. However, 30 watts/carrier is used as a final value to allow full investigation of sidelobe interference since only reduced number of spots are being used.

A mass and power summary is given below:

	Mass (kg)	Power (W)
12-GHz dish (7.5 m) x 2	360	200
20 Tx (30 W) x 2	960	5000
20 Rc x 2	480	400
20x20 switch x 1	200	150
Feed x 2	320	
14-GHz dish (6.4 m) x 2	300	200
	2620	5950

Sizing the data rate for TT&C using payload characteristics is not straightforward. What has been done is to use Shuttle orbiter and TDRS TT&C data rates since these facilities will already be in place and these vehicles can also serve as relays if need be.

Platform pointing should be accurate to within one-third beamwidth or so for acquisition. Thus, 0.1 degree is needed for 0.26 degree beamwidth. Alignment follows the same argument. Stability of the platform should be that in 100 seconds the total movement of the platform is less than 0.1 degree. In this way, simple tracking loops can easily follow this rate of drift with small error.

3.3.2 Scan Phased Array

The scanning beam system has a natural advantage over multiple fixed beam systems in the area of sidelobes and crosstalk. Since TDMA is used, -15 dB sidelobes are specified if only to prevent power waste. In the system under consideration here, some fixed beams are used to increase total system capacity. Hence, -20 dB sidelobes will probably be needed. Even so, this is a decided advantage over other payload configurations. Another great advantage the scanning beam inherently enjoys is that it serves a non-uniform traffic density much better than others. In fact, it adapts quickly to immediate changes in traffic. The areas of difficulty arise in the fields of switching, phase shifter tolerance, and buffering.

The scanning beam must be swept in synchronism with a similar time-division format of the fixed beams so that interconnections between all beams are possible. Efficient access in this system is aided by buffering. As the number of stations in the system grows, the complexity grows also.

The two-scanning beam and four fixed beams concept is shown in Figure 3.3.2-1. This pilot test program should be adequate to test the concept and provide a means of comparison to other systems. The two scanning beams (horizontally polarized) are pointed by means of 4-bit pin diode phase shifters. The four fixed beams (vertically polarized) are formed by means of conical feedhorns. The same aperture is used for all beams by using a polarization selective subreflector. The phased scanned beams are separated by a frequency diplexing grid and also use a subreflector to properly shape the beam. So, rather than being limited by sidelobes, this system tends to be limited by the degree of polarization isolation achievable.

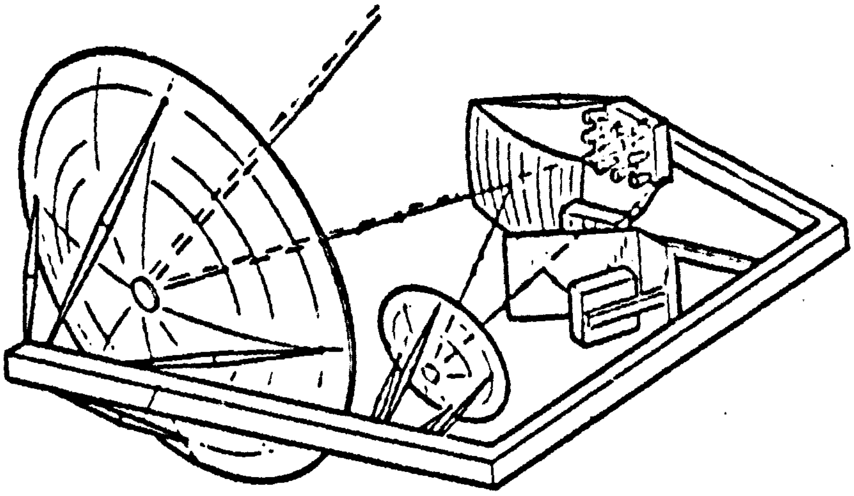
The antenna is sized to give a 0.5-degree beamwidth, or 3.7-m aperture at 12/14 GHz. Not as small a spot is required as in the previous concept, since more instantaneous bandwidth is available per spot. The antenna is sized, rather, to reduce the power of the transmitters to a reasonable value.

A mass and power summary is given below:

	Mass (kg)	Power (W)
12/14 GHz dish and drive	170	50
Subdish	50	
One-half reflector	50	
Four-horn assembly	40	
Scanning array	40	
One 200 Mb/s Tx (165 W)	215	660
Four 36 Mb/s Tx (4x30 W)	96	500
One 200 Mb/s Rc	20	30
Four 36 Mb/s Rc	48	40
6x6 switch and buffer	71	60
Feed struct. and gimbal	6	
	806	1340

The scanning system is less demanding on the platform than the other payloads in terms of pointing and alignment since (1) the beams are larger, (2) there are fewer beams, and (3) the scanning array easily acquires and remembers the proper phase for pointing. Once per hour or so, checking of alignment is performed to correct platform drift. The platform should drift less than one third of beamwidth per hour, or

$$\frac{\Delta\theta}{\Delta t} = \pm\frac{0.50\ \text{deg}}{3\ (\text{hr})} = \pm 0.17\ \text{deg/hr}$$



ADVANTAGES

- SCANNING BEAM EFFICIENTLY COVERS WIDE AREA
- FIXED BEAMS COVER HIGH USAGE POINTS: SINCE NOT CONTIGUOUS, SIDELOBE CONTROL EASIER

DISADVANTAGES

- PARTIAL LOSS OF ONE POLARIZATION
- FAST SWITCHING PROBLEMS
- DEPLOYMENT/ALIGNMENT

Figure 3.3.2-1. Scanning and Fixed Beam Concept

If this is too severe, the drift must be used by the scanning system to update the phase shifters or else drift corrections must be performed more often.

3.3.3 Beam-Forming Network

The beam-forming network (BFN) is a method of forming multiple beams by use of various hybrid-coupler configurations. A repetitive progression produces a pattern as typified in Figure 3.3.3-1. However, the pattern can easily be reconfigured into another pattern, as illustrated in Figure 3.3.3-2, to contour the antenna areas in accordance with traffic density. This adaptability is an important advantage for the BFN. The disadvantage is more loss and more complexity in the feed network. Otherwise, there is little difference between the BFN and the interleave method from a systems point of view.

The antenna is sized at 7 m for about 0.26-degree spot size as an average for 12 and 14 GHz. Both transmit and receive channels utilize the same antenna. This complicates the filtering process somewhat, but the BFN method promises good sidelobes and the system results may prove (or disprove) the combination feasible for expanded operation. The power calculations are similar to those for the interleave system. If large area spot combinations are used, then more power is needed since larger areas mean less gain. For the pilot test, ten area spots, 0.26 degree each, will be used so a close comparison to the interleave method can be made.

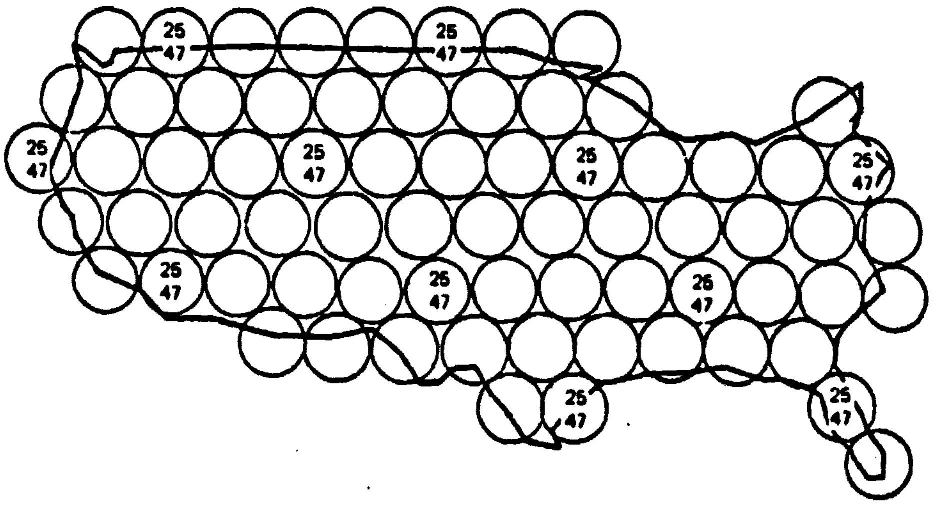
The power and mass summary is given below:

	Mass (kg)	Power (W)
12/14-GHz dish (7 m)	180	200
20 Tx (30 W)	480	2500
20 Rc	240	200
10x10 switch	100	80
Feed	100	
Feed struct. and gimbal	53	
	1153	2980

The pointing, alignment, and stability requirements are the same as the interleave case.

3.3.4 Non-Contiguous Beams

The non-continguous beam antenna operates in the 20/30-GHz band as opposed to the other payloads which operate in 12/14 GHz. The purpose for including this payload is not to check out the antenna operation, but to verify (1) system operation at 20/30-GHz during light rain, and (2) the switching of 20/30-GHz traffic to 12/14 GHz during heavy rain. The heavy rain attenuation at 20 and 30 GHz is so severe that a lower frequency band is needed as backup. The verification of communication system operation of the 20/30 and 12/14 GHz combination is one of the most important missions of the ETV platform.

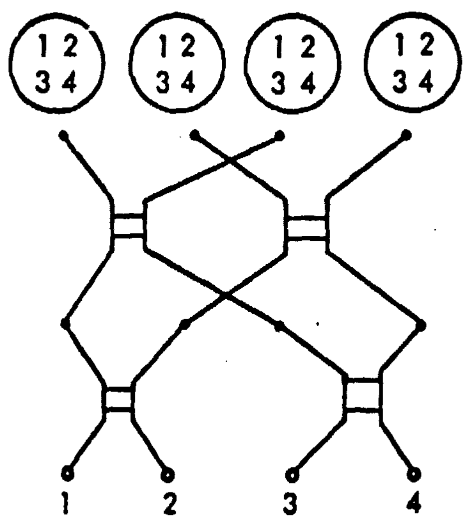


ADVANTAGES

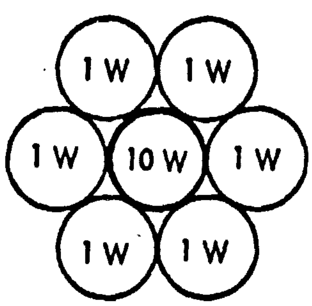
CONTIGUOUS BEAMS
HIGH FREQUENCY REUSE
REQUIRES ONE APERTURE
TOLERANCE REDUCTION

DISADVANTAGES

HIGH COMPLEXITY



BUTLER MATRIX



REDUCTION IN
SIDELOBES

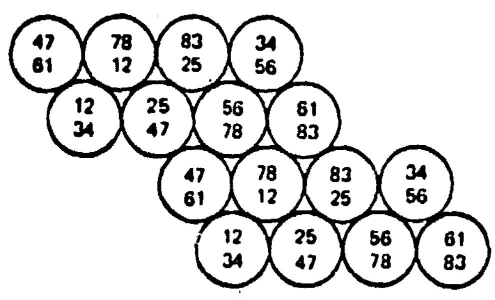


Figure 3.3.3-1. Beam-Forming Network Concept

C-3

EACH SECTOR HAS COMPARABLE TRAFFIC

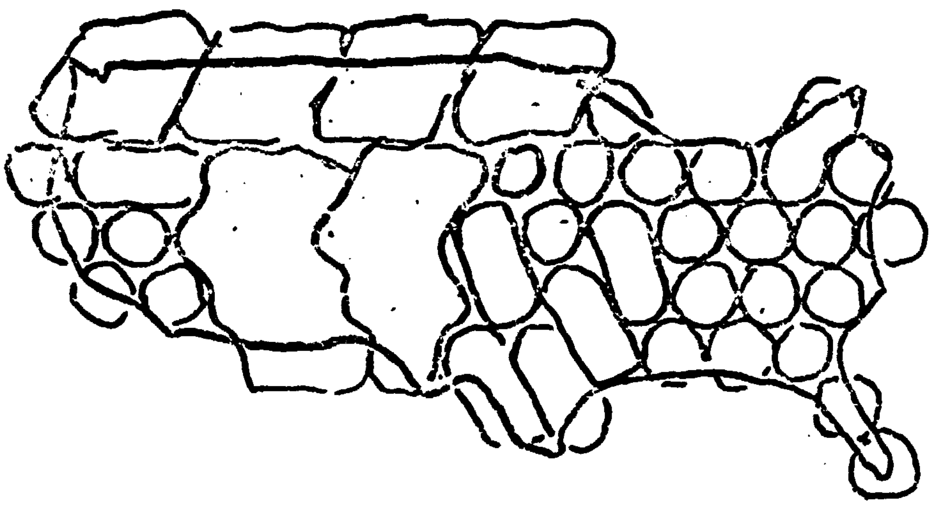


Figure 3.3.3-2. B.F.N. Allows Beam Shaping

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Since only 20 to 40 beams are envisioned for a full 20/30-GHz system, and contiguous beams are not needed, the pilot system proposed for testing on the ETVP utilizes one antenna for transmit and receive. Sidelobes should not present a serious problem. The antenna is sized at 4 m to give 0.26-degree beamwidth and about 56 dB gain at 20 GHz. Using 12 W per carrier, 36 Mb/s data rate with a 3.5-m ground antenna through 4 km (horizontal) of 16 mm/hr rain, gives satisfactory operation, theoretically. This calculation includes 10 dB for rain margin. The amount of allowance needed for rain is one of the test features for this payload. Operation under actual conditions will greatly reduce the uncertainty of this area.

A mass and power summary is given below:

	Mass (kg)	Power (W)
20/30-GHz dish (4 m)	240	150
20 Tx (12 W)	480	1440
20 Rc (36 Mb/s)	240	200
10x10 switch	100	80
Feed struct. and gimbal	53	
Bandwidth and cables to 12/14 GHz	300	130
	1413	2000

Pointing, alignment, and stability requirements are similar to the interleave system.

3.4 MASS PROPERTIES

Table 3.4.0-1 presents the mass summary statement for the platform configuration under study.

Rationale for Analysis

Basic structure masses were based on structures analysis and sizing.

The docking port masses and the rotary joint mass were calculated from layouts. The systems control module mass is based on nominal specific unit weights.

The solar panel mass was based on the standard Lockheed panel of 0.752 m by 4 m dimension with a mass of 3.147 kg. For the antenna platform configurations, additional deployment and canisters were required. The remainder of the electrical power and distribution mass was based on analyses and requirements.

The attitude control mass, consisting of CMG's and the RCS, was based on requirements reflected by the satellite mass properties and mission requirements.

The mass of the TT&C and thermal control system are estimates based on prior studies. The mass of the microwave and communication systems was based on scaling algorithms. Propellant mass was based on satellite mass properties and mission requirements.

Table 3.4.0-1. Mass Summary

ITEM	MASS (KG)
PLATFORM ARRAY (LESS P/L)	(5313)
STRUCTURE & MECHANISM	(4219)
FABRICATED BEAM, PRIMARY STRUCTURE	1252
CORD ASSEMBLY	15
STRUTS, ARRAY (24)	262
INTERSECT FITTINGS (66)	90
ATTACH PORTS (28)	2389
BRIDGE STRUCT. (SOLAR ARRAY END) (16)	34
BRIDGE THRUST STRUCTURE	147
MECH—THRUST STRUCTURE	20
MECH—FWD MOD. TRUSS STRUCTURE	10
ATTITUDE CONTROL	(78)
GUIDANCE, NAVIGATION & CONTROL	14
MAGNETOMETER	1
WIRE HARNESS	13
SUPPORT	NEG
SUN SENSOR	NEG
RENDEZVOUS & DOCKING AIDS (16)	64
TT&C	(111)
Ku-BAND	111
ANTENNA DISH (3-FT DIA) (1)	13
ELECTRONIC ASSY (1)	44
SIGNAL PROCESSOR (2)	45
STRUCTURAL HOUSING & MOUNT	9
ELECTRICAL POWER & DISTRIBUTION	(905)
SWITCH GEAR	8
CONDUCTOR—POWER	234
COAX & TSP (C&DM)	19
DC/DC CONVERTER/REGULATOR	109
DC/AC CONVERTER/REGULATOR	219
INSTALLATION	316
SYSTEM CONTROL MODULE	(8282)
STRUCTURE & MECHANISM	(2252)
PRIMARY STRUCTURE	(2134)
EQUIPMENT SUPPORT CRUCIFORM	1000
SUPPORT TRUNNION STRUCTURE (2)	1134
ATTACH PORT (1)	(68)
RENDEZVOUS, DOCK & BERTHING AIDS (TELEOP.)	(50)

Table 3.4.0-1. Mass Summary (Cont.)

ITEM	MASS (KG)
SYSTEM CONTROL MODULE (CONT.)	
ATTITUDE CONTROL	(1134)
GUIDANCE, NAVIGATION & CONTROL	(1134)
PRECISION ATT. REF. PACKAGE	(35)
INERTIAL REF. UNIT (1)	17
STAR TRACKER (2)	18
BACKUP ATT. REF.	(17)
INERTIAL REF. UNIT (1)	17
COMPUTER (2)	(26)
CONTROL MOMENT GYRO (CMG) (3)	(1056)
TT&C	(91)
S-BAND	(91)
PM TRANSPONDER (2)	16
PM PROCESSOR (2)	8
DOPPLER EXTRACTOR (1)	7
POWER AMPLIFIER (1)	14
PREAMPLIFIER (1)	12
FM TRANSMITTER & ANTENNA (2)	6
FM PROCESSOR (1)	5
SWITCH ASSY (1)	3
INSTALLATION	20
THERMAL CONTROL	(751)
RADIATOR (2) (200 FT^2 EA. SIDE)	294
RADIATOR DEPLOYMENT SYSTEM	30
FLUID LOOP	234
COLDPLATES (76)	193
ELECTRICAL POWER & DISTRIBUTION	(4054)
BATTERIES	3430
SWITCH GEAR, BATTERY (4)	2
BATTERY RACK	100
BATTERY CHARGERS, ELECTRONICS, SENSORS, HEATERS	198
DC/DC CONVERTER/REGULATORS	83
DC/AC CONVERTER/REGULATORS	165
CONDUCTOR	16
INSTALLATION	60
ROTARY JOINT ASSEMBLY	(439)
STRUCTURAL HOUSING SOLAR ARRAY DRIVE SHAFT MECHANISMS ACTUATING SYSTEM	308
ADAPTER, ROTARY TO S/A (2)	52
WIRE HARNESS	31
SLIP RINGS	48

Table 3.4.0-1. Mass Summary (Cont.)

ITEM	MASS (KG)
SOLAR ARRAY ASSEMBLY	(1226)
SOLAR ARRAY, 2 PEP SYSTEMS (4 PEP WINGS)	1224
SWITCH GEAR	2
REACTION CONTROL SYSTEM MODULE (DRY)	(2721)
MODULE (4)	2268
ATTACH PORT (4)	453
TOTAL—SATELLITE (DRY) LESS P/L & OTV	17,981
RCS PROPELLANT (7-YR SUPPLY	5,784
TOTAL—SATELLITE (WET) LESS P/L & OTV	23,765
PAYLOAD—TYPICAL COMMUNICATION	(6650)
INTERLEAVE	(2996)
7.5-m DISH ANTENNA (2) 12 GHz	360
TRANSMITTER (40)	960
RECEIVER (20)	480
SWITCH (20)	200
FEED	160
6.4-m DISH ANTENNA (2), 14 GHz	300
BERTHING PORTS (4)	374
FEED STRUCT. & GIMBAL, 7.5 m (2)	105
FEED STRUCTURE & GIMBAL, 6.4 m (2)	57
SCANNING PHASED ARRAY	(900)
3.7-m DISH ANTENNA (1) 12/14 GHz	170
SUBDISH	50
1/2 REFLECTOR	50
HORN ASSY (4)	40
SCANNING ARRAY	40
TRANSMITTER (1) 165 W	215
TRANSMITTER (4) 30 W EA	96
RECEIVER (1)	20
RECEIVER (4)	48
SWITCH & BUFFER	71
BERTHING PORT (1)	94
FEED STRUCT. & GIMBAL (1)	6
BEAM FORMING NETWORK	(1247)
7.5-m DISH ANTENNA (1) 12/14 GHz	180
TRANSMITTER (20)	480
RECEIVER (20)	240
SWITCH	100
FEED	100
BERTHING PORT (1)	94
FEED STRUCT. & GIMBAL	53

Table 3.4.0-1. Mass Summary (Cont.)

ITEM	MASS (KG)
PAYLOAD (CONT.)	
NON-CONTIGUOUS BEAMS	(1207)
4-m DISH ANTENNA (1)	240
TRANSMITTER (20)	480
RECEIVER (20)	240
SWITCH	100
BERTHING PORT (1)	94
FEED STRUCT. & GIMBAL	53
BAND SWITCH & CABLES TO 12/14 GHz	(300)
TOTAL—SATELLITE (WET) LESS OTV (GEO)	30,415
ORBIT TRANSFER PROPULSION (INERT) (3)	10,204
TOTAL—SATELLITE, GEO BURNOUT	40,619
OTV PROPELLANT (3 MOTORS)	69,687
TOTAL SATELLITE, LEO (INITIAL)	110,306

APPENDIX A

SYMMETRIC VERSUS ASYMMETRIC SOLAR ARRAY CONFIGURATION

SYMMETRIC VS. ASYMMETRIC SOLAR ARRAY CONFIGURATION

In the pursuit of this trade, comparative information was generated for each of six basic issues. Expedient "problem models" and/or assumptions were made in each analysis area to limit the scope to just the depth necessary for identifying distinguishing differences. The main objective was to make an intelligent solar array concept selection from which the ETVP preliminary design activity could proceed.

SELECTION SUMMARY

Briefly, the asymmetric configuration concept was selected because it offers (1) comparative simplicities in its design and construction, (2) the inherent design and servicing advantages of consolidated subsystems at a single location, and (3) the potential for additional payload accommodations on the open end which could provide wide unobstructed viewing. The symmetric case, while offering the advantages of downsized attitude control/CMG's, does pose design complexities for integrating the OTV thrust structure with the solar array rotary joint/mounting structure. Proximity to the propulsion vibration environment may also pose additional hazards. Thus, the asymmetric solar array configuration concept was selected.

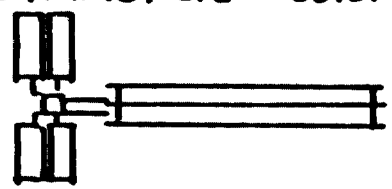
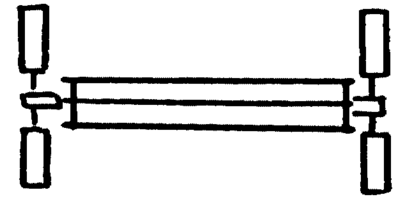
The six basic issues and their key comparative factors are summarized in Figure 1. Brief discussions of the individual issues highlighting the basic evaluation logic and the resulting important differences are presented in the accompanying paragraphs. The actual analysis packages from which these summary materials were derived are contained in the following enclosures:

- Enclosure (1) Attitude Control Analysis
- Enclosure (2) Configuration Analysis
- Enclosure (3) Construction Impact Analysis
- Enclosure (4) Power Distribution Impact Analysis

SYSTEM WEIGHTS IMPACT

The main factor affecting system weight is the lower disturbance torques inherent with the symmetric solar array configuration. In GEO, solar pressure torques are the dominant disturbance (for the asymmetric configuration). These could be reduced by 75% (or more, if careful attention is given toward payload and subsystem location) for the symmetric configuration. This would allow the use of smaller CMG's and reduced RCS propellants for CMG desaturation. The combination of these effects, based on data scaled from the advanced communications platform synthesized in Part I of the study, could be as much as 10 to 15% of the ATP system weight, but would likely fall into the 5% range. Reduced ATP system weight would also allow a corresponding reduction in orbit transfer propulsion weight. However, the magnitude of these potential weight reductions will not reduce the number of Shuttle flights required for assembly and propulsion delivery (a weight

FIGURE 1 COMPARISON SUMMARY

ISSUES \ CONFIGURATION OPTIONS	ASYMMETRIC CONFIG 	SYMMETRIC CONFIG 
• SYSTEM WT & ORBIT XFER PROPULSION REQ		✓ UP TO 10% LOWER SYST WT
• ELEC POWER REQ		✓ SMALL POWER REDUCTION (SMALLER CMGS)
• CONSTRUCTION IMPACTS	✓ SINGLE INSTALLATION CONSOLIDATED PLATFORM SUBSYSTEMS	
• COMPLEXITY OF ELEC POWER DISTRIBUTION	✓ SMALL ADVANTAGE WITH "TAPERED" LOADS	
• CARRYOVER OF PEP HARDWARE	✓ EQUAL CARRYOVER	✓ EQUAL CARRYOVER
• IMPACT ON LOGISTICS & COSTS	(NO CLEAR ADVANTAGE)	(NO CLEAR ADVANTAGE)

reduction of 25-30% would be required to assure a reduction in the number of Shuttle flights). Thus, the main advantage with the symmetrical configuration is the ability to use a smaller CMG system.

POWER REQUIREMENTS

The main effect on electrical power requirements is related to the reduced CMG's above. The ability to use smaller CMG's with the symmetric configuration means less power would be needed to operate the attitude control subsystem. The delta power is estimated to be as much as 500 to 1000 watts, but is not highly significant in a system sized to about 60 kW overall. The power requirements issue represents a small "plus" for the symmetric configuration.

CONSTRUCTION IMPACTS

The asymmetric configuration offers significant simplicities in construction over the symmetric concept. The power generation system installation occurs all at one location and can be performed in a smooth uninterrupted sequence. All major electrical power system elements can be handled, installed, and checked out in an efficient work pattern. With the symmetric configuration, power system elements must be installed at both ends of the platform. This implies, at the least, an extreme translation process to move the construction fixture to the opposite end of the platform. This time-consuming interruption in the EPS installation process may be further prolonged by sandwiching in other construction tasks along the way. Also, in the symmetric configuration case, a more complex thrust structure will likely be required to integrate the rotary joint/solar array drive system into the structural arrangement. This would add to the construction/installation complexity for the second solar array. Thus, the asymmetric configuration offers significant design and construction advantages over the symmetric concept.

COMPLEXITY OF POWER DISTRIBUTION

If full redundancy is assumed for the power distribution system, there are no basic differences between distribution networks for the symmetric and asymmetric configurations. Similar parallel power buses would be required along the length of the platform for both cases. This is especially true where the same level of service—say, 5 or 10 kW—is to be provided at all payload/user stations. In the event that user loads could be tapered along the bus (high loads at one end, reduced to low loads at the other end), the line sizes could be reduced in steps along the way. This would slightly favor the asymmetric case where all power is generated at one end. Thus, from a power distribution standpoint, the symmetric vs. asymmetric configurations is essentially a toss-up with, maybe, a very small "plus" favoring the asymmetric concept because of the tapered load possibility.

CARRYOVER OF PEP HARDWARE

No differences are apparent between the symmetric and asymmetric concepts in their ability to directly utilize PEP solar array hardware. Both concepts can be configured around the use of four solar arrays, canisters, and mast deployment mechanisms. The asymmetric configuration offers the possibility of "ganged" deployment design, but can equally well use the individual deployment scheme which would be employed with the symmetric arrangement. Thus, neither concept is favored from the standpoint of PEP hardware carryover.

IMPACT ON LOGISTICS AND COSTS

Logistics and costs impacts are judged to be a "toss-up". Reduced costs and logistics due to down-sized CMG's and RCS propellants with the symmetric configuration could easily be offset by cost increases for design and integration of the more complex thrust structure/rotary joint needed with this configuration and possibly a longer construction process. Thus, within the problem model definition for this task, no clearly definable cost/logistics superiority could be identified with either concept.

ENCLOSURE (1) ATTITUDE CONTROL ANALYSIS

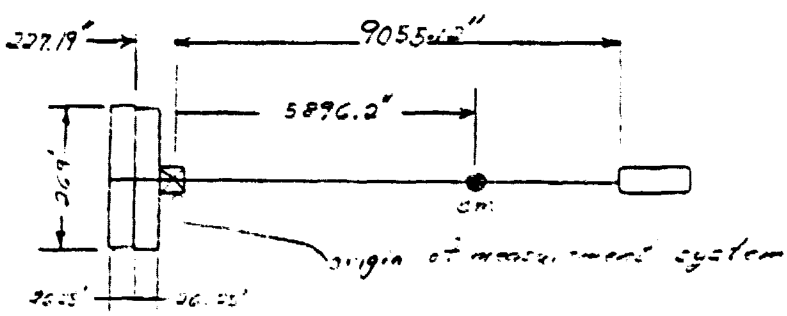
R. Abramson

ATTITUDE CONTROL ANALYSIS

SUBJECT: Balanced Solar Panels for the Advanced Technology Communications Platforms

A study was performed on the advanced technology communications platforms to determine the benefits in reduction of attitude control system weight and power of moving one-half the solar panels to the opposite end of the platform. The major environmental disturbance on the platform at geosynchronous altitude is solar pressure. This is due to the long lever arm between the solar panels and the platform center of mass (CM) in the asymmetric design.

The geometry and mass properties data used in the study are given in the figure below.



Mass of platform = 4125.44 slugs (132,732 lb)

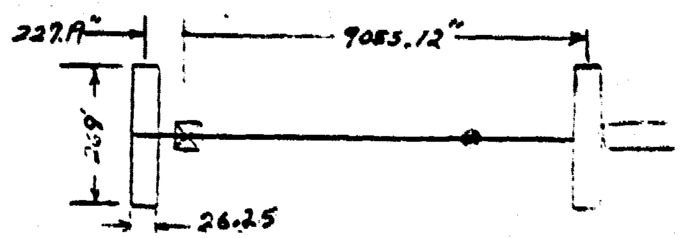
$$I_{XX_{CM}} = 404.60\times10^{6} \text{ sl-ft}^2$$

$$I_{YY_{CM}} = 3.476\times10^{6} \text{ sl-ft}^2$$

$$I_{ZZ_{CM}} = 405.03\times10^{6} \text{ sl-ft}^2$$

Mass of total solar array = 93.2 slugs (2998.3 lb).

The new configuration is given below:



The first task is to determine the new CM of the platform.

CM of platform without solar panels

$$\ell = \frac{(132,732)(5896.2)+(2998.3)(227.19)}{129,733.71} = 6037.72 \text{ in.}$$

CM of platform with balanced solar panels

$$\ell_{CM} = \frac{(129,733.71)(6037.72)+(1499.15)(9055.12)-(1499.15)(227.19)}{132,732} = 6001.04 \text{ in.}$$

The solar pressure torque on the balanced configuration is determined as follows:

$$T_{SP_{\text{BASIC VEHICLE}}} = F_{SP}(2A)(5896.2+227.19) = 12,246.78\ A\,F_{SP}$$

$$T_{SP_{\text{BALANCED VEHICLE}}} = F_{SP}[A(6001.04+227.19)-A(6001.04-9055.12)] = 3,174.15\ A\,F_{SP}$$

Therefore, the torque due to solar pressure on the balanced platform is 0.26 (26%) that of the basic platform. The configuration change results in the following momentum buildups:

Secular	Roll/Yaw	= 5800×0.26 = 1508 N-m-sec/hr
Secular	Pitch	No change since X and Z CM do not change
Cyclic	Pitch	No change

The other environmental disturbance on the platform is gravity gradient. The largest gravity gradient torque is the cyclic pitch torque. The gravity-gradient torques are a direct function of the difference of the off-axis inertias. Therefore, the change in I_{XX} and I_{ZZ} will be determined. The change in geometry does not affect the pitch axis inertia.

The panels are assumed to lie in the YZ plane of the platform. The $m\ell^2$ term of the complete solar array =

$$93.2' \times (\underset{12}{\underline{5896.2+227.19}}/\cancel{12})^2 = 24.27\times10^6 \text{ sl-ft}^2$$

The I_X of the solar array = $93.2\ (\underset{12}{\underline{52.5^2+269^2}}/\cancel{12}) = 0.583\times10^6$ sl-ft²

The I_X of one solar panel = $46.6\ (\underset{12}{\underline{26.25^2}} + 269^2/\cancel{12}) = 0.284\times10^6$ sl-ft²

I_{XX} of ATCP without solar panels about original CM

$$404.6\times10^6 - 24.27\times10^6 - 0.58\times10^6 = 379.75\times10^6 \text{ sl-ft}^2$$

I_{XX} of ATCP with balanced solar panels about original CM

$$379.75\times10^6 + 2(0.284\times10^6) + 46.6\ (5896.2+227.2/12)^2 + 46.6(263.24) = 395.68\times10^6 \text{ sl-ft}^2$$

I_{XX} of ATCP with balanced solar panels about new CM

$$I_{XX} = 395.68\times10^6 - 4125.44\ (5896.2-6037.72/12)^2 = 395.11\times10^6 \text{ sl-ft}^2$$

The I_Z of the solar array = $93.2\ (52.5^2/12) = 21{,}405$ sl-ft²

The I_Z of one solar panel = $46.6\ (26.25^2/12) = 2{,}676$ sl-ft²

I_{ZZ} of ATCP without solar panels about original CM

$$405.03\times10^6 - 21{,}405 - 24.27\times10^6 = 380.74\times10^6 \text{ sl-ft}^2$$

I_{ZZ} of ATCP with balanced solar panels about original CM

$$380.74\times10^6 + 2(2676)+46.6(5996.2+227.19/12)^2 + 46.6(263.24)^2 = 396.11\times10^6 \text{ sl-ft}^2$$

I_{ZZ} of ATCP with balanced solar panels about new CM

$$396.11\times10^6 - 4125.44(5396.2-6037.72/12)^2 = 395.54\times10^6 \text{ sl-ft}^2$$

The ratio of old inertia differences to new are

about X-axis $\dfrac{405.03\times10^6 - 3.476\times10^6}{395.54\times10^6 - 3.476\times10^6} = 1.02$

about Y-axis $\dfrac{404.6\times10^6 - 405.03\times10^6}{395.11\times10^6 - 395.54\times10^6} = 1.00$

The change in inertia differences is not significant, so that significant changes in gravity-gradient torques cannot be expected and the same secular and periodic momentum buildups can be expected with either configuration.

With the new or balanced configuration, the CMG sizing is as follows, assuming a 12-hr desaturation interval:

Pitch axis—no change

$H_Y = 67\times12+2020 = 2824$ N-m-sec/12 hr

Roll/yaw axis

$H_{X/Z} = (130+1508)\times12+20 = 19{,}676$ N-m-sec/12 hr

The RSS of these two momentum values is 19,878 N-m-sec/12 hr. This is 28% of the old configuration momentum value of 71,180 N-m-sec/12 hr (a reduction of 72%). A configuration of seven Skylab CMG's of 3118 N-m-sec (2300 ft-lb-sec) would accomplish this task. The CMG package would weigh $467\times7 = 3269$ lb.

The electrical power consumed by the CMG package would be (steady state) about $7\times150 = 1050$ watts.

The RCS system for desaturation is required for only secular momentum control. Assuming that stationkeeping does not aid in momentum dump, the RCS propellant required per 12 hours is computed to be, assuming an $I_{SP} = 2746$ N-kg/sec (280 sec),

Roll/Yaw = $(1508+130)\times12)/(230/2\times2746) = 0.0622$ kg every 12 hours.

The original configuration would use

Roll/Yaw = $(5930\times12)/(230/2\times2746) = 0.2254$ kg every 12 hours.

The ratio is 0.28, or a savings of 72%.

There is no difference in the pitch axis; the RCS propellant required is:

Pitch = $(67\times12/(24.2/2\times9746) = 0.0242$ kg every 12 hours.

For a year's operation, the balanced configuration would require 63 kg to the original configuration's 182.2 kg—a savings of 65%.

If the ATCP could be configured so that the roll/yaw secular solar pressure torque is eliminated, the control system size would reduce to the following:

Roll/Yaw momentum = 1580 N-m-sec/12 hr

Pitch momentum = 2164 N-m-sec/12 hr

Total momentum required = 2,679.4 N-m-sec.

This momentum requirement could be met by one Skylab CMG. At least two Skylab CMG's are required for three-axis control, or smaller CMG's could be used such as three Sperry CMG's of 1356 N-m-sec. Three of these units would weigh 525 lb and use and require about 48 watts of running power.

Propellant requirements for desaturation are

Roll/Yaw: 0.0050 kg/12 hr

Pitch: 0.0242 kg/12 hr

which results in 21.4 kg/year.

The original Task 1 study assumed that one half of the RCS attitude control propellants could be saved by judicious firings of the thrusters to combine momentum desaturation with the once-per-day stationkeeping maneuvers. Therefore, the propellant requirements computed in this analysis could be halved using the same assumptions.

If a balanced configuration is assumed, the use of an all-RCS system (no CMG's) might become feasible. The cyclic gravity torques are at twice orbit frequency, while the solar pressures are at orbit frequency.

Roll/yaw per day = $(130\times24+4\times20)/(230/2\times2746) = 10.14\times10^{-3}$ kg/day

Pitch per day = $(67\times24+4\times1900+120\times2)/(24.2/2\times2746) = 0.28$ kg/day

For a seven-year period this amounts to 742 kg (1634 lb) which is about 3.2 times the weight of the CMG system.

The Task 1 study allotted 2.8×10^6 N-sec for attitude control over a seven-year period solely dedicated to CMG desaturation. Another 2.8×10^6 N-sec were picked up by the stationkeeping maneuver, which gives 5.6×10^6 N-sec. The breakdown is 4.4×10^6 N-sec for roll/yaw and 1.2×10^6 N-sec for pitch.

The weight breakdown is 1602.33 kg for roll/yaw and 437 kg for pitch, or a total of 2039.33 kg. The yearly allocation is 291.33 kg (642.28 lb) which is greater by a factor of 3.2 over the calculations stated herein. The difference is not known at this time, but could be in the value of specific impulse or lever arms used.

ENCLOSURE (2) CONFIGURATION ANALYSIS

R. Hart

CONFIGURATION ANALYSIS

The following five figures summarize important options and implications associated with symmetric and asymmetric color array installations.

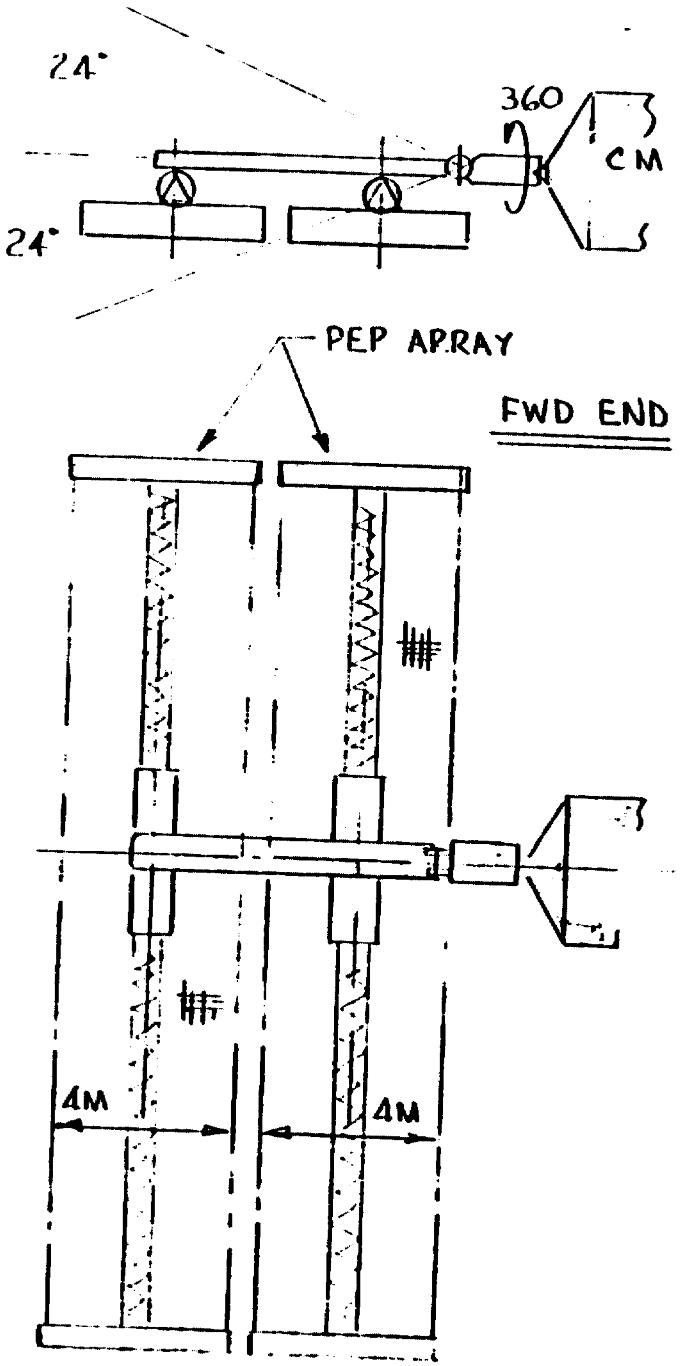
Figure 2-1 shows an asymmetric configuration concept in which the standard PEP arrays and their deployment masts are mounted to a common support boom. This concept offers the potential for virtually off-the-shelf use of PEP hardware.

Figure 2-2 also shows an asymmetric configuration concept, but reflects an option which has significantly less potential for direct use of PEP hardware. It uses the standard PEP solar arrays, but utilizes single deployment masts to extend the two PEP arrays making up each "wing." Thus, the mast deployment mechanism would be nonstandard.

Figure 2-3 shows the mounting concept for the aft array installation of a symmetric configuration. This concept requires a longer apex beam (platform structural modification) in order to provide clearance between the orbit transfer propulsion module and the solar array.

Figure 2-4 shows basically the same installation design as in Figure 2-3, but without the extended apex beam. In this case, the capability for safe fly-away must be added to the propulsion module. With the propulsion module gone, the solar array can be deployed with adequate clearance envelopes.

Figure 2-5 shows a third option for the symmetric solar array configuration. In this arrangement, the rotary joint is modified to accommodate a thrust load path. This eliminates the need for adding a fly-away capability in the propulsion module.



ASYMMETRIC ARRAY CONFIGURATION

- TWO PEP ARRAYS MOUNTED ON A COMMON BOOM AT THE FWD END OF THE PLATFORM
- UNBALANCED FOR SOLAR PRESSURE
- PEP ARRAYS 99% "OFF SHELF" (HIGH HARDWARE CARRYOVER)
- REDUNDANCY
  - FOUR INDEPENDENT ASTROMASTS
  - ALL POWER FROM SOLAR ARRAYS TRAVELS THROUGH ONE ROTARY JOINT

Figure 2-1. Standard PEP Arrays and Deployment Masts

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ASYMMETRIC ARRAY CONFIGURATION

- 2 PM ARRAYS MOUNTED ON A COMMON BOOM AT FWD END
- UNBALANCED FOR SOLAR PRESSURE
- PM ARRAYS TO BE DEVELOPED
- REDUNDANCY
  - ONE ASTROMAST FOR TWO ARRAYS
  - ALL POWER FROM SOLAR ARRAYS TRAVELS THROUGH ONE ROTARY JOINT

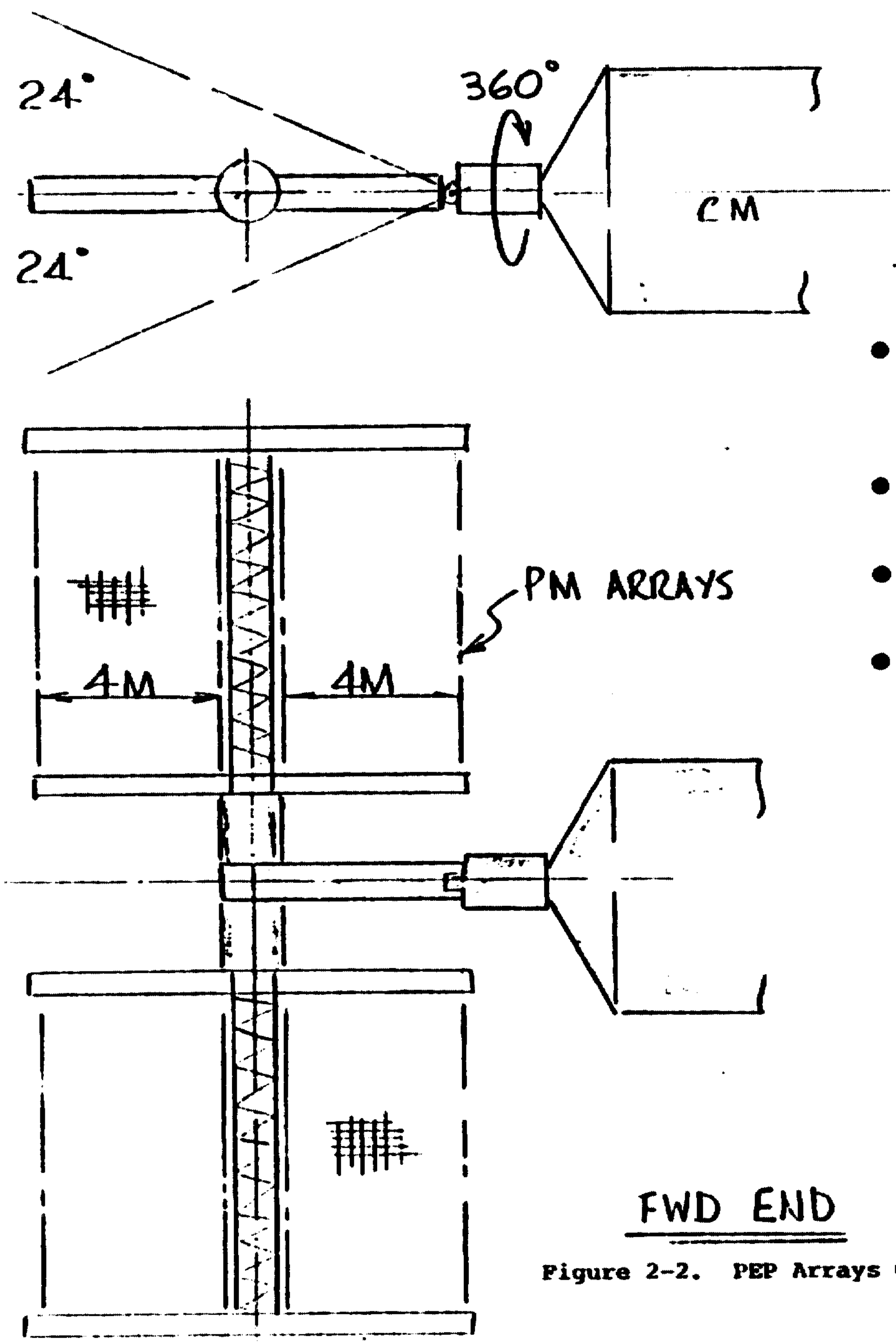
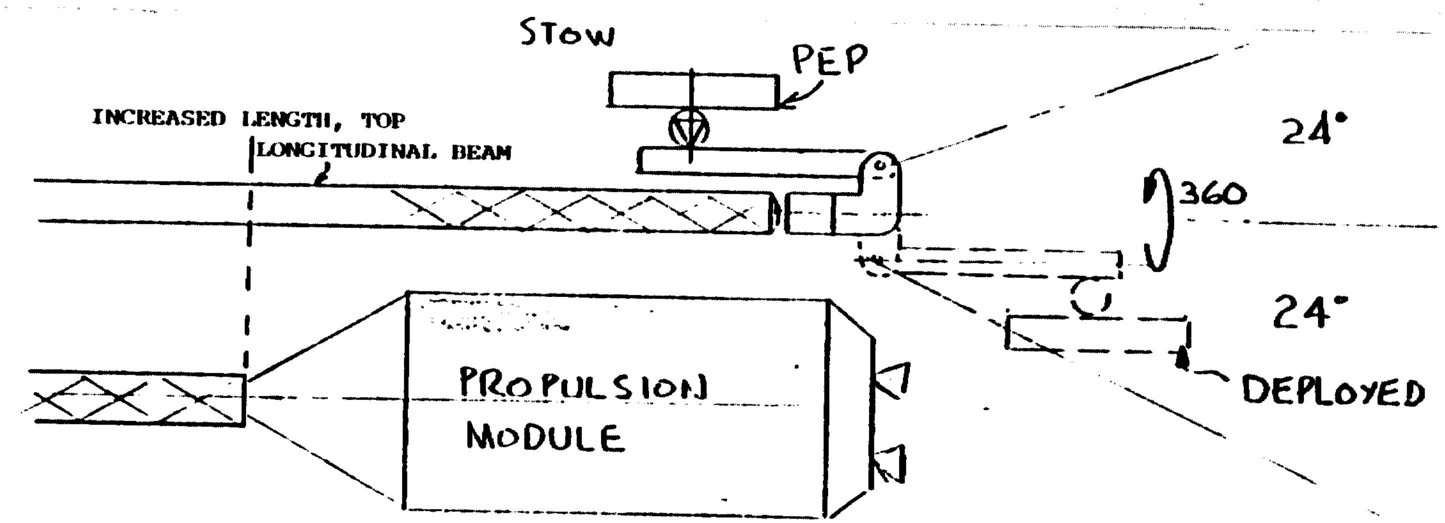


Figure 2-2. PEP Arrays Configured for Common Mast Deployment

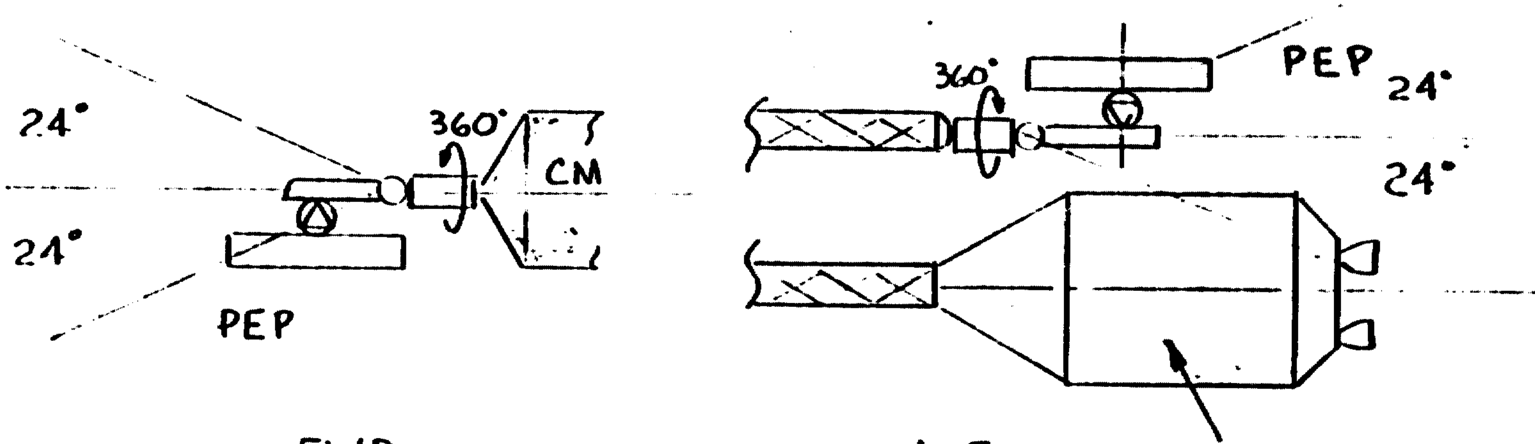


AFT END

SYMMETRIC ARRAY CONFIGURATION

- PEP AT EACH END OF PLATFORM
- BALANCED FOR SOLAR PRESSURE
- JETTISON 2 OUTBOARD PROPULSION MODULES DURING ORBIT TRANSFER
- RETAIN CENTER PROPULSION MODULE
  - NO REQUIREMENT FOR FLYAWAY PACKAGE
- AFT PEP ARRAY STOWED FOR ORBIT TRANSFER, DEPLOYED FOR OPERATIONS
- COMPLETE REDUNDANCY OF SOLAR ARRAYS
- REQUIRES 2 DESIGNS (DIFF. EACH END) FOR ROTATION JOINTS AND ARRAY MOUNTING
- 99% "OFF SHELF" PEP ARRAYS

Figure 2-3. Beam Extension Mounting Concept



SYMMETRIC ARRAY CONFIGURATION

- PEP AT EACH END OF PLATFORM (IDENTICAL MECH, BOOM, ETC.)
- BALANCED FOR SOLAR PRESSURE
- JETTISON 2 OUTBOARD PROPULSION MODULES DURING ORBIT TRANSFER
- JETTISON CENTER PROPULSION MODULE AFTER ACHIEVING GEO
  - REQUIRES CONTROLLED FLYAWAY PACKAGE
- COMPLETE REDUNDANCY OF SOLAR ARRAYS
- 99% "OFF SHELF" PEP ARRAYS (HIGH HARDWARE CARRYOVER)

Figure 2-4. Common PEP Mounting Concept

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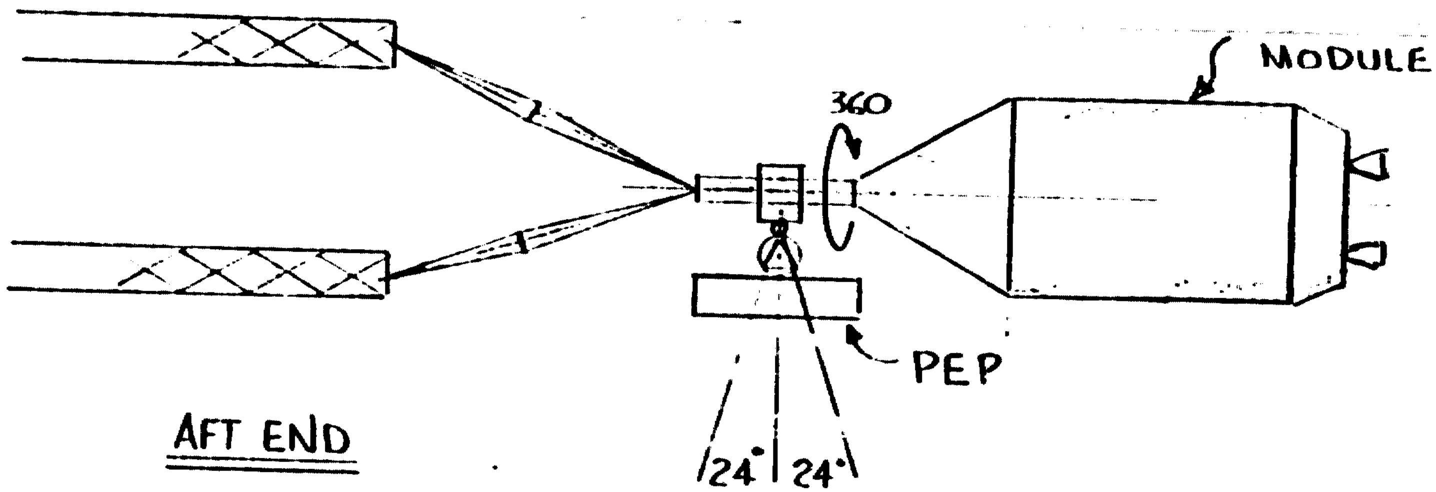


AFT END

SYMMETRIC ARRAY CONFIGURATION

- PEP ARRAY AT EACH END OF PLATFORM
- BALANCED FOR SOLAR PRESSURE
- JETTISON 2 OUTBOARD PROPULSION MODULES DURING ORBIT TRANSFER
- RETAIN CENTER PROPULSION MODULE
  - NO REQUIREMENT FOR FLYAWAY PACKAGE
- COMPLETE REDUNDANCY OF SOLAR ARRAYS
- REQUIRES 2 DESIGNS FOR ROTATING JOINTS & ARRAY MOUNTING
- 99% "OFF SHELF" PEP ARRAYS

Figure 2-5. Modified Rotary Joint PEP Mounting Concept

ENCLOSURE (3) CONSTRUCTION IMPACT ANALYSIS

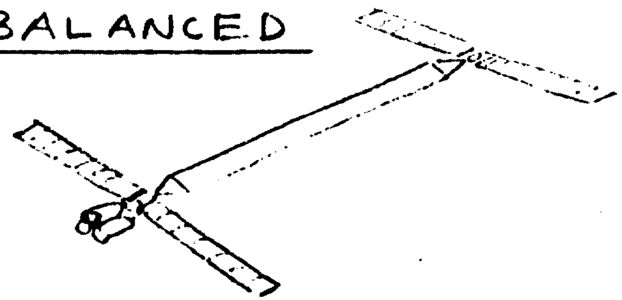
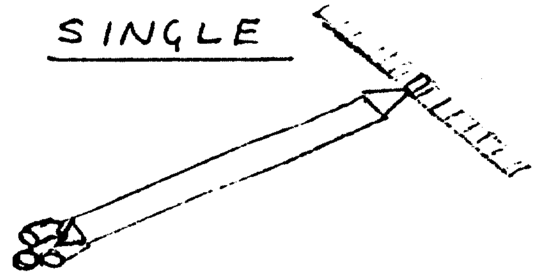
J. Roebuck

CONSTRUCTION IMPACT ANALYSIS

The major considerations relating to construction operations and resulting timelines are the duplications of effort needed to install the balanced solar array system. As shown in Figure 3-1 for the balanced array there is an unavoidable additional set of activities involving installation of two sets of solar arrays, two slip ring rotary joints and associated electrical power distribution wiring, switches, j-boxes and related equipment. Also there are two major installations to checkout. If the solar array system installation is substantially different at the propulsion end of the ATP (as compared to the opposite end), there may be added complications of construction handling, lighting, special TV viewing devices, as well as crew training.

The interactions of logistics with installation sequence are not clear at this time. However; there is a strong possibility that both the solar arrays will be installed near the end of the construction project, in close serial order. Thus, the balanced array may require an additional translation of the construction fixture from one end of the structure to the other.

FIGURE 3-1 BALANCED VS SINGLE SOLAR ARRAY CONSTRUCTION OPERATIONS/TIME

BALANCED	SINGLE
	

BALANCED

- DUPLICATES PROCESSES
  - INSTALLATION OF ROTARY JOINTS
  - INSTALLATION OF ARRAYS
  - CHECKOUT
- ADDS COMPLEXITY TO ELECT. DISTR. SYSTEM
- ADDS TRANSLATION TIME & CLEARANCE NEAR END OF CONSTRUCTION

SINGLE

- MINIMAL INSTALLATION PROCESSES & CHECKOUT
- SIMPLE ELECTR. PWR DISTR. SYSTEM
- SEQUENCE FOR MINIMAL TRANSLATION POSSIBLE (TBD)

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ENCLOSURE (4) POWER DISTRIBUTION SYSTEM ANALYSIS

Pieter de Jong

POWER DISTRIBUTION SYSTEM ANALYSIS

Topic: Solar Arrays located at both ends of the platform vs one S/A at one end, i.e., balanced vs unbalanced power sources.

Power Distribution and Switching issues affected are as listed on the Figure 4-1. Figures 4-2 and 4-3 show the main elements of an EPS system concept suited to the Applications Technology Platform.

The possible reduction in wiring weight and/or losses depends on location of the loads. One that basis the reduction can range from 0 to 50%.

Smaller power sources to be switched would be of minor impact where switching would be performed in stages due to limitations of available switch gear.

The requirement to switch power at both ends of the platform is considered a slight disadvantage of the balanced system because of remoteness (from the control module) of half of the power cource switch gear.

Other elements of power distribution and switching are anticipated to remain unchanged whether balanced or unbalanced power sources are employed.

In summation, there are no strong arguments in favor or against a balanced solar array system with regard to considerations of power switching and distribution.

FIG 4-1 POWER DISTRIBUTION AND SWITCHING

WITH BALANCED S/A CONCEPT, IS AFFECTED AS FOLLOWS:

- POSSIBLE REDUCTION IN:
  - WIRING WEIGHT
  - POWER LOSSES
- INCREASED REDUNDANCY IN SWITCHING OF POWER SOURCES
- SMALLER POWER SOURCES TO BE SWITCHED
- ABILITY TO SWITCH POWER SOURCES IS REQUIRED AT BOTH ENDS OF BUS RATHER THAN AT ONE END.

FIG 4-2 BALANCED SOLAR ARRAY POWER DISTRIBUTION CONCEPT

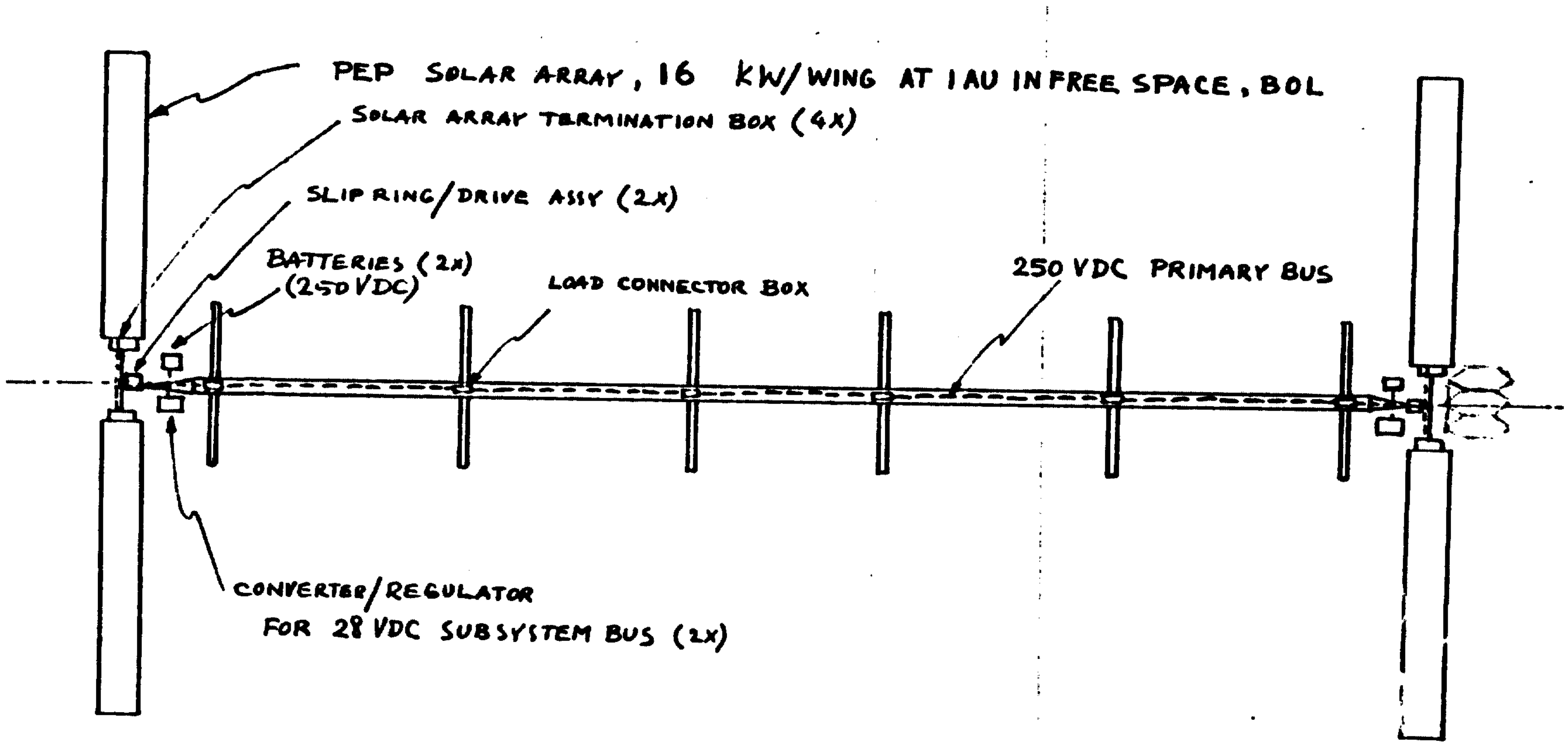
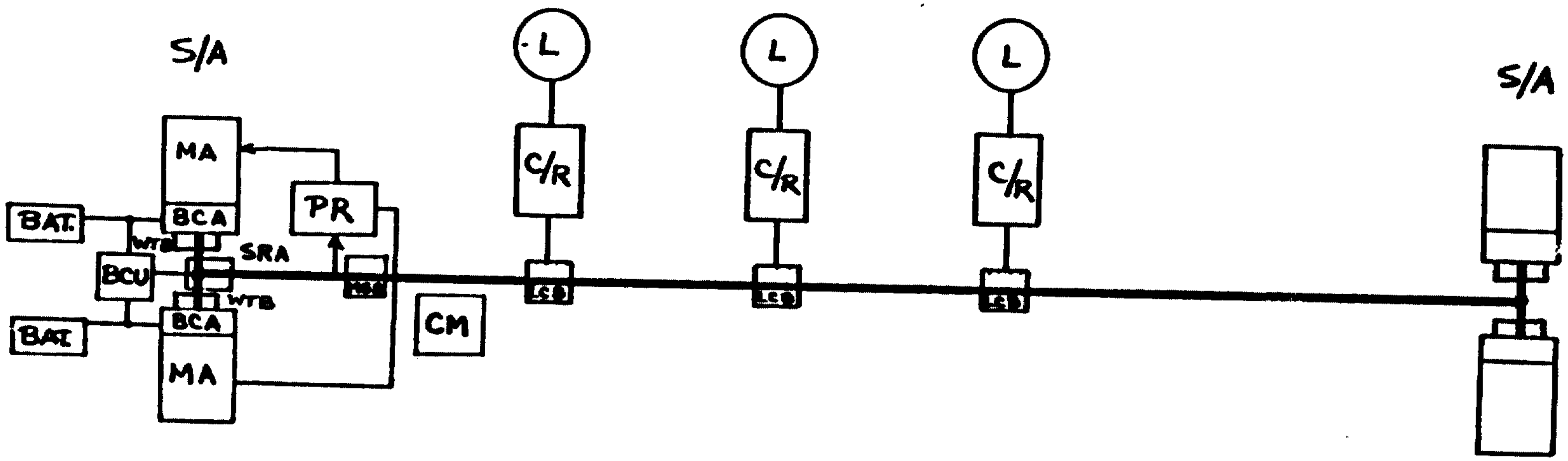


FIG 4-3 PRIMARY POWER BUS

EPS IS SYMMETRIC : S/A's, BATTERIES, REGULATORS, ETC. ARE LOCATED AT EACH END OF PLATFORM.



OUTSTANDING FEATURES

TWO S/A'S : BALANCED SYSTEM
ELECTRICALLY AND DYNAMICALLY
REDUNDANCY
REDUCED DISTRIBUTION WIRING LOSSES
BATTERY CHARGE BY SWITCHABLE ARRAYS

C/R	CONVERTER/REGULATOR SOLID STATE DEVELOPED FOR MMS AND OTHERS	SRA	SLIP RING ASSY
		WTB	WING TERMINATION BOX
CM	CONTROL MODULE.	MSB	MAIN SWITCH BOX

S/A	SOLAR ARRAY
MA	MAIN ARRAY
BCA	BATTERY CHARGE ARRAY
PR	PRE-REGULATOR, BY SWITCHING PORTIONS OF MA
LCB	LOAD CONTROL BOX HI VAC. PWR RELAYS SUCH AS ITT/JENNINGS WITH CURRENT SENSING TO CUT OFF OVER LOADS.
BCU	BATT. CHARGE UNIT-SELECTS CURRENT BY SWITCHING.

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APPENDIX B

IN-PLANE VERSUS STAGGERED BEAM ETVP CONFIGURATION TRADE

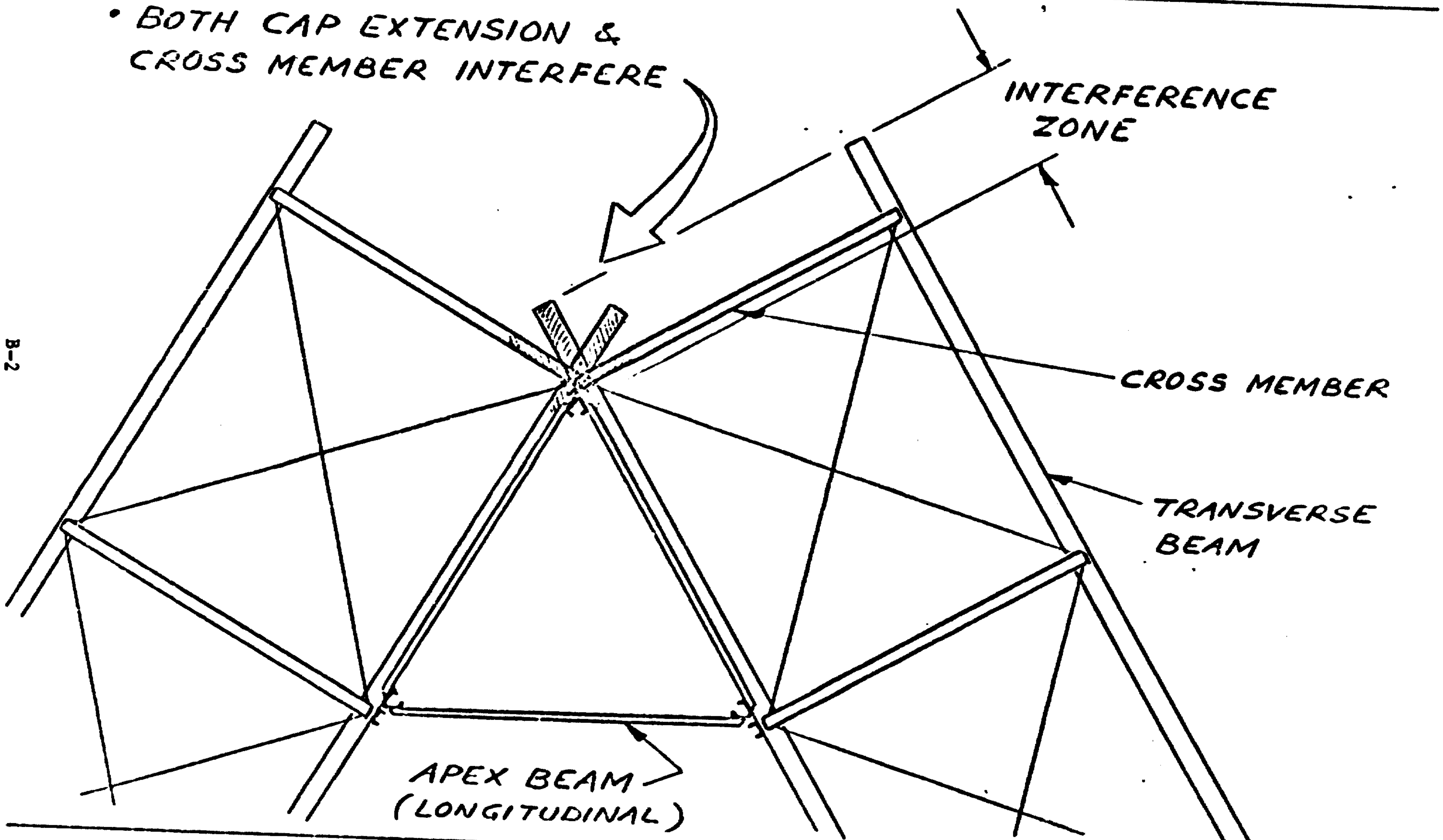
In-Plane Versus Staggered Beam ATP Configuration Trade

This trade was initiated to determine the characteristics and implications of an ATP structure configuration with in-plane mounted transverse members and of a structure configuration with staggered mounted transverse members. Earlier studies revealed an interference condition with the transverse members when utilized in a tri-beam arrangement. Figure 1 illustrates the interference. Two solutions to this condition were identified; (1) stagger the members, or (2) modify the end configuration of the transverse members. This trade addressed the characteristics and implications to the ATP for each of two solutions.

The enclosed briefing discusses the issues of structural capability, intersection configurations illustrating three variations of the revised beam ends, construction and construction strategy, systems installation, solar panel installations for the SPS Test Article arrangement, safety, and growth/legacy.

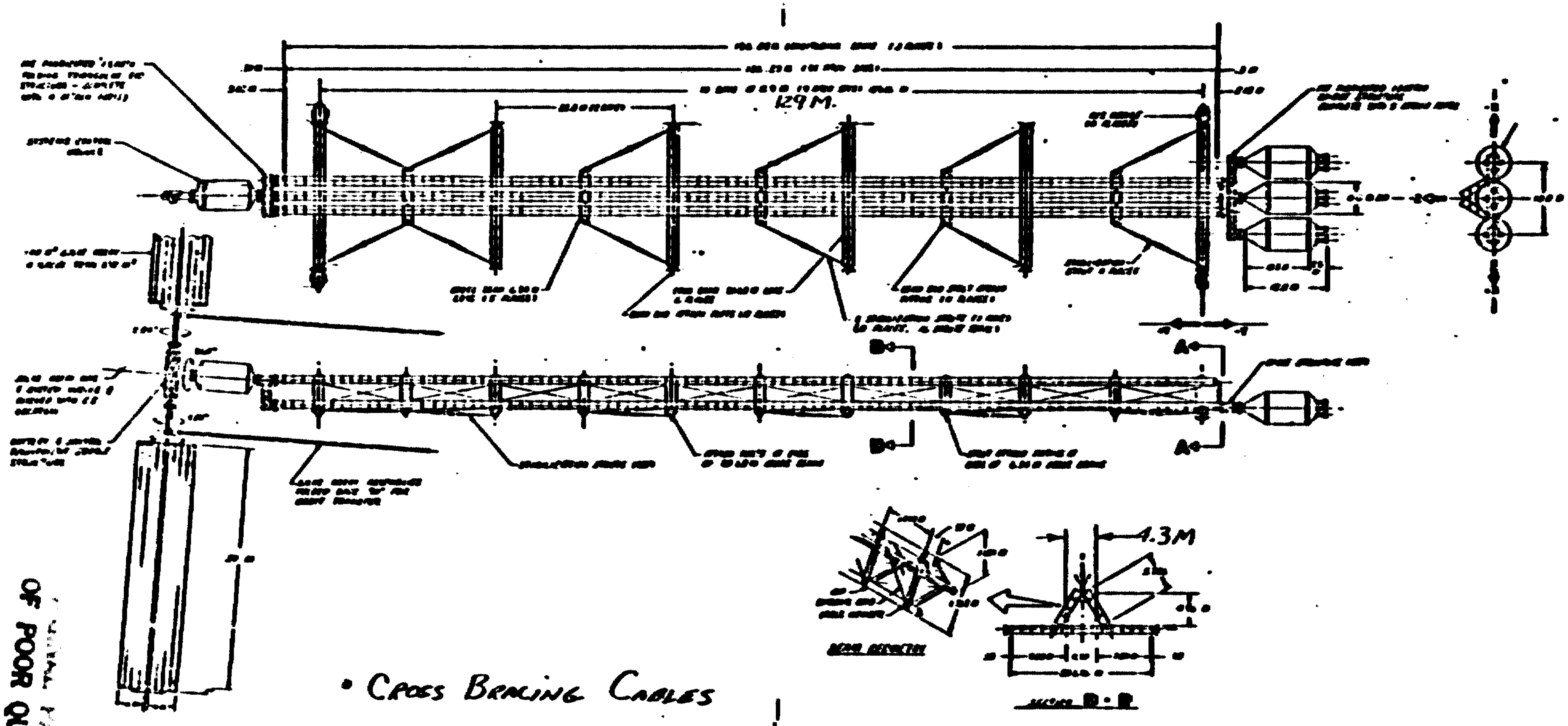
The in-plane arrangement was selected because of the advantages over the staggered arrangement in the construction operations. The staggered arrangement requires a three step translation for the attachment of the transverse members, and the installation of the solar panels for the SPS Test Article is more complex. The revision to the beam ends required for the in-plane arrangement involves only software changes to the beam builder to provide the shorter end bay. Consequently, the ATP structural arrangement will be as shown in the enclosed briefing chart entitled the "Rockwell Tri-Beam Concept."

FIGURE 1. – TRANSVERSE BEAM'S INTERSECTION



IN PLANE vs STAGGERED BEAM ATP CONFIGURATION

ROCKWELL TRI-BEAM CONCEPT



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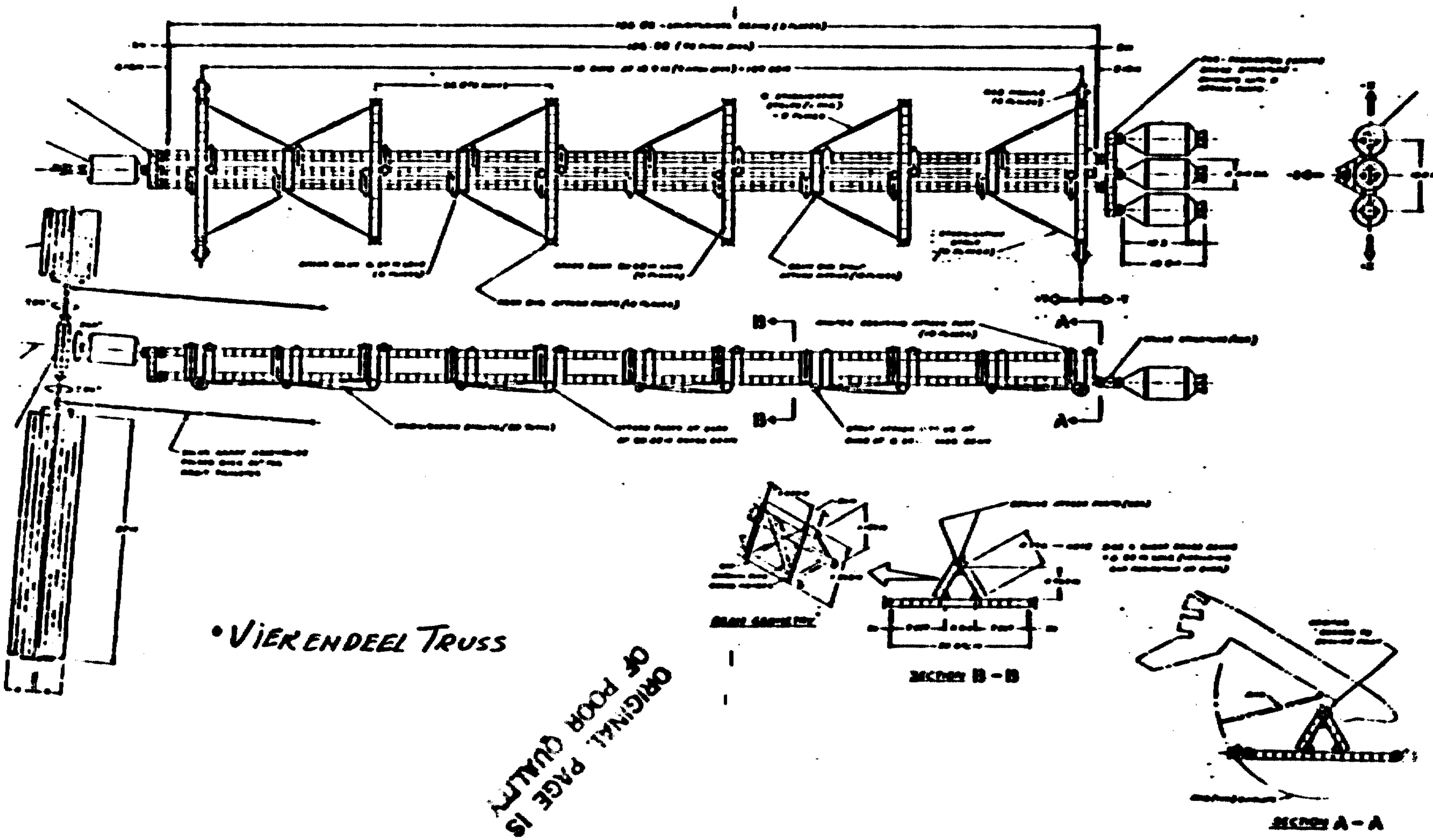
STAGGERED TRI-BEAM CONCEPT



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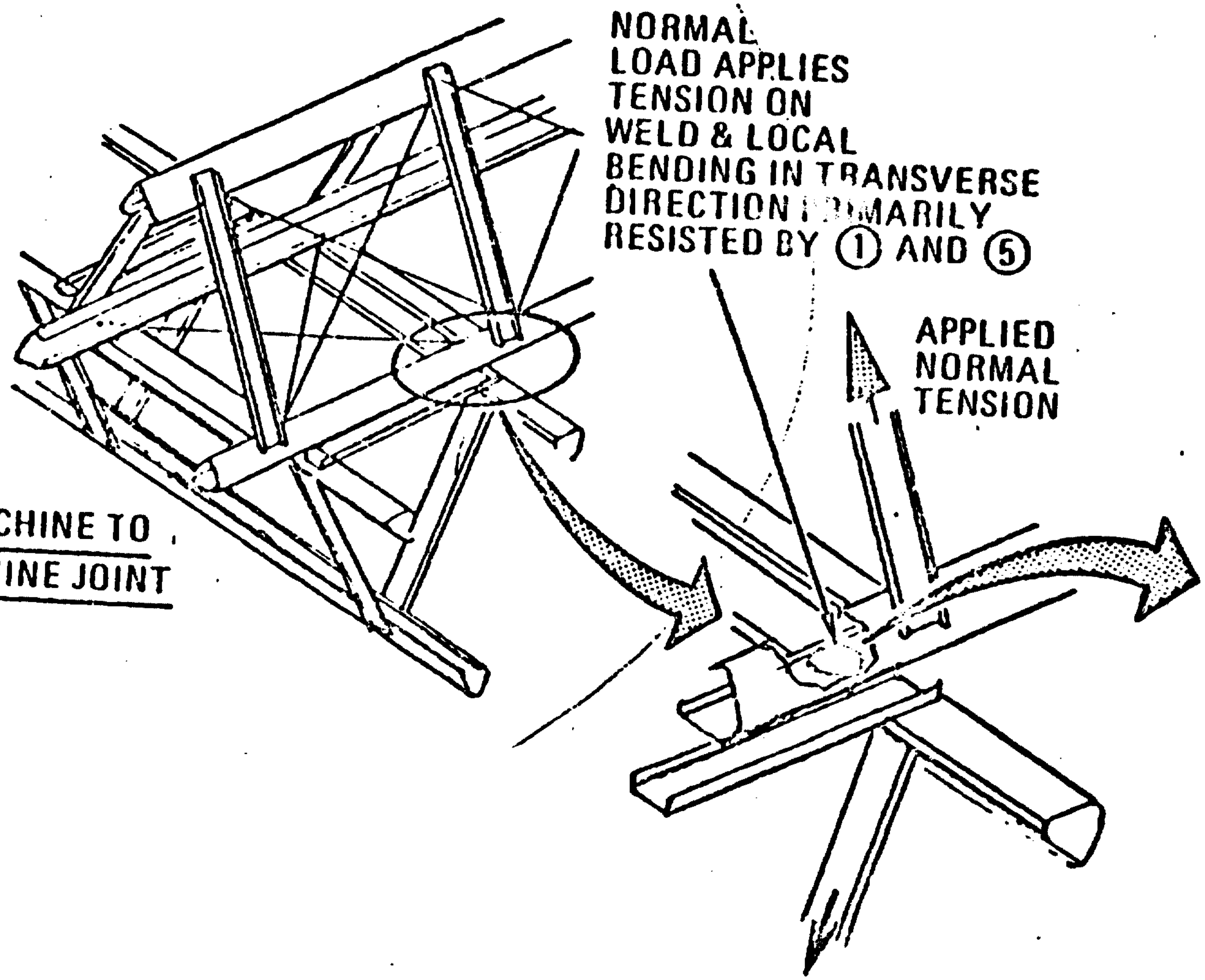
STRUCTURAL ISSUES

- JOINT LOADS
  - NOT SIGNIFICANTLY DIFFERENT.
  -
- DIAGONAL CORDS
  - REQUIRED FOR EITHER CONCEPT
  - INSTALLATION IDENTICAL FOR EACH CONFIG.

JOINT CONFIGURATION



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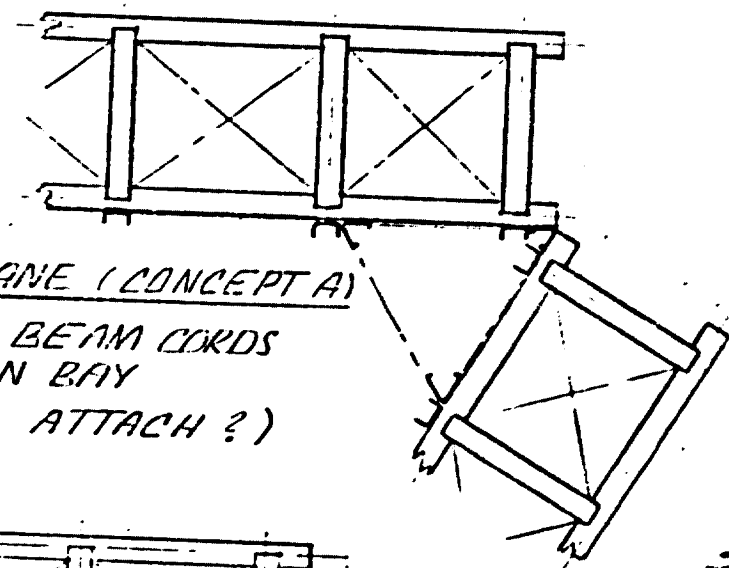
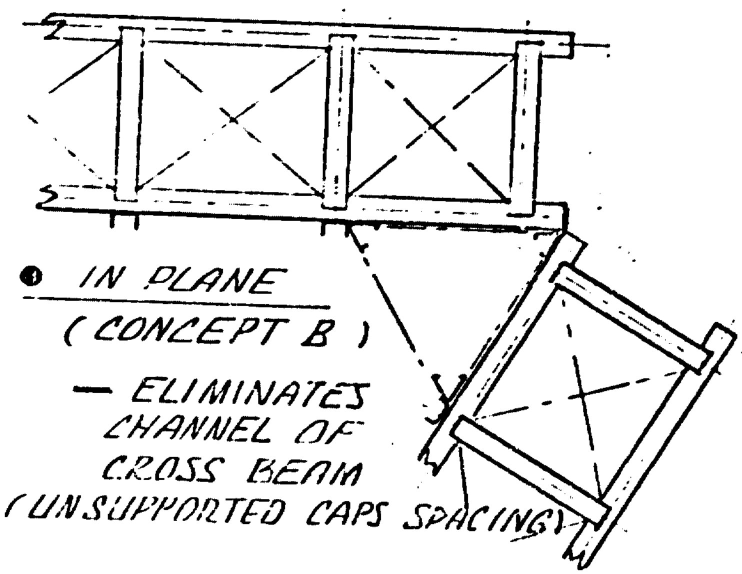
CONFIGURATION ISSUES

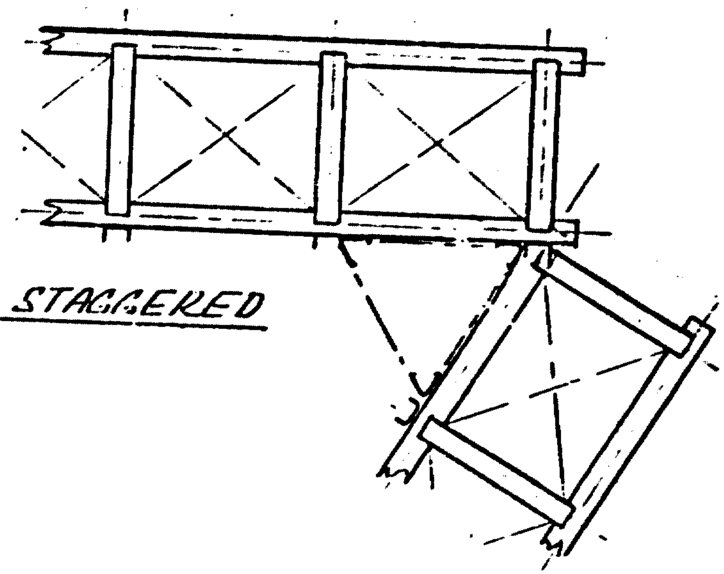


- IN PLANE (CONCEPT A)
  - LONG BEAM CORDS CUT IN BAY (CORD ATTACH ?)

- JOINT REINFORCEMENT
  - IN PLANE, 2 DESIGNS
  - STAGGERED, 1 DESIGN

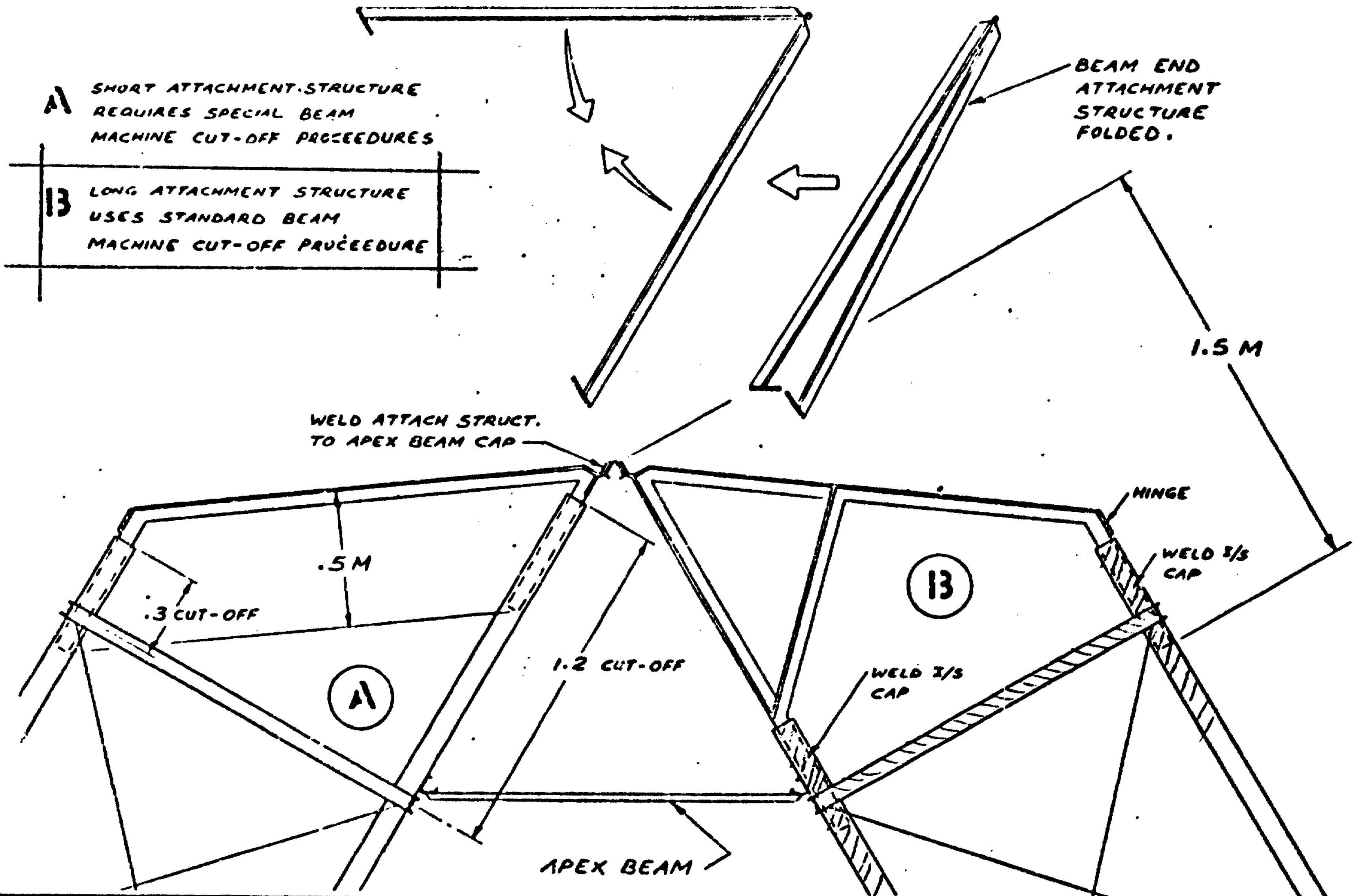


- IN PLANE (CONCEPT B)
  - ELIMINATES CHANNEL OF CROSS BEAM (UNSUPPORTED CAPS SPACING)



- STAGGERED

CROSS BEAM ATTACHMENT DEVICE



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CONSTRUCTION ISSUES

- **FIXTURE**
  - TRANSLATION CAPABILITY FOR BOTH CONCEPTS
  - R.I. FIXTURE
- **STRATEGY**
  - IDENTICAL ASSEMBLY SEQUENCE
  - STAGGERED CONCEPT-CROSS BEAMS

    INSTALLATION REQUIRES
    - THREE STOPS
    - ONE STOP WITH TRANSLATING BEAM POSITIONER

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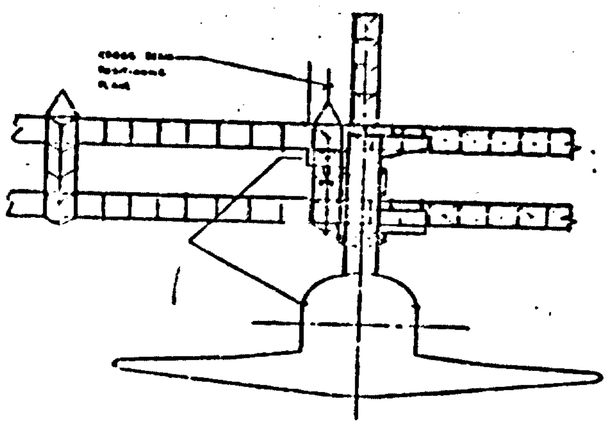
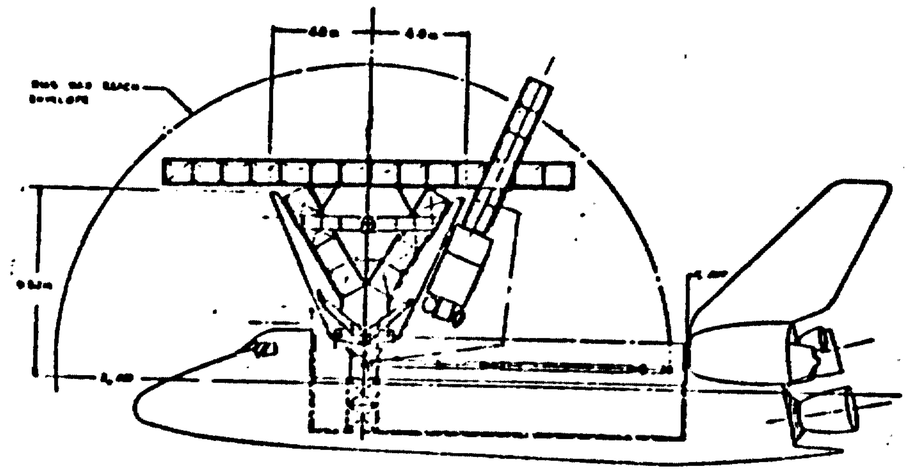
R.I FIXTURE SYSTEM - IN PLANE TRI BEAM CONCEPT





- BEAM MACHINE ATTACHED TO SIDE OF FIXTURE SUPPORT YOKE
- CROSS BEAMS ATTACHED DURING PLATFORM TRANSLATION AT EACH INTERFACE OF CROSS BEAM WITH LONGITUDINAL BEAMS (ONE PLANE , ONE STOP)

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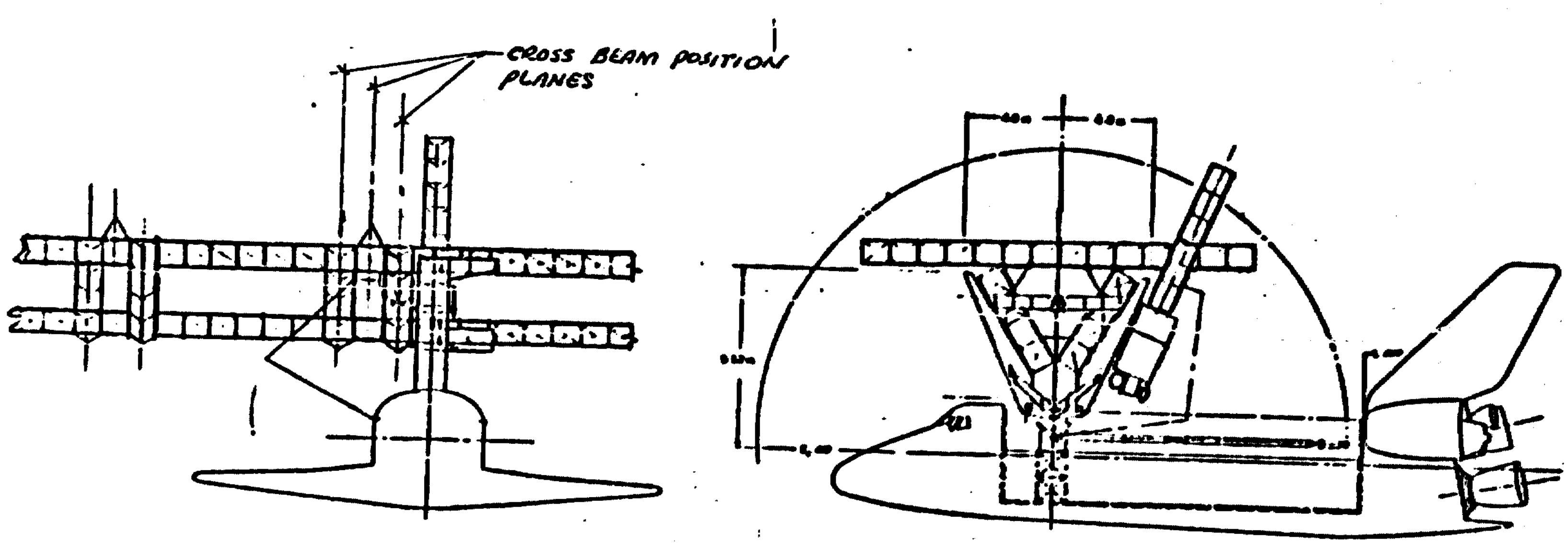
R.I. FIXTURE SYSTEM - STAGGERED TRI-BEAM CONCEPT



- BEAM MACHINE ATTACHED TO SIDE OF FIXTURE SUPPORT YOKE
- CROSS BEAMS ATTACHED DURING PLATFORM TRANSLATION AT EACH INTERFACE PLANE OF CROSS BEAM WITH LONGITUDINAL BEAMS (3 PLANES, 3 STOPS OR ONE STOP WITH TRANSLATING BEAM POSITIONER)

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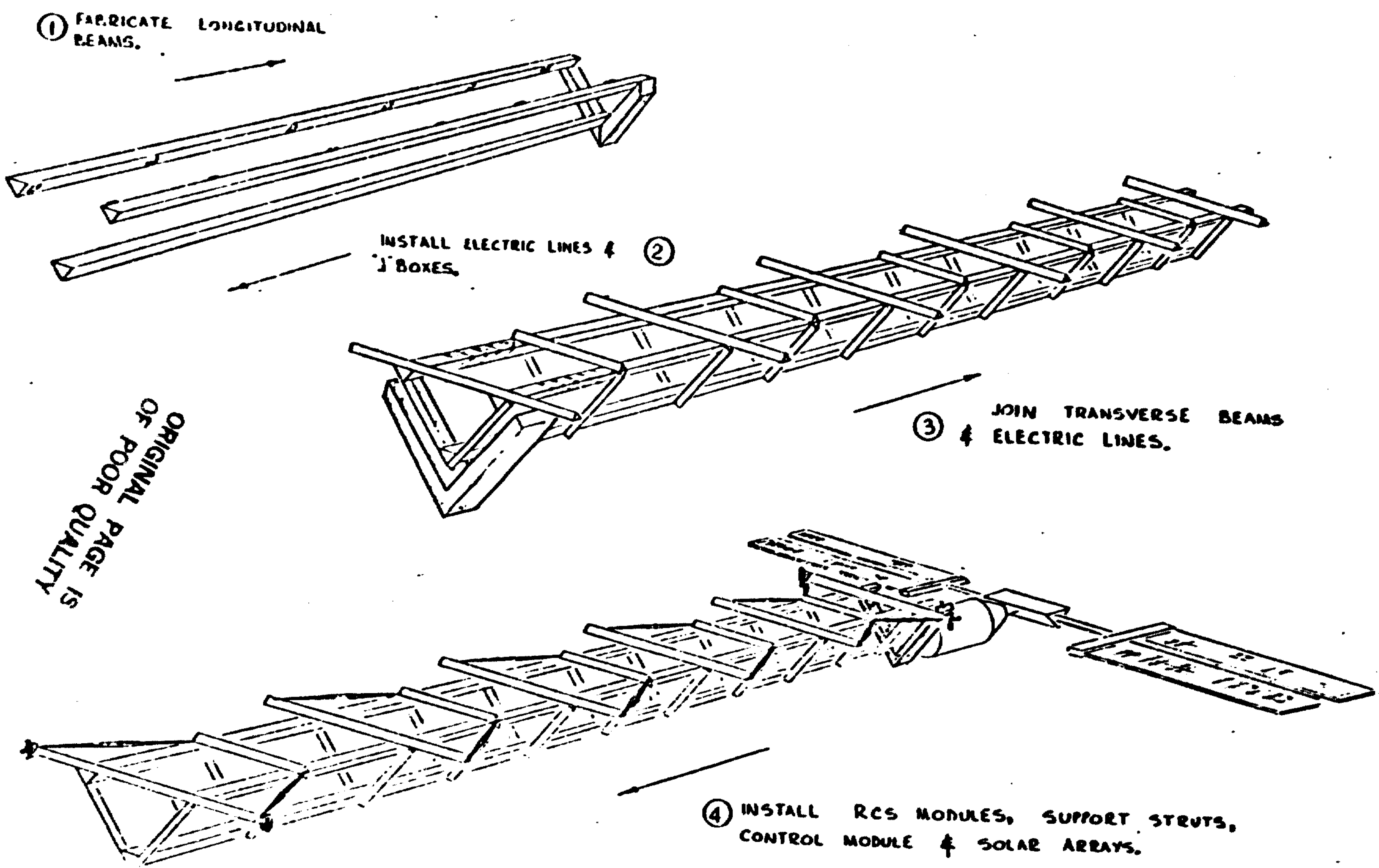
R.I TRI-BEAM CONSTRUCTION STRATEGY
ANTENNA PAYLOADS CONCEPT



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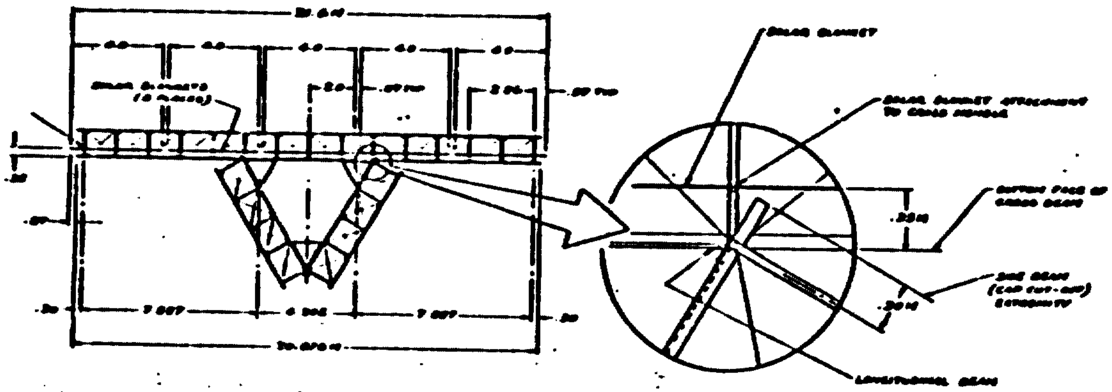
SYSTEM INSTALLATIONS

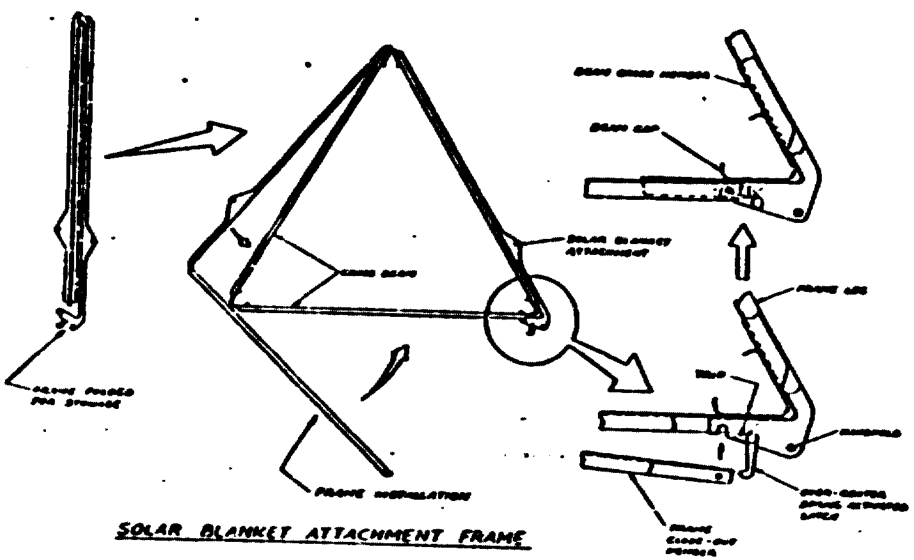
- STAGGERED CONCEPT PLATFORM HAS SLIGHTLY LONGER RMS REACH TO INSTALL ANTENNAS
- LINES INSTALLATION VERY SIMILAR

SPS T.A. SOLAR PANEL INSTALLATION

STAGGERED BEAM CONCEPT



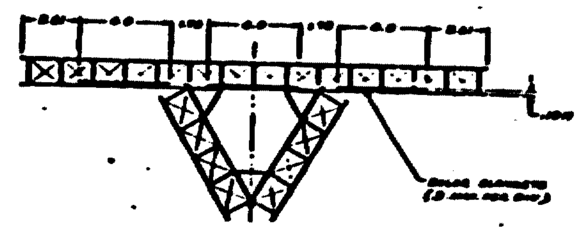
SECTION A – A



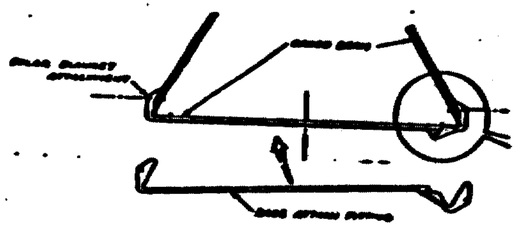
SOLAR BLANKET ATTACHMENT FRAME

SECTION B – B

ALTERNATE CONFIGURATION



SECTION A – A ALTERNATE



SECTION B – B ALTERNATE

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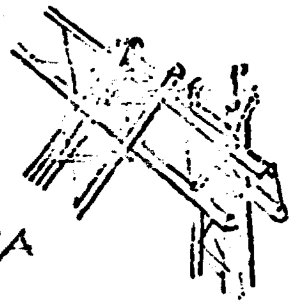
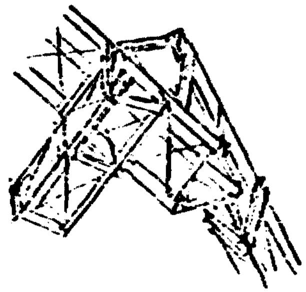
STAGGERED VS IN-PLANE TRI-BEAM CONFIG. SAFETY PROVISIONS AT BEAM ENDS

STAGGERED



- SIX PROJECTIONS AT JOINTS ENDANGER EVA PRESSURE SUIT & UMBILICALS (IF UNPROTECTED)
- ADD-ON COVERS/GUARDS CREATE COMPLEXITY & WEIGHT/TIME PENALTY

IN-PLANE



- FEWER PROJECTIONS AT JOINTS (2 OUTSIDE CORNERS ONLY)
- LIGHTER, SIMPLER GUARDS-IF REQUIRED

- NO OBVIOUS DIFFERENCES BETWEEN ELECT. CABLE INSTALLATIONS

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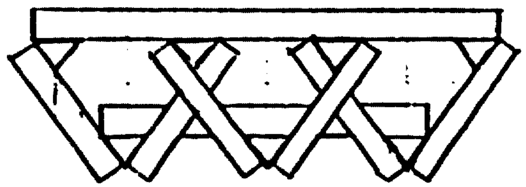
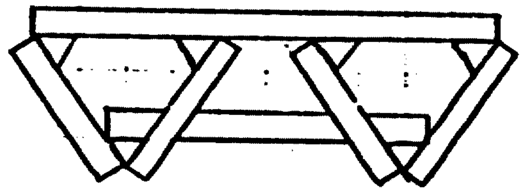
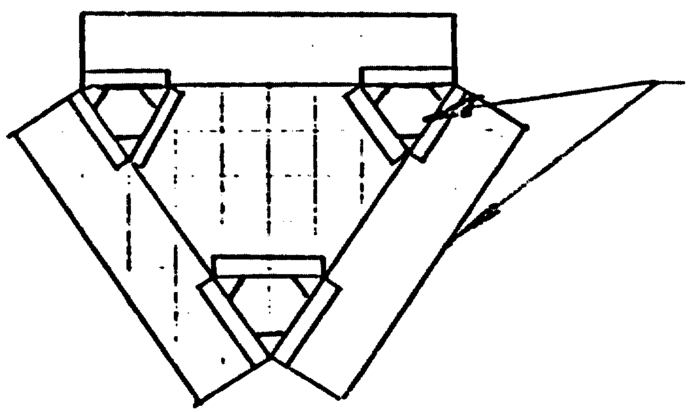
GROWTH / LEGACY

- GROWTH CONCEPTS FOR PLANER PLATFORMS - IN CONTRAST TO LINEAR PLATFORMS - REQUIRES SOME STAGGERED JOINTS. NUMBER AND LOCATION DEPEND ON DESIGN CONCEPT





- LEGACY TO SPS



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SUMMARY

- STRUCTURE — NO SIGNIFICANT DIFFERENCE
- CONFIGURATION — RECOMMEND CONCEPT "B" FOR IN PLANE BEAM END CONFIGURATION
- CONSTRUCTION — IDENTICAL FIXTURE - RECOMMEND 3 POSITION TRANSLATION FOR STAGGERED BEAM INSTAL.
  — IDENTICAL CONSTRUCTION STRATEGY
  — NO SIGNIFICANT DIFFERENCE FOR SYSTEMS INSTALL.
  — SOLAR BLANKET INSTALL. FOR SPS T.A. SIGNIFICANTLY MORE COMPLEX FOR STAGGERED BEAM ARRANGEMENT
- SAFETY — NO SIGNIFICANT DIFFERENCE
- GROWTH/LEGACY — STAGGERED ARRANGEMENT OCCURS IN VARYING DEGREES FOR PLANER PLATFORM CONCEPTS
  (EXCEPTION - BUTT JOINT CONCEPT)

APPENDIX C

TRADE STUDY, STRUCTURAL CONFIGURATION

Presented herewith is the trade study that compared a Vierendeel tri-beam with an "X"-braced tri-beam.

The decision factors favored the "X"-braced tri-beam rather than the Vierendeel. Briefly stated, the disadvantage of increased fixture, construction and structural analysis complexity inherent with the X-bracing are outweighed by the following significant advantages:

- Use of one less propulsion module - higher T/W allowable; lower gravity loss.
- Higher operational flexibility - use of the Vierendeel structure would require at least a 3½ hour settling time to reduce attitude control torque induced deformations to acceptable values. In contrast the settling time for the "X"-braced tri-beam is negligible.
- Higher versatility of design - the Vierendeel design is essentially at the maximum limit of its capability (p. C-28) within the constraint of the tri-beam depth. Both the longitudinal cap and diagonal cords were increased to the permissible limits anticipated by GD. Increased performance is available, however, with the "X"-braced design through increase of the beam cap gage or increase of the "X"-brace strip member cross sectional area.
- Least Control System Impact - The minimum structural frequency of the "X"-braced design is significantly higher.
- Legacy - The "X"-braced design will provide the legacy for future applications with more stringent structural requirements than those of the ATP.

An illustration of the foregoing is described on page C-27. The chart compares the variation of axial load capability as a function of the characteristic AE. For the truss (i.e., cable braced), the parameter refers to the "X" bracing with a constant value of 20,000 pounds used for the AE of the GD beam diagonal cords. For the Vierendeel design the parameter AE refers to the GD beam diagonal cords. The Vierendeel structure is competitive with the truss, i.e., the axial load capability was essentially the same for the same configuration depth, cap areas, and AE, BUT the Vierendeel capability is limited by the diagonal cord AE that is achieved with the current GD machine-made beam. Discussion with GD personnel have indicated .080 inch diameter "S-glass" for the diagonal cords are probably the upper limit of the present design (.040"). The .080 inch diameter cords have the AE value of 20×10^3 lb shown, and for this example, the axial load capability for a depth of 172 inches is = 10,000 pounds. The truss, however, using bracing of AE = 1.47×10^6 lb can provide an axial load capability greater than 40,000 pounds. The same conclusions can be drawn for torsional induced deformations.

With regard to the geodetic beam configurations, the "Geodetic Vierendeel" appeared to represent the best of the three geodetic beam applications studied. The basic geodetic beam shear and torsional stiffness is substantial - such that this Vierendeel does not require "X" bracing for the ATP application. The geodetic beam cluster structural performance was comparable to the Vierendeel; however, the construction complexities inherent with the joining of the beams (p. C-12 & C-23), while possibly resolvable with future study, were sufficient at this point to result in selection of the "Geodetic Vierendeel" for comparison with the "X"-braced tri-beam utilizing the GD machine-made beam.

In assessing the relative merits of the geodetic and triangular (GD) beams for application to the ETVP, it was concluded that both are viable options but that additional data (p. C-29) is required. For example, the joint capability of the lap joint utilizing the GD beam (p. C-13) needed to be understood as well as the capability of the Geodetic structure to receive joint loads (p. C-14).

The foregoing discussions in conjunction with the enclosed briefing charts present the major considerations that resulted in the selection of "X"-braced tri-beam (utilizing the GD machine-made beam) for the structural configuration ETVP. For this report the following text is attached for clarification of the figures presented on the briefing charts.

Page C-7. The attachment to the geodetic beam was based on the use of appropriate equipment mounted external to the geodetic beam. The openings between the pultruded rods permit access for backup of the attachment. Placement of an astronaut or equipment inside the beam was not considered.

Page C-11. The figure at the extreme lower right portion of the chart represents one potential concept for a saddle in a Vierendeel or "X"-braced configuration. The cross hatched regions denote the area of attachment to the geodetic beam. Three regions each are provided for the longitudinal beam and for the transverse beam. The regions can be a structural grid that matches the geodetic nodal point pattern on the beam, thereby providing numerous attach points to the beam (as required for load). The regions are joined together by appropriate bracing. The three regions of attachment permit transfer of all the potential axial forces and moments (p.C-14) in the geodetic beam through forces essentially tangent to the geodetic beam surface.

In this concept the separate sections above and below the plane of attachment can be folded into an essentially flat array to be subsequently and separately fastened to the geodetic beams. Later mechanical or welded joining at the plane of attachment is accomplished.

Page C-12. To develop the composite behavior of the individual geodetic beams the structural element (shaded) is provided (at discrete intervals, as required). This element attaches (tangentially) to each geodetic beam at two locations. The attachment must transfer the longitudinal VQ/I forces necessary to composite behavior. For the particular geometry shown these VQ/I forces exert couples that must be sustained by balancing tangential forces directed along the circumference. This structural element can be totally fabricated and contained in a magazine as shown on p. C-23. An alternate concept is shown at the lower left, for minimum stowage volume.

Page C-14. The potential axial loads and moments to be transmitted to the saddle are shown at the right side of the chart.

- The "X" loads are transmitted longitudinally along attachment lines 1-2 and 5-6.
- The combination of "Y" loads and moment about the "Z" axis is transmitted by the distributed transverse loading shown along attachment line 3-4.
- The combination of "Z" loads and moment about the "X" and "Y" axes are transmitted by the distributed transverse loading shown along attachment lines 1-2 and 5-6.

All of the above loads are essentially tangent to the geodetic beams.

VIERENDEEL VS. X-BRACED TRI-BEAM

FOR AN

ENGINEERING TEST & VERIFICATION PLATFORM (ETVP)

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99SSD04955

CANDIDATE CONCEPTS

- ROCKWELL TRI-BEAM WITH X-BRACING (GD BEAM)
- VIERENDEEL—WHICH IS A TRI-BEAM WITHOUT X-BRACING (GD BEAM)

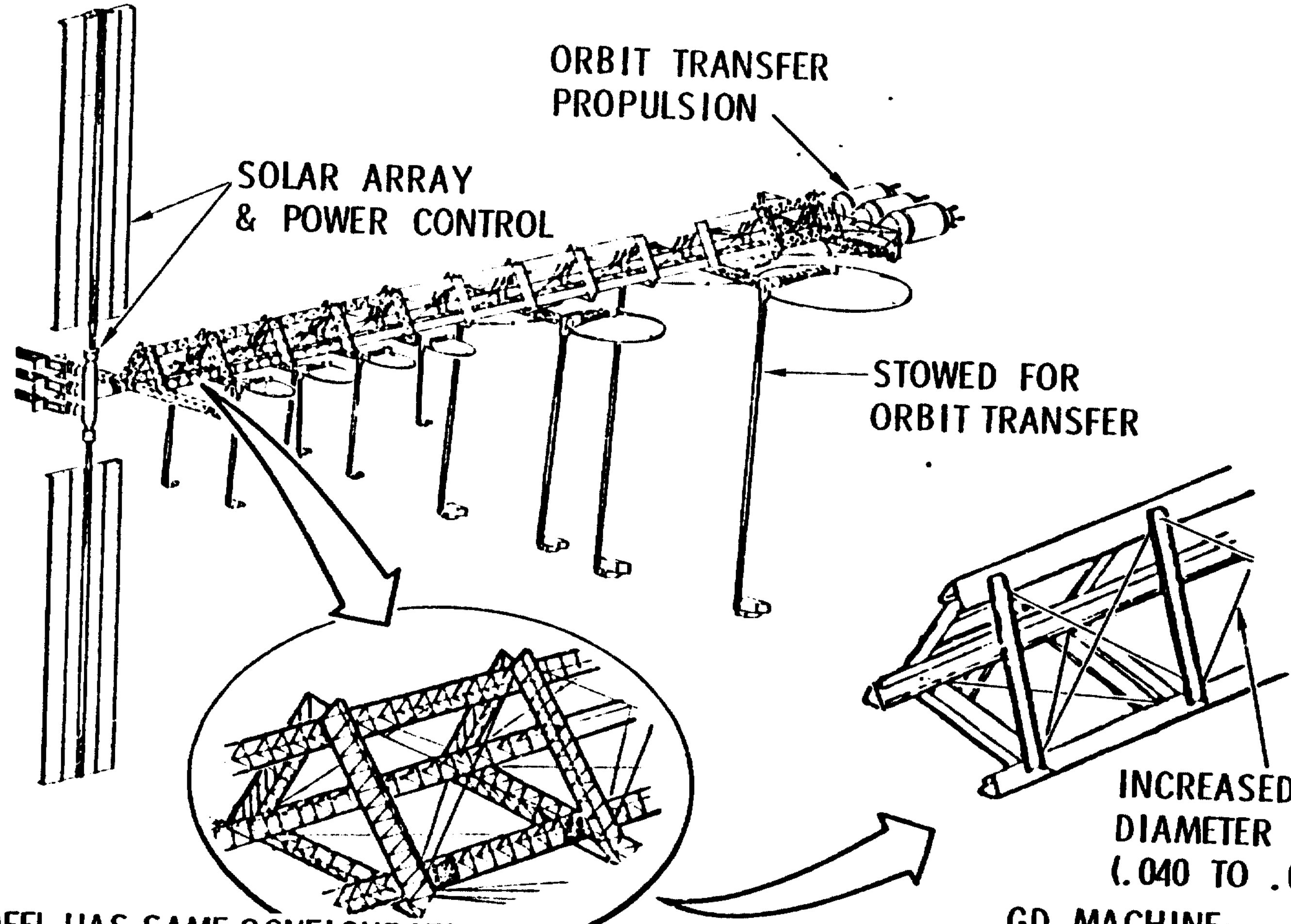
ISSUES

- COLUMN STABILITY DURING ORBIT TRANSFER
- MEMBER IMPOSED AXIAL FORCES AND MOMENTS
- ATTITUDE CONTROL/STRUCTURAL DEFORMATION ERRORS— (TWIST ABOUT PITCH AXIS)
- BEAM LOCAL LOAD SUITABILITY
    - SEPS BLANKET ATTACHMENTS
    - ELECTRICAL LINE MOUNTING AND J-BOXES
    - BEAM TO BEAM ATTACH
    - DOCKING PORT MOUNTING
    - HANDLING, MOVING, ETC.
- FIXTURE IMPLICATIONS
- CONSTRUCTION IMPLICATIONS
- GROWTH/LEGACY

GROUND RULES

- ATP ORBIT TRANSFER WEIGHT—80,000 LB
- PITCH ATTITUDE CONTROL THRUSTERS—ONE POUND
- ORBIT TRANSFER BURNS—ONE AT APOGEE, ONE AT PERIGEE
- ROCKWELL FIXTURE AS BASELINE
- ETVP LENGTH -- 130 METERS

ROCKWELL "X" BRACED TRI-BEAM



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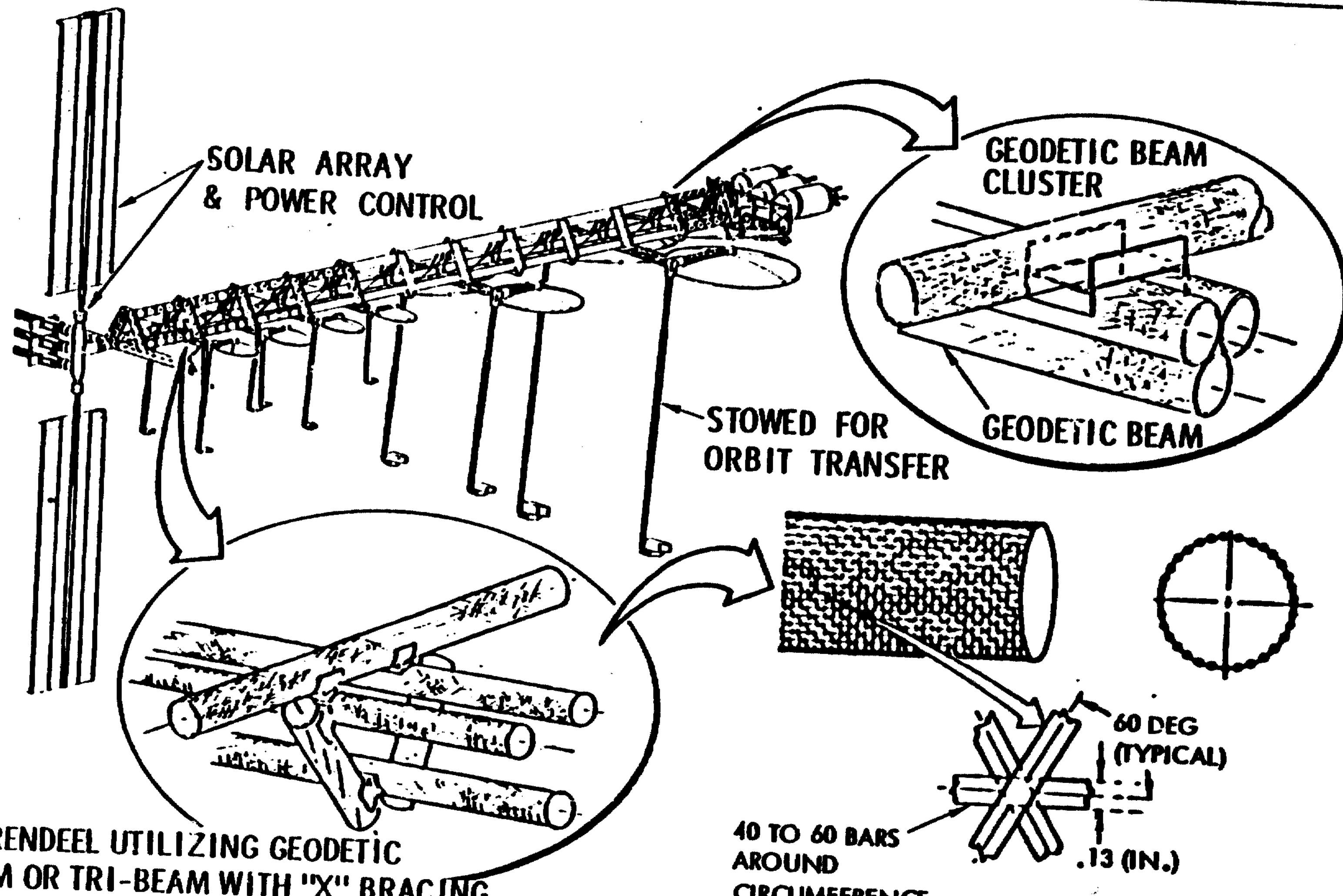
GEODETIC CONFIGURATIONS



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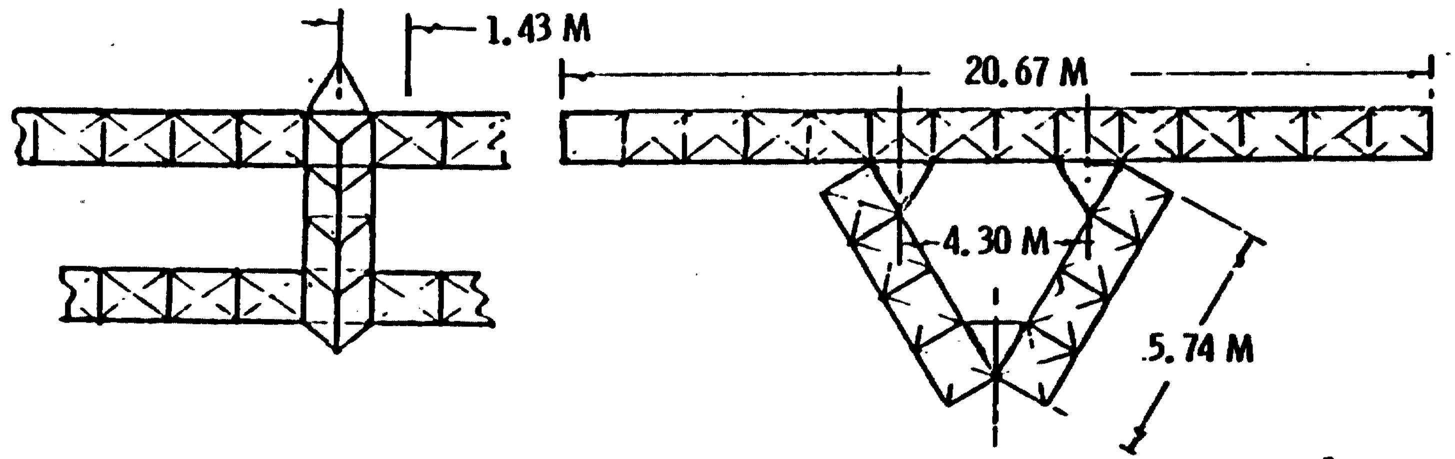
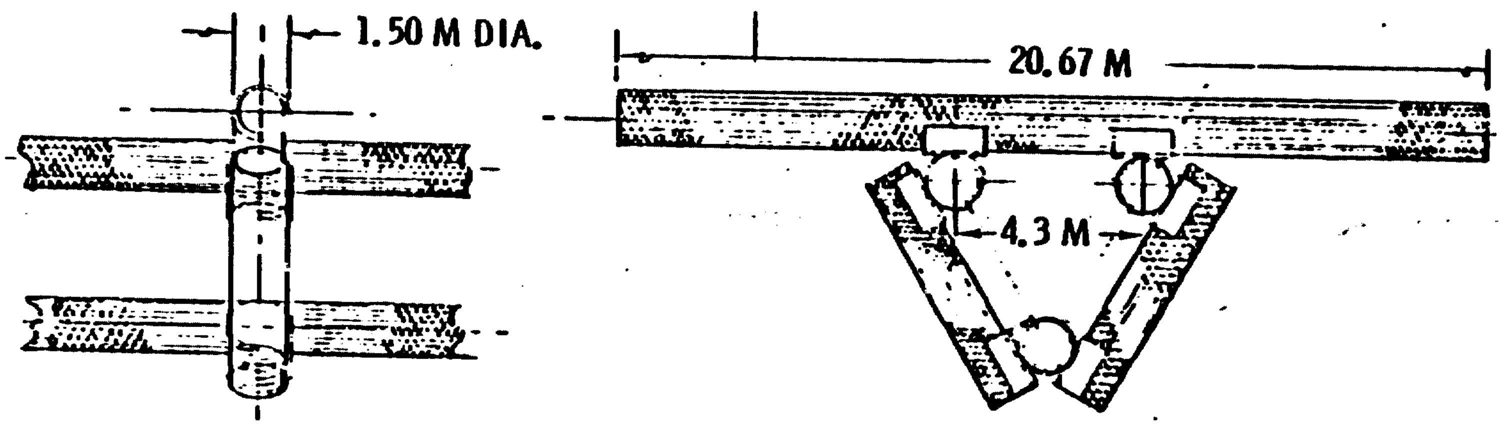
TRIANGULAR VS. GEODETIC TRI-BEAM CONFIGURATION

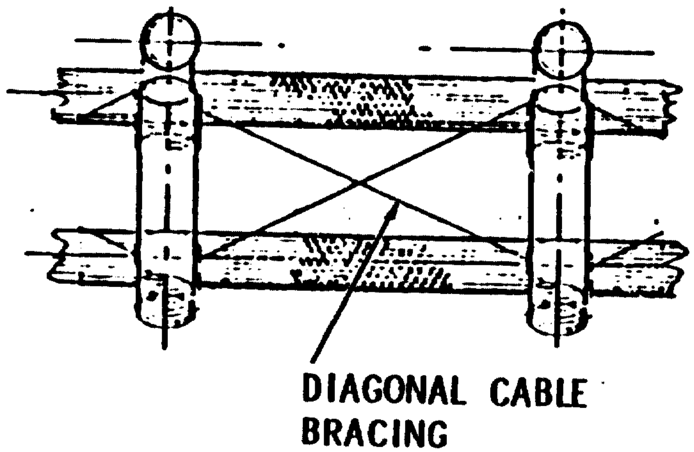


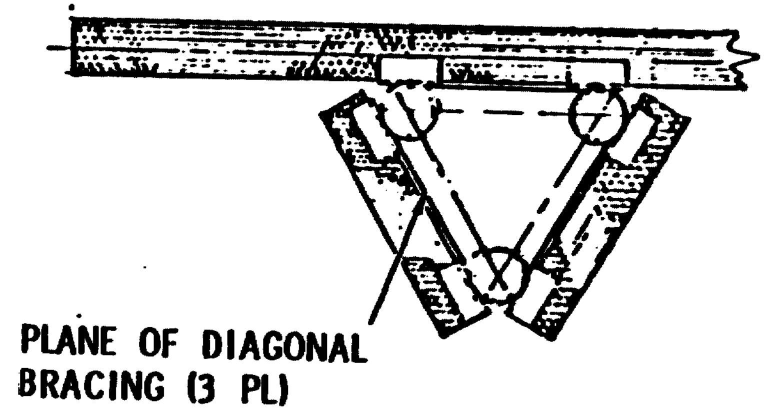
TRIANGULAR BEAM—TRI-BEAM CONFIGURATION

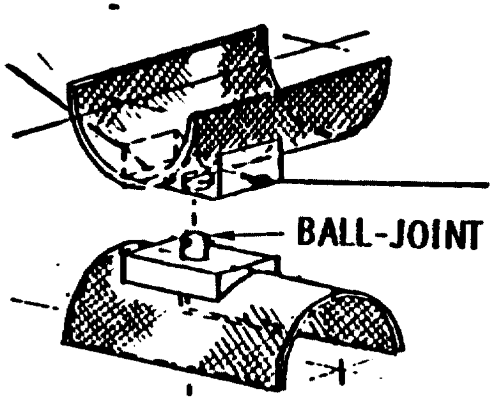


GEODETIC BEAM TRI-BEAM CONFIGURATION

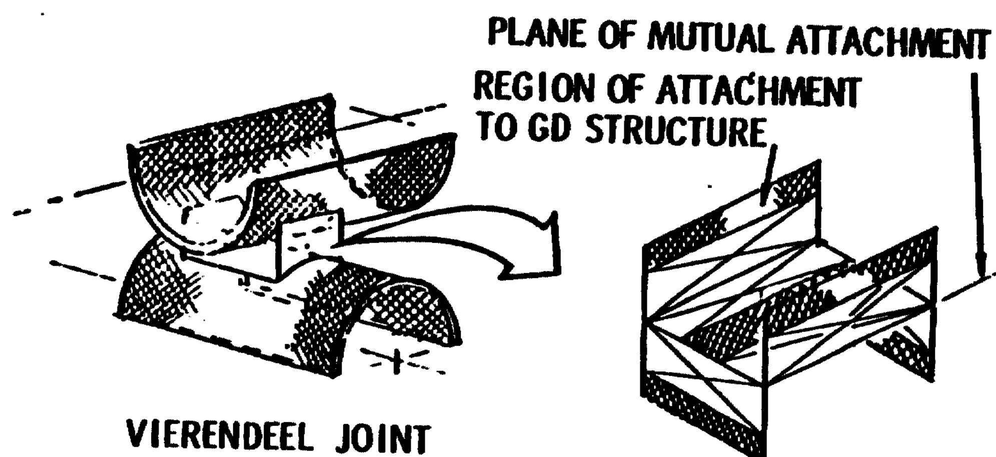
GEODETIC TRI-BEAM/VIERENDEEL JOINT CONCEPTS







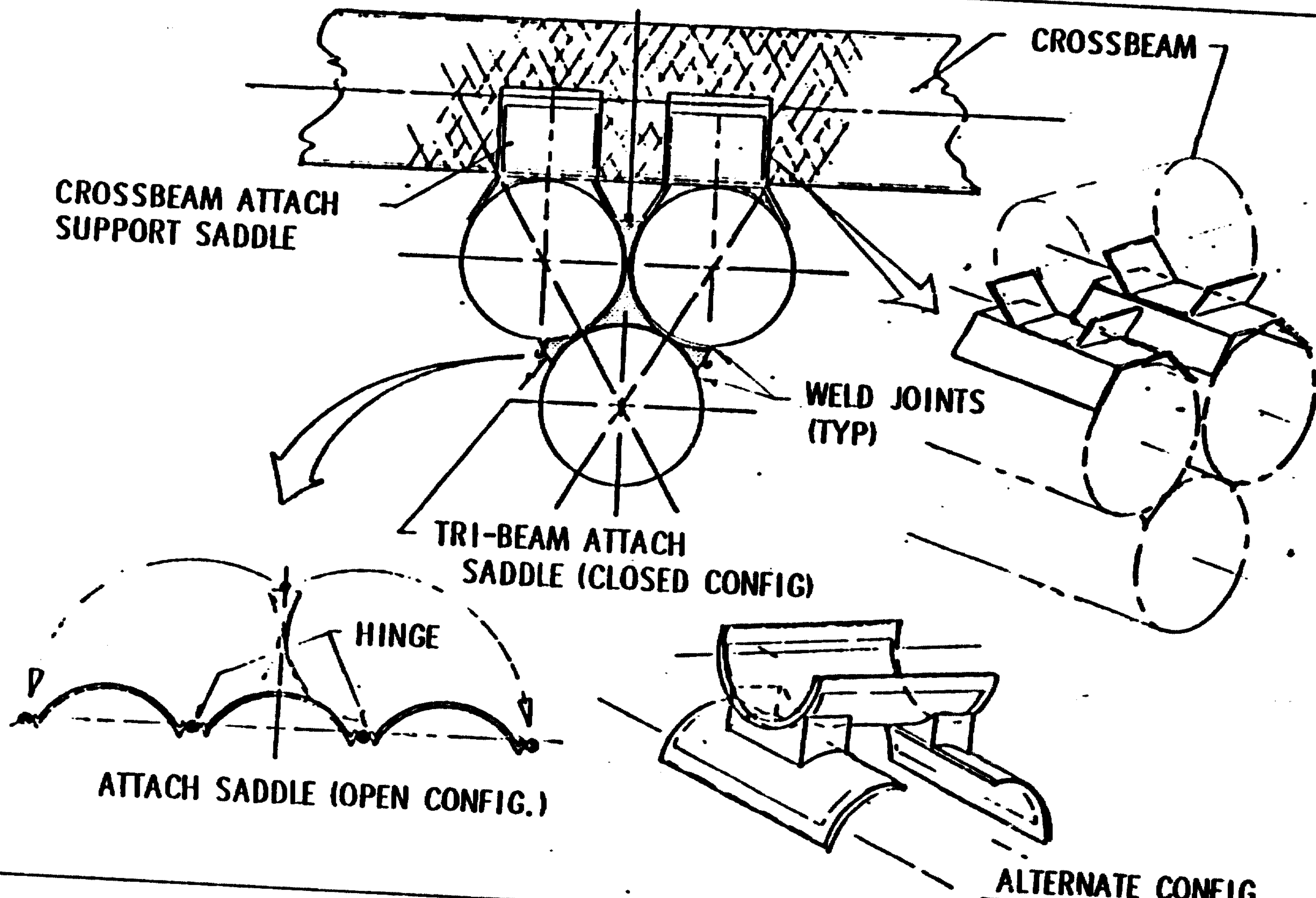
PIN ENDED CONCEPT



VIERENDEEL JOINT

POTENTIAL JOINING FRAME

CLUSTERED GEODETIC ATTACHMENT CONCEPTS



LAP JOINT STRENGTH CONSIDERATIONS

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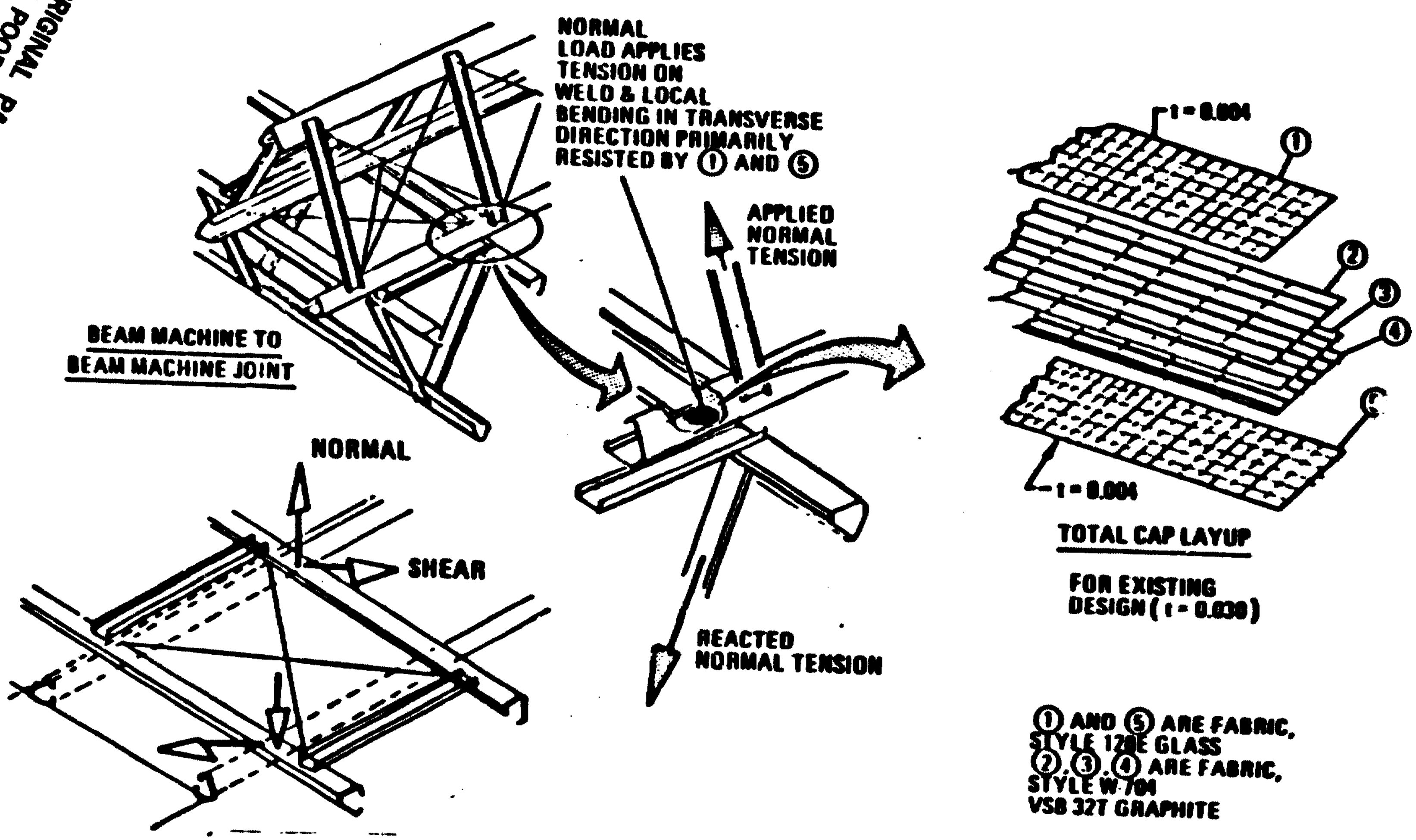
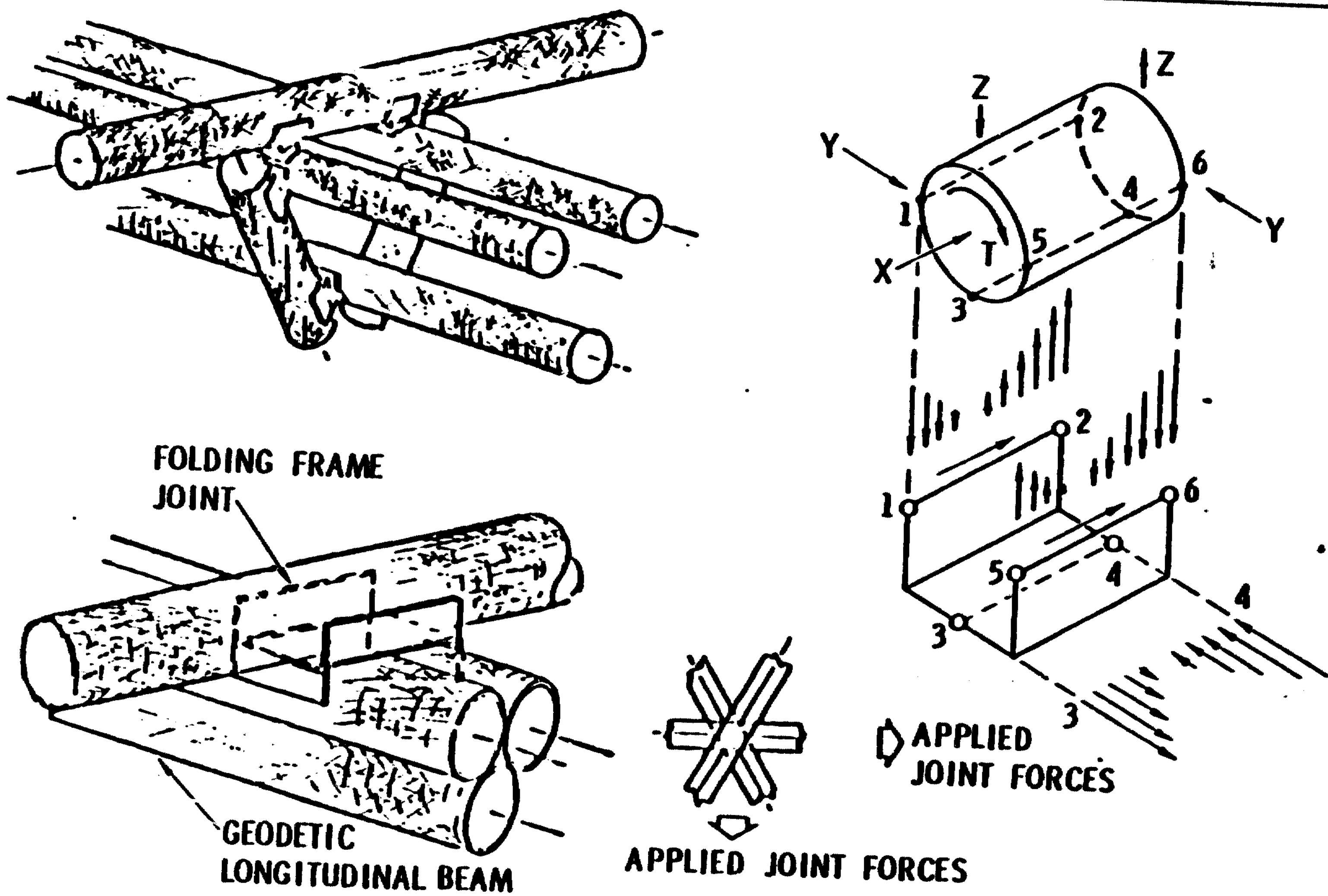


GEODETIC VIERENDEEL JOINT IMPLICATIONS



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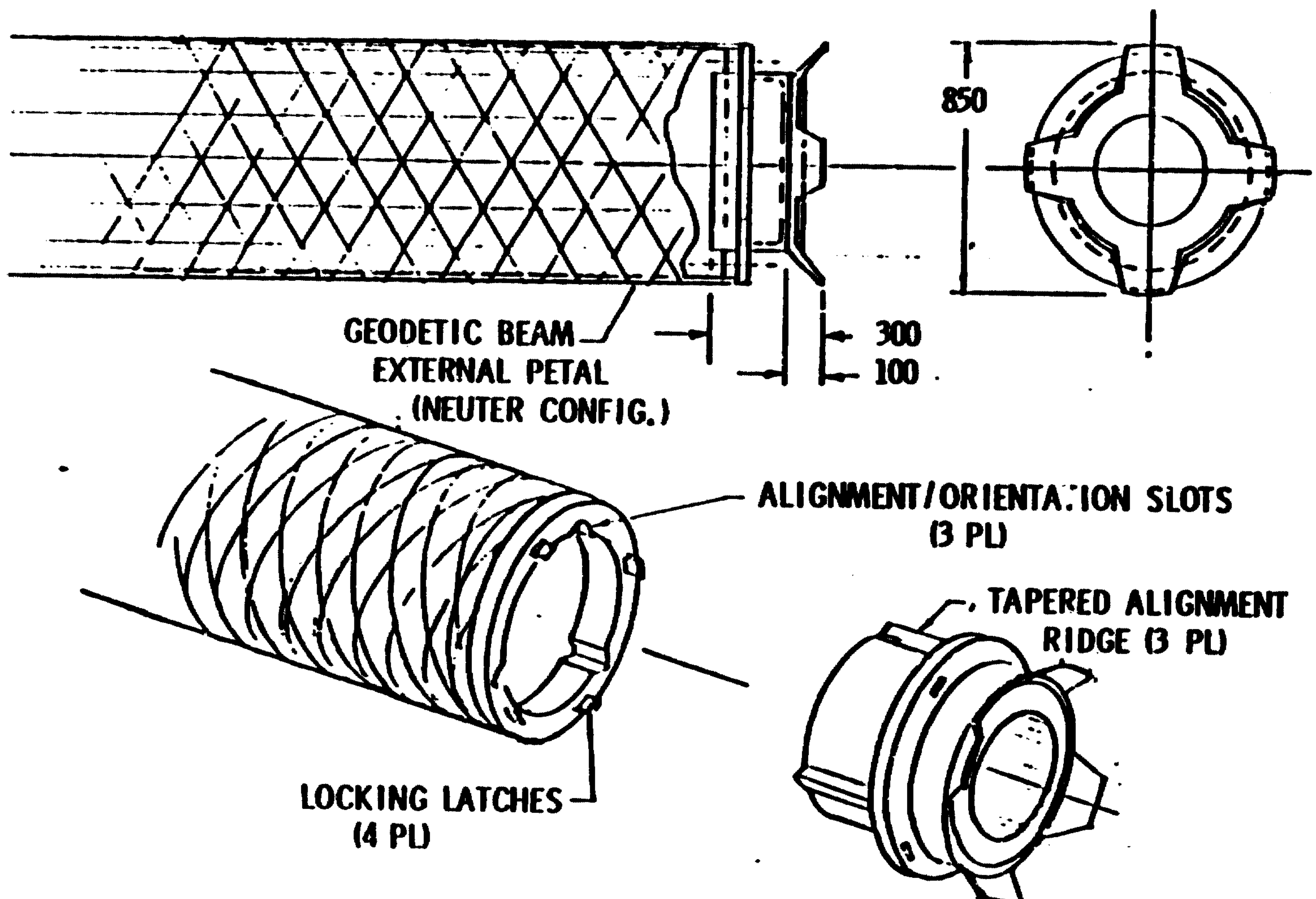
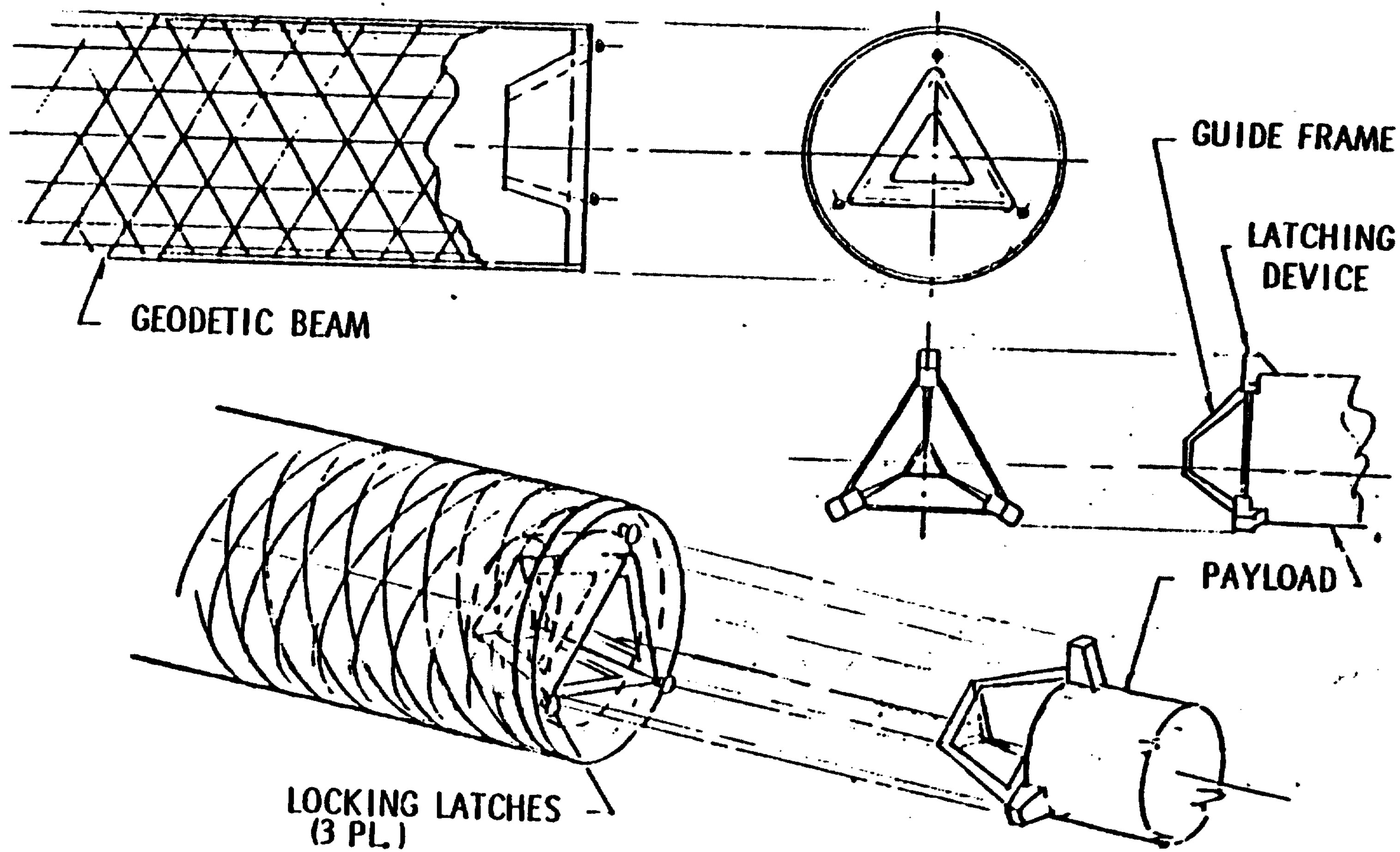
MODULE COMPONENT INSTALLATION



MODULE COMPONENT INSTALLATION



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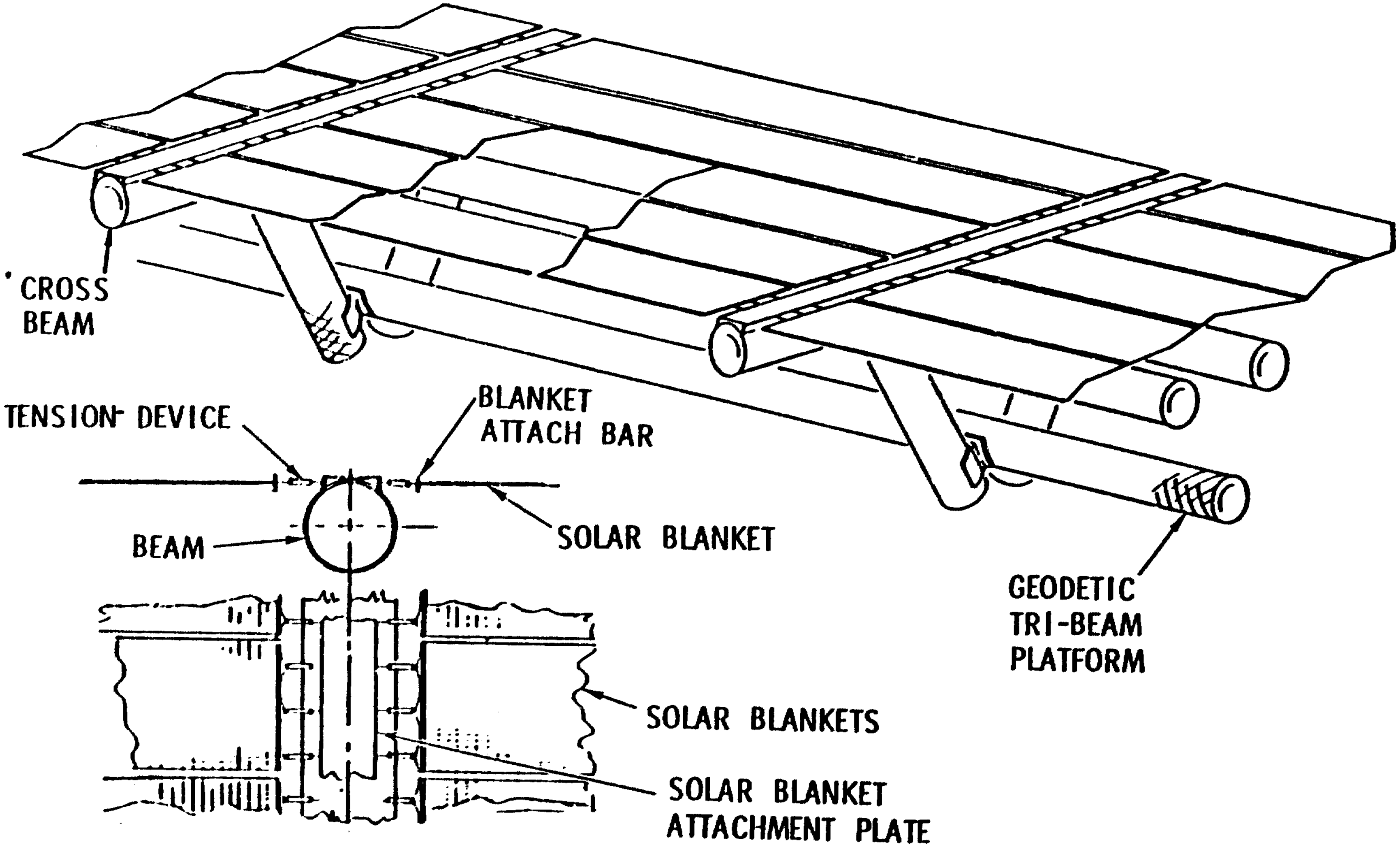
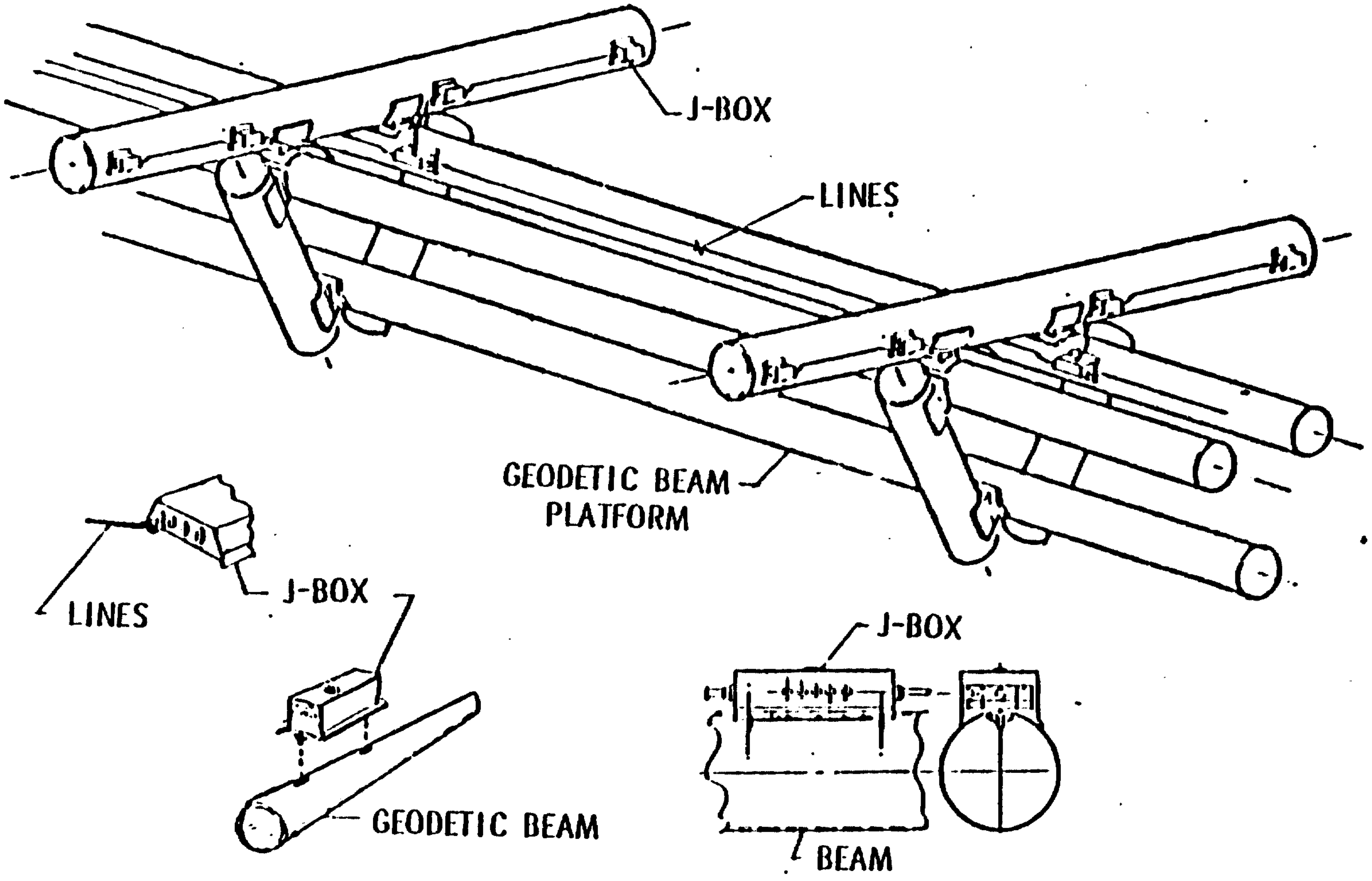
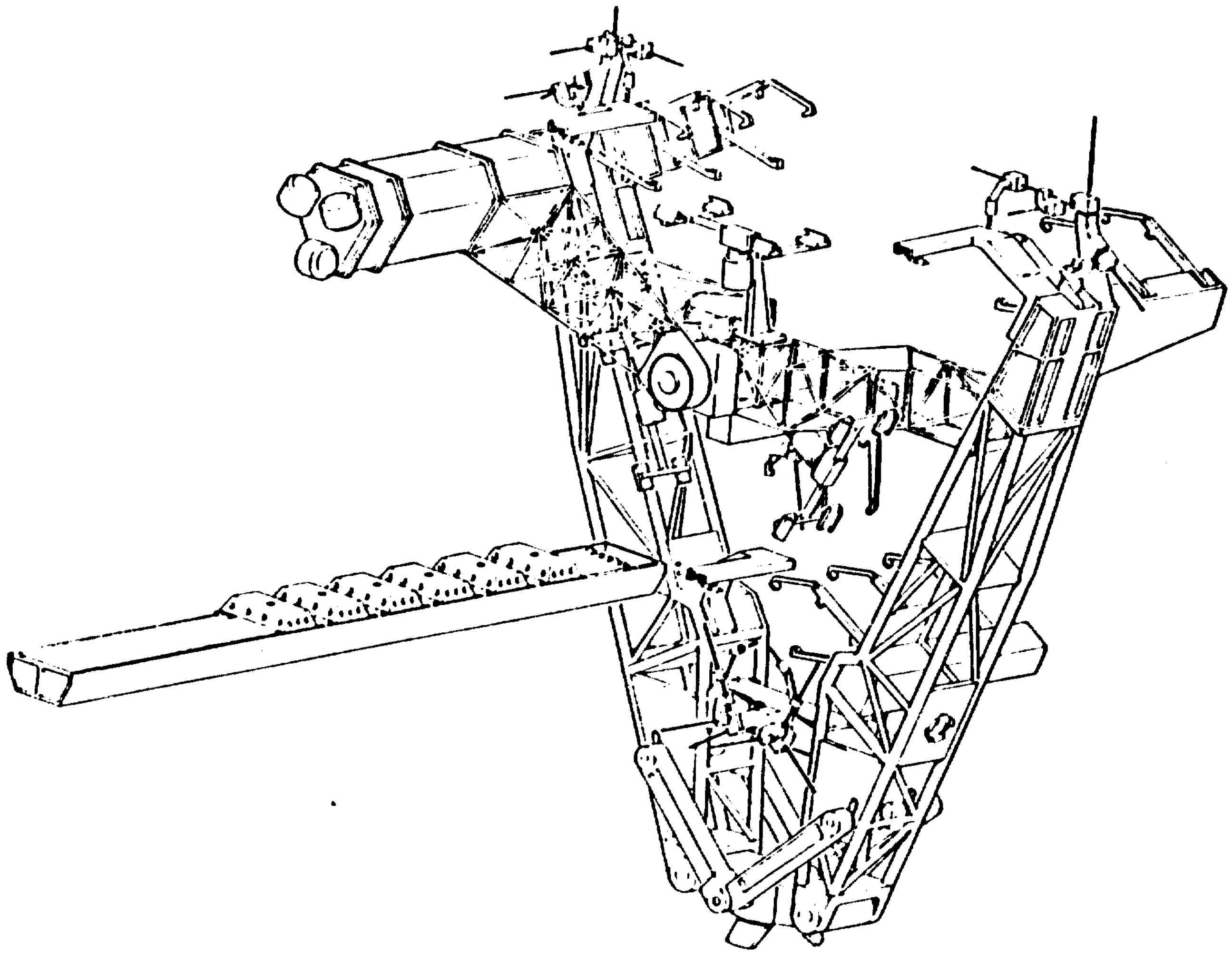
GEODETIC TRI-BEAM PLATFORM - SOLAR BLANKET INSTALL



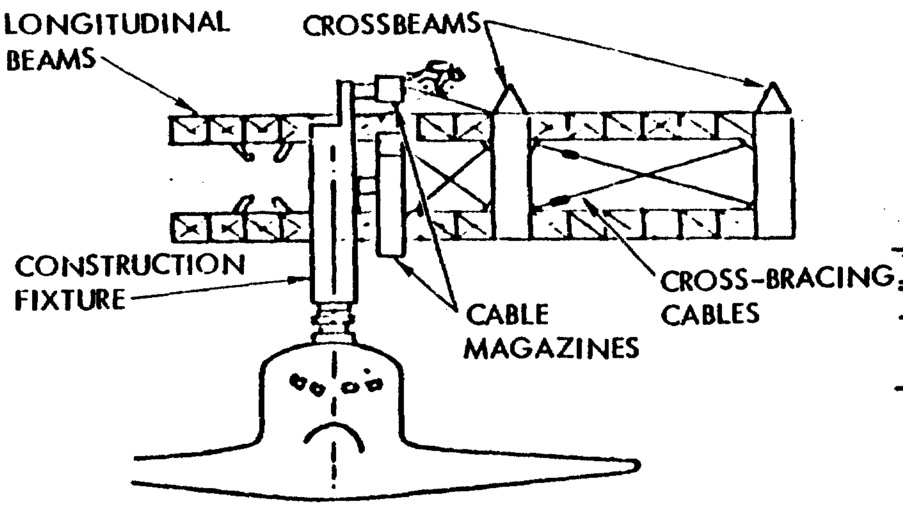
GEODETIC TRI-BEAM PLATFORM—LINES AND J-BOX INSTALLATION

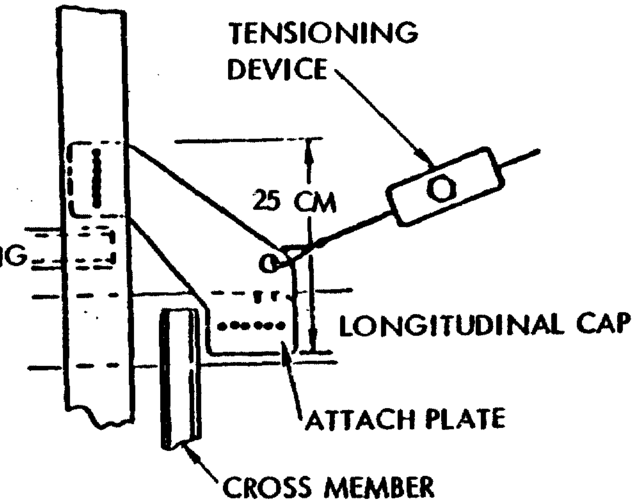


ROCKWELL TRI-BEAM FIXTURE SYSTEM



"X" BRACING CONSTRUCTION IMPACT





- WELD THE ATTACH PLATES TO THE LONGITUDINAL BEAMS AS THEY ARE FABRICATED BY THE BEAM BUILDER.
- WELD ATTACH BRACKET TO BEAMS.
- EVA ASTRONAUTS INSTALL THE FORWARD ENDS OF SIX CABLES TO THE ATTACH PLATES WHILE THE FIRST SET OF CROSS AND TRANSVERSE BEAMS ARE BEING FABRICATED AND INSTALLED.
- WELD ATTACH BRACKET TO BEAMS.
- EVA ASTRONAUTS TENSION THE SIX CABLES AND INSTALL THE FORWARD ENDS OF THE SECOND SET OF CABLES WHILE THE SECOND SET OF CROSS AND TRANSVERSE BEAMS ARE BEING FABRICATED AND INSTALLED. THE TWO CABLES IN EACH PLANE WILL BE TENSIONED SIMULTANEOUSLY.

C-20

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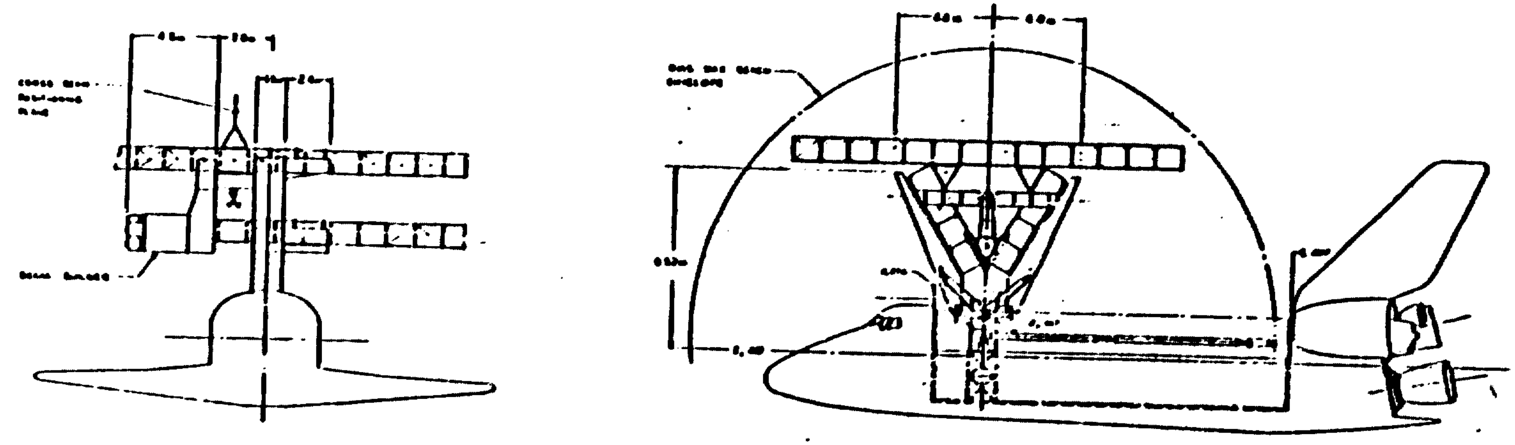
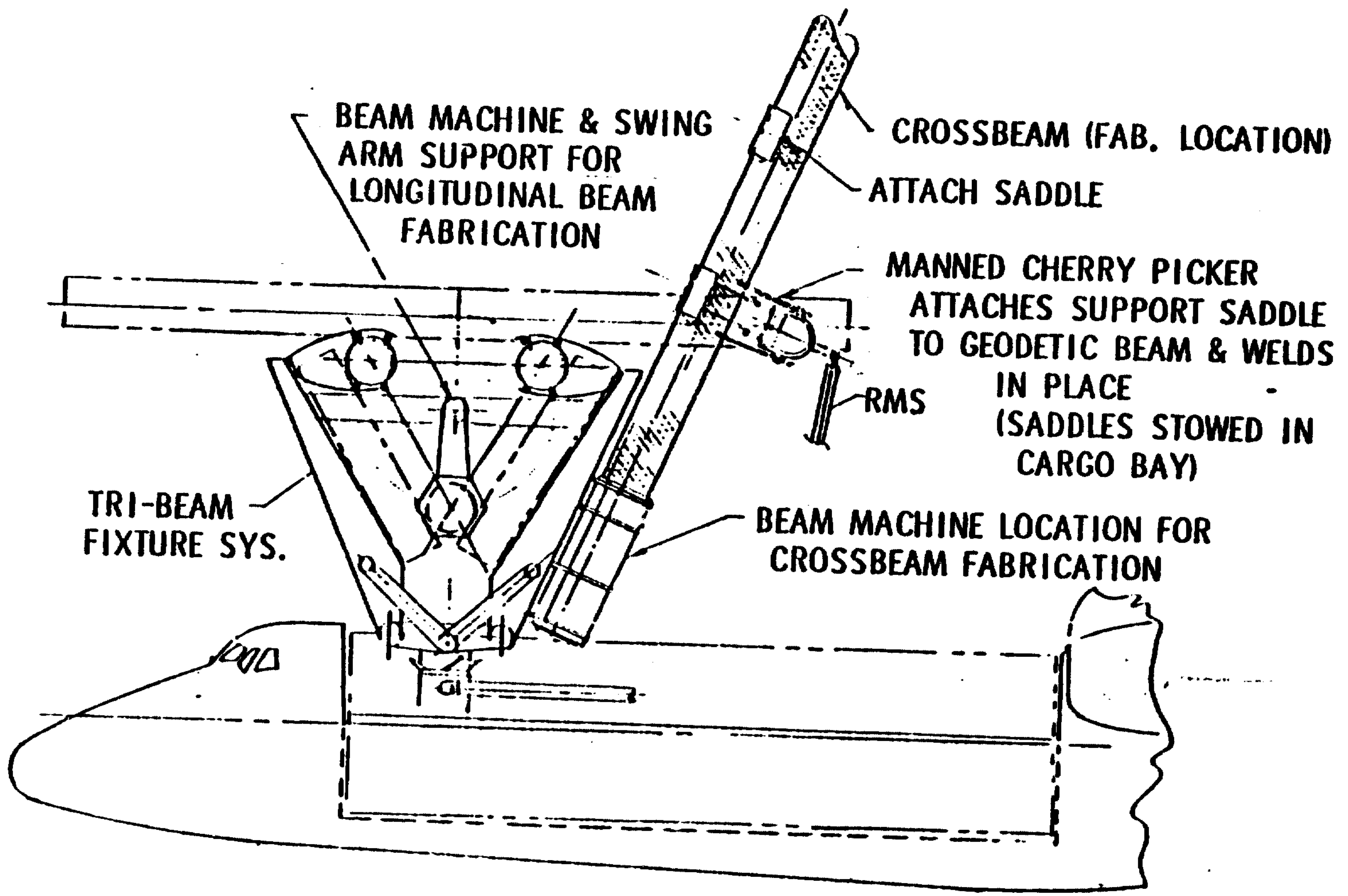
GEODETIC BEAM CONSTRUCTION IMPLICATIONS TO TRI-BEAM FIXTURE SYSTEM



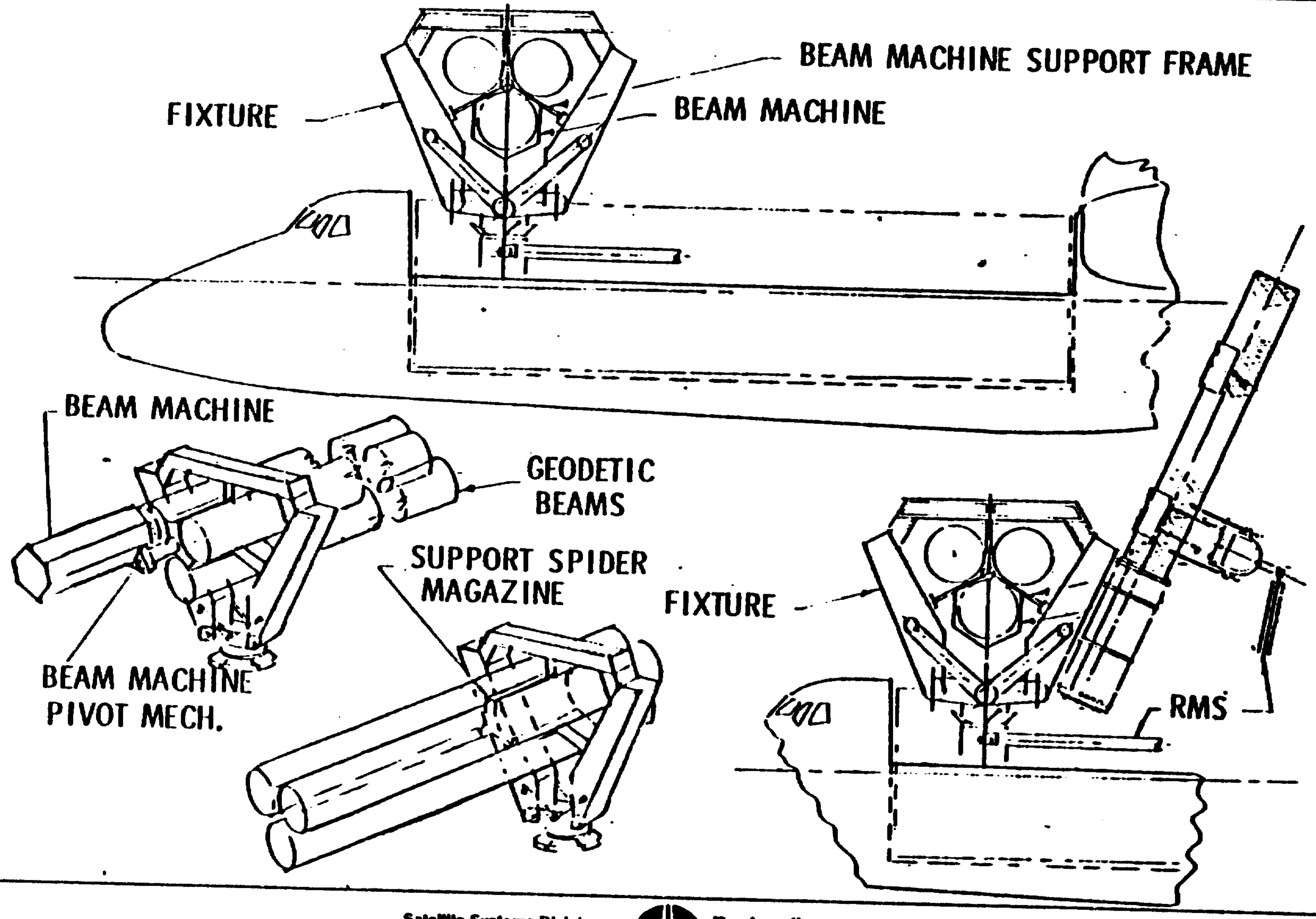
MODIFICATIONS REQUIRED

- ADAPTION OF SWING ARM TO SUPPORT BEAM MACHINE
- REDESIGN OF LONGITUDINAL BEAM ROLLER SUPPORT ARMS
- REDESIGN OF AND MODIFICATIONS TO WELDING HEAD ASSEMBLIES
- REDESIGN OF TRANSVERSE BEAM POSITIONERS
- PROVISIONS ADDED TO BASIC FRAME FOR BEAM MACHINE SUPPORT DURING CROSSBEAM FABRICATION
- ADDITIONAL SUPPORTS AND WELD HEAD ASSEMBLIES REQUIRED TO ATTACH SADDLES TO LONGITUDINAL BEAMS
- ALIGNMENT & SUPPORT PROVISION PLUS WELDING FACILITIES REQUIRED TO ATTACH SADDLES TO CROSSBEAMS DURING FABRICATION

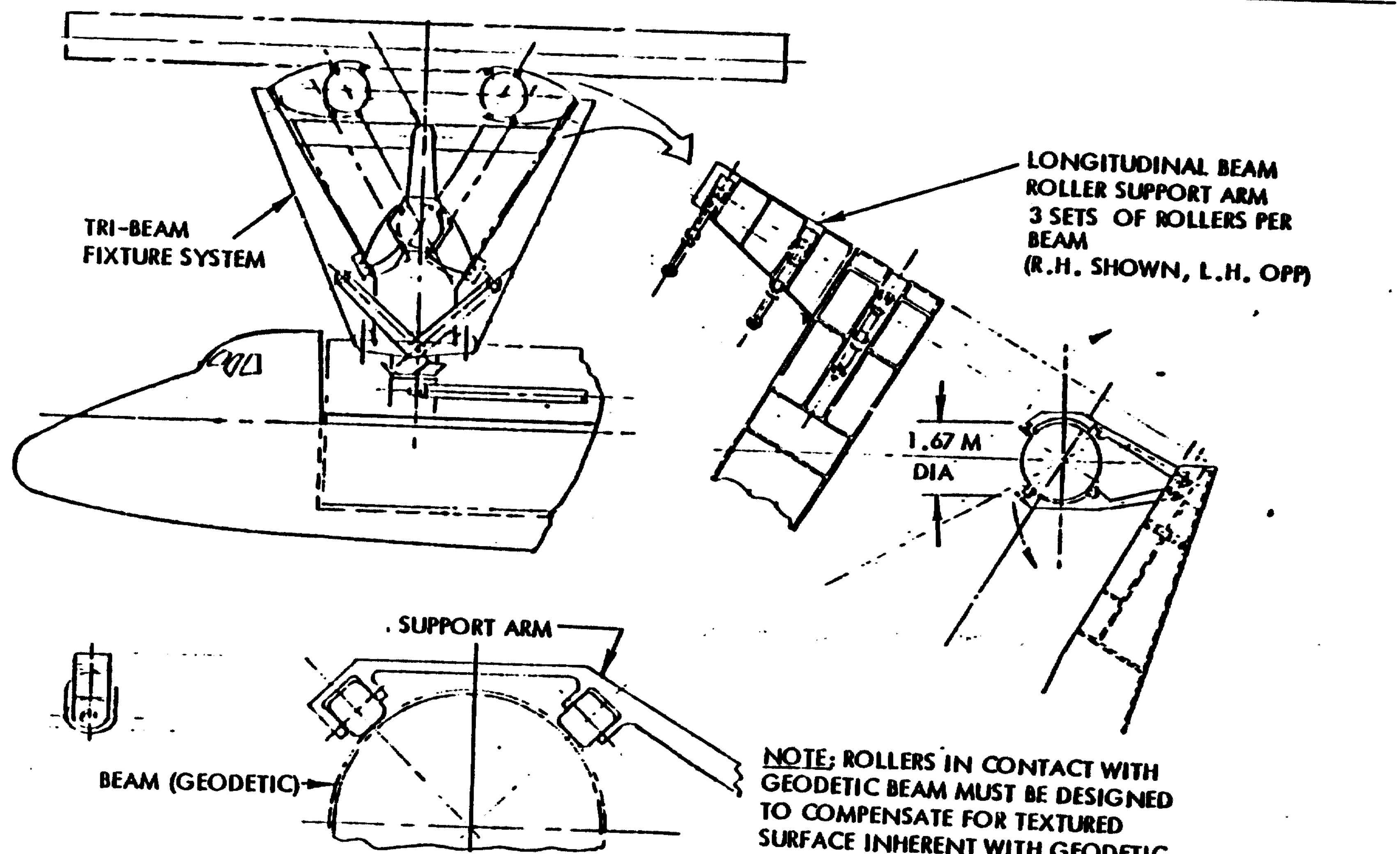
GEODETIC TRI-BEAM ASSEMBLY



GEODETIC BEAM—FIXTURE CONCEPT



FIXTURE SYSTEM—GEODETIC BEAM SUPPORT



NOTE: ROLLERS IN CONTACT WITH GEODETIC BEAM MUST BE DESIGNED TO COMPENSATE FOR TEXTURED SURFACE INHERENT WITH GEODETIC BEAM DESIGN

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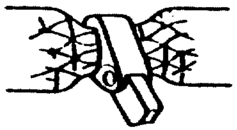
GEODETIC BEAM CONFIGURATION
IMPLICATIONS TO CONSTRUCTION OPERATIONS/EQUIPMENT

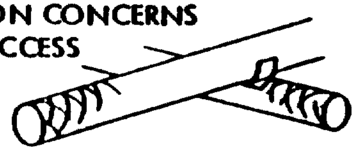
OPERATIONS

- HANDLING CIRCULAR BEAMS, SADDLE JOINTS, SPACERS
- AVOIDING DAMAGE TO THIN STRUCTURAL ELEMENTS



(MINIMIZE RADIAL PRESSURE)

- JOINING CYLINDRICAL BEAMS WITH SADDLE JOINTS
  - ATTACH POINT ACCESS
  - STOWAGE CONCERNS
  - ORIENTATION CONCERNS
  - WELDING ACCESS



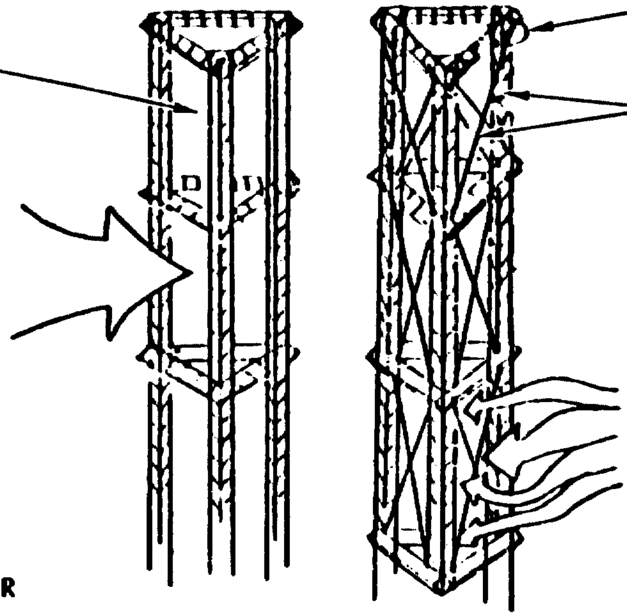
- COMBINATIONS OF ERECTABLE AND SPACE-FABRICATED CONSTRUCTION CONCEPTS

EQUIPMENT

- SPECIALLY SHAPED END EFFECTORS, BRUSHES, ROLLERS, GUIDES

- LOW-PRESSURE GRASP FORCES, SLOW SPEED EQUIPMENT, WITH LOW ACCELERATIONS
- SPECIAL SENSORS MAY BE REQUIRED
- SPECIAL JOINT-HANDLING END EFFECTORS
- SPECIAL WELDING DEVICES
- SPECIAL ALIGNMENT DEVICES, HOLDING FIXTURES
- SPECIAL ASSEMBLY FIXTURES

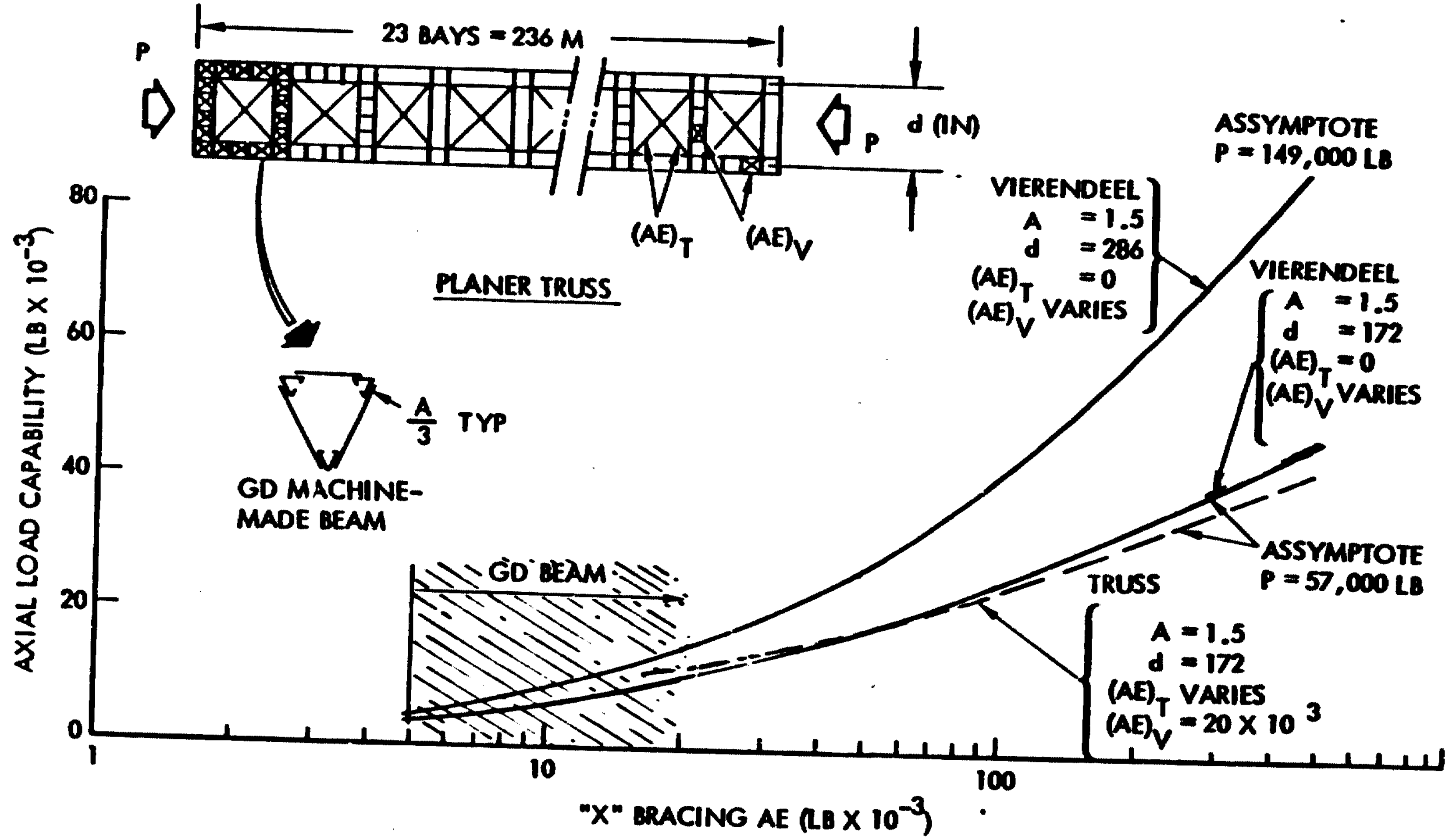
VIERENDEEL VS CORD-BRACED TRIBEAM CONSTRUCTION COMPLEXITY



VIERENDEEL	CORD-BRACED
NO CORD INSTALLATIONS	CORD ATTACH GUSSET PLATE INSTALLATION
	CORD INSTALLATION AND TENSIONING (SEQUENTIAL FORCE APPLICATION - SPECIAL TOOL)
CLEAR ACCESS INTO BAY	RESTRICTED ACCESS INTO BAY
MINIMAL VISIBILITY PROBLEMS	POTENTIAL PROBLEMS OF VISIBILITY OF SMALL DIAMETER CORDS OR STRAPS
NO CAPABILITY FOR FINAL ALIGNMENT	CAPABILITY FOR FINAL ALIGNMENT

CONCLUSION: ADDITIONAL TIME, COST & COMPLEXITY VS VIERENDEEL

VIERENDEEL / "X" BRACED TRUSS AXIAL CAPABILITY



COMPARISON OF CONFIGURATIONS (STRUCTURAL CONSIDERATIONS)

CONCEPT	MACIL MADE BEAM	ALLOW T/W	COMMENTS	REQ'D NO. OF PROPULSN MODULES	STATIC ATT'DE CONTROL ERROR (MIN)	ESTIM MODAL FREQ (HZ)	SETTLING TIME TO 2 MIN ERROR	STRUCT VERSAT-ILITY FOR ATP
ROCKWELL TRIBEAM WITH "X" BRACING		>.20	.040 CAPS .080 CORDS .012 IN.^{2} AREA OF "X" BRACING STRIPS	5 WITH SMALL MARGIN	2.2	.01	NEGL	HIGH
VIERENDEEL		.13	.070 CAPS .080 CORDS	6 WITH LARGE MARGIN	21.6	.003	3.5 HRS	MINIMAL
VIERENDEEL		.22	43 LONG BARS .13 IN STOCK 5 FT DIA	5 WITH SMALL MARGIN	.4	.025	0	HIGH
TRIBEAM WITH "X" BRACING		>.22	40 LONG BARS .13 IN STOCK 5 FT DIA	5 WITH SMALL MARGIN	< .4	> .025	0	HIGH
GEODETIC BEAM CLUSTER		.20	60 LONG BARS .13 IN STOCK 16 FT DIA	5 WITH SMALL MARGIN	.55	.030	0	LOW

CRITICAL UNKNOWNS

<u>GD MACHINE MADE BEAM</u>

- LAP JOINT STRENGTH - SHEAR & TENSION LOADS
- LOCAL CAP HANDLING LOADS

<u>MAC DAC GEODETIC BEAM</u>

- LOCAL TANGENTIAL LOAD THAT CAN BE APPLIED (LOCAL WELD & ADJACENT STRUCTURE BEHAVIOR)
- HANDLING LOADS
- POTENTIAL OF BEAM MACHINE TO FABRICATE INTERMEDIATE FRAMES (IF REQUIRED)
- TECHNIQUE FOR LOCAL ATTACHMENTS
- SUITABILITY TO SUSTAIN ROLLING, RUBBING LOADS ETC...
- CONCEPT SUITABILITY IN REGIME OF .10 TO .15 INCH PULTRUDED RODS

APPENDIX D

THRUST STRUCTURE

THRUST STRUCTURE CONCEPT SELECTION

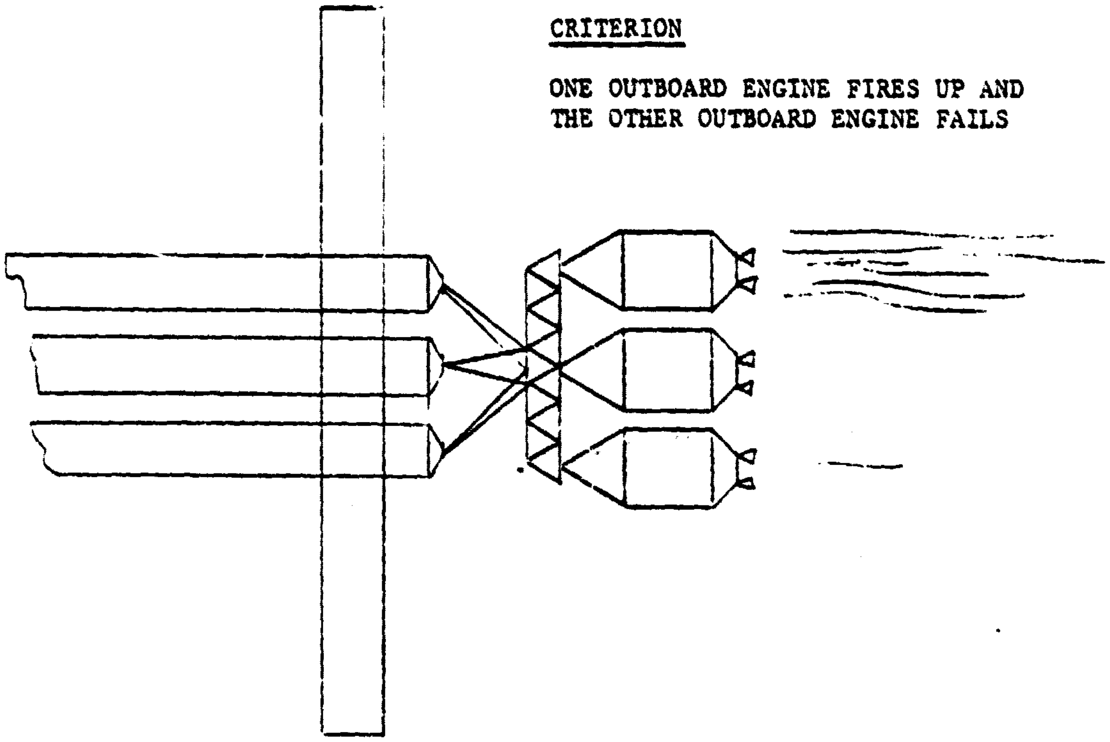
The Engineering Test and Verification Platform requires transport from its LEO construction orbit to its GEO mission orbit. A low thrust cryo propulsion concept is to be used for the required orbit transfer maneuvers. Three equally sized propulsion modules were assumed for the thrust structure analysis model used here. Two propulsion modules would be fired in parallel as a first stage and the third module would serve as the second stage. The platform is assumed to be a long (130 meters), slender, space-fabricated tri-beam configuration with various subsystem and payload modules attached. The thrust structure must be designed for thrusting along the long axis of the platform.

With this basic "problem model" four thrust structure concepts were synthesized and evaluated over gross comparative factors. The four concepts are shown schematically in Figures 1 through 4 along with key comparative features/factors.

The results are summarized in the table below.

Thrust Structure Concepts Comparison Summary

Concept / Factor	1 Truss & Tripod	2 Strap-On Propulsion	3 Strut Space Frame	4 Axial Propulsion
Stowage	Poor	Good	Good	Good
Design Complexity	Poor	Poor	Good	Good
Installation Opns Complexity	Good	Poor	Good	Poor
Impact on Propulsion	None	High	None	High
Selected Concept			✓	

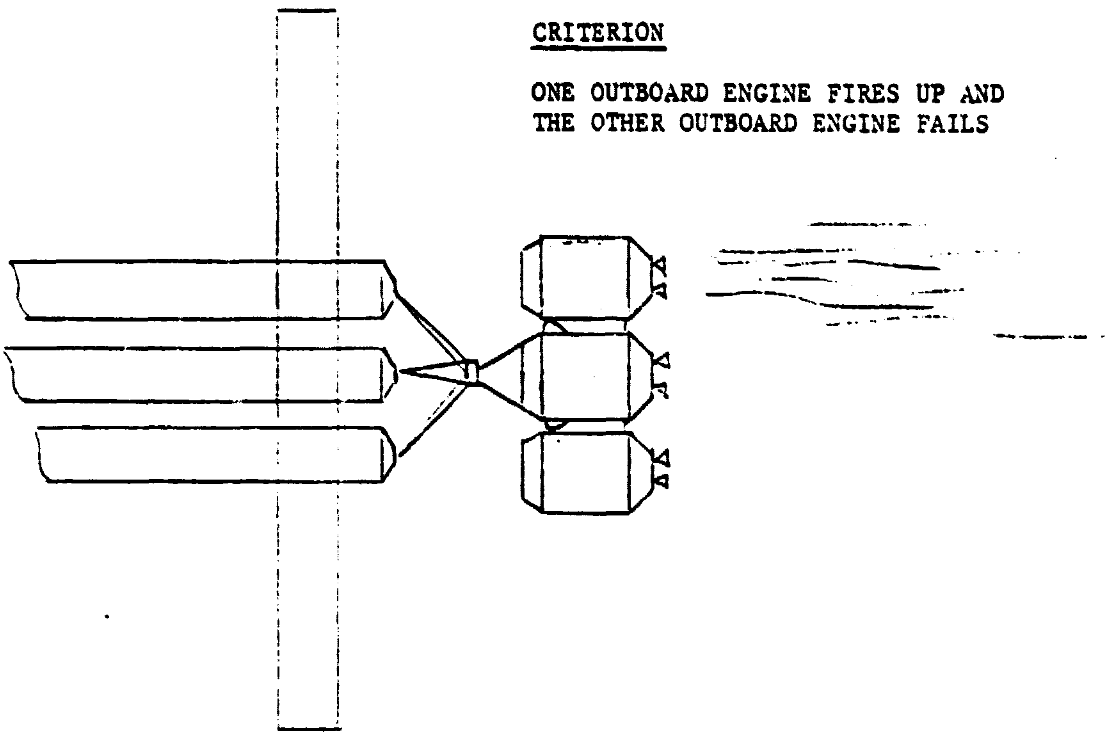


- RIGID TRUSS AND TRIPOD
- BULKY STOWAGE
- EASY INSTALLATION
- ENGINE FAILURE CRITERION - REQUIRES EXTRA STRUTS

Figure 1. Rigid Truss & Tripod

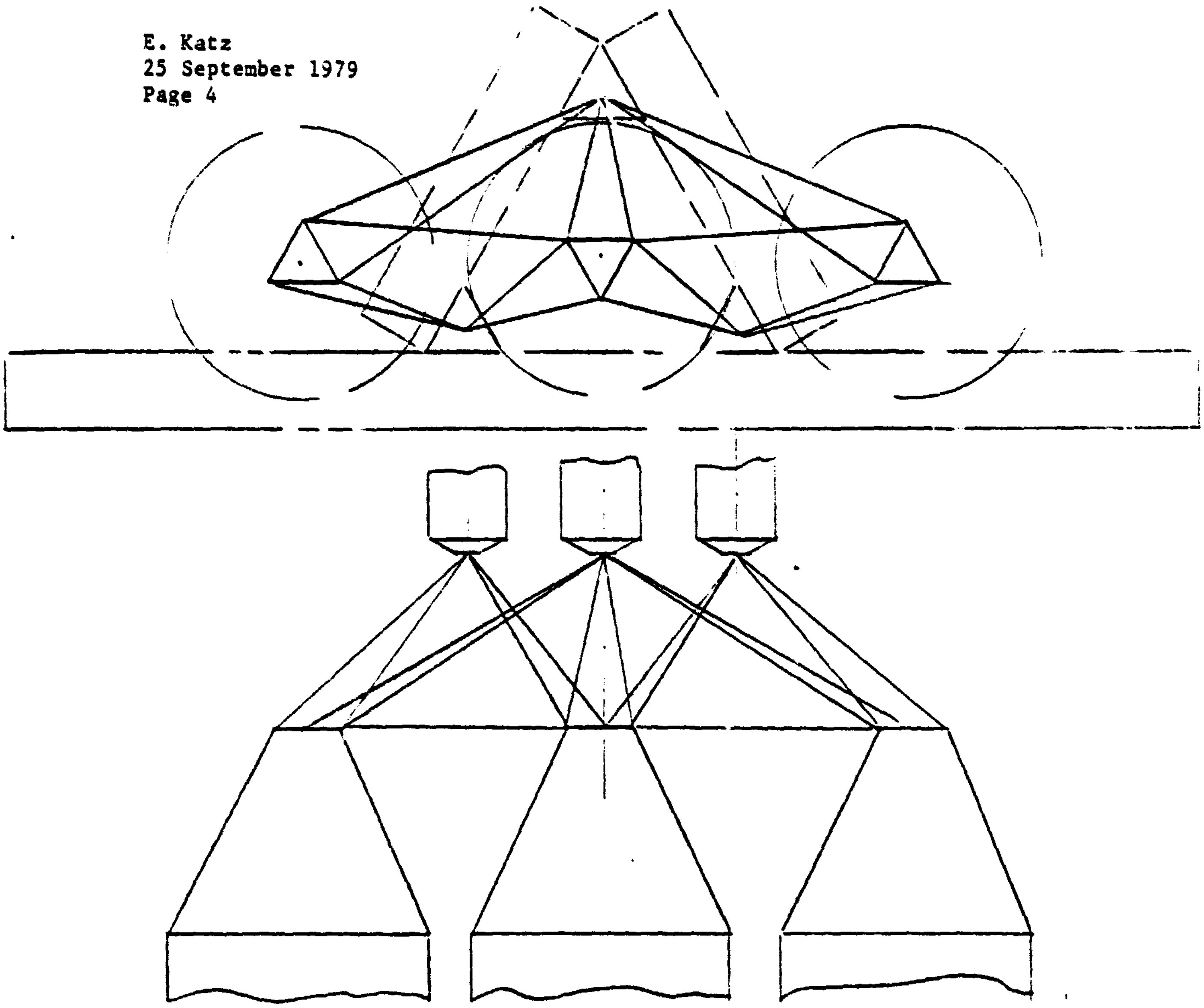
<u>CRITERION</u>

ONE OUTBOARD ENGINE FIRES UP AND THE OTHER OUTBOARD ENGINE FAILS



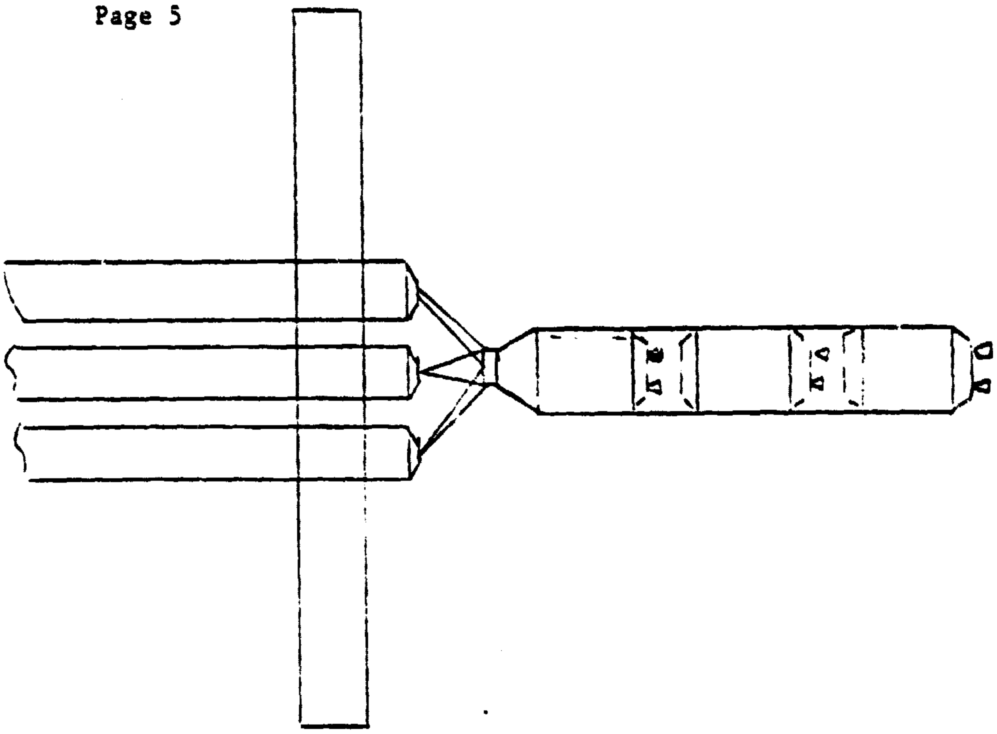
- STRAP ON ENGINES
- ENGINE FAILURE CRITERION - REQUIRES EXTRA STRUTS
- ENGINE MODULE REQUIRES MODIFICATION

Figure 2. Strap On Propulsion



- SPACE FRAME - STRUTS
- LIGHT WEIGHT
- ENGINE FAILURE CRITERION - REQUIRES NO CHANGES
- LOW VOLUME STOWAGE

Figure 3. Strut Formed Space Frame



- AXIAL ENGINE INSTALLATION
- DIFFICULT TO INSTALL ENGINES
- LOWER T/W CAUSES FUEL PENALTY
- REQUIRES PROPULSION INTERSTAGE ELEMENT
  POWER, SIGNAL, SEPARATION INTERFACE

Figure 4. Axial Propulsion Concept

Concept 3, the strut formed space frame was selected. It offers the option of being built up with individual strut members which can be efficiently packaged in the orbiter bay or of being formed with deployable subassemblies which would be easier to assemble, but possibly require more packaging volume. It is a relatively simple design of "pin-jointed" members and requires no modifications to the propulsion modules for strap-on or interstage capabilities. It also provides good access for propulsion module installation operations without the need for making special strap-on or interstage physical/electrical connections. Thus, Concept 3 was selected because it offers beneficial features in all comparative factors.

APPENDIX E

SCM STRUT SUPPORT CONCEPT

This analysis documents the rationale for the selection of the SCM support structure concept.

Three SCM support structure arrangements were developed and analyzed and are schematically illustrated in Figure 1. The prefabricated truss support concept was the arrangement that was represented on the baseline ETVP. The other two arrangements utilized erectable type strut configurations.

The evaluation factors that were utilized to evaluate and select a concept are listed below:

Evaluation Factors:

(1)	Support Structure -	Support structure concept to minimize installation complexity, be compatible with RMS capability, minimize weight and complexity of total assembly.
(2)	SCM Location -	Support structure to be compatible with acceptable locations of the SCM to minimize load impacts to the platform, and to also accommodate sensor viewing requirements.
(3)	Impact on SCM -	Support structure to be compatible with the least weight and least complex structure arrangement for the SCM.
(4)	Install Effort -	Installation to be accomplished from the orbiter utilizing the RMS. Complexity of operations and time to assemble to be minimized. Remote installation effort desirable - no EVA.
(5)	Packageability -	Support structure to be packageable for transport in the Shuttle Orbiter in the most efficient packaged configuration. Minimal complexity of restraint hardware desireable.

A description of the three SCM support structure concept follows:

Prefabricated Truss Support. This concept utilizes three truss type beams attached to the ends of the platform longitudinal members. The attachment is achieved by utilizing the standard attach ports installed on the ends of the longitudinal members. A single attach port which will accept the SCM is attached to the truss beams and located in the vicinity of the apex of the ETVP cross section. This location minimizes the bending moments imposed on the platform during GEO transfer by positioning the SCM on the opposite side of the ETVP payloads. This single point attachment arrangement, however, introduces deflections in the SCM that must be reacted by the primary structure of the SCM. Additional structural elements, therefore, will be required to accommodate the single point load reaction path.

The prefabricated beams will occupy approximately $40m^3$ of cargo bay volume. Each beam will be installed on the platform by utilizing the RMS to mate the beams to the platform attach ports. A nominal amount of time is anticipated to accomplish this installation/assembly.

Individually Assembled Strut Concept

The individually assembled strut SCM support structure concept utilizes tapered struts. These struts are completely assembled and need only to be installed on the platform. The strut lengths were adjusted in order to place the SCM on the centroid of the platform cross section. Four different length struts are required to achieve the SCM centroidal location. The centroid location was chosen because it does not introduce any additional bending loads on the ETVP.

Three attach points are located on the SCM to accept the six struts. These struts interface with the SCM structure in a compatible manner that minimizes SCM load reaction structure.

The installation procedure requires that each strut be installed individually. Consequently, the assembly time is greater than that anticipated for the baseline truss structure concept.

The individual struts package in a fairly efficient manner and require only .3m^3 of cargo bay volume.

Foldable Strut Concept

The foldable strut concept was developed in order to minimize the assembly time for the assembly and installation of the support structure and for the installation of the SCM to the support structure. For this concept the struts are assembled on the ground with appropriate folding joints which allow the assembly to be packaged in a reasonable volume. The structural arrangement has the same features as the individually assembled concept - three reaction points on the SCM. Each of the struts in this concept are of equal length. Consequently, the SCM is located slightly above the centroid (closer to the apex) of the platform so that it will slightly reduce the bending moments on the platform structure.

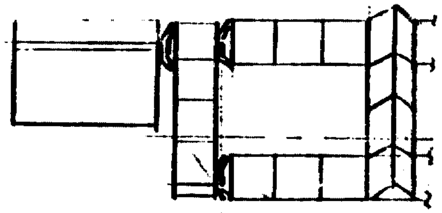
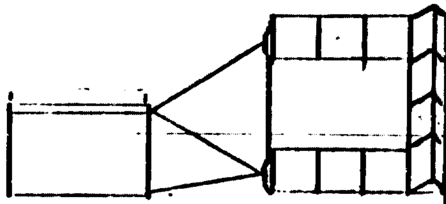
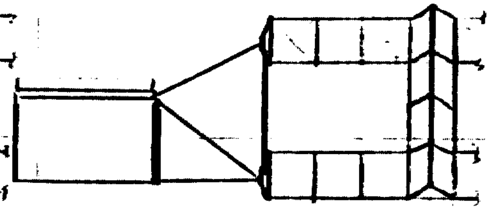
The total assembly is removed from the cargo bay by the RMS and attached to one end of the platform and then deployed in a controlled manner. The other two attachments are then mated, thus completing the SCM support structure installation. A minimum installation time is required for this concept with, however, additional complexity necessary to achieve the folding.

The volume occupied in the cargo bay is approximately 1m^3.

The foldable structural arrangement was selected for the ETVP to minimize construction time at the expense of some structural arrangement complexity. The equal length struts will minimize fabrication costs of the struts while the three point support structure attach concept will minimize the structure of the SCM. The packaging of the concept creates an acceptable arrangement and an acceptable volume utilization.

Figure 1 illustrates the three concepts and summarizes the characteristics of each.

FIGURE _ -
SCM SUMMARY

EVALUATION FACTORS	PREFAB TRUSS SUPPORT	STRUT SUPPORT INDIVIDUALLY INSTAL	STRUT S[illegible] UNIT FOLDABLE [illegible]MBLY
			
SUPPORT STRUCTURE	• TRUSS BEAMS - PREFAB, COMPOSITE, SQUARE CROSS-SECTION	• STRUTS - COMPOSITE, TAPERED, INDIVIDUAL	STRUTS - COMPOSITE, FOLDABLE ASSEMBLY
SCM LOCATION	ETVP APEX	ETVP CENTROID	ABOVE ETVP CENTROID
IMPACT ON SCM	SINGLE POINT ATTACH	THREE POINT ATTACH	THREE ATTACH POINTS
INSTALLATION EFFORT	NOMINAL	COMPLEX, TIME CONSUMING	minimum
PACKAGEABILITY	40 m^3 ENVELOPE	.3 m^3 ENVELOPE	1 m^3 ENVELOPE
SELECTED CONCEPT			✓

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APPENDIX F

ATTACH PORT CONCEPTS TRADE

ATTACH PORT CONCEPTS TRADE

The attachment concept for large subsystems (RCS modules, system control module, etc.) and for the antenna payloads is achieved with an active latching arrangement. The modules are guided into their proper position to affect the attachment. The orbiter RMS provides the berthing movement which activates this attachment process. The Space Construction Systems Analysis performed in Part I assumed a port design concept similar to the three petal neuter concept which is considered to be the baseline docking concept for the orbiter. This concept - the neuter docking arrangement - is driven by the requirement to permit manned passage thru the docking interface. The attachment of the subsystem modules and of the antennas does not require this capability and, consequently, a simpler, lighter concept would appear to be more compatible and desirable. This trade study addresses that issue - what module attaching port concept is most compatible for the installation operation?

The enclosed briefing charts define the principle requirements of an attaching port and discusses the issues of structural joining of the port to the ETVP beams, various alignment concepts (probe/drogue and petal), implications to payload packaging, and latching considerations.

The truss to truss attach port concept was selected because of its relative simplicity, good visibility of the latching interface, and good packaging density. This concept will be defined in sufficient detail required to analyze the construction process for joining the attach port to the structure, the installation procedure of modules, and the orbiter packaging arrangement.

PAYLOAD / SYSTEMS ATTACH PORTS

- REQUIREMENT

  TO BE ABLE TO ATTACH MODULES, EQUIP., ETC., TO AN ETVP CONSTRUCTED FROM BEAM MACHINE ELEMENTS.

- APPROACH

  AN ATTACH PORT JOINED TO THE ETVP'S STRUCTURAL ELEMENTS (BEAMS)

  THIS ATTACH PORT SHOULD COMPLY TO THE FOLLOWING CONSTRAINTS —

- EASE OF APPROACH
  - VISUAL SIGHT ALIGNMENT
  - MISSALIGNMENT CORRECTED DURING APPROACH
- INTERFACE
  - POSITIVE CLOCKING ORIENTATION
  - POSITIVE LOCKING & UNLOCKING CAPABILITY
- STRUCTURALLY COMPATIBLE WITH ETVP STRUCTURE
- MAINTAIN INTERFACE & APPROACH OPERATIONS SAFELY
- VISUAL VERIFICATION OF INTERFACES MATED (LATCHES LOCKED)

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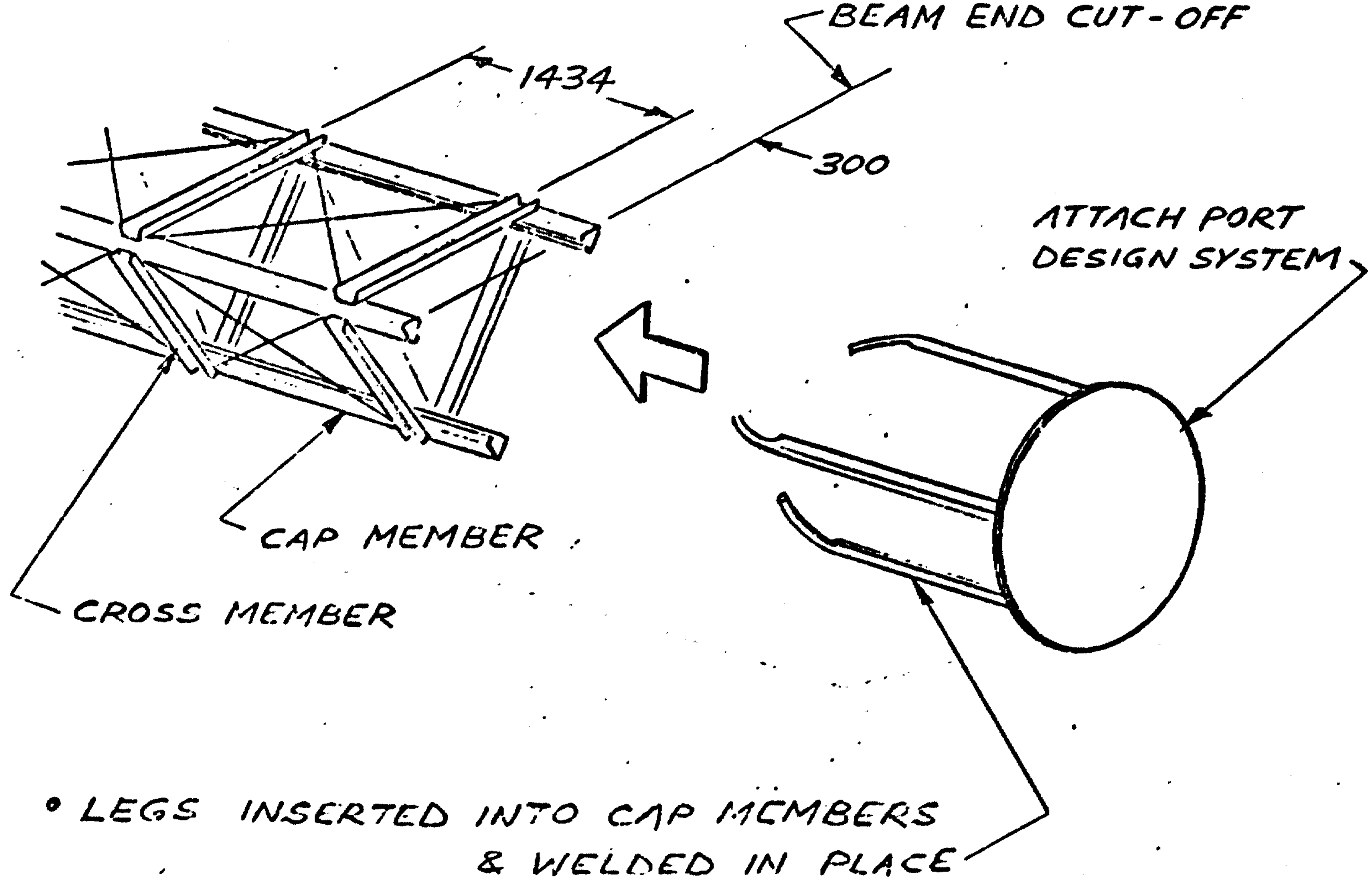
C-4

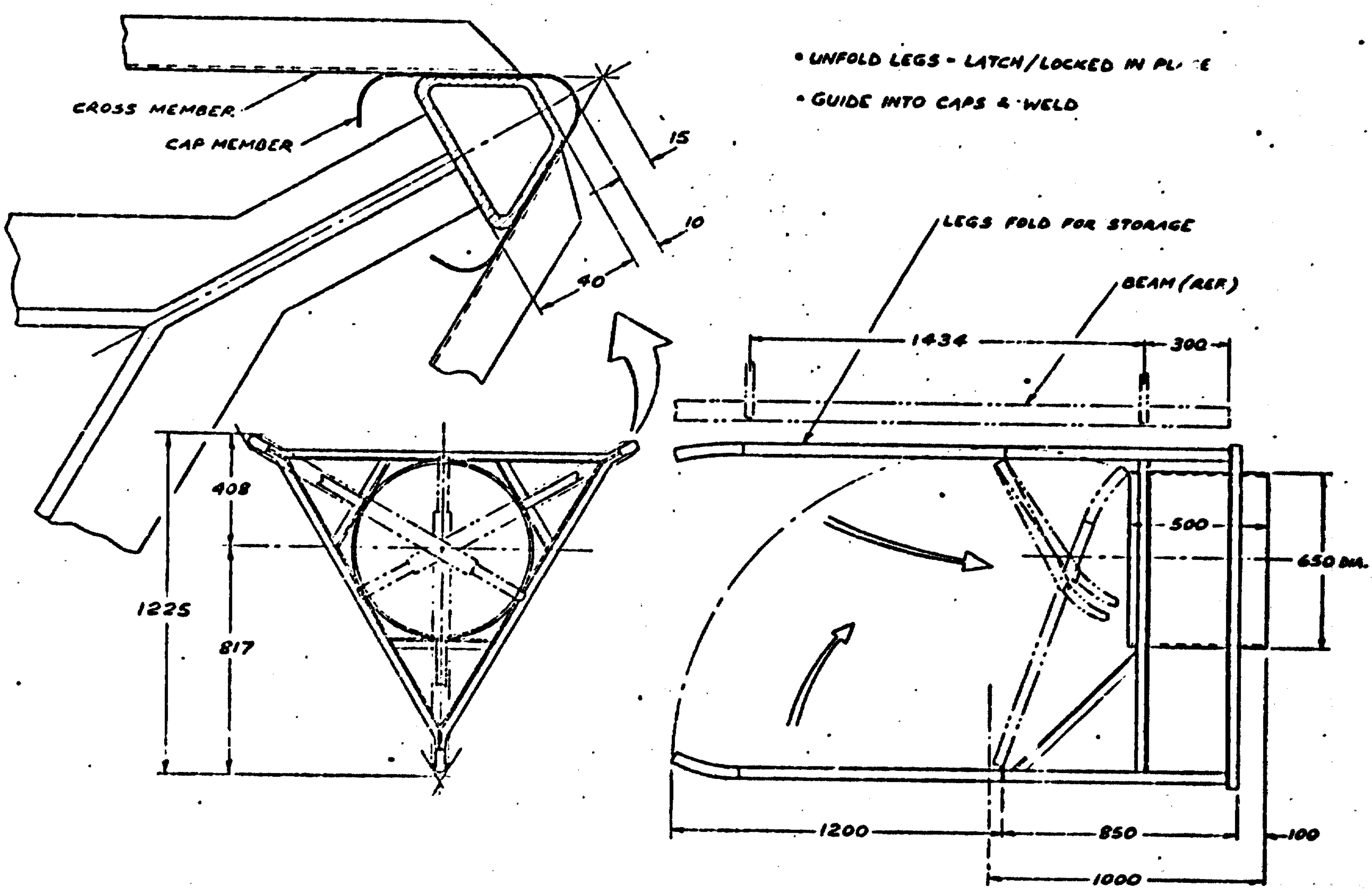
METHOD APPROACH



- LEGS INSERTED INTO CAP MEMBERS & WELDED IN PLACE

BASIC BUILDING BLOCK

- UNFOLD LEGS - LATCH/LOCKED IN PL·CE
- GUIDE INTO CAPS & WELD



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Rockwell International

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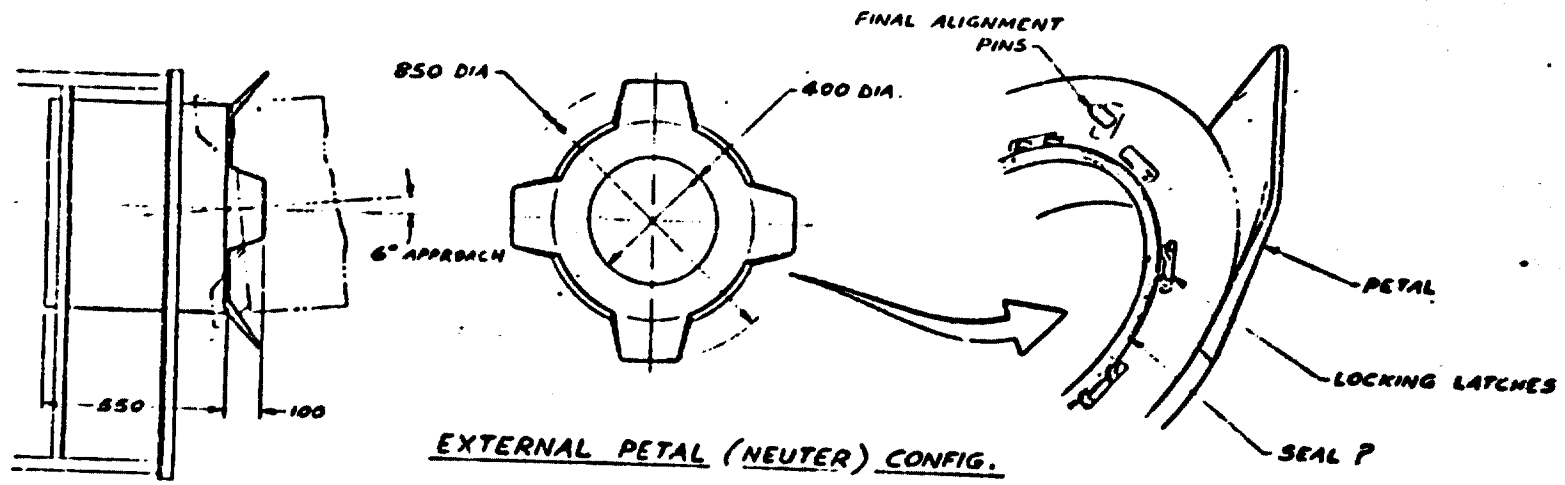
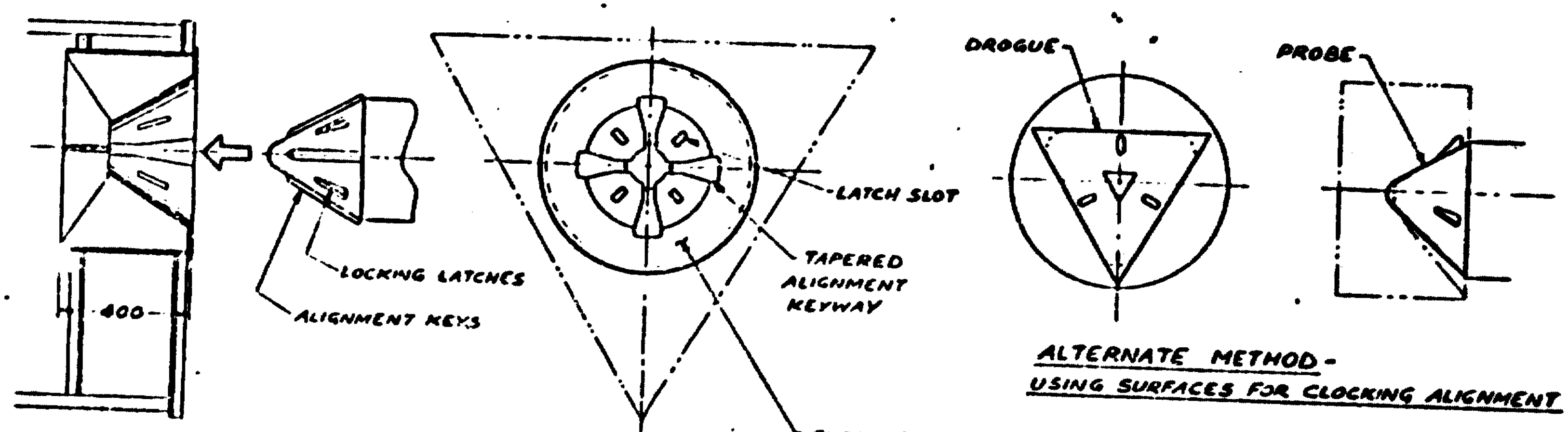
EXTERNAL VERSUS INTERNAL PORT



EXTERNAL PETAL (NEUTER) CONFIG.



ALTERNATE METHOD - USING SURFACES FOR CLOCKING ALIGNMENT

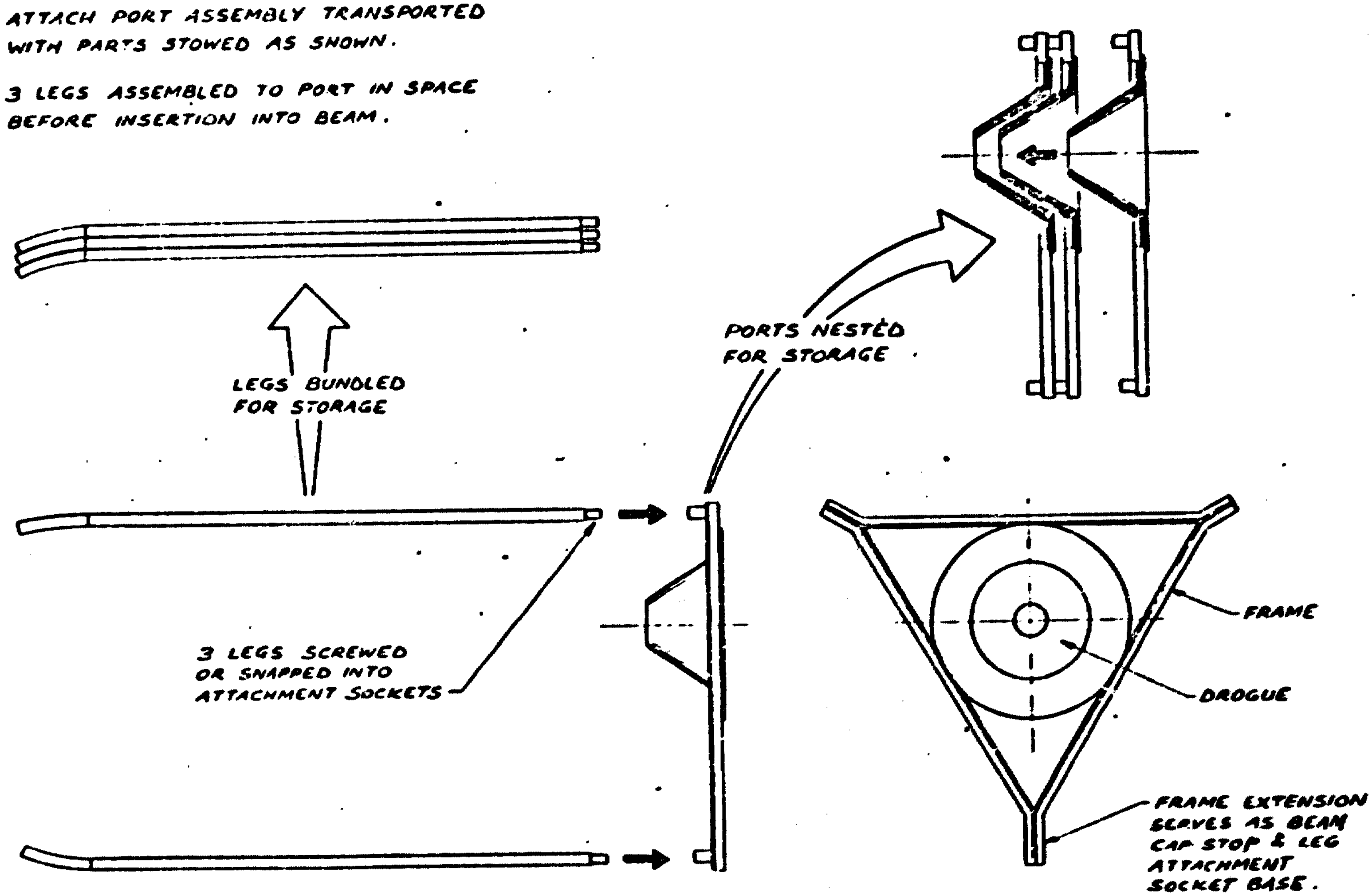
PROBE & DROGUE CONFIG.

Space Systems Group Rockwell International

(2)

PAYLOAD BAY PACKAGING CONCEPT

- ATTACH PORT ASSEMBLY TRANSPORTED WITH PARTS STOWED AS SHOWN.
- 3 LEGS ASSEMBLED TO PORT IN SPACE BEFORE INSERTION INTO BEAM.



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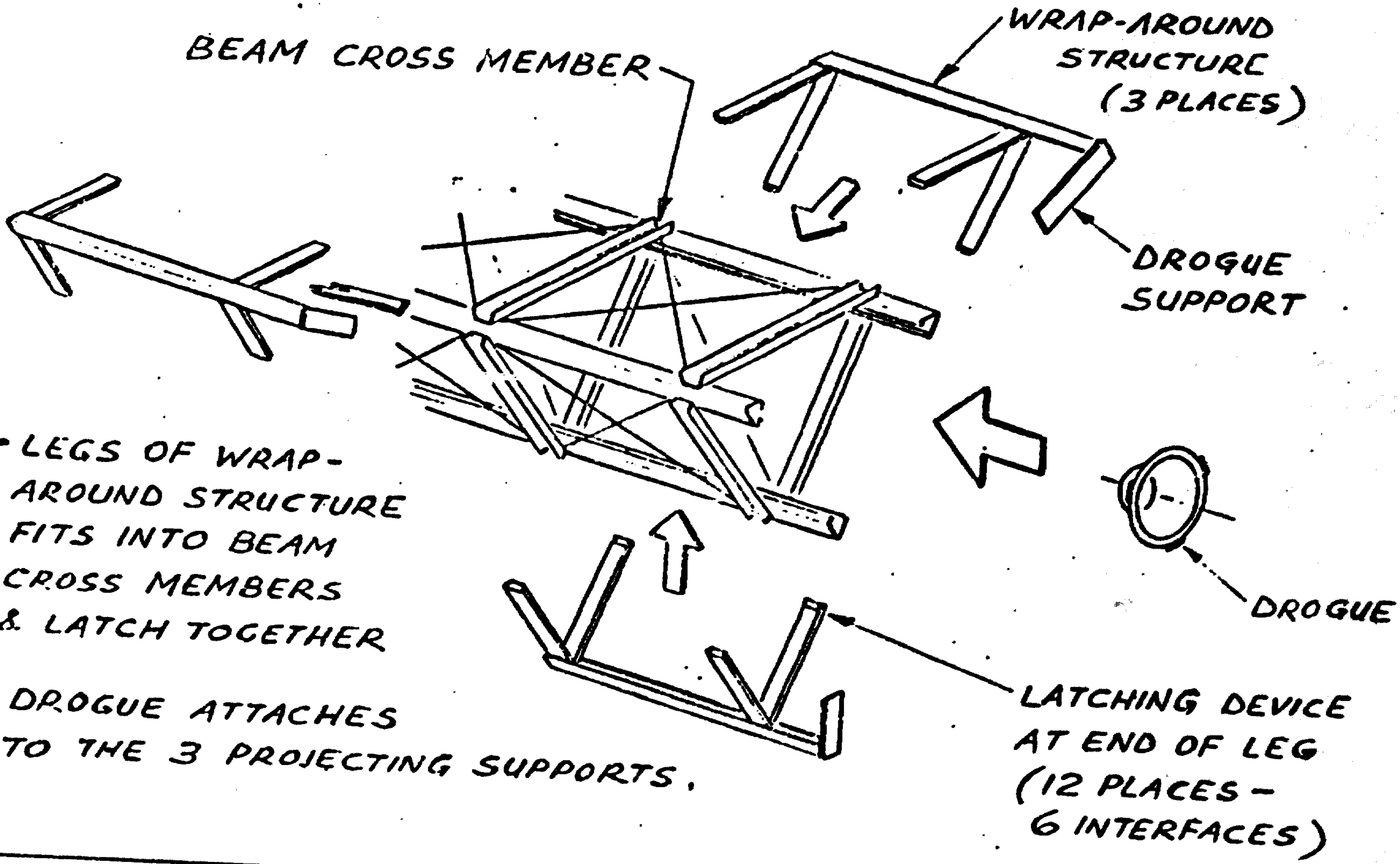
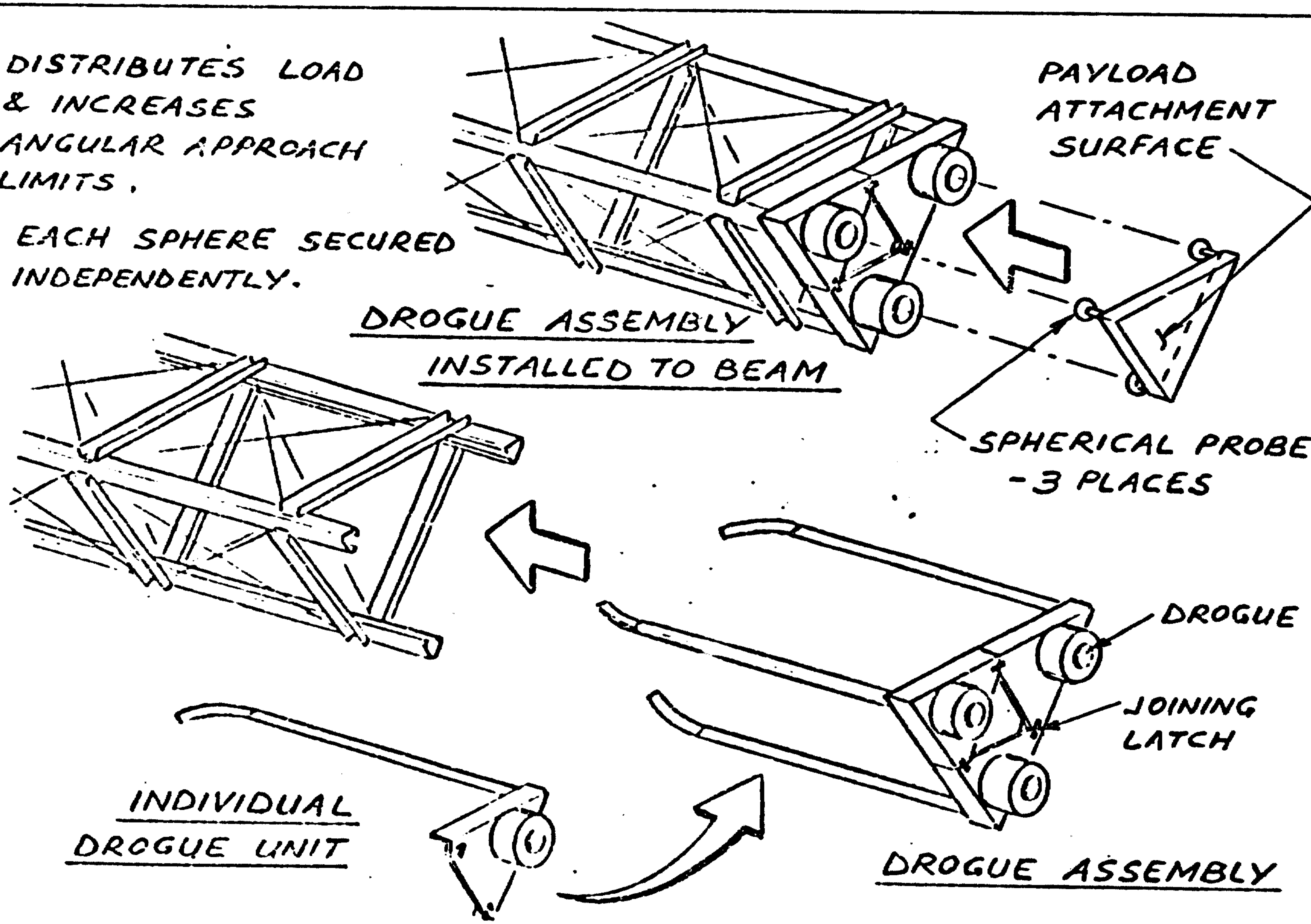
WELDLESS PORT ASSEMBLY CONCEPT



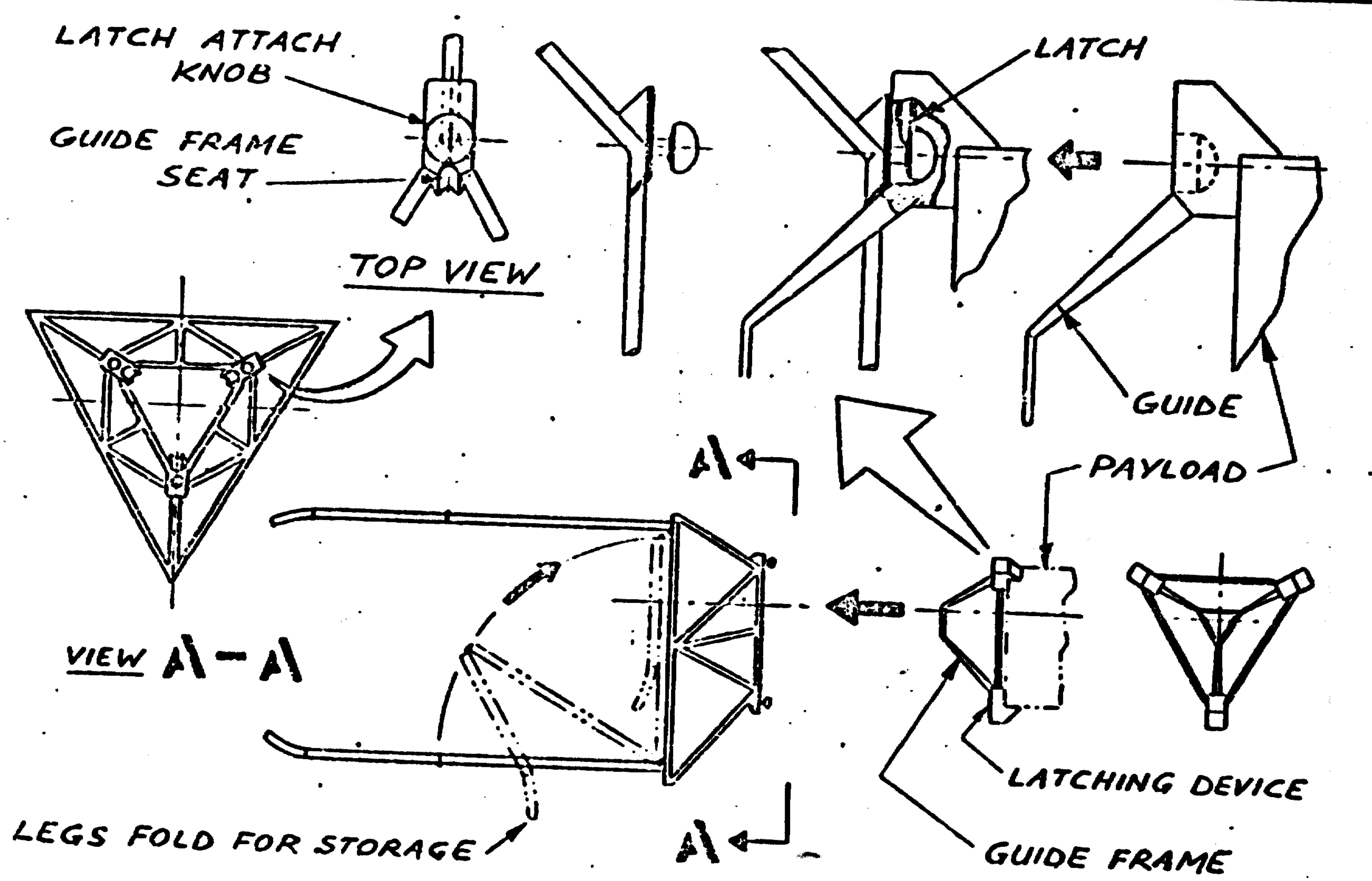
- LEGS OF WRAP-AROUND STRUCTURE FITS INTO BEAM CROSS MEMBERS & LATCH TOGETHER
- DROGUE ATTACHES TO THE 3 PROJECTING SUPPORTS.

3 CORNERED BEAM END ATTACHMENT CONCEPT

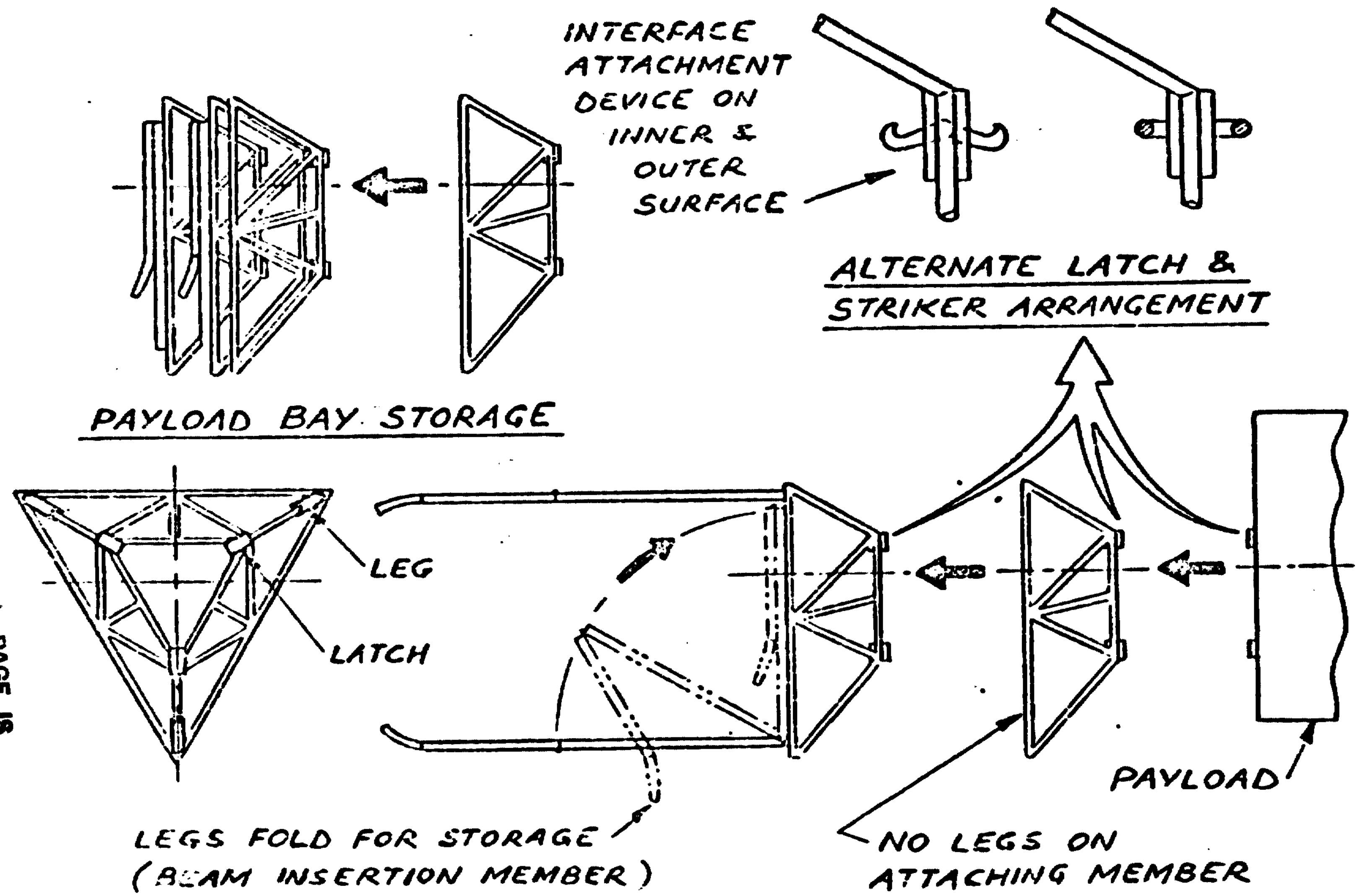
- DISTRIBUTES LOAD & INCREASES ANGULAR APPROACH LIMITS.
- EACH SPHERE SECURED INDEPENDENTLY.



BEAM END TRUSS ATTACHMENT PORT



TRUSS TO TRUSS ATTACH PORT CONCEPT



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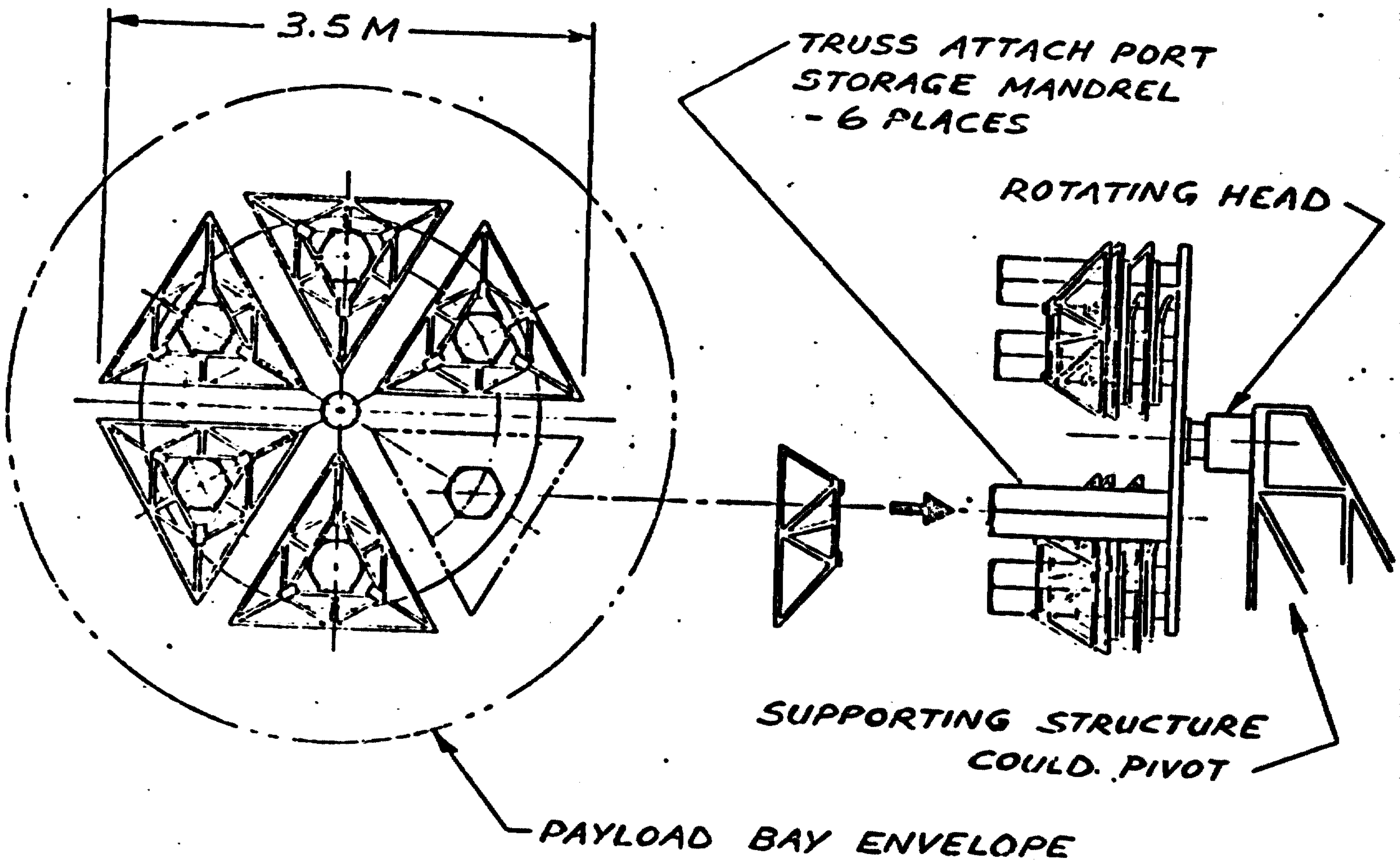
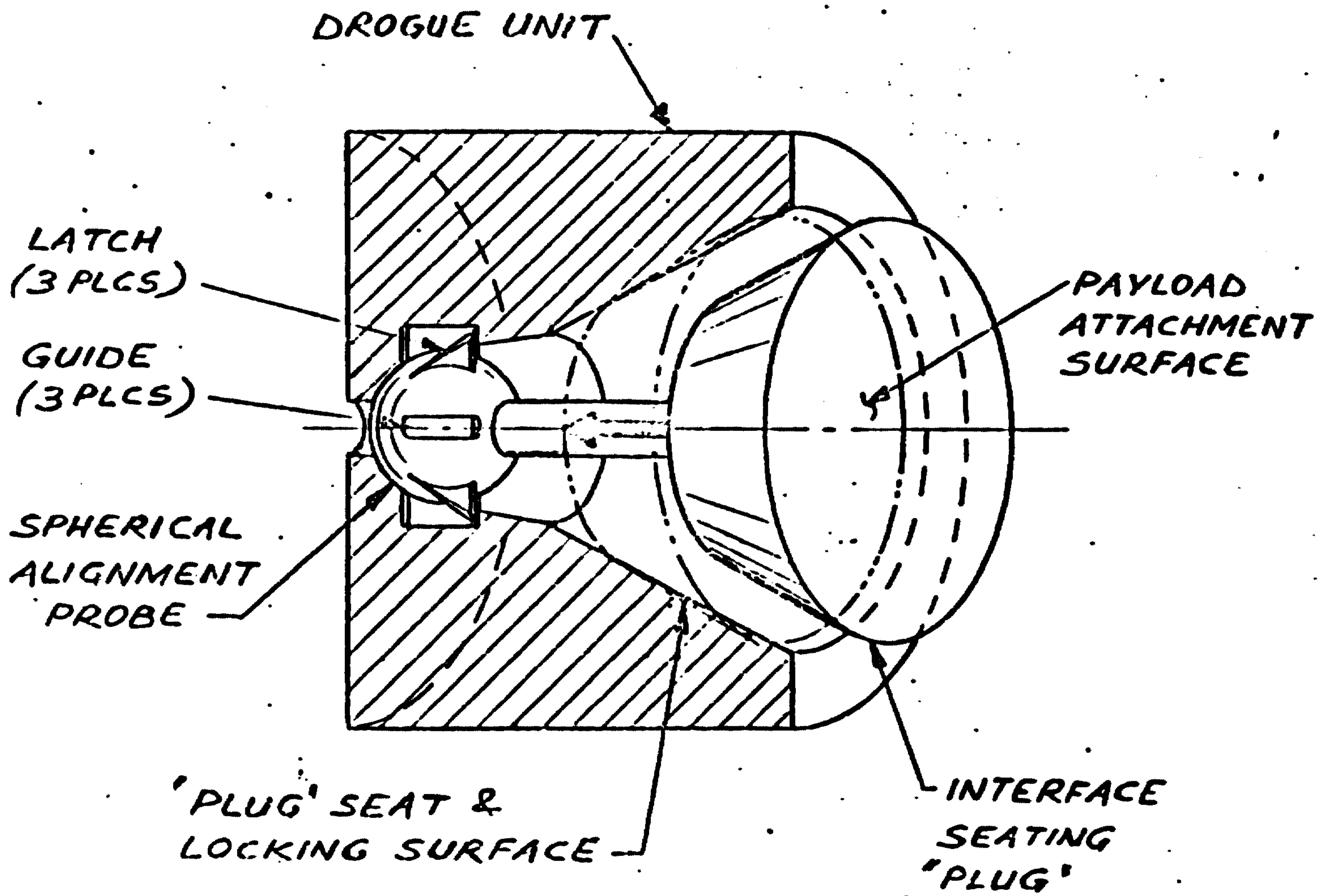
TRUSS ATTACH PORT - PAYLOAD BAY STORAGE CONCEPT



PILOT ATTACH & PULL-IN APPROACH
2 STAGE LATCHING CONCEPT



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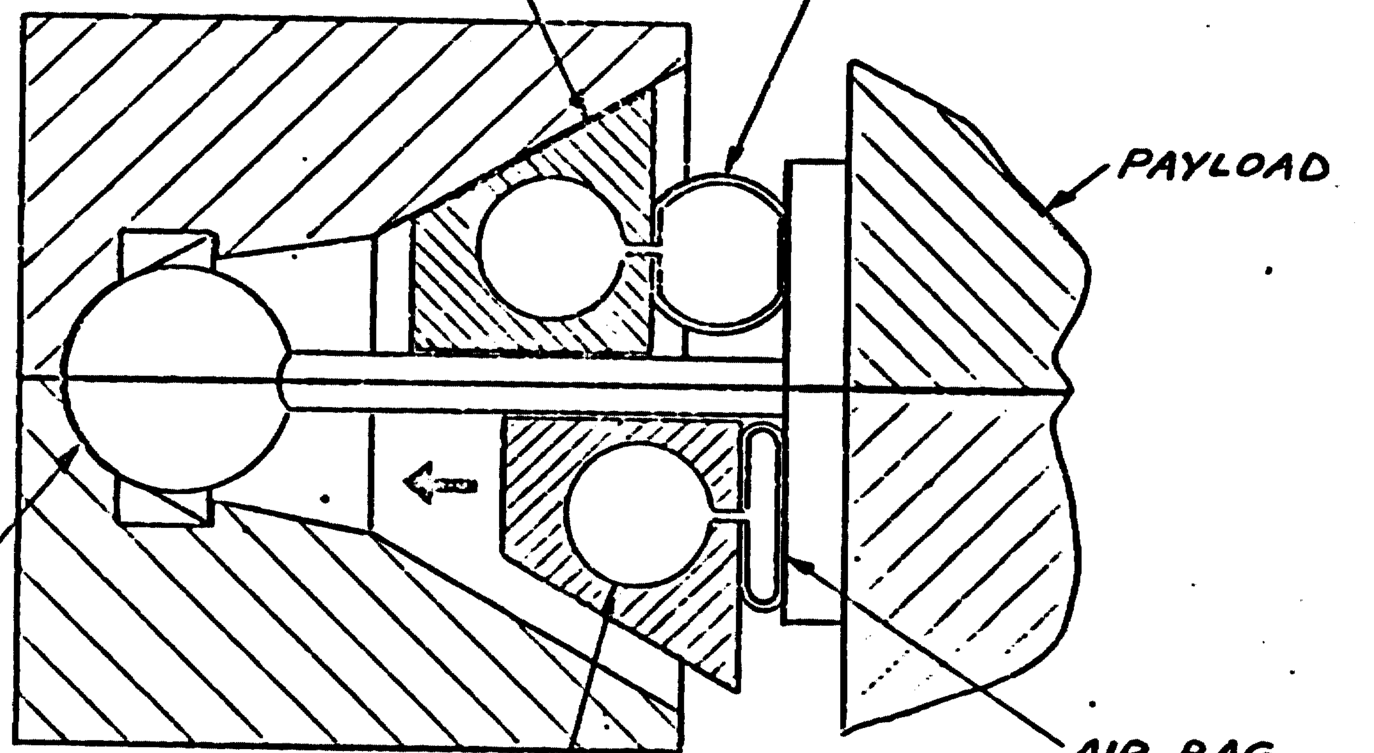
PNEUMATIC INTERFACE SEATING CONCEPT



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APPENDIX G

USER SURVEY RESULTS

USER RESPONSE

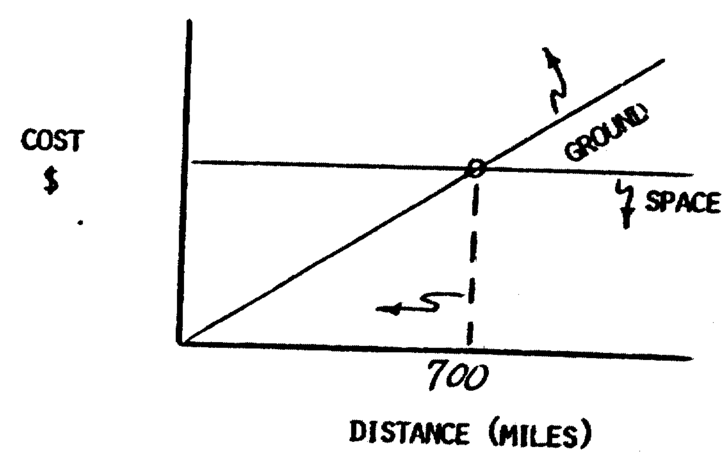
DON NOWAKOSKI -- WESTERN UNION

ATP CONCEPT	PROBLEMS	COMMENTS
• SUPPORTS CONCEPT	• SWITCHING • TIMING • BUFFERING • COST - JUST ADV WESTAR/TDRSS OVER 10 YEARS IS $500 M 2 NASA 1 WU 1 SPARE • NO DEVELOPMENT - OFF SHELF	• FAVORS NON-CONTIGUOUS BEAMS

USER RESPONSE

. WALTER MORGAN -- COMSAT

ATP CONCEPT	PROBLEMS	COMMENTS
• SUPPORTS	• DUBIOUS THAT NASA WILL FUND. BELIEVES CONSORTIUM OR COMSAT, WU, ITT, SBS, ASC WILL FIRST. • DUBIOUS OVER TELEOPERATOR - TOO COSTLY	• PREFERS INTERLEAVE & BFN OVER SCANNING SPOT



USER RESPONSE

JOHN McELROY -- NASA HEADQUARTERS

ATP CONCEPT	PROBLEMS	COMMENTS
• LIKES TESTING CONCEPT • LIKES MULTI-BEAM ANTENNA DEVELOPMENT • WANTS SWITCHING BETWEEN 20/30 AND 12/14 GHz	• MAY NOT BE NECESSARY IF 20/30 BAND WORKS OUT (X5 REUSE) AND 12/14 (X8) WITH 0.5 DEGREE ANTENNAS. 2 METERS @ 20/30 3.5 METERS @ 12/14	• ONLY HAVE 0.5 GHz AT 12/14 TO BACK-UP 2.5 GHz BANDWIDTH AT 20/30. • NEW NASA GUIDELINES (1) JOINT WITH COMMERCIAL (2) NASA TIME LIMITED IN VENTURE (3) FULL TRANSFER • KEY TO ADEQUATE CAPACITY IS 20/30 GHz DEVELOPMENT

USER RESPONSE

. FRANK NORWOOD -- JOINT COUNCIL ON EDUCATION TV

ATP CONCEPT	PROBLEMS	COMMENTS
• LIKES ANY VEHICLE WHICH WOULD DO EDUCATIONAL JOB.	• COMPARED TO ATS-6 BUT EDUCATIONAL TV NOW HAS TO BE SELF-SUSTAINING. • LOT OF POTENTIAL TRAFFIC, BUT SCHOOL TIMING PROBLEMS - LARGE USE OF VIDEO RECORDERS.	• WOULD LIKE STUDENT TALK-BACK CAPABILITY.

USER RESPONSE

PETER FOLDES -- GENERAL ELECTRIC

ATP CONCEPT	PROBLEMS	COMMENTS
• ATP - "FINE IDEA"		• REVIEWED BFN • GOOD SIDELOBES (-24 dB) • GOOD CROSSOVER • SHAPED AREA COVERAGE • RECONFIGURABILITY
• FINAL SYSTEM (OPERATIONAL) "RATHER NEUTRAL"		
SINGLE PWR SOURCE	• SINGLE POINT FAILURE	LANDSAT D1 1 YEAR 3 MONTHS
ECONOMY OF SCALE	• ISOLATION BETWEEN ANTENNAS	
EASY DATA TRANSFER	• MAY NOT BE NECESSARY - 20/30 GHz ON ON SMALL SATELLITES WITH 2.5 METER ANTENNAS	LANDSAT D2 20 METER APERTURE - TOO SMALL FOR SOIL MEASUREMENTS

USER RESPONSE

AL. YEH -- BELL LABORATORIES

ATP CONCEPT	PROBLEMS	COMMENTS

• "BENEFICIAL FOR COMPARISON OF ANTENNA CONCEPTS"	• FIXED BEAM SIDELOBE PROBLEM →	• SCANNING SYSTEM EASES SIDELOBE PROBLEM
		• LAB DEMONSTRATION
		  • AUTOMATIC ACQUISITION AND LEARNING
		  • QUICK DIAGNOSTICS

USER RESPONSE

. BOB HARRIS -- AT&T LONG LINES

ATP CONCEPT	PROBLEMS	COMMENTS
• NOT DESIRABLE • NOT FEASIBLE • TOO COSTLY • 1000 - 2000 TRANSPONDERS IN FINAL VERSION IS "ONLY SPECULATION"	• ATS-6 LEFT IMPRESSION THAT CERTAIN COMMUNICATION CHORES ARE ECONOMICAL WHEN THEY ARE NOT	• AT&T KEENLY INTERESTED IN SPACE DEVELOPMENTS