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LINE ITEM 3

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Solar Power Satellite

SYSTEM DEFINITION STUDY PART II

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CONTRACT NAS9-15196
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LINE ITEM 3

Solar Power Satellite

SYSTEM DEFINITION STUDY PART II

VOLUME V
SPACE OPERATIONS
(CONSTRUCTION AND TRANSPORTATION)
D180-22876-5
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FOREWORD

The SPS system definition study was initiated in December 1976. Part I was completed on May 1, 1977. Part II technical work was completed October 31, 1977.

The study was managed by the Lyndon B. Johnson Space Center (JSC) of the National Aeronautics and Space Administration (NASA). The Contracting Officer's Representative (COR) was Clarke Covington of JSC. JSC study management team members included:

Dickey Arndt	Microwave System Analysis	Andrei Konradi	Space Radiation Environment
Harold Benson	Cost Analysis	Jim Kelley	Microwave Antenna
Bob Bond	Man-Machine Interface	Don Kessler	Collision Probability
Jim Cioni	Photovoltaic Systems	Lou Leopold	Microwave Generators
Hu Davis	Transportation Systems	Lou Livingston	System Engineering and
R. H. Dietz	Microwave Transmitter and Rectenna	Jim Meany	MPTS Computer Program
		Stu Nachtwey	Microwave Biological Effects
Bill Dusenbury	Energy Conversion	Sam Nassiff	Construction Base
Bob Gundersen	Man-Machine Interface	Bob Ried	Structure and Thermal Analysis
Alva Hardy	Radiation Shielding		
Buddy Heineman	Mass Properties	Jack Seyl	Phase Control
Lyle Jenkins	Space Construction	Bill Simon	Thermal Cycle Systems
Jim Jones	Design	Fred Stebbins	Structural Analysis
Dick Kennedy	Power Distribution		

The study was performed by the Boeing Aerospace Company. The Boeing study manager was Gordon Woodcock. Boeing Commercial Airplane Company assisted in the analysis of launch vehicle noise and overpressures. Boeing technical leaders were:

Ottis Bullock	Structural Design	Don Grim	Electrical Propulsion
Vince Caluori	Photovoltaic SPS's	Henry Hillbrath	Propulsion
Bob Conrad	Mass Properties	Dr. Ted Kramer	Thermal Analysis and Optics
Eldon Davis	Construction and Orbit-to-Orbit Transportation	Frank Kilburg	Alternate Antenna Concepts
		Walt Lund	Microwave Antenna
Rod Darrow	Operations	Keith Miller	Human Factors and Construction Operations
Owen Denman	Microwave Design Integration		
		Dr. Ervin Nalos	Microwave Subsystem
Hal DiRamio	Earth-to-Orbit Transportation	Jack Olson	Configuration Design
		Dr. Henry Oman	Photovoltaics
Bill Emsley	Flight Control	John Perry	Structures
Dr. Joe Gauger	Cost	Scott Rathjen	MPTS Computer Program Development
Jack Gewin	Power Distribution		
Dan Gregory	Thermal Engine SPS's		

The General Electric Company Space Division was the major subcontractor for the study. Their contributions included Rankine cycle power generation, power processing and switchgear, microwave transmitter phase control and alternative transmitter configurations, remote manipulators, and thin-film silicon photovoltaics.

Other subcontractors were Hughes Research Center gallium arsenide photovoltaics; Varian klystrons and klystron production; SPIRE silicon solar cell directed energy annealing.

This report was prepared in 8 volumes as follows:

- I — Executive Summary
- II — Technical Summary
- III — SPS Satellite Systems
- IV — Microwave Power Transmission Systems
- V — Space Operations
- VI — Evaluation Data Book
- VII — Study Part II Final Briefing Book
- VIII — SPS Launch Vehicle Ascent and Entry Sonic Overpressure and Noise Effects

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1.0 INTRODUCTION AND OVERVIEW

This volume includes both construction and transportation data since these two topics have many interrelated factors when comparisons are made concerning the selection of power generation systems and construction locations.

This document will primarily cover the material developed during Part 2 of the SPS System Definition Study. Accordingly, construction and transportation systems and operations are described for the following combinations: 1) silicon photovoltaic CR=1 satellite constructed primarily in LEO, 2) silicon photovoltaic CR=1 satellite constructed in GEO, 3) Rankine thermal engine satellite constructed primarily in LEO and 4) Rankine thermal engine satellite constructed in GEO. Alternate photovoltaic satellites incorporating a CR=2 design and a Brayton thermal engine satellite were discussed in Part 1 documentation (D180-20689-3) and are not repeated.

Section 2 of this document consists of a summary presented in a manner to emphasize the key differences between the two power generation system options followed by differences between the two construction location options as measured by various construction and transportation factors. Data resulting from these comparisons indicate a photovoltaic satellite constructed in LEO offers the most desirable features in terms of construction and transportation factors. Recommendations for construction and transportation technology demonstrations are presented at the end of the summary.

Section 3 of this document presents detailed construction analysis in terms of the construction operations and construction base definition associated with both power generation systems concepts and both construction location options. No construction comparison of the options is included in this section since it has been incorporated in the overall summary of Section 2.

Section 4 contains the system descriptions of the Earth-to-LEO and LEO-to-GEO transportation systems used to support the various combinations of power generation systems and construction location options. The material primarily consists of the reference system descriptions since alternatives for the various transportation systems were discussed in the Part 1 documentation (D180-20689-5).

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2.0 SUMMARY

In this summary section, the construction and transportation systems are discussed together from the standpoint of how they relate to the two main issues of the study which are the comparison of 1) the power generation systems and 2) the location for their construction. Both LEO and GEO construction options have been studied for both power generation systems. In order to focus more clearly on the differences in the construction and transportation characteristics for the two pow r generation options, this first portion of the summary will be confined to the LEO construction approach. It should be noted however, the outcome of the power generation comparison is not influenced by the the construction location.

Resulting from the power generation comparison will be a judgment as to which is the preferred system from the construction and transportation standpoint. This concept will then be used in the comparison of the construction location options found in the second section of the summary. Again, both power generation systems have been investigated for both construction locations.

Assumptions and Philosophy

The key assumptions and philosophy used in the construction and transportation analysis are indicated in Table 2-1. Most of these items are self explanatory but a few require a brief explanation. Item 1 was specified in the Statement of Work. Item 2 deals with the actual amount of useful time available for construction taking into account that personnel do not work literally an entire shift (coffee breaks, etc.), and allowances also included for machine down time. Item 3 is specified to indicate no construction options were investigated which used the satellite itself to support construction equipment. Item 5 relates to the case where a given type of machine operation such as a solar array deployment, was analyzed to determine its required construction rate in LEO construction and then this same rate was used for the GEO construction approach. Item 6 deals with the thought that wherever practical, parallel construction operations were performed in order to reduce the construction rates of the equipment and at all times an attempt was made to eliminate the cases where several operations had to occur simultaneously to finish a given task. Item 9 primarily deals with the task of indexing the satellite or the terminal phase of bringing together large items such as satellite modules or antennas using propulsive devices. Item 10 identifies the two stage ballistic ballistic system as the reference cargo launch vehicle although two stage winged/winged systems were also investigated.

2.1 POWER GENERATION SYSTEM COMPARISON

The overall configuration characteristics that influence the construction and transportation of the reference photovoltaic and thermal engine satellites are shown respectively in Figure 2-1 and 2-2. The referenced 10 GW photovoltaic satellite shown in Figure 2-1 consists of 8 satellite modules, which when assembled have an overall length of 21.6 kilometers. Approximately 1300 kilometers of 20 meter beam is assembled. 112 square kilometers of solar array is installed along with 65 kilometers of powerbus. Construction of two antennas involves fabrication of structure and the placement of 1.6 square kilometers of radiating surface.

SPS-1630

Table 2-1. Assumptions and Philosophy Construction and Transportation

1. ONE YEAR CONST TIME (INCL 30 DAYS TEST AND C/O)
2. PRODUCTIVITY FACTOR OF 0.75
3. FACILITIZED CONSTRUCTION WITH ASSEMBLY LINE TYPE OPERATIONS
4. COMPONENTS MANUFACTURED ON EARTH, ASSEMBLED IN SPACE
5. SIMILAR CONST EQUIP USE SAME RATES FOR ALL CONST OPTIONS
6. PARALLEL AND DECOUPLED CONSTRUCTION WHEREVER PRACTICAL
7. CONST. ACCOMPLISHED USING CREW OPERATED OR MONITORED EQUIP/MACHINES - NO "HANDS ON" OPERATIONS
8. CREW WORK SCHEDULE
   10 HOURS PER DAY
   6 DAYS PER WEEK
   90 DAY STAYTIMES
9. NO FREE FLYING INDEXING OR DOCKING OF LARGE SYSTEMS OR MOVEMENT OF CARGO AROUND FACILITY
10. REFERENCE CARGO LAUNCH VEHICLE–TWO STAGE BALLISTIC/BALLISTIC
11. SHUTTLE GROWTH (LIQUID BOOSTER) USED FOR LEO CREW DELIVERY
12. ORBIT TRANSFER SYSTEMS USED ION ELECTRIC OR LO_2/LH_2 PROPULSION

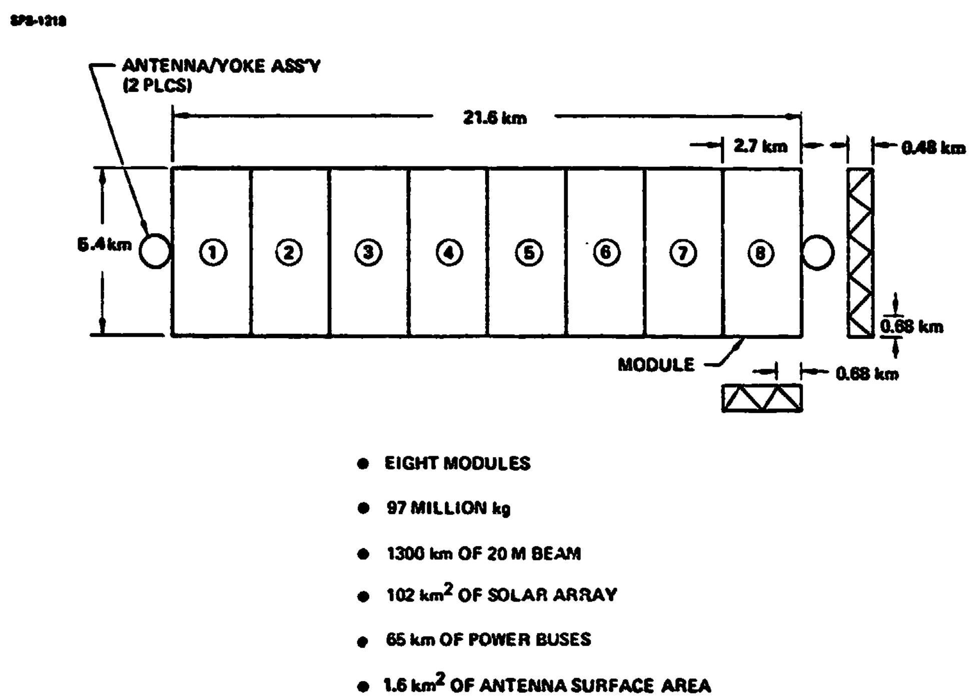

Figure 2-1. Photovoltaic Satellite Configuration

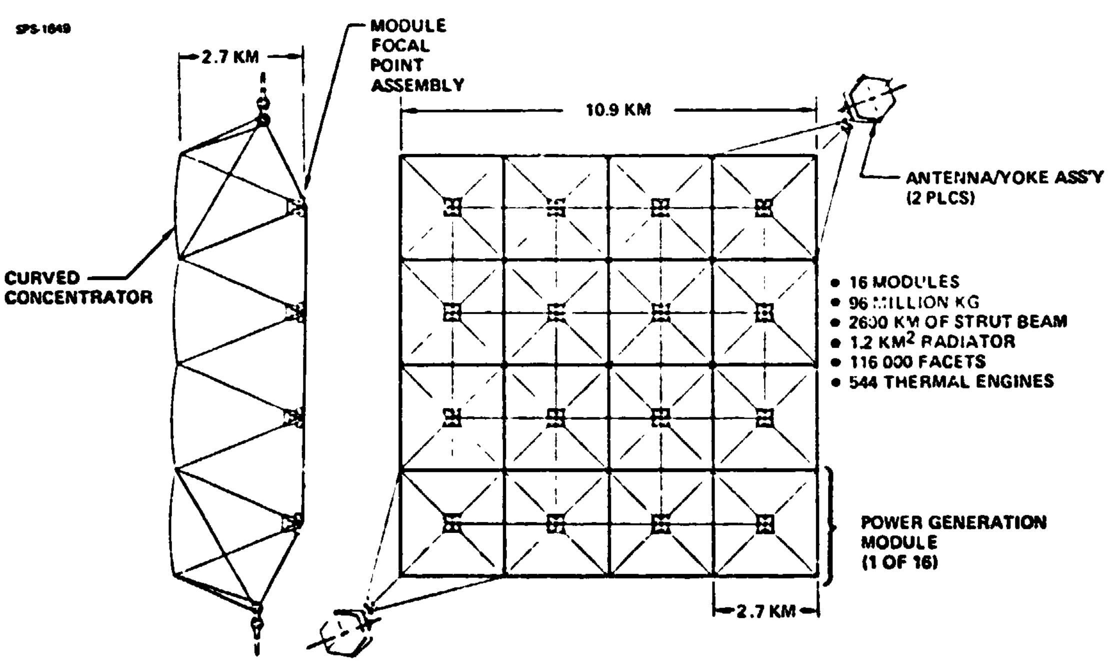

Figure 2-2. Thermal Engine Satellite Configuration

The thermal engine satellite shown in Figure 2-2 consists of 16 modules which when assembled have a planform dimension of 10.9 kilometers on a side resulting in approximately the same area as the photovoltaic satellite. A key distinguishing feature of this configuration relative to the photovoltaic satellite is that the depth of the satellite is considerably greater. Key component characteristics are also indicated, the only component directly comparable to the photovoltaic satellite being that of the structure which is approximately 2.5 times greater in length, although in this case the majority of this beam is 10 meter size rather than 20 meter.

The principal areas which will be used to compare the two power generation systems are indicated in Table 2-2. These areas have been selected to emphasize the differences between the two satellites. The approach used in the summary will be to compare both power generation system options for a given topic in two consecutive charts rather than going all the way through the photovoltaic satellite and then the thermal engine satellite followed by a comparison at the end.

2.1.1 Construction Concept

The first comparison to be made is that of the overall construction transportation concept for each satellite. As indicated earlier, the LEO construction approach will be used in making the power generation system comparison.

In the case of the photovoltaic satellite shown in Figure 2-3, eight modules and two antennas are constructed at the LEO base. All modules are transported to GEO using self-power electric propulsion. Two of the modules will transport an antenna while the remaining six modules will be transported alone. The GEO operation requires berthing (docking) the modules to form the satellite and deployment of the solar arrays not used for the transfer, followed by the rotation of the antenna into its desired operating position.

The thermal engine LEO construction concept is shown in Figure 2-4 and is similar to the photovoltaic satellite with the exception that 16 modules are constructed in LEO with 14 of these being transported alone and again 2 modules each taking up an antenna. Berthing is again required at GEO, however, no reflector facets require deployment since they are not affected by radiation when passing through the Van Allen belt so consequently are deployed while in LEO in order to simplify the construction operations at GEO.

The construction operations to be performed at the LEO construction base are as follows. In the case of the photovoltaic satellite the operations illustrated in Figure 2-5 include 1) assemble the structure to form a module 1/8 the size of the total satellite, 2) install solar arrays, 3) install power bus system, 4) install orbit transfer system, 5) install subsystems and 6) construct two antennas with their yoke and rotary joints.

Table 2-2. Power Generation System Comparison

SPS-1614

AREAS OF COMPARISON

- CONSTRUCTION BASE CONFIGURATION
- SATELLITE AND ANTENNA CONSTRUCTION OPERATIONS
- FINAL ASSEMBLY OPERATIONS
- CONSTRUCTION EQUIPMENT
- CREW REQUIREMENTS
- CONSTRUCTION SYSTEM MASS AND COST
- LAUNCH SYSTEM
- ORBIT TRANSFER SYSTEM
- TRANSPORTATION COST

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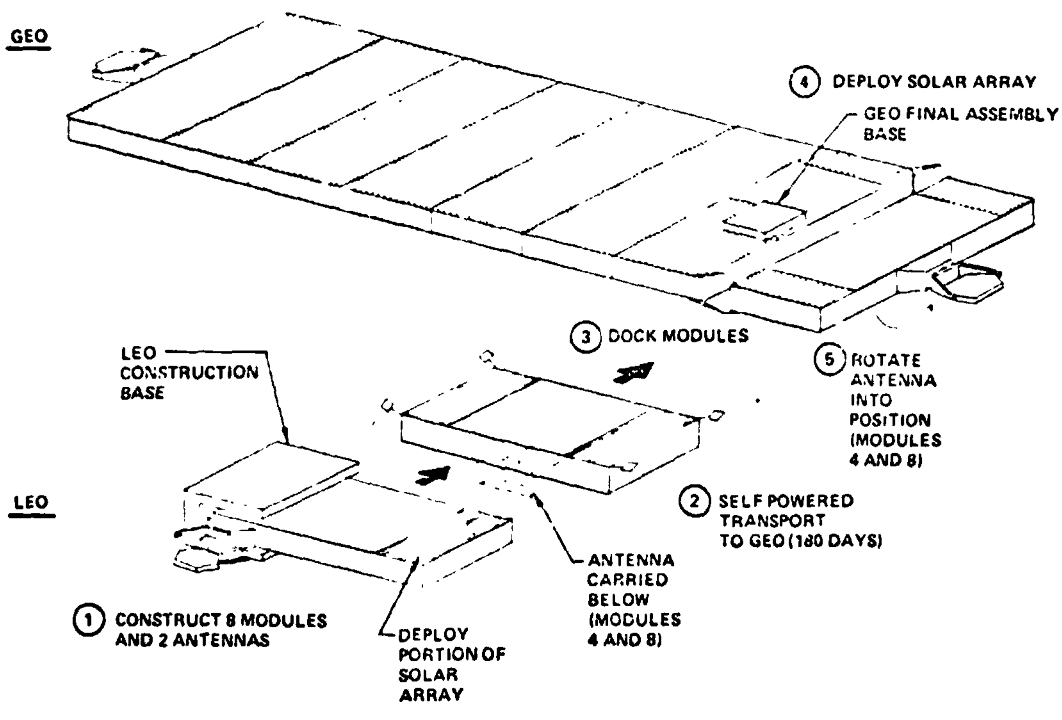

Figure 2-3. LEO Construction Concept
Photovoltaic Satellite

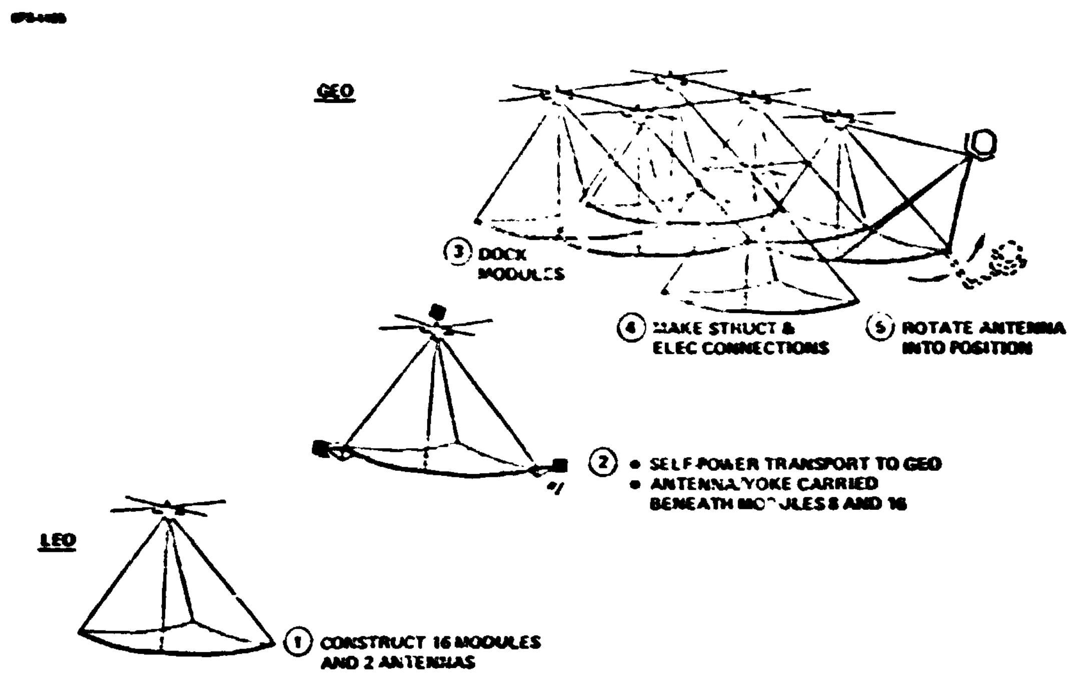

Figure 2-4. LEO Construction Concept Thermal Engine Satellite

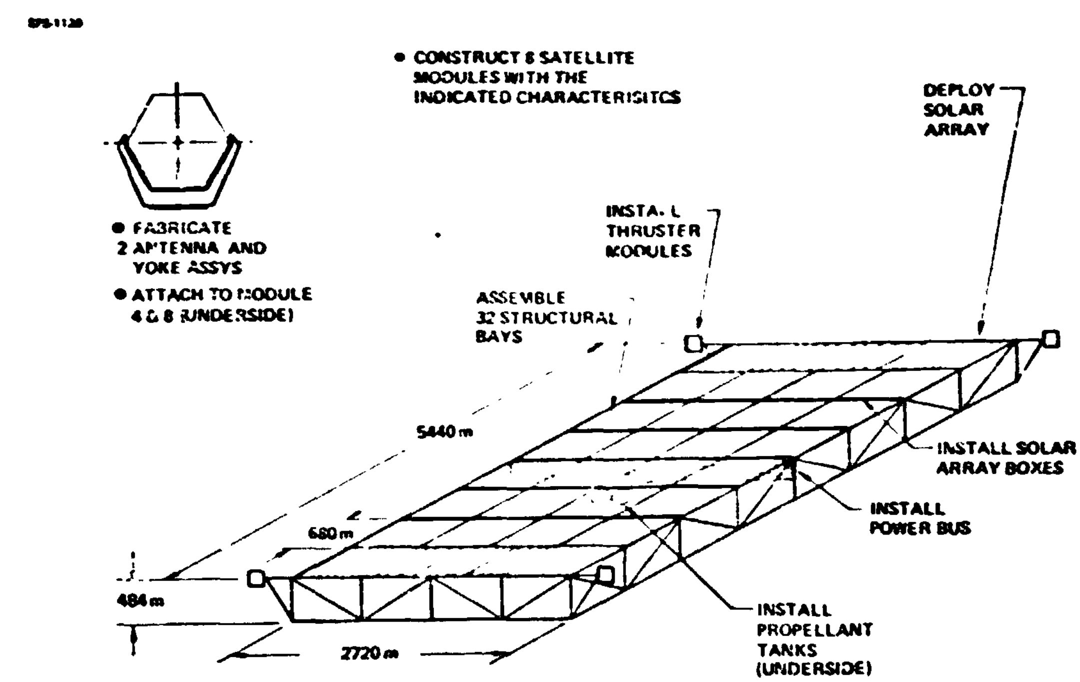

Figure 2-5. LEO Base Construction Tasks

The construction tasks associated with thermal engine satellite modules are indicated in Figure 2-6. The key difference compared to the photovoltaic construction task primarily relates to the difference in the power generation devices (i.e., reflectors and thermal engines/radiators instead of solar arrays).

2.1.2 LEO Construction Bases

The construction base for the photovoltaic satellite is illustrated in Figure 2-7 and consists of two connecting facilities with one used to build the modules and the other to build the antenna. The module construction facility is an open ended structure which allows the four bay wide module to be constructed with only longitudinal indexing. There are two basic work areas. The aft area is used for structural assembly using beam machines and joint assembly machines attached to both the upper and lower surfaces of the facility. Solar array and power distribution are installed from equipment attached to the upper facility surface in the forward area. The satellite module is supported by movable towers located on the lower surface of the facility. These towers are also used to index the module as it is being fabricated.

The antenna facility is configured to enclose four bays of antenna in width and four rows of bays in length. The minimum planview shape of the facility is obtained through use of a 60 degree parallelogram. This shape is the result of the basic unit of the primary structure being triangular in shape and the resulting angular indexing. The lower surface of the facility is used to support beam machines, joint assembly machines, support indexing machines and bus deployment equipment. The upper surface is used to support beam machines, joint assembly machines and a deployment platform that is used to deploy the secondary structures and antenna subarrays.

The thermal engine satellite construction base has been designed to surround the thermal engine satellite module and as a result consists of some rather large dimensions as shown in Figure 2-8. The construction operations are performed in three separate levels or areas of the base. At the lower level is located the antenna construction facilities and those provisions necessary to construct the antenna yoke. Immediately above this area is the reflector construction factory which includes equipment necessary to construct reflector structure and install reflecting facets. Support of the constructed reflectors is accomplished using indexing devices moving down two side rails. These rails are also used to support beam machines used to construct the four supporting legs between the reflector surface and the focal point. At the upper level of the construction base is located the focal point factory which has the task of constructing the CPC, cavity, installing the thermal engines, constructing radiators and the spine which serves as the power distribution system. A fourth area, although only used in the construction of two modules is the assembly platform used to form the antenna structure support point for the antenna.

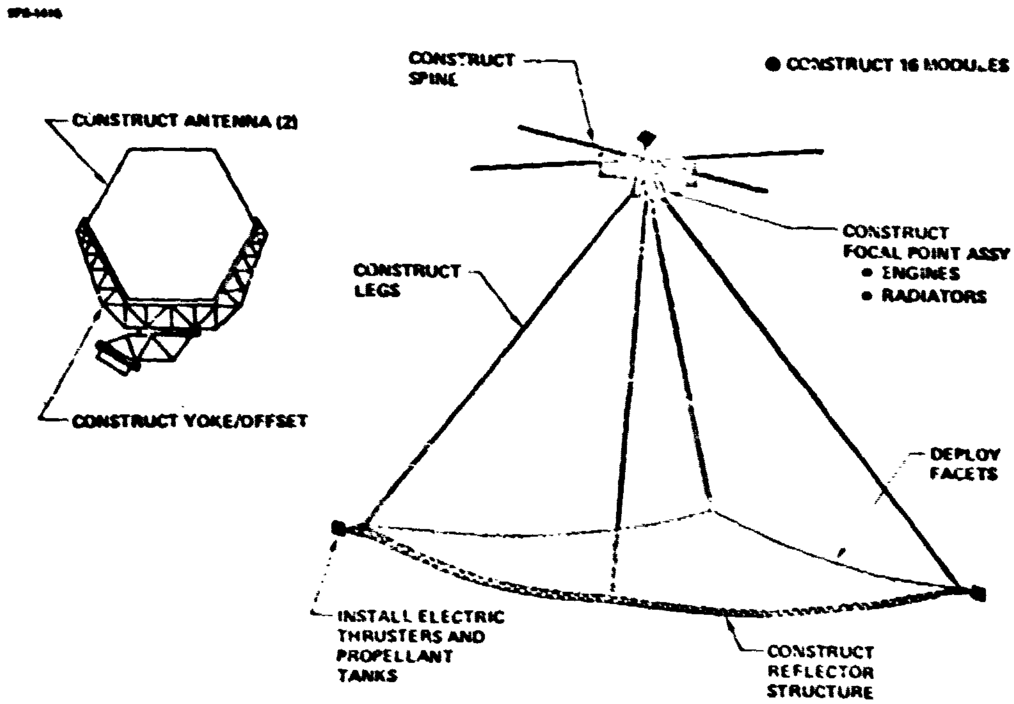

Figure 2-6. LEO Base Construction Tasks Thermal Engine Satellite

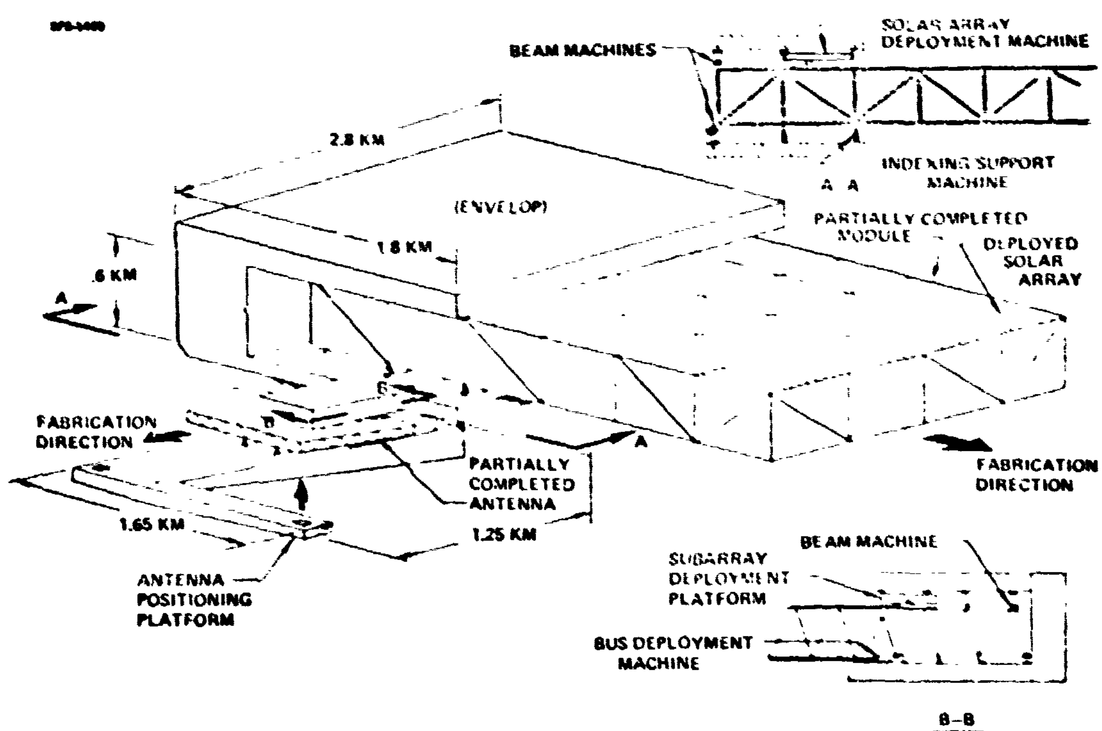

Figure 2-7. LEO Construction Base Photovoltaic Satellite

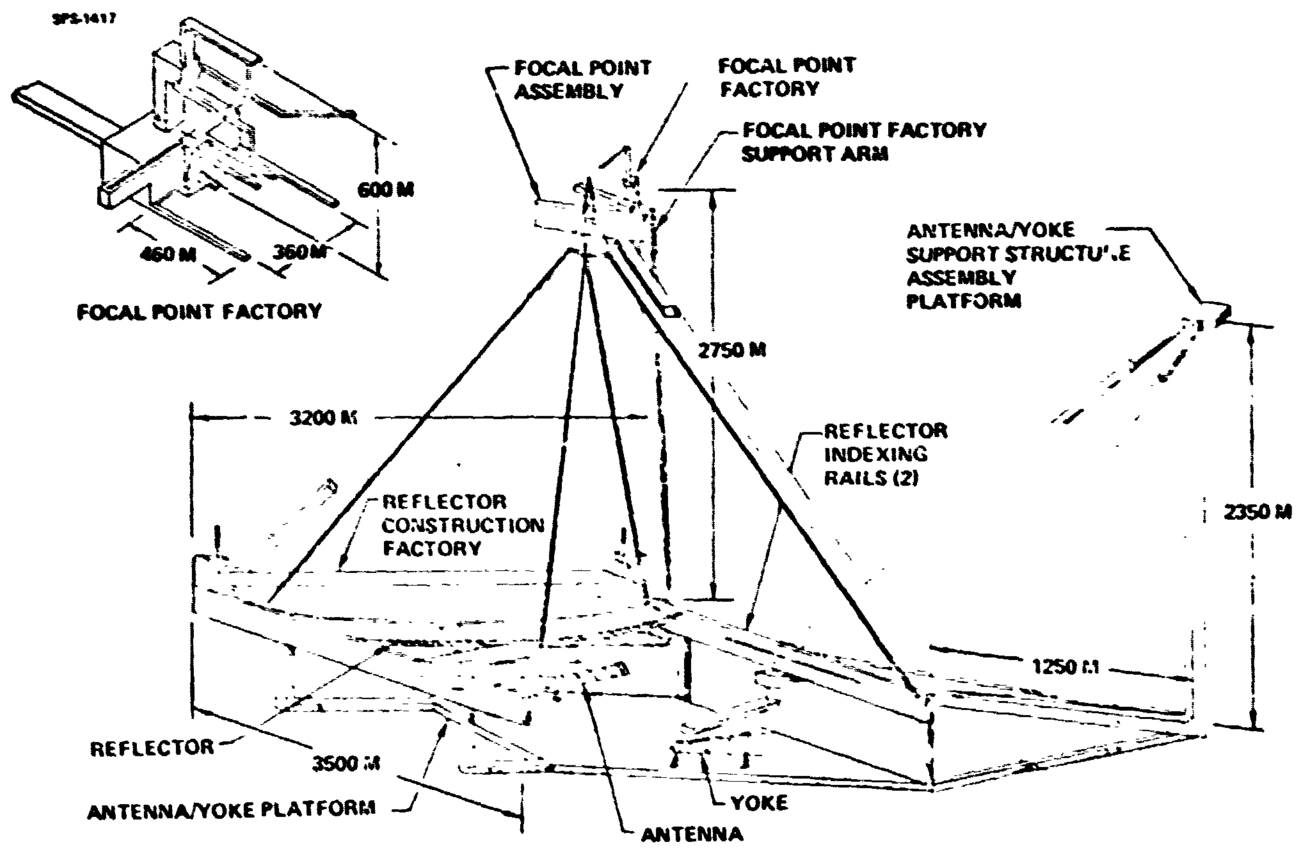

Figure 2-8. LEO Construction Base
Thermal Engine Satellite

2.1.3 Satellite and Antenna Construction Operations

The overall construction sequence of a photovoltaic satellite module is shown in Figure 2-9. The construction sequence associated with the structure, solar array and power buses consists of initially building the first end frame of the structure. This end frame is indexed forward one structural bay length at which time machines can then form the remainder of the structure in each of the bays. The first row of four bays is then indexed forward to allow construction of the second row of structural bays in parallel with installation of solar arrays in bay 1 through 4. Solar array installation and construction of structure occurs simultaneously across the width of the module, although neither operation depends on the other. At the completion of 16 bays or four rows of bays in length, the power buses and propellant tanks are installed. Construction of the structure and installation of solar arrays of the remaining four bay lengths of the module are done in a similar manner to that previously described. Thruster modules for the self-power system are attached to each of the four corners of the module.

The construction sequence for the power generation portion of the thermal engine satellite module is presented in Figures 2-10 and 2-11. The first of these construction operations deals with the formation of the reflector surface and is shown in Figure 2-10. The principal elements involved in this operation are the factory itself and the structural machines, reflector deployment machines and indexing devices. The complexity of this operation and the machines themselves is better appreciated by the fact that the shape of the reflector surface is a portion of a sphere and in addition, the structure forming the shape consists of interconnecting tetrahedrons. To accomplish this task, the structure and reflector machines are attached to the underside of the reflector factory and run on tracks. The spherical reflector shape is obtained by having the reflector factory move up and down in elevation and rotate about its longitude axis. Movement of the factory occurs after the machines make each transit across the factory length. Five structural and reflector machines are required in order to satisfy the timeline requirements. The other major operation in constructing the thermal engine satellite occurs at the top of the construction base where the focal point equipment is constructed and installed. The major individual operations to occur are shown in Figure 2-11 against a background of the focal point assembly factory. The point to be kept in mind is that all of these operations are going on simultaneously. At several points in time, major subassemblies are brought together and finally all elements are then connected to form the complete unit. At that point, the factory is moved away and the focal point can be attached to the support legs coming up from the reflector surface.

Construction of antenna and yoke for each satellite is essentially the same, and for that reason specific operations associated with this task are not covered at this time. In both cases, each antenna requires six months of construction time. A point of difference however, is where and when these elements are constructed and how the assembled antenna/yoke is attached to the satellite for transportation.

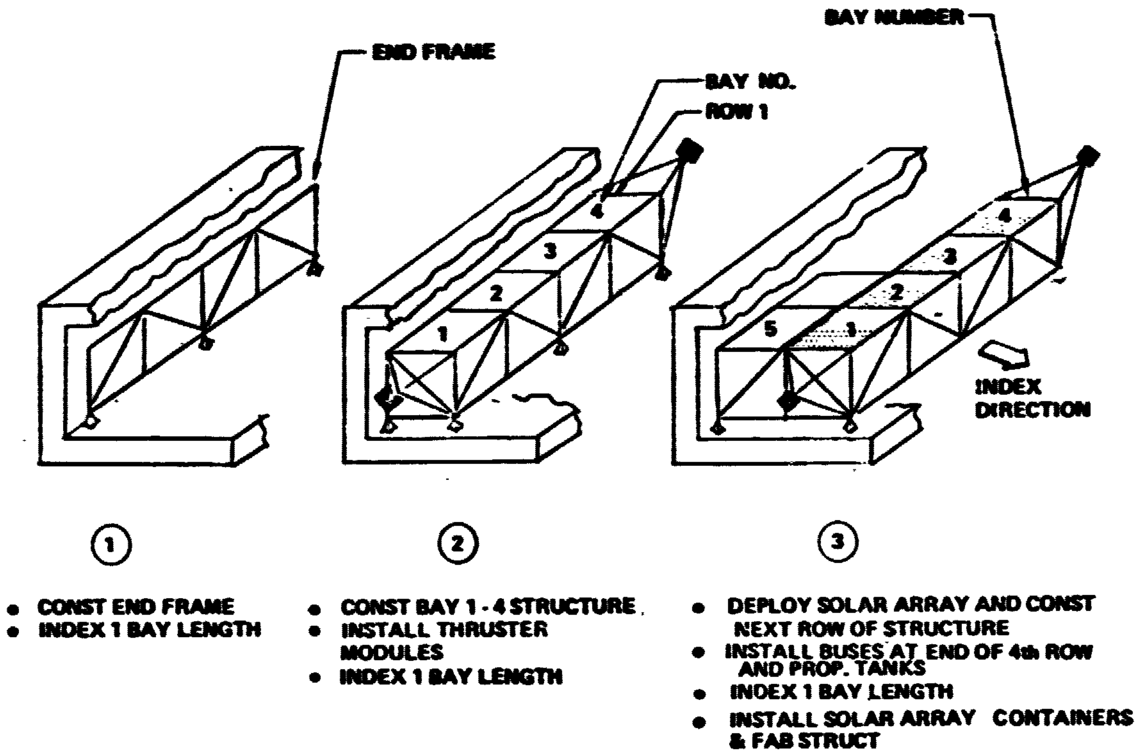

Figure 2-9. Module Construction Sequence Photovoltaic Satellite

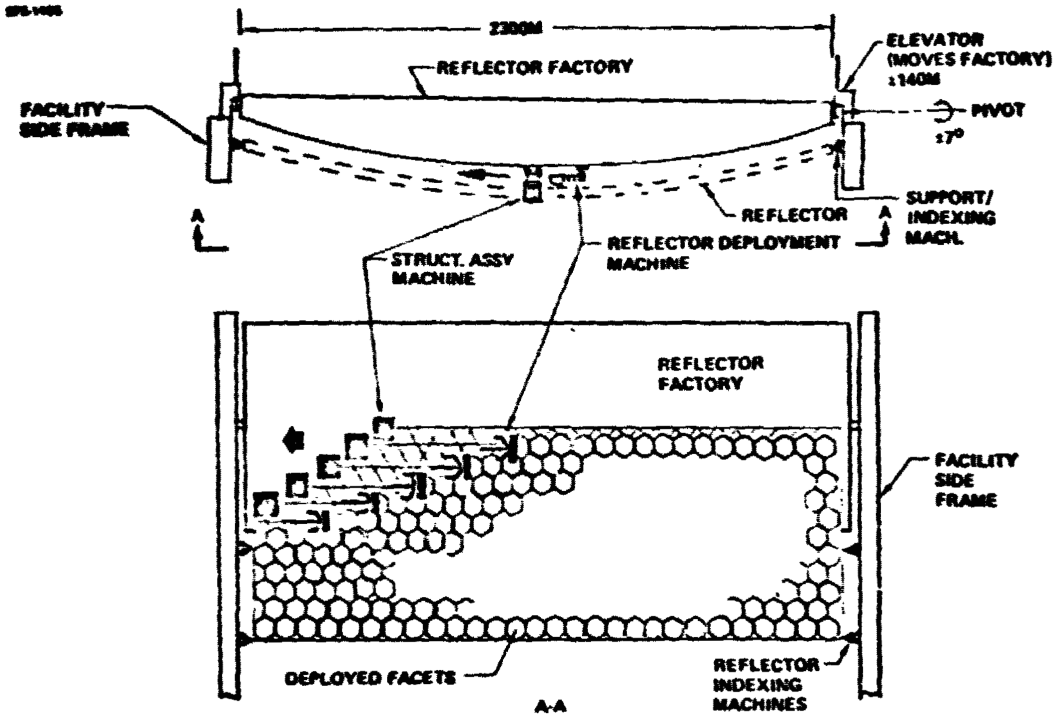

Figure 2-10. Reflector Construction Operations Thermal Engine Satellite

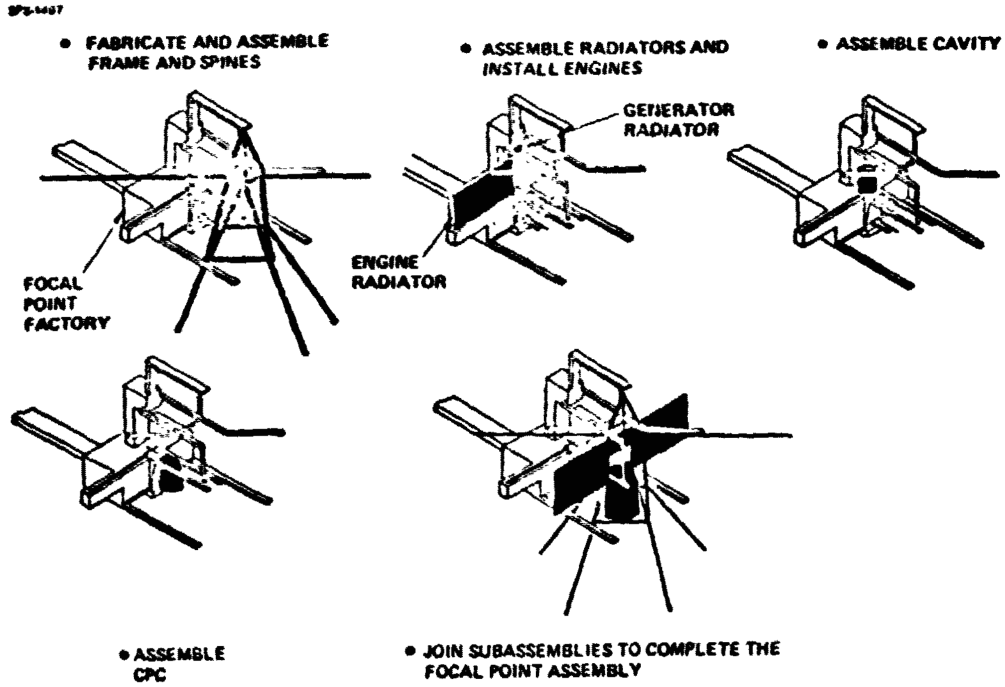

Figure 2-11. Focal Point Assembly Operations Thermal Engine Satellite

The antenna and yoke operations associated with the photovoltaic satellite are shown in Figure 2-12. The yoke for the antenna is constructed in the module construction facility because of its large dimensions. When using this approach however, it requires the yoke to be made in between the third and fourth module and between the seventh and eighth modules. Following yoke construction it is moved to the side of the module facility. At that time either the fourth or the eighth module will be constructed. During the construction of these modules, the antenna is completed so that it can then be attached to the yoke. After five rows of bays have been completed in the fourth and eighth modules, the antenna/yoke combination can then be attached to the module in its required location. Construction of two more rows of bays puts the antenna outside the facility where it then can be hinged under the module for its transfer to GEO.

Construction of the antenna elements of the thermal engine satellite occur at the lower level of the construction base as indicated in Figure 2-13. The support structure for the yoke and the hinge linkage used to position the antenna are different from the photovoltaic satellite. Another difference although not having a major impact is that the antenna is constructed with the radiating surface down. In the case of the support structure, there is an offset which allows the proper pointing of the antenna while the satellite flies PEP rather than POP as in the case of the photovoltaic satellite. The hinge linkage used to position the satellite is made following the yoke. Assembly of the antenna, yoke and hinge linkage into one unit is followed by the attachment of this unit to the underside of the reflector surface for the transfer to GEO.

2.1.4 Final Assembly Operations

Several key and distinguishing construction tasks are required by each satellite once GEO is reached. (The transportation of the satellite modules from LEO to GEO are discussed in Sec. 2.1.10.) In the case of the photovoltaic satellite, an additional task is required in that those solar arrays not deployed for transfer now require deployment. This operation requires a final assembly platform that can support four solar array deployment machines as shown in Figure 2-14. The other tasks to be performed at GEO are shown on subsequent charts.

The first operation to occur once the photovoltaic satellite modules reach GEO is that of the berthing (or docking) of the modules. In the case of the photovoltaic satellite, the modules are berthed along a single edge as indicated in Figure 2-15. The major equipment used to perform these berthing operations are shown. The concept employs the use of four docking systems with each involving a crane and three control cables. Variations in the applied tension to the cables allows the modules to be pulled in, provide stopping control and provides attitude control capability. Also required in this concept is an attitude control system involving thrusters which are not shown.

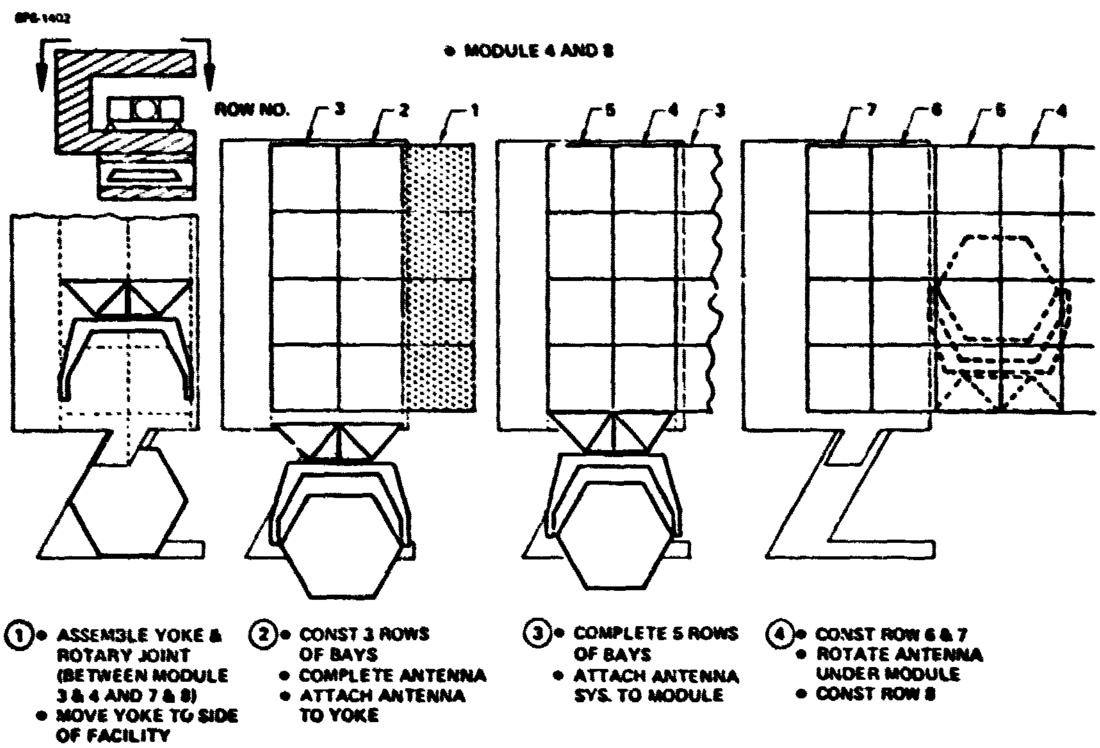

Figure 2-12. Antenna/Yoke/Module Assembly
Photovoltaic Satellite

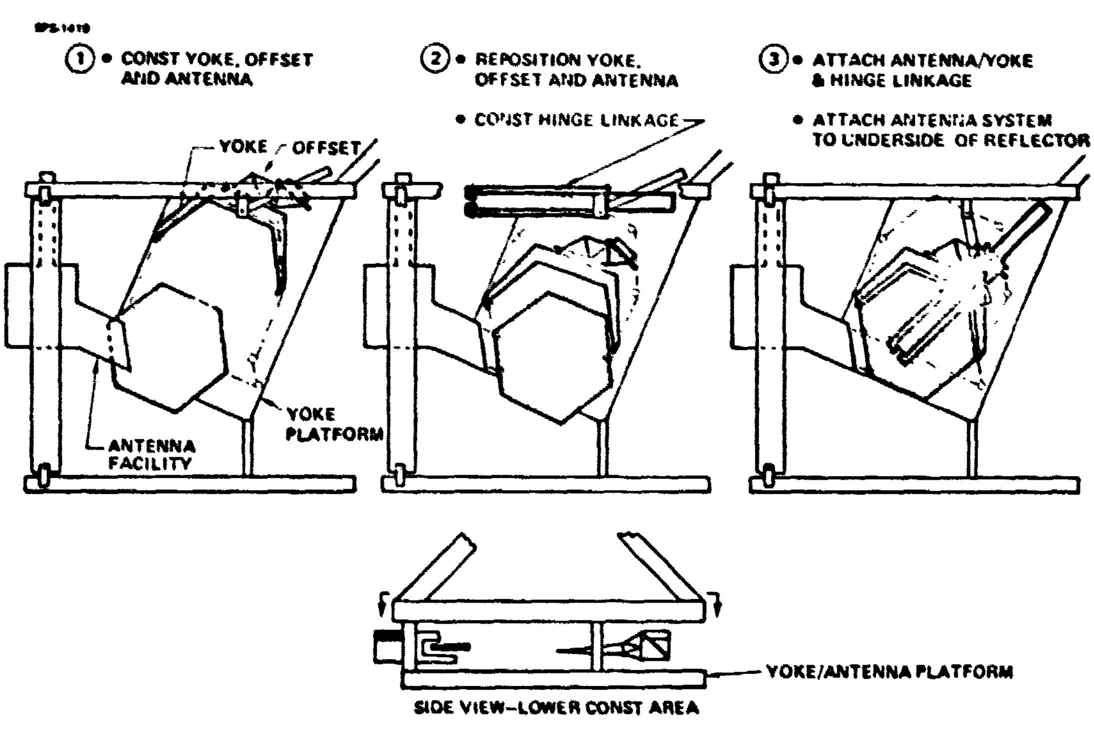

Figure 2-13. Antenna/Yoke/Module Assembly
Thermal Engine Satellite

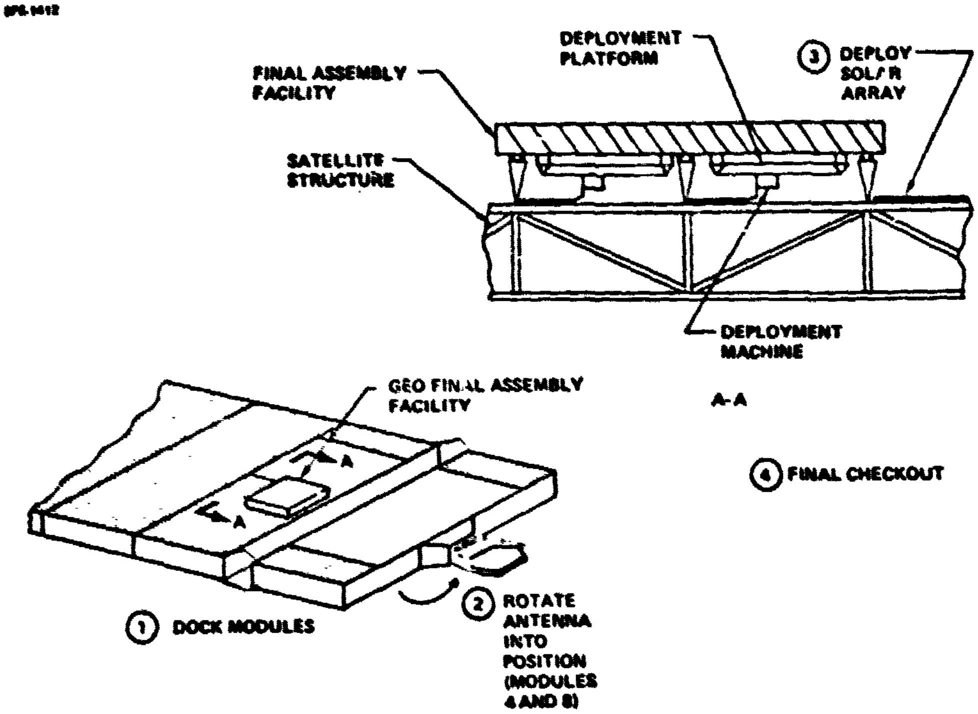

Figure 2-14. GEO Base Construction Tasks Photovoltaic Satellite

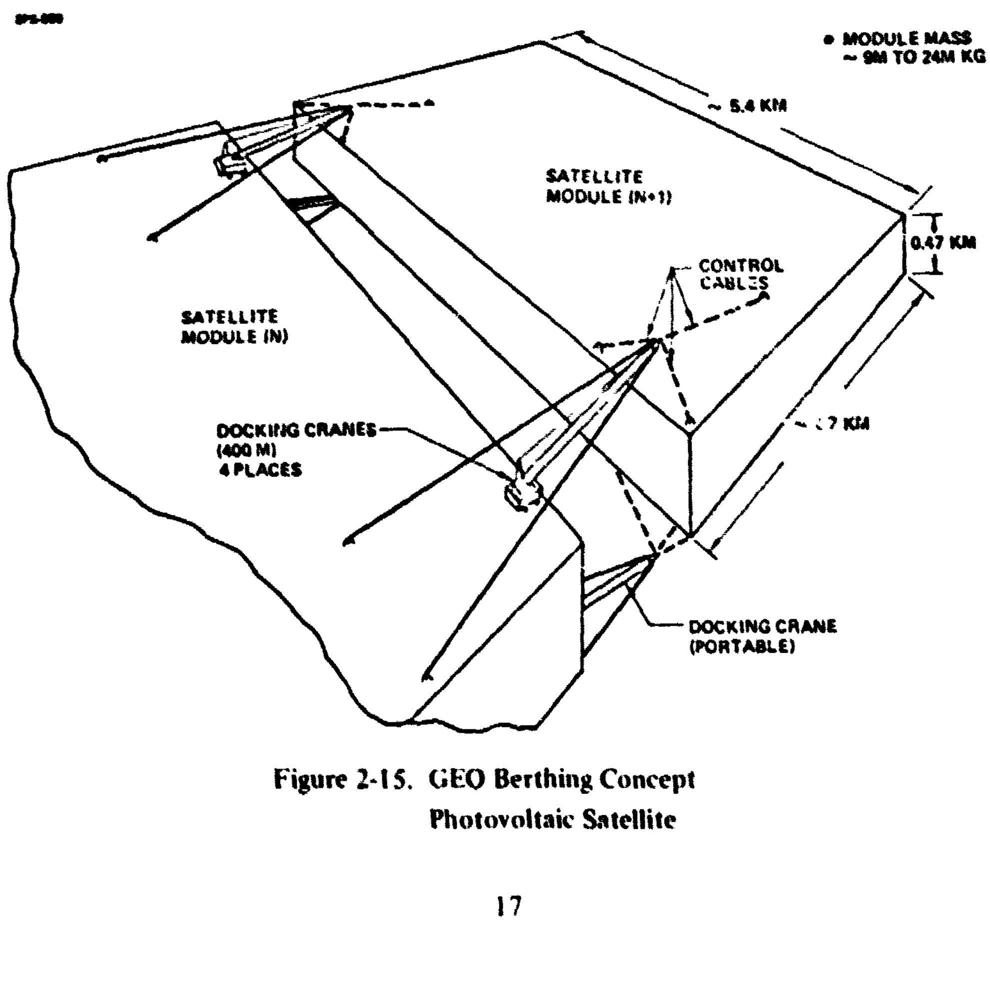

Figure 2-15. GEO Berthing Concept Photovoltaic Satellite

Berthing operations associated with the thermal engine satellite modules require both single edge and two edge (corner) berthing as indicated in Figure 2-16. To accomplish the berthing operation, two facilities are employed. Each is provided with a crane system similar to those described for the photovoltaic concept. One of the facilities is located at the upper portion of the module while the second is near the plane of the reflector so that forces can be applied around the cg of the module. Movement of one of these facilities from module to module occurs by releasing one attachment point and pivoting around the other until the desired location is reached.

Comparison of the antenna final installation operations associated with the satellite also illustrates some differences in terms of complexity of the required mechanisms. In the case of the photovoltaic satellite, the antenna is attached below the module and uses a single hinge line. Once GEO is reached, the antenna is rotated into position followed by the final structural and electrical connections. These operations are illustrated in Figure 2-17.

Placement of the thermal engine satellite antenna requires similar operations except that 3 hinge lines are required rather than one as shown in Figure 2-18. This condition is a result of the long distance between the transfer position of the antenna and the final position for the operational phase.

2.1.5 Construction Equipment

The major construction equipment associated with the photovoltaic satellite are illustrated in Figure 2-19 along with some of the key characteristics such as quantity, mass and dimensions. Again, because the antenna itself is common to both satellite systems, its special equipment is not shown although this material is presented in the detail construction section of this document. The beam machine shown is indicative of the structural concept which uses two beam machines to form all the main structure. Accordingly, it has both translation as well as rotational capability. The dimensions and mass indicated are indicative of the segmented beam approach although machines fabricating thermally formed continuous cord structure could also be attached to the same frame.

Crane/manipulator systems are primarily used to form the structural beam joints. Although the size indicated is most common, several 250 meter units are also required in the construction of the antenna yoke as well as several 20 meter cranes. Two man control cabins with manipulators are located at the end of the crane which is itself attached to a moving platform.

The principal difference between the indicated solar array machine and those illustrated in previous briefings is that the gantry itself is located approximately 50 meters below the facility beams since that is the location of the upper surface of the satellite. Further discussion on these machines occurs in Section 3.2.

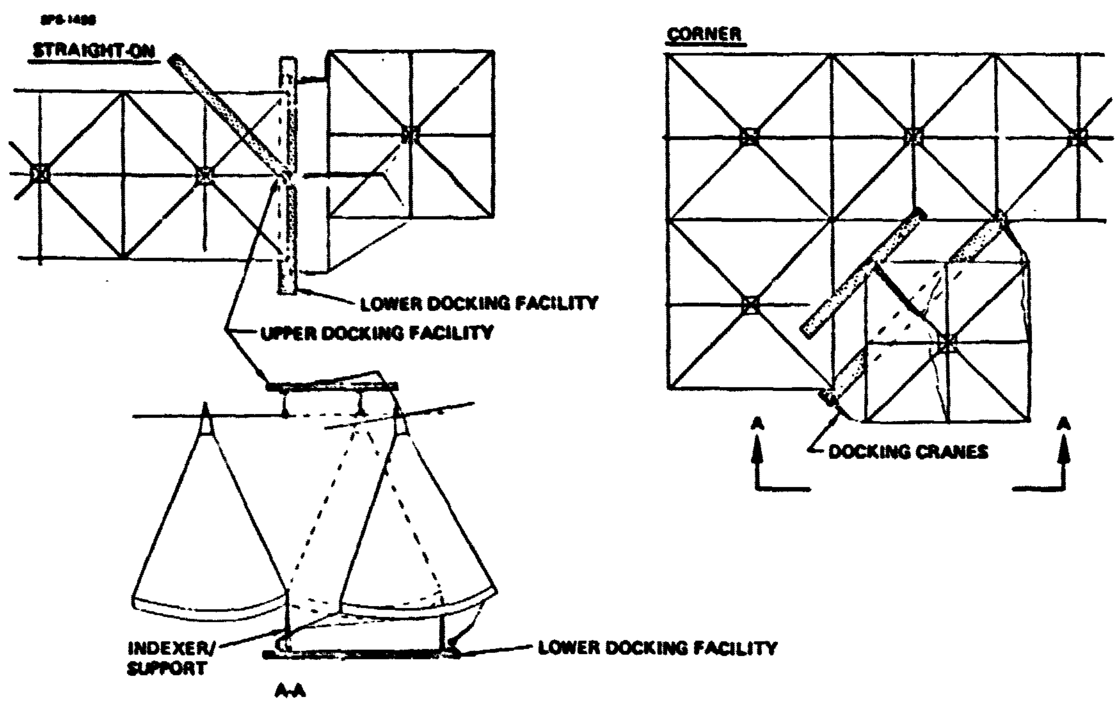

Figure 2-16. GEO Berthing Concept
Thermal Engine Satellite

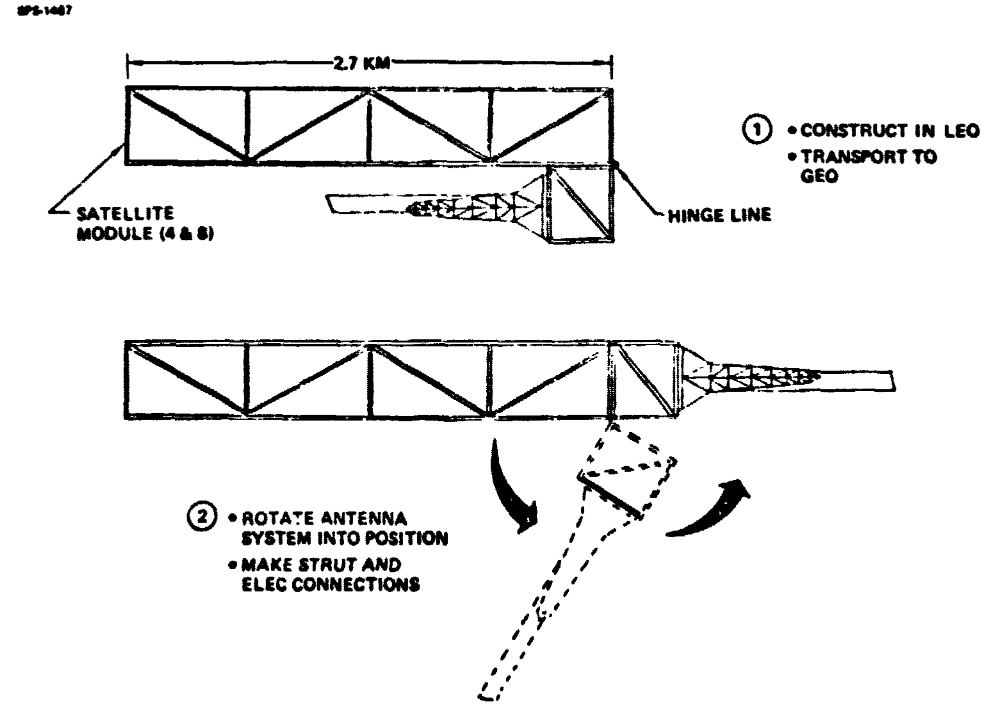

Figure 2-17. Antenna Final Installation
Photovoltaic Satellite

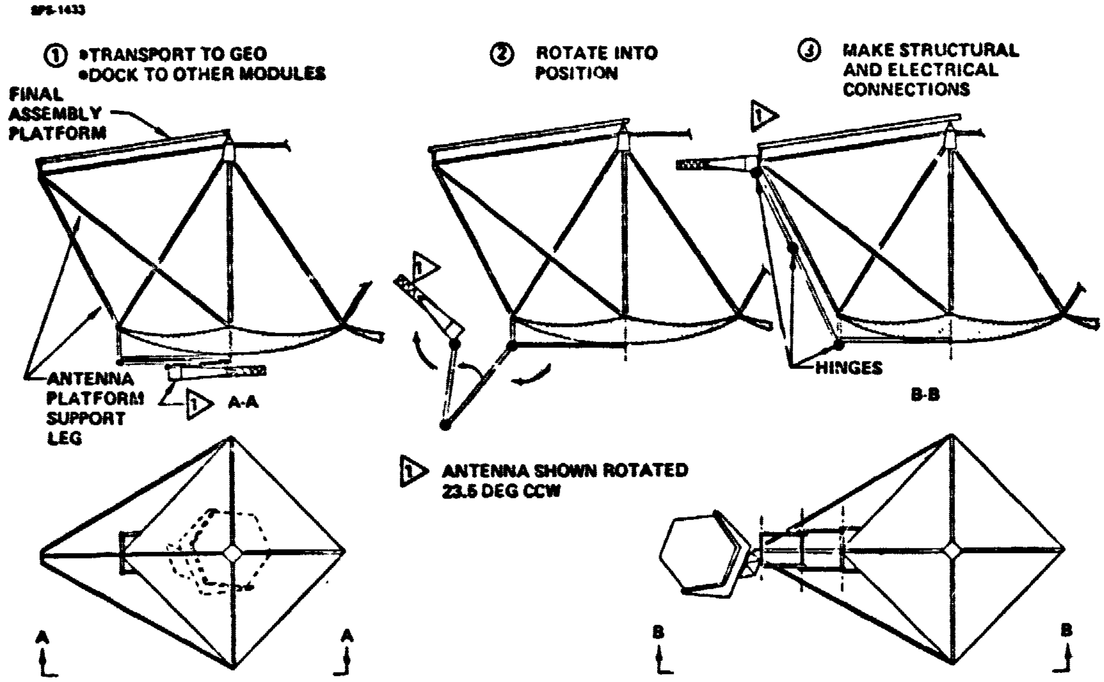

Figure 2-18. Antenna Final Installation
Thermal Engine Satellite

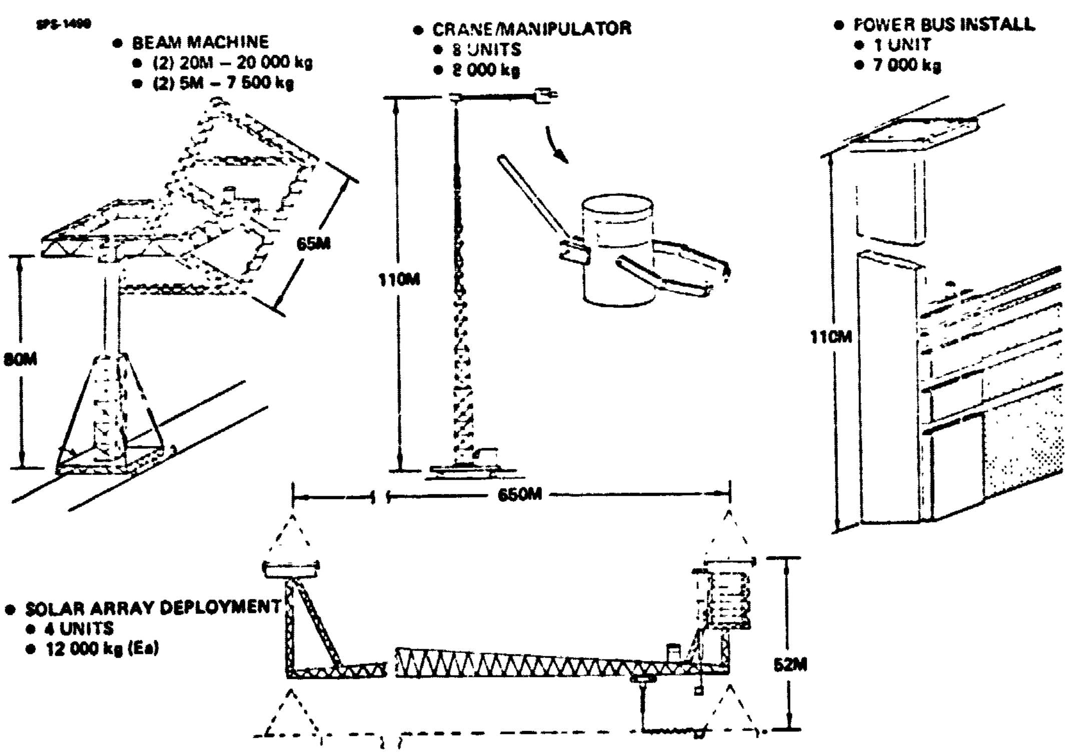

Figure 2-19. Major Construction Equipment
Photovoltaic Satellite

The thermal engine satellite requires several machines similar to the photovoltaic equipment but in addition requires several different units. Some of this equipment is depicted in Figure 2-20. Beam machines are also required with the key difference being th[illegible]ntity and also the need of a 10 meter beam machine. Crane/manipulator units are approximately the same. Formation of the reflector (facets surface) requires a special structure machine and a facet deployment machine. In addition to individual machines, the thermal engine satellite construction operation requires several mini-factories involving numerous pieces of equipment as shown in Figure 2-21. Examples of these small factories are as follows: the formation of the CPC and cavity where cranes, manipulators, welders, conveyors and control cabins are required; a radiator factory that welds the 20 meter length sections of pipe into 350 meter lengths and then attaches the radiators (heat pipe) panels to the main pipes; in addition, engine installation is required including a connection of power busses between the engines and finally, the spine assembly that consists of machines to build structure running between module focal points and machines to assemble and attach the major power busses to that structure.

2.1.6 Crew Requirements

The difference in crew size and distribution of crew is compared for the two satellite concepts in Figure 2-22. The crew size for all orbital personnel indicates the photovoltaic satellite requires approximately 300 fewer people with all this difference occurring in the low Earth orbit construction base. The principal reason for the larger crew requirements for the thermal engine satellite is due to more construction operations required and of course this then contributes to the construction (indirect) personnel and the support personnel manloadings.

2.1.7 Construction System Mass and Cost

ROM e[illegible]timates are presented for the construction bases as well as crew rotation resupply in Figure [illegible]. In the case of the LEO construction bases, the photovoltaic satellite is lighter by approximately 3 million kilograms. The major contributors [illegible] the thermal engine mass is the large foundation (structure) along with three extra crew modules due to the 300 additional people and additional construction equipment. GEO final assembly bases are approximately equal. Differences in the annual crew rotation resupply requirements reflects the difference in the 300 man crew size.

Comparison of the unit cost of the first set of construction bases indicates over a 4 billion dollar savings for the photovoltaic satellite as shown in Figure 2-24. These values reflect a 90[illegible] learning factor applied to each major end item (i.e. beam machine, crew module). Tr[illegible]sportation costs are not included in this particular chart. In the case of the thermal engine satellite the principal difference in the facility cost is the three extra crew modules. The large difference in construction equipment quantity and mass contributes to the difference in construction equipment cost. The wrap-around factor is applied to the sum of the facility and construction equipment costs.

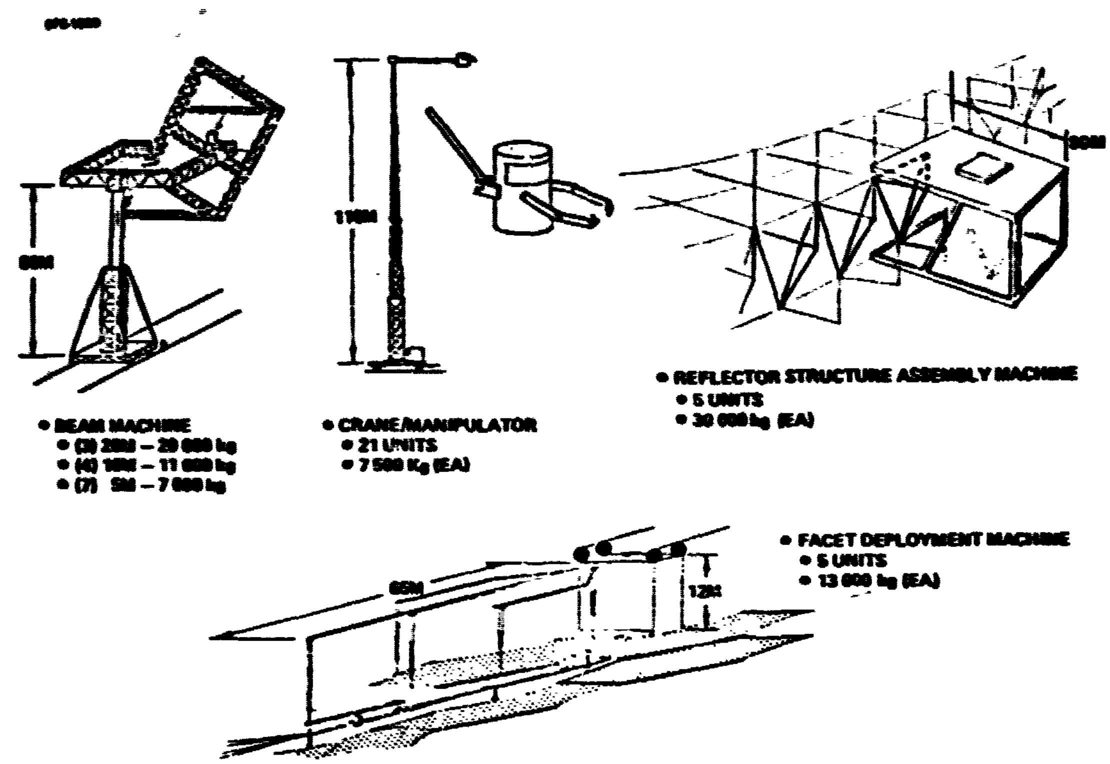

Figure 2-20. Major Construction Equipment Thermal Engine Satellite

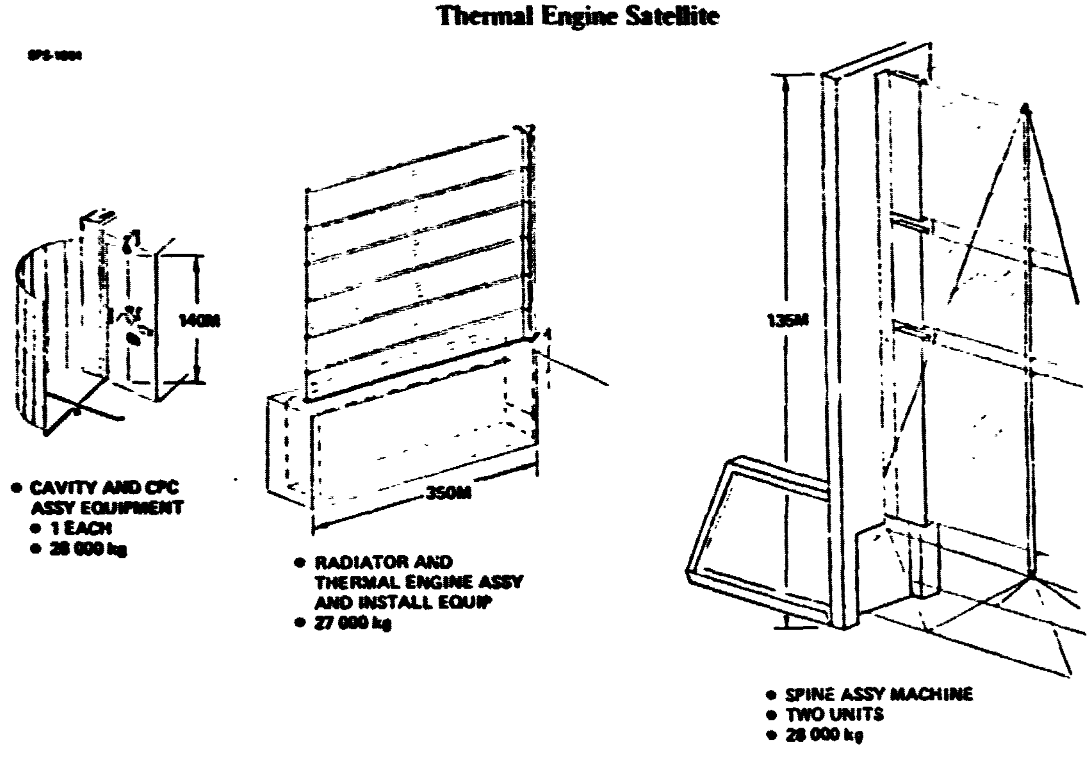

Figure 2-21. Major Construction Equipmer Thermal Engine Satellite

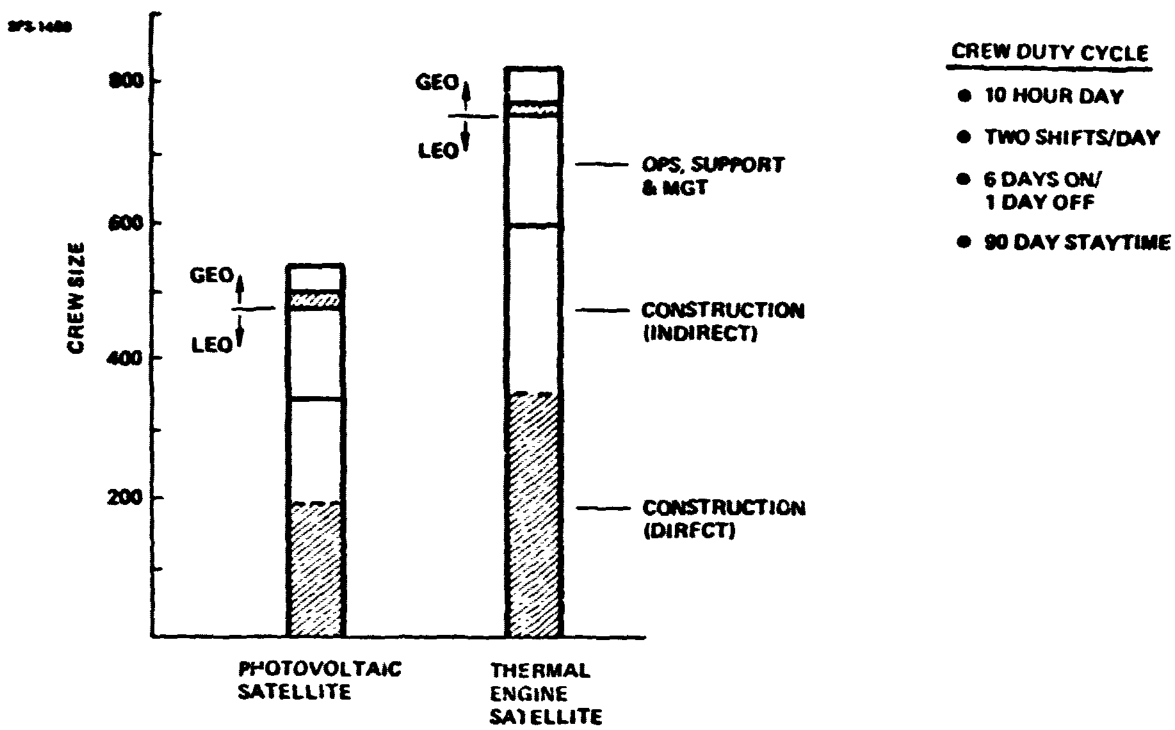

Figure 2-22. Crew Size and Distribution
LEO Construction

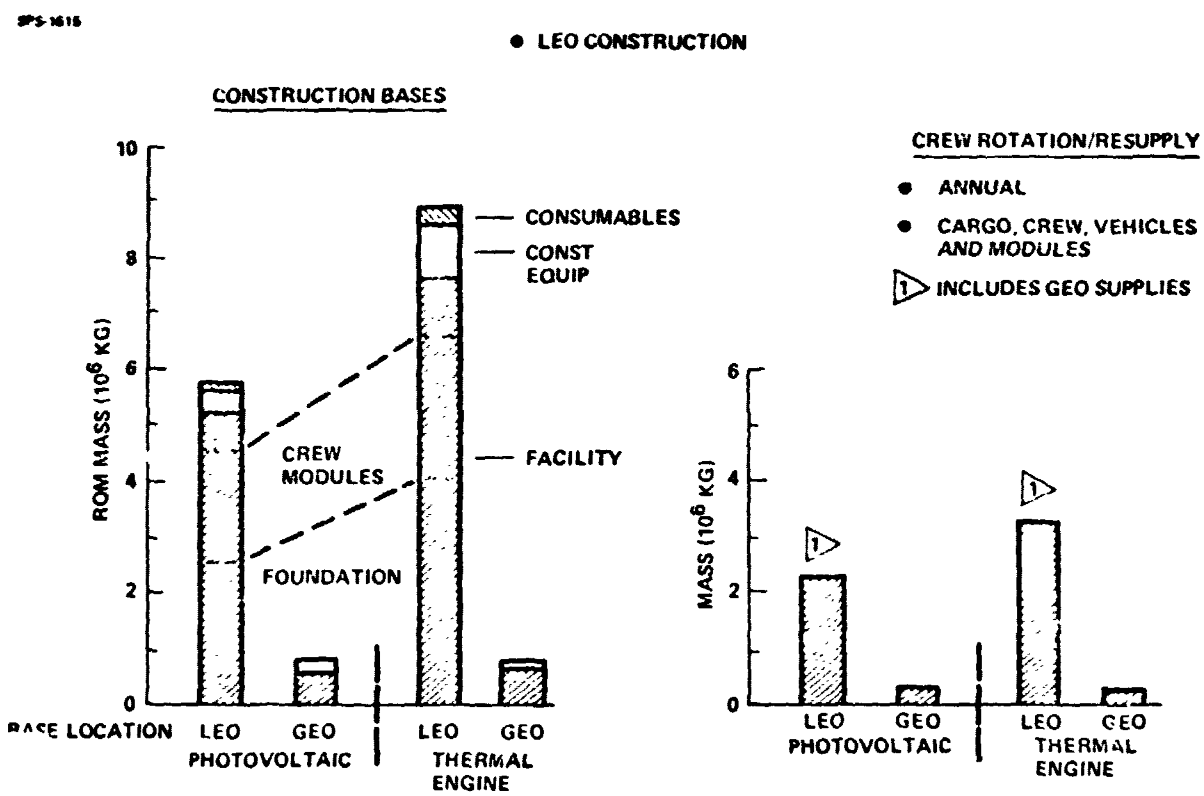

Figure 2-23. Construction Mass Summary

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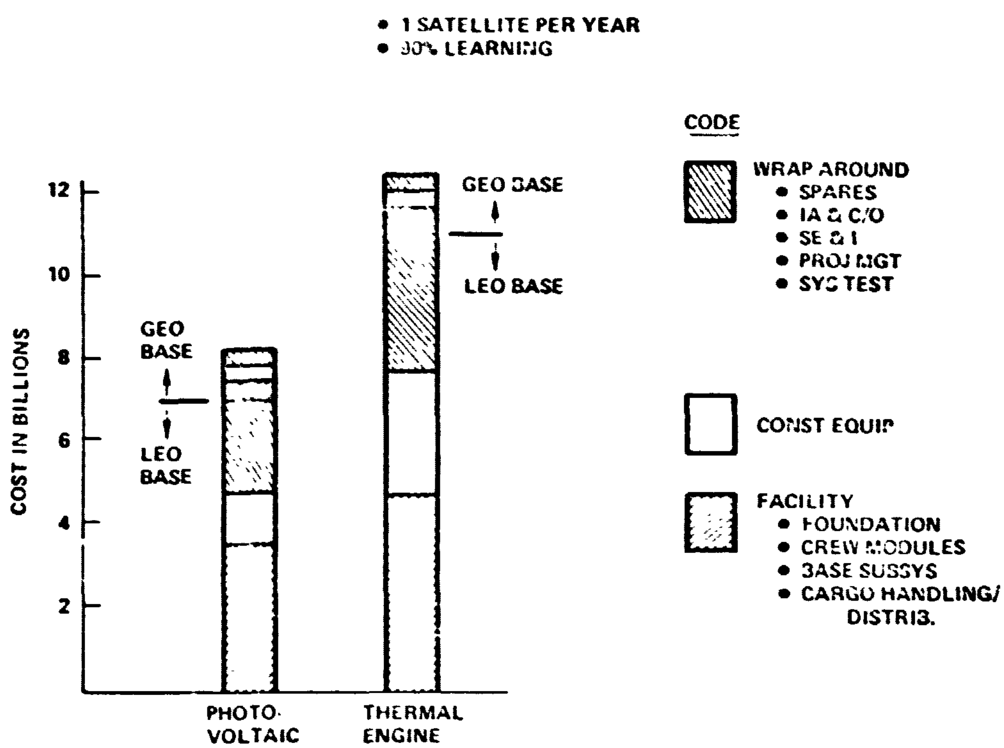


Figure 2-24. Construction Base ROM First Set Cost

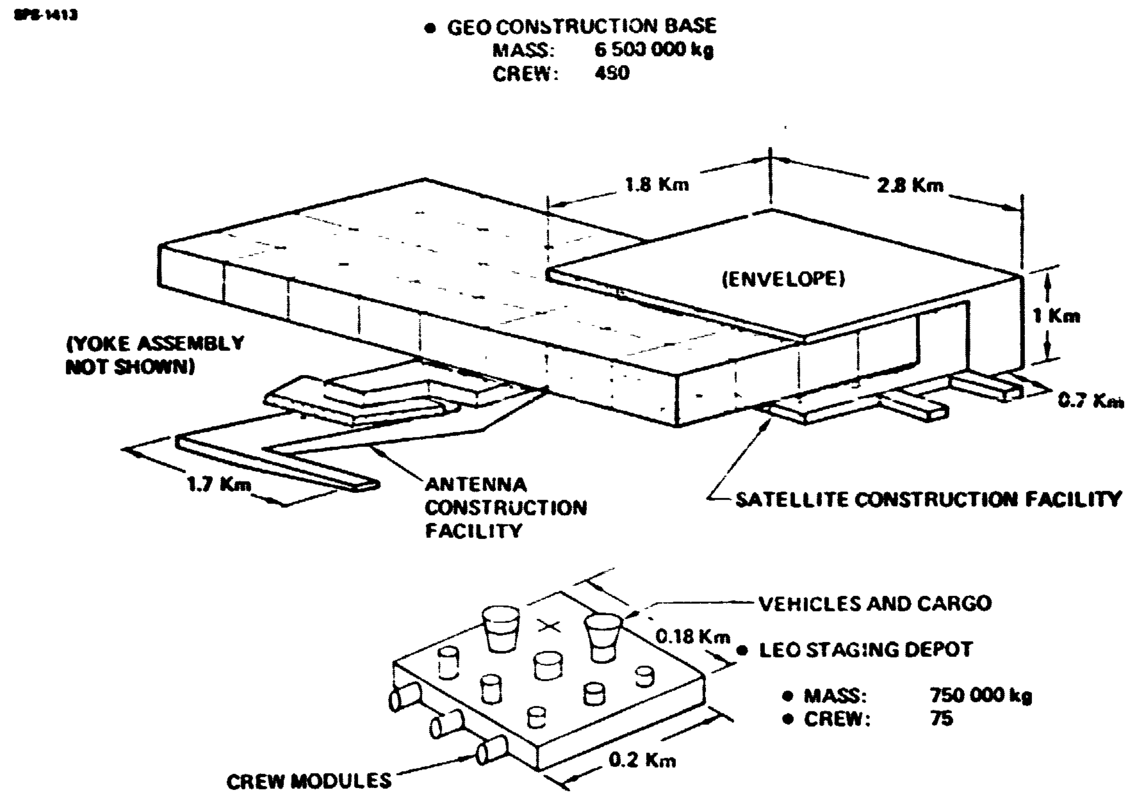


Figure 2-35. Orbital Bases
GEO Construction

SPS-1613

TRANSPORTATION FUNCTION	DIFFERENCE	REASON
• SATELLITE LAUNCH SYSTEM • TWO STAGE BALLISTIC	• PAYLOAD SHROUD P/V – REUSABLE T/E – EXPENDABLE	• SATELLITE COMPONENT DENSITY
• CREW LAUNCH SYSTEM • SHUTTLE GROWTH	• NUMBER OF FLTS	• MORE PEOPLE IN ORBIT
• SATELLITE LEO-GEO SYSTEM • SELF POWER	• T/E HAS LESS SATELLITE DESIGN IMPACT • LESS GRAVITY GRADIENT TORQUE	• NO OVERSIZING AND USE OPER. VOLT. • LOWER INERTIAS
• CREW/SUPPLIES LEO-GEO • TWO STAGE LO_2/LH_2 OTV	• NONE	

Figure 2-25. Transportation System Differences

SPS-1666

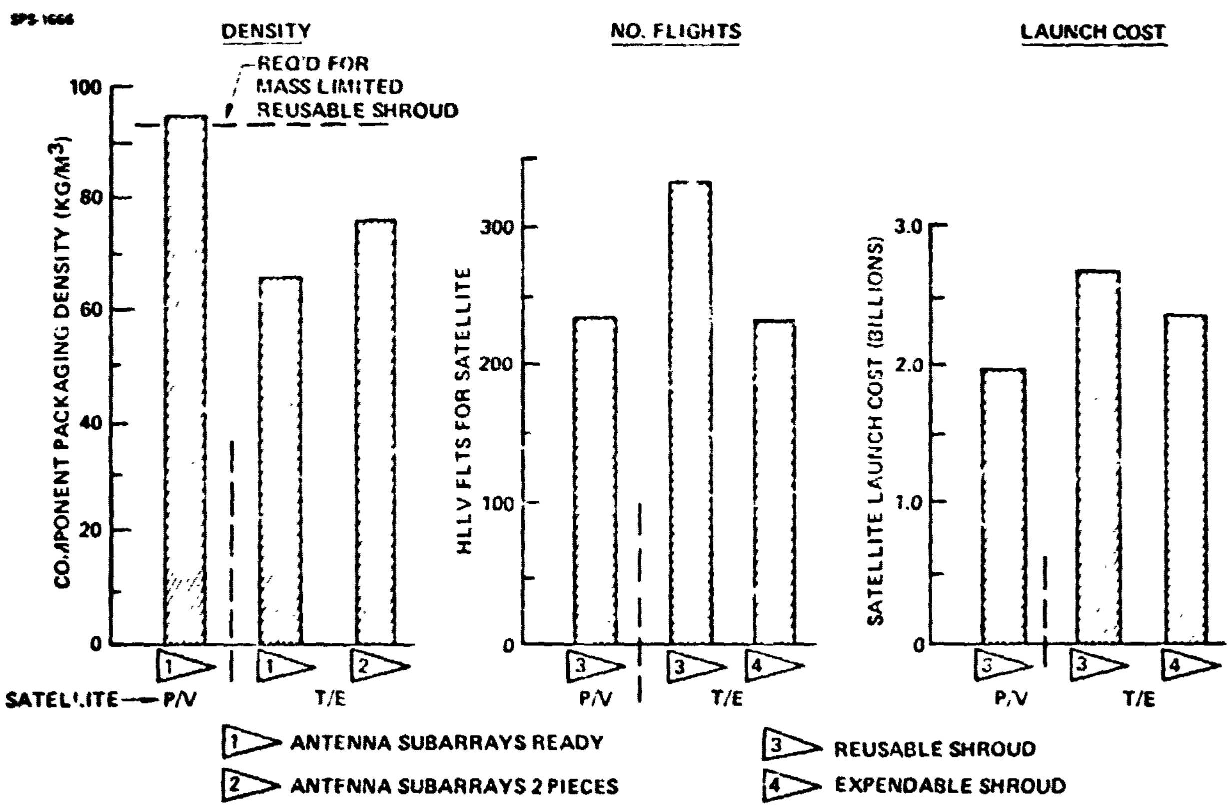


Figure 2-26. Component Packaging Density Impact

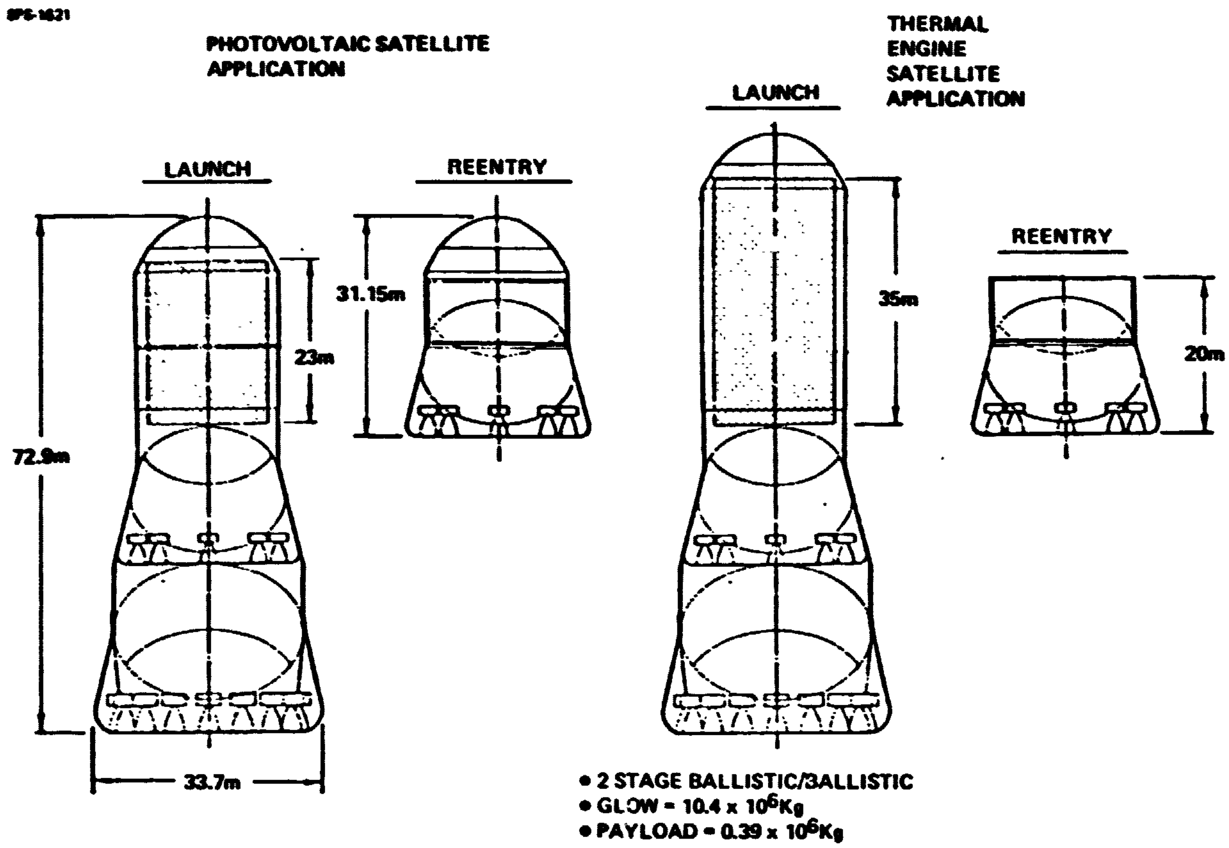


Figure 2-27. Satellite Launch Vehicle

2.1.10 Satellite Orbit Transfer System

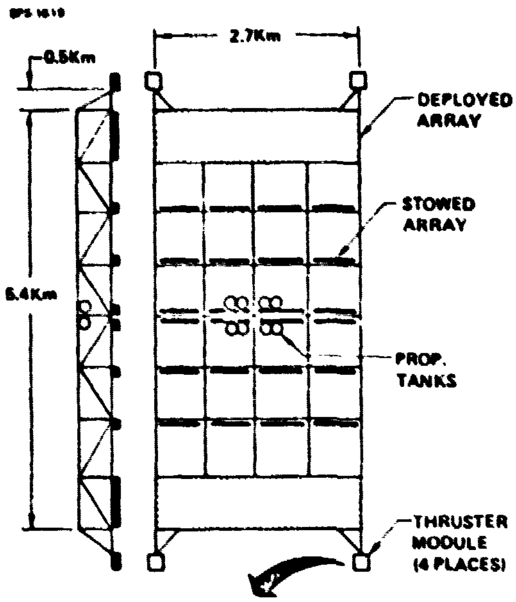
The transfer of the satellite modules from LEO to GEO involves the use of electric propulsion using power provided by the module (thus the name self-power). The characteristics associated with self-power of a photovoltaic module are shown in Figure 2-28 for both those modules transferring antennas and those that do not. The general characteristics indicate a 5% oversizing of the satellite to compensate for the radiation degradation occurring during passage through the Van Allen belt and the inability to anneal out all of the damage after reaching GEO. It should also be emphasized at this point, only the arrays needed to provide the required power for transfer are deployed. The remainder of arrays are stowed within radiation proof containers. Cost optimum trip times and I_{sp} values are respectively 180 days and 7,000 seconds. Flight control of the module when flying a PEP attitude during transfer results in large gravity gradient torques at several positions in each revolution. Rather than provide the entire control capability with electric thrusters which are quite expensive, the electric system is sized only for the optimum transfer time with the additional thrust provided by LO_2 LH_2 thrusters. This penalty actually is quite small since by the time an altitude of 2,500 kilometer is reached the gravity gradient torque is no longer a dominating factor.

Self-power of the thermal engine satellite modules are for the most part similar to the photovoltaic modules from a performance standpoint although there are some distinguishing differences in terms of satellite design impact as identified in Figure 2-29. One example of this is that no oversizing of the thermal engine modules is required since the reflector facets and engines are not sensitive to radiation as are the solar arrays. A second point is that the voltage generated by the satellite can be the same as the operating satellite voltage (since no plasma losses occur as in the case of solar arrays) and thus a minimum power distribution penalty occurs. From a propulsion standpoint, three thruster modules are used rather than four and although all facets are deployed in LEO, only a portion of these are required for the transfer. Gravity gradient torque associated with this configuration are considerably lower due to the inertia characteristics of the module and consequently the chemical thrust required and the amount of LO_2 LH_2 propellant are considerably less than in the case of the photovoltaic satellite module.

2.1.11 Crew Rotation/Resupply Transportation

The major transportation system elements and the number of flights associated with crew rotation/resupply is presented in Figure 2-30. A shuttle growth vehicle using a liquid booster delivers up to 75 crewmen per flight to LEO. Cargo in terms of crew and base supplies as well as propellant and OTV hardware is delivered by the satellite launch vehicle. The OTV used for crew rotation/resupply is a two-stage LO_2/LH_2 system with each stage having identical propellant capacity.

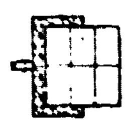
Crew LEO delivery flights and supply flights are different as a result of the difference of 300 people required to construct the two satellites. The GEO bases have nearly the same crew sizes and consequently no difference occurs in the OTV operation.



GENERAL CHARACTERISTICS

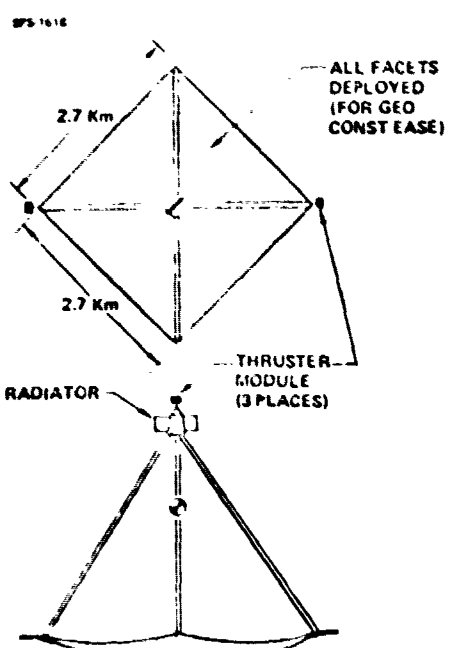
- 5% OVERSIZING (RADIATION)
- TRIP TIME - 180 DAYS
- ISP - 7000 SEC

MODULE CHARACTERISTICS	NO ANTENNA	WITH ANTENNA
• NO MODULES	6	2
• MODULE MASS (10^6KG)	8.7	23.7
• POWER REQ'D (10^6Kw)	0.3	0.81
• ARRAY %	13	36
• OTS DRY (10^6KG)	1.1	2.9
• ARGON (10^6KG)	2.0	5.6
• LO_2/LH_2 (10^6KG)	1.0	2.8
• ELEC THRUST (10^3N)	4.5	12.2
• CHEM THRUST (10^3N)	12.0	5.0



	NO ANTENNA	WITH ANTENNA
PANEL SIZE:	24x38m	48x57m
NO. THRUSTERS:	560	1680

Figure 2-28. Self Power Configuration
Photovoltaic Satellite



GENERAL CHARACTERISTICS

- NO OVERSIZING
- TRIP TIME - 180 DAYS
- ISP - 7000 SEC

MODULE CHARACTERISTICS	NO ANTENNA	WITH ANTENNA
• MODULES	14	2
• MODULE MASS	4.1	19.1
• POWER REQ D (10^6 Kw)	0.14	0 65
• FACETS REQ D % [1]	27	74
• OTS DRY (10^6 Kg)	0.5	2.35
• ARGON (10^6 Kg)	1.0	4.5
• LO_2/LH_2 (10^6 Kg)	0.2	0.94
• ELEC THRUST (10^3 N)	2.1	9.8
• CHEM THRUST (10^3 N)	2.1	9.8

[1] INCLUDES 14% TO COVER LOSSES

Figure 2-29. Self Power Configuration
Thermal Engine Satellite

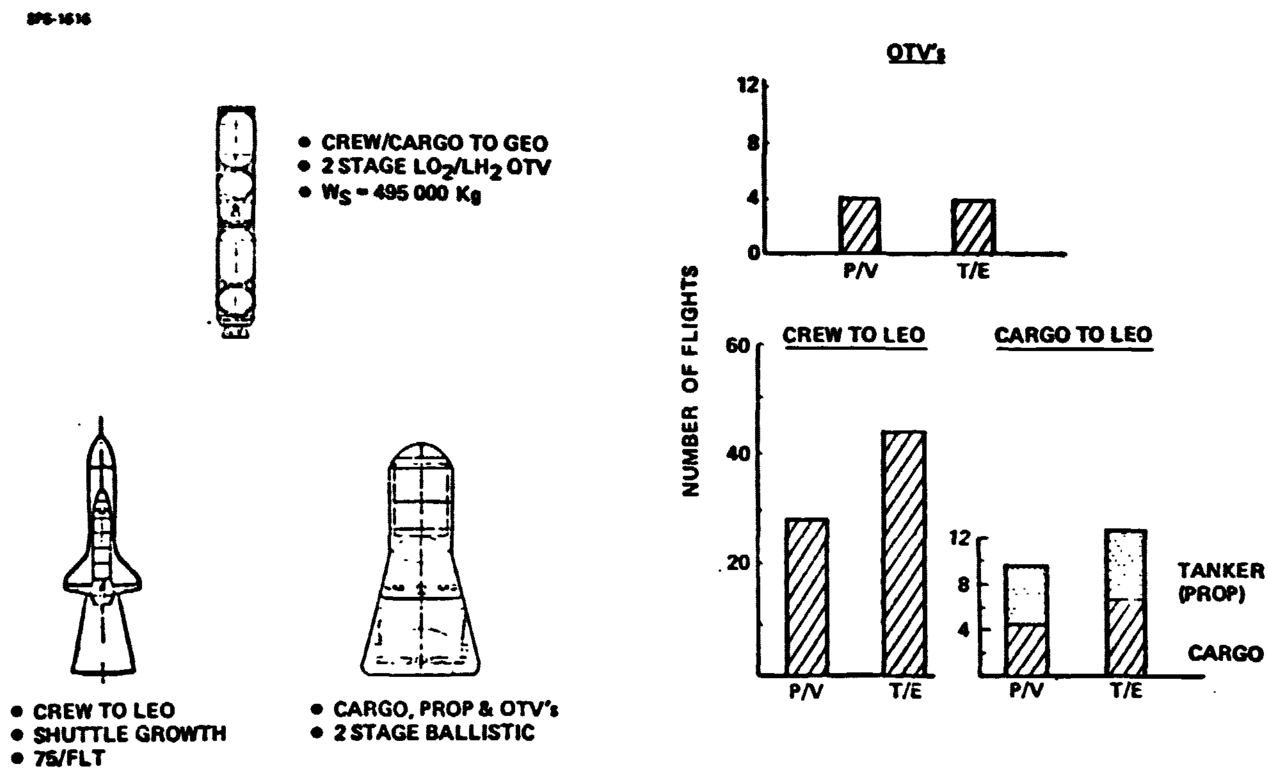


Figure 2-30. Crew Rotation/Resupply Transportation Power Generation System

2.1.12 Transportation Cost

The toal transportation cost of the photovoltaic and thermal engine satellite effort is presented in Figure 2-31. The costs are broken down to illustrate the differences for the three major transportation operations although the magnitude of the cost of the three are quite different. In the case of the satellite transportation costs, the primary reason for the thermal engine being greater is its need to use an expendable shroud in order to achieve a mass limited launch condition. Crew rotation resupply differences are reflecting the difference in numbers of flights to get an extra 300 people to LEO in the case of thermal engine satellites. Construction base transportation differences are primarily due to the larger mass of the thermal engine construction base as well as the volume limited condition of the construction equipment itself and the fact that the thermal engine concept uses considerably more equipment. It should be remembered however, that this initial placement will most likely last for 20 years in terms of the facility and 10 years for the construction equipment so that facility transportation costs can be considered as amortized.

2.1.13 Construction Transportation Summary

A summary comparison of the photovoltaic and thermal engine satellite is presented in Table 2-3 along with an indication of which concept is preferred relative to the various construction and transportation parameters discussed in preceding paragraphs. Compared in this manner, it appears that the photovoltaic satellite has a clear advantage in terms of less complex facilities, construction operations and construction equipment, all leading to a lower construction cost and in addition has lower transportation costs.

2.2 CONSTRUCTION LOCATION COMPARISON

Comparison of the major construction and transportation parameters associated with LEO and GEO construction will be done using the photovoltaic satellite due to it being judged to offer the best characteristics in terms of construction and transportation. The principle areas to be used in comparing the two construction locations options are indicated in Table 2-4. As in the case of comparing the power generation system options, the two construction location concepts will be compared at the same time or on consecutive charts for a given item of comparison.

2.2.1 CONSTRUCTION CONCEPTS

To establish a framework from which to conduct the comparison of LEO vs GEO construction, an overall summary of each construction concept is presented. The LEO construction concept of the photovoltaic satellite is shown in Figure 2-32 and consists of eight modules and two antennas all constructed in LEO facilities. All modules are transported to GEO using self-power. Two of the modules will transport an antenna while the remaining six modules go up alone. GEO operations require berthing of the modules to form the complete satellite and the deployment of the solar arrays not used for the transfer

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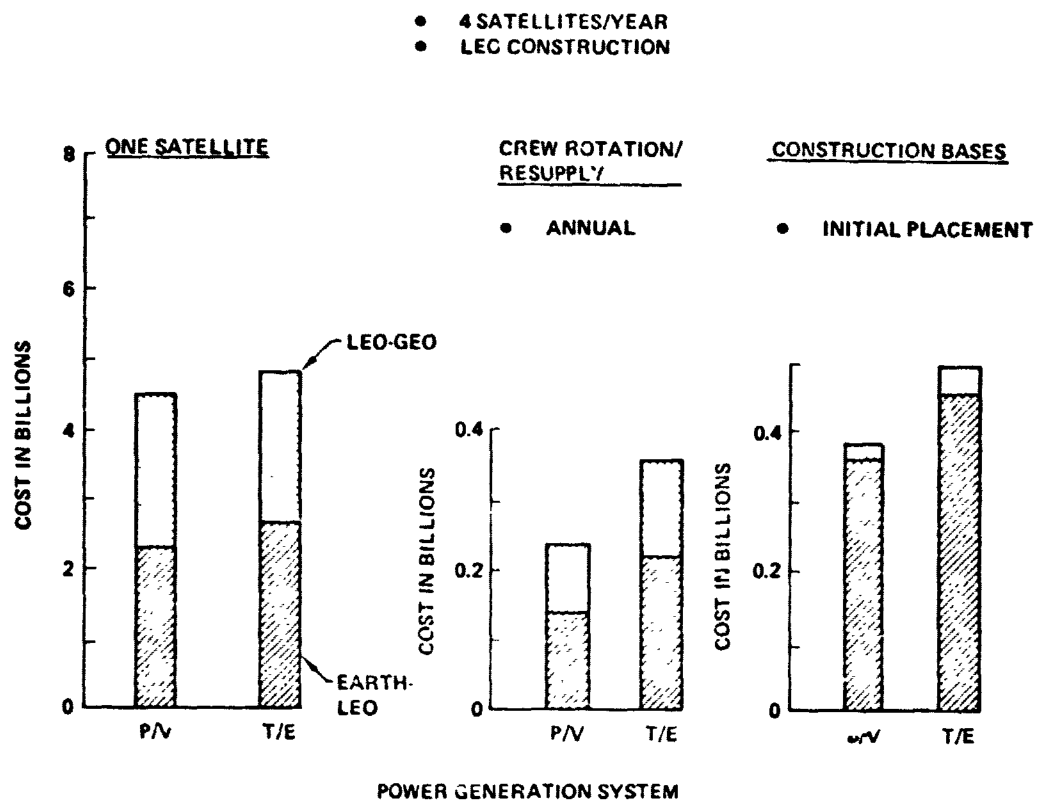

Figure 2-31. Transportation Cost Photovoltaic vs Thermal Engine

Table 2-3. Construction/Transportation Summary Power Generation Comparison

SPS-1625

✓ FOR MOST PROMISING CONCEPT

COMPARISON PARAMETER	PHOTO-VOLTAIC	THERMAL ENGINE	RATIONALE
1. BASE CONFIGURATION	✓		• SMALLER • LESS COMPLEX
2. SATELLITE CONSTRUCTION	✓		• LESS COMPLEX • FEWER OPERATIONS
3. ANTENNA CONSTRUCTION	NO DIFFERENCE		
4. FINAL ASSEMBLY OPS	✓		• DOCKING & ANTENNA INSTALL LESS COMPLEX
5. CONSTRUCTION EQUIP	✓		• FEWER TYPES AND LESS COMPLEX
6. CONSTRUCTION SYSTEM MASS AND COST (UNIT)	✓		• LIGHTER (3.2M Kg, 33%) • CHEAPER ($4.0B, 33%)
7. CREW REQUIREMENTS	✓		• 300 FEWER PEOPLE • $110M LESS/YR (33%)
8. LAUNCH SYSTEM	✓		• HIGH DENSITY COMPONENTS ALLOW REUSABLE SHROUD
9. SATELLITE ORBIT TRANSFER		✓	• LESS IMPACT ON SATELLITE DESIGN
10. SATELLITE TRANSPORTATION COST	✓		• CHEAPER ($300M, 6%)

Table 2-4. Construction Location Comparison
Photovoltaic Satellite

SPS-1634

- ORBITAL BASES
- SATELLITE (MODULE) AND ANTENNA CONST. OPERATIONS
- CONSTRUCTION EQUIPMENT
- CREW REQUIREMENTS
- ENVIRONMENTAL FACTORS
- CONSTRUCTION MASS AND COST
- SATELLITE DESIGN IMPACT
- ORBIT TRANSFER COMPLEXITY
- LAUNCH OPERATIONS
- TRANSPORTATION COST

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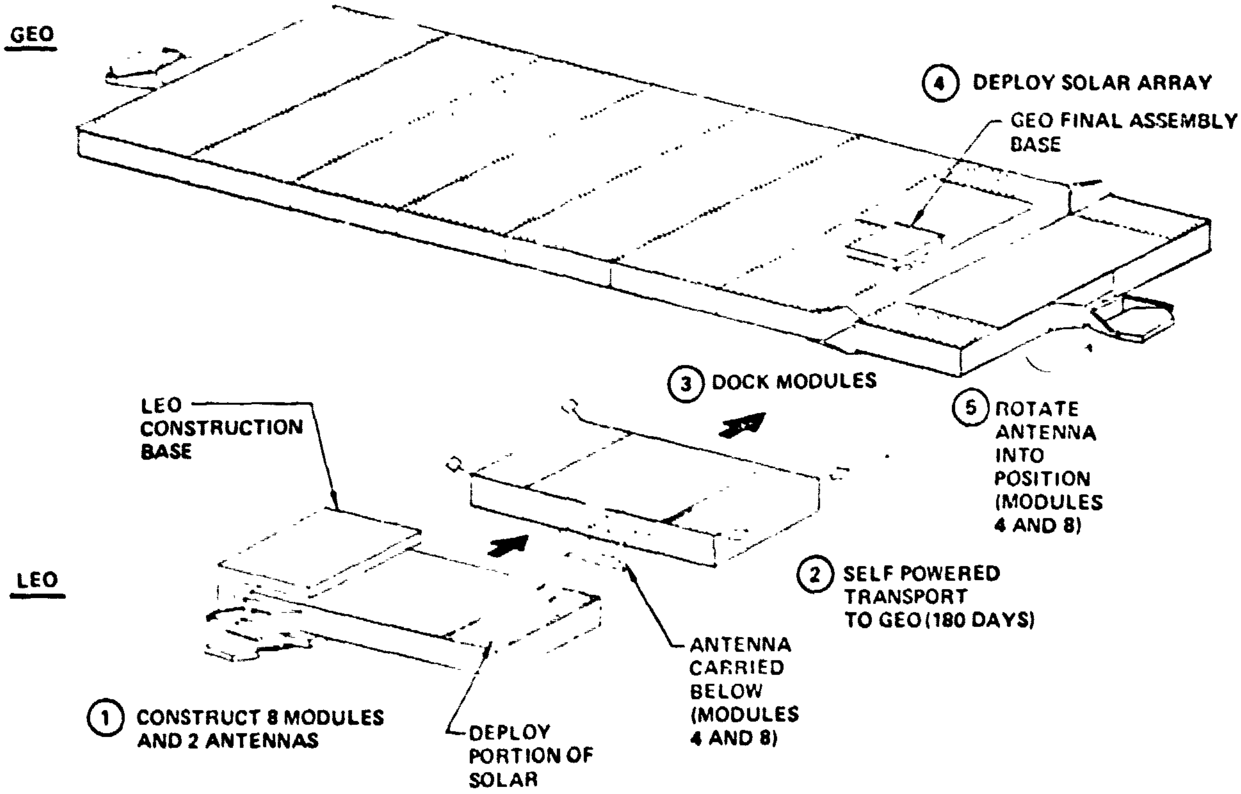

Figure 2-32. LEO Construction Concept
Photovoltaic Satellite

The GEO construction concept is illustrated in Figure 2-33 and begins with a staging depot, which has the capability to transfer payloads from a launch vehicle to orbit transfer vehicles and to house and maintain the orbit transfer vehicle fleet. Transfer of all payloads between LEO and GEO is accomplished using LO_2 LH_2 OTV's. Construction of the entire satellite including antenna is done at GEO. The reference satellite for the GEO construction option is a monolithic design rather than modular as in the case of LEO construction. The effect of this difference as well as others is discussed on subsequent charts.

2.2.2 Orbital Bases

Two principal bases are required for the construction of each satellite in the LEO construction option as illustrated in Figure 2-34. The bases for the photovoltaic option have been described earlier in the comparison of the two power generation system concepts. In summary, however, the LEO construction base consists of two connecting facilities, with one used for construction of satellite modules, while the other is used to construct the antennas. The GEO base provides basing for cranes used in the berthing of the modules and supports solar array deployment machines.

Two orbital bases are also required for the GEO construction option as shown in Figure 2-35. The GEO construction base has been sized to construct a satellite in one year and consequently, results in the same overall size as the base for LEO construction. This approach does result in moving the satellite construction facility in two directions rather than one. This has been judged to be more cost effective than having a full width facility and additional construction equipment and have this equipment sit idle half of the time. Additional discussion on this subject will occur in subsequent charts. Mass difference for this construction base compared to the base for LEO construction primarily reflects the additional mass required for shielding protection against solar flares. Other significant differences in the GEO construction base are the outriggers on the satellite facility to allow lateral direction indexing in addition to the movement of the antenna facility from one end of the satellite to the other. Again, both of these differences are the subject of subsequent charts.

The staging depot located in LEO for the GEO construction option is sized to support the construction of one satellite per year, and accordingly requires one SPS component OTV flight per day, based on a five day a week launch and flight schedule. As such, the depot must provide accommodations for three launch vehicle payloads, one being the SPS components and the other two being propellant tankers used to refuel the orbit transfer vehicles. Since the orbit transfer vehicle propellant loading requires slightly more propellant than can be provided by two tankers, a storage tank is also provided at the staging depot and is refuled every fourth OTV flight. Other docking accommodations are provided for a dedicated OTV used for GEO crew rotation resupply on a once per month basis. This operation also requires docking for supply modules and crew transfer vehicles. The operational crew size for the staging depot is 75 which can be accommodated in one module similar to the crew modules used in the GEO construction base. A transient crew quarters module is also provided to accommodate the 160 personnel rotated with each crew flight to the GEO base. A maintenance module is also included at this base for repair work primarily on the transportation systems.

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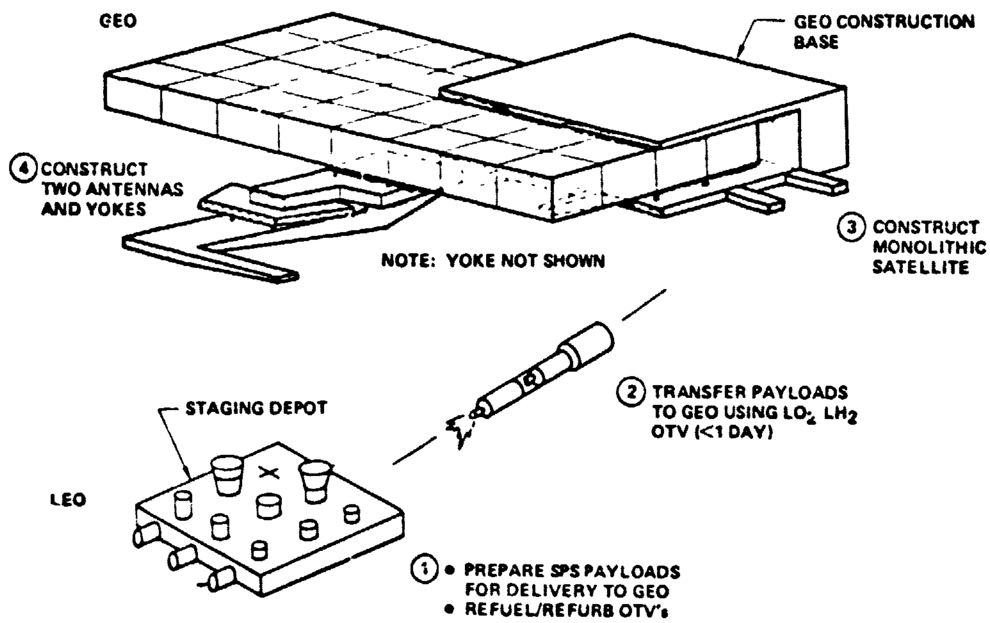

Figure 2-33. GEO Construction Concept

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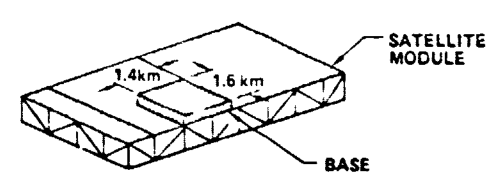

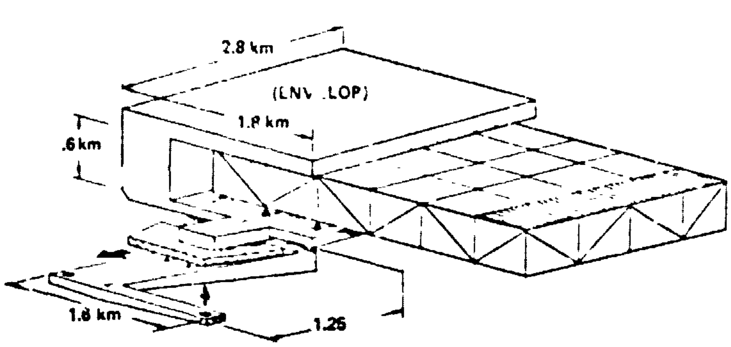

Figure 2-34. Orbital Bases
LEO Construction Concept

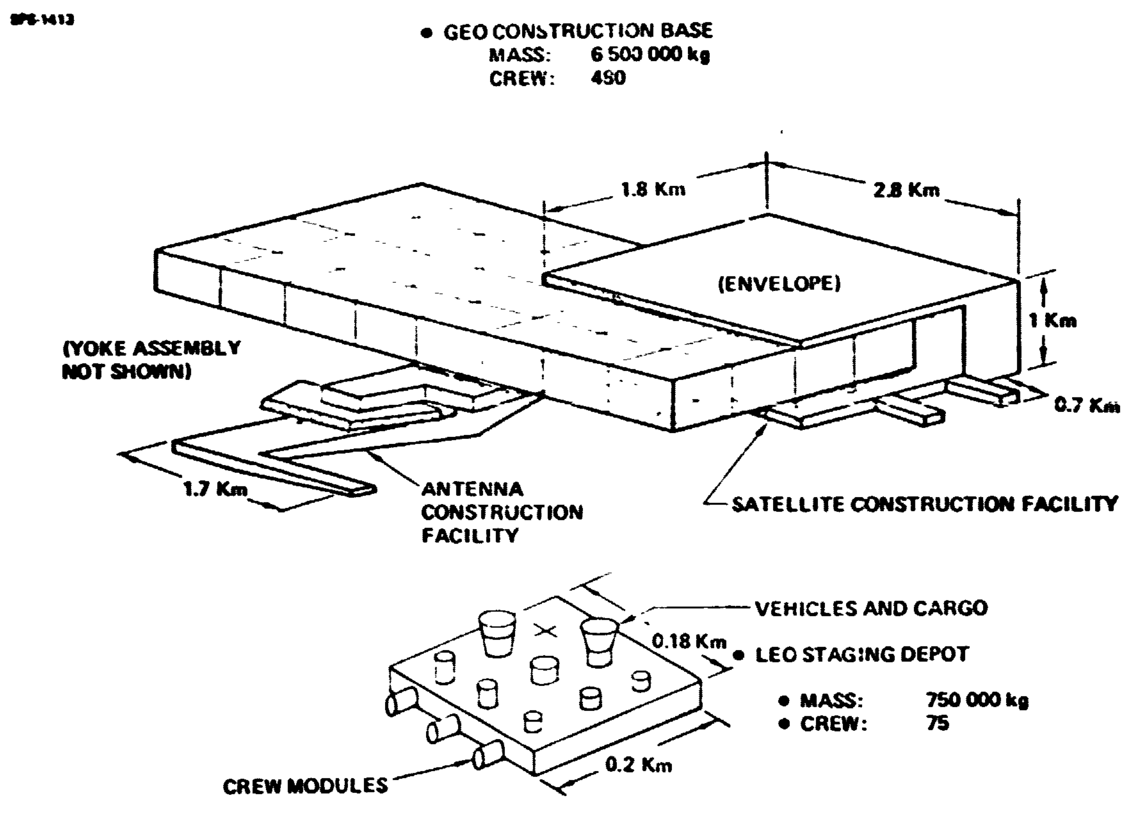

Figure 2-35. Orbital Bases
GEO Construction

2.2.3 Satellite and Antenna Construction Operations

Satellite Construction

The LEO construction operations associated with the photovoltaic satellite have previously been shown and described. In Figure 2-36, the operations are illustrated in a slightly different manner in order to show a more direct comparison with the GEO constructed satellite. Each satellite module is four bays wide and eight bays long. The module facility is four bays wide and consequently can construct a complete width of the module and results in indexing the module only in the longitudinal direction.

The GEO constructed satellite is monolithic in design (although it could also be modular, if so desired) and as a result has a construction width of eight bays. In order to obtain this width with the same size facility (least mass and cost) as a LEO construction base, indexing of the satellite is required in two directions as indicated in Figure 2-37. In general, four bays of the satellite are under construction at one time. With their completion, those bays are moved laterally and the remaining four bays of that row are constructed. When a given row is completed, it is then indexed in a longitudinal direction and the construction operation is repeated. In order to accomplish the lateral indexing in only two steps, outriggers have been added to the side of the satellite facility to enable indexing of four bays outside the construction envelope.

Antenna Construction and Installation

Antenna construction and installation also presents some significant differences in the two construction location options. Again, the photovoltaic LEO construction approach has been presented in the power generation system comparison, but is shown in Figure 2-38 in a manner to make a more direct comparison with the GEO construction approach. In summary, the yoke support structure of an antenna is made in the module facility and in between the third and fourth modules or between the seventh or eighth modules depending on whether it is the first antenna or second antenna being built. The antenna is made in its facility which remains permanently attached to the module facility. Construction of either the fourth or eighth module is then partially completed and the antenna and yoke attached at its proper location. Following module construction completion, the antenna is rotated under the module for transfer to GEO. Once GEO is reached, the antenna is rotated back up to its operating position.

The GEO construction concept also utilizes separate satellite and antenna facilities. However, in the reference case indicated, the antenna facility with antenna is required to free fly to the opposite end of the satellite and back as illustrated in Figure 2-39. It should be noted at this point, that the antenna construction installation approach indicated has been judged to be one of the best, if not the best, option for this particular task. Eight other options, involving variations of the antenna facility remaining attached, others with it independent, and also two separate antenna facilities were investigated and are reported in the final documentation.

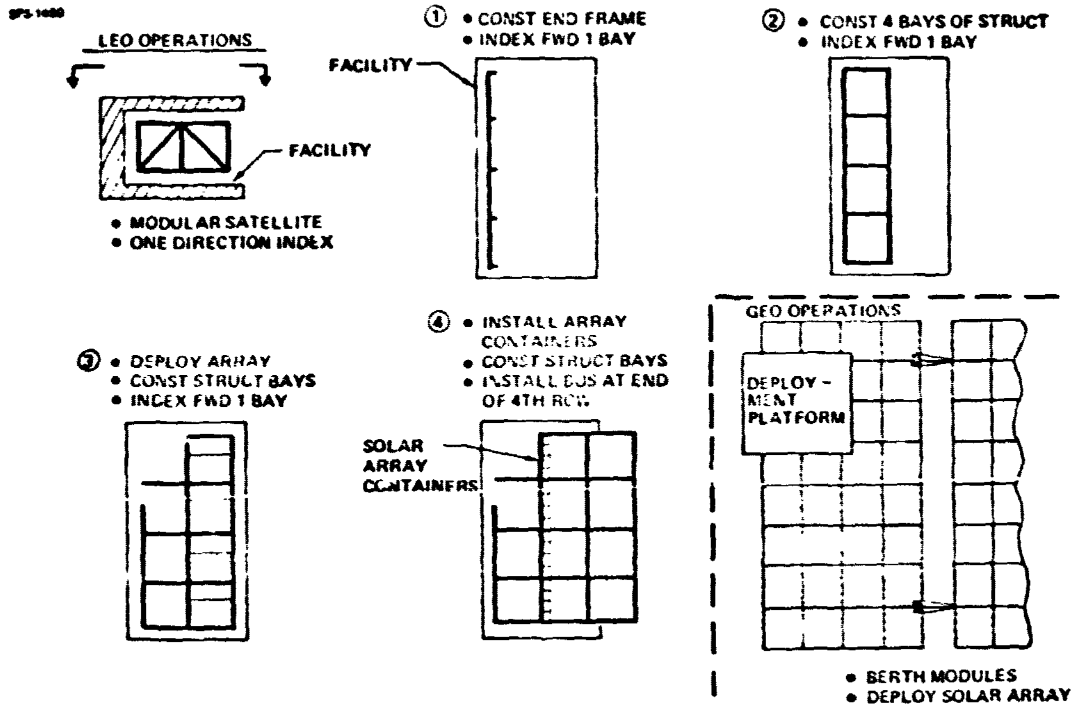

Figure 2-36. Satellite Construction Operations
LEO Construction Concept

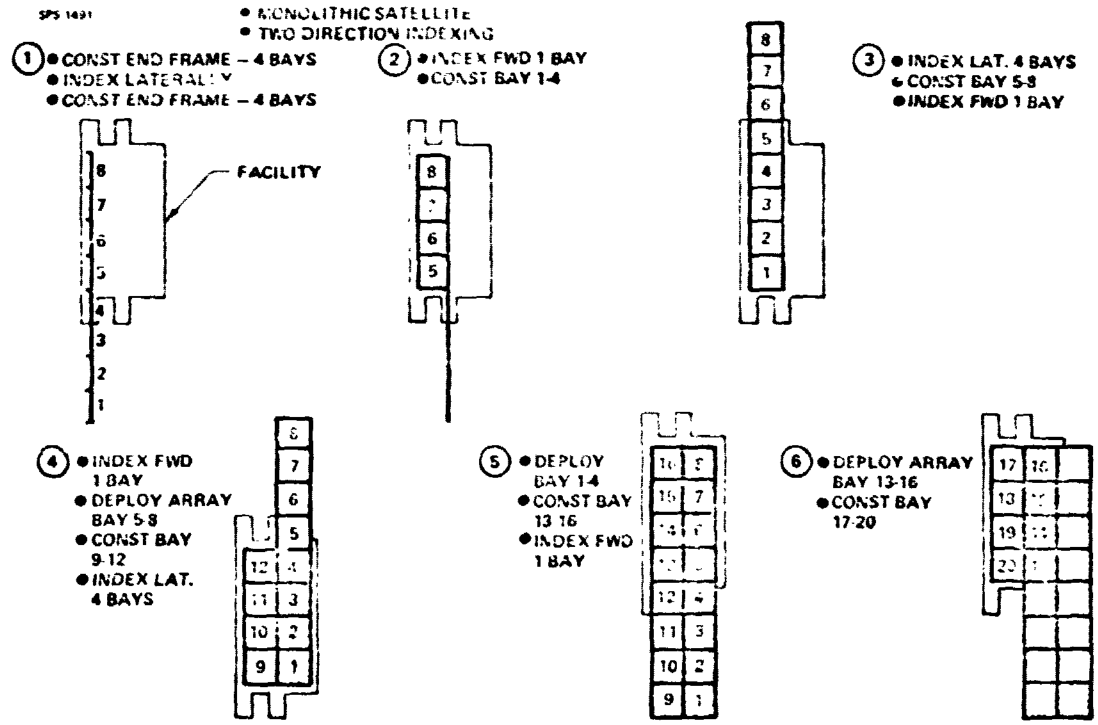

Figure 2-37. Satellite Construction Operations
GEO Construction Concept

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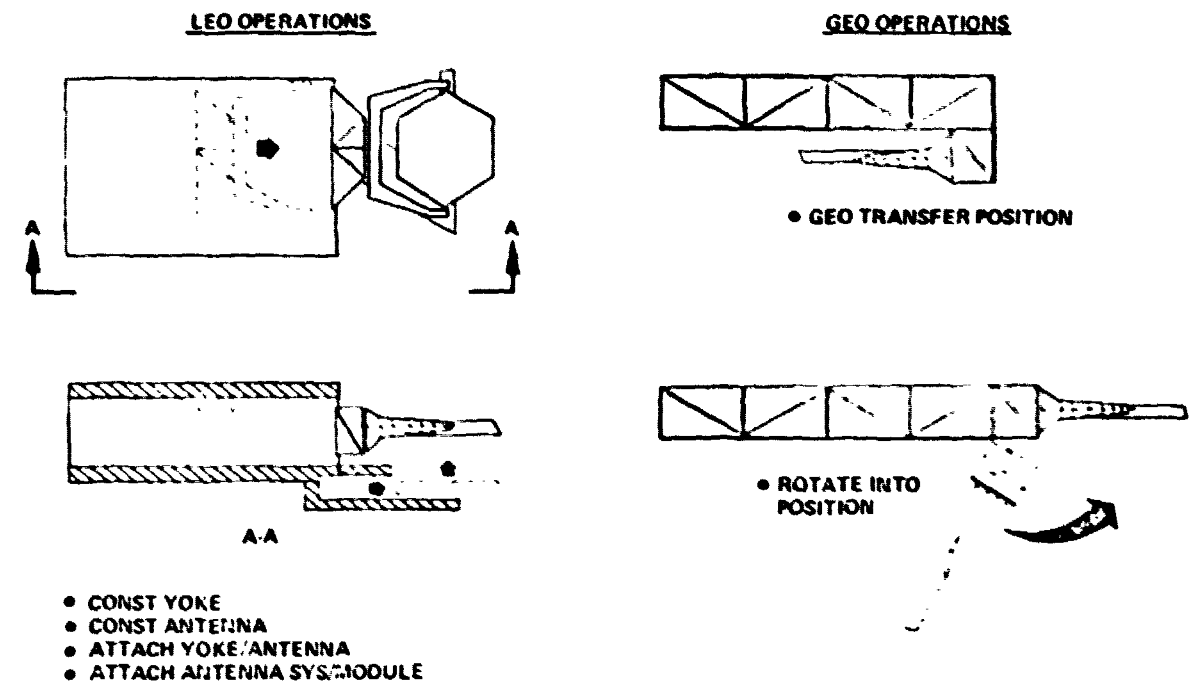

Figure 2-38. Antenna Construction and Installation LEO Construction

SPS-1483

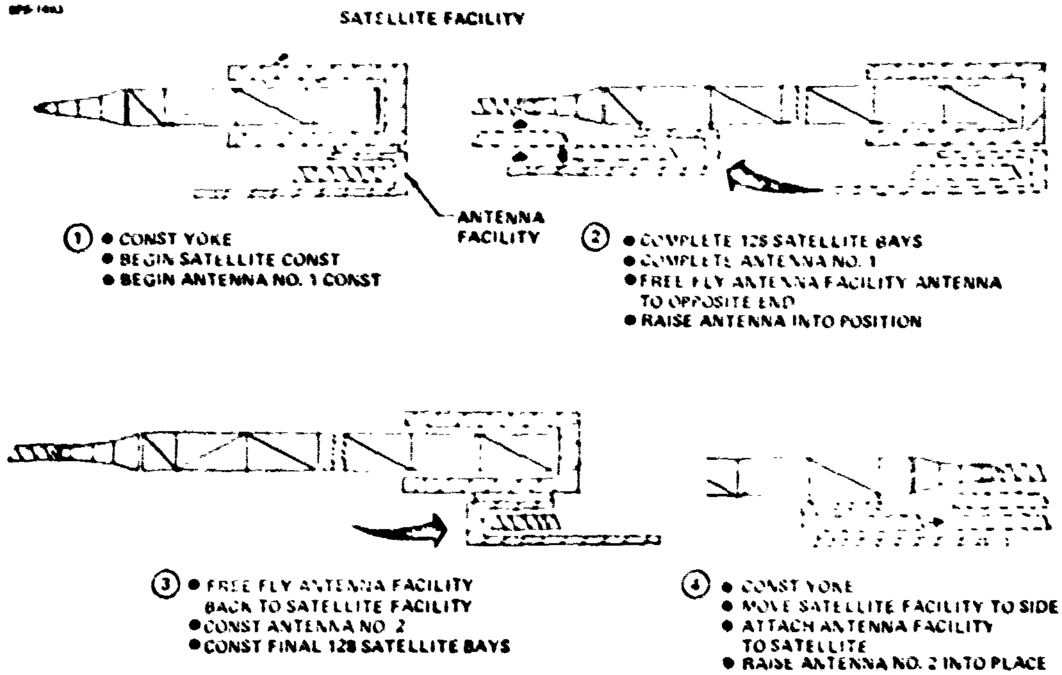

Figure 2-39. Antenna Construction and Installation GEO Construction Concept

In summary, the reference approach consists of the first antenna being made while the first half of the satellite is constructed including the yoke and support structure. At that point, the antenna facility with antenna is flown to the end of the satellite and docked and the antenna then attached to the yoke. The antenna facility is then flown back to the satellite facility (the short term separation of the two facilities simplifies the logistics problem in terms of supplying antenna components as well as living quarters for the crew). The remaining half of the satellite is then constructed, including a second yoke, while the antenna facility constructs the second antenna. Indexing of the satellite facility to the extreme edge of the satellite allows the antenna facility to be positioned to enable placement of the antenna into the yoke.

2.2.4 Crew Requirements

There is essentially no difference in orbital crew size between the two construction location concepts, although the distribution of personnel is considerably different as shown in Figure 2-40. Crew size for the main construction base indicates 480 people in LEO concept while this same number is required in GEO for the GEO concept. Staging depot and final assembly manning requirements are also found to be nearly the same.

2.2.5 Environmental Factors

Several key environmental factors should be considered when comparing the two construction location options. A summary of these factors is presented in Table 2-5 plus an additional chart dedicated to the topic of collision with man-made objects.

The principal difference between the two construction location options, in terms of natural radiation, is the large amounts of solar flare shielding which must be provided for all crew modules located at GEO. Steady-state radiation would make EVA at GEO considerably worse than at LEO although only a bare minimum of suit EVA is anticipated in either case.

Occulations of the construction base at LEO occur 15 times a day, while a base at GEO is only occulted 88 times per year. The principal effects of occultation are on the electrical power supply and thermal aspects of the structure. In the case of power requirement, the GEO option requires less power due to not having to recharge nickel hydrogen batteries used for the occultation. The penalty for the larger power system is relatively small, however, when one is in the era of low mass, low cost solar arrays. Although a GEO base is certainly more continuously illuminated, the construction base itself produces shadows. Consequently, both construction locations require a large amount of power for lighting purposes. Use of graphite epoxy structure in both the satellite as well as the construction base structure should minimize the impact of thermal effects.

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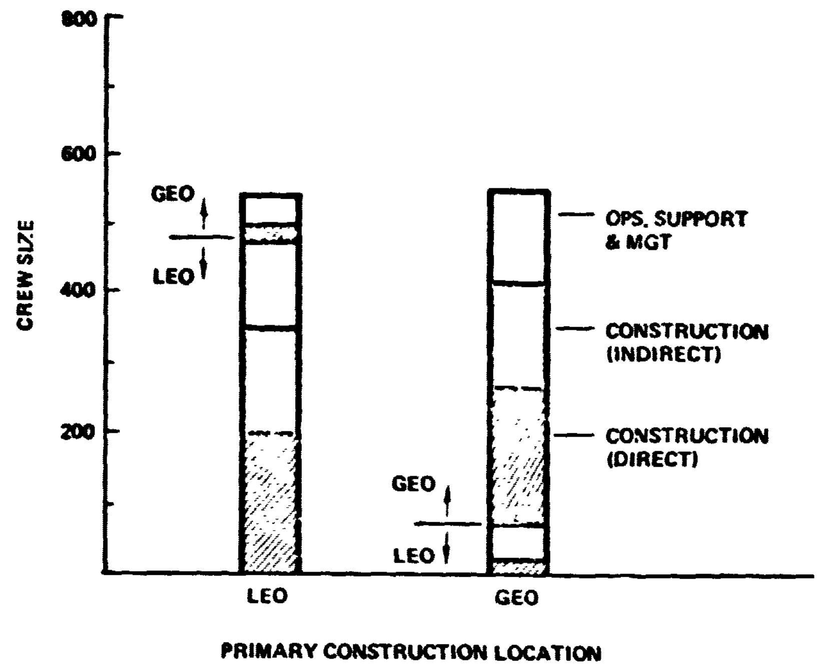


Figure 2-40. Crew Size and Distribution

Table 2-5. Environmental Factors Summary

SPS-1611

FACTOR	LEO BASE	GEO BASE
• RADIATION		
• SOLAR FLARE	2-3 GM/CM^2	20-25 GM/CM^2 (115 000 KG/100 PEOPLE)
• EVA	SO. ATLANTIC ANOMALY RESTRICTION	STEADY STATE IS WORSE
• OCCULTATION		
• BASE POWER REQ'TS:	3600 KW	2500 KW
• LIGHTING:	• REQ'D AT BOTH LOCATIONS (Δ OF 100-150 KW)	
• THERMAL EFFECTS:	• NO SIGNIFICANT DIFFERENCE IF GRAPHITE EPOXY IS USED	
• GRAVITY GRADIENT & DRAG:	• GRAVITY GRADIENT CONST MODE USED FOR BOTH LOCATIONS • LEO PROP REQ'T GREATER BY 800 KG/DAY	
• COLLISION WITH MAN-MADE OBJECTS	• POTENTIAL GREATER FOR LEO BUT AVOIDANCE MANEUVERS • REDUCE PROBABILITY TO NEAR ZERO	

Most construction concepts will orient the construction base so it is passively stable for attitude control and minimize gravity gradient torque. Although the LEO construction case required considerably more orbit keeping attitude control propellant per day, it still results in less than one HLLV launch per year for this propellant makeup.

Large amounts of debris from man-made space systems have resulted in some concern regarding LEO construction. The analysis conducted has indicated the potential is greater with construction in LEO however, simple avoidance maneuvers can reduce the probability of being hit to near zero. Further discussion on this topic is presented in the following paragraph.

Collision Analysis

The collision analysis has been done for an environment predicted for the year 2000, including an addition of 500 objects per year since 1975. Results of this analysis are shown in Figure 2-41. This data indicates that the LEO construction approach could have forty additional collisions if no preventive action is taken. However rescheduled orbit altitude corrections can essentially eliminate the problem of collision with little or no additional penalty. Thrust initiation or termination during orbit transfer can also be used to prevent collisions. In summary, there should be no difference between the two concepts regarding the number of collisions although the LEO construction approach does require slightly different operations, including the use of the tracking and warning systems.

2.2.6 Construction Mass and Cost

The comparison of the mass and cost associated with the orbital bases and their equipment is presented in Figure 2-42. The LEO construction approach has orbital bases which are approximately 0.5 million kg lighter and cheaper by 0.5 billion

2.2.7 Satellite Design Impact

The design impact on the satellite for the case of LEO construction and self-power has been described earlier in the description of the photovoltaic satellite. A summary of the key impact areas is presented in Table 2-6. In the area of solar array, an oversizing of 5 percent has been included to compensate for the inability to completely anneal out all the damage to the cells caused by radiation occurring during transfer and for the mismatch in voltage output between the damaged and undamaged cells

The structural impact includes both that of modularity and oversizing. Modularity includes additional vertical members used around the perimeter of the satellite module and lateral beams at the end of the modules as well as the penalties for the transfer of the 15 million kg antenna supported underneath the module. (It should be noted that all module structure has been sized to that dictated by the modules used to transfer the antenna)

SPS-1606

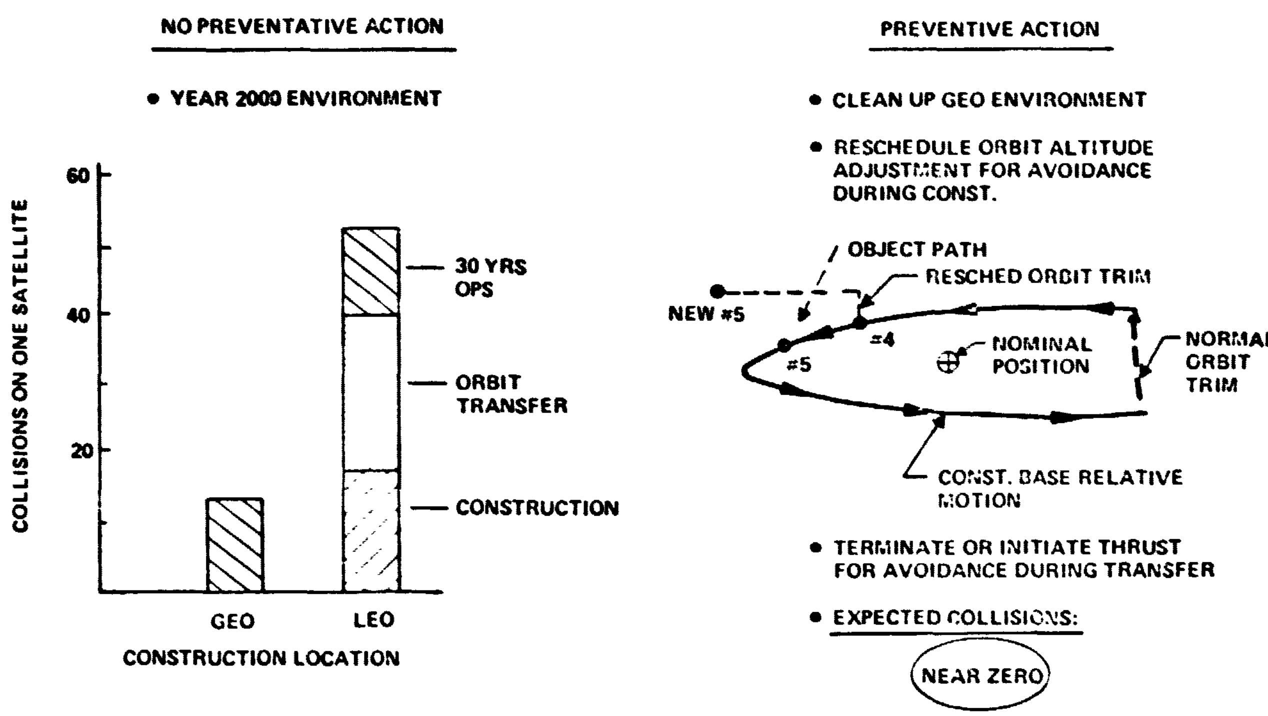


Figure 2-41. Collisions with Man Made Objects

SPS-1708

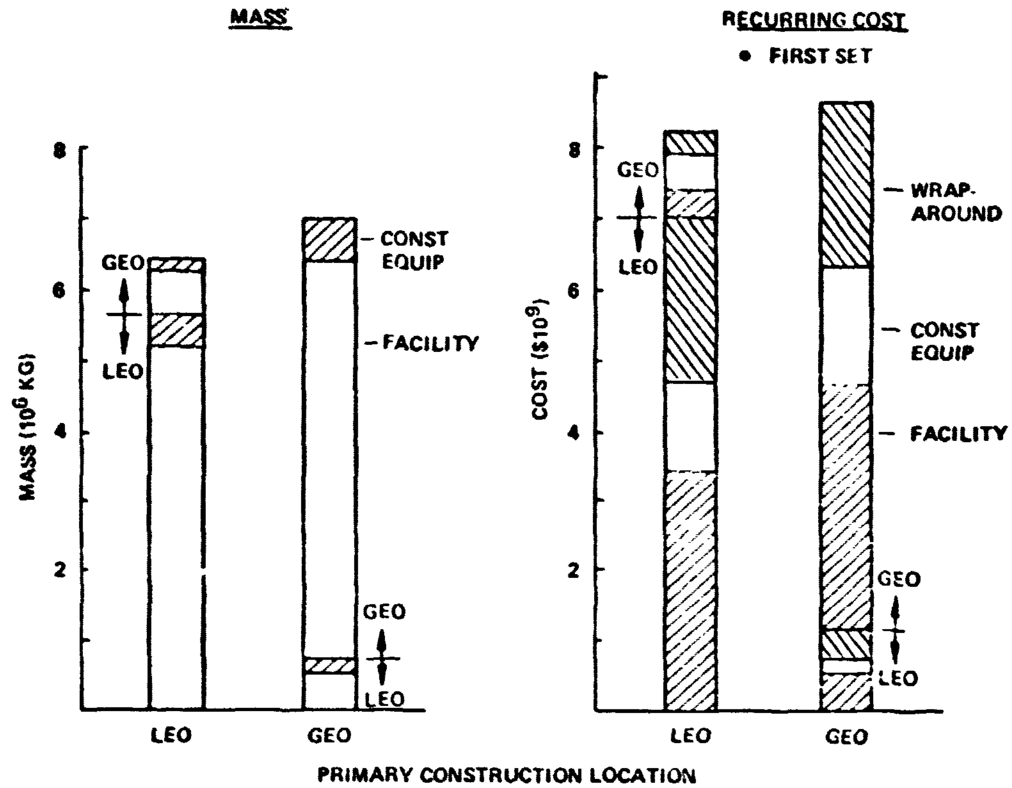


Figure 2-42. Construction Mass and Cost Photovoltaic Satellite

Table 2-6. Satellite Design Impact Summary LEO Construction

SPS 1620

SELF POWER TRANSFER

IMPACT	REASON	PENALTY
• SOLAR ARRAY	• OVERSIZING FOR RADIATION DEGRADATION	• 2.75M Kg [1]
• STRUCTURE	• MODULARITY • OVERSIZING	• 1.07M Kg • 0.34M Kg [1]
• POWER DISTRIBUTION	• EXTRA LENGTH DUE TO OVERSIZING	• 0.07M Kg [1]

[1] FUNCTION OF SELF POWER PERFORMANCE CHARACTERISTICS

The power distribution penalty is related to the additional length of bus caused by the oversizing of the array. The total mass penalty for a LEO consructed satellite is approximately 4.2 million kg for the selected self power transportation system. It should be noted however that the array oversizing and power distribution penalty depend on the particular performance characteristics selected for the self power system.

2.2.8 Transportation Requirements

Transportation requirements associated with the payloads of each construction location concept are shown in Figure 2-43; there is no OTV propellant mass included.

The difference in satellite mass only reflects the structural mass penalty of the additional vertical and lateral members and loads caused by transfer of the antenna. Oversizing and power distribution penalties are all a function of orbit transfer characteristics and consequently are chargeable to the orbit transfer system itself.

Differences in crew and supply requirements delivered to LEO primarily reflect additional orbit keeping attitude control propellant requirements. The key difference, however, is in the mass which must be delivered to GEO.

Facility transportation requirements reflect the initial placement task as well as in the case of the GEO bases (both options), that mass that must be moved to the longitude location where the next satellite is to be constructed. The principal difference in the two main construction bases is that the six crew modules in the GEO concept each have approximately 115 000 kg of additional mass for solar flares shelters.

2.2.9 Satellite Orbit Transfer Complexity

Concepts

Prior to comparing the overall configuration characteristics of the orbit transfer options, a review of the basic propulsion systems seems appropriate. A simplified schematic and key operation characteristics are presented in Figure 2-44. Chemical systems using LO_2 LH_2 have demonstrated an I_{SP} of 470 sec. An important factor in the consideration of a LO_2/LH_2 system for SPS application is that it must be reusable because of economic considerations.

The electric propulsion system involves more system elements but has several significant benefits in terms of offering an I_{SP} of 7500 sec and the system does not have to be recoverable for economic viability. Although 120 cm argon thrusters and baseline power processing system has not been demonstrated, ion propulsion systems using mercury and 15 cm dia thrusters have been flight tested and thrusters up to 150 cm have been ground tested.

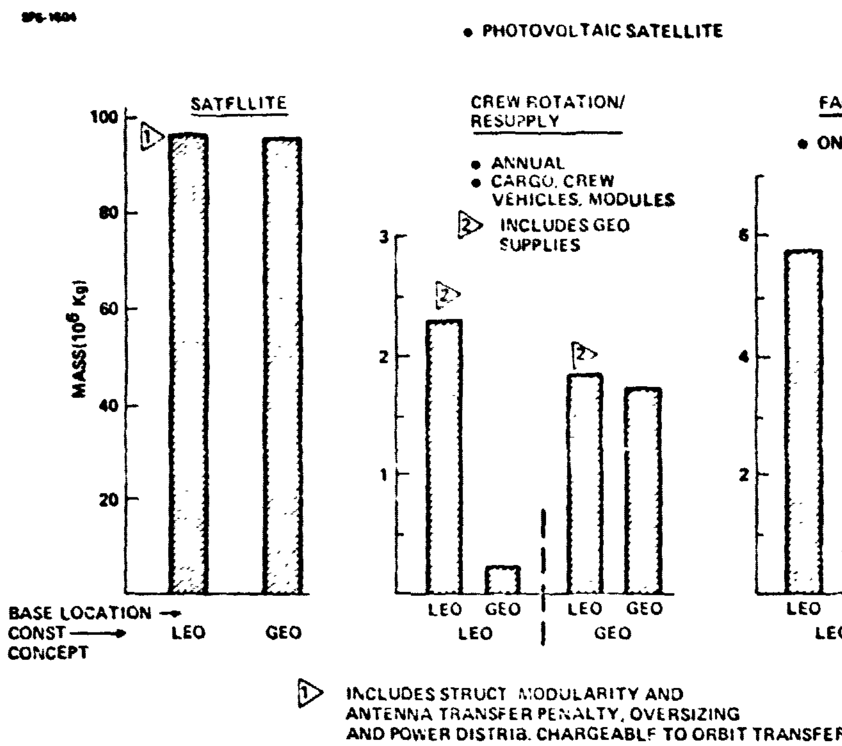


Figure 2-43. Transportation Requirements LEO vs GEO

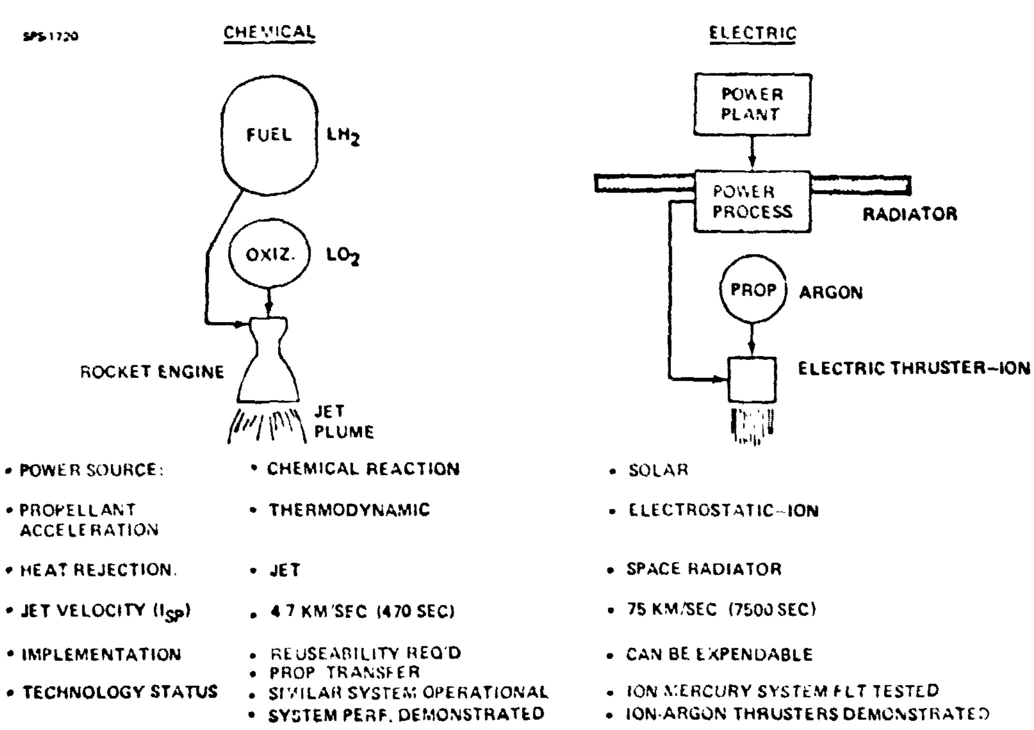


Figure 2-44. Orbital Transfer Propulsion Options

Configurations

The self power orbit transfer system used in the LEO construction approach has previously been shown in the power generation system comparison and is shown again in Figure 2-45. In summary, electrical power generated by the solar arrays is used to power ion electric thrusters which use argon propellant. LO_2/LH_2 thrusters are also included to provide attitude control during all occultations and during short periods of time early in the transfer (up to 2 500 km altitude) when thrust required to counter gravity grading torque is greater than that provided by electric thrusters.

The cost optimum trip time and I_{sp} are respectively 180 days and 7,000 seconds. Variation in number of thrusters, propellant tanks, etc. do occur in the design to compensate for the case of whether a module is being transported alone or with an antenna.

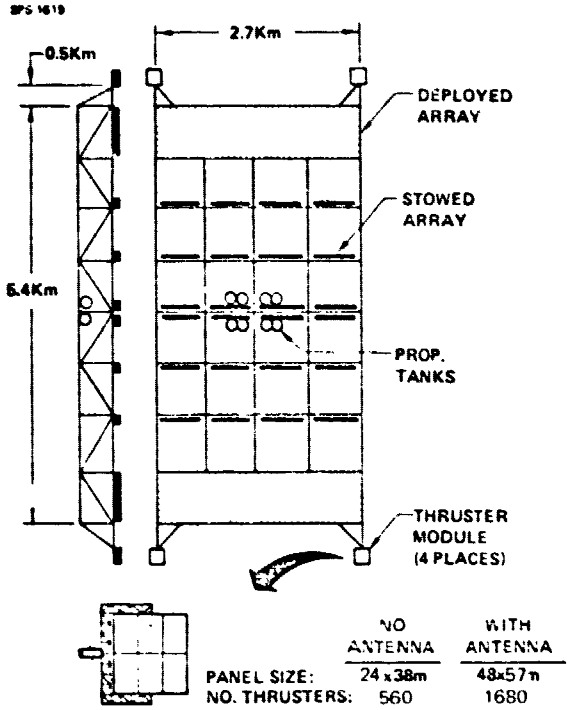
The GEO construction OTV is a space based common stage (2-stage) system with both stages having identical propellant capacity. The configuration and mass characteristics for this vehicle are shown in Figure 2-46. The first stage provides approximately 2/3 of the delta V requirement for boost out of low earth orbit at which point it is jettisoned for return to the low earth orbit staging depot. The second stage completes the boost from low earth orbit as well as providing the remainder of the other delta V requirements to place the payload at GEO and the required delta V to return the stage to the LEO staging depot. Subsystems for each stage are identical in terms of design approach. The basic difference includes the use of four engines in the first stage due to thrust-to-weight requirements of approximately 0.15. The second stage requires additional auxiliary propulsion due to its maneuvering requirements in the docking of the payload to the construction base at GEO. The OTV shown has been sized to deliver a payload taken directly from the launch vehicle (400 000 kg). As a result, the OTV startburn mass is approximately 890 000 kg with the vehicle having an overall length 56 meters.

Flight Operations

Flight operation differences between the two orbit transfer vehicle options is influenced by their orbit transfer time. In the case of the self power system illustrated in Figure 2-47, as many as 1200 revolutions around the Earth occur prior to reaching GEO when using a 180 day transfer. The flight schedule including a 40 day construction phase indicates as many as five modules can be in transit at any one time for the case of 8 modules per satellite.

The mission profile for the common stage LO_2/LH_2 OTV for GEO construction is shown in Figure 2-48 and indicates a 40 hour mission requirement for the first stage and 85 hours for a second stage which delivers the payload. These times include about 12 hours for refueling and refurbishment of each stage. With the requirements of one OTV flight per day with the GEO construction option, a total of two lower stages and four upper stages are required. Operated in this manner, as many as six independently operating stages can be in flight at one time.

SPS 1619



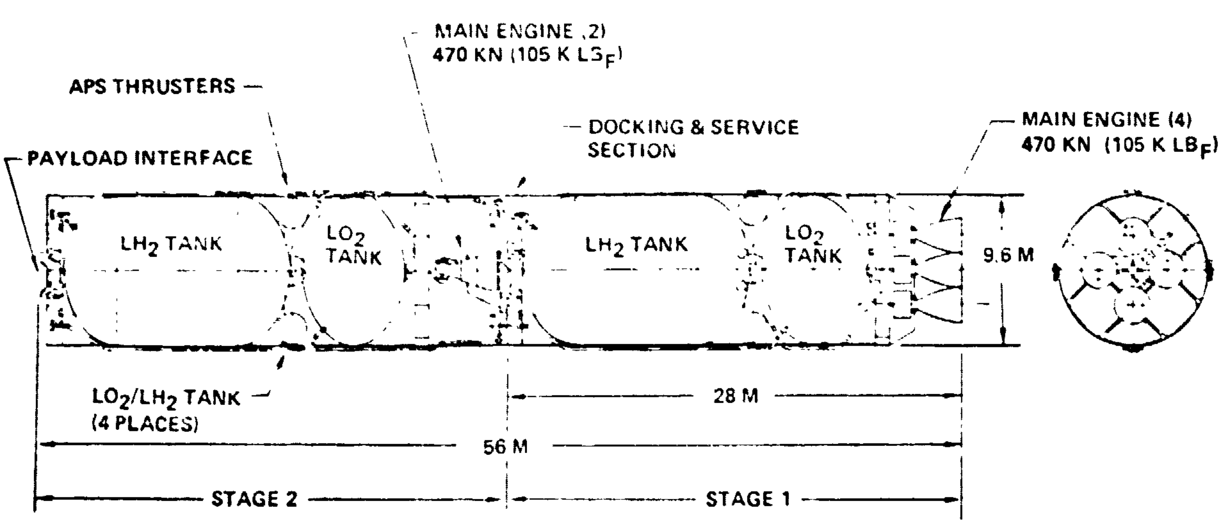
GENERAL CHARACTERISTICS

- 5% OVERSIZING (RADIATION)
- TRIP TIME = 180 DAYS
- ISP = 7000 SEC

MODULE CHARACTERISTICS	NO ANTENNA	WITH ANTENNA
• NO MODULES	6	2
• MODULE MASS (10^6KG)	8.7	23.7
• POWER REQ'D (10^6Kw)	0.3	0.81
• ARRAY %	13	36
• OTS DRY (10^6KG)	1.1	2.9
• ARGON (10^6KG)	2.0	5.6
• LO_2/LH_2 (10^6KG)	1.0	2.8
• ELEC THRUST (10^3N)	4.5	12.2
• CHEM THRUST (10^3N)	12.0	6.0

Figure 2-45 Self Power Configuration
Photovoltaic Satellite

SPS 203



- PAYLOAD CAPABILITY = 400,000 KG
- OTV STARTBURN MASS = 890,000 KG
- STAGE CHARACTERISTICS (EACH
  - PROPELLANT = 415,000 KG
  - INERTS = 29,000 KG
    (INCLUDING NONIMPULSE PROPELLANT)
- 280 OTV FLIGHTS PER SATELLITE

Figure 2-46. Space Based Common Stage OTV
GEO Construction

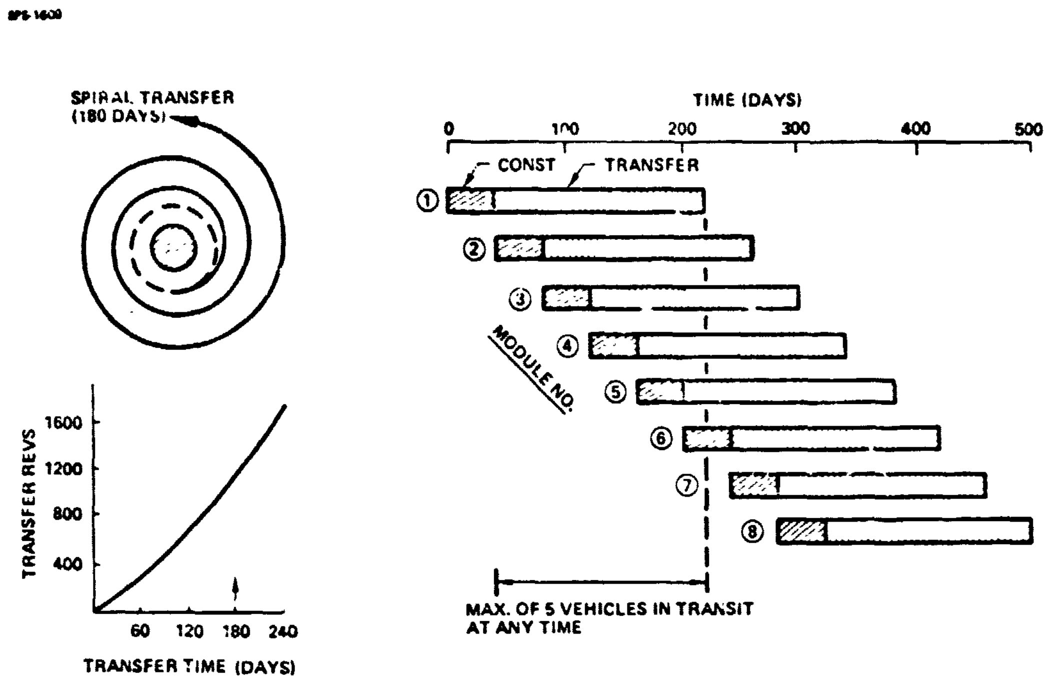


Figure 2-47. Flight Operations
Self Power Orbit Transfer

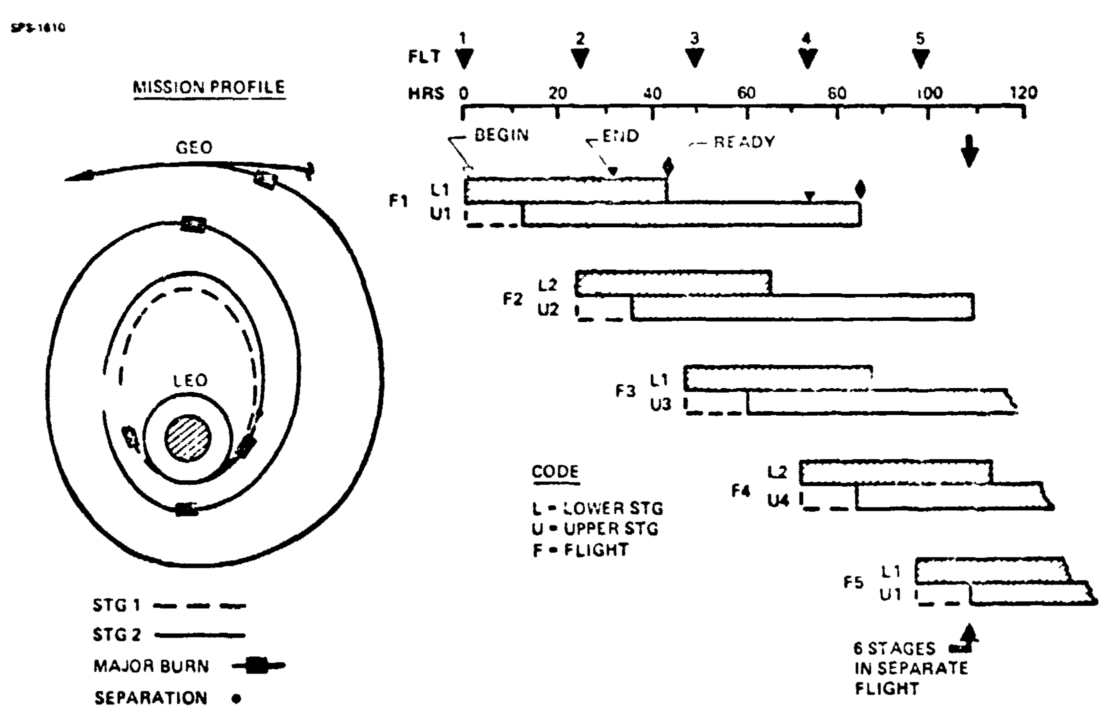


Figure 2-48. Flight Operations
Chemical Orbit Transfer

2.2.10 Crew Rotation/Resupply Transportation

Crew rotation/resupply systems consist of a shuttle growth vehicle for delivery of personnel to LEO and the standard two-stage ballistic ballistic launch vehicle for delivery of supplies and propellant to LEO. The utilization of these vehicles is shown in Figure 2-49. Crew and supply delivery between LEO and GEO use a two-stage LO_2/LH_2 OTV. The OTV for the LEO concept is about ½ as large as that for the GEO concept and requires one-third as many flights because of the significantly fewer people at GEO. Since the total orbital crew size for the two concepts is about the same, the number of delivery flights to LEO are also the same. Cargo flights to LEO, however, are three times greater for the GEO approach primarily due to the large OTV propellant requirements.

2.2.11 Launch Operations

Total cargo mass which must be handled by the launch vehicle are shown in Figure 2-50 and reflect both the payload requirements indicated earlier and the OTV propellant and hardware requirements. For the three system elements that require transportation, payload requirements are not too different, however, the inclusion of the orbit transfer system requirements add significantly to the total mass which must be delivered by the HLLV. Again, it should be emphasized that the satellite transportation requirements are by far the most dominating.

As previously stated, the reference cargo launch vehicle is a two-stage ballistics ballistic device using LO_2/RP in stage one and LO_2/LH_2 in stage two. The configuration and performance associated with this vehicle is shown in Figure 2-51. GLOW for this system is approximately 10.5 million kg for the case of delivering 391 000 kg to the construction base or the staging depot located in LEO with orbit characteristics of 477 km altitude and 31 degrees inclination. Vehicle operations include first stage separation at a relative velocity of 2970 meters per second and downrange water landing approximately 815 km. The second stage delivers the payload to the LEO base; docks and returns one day later and also uses a water landing.

A most significant impact in the area of launch operations is the difference in the number of launches required to support each construction location option. The number of flights indicated in Figure 2-52 are only those relating to the delivery of satellite components and orbit transfer provisions for the satellite and are for the case of constructing four satellites per year. As would be expected from the transportation requirements chart presented earlier, the LEO construction option requires only one half as many Earth launches as the GEO construction option.

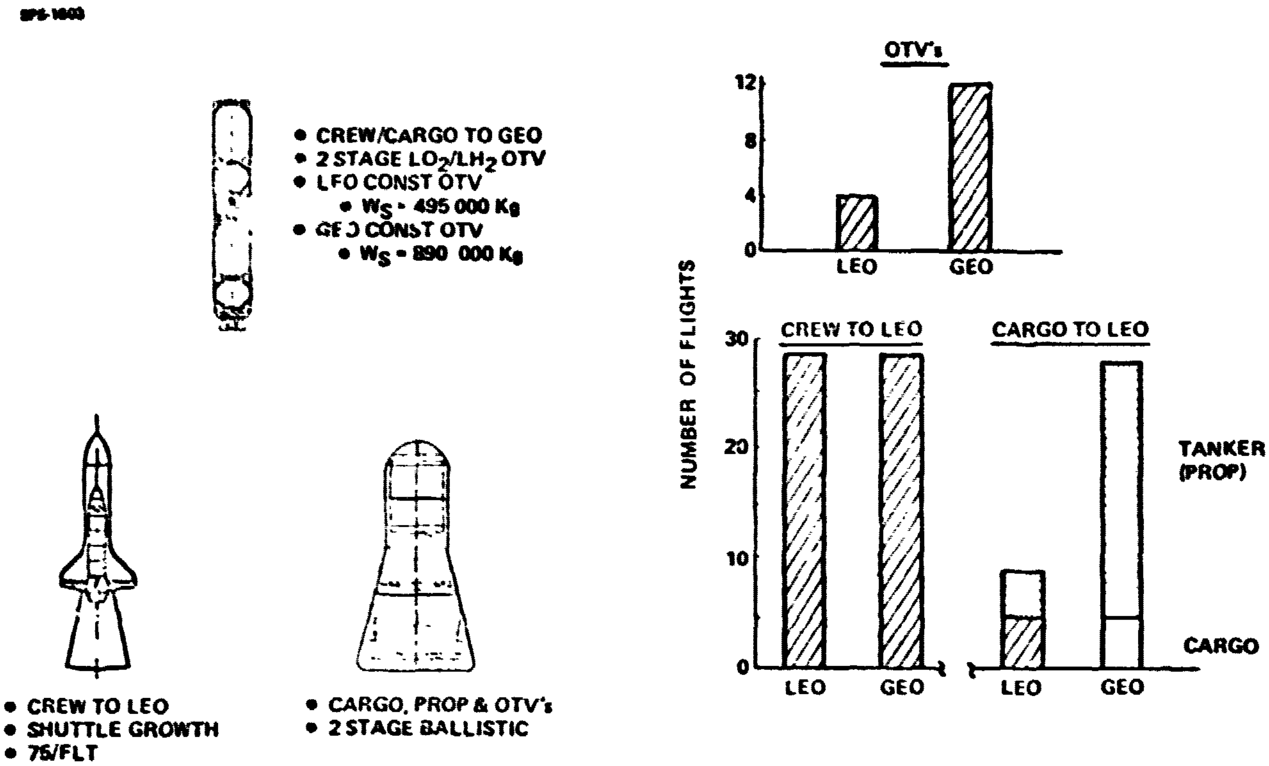


Figure 2-49. Crew Rotation/Resupply Transportation

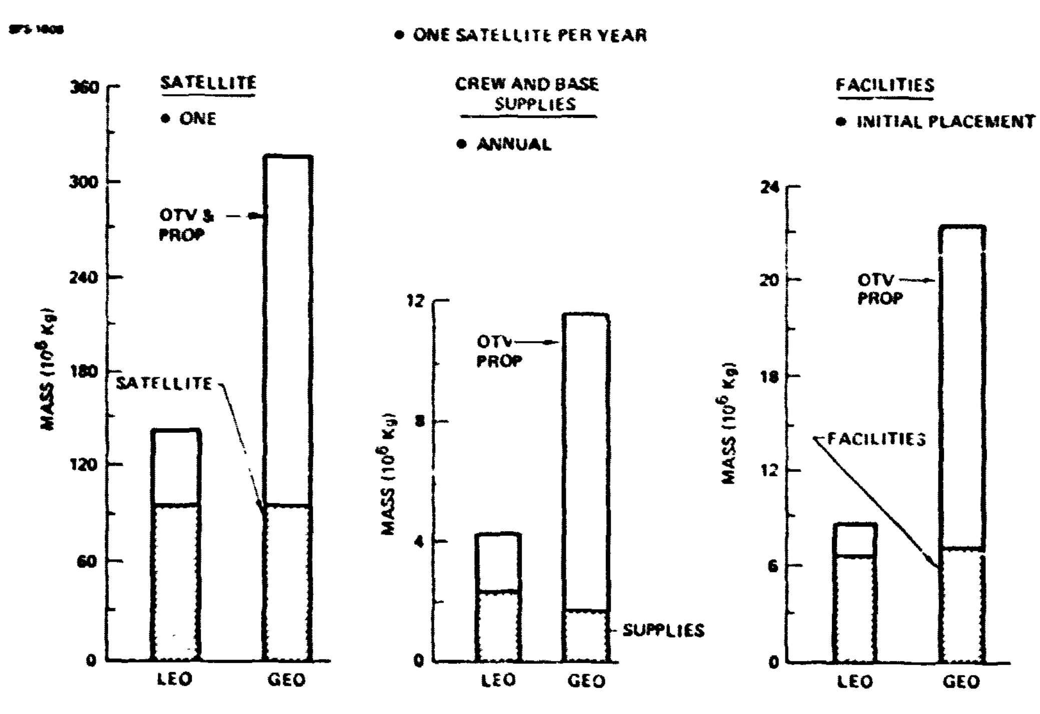


Figure 2-50. Total Cargo Mass to LEO

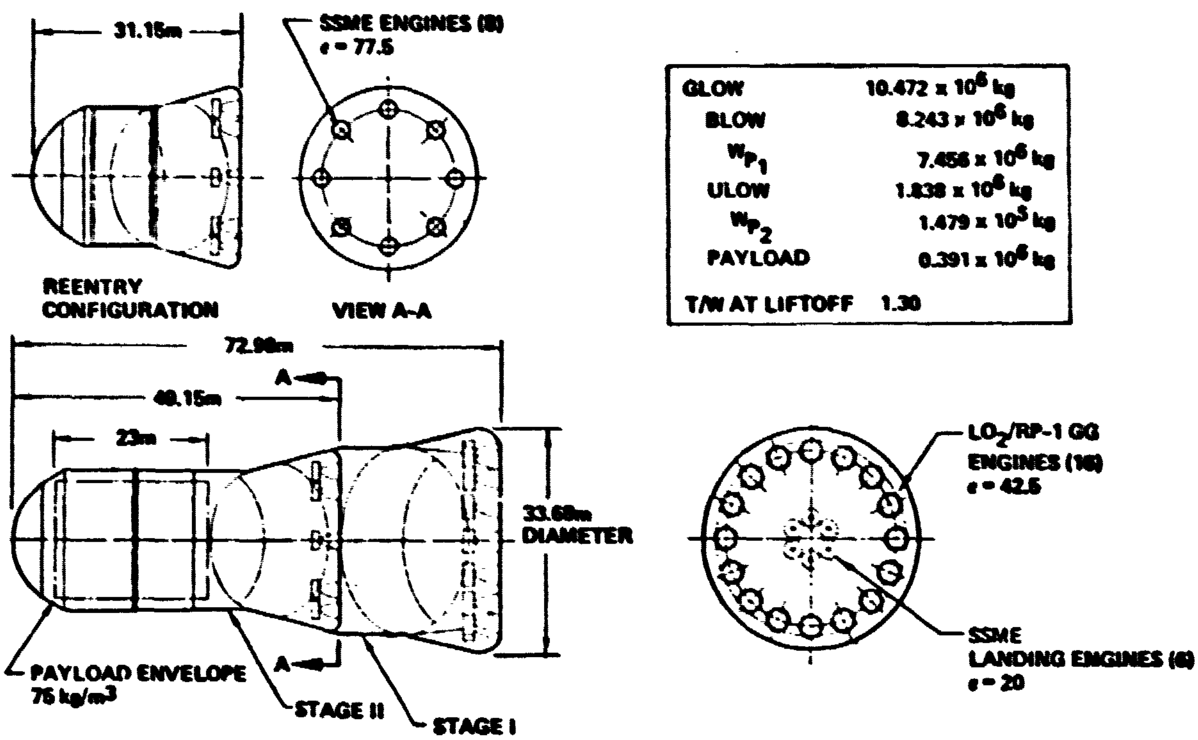


Figure 2-51. SPS Cargo Launch Vehicle

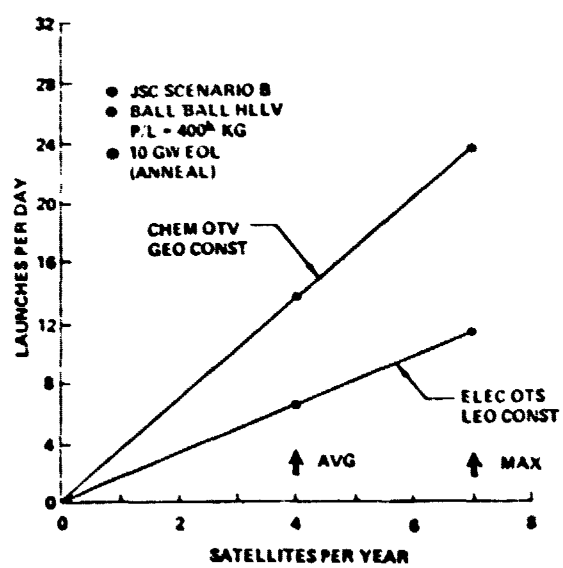


Figure 2-52. Number of HLLV Launches

2.2.12 Transportation Costs

Total transportation cost for the three major system elements is presented in Figure 2-53. Cost is related to that associated with one satellite, but reflect rates associated with four satellites per year. The Earth-LEO bar increments reflect the cost of getting payloads to LEO. Accordingly, the LEO-GEO increment relates to cost of refueling orbit transfer vehicles and their unit cost. In the case of satellite delivery, the interest increment relates to the self power trip time of 180 days and the additional interest accrued. (Note: Revenue is not lost, only delayed 180 days - the same revenue period still exists.)

The dominating factor in this comparison is that satellite transportation with LEO construction using self-power provides a $2 billion (33% savings) over the GEO construction approach. Crew rotation/resupply transportation cost are also $150 million (36%) lower for the LEO construction concept along with a $200 million savings for the initial placement of the construction bases.

2.2.13 Construction Location Summary

A summary comparison of the LEO and GEO construction locations is presented in Table 2-7 with an indication of which approach is most desirable. Compared in this manner, a number of parameters result in no significant differences between the two construction location options. However, a number of parameters give a clear indication that LEO construction is most desirable. Most notable among these being transportation costs, simplified launch operations, and reduced construction base mass and costs. One parameter has been judged to be in favor of the GEO construction approach (the impact on satellite design) although this data is then fed into the transportation comparison which still favors the LEO construction approach.

2.3 CONSTRUCTION TRANSPORTATION CONCLUSIONS

The conclusions regarding the issues of power generation system comparison and construction location comparison as influenced by construction and transportation factors are shown in Table 2-8. This data indicate a distinct advantage for a photovoltaic satellite (CR=1) constructed in low earth orbit and transported to GEO using self power.

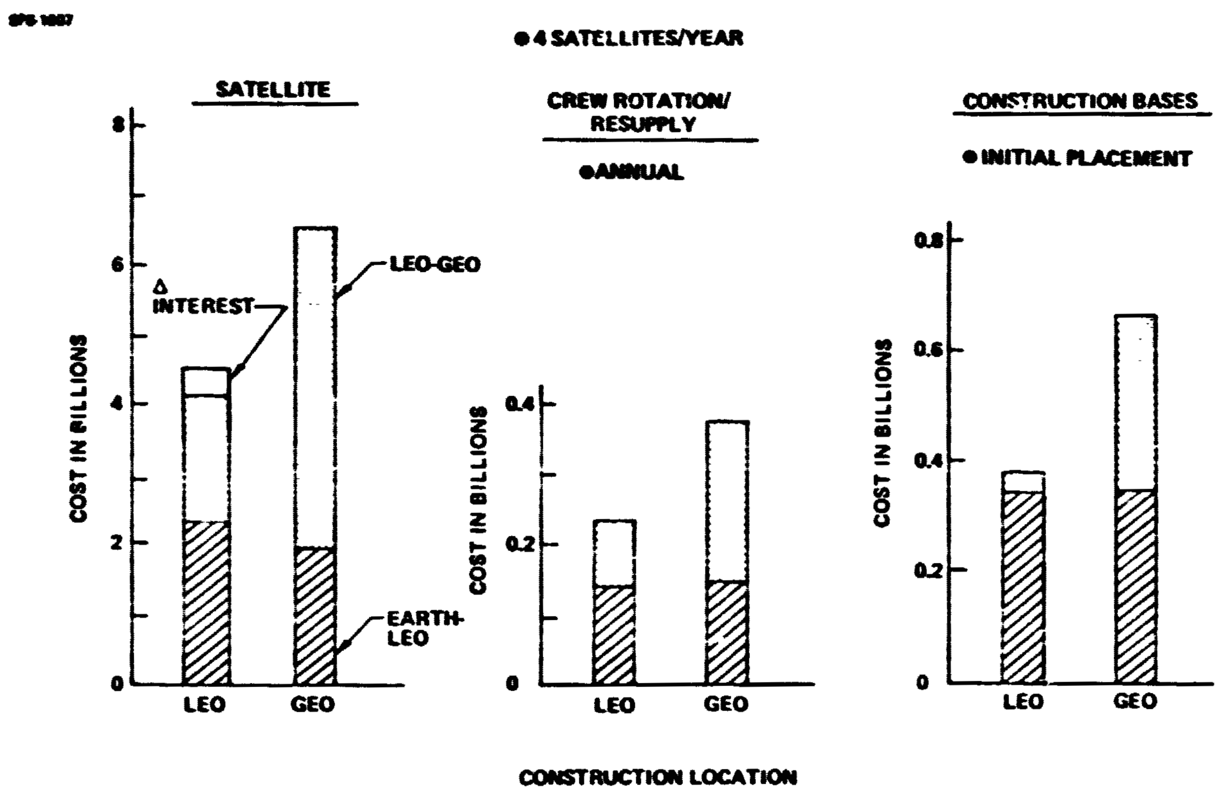


Figure 2-53. Transportation Cost LEO vs GEO

Table 2-7. Construction Location Summary

SPS-1623

COMPARISON PARAMETER	LOCATION: LEO	LOCATION: GEO	RATIONALE
• CONST. BASES	NO SIGNIF. DIFFERENCE		• SAME CONST BASE • STAGING DEPOT vs FINAL ASSY FACILITY
• SATELLITE (PWR GEN) CONST	NO SIGNIF. DIFF.		• 1 vs 2 DIRECTION INDEXING OFFSET BY MODULE DOCKING
• ANTENNA INSTALLATION	✓		• ANTENNA FACILITY DOESN'T MOVE
• CONSTRUCTION EQUIP	NO SIGNIF. DIFF.		
• CREW REQUIREMENTS	✓		• SAME SIZE BUT MAJORITY AT LEO
• ENVIRONMENTAL FACTORS	NO SIGNIF. DIFF.		• ALL FACTORS CAN BE HANDLED WITH ACCEPTABLE SOLUTIONS
• CONSTRUCTION MASS & COST	✓		• LIGHTER (0.6M Kg; 7%) • CHEAPER ($0.5B, 6%)
• SATELLITE DESIGN IMPACT		✓	• NO Δ OVERSIZING, MODULARITY OR POWER DIST. PENALTY • BOTH USE ELEC. PROPUL. ACS
• ORBIT TRANSFER COMPLEX		✓	• APPROX SAME NO. VEHICLES IN FLT • TECHNOLOGY MORE ADVANCED
• LAUNCH OPERATIONS	✓		• ONE-HALF AS MANY LAUNCHES
• TRANSPORTATION COST	✓		• CHEAPER ($2B; 33%)

✓ INDICATES MOST PROMISING CONCEPT

Table 2-8. Construction/Transportation Conclusions

SPS-1622

- THE PHOTOVOLTAIC SATELLITE (CR = 1) OFFERS SIGNIFICANT ADVANTAGES
  - LESS COMPLEXITY IN FACILITIES AND CONSTRUCTION EQUIP
  - SMALLER CONSTRUCTION CREW
  - LOWER CONSTRUCTION COST
  - LOWER TRANSPORTATION COST
- LEO CONSTRUCTION OFFERS A SIGNIFICANTLY LOWER TRANSPORTATION COST. OTHER FACTORS ARE COMPARABLE:
  - CONSTRUCTION OPERATIONS
  - SATELLITES IN EITHER CASE REQUIRE ELECTRICAL PROPULSION AND 3 AXIS ATTITUDE CONTROL
  - ENVIRONMENTAL FACTORS CAN BE HANDLED WITHOUT EXCESSIVE PENALTIES

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3.0 CONSTRUCTION

3.1 INTRODUCTION

This section contains the construction analysis for the two types of satellites and construction location alternatives:

- Photovoltaic satellite LEO construction (Sec. 3.2.1)
- Photovoltaic satellite GEO construction (Sec. 3.2.2)
- Thermal Engine satellite LEO construction (Sec. 3.3.1)
- Thermal Engine satellite GEO construction (Sec. 3.3.2)

3.1.1 Objectives

The objectives of the construction analysis were the following:

- Assist the satellite system designers evolve configuration designs that incorporate in-orbit construction considerations.
- Analyze the configurations to determine a rational construction task breakdown.
- Design facilitized construction approaches.
- Define the overall integrated construction sequence.
- Develop detailed construction approaches for each major construction task to a sufficient level of detail that feasibility is obvious.
- Determine the envelope and functional requirements for the major construction equipment items.
- Design the base cargo handling and distribution system.
- Define a crew organizational concept and estimate the crew size.
- Compare the construction differences between the two orbital construction locations.
- Estimate construction base and equipment mass and cost.
- Provide inputs to the component packaging analysis that determines how many Earth launches are required.
- Refine the collision with manmade objects analysis of Part I.

3.1.2 Construction Philosophy

During the course of the construction analysis a distinct "style" or "construction philosophy" began to emerge that characterizes our approach to orbital construction (Table 3.1-1). This philosophy will be evident in both the photovoltaic and the thermal engine satellite construction analyses.

Table 3.1-1 Construction Philosophy [1]

CONCEPT	RATIONALE
• USE CONSTRUCTION FACILITIES	• NON-FACILITIZED APPROACH WOULD REQUIRE CONSTRUCTION EQUIPMENT TO BE MOUNTED ON SATELLITE STRUCTURE. THIS WOULD REQUIRE EACH SATELLITE STRUCTURE TO INCORPORATE EXTRA STRENGTH (EXTRA MASS) AND BRING ABOUT COMPETITION FOR THE SAME SPACE AT THE SAME TIME BY THE CONSTRUCTION EQUIPMENT.
• DECOUPLE CONSTRUCTION OPERATIONS	• CONSTRUCTION OPERATIONS MADE AS INDEPENDENT OF EACH OTHER AS PRACTICAL SO THAT SLOW DOWNS IN ONE OPERATION DO NOT IMPEDE PROGRESS OF OTHERS.
• FABRICATE MAJOR SUB-ASSEMBLIES IN PARALLEL	• MAJOR SUBASSEMBLIES SUCH AS THE ANTENNA AND THE MODULES ARE ASSEMBLED IN SEPARATE BASE AREAS SO THAT THE MAXIMUM CONSTRUCTION TIME IS AVAILABLE FOR EACH ASSEMBLY.
• MAJOR SUB-ASSEMBLY BASE AREAS ARE CONTIGUOUS (THIS WAS NOT POSSIBLE FOR ALL CASES)	• ALL OF THE MAJOR SUBASSEMBLY AREAS ARE INCORPORATED INTO A CONTIGUOUS BASE. • ALLOWS MATERIAL AND PERSONNEL LOGISTICS TO EMANATE FROM A COMMON CARGO HANDLING OR WAREHOUSE AREA. • ELIMINATES NEED FOR FREE-FLYING DOCKING OF SUBASSEMBLIES .
• USE INDEXING/SUPPORT MACHINES	• THE SUBASSEMBLIES ARE SUPPORTED AND INDEXED BY INDEXING-SUPPORT MACHINES THAT MOVE ON FACILITY TRACKS. THE SUBASSEMBLIES ARE NOT SUPPORTED OR INDEXED BY THE CONSTRUCTION MACHINERY

[1] SEQUENCE OF LISTING DOES NOT IMPLY RELATIVE IMPORTANCE OF THE CONCEPTS

Table 3.1-1 (Continued)
Construction Philosophy ▷1

CONCEPT	RATIONALE
• USE FACILITY TRACK SYSTEM	• ALL MOVING MACHINERY TRAVELS ON A COMMON GAGE FACILITY TRACK SYSTEM. THIS ALSO PROVIDES ACCESS TO ALL MACHINES BY RESUPPLY AND PERSONNEL TRANSPORTERS. THIS AVOIDS USING FREE FLYERS OR COMPLICATED CABLE TRANSPORT SYSTEMS. USING A COMMON GAGE TRACK SYSTEM WOULD ALLOW A COMMON SELF PROPELLED TRANSPORTER MODULE TO BE DESIGNED THAT WOULD BE CONFIGURED TO ADAPT TO SPECIFIC PAYLOADS.
• USE HIGHLY AUTOMATED CONSTRUCTION MACHINERY	• THE LARGE QUANTITY, HIGHLY REPETITIVE ASSEMBLY OPERATIONS WILL REQUIRE A HIGH DEGREE OF AUTOMATION IN ORDER TO ATTAIN A SATISFACTORY PRODUCTION RATE USING A MINIMUM OF IN ORBIT CONSTRUCTION PERSONNEL.
• USE CASSETTES FOR THE LARGE QUANTITY PARTS	• TO ADAPT TO THE HIGHLY AUTOMATED PROCESSES AND TO REDUCE COMPONENT HANDLING, IT WILL BE NECESSARY TO HAVE THE PARTS FABRICATED ON EARTH PACKAGED FOR SHIPMENT TO ORBIT IN CASSETTES THAT CAN BE LOADED DIRECTLY INTO THE ASSEMBLY MACHINES WITHOUT REPACKAGING.
• OPERATORS USED FOR SUPERVISORY CONTROL	• THE OPERATORS WILL EXERCISE SUPERVISORY CONTROL OF THE CONSTRUCTION MACHINERY WHEREVER PRACTICAL IN ORDER TO GET MAN OUT OF THE LOOP SO THAT THE HIGH PRODUCTION RATES CAN BE ATTAINED. NO "HANDS ON" ASSEMBLY OPERATIONS TO BE USED.
• COMPONENTS FABRICATED ON EARTH AND THEN ASSEMBLED IN ORBIT	• FOR THE FIRST FEW SPS'S, THIS APPROACH IS MORE FEASIBLE THAN BRINGING RAW OR PARTIALLY PROCESSED MATERIALS INTO ORBIT AND THEN CONVERTING THEM TO COMPONENTS.

▷1 SEQUENCE OF LISTING DOES NOT IMPLY RELATIVE IMPORTANCE OF THE CONCEPTS

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3.2 PHOTOVOLTAIC SATELLITE CONSTRUCTION

This section contains the construction analysis of the photovoltaic satellite. There are two construction approaches that were analyzed: 1) LEO construction–wherein the satellite is assembled in modules at LEO and then the modules are self-powered to GEO and joined together to form the total satellite (Section 3.2.1), and 2) GEO construction–wherein the satellite is constructed as a contiguous unit at GEO (see Sec. 3.2.2).

In the following sections, for both of the satellite types, the LEO construction approach is addressed in detail and then the GEO construction approacn is analyzed to illustrate its differences. In the detailed construction analysis sections, the construction tasks, construction facilities, construction sequences, construction machinery, logistics systems, manning and mass and cost estimates are described in detail.

3.2.1 LEO Construction Concept

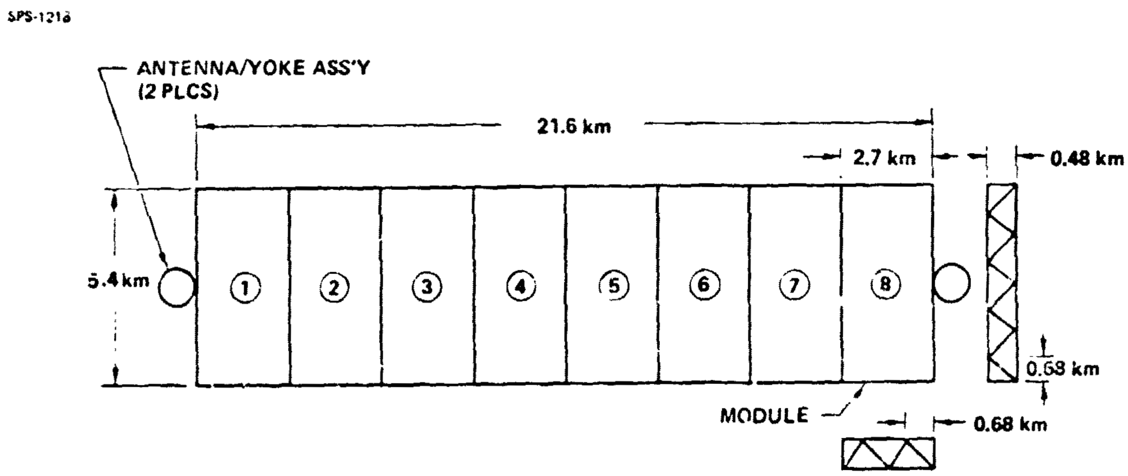
The reference photovoltaic satellite configuration for construction in LEO is shown in Figure 3.2-1. The LEO construction concept is illustrated in Figure 3.2-2 and entails constructing 8 modules and 2 antennas at a LEO construction base, deploying a portion of the solar array on each of the modules to provide power for the self-power transfer to a GEO base where the modules are joined together and the antennas erected.

The construction operations at the LEO base are described in Section 3.2.1.1.1 and those at the GEO base are described in Section 3.2.1.1.2.

The top-level construction timeline is shown in Figure 3.2-3. This timeline shows that the LEO construction operations are completed after 340 days have elapsed and that the completed satellite is ready to generate power after 580 days.

Other Options
Other LEO construction concepts were considered but were discarded:

- 16 Modules This concept was the baseline at Part 1 of this study. It was discarded after considering the difficulties of docking modules and controlling two edges as well as reducing the modularity mass penalty.



- EIGHT MODULES
- 97 MILLION kg
- 1300 km OF 20 M BEAM
- 102 km² OF SOLAR ARRAY
- 65 km OF POWER BUSES
- 1.6 km² OF ANTENNA SURFACE AREA

Figure 3.2-1 Photovoltaic Satellite Configuration

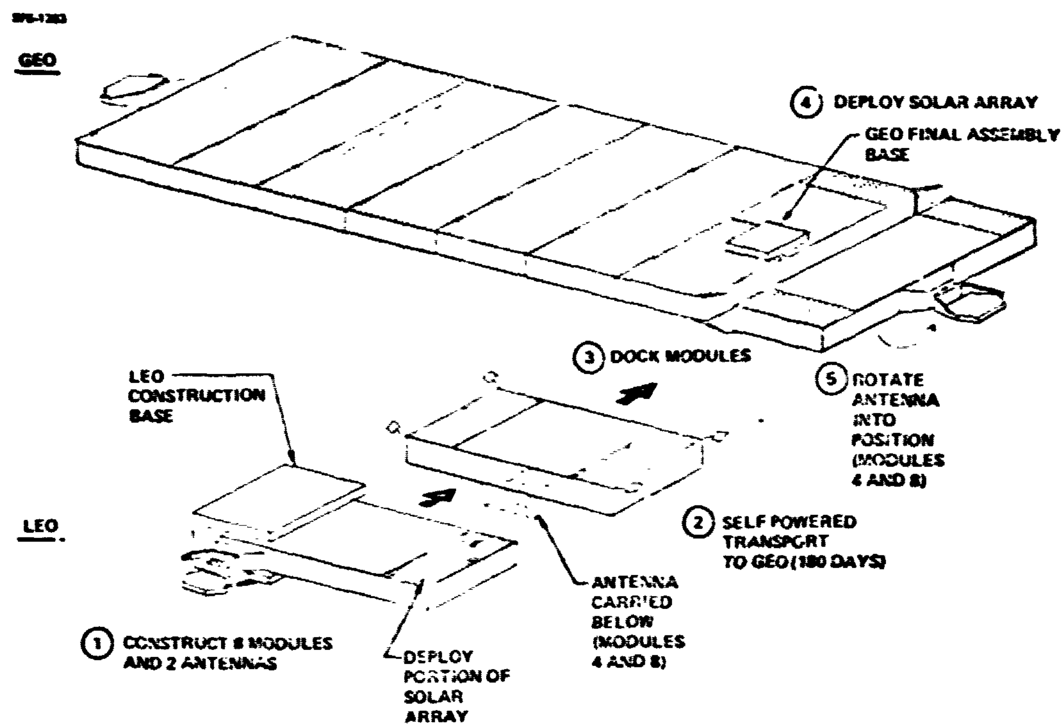


Figure 3.2-2 LEO Construction Concept
Photovoltaic Satellite

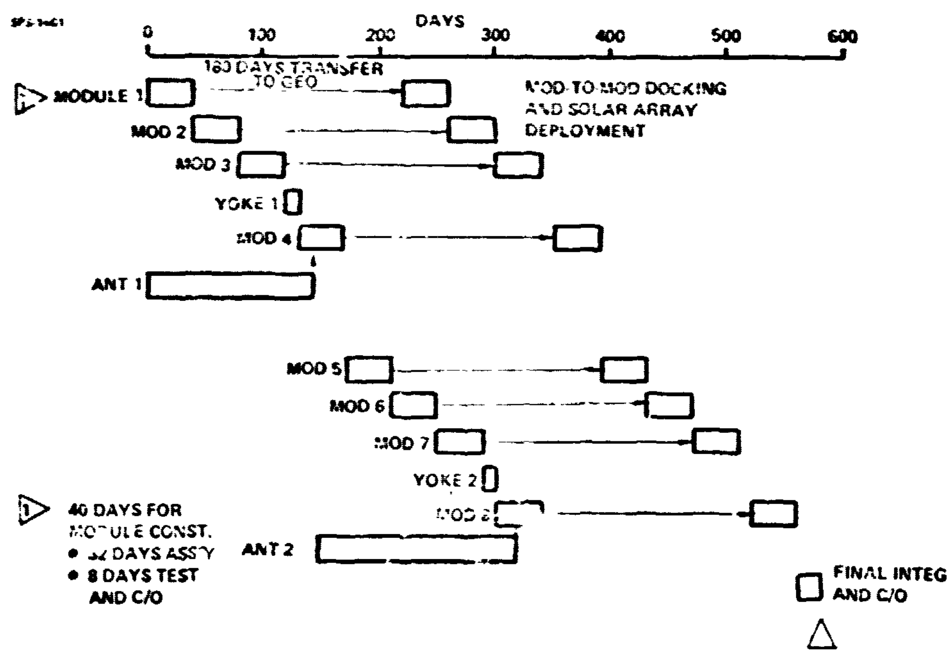


Figure 3.2-3 Photovoltaic Satellite LEO Construction Timeline

- Construct antenna at GEO—this was considered as an option to avoid having the difficulties in attaching the assembled antenna to a module at LEO and to avoid installation problems at GEO. This approach was abandoned when satisfactory solutions to these problems were found. It was also determined that it would be significantly less expensive to construct the antenna at LEO due to the crew transportation and resupply cost associated with placing large numbers of people at GEO. This option is discussed in greater detail in Section 4.3.

3.2.1.1 LEO Base Construction Analysis

In this section, the LEO operations are described in detail. This is followed by discussions of the LEO base, environmental factors, crew operations, and the cost and mass summaries.

3.2.1.1.1 LEO Construction Operations

3.2.1.1.1.1 Top-Level LEO Construction Tasks

The construction tasks to be accomplished at the LEO base are summarized in Figure 3.2-4. Each of these construction tasks are described in detail in the following sections.

3.2.1.1.1.2 LEO Construction Base

The construction base to be used at LEO to accomplish the tasks described in the previous section is shown in Figure 3.2-5. There are two primary construction areas: 1) a module construction area (also used to construct the yoke) and 2) an antenna construction area. These two areas are connected into a contiguous structure. Section 3.2.1.1.2 describes the base in greater detail.

Other Options

Other base concepts were considered but were discarded:

- 2 X 2 bay facility—This concept would entail the use of a facility that looks very much the same as shown only it would be 2 bays wide instead of 4. This would require the module to be indexed laterally through the facility as well as longitudinally. This concept would be a good candidate if it were not for the need to use 4 solar array deployment machines. If the solar array deployment rate could be doubled, then only 2 machines would be required and, hence, the 2 x 2 bay facility would be a good choice.

- 8 X 2 bay facility—This concept would entail the use of a facility that looks very much like the one shown except that it would be twice as long. It would, therefore, make the satellite module along the 8-bay width instead of the 4-bay width. A facility this large would be required if it were necessary to double the production rate or if it was found that solar array deployment could proceed only half as fast as predicted.

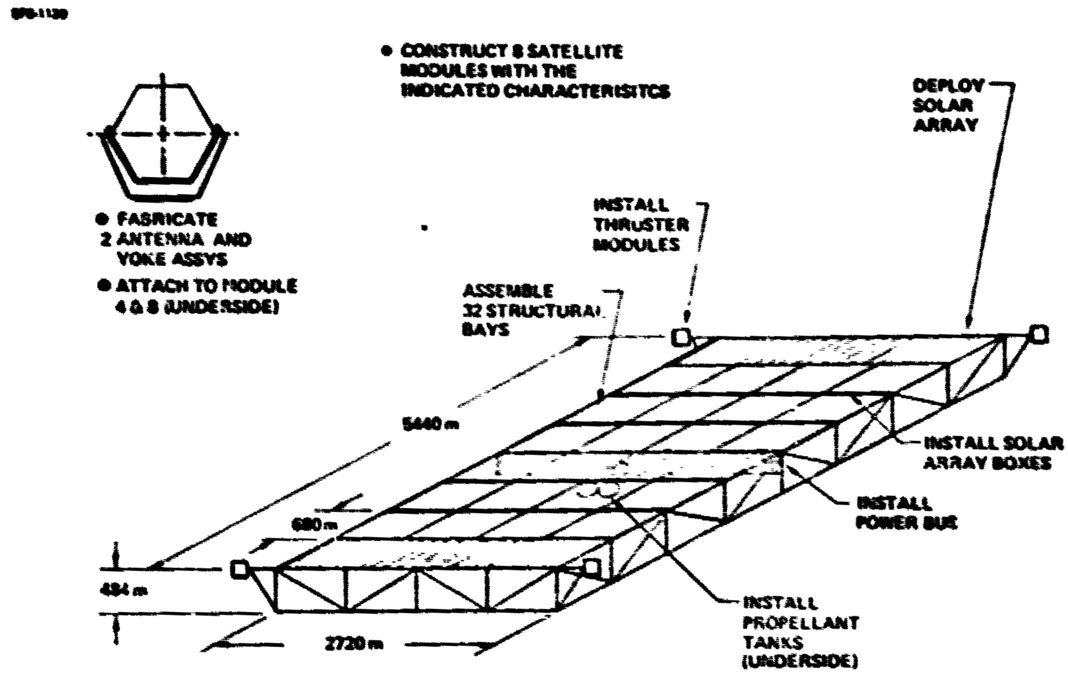


Figure 3.2-4 LEO Base Construction Tasks

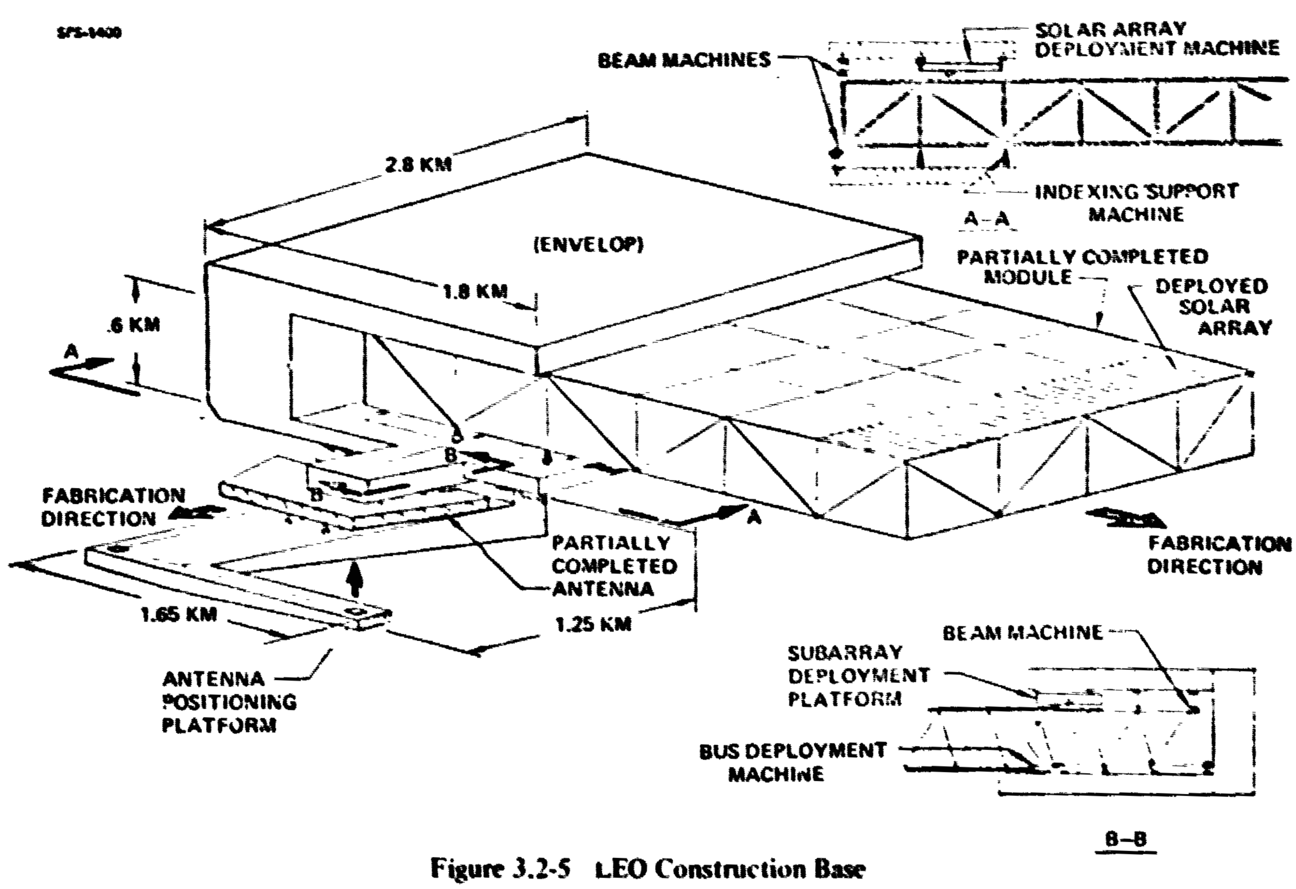


Figure 3.2-5 LEO Construction Base

3.2.1.1.1.3 Top-Level Construction Sequence and Timeline

The top-level construction sequence for the module is shown in Figure 3.2-6. The module construction timeline is shown in Figure 3.2-7. There are 40 days allotted to construct each module.

The top-level construction sequence for the antenna/yoke assembly is shown in Figure 3.2-8. The top-level construction sequence for the antenna is shown in Figure 3.2-9. Each of these major construction sequences is described in detail in the following sections.

3.2.1.1.1.4 Module Detailed Construction Analysis

3.2.1.1.1.4.1 Structural Assembly

The structural assembly concept to be described below is only one of several competing structural concepts. The rationale for the selection of the structural concept is addressed in Volume 3, Section 5.1.1.5. Data pertaining to the sizing, mass, and cost estimate of the selected structural concept is addressed in Volume 3, Section 6.1.1.1.2.2.

As the details of the construction of the selected structural concept are discussed below, some of the alternative concepts will be addressed.

The structural concept to be discussed in this section is described by the following nomenclature:

Beam Type	Pentahedral or Pyramid Beam
Strut Type	Tapered, Nested Struts
Strut-to-Strut Joint Type	Stud-in-socket, lateral assembly
Frame Type	Segmented Frame
Beam-to-Beam Joint Type	Lap Joint
Beam Machine Type	Movable, Articulated 20 meter Beam Machine

The beam configuration is shown in Figure 3.2-10. The beam shown is referred to as a 20m beam. All frame, equipment, and facility dimensions and all timeline analyses are based on this 20m beam.

Strut Assembly

The tapered half-struts will be manufactured on Earth. There will be 3 different strut half lengths (10m, 11.2m, and 14.15m). These half struts are nested to form 100 piece units, with several of these units then put into a dispensing magazine as shown in Figure 3.2-11. (The number of units to a magazine would be set so that the magazine has to be changed out of the beam machine only once a day at most.) These magazines would be delivered to the LEO construction base where they would then be delivered to the beam machine and inserted into strut assembly machines that are part of the beam machine.

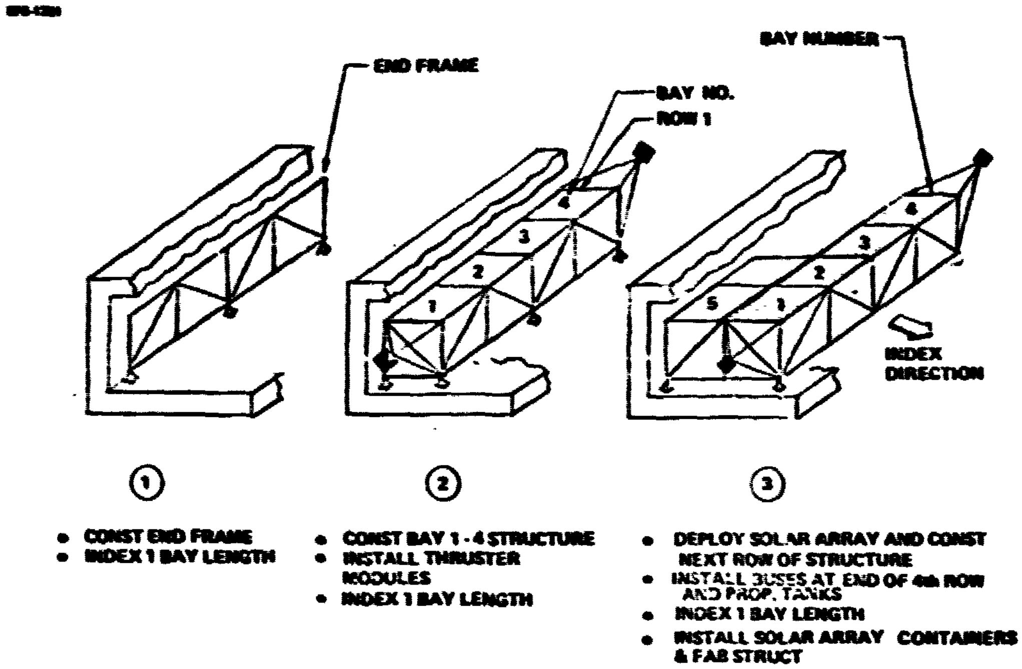


Figure 3.2-6 Module Construction Sequence—Photovoltaic Satellite

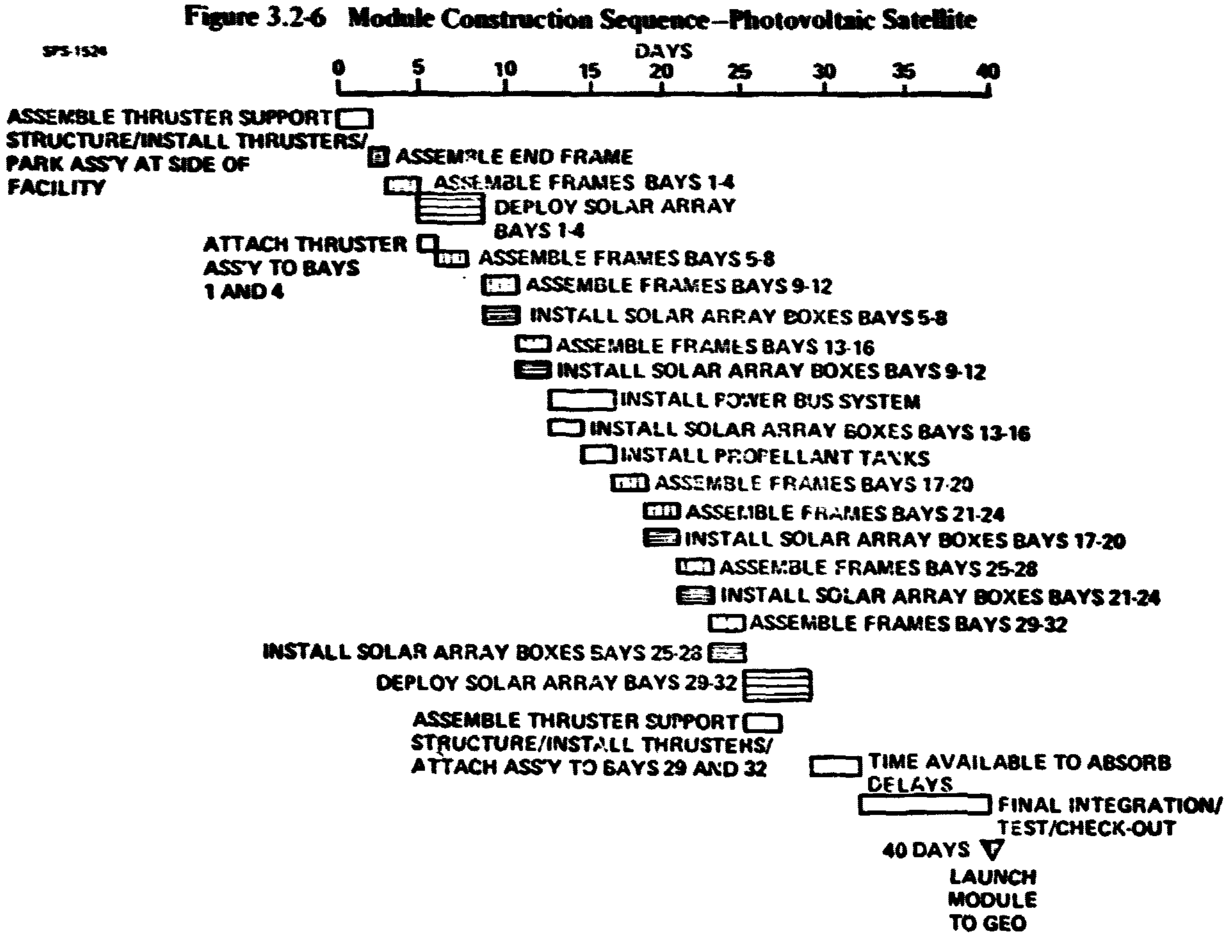


Figure 3.2-7 Module Construction Timeline

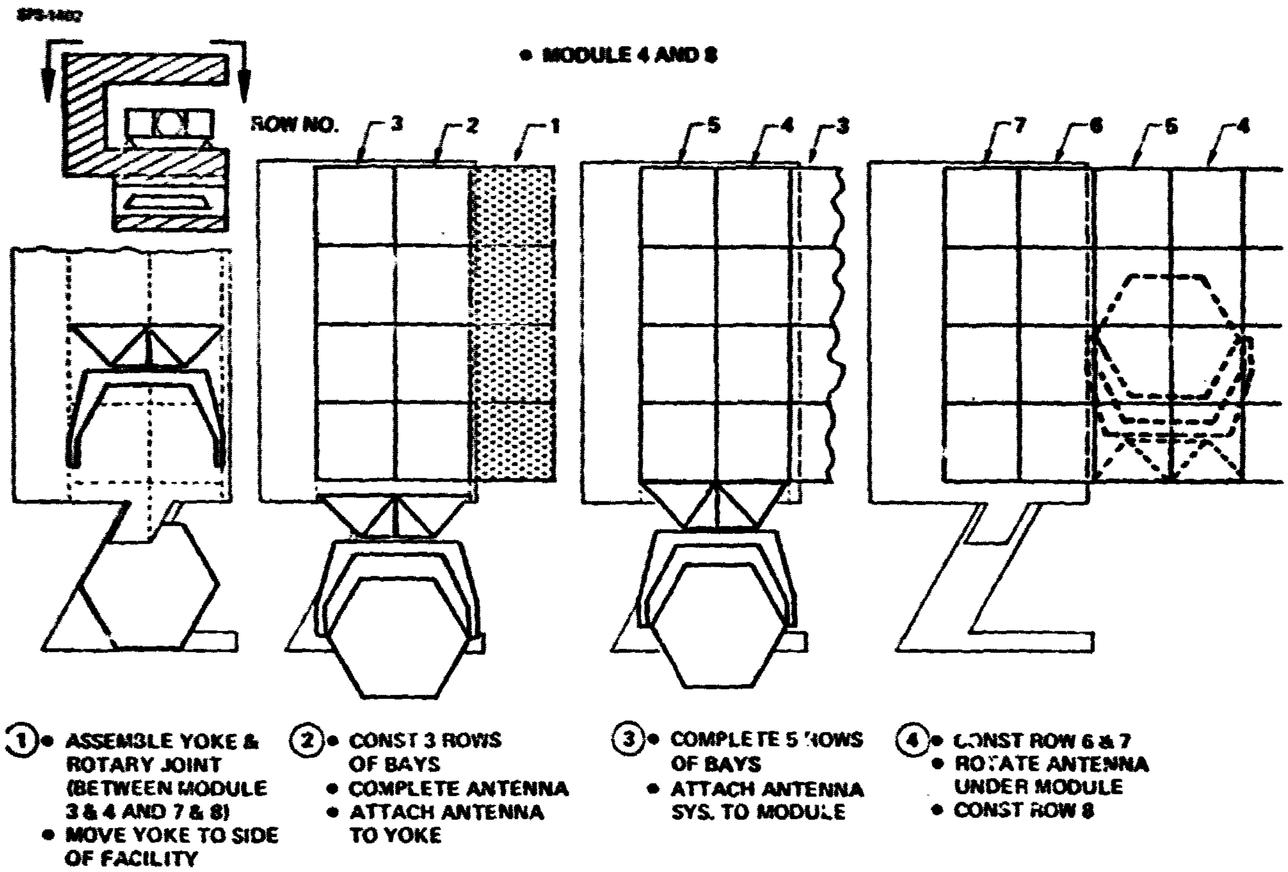


Figure 3.2-8 Antenna/Yoke/Module Assembly
Photovoltaic Satellite

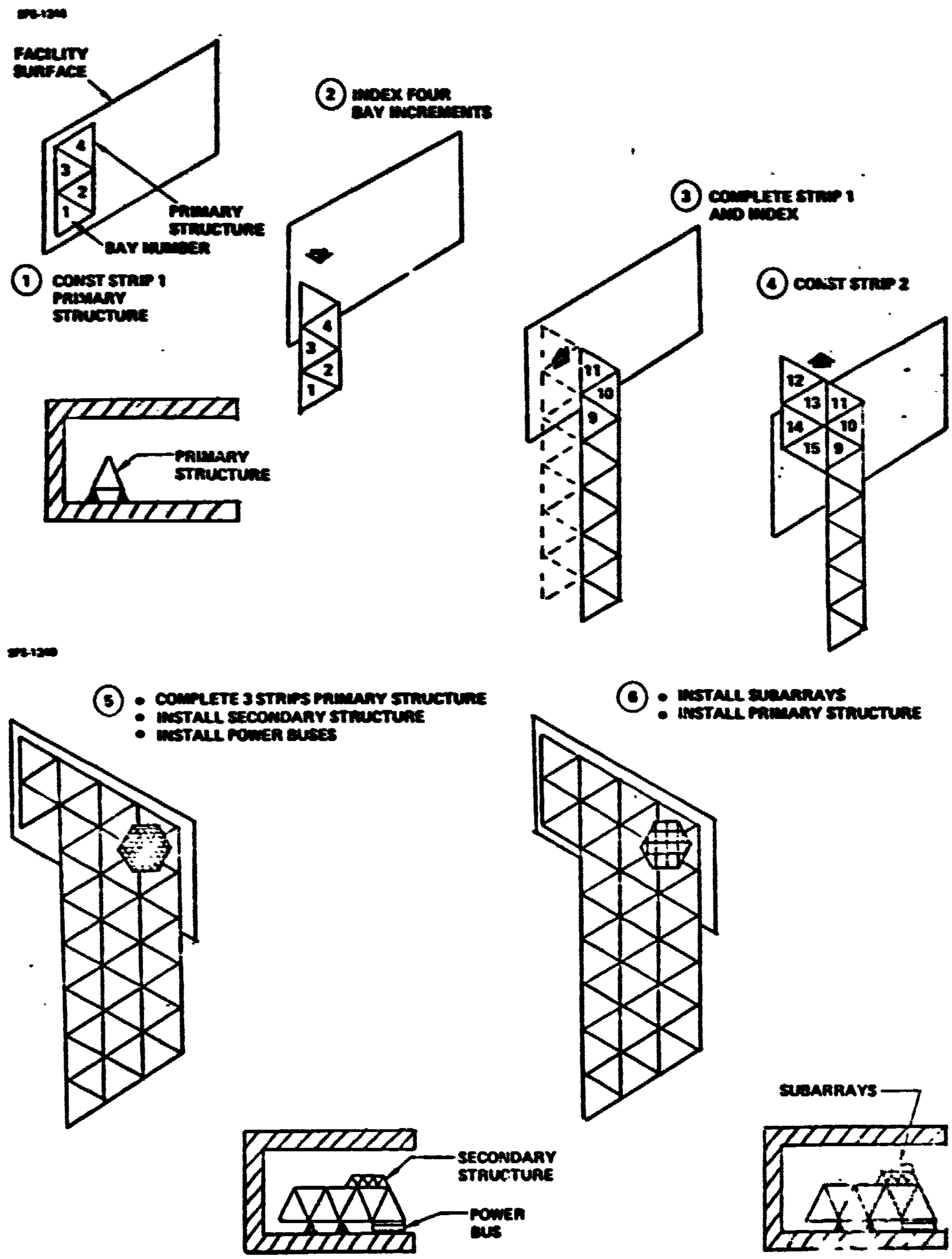


Figure 3.2-9 Antenna Construction Sequence

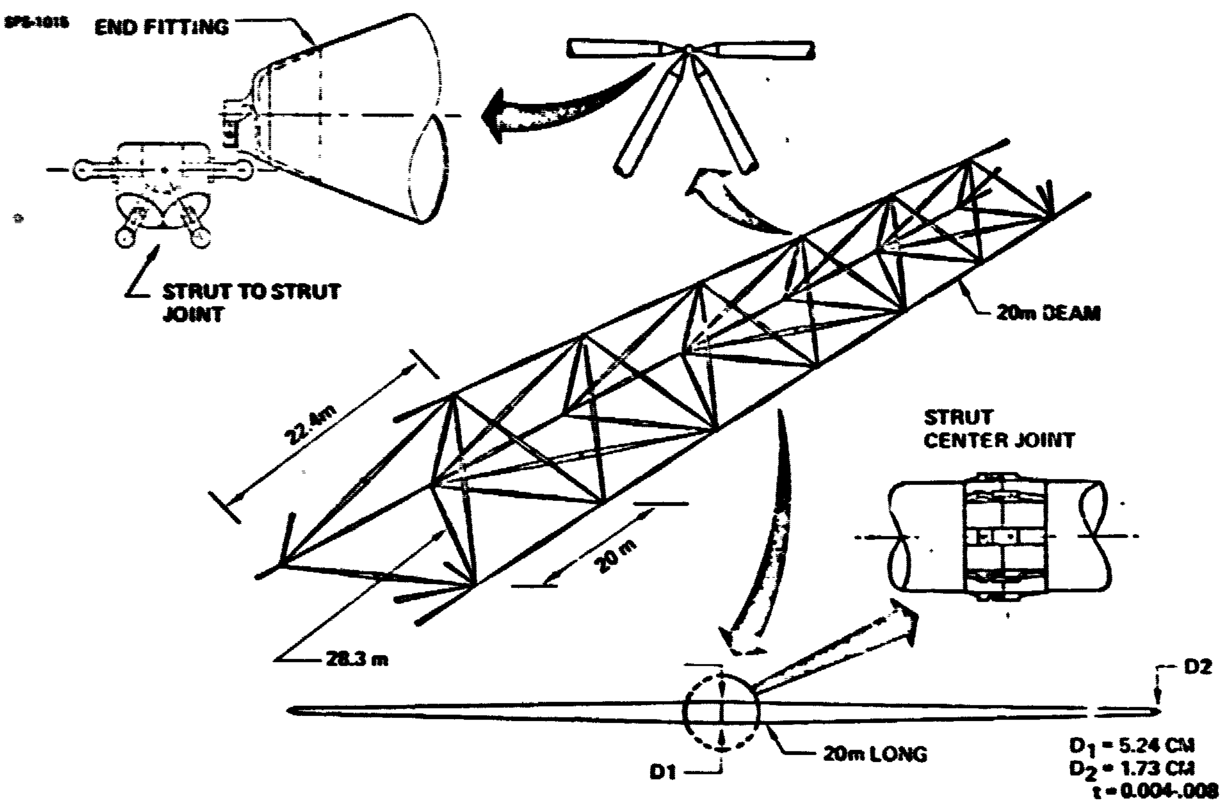


Figure 3.2-10 Primary Beam

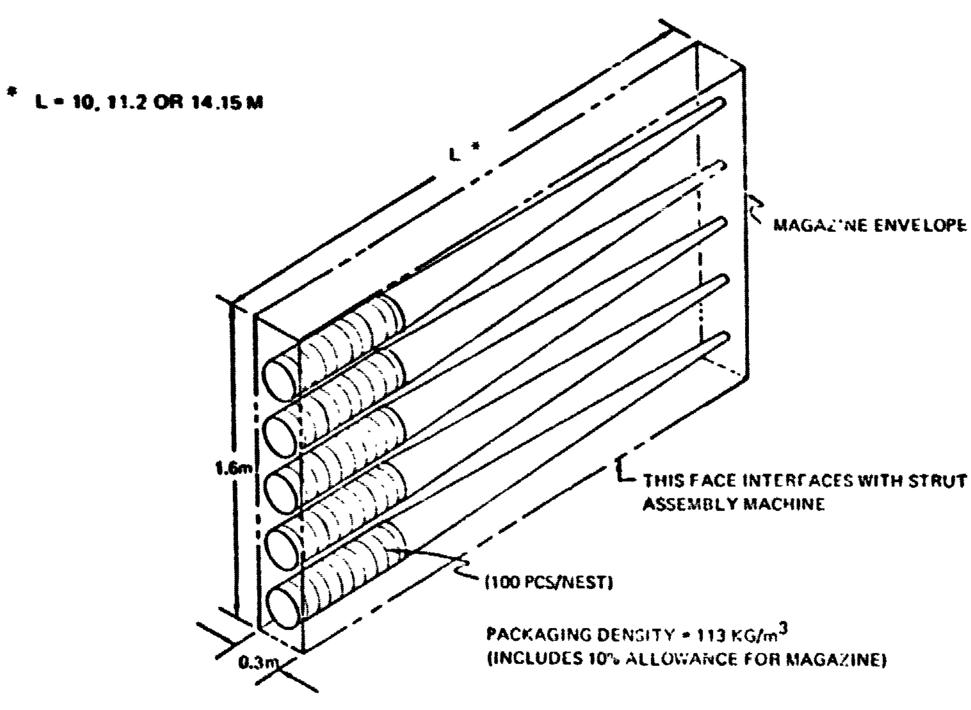


Figure 3.2-11 Strut Magazine Concept

The strut assembly machine performs the operations depicted in Figure 3.2-12. Figure 3.2-13 shows a concept of a strut assembly machine used to form 20m long struts. The other length struts would be assembled by similar machines.

Strut-to-Strut Joints

There are three variations of the strut-to-strut joint fitting previously shown in Figure 3.2-10. Each type would be manufactured on Earth and then packaged into a dispensing carrousel such as is shown in Figure 3.2-14. The number of joint fittings to be installed in a carrousel would be sufficient to provide a one-day supply. These joint fitting carrousels are delivered to LEO ready to be inserted into the beam machine.

Beam Assembly

To assemble the 20m pentahedral beam, it will be necessary to use 9 strut assembly machines (5 of the 20m size, 4 of the 22.4m size, and 1 of the 28.3m size) configured as shown in Figure 3.2-15. Three joint installation mechanisms such as is shown in Figure 3.2-16 will be required. These mechanisms will extract fittings from the carrousel and attach them to indexing carriages.

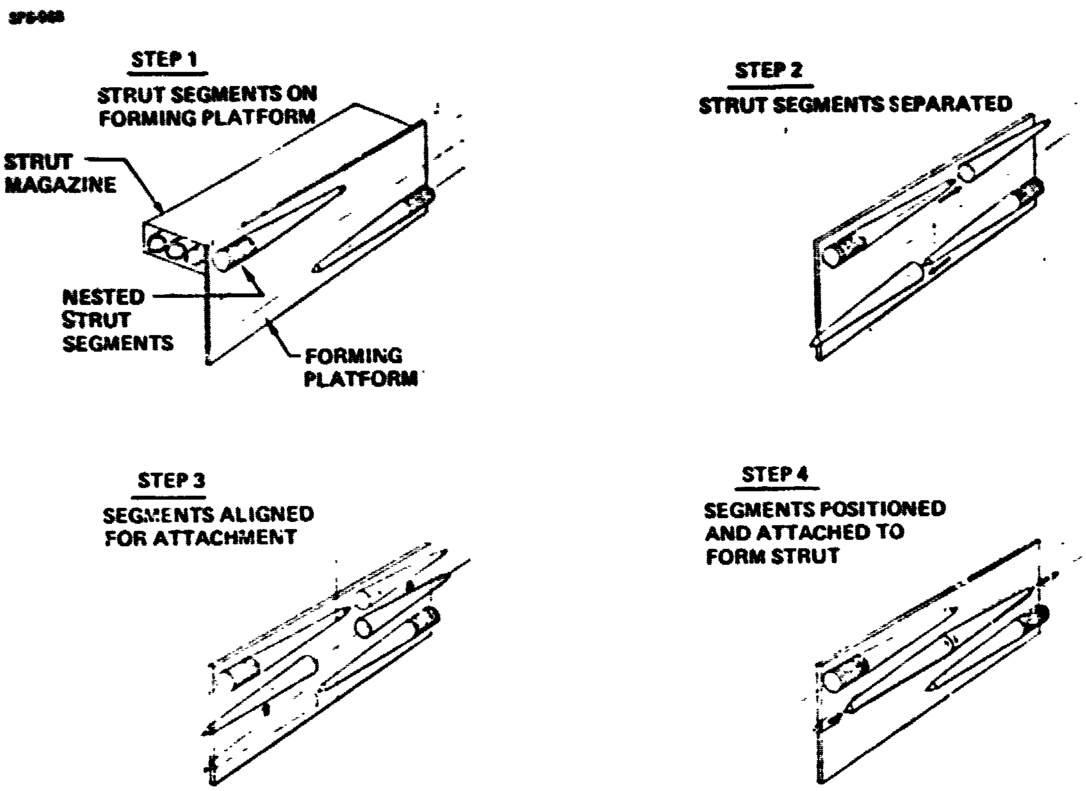
The beam assembly operations are shown in Figures 3.2-17 and 3.2-18. Figure 3.2-19 shows a timeline for this assembly operation. The 5.3m/minute rate is considered to be conservative. Other timeline estimates have ranged as high as 15m/minute. Figure 3.2-20 shows that a single beam machine would easily be capable of making all of the necessary beams within the one year construction time at the 5.3m/minute rate. However, for reasons discussed below, two beam machines will be required for operational use.

Beam Machine

The beam assembly equipment described above can be configured into a support frame as shown in Figures 3.2-21 and 3.2-22. It has been determined that it is necessary to have 2 beam machines. One of these will operate on the lower surface of the facility while the other will operate from the roof of the facility. These two beam machines will be capable of making all of the necessary beams by giving the machine translation and rotation capabilities. Figure 3.2-23 shows the beam machine with the necessary functional capabilities. Two operators/shift are assigned to each beam machine.

Frame Assembly

The frame configuration chosen as the reference structure is referred to as a lap joint/segmented configuration (as opposed to centroid joint/segmented or lap joint/continuous configuration). This frame configuration requires that all of the beams be made in one-bay long pieces. Figure 3.2-24 shows this frame assembly concept. Figure 3.2-25 shows details of how beams are joined.



NOTE: ASSEMBLY EQUIPMENT NOT SHOWN

Figure 3.2-12 Strut Assembly Sequence

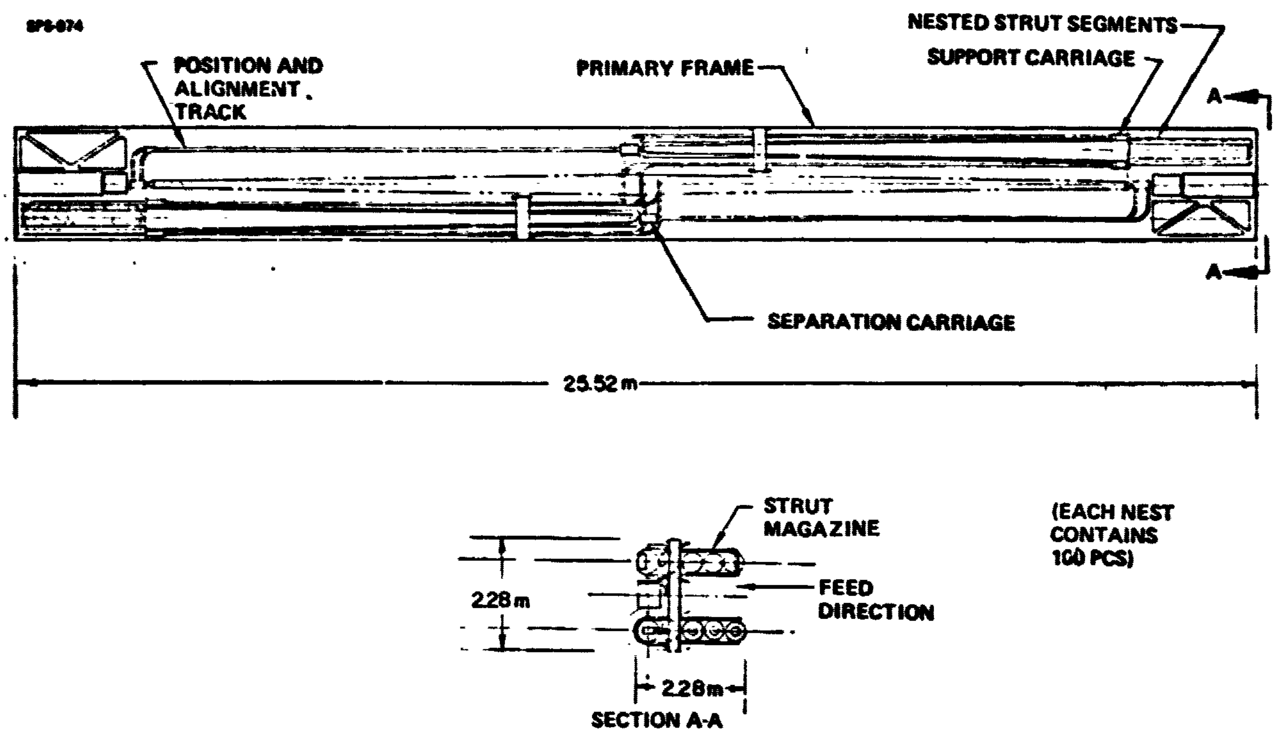


Figure 3.2-13 Strut Assembly Machine

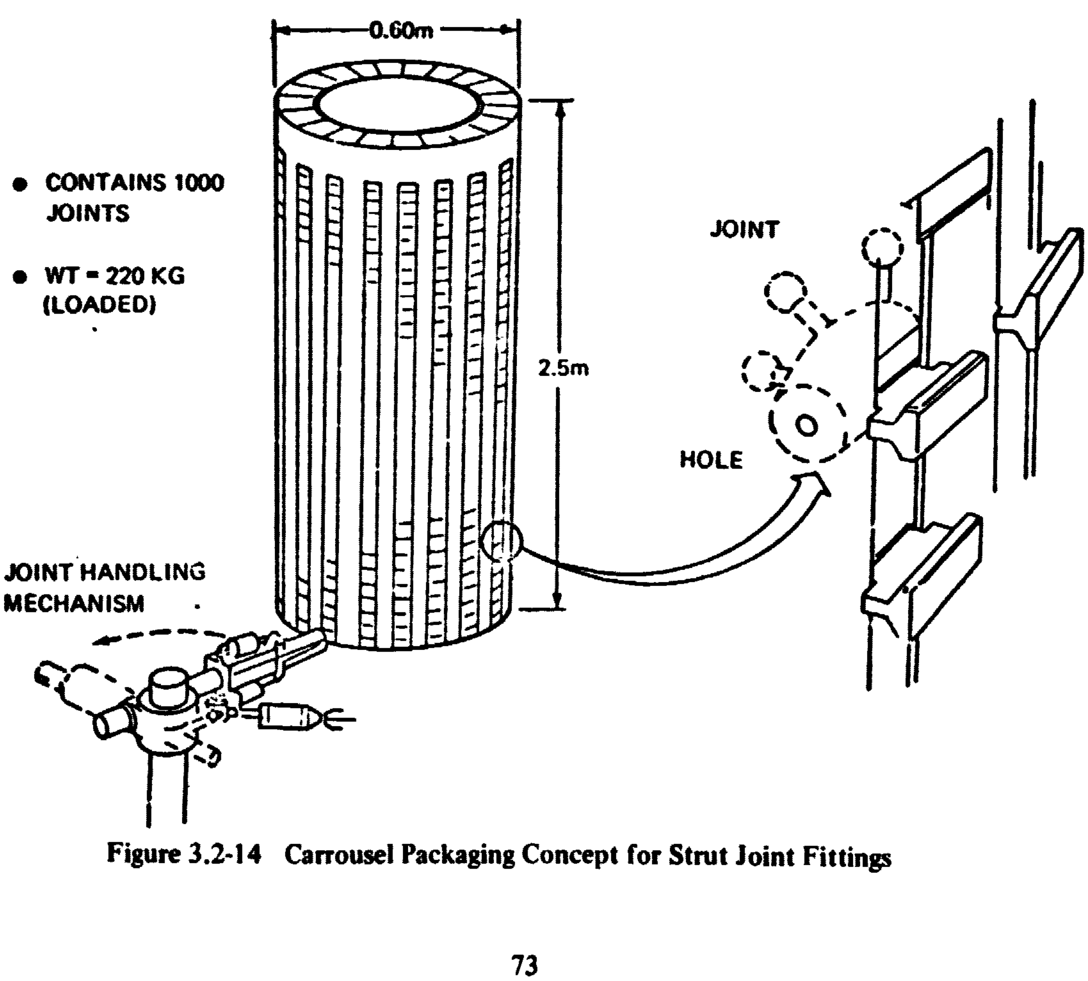


Figure 3.2-14 Carrousel Packaging Concept for Strut Joint Fittings

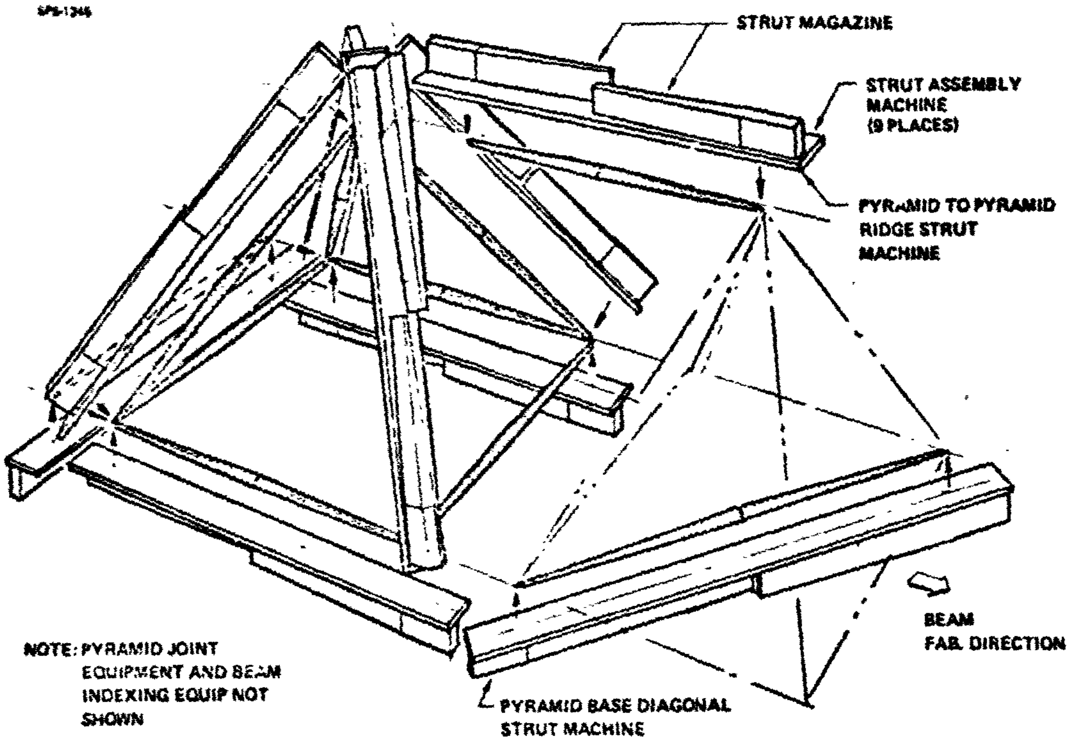


Figure 3.2-15 Beam Assembly Concept

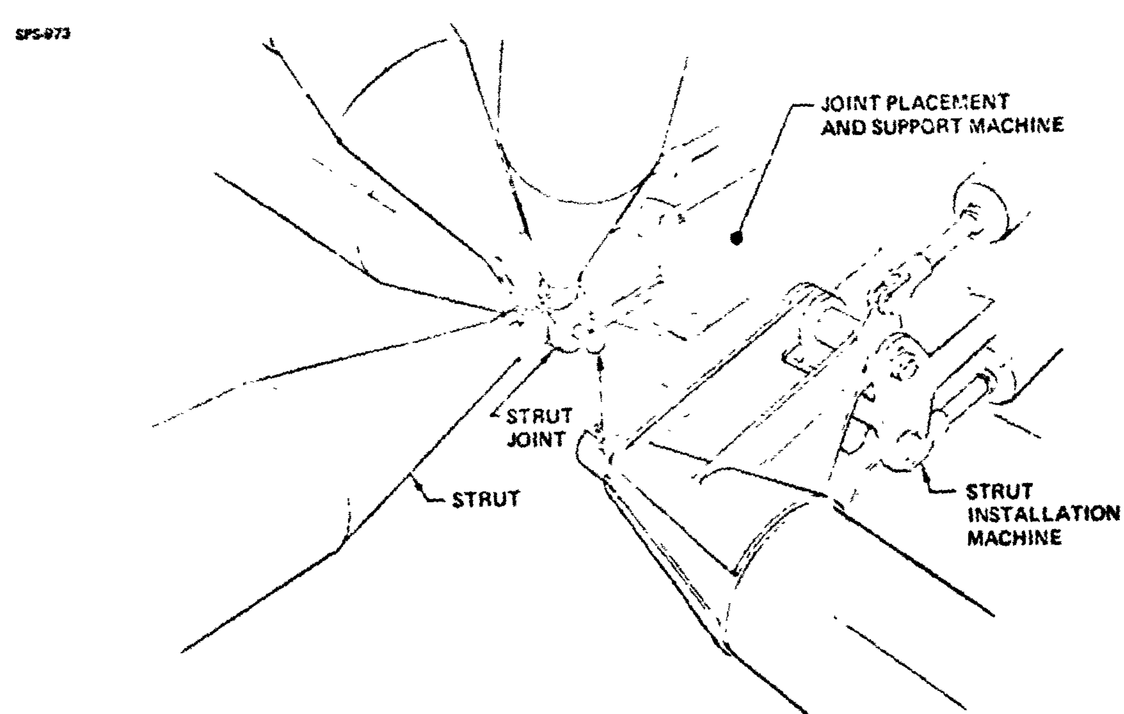


Figure 3.2-16 Strut/Joint Assembly

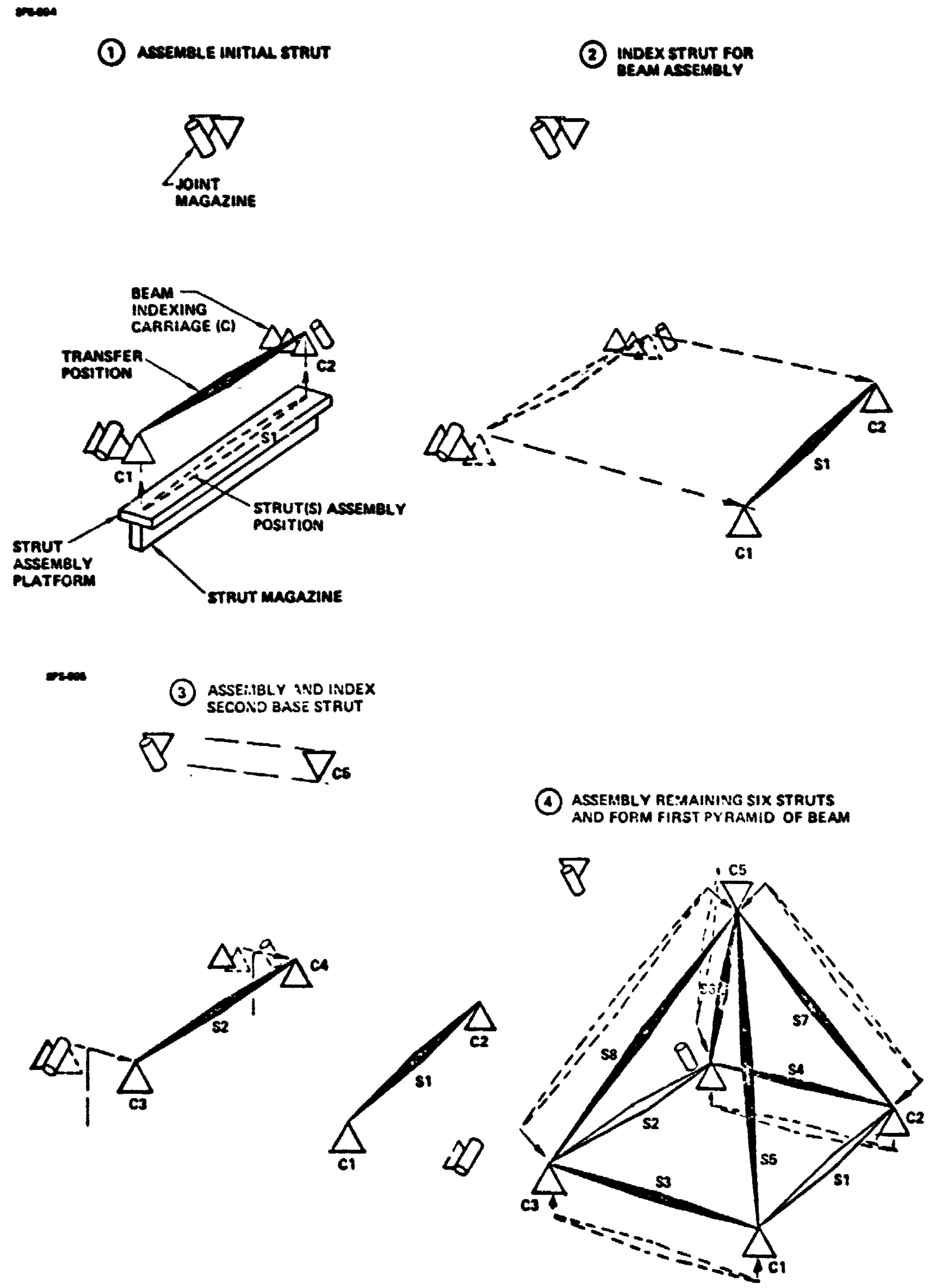


Figure 3.2-17 Beam Assembly Operations

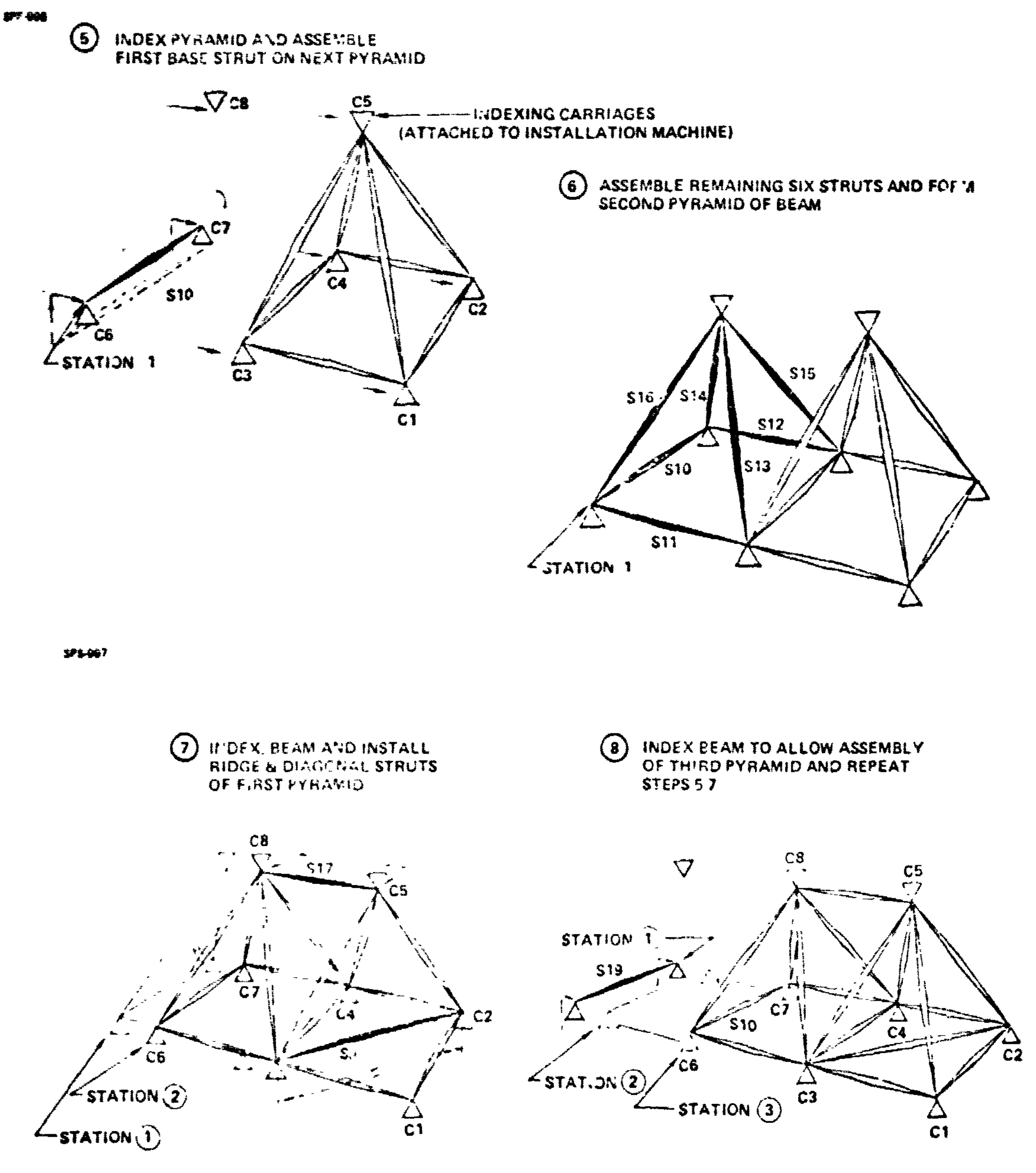


Figure 3.2-18 Beam Assembly Operations (Continued)

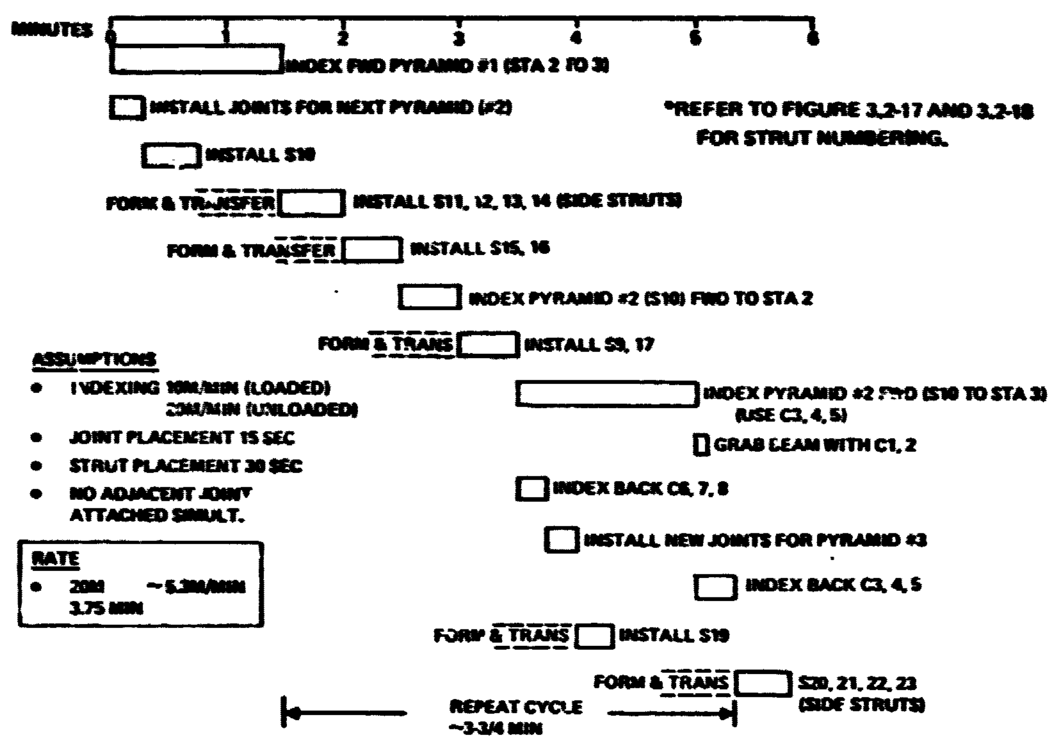


Figure 3.2-19 Beam Assembly Timeline

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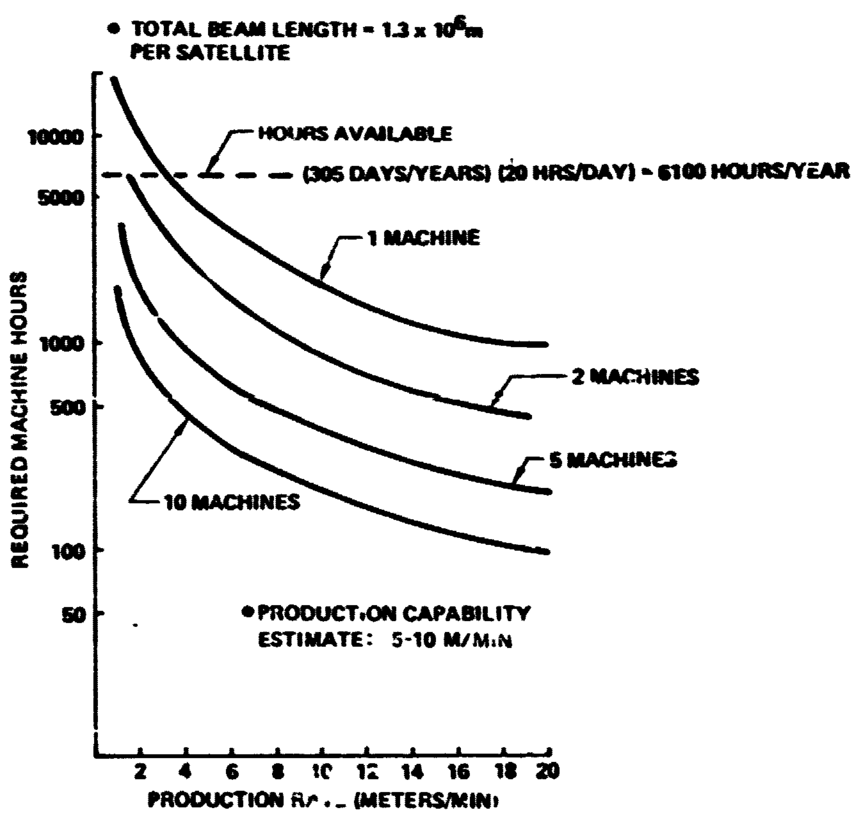

Figure 3.2-20 Beam Production Requirements

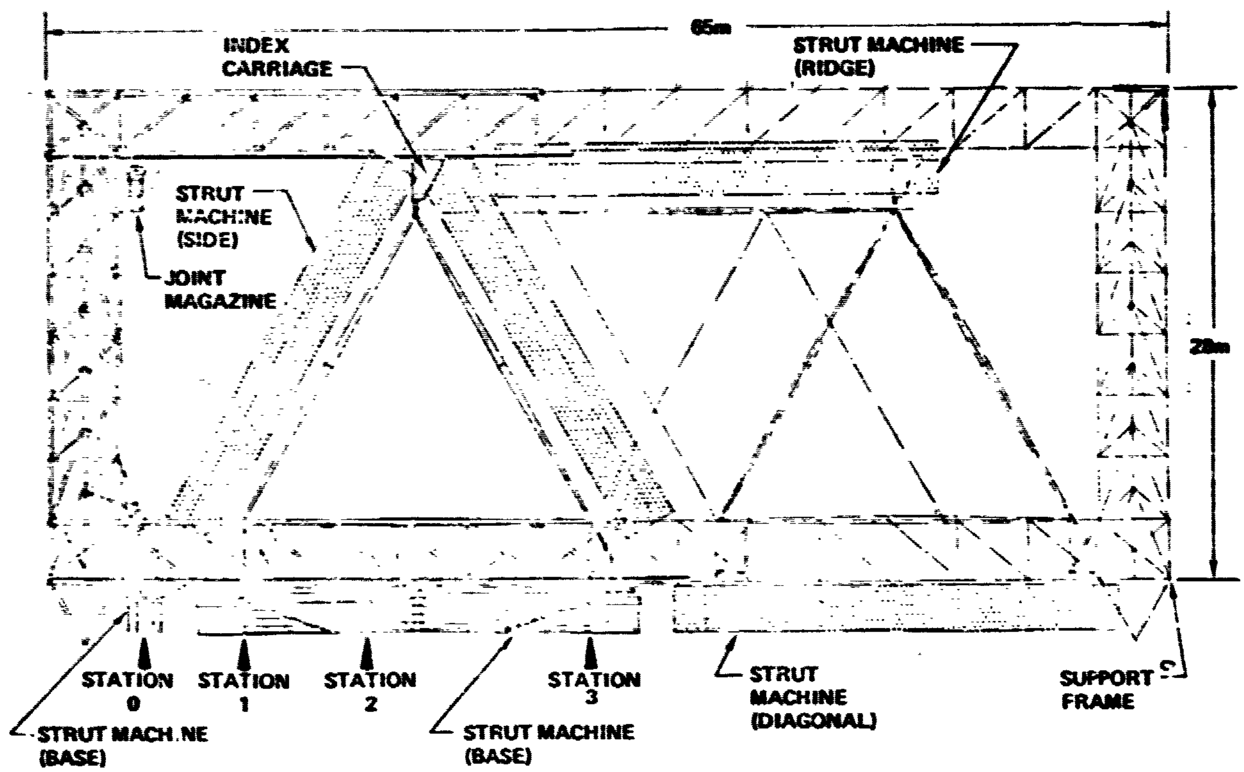

Figure 3.2-21 Beam Assembly Machine Side View

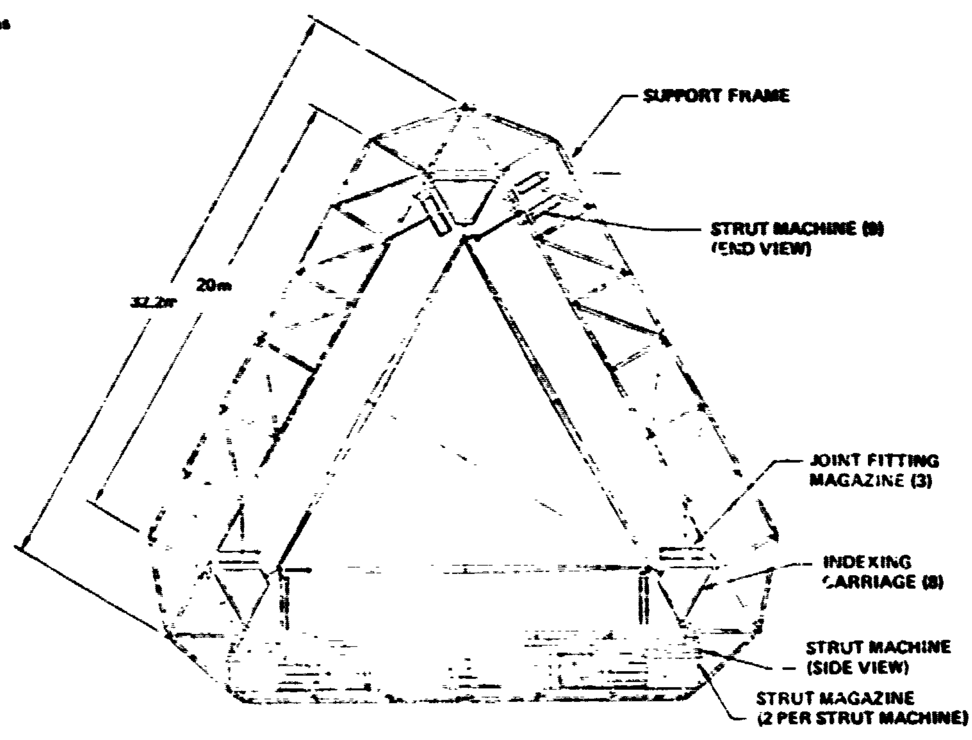

Figure 3.2-22 Beam Assembly Machine End View

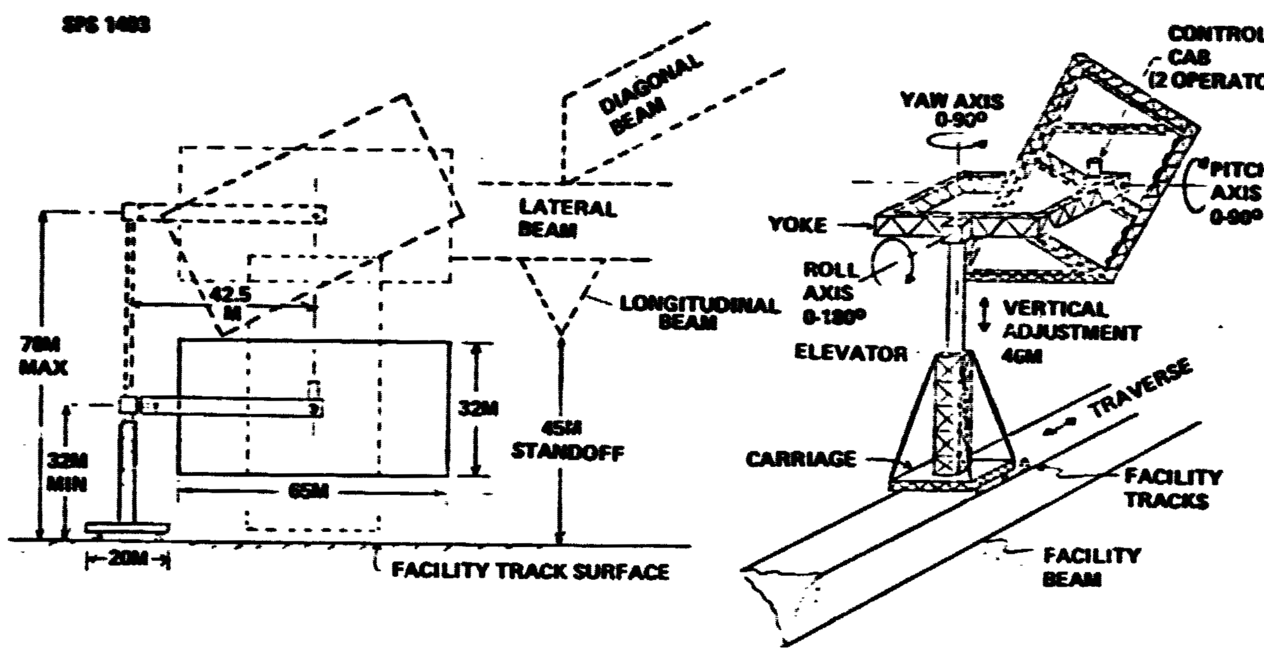

Figure 3.2-23 Articulating Beam Machine

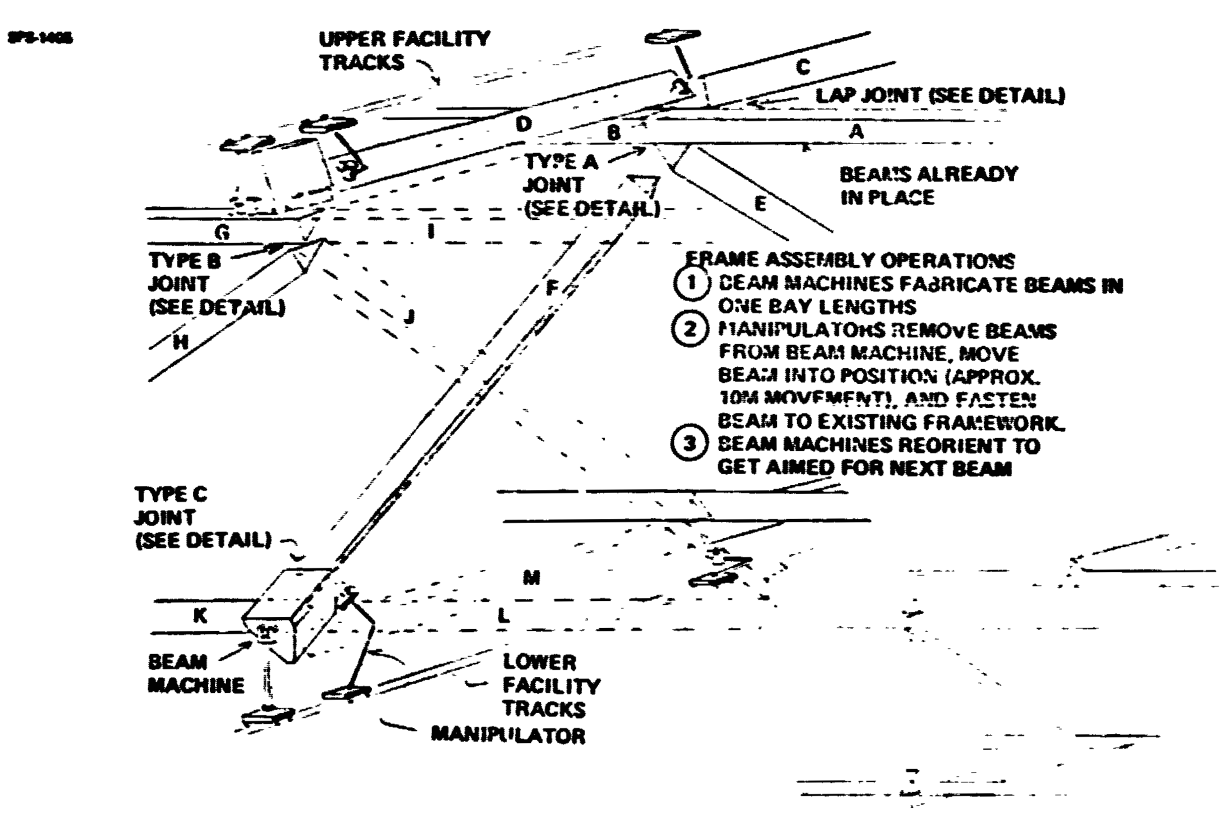

Figure 3.2-24 Lap Joint Frame Assembly (Using One Bay Long Beams)

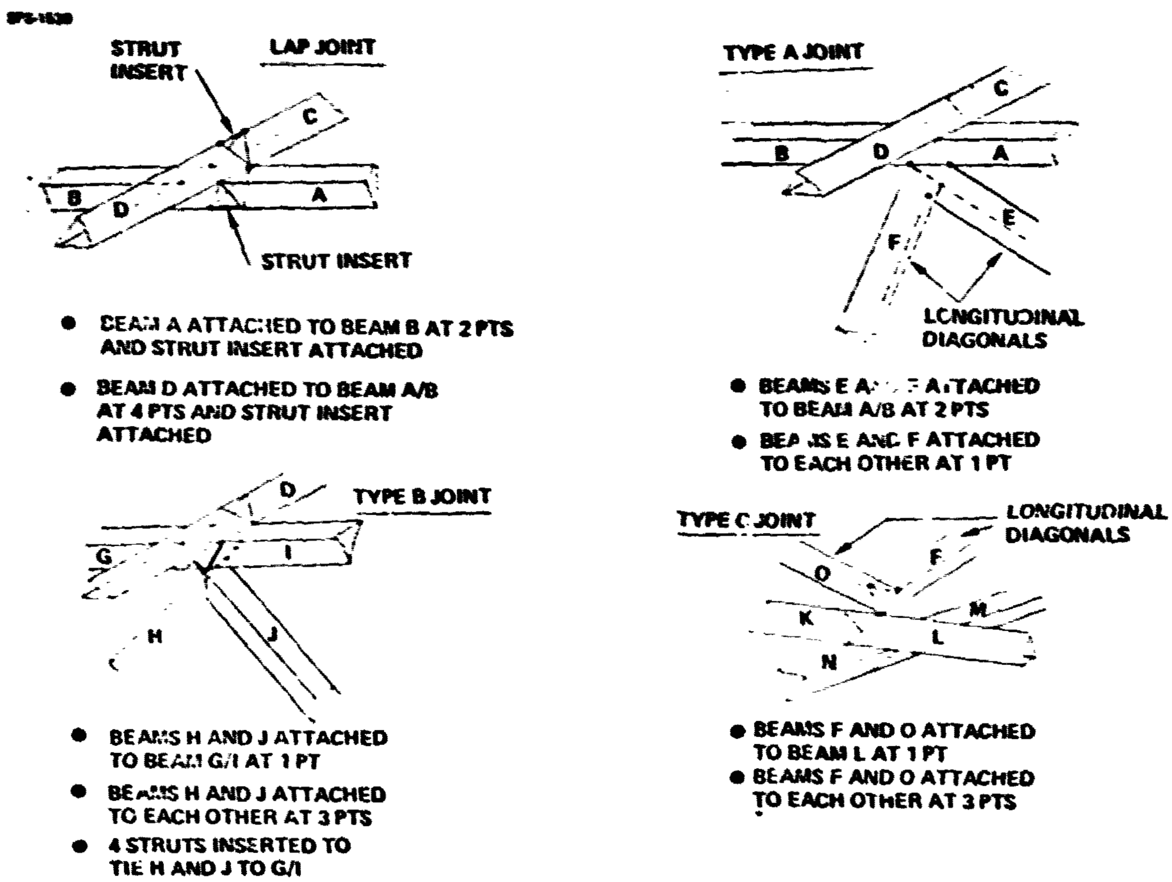

Figure 3.2-25 Frame Joint Assembly Details

Crane/Manipulator

The assembly of the frame requires the use of crane/manipulators with characteristics as shown in Figure 3.2-26. The item shown is hereafter referred to as a 110m crane/manipulator. It has been determined that there are also requirements for a 20m and a 250m version of this machine. Two operators/shift are assigned to the 110m and 250m machines and 1 operator to the 20m crane/ manipulator.

Other Options

As was mentioned in the introduction of this section, other structural configurations have been considered. One option is a concept that would employ a 20m continuous chord, thermally formed beam. One application for this structure would be in a so-called continuous beam frame which is defined as one which literally has the beams running the entire length and width of the satellite or module.

Figure 3.2-27 shows a comparison of the continuous beam frame concept to the segmented beam concept previously described.

Figure 3.2-28 shows a comparison of the beam types and beam joints that pertain to these two frame options.

Figure 3.2-29 shows a comparison of how many beam machines and manipulators are required to implement the two frame configurations.

Figure 3.2-30 shows a comparison of the size of facility required to construct the two frame configurations.

From these comparisons it is shown that a continuous beam frame requires more beam machines (recall that a single beam machine could make all of the necessary beam within one year's time), more manipulators, twice as many solar array deployers and a facility twice the size. Based on the comparison, the segmented beam frame concept is preferred.

A continuous chord beam is still a likely candidate (even though it weighs more than the pentahedral beam) in an application where it is used in one-bay lengths to make the segmented beam frame.

If weight savings was of paramount importance, a frame employing a centroid joint segmented beam is a strong contender. This frame could be made from either continuous chord or pentahedral beams. It would require that "joint plug" subassemblies to be made that would match the ends of beams to make a centroid joint. This requires more crewmembers and equipment and also presents a more challenging frame assembly operation. The centroid joint frame is used to make the antenna primary structure.

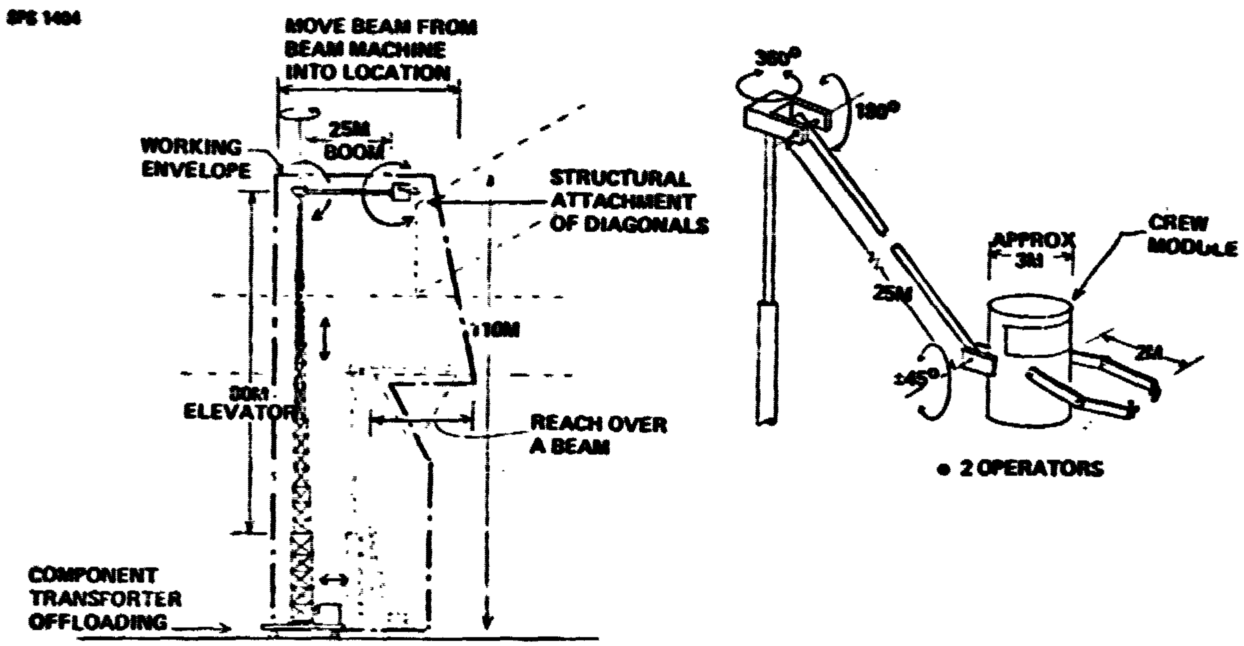

Figure 3.2-26 110M Crane/Manipulator

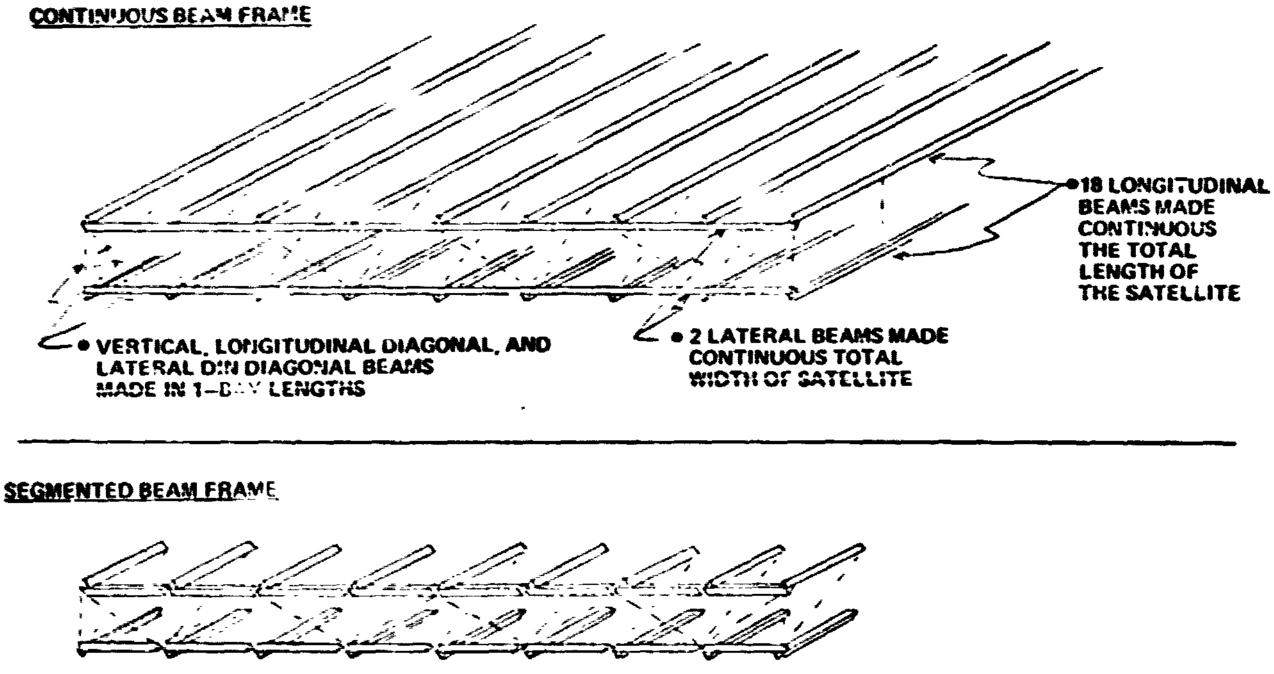

Figure 3.2-27 Structural Frame Concepts

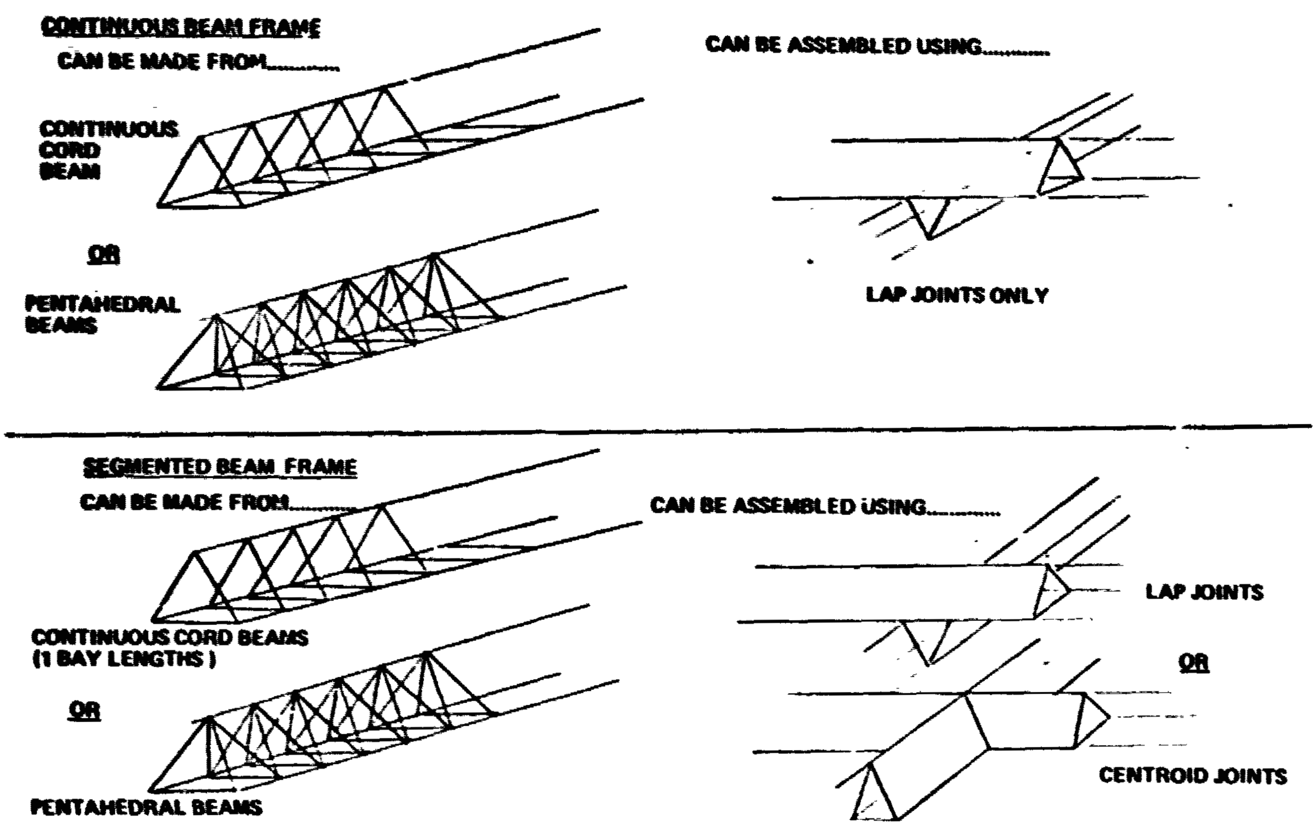

Figure 3.2-28 Beam and Beam Joint Concepts

CONTINUOUS BEAM FRAME

DEDICATED, NON-MC ..
BEAM MACHINES

- REQUIRES
  - 48 BEAM MACHINES
  - 36 MANIPULATORS
  - 8 SOLAR ARRAY DEPLOYERS
- BEAMS CANNOT INTERSECT AT A COMMON JUNCTION
- THE LONGITUDINAL BEAM MUST BE SYNCRONIZED

MOVING BEAM MACHINES

- REQUIRES
  - 38 BEAM MACHINES (20 FIXED, 18MOVING)
  - 36 MANIPULATORS
  - 8 SOLAR ARRAY DEPLOYERS

SEGMENTED BEAM FRAME

- REQUIRES
  - 2BEAM MACHINES
  - 8 MANIPULATORS
  - 4 SOLAR ARRAY DEPLOYERS

Figure 3.2-29 Frame Concept Comparison

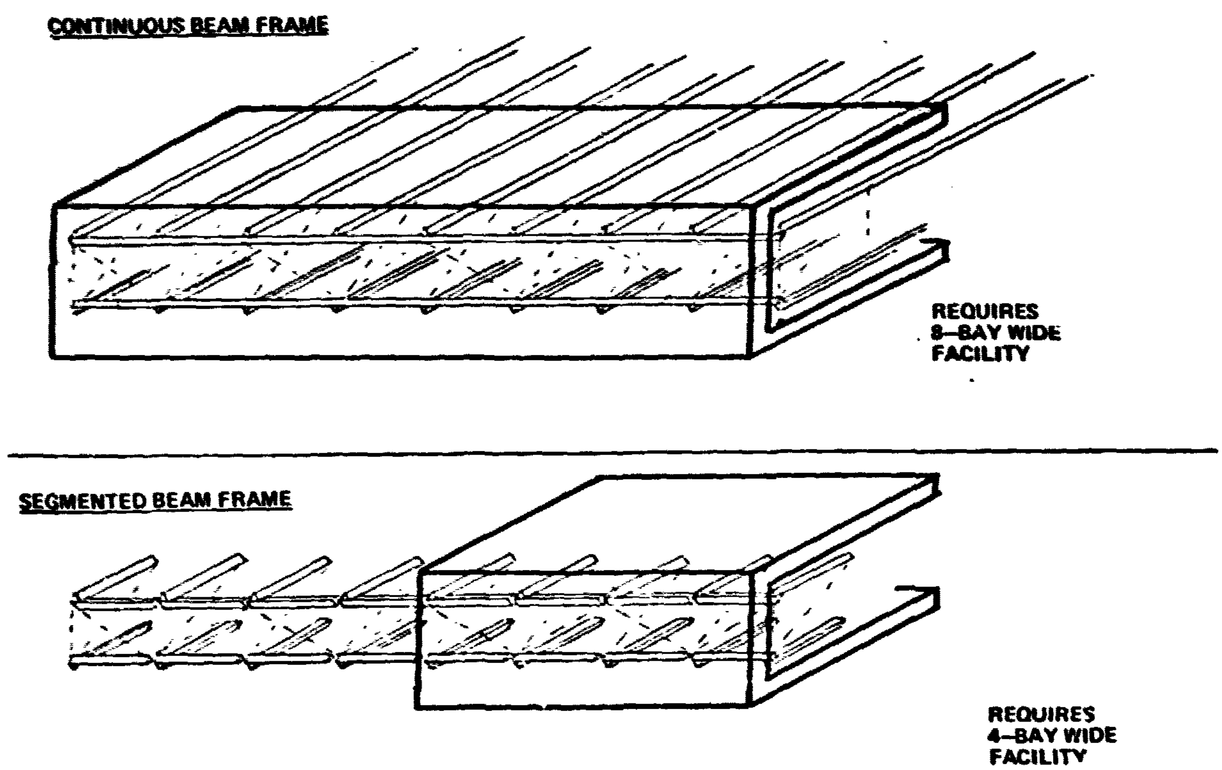

Figure 3.2-30 Frame Facility Concepts

3.2.1.1.1.4.2 Solar Array Deployment

(Note: the baseline structure has 20m hardpoint spacings but with another iteration will be changed to match the 15m solar array width which is the largest possible with the payload shroud assuming the arrays must be launched with the long axis of the package perpendicular to the launch axis. The solar array is delivered to the LEO base as 15m wide by 650m long blanket strips. The strips are accordion-folded in the 650m direction to form blanket packages. Forty-three of these blanket strips are to be installed within each structural bay as shown in Figure 3.2-31. For the LEO construction approach, only a portion of the array will be deployed at LEO. The exact amount depends on whether the module is self-powered to GEO alone or with an antenna. In the remaining bays, the solar array blanket packages will be attached to the structure at one end of each bay nearest the power bus. These blankets will be deployed later at GEO.

The machine shown in Figures 3.2-32 and 3.2-33 will be used to: 1) install the solar array packages. 2) deploy the blankets across the bay, 3) attach the adjacent edges of blankets and, 4) to attach the blankets via catenary spring assemblies to the structure.

Four of these machines will be required--one in each of the four solar array deployment bays of the facility. These machines each have 8 days to deploy the 43 blankets within their bay. A timeline analysis has shown that the deployment assembly has to move at a rate of 7.8 meters per minute.

Two operators per shift are allocated per solar array deployment machine. One of them controls the blanket package installation and gantry indexing operations. The other operator controls the deployer.

Other Options

Other solar array deployment concepts were considered but were discarded:

- Full bay width catenary In this concept, the blankets would be attached to a single corner-to-corner catenary. The construction problems associated with this were too intimidating to seriously consider.

- Cable Winch Deployment In this concept, the blankets would be extracted simultaneously from their packages by a winch and cable system. This was discarded as the deployment of the cables and winches would be at least as complicated as deploying the blankets.

- Folded and Rolled Blankets In this concept, the full 650m width of solar array would be accordion folded in one dimension and then the accordion folded package would be rolled into a single package. This would avoid the blanket-to-blanket edge attachment problem. However, it was found that there was no feasible way to roll the accordion folded package without resulting in considerable stress placed on the individual solar cells and most likely result in popping off the cells.

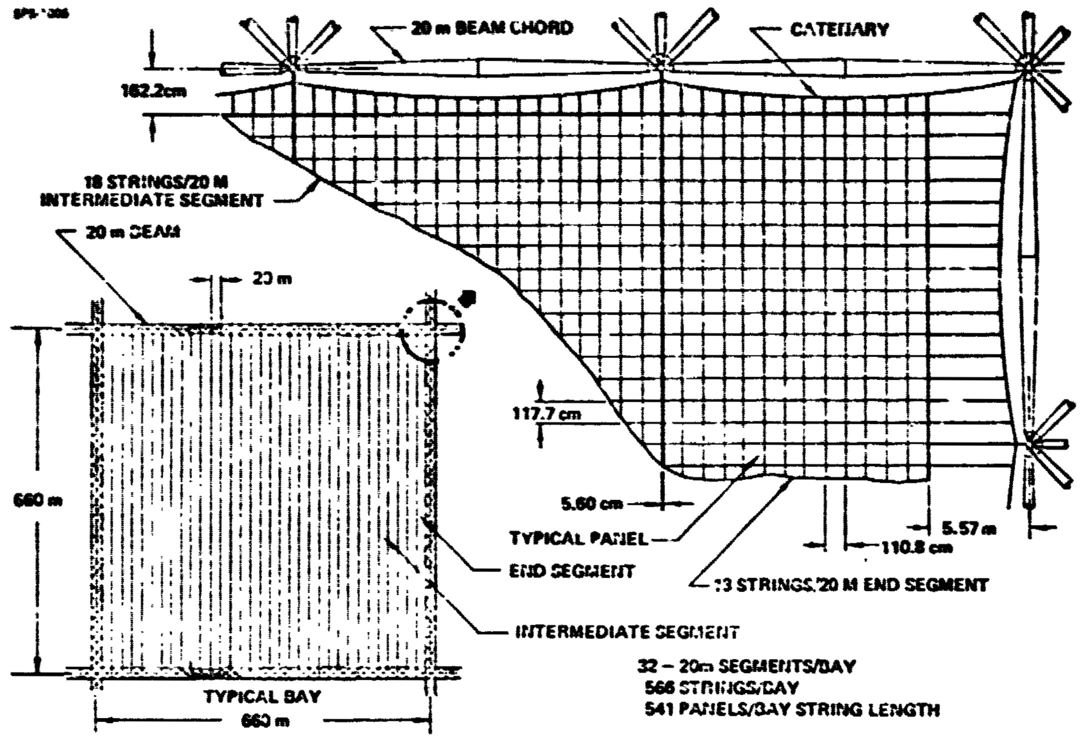

Figure 3.2-31 Solar Array Arrangement and Attachment

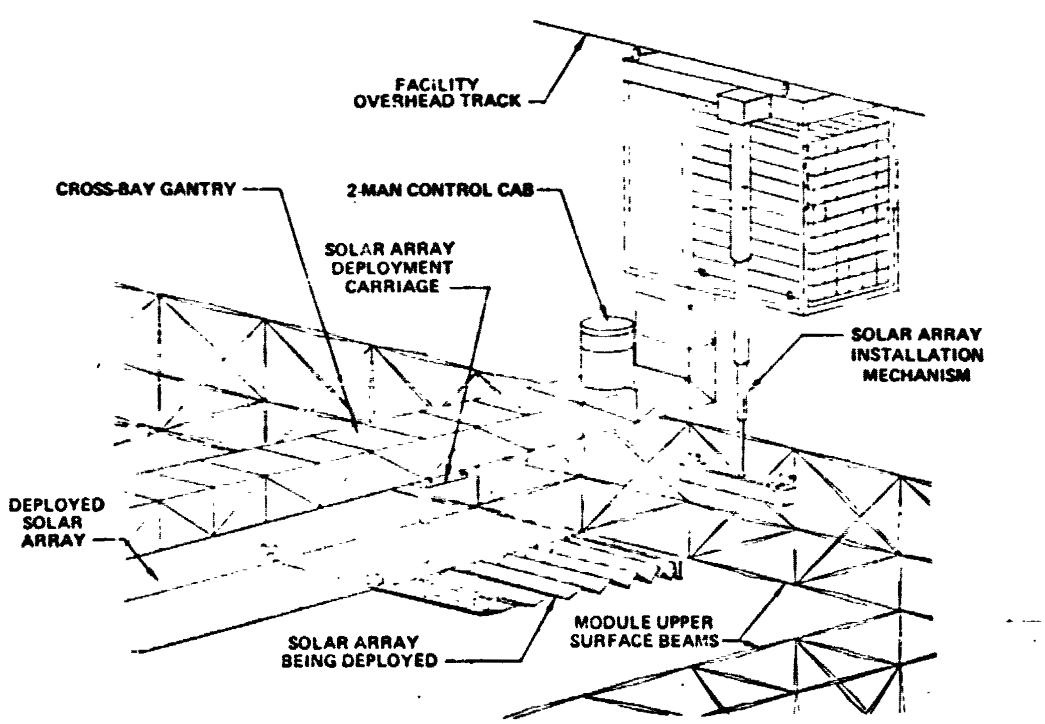

Figure 3.2-32 Solar Array Deployment Concept

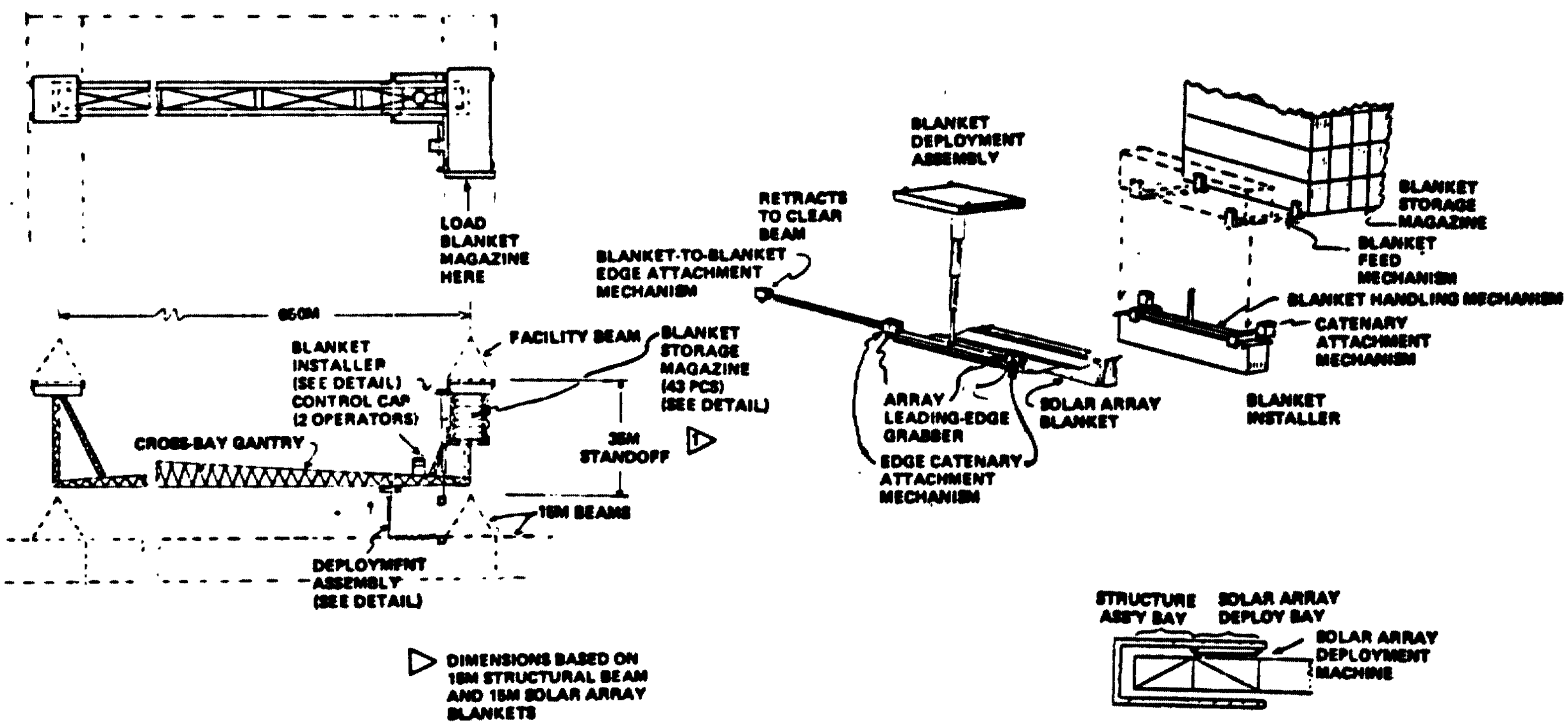

Figure 3.2-33 Solar Array Blanket Installer

3.2.1.1.1.4.3 Annealing System Installation

A solar array annealing system will be installed at LEO if such a system is required. This system would be used at GEO to anneal the solar array that was deployed for self-powered LEO-to-GEO transport. This system would also be used on the operational satellite to periodically anneal the solar array. This system has yet to be defined; therefore no installation concept has been developed.

3.2.1.1.1.4.4 Power Bus System Installation

The power bus system will be installed along the top face of the fourth row of structural bays. Figure 3.2-34 shows the configuration of the bus system and the machine used to deploy the busses.

The switch gear subassemblies and the bus support subassemblies will be made in the subassembly area at the central warehouse (see Section 3.2.1.1.1.7) and will be delivered to the installation site ready to install.

The bus support cable assembly will be installed by the use of two 250m boom manipulator/cranes. The switch gear assembly will be installed by one of the 110m manipulator/cranes. The busses will be installed by the bus assembly machine shown in the figure.

Similar versions of this bus deployment machine are used to deploy the busses on the yoke and antenna.

As the structural assembly operations are interrupted while the bus system is installed, it is reasonable to assume that two of the frame assembly crane/manipulator operators would be available to operate the bus deployer. The manipulator/cranes used to install the bus system would be operated by the other crane operators from the frame construction crew.

3.2.1.1.1.4.5 Thruster System Installation

The thruster system is composed of four thruster modules mounted on support structures at the four corners of the module, propellant tanks located beneath the center of the module, and the plumbing and control subsystems interconnecting the system.

The thruster modules are assembled in the subassembly factory area (see Section 3.2.1.1.1.7). Four of these subassemblies are constructed per module. These units are delivered to the structural assembly bay ready to be installed on the support structure.

The support structures are assembled from 20m beams. The structures to be attached to the leading edge of the module are fabricated prior to building the first end frames of the module by the frame construction machinery and operators. The thruster modules are attached to the support structure and the plumbing and control circuitry are installed. The support structure assemblies are then

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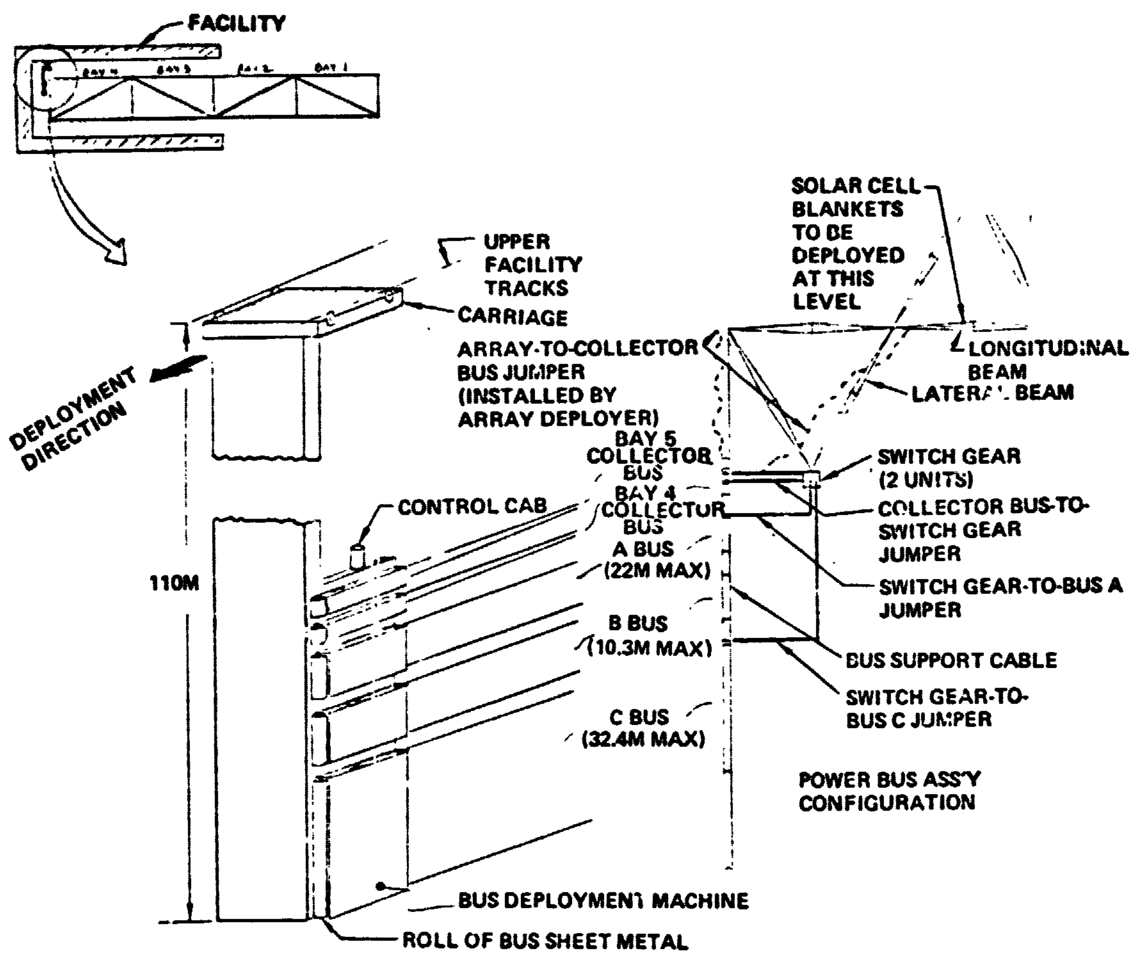


Figure 3.2-34 Satellite Power Bus Installation

moved off to the side of the facility and parked out of the way. When the first row of four bays has been advanced into the solar array deployment section of the facility, the thruster support structures are moved into position and attached to the module frame.

The plumbing and control circuitry are installed along the module frame simultaneously with the solar array deployment operations.

After the fifth row of module frames are assembled, the thruster propellant tanks are installed under the center point of the module.

After the aft end of the module has been advanced into the solar array deployment bay, the thruster support structures are assembled and attached to the module frame and the thrusters installed.

Eight crewmembers (during two shifts) have been allocated for thruster system installation and assembly. Eight crewmembers are allocated to the thruster subassembly fabrication in the subassembly factory.

3.2.1.1.1.4.6 Subsystem Installation

There are a variety of satellite subsystems that are to be installed (switch gear, sensors, control lines, data management and communication equipment, etc.). The configurations of these items have not been established as yet so detailed installation data was not developed. However, eight crewmembers (during two shifts) have been allocated for this activity.

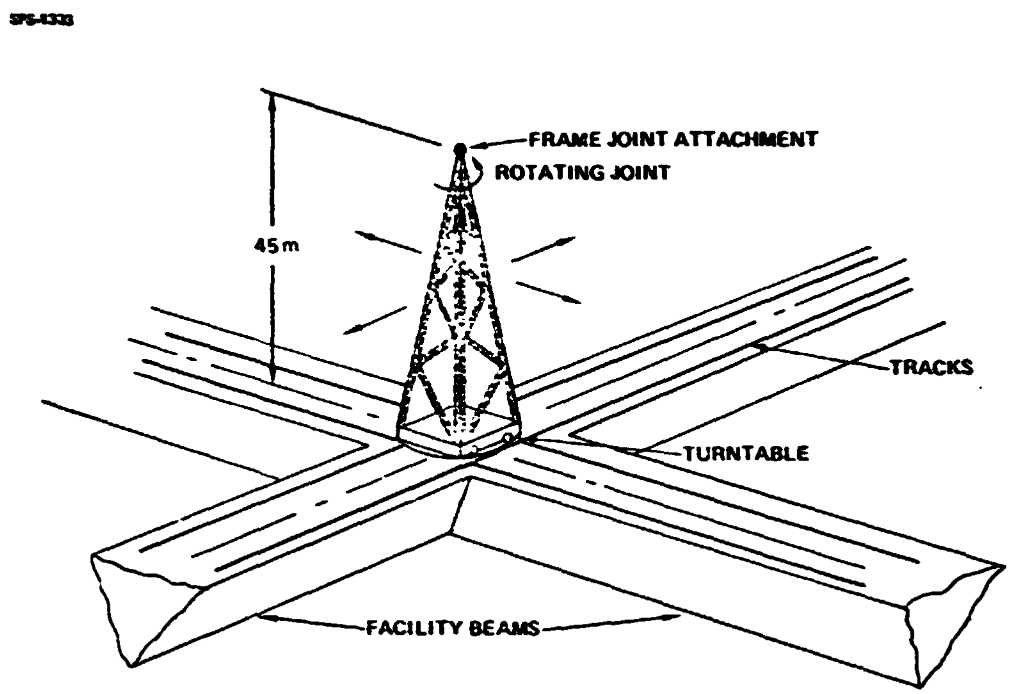
3.2.1.1.1.4.7 Module Indexing

As the module is constructed, it will be supported and indexed by the indexing support machine shown in Figure 3.2-35. A 200m tall version of this machine will be used to support the yoke during its construction.

An indexing speed of one meter per minute has been assumed.

These indexing machines would be operated by remote control from the command and control center.

This indexing concept has been applied in the antenna construction and in the thermal engine satellite construction.



- MINIMUM OF 3 OF THESE MACHINES ATTACHED TO SATTELITE FRAME AT ANY TIME

Figure 3.2-35 Satellite Support/Indexing Machine

3.2.1.1.1.5 Antenna Detailed Construction Task Analysis

3.2.1.1.1.5.1 Antenna Construction Tasks

The tasks involved in constructing the antenna are summarized in Figure 3.2-36. Figure 3.2-37 shows the antenna facility concept.

3.2.1.1.1.5.2 Primary Structure Assembly

The primary structure is composed of 130m long 5m beams that are fabricated by 5m beam machines that are similar in operation to the 20m beam machines previously described. These beams are assembled to form the structure as shown in Figure 3.2-38. Two beam machines and four 110m crane/manipulators are required. This structure requires the installation of joint plugs that will be preassembled in the central subassembly area (see Section 3.2.1.1.1.7).

3.2.1.1.1.5.3 Secondary Structure/Subarray Installation

The secondary structure is delivered to LEO as telescoped/compressed self-expanding packages. These structures are deployed as shown in Figure 3.2-39. The deployment platform to install the secondary structure and subarrays is shown in greater detail in Figures 3.2-40 and 3.2-41. Before attaching these structures to the primary structure, it is necessary to install wiring harnesses on the bottom surface. Two operators are assigned to the secondary structure assembly and deployment tasks.

The antenna subarrays are delivered to LEO preassembled. At LEO, these assemblies are tested and then stacked onto a transporter as illustrated in Figure 3.2-42. The subarray stack is then transferred to the deployment machine as shown in Figure 3.2-43. The subarrays are then installed onto the secondary structure and the wiring attached to the wiring harnesses on the secondary structure as shown in Figure 3.2-44. Two operators are assigned to the subarray deployment tasks.

3.2.1.1.1.5.4 Power Distribution System Installation

The installation of the power distribution system on the antenna is shown in Figure 3.2-45. The bus support structures and the switch gear assemblies are preassembled in the subassembly factory (see Section 3.2.1.1.1.7) and are delivered to the antenna factory lower level ready for installation.

The power bus system installation operations requires three 110m manipulator/cranes and a bus deployment machine.

3.2.1.1.1.6 Yoke Detailed Construction Analysis

The antenna yoke assembly is constructed within the module construction facility prior to constructing modules 4 and 8 as was shown in Figure 3.2-8 in Section 3.2.1.1.1.3. The yoke construction tasks and machinery used is shown in Figure 3.2-46. The equipment is operated by beam machine and crane/manipulator operators from the frame construction crew.

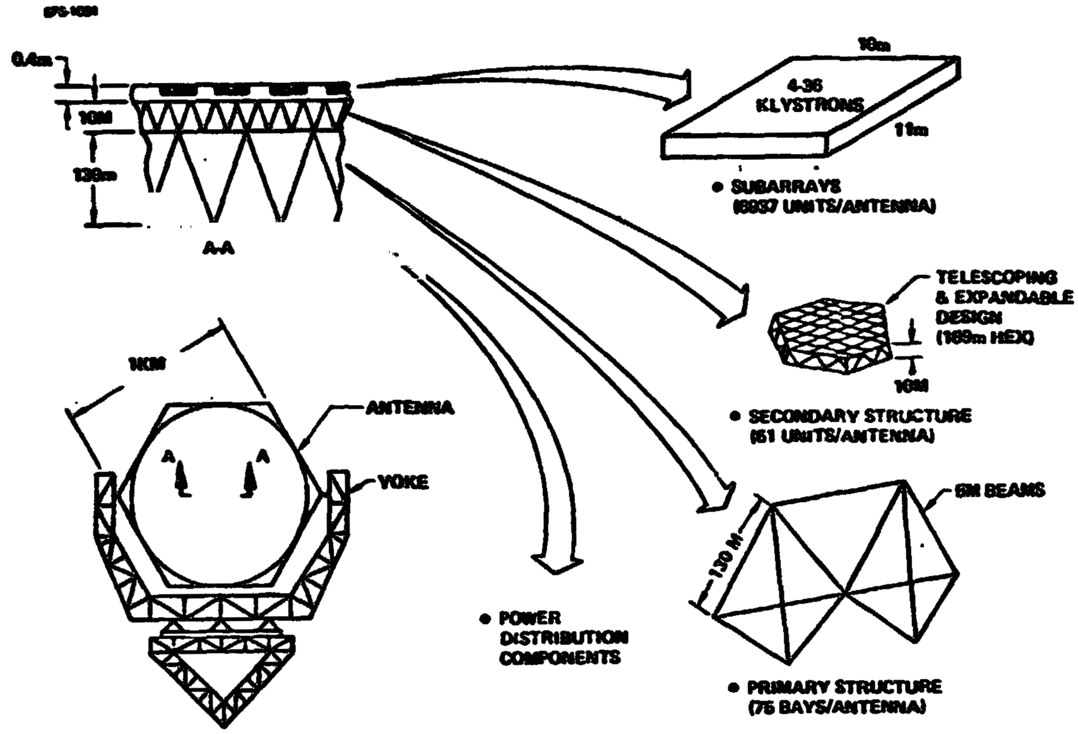


Figure 3.2-36 Antenna Construction Task

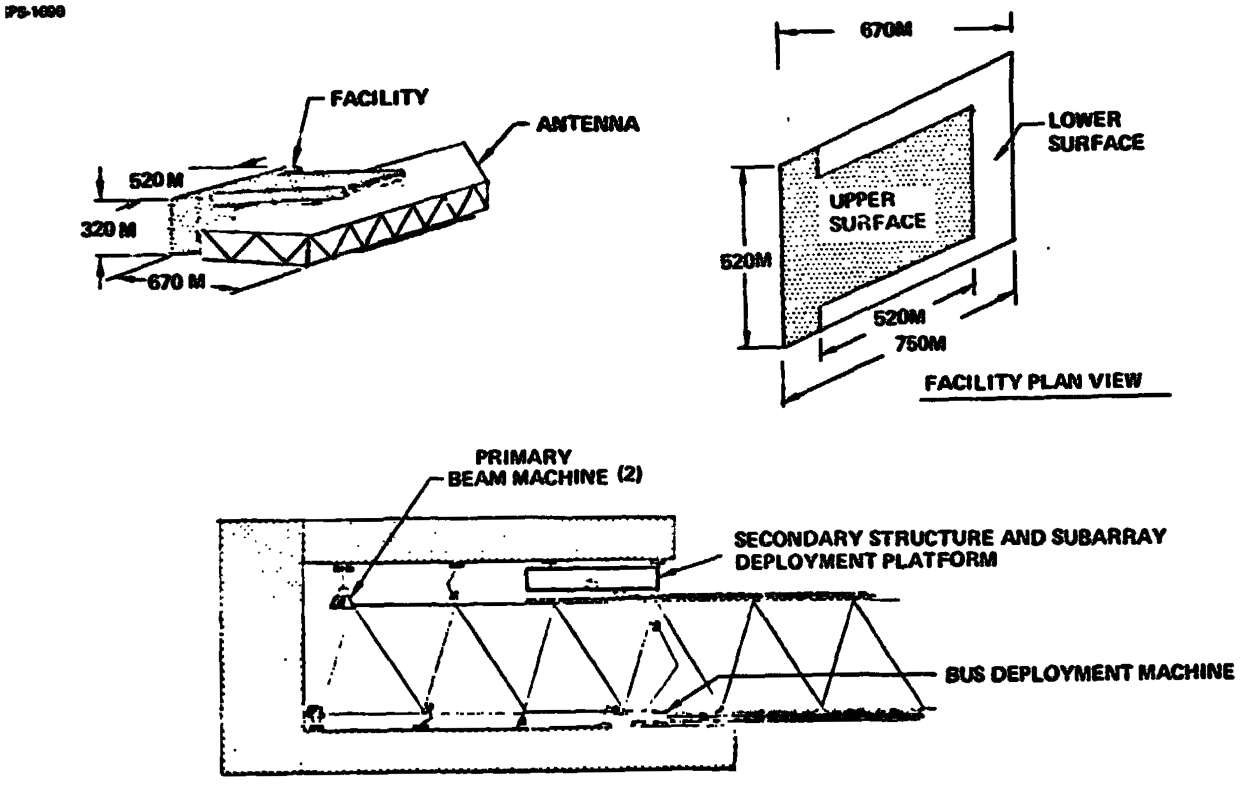


Figure 3.2-37 Antenna Construction Facility

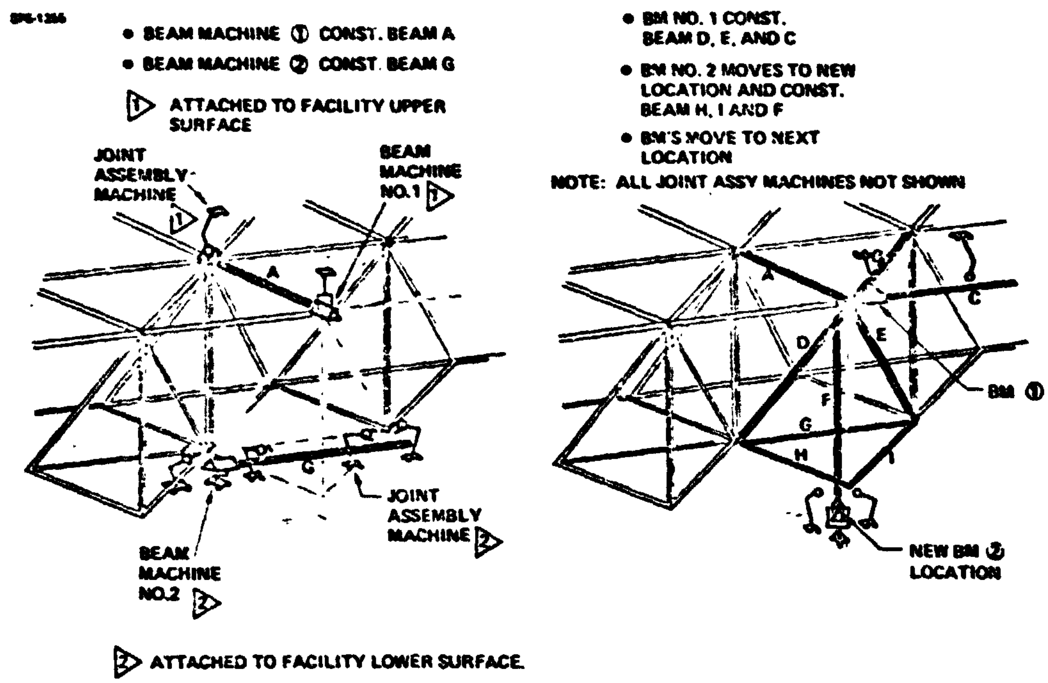


Figure 3.2-38 Antenna Primary Structure Construction Operations

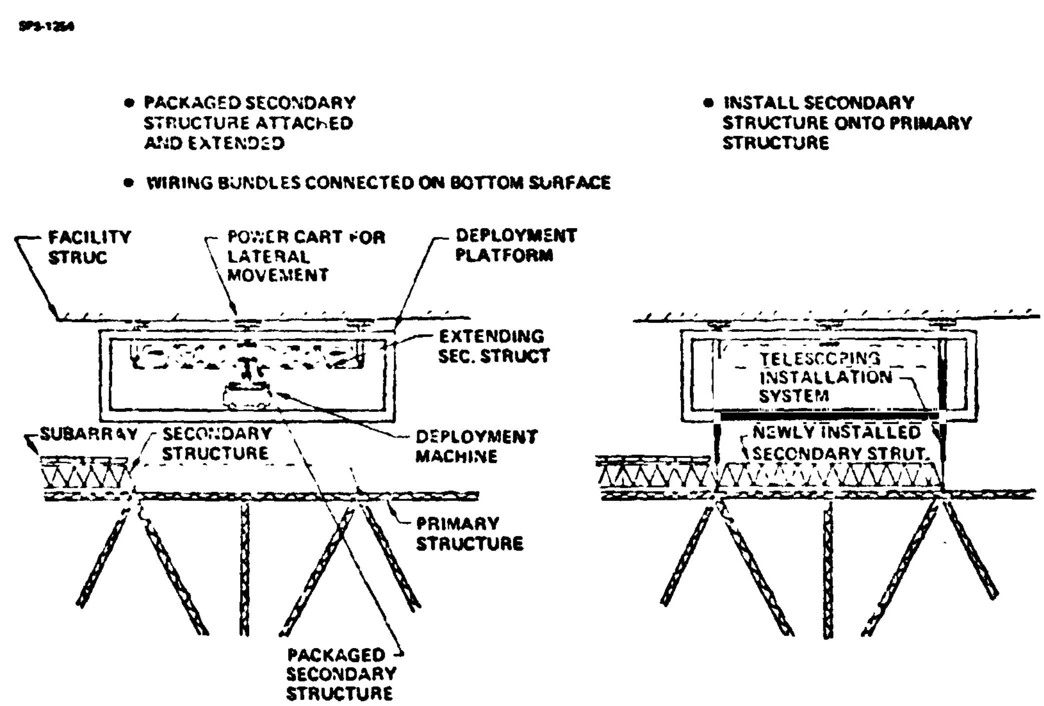


Figure 3.2-39 Antenna Secondary Structure Construction Operations

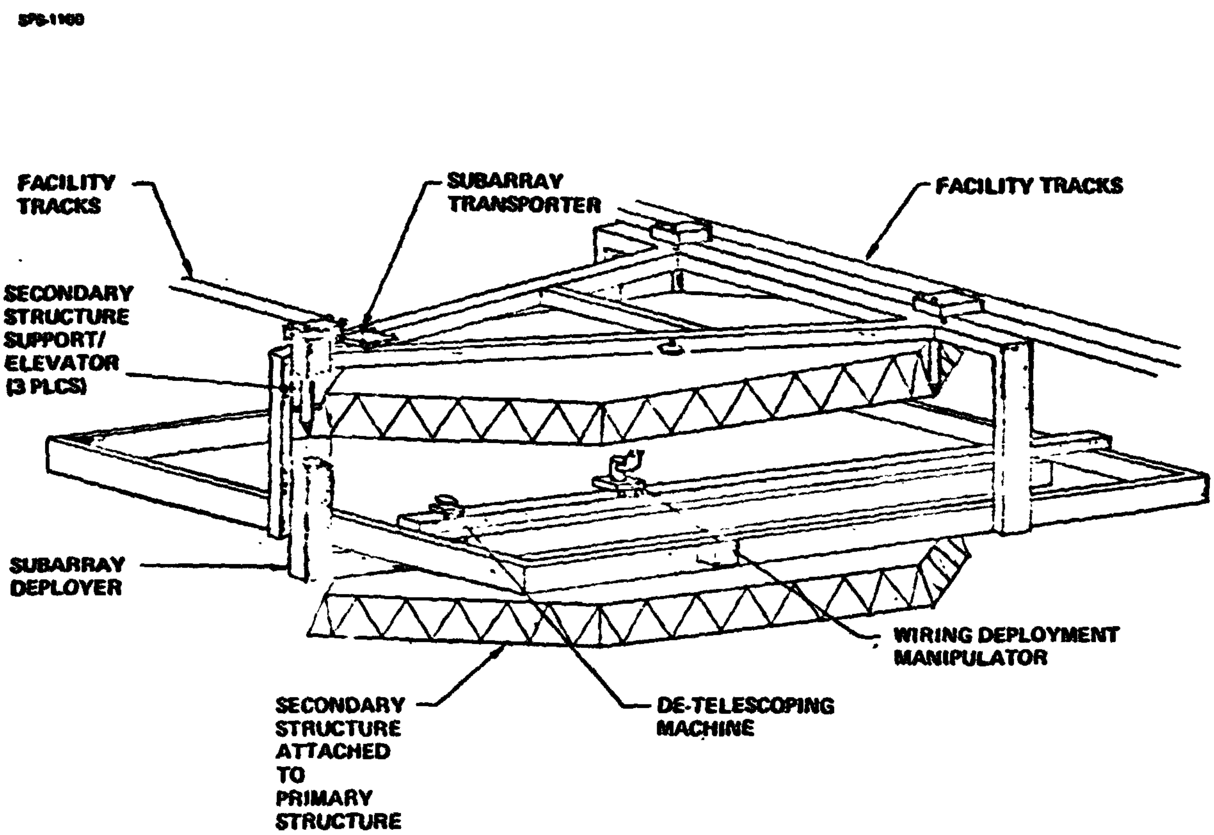


Figure 3.2-40 Deployment Platform

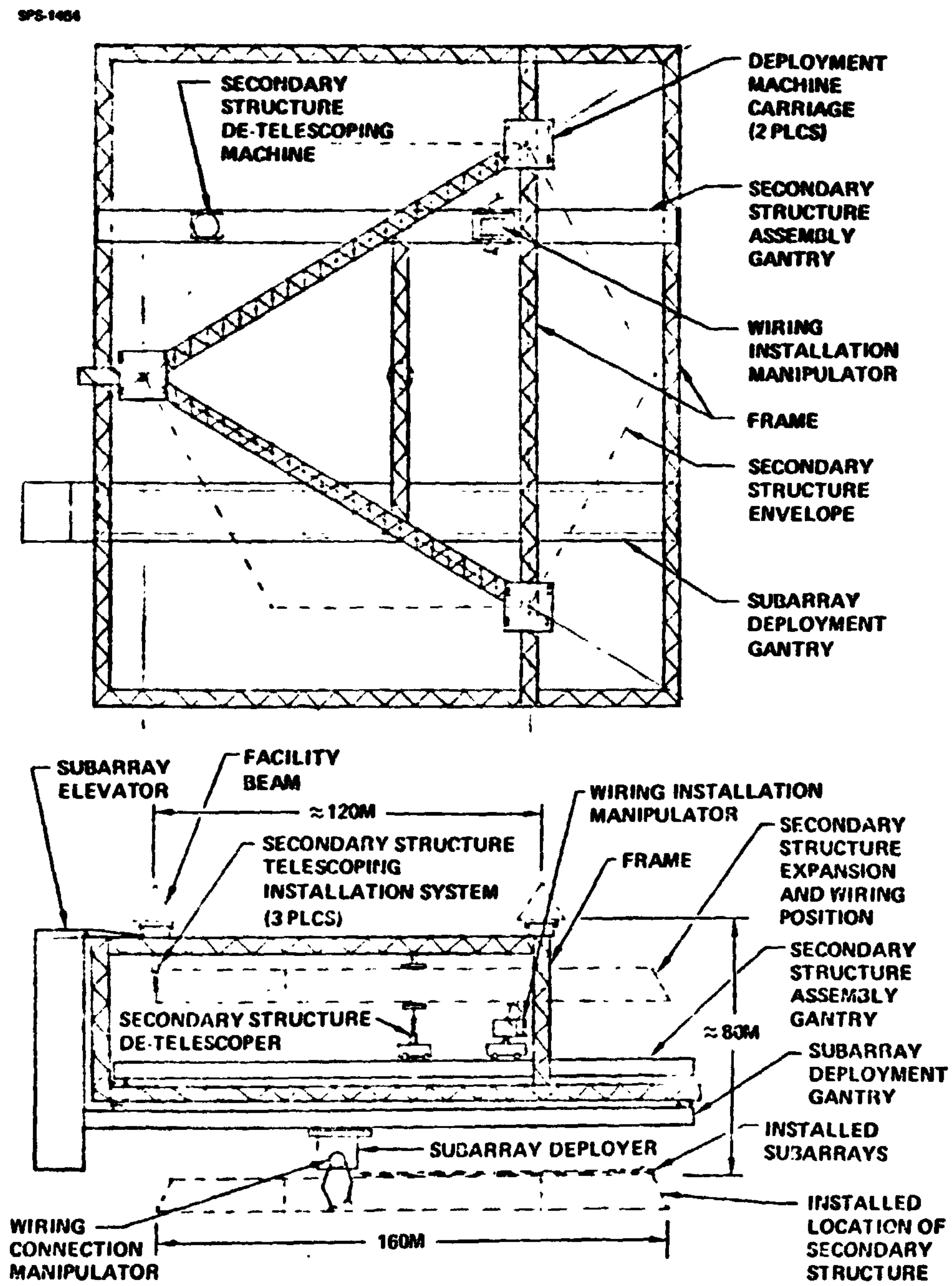


Figure 3.2-41 Deployment Platform Detail

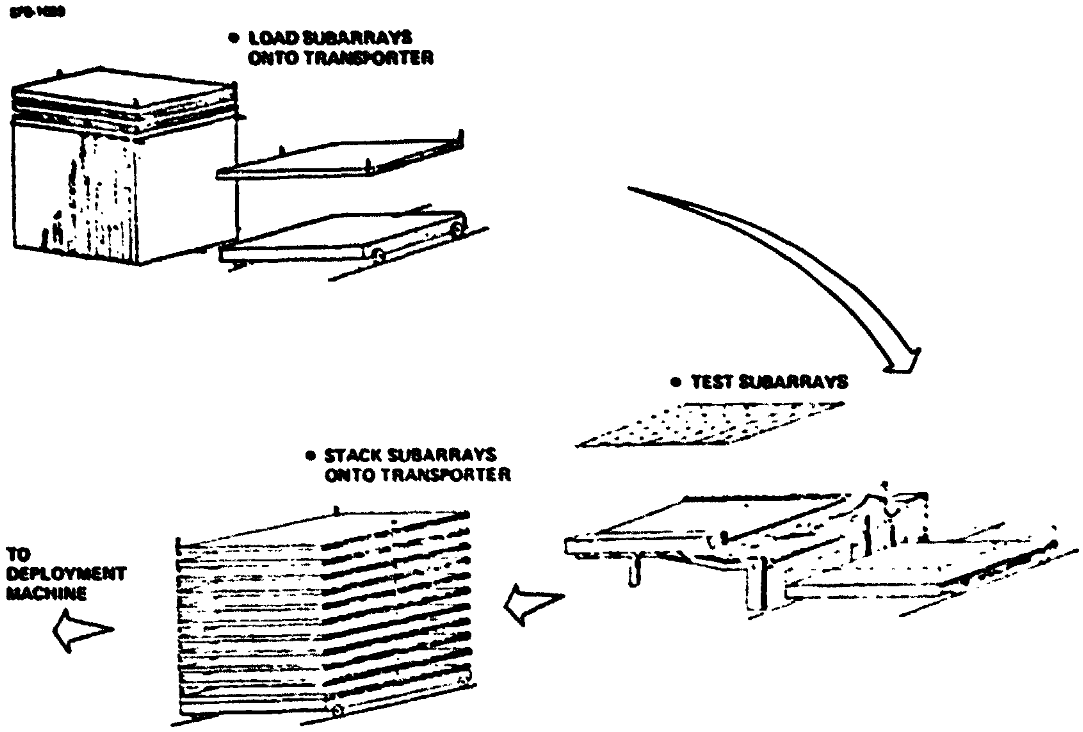


Figure 3.2-42 Subarray Preparation Process

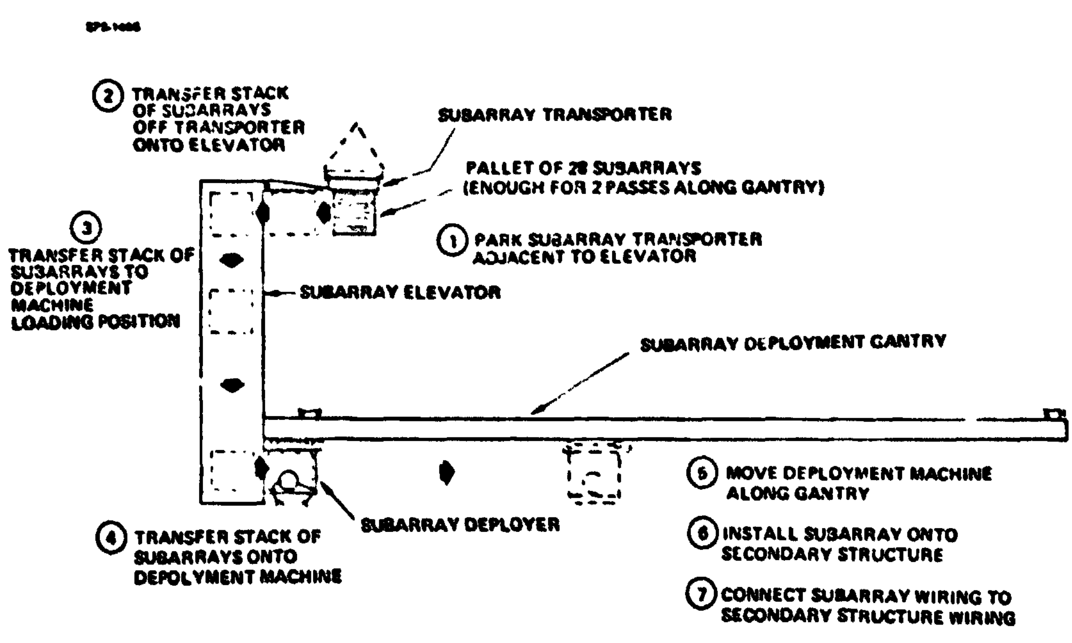


Figure 3.2-43 Subarray Deployment Sequence

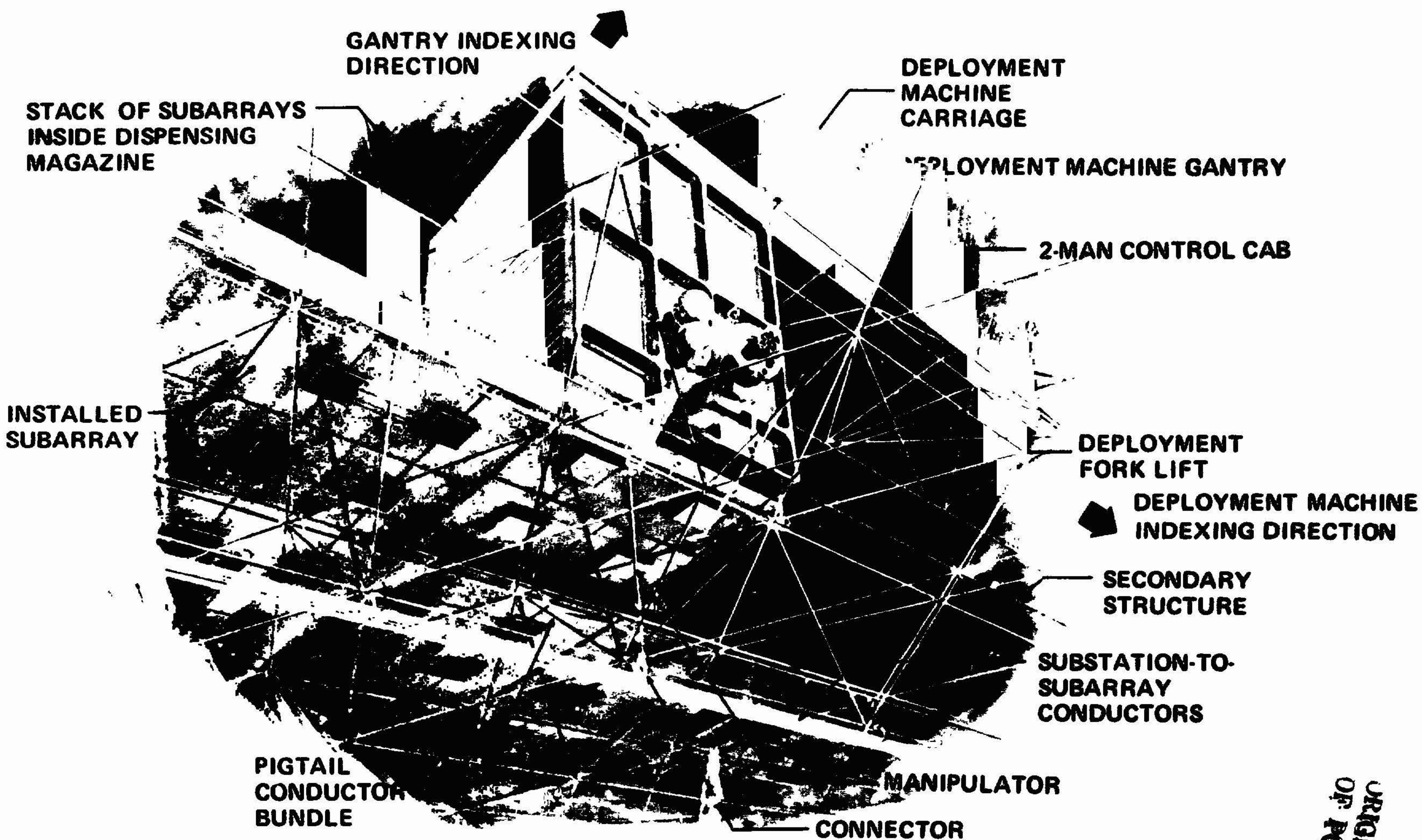


Figure 3.2-44 Subarray Deployment Machine Details

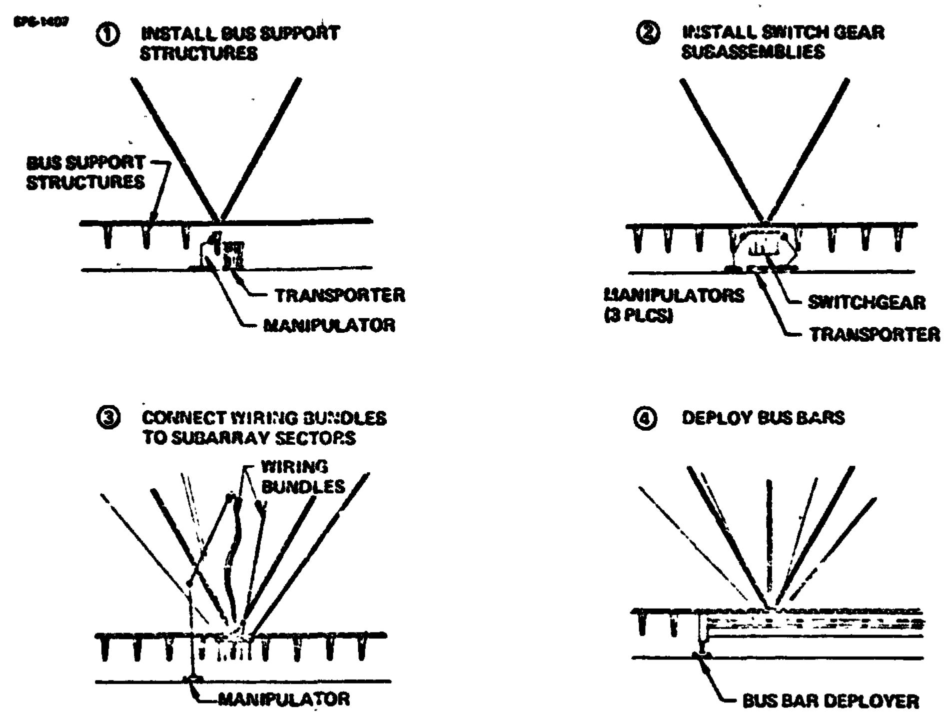


Figure 3.2-45 Power Distribution System Installation

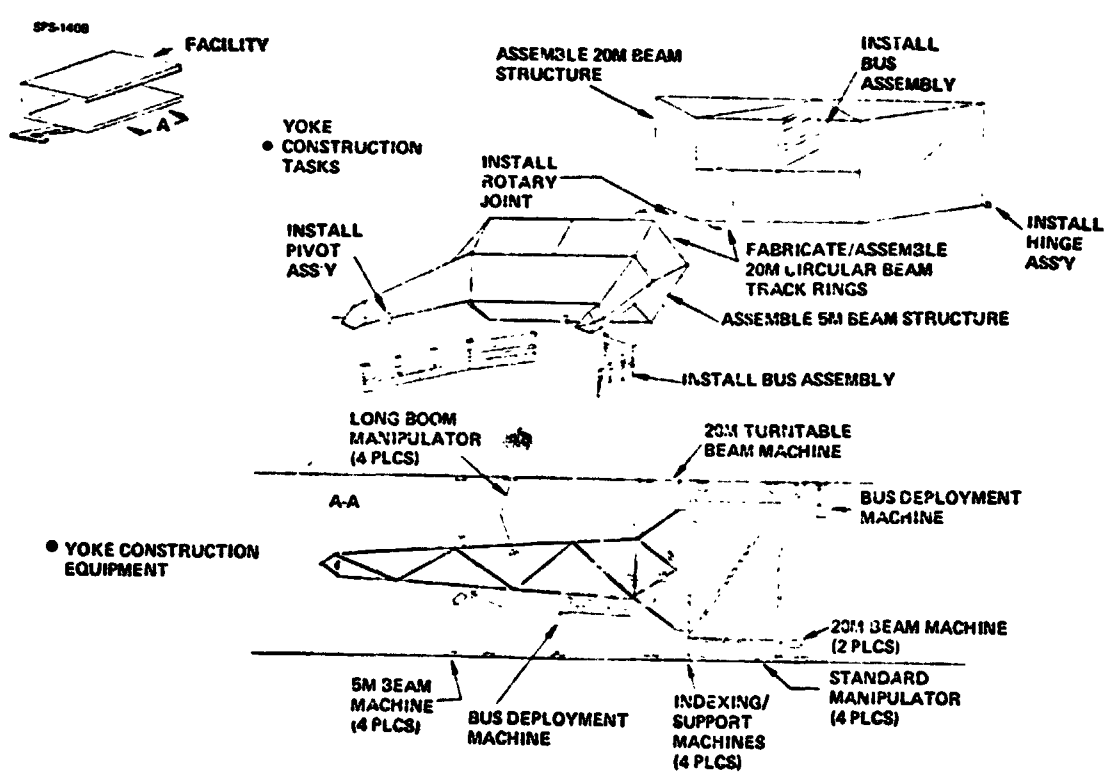


Figure 3.2-46 Antenna Yoke Assembly

3.2.1.1.1.7 Subassemblies

The various subassemblies described in the preceding sections will be assembled in a factory area that is adjacent to the central warehouse (see Figure 3.2-47). Crewmembers and assembly equipment have been allocated for the subassembly operations.

3.2.1.1.1.8 Construction Equipment Summary

The construction equipment described in the previous sections have been summarized in Table 3.2-1. Spares have not been included in this summary.

3.2.1.1.2 Construction Base Definition

This section describes the LEO base in detail. The overall configuration, foundation, cargo handling/distribution system, crew modules and subsystems are discussed.

3.2.1.1.2.1 Configuration

The general arrangement of the construction base has been described in Section 3.2.1.1.1.2. In summary, the base is divided into two major facilities with one used to construct the satellite and the other to construct the antennas.

The overall construction base is shown in greater detail in Figure 3.2-48. The principal elements of the base include the foundation (structural framework), cargo handling and distribution system, crew modules and base subsystems.

The foundation for both the module and antenna facilities include upper and lower surfaces to which construction equipment is attached, the satellite is supported and other base elements are attached.

Ten primary crew modules are located in an area where the greatest concentration of personnel are involved while performing their daily duties. Six of the modules serve as crew quarters and four as work centers. Other pressurized shirt sleeve environment modules or cabins are also present but serve only as small work quarters sometimes referred to as remote work stations or control cabs.

Docking provisions for all transportation vehicles are located along the back edge of the module facility. The orbit transfer vehicle operations center is located at the opposite end of the base from the crew modules due to the required propellant transfer operations.

Each of the base elements is described in additional detail in subsequent paragraphs.

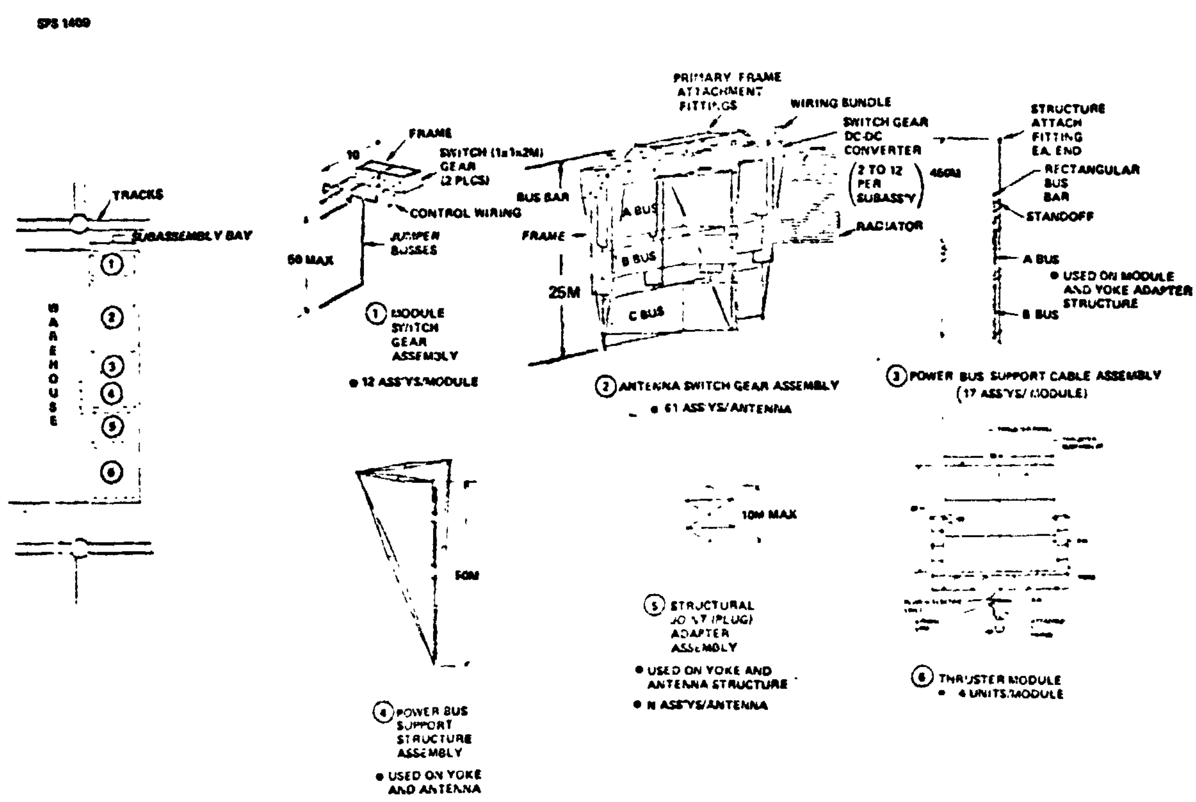


Figure 3.2-47 Subassemblies

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Table 3.2-1 LEO Base Construction Equipment – Photovoltaic Satellite

EQUIPMENT ITEM	NUMBER REQ'D MOD	ANT	YOKE	SUBASSY	EQUIPMENT ITEM MAJOR ELEMENT	NO. REQ'D ITEM
• 20M BEAM MACHINE 1 ALSO USED TO MAKE MODULE-TO VOICE INTERFACE STRUCTURE (MASS 20^K Kg) (COST $100M) A	2 1				• CARRIAGE • YOKE ASSY • STRUT ASSY MACHINE • JOINT FITTING FEED MECH • INDEXING CARRIAGES • STRUT MAGAZINES • JOINT FITTING CARROUSEL • CONTROL CAB (2 MAN)	1 1 9 3 6 18 3 1
• 5M BEAM MACHINE (MASS 7^K Kg) (COST $35M)		2	2			
• 5M [illegible] BEAM MACHINE (MASS [illegible] Kg) (COST [illegible])			1			
• 20M MANIPULATOR/CRANES 2 THRUSTERS 2 SW GEAR 4 STRUCTURES 4 CABLE ASSY 1 (MASS [illegible] Kg) (COST [illegible])		2		11 2	• CARRIAGE • ELEVATOR BOOM • TRANSVERSE BOOM • CONTROL CAB (1 MAN) • MANIPULATOR ARM	1 1 1 1 2
• 110M MANIPULATOR/CRANE (MASS [illegible] Kg) (COST [illegible])	8	8				
• 250M MANIPULATOR/CRANES (MASS 12^K Kg) (COST [illegible])			2			

A ALL COST REFLECT AVG UNIT COST AFTER APPLYING LEARNING FACTOR OF 0.9.

Table 3.2-1 (Continued)

SPS 1477

EQUIPMENT ITEM	NUMBER REQ'D MOD	ANT	YOKE	SUBASSY	EQUIPMENT ITEM MAJOR ELEMENT	NO. REQ'D ITEM
• 45M INDEXING/SUPPORT MACHINE (MASS [illegible] Kg) (COST [illegible])	6	6	2		• CARRIAGE • BOOM	1 1
• 200M INDEXING/SUPPORT MACHINE (MASS [illegible] Kg) (COST [illegible])			2		• CARRIAGE • BOOM	1 1
• BUS DEPLOYMENT MACHINE • [illegible] BOOM • [illegible] BOOM • 110M ARTICULATING BOOM 2 NOT REQ'D ON YOKE AND ANTENNA MACHINES (MASS [illegible] Kg) (COST [illegible]) { AVG FOR THE 3 SIZES	1	1	1		• CARRIAGE • BOOM • BUS DEPLOYMENT MACHINES • A BUS • B BUS • C BUS • COLLECTOR BUS • CONTROL CAB (2 MAN)	1 1 1 1 1 2 1
• SOLAR ARRAY DEPLOYMENT MACHINE (MASS [illegible] Kg) (COST [illegible])	4				• CARRIAGE/GANTRY • BLANKET MAGAZINE • BLANKET FEED MECH • BLANKET PACKAGE INST MACH • BLANKET DEPLOYER • CARRIAGE • BLANKET END HANDLER MECH • EDGE CLAMPER • CONTROL CAB (2 MAN)	1 1 1 1 1 1 1 1

Table 3.2-1 (Continued)

SPS 1478

EQUIPMENT ITEM	NUMBER REQ'D				EQUIPMENT ITEM MAJOR ELEMENT	NO. REQ'D ITEM
	MOD	ANT	YOKE	SUBASSY		
• DEPLOYMENT PLATFORM [3] INCLUDED IN 20M MANIP/CRANE COUNT (MASS 28^{K} Kg) (COST $80M)					• CARRIAGE/FRAME ASSY • SECONDARY STRUCTURE INST TELESCOPES • SECONDARY STRUCTURE DEPLOYMENT GANTRY • GANTRY/CARRIAGE • DETELESCOPING MACH • 20M MANIPULATOR/CRANE • SUBARRAY DEPLOYMENT GANTRY • GANTRY/CARRIAGE • ELEVATOR • SUBARRAY DEPLOYER • CARRIAGE • MAGAZINE • DEPLOYMENT MECH • 20M MANIP/CRANE • CONTROL CAB (2 MAN)	1 3 1 1 1 [3] 1 1 1 1 1 1 [3] 1
BUS BAR ROD BENDER				1		
BUS BAR WELDER				2		
RADIATOR PIPE WELDER				1		
CABLE FITTING MACH				1		
STRUT ASSY MACH (4 SIZES)				11		
SUBARRAY TESTING MACH				1		

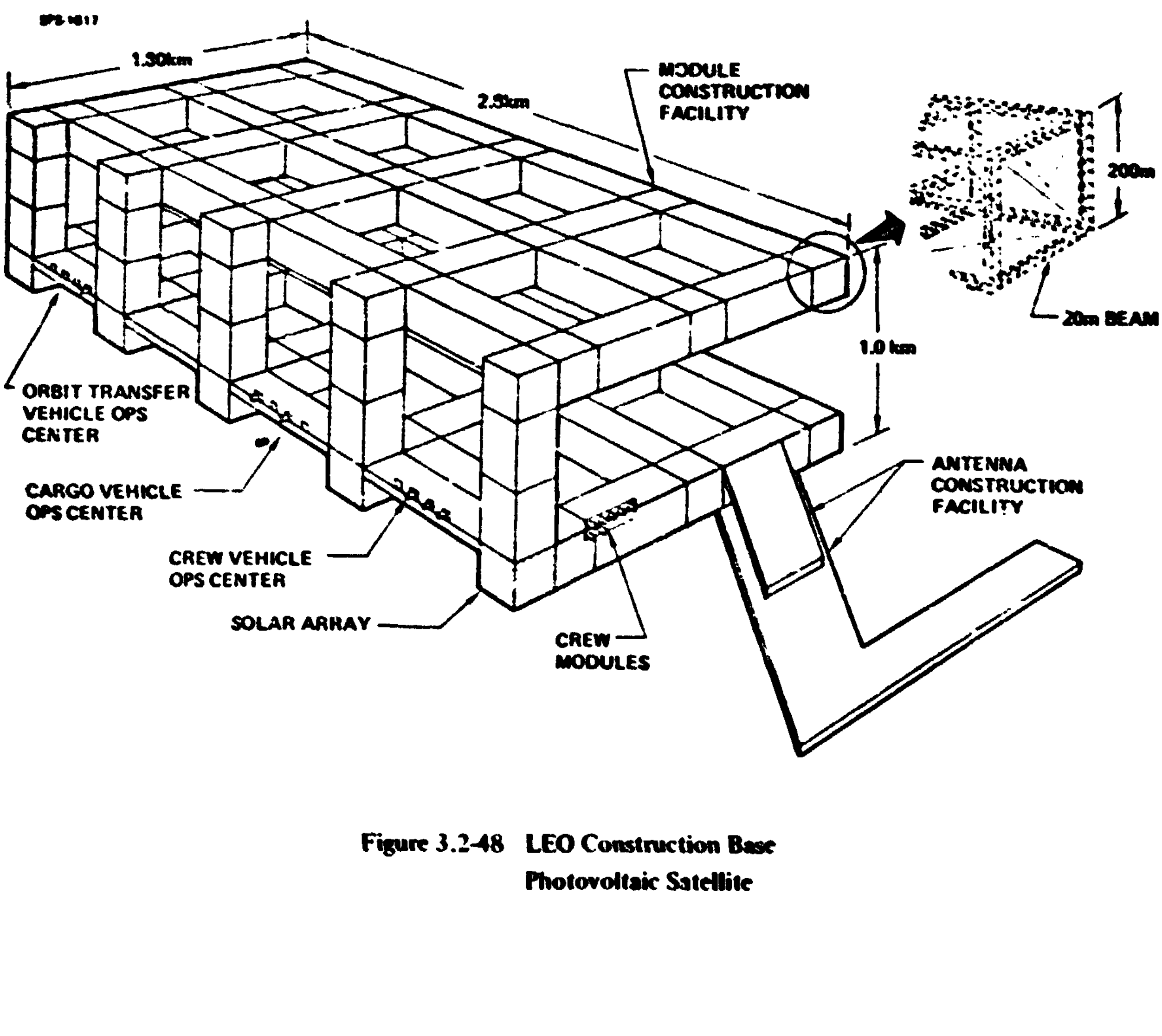

Figure 3.2-48 LEO Construction Base
Photovoltaic Satellite

3.2.1.1.2.2 Foundation

Function

The foundation or structural framework of the construction base must provide a mounting/attachment surface for all construction equipment as well as mounting provision for other base elements such as crew modules, cargo handling and distribution systems and base subsystems.

Design Criteria and Analysis

The principal loading conditions which should be considered in sizing the structure include the following:

- **a. Altitude or attitude corrections at 10^{-4} g's**
- **b. Movement of construction equipment at their required construction rates**
- **c. Indexing of the satellite module antenna**
- **d. Gravity gradient**
- **e. Docking of transportation vehicles.**

A preliminary analysis was done considering items a, b, and c and resulted in item "a" having the largest loading condition as applied through a 20m beam which was assumed for the structure. This loading however is less than 35% of the load carrying capability of 20m beams typical of those used in the satellite where strut wall thicknesses of 0.05 cm (0.020 in.) are used.

The final criteria used in establishing the base structure was that of having the proper natural frequency relationship relative to the constructed element (either a module or antenna). In general, this means having different frequencies. In the case of the module, the natural frequency is lowest when it is finished and accordingly it has a much higher frequency when construction has just begun.

Frequency considerations for the module construction facility would initially indicate a higher frequency (greater stiffness) would be desirable relative to the constructed module. However as previously indicated the module frequency goes from high to low as it is constructed, consequently, in order to prevent a frequency crossover during construction and to minimize the depth of the structure the facility has been designed to have a lower frequency than the module at any time during the construction phase.

Frequency calculations for a module (one-eighth of a satellite) and module construction facility were done assuming each to act as if it is cantilevered and pin-ended relative to the other. Furthermore, the natural frequency of the construction facility was determined by assuming the "U" shaped facility was spread out as a long flat truss. Analyzed in this manner and assuming five 200m deep trusses each consisting of four 20m beams resulted in a natural frequency of approximately 40 cphr which was lower than the satellite module frequency of approximately 45 cphr at its completion.

The antenna facility was not analyzed in this manner since the natural frequency of the antenna itself was not determined.

Selected System Description

The foundation or structure of the module facility consists of five 200m trusses formed in the shape of a "U" and tied together with three 200m lateral trusses in both the upper and lower surfaces of the facility as was shown in Figure 3.2-48. Each truss consists of four 20m beams running its entire length. The truss also includes perpendicular and diagonal members which are also 20m beams. The 20m beams are the same type as used in the satellite with all individual struts having a wall thickness of 0.05 cm (0.020 in.) resulting in a mass of 5 kg per meter. This sizing appears to be rather conservative but seems justified at this point in the analyses. The module construction facility was found to have approximately 435,000m of 20m beam.

The antenna construction facility was assumed to have truss depths of 50m in its upper and lower surfaces. As a result, a total length of 53,000m of 20m beam was estimated. Again, the 20m beam was assumed to have a mass of 5 kg/meters.

3.2.1.1.2.3 Cargo Handling/Distribution System

One of the keys to high productivity is an efficient base logistics system designed to move the large quantities of materials from the receiving area to the user machines. The base logistics system is shown in Figure 3.2-49. The central receiving and warehousing area is shown in Figure 3.2-50. A concept for transporting personnel around the base and between the bus and a control cab is shown in Figure 3.2-51.

A summary listing of the equipment used in cargo handling and distribution is presented in Table 3.2-2.

3.2.1.1.2.4 Crew Modules

Module Definition

A total of ten primary crew modules have been included in the LEO construction base. The modules have an Earth atmosphere environment and have been sized to accommodate crew sizes between 50 and 100 or to serve as large work areas. Accordingly, the modules have dimensions of 17m diameter and up to 23m length. Excluded from this category of modules then are the crew buses used to transfer personnel around the base and the small two-man control cabins used in conjunction with the construction equipment and cargo handling and distribution equipment.

A summary listing of these modules and their functions are presented in Table 3.2-3. All modules are self-sufficient in terms of environmental control provisions and emergency power. Primary power is obtained through a common power supply provided by the base. Functions peculiar to each module have been identified in Table 3.2-3. Five crew quarter modules have been provided

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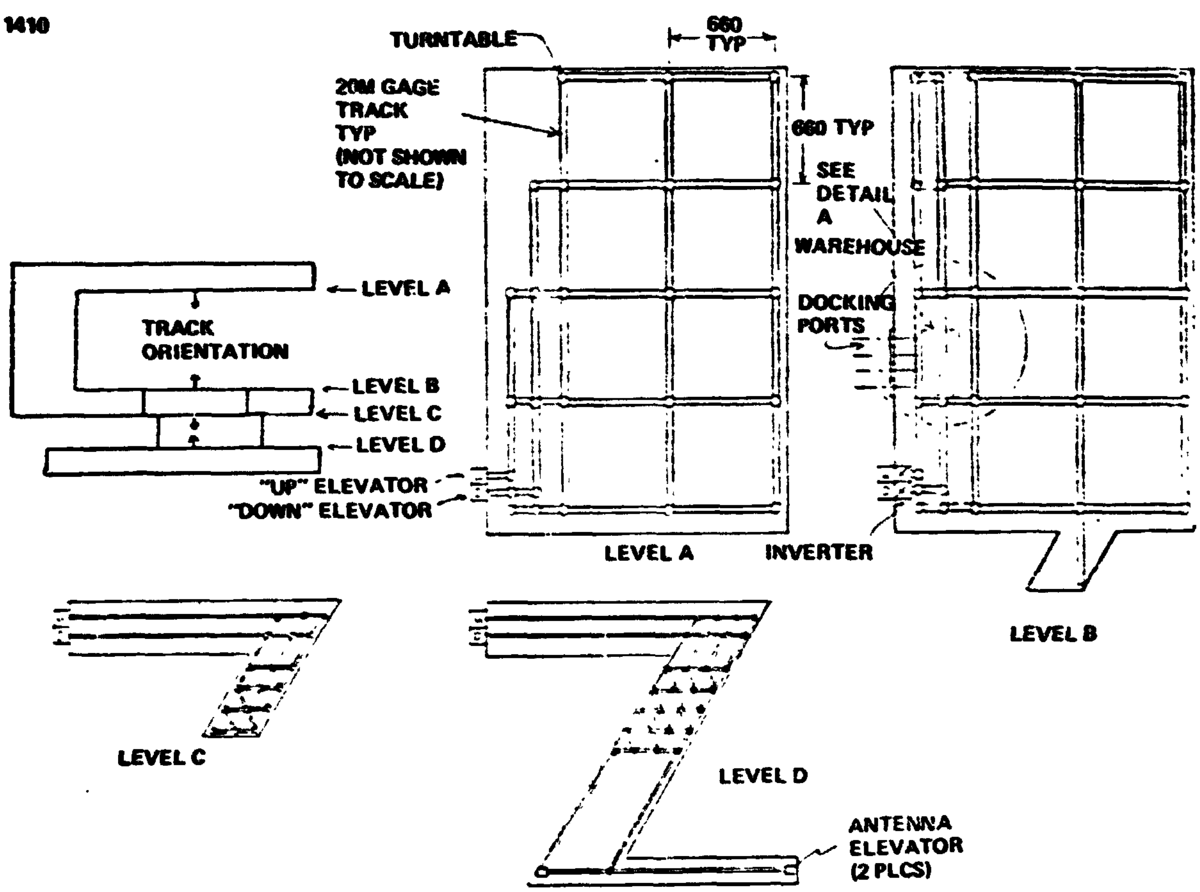

Figure 3.2-49 LEO Base Facility Logistics System

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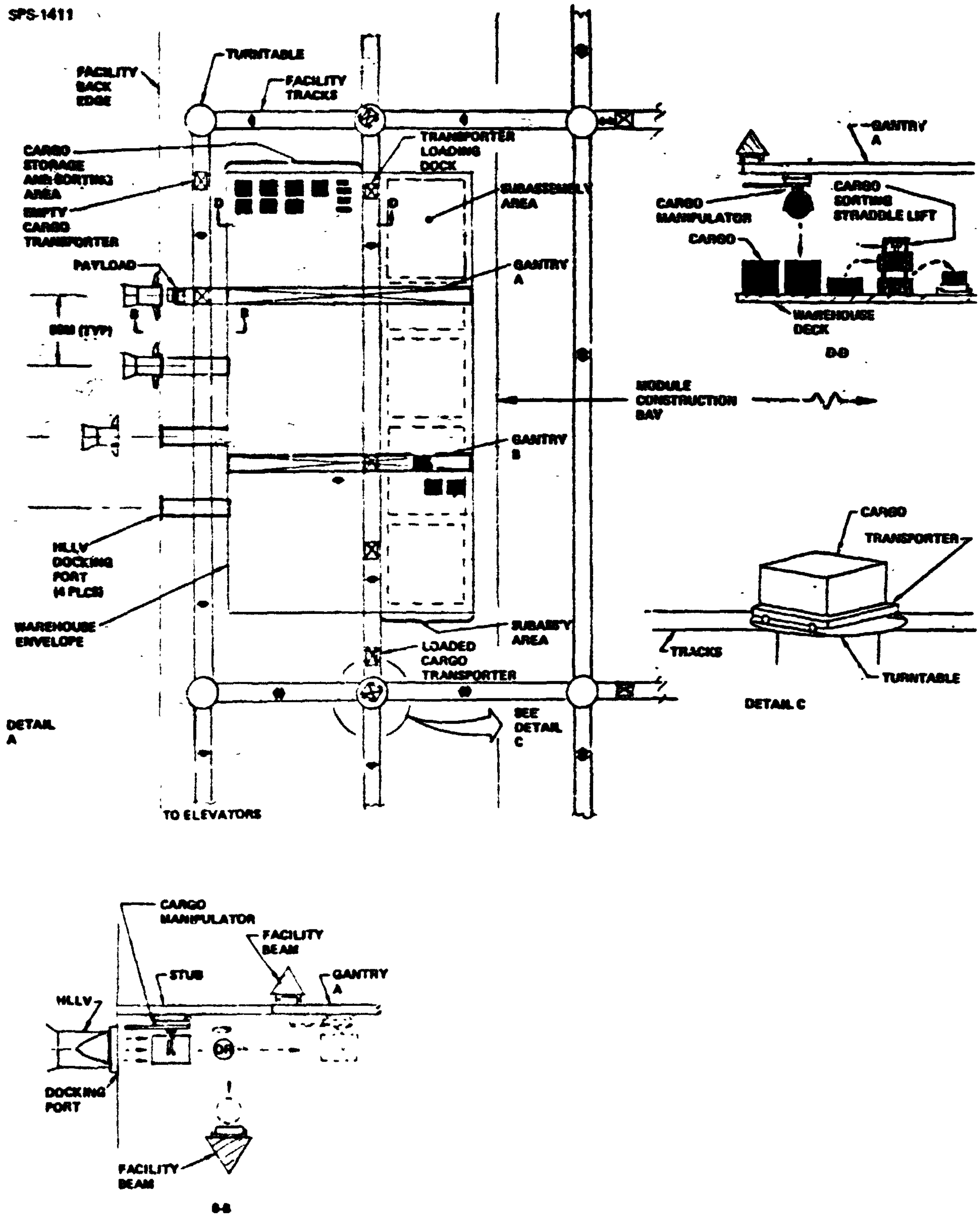

Figure 3.2-50 Central Receiving and Warehousing

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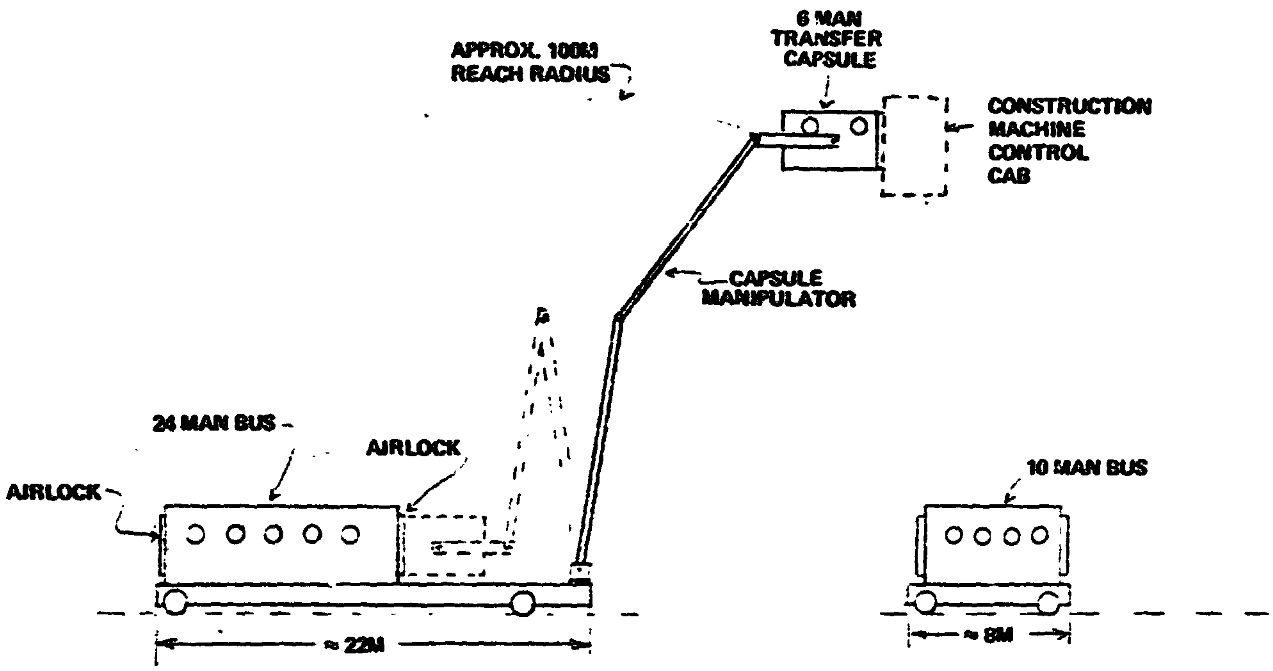


Figure 3.2-51 Crew Bus

SPS 1479

Table 3.2-2 LEO Base Cargo Handling & Distribution Equipment Photovoltaic Satellite

EQUIPMENT ITEM	NO REQ'D	MASS (EA) 10^3 Kg	COST (EA) $\$10^6$
• HLLV CARGO DOCKING PORT	4	4	10
• HLLV CARGO EXTRACTION SYS	4		6
• HLLV TANKER DOCKING PORT	3	4	10
• HLLV TANKER CARGO EXTRACTION SYS	3		6
• OTV TANKER DOCKING PORT	2		
• OTV TANKER LOADING SYS	2		
• SHUTTLE DOCKING PORT	3		
• GROWTH SHUTTLE DOCKING PORT	2		
• PERSONNEL TRANSFER AIRLOCK SYS	6		
• GANTRY CRANE	2		
• CARGO SORTING MANIPULATOR/TRANSPORTER	2		6
• TRANSPORTER ELEVATOR	2	8	7
• TRANSPORTER INVERTER	2		7
• ANTENNA ELEVATOR	2		
• 24 MAN CREW BUS	2	12	7
• 10 MAN CREW BUS	2	5	4

SPS 1480

Table 3.2-2 (Continued)

EQUIPMENT ITEM	NO REQ'D	MASS (EA) 10^3Kg	COST (EA) $\$10^6$
• TURNTABLES	98	0.2	0.1
• CONTROL CABS FOR LOGISTICS EQUIP	7		
• HLLV CARGO	1		
• HLLV/OTV TANKER	1		
• SHUTTLE/SHUT GROWTH	1		
• GANTRY CRANES	2		
• CARGO SORTER	2		

Table 3.2-3 Construction Base Crew Modules

MODULE	QUANTITY	FUNCTION (PROVISIONS)
• CREW QUARTERS	5	• PERSONAL QUARTERS/HYGIENE • PHYSICAL FITNESS/RECREATION • DINING
• TRANSIENT CREW QUARTERS	1	• USED DURING CREW ROTATION PERIODS • HOUSE VIP'S • EMERGENCY QUARTERS
• OPERATIONS CENTER	1	• BASE OPERATIONS • CONSTRUCTION OPERATIONS
• MAINTENANCE, TEST AND CHECKOUT	1	• CONSTRUCTION EQUIPMENT • SATELLITE COMPONENTS
• TRAINING & SIMULATION	1	• NEW PERSONNEL • NEW CONSTRUCTION OPERATIONS
• UNDEFINED	1	• CLINIC (COULD BE IN SPS CENTER)

NOTE: ALL MODULES SELF-SUFFICIENT EXCEPT PRIMARY POWER AND FLIGHT CONTROL.

with each sized for a crew of 100. These modules provide all of the off-work functions associated with living. Further information concerning the sizing of each module is presented in subsequent paragraphs.

As indicated, a transient crew quarters has been provided. The logic associated with this module relates to crew rotation periods where the overlapping of the crews could occur without causing inconvenience in terms of quartering etc., and also allows for time to clean up the rooms or modules of the departing crew. An additional feature of this module concerns itself with an emergency situation where one of the primary crew quarters has a failure or in the event a crew scheduled to move from the LEO base up to GEO or back to Earth are unable to do so due to weather, vehicle trouble, etc.

The operations module serves as the control center for all base operations and construction operations. Typical base operations to be controlled from this module include that associated with the primary power supply and flight control system (attitude and station keeping), communication system within the base as well as that with Earth, other bases and transportation vehicles in transit. Overall crew scheduling and consumables management functions are also included under base operations. Construction operations controlled from the module include those functions associated with scheduling, briefings, troubleshooting or identifying workarounds, monitoring of the actual construction operations being conducted and the operations associated with cargo handling and distribution. Another function provided by the operations module is that of housing the central data management and processing center.

The maintenance, test and checkout module provides the capability to work on large pieces of construction or base equipment or satellite components while in an Earth atmosphere environment.

A training and simulation module has been included with its primary purpose being to train new personnel and to establish and/or demonstrate certain construction tasks while in a controlled environment.

A tenth module has been included primarily to cover the volume requirements of functions not included in other modules at this time. Examples of such functions include clinic type provisions in terms of medical, dental and sickbay provisions as well as for the temporary containment of personnel who have died while on duty. Isolation of the sickbay from the other base crew quarters seems to be particularly important due to relatively confined volume that is available.

Crew Quarters Sizing and Design

Selection of a crew quarters module to accommodate 100 people came about as a result of the following factors. First, there was the capability to have a very large module due to the large payload envelope when using the reference two stage cargo launch vehicle. Secondly, when the floor area requirements for 100 people were matched with the available module envelope there was adequate space (actually space is available for close to 130 people). The mass of a 100 person module was found to be well within the capability of the launch vehicle. In summary, a larger crew could be accommodated within the volume and mass constraints of the launch vehicle, however, 100 people in one basic living volume appears to be quite sufficient.

Floor area requirements associated with a 100 person module and the division of functions among the decks of the module are shown in Figure 3.2-52. The indicated area allocations are based to a large degree on the Rockwell Integral Space Station Study (NAS9-9953). It should also be pointed out that the indicated areas reflect having all 100 people present which is a case which occurs one day per week when both shifts are off-duty.

The size of the module to contain the required floor space is 17m in diameter and approximately 20m in length including the spherical end domes. The module is divided into seven decks with the indicated functions performed on each deck. General arrangement within each deck was not performed at this time.

3.2.1.1.2.5 Subsystem Definition

The design approach used for each subsystem was generally the same as defined by Rockwell in their solar powered integral Earth orbit space station study (NAS9-9953) for JSC in 1970. A summary of these subsystems is provided in Table 3.2-4 and described below.

Structure

Crew module structure primarily consists of aluminum alloy. The pressure compartment is designed for an operating pressure of 101000 n/m^2 (14.7 psia). The outer shell of each module consists of a double bumper micrometeroid protection system that was designed to give a 0.9 probability of no penetration in 10 years. Quite possibly, this particular design criteria will merit further examination in the future. Also included in the outer bumper system is the thermal radiator for internal heat rejection. An aerothermal shroud for the crew modules is not required since they will be launched within the payload shroud of the launch vehicle.

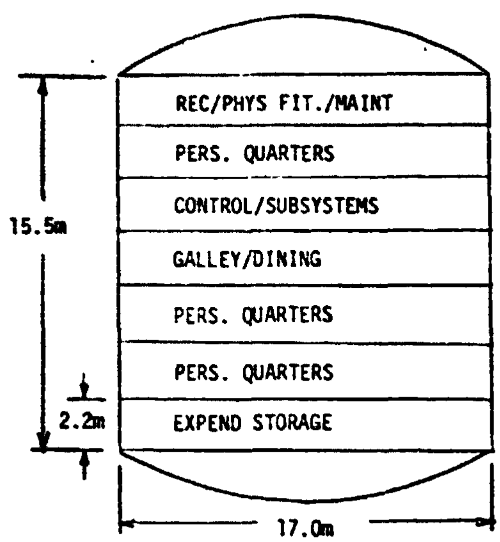
Electrical Power

The primary electrical power system is discussed under the Base Subsystem Section 3.2.1.1.2.5. Each crew module however incorporates an emergency power system consisting of fuel cells. Distribution, wiring and special power conditioning equipment is also included in each module.

Environmental Control

All modules have an independent ECS. The system provides an Earth atmosphere environment. Oxygen makeup for leakage and usage is provided through electrolysis of water which is obtained by reduction of CO_2 using Sabatier reactor with CO_2 itself is removed using molecular sieves.

- o SIZED FOR 100
- o FLOOR AREAS SCALED FROM 12 MAN UNITARY SPACE STATION (BASED ON ROCKWELL 1970 STUDY - NAS 9-9953)
- o AREAS BASED ON ENTIRE CREW BEING PRESENT



- o GEO MODULE MODIF
  - o ADD 1 DECK FOR RADIATION SHELTER

ALLOCATIONS PER MODULE

FUNCTION		FLOOR AREA M^2	(FT^2)
o PERSONAL QTRS		512	(5500)
o PHYSICAL HYGIENE		89	(960)
o RECREATION		107	(1150)
o PHYSICAL FITNESS		53	(570
o GALLEY		53	(570
o DINING		116	(1250)
o CONTROL CENTER		37	(400)
o SUBSYSTEMS		149	(1600)
o MAINTENANCE SHOP		9	(100)
o EXPENDABLE STORAGE (90 DAYS)		193	(2080)
o TUNNELS/AISLES		163	(1750)
	TOTAL	1480	15930
	MARGIN	71	760

Figure 3.2-52 Crew Quarters Sizing

Table 3.2-4 Subsystem Summary

CREW MODULES	

• STRUCTURE	• ALUMINUM ALLOY
	• METEOROID PROTECTION
	  • $P_{(O)} = 0$ FOR 10 YRS.
	  • DOUBLE BUMPER
	• PRESSURE COMPARTMENT
	  • 101000 n/m^2 (14.7 psia)

• ELECTRICAL POWER	• EMERGENCY - FUEL CELLS

• ENVIRONMENTAL CONTROL	• EACH INDEPENDENT
	• LEAKAGE
	  • OXYGEN - WATER ELECTROLYSIS
	  • NITROGEN - CRYOGENIC
	• REPRESSURIZATION
	  • OXYGEN - HIGH PRESS
	  • NITROGEN - CRYOGENIC
	• WATER - SABATIER REACTOR
	• CO_2 REMOVAL - MOLECULAR SIEVES
	• THERMAL - WATER AND FREON LOOPS

• LIFE SUPPORT	• URINE AND WASH WATER RECOVERY • DRIED AND FROZEN FOOD • WASTE MANAGEMENT • PERSONAL HYGIENE
• CREW ACCOMMODATIONS	• PERSONAL EQUIPMENT • FURNISHINGS • RECREATION • PHYSICAL FITNESS
• INFORMATION SYSTEM	• COMMUNICATIONS - S BAND • DATA PROCESSING • DISPLAYS AND CONTROLS

Nitrogen to supply leakage and repressurization is stored as a cryogenic. Oxygen for repressurization is stored as a cryogenic while the emergency oxygen system uses high pressure storage. Thermal control of the modules makes use of water and freon loops.

Life Support
Both urine and wash water are recovered. The urine is reprocessed using vapor compression while wash water recovery utilizes reverse osmosis. Dried and frozen food was used. Also included under life support are the waste management and personal hygiene systems.

Crew Accommodations
Included under this category are the personal equipment, furnishings, recreation and physical fitness equipment. Again these systems are located only in the crew quarters.

Information System
The principal systems included are communications, data processing and displays and controls. Each module will have its own internal communication system as well as contact with the main communication center located in the operations module. The principal link between the base and Earth or transportation vehicles is S-band. Each module has data processing capability suitable for its needs. However, again the principal data processing center is located in the Operations module. Each module also has the appropriate set of displays and controls although the Operations module contains all displays and controls associated with overall base operation.

Guidance and Control
Displays and controls for these systems are located in the Operations module although the equipment itself is located throughout the base and consequently are discussed under Base Subsystems.

Reaction Control
Again, this is a base level subsystem and is discussed under Section 3.2.1.1.2.5.

Special Equipment
This is equipment that is peculiar to the maintenance/test/checkout and training/simulation modules.

Mass Estimate
A mass estimate for each of the modules down to the major subsystem is shown in Table 3.2-6. These masses reflect the subsystem design approaches discussed in the previous paragraph. Differences in the subsystem mass for the various modules is reflecting the variation in number of personnel present and consequently power levels as well as the function of the module itself. Included in the mass of each module is a growth/contingency allowance of 33% on the estimated mass.

As indicated, all modules are well within the mass capability of the launch vehicle although in the case of a crew quarters module, the addition of a radiation shelter for GEO application would add another 115000 Kg and consequently be quite close to the mass limit.

Table 3.2-6 Crew Module Mass Summary

o Mass In 10^3kg

SYSTEM	CREW QUARTERS (EA)	OPERATIONS CENTER	MAINTENANCE TEST & C/O	TRAINING & SIMUL.	MISC.
STRUCTURE	80	80	80	80	
ELEC. POWER	5	7	3	7	
ENVIRON. CONT./ LIFE SUPPORT	60	42	24	11	
CREW ACCOMMODATIONS	11	4	3	3	
INFORMATION	6	30	5	0	
GUID & CONT	0	1	0	0	
REACTION CONT	0	0	0	0	
SPECIAL EQUIPMENT	0	0	5	0	
SUBTOTAL	162	164	120	108	110
GROWTH/ CONTINGENCY	53	54	40	35	36
TOTAL DRY	215	218	164	143	146
CONSUMABLES (90 DAYS)	45	0	0	0	0
TOTAL	260	218	164	143	146

3.2.1.1.2.6 Base Subsystems

As indicated previously, several subsystems do not relate specifically to anyone of the crew modules, but instead are associated with operating the base as a total entity. Such subsystems include primary power and flight control.

Electrical Power

Requirements–Basic operating power requirements have been grouped into the categories associated with crew modules, construction equipment and external lighting as shown in Table 3.2-7. The average operating power level required is estimated at over 1600 KW. This load does not include recharging of the secondary power supply or losses.

Under the category of crew module, considerable use was made of the estimates identified for a 12 man space station as defined by Rockwell. These estimates were then scaled up both to account for the difference in crew size and the number of modules involved.

Construction equipment power estimates were made using both Boeing generated data and data from recent space station studies. Typical examples per machine include the 20m beam machine at 5 KW, solar array deployer at 5 KW, crane/manipulator at 3 KW. All of these estimates include the power for a two man control cabin

External lighting estimates are based on providing 216 lumens/m^2 as specified by McDonnell Douglas in the Space Station Systems Analysis Study (NAS9-14958). Typical construction areas in this study covered 2000 m^2 and required 10 KW to provide the specified illumination. A total of 32 areas of this size have been estimated for the SPS construction base.

The total power requirement to be used in sizing the primary power supply is 3725 KW as shown in Table 3.2-8. The secondary power recharging load is for a nickel hydrogen system that produces the operating loads during 37% of the orbit. The allowance for oversizing is that associated with 50 μm cells and 75 μm cover slips. No thermal annealing is assumed.

System Description–The primary power generation system is solar arrays similar to those used in the satellite, with a nickel hydrogen battery system used for occultation periods. An array voltage of 1500 volts has been selected and appears to be the highest practical when considering plasma losses.

The selected installation approach for the array is a fixed body mounted concept, with an array located on three sides of the construction base so that the necessary power can be generated by any one array with the base at any location in orbit. Figure 3.2-48 shown previsouly, illustrates the location of two of these arrays. Each array al[illegible] has been sized to account for sun incidence angle

Table 3.2-7 Base Operating Power Requirements
Photovoltaic Satellite

OPERATING POWER		KW
CREW MODULES		(1175)
ENVIRONMENT CONT/LIFE SUPPORT	750	
INTERNAL LIGHTING	350	
INFORMATION SYSTEM	70	
GUID. & CONT.	5	
CONSTRUCTION EQUIPMENT		(150)
SATELLITE EQUIPMENT	50	
ANTENNA EQUIPMENT	50	
SUBASSEMBLY	50	
EXTERNAL LIGHTING		(320)
SATELLITE CONST.	120	
ANTENNA CONST.	120	
SUBASSY/WAREHOUSE	80	
	TOTAL	1645

Table 3.2-8 Solar Array Sizing

• REQUIREMENTS (KW)		(3645)
• OPERATING LOAD		1645
• SECONDARY POWER SUPPLY RECHARGING		960
• POWER CONDITIONING		330
• POWER DISTRIBUTION		540
• RADIATION DEGRADATION (5%)		170
• SIZING		
• CONTINUOUSLY SUN ORIENTED ARRAY: (SATELLITE TYPE CELLS, 140 w/m^2)	26000 m^2	
• FIXED BODY MOUNTED ARRAY WITH EARTH ORIENTED CONST. BASE		
• ARRAYS ON 3 SIDES OF BASE		
• MAX SUN INCIDENCE ANGLE OF 54.5 DEG		
• TOTAL ARRAY SIZE:	≈130000 m^2 205m x 205m PER SIDE	

penalties so the combined net affect is a total array that is approximately five times as large as an array that was always PEP. By past space system standards, this excess would be prohibitive. However, in the era of power satellite with low mass and low cost cells, the penalty is quite small.

An alternate solar array installation approach was also considered in the form of providing the array with two axis control so it could always be directed toward the sun. With the physical dimensions of the base however, the boom to which the solar array would be attached had to extend approximately 600m from the base. The operational disadvantages of this approach in terms of the impact on vehicle traffic as well as the relatively small savings and additional complication of a large two axis system resulted in not selecting this approach for the reference system.

The mass for the selected power system is estimated at approximately 120000 Kg as indicated in Table 3.2-9.

Table 3.2-9. Primary Power System Mass

Fixed Array	65000 Kg
N_iH_2 Batteries	20000 Kg
Distribution (Power Bus)	20000 Kg
Switchgear	2000 Kg
DC-DC Conversion	2000 Kg
Radiator (for N_iH_2 System)	1000 Kg
Structure	9000 Kg
	119000 Kg

Flight Control

Included under the category of flight control are the guidance navigation attitude type sensors such as IRU, star trackers and horizon sensors and the propulsion system to perform attitude and orbit maintenance maneuvers.

A LO_2 LH_2 propulsion system has been selected to provide attitude control of the base rather than CMG's due to the large disturbances involved and the relatively easy pointing requirements (between ±1 deg and ±5 deg). An Earth oriented attitude has been selected which has the satellite being constructed toward the center of the Earth.

Orbit maintenance in terms of maintaining a fairly constant altitude 477 Km ± 1 Km obviously requires the use of a propulsion system.

The selected propulsion system uses LO_2 LH_2 and for an acceleration level of 10^{-4} g's (satellite design conditions), a total thrust of 27000N is required for moving the combined mass of the base.

one module and one antenna. Distribution of the thrusters around the base will occur to minimize the impact on the structure. The average orbit decay rate is estimated at 1 Km/day. Orbit makeup is performed daily with the average propellant requirement being 800 Kg/day.

The propulsion system mass including 90 days of propellant is estimated at 80000 Kg.

3.2.1.1.3 Environmental Factors

The principal environmental factors that influence the design of the construction base or its operations include radiation, meteoroids, occultations, gravity gradient and drag and collision with manmade objects. All of these factors have been discussed in prior paragraphs, except collision Consequently, the previously discussed items will be summarized while collision will be discussed in more detail.

Radiation

Radiation effects on personnel at the LEO construction base are primarily in the area of EVA activity since the 3 gm/cm^2 wall density in the manned compartments is more than adequate to allow the 90 day stay times. Although a bare minimum of EVA activity is anticipated, should it occur, it most likely would be restricted during passages through the South Atlantic anomaly.

Meteoroids

Protection against meteoroids is provided by the double wall bumper used around all manned habitats.

Occultations

The principal impact of the occultations of the base which occur 15 times per day are in the areas of electric power supply and thermal aspects of the structure. In the case of the impact on the electric power supply, it means sizing the primary system so that the system used during occultation can be recharged. The penalty for the larger power system is relatively small however, in the era of low mass, low cost solar arrays. Use of graphite/epoxy structure in both the satellite and construction base structure should minimize the impact of thermal effects.

Gravity Gradient and Drag

Most construction concepts will orient the construction base so it is passively stable for attitude control and minimize gravity gradient torque. Although the LEO construction base required considerably more orbit keeping/attitude control propellant per day, it still results in less than one HLLV launch per year for this propellant makeup.

Collision

Large amounts of debris from manmade space systems have resulted in some concern regarding LEO construction. The approach used in establishing the number of potential collisions and an initial estimate was presented in the Part I Final Documentation Volumes III and V (D180-20689-3 and

-5, respectively). The number of potential collisions has been since updated as well as the identification of methods which can be used to avoid collision. Prior to this discussion however, it should be noted that the problem of collision involves both the construction phase and the transfer phase from LEO to GEO. Rather than split this discussion between construction and transportation, it will all be discussed as one subject at this time.

The number of collisions expected on a total satellite during LEO construction, transfer to GEO and thirty years of operating life at GEO are shown in Figure 3.2-53. The most dominating portion of the flight is that associated with construction phase and the transfer through the first 3000 kilometers, where 17 and 22 collisions are estimated, respectively. The assumptions used in this estimate are shown, with key importance given to the use of the object model as of the year 2000 including 500 objects added per year since 1975. The other key point to be considered with this data is that no attempt was made at cleaning up the debris or taking avoidance action to prevent collision. The method used to prevent collisions is presented in the following paragraphs.

The method used to eliminate or considerably reduce the number of collisions during construction simply involves a rescheduling of the orbit trim (drag makeup) maneuver of the construction base. A simplified diagram of this operation is illustrated in Figure 3.2-54. For general planning purposes, the construction base has a nominal position. The actual position of the base relative to the nominal position is shown at the completion of each of the 15 revolutions (1 day) around the Earth. At the completion of the 15th revolution, an orbit trim maneuver is performed and the gradual decay begins again. Collision avoidance operations take place in the following manner. At a given revolution (such as number 4) it is determined that on revolution 5, the construction base/satellite will be hit by an object (approaching perpendicular to orbit track) if no corrective action is taken. At that time, however, an unscheduled orbit trim maneuver will be initiated which will increase the altitude of the base and as such results in lower orbital velocity, and on a relative position basis, puts the construction base at a new position for revolution 5, which is approximately 7 kilometers downtrack from the original position of revolution 5, and consequently should eliminate the possible collision. The key factor in this avoidance operation is a need for approximately 1 revolution of warning time, that could be obtained from both on-orbit and ground tracking systems.

Objects coming into the construction base, along a more tangential path can also be avoided using a similar technique, but requiring a greater change in altitude and consequently more propellant. This operation is shown in Figure 3.2-55. For example, a change of 6 kilometers in altitude requires approximately 16,000 kilograms of propellant. Since the large change in altitude also results in excessively large changes in along track position, a deorbit maneuver is also required (16,000 kilograms of propellant), thus bringing the construction base back to its nominal position.

Avoidance during transfer can also be accomplished in a similar manner with approximately 1 revolution of warning time. In this case, however, should a collision be predicted for the satellite

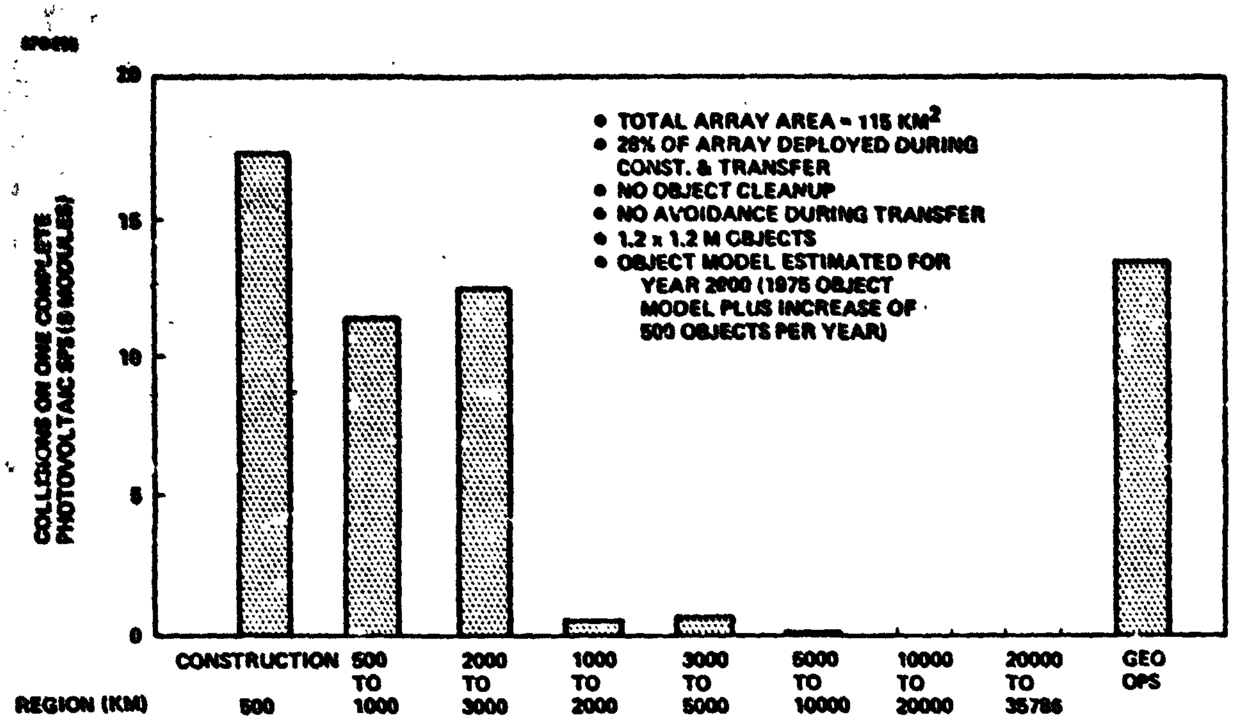


Figure 3.2-53 Number of Collisions Photovoltaic CR=1 Satellite

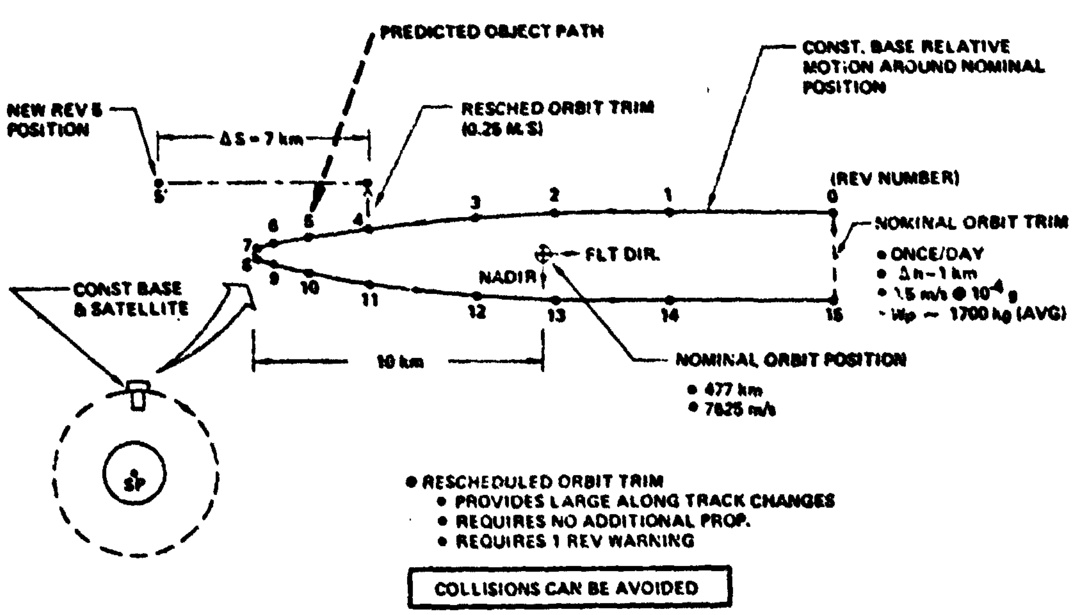


Figure 3.2-54 Collision Avoidance During Construction Perpendicular Objects

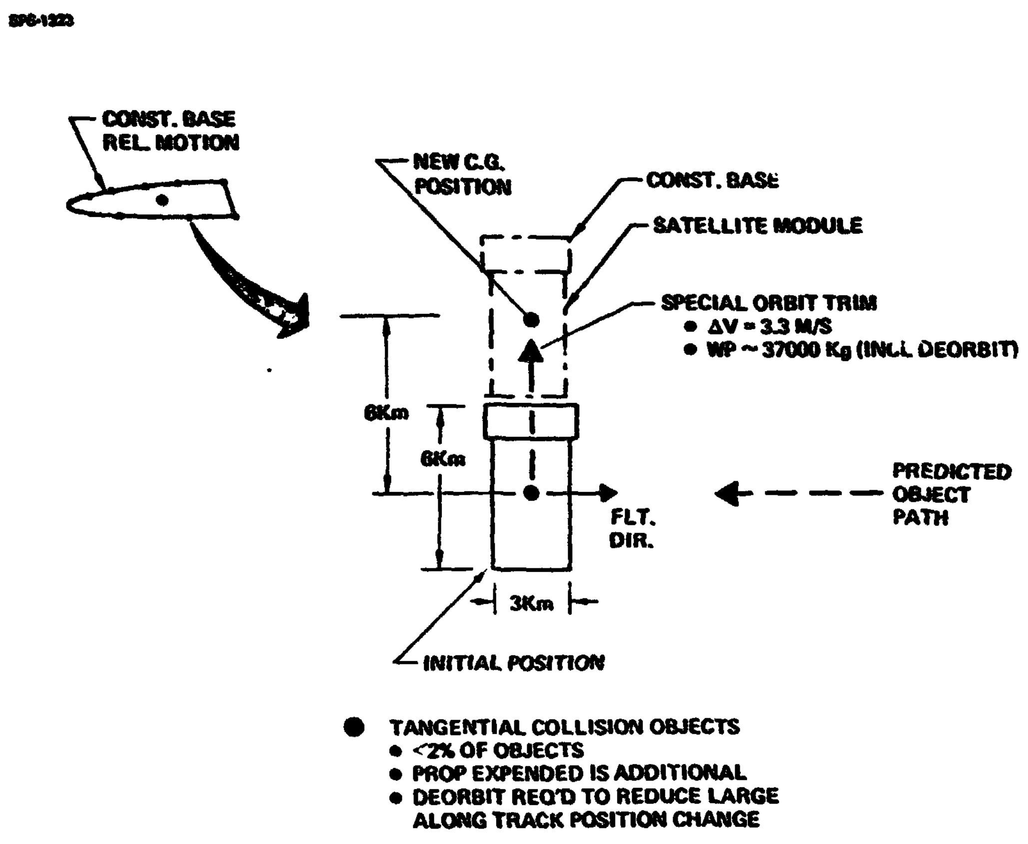


Figure 3.2-55 Collision Avoidance During Construction Along Track Objects

module, at a point in its next revolution, the transfer thrust can merely be terminated for a time period, thus placing the satellite behind its previously anticipated position on the next revolution and consequently eliminate or avoid the collision.

In summary, the described operational procedures appear to offer an approach to reduce the number of collisions to zero with a minimum of penalty.

3.2.1.1.4 Crew Operations

This section addresses the following crew-related topics: scheduling, productivity, and organization. The crew schedule and productivity factors will be applied in all other crew sizing estimates.

3.2.1.1.4.1 Crew Schedule

The crew scheduling concept that was used in all of the construction task timelines, crew sizing, and crew transportation analyses was the following:

90 Day Staytime
6 Days On/1 Day Off Per Week
10 Hrs Per Day Work Shift Using a 5/1/5/13 Work Rest Cycle
2 Shifts Per Day (2 Crews)

The selection of this scheduling concept was described in Section 3.4.2 of the Part I Final Report.

3.2.1.1.4.2 Operator Productivity

When considering the amount of work time per day, it is necessary to take into account the fact that operators do not work at 100% of their capacity throughout a work shift. It is necessary to take into account operator fatigue, delays and personal factors.

The data shown in Figures 3.2-56 and 3.2-57 is based on Boeing manufacturing experience data. For the purposes of establishing machine operating rates, a productivity factor of 75% over a 10 hour work shift was applied. A typical application of this productivity factor is shown below:

$$\begin{pmatrix}\text{32 Days Available}\\ \text{Per Module}\end{pmatrix}\begin{pmatrix}\text{20 Hours}\\ \text{Per Day}\end{pmatrix}^{\triangleright 1}(.75) = \begin{matrix}\text{480 Hours Available}\\ \text{Per Module}\end{matrix}$$

$$\triangleright 1 \quad \begin{pmatrix}\text{2 Shifts/}\\ \text{Day}\end{pmatrix}\begin{pmatrix}\text{10 Hours/}\\ \text{Shift}\end{pmatrix} = \begin{matrix}\text{20 Hours/}\\ \text{Day}\end{matrix}$$

3.2.1.1.4.3 LEO Base Crew Organization

The operators and associated personnel have been combined into the organizational structure shown in Figures 3.2-58 through 3.2-67. A total of 478 people will be at the LEO base.

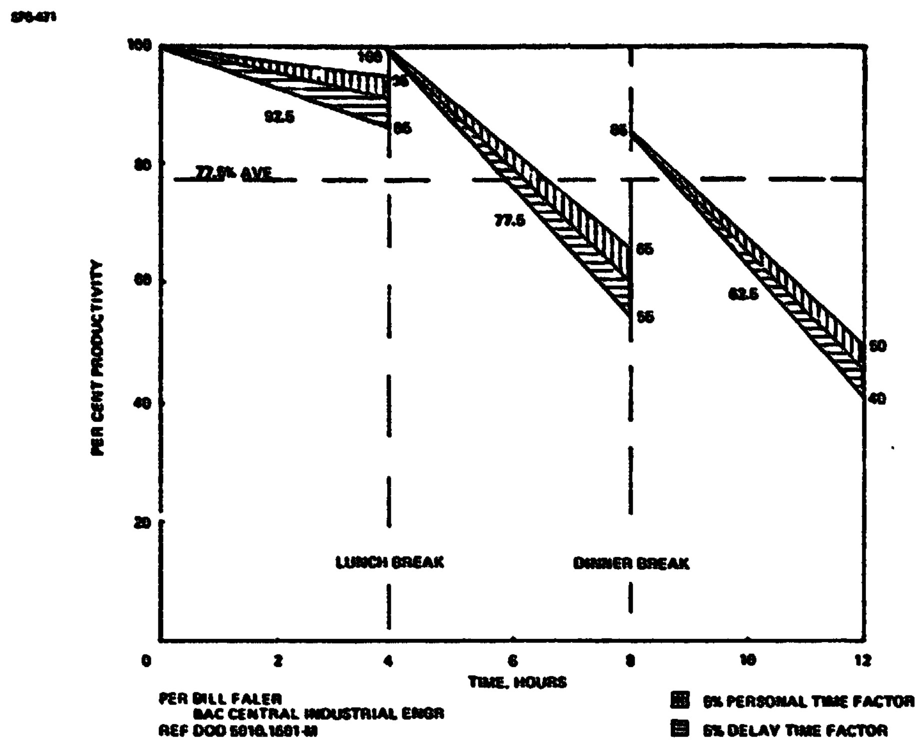


Figure 3.2-56 Machine Operators

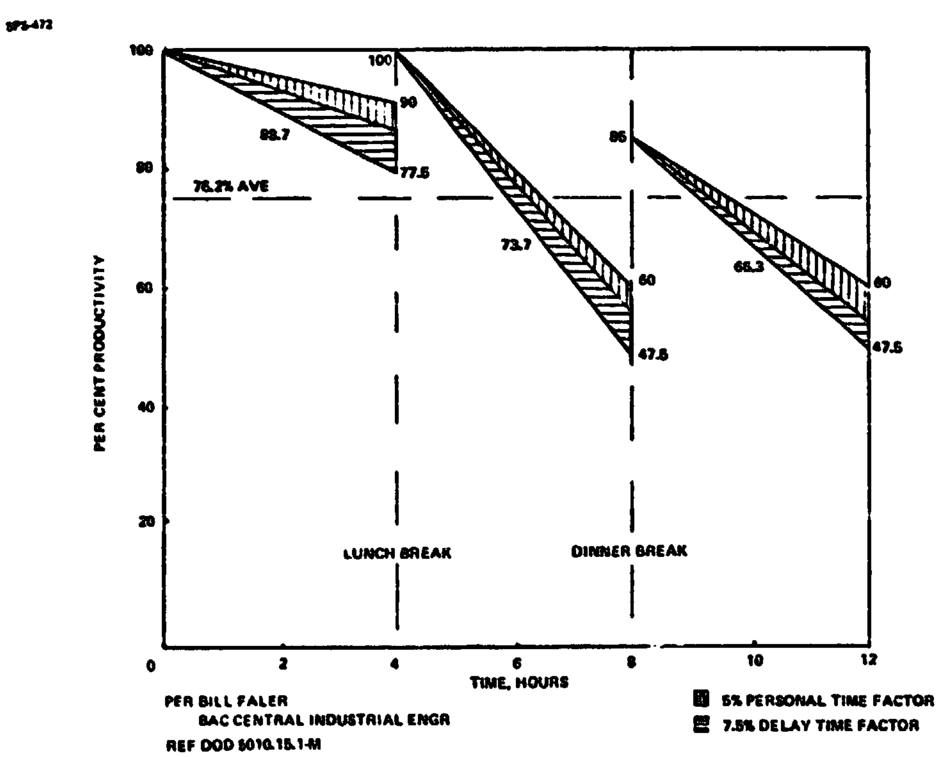
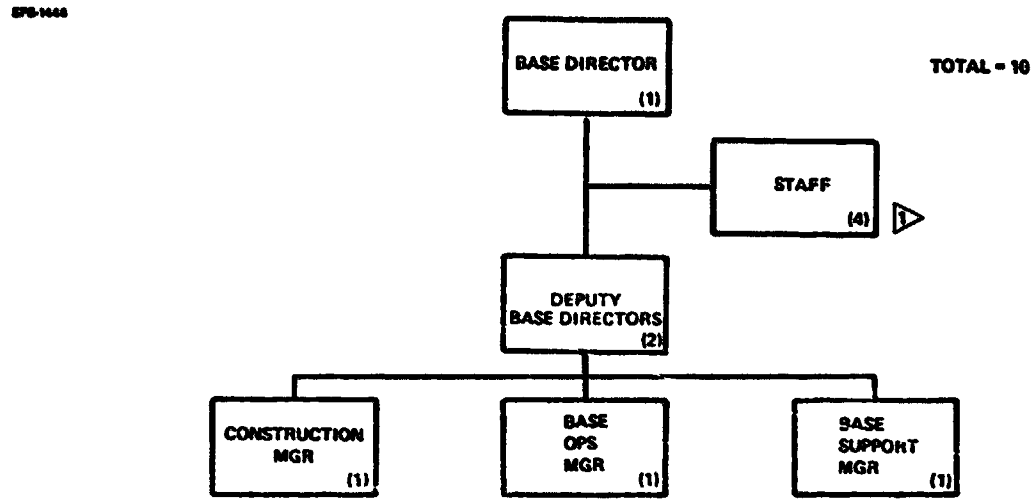
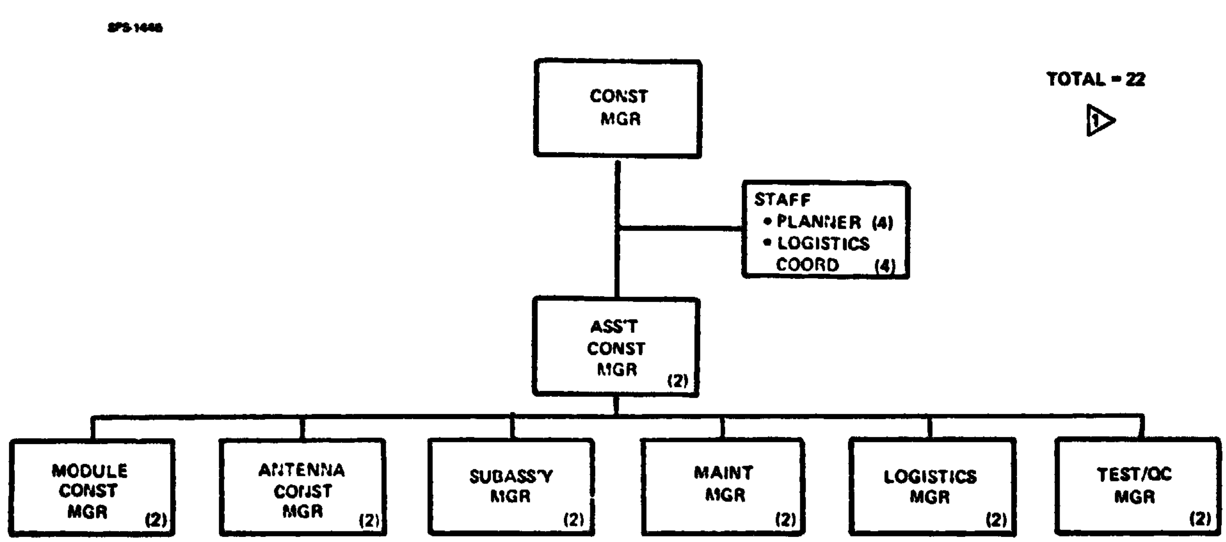


Figure 3.2-57 Assembly Workers



1 NUMBERS IN ( ) INDICATE THE NUMBER OF CREW MEMBERS REQUIRED TO STAFF THE INDICATED JOB OVER 2 SHIFTS

Figure 3.2-58 LEO Base Personnel



1 IN ALL THIS AND ALL OTHER ORGANIZATION CHARTS, A MANAGER WHO WAS COUNTED IN A HIGHER LEVEL ORGANIZATION CHART IS NOT INCLUDED IN THE TOTAL SHOWN HERE

Figure 3.2-59 LEO Base Personnel (Continued)

SPS-1488

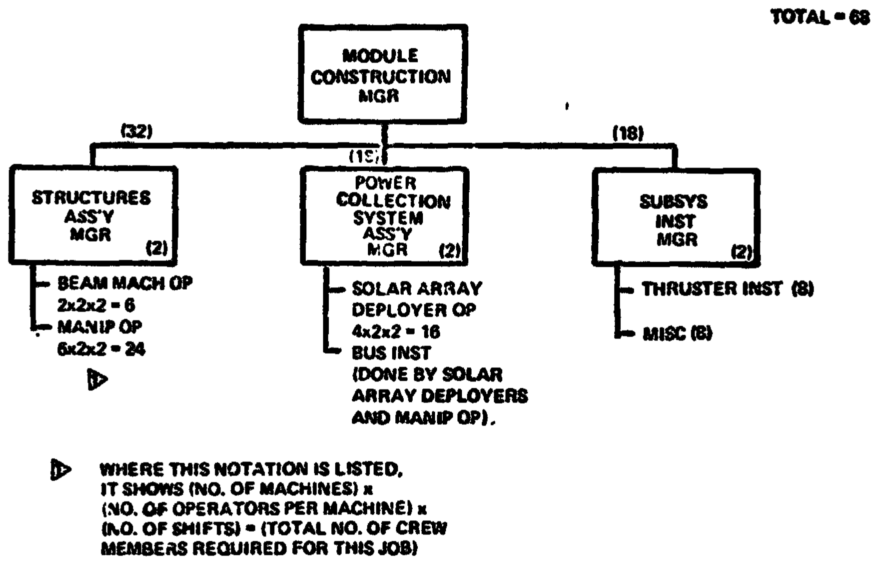


Figure 3.2-60 LEO Base Personnel (Continued)

SPS-1573

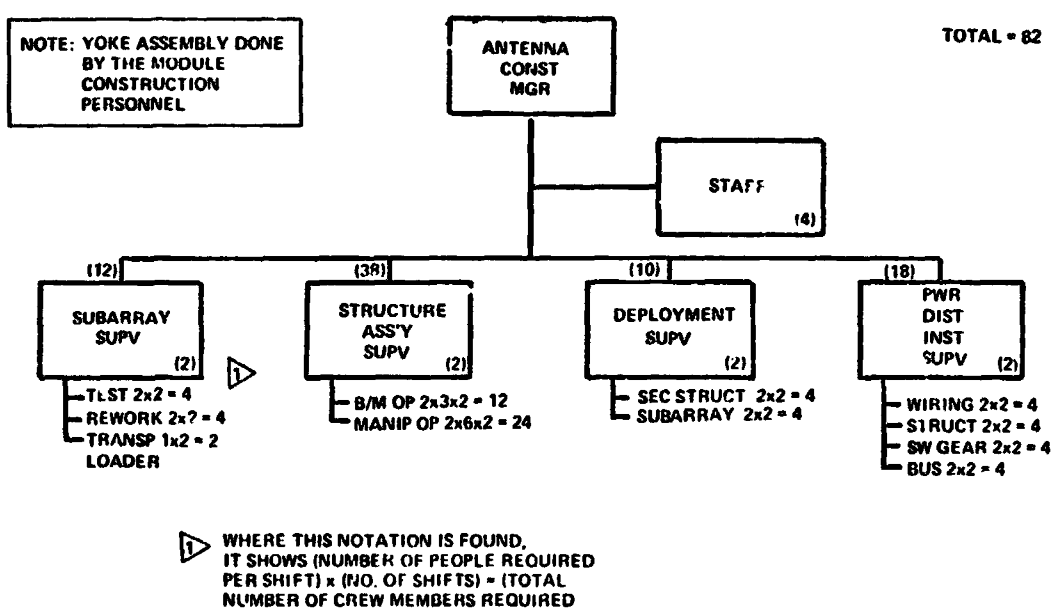


Figure 3.2-61 LEO Base Personnel (Continued)

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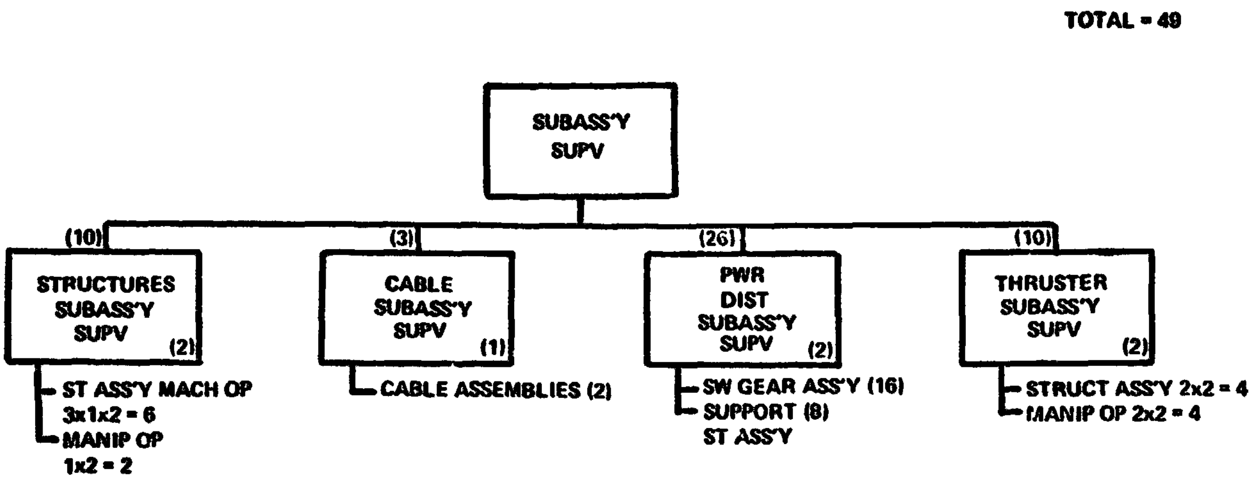

Figure 3.2-62 LEO Base Personnel (Continued)

SPS-1468

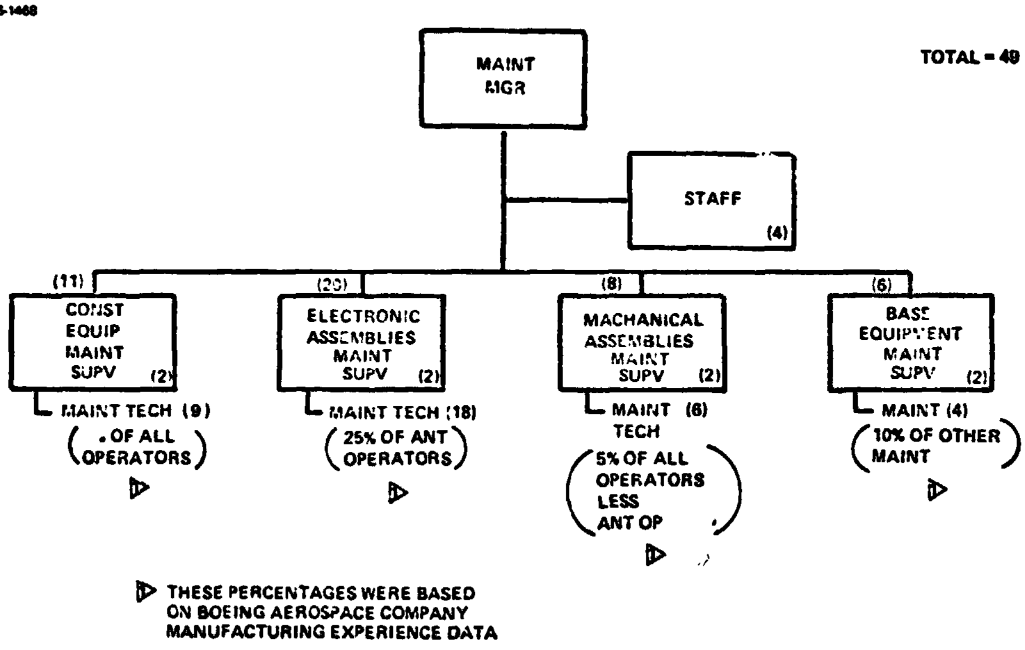

Figure 3.2-63 LEO Base Personnel (Continued)

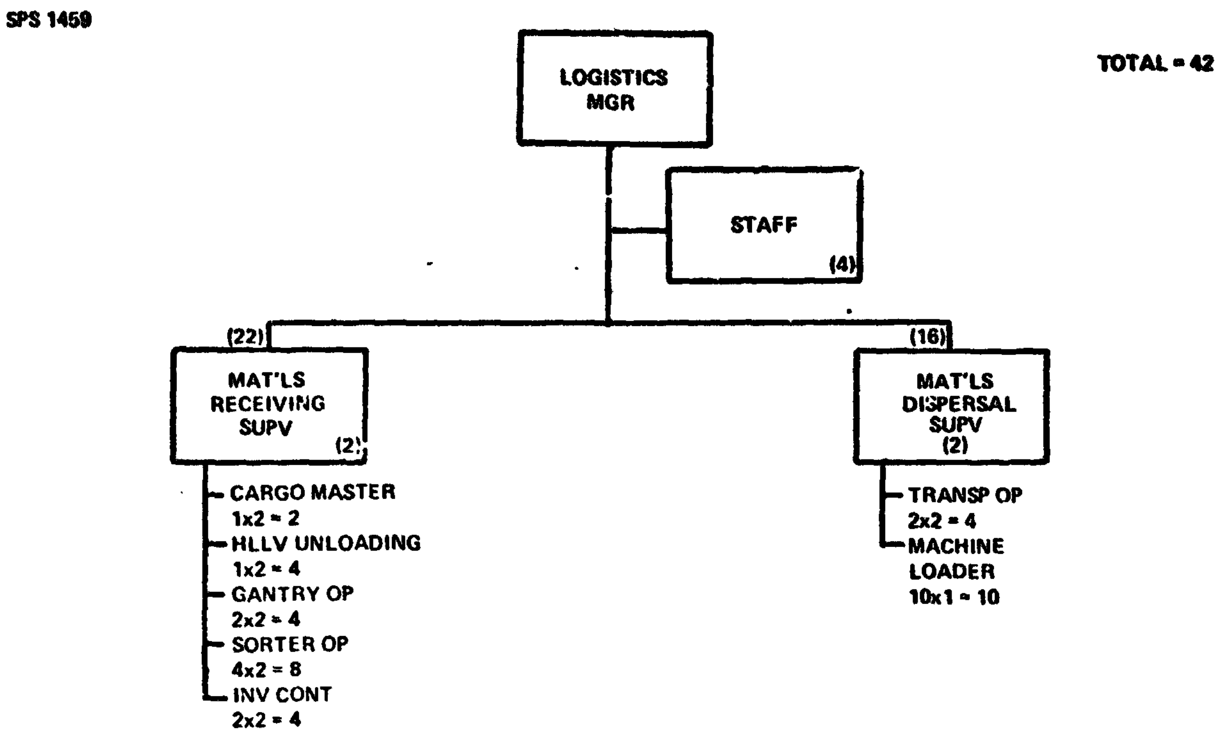

Figure 3.2-64 LEO Base Personnel (Continued)

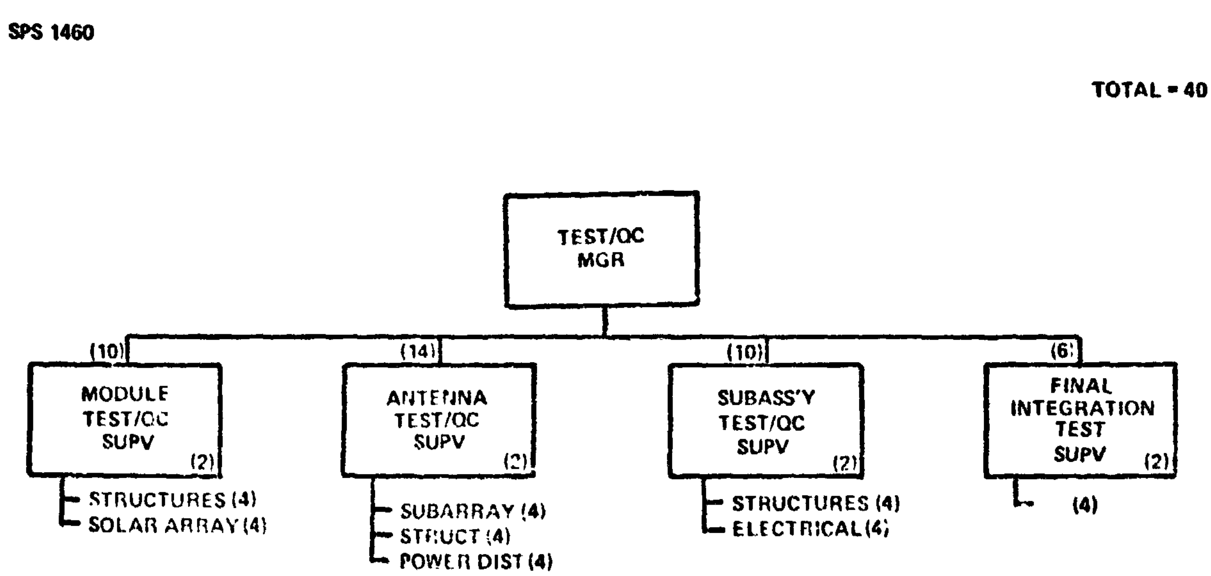

Figure 3.2-65 LEO Base Personnel (Continued)

SPS 1452

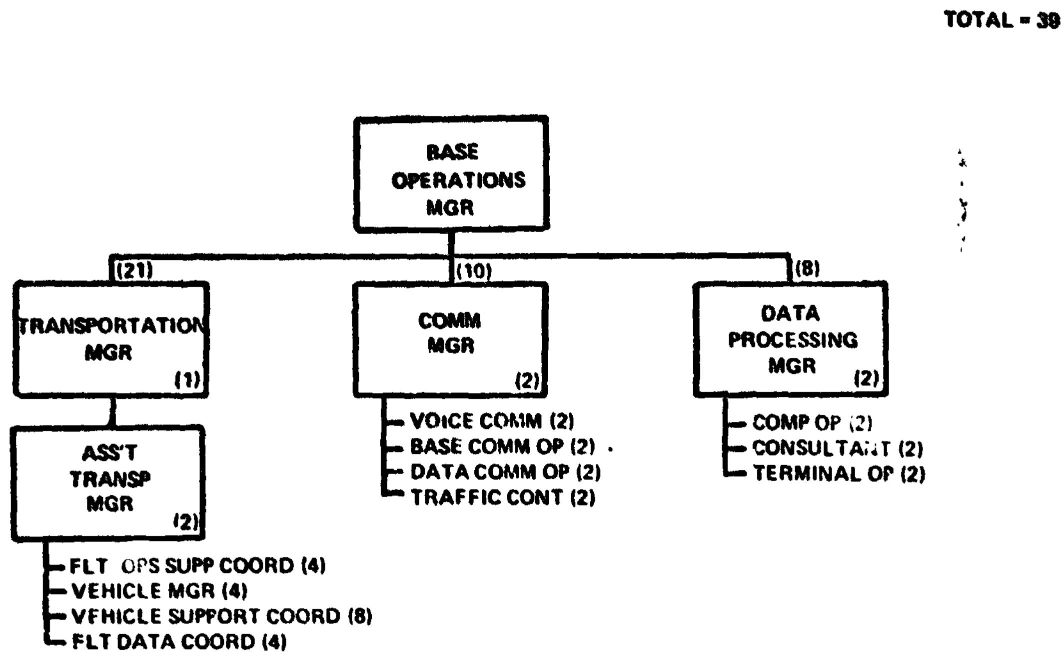

Figure 3.2-66 LEO Base Personnel (Continued)

SPS 1461

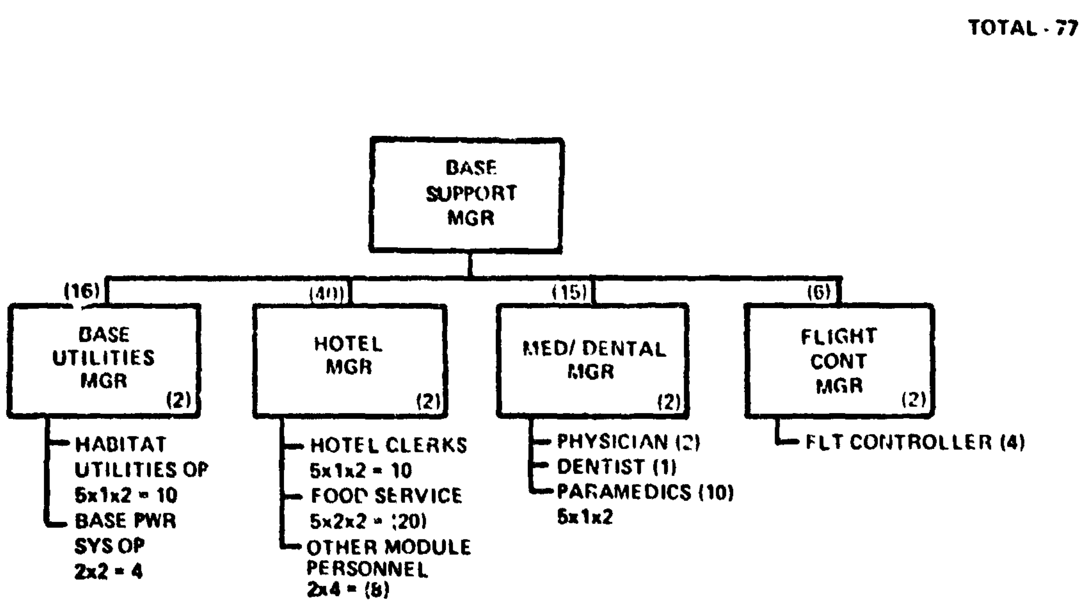

Figure 3.2-67 LEO Base Personnel (Continued)

3.2.1.1.5 **Mass Summary**

A mass summary for a complete construction base capable of constructing one satellite per year is presented in Table 3.2-10. The dry mass of the base is 5.6 million Kg and with 90 days of consumable included the total mass becomes 5.87 million Kg. The largest contribution to the mass is the foundation (structure) and the ten crew modules.

3.2.1.1.6 **Cost Summary**

The cost summary for the construction base is presented in Table 3.2-11. Basic hardware costs are estimated at approximately $4.8 billion. An additional $2.2 billion also exists in the form of various wraparound cost items. The indicated costs were developed using mass statements and unit quantities developed by Boeing and cost CER's of JSC and Boeing. A learning factor of 0.9 was applied to all systems having more than four basic units.

3.2.1.2 **GEO Final Assembly Base Construction Analysis**

The following construction data pertains to the GEO final assembly base used in the LEO construction concept.

3.2.1.2.1 **GEO Construction Operations**

3.2.1.2.1.1 **Top-Level GEO Construction Tasks**

The construction tasks to be accomplished at the GEO base are summarized in Figure 3.2-68. Each of the tasks are discussed in the following sections.

3.2.1.2.1.2 **GEO Final Assembly Base**

The GEO base shown in Figure 3.2-68 consists of one platform 1600m x 1400m that supports all of the final assembly and deployment equipment. A more complete discussion of the base is presented in Section 3.2.1.2.2.

3.2.1.2.1.3 **Detailed Construction Task Analysis**

3.2.1.2.1.3.1 **Module Berthing (Docking)**

The first task to be performed after modules reach GEO is that of their berthing (docking) to form the complete satellite. The concept employed to perform this operation is illustrated in Figure 3.2-69.

Four docking systems are used with each involving a crane and three control cables. Tension applied to the cables allows the modules to be pulled in, provides stopping control and attitude capability. Also required in this concept is an attitude control system including thrusters which are not shown. Additional characteristics of the docking crane are shown in Figure 3.2-70.

The major docking operations associated with the modules at GEO are illustrated in Figures 3.2-71 and 3.2-72. The initial step at T = 0 hours has the N+1 module having arrived in the near vicinity of module N which is already at GEO. Module N+1 is at a position below module N and

Table 3.2-10 LEO Construction Base ROM Mass

		10^3kg	
FACILITY		(5200)	
FOUNDATION		2500	
CREW MODULES		2000	1
CARGO HANDLING/DISTRIBUTION		400	
BASE SUBSYSTEMS		200	
MAINTENANCE PROVISIONS		100	
CONSTRUCTION AND SUPPORT EQUIPMENT		(400)	
STRUCTURAL ASSEMBLY		80	
ENERGY COLLECTION/CONVERSION INSTALL.		60	
POWER DISTRIBUTION INSTALL.		20	
ANTENNA SUBARRAY/SEC. STRUCT INSTALL.		30	
CRANES/MANIPULATORS		180	
INDEXERS		30	
	DRY TOTAL	(5600)	
CONSUMABLES (90 DAYS)		(270)	
	TOTAL	(5870)	

1 INCLUDES 33% GROWTH ALLOWANCE.
OTHER ITEMS DO NOT INCL. GROWTH.

Table 3.2-11 LEO Construction Base ROM Cost

			$\$10^6$
FACILITY			(3465)
FOUNDATION		360	
CREW MODULES		2870	
CARGO HANDLING/DISTRIBUTION		330	
BASE SUBSYSTEM		15	
MAINTENANCE PROVISIONS		–	
CONSTRUCTION AND SUPPORT EQUIPMENT			(1310)
STRUCTURAL ASSEMBLY		350	
ENERGY COLLECTION CONVERSION INSTALL.		165	
POWER DISTRIBUTION		75	
SUBARARY INSTALL		80	
CRANES/MANIPULATORS		560	
INDEXERS		80	
	BASIC HARDWARE		(4775)
SPARES (15%) [1]			715
INSTALL, ASSY, C/O (16%)			765
SE & I (7%)			335
PROJ MGT (2%)			95
SYS TEST (3%)			145
GSE (4%)			190
		TOTAL	(7020)

[1] % OF BASIC HARDWARE

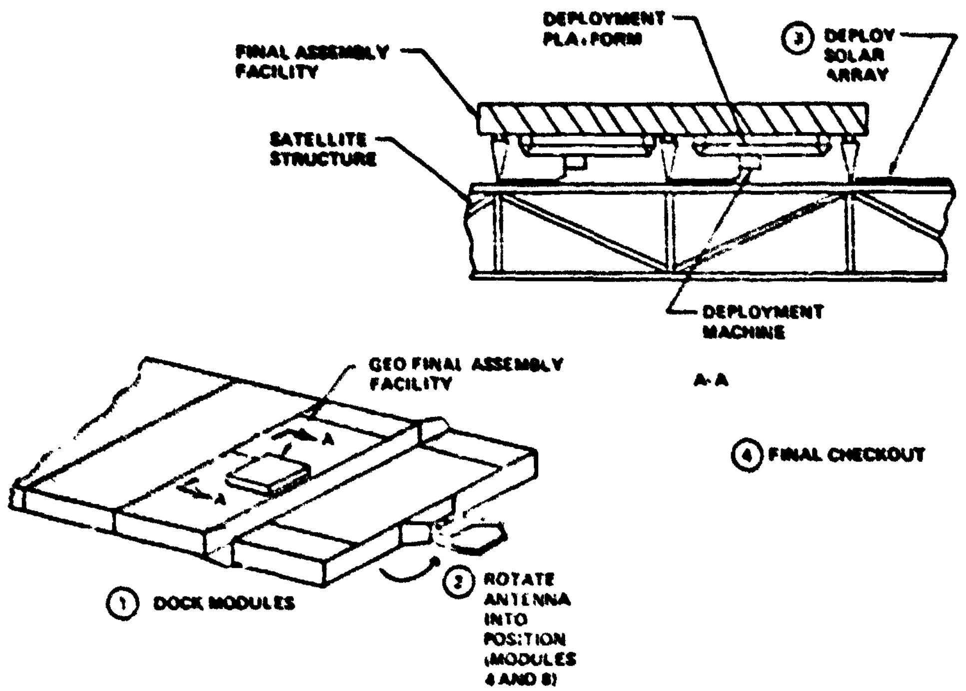

Figure 3.2-68 GEO Base Construction Tasks
LEO Construction Concept

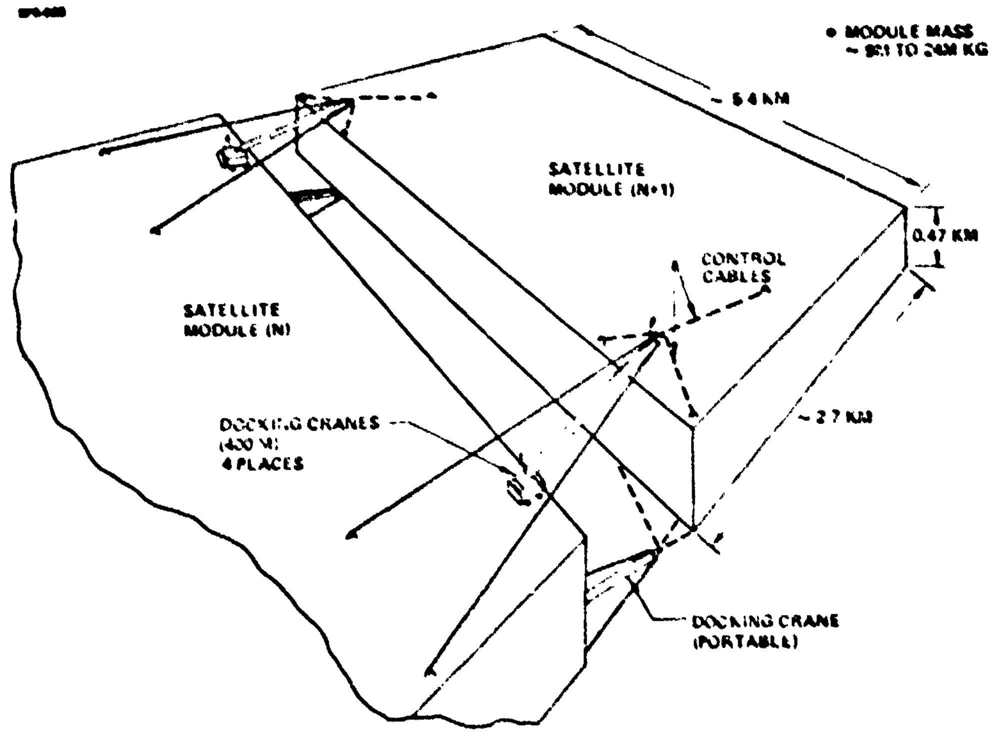

Figure 3.2-69 Berthing Concept
Photovoltaic Satellite

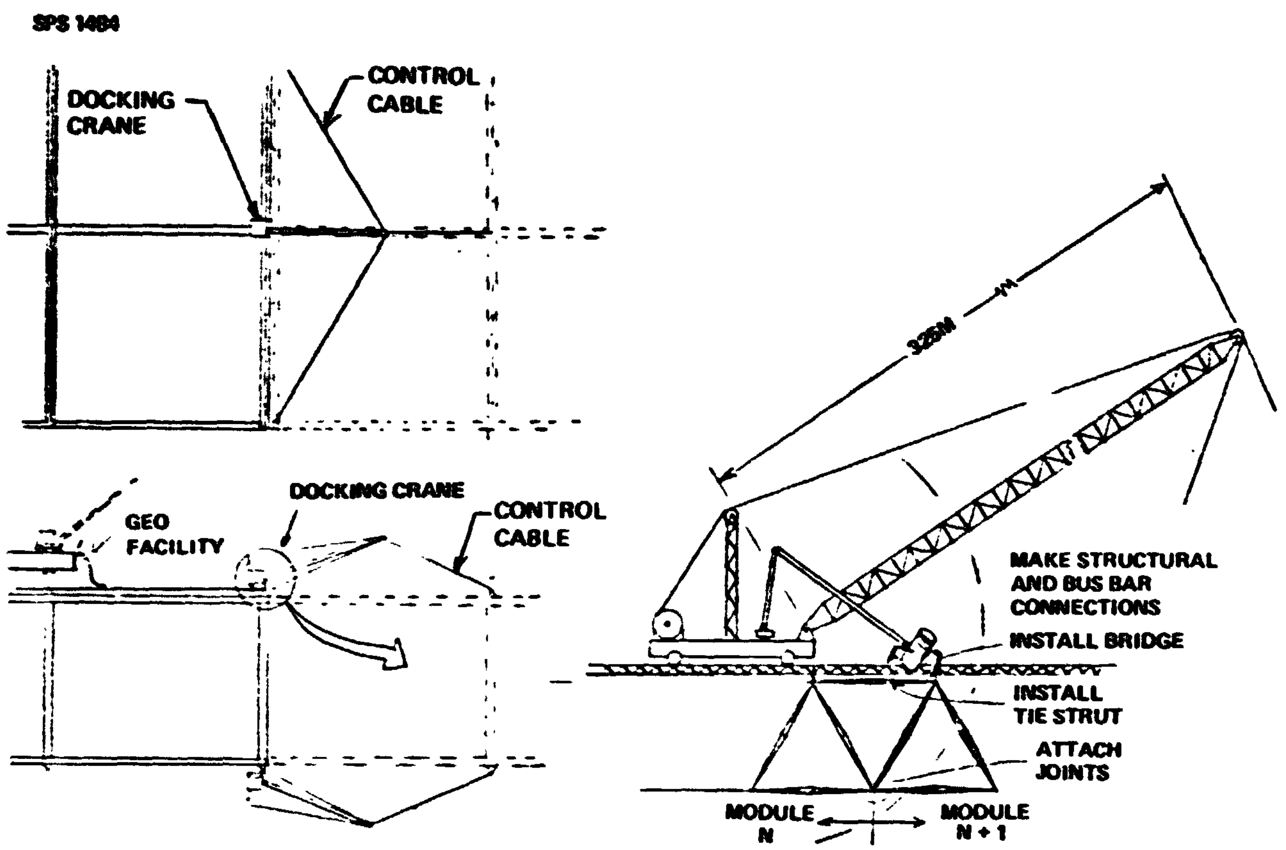

Figure 3.2-70 Docking Crane

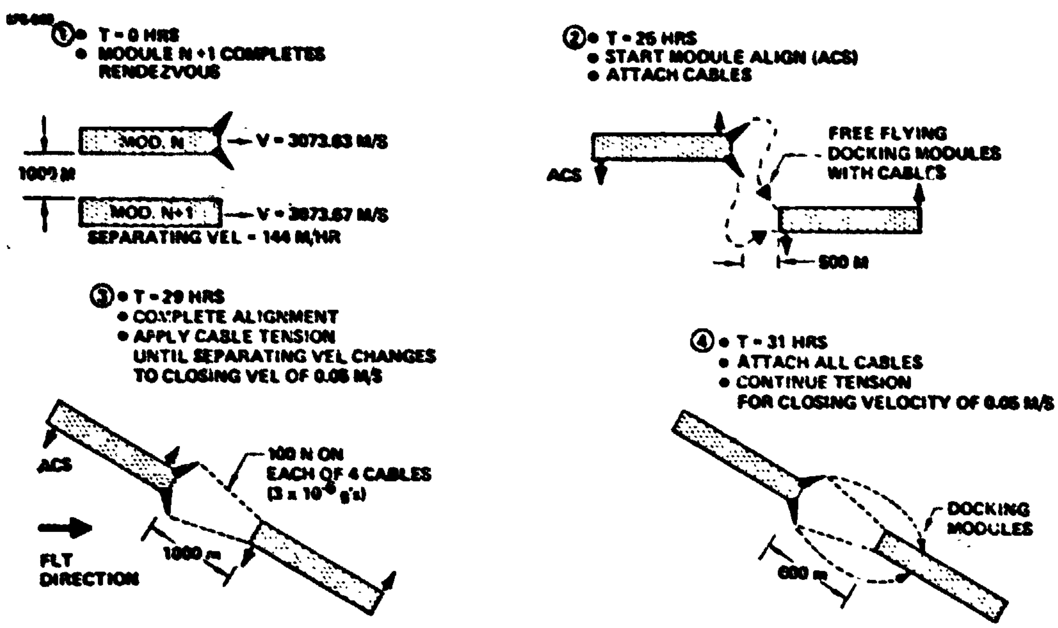

Figure 3.2-71 GEO Docking Operations

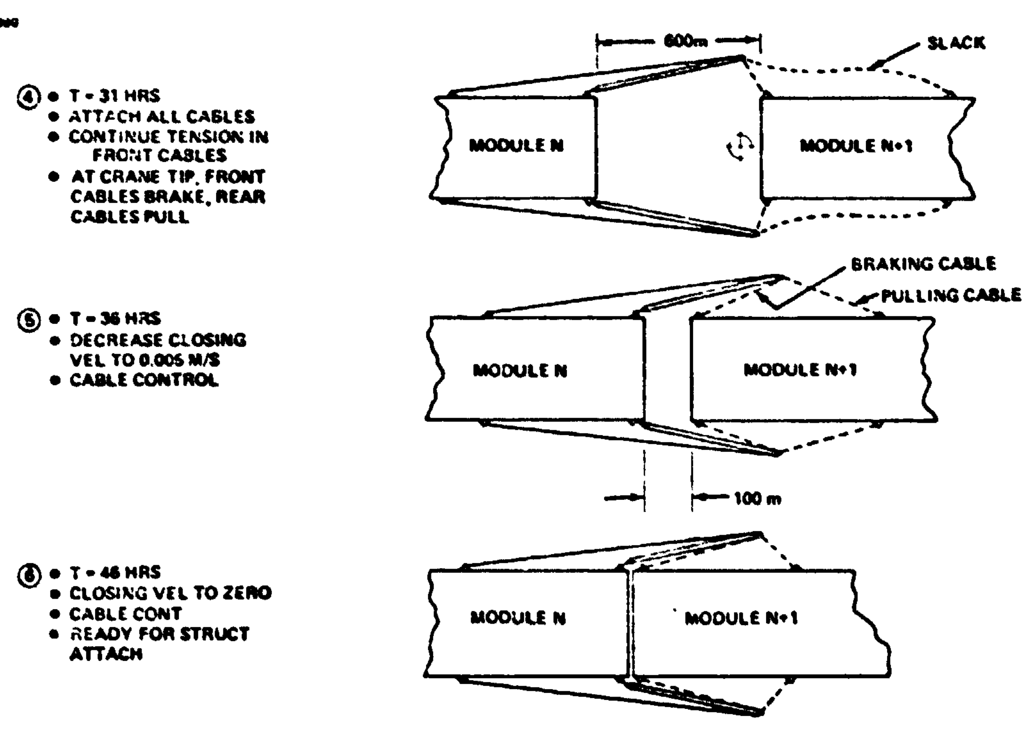

Figure 3.2-72 GEO Docking Operations

consequently has a greater orbital velocity and for the specified altitude difference, a separating velocity of approximately 0.04 meter per second (144 meters per hour). Step 2 illustrates module N+1 having moved ahead of module N approximately 500 meters due to the separating velocities. At this point, free flying docking modules carry cables to module N+1 for the initial attachment. Also, realignment of the attitude of the two modules occurs to simplify the docking operation. In Step 3 and 29 hours after the operation has begun, alignment has been completed and a tension of 100 Newtons has been applied to each of the four lines so that the separating velocity has been changed to a closing velocity of 0.05 meter per second. Tension applied over this time period results in loads of 3×10^{-6} g's. At T = 31 hours, the remaining cables are attached again using docking modules, however, these cables remain slack. Tension of 100 Newtons is applied until a separation distance of approximately 600 meters is reached.

The 4th major step, T = 31 hours, is again illustrated but with additional detail. The key point to be made is that the four cables remain under tension and pull the modules together until the leading edge of the module N+1 passes the end of the crane extending from module N. At this point, the front cables switch to serve as a braking device and the back cables come under tension and begin to pull the module together. At T = 36 hours, the modules have closed to 100 meters of separation and the closing velocity is decreased to 0.005 meter per second. Finally, at T = 46 hours, the closing velocity has been reduced to 0 and the modules are ready for structural attachment.

3.2.1.2.1.3.2 Module Attachment

After the modules are docked, the adjacent edges of the modules are interlocked using strut inserts that are put into position by the crane/manipulators that are carried by the docking cranes.

The bus bar connections between modules are accomplished by a 110m crane/manipulator that is maneuvered on the facility track system.

3.2.1.2.1.3.3 Solar Array Deployment

The solar array blanket packages that are attached to the edges of bays are deployed by four solar array deployment machines that operate in parallel. The solar array deployment machines used at GEO do not have the blanket magazine or blanket installation mechanisms required by the LEO solar array machines so consequently only one operator per machine is allocated.

The solar array that was deployed to provide self-transport power will be annealed by the annealing system previously installed at LEO.

3.2.1.2.1.3.4 Antenna Installation

On modules 4 and 8, the antenna/yoke assembly is transported to GEO by being hinged under the module as shown in Figure 3.2-73. At GEO, the antenna is rotated up to its proper location using electric motor actuators that were used to initially hinge the assembly underneath the module for

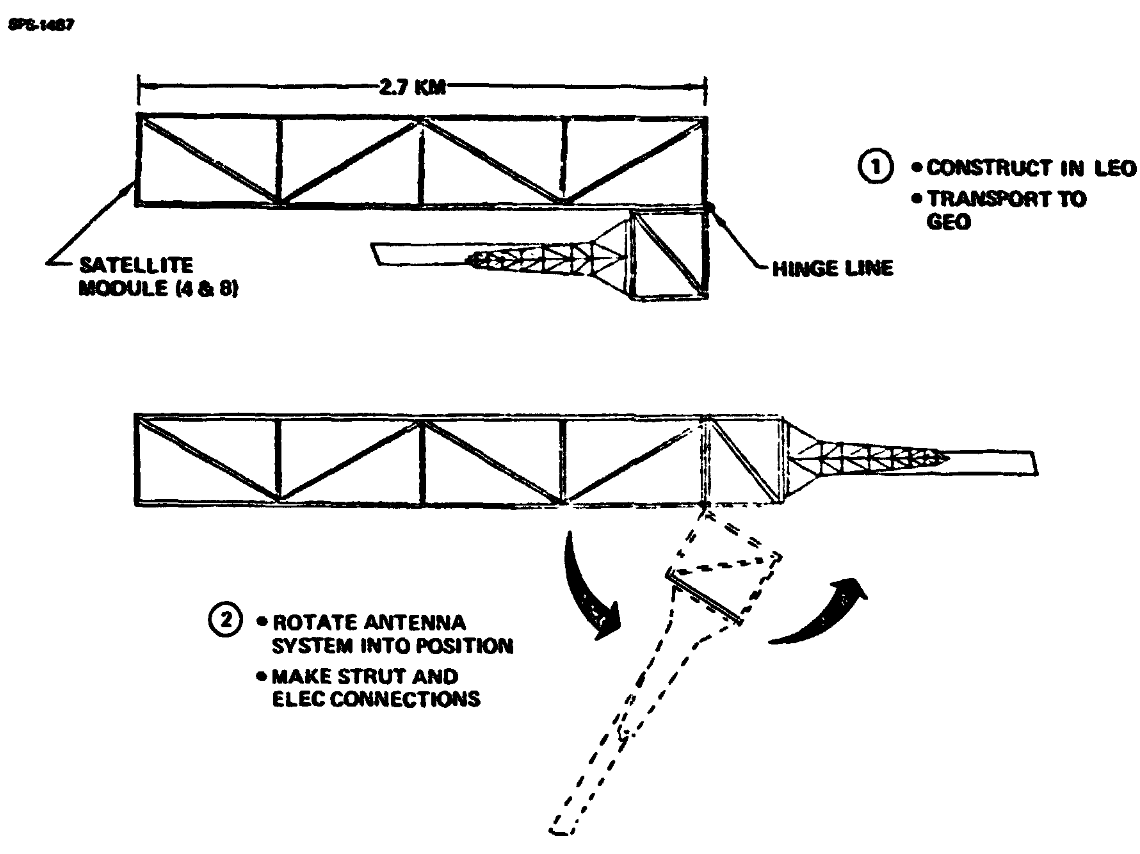

Figure 3.2-73 Antenna Final Installation
Photovoltaic Satellite

transport. Once the antenna/yoke assembly is in position, structural ties are made between the yoke/module interface and the power busses are spliced. The structural and power bus interfaces are accomplished by 110m crane/manipulator operating from the facility.

3.2.1.2.1.4 Equipment Summary

The GEO base construction equipment is summarized in Table 3.2-12. Spares are not included in this summary.

3.2.1.2.2 GEO Final Assembly Base Description

3.2.1.2.2.1 Configuration

The overall configuration of the GEO final assembly base is shown in Figure 3.2-74. The base has overall dimensions of 1400m x 1600m x 100m with two decks of operation. The upper deck supports the crew and maintenance modules and docking facilities for transportation systems and payloads. The lower surface of the facility supports the four solar array deployment machines. Docking cranes used in berthing the modules are also attached to the base when not in use or when the GEO base is transferred to another longitudinal location.

3.2.1.2.2.2 Foundation

The foundation (structure) of the base has been sized to provide a natural frequency of 50 cphr which is greater than that of a single satellite module. The primary structure consists of 20m beams forming a grid pattern for both the upper and lower surfaces of the base. A total beam length of 55,000m has been estimated.

3.2.1.2.2.3 Cargo Handling and Distribution

Movement of satellite components from central receiving to the module will not be required except in the case of replacing items that may have been damaged during the transfer from LEO or during the berthing of the modules. A logistic network and personnel bus is necessary, however, to move the crews from the living quarters to such remote work stations as the solar array deployers. A listing of all the cargo handling and distribution equipment is presented in Table 3.2-13.

3.2.1.2.2.4 Crew Modules

The GEO base has a crew size of 65 and only a minimum of construction operations so consequently, all functions can be incorporated into a single crew module. The module is similar in design to the crew quarters modules used at the LEO construction base. The major modifications to the LEO modules are as follows: 1) incorporation of an operations deck in place of one of the three personnel decks since only 65 rather than 100 people are housed in the module, 2) add an eighth deck which serves as a solar flare radiation shelter. Assuming a shielding requirement of 20 to 25 gm/cm^2, the shelter will add an additional 115,000 Kg to the basic module mass. Within the shelter will be provisions for up to five days and controls to operate the complete base on a standby status

Table 3.2-12 GEO Base Construction Equipment Photovoltaic Satellite

SPS 1481

EQUIPMENT ITEM	NUMBER REQ'D	EQUIPMENT ITEM MAJOR ELEMENTS	NO. REQ'D ITEM
• DOCKING CRANES (MASS 25^{K} Kg) (COST \$52M) A	4	• CARRIAGE • BOOM • WINCH SYSTEM • DOCKING PROBES • CONTROL CAB (2 MAN)	
• 25M INDEXING/SUPPORT MACHINES (MASS 1^{K} Kg) (COST \$3M)	6	• CARRIAGE • BOOM	
• 60M MANIPULATOR/CRANES (MASS 7^{K} Kg) (COST \$18M)	4	• CARRIAGE • ELEVATOR BOOM • TRANSVERSE BOOM • CONTROL CAB (1 MAN) • MANIPULATOR ARMS	
• SOLAR ARRAY DEPLOYMENT MACHINE (MASS 12^{K} Kg) (COST \$45M)	4	• GANTRY/CARRIAGE • DEPLOYMENT CARRIAGE • BLANKET END HANDLER MECH • EDGE ATTACHMENT MECH • CONTROL CAB (2 MAN)	
• SOLAR ARRAY ANNEALING MACHINE	(TBD)	(TBD)	

A ALL COST REFLECT AVG UNIT COST AFTER APPLYING LEARNING FACTOR OF 0.9.

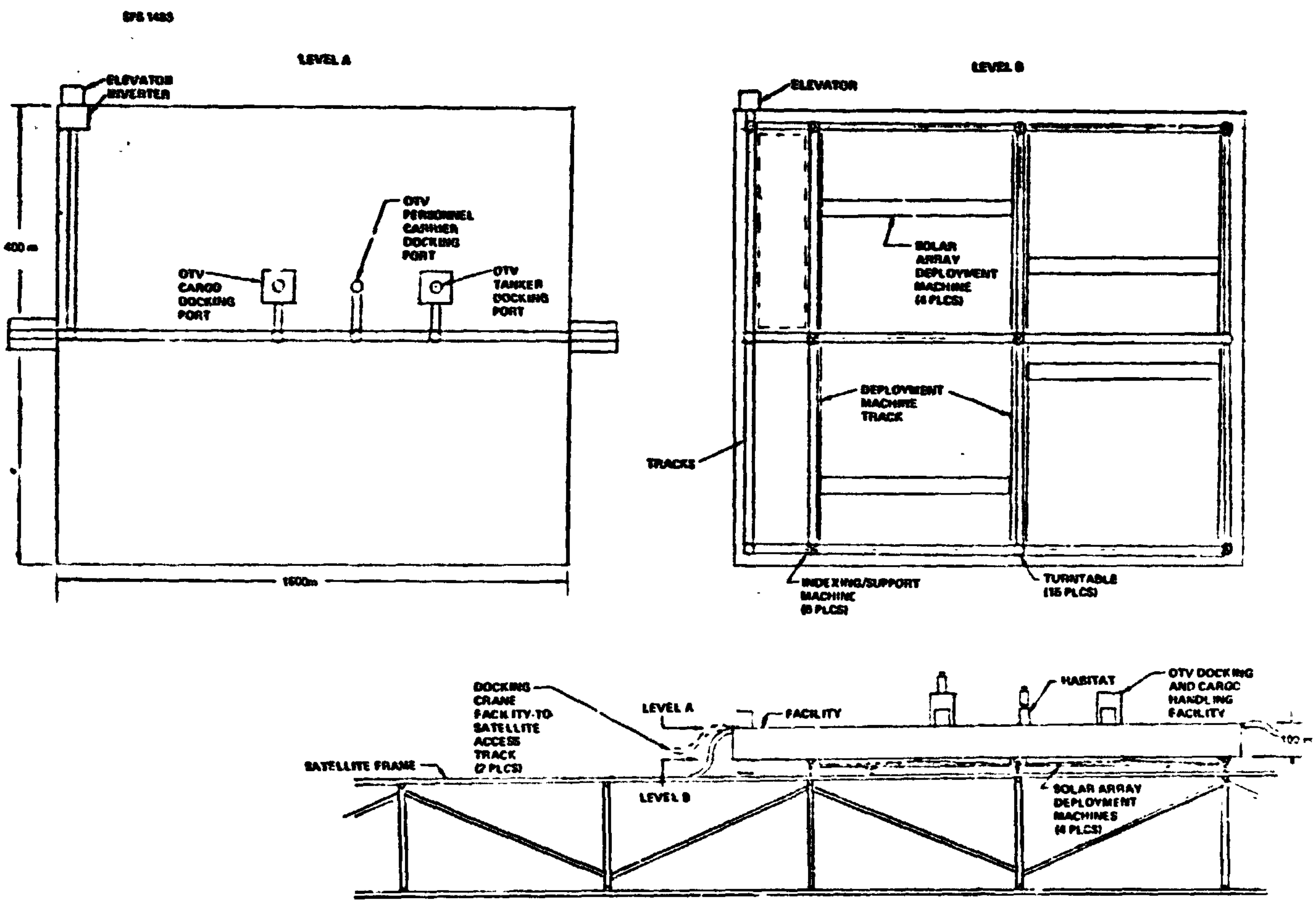

Figure 3.2-74 GEO Final Assembly Base

SPS 1482

Table 3.2-13 Cargo Handling and Distribution Equipment
GEO Final Assembly Base

EQUIPMENT ITEM	NO. REQ'D	MASS (EA) 10^3 Kg	COST (EA) $\$10^6$
• OTV CARGO DOCKING PORT	1	1	4
• OTV CARGO EXTRACTION SYS	1		
• OTV TANKER PORT	1	1	4
• OTV TANKER CARGO EXT SYS	1		
• OTV PERSONNEL DOCKING PORT	1		
• PERSONNEL AIRLOCK SYS	2		
• CARGO SORTING MANIP/CRANE	1	3	6
• CARGO TRANSPORTER	2	0.5	2
• 10 MAN CREW BUS	2	5	4
• TURNTABLES	15		
• CONTROL CABS FOR LOGISTICS EQUIP	2		
• OTV DOCKING	1		
• CARGO SORTER	1		

Subsystems used within the module are the same as for the LEO construction crew quarters modu:

3.2.1.2.2.5 Base Subsystems

The two key base subsystems are the electrical power and flight control systems. An operating electrical load of 260 Kw has been estimated. Use of satellite type solar arrays results in an array size of 1700 square meters. Flight control in terms of attitude control, station keeping and transfer of the base to the longitude location of the next satellite will make use of a LO_2/LH_2 propulsion system.

3.2.1.2.3 Environmental Factors

The principal environmental factors that influence the design of the final assembly base or its operations include radiation, meteoroids, occultations, gravity gradient and drag and collision with manmade objects.

Radiation

The major impact of the radiation environment at GEO is the provision of a solar flare radiation shelter as described in Section 3.2.1.2.4 and revisions to EVA operations should they be necessary. Since the natural GEO radiation environment is more severe than in LEO when EVA is required consideration must be given to either EVA suits with more shielding or shorter EVA periods must be used in order to not exceed the allowable radiation levels for crewmen.

Meteoroids

At this time, no difference has been defined for the meteoroid environment at GEO versus LEO. In both cases, bumpers will be used around all manned modules.

Occultations

The impact of occultations on the GEO base solar array design is less than for the LEO base since only 88 occultations occur per year and with the maximum durations being approximately 70 minutes at the time of the two equinoxes. Shadowing from the base will necessitate external lighting.

Gravity Gradient and Drag

Gravity gradient at GEO has been estimated to be only 0.4% as great as at LEO and drag is essentially negligible.

Collision

The complete collision analysis has been presented in Section 3.2.1.1.3. The one year final assembly of the total satellite adds approximately 0.5 of a collision to the total number of collisions.

3.2.1.2.4 Crew Summary

The GEO base personnel are summarized in the organizational chart shown in Figure 3.2-75. A total of 63 crew members are required at the GEO base. Table 3.2-14 summarizes the crew size at both the LEO and the GEO facilities.

3.2.1.2.5 Mass Summary

A ROM mass for the GEO final assembly base including 90 days of consumables is estimated at 880,000 Kg. A mass breakdown of base is presented in Table 3.2-15.

3.2.1.2.6 Cost Summary

The total unit cost (ROM) of the GEO base for the LEO construction option is estimated at $1,172 million of which approximately $800 million is the basic hardware with the remainder associated with the wraparound items. Unlike the LEO base cost, the construction equipment cost is greater than that of the facility primarily because there is only one crew module. A cost breakdown is presented in Table 3.2-16.

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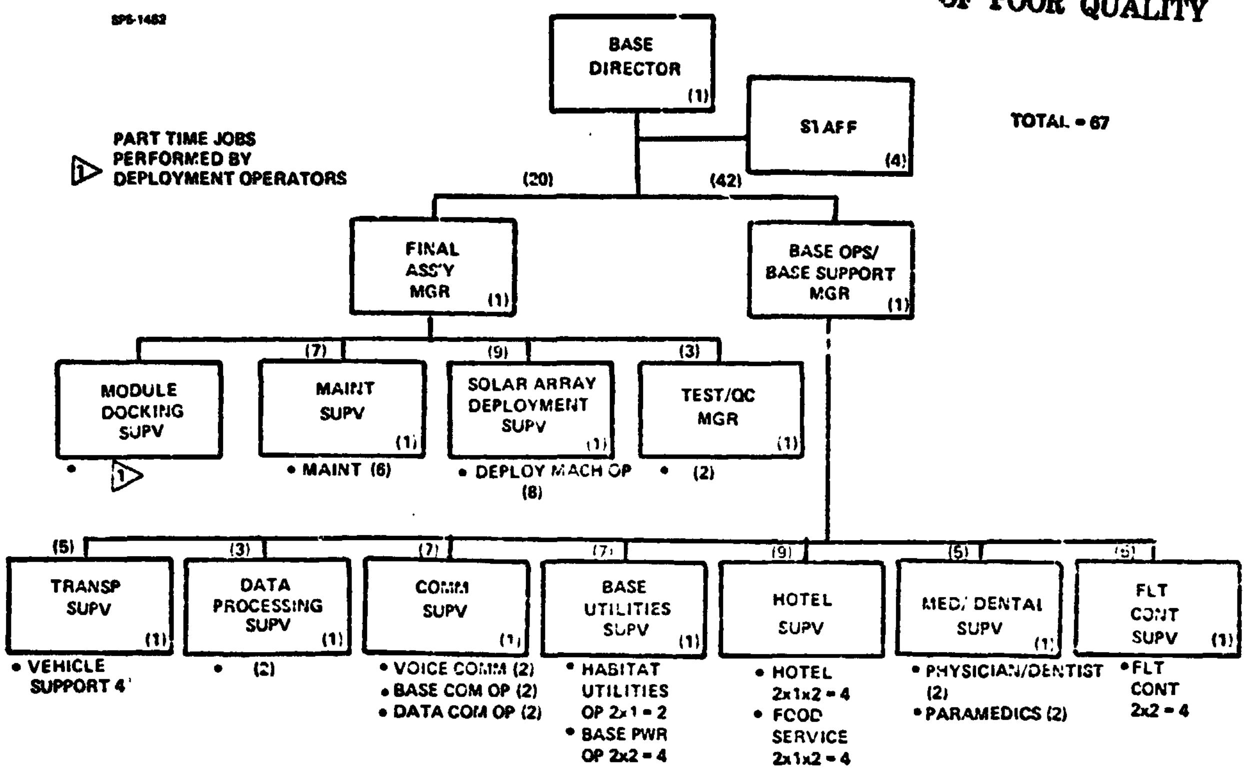

Figure 3.2-75 GEO Base Personnel

Table 3.2-14 LEO Construction Manpower Estimate

SPS 1455

	LEO BASE	GEO BASE
BASE MGMT	(10)	(5)
CONSTRUCTION	(352)	(20)
MGMT	22	4
MODULE CONST	66	8
ANTENNA CONST	82	
SUBASSEMBLY	49	
MAINT	49	6
LOGISTICS	42	
TEST/QC	40	2
BASE OPS	(39)	(16)
MGMT	7	4
TRANSPORTATION	18	4
COMM	8	6
DATA PROCESSING	6	2
BASE SUPPORT	(77)	(26)
MGMT	8	4
BASE UTILITIES	14	6
HOTEL	38	8
MEDICAL	13	4
FLT CONT	4	4
BASE TOTAL	478	67
TOTAL	545	

Table 3.2-15 GEO Final Assembly Base ROM Mass

	10^3Kg	
FACILITY		(680)
FOUNDATION	280	
CREW MODULE	335 [1]	
CARGO HANDLING/DISTRIBUTION	55	
BASE SUBSYSTEMS	10	
CONSTRUCTION & SUPPORT EQUIPMENT		(175)
SOLAR ARRAY INST	50	
CRANE/MANIPULATOR	15	
INDEXERS	6	
DOCKING CRANES	104	
DRY TOTAL		(855)
CONSUMABLES (90 DAYS)		(35)
TOTAL		(880)

[1] INCLUDES RADIATION SHELTER

Table 3.2-16 GEO Final Assembly Base ROM Cost

		$\$10^6$
FACILITY		(382)
FOUNDATION	30	
CREW MODULES	300	
CARGO HANDLING/DISTRIBUTION	50	
BASE SUBSYSTEMS	2	
CONSTRUCTION EQUIPMENT		(425)
SOLAR ARRAY INSTALLATION	165	
CRANE/MANIPULATOR	35	
INDEXERS	15	
BERTHING CRANES	210	
BASIC HARDWARE		(807)
SPARES		120
INSTALL, ASSEMBLE, C/O		123
SE&I		55
PROJECT MANAGEMENT		15
SYSTEM TEST		17
GSE		30
TOTAL		(1172)

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3.2.2 GEO Construction Concept

The GEO construction concept is illustrated in Figure 3.2-76 and begins with a LEO staging depot, which has the capability to transfer payloads from a launch vehicle to orbit transfer vehicles and to house and maintain the orbit transfer vehicle fleet. Transfer of all payloads between LEO and GEO is accomplished using LO_2/LH_2 OTV's. Construction of the entire satellite including antenna is done at GEO. The reference satellite for the GEO construction option is a monolithic design rather than modular as in the case of LEO construction.

The following sections describe the construction tasks, facilities, construction sequences and the differences between the GEO construction concept and the LEO construction concept described in Section 3.2.1.

3.2.2.1 LEO Staging Depot Analysis

3.2.2.1.1 Operations
The LEO staging depot is sized to support the construction of one satellite per year. The principal functions of the depot and the number of docking ports required are presented in Table 3.2-17. One SPS component OTV flight per day is based on a five day a week launch and flight schedule. As such, the depot must provide accommodations for three launch vehicle payloads: one being the SPS components and the other two being propellant tankers used to refuel the orbit transfer vehicles. Since the orbit transfer vehicle propellant loading requires slightly more propellant than can be provided by two tankers, a storage tank is also provided at the staging depot and is refueled

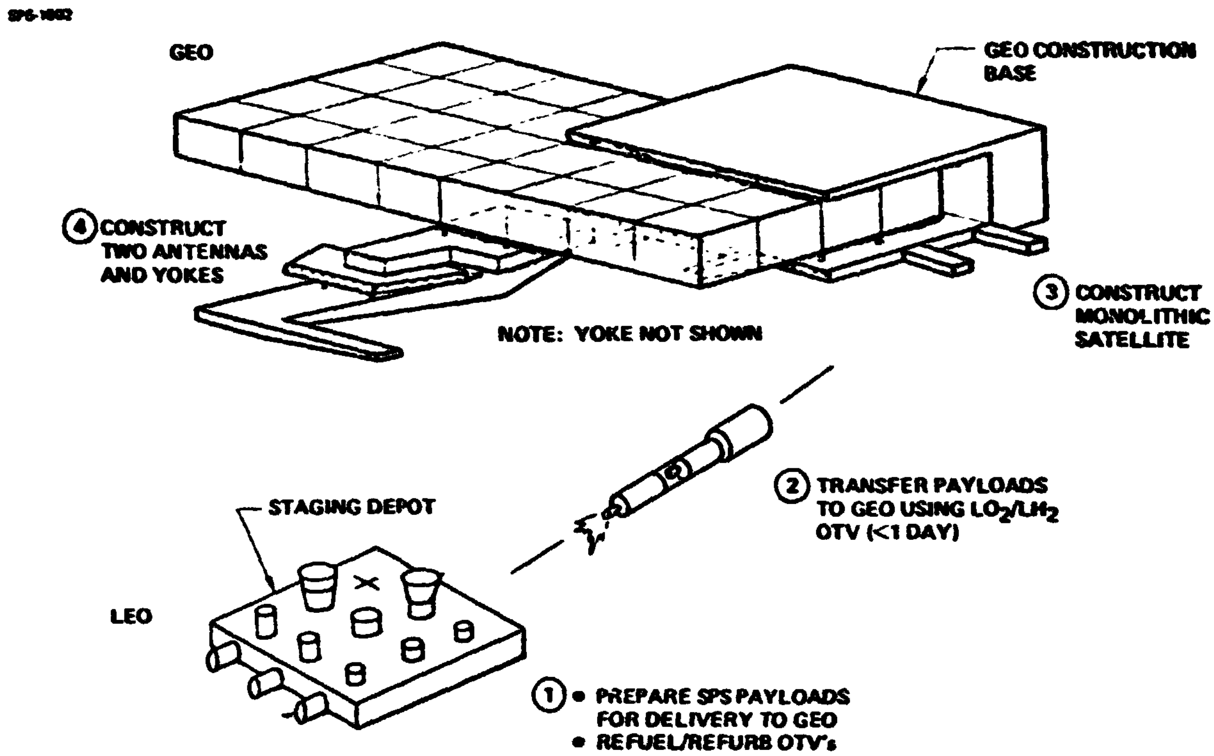

Figure 3.2-76 GEO Construction Concept

Table 3.2-17 Staging Depot Requirements

SPS-1775

FUNCTION	DOCKING PORT	
• 1 CARGO OTV FLT/DAY	1	OTV STG 1
	1	OTV STG 2
	1	CARGO MODULE (SPS COMPONENTS)
	1	TANKER 1
	1	TANKER 2 15 HOURS FOR TRANSFER
	–	TANKER 3 (ARRIVES AFTER T-1 LEAVES)
• 1 CREW ROTATION/RESUPPLY FLT/30 DAYS (SIMULTANEOUS WITH CARGO FLT)	1	OTV STG 1
	1	OTV STG 2
	1	SUPPLY MODULE
	1	CREW MODULE (FLT CONT & PERS. MOD)
	1	TANKER 1
	1	TANKER 2
	–	TANKER 3 (ARRIVES AFTER T-1 DEPARTS)
	–	LEO CREW MODULE (DOCKS TO TRANSIENT CREW QUARTERS)
• SPARES	1	OTV STG 1
	1	OTV STG 2
	1	CREW MODULE
PROPELLANT STORAGE	1	COMBINATION LO_2/LH_2 TANK (< 1 M KG CAPACITY)
	TOTAL 15	

every fourth OTV flight. Other docking accommodations are provided for a dedicated OTV used for GEO crew rotation/resupply on a once per month basis. This operation also requires docking for supply modules and crew transfer vehicles.

The crew rotation operations associated with the GEO construction concept are illustrated in Figure 3.2-77. Two shuttle growth vehicles are used to deliver 160 people to the LEO staging depot once per month. A crew rotation/resupply OTV delivers the new crew to the GEO construction base. The OTV then returns to the LEO staging depot with 160 people who have completed their duty. The two orbiters which delivered the new crew to the staging depot then return the old crew to Earth.

3.2.2.1.2 System Description

3.2.2.1.2.1 Configuration

The LEO staging depot consists of three crew modules and 15 docking ports for the various system elements. The functional arrangement including identification of those elements having functional interfaces is shown in Figure 3.2-78. The configuration of the staging depot is presented in Figures 3.2-79 and 3.2-80. (NOTE: The arrangement of the docking ports is slightly different from the functional arrangement which is the most updated.)

3.2.2.1.2.2 Foundation

The foundation of the staging depot consists of 20m beams forming an overlap grid pattern consisting of 2300m of beam.

3.2.2.1.2.3 Crew Modules

Three crew modules have been provided at the staging depot. One module serves as a combination crew quarters for the 75 people and the operations center for the depot. A second module is used as a transient crew quarters to accommodate 160 personnel which is the quantity that is rotated at the GEO base each month. A maintenance module is also included at the depot for repair work primarily on the transportation systems and payload handling systems. The crew modules have the same design approach as described in Section 3.2.1.1.2.4.

3.2.2.1.2.4 Vehicle and Payload Handling

The overall concept of vehicle and payload handling consist of having incoming vehicles dock directly to the depot and then utilize a crane/manipulator to stack the various elements to form the desired vehicle as shown in Figure 3.2-80. For payloads that are removed from the payload shroud of the launch vehicle, the docking system is located on a pedestal to allow the removal as shown in Figure 3.2-80. Payloads are moved around the depot through the use of cargo transporters or the crane/manipulator.

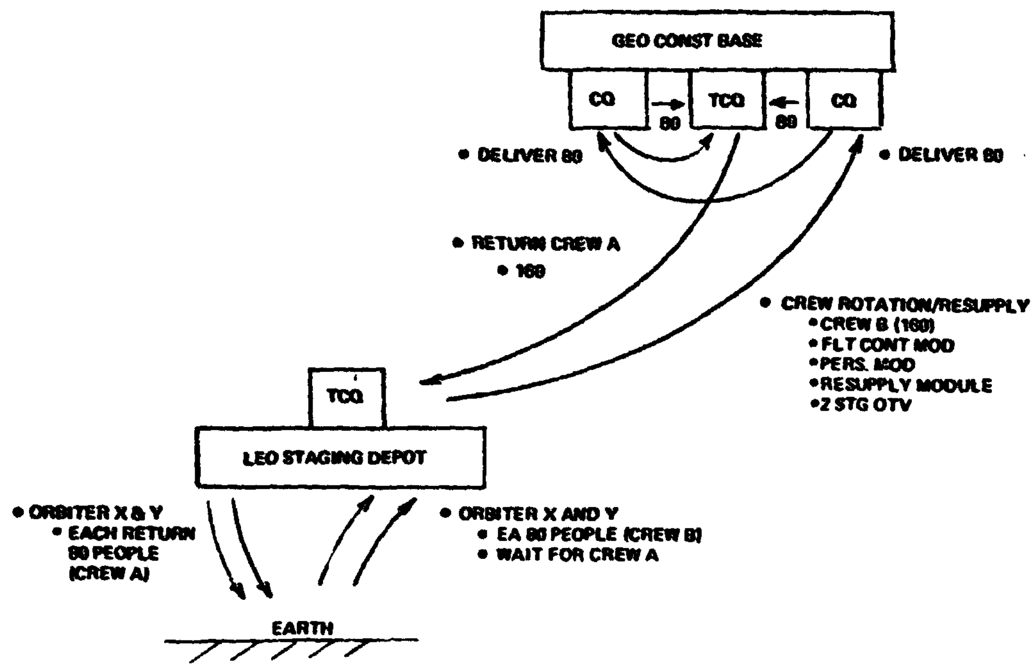

Figure 3.2-77 Crew Rotation Operations GEO Construction

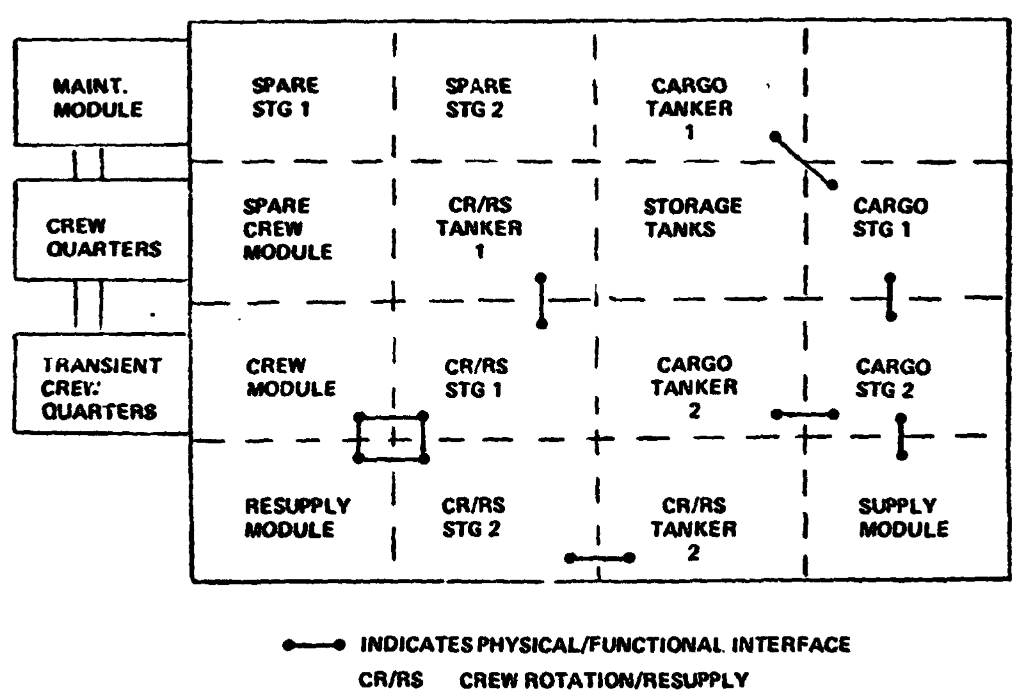

Figure 3.2-78 Staging Depot Functional Arrangement

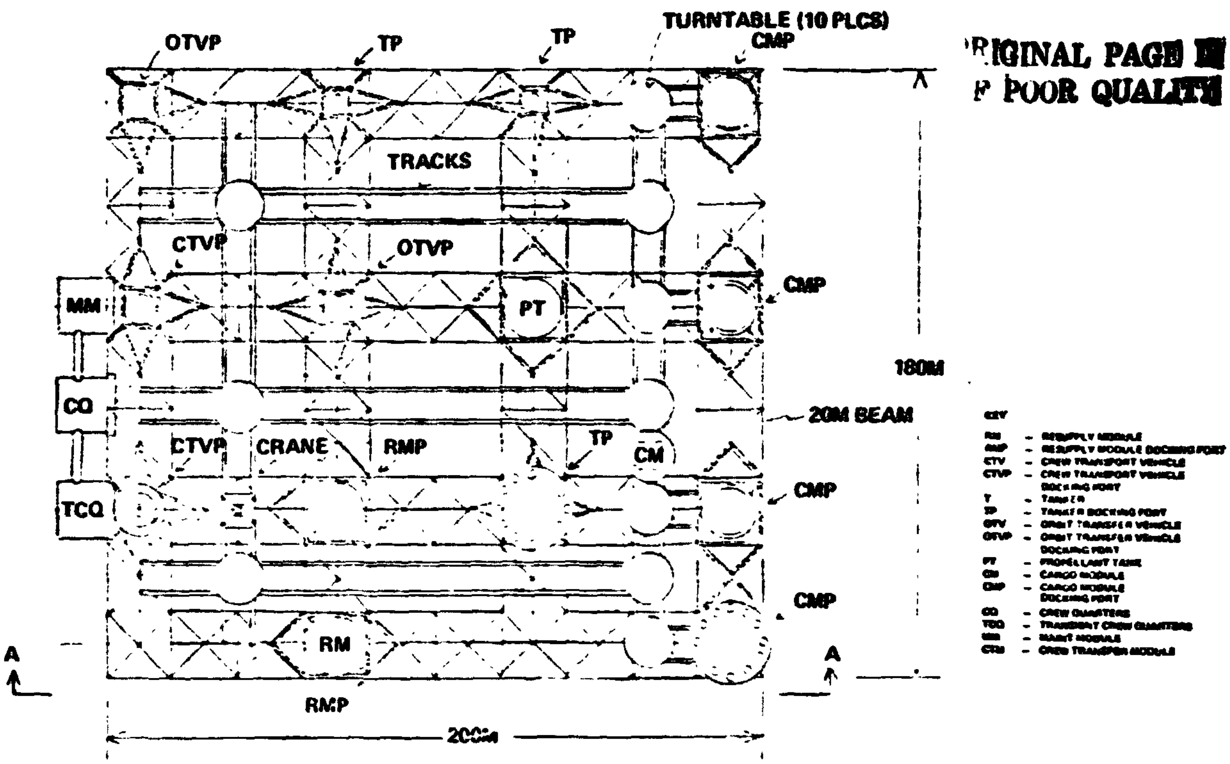

Figure 3.2-79 LEO Staging Depot

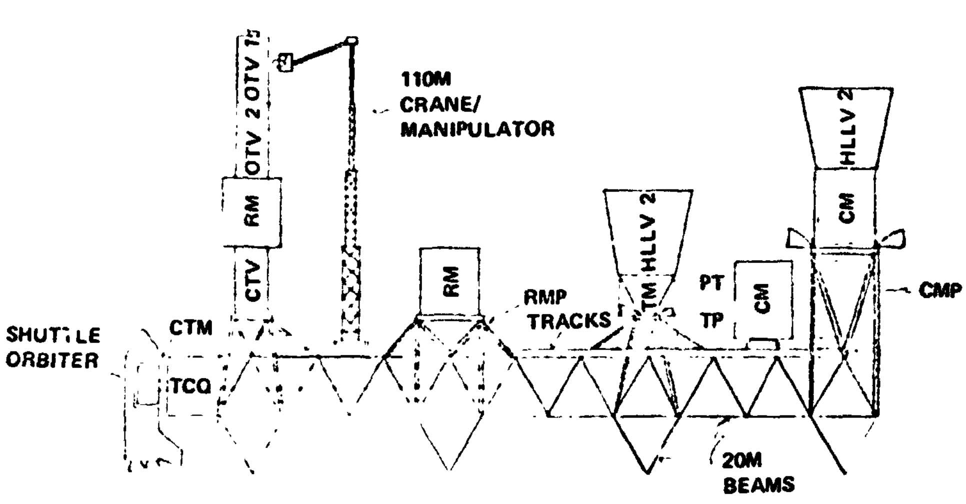

Figure 3.2-80 LEO Staging Depot

3.2.2.1.2.5 Propellant Storage and Distribution

Under normal conditions, one cargo OTV flight per day requires 830,000 Kg of propellant. Two propellant tankers can deliver and transfer directly to the OTV 720,000 Kg of propellant thus resulting in a 110,000 Kg deficit. The remaining propellant is then provided by a third tanker which can again transfer directly to the OTV or to a storage tank which can in turn transfer the propellant to the OTV's. This results in the storage tanks being sized to hold all of the propellant from one tanker (360,000 Kg). A larger storage capacity has not been provided since if the tankers are unable to reach the staging depot due to weather, etc., launch vehicles delivering satellite components would also be cancelled. At this point, the only justification for more storage is for the case of the ground propellant production capability being restricted or shut down for a short time period while the payloads could still be launched. This case, however, is speculation. A more complete analysis of the logistics associated with material and propellant will be conducted in Part III of the SPS study.

The method used to transfer propellant from one vehicle to another is that of capillary acquisition using suspended screen channels as defined by General Dynamics in study NAS9-15305. A total of 15 hours is assumed for the transfer of 830,000 Kg of propellant when using this transfer approach. A liquifaction unit is also included to minimize the effect of boil-off.

3.2.2.1.2.6 Base Subsystems

Primary Power is provided by a body mounted solar array with blanket characteristics the same as for the satellite. The array is sized for a load of 1000 Kw as shown in Table 3.2-18.

Table 3.2-18. Staging Depot Power Requirements

ECLS	300
Internal Lighting	90
External Lighting	60
Information System	15
Propellant Management	35
Base Load	500 Kw
Sec. Power Recharging	300
Conditioning	100
Distribution	100
Total	1000 Kw

The basic solar array for the load would require 7000 m^2 however with fixed body mounted cells a total of 35,000 m^2 is required.

Nickel hydrogen batteries are used for the secondary power system used during occultations.

A total power system mass of 30,000 Kg has been estimated.

The flight control/station keeping capability is provided by a LO_2/LH_2 system.

3.2.2.1.3 Environmental Factors

The environmental factors for the LEO staging depot are the same as discussed for the LEO construction base in Section 3.2.1.1.3.

3.2.2.1.4 Crew Summary

A crew size of 72 has been estimated to operate the LEO staging depot. Again this value is based on work schedules of two shifts each 10 hours per day and 6 days per week. A breakdown of the crew is presented in Table 3.2-19

3.2.2.1.5 Mass Summary

A ROM mass of 750,000 Kg has been estimated. A breakdown of the mass is presented in Table 3.2-20. Mass estimates for the propellant storage and distribution were based to some degree on the General Dynamics study results.

3.2.2.1.6 Cost Summary

A total ROM unit cost for the LEO staging depot has been estimated at $1130 million with the basic hardware contributing $785 million and wraparound cost the remainder of the total. A breakdown of the cost estimate is presented in Table 3.2-21.

3.2.2.2 GEO Base Construction Analysis

3.2.2.2.1 GEO Operations

3.2.2.2.1.1 Top-Level GEO Construction Tasks

The top-level construction tasks to be performed using the GEO construction approach are the following.

- Construct frame assembly
- Install solar array
- Install power busses
- Construct install 2 yoke assemblies
- Construct/install 2 antennas
- Install attitude control system
- Install subsystems

Table 3.2-19 LEO Staging Depot Crew Summary
GEO Construction

FUNCTION		QUANTITY
MANAGEMENT		6
DEPOT OPERATIONS		12
FLIGHT CONTROL	(2)	
COMMUNICATIONS	(2)	
DATA MANAGEMENT	(2)	
UTILITIES	(4)	
SUPERVISOR	(2)	
HOTEL MANAGEMENT		12
MANAGERS	(6)	
COOKS	(6)	
VEHICLE AND DEPOT MAINTENANCE		16
STRUCTURE	(2)	
MECHANICAL	(6)	
AVIONICS	(6)	
SUPERVISOR	(6)	
VEHICLE AND PAYLOAD HANDLING		14
STATUS	(4)	
DOCKING	(4)	
ASSEMBLY	(4)	
SUPERVISOR	(2)	
PROPELLANT MANAGEMENT		14
TANKER OR HOLDING TANK		
SYSTEM STATUS	(2)	
TRANSFER OPERATION	(2)	
MISCELLANEOUS	(2)	
ORBIT TRANSFER VEHICLE		
SYSTEM STATUS	(2)	
TRANSFER OPERATION	(2)	
MISCELLANEOUS	(2)	
SUPERVISOR	(2)	
TOTAL		72

Table 3.2-20 LEO Staging Depot ROM Mass

		10^3KG
FOUNDATION		15
CREW MODULES		590
BASE SUBSYSTEMS		30
VEHICLE AND PAYLOAD HANDLING		40
PROPELLANT STORAGE AND DISTRIBUTION		55
	DRY TOTAL	730
CONSUMABLES (90 DAYS)		20
	TOTAL	750

Table 3.2-21 LEO Staging Depot ROM Cost

FIRST SET

		($\$10^6$)
FOUNDATION		1
CREW MODULES		645
BASE SUBSYSTEMS		4
VEHICLE AND PAYLOAD HANDLING		120
PROPELLANT STORAGE AND DISTRIBUTION		15
	BASIC HARDWARE	\$785
SPARES (15%) [1]		115
INSTALL, ASSEMBLY, C/O (16%)		125
SE&I (7%)		55
PROGRAM MANAGEMENT (2%)		15
SYSTEM TEST (3%)		25
GSE (4%)		30
	TOTAL	\$1130

[1] % OF BASIC HARDWARE

3.2.2.2.1.2 GEO Construction Base

GEO construction of the photovoltaic satellite will require two separate facilities at GEO as shown in Figure 3.2-81. One facility is used to construct the satellite while the other is used to construct the antenna. These facilities are similar in size and in function to the LEO construction facility described in Section 3.2.1. Therefore, these facilities will only be discussed in terms of their differences relative to LEO construction base facilities.

It should be noted that, while in the figure these two facilities are shown to be separated, in operational use, the antenna facility will be attached beneath the satellite facility most of the time. It will be detached only when the antennas are mated to the yoke. This is described in the following sections.

3.2.2.2.1.3 Top-Level Construction Sequence

The top-level construction sequence is shown in Figure 3.2-82. Note that the satellite facility must index both laterally and longitudinally. The operational sequence involved in mating the antenna to the yoke is shown in Figure 3.2-83. The overall timeline for the GEO construction concept is presented in Figure 3.2-84.

3.2.2.2.1.3.1 Alternate Antenna Construction Options

Several alternate antenna construction/installation options were investigated for the GEO construction option. These options are illustrated in Figures 3.2-85 and 3.2-86. Two main categories of options were identified. The first considers the case where there is only one antenna facility but has two suboptions in the form of: 1) IA, the antenna facility is always contiguous with the satellite facility as in Figure 3.2-85 and 2) IB, the antenna facility is independent of the satellite facility as in Figure 3.2-86. The second main option, II, makes use of two antenna facilities as shown in Figure 3.2-86.

One of the most significant criteria used in assessing the options is that of the logistics problem in getting satellite and antenna components to their required location. This factor becomes a major issue since launch economics (payload component densities) forces both types of components to be in each delivery from Earth. Other factors considered in the assessment include antenna installation complexity, redundancy in orbital systems and facility design. The particular problem(s) with each of the investigated alternatives is indicated in the figures.

3.2.2.2.1.4 Detailed Construction Task Analysis

The construction of the satellite, antenna, and yoke will be accomplished identically to that described in detail in Section 3.2.1.1.4.

3.2.2.2.1.5 Construction Equipment Summary

Since the construction operations are virtually identical to that described for the LEO construction concept the construction equipment previously given in Table 3.2-1 also applies.

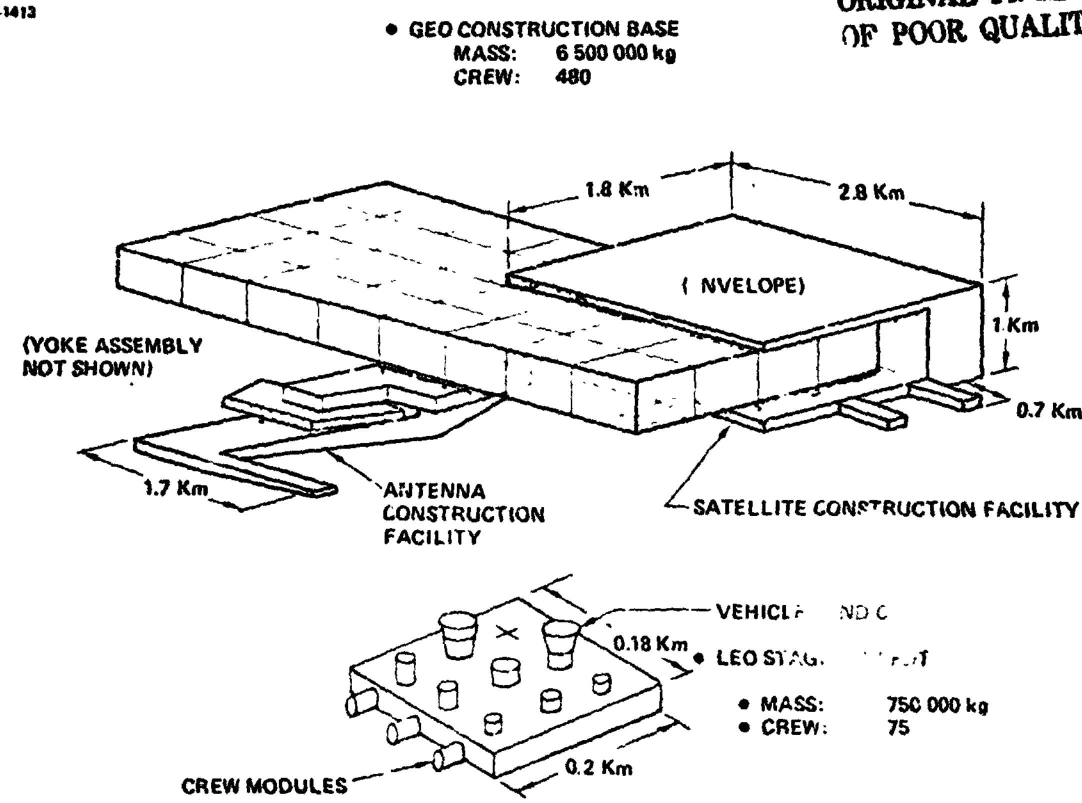

Figure 3.2-81 GEO Construction Base

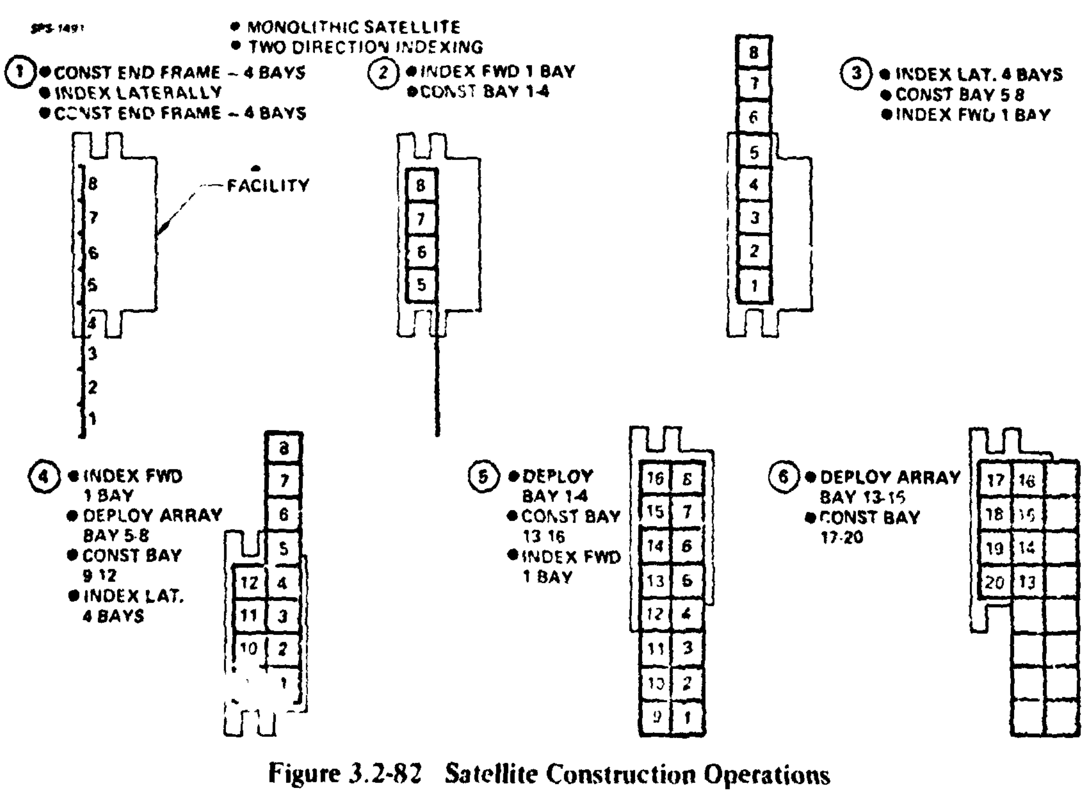

Figure 3.2-82 Satellite Construction Operations
GEO Construction Concept

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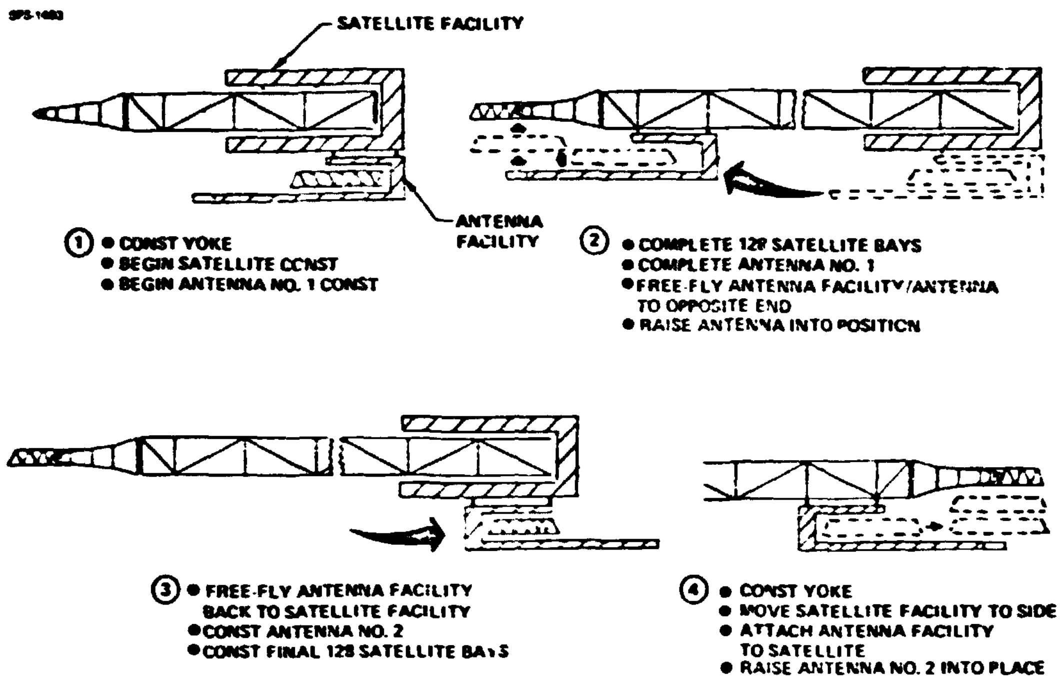

Figure 3.2-83 Antenna Construction and Installation GEO Construction Concept

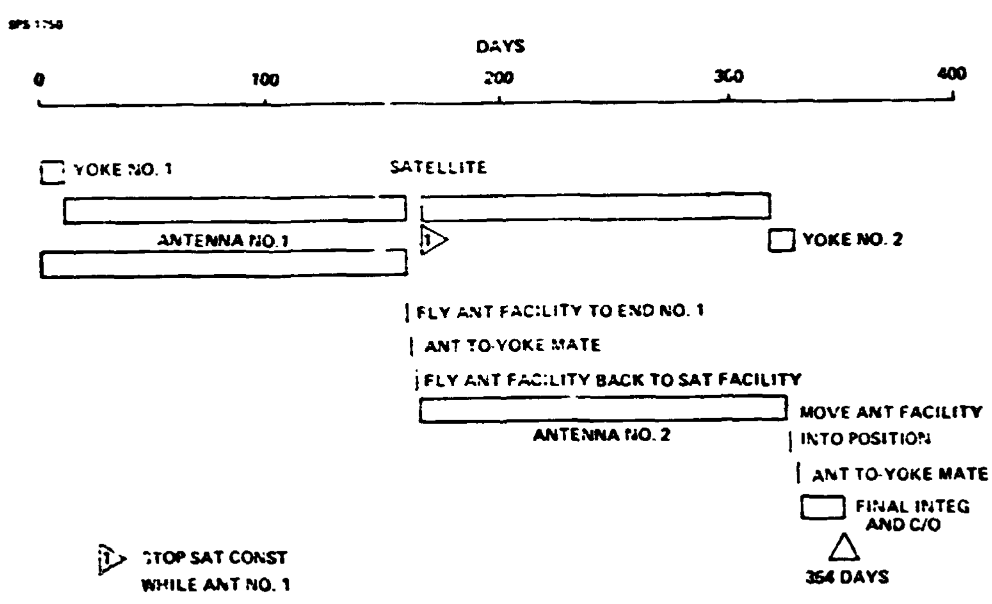

Figure 3.2-84 GEO Construction Timeline

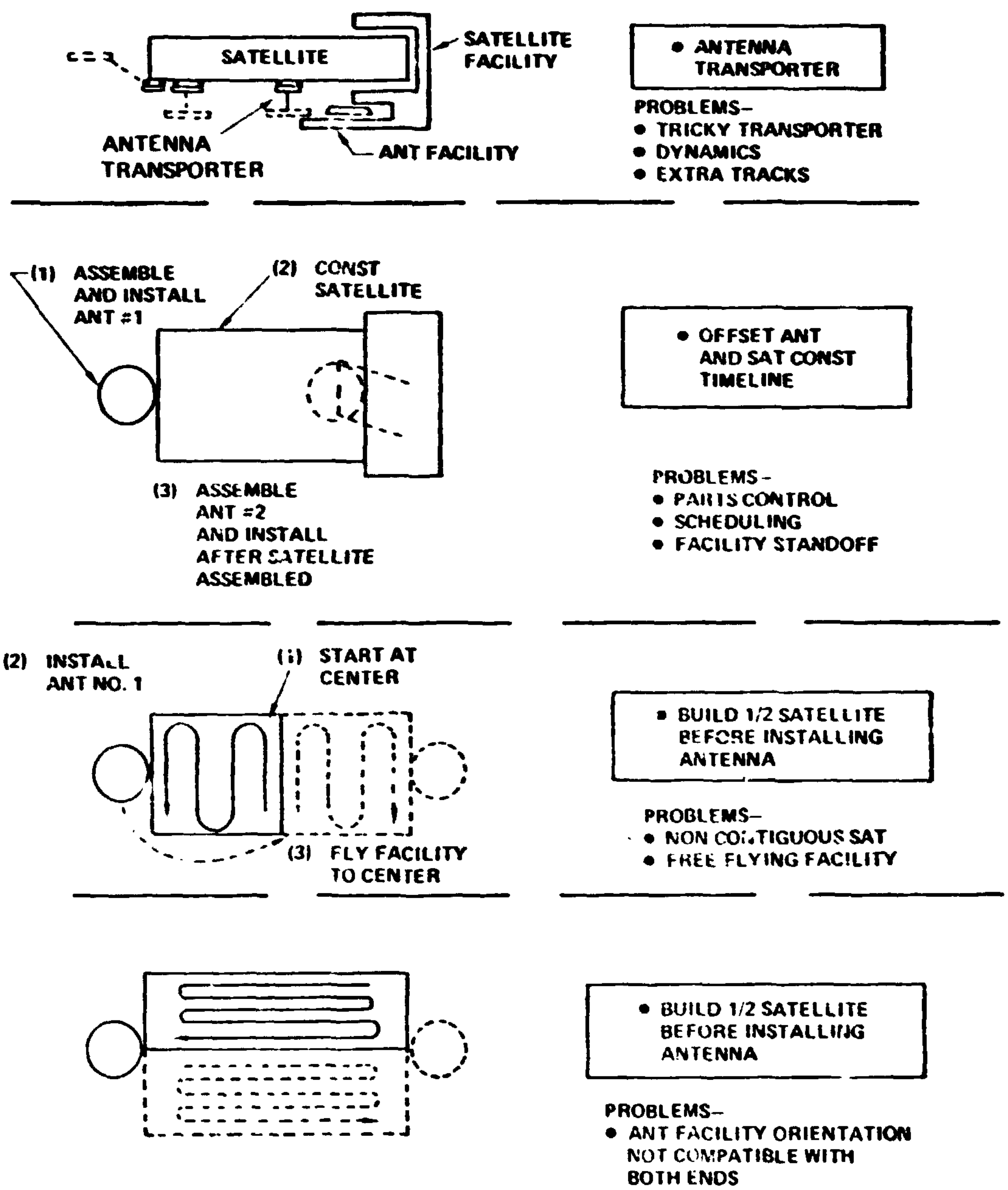

Figure 3.2-85 IA Antenna Facility Contiguous with Satellite Facility

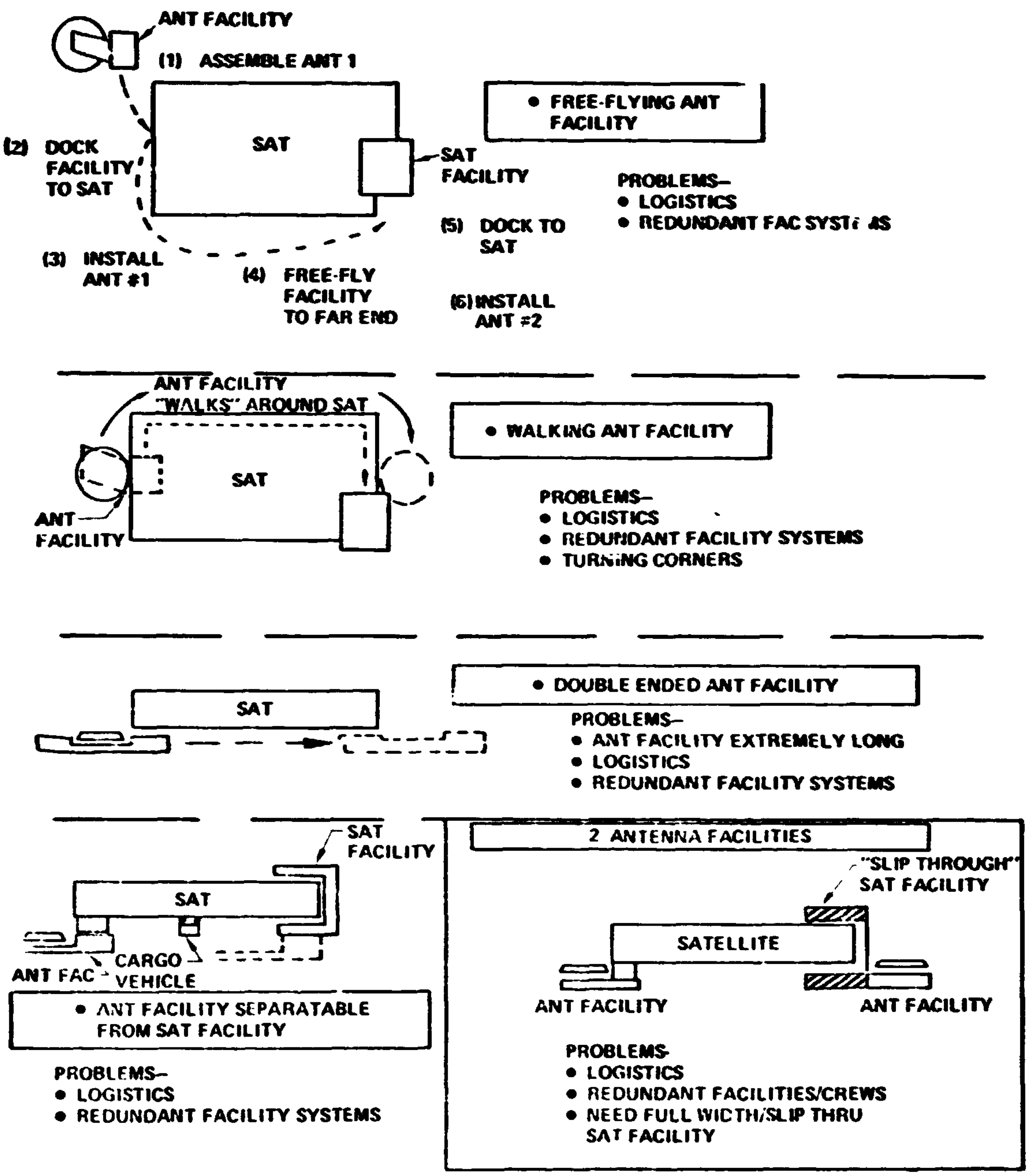

Figure 3.2-86 IB Antenna Facility Independent from Satellite Facility

3.2.2.2.2 GEO Construction Base Description

3.2.2.2.2.1 Configuration

The general configuration of the construction base for the GEO construction option is essentially the same as for the LEO construction base described in Section 3.2.1.1.2 but has a few small variations. One of these variations occurs in the form of outriggers on the side of the satellite construction facility. Each of these four outriggers is approximately 700m in length which allows a completed portion of the satellite to be indexed laterally one bay length so it is outside construction area so additional sections of the satellite can be constructed. This particular operation was shown previously in Figure 3.2-82.

Another deviation of the GEO construction base is that the antenna construction facility is designed so that it can be separated from and reattached to the satellite construction facility. This operation is necessary in the installation of the antenna as previously discussed in Section 3.2.2.2.1.3.

3.2.2.2.2.2 Foundation

The foundation or structure of the construction base has been assumed to be the same as for the LEO construction base described in Section 3.2.1.1.2.2.

3.2.2.2.2.3 Cargo Handling Distribution

This system moves cargo and personnel around the base and is essentially the same as described for the LEO construction base in Section 3.2.1.1.2.3.

3.2.2.2.2.4 Crew Modules

Ten crew modules including six crew quarters modules are also used at the GEO construction base. The definition of these modules is essentially the same as for those described for the LEO construction base in Section 3.2.1.1.2.4 with the exception that each of the crew quarters modules has an eighth deck that serves as a solar flare radiation shelter. The mass penalty for each of these shelters is estimated to be 115,000 Kg.

3.2.2.2.2.5 Base Subsystems

The electrical power system for the base consists of satellite type solar arrays for primary power and nickel hydrogen batteries during occultation. Since occultations at GEO are of very short duration and occur very infrequently (88 times per year) it has been assumed the construction work will shut down during these periods and have no appreciable effect on the timeline. Because the nickel hydrogen system does not need recharging every day the total power requirement is 2600 Kw rather than 3200 Kw as for the LEO construction base.

A LO_2/LH_2 flight control system is used to satisfy the functions of attitude control, station keeping and movement of the base to the next satellite location.

3.2.2.2.3 Environmental Factors

Environmental factors including radiation, meteoroids, occultations, gravity gradient and drag and collision with manmade objects are the same as discussed for the GEO final assembly base in Section 3.2.1.2.3.

3.2.2.2.4 Crew Summary

The crew size for the GEO construction base is judged to be the same as for the LEO construction base or 480 since essentially the same tasks are being performed during the same time period. A breakdown of the crew would be similar to that defined in Section 3.2.1.1.4.3.

3.2.2.2.5 Mass Summary

A ROM mass of approximately 6.5 million kg has been estimated for the GEO construction base. A summary of the estimate is presented in Table 3.2-22. This mass is approximately 0.7 million kg heavier than the comparable LEO construction base primarily as a result of the additional mass of the solar flare radiation shelters.

3.2.2.2.6 Cost Summary

A ROM unit cost of approximately $7.5 billion has been estimated for the GEO construction base. A summary breakdown is presented in Table 3.2-23. This base is approximately $0.5 billion greater than the LEO construction base again primarily due to the cost associated with the radiation shelters.

Table 3.2-22 GEO Construction Base ROM Mass

		(10^3KG)	
FACILITY			(5750)
FOUNDATION		2500	
CREW MODULES		2690	
CARGO HANDLING/DISTRIBUTION		400	
BASE SUBSYSTEMS		60	
MAINTENANCE PROVISION		100	
CONSTRUCTION & SUPPORT EQUIPMENT			(515)
STRUCTURAL ASSEMBLY		80	
SOLAR ARRAY INSTALLATION		60	
POWER DISTRIBUTION INSTALLATION		20	
SUBARRAY INSTALLATION		30	
CRANES/MANIPULATORS		255	
INDEXERS		70	
	DRY TOTAL	6265	
CONSUMABLES (90 DAYS)		270	
	TOTAL	6535	

Table 3.2-23 GEO Construction Base ROM Cost

FIRST SET

	$\$10^6$	
FACILITY		(3610)
FOUNDATION	250	
CREW MODULES	3020	
CARGO HANDLING/DISTRIBUTION	330	
BASE SUBSYSTEMS	10	
MAINTENANCE PROVISIONS	–	
CONSTRUCTION AND SUPPORT EQUIPMENT		(1555)
STRUCTURAL ASSEMBLY	350	
ENERGY COLLECTION AND CONVERSION	165	
POWER DISTRIBUTION INSTALLATION	75	
SUBARRAY INSTALLATION	80	
CRANES/MANIPULATORS	760	
INDEXERS	125	
BASIC HARDWARE		(5165)
SPARES		775
INSTALLATION, ASSEMBLY, C/O		825
SE&I		360
PROJ MANAGEMENT		100
SYSTEM TEST		155
GSE		205
TOTAL		(7585)

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3.3 THERMAL ENGINE SATELLITE CONSTRUCTION

This section contains the construction analysis of the thermal engine satellite configuration shown in Figure 3.3-1. There are two construction approaches that were analyzed: 1) LEO construction–the satellite is assembled in modules at LEO and transported to GEO using self-powered electric thrusters, and then the modules are attached at GEO, and 2) GEO Construction–the satellite is constructed in modular fashion at GEO.

The LEO construction approach is developed in detail while the GEO approach is examined for its differences relative to the LEO construction approach. In the following sections, the construction tasks, facilities, assembly sequences, machinery, logistics and personnel are discussed.

3.3.1 LEO Construction Concept

For the thermal engine satellite, the LEO construction concept s' in Figure 3.3-2 entails constructing 16 modules and 2 antennas at LEO, individually transporting each module to GEO using self-powered electric propulsion, and at the GEO the modules are attached to form the complete satellite and the antennas erected.

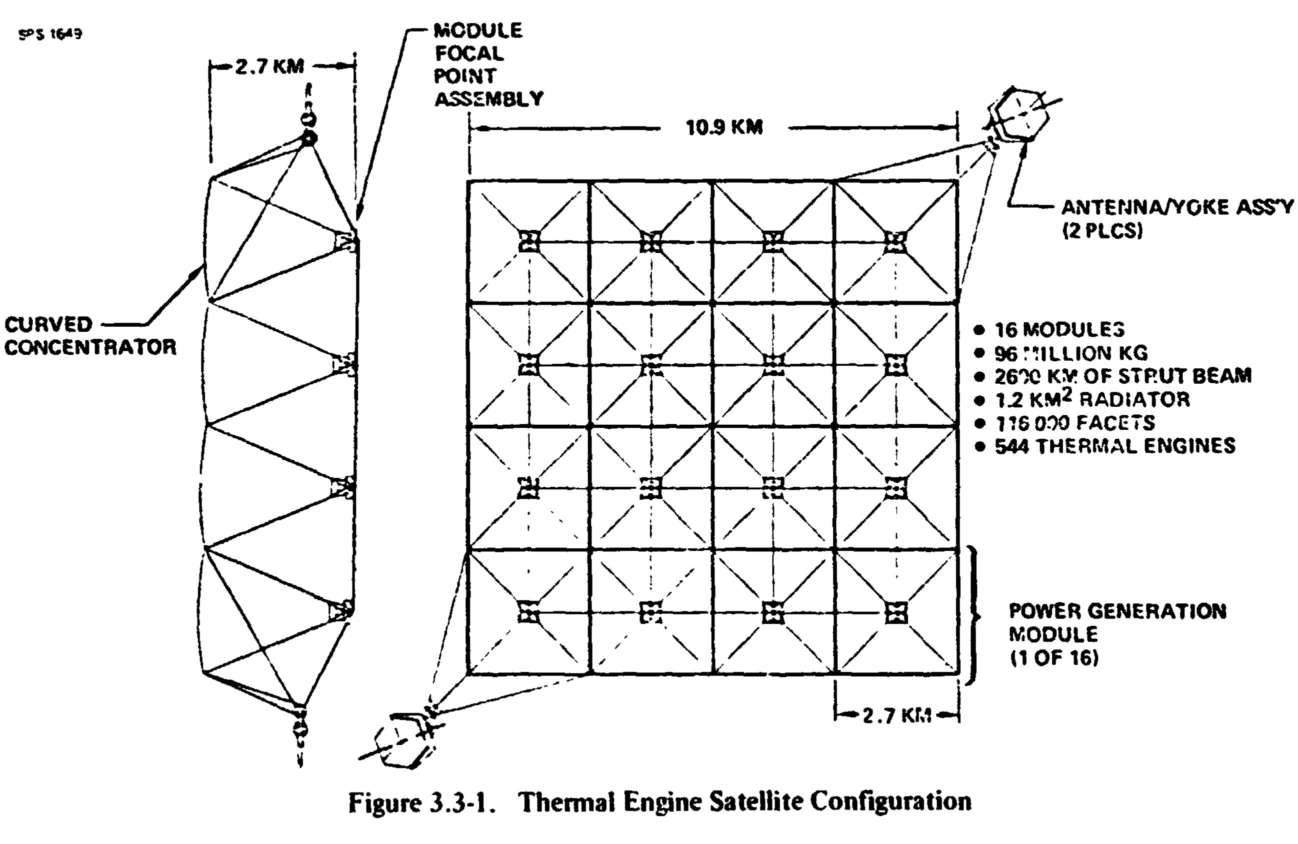

Figure 3.3-1. Thermal Engine Satellite Configuration

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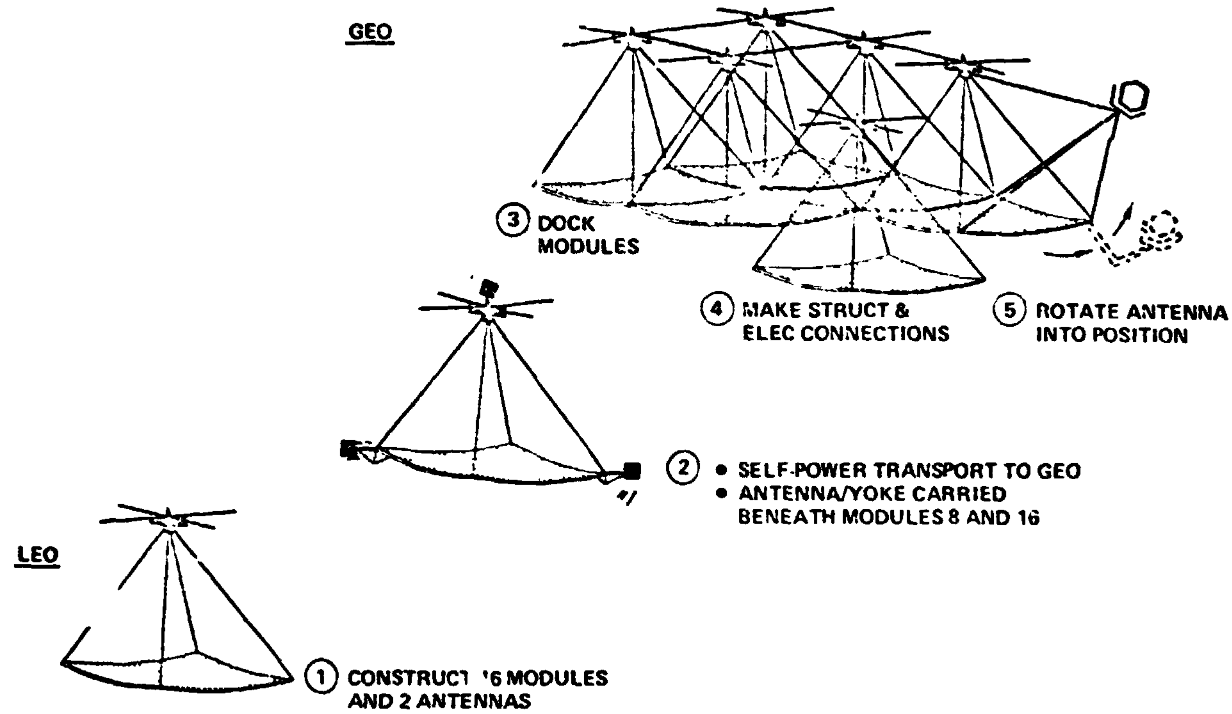

Figure 3.3-2. LEO Construction Concept Thermal Engine Satellite

The construction operations at the LEO base are described in Section 3.3.1.1. The GEO base operations are described in Section 3.3.1.2.

3.3.1.1 LEO Base Construction Analysis

3.3.1.1.1 LEO Construction Operations

3.3.1.1.1.1 Top Level LEO Construction Tasks

The construction tasks to be accomplished at the LEO base are summarized in Figure 3.3-3. Each of these tasks are discussed in detail in following sections.

3.3.1.1.1.2 LEO Construction Base

The construction base to be used at LEO is shown in Figure 3.3-4. The base consists of three construction areas: 1) the yoke and antenna construction area, 2) the reflector construction factory, and 3) the focal point factory. The base is described in more detail in Section 3.3.1.1.2.

3.3.1.1.1.3 Top-Level Construction Sequence and Timeline

The reflector, the focal point and the antenna yoke assemblies are constructed in parallel.

Figure 3.3-5 shows how the reflector is assembled. This assembly operation is discussed in detail in Section 3.3.1.1.1.4.1.

Figure 3.3-6 shows how the focal point assembly is constructed. The details of this assembly operation is discussed in Section 3.3.1.1.1.4.2.

Figure 3.3-7 shows how the yoke antenna assembly is constructed. The details of these assembly operations are discussed in Sections 3.3.1.1.1.5 and 3.3.1.1.1.6.

These reflector and focal point subassemblies are connected to form a complete module and then the module is indexed out of the construction facility as shown in Figure 3.3-8.

The yoke and antenna construction proceed concurrently with the construction of eight modules. The antenna and yoke assemblies are then attached to the 8th and 16th modules via a hinge assembly.

The top-level construction timeline is shown in Figure 3.3-9.

3.3.1.1.1.4 Module - Detailed Construction Task Analysis

The module is composed of the reflector and the focal point assemblies. The reflector construction details are given in Section 5.2.4.1.1.4.1 and the focal point construction details are given in Section 3.3.1.1.4.2.

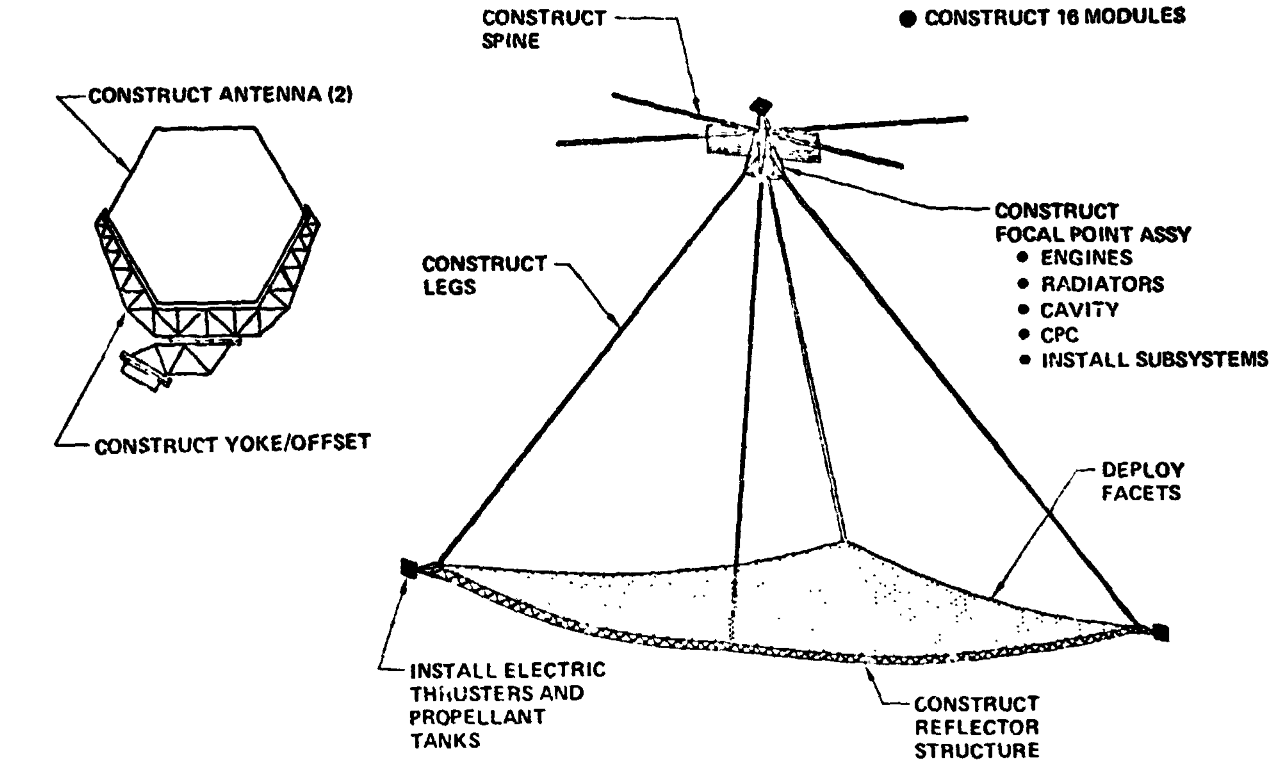

Figure 3.3-3. LEO Base Construction Tasks
Thermal Engine Satellite

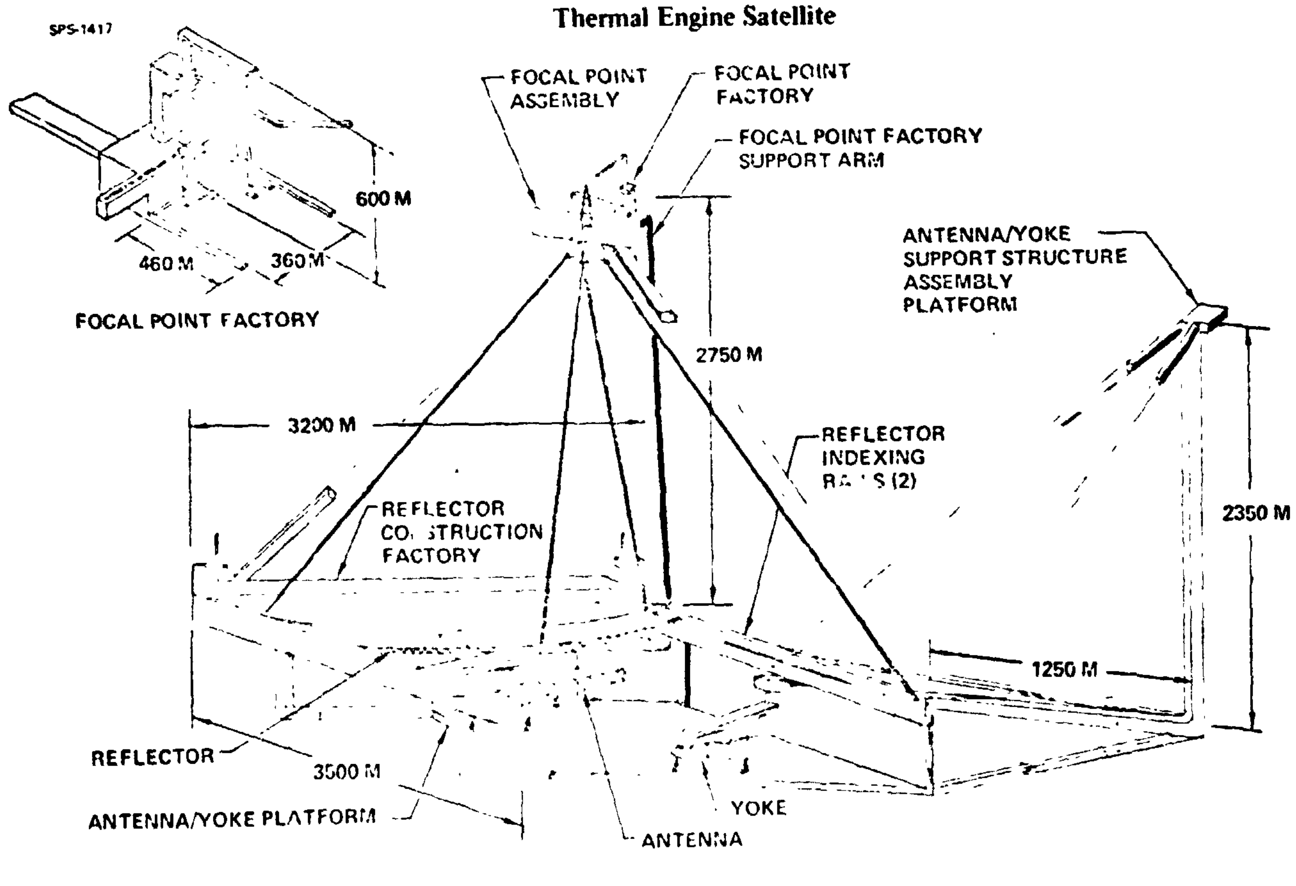

Figure 3.3-4. LEO Construction Base
Thermal Engine Satellite

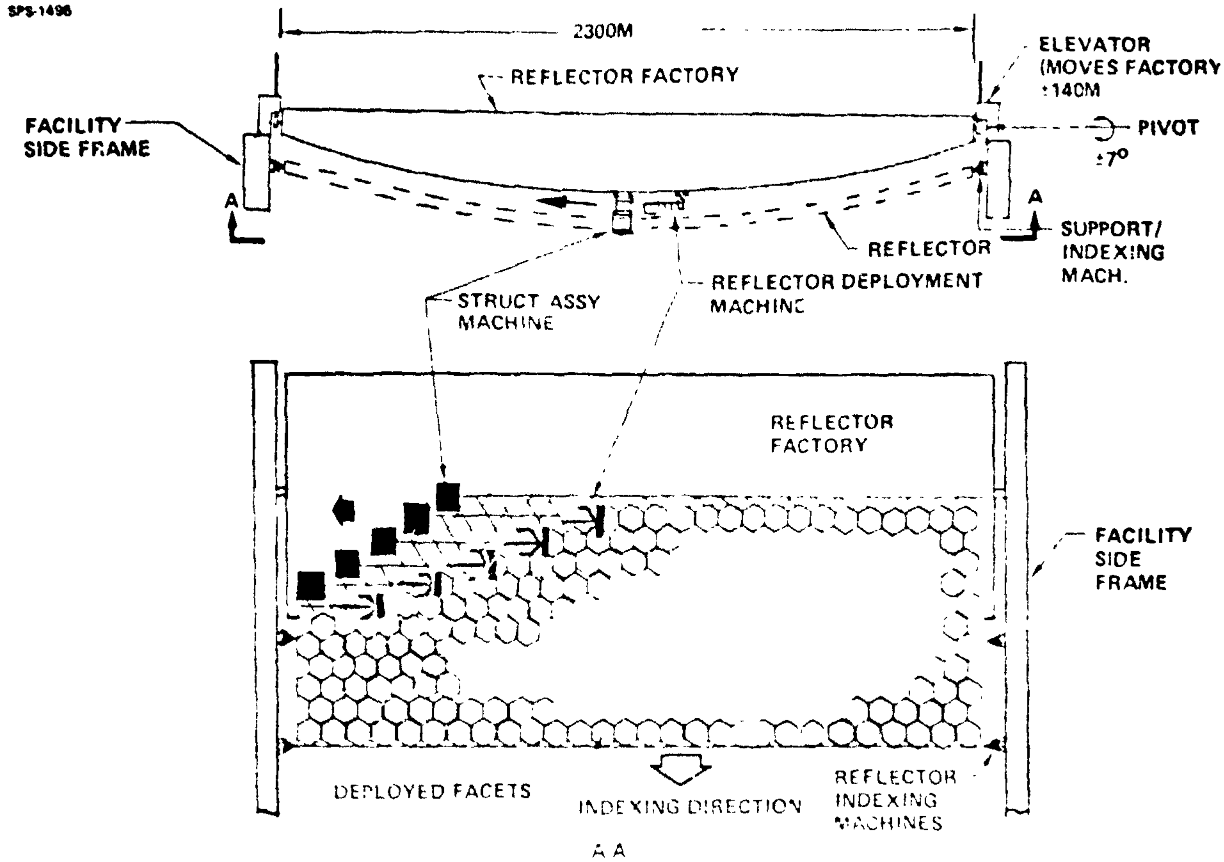

Figure 3.3-5 Reflector Construction Operations

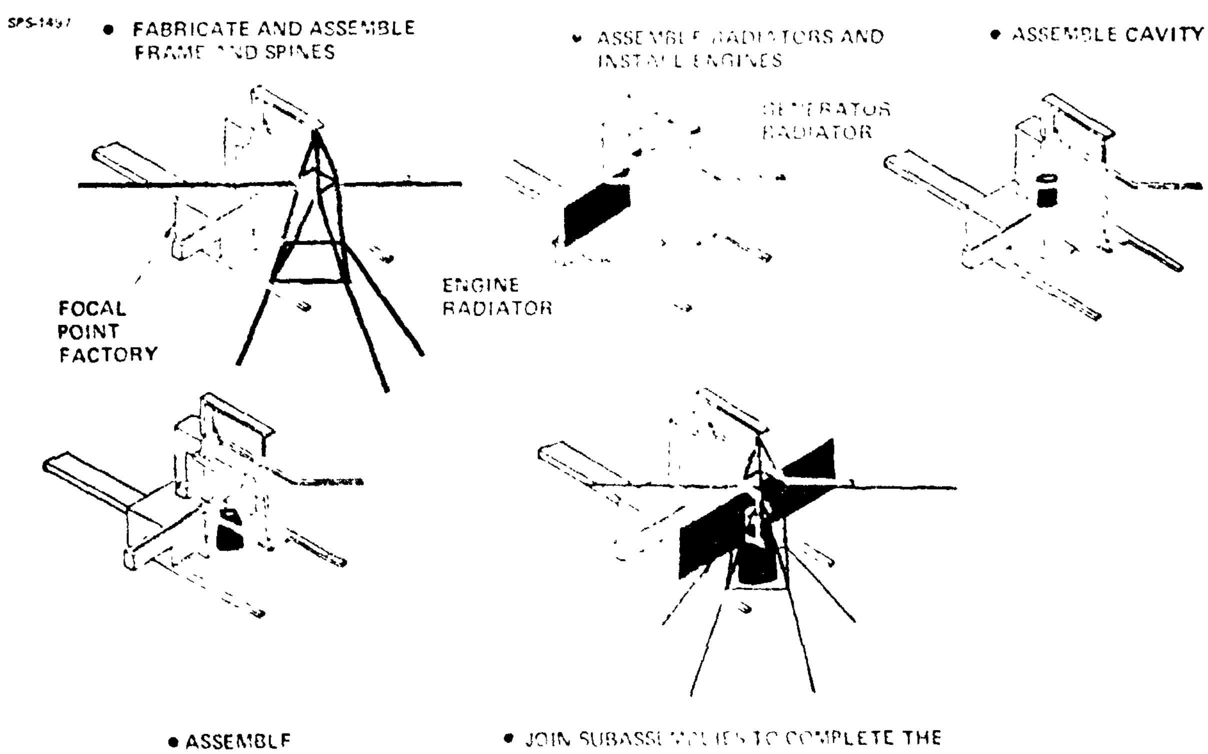

Figure 3.3-6 Focal Point Assembly Operations

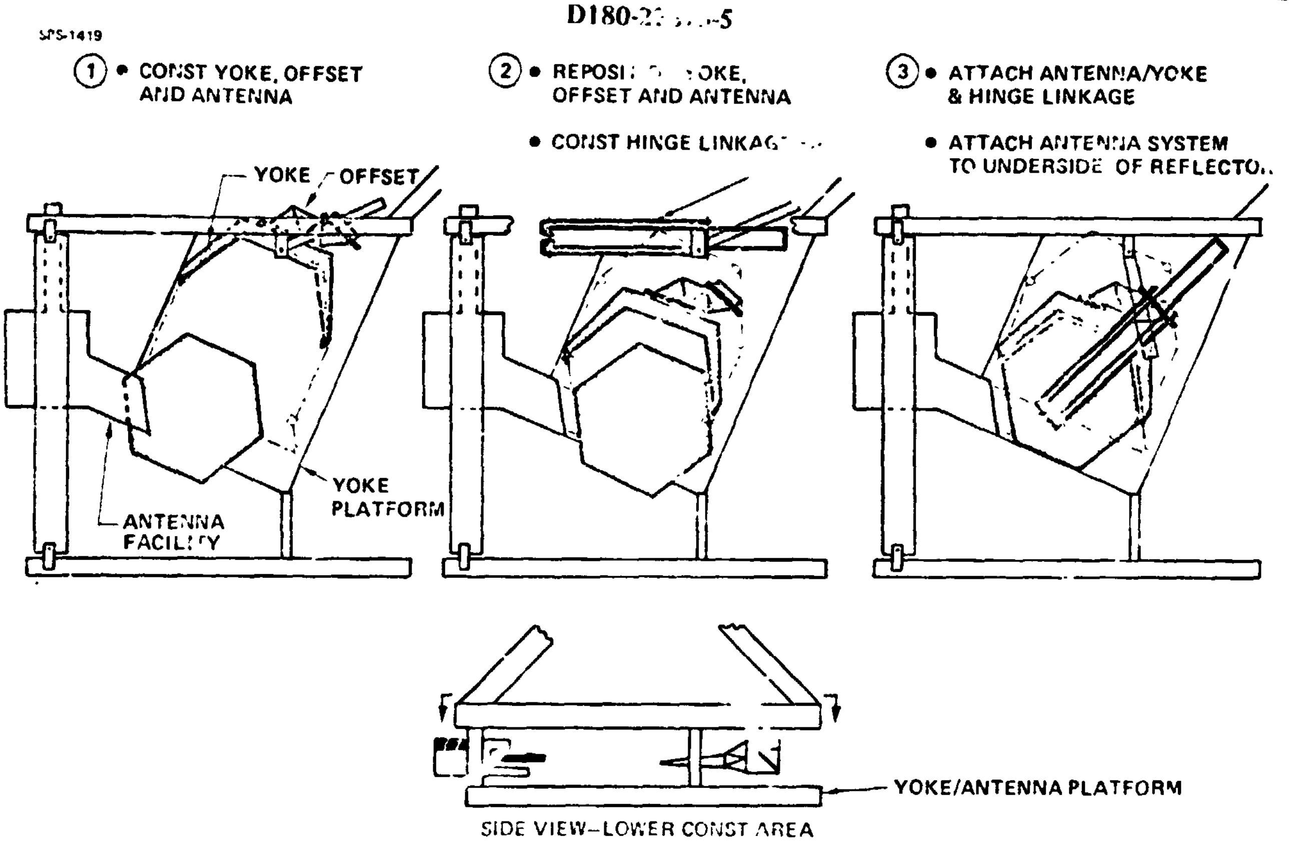

Figure 3.3-7. Antenna/Yoke/Module Assembly

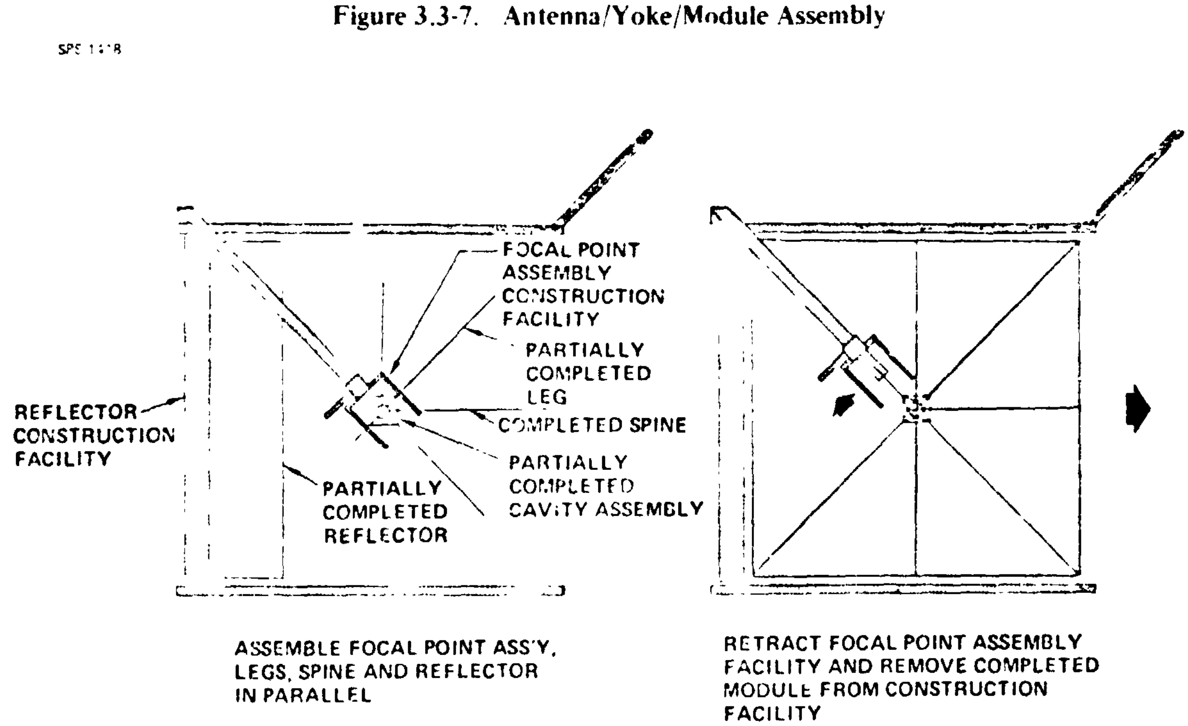

Figure 3.3-8. Mo... Separation

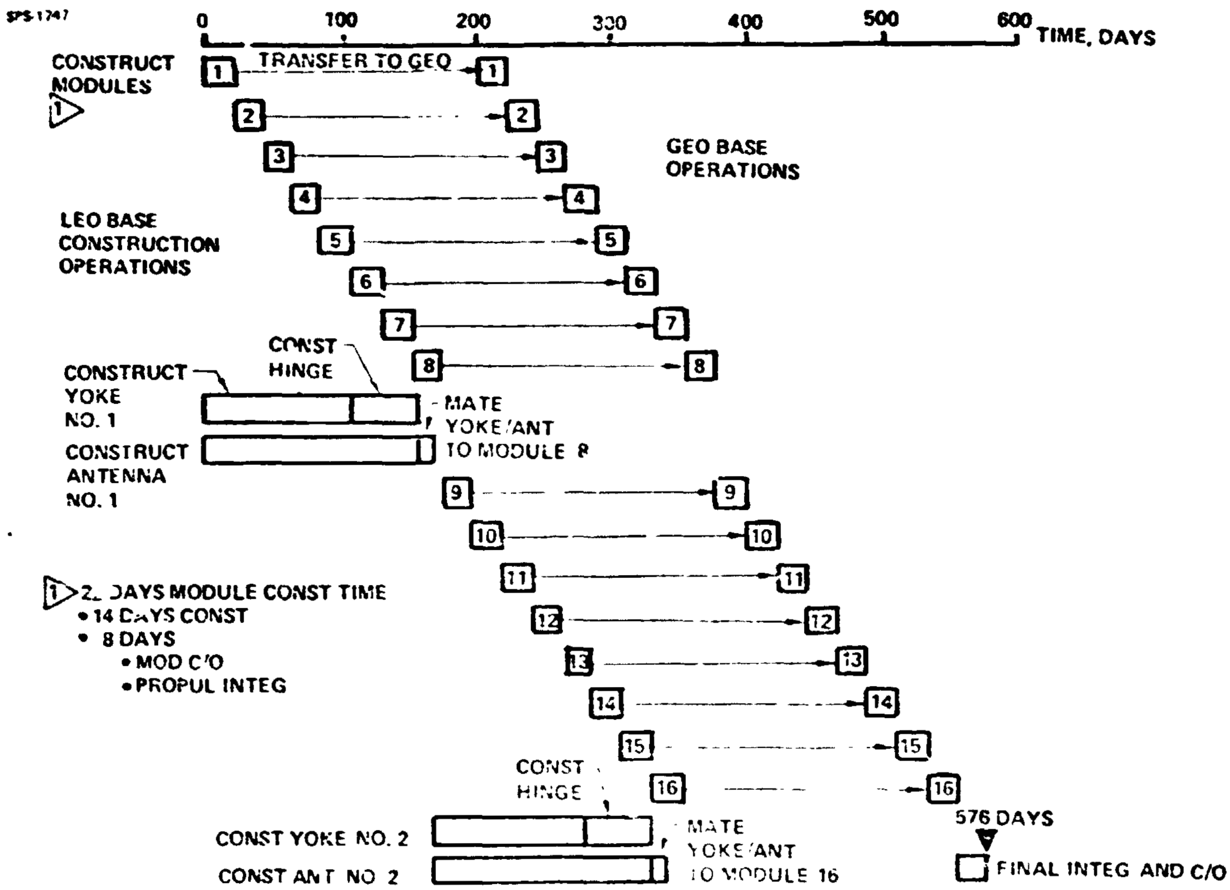

Figure 3.3-9 LEO Construction Timeline

3.3.1.1.1.4.1 Reflector Assembly

The reflector structure is a spherical disk constructed from tetrahedral structural units. The basic structural unit is shown in Figure 3.3-10 along with the structure assembly machine. This machine incorporates 9 strut assembly machines and 7 joint fitting installation mechanisms similar to those described in detail in the photovoltaic satellite structure construction section. A pyramid and a triangle are assembled in two separate areas of the machine. The triangle is assembled directly to previously installed structure. After its assembly, the pyramid section is transported and connected to the triangle and to previously installed structure as illustrated in Figure 3.3-11.

Using the timing constraints imposed on the beam machine described previously, it was found that five of these structure assembly machines must operate in parallel in order to finish the reflector within 14 days. Two operators are assigned to each machine each shift

Facet Deployment

The hexagonal reflector facets are attached directly to the reflector structure via spring-tensioned rocker arms. Figure 3.3-12 shows the facet deployment machine that could be used to perform this task. Figure 3.3-13 shows details of this machine. Figure 3.3-14 shows the facet deployment sequence.

Five of these facet deployment machines are required. Two operators are assigned to each machine each shift.

Integrated Structure Assembly and Facet Deployment Sequence

The five structure assembly machines and the five facet deployment machines operate along the curved surface of the reflector factory as was shown in Figure 3.3-5 and in a staggered alignment as shown in Figure 3.3-15.

In order to construct the curved dish, it is necessary to move the factory up and down through a range of 140m and to rotate it through 7° as shown in Figure 3.3-16. This factory reorientation is accomplished at the end of each pass of the machines.

3.3.1.1.1.4.2 Focal Point Assembly

The focal point assembly is composed of the following subassemblies

- Structure (including legs)
- Cavity
- Compound Parabolic Concentrator (CPC)
- Engine Assembly
- Engine Radiator
- Generator Radiator
- Spine (includes power busses)

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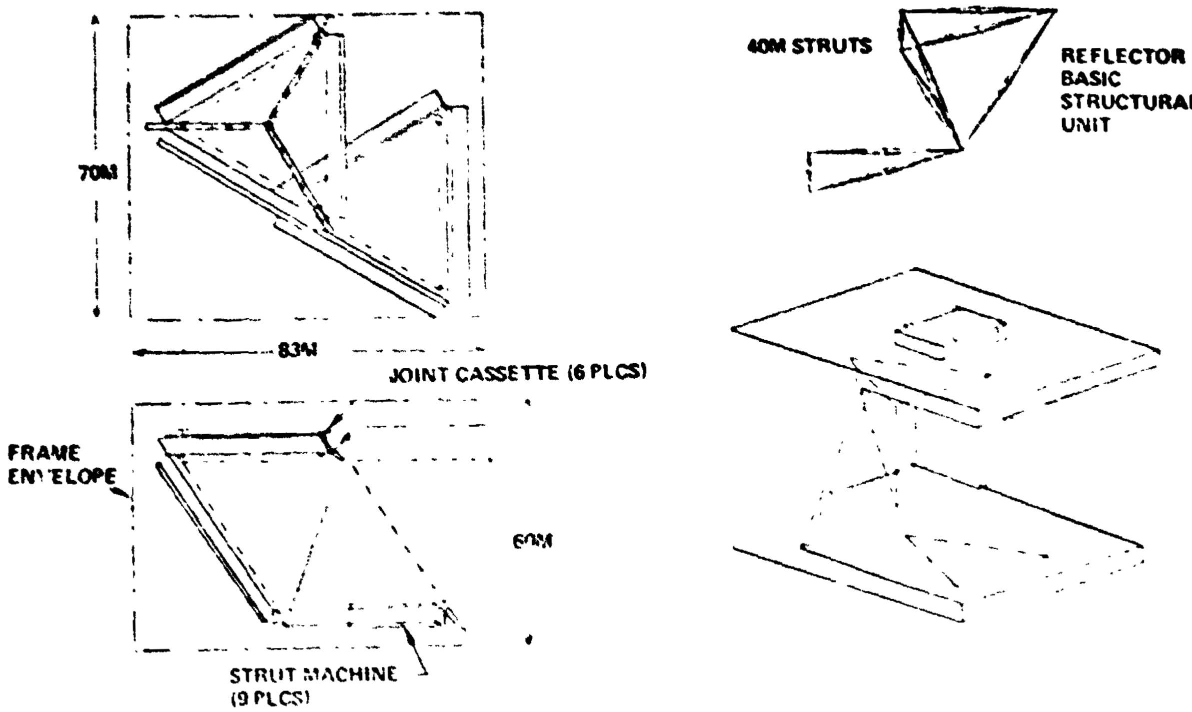

Figure 3.3-10. Structure Assembly Machine

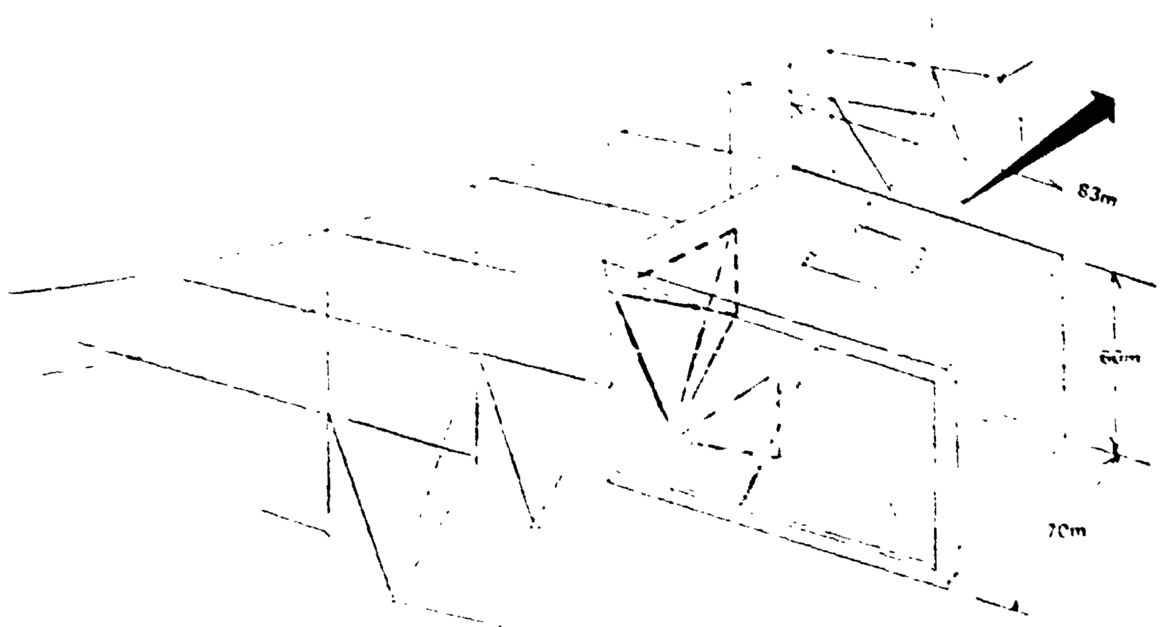

Figure 3.3-11. Structure Assembly Machine Interface

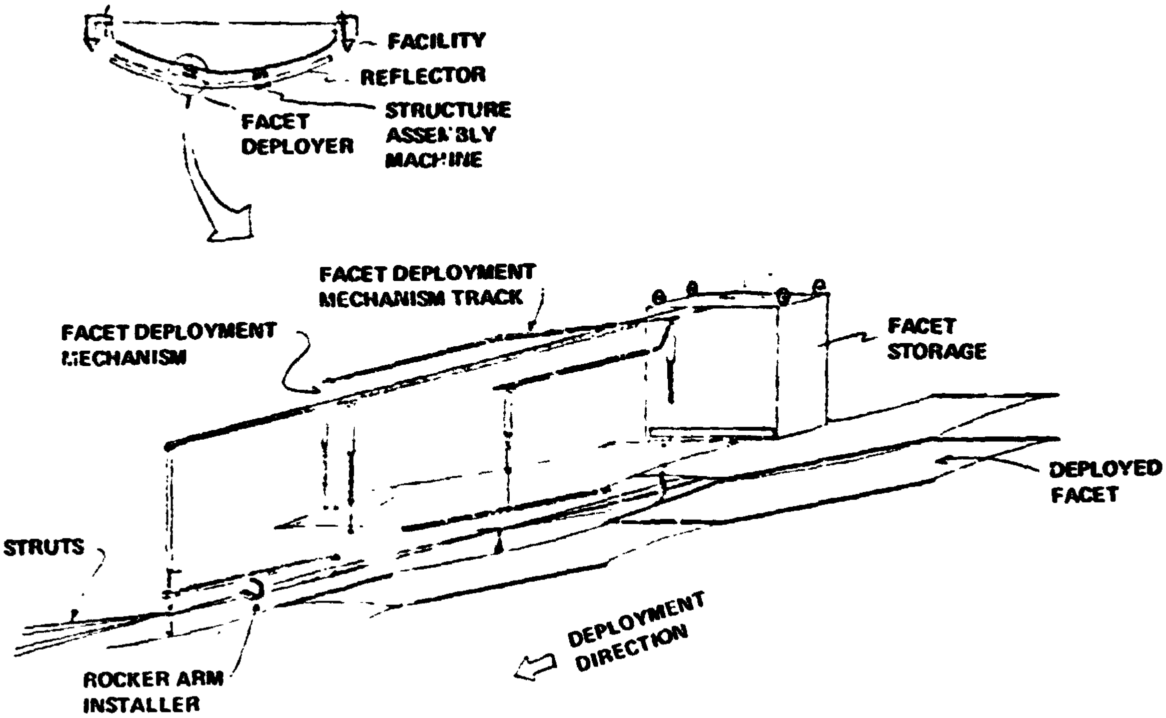

Figure 3.3-12. Facet Deployment Machine

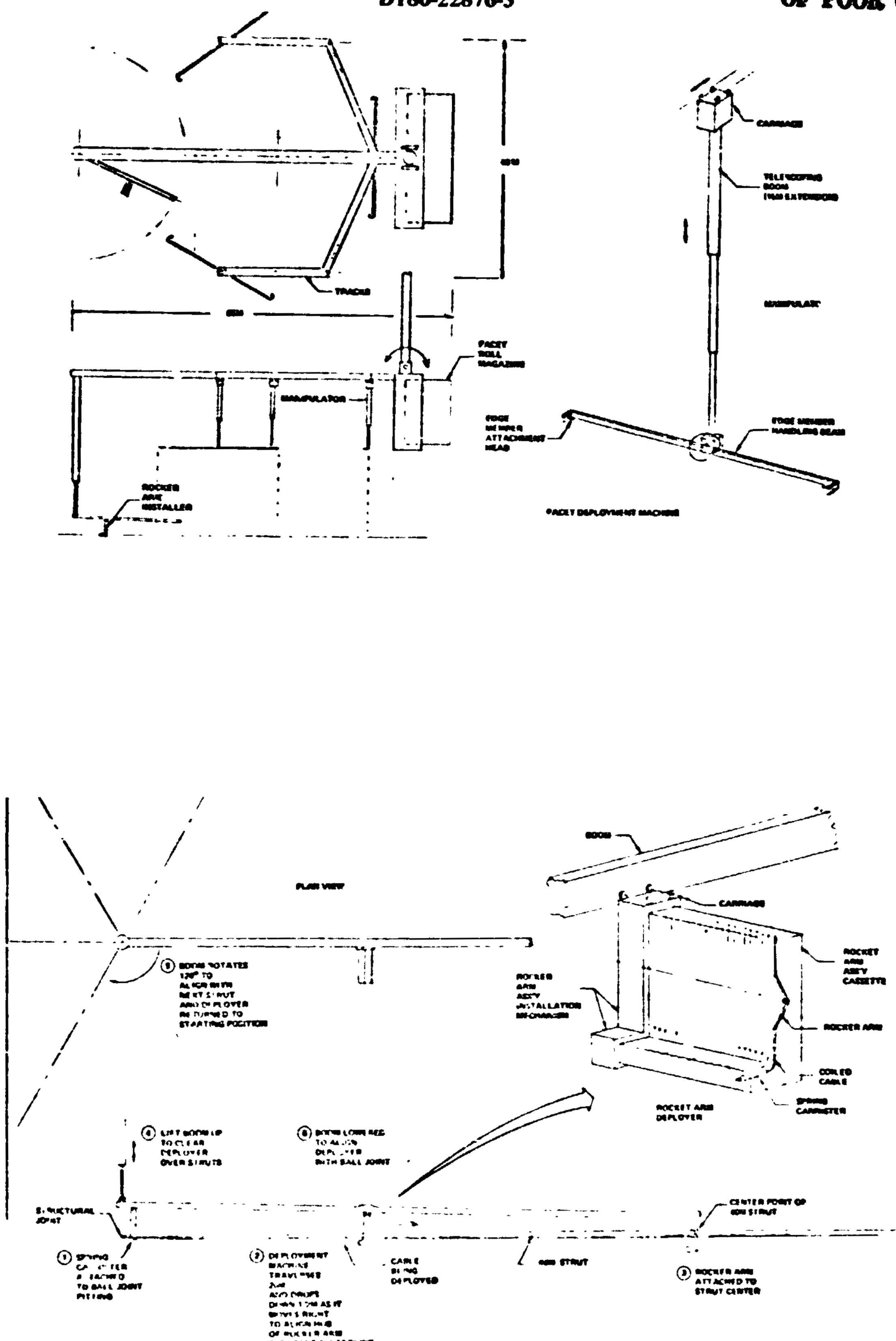

Figure 3.3-13. Facet Deployment Machine Details

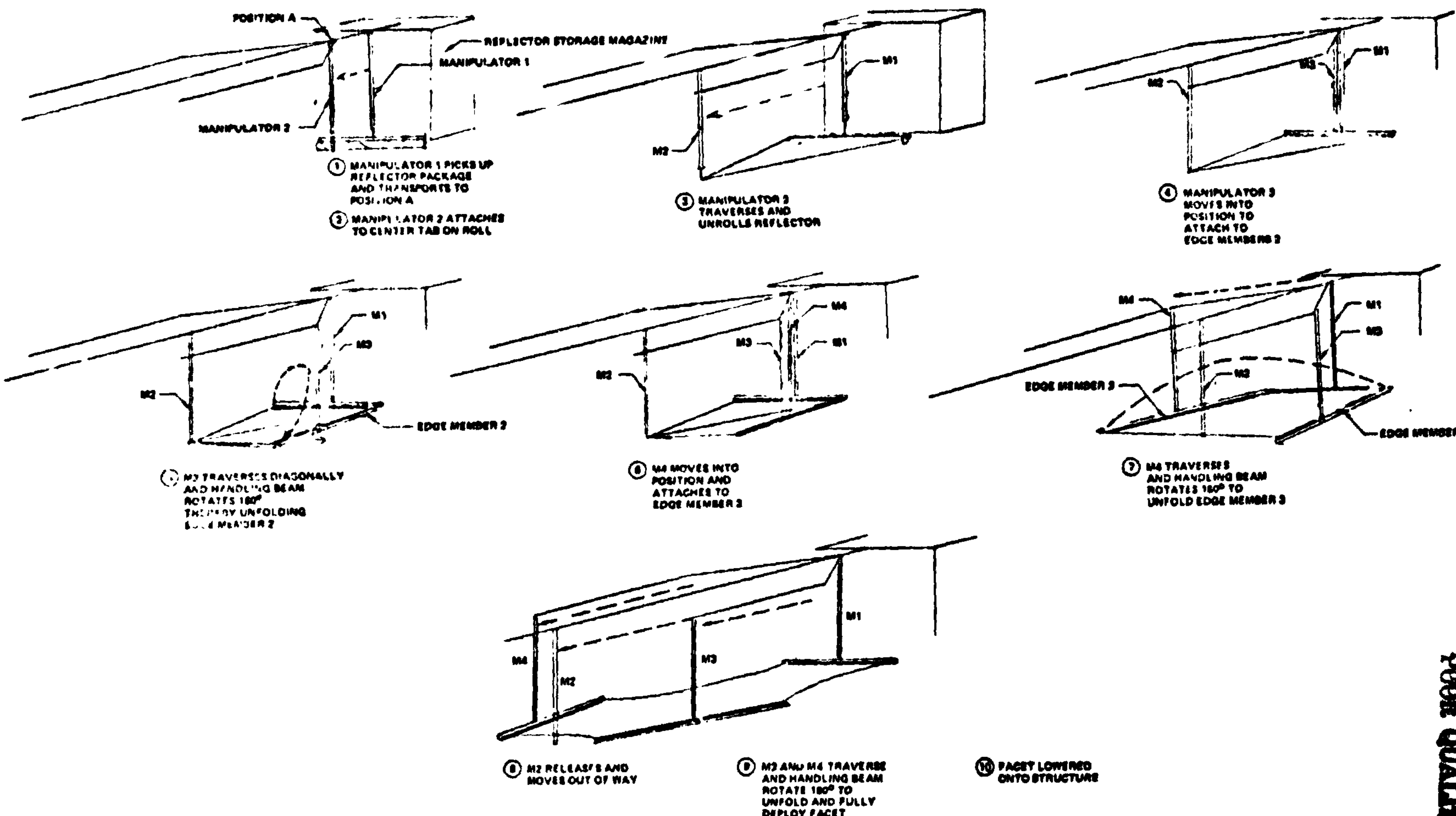

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Figure 3.3-14. Facet Deployment Sequence

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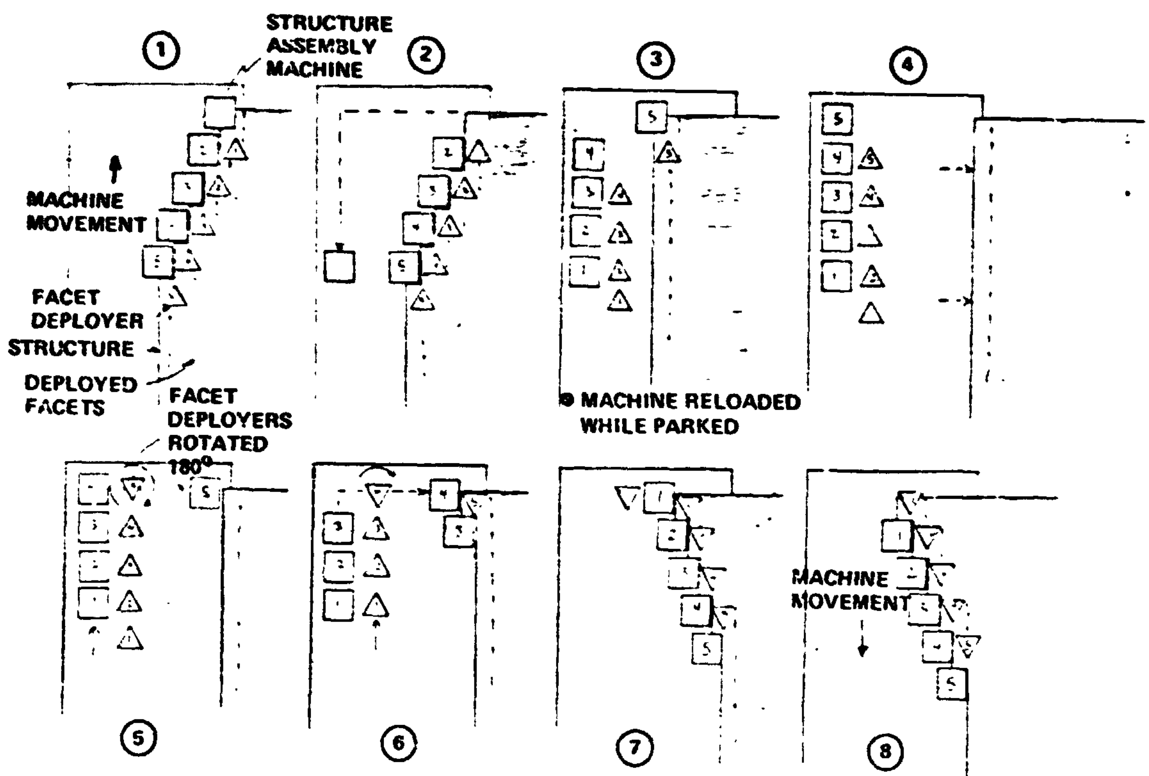

Figure 3.3-15. Reflector Assembly Sequence

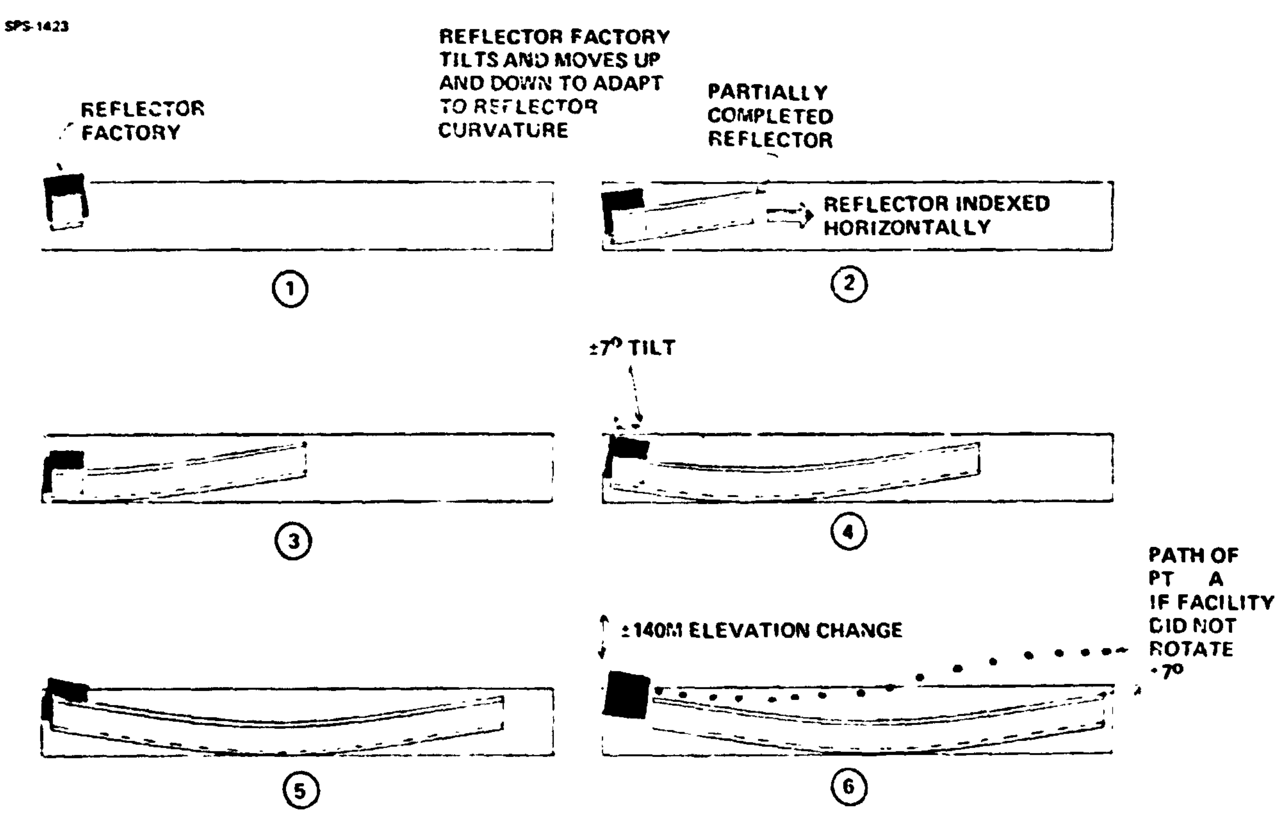

Figure 3.3-16. Reflector Factory Operations

These subassemblies are constructed concurrently in dedicated assembly areas on the focal point factory as shown in Figure 3.3-17

In the following sections, the construction of each of these subassemblies is described separately and then an integrated construction sequence is shown.

Major Subassemblies Construction Concepts

Structure—The structure is fabricated and assembled by a collection of beam machines and manipulators operating from the various surfaces of the focal point factory as shown in Figure 3.3-18.

The 20m beam legs and the antenna support structure are assembled by beam machines and crane/manipulators that operate from the side rails of the facility as shown in Figure 3.3-19.

Cavity and CPC—The cavity and CPC are constructed on turntables in their respective assembly areas. These assemblies are constructed in a two-step, two-station sequence using the construction equipment shown in Figure 3.3-20.

At Station 1, the exoskeleton is fabricated by two 5m beam machines and a 20m crane/manipulator. One of the beam machines makes the longitudinal beams and the other beam machine makes the laterals. These beams are interconnected by the manipulator. The frame is periodically attached to the turntable which is used to index the assembly through the assembly stations.

At Station 2, the panels are transported to their assembly location by a conveyor system that can be inserted into and retracted out of the inner recesses of the cavity or CPC. The panels are removed from the conveyor and attached to the frame by a panel installation machine. On the cavity panel installation machine there is incorporated another manipulator that is used to weld the manifold pipes.

Engine Rib Assembly—The thermal engine assembly is composed of 10m beam to which is attached the thermal engine pallets, the piping associated with the radiators, and the collector power busses.

This integrated assembly can be produced in a 3-bay subassembly factory as shown in Figure 3.3-21. The completed assembly will be "extruded" from this factory at a rate that matches the "extrusion" rate of the engine radiator factory discussed in the next paragraph. This allows the radiator manifold pipes to be welded to the engine manifolds within the facility.

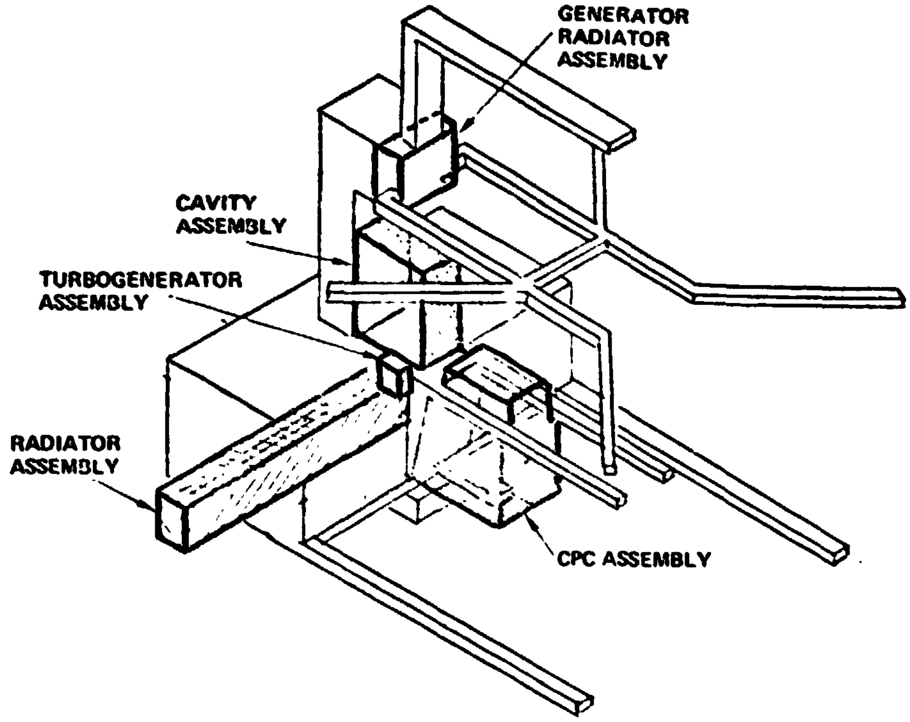

Figure 3.3-17. Focal Point Assembly Areas

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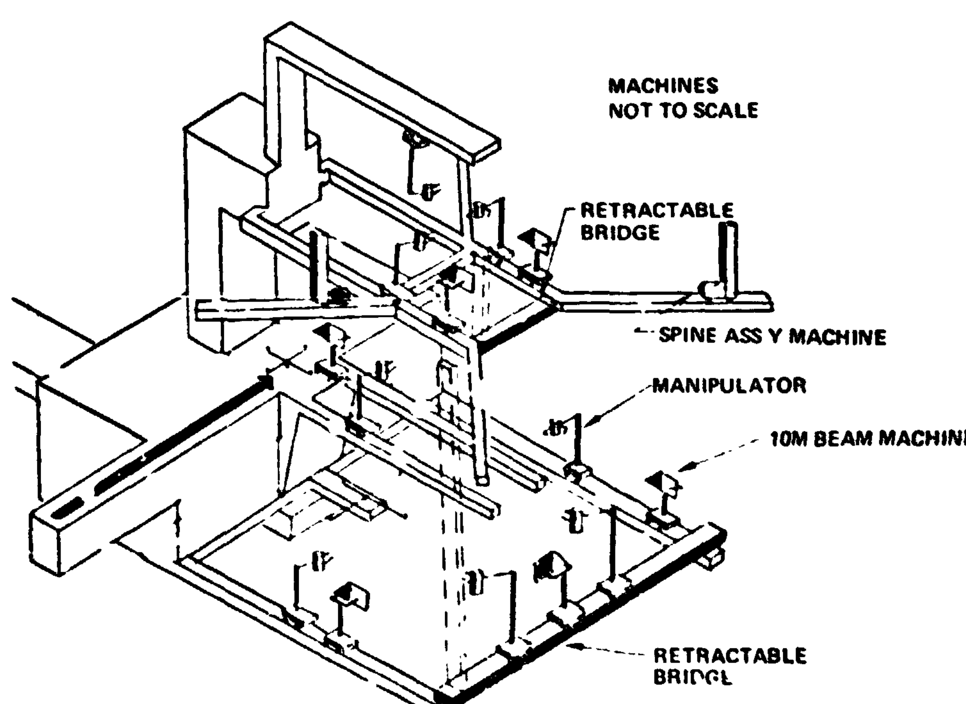

Figure 3.3-18. Super :ructure and Spine Construction Equipment

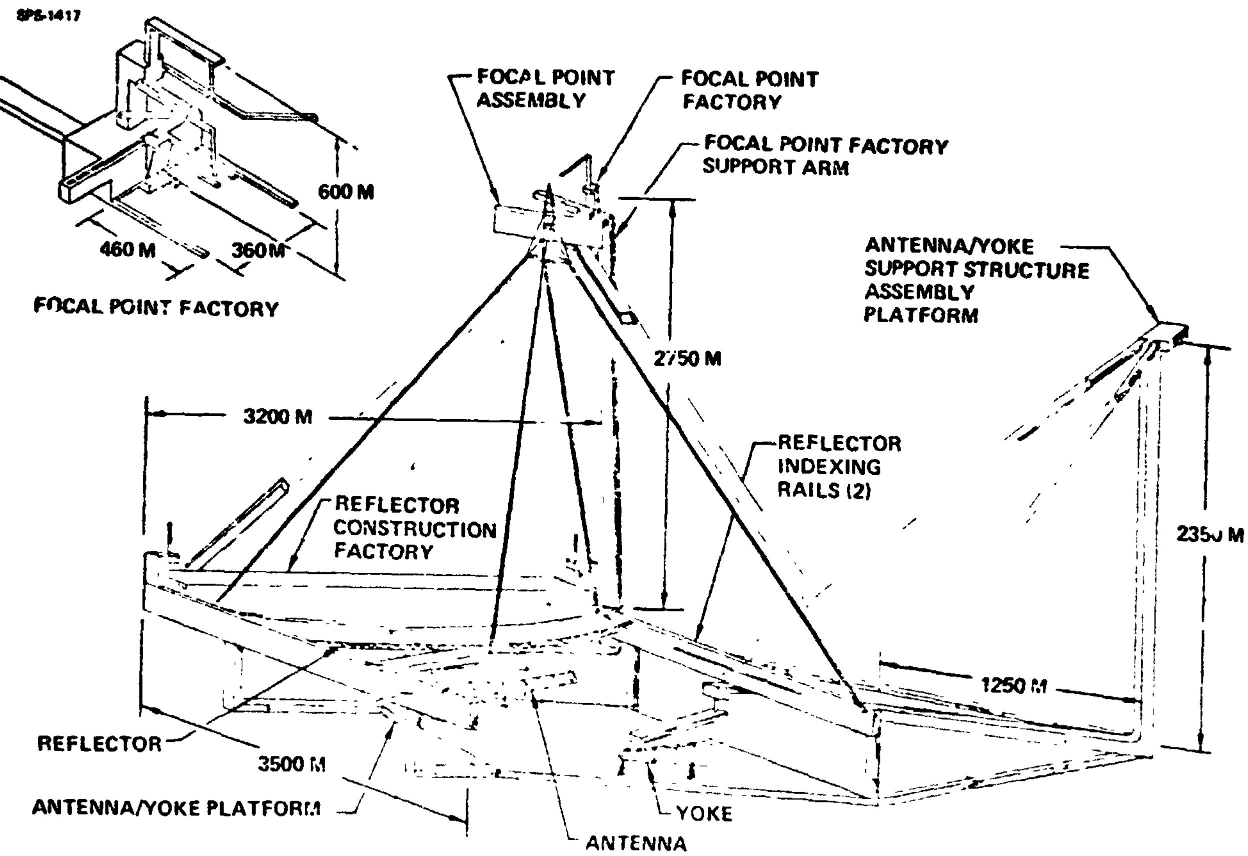

Figure 3.3-19. LEO Construction Base
Thermal Engine Satellite

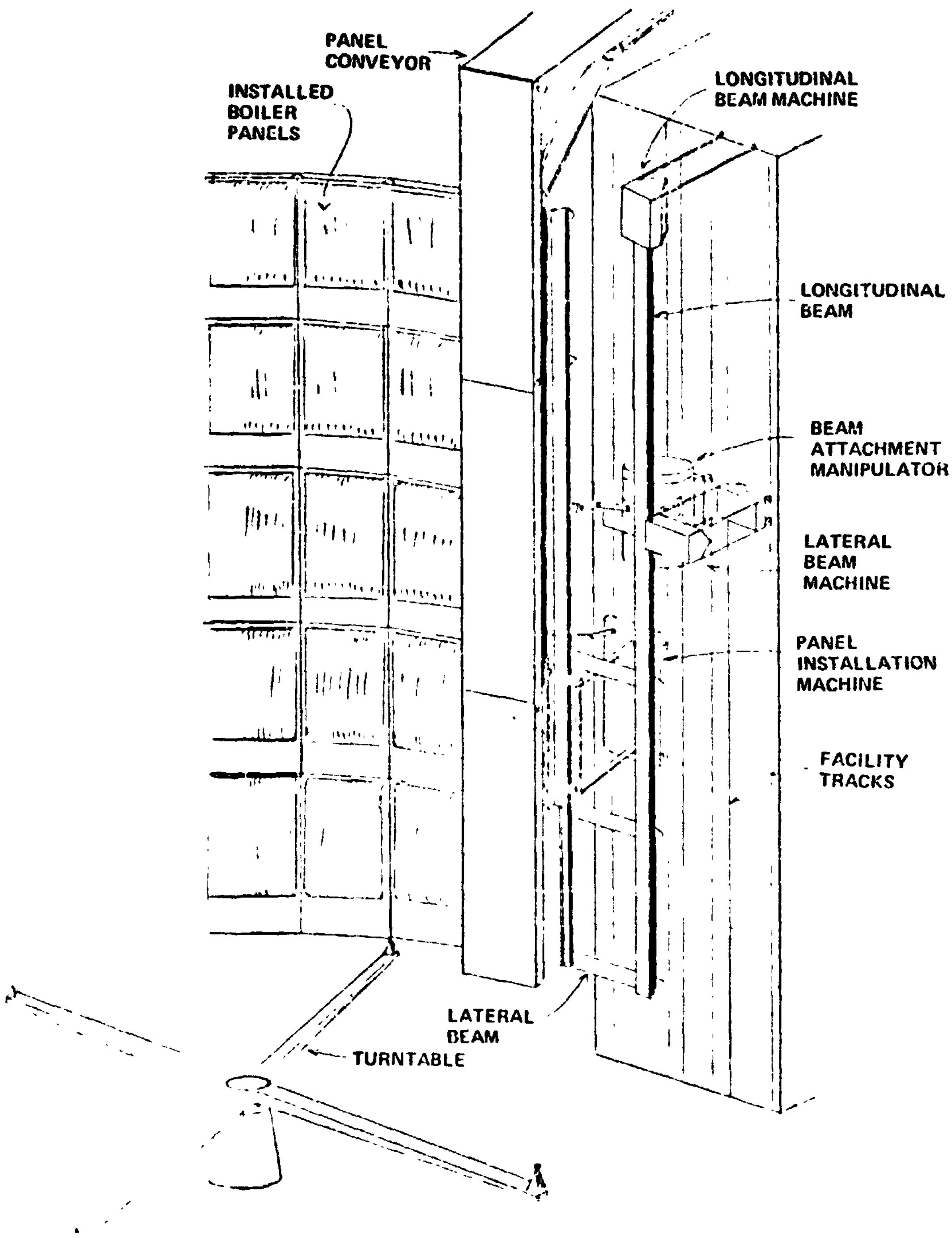

Figure 3.3-20. Cavitv Assembly Equipment

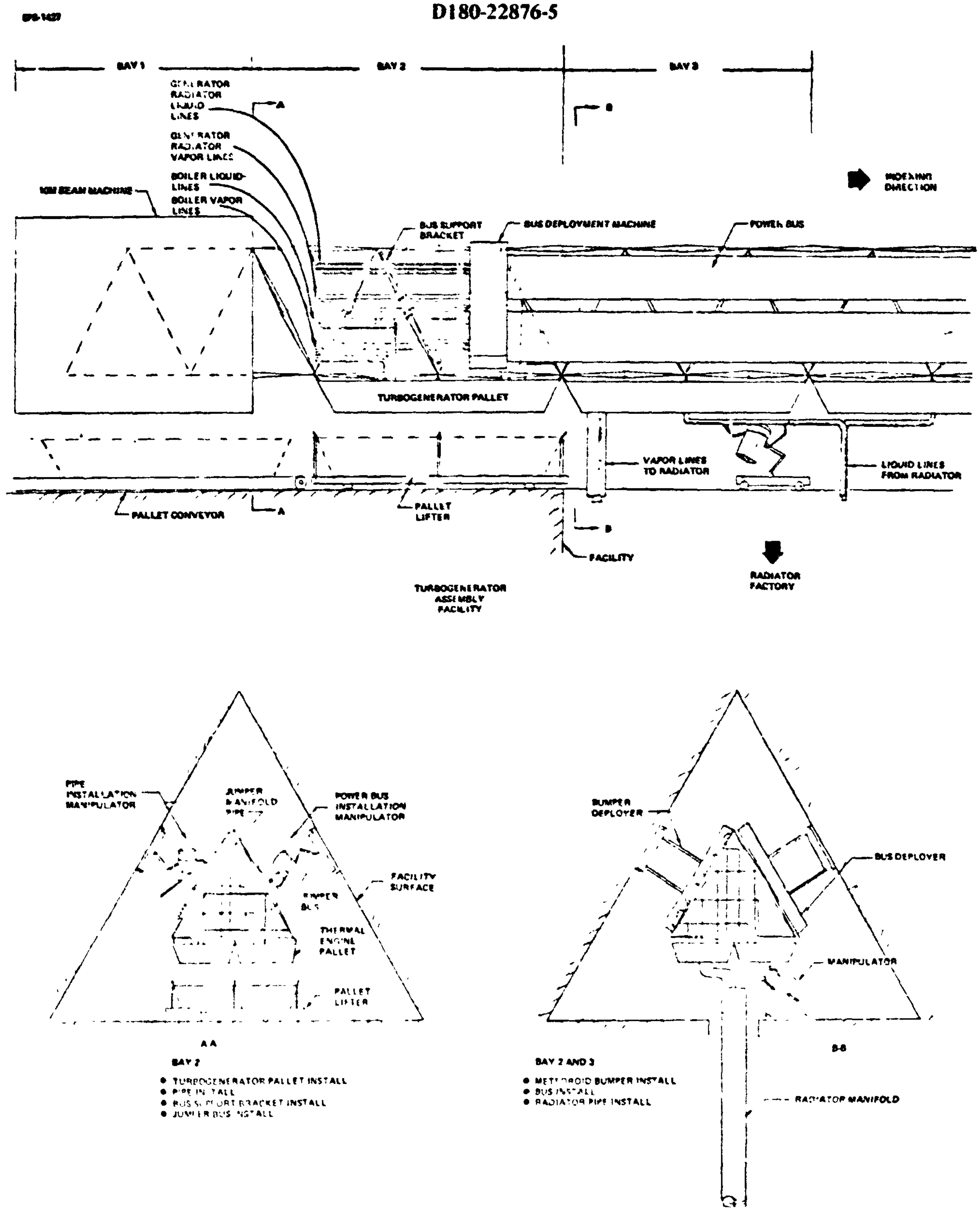

Figure 3.3-21. Engine Rib Assembly

Radiators—The thermal engine and the generator radiators can be produced using a 3-station assembly line as illustrated in Figure 3.3-22.

At the first station, the pipe sections are deliver ·l by a conveyor system to assembly positions where pipe welders attach the pipes end-to-end. The completed pipe is index:d ahead a short distance. The second pipe is then similarly assembled. These two pipes are then attached together by pipe brackets installed by pipe bracket installation machines. The completed 2-pipe set is then indexed to Station 2.

At Station 2, the meteoroid bumpers are attached to the pipes in a 2-step assembly operation. After the bumpers are installed, the pipe set is advanced to Station 3.

At Station 3, the radiator panels are conveyed to their installation positions. A panel installation machine extracts panels from the conveyor and moves the panel to the final assembly position where the ends of the panels are welded to the manifoiu pipes. The completed assembly is then indexed out of the radiator factory.

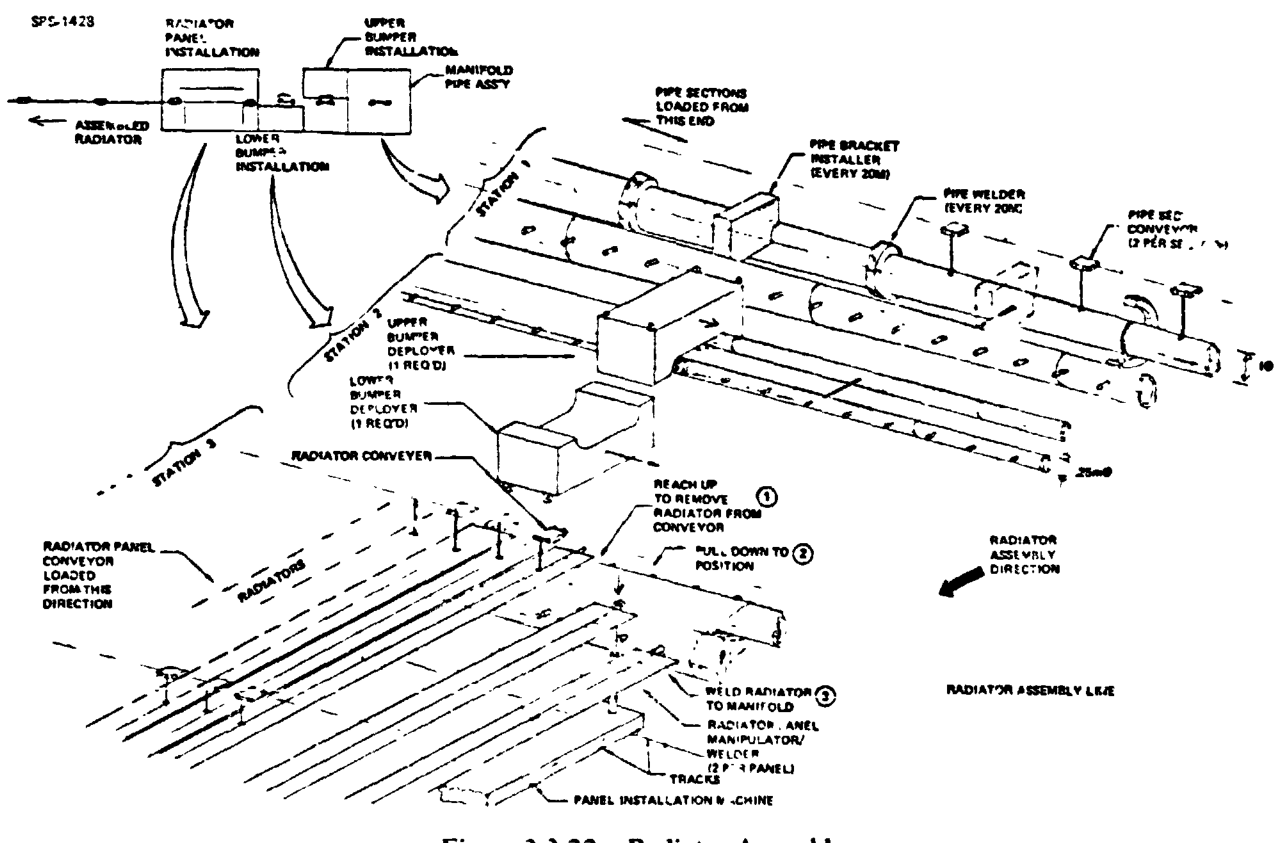

Figure 3.3-22. Radiator A.sembly

SPINE—The spine is composed of a 20 m beam. Some of these spine beams will have power busses attached. This spine assembly can be produced by a machine concept as shown in Figure 3.3-23.

Integrated Focal Point Construction Sequence—The various subassembly operations just described all proceed in parallel in an integrated sequence as shown in Figures 3.3-24 and 3.3-25.

3.3.1.1.1.5 Antenna—Detailed Construction Task Analysis

The antennas are constructed in the antenna factory located below the reflector factory. The antenna construction details are identical to that described in the photovoltaic satellite antenna construction (Sec. 3.2.1.1.1.5) with the exception that for the thermal engine satellite, the antenna is constructed upside down. This difference is due to the thermal engine satellite's unique antenna hinge system.

3.3.1.1.1.6 Yoke—Detailed Construction Task Analysis

The yoke configuration is similar to that used on the photovoltaic satellite. The significant difference is that the thermal engine satellite yoke has an "offset" in it with an extra turntable. A hinge assembly is also required.

Construction of the yoke assembly for the thermal engine satellite requires a special platform and rotating boom as shown in Figure 3.3-26 since no overhead construction provisions are available in the lower area of the construction base.

The sequential mating of the antenna-to-yoke, hinge-to-yoke, and antenna yoke hinge-to-reflector was previously shown in Figure 3.3-7.

3.3.1.1.1.7 Subassemblies

Many of the subassemblies discussed in Section 3.2.1.1.1.7 are required [illegible] is satelli[illegible] Figure 3.3-3.3-27 shows the subassemblies required.

3.3.1.1.1.8 Construction Equipment Summary

The major equipment used to construct the thermal engine satellite and their characteristics is presented in Table 3.3-1, Sheets 1 through 6.

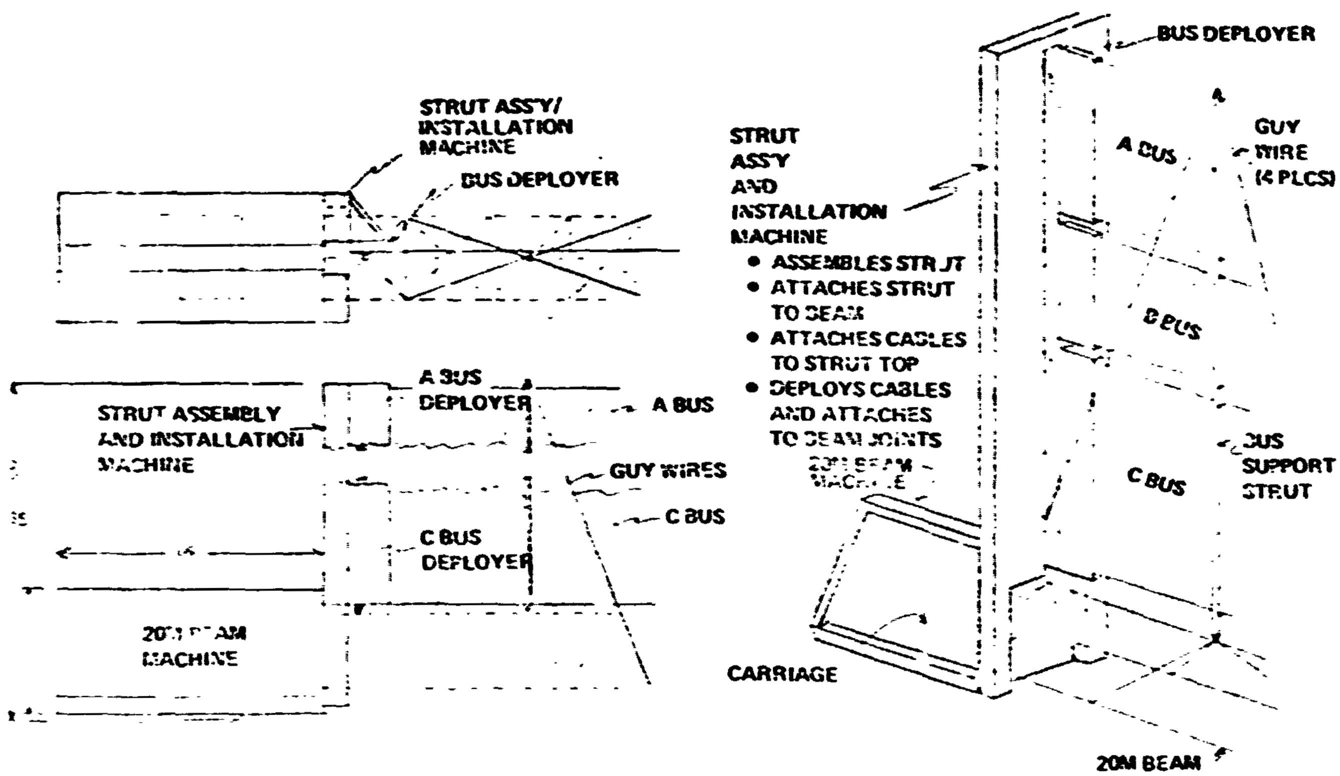

Figure 3.3-23. Spine Assembly Machine

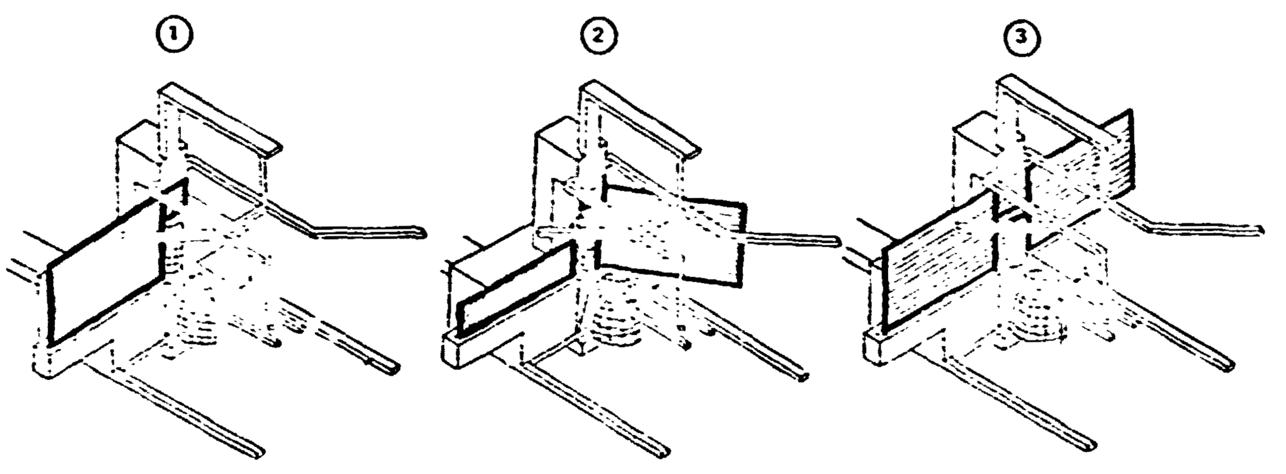

Figure 3.3-24. Focal Point Construction Sequence

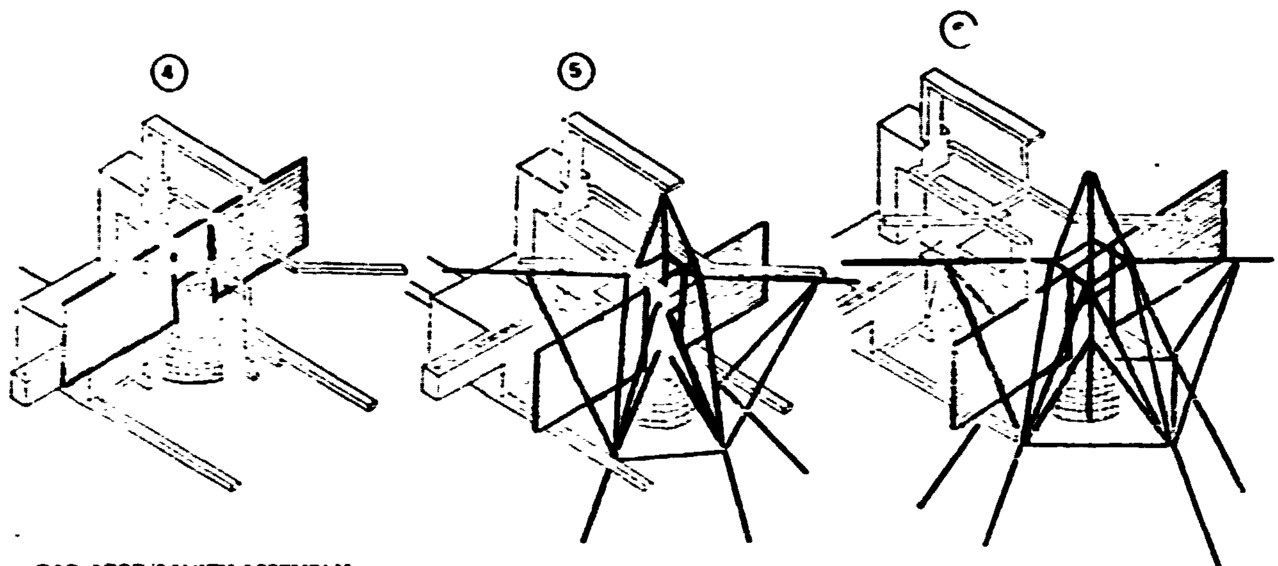

Figure 3.3-25. Focal Point Construction Sequence

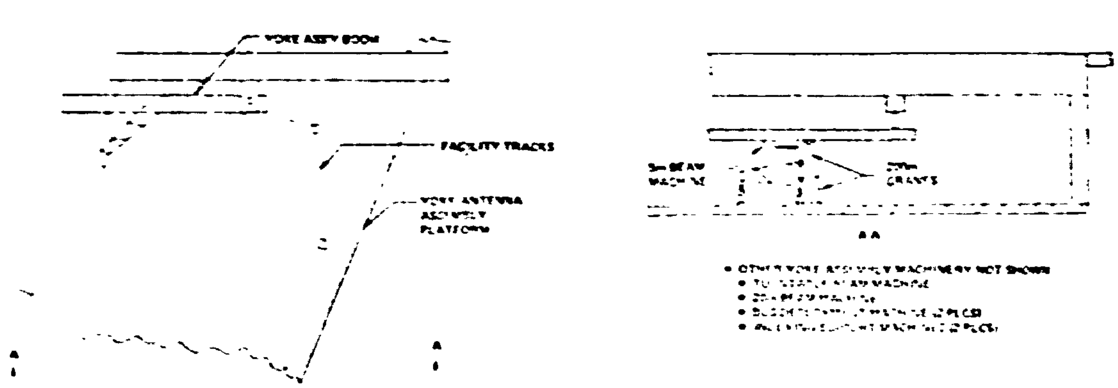

Figure 3.3-26. Yoke Assembly

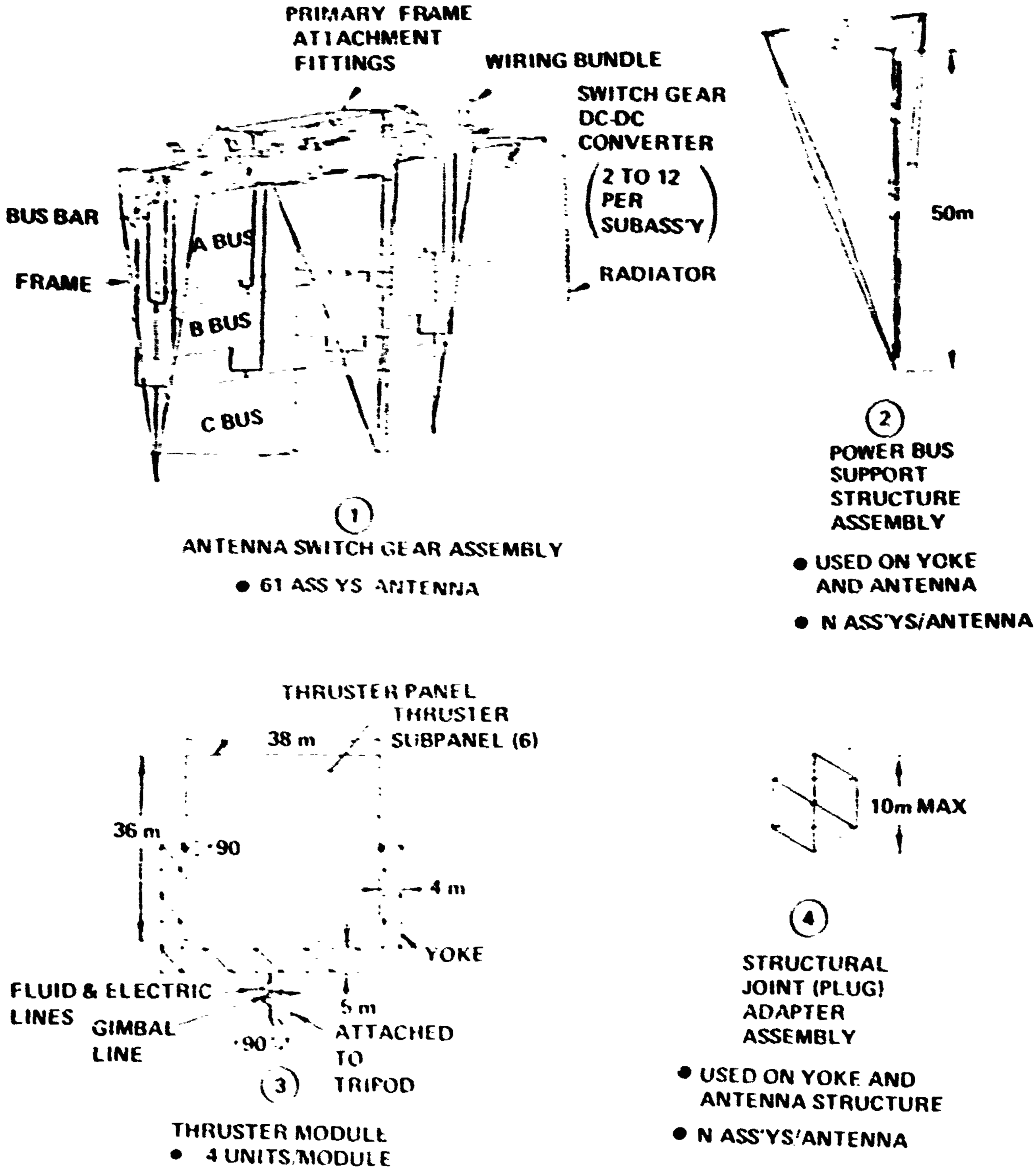

Figure 3.3-27 Thermal Engine Subassemblies

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Table 3.3-1. LEO Base Construction Equipment Sheet 1

EQUIPMENT ITEM	NUMBER REQ'D REFL	FOC PT	ANT	YOKE	SUBASSY	EQUIPMENT ITEM MAJOR ELEMENTS	NO. REQ'D ITEM
• STRUCTURE ASSEMBLY MACHINE (MASS 31^K Kg) (COST $130M) ▷A	5					• CARRIAGE • STRUT ASSY MACH • JOINT FITTING MACH • INDEXING CARRIAGES • JOINT FITTING CARROUSELS • CONTROL CABIN (2 MAN) • STRUT MAGAZINES	1 9 7 6 7 1 9
• FACET DEPLOYMENT MACHINE (MASS 13^K Kg) (COST $48M)	5					• CARRIAGE • FACET STORAGE MAGAZINE • FACET FEED MECH. • FACET DEPLOYMENT MECH. • ROCKER ARM DEPLOY MECH. • CONTROL CABIN (2 MAN)	1 1 1 4 1 1
• 20M MANIPULATOR/CRANE ▷1 CPC 1 CAVITY 1 ENGINES 3 SPINE 2 ▷2 THRUSTERS 2 SA GEAR 4 STRUCTURES 4 (MASS 3^K Kg) (COST [illegible])		7 ▷1	2		10 ▷2	• CARRIAGE • ELEVATOR BOOM • TRANSVERSE BOOM • MANIPULATOR ARMS • CONTROL CABIN (1 MAN)	1 1 1 2 1
• 110M MANIPULATOR/CRANE ▷3 SIDE ARMS 2 FACTORY 1 (MASS [illegible]) (COST [illegible])	3 ▷3	8	10			(SAME AS ABOVE EXCEPT 2 MAN CABS)	
• 2[illegible] MANIPULATOR/CRANE (MASS [illegible]) (COST [illegible])				2		(SAME AS ABOVE EXCEPT 2 MAN CABS)	

▷A ALL COST IN TABLE REFLECT AVG COST PER UNIT AFTER APPLYING LEARNING FACTOR OF 0.9.

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Table 3.3-1. LEO Base Construction Equipment Sheet 2

EQUIPMENT ITEM	NUMBER REQ'D REFL	FOC P4	ANT	YOKE	SUBASSY	EQUIPMENT ITEM MAJOR ELEMENTS	
• 20M BEAM MACHINE ▷4 SPINE 2 LEGS 2 (MASS 20^K Kg) (COST $94M)		4 ▷4		1		• CARRIAGE • YOKE ASSY • STRUT ASSY MACH • JOINT FITTING MECH • INDEXING CARRIAGES • STRUT MAGAZINES • JOINT FITTING CARROUSELS • CONTROL CAB (2 MAN)	1 1 9 3 6 8 3 1
• 10M BEAM MACHINE ▷5 SUPERSTRUCTURE 6 ENGINE RIB 1 CAVITY RING 1 (MASS 11^K Kg) (COST $53M)		8 ▷5				(SAME AS ABOVE)	( )
• 5M BEAM MACHINE ▷6 CAVITY 2 CPC 2 (MASS 7^K Kg) (COST [illegible])		4 ▷6	2	1		(SAME AS ABOVE EXCEPT 1 MAN CAB)	( )
• 5M TURNTABLE BEAM MACHINE (MASS 10^K Kg) (COST [illegible])				1		(SAME AS ABOVE EXCEPT 1 MAN CAB)	( )
• 350M RETRACTABLE BRIDGE (MASS) (COST)		1				• CARRIAGE • STRUCTURE	1 1
• 100M RETRACTABLE BRIDGE (MASS) (COST)		1				• CARRIAGE • STRUCTURE	1 1

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Table 3.3-1. LEO Base Construction Equipment Sheet 3

EQUIPMENT ITEM	NUMBER REQ'D REFL	FOC PT	ANT	YOKE	SUBASSY	EQUIPMENT ITEM MAJOR ELEMENTS	NO. REQ'D ITEM
• YOKE ASSY BOOM (MASS) (COST)				1		• HINGE ASSY • BOOM STRUCTURE	1 1
• BUS DEPLOYMENT MACHINE 7▷ REQ'D ON YOKE MACH ONLY 8▷ • BOOM NOT REQ'D ON SPINE MACH OR ENG ASSY LINE • ARTICULATING BOOM REQ'D ON ONE OF THE YOKE MACHINES (MASS 3ᴷ Kg) (COST $21M)		3	1	2		• CARRIAGE 7▷ • BOOM 8▷ • A BUS DEPLOY MACH • B BUS DEPLOY MACH • C BUS DEPLOY MACH • CONTROL CAB (2 MAN)	1 1 1 1 1 1
• SPINE ASSY MACHINE 9▷ INCLUDED IN 20M BEAM MACH COUNT 10▷ INCLUDED IN 20M MANIP/CRANE COUNT 11▷ INCLUDED IN BUS DEPLOYMENT MACH COUNT (MASS 5ᴷ Kg) (COST $25M)		2				• CARRIAGE • 20M BEAM MACH • STRUT ASSY MACH • 20M MANIPULATOR/CRANE • BUS DEPLOYMENT MACH • CONTROL CAB (2 MAN)	1 1 9▷ 1 1 10▷ 3 11▷ 1
• CAVITY ASSEMBLY AREA EQUIP 12▷ INCLUDED IN 5M BEAM MACH COUNT 13▷ PANEL CONVEYOR AND PANEL INSTALLER OPERATORS USE THIS CAB (MASS 10ᴷ Kg) (COST $40M)						• 5M BEAM MACH • 20M MANIP/CRANE • PANEL INST MACH • PANEL CONVEYOR • CONTROL CAB (2 MAN) • TURNTABLE	2 12▷ 1 10▷ 1 1 1 13▷ 1

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Table 3.3-1. LEO Base Construction Equipment Sheet 4

EQUIPMENT ITEM	NUMBER REQ'D REFL	FOC PT	ANT	YOKE	SUBASSY	EQUIPMENT ITEM MAJOR ELEMENTS	NO. REQ'D ITEM
• CPC ASSEMBLY AREA EQUIP (MASS 10ᴷ Kg) (COST $40M)						(SAME AS CAVITY ASS'Y AREA)	( )
• THERMAL ENGINE ASSEMBLY LINE 14▷ INCLUDED IN 10M BEAM MACH COUNT 15▷ INCLUDED IN 20M MANIP/CRANE COUNT 16▷ INCLUDED IN BUS DEPLOYER COUNT 17▷ ATTACHMENT ON MANIP/CRANE (MASS 14ᴷ Kg) (COST $45M)						• ENGINE PALLET CONVEYOR • ENGINE PALLET ELEVATOR • 10M BEAM MACH • 20M MANIP/CRANE • BUMPER INSTALLER • BUS DEPLOYER • PIPE WELDER 17▷ • BUS WELDER 17▷ • CONTROL CAB (2 MAN)	1 1 1 14▷ 3 15▷ 1 1 16▷ 2 1 1
• ENGINE RADIATOR ASSEMBLY LINE (MASS 27ᴷ Kg) (COST $130M)						• CONTROL CAB (15 MAN) • PIPE WELDERS • BRACKET INSTALLERS • BUMPER DEPLOYER • PANEL INST MACH • INDEXING GANTRY • WELD TESTER	1 17 17 2 2 1 3

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Table 3.3-1. LEO Base Construction Equipment Sheet 5

EQUIPMENT ITEM	NUMBER REQ'D					EQUIPMENT ITEM MAJOR ELEMENTS	NO. REQ'D
	REFL	FOC PT	ANT	YOKE	SUBASSY		ITEM
• GENERATOR RADIATOR ASSEMBLY LINE EQUIP (MASS 11^K Kg) (COST $30M)						• CONTROL CAB (5 MAN) • PIPE WELDER • BRACKET INSTALLER • BUMPER DEPLOYER • PANEL INST MACH • INDEXING GANTRY • WELD TESTER	1 1 1 2 1 1 1
• SUB-ARRAY TEST MACHINE					1		
• 10M INDEXING/SUPPORT MACH		4					
• 40M INDEXING/SUPPORT MACH	8		6	2		• 1.3^K Kg • $3.5M	
• 200M INDEXING/SUPPORT MACH				2		• 5^K Kg • $10M	
• BUS BAR ROD BENDER					1		
• BUS BAR WELDER					2		
• RADIATOR PIPE WELDER					1		
• STRUT ASSY MACHINE (4 SIZES)					11		
• CONTROL CABS (2 MAN)					4		

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Table 3.3-1. LEO Base Construction Equipment Sheet 6

EQUIPMENT ITEM	NUMBER REQUIRED					EQUIPMENT ITEM MAJOR ELEMENTS	NO. REQ'D
	REFL	FOC P4	ANT	YOKE	SUBASSY		ITEM

• DEPLOYMENT PLATFORM
18 INCLUDED IN 20M MANIPULATOR/CRANE COUNT (MASS 28^K Kg) (COST $50M)
						• CONTROL CAB (2 MAN)
						• CARRIAGE/FRAME ASSY
						• SECONDARY STRUCTURE INST. TELESCOPES
						• SECONDARY STRUCTURE GANTRY
						  • GANTRY
						  • 20M MANIPULATOR
						  • DETELESCOPING MACH
						• SUBARRAY DEPLOYER
						  • GANTRY
						  • ELEVATOR
						  • DEPLOYER
						    • MAGAZINE
						    • CARRIAGE
						    • DEPLOYMENT MECH
						    • CONTROL CAB (2 MAN)
						    • MANIPULATOR
							1
							1
							3
							
							1
							1 18
							1
							
							1
							1
							
							1
							1
							1
							1
							1 18

3.3.1.1.2 LEO Construction Base

3.3.1.1.2.1 Configuration

The overall configuration of the base has previously been presented in Figure 3.3-4. No additional detail was developed. Crew modules are located near the antenna construction facility.

3.3.1.1.2.2 Foundation

The foundation or structure of the base generally uses 20 m beams forming either 100 m or 200 m trusses. A total requirement of 800,000 m of 20 m beam was estimated.

3.3.1.1.2.3 Cargo Handling-Distribution System

The peculiar configuration of the thermal engine satellite facilities has created a need for a significantly more complex logistics system than was described for the photovoltaic satellite.

The thermal engine satellite facility logistics network for the lower construction areas and focal point are shown respectively in Figures 3.3-28 and 3.3-29. A logistic network also is required on the reflector factory and up and down one of the construction base legs.

A summary of the cargo handling and distribution equipment is presented in Table 3.3-2.

3.3.1.1.2.4 Crew Modules

The types of crew modules used for the thermal engine satellite construction are the same as for the photovoltaic satellite described in Section 3.2.1.1.2.4. Three additional crew quarters modules (total of 9) are required however, since the thermal engine satellite requires 300 additional people for construction.

3.3.1.1.2.5 Base Subsystems

The electrical power system again relies on solar arrays for primary power and nickel hydrogen batteries during the occultation periods. Solar cells having 14% efficiency (10 m^2/Kw) were used for the thermal engine satellite construction base rather than 17% as for the photovoltaic satellite since for a thermal engine satellite program, it was judged that not as much emphasis would be placed on improving the performance of solar cells.

The power load for the LEO thermal engine construction base is 6,000 Kw rather than 3,700 Kw as for the photovoltaic satellite construction base due to the 300 additional people and more construction equipment.

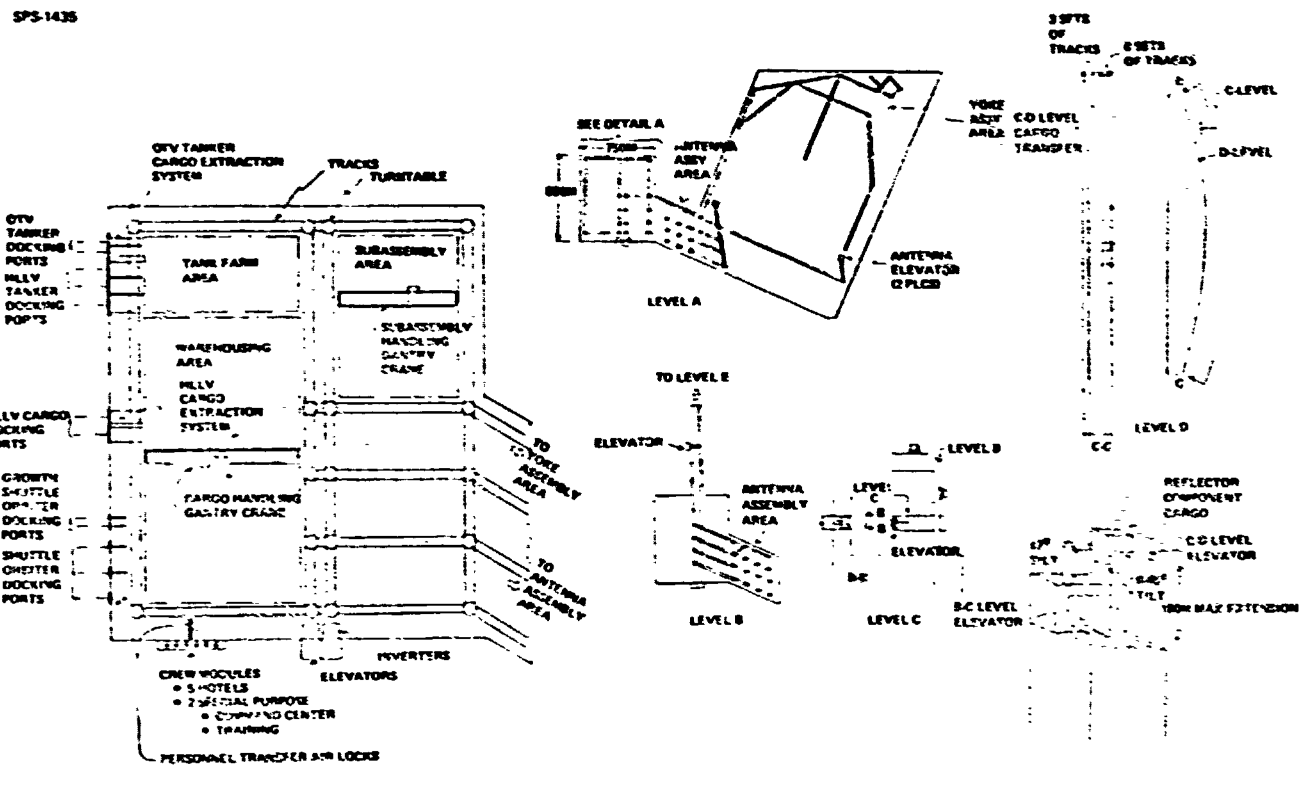

Figure 3.3-28. Lower Construction Area Logistics Network

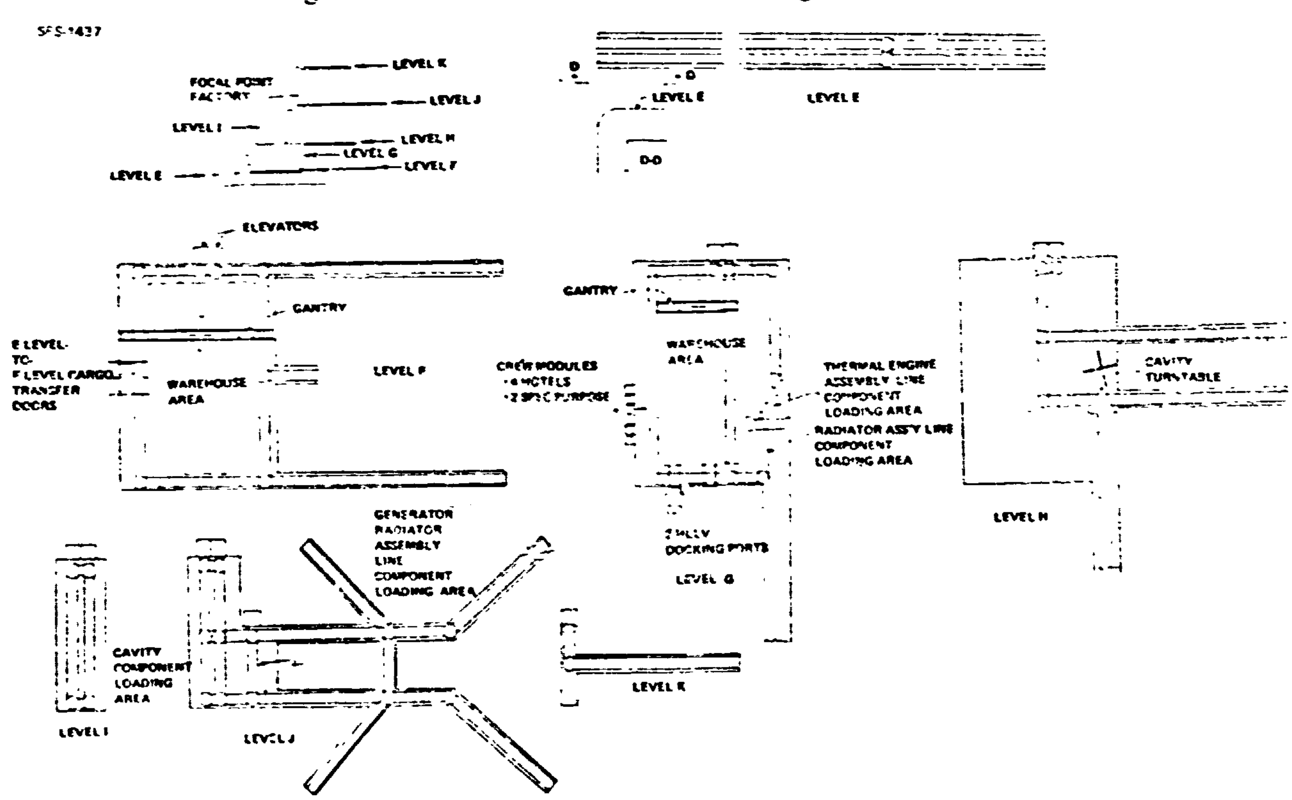

Figure 3.3-29. Focal Point Logistics Network

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Table 3.3-2. LEO Base Logistics Equipment

EQUIPMENT ITEM	NUMBER REQ'D					MASS (EA) 10^3 Kg	COST (EA) $\$10^6$M
	REFL	FOC PT	ANT	YOKE	SUBASSY		
• HLLV CARGO DOCKING PORT		2	2			6	11
• HLLV CARGO EXTRACTION SYS		2	2			3	6
• HLLV TANKER DOCKING PORT			3			6	11
• HLLV TANKER CARGO EXTRACTION SYSTEM			3			3	6
• OTV TANKER DOCKING PORT			2			1	7
• OTV TANKER LOADING SYS			2				
• SHUTTLE DOCKING PORT			3				
• SHUTTLE GROWTH DOCKING PORT			2				
• PERSONNEL TRANSFER AIRLOCK SYS			6				
• GANTRY CRANES		2	2			3	6
• TRANSPORTER ELEVATOR		3	3				
• TRANSPORTER INVERTER		2	3				
• CARGO SORTING MANIPULATOR/ TRANSPORTER		1	1				
• CARGO TRANSFER MANIPULATOR		2					
• ANTENNA ELEVATOR			2				
• CARGO TRANSPORTERS		10	10			0.5	2
• 24 MAN CREW BUS		1	1			12	7
• 10 MAN CREW BUS		1	1			5	3
• TURNTABLES	20	41	25	13	32		
• CONTROL CABS (2 MAN)		3	6				
• (FOR LOGISTICS EQUIPMENT)							
• HLLV CARGO 1							
• HLLV/OTV TANKER 1							
• SHUTTLE/SHUTTLE GROWTH 1							
• GANTRY CRANES 4							
• CARGO SORTER 2							

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OF POOR QUALITY

A LO_2/LH_2 three axis system is again used for flight control.

3.3.1.1.3 **Environmental Factors**

The environmental factors associated with LEO construction of the thermal engine satellite are the same as described for the LEO constructed photovoltaic satellite described in Section 3.2.1.1.3.

3.3.1.1.4 **Crew Summary**

A total LEO crew size of 760 has been estimated to construct the thermal engine satellite. A breakdown of the crew in terms of organization and functions is presented in Figures 3.3-30 through 3.3-3.3-39.

3.3.1.1.5 **Mass Summary**

A ROM mass of nearly 9 million kg has been estimated for the thermal engine LEO construction base. A breakdown of this estimate is presented in Table 3.3-3. This value is nearly 3 million kg heavier than the LEO photovoltaic construction base with the main contributions to the difference being in the mass of the foundation and crew modules.

3.3.1.1.6 **Cost Summary**

A ROM cost for the first thermal engine construction base is estimated at over $11 billion. Approximately $7.6 billion is for the basic hardware with the remainder for the wrap-around cost. A breakdown of the cost is presented in Table 3.3-4. The total cost is approximately $4 billion greater than a LEO photovoltaic construction base.

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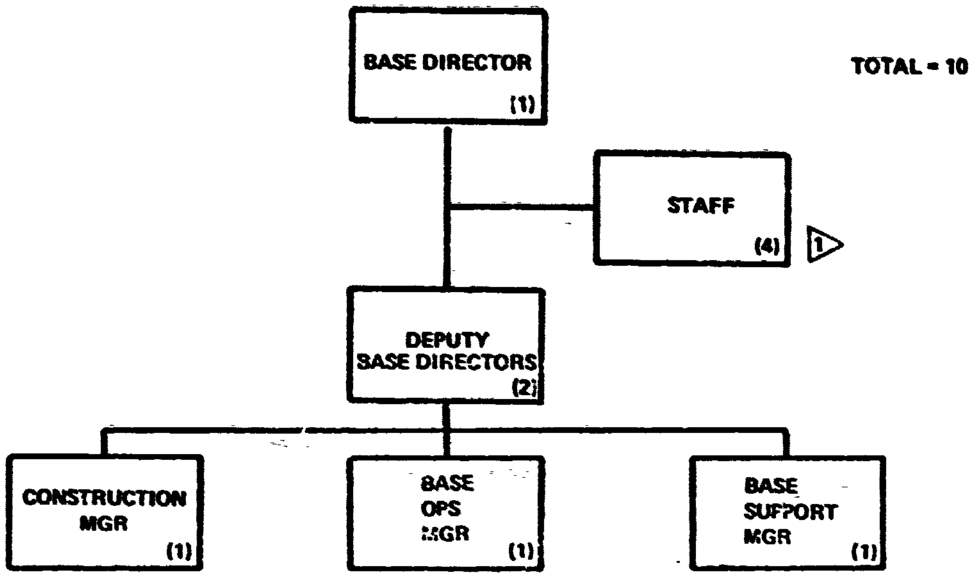
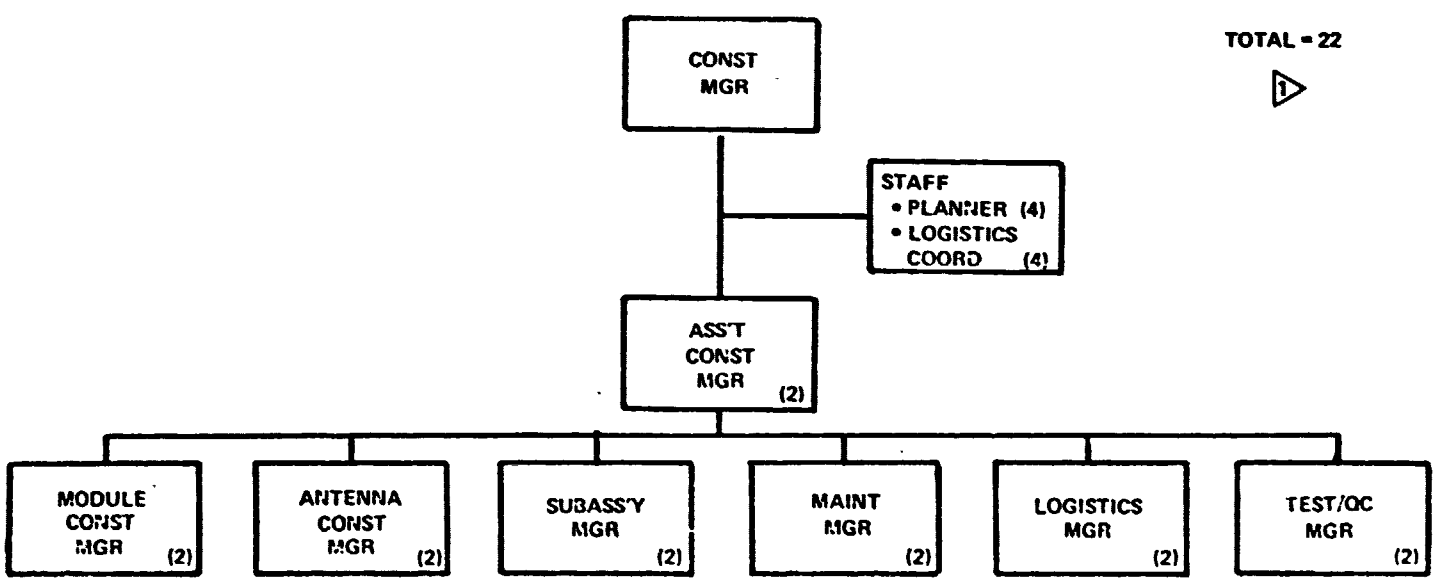


NUMBERS IN ( ) INDICATE THE NUMBER OF CREW MEMBERS REQUIRED TO STAFF THE INDICATED JOB OVER 2 SHIFTS

Figure 3.3-30. LEO Base Personnel Thermal Engine Satellite

SPS-1445



IN ALL THIS AND ALL OTHER ORGANIZATION CHARTS, A MANAGER WHO WAS COUNTED IN A HIGHER LEVEL ORGANIZATION CHART IS NOT INCLUDED IN THE TOTAL SHOWN HERE

Figure 3.3-31. LEO Base Personnel Thermal Engine Satellite

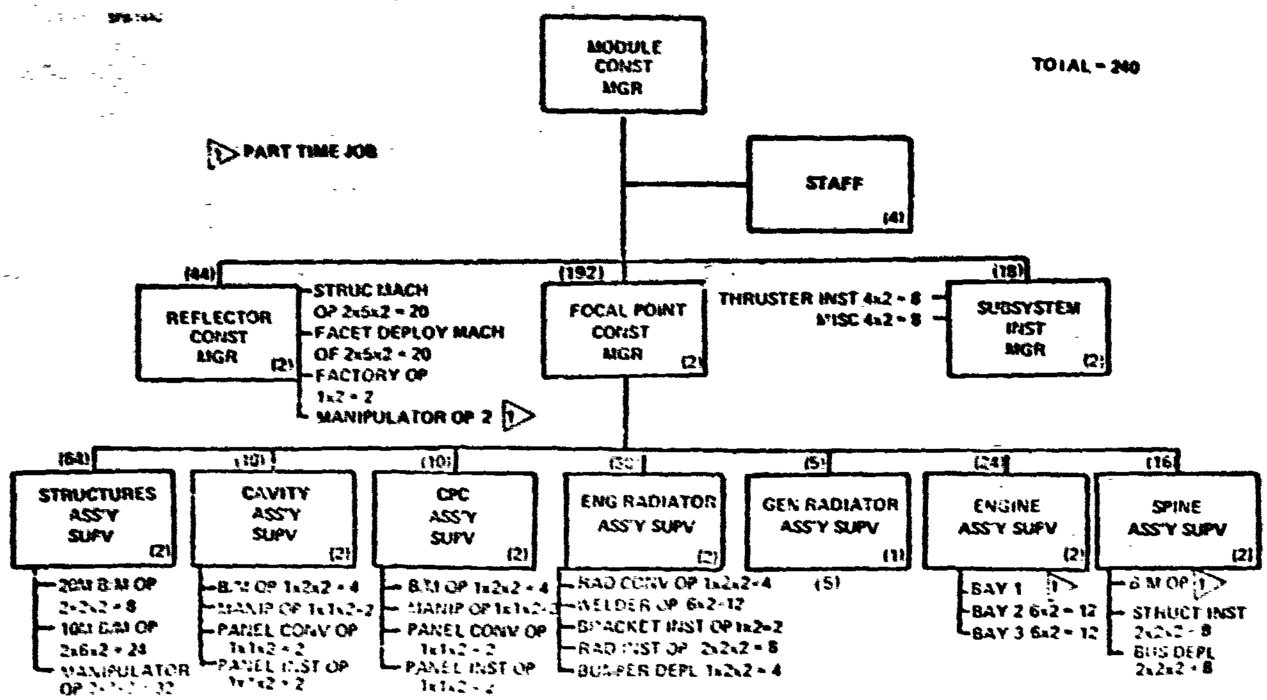


Figure 3.3-32. LEO Base Personnel
Thermal Engine Satellite

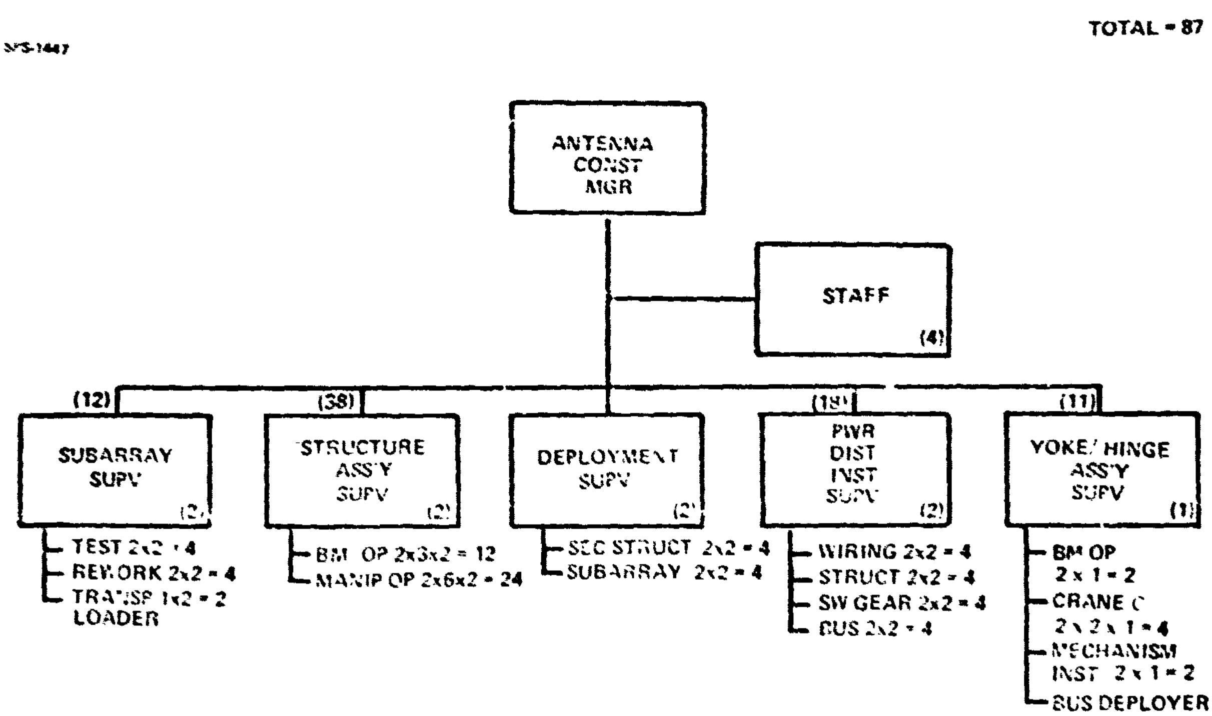


Figure 3.3-33. LEO Base Personnel
Thermal Engine Satellite

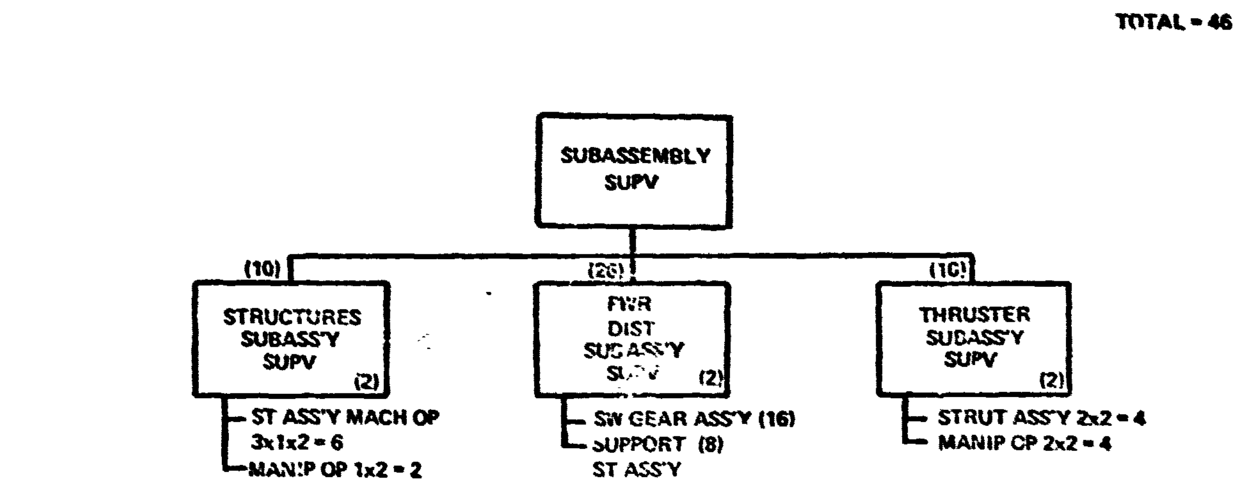


Figure 3.3-34. LEO Base Personnel
Thermal Engine Satellite

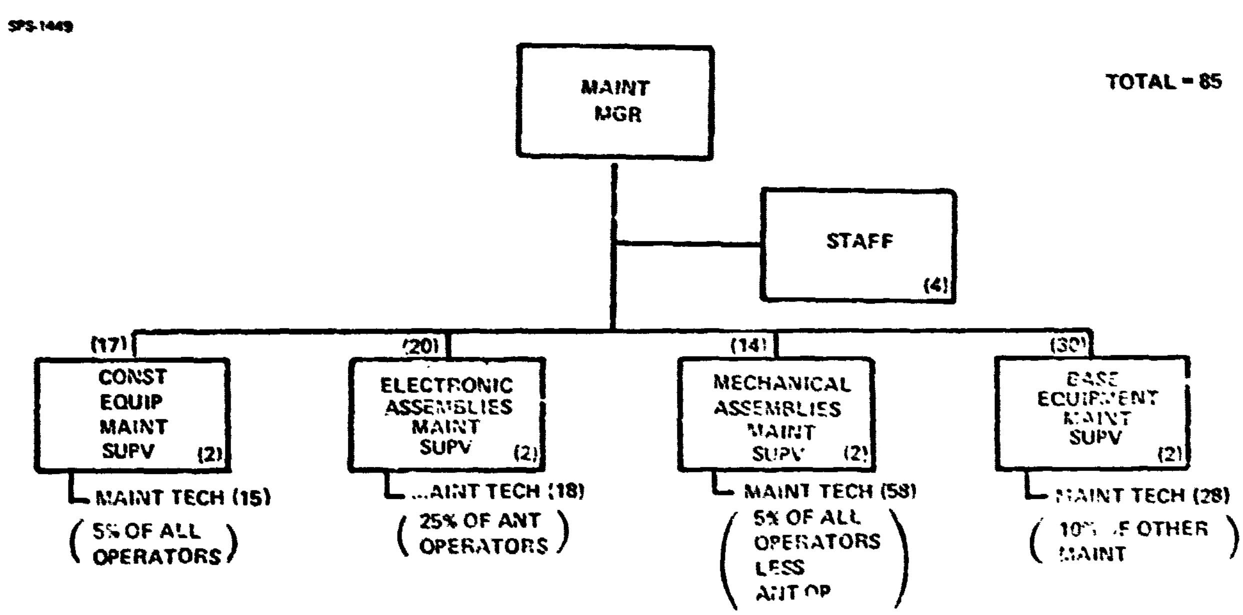


Figure 3.3-35. LEO Base Personnel
Thermal Engine Satellite

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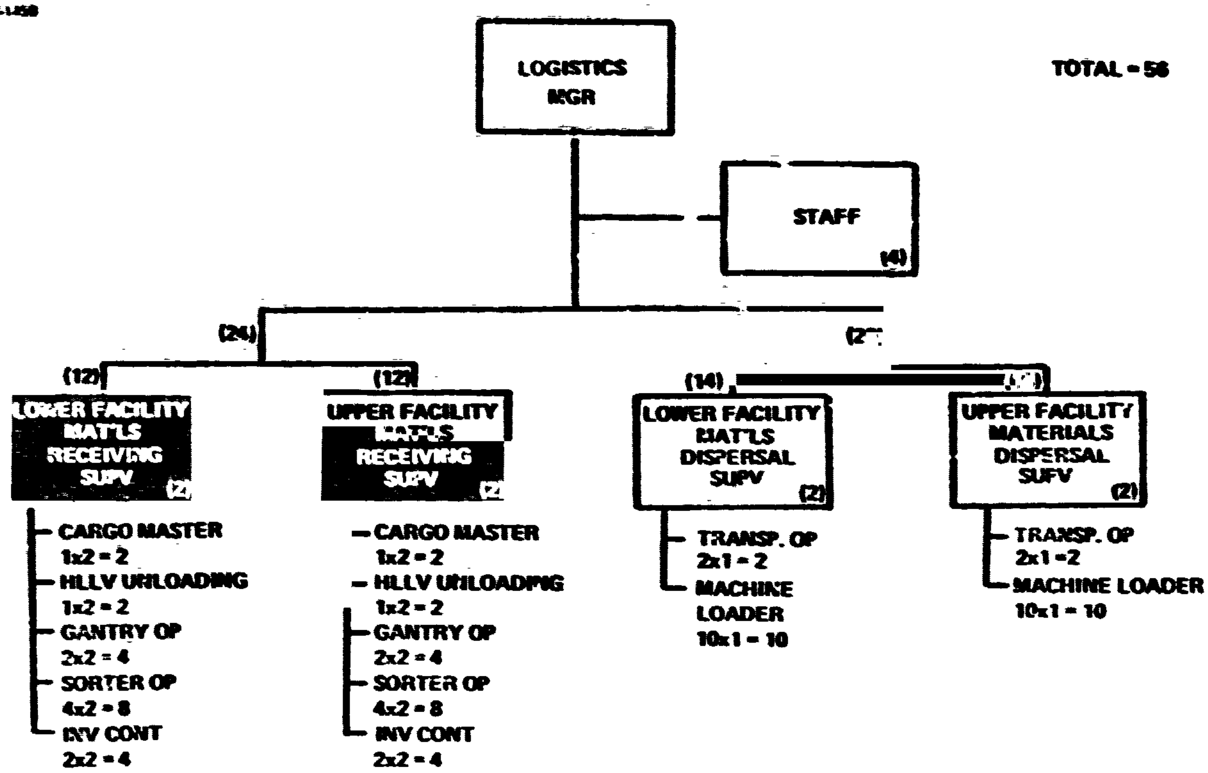

Figure 3.3-36. LEO Base Personnel
Thermal Engine Satellite

SPS-1451

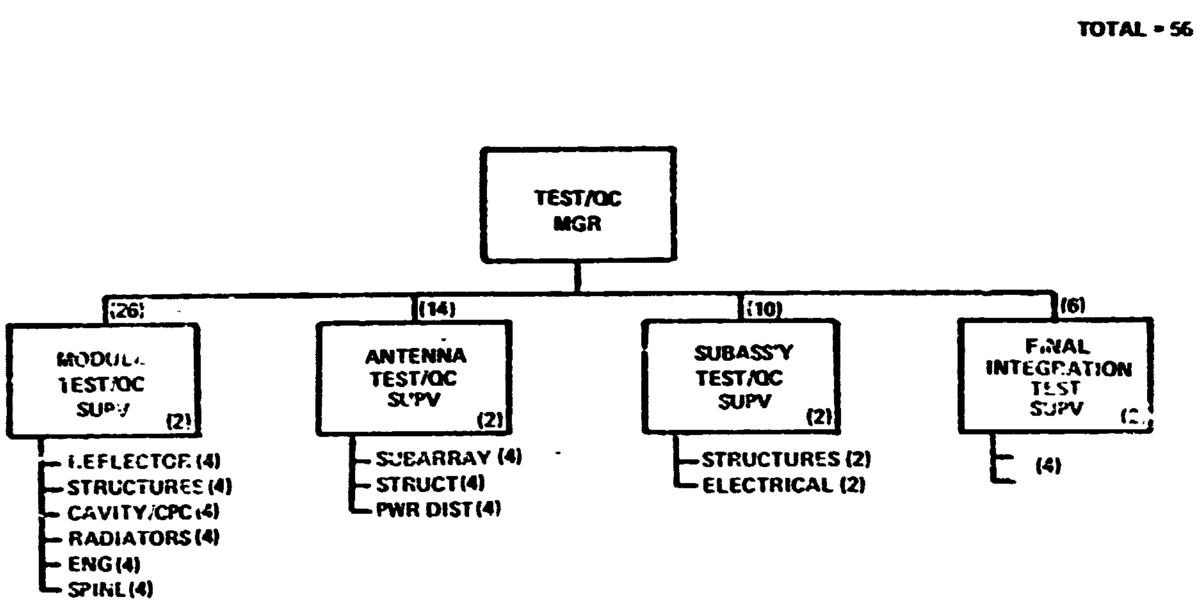

Figure 3.3-37. LEO Base Personnel
Thermal Engine Satellite

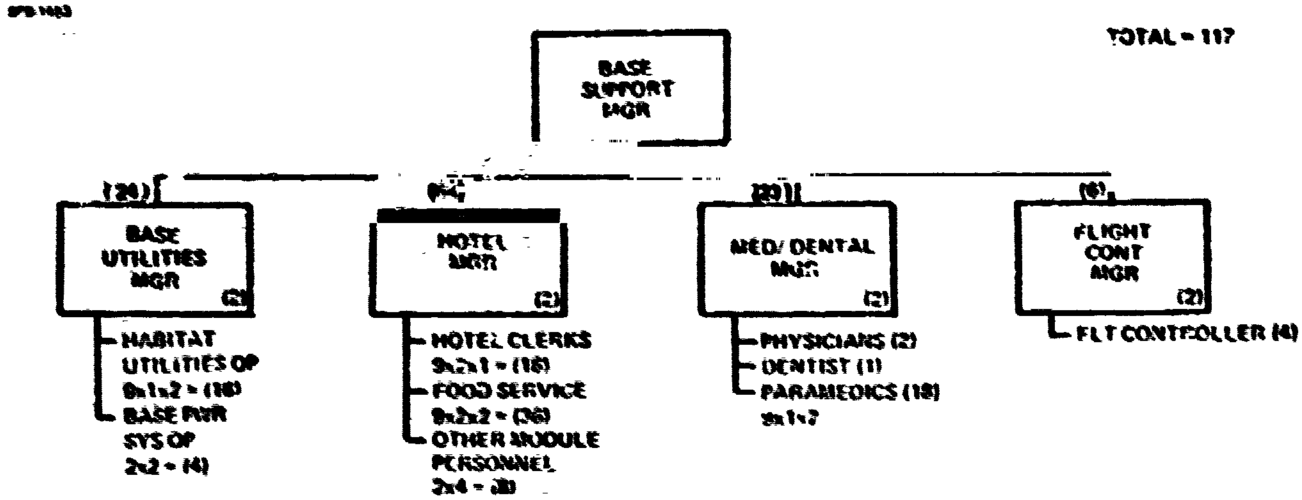

Figure 3.3-38. LEO Base Personnel
Thermal Engine Satellite

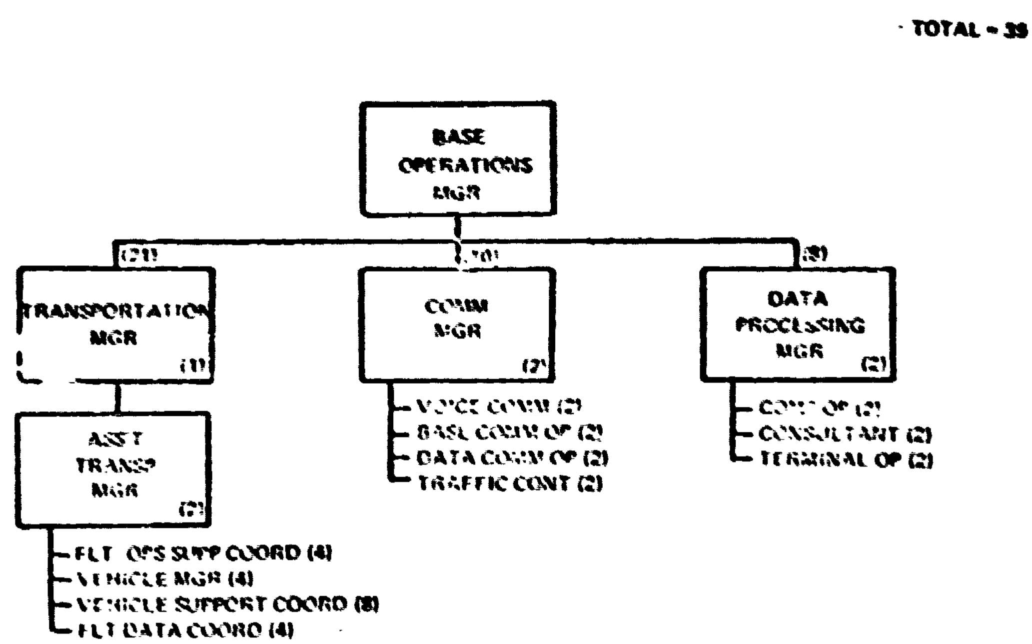

Figure 3.3-39. LEO Base Personnel
Thermal Engine Satellite

Table 3.3-3. LEO Construction Base ROM Mass
Thermal Engine Satellite

		10^3 kg
		(7615)
FACILITY		
FOUNDATION		4000
CREW MODULES [1]		2600
CARGO HANDLING/DISTRIBUTION		515
BASE SUBSYSTEMS		400
MAINT. PROVISION		100
CONSTRUCTION AND SUPPORT EQUIP.		(945)
STRUCTURAL ASSEMBLY		500
ENERGY COLLECTION INSTALL.		75
ENERGY CONVERSION INSTALL.		65
POWER DISTRIBUTION INSTALL.		70
SUBARRAY INSTALL.		30
CRANES/MANIPULATORS		240
INDEXERS		30
	TOTAL DRY	(8560)
CONSUMABLES (90 DAYS)		(400)
	TOTAL	(8960)

[1] INCLUDES 33% GROWTH ALLOWANCE

Table 3.3-4. LEO Construction Base ROM Cost
Thermal Engine Satellite

FIRST SET-UNIT			$\$10^6$
FACILITY			(4670)
FOUNDATION		400	
CREW MODULES		3600	
CARGO HANDLING/DISTRIBUTION		470	
BASE SUBSYSTEMS		200	
MAINTENANCE PROVISIONS		-	
CONSTRUCTION & SUPPORT EQUIPMENT			(2920)
STRUCTURAL ASSEMBLY		1420	
ENERGY COLLECTION		280	
ENERGY CONVERSION		250	
POWER DISTRIBUTION		190	
SUBARRAY INSTALLATION		80	
CRANES/MANIPULATORS		600	
INDEXERS		90	
	BASIC HARDWARE		(7590)
SPARES (15%) [1]			1135
INSTALLATION, ASSEMBLY, C/O (16%)			1210
SE&I (7%)			530
PROJ MANAGEMENT (2%)			150
SYSTEMS TEST (3%)			225
GSE (4%)			305
	TOTAL		$11,145

[1] % OF BASIC HARDWARE

3.3.1.2 GEO Final Assembly Base Construction Analysis

The following data pertains to the GEO final assembly based used in the LEO construction concept.

3.3.1.2.1 GEO Construction Operations

3.3.1.2.1.1 Top-Level GEO Construction Tasks

The construction tasks to be accomplished at the GEO base are summarized in Figure 3.3-40. Each of these tasks are discussed in the following sections.

3.3.1.2.1.2 GEO Base

The GEO final assembly base consists of two facilities as shown in Figure 3.3-41.

Upper Facility
The upper facility attaches itself to the ends of the spines. This facility incorporates a docking crane 2 facility docking fixtures, and a docking port for a intra-facility free flyer vehicle.

A typical facility docking fixture is illustrated in Figure 3.3-42. This fixture provide the means by which the upper facility is attached to the spine. A crane/manipulator attached to these fixtures is used to make the structural and electrical connections between modules.

The upper facility is indexed by having one of the facility docking fixtures detach from the spine. This fixture is then retracted so that it will be clean of obstructions during the indexing maneuver. The facility docking fixture that is still attached to a spine is then used to swing the upper facility through an arc such that the opposite facility docking fixture is located above a new spine end. Once in position, the retracted facility docking fixture is extended and attached to the spine.

The docking crane and the facility docking fixtures can be relocated anywhere along the length of the upper facility via facility tracks.

Lower Facility
The lower facility attaches itself to the corners of the reflector. This facility incorporates two docking cranes, two facility docking fixtures, vehicle docking ports and a crew module.

The facility docking fixtures incorporate crane/manipulators that are used to make the structural attachments between module reflectors. The fixtures provide indexing capability as described above.

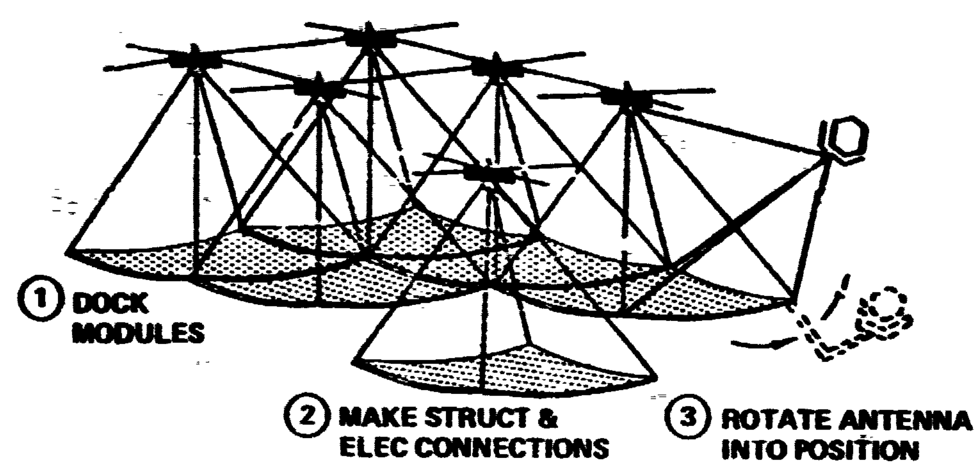

Figure 3.3-40. GEO Constructi n Tasks
LEO Construc tion Concept

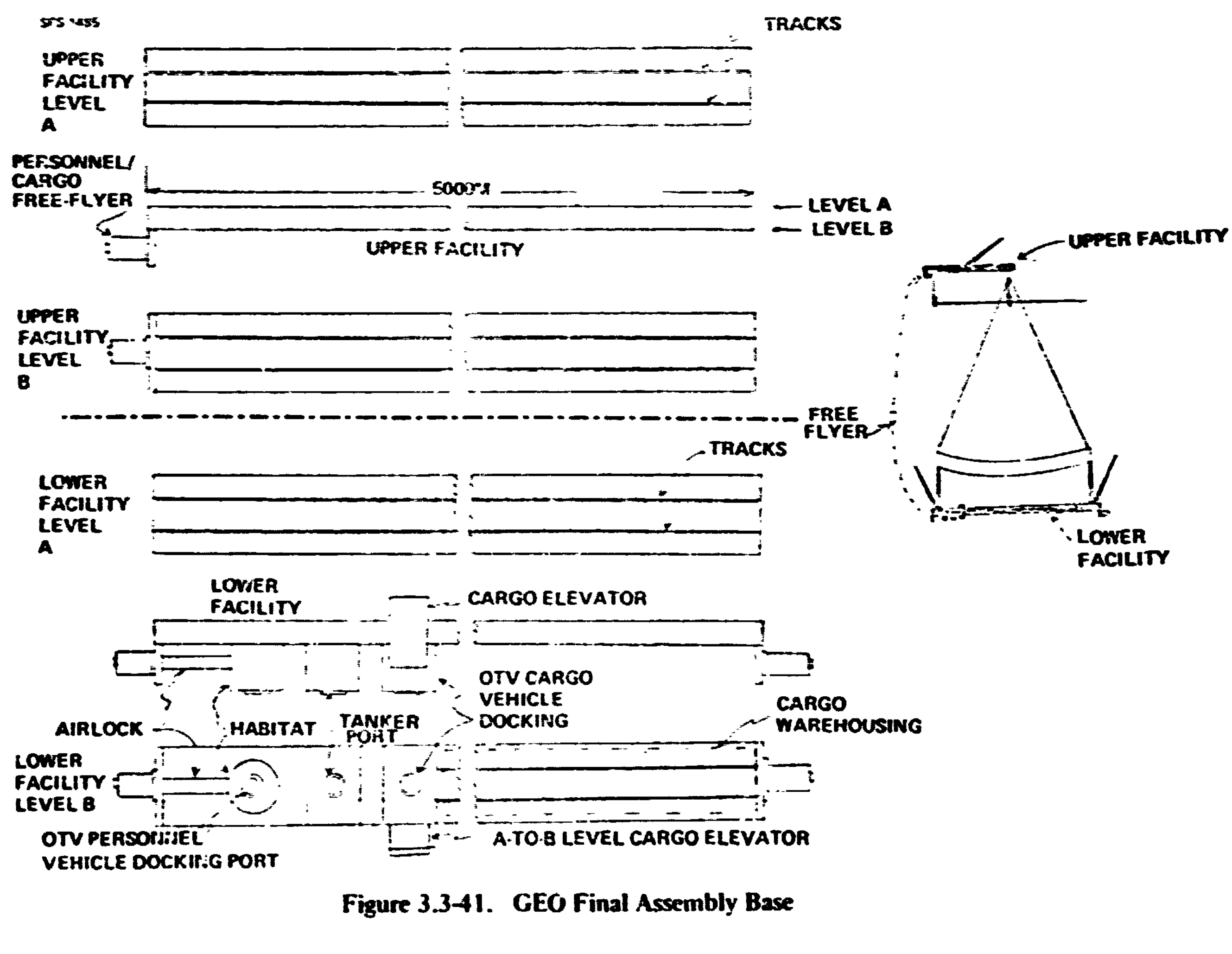

Figure 3.3-41. GEO Final Assembly Base

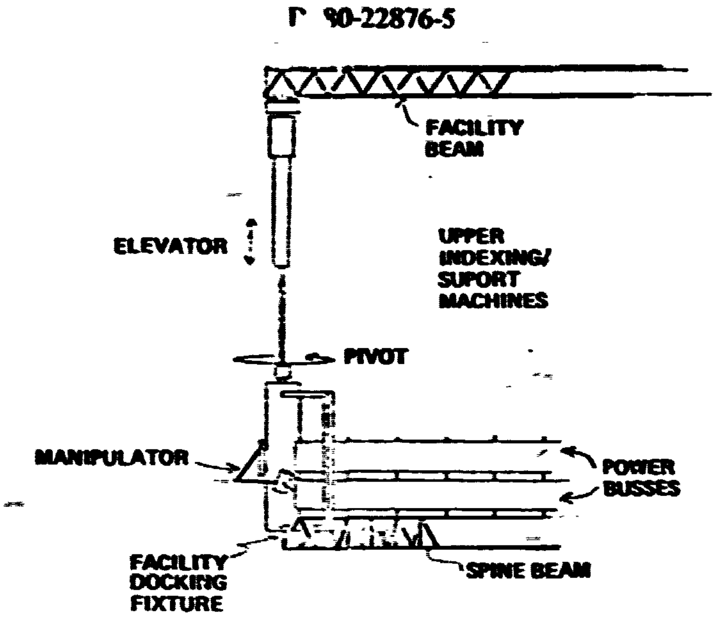

Figure 3.3-42. GEO Base Docking Fixture

3.3.1.2.1.3 Detailed Construction Tasks Analysis

3.3.1.2.1.3.1 Module Berthing (Docking)

The berthing (docking) cranes on the upper and lower facilities are identical in concept to the docking crane described in the photovoltaic system (the booms will be longer). The docking scheme is the same except that only three docking cranes are used to dock the thermal engine satellite modules.

Figure 3.3-43 shows how these docking cranes would be employed to achieve module docking. This figure also shows how the facilities have to be oriented in order to get the crane/manipulators that will be used to attach modules in the right position.

When docking an incoming module in a corner, it will be necessary for the lower facility to make an extra indexing maneuver in order to get the inner corner of the reflector attached to adjacent modules.

3.3.1.2.1.3.2 Antenna Installation

As was described in preceding sections, the antennas are transported to GEO with modules 8 and 16. The antenna is connected via its yoke to the underside of the reflector using a hinged linkage assembly. Figure 3.3-44 shows this arrangement and how the antenna is raised into position using the linkage once GEO is reached.

Note that the upper facility has been indexed such that it is attached on one end to the antenna support structure. This gets a crane/manipulator out to the yoke attachment position where structural and bus bar connections need to be made after the antenna assembly has been moved into position.

3.3.1.2.1.3.3 Construction Equipment Summary

A listing of the GEO base construction equipment and their characteristics is presented in Table 3.3-5.

3.3.1.2.2 GEO Base Description

3.3.1.2.2.1 Configuration

The configuration of the base has previously been shown in Figure 3.3-41. Placement of the crew module at the lower facility was somewhat arbitrary since the activity is about equal between the upper and lower facilities. Regardless of its location however, a small manned free-flying is necessary to transport crewmen to their work stations at the opposite facility.

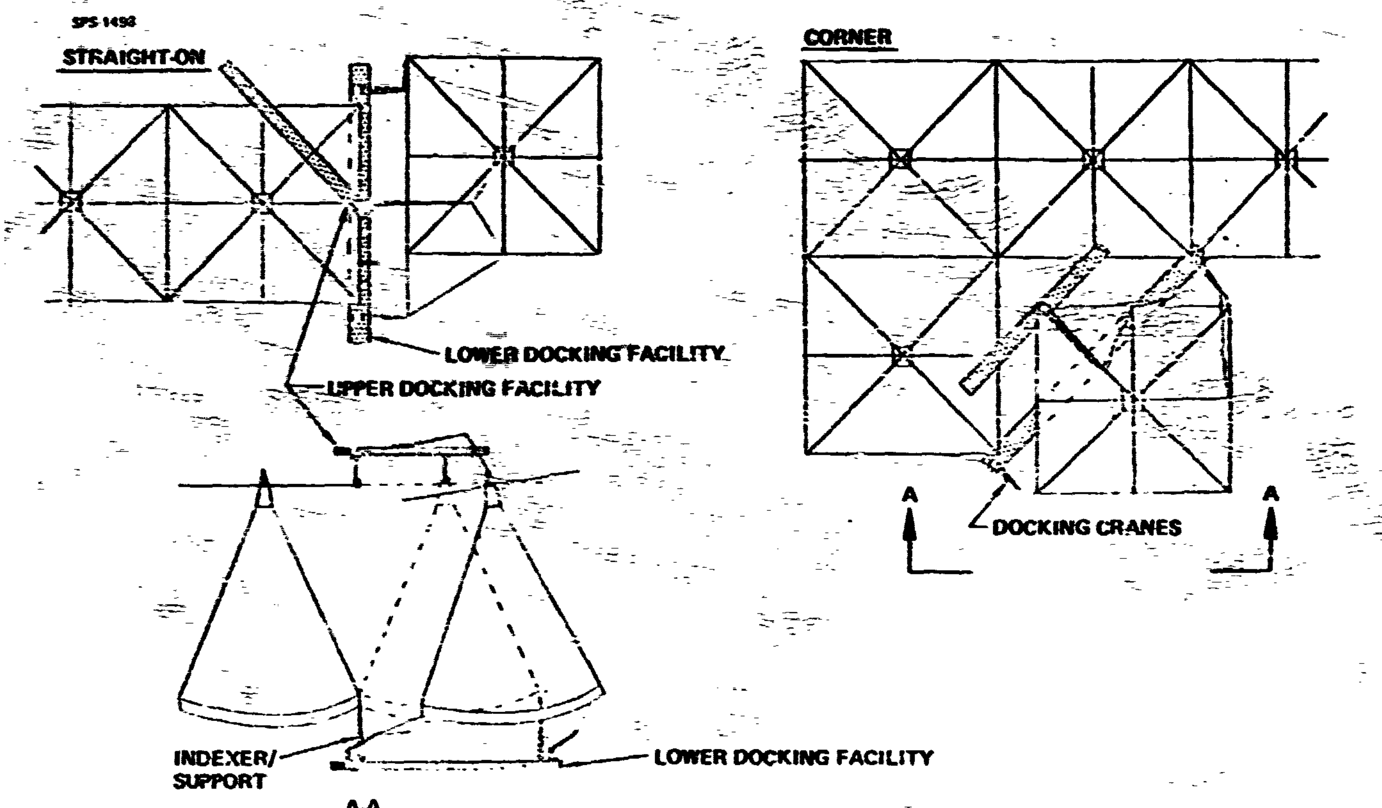

Figure 3.3-43. GEO Berthing Concept
Thermal Engine Satellite

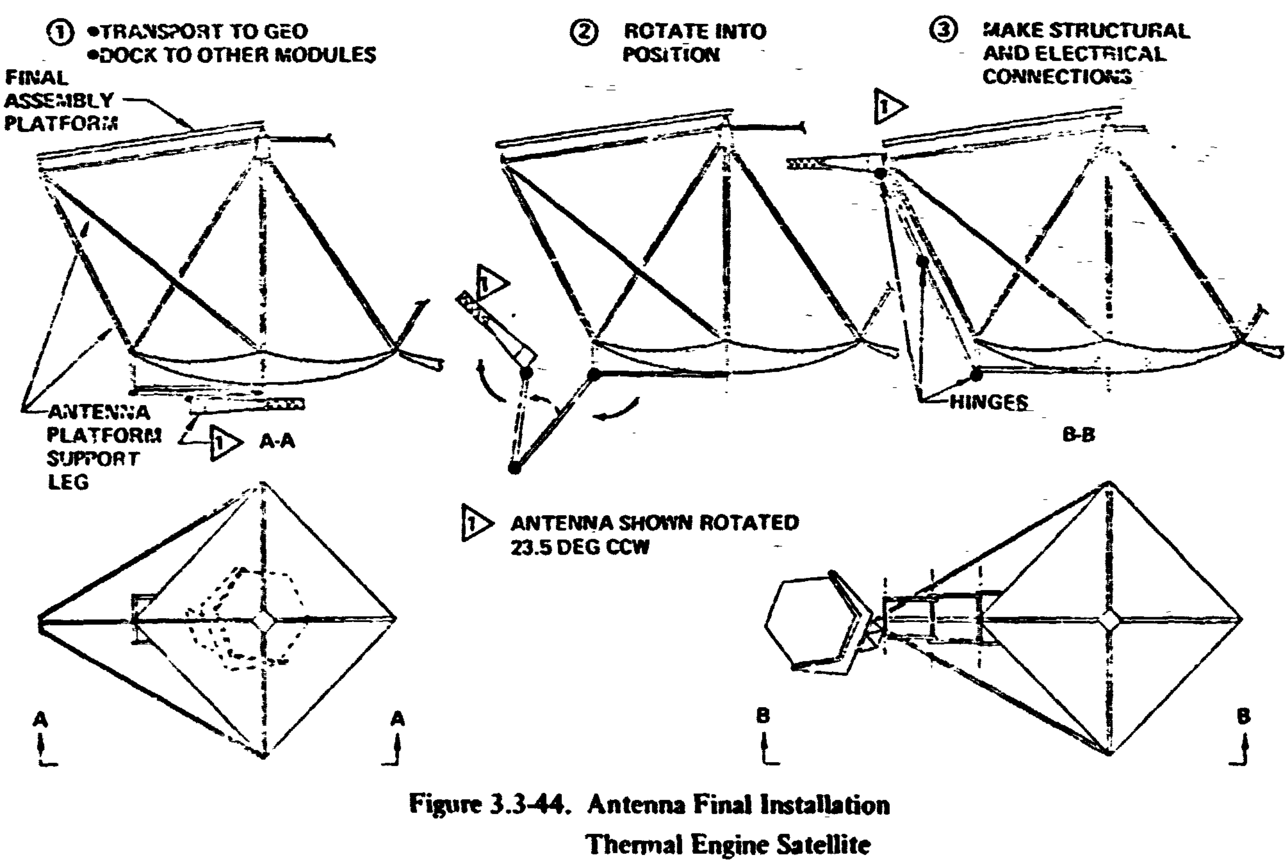

Figure 3.3-44. Antenna Final Installation
Thermal Engine Satellite

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Table 3.3-5. GEO Base Construction Equipment-Thermal Engine Satellite

EQUIPMENT ITEM	NUMBER REQ'D		EQUIPMENT ITEM MAJOR ELEMENTS	NO. REQ'D
	UPPER FAC	LOWER FAC		ITEM
• DOCKING CRANE (MASS [illegible]^{K} Kg) (COST $53M) A	1	2	• CARRIAGE • BOOM • WINCHES • CONTROL CAB (2 MAN) • FREE-FLYING DOCKING PROBES	1 1 3 1 [illegible]
• UPPER FACILITY INDEXING/ SUPPORT MACH 1 INCLUDED IN 20M MANIPULATOR/CRANE COUNT (MASS 7^{K} Kg) (COST $10M)	2		• CARRIAGE • BOOM ASSY • 20M MANIPULATOR/CRANE • CONTROL CAB (2 MAN)	1 1 1 1 1
• LOWER FACILITY INDEXING/ SUPPORT MACHINE (MASS 10^{K} Kg) (COST $11M)		2	• CARRIAGE • BOOM ASSY • CONTROL CAB (2 MAN)	1 1 1
• 110M MANIPULATOR/CRANE (MASS 8^{K} Kg) (COST $18M)		2	• CONTROL CAB (2 MAN) • CARRIAGE • ELEVATOR BOOM • TRANSVERSE BOOM • MANIPULATOR ARMS	1 1 1 1 2
• 20M MANIPULATOR/CRANE (MASS 6^{K} Kg) (COST $19M)	2		• CONTROL CAB ([illegible] MAN) • CARRIAGE • ELEVATOR BOOM • TRANSVERSE BOOM • MANIPULATOR ARMS	1 1 1 1 2

A ALL COST IN TABLE REFLECT AVG COST OBTAINED FROM 0.9 LEARNING.

3.3.1.2.2.2 **Foundation**

The foundation consists of 20m beams forming trusses.

3.3.1.2.2.3 **Cargo Handling/Distribution**

The equipment used to handle and distribute cargo and transport personnel are presented in Table 3.3-6.

3.3.1.2.2.4 **Crew Module**

The one crew module serves as both crew quarters and operations center. The module itself is similar in design to the module used at the GEO final assembly base for the photovoltaic satellite including a radiation shelter.

3.3.1.2.2.5 **Base Subsystems**

The systems are the same as described for the photovoltaic GEO final assembly base in Section 3.2.1.2.2.5.

3.3.1.2.3 **Environmental Factors**

The environmental factors concerning the thermal engine GEO base are the same as for the photovoltaic GEO base described in Section 3.2.1.2.3.

3.3.1.2.4 **Crew Summary**

A crew size of 53 has been estimated. The organization of this crew is shown in Figure 3.3-45. Table 3.3-7 presents a crew functional breakdown for both the LEO and GEO bases associated with the LEO construction concept of the thermal engine satellite.

3.3.1.2.5 **Mass Summary**

A ROM mass of approximately 960 000 Kg has been estimated. A breakdown of this estimated is presented in Table 3.3-8.

3.3.1.2.6 **Cost Summary**

A ROM cost of approximately $1.2 billion has been estimated for the GEO base. A breakdown of this estimate is presented in Table 3.3-9.

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Table 3.3-6. GEO Base Logistics Equipment -Thermal Engine Satellite

EQUIPMENT ITEM	NUMBER REQ'D UPPER	NUMBER REQ'D LOWER	MASS (EA) 10^3 Kg	COST (EA) $\$10^6$
• OTV CARGO DOCKING PORT		1	1	7
• CARGO EXTRACTION SYS		1	3	6
• OTV TANKER DOCKING PORT		1	1	7
• TANKER CARGO EXTRACT SYS		1	3	6
• OTV PERSONNEL DOCKING PORT		1		
• PERSONNEL AIRLOCK SYS		1		
• CARGO SORTING MANIP/CRANE		1		
• CARGO TRANSPORTER	1	1		
• 10 MAN CREW BUS	1	1	5	3
• TURNTABLES		4		
• CONTROL CAB (FOR LOGISTICS EQUIP)		2		
• CARGO/TANKER HANDLING				
• CARGO SORTER				
• PERSONNEL FREE-FLYER		1	7	6

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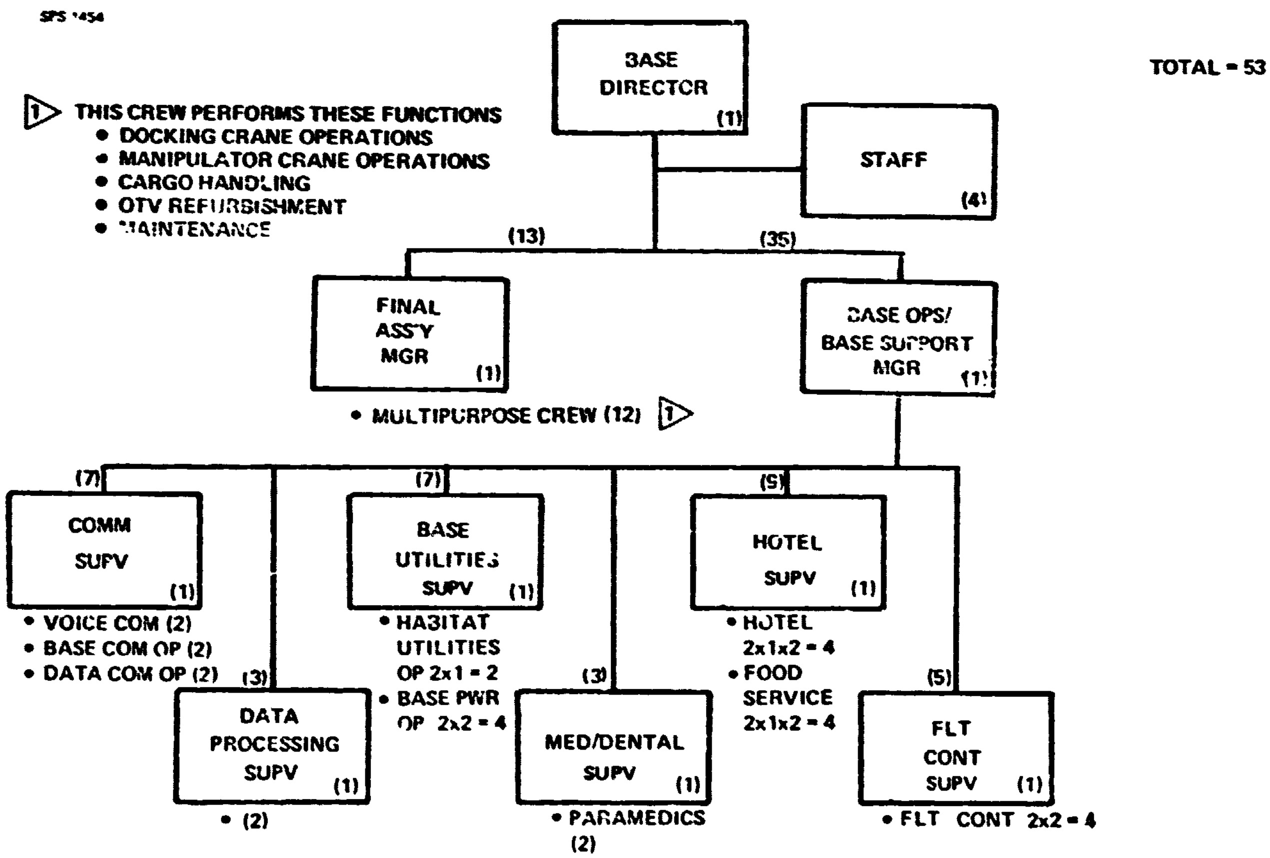

Figure 3.3-45. GEO Base Personnel Thermal Engine Satellite

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Table 3.3-7. LEO Construction Manpower Thermal Engine Satellite

	LEO BASE	GEO BASE
BASE MANAGEMENT	(10)	(7)
CONSTRUCTION	(592)	(12)
MANAGEMENT	22	
MODULE CONST	240	12
ANTENNA CONST	87	
SUBASSEMBLY	46	
MAINTENANCE	85	
LOGISTICS	56	
TEST/QC	56	
BASE OPERATIONS	(39)	(10)
MANAGEMENT	7	2
TRANSPORTATION	18	
COMM	8	6
DATA PROCESSING	6	2
BASE SUPPORT	(117)	(24)
MANAGEMENT	8	4
BASE UTILITIES	22	6
HOTEL	62	8
MEDICAL	21	2
FLIGHT CONT	4	4
BASE TOTAL	758	53

TOTAL 811

Table 3.3-8 GEO Final Assembly Base ROM Mass Thermal Engine Satellite

	10^3KG	
FACILITY		(805)
FOUNDATION	390	
CREW MODULES	335	
CARGO HANDLING/DISTRIBUTION	60	
BASE SUBSYSTEMS	20	
MAINTENANCE PROVISIONS	–	
CONSTRUCTION AND SUPPORT EQUIPMENT		(130)
CRANES/MANIPULATORS	35	
DOCKING CRANES	60	
INDEXERS	35	
DRY TOTAL		(935)
CONSUMABLES (90 DAYS)		(28)
TOTAL		(963)

Table 3.3-9 GEO Final Assembly Base ROM Cost Thermal Engine Satellite

	$\$10^6$	
FACILITY		(600)
FOUNDATION	40	
CREW MODULE	400	
CARGO HANDLING/DIST	160	
BASE SUBSYSTEM	1	
CONSTRUCTION AND SUPPORT EQUIPMENT		(250)
DOCKING CRANES	160	
INDEXERS	20	
CRANES/MANIPULATORS	70	
	BASIC HARDWARE	($850)
SPARES		125
INST. ASSY. C O		135
SE&I		60
PROJ MGT		15
SYS TEST		25
GSE		35
	TOTAL	($1245)

3.3.2 GEO Construction Concept Overview

For the thermal engine satellite, the GEO construction concept entails constructing 16 modules and 2 antennas of the configuration described in Section 5.2.4.1. The construction facilities, operational sequences, task breakdowns, construction machinery, etc will be identical to that described under the thermal engine satellite LEO construction section. A LEO staging depot would also be required with characteristics similar to the staging depot described in Section 3.2.2.1.

The only detectable difference will be a small change in crew size due to the elimination of the redundant base management, base support and base operations personnel associated with what was the GEO base for the LEO construction concept. However, this delta will almost totally be offset by the need for a similar redundancy required to staff a LEO staging depot.

Mass and cost estimates were not prepared specifically for this particular option. A preliminary estimate can be made by using data from the other construction bases described in this document. Using this approach, the thermal engine satellite GEO construction base is estimated to have a mass of 10 million Kg and a unit cost of $12 billion.

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3.4 KEY CONSTRUCTION TECHNOLOGY DEVELOPMENT REQUIREMENTS

This section presents brief descriptions of key construction operations, techniques, and support systems that will require development and demonstration early in the SPS precursor programs in order to verify that it will indeed be possible to construct SPS's as described. These are items for which no known study program has been initiated or for which previous space flight data is not available or pertinent.

3.4.1 Beam Assembly

Demonstrate the capability to make 15 m to 20 m beams from an automated beam machine at 5 to 10 meter per minute rates

3.4.2 Frame Assembly

Demonstrate the capability of crane manipulators and accessory equipment to move long (650 m) beams into position and to attach these beams into a rigid framework.

3.4.3 Solar Array Deployment

Demonstrate the capability to deploy accordion folded solar array blankets, attach blankets to structure and attach blankets edge-to-edge using deployment equipment of the type described in this document.

3.4.4 Power Bus Deployment

Demonstrate the capability to deploy 5 to 30 m wide by 1 millimeter thick sheet metal busses from roll stock.

3.4.5 Module Support Indexing

Demonstrate the capability to support large structures using several indexing support machines operating simultaneously from a facility surface.

3.4.6 Attitude Control During Construction

Demonstrate the capability to control the attitude of large facility product combination as the product is grown.

3.4.7 Large Module Docking

Demonstrate the capability to dock large structures using the technique described in this document.

3.4.8 Large Machine Assembly

Demonstrate the capability to assemble in-orbit large construction machines that are delivered as large subassemblies.

3.4.9 Antenna Subarray Installation/Maintenance

Develop capability to install, repair and replace complicated subarrays and components.

3.4.10 Environmental Control

Demonstrate environmental control and life support systems suitable for long durations and large crew sizes as specified in this document.

3.4.11 EVA Suit

Develop a highly mobile 14.7 psia EVA suit compatible with the atmosphere of the primary crew modules and would consequently eliminate long pre-breathing and post-breathing periods.

3.4.12 External Lighting

Establish external lighting levels required for the continuous construction operations involving large areas and monitored or controlled by man and/or sensors.

3.4.13 Space Debris Tracking System

Develop capability to accurately predict the size and trajectories of space debris that may pose a problem for the construction and operation of an SPS.

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4.0 TRANSPORTATION SYSTEM ANALYSIS

A major portion of Part 1 of the SPS System Definition Study was devoted to the definition of SPS transportation systems. This data is documented in Volume IV (D180-20689-4) and Volume V (D180-20689-5). These documents identify requirements, system trades and system descriptions.

The transportation effort of Part 2 was limited to making refinements to the Part 1 data to reflect changes in satellite design, improve the system definition and update the cost estimate. A system description of the launch vehicles and orbit transfer vehicles is presented in the following paragraphs.

4.1 TRANSPORTATION REQUIREMENTS

Detail SPS transportation system requirements were presented in Volume IV of the Part 1 documentation. A summary of the transportation requirements associated with the payloads of the reference photovoltaic satellite constructed in either LEO or GEO is shown in Figure 4.1-1. There is no OTV propellant mass included.

The difference in satellite mass only reflects the structural mass penalty of the additional vertical and lateral members and loads caused by transfer of the antenna. Oversizing and power distribution penalties are all a function of orbit transfer characteristics and consequently are chargeable to the orbit transfer system itself.

Differences in crew and supply requirements delivered to LEO primarily reflect additional orbit keeping attitude control propellant requirements. The key difference, however, is in the mass which must be delivered to GEO.

Facility transportation requirements reflect the initial placement task as well as in the case of the GEO bases (both options), that mass that must be moved to the longitude location where the next satellite is to be constructed. The principal difference in the two main construction bases is that the six crew modules in the GEO concept each have approximately 115 000 kg of additional mass for solar flares shelters.

Total cargo mass which must be handled by the launch vehicle are shown in Figure 4.1-2 and reflect both the payload requirements indicated earlier and the OTV propellant and hardware requirements. For the three system elements that require transportation, payload requirements are not too different between LEO and GEO construction options, however, the inclusion of the orbit transfer system requirements add significantly to the total mass which must be delivered by the HLLV. The LEO construction location assumes electric propulsion for the satellite LEO to GEO transportation while LO_2 LH_2 is used for the GEO construction option.

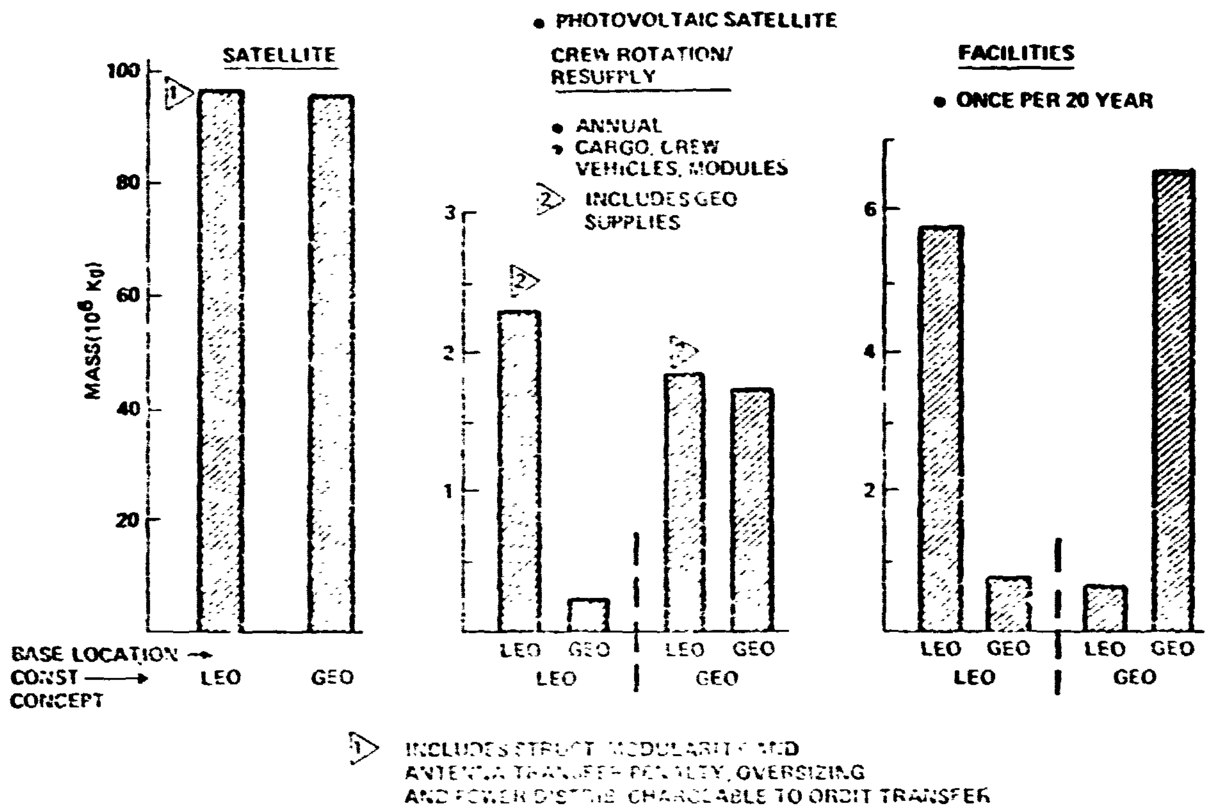

Figure 4.1-1. Transportation Requirements LEO vs GEO

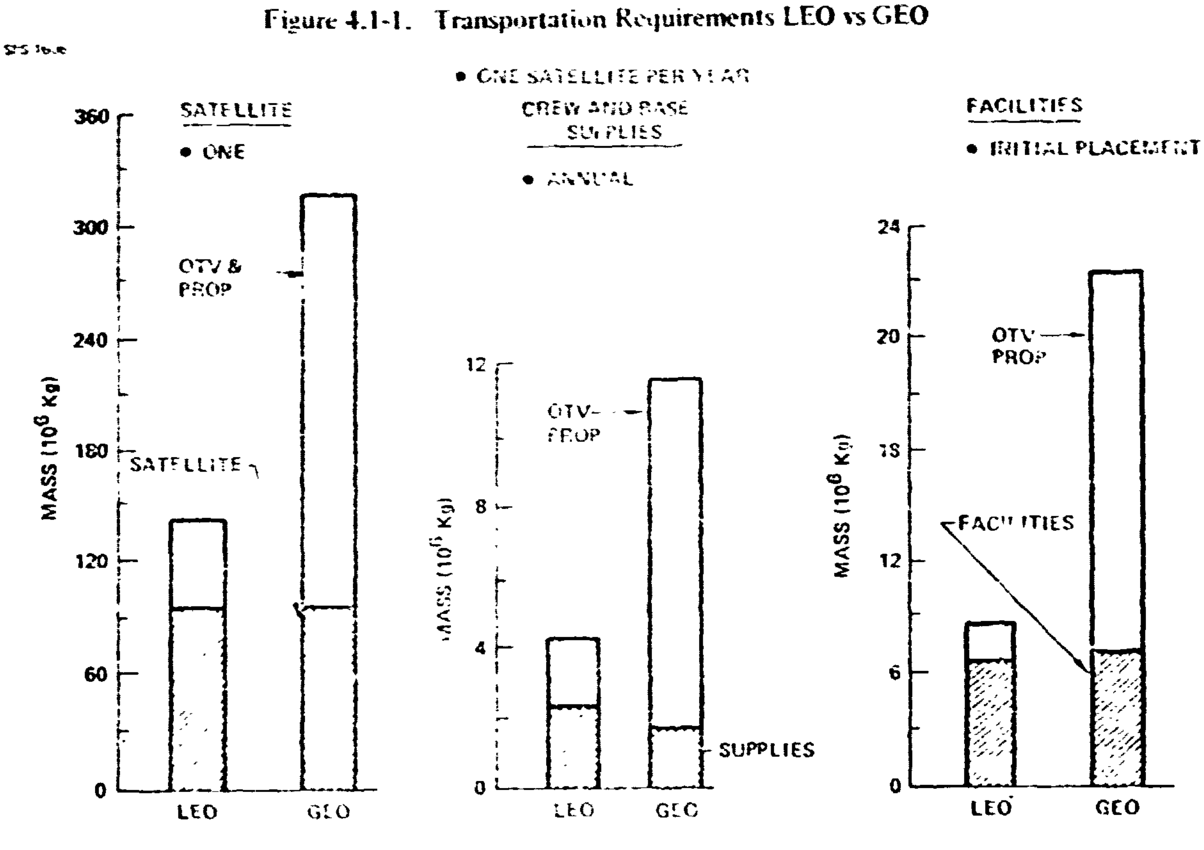

Figure 4.1-2. Total Cargo Mass to LEO

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4.2 LAUNCH SYSTEMS

4.2.1 System Descriptions

Detail system descriptions for the launch systems are presented in Volume V (D180-20689-5) of the Part I Report. The following paragraphs present a summary description.

4.2.1.1 Cargo Delivery System

Both two stage ballistic/ballistic and two stage winged vehicles have been investigated. In both cases, the reference payload capability was 400 000 kg. Other common characteristics included the launch site at KSC and a delivery orbit of 477.5 KM and 31 degrees.

4.2.1.1.1 Two-Stage Ballistic Vehicle

Configuration

The baseline engine is a scaled up version of the Alternate Mode 1 engine defined by Aerojet Liquid Rocket Company under contract to NASA Lewis Research Center. The following engine characteristics were used in the analysis.

The baseline engine is a scaled up version of the Alternate Mode 1 engine defined by Aerojet Liquid Rocket Company under contract to NASA Lewis Research Center (see attached appendix). The following engine characteristics were used in the analysis.

Propellants	RP-1 LO_2/LH_2	
Thrust Vacuum	$9.059 \times 10^6 N$	(2.037×10^6 lbf)
Chamber Pressure	29300 kpa	(4250 psia)
Mixture Ratio	2.9:1	
Specific Impulse (SL/Vac.)	323.5/350.7 sec.	
Total Flow Rate/Engine	2635 kg/sec	(5808 lbm/sec)

Engine overall length is 5.44m and the power head and exit diameters are 3.51m and 2.97m, respectively. The total mass of the engines including accessories is estimated to be 138322 kg.

Flight Characteristics

The ascent trajectory characteristics for the vehicle are shown in Figure 4.2-2. The major characteristics are summarized as follows:

First Stage

- T/W @ Ignition = 1.30
- Maximum Dynamic Pressure = 32.125 kpa
- Maximum Acceleration = 4.90 g's
- Stage Burn Time = 176.89 sec.
- Dynamic Pressure at Staging = 405 pa

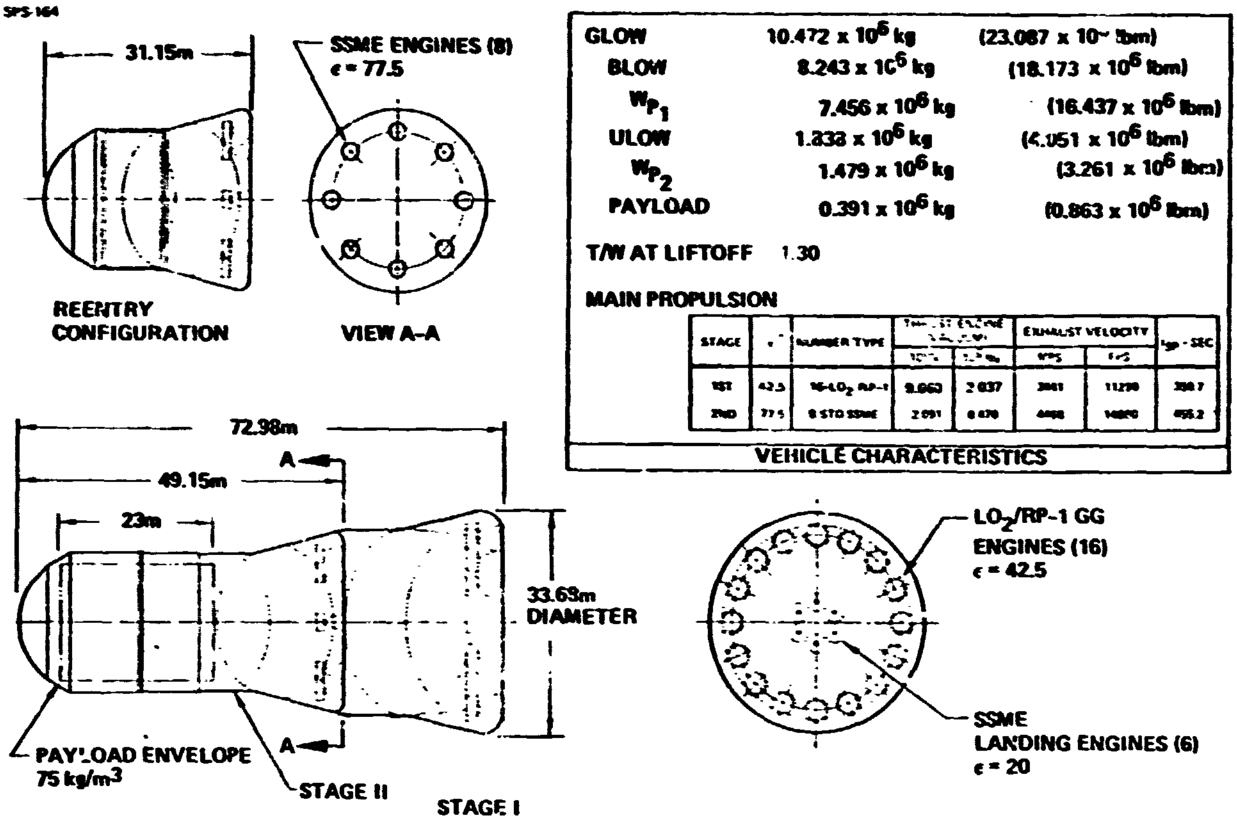


Figure 4.2-1. SPS Launch Vehicle–Cargo Version

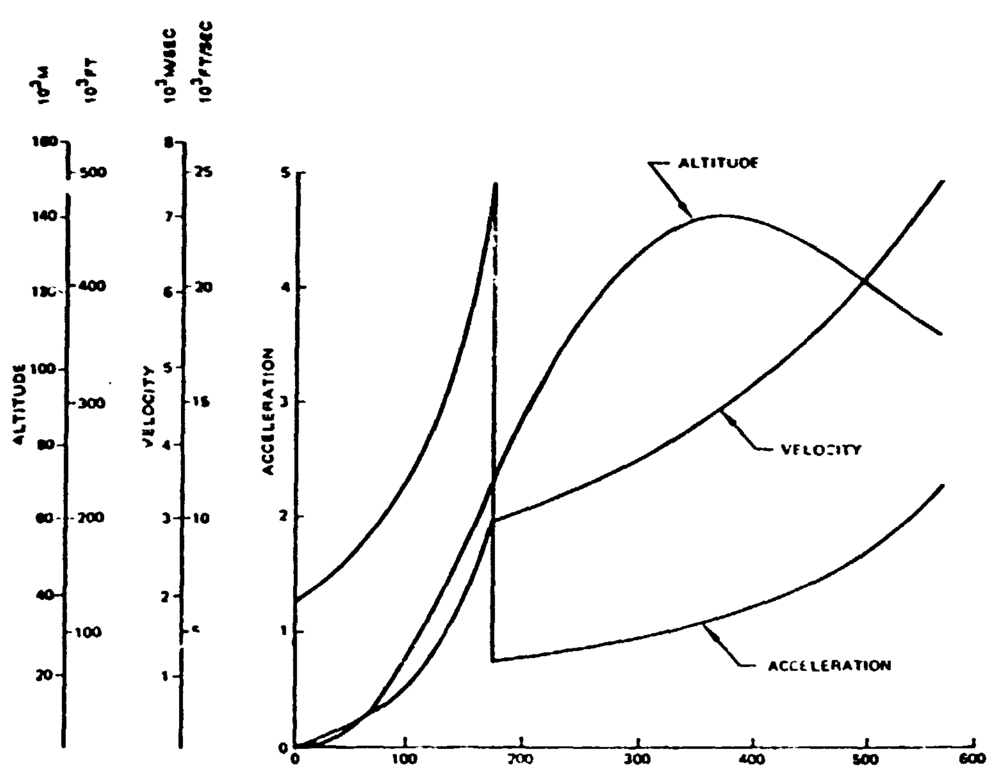


Figure 4.2-2. 2-Stage Ballistic Vehicle Ascent Performance Characteristics

Second Stage

T/W @ Ignition = 0.76

Maximum Acceleration = 2.28 g's

Stage Burn Time = 394.84 sec.

At main engine cutoff (MECO) the trajectory characteristics are as follows:

Altitude = 110948 m

Relative Velocity = 7540 m/sec

Burnout Mass = 749583 kg

Cost

A DDT&E cost of slightly more than $7.1B and a 1st unit cost of $895.8M are estimated for both the vehicle. The DDT&E estimate includes the equivalent of 2.5 ground test and 2.0 flight test units. The cost per flight for LEO construction is estimated at $8.3 million based on 1875 flights over a 14 year period. A breakdown of this cost is presented in Table 4.2-1.

4.2.1.1.2 Two-Stage Winged Vehicle

Configuration

The reference concept for the winged recoverable vehicle is shown in Figure 4.2-3. Main propulsion is provided by sixteen (16) RP-1/LO_2/LH_2 gas generator cycle engines similar to those on the two-stage ballistic vehicle. The following engine characteristics were used in the analysis:

Propellants	RP-1/LO_2/LH_2
Thrust-Vacuum	8.275×10^6 N
Chamber Pressure	29300 kpa
Mixture Ratio	2.9:1
Specific Impulse (S.L./Vac)	323.5/350.7 sec

The total mass of the sixteen engines and the associated accessories and gimbals is 128090 kg.

Flight Characteristics

The ascent trajectory characteristics for the vehicle are shown in Figure 4.2-4. The major characteristics are summarized as follows:

First Stage

T/W @ Ignition = 1.30

Maximum Dynamic Pressure = 34.446 kpa

Maximum Acceleration = 3.49 g's

Stage Burn Time = 147.96 sec.

Dynamic Pressure at Staging = 1819 pa

Second Stage

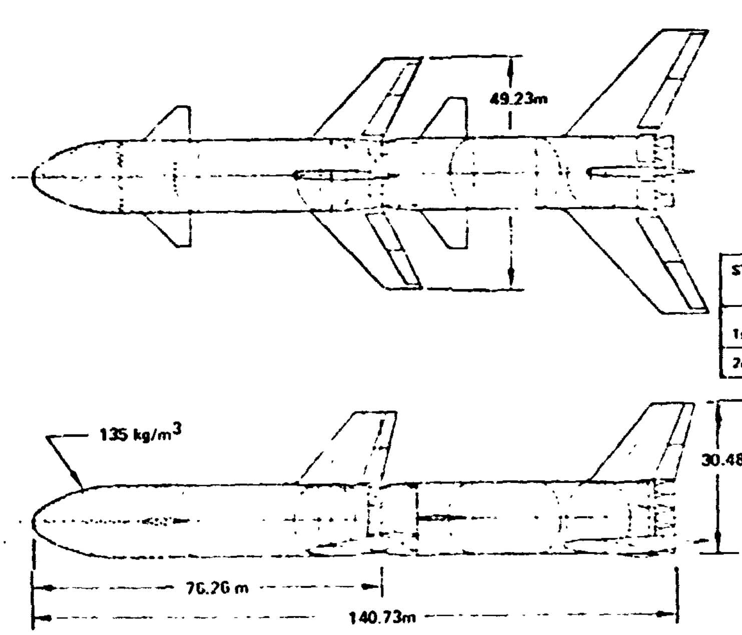
T/W @ Ignition = 0.95

Maximum Acceleration = 3.07 g's

Stage Burn Time = 351.78 sec.

Table 4.2-1. 2-Stage Ballistic Vehicle Average Operating Cost/Flight–LEO Assembly

WBS ELEMENT	COST BY WBS LEVEL – $M				
	1	2	3	4	5
OPERATIONS COST	8.332				
PROGRAM DIRECT		6.755			
PROGRAM SUPPORT			0.317		
PRODUCTION AND SPARES			3.342		
STAGE 1				2.032	
AIRFRAME					1.061
ENGINES					0.971
STAGE 2				1.097	
AIRFRAME					0.581
ENGINES					0.516
PAYLOAD SHROUD				0.213	
TOOLING			0.466		
STAGE 1				0.318	
STAGE 2				0.132	
PAYLOAD SHROUD				0.016	
GROUND OPS/SYS			2.630		
GROUND OPS				0.426	
GROUND SYS				0.056	
GSE SUSTAINING ENGR				0.053	
GSE SPARES				0.112	
PROPELLANT				1.964	
OTHER				0.019	
DIRECT MANPOWER		0.768			
CIVIL SERVICE			0.402		
SUPPORT CONTRACTOR			0.366		
INDIRECT MANPOWER		0.809			
CIVIL SERVICE			0.451		
SUPPORT CONTRACTOR			0.358		



VEHICLE CHARACTERISTICS

- GLOW = 9.566 X 10^6 kg
- BLOW = 6.445 X 10^6 kg
  W_{p_1} = 5.696 X 10^6 kg
- ULOW = 2.739 X 10^6 kg
  W_{p_2} = 2.306 X 10^6 kg
- PAYLOAD = 0.381 X 10^6 kg
- T/W AT LIFTOFF = 1.30
- MAIN PROPULSION

STAGE	NUMBER & TYPE	ϵ	THRUST/ENG. (10^6N VAC.)	I_{sp} – Vac. (Sec.)
1st	16–LO_2/RP 1	42.5	8.275	350.7
2nd	14–SSME	77.5	2.091	455.2

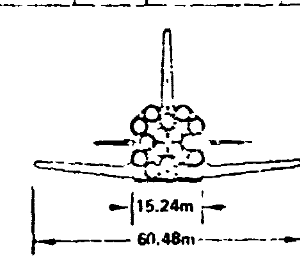


Figure 4.2-3. 2-Stage Winged SPS Launch Vehicle

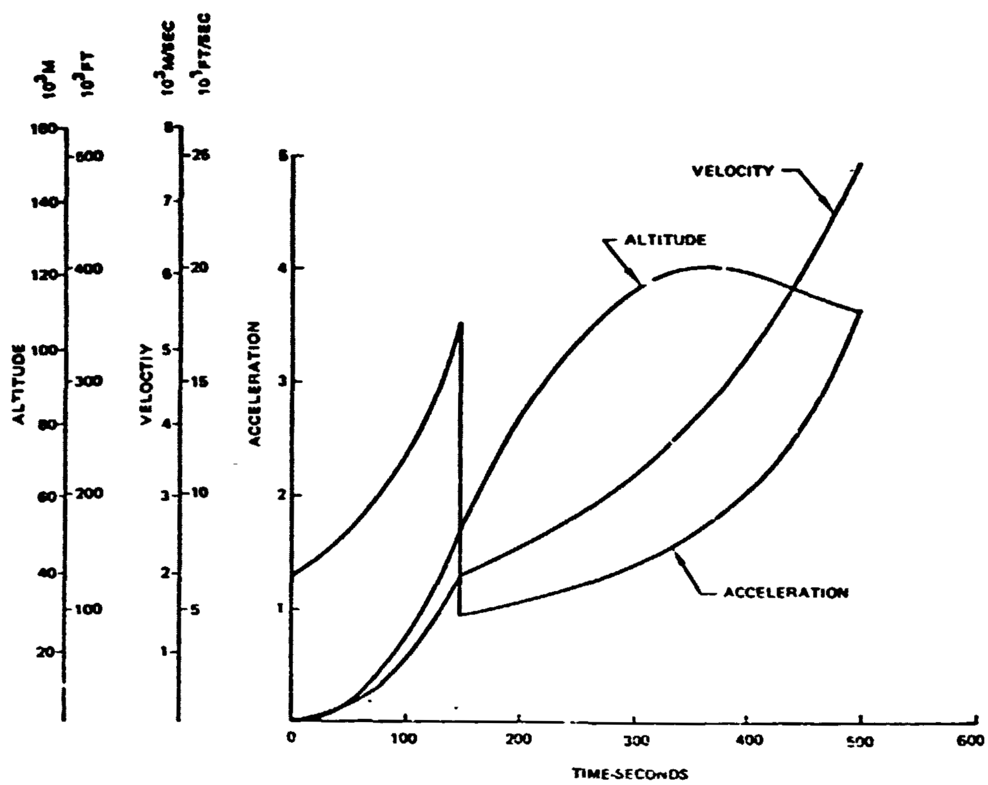


Figure 4.2-4. 2-Stage Winged Vehicle Ascent Performance Characteristics

Cost

A DDT&E cost of \$9.1B is estimated for the vehicle with the booster contributing \$5.2B and the upper stage the remainder. TFU for the vehicle is slightly over \$1 billion with the cost about equally split between the booster and upper stage. The cost per flight for the two stage winged vehicle for GEO construction was estimated at \$7.9 million. A breakdown of these costs are presented in Table 4.2-2. A cost per flight was not derived for LEO construction although a reasonable estimate would be \$8.5 to \$8.8 million.

4.2.1.2 Personnel Carrier Vehicle

The personnel carrier vehicle provides for the transportation of the crews between earth and low earth orbit. The vehicle is a derivative of the current Space Shuttle system which incorporates a liquid propellant booster in place of the Solid Rocket Boosters (SRB's). A series-burn ascent mode was selected and as a result a reduced External Tank (ET) propellant load is required.

Configuration

The personnel launch vehicle, shown in Figure 4.2-5, incorporates a propane fueled booster, External Tank and Space Shuttle Orbiter. Overall vehicle geometry and characteristics are shown on the figure. The overall length of 60.92 m is due to the tandem arrangement rather than the side-mounted concept in the current Shuttle system.

The booster stage is powered by four C_3H_8/LO_2 engines which provide 8.523×10^6N of vacuum thrust. The following engine characteristics were used in the analysis:

Propellants	C_3H_8/LO_2
Thrust - Vacuum	8.523×10^6N
Chamber Pressure	20685 kpa
Mixture Ratio	2.68:1
Specific Impulse (S.L./Vac.)	304.1/340.0 sec
Total Flow Rate/Engine	2556.5 kg/sec

The pressurization gases are heated GH_e and GO_2 for the main tanks. Individual propellant delivery lines are provided to each engine. The total mass of the ascent propulsion system is 47 138 kg.

Flight Characteristics

The personnel carrier vehicle performance was calculated based on the following ground rules:

- Kennedy Space Center (KSC) was the launch site (latitude = 28.5°)
- ΔV Reserves = .85% V_I
- Delivery Orbit

  Altitude = 477 km circular

  Inclination = 31°

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Table 4.2-2. Average Operating Cost/Flight–GEO Assembly

WBS ELEMENT	COST BY WBS LEVEL – $M				
	1	2	3	4	5
OPERATIONS COST					
PROGRAM DIRECT	7.934	6.517			
PROGRAM SUPPORT			0.281		
PRODUCTION AND SPARES			3.239		
STAGE 1				1.575	
AIRFRAME					0.999
ENGINES					0.576
STAGE 2				1.664	
AIRFRAME					0.903
ENGINES					0.761
TOOLING			0.421		
STAGE 1				0.259	
STAGE 2				0.162	
GROUND OPS/SYS			2.576		
GROUND OPS				0.355	
GROUND SYS				0.050	
GSE SUSTAINING ENGR				0.047	
GSE SPARES				0.106	
PROPELLANT				2.001	
OTHER				0.017	
DIRECT MANPOWER		0.682			
CIVIL SERVICE			0.357		
SUPPORT CONTRACTOR			0.325		
INDIRECT MANPOWER		0.735			
CIVIL SERVICE			0.400		
SUPPORT CONTRACTOR			0.335		

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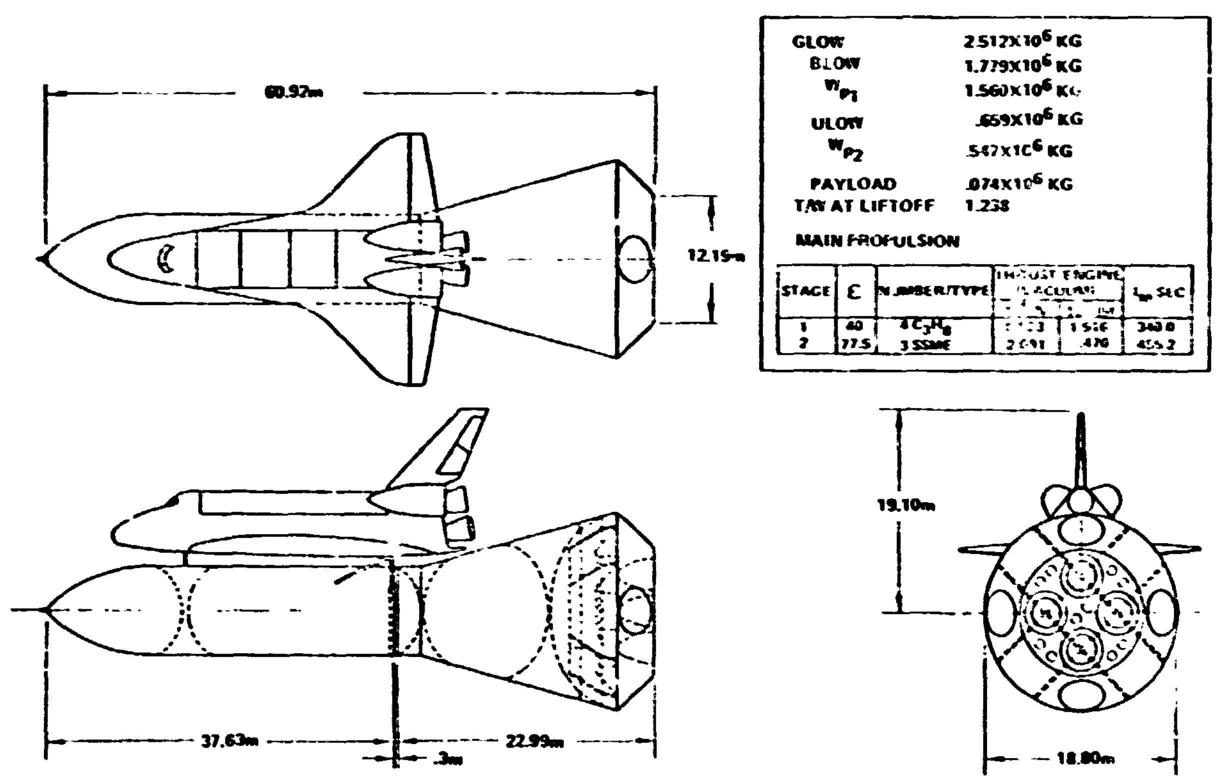

Figure 4.2-5. Personnel Launch Vehicle

The ascent trajectory characteristics are summarized as follows:

T/W @ ignition = 1.24

Maximum Dynamic Pressure = 29.733 kpa

Maximum Acceleration = 3.0 g's

Burn Time = 541.9 seconds

The personnel carrier payload performance is summarized in Table 4.2-3. A net payload of 73550 kg is delivered to the 477 km orbit. The orbiter events including the suborbital jettison of the ET and the resulting vehicle mass by event are also noted. The Shuttle orbiter OMS system performs the majority of the orbiter maneuvers.

Cost

DDT&E cost for the personnel vehicle are estimated at $2.56B with the booster making up $2.5B. The TFU of the reusable booster is estimated at $220 million while the ET contributes $4.9 million. Cost per flight based on [illegible] flights per year and a total of 14 years was estimated at $12.6 million. A breakdown of the cost per flight is presented in Table 4.2-4.

4.2.2 Payload Packaging

A significant factor in the launch aspect of power satellites is the component packaging density and its impact on the number of launches required and/or the type of payload shroud that is used. Typical characteristics for the reference photovoltaic satellite components are shown in Figure 4.2-6. The component presenting the greatest concern is that of the antenna subarrays which have a median packaging density of only 28 kg/m^3. Similar packaging characteristics were defined for the thermal engine satellite. The significance of this density can be fully appreciated by realizing that the payload shroud had previously been sized for a payload density of 75 kg/m^3.

The impact of the component packaging density is illustrated in Figure 4.2-7. The photovoltaic satellite results in an average density of 95 kg/m^3. A mass limited launch condition requires a packaging density of approximately 93 kg/m^3 when applying a 0.7 volume utilization factor to the 17.5 m dia. X 23 m height payload shroud that has a wet volume density of 75 kg/m^3.

The thermal engine satellite has a density of approximately 60 kilograms per cubic meter primarily due to the low density of the radiators, reflecting facets and antenna subarrays. Should the antenna subarrays be divided into a waveguide structure section and klystron tube section, the density would go up to 76 kilograms per cubic meter. This approach however, requires assembly of the subarrays in orbit which is not deemed desirable at this time. Consequently, the thermal engine concept presents a difficult case for achieving mass limited launch conditions. The number of flights for the photovoltaic satellite reflect mass limited launch conditions. The thermal engine system is shown for both an expendable shroud large enough to reach a mass limited condition and a reusable shroud option. Launch cost for these options are compared in the third set of bars. For the

575-001

Table 4.2-3. Personnel Launch Vehicle Performance Mass Statement

DRY MASS	
VEHICLE ELEMENT	10^3 KG
BOOSTER	(164.68)
STRUCTURE	80.52
THERMAL PROTECTION SYSTEM	10.41
LANDING SYSTEM & RCS	5.48
ASCENT PROPULSION	47.14
PRIME POWER	.82
POWER CONV/DIST	1.73
ECS	.86
AVIONICS	2.74
GROWTH	14.98
EXTERNAL TANK	(26.73)
ORBITER	(68.56)
DRY MASS =	(259.97)

SECOND STAGE SEQUENCE	
EVENT	MASS AFTER EVENT 10^3 KG
STAGE AT MECO	187.29
ΔV RESERVE	183.98
DROP ET	155.72
PERIGEE BURN	154.17
APOGEE CIRCULARIZATION	148.94
RCS TRIM	148.05
OMS TRIM	147.54
DEPLOY PAYLOAD (P/L = 73 550 kg)	73.99
DEORBIT ΔV	71.21

Table 4.2-4. Personnel Carrier Average Cost/Flight (256 Flights/Year For 14 Years)

WBS ELEMENT	COST BY WBS LEVEL - $M (1977 $)			
	1	2	3	4
TOTAL PROGRAM OPERATING COST	12.619			
PROGRAM DIRECT		9.388		
PROGRAM SUPPORT			0.908	
PRODUCTION & SPARES			3.436	
ORBITER PRODUCTION				1.536
ORBITER SPARES				0.342
SSME'S				0.325
BOOSTER AIRFRAME				0.279
BOOSTER ENGINES				0.290
CREW RELATED GFE				0.165
EXPENDABLE HARDWARE - E.T.			1.858	
TOOLING			0.437	
GROUND OPS/SYS			2.759	
GROUND OPS				1.413
GSE SPARES				0.236
PROPELLANT				0.886
OTHER				0.074
DIRECT MANPOWER		1.568		
CIVIL SERVICE			0.861	
SUPPORT CONTRACTOR			0.707	
INDIRECT MANPOWER		1.663		
CIVIL SERVICE			0.755	
SUPPORT CONTRACTOR			0.908	

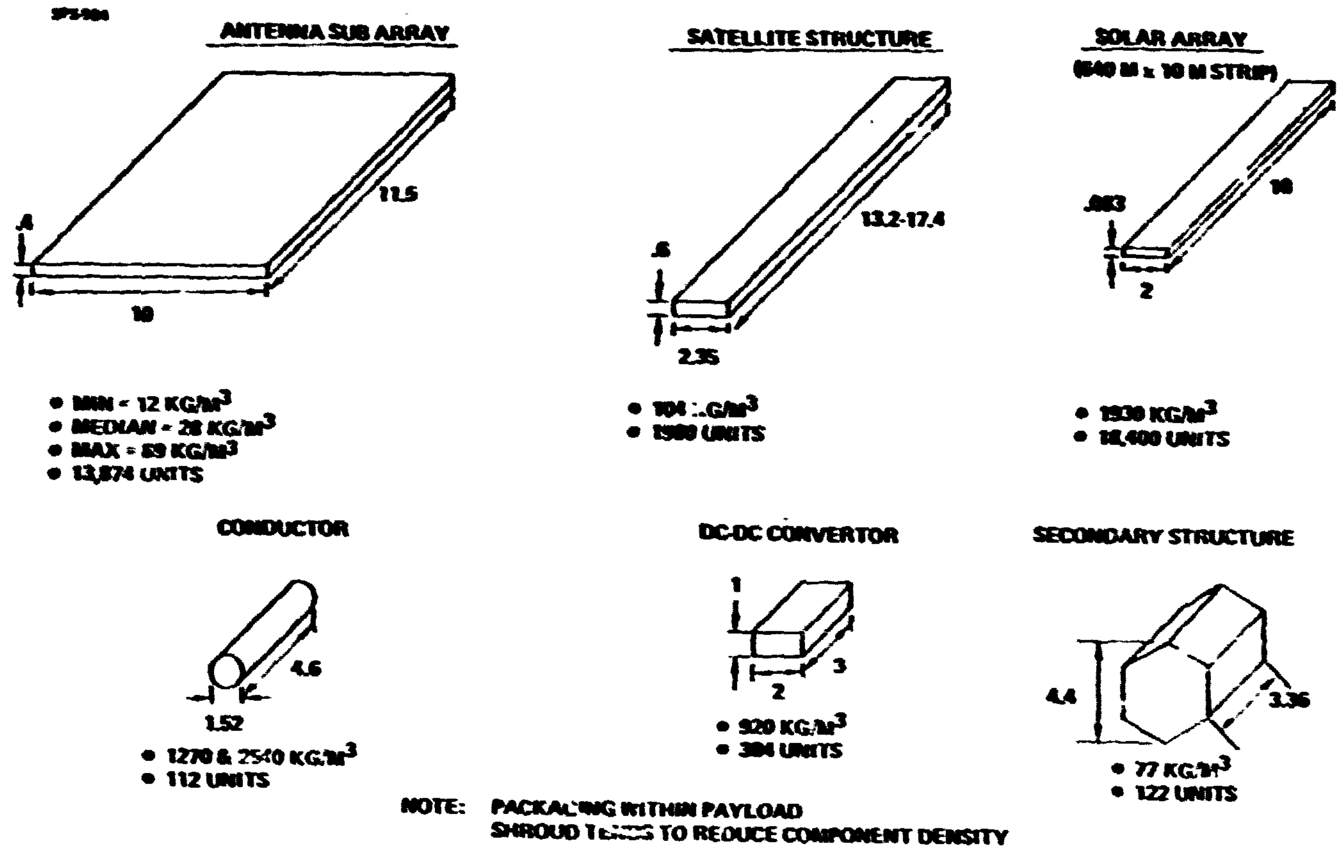

Figure 4.2-6. Component Packaging Characteristics

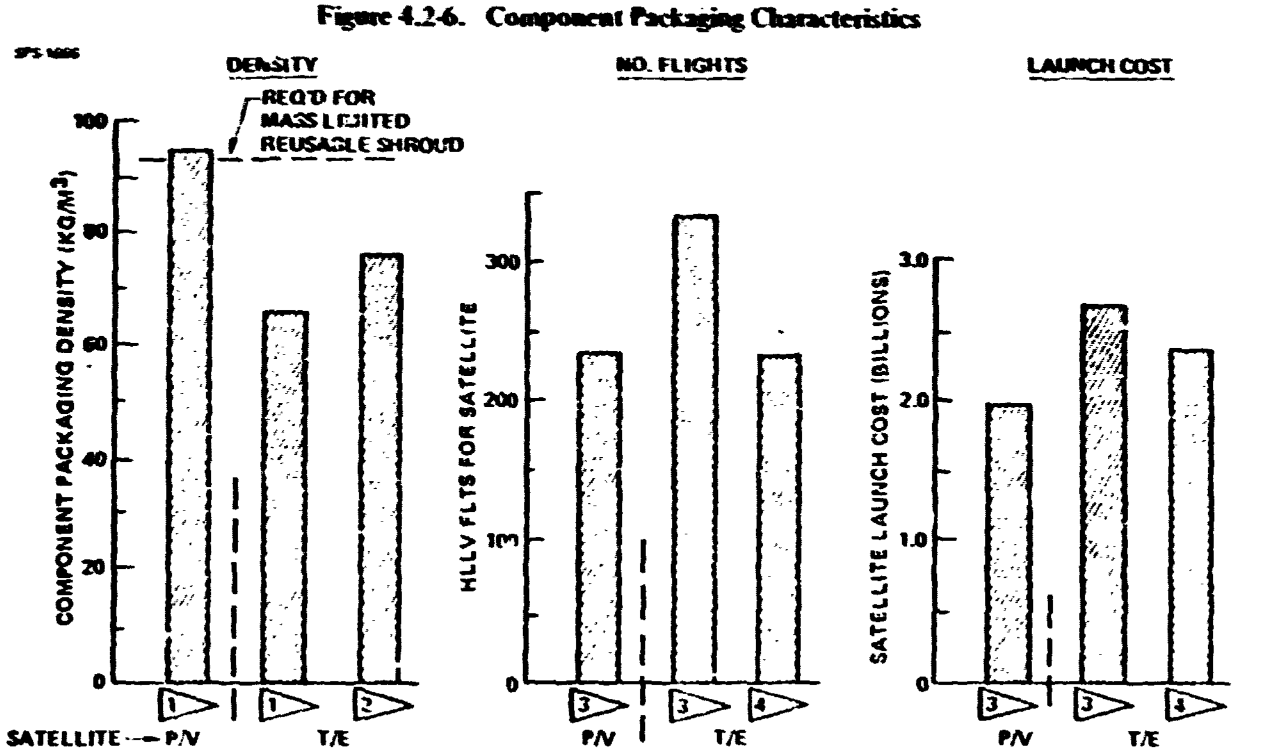

Figure 4.2-7. Component Packaging Density Impact

thermal engine system, the expendable shroud shows approximately a 300 million dollar savings per satellite as compared with a reusable shroud due to the low unit cost (2 million dollars) for the expendable shroud when large quantities are procured. It should be mentioned however, that the thermal engine satellite will also utilize reusable shrouds for the delivery of crew and supplies and delivery of construction requirements.

One possible mix of the various components for delivery to LEO is illustrated in Figure 4.2-8. The number of flights indicated is that associated with the mix of components and is not meant to be indicative of the actual launch sequence. It should also be noted, the number of flights is associated with a photovoltaic satellite with a mass of 110 million kg rather than 100 million kg for the final Part II satellite. As indicated, the dominating component was the antenna subarrays, which was included in 246 out of 247 total flights (of identifiable hardware). Fortunately, the high density solar arrays can be used to offset the lower density subarrays during most of their launches. In summary, unlike the Part I analysis where only about 25 to 30% of the payload shroud was used (antenna undefined), a more complete understanding of the antenna and desire to deliver subarrays fully assembled has resulted in using the full length of the payload shroud in order to achieve a mass limited launch condition.

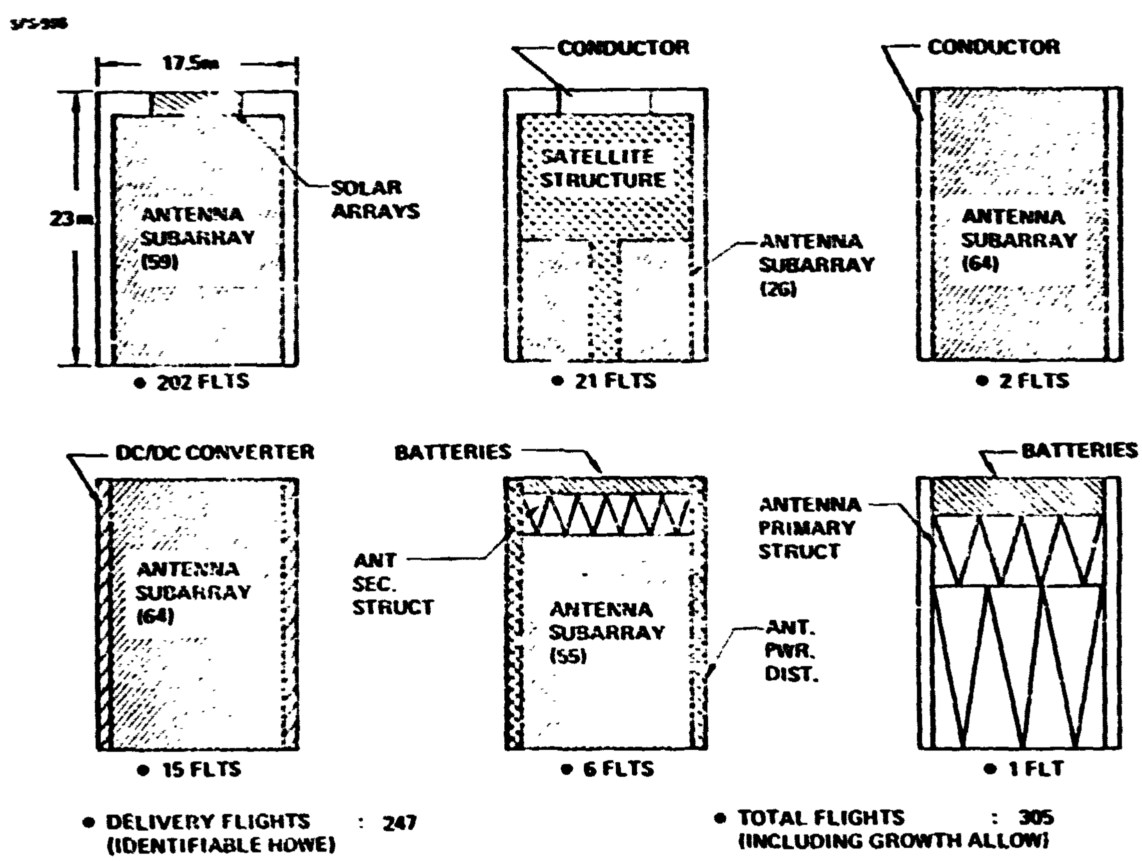

Figure 4.2-8. Component Mix Per Delivery Flight

4.2.3 Exhaust Product Analysis

The exhaust products of the launch vehicles can be an environmental concern due to the release of chemical pollutants into the atmosphere. The results of the effort reported in this section supercedes the data reported in the Part I final report (Section 8.0 of D180-20689-5). The proposed launch vehicle concepts are two stage devices, in which the first or booster stage burns a hydrocarbon fuel and oxygen and the second stage uses hydrogen and oxygen as the propellant combination. Two types of launch vehicles have been identified, either the two-stage ballistic recoverable or the two-stage winged recoverable concepts. In addition, the following two variations of the booster engine have been proposed:

1. A 29 300 kPa (4250 psi) chamber pressure gas generator cycle engine using RP-1/LO_2/LH_2 propellants with a 2.9:1 oxidizer to fuel mixture ratio.

2. A 34 475 kPa (5000 psi) chamber pressure gas generator cycle engine using LCH_4/LO_2 propellants with a 3.0:1 oxidizer to fuel mixture ratio.

Either booster engine concept is applicable for the SPS application. However, for purposes of this report, the ballistic recoverable booster was sized for the RP-1 LO_2/LH_2 type engine and the winged booster uses the LCH_4 LO_2 propellant combination.

Therefore, the exhaust products of both booster engine types could be assessed on a per flight basis. The second stages of both vehicle concepts are powered by Space Shuttle Main Engines (SSME's) which use a LH_2 LO_2 propellant combination at a 6:1 oxidizer to fuel mixture ratio.

The launch vehicle exhaust products are distributed through the various regions of the atmosphere. Using the definition of the various layers of the atmosphere shown in Figure 4.2-9 and the propellant flow rates as a function of altitude time, the propellant usage in each layer can be established. Table 4.2-5 identifies the predicted amount of propellant consumed in each layer of the atmosphere.

Table 4.2-5

Propellant Consumption Per Flight in the Various Layers of the Atmosphere

Region	Ballistic Recoverable		Winged Recoverable	
Troposphere	3.012×10^6 kg	RP-1 LO_2 LH_2	3.196×10^6 kg	LCH_4/LO_2
Stratosphere	3.172×10^6 kg	RP-1 LO_2/LH_2	3.368×10^6 kg	LCH_4/LO_2
Mesosphere [1]	1.273×10^6 kg	RP-1 LO_2 LH_2	0.272×10^6 kg	LCH_4 LO_2
	0.055×10^6 kg	LH_2 LO_2	0.289×10^6 kg	LH_2 LO_2
Ionosphere	1.425×10^6 kg	LH_2 LO_2	2.006×10^6 kg	LH_2 LO_2

[1] The booster burnout condition occurs in the mesosphere for both launch vehicle concepts.

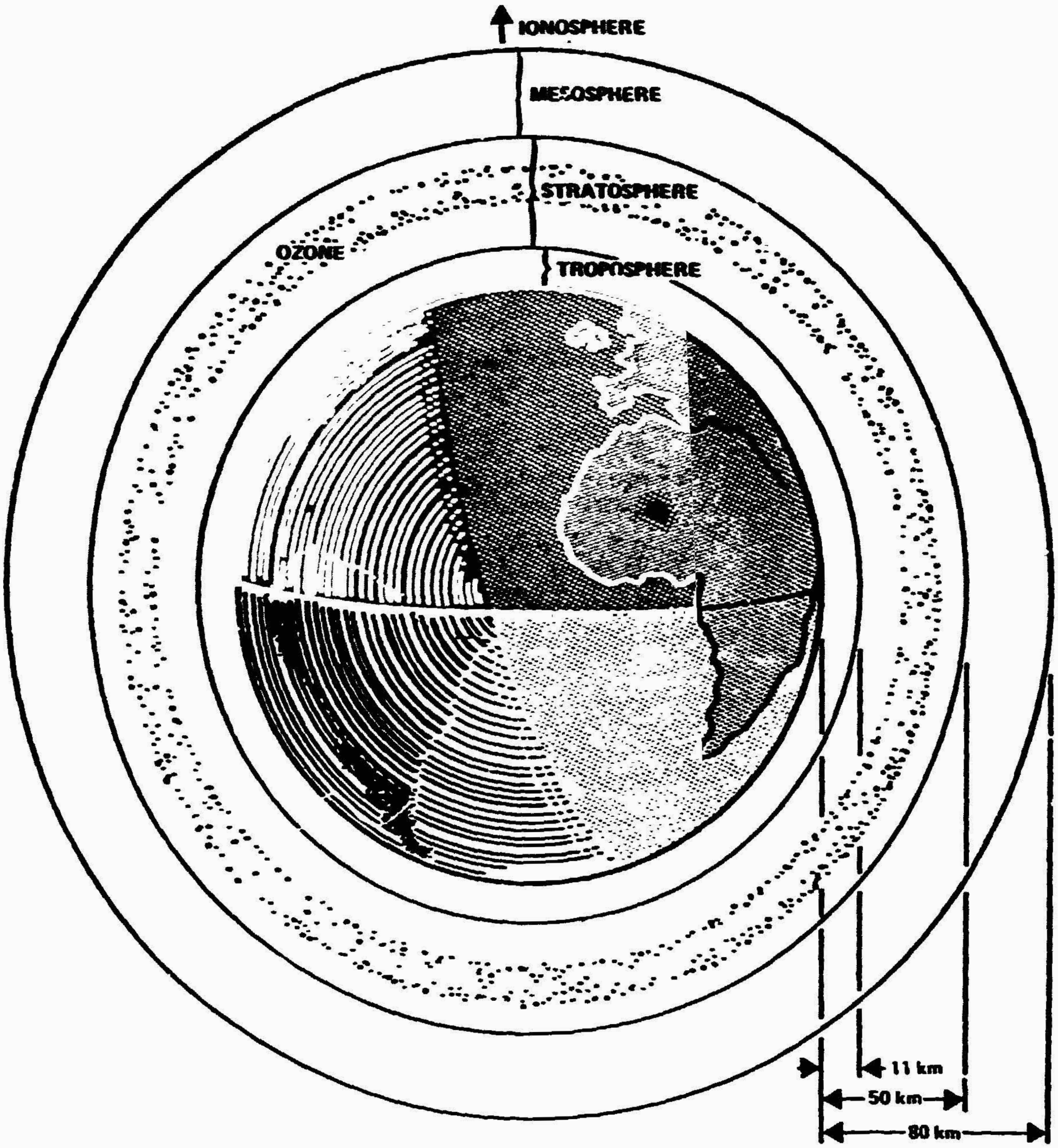

Figure 4.2-9 Locations of the Troposphere, Stratosphere, Mesosphere, and Ionosphere Atmosphereic Layers

The exhaust products at the nozzle exit for both types of booster engines are listed in Table 4.2-6 including the magnitude of each combustion product. These values "do not" include the effects of atmospheric reactions and recombinations due to "afterburning" phenomena. Future studies on the effects of exhaust products should include these phenomena. The SSME exhaust products are water (96.5%) and free hydrogen (3.5%). The booster engine exhaust products are mainly CO_2, H_2O, CO, and H_2 which account for 99.93% of the total.

The distribution of the exhaust products, in the atmosphere on a per flight basis is shown in Table 4.2-7, for both vehicle concepts. It is recognized that due to the afterburning effect the exhaust products will react with air to form oxides of nitrogen and possibly very small amounts of organic nitrogen compounds. For example, the amount of $(NO)_x$ produced by the F-1 engines on the Saturn V has been estimated to be 0.4% of the exhaust gas mass in the troposphere and about 0.002% in the stratosphere. The lower production of $(NO)_x$ in the stratosphere is partly due to the lower density and partly due to the lower temperature of the more expanded plume.

The booster engines proposed for use on the SPS Launch Vehicle will be of more advanced design than the F-1. Even if no environmental constraints are placed on the engine, performance considerations will tend to reduce the pollutant levels. Two of the changes which will be significant are higher chamber pressure and a different operating cycle.

Since the combustion temperature is essentially independent of the chamber pressure and since the higher chamber pressure results in a higher optimum expansion ratio, the plume boundary temperature will be lower, especially at altitude. For a 40:1 expansion ratio the exit static temperature is about 2000°K. The production of $(NO)_x$ will, therefore, be reduced. The $(NO)_x$ level for the candidate booster engines should be near or below the values for the Space Shuttle Main Engine which have been estimated at 0.01% in the troposphere and 0.001% in the stratosphere.

Table 4.2-6 Rocket Engine Exhaust Products–Percent of Propellant Flow Mass

	Species of Exhaust Products at the Nozzle Exit–Percent of Propellant Flow								
	H_2O	CHO	CH_4	H	CO	OH	CH_2O	CO_2	H_2
Booster Engine									
RP-1/LO_2/LH_2	34.538	3.127×10^{-5}	2.881×10^{-6}	1.649×10^{-4}	24.590	4.089×10^{-4}	1.668×10^{-6}	39.618	1.193
LCH_4/LO_2	39.881	3.623×10^{-6}	2.229×10^{-6}	3.222×10^{-6}	18.053	1.803×10^{-6}	1.537×10^{-6}	40.197	1.802

Table 4.2-7 Distribution of Exhaust Products in the Various Regions of the Atmosphere

Region	Exhaust Product	Magnitude of Exhaust Species/Flight – 10^3kg								
		H_2O	CHO	CH_4	H	CO	OH	CH_2O	CO_2	H_2
2-Stage Ballistic Recoverable										
	Troposphere	1040	*	*	0.005	740	0.012	*	1193	36
Booster	Stratosphere	1096			0.005	780	0.013		1257	38
	Mesosphere	440			0.002	313	0.005		504	15
Upper	Mesosphere	53			*	*	*		*	2
Stage	Ionosphere	1375	*	*	*	*	*	*	*	50
2-Stage Winged										
	Troposphere	1275	*	*	*	577	*	*	1285	58
Booster	Stratosphere	1343				608			1354	61
	Mesosphere	108				49			109	5
Upper	Mesosphere	279				*			*	10
Stage	Ionosphere	1936	*	*	*	*	*	*	*	70

Less than 1 kg per flight.

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4.3 ORBIT TRANSFER SYSTEMS

4.3.1 Satellite Delivery System

Analysis of the two construction location options have resulted in the detail analysis of two satellite delivery systems. The LEO construction concept for the reference photovoltaic satellite is illustrated in Figure 4.3-1. Eight modules and two antennas are constructed in LEO. All modules are transported to GEO using self power electric propulsion. Two of the modules will transport an antenna while the remaining six modules are transported alone.

The GEO construction concept is illustrated in Figure 4.3-2. This concept involves a staging depot in LEO which has the capability to transfer payloads from the launch vehicles to orbit transfer vehicles (OTV) and maintain the OTV fleet. Transfer of all payloads from LEO to GEO is accomplished using LO_2/LH_2 OTV's. Construction of the entire satellite including antennas is done in GEO.

Subsequent sections will discuss each of these delivery systems in terms of their characteristics which have been updated during Part II. Numerous trades and characteristics have not been updated during Part II but may be found in Part I Volume V.

4.3.1.1 Photovoltaic Satellite Self Power System

The reference photovoltaic satellite uses annealable silicon cells with a concentration ratio of one. The self power elements required to transfer each module are discussed in the following sections.

4.3.1.1.1 Configuration

The configuration arrangement and characteristics of the system elements used in the transfer of each satellite module is shown in Figure 4.3-3. The general characteristics indicate a 5% oversizing of the satellite to compensate for the radiation degradation occurring during passage through the Van Allen belt and the inability to anneal out all of the damage after reaching GEO. It should also be emphasized at this point, only the arrays needed to provide the required power for transfer are deployed. The remainder of arrays are stowed within radiation proof containers. Cost optimum trip times and I_{sp} values are respectively 180 days and 7,000 seconds.

Thruster modules are located at four corners of the module to provide the most effective thrust vector and satisfy control requirements. Further discussion of the thruster module is provided later in this section. A two axis gimbal system correctly positions the panel. Installation of the thruster module approximately 500 meters from the satellite in conjunction with gimbal limits prevents high velocity ions from impinging on the satellite and causing erosion. Propellant tanks for the thrusters have been located at the center of the satellite module and at the lower surface to provide a more desirable inertia characteristic (the dominating factor in the amount of gravity gradient torque). Radiators dissipate the waste heat from the power processing units.

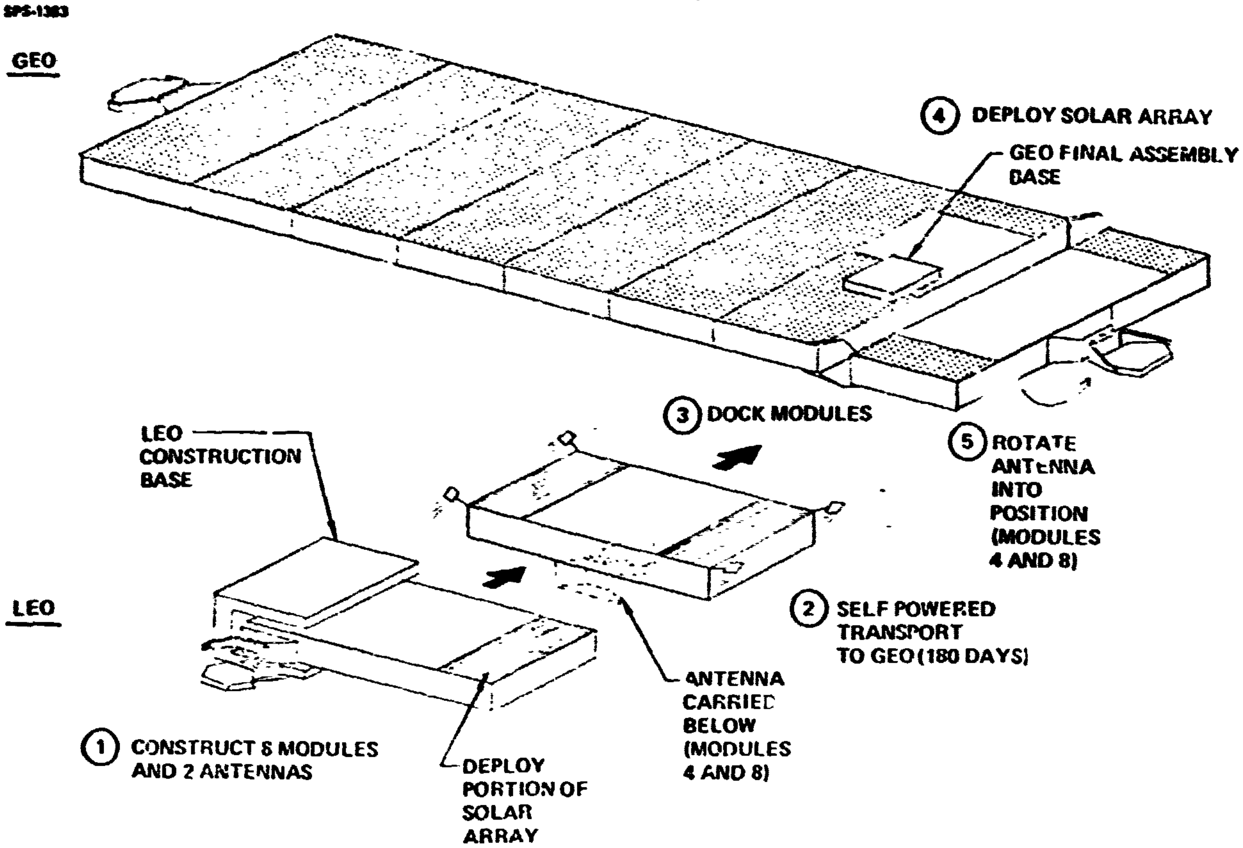

Figure 4.3-1. LEO Construction Concept Photovoltaic Satellite

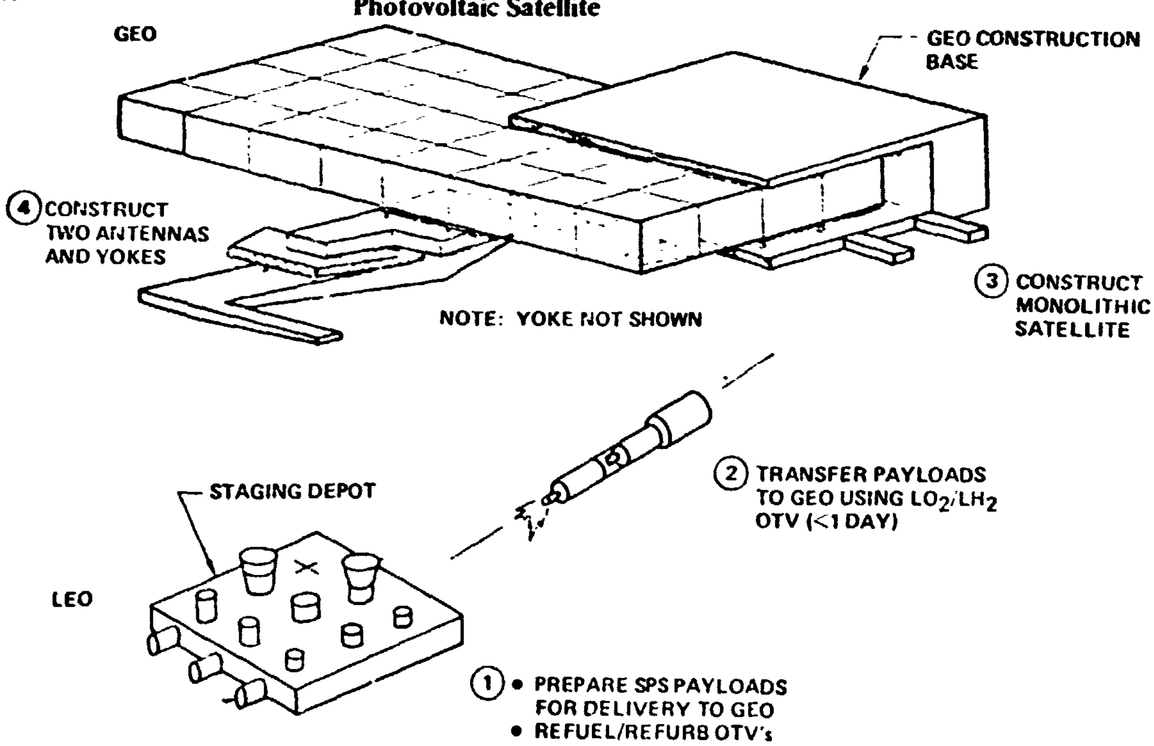

Figure 4.3-2. GEO Construction Concept

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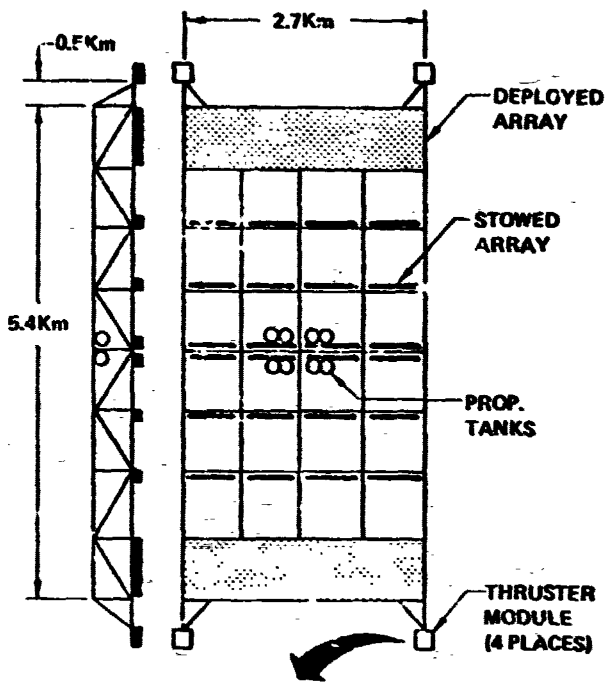
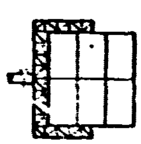


GENERAL CHARACTERISTICS

- 5% OVERSIZING (RADIATION)
- TRIP TIME = 180 DAYS
- ISP = 7000 SEC

MODULE CHARACTERISTICS	NO ANTENNA	WITH ANTENNA
• NO. MODULES	6	2
• MODULE MASS (10^6KG)	8.7	23.7
• POWER REQ'D (10^6Kw)	0.3	0.81
• ARRAY %	13	36
• OTS DRY (10^6KG)	1.1	2.9
• ARGON (10^6KG)	2.0	5.6
• LO_2/LH_2 (10^6KG)	1.0	2.8
• ELEC THRUST (10^3N)	4.5	12.2
• CHEM THRUST (10^3N)	12.0	5.0



	NO ANTENNA	WITH ANTENNA
PANEL SIZE:	24x38m	48x57m
NO. THRUSTERS:	560	1680

Figure 4.3-3. Self Power Configuration Photovoltaic Satellite

Flight control of the module when flying a PEP attitude during transfer results in large gravity gradient torques at several positions in each revolution. Rather than provide the entire control capability with electric thrusters which are quite expensive, the electric system is sized only for the optimum transfer time with the additional required thrust provided by LO_2/LH_2 thrusters. The performance penalty for this approach is actually quite small since by the time 2,500 kilometer altitude is reached the gravity gradient torque is no longer a dominating force.

The mass characteristics of the electric propulsion system elements are directly proportional to the mass of the payload being transferred for the case of fixed trip time and I_{sp}. Consequently, the modules transporting the antennas require considerably more OTS hardware and propellant. The total mass of the satellite in this case is approximately 100 million kg which includes the following penalties for self power transfer. In the area of solar array, an oversizing of 5 percent has been included to compensate for the inability to completely anneal out all the damage to the cells caused by radiation occurring during transfer and for the mismatch in voltage output between the damaged and undamaged cells. The structural impact includes both that of modularity and oversizing. Modularity includes additional vertical members used around the perimeter of the satellite module and lateral beams at the end of the modules as well as the penalties for the transfer of the 15 million kg antenna (includes growth) supported underneath the module. (It should be noted that all module structure has been sized to that dictated by the modules used to transfer the antenna.) The power distribution penalty is related to the additional length of bus caused by the oversizing of the array. The total mass penalty for a LEO constructed satellite is approximately 4.2 million kg for the selected self power transportation system. It should be noted however that the array oversizing and power distribution penalty depend on the particular performance characteristics selected for the self power system.

A typical thruster module configuration consists of a yoke and a thruster panel as shown in Figure 4.3-4. This particular configuration is representative of the equipment involved and their relative physical interfaces, however, the number of subpanel are different from the selected configuration previously shown in Figure 4.3-3. For the selected configuration, modules without antenna have four subpanels, while modules with antenna have 12 subpanels per corner. Included within the yoke are gimbals both at the tripod interface and at the thruster panel interface. Across the tripod gimbal are found two 25 cm dia. lines for PPU coolant, one eight centimeter diameter line for propellant and a 1 m x 0.4 cm electric power conductor. Gimbal capability of ±90 degrees is provided at each gimbal to provide the necessary thrust vector direction. The thruster panel is divided into six subpanels for compatibility with the payload shroud dimension constraint. Each subpanel consists of a basic structural framework with thrusters mounted on one surface and PPU's, fluid lines and electrical lines on the opposite surface. Reinforced edges on the subpanels provide the required stiffness to attach subpanels to each other to form the total thruster panel. Each subpanel has (140) 120 cm diameter thrusters arranged in a pattern of 10 rows with 14 thrusters per row. A 0.15 meter spacing between thrusters has been provided for installation. Two PPU's consisting of DC DC-converter and associated switch gear provides power processing. Each PPU system processes power for 70 thrusters. Radiators for dissipating PPU waste heat are located on the support tripod.

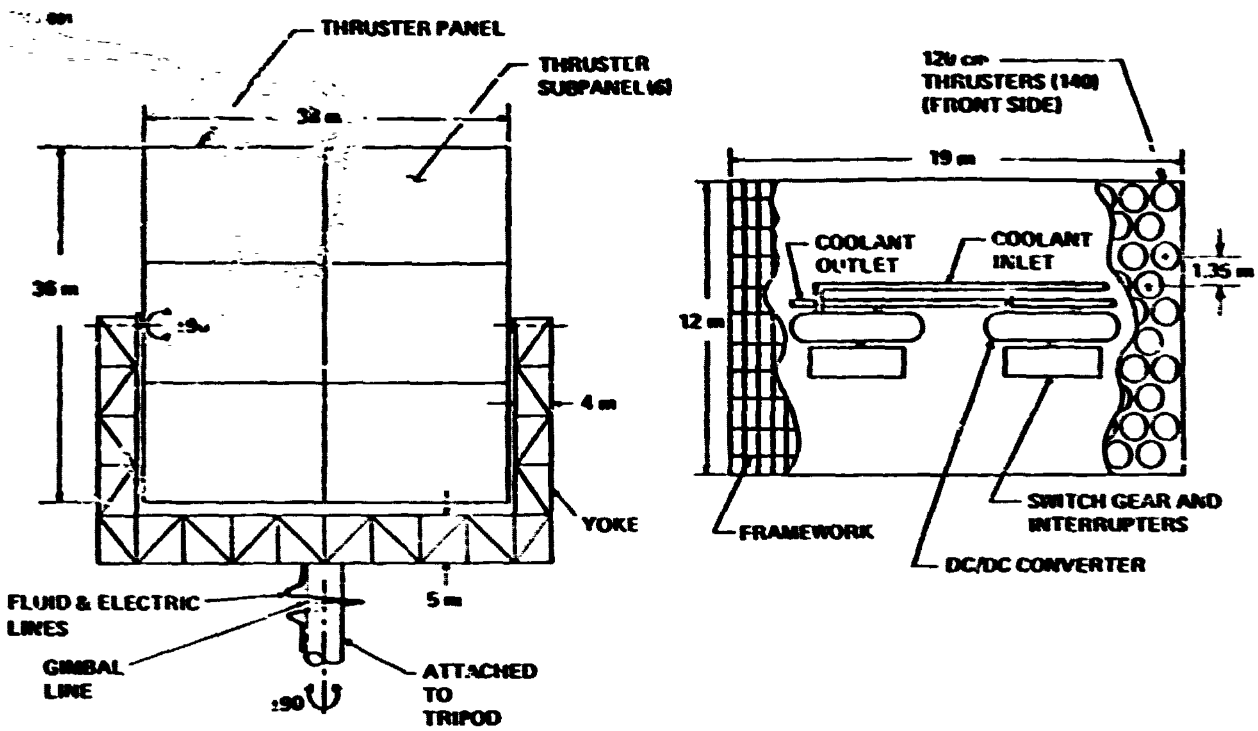


Figure 4.3-4. Thruster Module Configuration

4.3.1.1.2 Subsystems

Seven major system elements are used in the electric propulsion system as shown in Figure 4.3-5. These are the generation of power by the satellite, the distribution of the power to the electric thruster system, conditioning the power by power processing equipment, thrusters and propellant storage. Power processing is estimated at 95% to 96% efficiency, therefore necessitating a thermal control system. Finally, in order to get the required pointing of the thrusters, a gimbal system is required. Each of these systems has been characterized in terms of mass and cost characteristics and incorporated into a cost optimization model. Further discussion on each of these elements follows.

Thrusters

The reference 120 cm ion thruster is illustrated in Figure 4.3-6 with design and selected operating characteristics (resulting from transportation optimization shown in Table 4.3-1. Parametric performance predictions for this thruster are shown in Figure 4.3-7. The parametric data are based on extrapolations from current 30 cm mercury ion thruster technology, including the recent 4A (beam current) demonstration tests which showed that the double current density was feasible, but that thruster life would be reduced roughly 50%. This should be compatible with SPS transfer requirements and is the basis of the selection of a beam current of 80 amperes.

Table 4.3-1. Selected 1.2 M Argon Ion Thruster Characteristics

Fixed Characteristics	
Beam Current	80.0 Amps.
Accel. Voltage	500.0 V
Discharge Voltage	30.0 V (Floating)
Coupling Voltage	11.0 V
Dbl Ion Rates	0.16 (J2/J1)
Neutral Efflux	4.8384 Amp Equiv.
Divergence	0.98
Discharge Loss	187.3 ev/ion
Other Loss	1758.0 W
Utilization	0.892 W
Life	8000 hr
*Weight	50 Kg
Selected Characteristics	
Screen (Beam) Voltage	1500 V
Input Power	120 KW
Thrust	2.7 N
Efficiency	73

*Weight prediction courtesy of T. Masek of HRL

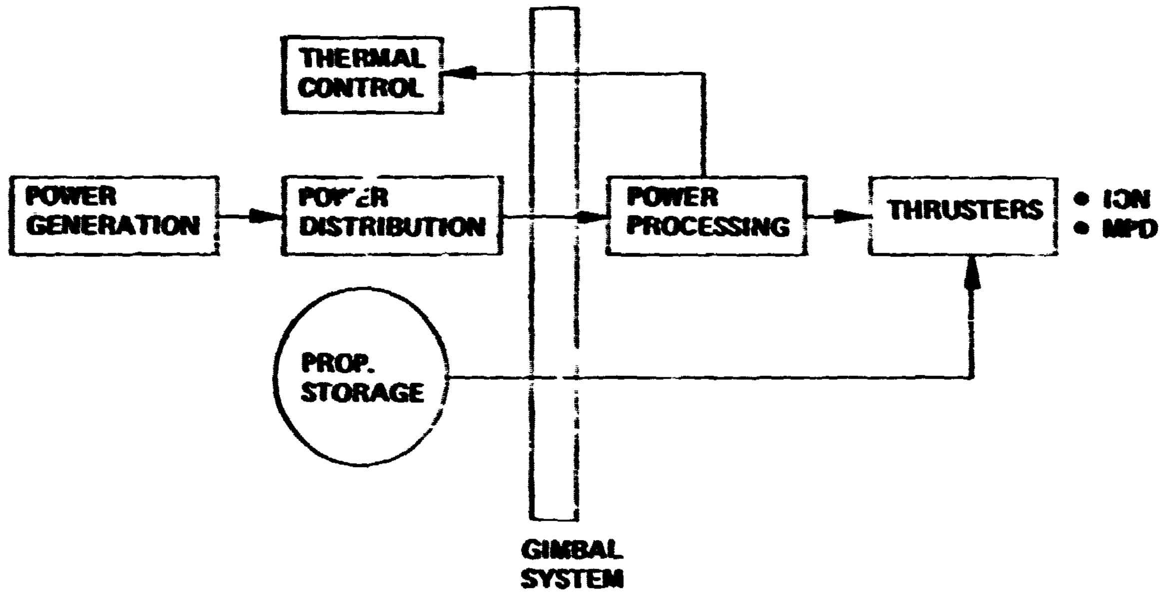


Figure 4.3-5. Electric Propulsion System Elements

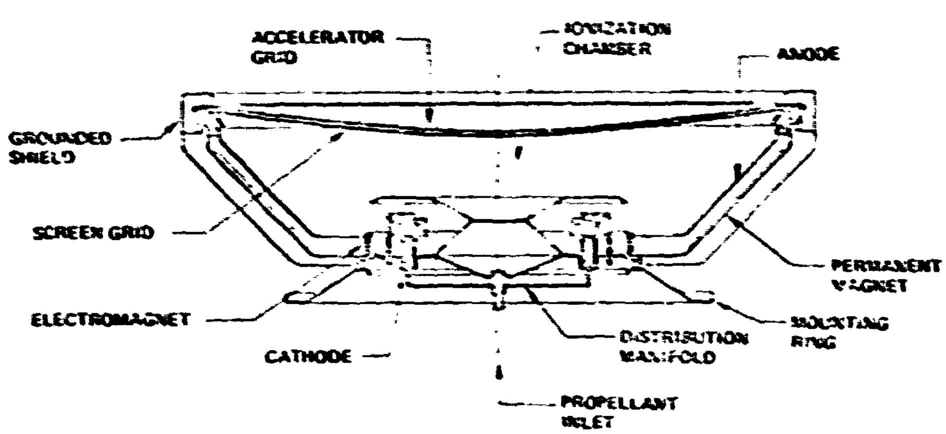


Figure 4.3-6. 120 CM Argon Ion Thruster

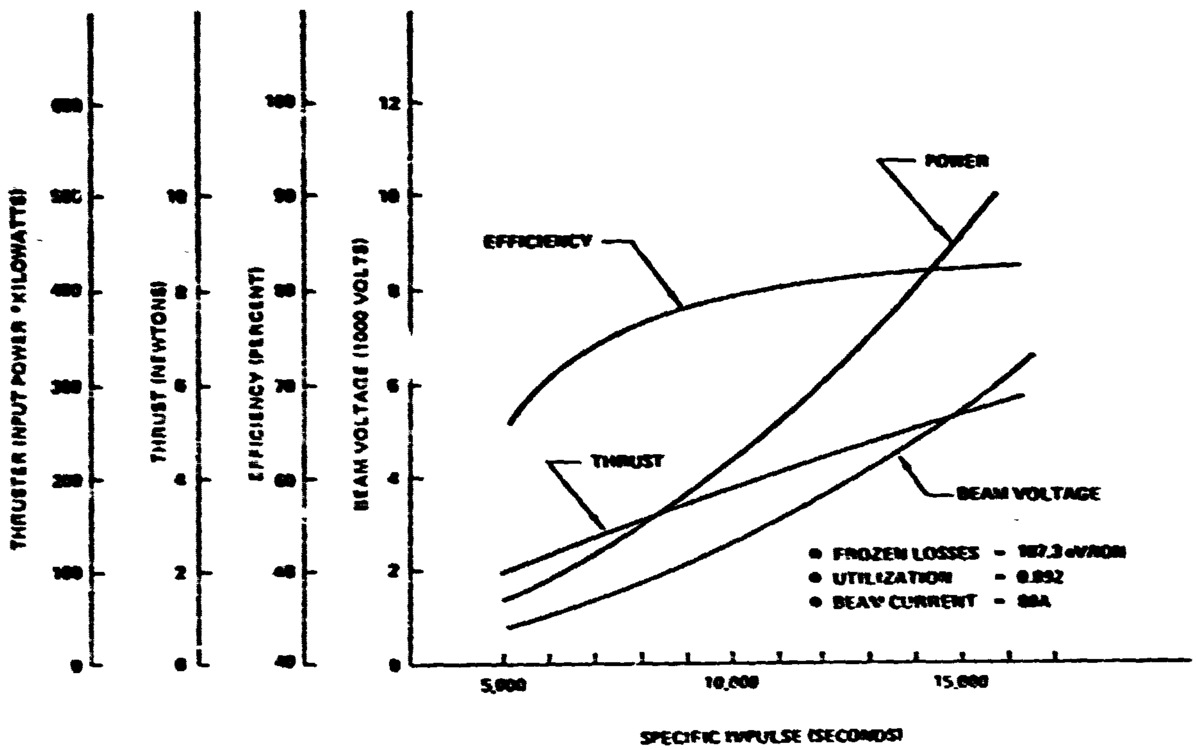


Figure 4.3-7. 120-CM Argon Ion Thruster Performance

Power Processing

The basic power processing concept is to provide each propulsion subarray with its own power processing center. It utilizes a motor generator system to provide the DC/DC conversion and is schematically illustrated in Figure 4.3-8. This approach assumes that multiple thrusters can be operated from common power supplies and that arcing can be controlled by quick acting switches.

Since the current ion thruster technology requires electrical independence among clusters of thrusters to prevent destabilizing electrodynamic interactions among thrusters (principally during grid arcing), quick acting interrupter switches (8) have been placed in the screen and accelerator grid circuits of each of the thrusters in a subarray. Discharge current controllers for each thruster may also be required. These can be "small" motor generators dedicated to each thruster. An isolation switch will be required to effectively remove a failed thruster from the system.

Thermal Control

The DC DC converter used in power processing requires an active thermal control system in order to control its operating temperature to a maximum of 200°C. A heat exchanger transfers heat from the gas circulating in the DC/DC converter to the Therminal 60 coolant loop. A general layout of the thermal control loop including radiator is shown in Figure 4.3-9. Waste heat from all six subpanels are collected into one 24 cm diameter line that passes out to the radiator which is mounted on the support tripod. The radiator rejects 330 KW of waste heat and requires a projected area of $1110m^2$. Power required for pumping the coolant is estimated at 100 KW. Switchgear and interrupter equipment are radiative cooled because of lower waste heat levels.

Electric Power and Distribution

Primary electric power for the propulsion system is obtained from the satellite. The principal voltage requirement during the orbit transfer is that associated with the thrusters. The cost optimum Isp of 7000-7500 seconds requires a 1300-1500 volt input to the thrusters. The GEO operational voltage of the satellite however is 40,000 volts. High voltage generation results in low I^2R losses but high plasma losses at altitudes below 1000 km. Taking these factors into consideration the voltage resulting in the least oversizing and power distribution mass penalty is between 3500 and 4000 volts. A one time switchgear system is included in the power distribution system to enable the arrays utilized during the transfer to switch to providing the operational voltage of 40,000 volts.

Propellant Storage and Delivery Systems

Argon propellant for the electric thrusters is stored in 8 m diameter tanks. Six tanks are required for modules without an antenna and 16 tanks for modules that transfer antennas. Tanks are manifolded and individual supply lines go to each thruster module. The maximum flow rate is 0.2 Kg/sec and 0.6 Kg/sec for the cases of transferring a module without and with an antenna. Propellant delivery from the tanks to each module is accomplished by heating the liquid argon and using the boil-off pressure to drive the gaseous argon. An example of a typical propellant delivery system at the thruster panel is illustrated in Figure 4.3-10 (Note: This is for a 6 subpanel module rather

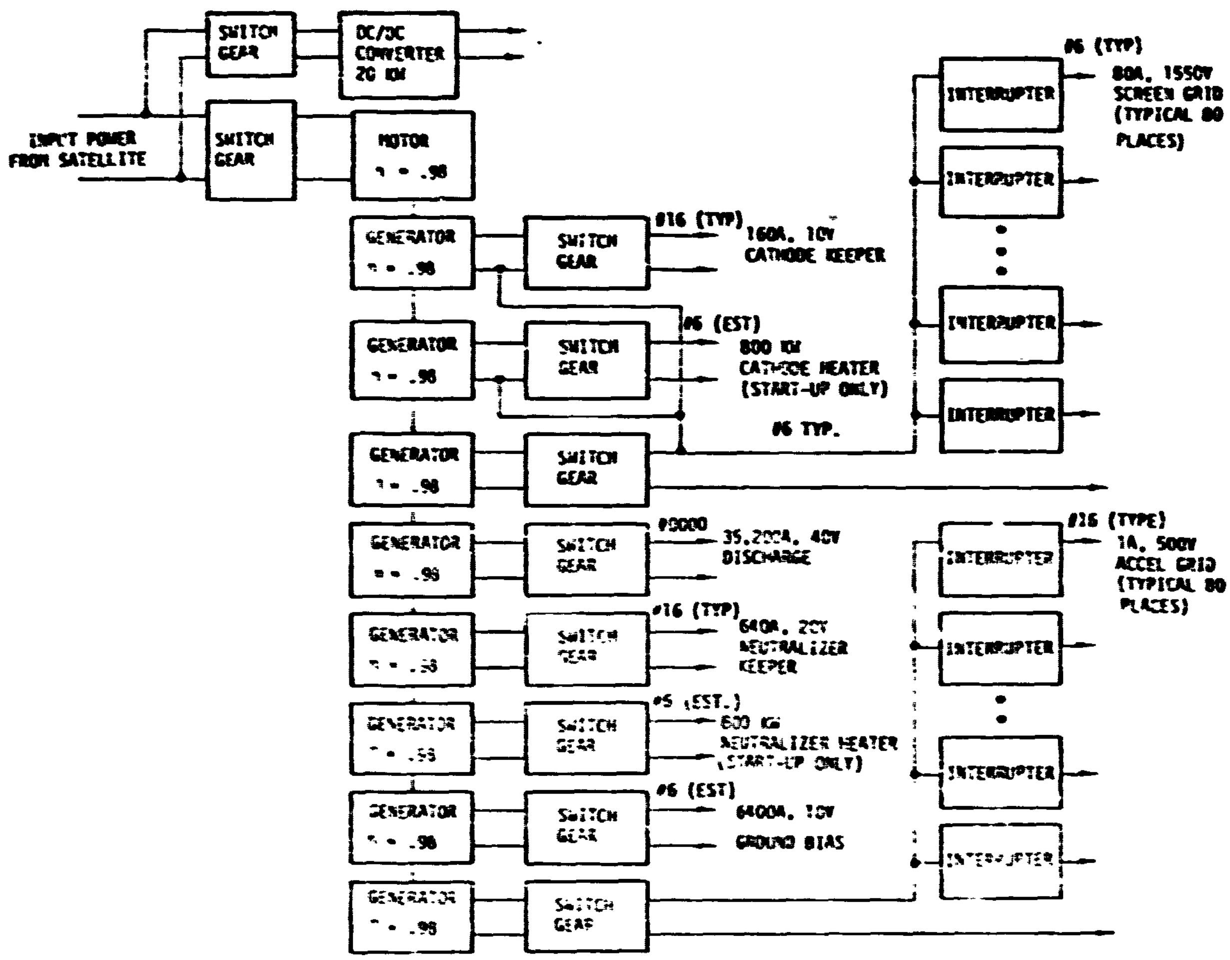


Figure 4.3-8. Propulsion Subarray Power Processing

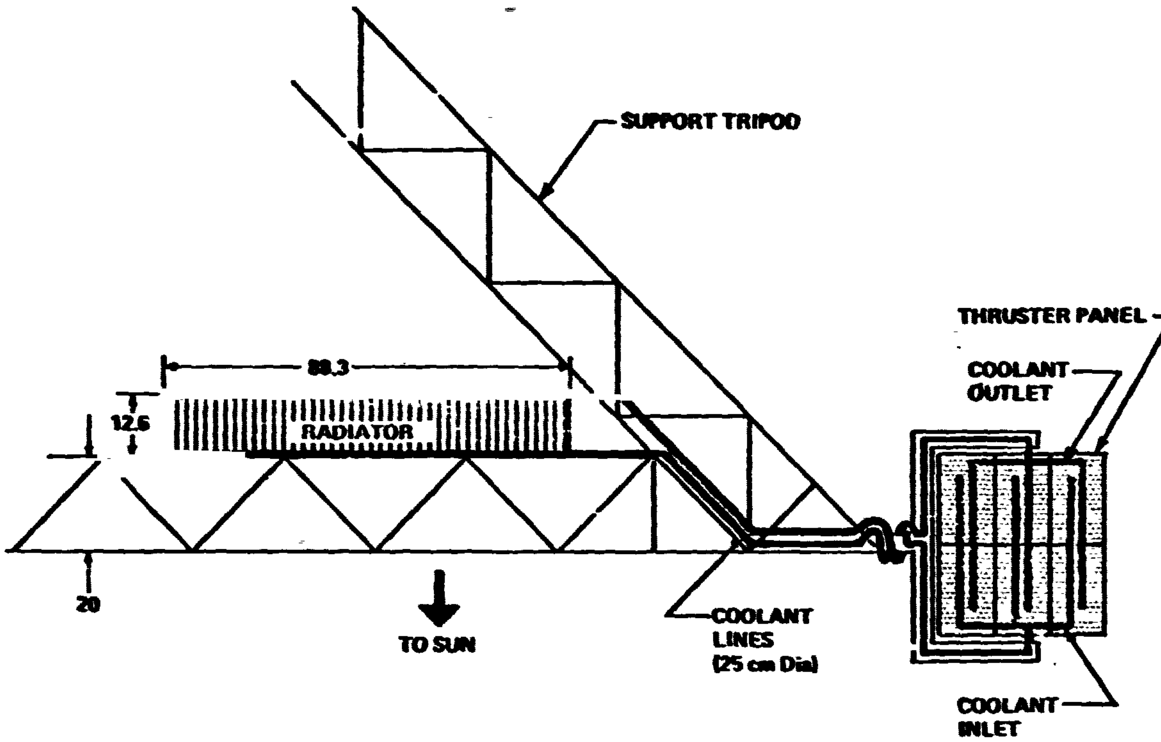


Figure 4.3-9. PPU Thermal Control

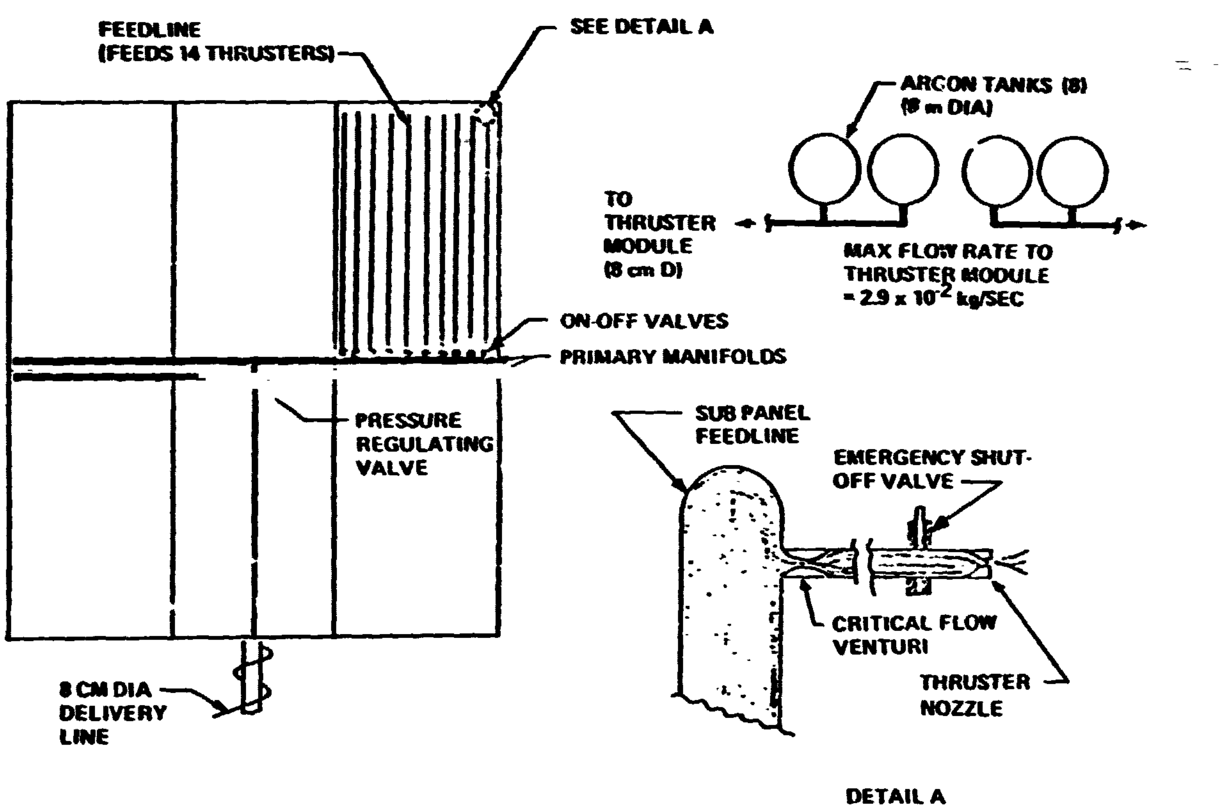


Figure 4.3-10. Typical Propellant Storage/Delivery System

than 4 or 12 subpanel module used for the preferred system, however, the overall concept is the same in all cases.). The propellant delivery line at the thrust module splits into two primary manifolds which in turn divides into 10 feed lines for each subpanel. Each feed line supplies propellant for 14 thrusters and is sized to provide a negligible pressure drop and a uniform pressure to each thruster. The individual thruster flow rate is fixed by using a critical flow venturi at the inlet of each feed line. With this arrangement, the flow rate is only a function of the manifold (plenum) pressure. The only line connections made in orbit will be attachment of the subpanel feed line with the main manifold and the attachment of propellant tank lines to the thruster module delivery lines. The thrust level produced from a module can be varied by turning on and off rows of 14 thrusters. No individual thruster control is necessary except for a one-way shutoff valve in the case of thruster failure. There is no individual thruster throttling capability. As the number of thrusters operating varies, the flow rate into the module primary manifold is regulated to maintain the correct pressure and corresponding thruster flow rate. The boil-off rate in the argon tanks is set accordingly by changing the rate of heat input.

Auxiliary Propulsion

An auxiliary propulsion system is required for the following functions during the transfer: 1) attitude control during the orbit transfer occultation periods, 2) to provide object collision avoidance thrust during occultations, 3) during the periods of high gravity gradient torque, 4) during the terminal docking maneuvers at GEO and 5) most likely during the initial maneuvers to move away from the construction base. A LO_2 LH_2 system is used providing an Isp of 400 sec. The thrust level is established by the requirement to supplement the electric thrusters during periods of high gravity gradient. For the case of a module transferring without an antenna a thrust level of 12000 N is required while a module with an antenna requires 500 N of supplemental thrust.

Avionics

Avionics functions include onboard autonomous guidance and navigation, data management and S-band telemetry and command communications. Navigation employs Earth horizon, star and Sun sensors with an advanced high performance inertial measurement system. Cross-strapped LSI computers provide required computational capability including data management, control and configuration control. The command and telemetry system employs remote-addressable date bussing and its own multiplexing. An additional factor that may need consideration is the need for radiation shielding due to the passage through the Van Allen belts. Although the shielding density may be quite high, the volume to be shielded is small and consequently the mass penalty should not be too severe.

4.3.1.1.3 Performance Optimization

Performance optimization for self power electric propulsion systems is focused on the parameters of specific impulse and trip time. Additional considerations used in selecting optimum trip time and Isp are radiation effects and interest rate (cost of money).

The results of optimizing I_{sp} as a function of radiation degradation with fixed trip time and interest rate is shown in Figure 4.3-11. Current cell and annealing characteristics indicates a 5% degradation. Although the curves for the low degradation levels are quite flat, the minimum cost per Kw is in the 7000 to 7500 sec region.

Optimization of the trip time for the optimum I_{sp} and several interst rates is shown in Figure 4.3-12. As indicated from this data, the assumed interest rate has a significant impact on the transportation cost as well as the optimum trip time. The interest rate judged to be most compatible with current utility philosophy is 7½%. The optimum trip time for this case then is approximately 220 days, however for analysis purposes the selected trip time is 180 days.

4.3.1.1.4 Mass Properties

Mass characteristics associated with the optimum self-power orbit transfer system is presented in Table 4.3-2. The values are related to the transfer of each satellite module.

Table 4.3-2 Reference Photovoltaic Self-Power Mass Summary

ITEM	MASS (10^6 Kg) WITHOUT ANTENNA	WITH ANTENNA
Orbit Transfer System	(0.76)	(2.08)
Power Processing Units	0.39	1.07
Electric Thrusters	0.32	0.32
Chemical Thrusters	NIL	NIL
Propellant Storage/Distribution	0.08	0.23
Thermal	0.14	0.39
Structural Installation	0.03	0.07
Usable Propellants	(3.06)	(8.26)
Argon	2.06	5.46
LO_2/LH_2	1.00	2.80
Satellite Modifications	(0.66)	(1.79)
Oversizing	0.24	0.65
Power Distribution	0.30	0.81
Structure (for Modularity)	0.12	0.33

Inertia characteristics for a module transferring without an antenna are presented in Figure 4.3-13.

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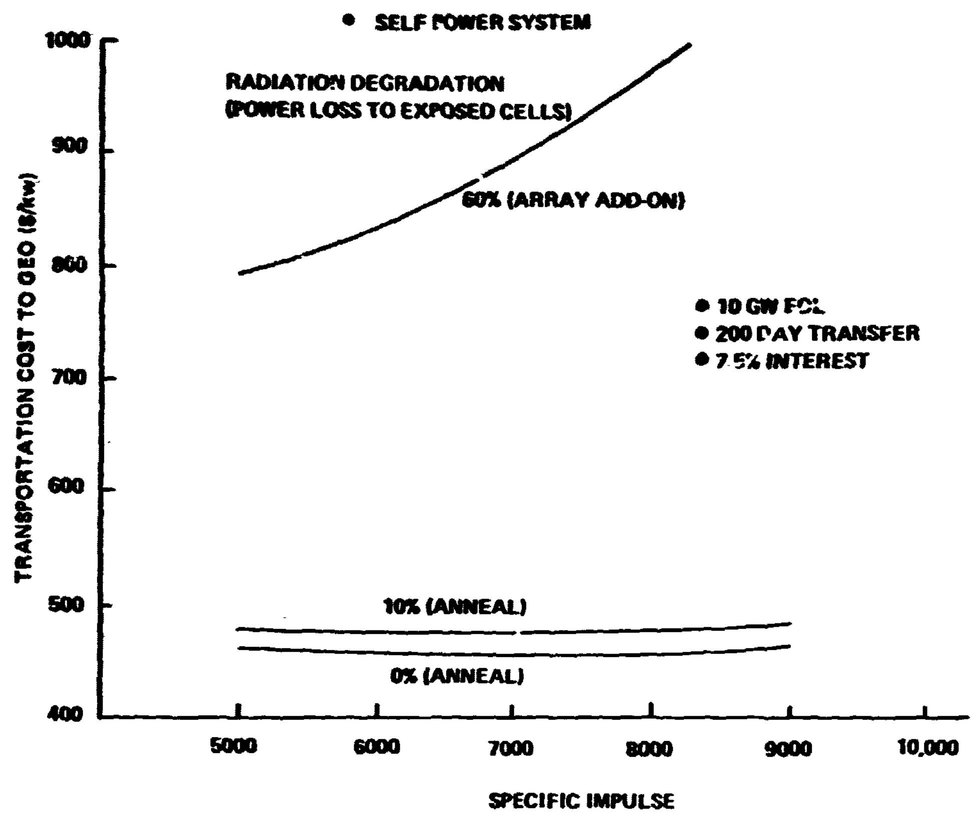

Figure 4.3-11. Transportation Cost Sensitivity Radiation Degradation

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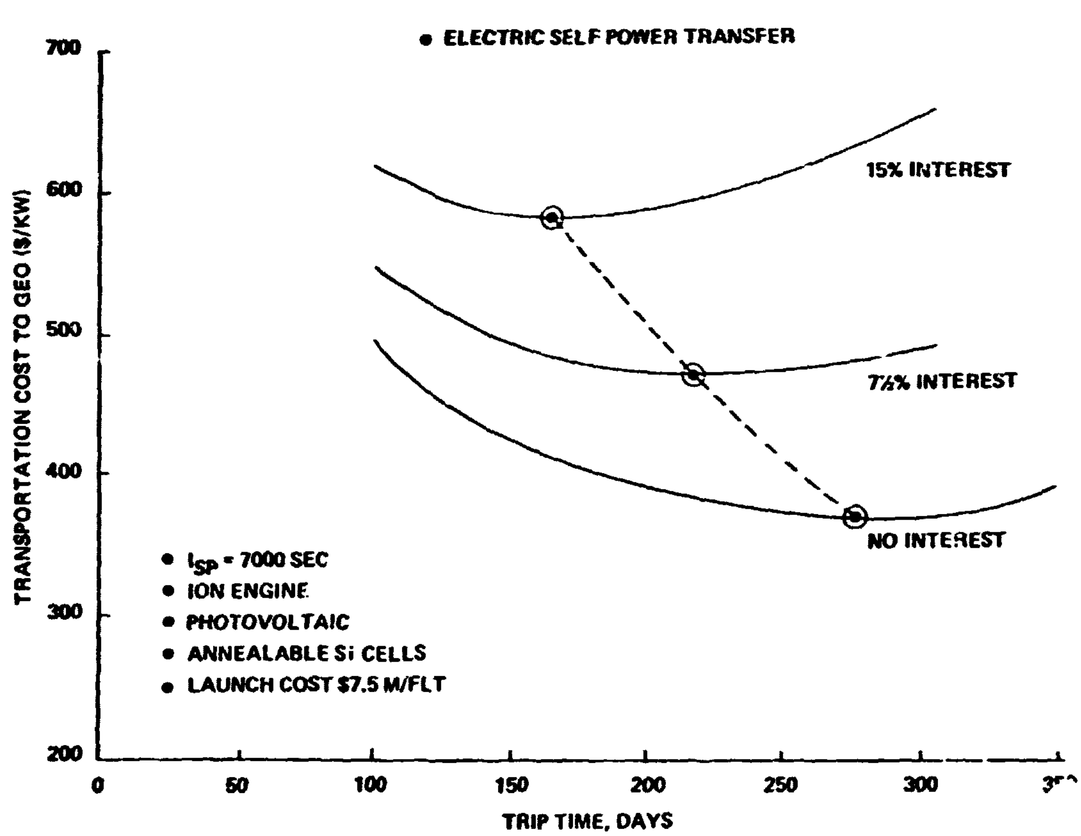


Figure 4.3-12. Transportation Cost Sensitivity Interest Rate

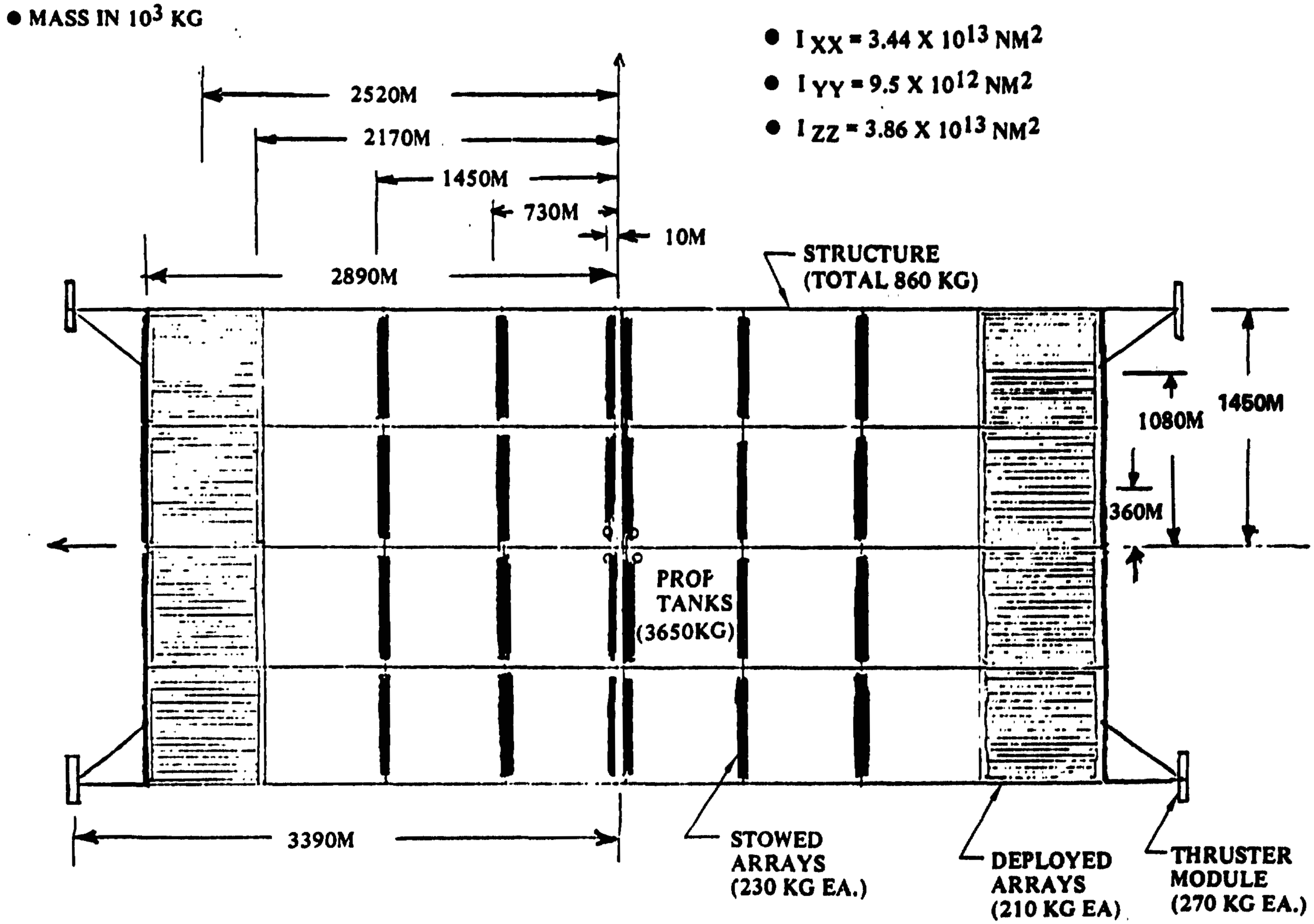


Figure 4.3-13. Configuration Inertia Characteristics

4.3.1.1 5 Mission Profile and Flight Operations

Mission Profile and Flight Sequence

Mission profile characteristics in terms of the relationships between orbit plane, altitude and elapsed time for a typical any time departure transfer are shown in Figure 4.3-14. A significant point that can be seen from this data is that a great deal of time is spent traveling through the Van Allen belts which have their main contributions below 10,000 km.

Since the self-power concept does involve low acceleration levels, the altitude increase per revolution is quite small particularly at the lower altitudes where a stronger gravity field is present. Each of these revolutions includes an occultation or shadow period when the satellite will be passing on the backside of the Earth and out of sunlight. The number of occultations that can be expected as a function of transfer time is presented in Figure 4.3-15. The band indicates the range in number of occultations depending on whether the transfer is initiated at the best or worst time of the year relative to the orbit and sun position. Therefore, for typical transfer times of 180 days, as many as 1000 occultations can be expected.

Also shown in Figure 4.3-15 is the fraction of time a vehicle in orbit is occulted as a function of the time from departure; the decrease with time is the result of the orbit getting larger and the shadow zone staying constant.

The flight sequence for the transfer of eight satellite modules is shown in Figure 4.3-16. Allowing 40 days for the construction and 180 days for transfer of each module results in a maximum of five satellite modules being in transit at one time after the fifth module has departed.

Flight Control

The flight control task associated with the self-power transfer of a satellite module from LEO to GEO involves directing the thrust vector in a manner to change the plane of the orbit and raise the altitude while maintaining the attitude of the satellite so that electric power can be generated for the thrusters. The flight attitude selected for the reference case consists of directing the solar arrays toward the sun during the entire transfer. The principal disturbance to the attitude is that of gravity gradient torque. As indicated in Figure 4.3-17, gravity gradient torques result because the satellite module is always flown PEP during the transfer as shown in the left hand portion of the figure and also because the flight begins from an incline orbit as illustrated on the right hand sketch. An additional consideration in the flight control analysis is the changing line of nodes which results from having the capability to begin the departure at any time during the year.

Determination of inertia characteristics, gravity gradient torque and thrust requirements has been done using a 6 DOF simulation. A typical result of this simulation is presented in Figure 4.3-18 for the case of the first revolution of a satellite module transferring without an antenna.

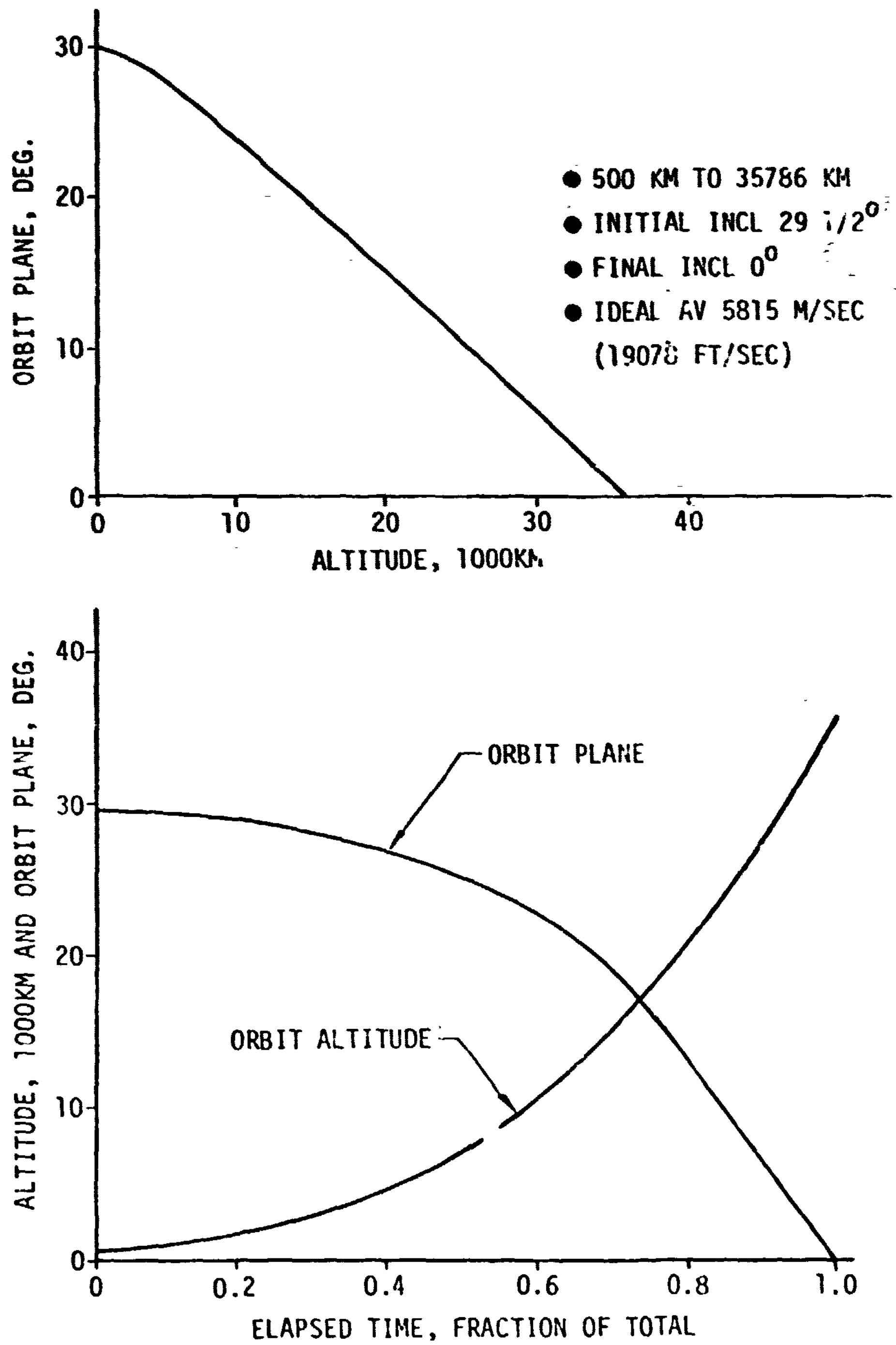


Figure 4.3-14. Low-Thrust Orbit Transfer Characteristics

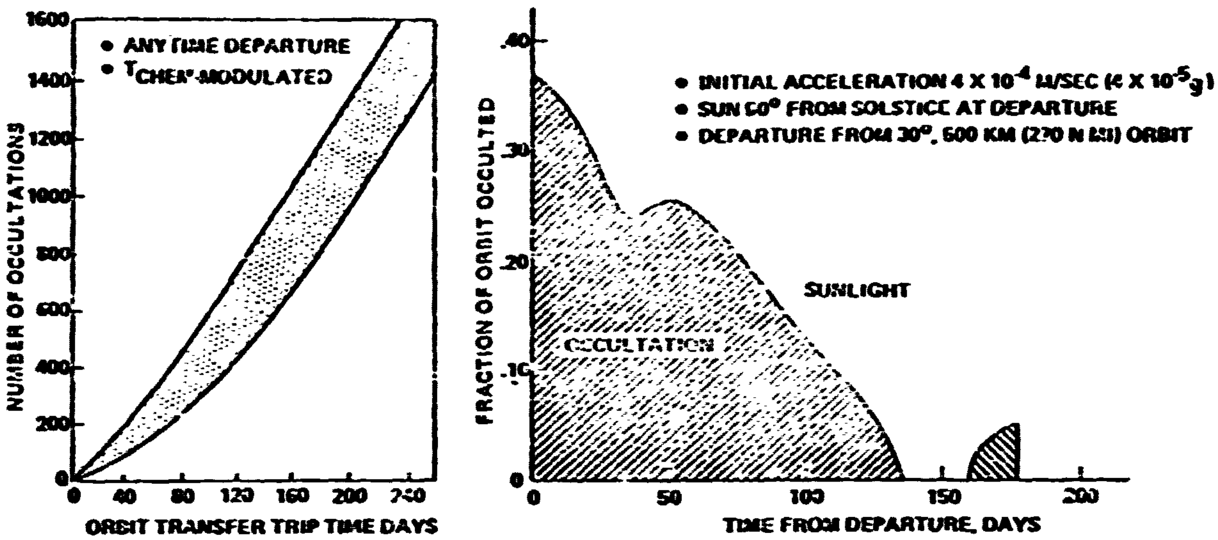


Figure 4.3-15. Orbit Transfer Occultations

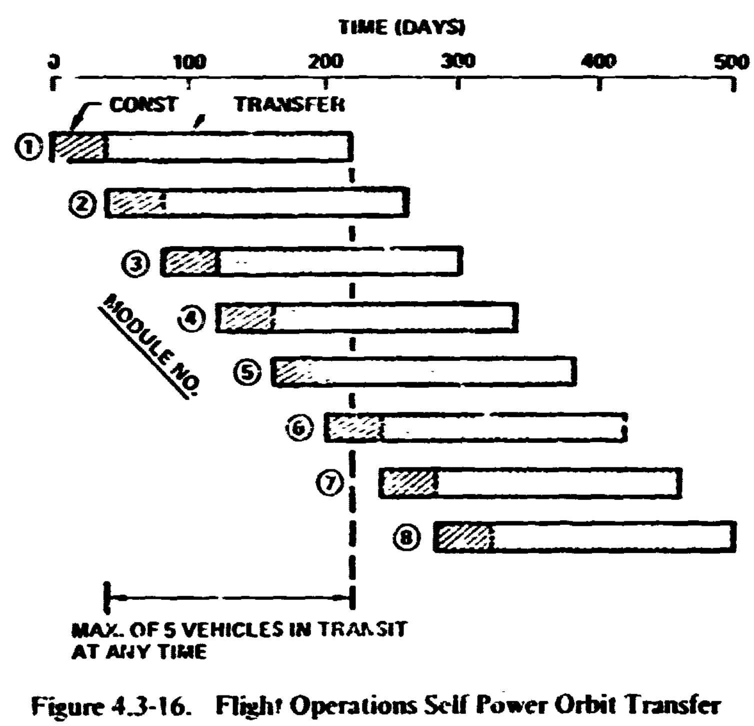


Figure 4.3-16. Flight Operations Self Power Orbit Transfer

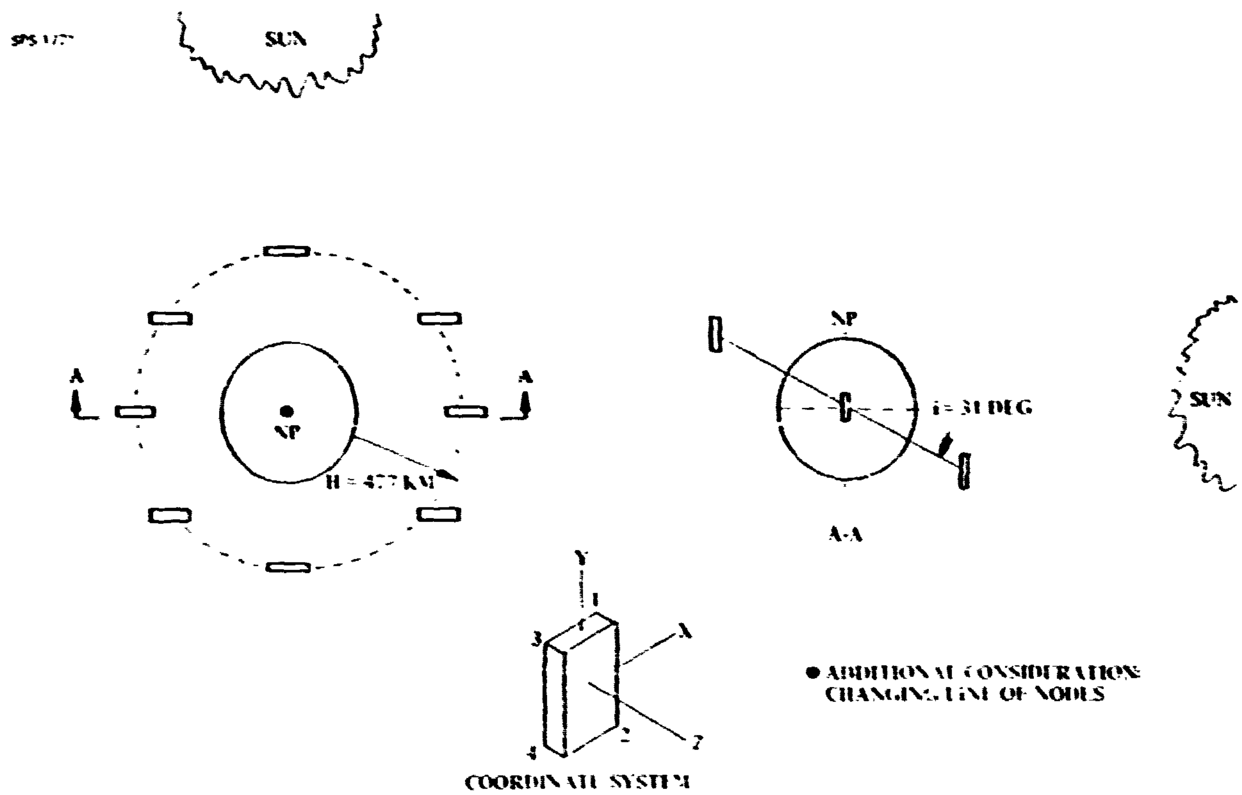


Figure 4.3-17. Self Power Flight Control Problem

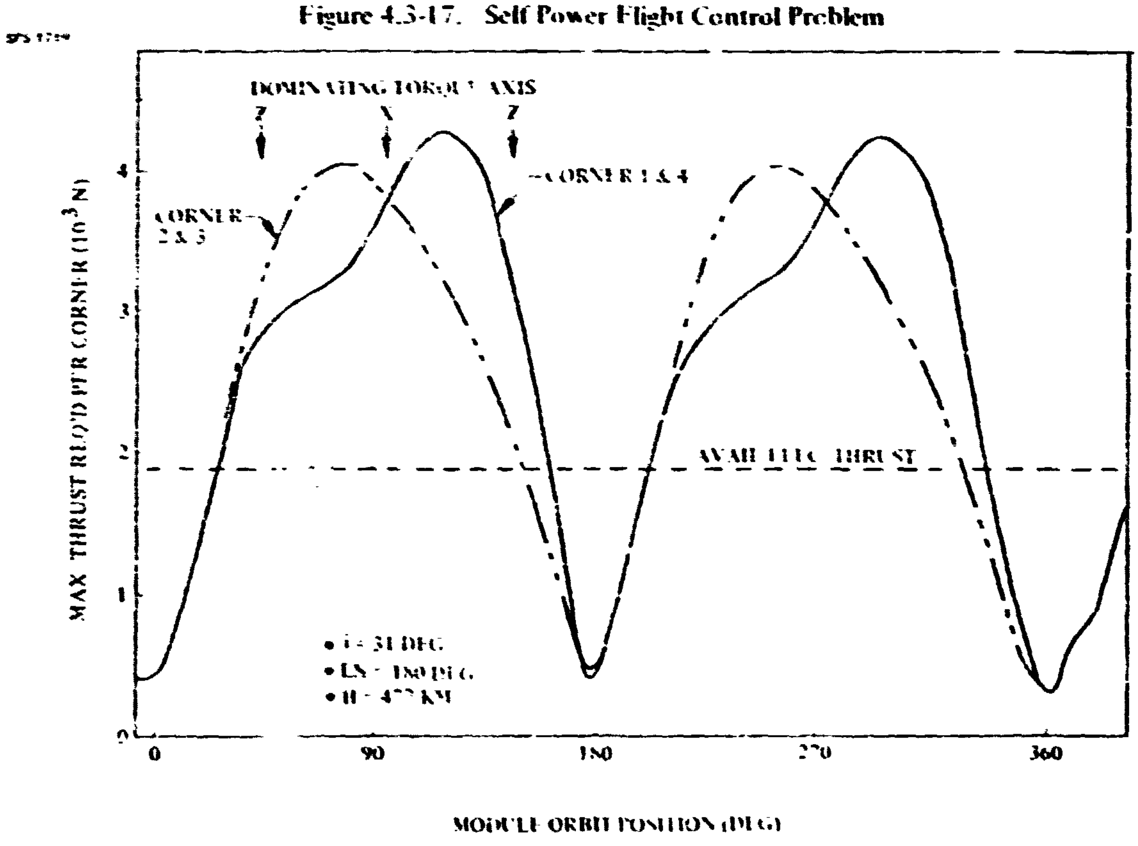


Figure 4.3-18. Control Authority Requirements

As indicated by this data, the maximum thrust required at each corner of the module is over 4000 N. The available thrust which is established by the optimum trip time is slightly under 2000 N per corner thereby necessitating supplemental thrust capability. The sensitivity of altitude to the control thrust requirement is presented in Figure 4.3-19. This data indicates that the available electric thrust is sufficient to control the gravity gradient torque once 2500 Km is reached (NOTE: GEO is 35786 Km). The line of node sensitivity to control thrust requirement is also presented in Figure 4.3-19. Again, this parameter deals with the time of year the departure is initiated.

In all of these flight control cases, a supplemental amount of thrust must be added to that provided by the electric thrusters. The approach used to provide the supplemental thrust and total impulse is that of a LO_2/LH_2 system. The magnitude of the thrust and propellant quantity has previously been identified in Figure 4.3-3.

4.3.1.1.6 Cost

The total DDTE cost for the self-power system is estimated at $1.39 billion. The flight system portion of this cost is estimated at $235 million. A breakdown of the DDTE cost is presented in Table 4.3-3

Table 4.2-3 Self-Power DDTE Cost

		$\$10^6$
Program Management		40
System Test		885
Software		95
SE & I		20
GSE		25
Tooling		90
Flight System		235
Power Processing System	60	
Electric Thrusters	10	
Chemical Thrusters	10	
Prop Storage & Distribution	60	
Thermal Control	10	
Structure and Mechanisms	60	
Avionics	20	
Power Distribution	5	
		$1390

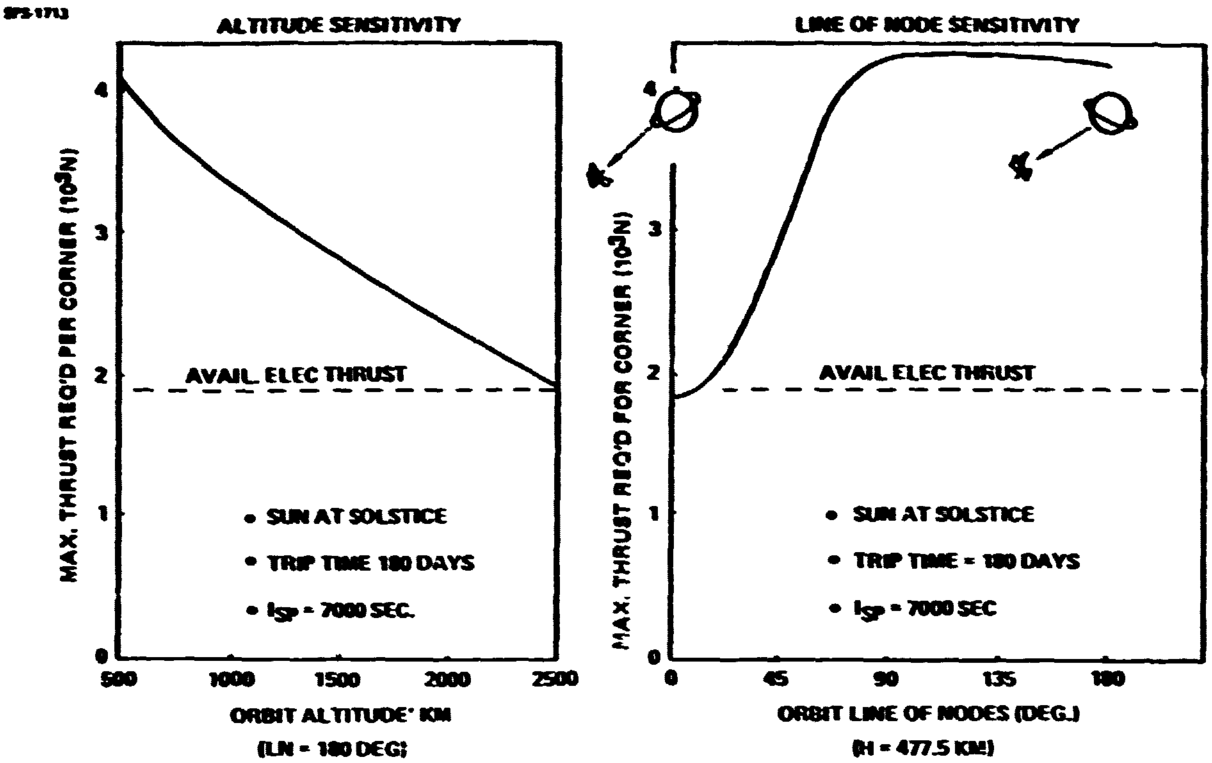


Figure 4.3-19. Control Authority Requirements

The recurring cost for the self-power system to transfer one complete satellite is estimated at $870 million. This estimate is based on a mature industry costing approach and delivery of four satellites per year. A breakdown of the recurring cost for the complete satellite as well as for modules being transported with and without an antenna is presented in Table 4.3-4.

Table 4.3-4 Self-Power Recurring Cost
(cost in millions)

SYSTEM	TOTAL SATELLITE	MODULE W/O ANTENNA	MODULE WITH ANTENNA
Flight System	(790)	(69)	(188)
Power Processing System	335	31	84
Electric Thrusters	65	6	16
Chemical Thrusters	Nil	--	--
Propellant Storage & Distribution	130	11	31
Thermal Control	80	7	18
Structure & Mechanisms	105	9	25
Avionics	20	2	5
Power Distribution	35	3	9
Program Management	(40)	(3)	(9)
Sustaining Engineering	(40)	(3)	(9)
TOTAL	870	75	206

4.3.1.1.7 Supporting System Level Trades

Supporting system level trades conducted during Part II involved the comparison of several locations for constructing the antennas for the LEO construction option and the cost effectiveness of recovery and reuse of the electric propulsion system when using self-power for LEO to GEO transfer. Each of these trades will be discussed in additional detail in the subsequent paragraphs.

Antenna Construction/Transportation Trade

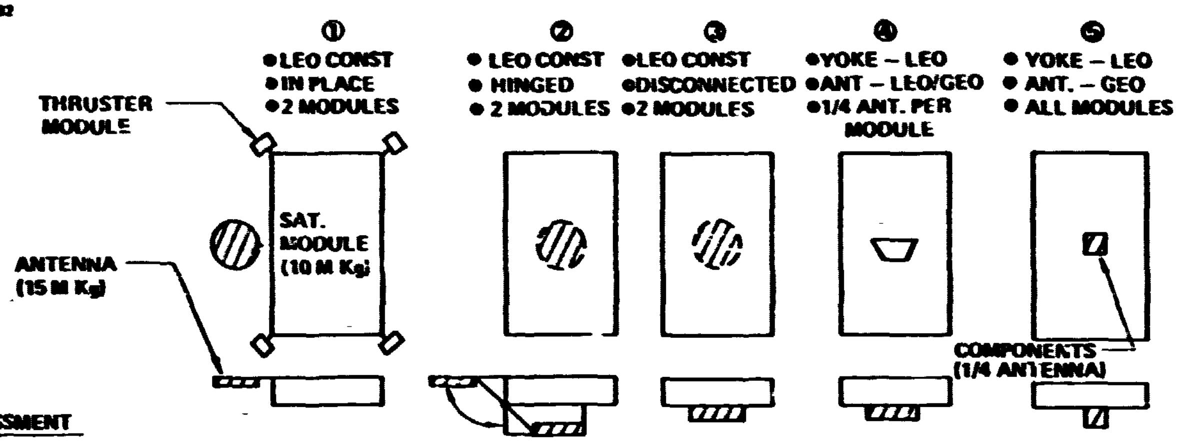
The early part of Part II involved the preliminary investigation of five different antenna construction/transportation methods when considering LEO construction of the satellite. These options are illustrated in Figure 4.3-20. The first three concepts involve construction of the entire antenna (including yoke) at LEO but vary in the location or method of attaching the antenna to the satellite module. Option 4 also has the antenna made in LEO but is transferred in sections, with each of the eight modules taking up one-fourth of an antenna. Option 5 has the yoke constructed at LEO (size requires a facility height similar to that required for the satellite construction) but has the antenna delivered in the form of components, which are assembled in GEO. Assessment of these options involved a number of criteria and resulted in selecting Options 2 and 5 as the most promising and requiring further depth in the analysis in order to make a selection.

Option 1 was eliminated from further consideration since it had the worst characteristics in terms of gravity gradient torque (GGT), restricted the thrust vector due to exhaust impingement on the antenna and had a large structural impact on a satellite due to eccentric loading during transfer and required two different propulsion system designs since the satellite modules without antennas would involve considerably less mass. Option 3 involved a structural impact on the satellite and required movement of the antenna to its operating position. The most significant disadvantage of Option 4 is that it required facilities at both LEO and GEO. The GEO facility was required to hook up the primary structure of the antenna sections plus install secondary structure and subarrays in between the already present secondary structure and subarrays that had been previously installed at LEO.

Further analysis of Options 2 and 5 involved a closer examination of the methods employed to construct and position the antenna as well as factors resulting in a difference in cost.

Construction Operations—Construction and installation of the antennas in LEO for Option 2 is illustrated in Figure 4.3-21. This concept has the antenna constructed in a facility that is attached to the module construction facility. Following completion of either the third and seventh modules, the antenna yoke is constructed in the module facility and parked at the side. Midway through the construction of the fourth and eighth modules, the antenna has been completed, attached to the yoke and this combination attached to the module in its operating position. After the module is completed and prior to release, the antenna is rotated down under the module for the transfer to GEO (this location gets the antenna out of the exhaust of the thrusters and also results in better inertia characteristics for attitude control). Once GEO is reached, the antenna is rotated back to its operating position through use of a single hinge line.

SPS-902



ASSESSMENT	①	②	③	④	⑤
• GGT CONTROL	WORST	BEST	BEST	GOOD	GOOD
• THRUST VECTOR	PARTIALLY RESTRICTED	–	–	–	–
• SAT. STRUCT IMPACT	WORST	MODERATE	MODERATE	LEAST	LEAST
• ANT. STRUCT DESIGN IMPACT	NONE	SUPP STRUCT	NONE	MODERATE	NONE
• PROPUL. SYS. DESIGNS	TWO	TWO	TWO	ONE	ONE
• GEO OPERATIONS	NONE	HINGE & HOOK-UP	DOCK & HOOK-UP	FINAL ASSY & FILL-IN	FULL ASSY
• LOGISTICS (CREW/SUPPLIES)	BEST	GOOD	GOOD	FAIR	WORST

Figure 4.3-20. Antenna Construction/Transportation Options—Self Power OTS

SPS-1492

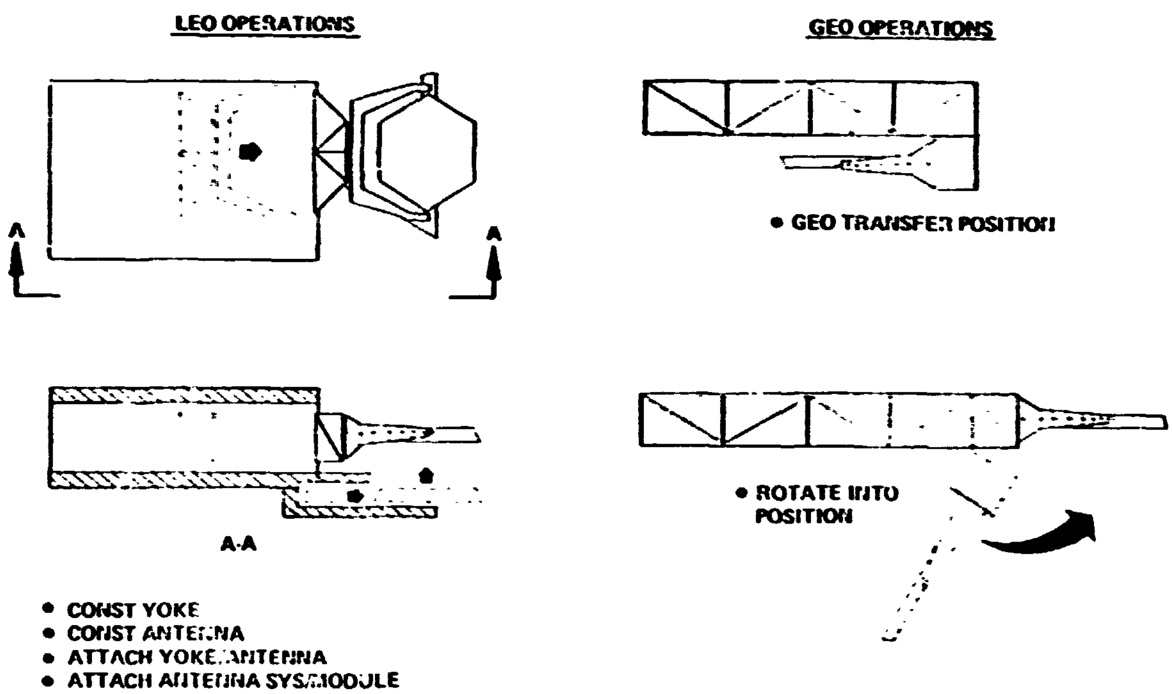


Figure 4.3-21. Antenna Construction and Installation LEO Construction

The antenna construction/installation concept for the option of constructing the antenna at GEO is shown in Figure 4.3-22. In this option, each of the eight satellite modules brings to GEO one-fourth of the components required for an antenna. These components must then be transferred to the antenna construction facility as indicated in picture 2. The actual installation of the antenna into the yoke is the same as in the previously described option. Construction of the second antenna is done in a manner similar to the first and when complete, the antenna facility is flown to the opposite end of the satellite so the antenna can be installed.

Compared from this standpoint, LEO construction of the antenna is judged to be the most desirable since the antenna facility does not have to be moved nor are antenna components required to be transferred the length of the satellite.

Cost—The principal cost differences between the two options are compared in Table 4.3-5. This data indicates a cost savings of $120 million per satellite when using LEO construction for the antenna. Satellite and transportation costs are greater for the LEO constructed option since the structure of a module must be sized to support a 15 million kg antenna. Consequently, the cost reflects both the cost of the extra structure plus its transportation. The antenna facility and its transportation is cheaper for the LEO construction approach primarily because only one crew module has a radiation shelter as opposed to three. In addition, the antenna facility does not have to be transported to GEO although this cost contribution is amortized over 10 years. The most significant difference, however, is the crew rotation resupply associated with having 200 fewer people at GEO.

In summary, LEO construction of the antenna is recommended because of the $120 million per satellite saving and less complex operations associated with installation of the antenna.

Electric OTS Reusability

Motivation for considering the reusability of the electric propulsion system has been the $620 million unit cost associated with the system at the midterm of Part 2 (note the final Part 2 value was $870 million). A breakdown of the system components and reusability rationale is shown in Figure 4.3-23. Those components considered as prime candidates for reusability include the thruster modules and propellant tanks due to their high cost and or high cost per kilogram, as well as being convenient in terms of disassembly reassembly and return back to low Earth orbit.

Life considerations of approximately 8,000 hours for the thruster units prevent more than two uses unless refurbishment is provided which amounts to approximately 50% of the unit cost.

SPS-1714

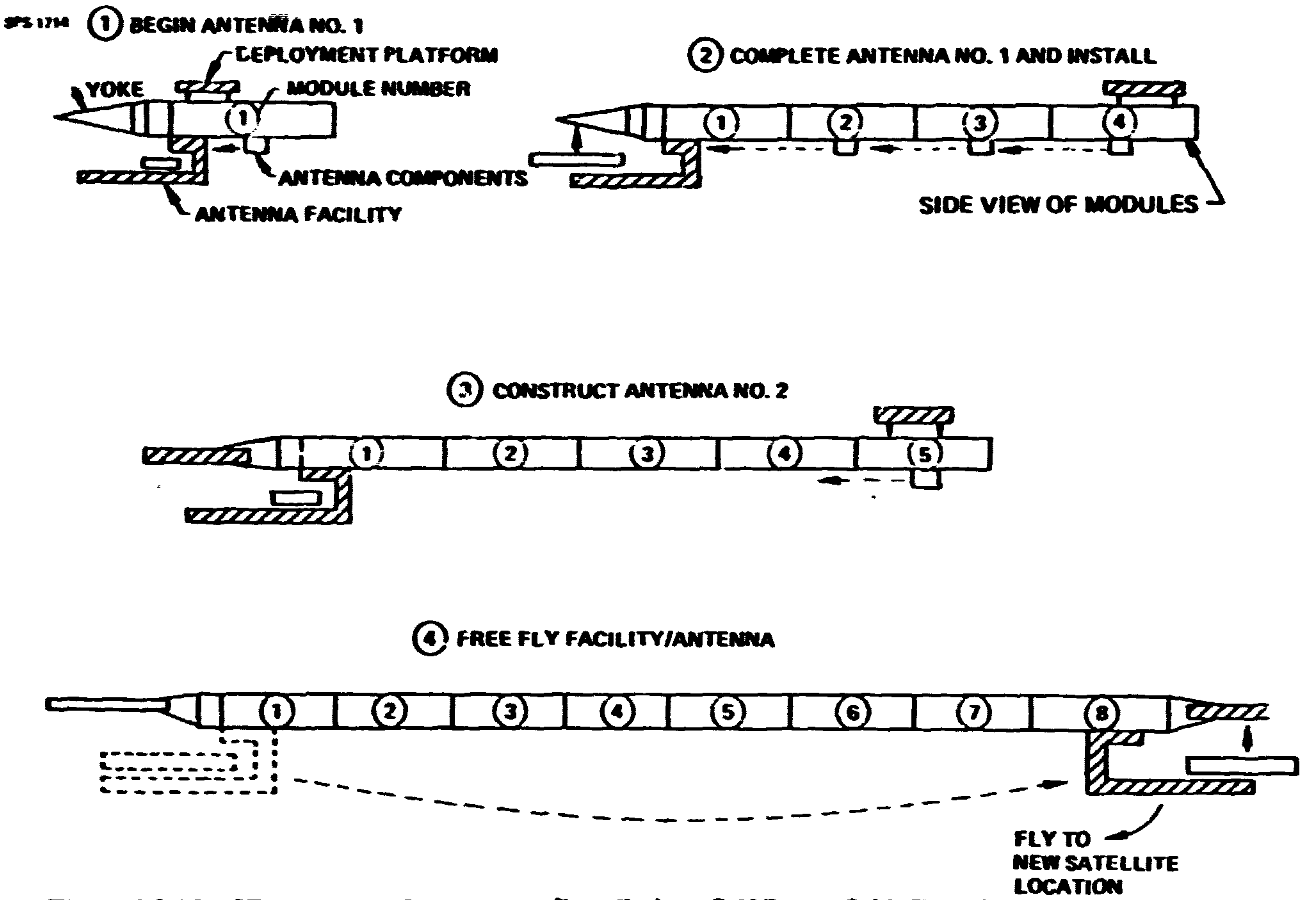
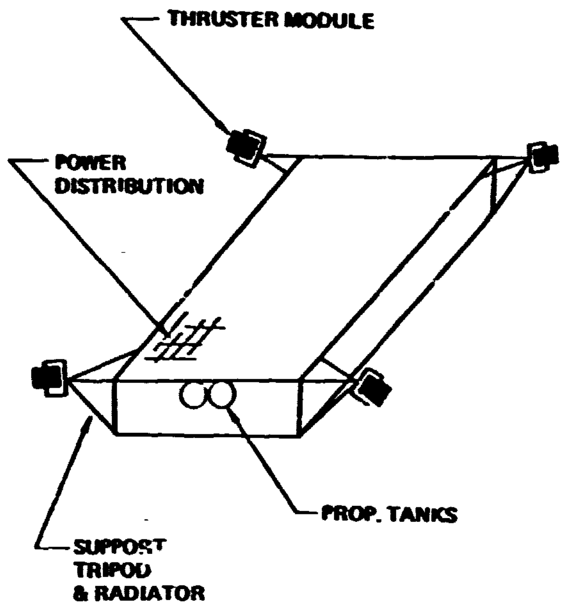


Figure 4.3-22. GEO Antenna Construction/Installation—Self Power Orbit Transfer

SPS-1716 **Table 4.3-5. Antenna Construction Location Comparison—LEO vs GEO**

ASSESSMENT CRITERIA	RECURRING COST CHANGE (LEO VS GEO)	EXPLANATION
• SATELLITE AND TRANSPORTATION	+ $40M	• Δ 0.4M KG
• ANTENNA FACILITY AND TRANSPORTATION	– $25M	• CHEAPER CREW MODULES • LESS DEMANDING TRANSP.
• LOGISTICS (CREW ROTATION/RESUPPLY)	– $135M	• 200 FEWER PEOPLE AT GEO
• ORBIT TRANSFER SYSTEM	0	• EACH SYSTEM SIZED TO ITS SPECIFIC PAYLOAD • SMALL DDTE PENALTY FOR TWO THRUSTER MODULE SIZES
SAVINGS	$120M	

- MOTIVATION: $620 MILLION/SATELLITE (77.5 MILLION/MODULE)



REUSE AND RATIONALE		
• THRUSTER MODULE THRUSTERS PPU YOKE/GIMBAL	YES	HIGH COST
• SUPPORT TRIPOD STRUCT RADIATOR	NO	AWKWARD
• PROP TANKS	YES	HIGH $/KG
• POWER DISTRIB	NO	INTEGRAL WITH SATELLITE
• CHEM SYS	NO	LOW COST

Figure 4.3-23. Electric OTS Reuseability

Recovery Concept for Electric OTS—The concept for recovery of the electric OTS is presented in Figure 4.3-24. The system analyzed for the return of the electric components to LEO is a single-stage LO_2/LH_2 OTV. Return of 210,000 kilograms of payload requires a propellant loading of approximately 400,000 kilograms. (Note: The final Part 2 chemical OTV for crew rotation/resupply is a common two stage system with total propellant loading of 460,000 kg.) Delivery of the LO_2/LH_2 stage to GEO involves mounting the stages below the satellite module. The resulting impact on the electric propulsion system of transporting an additional 1.6 million kilograms of LO_2/LH_2 stages is relatively minor due to its high performance characteristics.

Reusability of the electric components used on the first module is not possible before transfer of the seventh module due to delivery times of 180 to 200 days.

Cost Assessment—Cost for recovery of the electric components compared to their value is nearly the same as indicated in Table 4.3-6. Consequently, recovery is not suggested for the Part 2 mid-term cost characteristics, although it is recommended that consideration be given for using these thrusters for operational attitude control and station-keeping associated with the satellite, which would also effectively reduce the cost of the self-power system and also to investigate recovery using an independent solar electric tug. Note: The final Part 2 OTS cost (hardware) was estimated at $790 million rather than $620 million for the Part 2 midterm for which this trade was conducted. Using the final Part 2 cost however results in a savings of $140 million per satellite which means further consideration of recovery should occur in Part 3.

SPS 1275

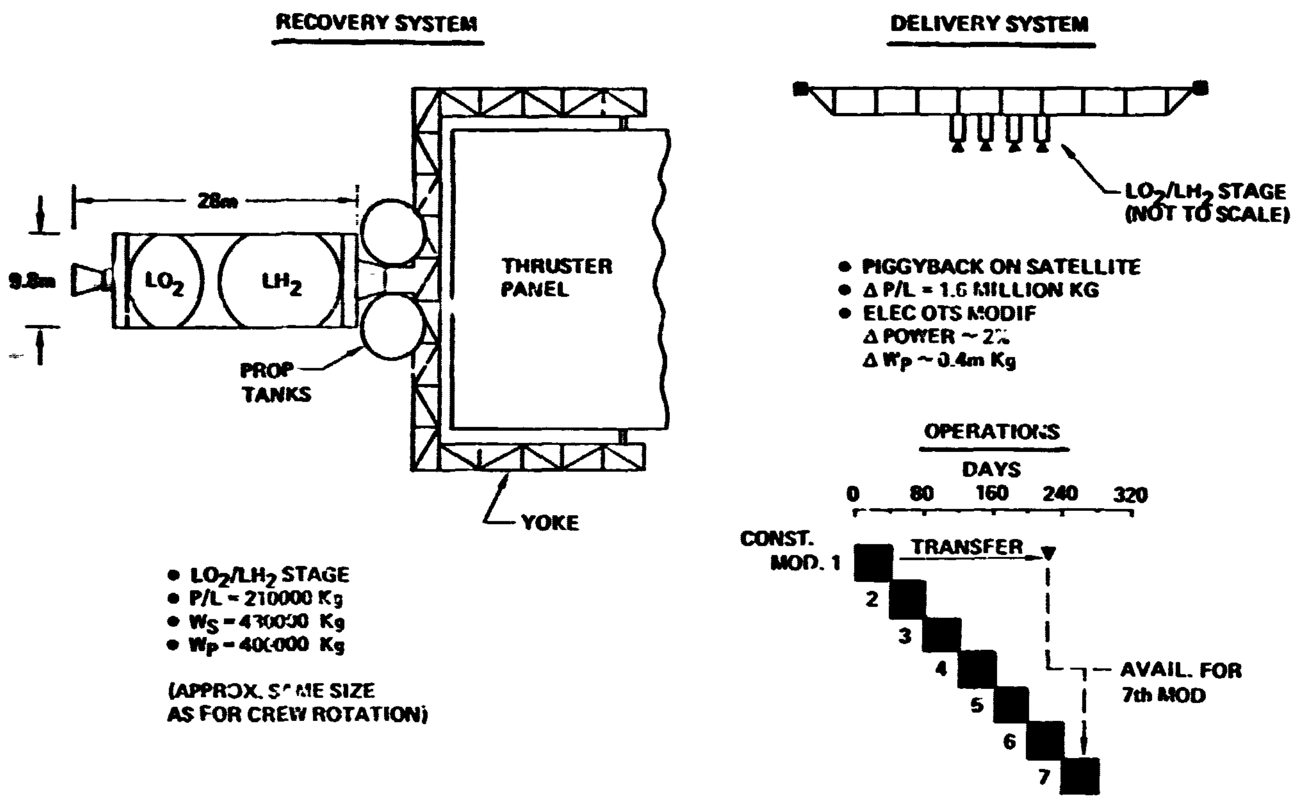


Figure 4.3-24. Electric OTS Recovery Concept

SPS-1226

Table 4.3-6. Cost Assessment Electric OTS Reuseability

• COST PER SATELLITE MODULE

Δ COST FOR RECOVERY ($M)	
• ELEC OTS (Δ UNIT)	9
• ELEC OTS TO LEO	11
• CHEM (Δ UNIT)	10
• CHEM TO LEO	33
TOTAL	$63M

VALUE OF RECOVERY	
• THRUSTER MODULES (UNIT)	43
• LAUNCH SAVINGS (ASSOC. WITH THRUSTERS)	16
• TANKS	5
TOTAL	$64M

CONCLUSION: NOT ECONOMICALLY ATTRACTIVE

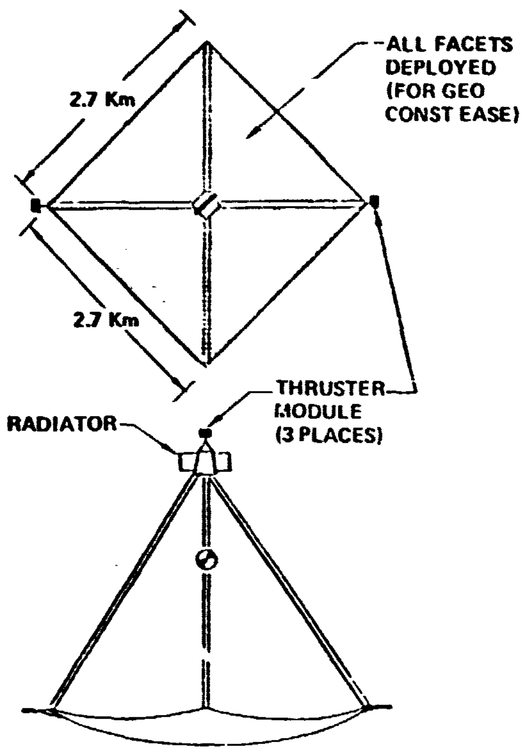
RECOMMENDATION: CONSIDER
1) ELECTRIC TUG RETURN
2) ELEC OTS FOR SATELLITE OPERATIONAL ACS & STATION KEEPING

4.3.1.2 Thermal Engine Satellite Self-Power System

Self-power of the thermal engine satellite modules are for the most part similar to the photovoltaic modules from a performance standpoint and basic hardware although there are some distinguishing differences in terms of satellite design impact as identified in Figure 4.3-25. One example of this is that no oversizing of the thermal engine modules is required since the reflector facets and engines are not sensitive to radiation as are the solar arrays. A second difference is that the voltage generated by the satellite can be the same as the operating satellite voltage (since no plasma losses occur as in the case of solar arrays) and thus a minimum power distribution penalty occurs. From a propulsion standpoint, three thruster modules are used rather than four and although all facets are deployed in LEO, only a portion of these are required for the transfer. Gravity gradient torque associated with this configuration are considerably lower due to the inertia characteristics of the module and consequencly the chemical thrust required and the amount of LO_2/LH_2 propellant are considerably less than in the case of the photovoltaic satellite module.

Flight operations such as trip time are the same as for the photovoltaic satellite. A difference in the operations, however, is that the thermal engine satellite consists of 16 modules as compared with eight modules for the photovoltaic satellite. As a result, 10 modules are in transit at any one time after the tenth module of the first satellite has begun its trip as shown in Figure 4.3-26.

Since the mass of the thermal engine satellite is essentially the same as the photovoltaic voltaic, the total electric OTS system DDTE and satellite cost is the same: approximately $1.4 billion for DDT&E and $870 million recurring per satellite.



GENERAL CHARACTERISTICS

- NO OVERSIZING
- TRIP TIME = 180 DAYS
- ISP = 7000 SEC

MODULE CHARACTERISTICS	NO ANTENNA	WITH ANTENNA
• MODULES	14	2
• MODULE MASS	4.1	19.1
• POWER REQ'D (10^6 Kw)	0.14	0.65
• FACETS REQ'D % [1]	27	74
• OTS DRY (10^6 Kg)	0.5	2.35
• ARGON (10^6 Kg)	1.0	4.5
• LO_2/LH_2 (10^6 Kg)	0.2	0.94
• ELEC THRUST (10^3 N)	2.1	9.8
• CHEM THRUST (10^3 N)	2.1	9.8

[1] INCLUDES 14% TO COVER CAVITY APERATURE OSSES

Figure 4.3-25. Self Power Configuration–Thermal Engine Satellite

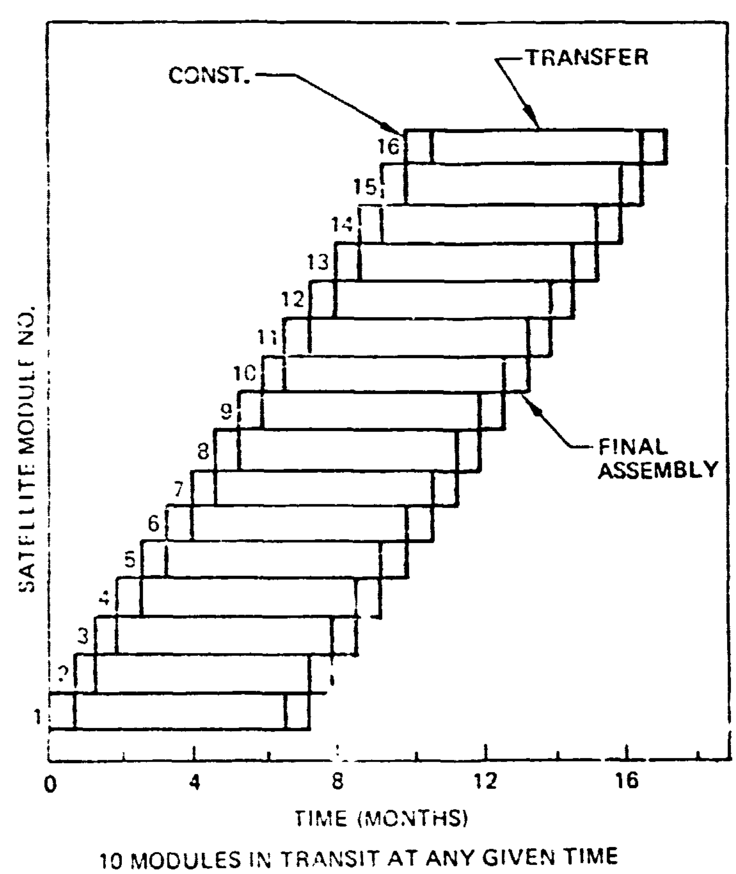


Figure 4.3-26. Flight Sequence for Thermal Engine Self Power Transfer

4.3.1.3 Chemical Orbit Transfer Vehicle

The chemical orbit transfer vehicle (OTV) is used in the satellite GEO construction concept. In this concept, satellite components are delivered to a LEO staging depot and then transferred to chemical OTV's for delivery to GEO where the construction occurs.

Various types of chemical OTV's have been investigated in the FSTS study and Part I of the SPS system definition study. The results of these studies have indicated a LO_2/LH_2 common stage (two stage) system to be the most desirable. This system will be summarized in the following sections and is applicable to either the photovoltaic or thermal engine satellites.

4.3.1.3.1 Configuration

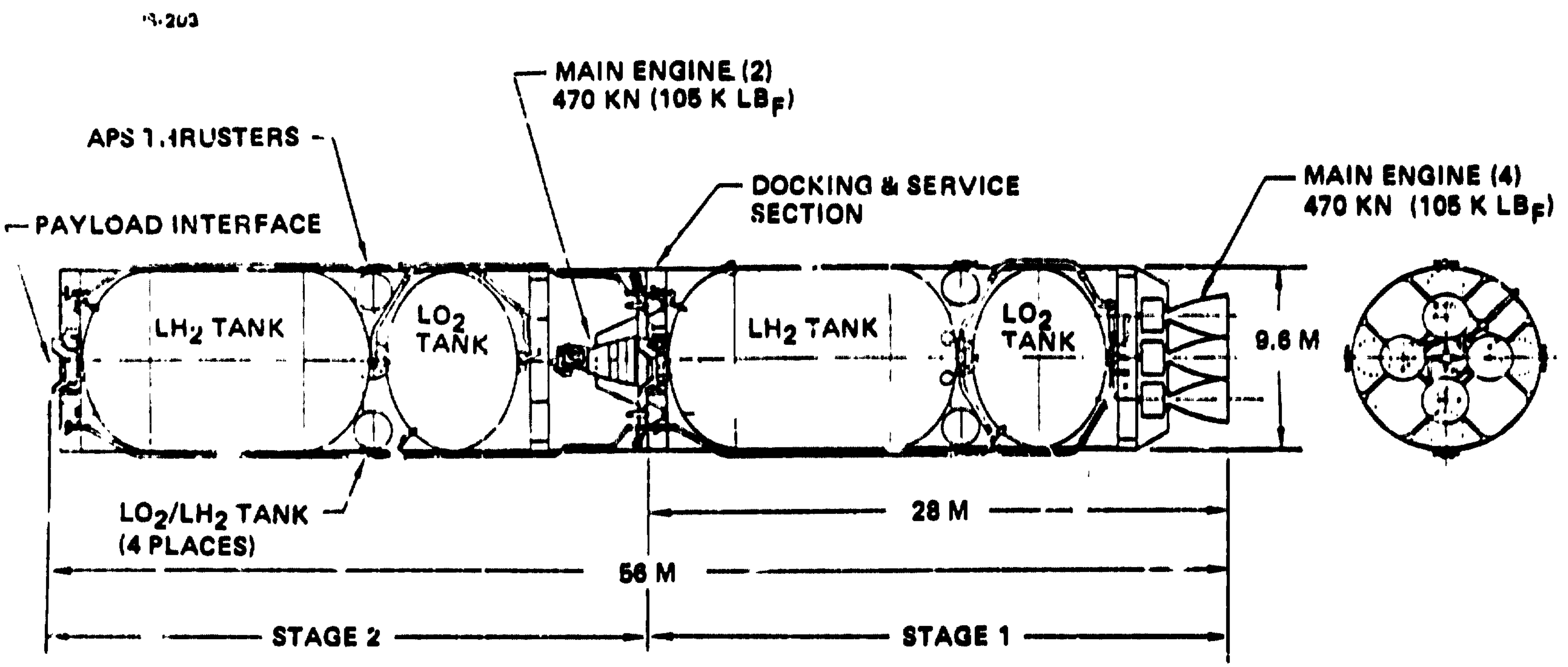
The space-based common stage OTV is a two-stage system with both stages having identical propellant capacity as shown in Figure 4.3-27. The first stage provides approximately 2/3 of the delta V requirement for boost out of low Earth orbit at which point it is jettisioned for return to the low Earth orbit staging depot.

The second stage completes the boost from low Earth orbit as well as the remainder of the other delta V requirements to place the payload at GEO and also provides the required delta V to return the stage to the LEO staging depot. Subsystems for each stage are identical in design approach. The primary difference is the use of four engines in the first stage due to thrust-to-weight requirements. Also, the second stage requires additional auxiliary propulsion due to its maneuvering requirements including docking of the payload to the construction base at GEO. The vehicle has been sized to deliver a payload of 400 000 kilograms. As a result, the stage startburn mass without payload is approximately 890 000 kilograms with the vehicle having an overall length of 56 meters.

4.3.1.3.2 Subsystems

Structure and Mechanisms

Main propellant containers are welded aluminum with integral stiffening as required to carry flight loads. Intertank, forward and aft skirts, and thrust structures employ graphite/epoxy composites. An Apollo/Soyuz type docking system is provided at the front end of each stage for docking with payloads, refueling tankers and orbital bases. The stage-to-stage docking system provides for docking the stages together with flight loads carried through full-diameter structures. Propellant transfer connections allow either stage to be fueled independently with the stages either separated or docked together. Structure of the two stages is identical to the extent practicable.



- PAYLOAD CAPABILITY = 400,000 KG
- OTV STARTBURN MASS = 890,000 KG
- STAGE CHARACTERISTICS (EACH
  - PROPELLANT = 415,000 KG
  - INERTS = 29,000 KG (INCLUDING NONIMPULSE PROPELLANT)
- 280 OTV FLIGHTS PER SATELLITE

Figure 4.3-27. Space Based Common Stage OTV GEO Construction

D180-22876-5

Main Propulsion

Main engines are based on shuttle engine technology, operating with a staged-combustion cycle at 20 Mn/m^2 (3000 psia) chamber pressure, a LO_2/LH_2 mixture ratio of 5.5 to 1.0 and a retractable nozzle with extension expansion area ratio of 400 providing a specific impulse of 470 seconds. Advanced low NPSH pumps are used to minimize feed pressures. A 6 degree square gimbal pattern is employed. The engines are capable of operating in a tank-head idle (THI) mode (pumps not turning; mixed-phase propellants) for cool-down and self-ullaging at a specific impulse of 350 seconds; 60 seconds (time) in self-ullaging mode is assumed needed prior to bootstrapping to full thrust. Throttling between tank-head idle and full thrust is not required. Main propellant pressurization is derived from engine tap off after an onboard helium prepressurization.

Auxiliary Propulsion

Auxiliary propulsion is used for attitude control and low delta V maneuvers during coast periods and for terminal docking maneuvers. An independent LO_2/LH_2 system is used and provides an Isp of 375 seconds averaged over pulsing and steady state operating modes. Thrusters are mounted in quad packages analogous to the Apollo Service Module installation. Each quad has its own propellant supply to facilitate change out. Auxiliary propulsion for the two stages uses common technology but capacities and thrust levels are tailored.

Electric Power

Primary electric power is provided by fuel cells based on shuttle technology, tailored to the OTV requirement. Reactants are stored in vacuum-jacketed pressure vessels. Product water is assumed retained onboard to minimize payload contamination potential. Ni-Cad batteries are employed for peaking and smoothing. 28 VDC power is rough-regulated and filtered with fine regulation provided by power using subsystems as needed. A potential inert mass saving (not assumed) would use low pressure reactants provided from main propellant tanks. Electric power systems for the two stages are identical except for reactant capacity and harnesses

Avionics

Avionics functions include onboard autonomous guidance and navigation, data management and S-band telemetry and command communications. Navigation employs Earth horizon, star and Sun sensors with an advanced high performance inertial measurement system. Cross-strapped LSI computers provide required computational capability including data management, control and configuration control. The command and telemetry system employs remote-addressable data bussing and its own multiplexing. Although the avionics systems in the two stages are identical, software for each stage is tailored to the stage functions.

Thermal Environment Control

Main propellant tanks are insulated by aluminized mylar multilayer insulations contained within a purge bag. The insulation system is helium purged on the ground and during Earth launch. The avionics systems semi-active louvered radiators and cold plates. Active fluid loops and radiators are required for the fuel cell systems. Superalloy metal base heat shields are employed to protect the base areas from recirculating engine plume gas.

4.3.1.3.3 Performance

Performance characteristics associated with the common stage LO_2 LH_2 OTV are shown in Figure 4.3-28. Propellant requirements are shown as a function of the payload return and delivery capability. Performance ground rules used in these parametrics are as follows (values are main propellant quantities):

- THI mode — Stg 1 100 kg per start; Stg 2 50 kg per start
- Stop loss — Stg 1 20 kg; Stg 2 10 kg
- Boiloff rate — 6 kg/hr each stage
- Burnout mass scaling equations:

  Stg 1 $3430\ kg + 0.05567\ WP_1 + 0.1727\ WP_2$

  Stg 2 $3800\ kg + 0.05317\ WP_1 + 0.1725\ WP_2$

  Where WP_1 and WP_2 are main and auxiliary propellant capacities respectively
- Stage λ' of 0.93
- Staging base at 477 Km, 31 degrees

4.3.1.3.4 Mass

Summary level mass estimates are presented in Table 4.3-7 for the selected satellite OTV. A weight growth factor of 10% was used rather than 15% as in LSLS based on the judgment that the SPS LO_2 LH_2 OTV would be a second generation vehicle. Mass estimates for the systems reflect the design approach previously described.

4.3.1.3.5 Mission Profile and Flight Operations

Typical orbit transfer operations from LEO to GEO for the common stage OTV are illustrated in Figure 4.3-29. The majority of the delta V for boosting from LEO is provided by Stage 1. Stage 1 then separates and returns to the staging depot following an elliptical return phasing orbit. Stage 2 completes the boost and puts the payload into a GEO transfer and phasing orbit, as well as injecting the payload into GEO and performing the terminal rendezvous maneuver with the GEO construction base. Following removal of the payload, stage 2 uses two primary burns in returning to the LEO staging depot. A detail mission profile indicating events, time and delta V is presented in Table 4.3-8.

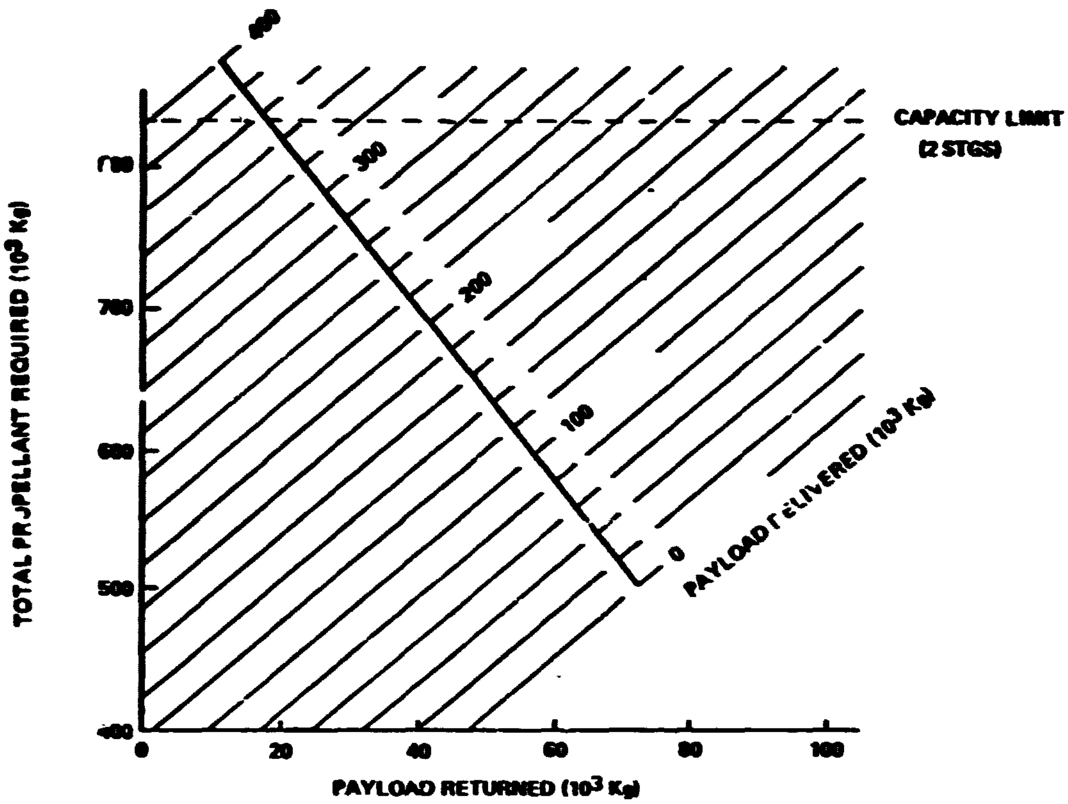


Figure 4.3-28. Two Stage LO_2/LH_2 OTV Performance

SPS-358

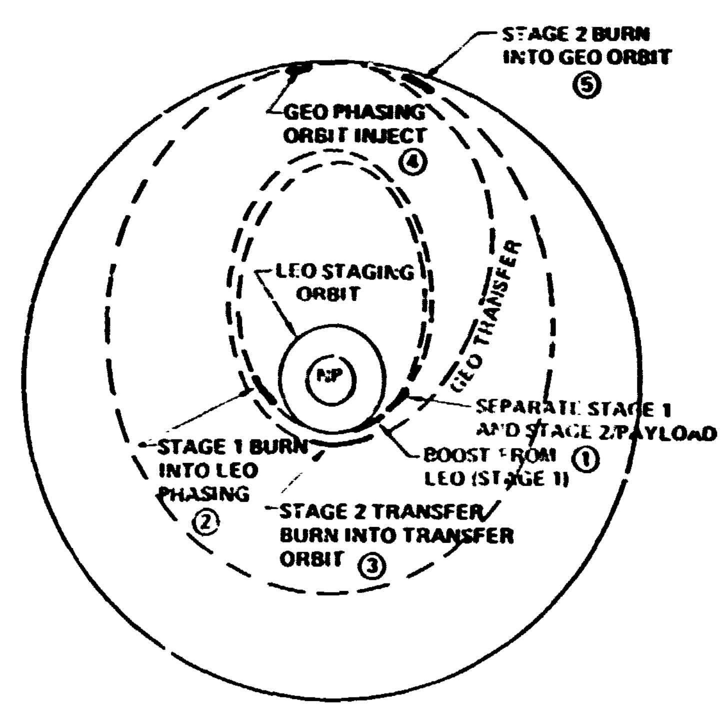


Figure 4.3-29. Chemical OTV Orbit Transfer Operations

Table 4.3-7. Chemical OTV Mass Summary

	Stage 1 (KG)	Stage 2 (KG)
Struct and Mechanisms	13,300	14,780
Main Propulsion	7,090	4,020
Auxiliary Propulsion	820	1,120
Avionics	300	310
Electrical Power	850	820
Thermal Control	1,850	2,310
Weight Growth (10%)	2,420	2,340
Dry	26,630	25,790
Fuel Bias	640	640
Unusable LO_2/LH_2	1,810	1,810
Unusable and Reserve APS	290	660
Burnout	29,370	28,990
Main Impulse Prop	415,000	407,000
APS	2,700	6,100
Startburn	447,070	442,090

A total mission timeline for each stage is presented in Figure 4.3-30. Allowing approximately eight hours for refueling and refurb results in 40 hours elapsed time before a given Stage 1 can be reused. A typical Stage 2, however, has an elapsed time of 85 hours before reuse including time for assembly between stages and between OTV and payload.

With the indicated turnaround times for each stage of an OTV, it is possible to establish the total stage fleet size as shown in Figure 4.3-31. The first two bars are associated with the first OTV flight. At the end of approximately 12 hours the second or upper stage (U1) separates from the first (lower) stage (L1). The first stage complete its operations and is available in time for the third OTV flight. The first upper stage finishes its mission and is available for another flight at the end of approximately 85 hours which allows it to be used on the flight scheduled for the fifth day. With operations conducted in this manner and the requirements for one OTV flight per day for five consecutive days per week (corresponds to launch vehicle operations) a total of two lower and four upper stages are required in the fleet in order to conduct day to day operations. Operated in this manner, as many as six independently operating stages can be in flight at one time during the construction of each satellite.

Table 4.3-8. Mission Profile

MISSION EVENT NO. & NAME	REQUIRED TIME (HR)	DELTA V M/SEC	PROPULSION (MAIN OR AUXILIARY)	REMARK
		MISSION		
1. STANDOFF	0	3	A	PROVIDES SAFE SEPARATION DISTANCE BETWEEN FACILITY & VEHICLE
2. PHASE	12	3	A	ΔV IS ATTITUDE CONTROL
3. COAST	.5	1715	M	OTV FIRST STAGE SEPARATES AFTER THIS ΔV
4. COAST	4.2	3	A	ELLIPTIC REV
5. INJECT	.1	750	M	INCLUDES 60 M/SEC ACCUMULATED FINITE - BURN LOSS
6. COAST	5.4	3	A	TRANSFER TO GEO
7. PHASE INJ	.1	1780	M	REPRESENTATIVE FOR 15° PHASING
8. PHASE	22	3	A	
9. TPI (TERMINAL PHASE INITIATION)	.1	55	M	INCLUDES 15 M/SEC OVER IDEAL TO ALLOW FOR CORRECTIONS
10. RENDEZVOUS	2	10	A	TPI ASSUMED TO OCCUR WITHIN 50 KM OF TARGET
11. DOCK	1	10	A	
12. WAIT	8	0	-	ASSUMED DOCKED
13. STANDOFF	.1	3	A	
14. DEORBIT	.1	1820	M	
15. COAST	5.4	10	A	TRANSFER TO LEO
16. PHASE INJECT	.1	2356	M	
17. PHASE	12	3	A	ORBIT PERIGEE AT STAGING BASE ALTITUDE
18. TPI	.1	60	M	
19. RENDEZVOUS	2	20	A	
20. DOCK	1	10	A	
21. RESERVE	-	130	M	2% OF STAGE MAIN PROPULSION V BUDGET
		FIRST STAGE RECOVERY		
1. COAST	4.2	30	A	ΔV TO CORRECT DIFFERENTIAL NODAL REGRESSION BETWEEN COAST ORBIT AND STAGING BASE
2. PHASE INJECT	.1	1645	M	ELLIPTIC ORBIT PERIGEE AT STAGING BASE ALT.
4. TPI	12	3	A	ALTITUDE CONTROL
3. PHASE	.1	50	M	
5. RENDEZVOUS	2	20	A	
6. DOCK	1	10	A	
7. RESERVE	–	85	M	2% OF STAGE MAIN PROPULSION V BUDGET

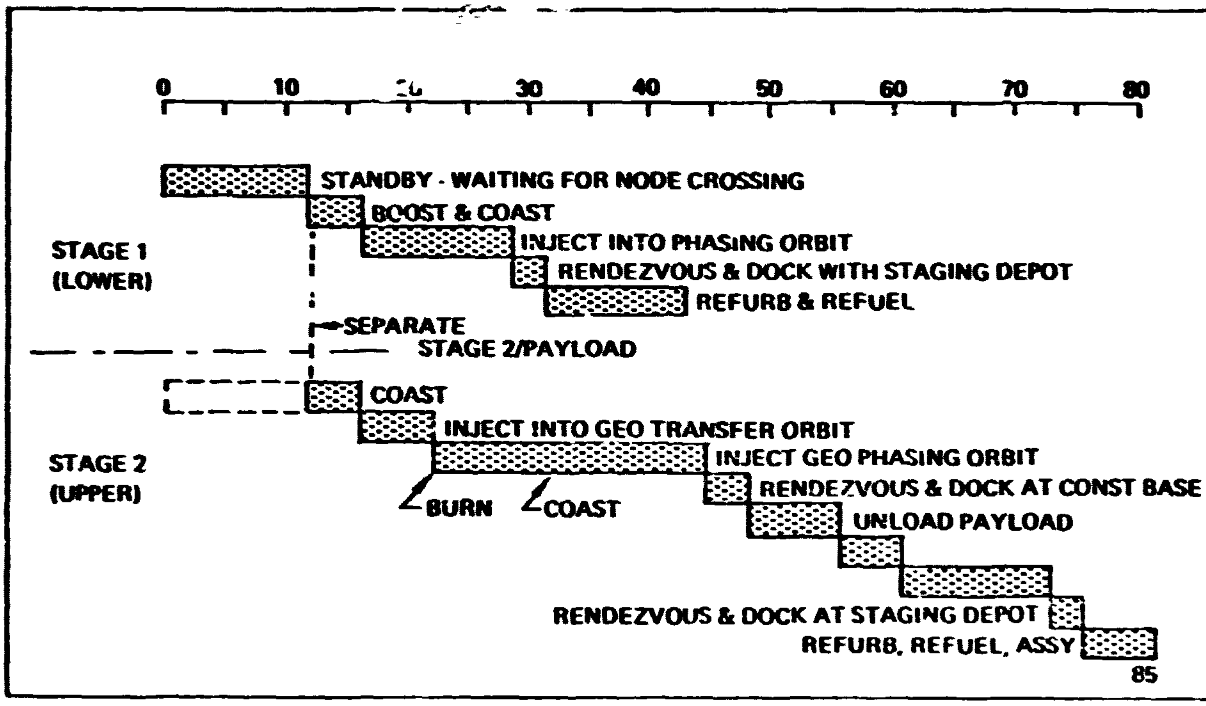


Figure 4.3-30. Chemical OTV Flight Operation Timeline

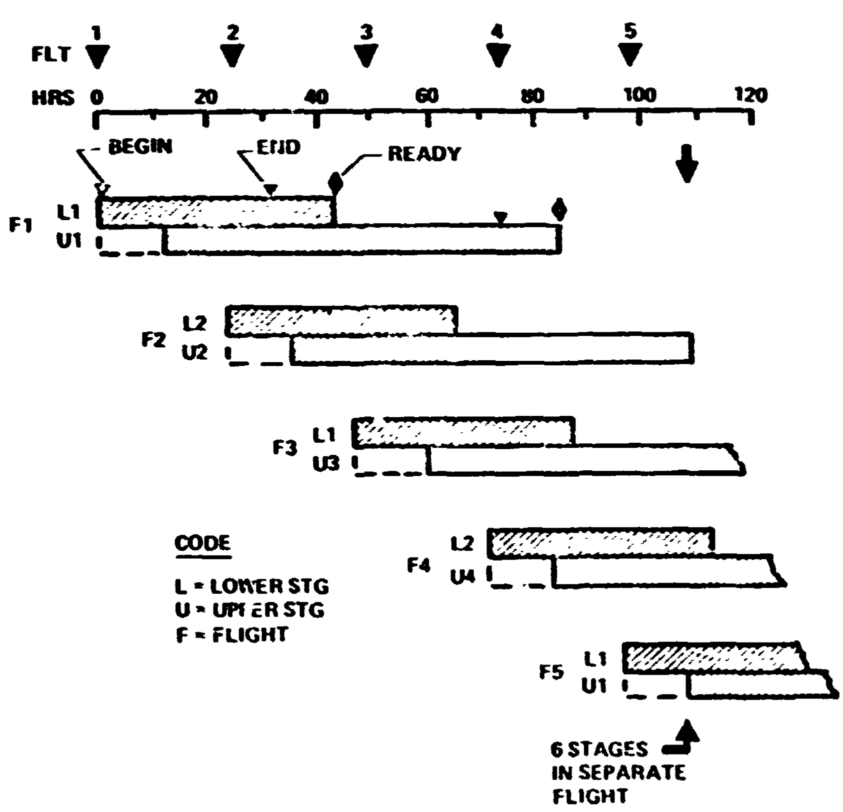


Figure 4.3-31. Flight Operations-Chemical Orbit Transfer

4.3.1.3.6 Cost

DDTE cost for the common stage LO_2/LH_2 OTV with a start burn mass of 900 000 kg is estimated at $950 million (1977 dollars) based on cost parametrics developed in the FSTSA study. The average TFU cost for the two stages is estimated at $82 million (1977 dollars) again using FSTSA parametrics.

Cost per flight for the LO_2/LH_2 OTV is based on the following ground rules:

- Space Based LO_2/LH_2 Common Stage
- Startburn Stage Mass of 445 K kg
- Stage TFU Equal $82M (1977 Dollars)
- 280 OTV Flights Per Satellite
- 4 Satellites Constructed Per Year
- 14 Year Program Life
- 50 Flight Design Life
- Stage Learning Factor of 0.88
- LO_2/LH_2 Bulk Cost of $0 10 per kg
- Spares Equal 50% of Operational Units

The majority of these ground rules are self-explanatory. However, several merit further explanation. The 280 flights for the orbit transfer vehicle is the number required for one satellite. A 14-year program has been assumed for the orbit transfer vehicle, since beyond that point in time it is generally assumed that a different generation of orbit transfer vehicle would be developed. A 50-flight design life has been assumed for the space based orbit transfer vehicle. This value is based on the MSFC Tug Study which assumed 50 uses for a ground based system. Assuming that the SPS OTV is a second generation vehicle, it was assumed 50 uses could be projected for a space based system.

Based on the above ground rules a total of 624 stages (upper and lower) are required resulting in an average stage cost of approximately $31 million. Cost per flight for a complete two stage OTV was estimated as $2.26 million with the following breakdown.

- Operational Units $1.24M
- Propellant $0.40M
- Spares $0.62M

4.3.2 Crew Rotation/Resupply Transportation System

The crew rotation resupply OTV for a photovoltaic or thermal engine satellite or LEO or GEO construction makes use of a common stage LO_2/LH_2 OTV. The system description of this OTV is essentially the same as described in Section 4.3.1.3 although the size of the system does vary with its application.

The complete crew rotation/resupply transportation system required for a photovoltaic satellite is presented in Figure 4.3-32. In the case of LEO construction, the crew rotation resupply concept volves rotating all of the personnel (75) at the GEO base every 90 days and providing supplies for 90 days. As a result, the OTV has a startburn mass of 495,000 kg.

Should the satellite be constructed in GEO, the same OTV as used to deliver the satellite components is employed. As a result, a crew rotation/resupply flight is flown once a month involving 160 personnel and supplies for 480 people and 30 days. Accordingly, the OTV has a startburn mass of 890,000 kg.

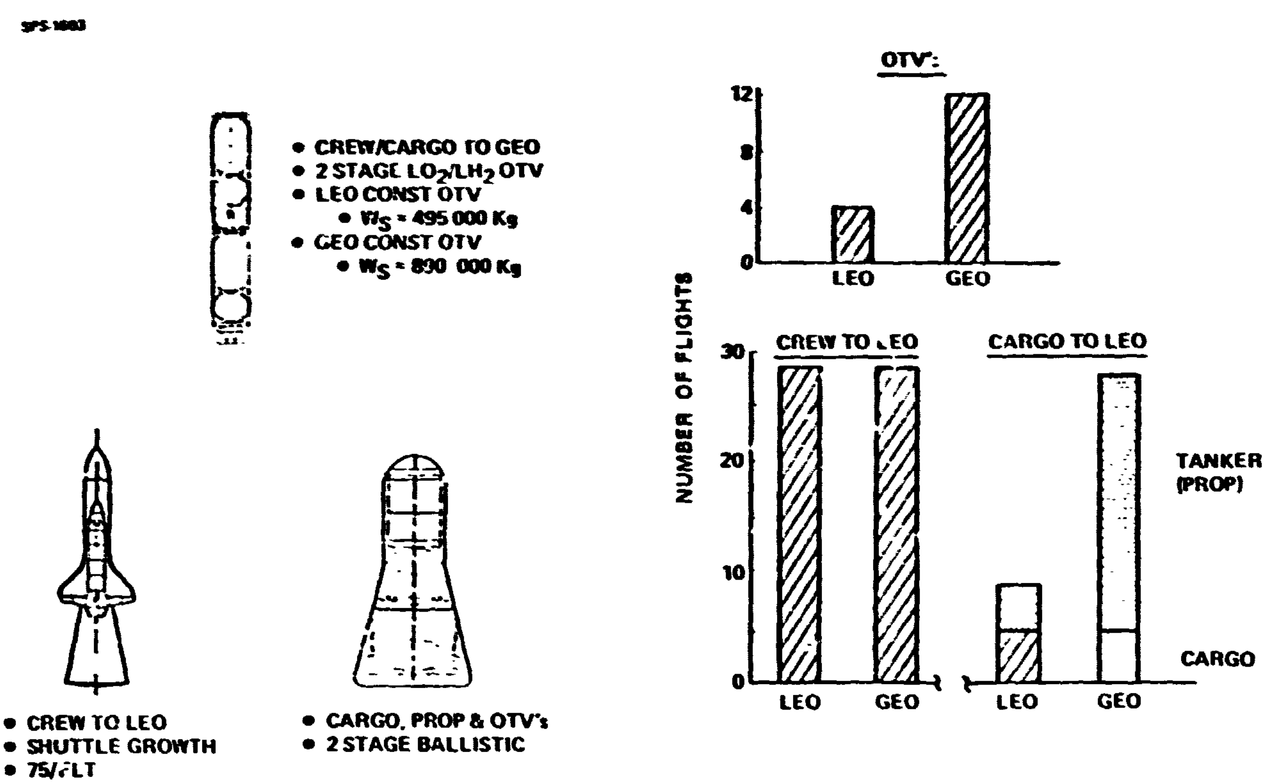


Figure 4.3-32. Crew Rotation/Resupply Transportation

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4.4 INTEGRATED RESULTS

Sections 4.2 and 4.3 have presented data related to either the launch system or orbit transfer system. This section presents total transportation flight and cost data for the preferred transportation system for the reference photovoltaic satellite. The transportation elements consist of a two stage ballistic/ballistic launch vehicle for cargo, self power electric propulsion for the satellite, shuttle growth (reusable liquid booster) vehicle for crew delivery to LEO and LO_2/LH_2 common two stage OTV for crew/supply delivery between LEO and GEO.

The flight schedule associated with the construction and delivery of one satellite per year is presented in Table 4.4-1. The maximum launch rate for the cargo delivery system is expected to be three per day for the case of constructing each of eight modules in 32 days and delivery of all of the OTS propellant for a module in the first 16 days of the 32 day construction period.

Table 4.4-1. Flight Schedule Per Satellite

	Per Satellite	Max. Rate
Earth-LEO Delivery		
2 Stage Ballistic/Ballistic		
Satellite Components	277	35/32 days
Satellite OTS Hardware	33	8/32 days
Satellite OTS Prop	83	24/16 days [1]
Base Crew Supplies	4	1/90 days
Chemical OTV Prop	5	~1/90 days
Shuttle Growth		
Personnel	28	~1/13 days
LEO-GEO Deliver		
Satellite	8	1/40 days
Crew/Supplies	4	1/90 days

[1] Relates to module transporting an antenna.

The total transportation cost for one satellite (based on four satellites/year construction rate) is estimated at slightly under $4.8 billion. A breakdown of this cost is presented in Table 4.4-2.

Table 4.4-2. Transportation Cost (Cost in Millions)

Satellite Transportation		(4250)
Components to LEO	2325	
OTS Hardware to LEO	355	
OTS Prop to LEO	795	
OTS Hardware	775	
Crew Rotation Resupply Transportation		(240)
Supplies to LEO	45	
OTV Prop to LEO	45	
OTV Hardware	10	
Crew to LEO	140	
Interest (Self Power Trip Time)		(300)
Total		$4790