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D180-20689-3
Part 1 Volume III

Construction, Transportation and Cost Analyses

Solar Power Satellite

SYSTEM DEFINITION STUDY

D180-20689-3

Contract NAS9-15196
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Part 1 Volume III

Construction, Transportation and Cost Analyses

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Approved By: G.R. Woodcock

G.R. Woodcock
Study Manager

Boeing Aerospace Company
Missiles A... Space Group
Space Div...
P.O. Box ...
Seattle, Washington

FOREWORD

The SPS systems definition study was initiated in December 1976. Part I was completed on May 1, 1977. Part I included a principal analysis effort to evaluate SPS energy conversion options and space construction locations. A transportation add-on task provided for further analysis of transportation options, operations, and costs.

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:

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

The Boeing study manager was Gordon Woodcock. Boeing technical leaders were:

Vince Caluori	Photovoltaic SPS's
Dan Gregory	Thermal Engine SPS's
Eldon Davis	Construction and Orbit-to-Orbit Transportation
Hal DiRamio	Earth-to-Orbit Transportation
Dr. Joe Gauger	Cost
Bob Conrad	Mass Properties
Rod Darrow	Operations
Bill Emsley	Flight Control
Jack Gewin	Power Distribution
Don Grim	Electric Propulsion
Henry Hillbrath	Propulsion
Dr. Ted Kramer	Thermal Analysis and Optics
Keith Miller	Human Factors and Construction Operations
Jack Olson	Configuration Design
Dr. Henry Oman	Photovoltaics
John Perry	Structures

The Part I Report includes a total of five volumes:

Vol. I	D180-20689-1	Executive Summary
Vol. II	D180-20689-2	System Requirements and Energy Conversion Options
Vol. III	D180-20689-3	Construction, Transportation, and Cost Analyses
Vol. IV	D180-20689-4	SPS Transportation System Requirements
Vol. V	D180-20689-5	SPS Transportation: Representative System Descriptions

Requests for information should be directed to Gordon R. Woodcock of the Boeing Aerospace Company in Seattle or Clarke Covington of the Spacecraft Design Division of the Johnson Space Center in Houston.

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3.4 CONSTRUCTION ANALYSIS

3.4.1 Satellite Construction Concepts

3.4.1.1 Approach

3.4.1.1.1 Construction Analysis Objectives–Unlike the other analyses performed during Part I, the construction analysis was not directed at determining the mass and cost associated with the construction of the alternative satellite types or alternative construction sites. The objectives of the Part I construction analysis were the following:

For each satellite type and each construction location,
- define workable construction concepts
- define associated types of facilities to be used
- define the construction sequences
- define the time allocations for each major construction task
- define the functional requirements for the construction machinery
- estimate how many of each type of construction machine is required and what operating rate is quired
- imate the number of personnel required

3.4.1.1.2 Construction Analysis Constraints and Assumptions–Due to the limited analytical time available, the number of satellite types/construction locations, and the lack of a data base from which to start, it was necessary to adopt some simplifying assumptions, constraints, and groundrules:

1. The total satellite was to be constructed in one year (excluding LEO-to-GEO transportation time).

2. construction machines of a given type were to have the same operating rate.

3. Antenna construction was not analyzed, but time for attaching the antennas to the satellite was allowed and an antenna construction crew size was estimated.

4. port equipment required to deliver parts to the construction machines was not considered, however, the support crew size was estimated.

5. Subsystem installation was not considered, however a subsystem installation crew size was estimated.

6. In most cases, only a single design solution was pursued. There was not time to analyze attenative machine or facility approaches.

3.4.1.1.3 **Construction Philosophy**—As the various satellite types were analyzed, a set of underlying principles (objectives, goals, guidelines) evolved that were incorporated into all of the various construction concepts. The collection of these principles could be called our "construction philsophy", see Table 3.4-1. In some cases, not all of these principles could be satisfied due to peculiar satellite configuration details. These principles will be reevaluated during Part 2.

TABLE 3.4-1
CONSTRUCTION PHILOSOPHY

CONCEPT	RATIONALE
• FACILITIZED CONSTRUCTION	• DO NOT HAVE TO BUILD IN EXTRA STRENGTH (MASS) INTO EVERY SATELLITE IN ORDER TO SUPPORT CONSTRUCTION EQUIPMENT • CONSTRUCTION OPERATIONS CAN BE DECOUPLED
• DECOUPLED OPERATIONS	• CONSTRUCTION OPERATIONS SHOULD BE INDEPENDENT AS POSSIBLE SO THAT A SLOW DOWN OR SHUTDOWN IN ONE OPERATION HAS MINIMUM IMPACT ON OTHERS
• MAJOR SUBASSEMBLIES IN PARALLEL	• FABRICATE MAJOR SUBASSEMBLIES IN PARALLEL IN SEPARATE FACILITY LOCATIONS SO THAT MAXIMUM TIME CAN BE ALLOTTED TO EACH SUBASSEMBLY FABRICATION
• CONTINUOUS BEAMS	• CONTINUOUS BEAMS, WHETHER CURVED OR STRAIGHT, - MINIMIZE THE NUMBER OF JOINTS - ELIMINATES THE NEED FOR SOME JOINT PLUG ASSEMBLIES
• MOVING BEAM MACHINES	• PLACING BEAM MACHINES ON TRACKS SUCH THAT THE MACHINE BACKS AWAY FROM "EXTRUDED" BEAM IS PREFERRED OVER FIXED BEAM MACHINES: - CONTINUOUS LONGITUDINAL BEAMS CAN BE MADE (NO LONGITUDINAL BUTT JOINTS REQUIRED) - CROSS FRAMES CAN BE STARTED AS SOON AS LONGITUDINAL BEAM MACHINES PASS THE JOINT AREA
• SUPPORT THE BEAMS	• LONG BEAMS SHOULD BE SUPPORTED AS THEY ARE FABRICATED TO ELIMINATE UNDESIRED STRESS AND UNGUIDED END POSITIONS
• MINIMIZE USE OF FREE FLYERS	• MACHINES THAT FREE FLY ARE NOT DESIRED. THE SATELLITE COMPONENTS ARE TOO FRANGIBLE TO TOLERATE ACCIDENTAL COLLISIONS. PROPELLANT CONSUMPTION, EXHAUST PRODUCT CONTAMINATION, AND PLUME IMPINGEMENT WOULD PRESENT PROBLEMS

3.4.1.2 **Summary of Results**–The detailed construction analysis of each of the six satellite types/ construction base alternates are included in Section 3.4.1.3. In this section, the significant results of the various construction analysis are summarized and evaluated.

3.4.1.2.1 **Construction Machinery Requirements**–The number of construction machines required for each concept is summarized in Figure 3.4-1.

3.4.1.2.2 **Manning Requirements**–The total crew size estimates for each satellite type/construction location is summarized in Figure 3.4-1 and is given in detail in Table 3.4-2.

3.4.1.2.3 **Construction Facilities**–The construction facilities derived for the various satellite types/ construction locations are shown in Figure 3.4-2.

3.4.1.2.4 **Constructability Rating**–An attempt was made to integrate the various construction factors into a "constructability rating". Figure 3.4-3 shows the results. The process used to establish the rating scores is described in Section 3.4.7.

(NOTE: This preliminary constructability rating does not include cost or mass factors.)

The relative ratings show that for a given satellite type, there is very little difference in the constructability due to construction base location.

The CR=1 Photovoltaic satellite should be the most constructable and the thermal engine satellite should be more difficult. However, the thermal engine satellite is constructable.

3.4.1.3 Construction Analysis

3.4.1.3.1 Photovoltaic Satellite (CR=2) GEO Base Construction Analysis

3.4.1.3.1.1 **Introduction**–The CR=2 Photovoltaic satellite is one of the two reference satellite configurations. Its distinguishing feature is that it employs large reflectors to concentrate the sunlight on the solar cells.

This was the first construction analysis performed. The construction philosophy, machine operating rates, and assumptions that evolved from this analysis were applied during the analysis of other satellite types.

Table 3.4-2 Manning Requirements Summary

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	CR - 2.0 Photovoltaic satellite				Thermal engine satellite				CR = 1.0 Photovoltaic satellite			
	LEO construction		GEO construction		LEO construction		GEO construction		LEO construction		GEO construction	
	LEO base	GEO base	LEO base	GEO base	LEO base	GEO base	LEO base	GEO base	LEO base	GEO base	LEO base	GEO base
Base management	(10)	(5)	(5)	(10)	(10)	(5)	(5)	(10)	(10)	(5)	(5)	(10)
Satellite construction	(302)	(135)	(0)	(414)	(337)	(119)		(331)	(186)	(95)	—	(220)
Management	72	22	—	80	21	14	—	21	46	22	—	42
Machine operators	152	32	—	170	146	20	—	140	78	20	—	57
Subsystems	12	15	—	24	30	30	—	30	12	15	—	24
Maintenance	28	28	—	56	68	30	—	68	28	16	—	48
Test and checkout	38	38	—	78	72	25	—	72	22	22	—	54
Antenna construction	(84)	(54)	—	(84)	(84)	(54)	—	(84)	(84)	(54)	—	(84)
Base operations	(138)	(68)	(82)	(124)	(138)	(68)	(82)	(124)	(138)	(68)	(82)	(124)
Management	12	8	8	12	12	8	8	12	12	8	8	12
Data processing	6	4	4	6	6	4	4	6	6	4	4	6
Base maintenance	42	19	19	42	42	19	19	42	42	19	19	42
Transportation	24	10	24	10	24	10	24	10	24	10	24	10
Materials handling	46	19	19	46	46	19	19	46	46	19	19	46
Communications	8	8	8	8	8	8	8	8	8	8	8	8
Base support	(64)	(37)	(23)	(64)	(64)	(37)	(23)	(64)	(64)	(37)	23)	(64)
Management	7	5	5	7	7	5	5	7	7	5	5	7
Utilities	14	8	2	14	14	8	2	14	14	8	2	14
Hotel/food service	24	12	4	24	24	12	4	24	24	12	4	24
Medical/dental	13	6	6	13	13	6	6	13	13	6	6	13
Safety	2	2	2	2	2	2	2	2	2	2	2	2
Chaplain	2	2	2	2	2	2	2	2	2	2	2	2
[illegible] control	2	2	2	2	2	2	2	2	2	2	2	2
Totals	598	299	110	692	633	283	110	613	477	259	110	502
Total	897		806		916		720		736		612	

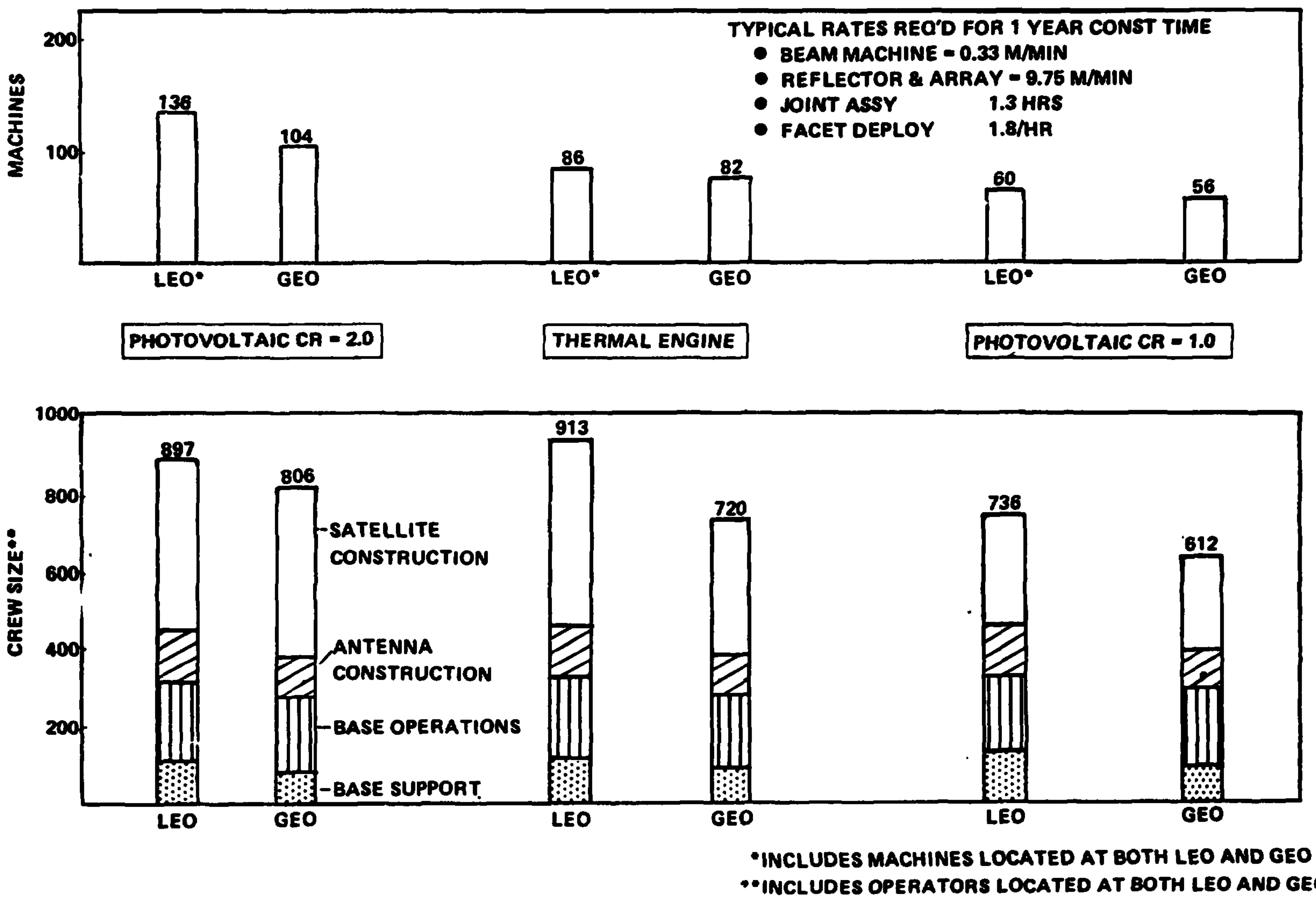

Figure 3.4-1 Construction Machine and Crew Size Comparison

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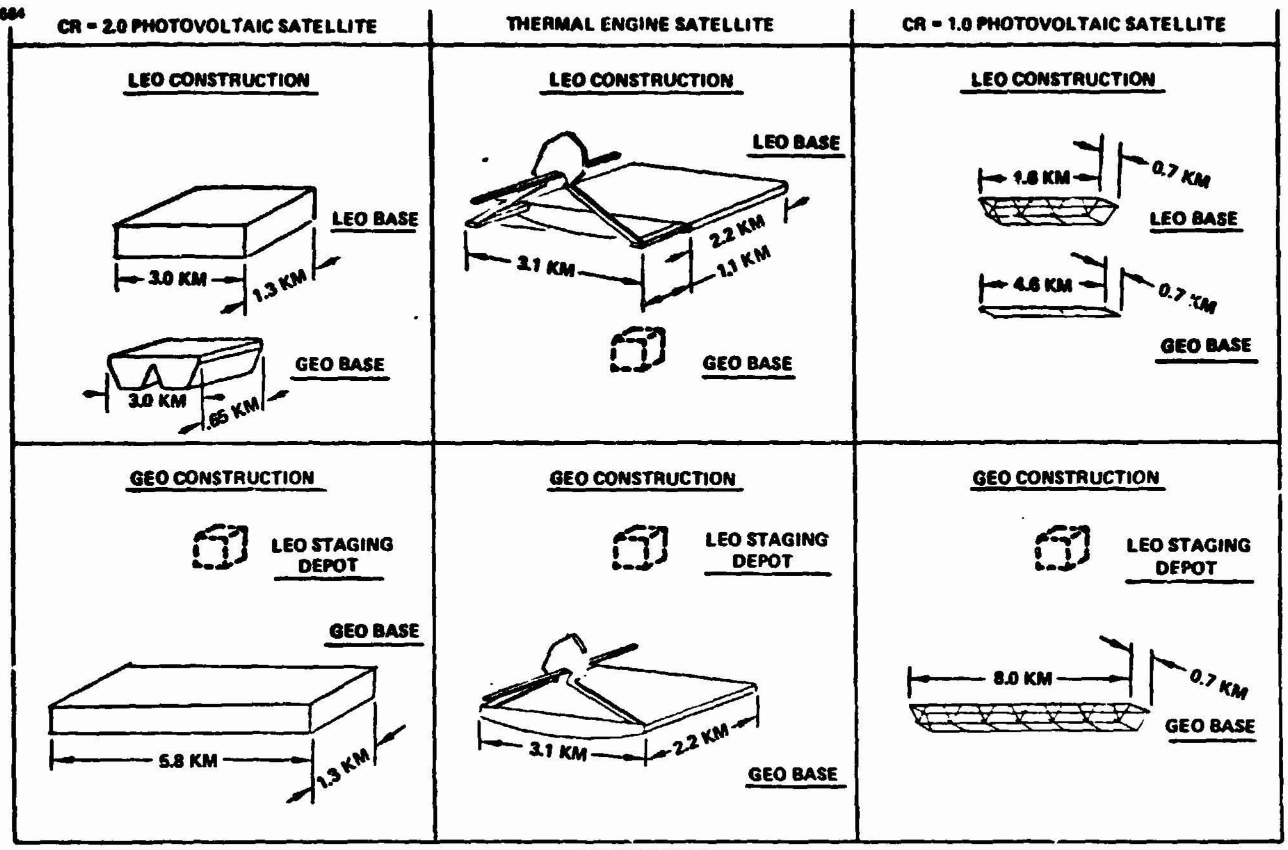

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Figure 3.4-2 Satellite Construction Facility Comparison

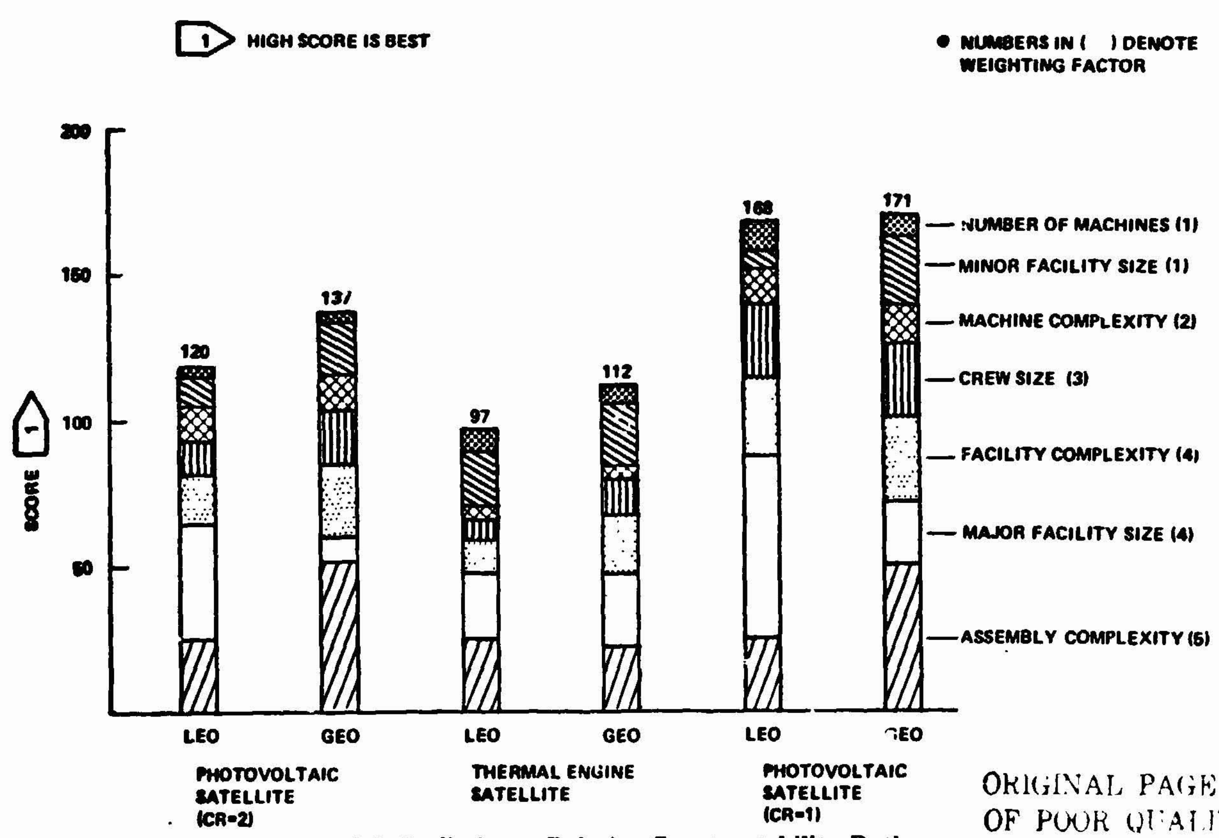

Figure 3.4-3 Preliminary Relative Constructability Rating

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3.4.1.3.1.2 **Overview**–In this section, the reference satellite configuration will be described and the top-level timeline will be developed.

Reference Satellite Configuration–The CR=2 photovoltaic satellite configuration that will be used as the reference for the construction analysis is shown in Figure 3.4-4.

Top-level Timeline Analysis

Assumption #1 - Total construction time/satellite = 365 days.
Assumption #2 - Allow 30 days for final integration and checkout.
Assumption #3 - Allow 20 days to fabricate end structures and miscellaneous components (10 days for each end).
Assumption #4 - Allow 20 days to install antennas (10 days each).
Assumption #5 - Antennas constructed in parallel with satellite.
Assumption #6 - Although 1 day/week will be allotted as an off duty day for each crewmember, the work phasing can be organized such that a common shutdown day is not required.

Given these assumptions, it is found that there are 305 days available for fabricating/assembling 44 bays (see Figure 3.4-5); approximately 7 days per bay.

Assumption #7 - Allow 1 day/bay for:
- catch up if machines break down
- final inspection
- coordination between all bays
- moving machinery out of way
- indexing satellite to next bay
- maintenance on machinery

Allow 6 days/bay available for primary fabrication/assembly work.

Assumption #8 - Two-bay construction facility.

The construction of the frame will be done in parallel with the other assembly operations.

Assumption #9 - Allow 10 hours to index the next bay (1.1 meter/minute)

Taking these factors into account, the top-level timeline for each bay is as shown in Figure 3.4-6.

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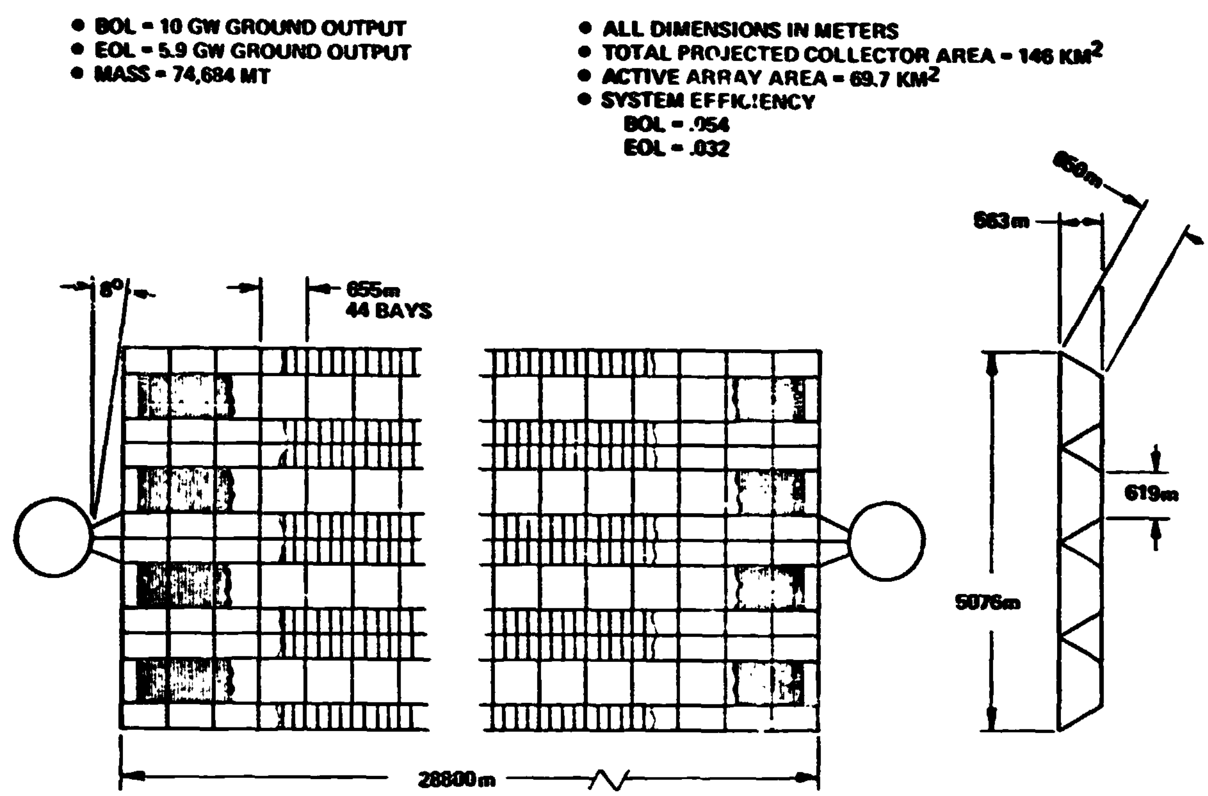

Figure 3.4-4 Reference Silicon Configuration

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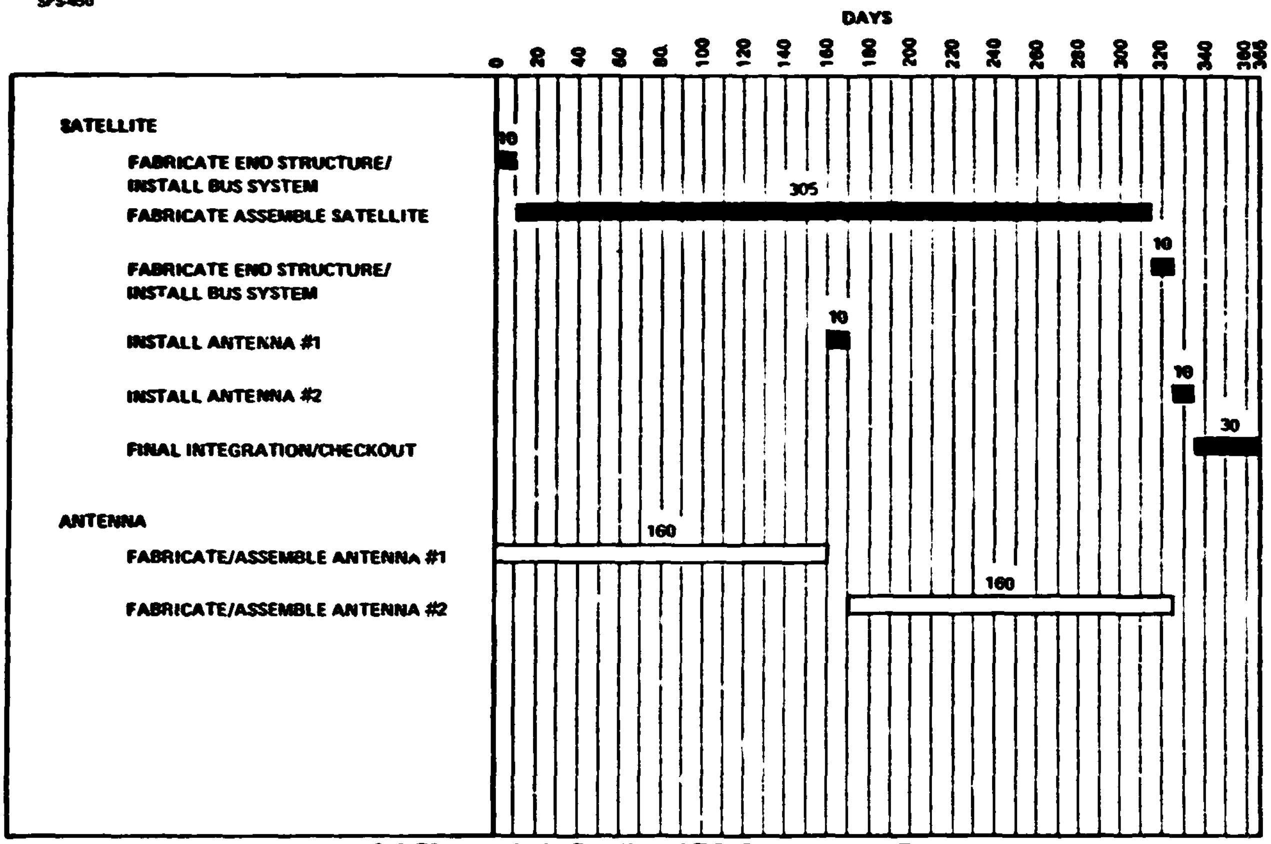

Figure 3.4-5 CR = 2.0 Photovoltaic Satellite GEO Construction Top Level Timeline

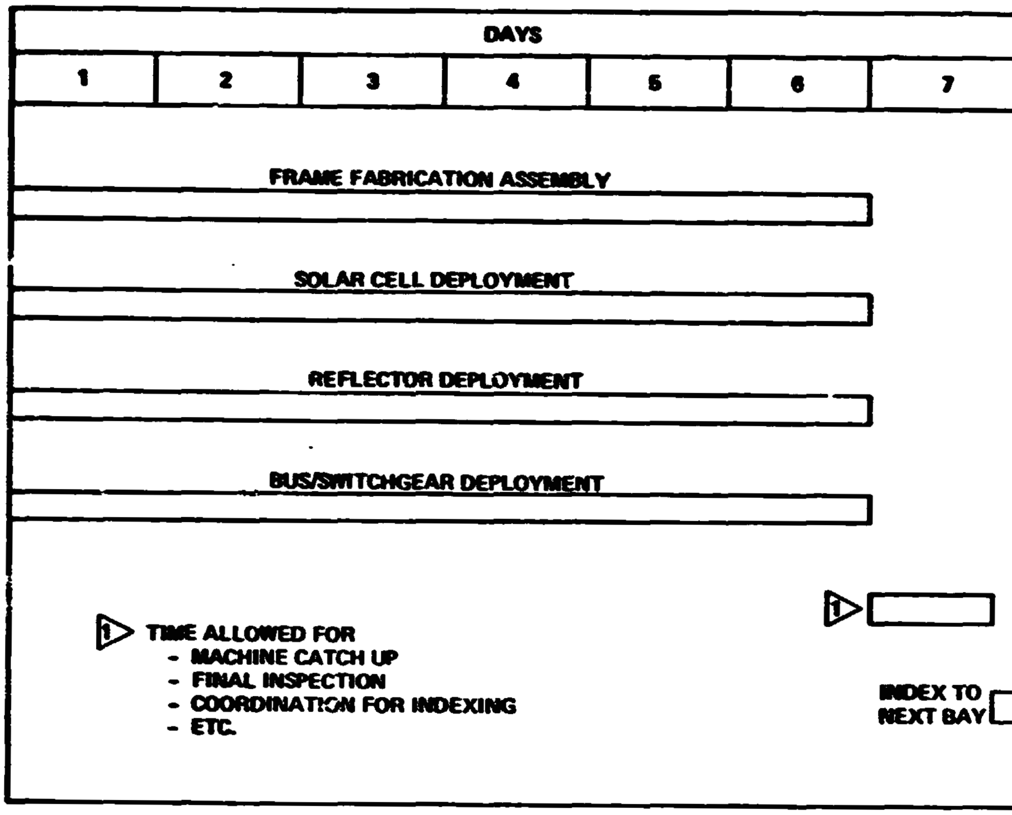

Figure 3.4-6 Top-Level Timeline for the Construction of Each Bay of the CR = 2.0 Photovoltaic Satellite

3.4.1.3.1.3 Construction Concept

Configurations

Frame Configuration–The frame configuration to be built within the 7 day time allotment is shown in Figure 3.4-7. The frame nomenclature shown in this figure will be used throughout this analysis.

To assemble the framework, it will be necessary to adapt the ends of some of the frames (as made by beam machines) to the angular surfaces of other frames to which they are attached. To accommodate this, joint plugs will be used (see Figure 3.4-8).

Solar Cell Blanket Assembly Configuration–The solar cell blankets will be delivered to the installation area in 20m wide accordion-folded blanket packages which will be attached to the frame. The blankets have a preattached bridle that will be used to unfold and deploy the blanket across the trough. The bridle will be attached to a preinstalled tensioning device. The edges of adjacent solar cell blankets are then attached. Figure 3.4-9 depicts the configuration described.

Reflector Assembly Configuration–The reflector assembly is similar to the solar cell blanket assembly. Figure 3.4-10 depicts the configuration. Note that the reflector is assumed to be delivered in rolls.

Bus/Switch Gear Assembly–The electrical power collection system is shown in Figure 3.4-11 and 3.4-12. These figures show that on both sides of each trough that there are two busses that have to be installed: 1) Module bus (triangular in shape-up to .415 m at highest point), and 2) the Main Bus (rectangular in shape - .415 incremental jumps - 9.13 m maximum of end bays.)

At one end of each bay, switch gear will be installed.

3.4.1.3.1.4 Construction Timeline Analysis

Frame Assembly

Assumption #10 - Twelve beam machines used (see Figure 3.4-13) per trough
- Some of the beam machines are used to make more than one beam (see Table 3.4-3).
- Some of the beam machines can be moved in and out of position in order to 1) use a machine in two positions and/or 2) to get the machines out of the way so the satellite can be indexed.

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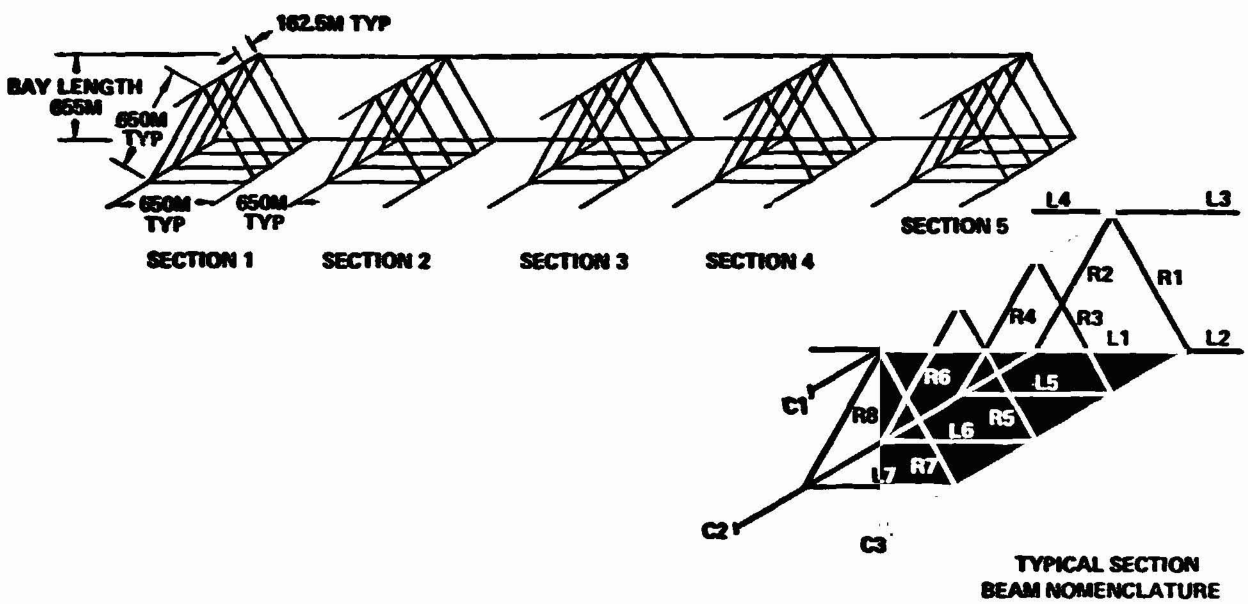

Figure 3.4-7 Satellite Frame Concept

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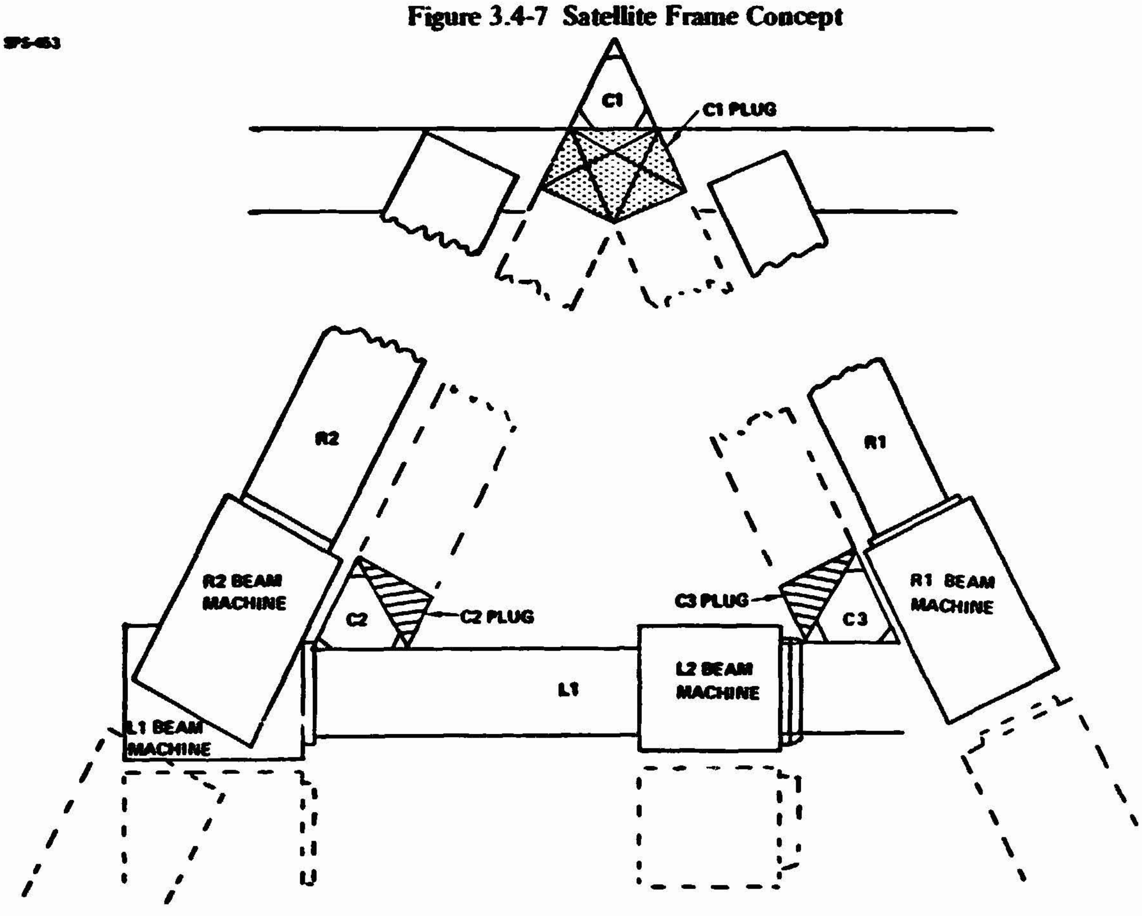

Figure 3.4-8 Joint Plugs

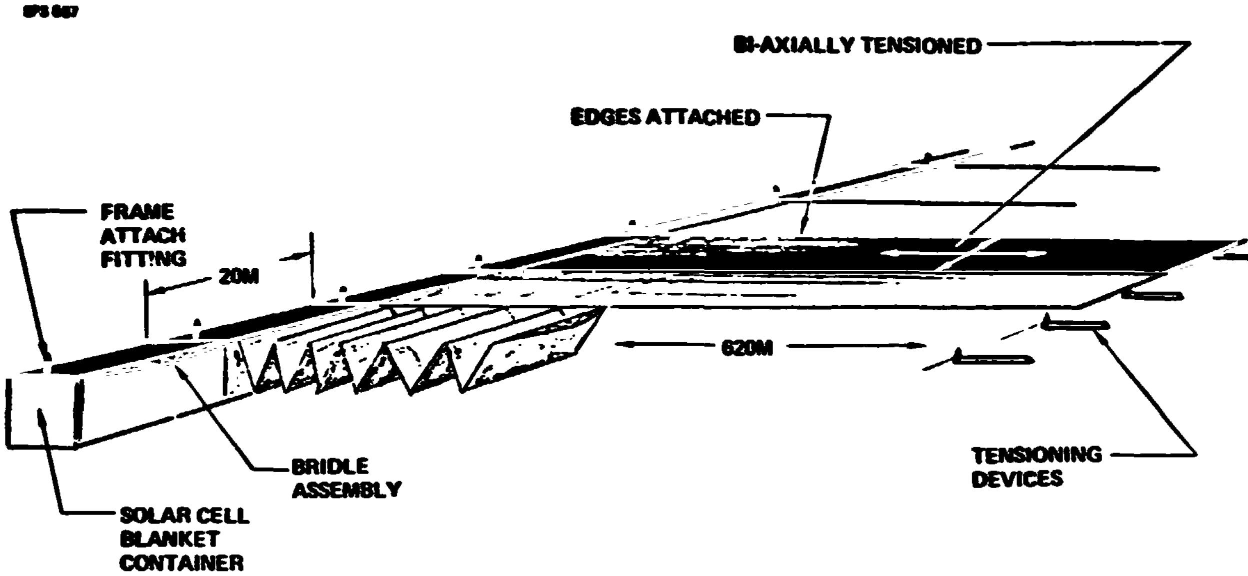

Figure 3.4-9 Solar Cell Installed Configuration

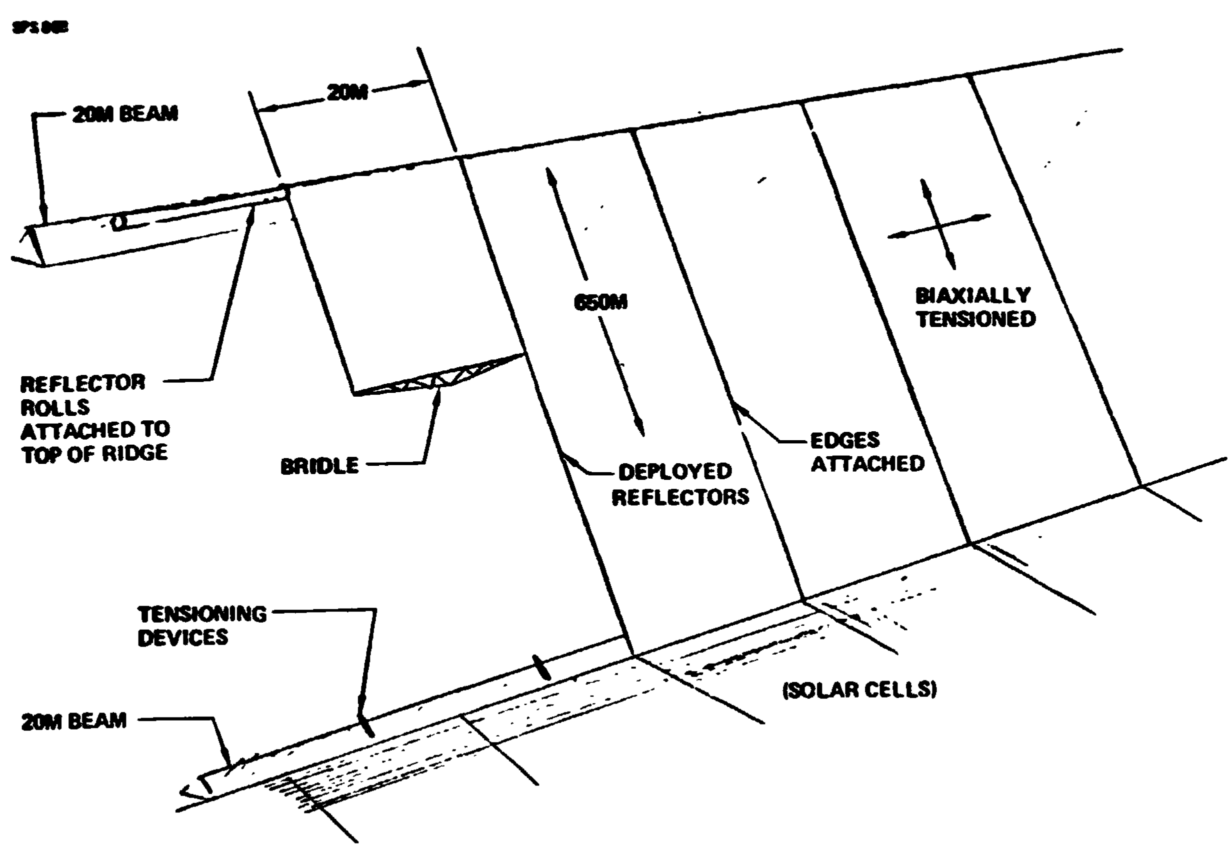

Figure 3.4-10 Reflector Installed Configuration

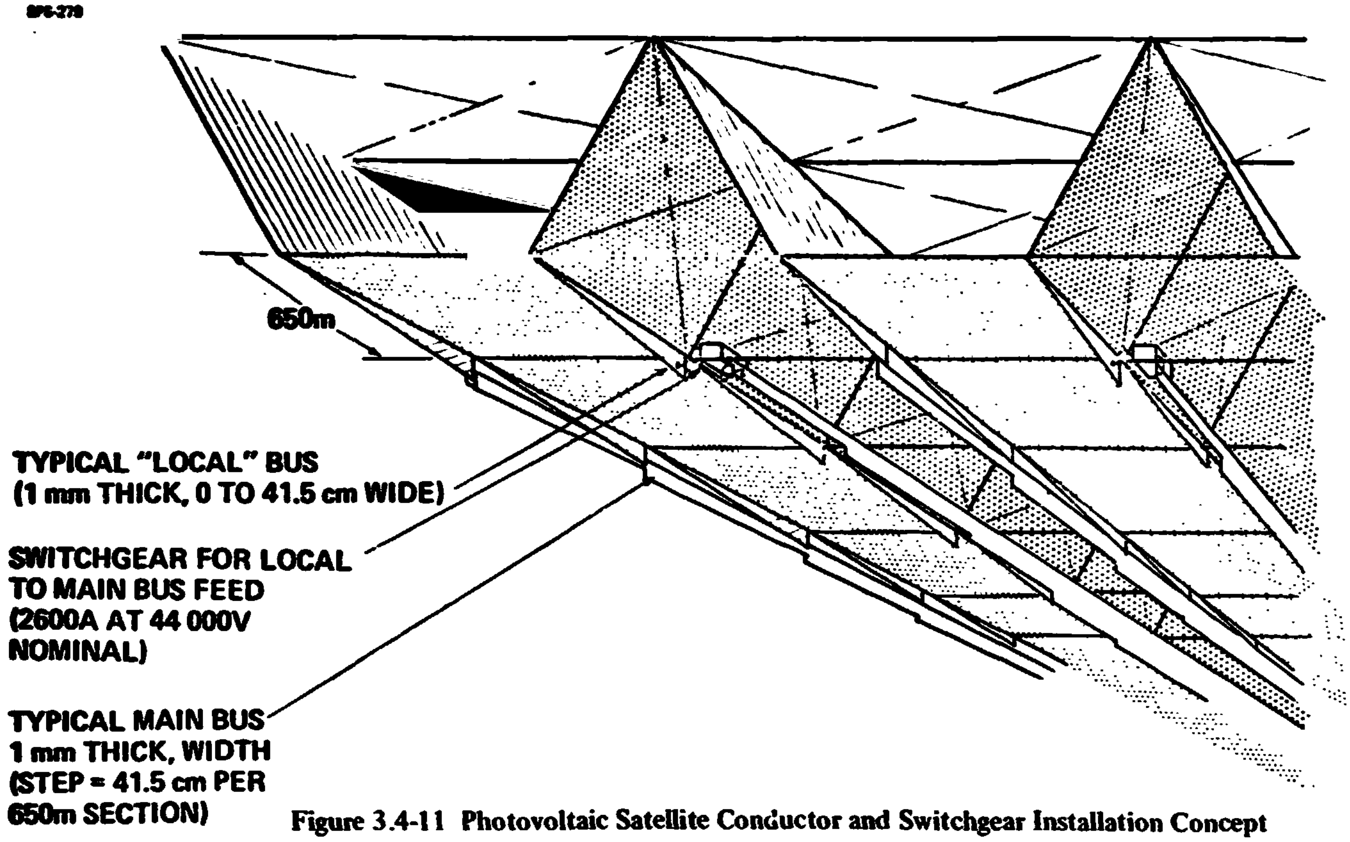

Figure 3.4-11 Photovoltaic Satellite Conductor and Switchgear Installation Concept

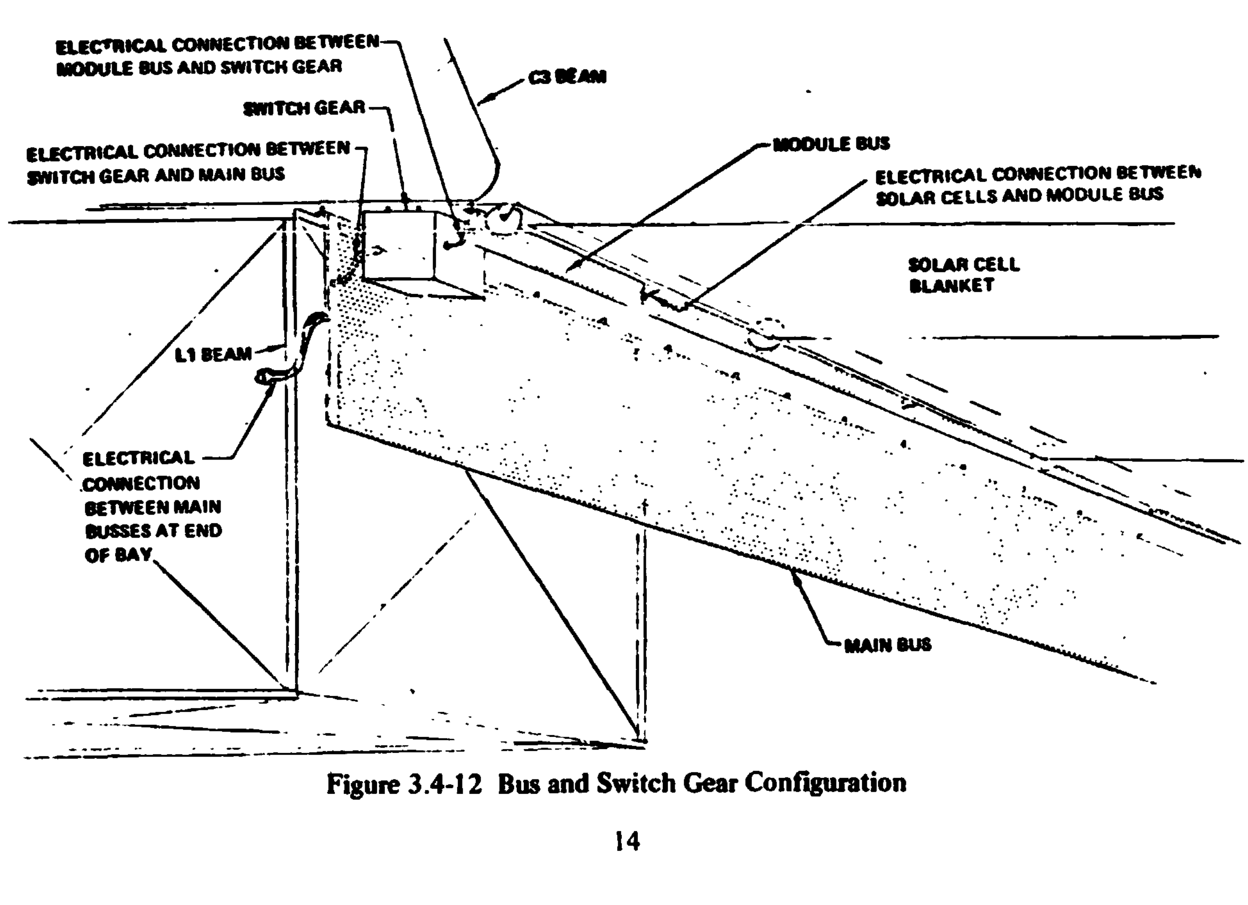

Figure 3.4-12 Bus and Switch Gear Configuration

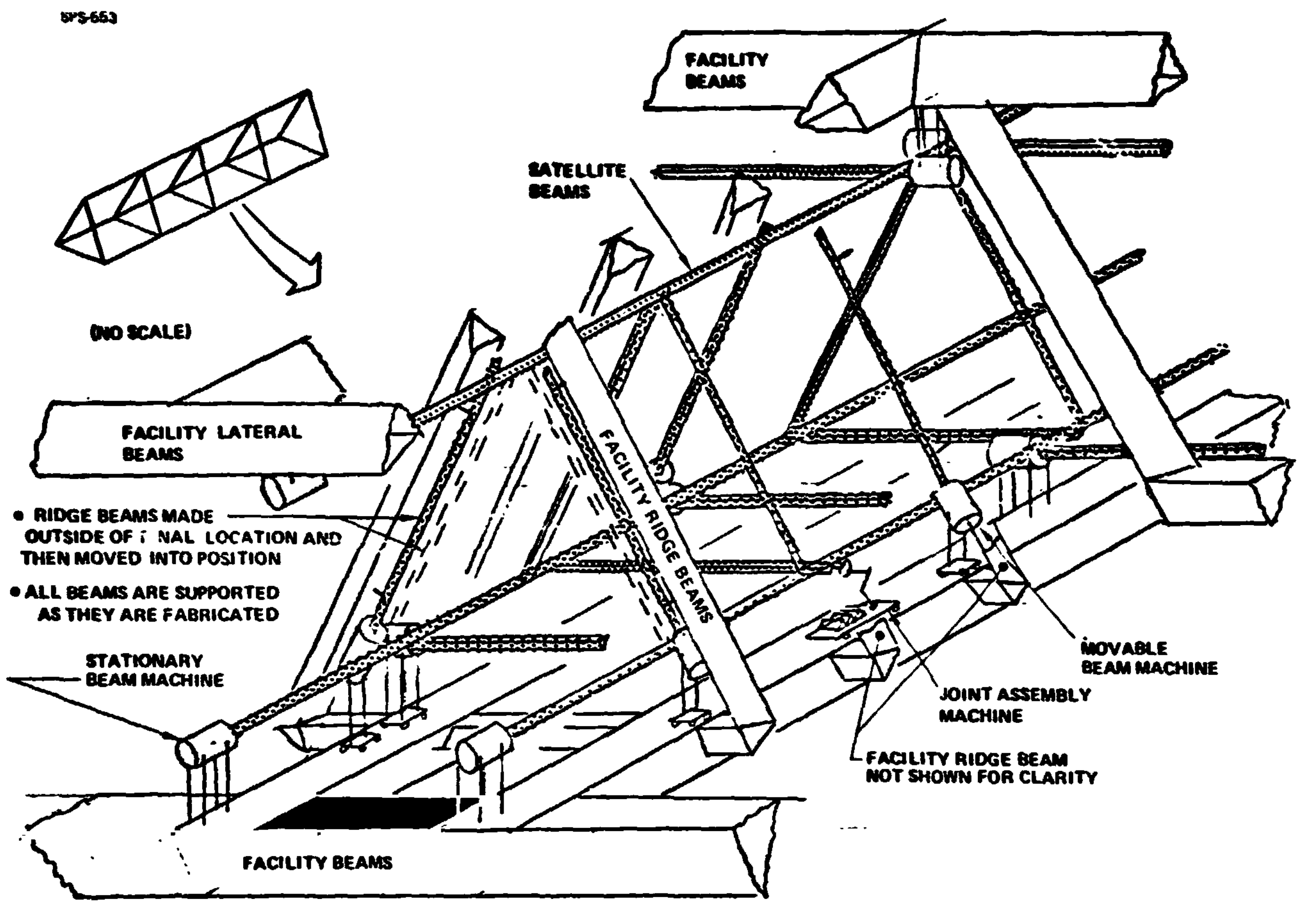

Figure 3.4-13 Frame Fabrication/Assembly Concept Photovoltaic Satellite (CR=2)

- The ridge beams are made outside of their final location as there is no way to make them in their final position due to the longitudinal beams C1, C2, and C3 occupying the joint area (see Figure 3.4-13).

Assumption #11 - Six beam machine operators used to operate the twelve beam machines (see Table 3.4-3).

TABLE 3.4-3

Operator	Beam Machine											
	C1	C2	C3	R1	R2	R5	R6	L1	L2	L3	L4	L6
1	X			▷1	▷1	▷1	▷1	▷1			X	▷1
2		X								X		
3			X						X			
4				X		X						
5					X		X					
6								X				X

▷1 Beam machine used in two places

See Figure 3.4-14 for construction sequence.

The three steps shown in Figure 3.4-14 are to be accomplished in six days.

Assumption #12 - Twenty-two hours/day available using 12-hour, 2-shift schedule with 4/ .5/4/ .5/4/11 work/rest cycle)

Assumption #13 - Allow one hour to move the beam machines into next position (2.7 m/min).

The required Beam machine rate is .25 m/min.

Other stationary beam machines have most of the 44 hours available (deduct five minutes to move machine into position from its retracted location). These also require a rate of 0.25 m/min.

These rates have to be adjusted to reflect the operator productivity; (see Section 3.4.4):

A rate of 0.33 m/min is used hereafter in this discussion.

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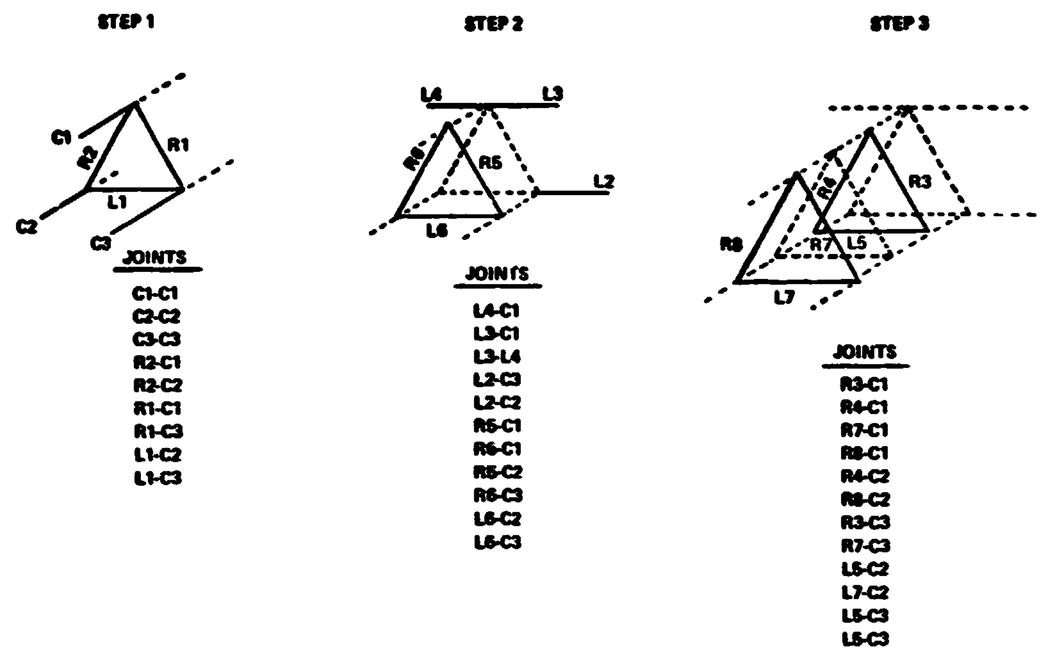

NOTE: DASHED LINES REPRESENT
BEAMS ALREADY COMPLETED

Figure 3.4-14 Beam Construction Sequence

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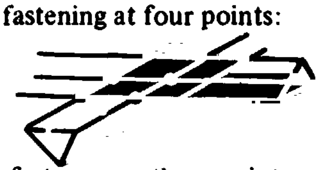
Assumption #14 - Lap joints require fastening at four points:

Lap Joint 

Assumption #15 - Butt joints require fastening at three points:

 Butt Joint

The detailed timeline for joint assembly is shown in Figure 3.4-15
- 60 min/lap joint and 50 min/butt joint

The C1, C2 and C3 plugs can be made at a subassembly factory. The plugs are transported from the factory and placed on joint assembly machines.

The timeline for the plug joint assembly is shown in Figure 3.4-16. After the plug is into position, the joint assembly sequence shown in Figure 3.4-14 is used (minus the first 15 minutes) increment allocated for moving the machine into place).

Total time for ridge frame joint assembly is 2 hrs.

The sequence of joint making is shown in Figure 3.4-14 and are further tabulated in Table 3.4-4.

Table 3.4-4

	Top		Bottom Left		Bottom Right	
	Butt	Lap	Butt	Lap	Butt	Lap
Step 1	C1-C1	R2-C1	C2-C2	R2-C2	C3-C3	R1-C3
		R1-C1		L1-C2		L1-C3
Step 2	L4-L3	L4-C1		L2-C2		L2-C3
		L3-C1		R6-C2		R5-C3
		R3-C1		L6-C2		L6-C3
Step 3		R4-C1		R4-C2		R3-C3
		R3-C1		L5-C2		L5-C3
		R7-C1		R8-C2		R7-C3
		R8-C1		L7-C2		L7-C3

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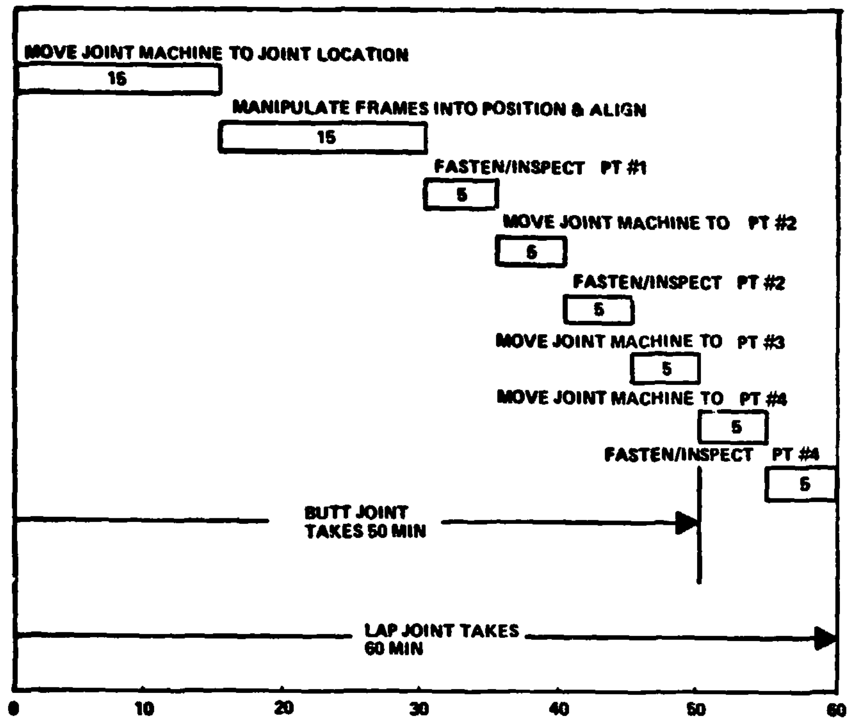

Figure 3.4-15 Lap and Butt Joint Ass'y Timeline

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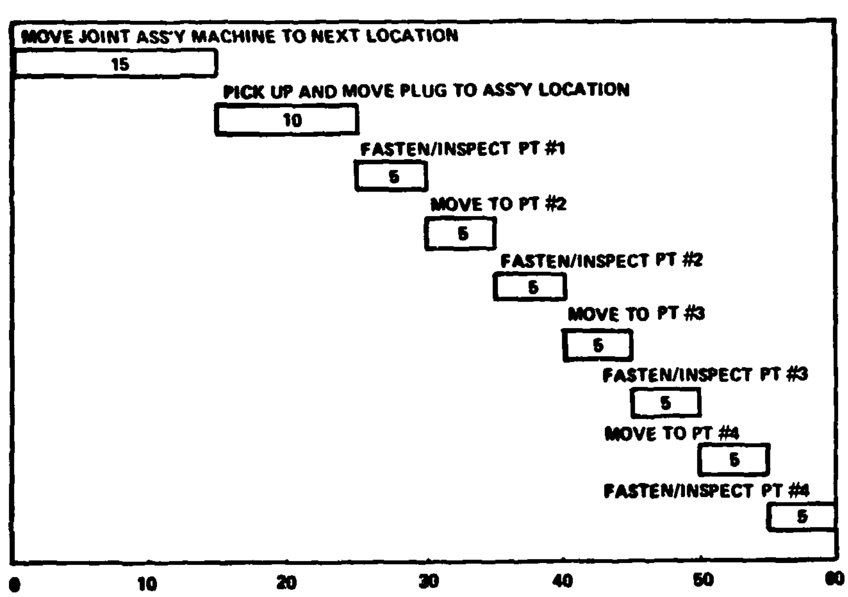

Figure 3.4-16 Plug Joint Ass'y Timeline Reflector Installation

Beam Machine Reloading

Based on the beam machine usage described above, it is necessary to reload the R1, R2, L1, R6 and L6 machines within one day (21 hours) so that they will be ready to go after the satellite has been indexed: There will be a total of 30 beam machines to be loaded within 21 hours (six machines per section). The machines should be designed to reload much faster than 21 hours in order to allow for machine breakdown delays.

Frame Fab/Assembly Timeline Summary

The assumptions, sequences, and rates described above have been used to create the detailed frame assembly timeline shown in Figure 3.4-17.

Reflector Deployment

Assumption # 16 - Reflector width = 20 m

Functions to be performed.

1. Deploy/attach reflector rolls (edge of C1 frame).
2. Deploy/attach reflector stretcher (edge of C3, C2).
3. Position deployment machine so that deployment book arm engages bridle.
4. Translate deployment machine down to unroll reflector sheet.
5. Position birdle to engage reflectors stretcher hook.
6. Position edge sealing head to engage both edges.
7. Translate deployment machine to move edge scales while it seals the edges.
8. Disengage edge sealer head.
9. Return to Step 3 and repeat.

See Figure 3.4-18 for detailed functional flow. Four hours per strip is available.

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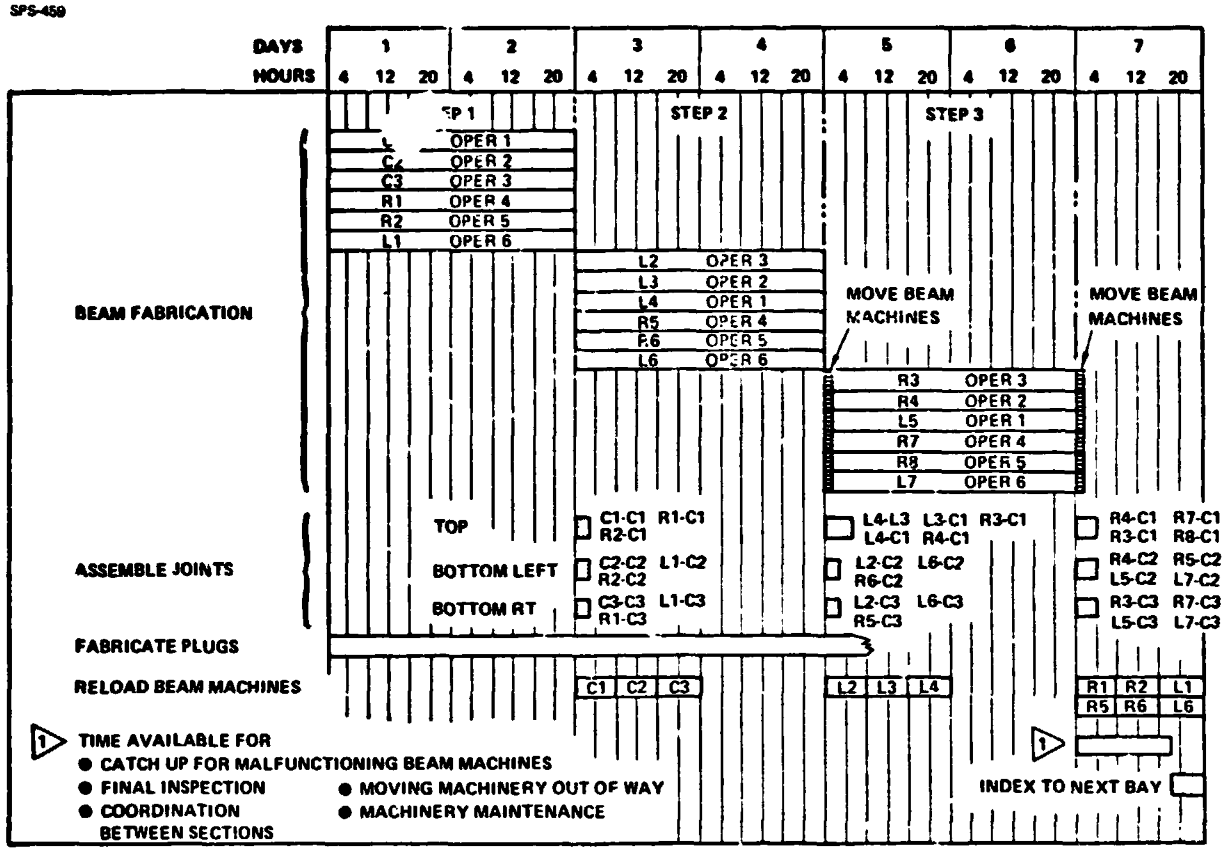

Figure 3.4-17 CR = 2.0 Photovoltaic Satellite Frame Fabrication and Assembly Timeline

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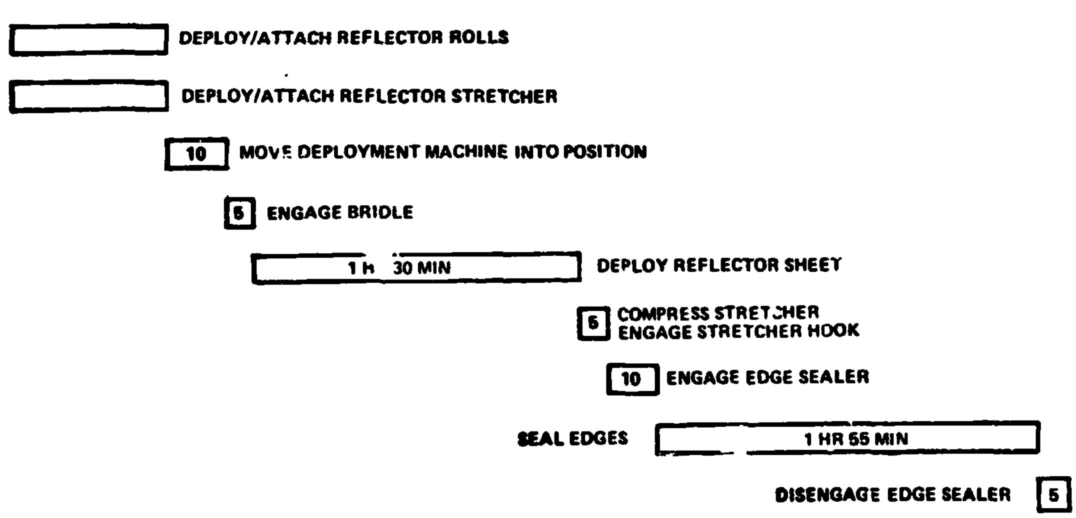

Figure 3.4-18 Reflector Deployment Function Flow

Alternatives

1. Use a separate gantry to deploy/attach reflector rolls and stretchers ahead of deployer/edge sealer.

2. Set aside some time to deploy the rolls and stretchers before going back to deploy the reflector (uses same gantry).

3. Carry the rolls and reflectors with deployment gantry and attach just prior to deployment

	Evaluation Criteria	Alternative 1	Alternative 2	Alternative 3
1.	Number of gantries	Most	Fewest	Fewest
2.	Deployment speed	Slowest	Fastest	Fastest
3.	Number of operators	Most	Fewest	Fewest
4.	Complexity of deployment machine	Least	Medium	Most
5.	Decoupled	Best	Medium	Least
6.	End Frame Clearance	Best	Worst	Worst

The choice is alternative 1 because of the complexity of trying to deploy the cannisters and tensioning devices using the same gantry as used to deploy the reflector.

Assumption # 17 - Allow 60 minutes each to install the rolls and stretchers and to move to next and to move to next position.

There are 33 strips to deploy per bay.

The component deployment gantry can finish its job within 33 hours.

The reflector deployment machine can begin work after the second roll is installed by the component deployment machine.

The detailed timeline for reflector installation operations shown in Figure 3.4-19. Rates are 7.2 in/min for reflector deployment and 5.65 in/min for edge attachment.

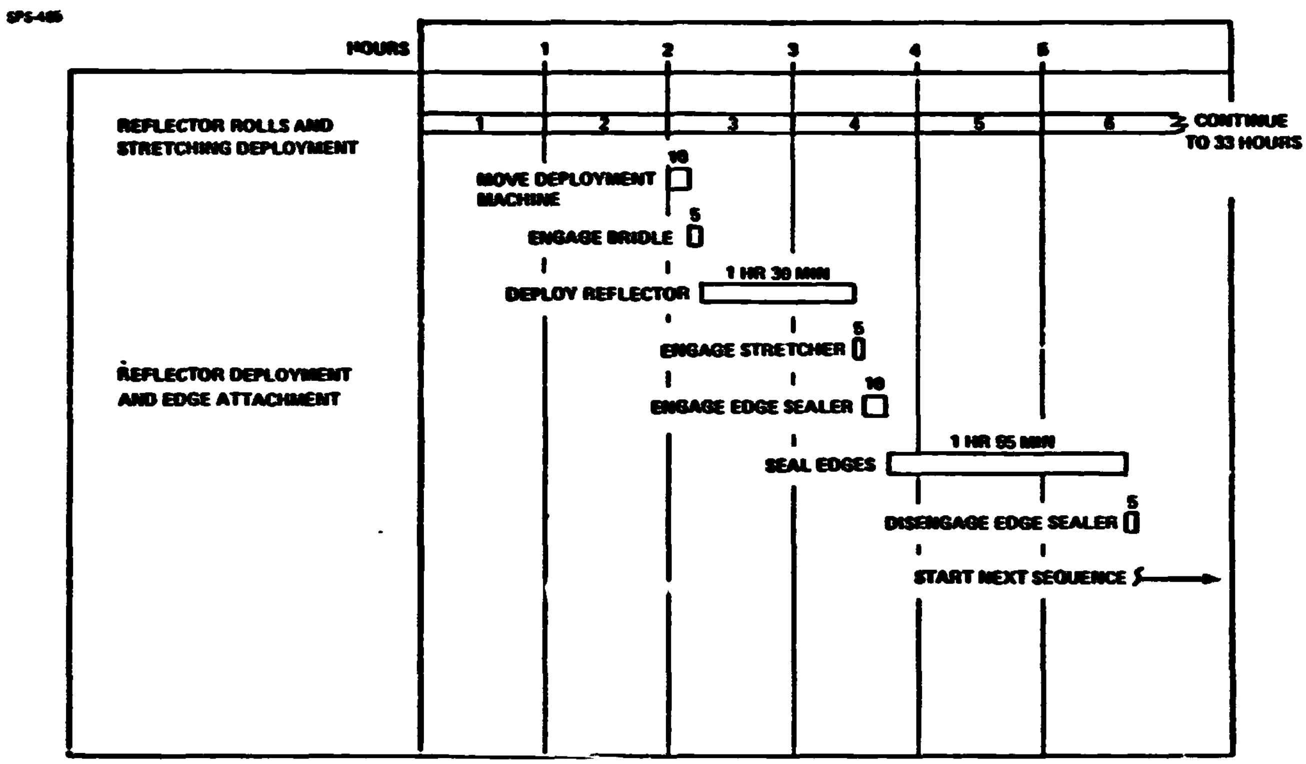

Figure 3.4-19 Reflector Installation

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Solar Cell Deployment

Assumption # 18 - Solar cell blankets 20 meters wide.

The functional flow and timelines for solar cell deployment activities are almost identical to the reflector functional flow and timeline.

Solar cell deployment rate = 7.2 m/min
Solar cell edge attach rate = 5.65 m/min

Bus and Switch Gear Installation–The functions to be performed include the following:

1. Deploy module bus
2. Deploy main bus
3. Deploy switch gear
4. Connect solar cells to module bus
5. Conncet module bus to switch gear
6. Connect switch gear to main bus
7. Connect main bus to main bus

The timeline for the required functions is shown in Figure 3.4-20.

3.4.1.3.1.5 Machinery Requirements

Beam Machines–The following numbers of 20 m beam machines will be required in each longitudinal section:

Section 1 - 11 beam machines
Section 2 - 12
Section 3 - 12
Section 4 - 12
Section 5 - 10

57 beam machines required.

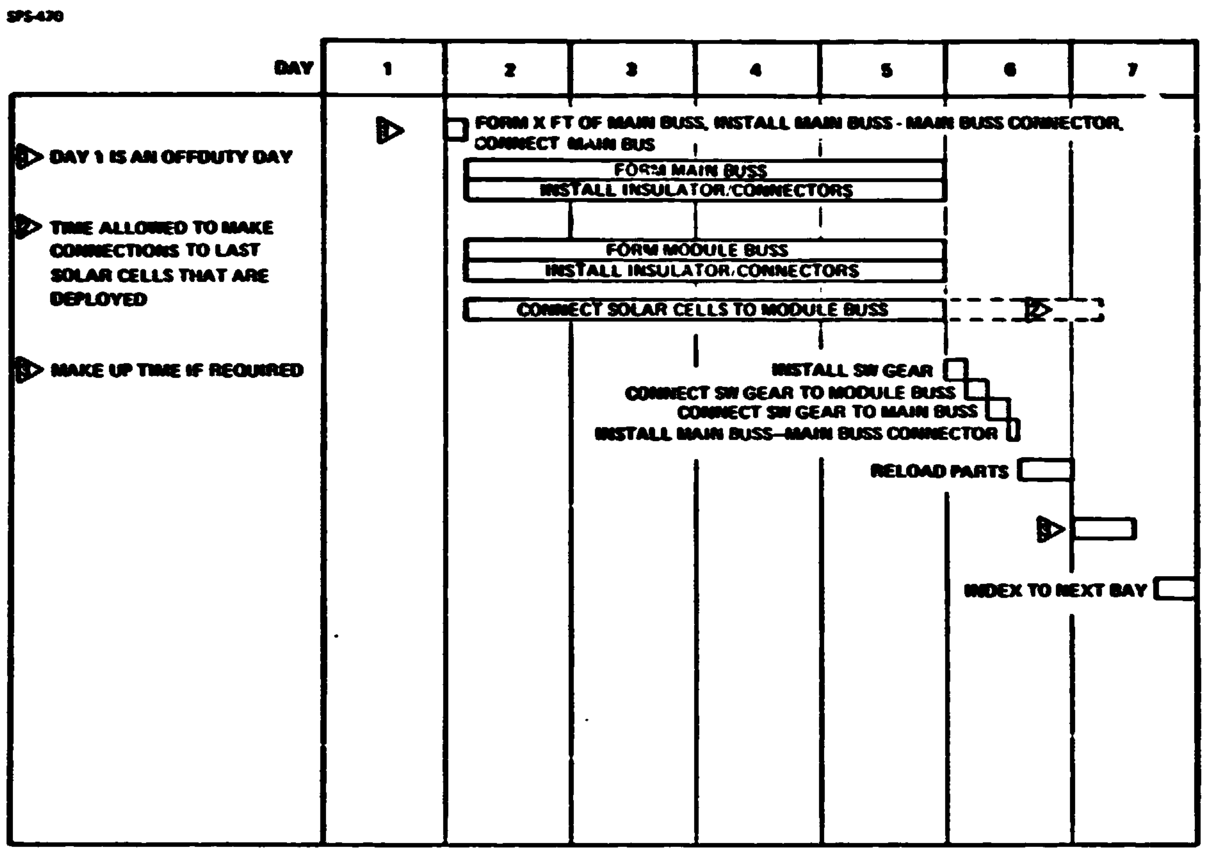

Figure 3.4-20 Electrical Power Collection System Installation

The C1/C2/C3 beam machines in each section will be fixed to the facility (15 total).

The L2/L3/L4 beam machines in each section will be mounted on retractable platforms (15 total).

The other beam machines (27 total) will have to be mounted on carriages such that they can be relocated.

Beam End Holder Machines—As each beam machine will remain fixed in place as it fabricates a beam, it will be necessary to attach the free end of the beam to a traveling beam and holder machine to restrain and guide the beam as it is being made. Therefore, 57 beam end holder devices will be required.

Beam Supports—Facility-mounted, retractable beam supports will be required every 200m along each beam (3 required each frame). Therefore, a total of 261 beam supports will be required.

Joint Assembly Machines—Joint assembly machines will be required at each of the ridge apexes and at mid span of the L3/L4 frame (see Figure 3.4-21). Figure 3.4-22 shows a concept for a joint assembly machine.

Component Deployment Machine—A gantry will be required to operate along each of the reflector ridges. This gantry will perform 3 major functions:

- Deploy solar cell blanket cannisters
- Deploy reflector roll cannisters
- Deploy tensioning devices

Figure 3.4-23 shows a concept for this machine. Eight of these machines will be required.

Solar Cell Deployment Machine—A solar cell deployment machine will be required that performs the following major functions.

- Deploy solar cell blanket across the trough
- Attach solar cell blanket to tensioning device
- Attach edges of adjacent solar cell strips

Figure 3.4-24 shows a concept for this machine. Five machines are required.

Reflector Deployment Machine—A machine will be required to deploy, tension, and edge-attach reflector rolls. A concept for this machine is shown in Figure 3.4-25. Eight of these machines will be required.

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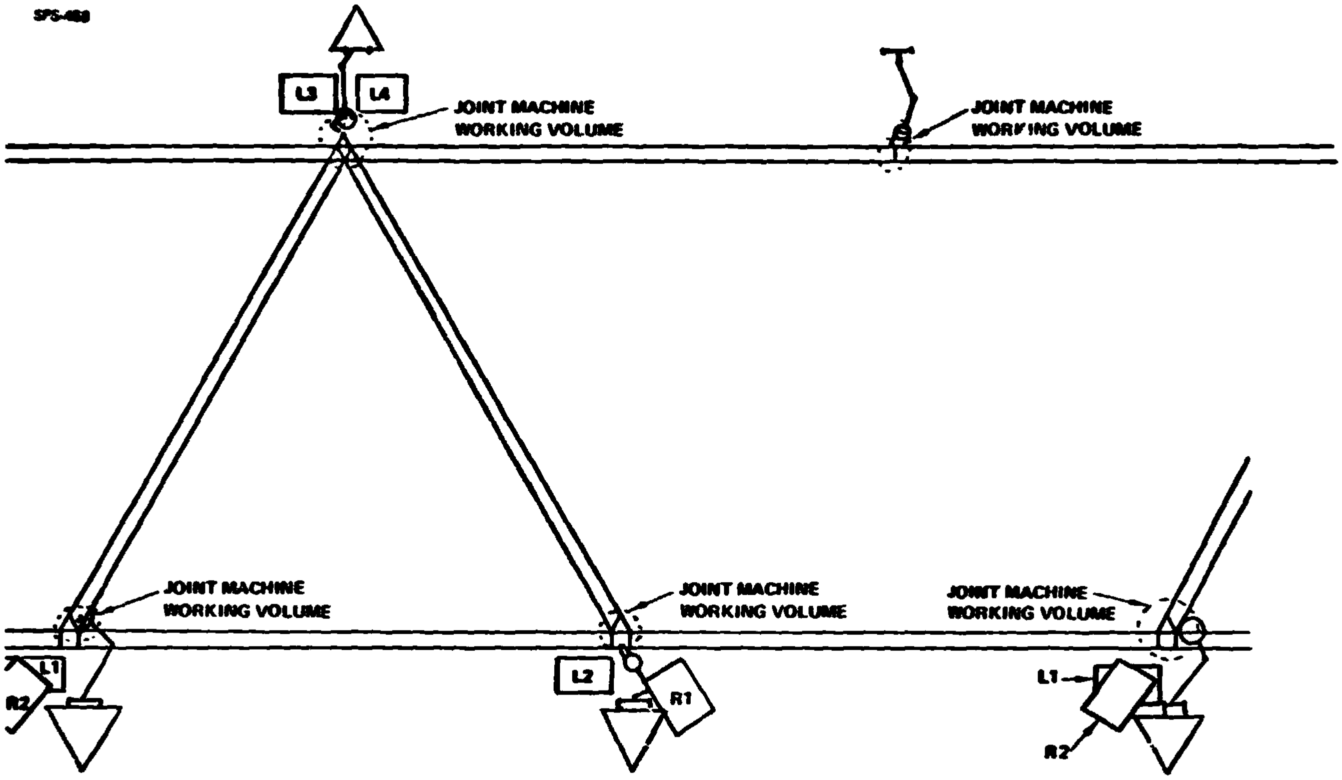

Figure 3.4-21 Joint Assembly Machine Locations

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REQUIREMENTS

- MOVES ALONG LONGITUDINAL MEMBERS
- MUST BE ABLE TO MOVE BETWEEN RETRACTED BEAM MACHINES
- MUST BE ABLE TO RELOCATE TO JOINT LOCATION IN 15 MINUTES
- MUST BE CAPABLE OF MANIPULATING AND HOLDING FRAMES PRIOR TO JOINING
- MUST BE ABLE TO ALIGN FRAMES
- MUST BE ABLE TO PRECISELY LOCATE FASTING DEVICE AT JOINT POINTS
- MUST BE ABLE TO FASTEN FRAMES
- MUST BE ABLE TO RELOCATE TO NEXT FASTENING POINT IN 5 MINUTES
- MUST BE ABLE TO INSPECT FASTENED JOINT
- MUST BE ABLE TO WORK AROUND STRUCTURE WITHOUT DAMAGING FRAMES
- MUST BE A MANNED MANIPULATOR
- TRANSPORT PLUG STRUCTURES

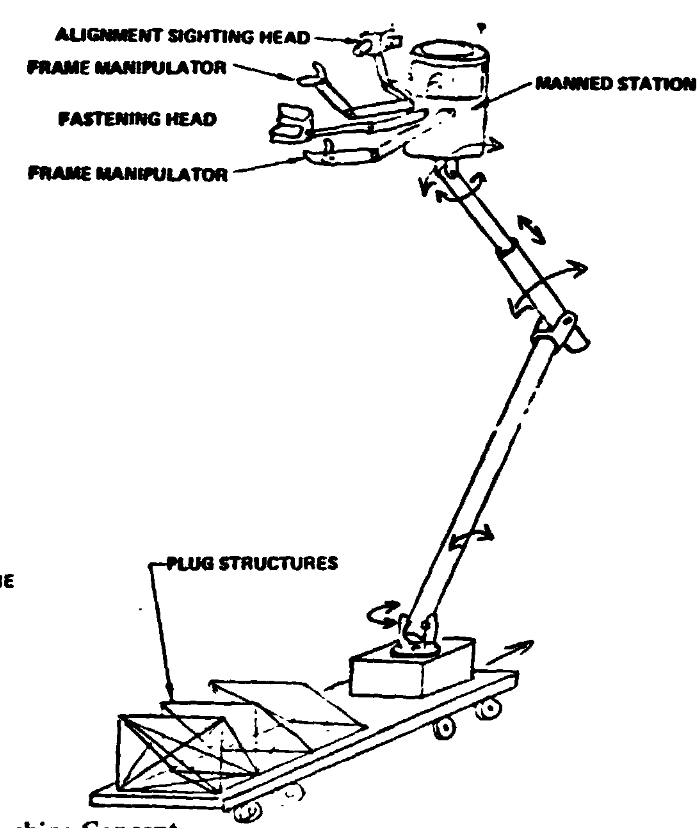

Figure 3.4-22 Joint Machine Concept

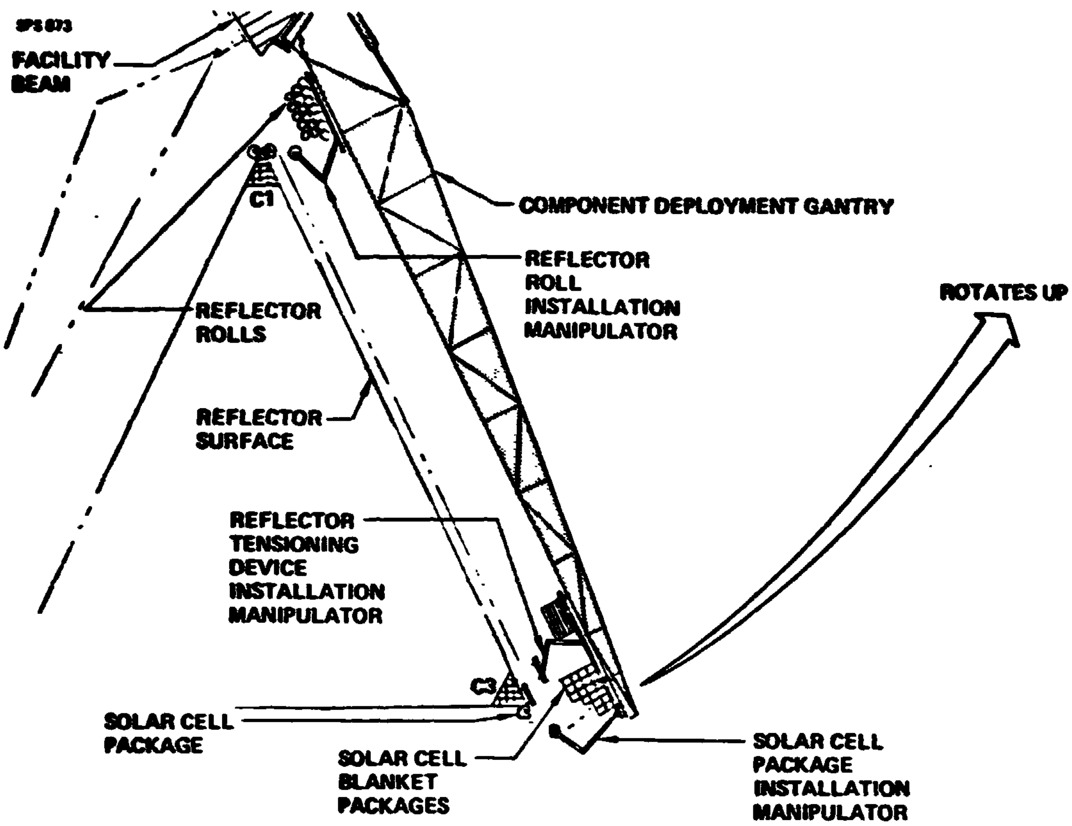

Figure 3.4-23 Component Deployment Machine

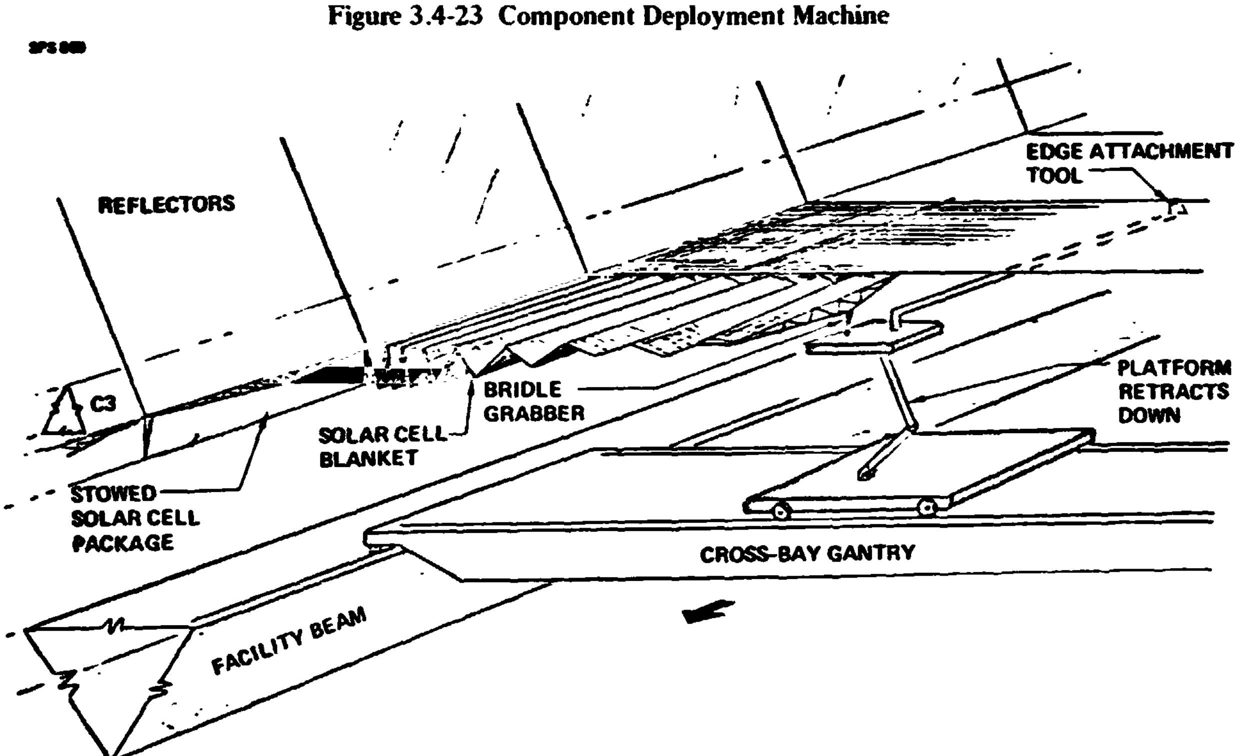

Figure 3.4-24 Solar Cell Deployment Machine

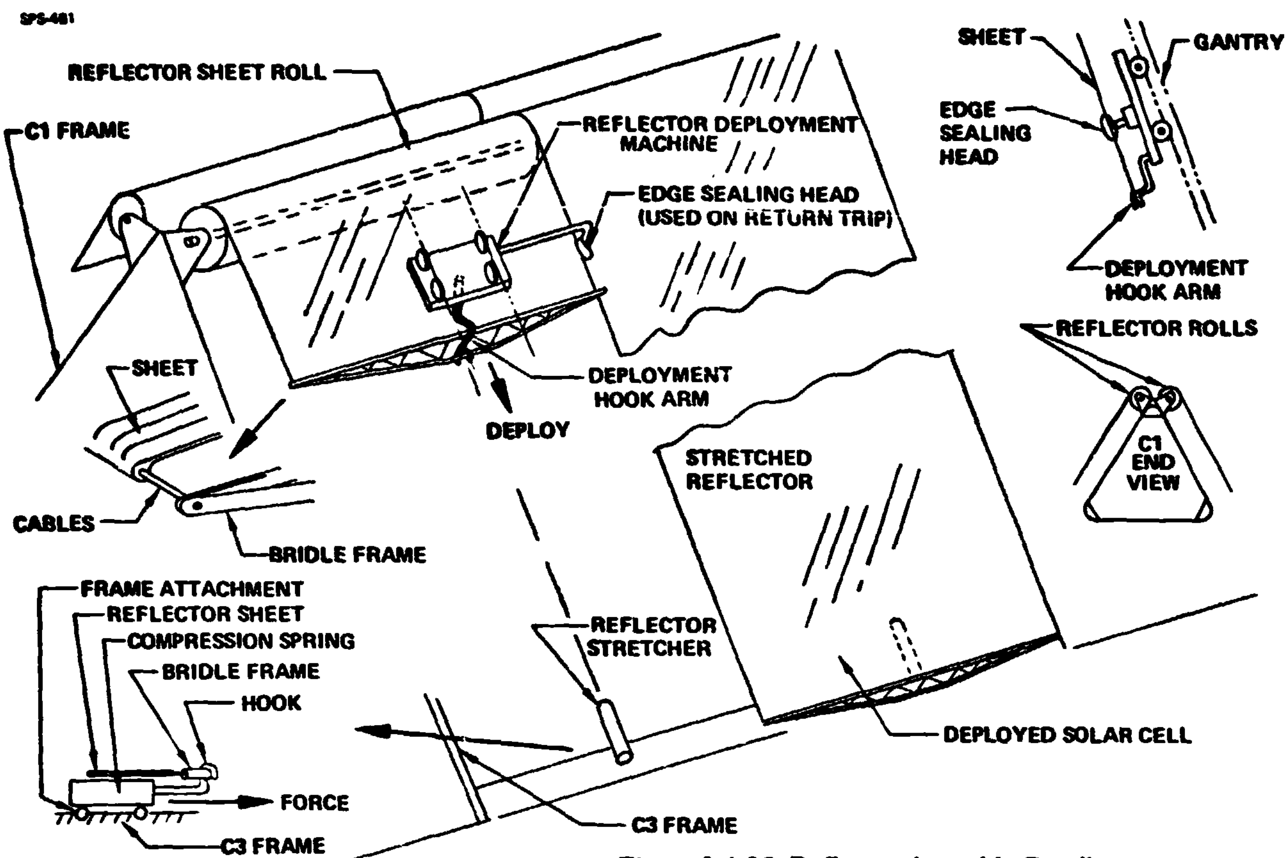

Figure 3.4-25 Reflector Assembly Details

Bus/Switch Gear Installation Machine–At the bottom corners of each trough, a machine will be required to install the busses and switch gear. Figure 3.4-26 shows a concept for this machine. Eight of these machines will be required.

Machine Requirements Summary–The quantity of each type of machine and the required operating rates are summarized in Figure 3.4-27.

3.4.1.3.1.6 **Manning Requirements**–The manning required for the GEO construction base is summarized in Figure 3.4-28.

3.4.1.3.1.7 **Facility**–The concept for the construction facility is shown in Figure 3.4-29. This facility is as wide as the total satellite and as long as two structural bays.

In this concept the structure is fabricated first in Facility Bay A, see Figure 3.4-30. After the one bay of the frame is assembled, the frame assembly is indexed in to Facility Bay B wherein the solar cells, reflectors, and power collection system will be installed, see Figures 3.4-31, -32, and -33.

3.4.1.3.2 Photovoltaic Satellite (CR=2) LEO Base Construction Analysis

3.4.1.3.2.1 Introduction

This analysis was directed at determining how photovoltaic satellite modules can be constructed in LEO, transported to GEO and assembled into a complete satellite. The construction concept will be described in terms of the numbers and types of construction machinery, the crew required at each location, LEO or GEO, and construction facility concept.

The results of this analysis are compared to the results obtained in the Photovoltaic Satellite GEO Base Construction Analysis presented earlier.

3.4.1.3.2.2 Overview

The analysis in this report will use data derived in the GEO construction analysis.

Reference Satellite Configuration

The photovoltaic satellite is to be constructed using modules that are 1/16 the size of the total satellite. Each module is constructed at the LEO base. These moduels are then transported to GEO where they will be mated together to make up the complete satellite. Figure 3.4-34 shows the completed satellite configuration. Figure 3.4-35 shows a typical 1/16 size module. Note that the module dimensions are larger than those used in the GEO construction analysis. This size increase is required to compensate for radiation degradation incurred by the transfer; see section 3.5.

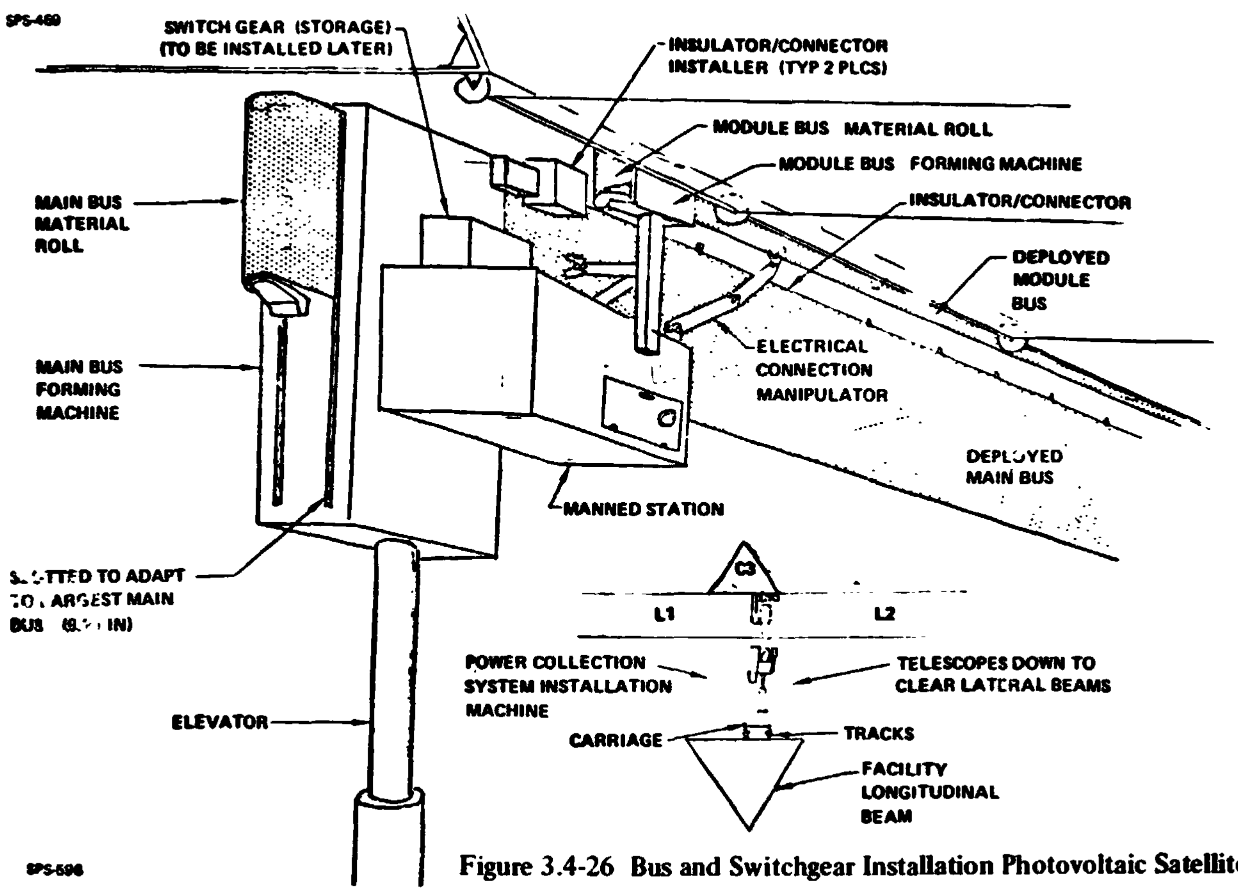

Figure 3.4-26 Bus and Switchgear Installation Photovoltaic Satellite

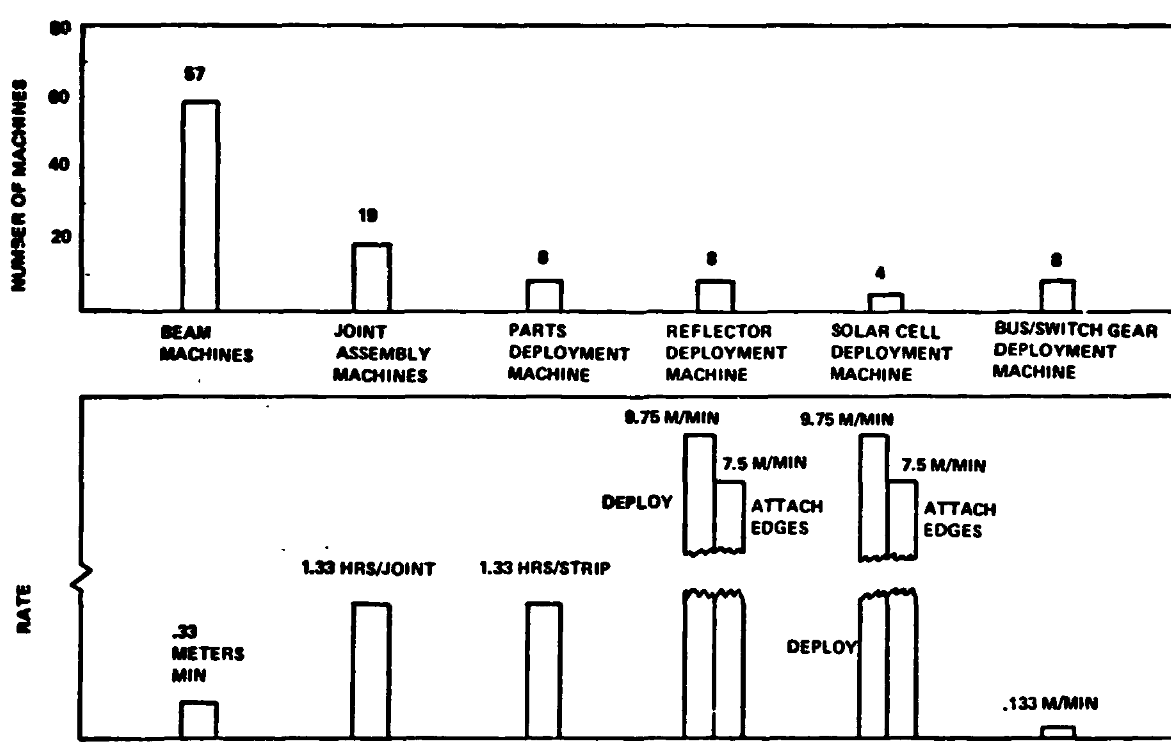

Figure 3.4-27 Construction Machine and Rate Comparison Photovoltaic Satellite (CR = 2)

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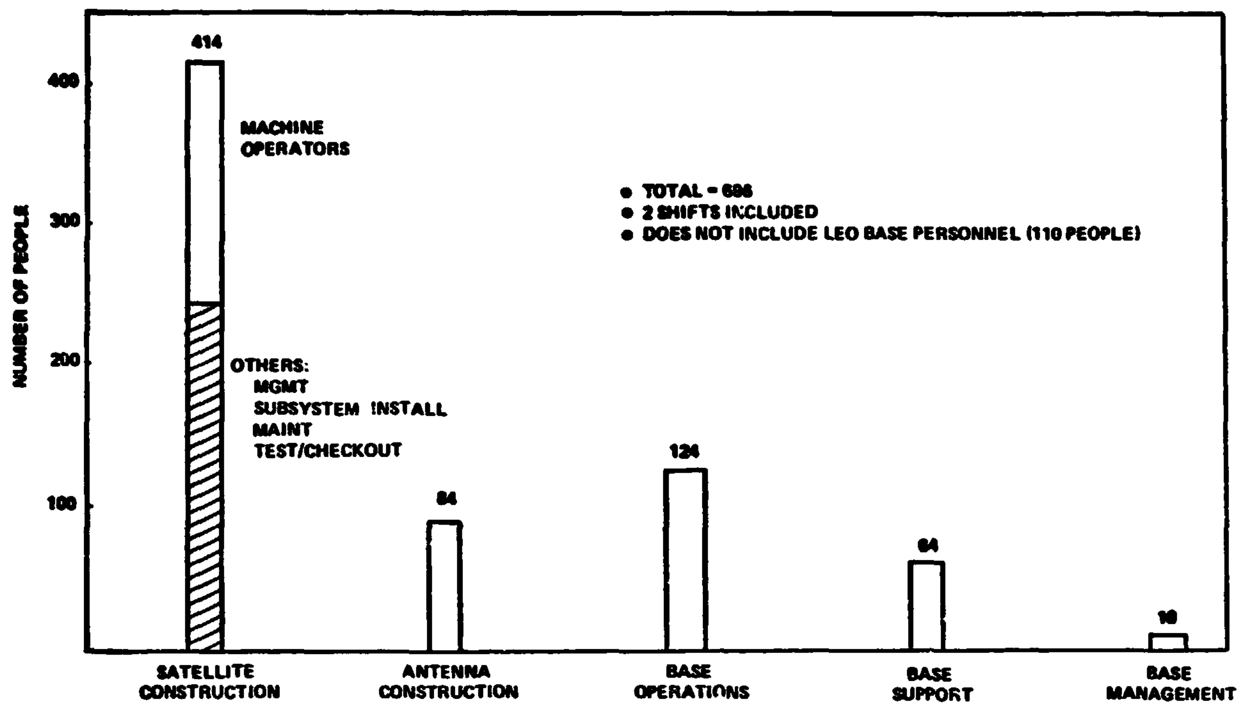

Figure 3.4-28 GEO Construction Manpower Summary Photovoltaic Satellite (CR = 2)

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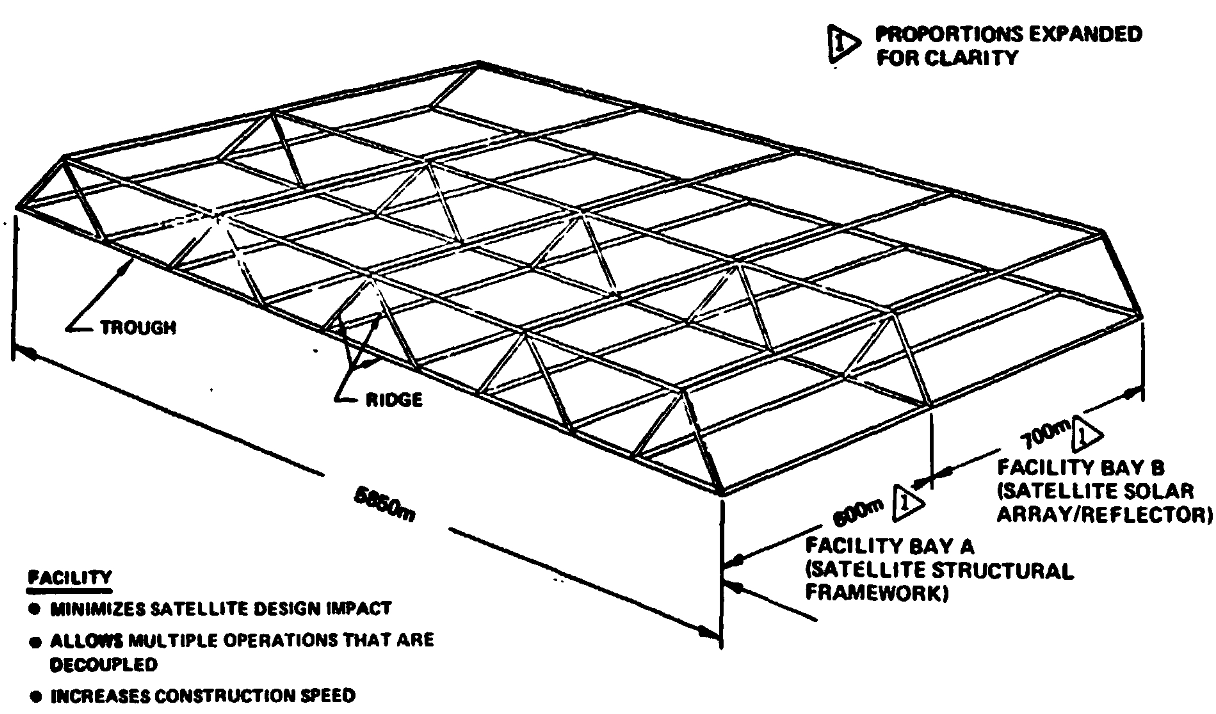

Figure 3.4-29 Photovoltaic Satellite Construction Facility

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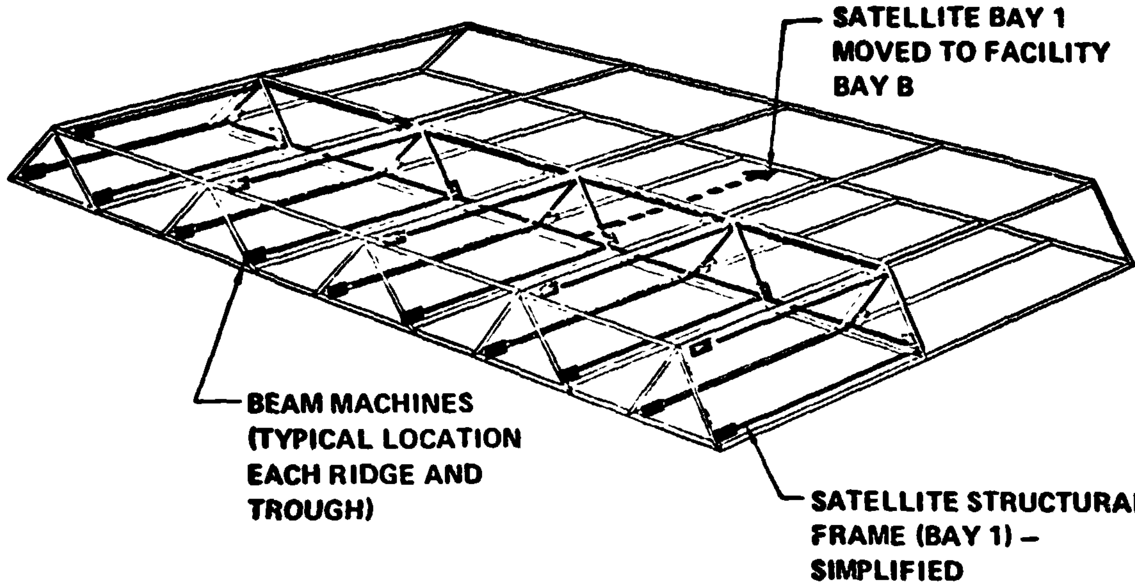

Figure 3.4-30 Satellite Bay 1 Framework Facility Bay A

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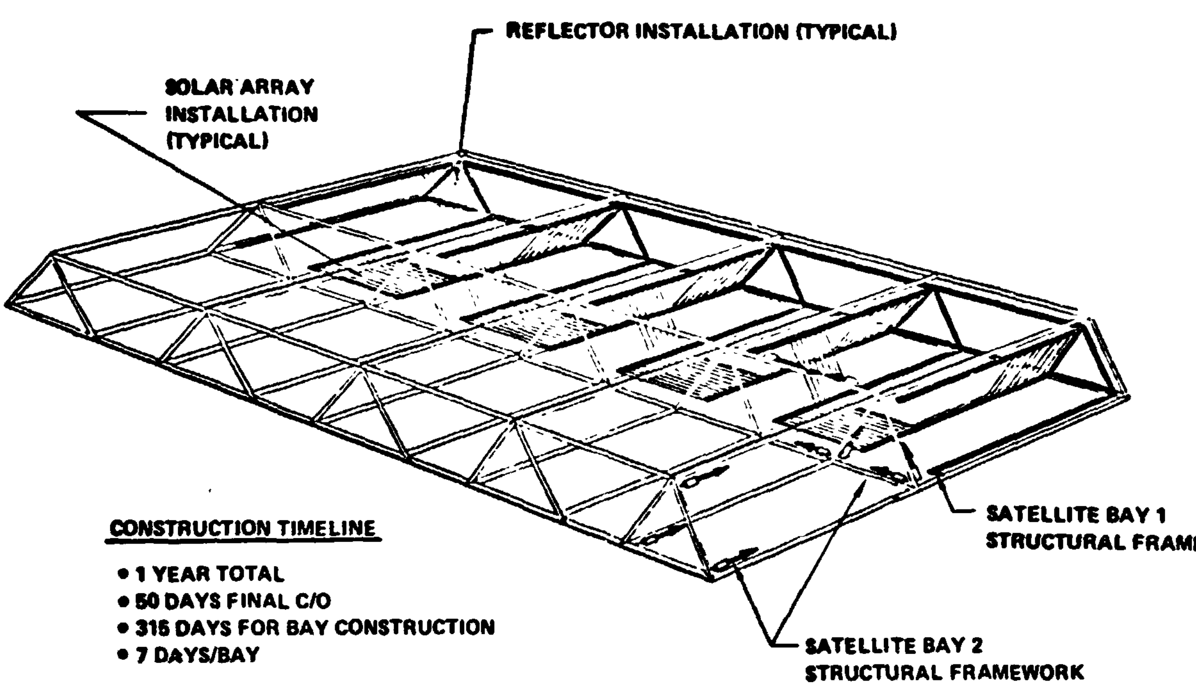

Figure 3.4-31 Satellite Solar Array and Reflector Installation Facility Bay B Operations

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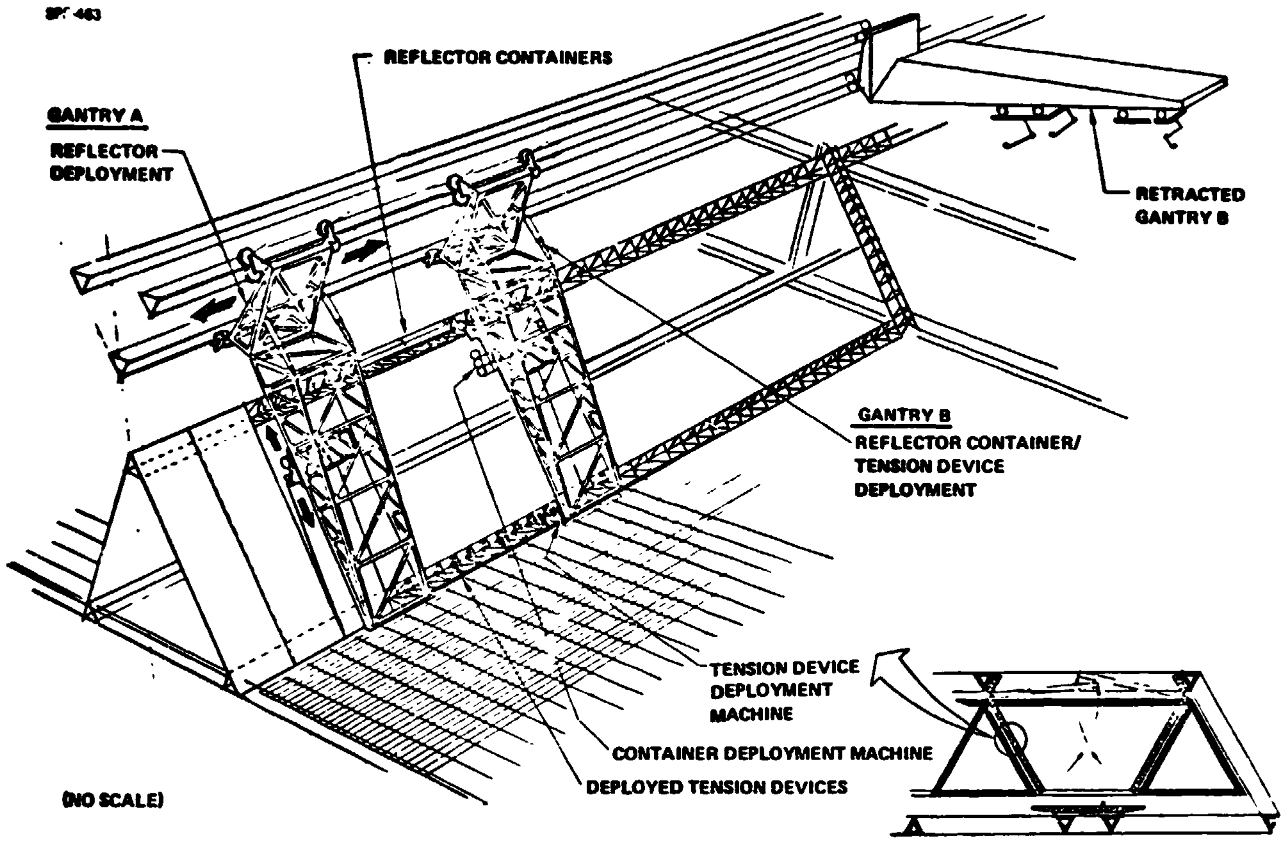

Figure 3.4-32 Reflector Installation

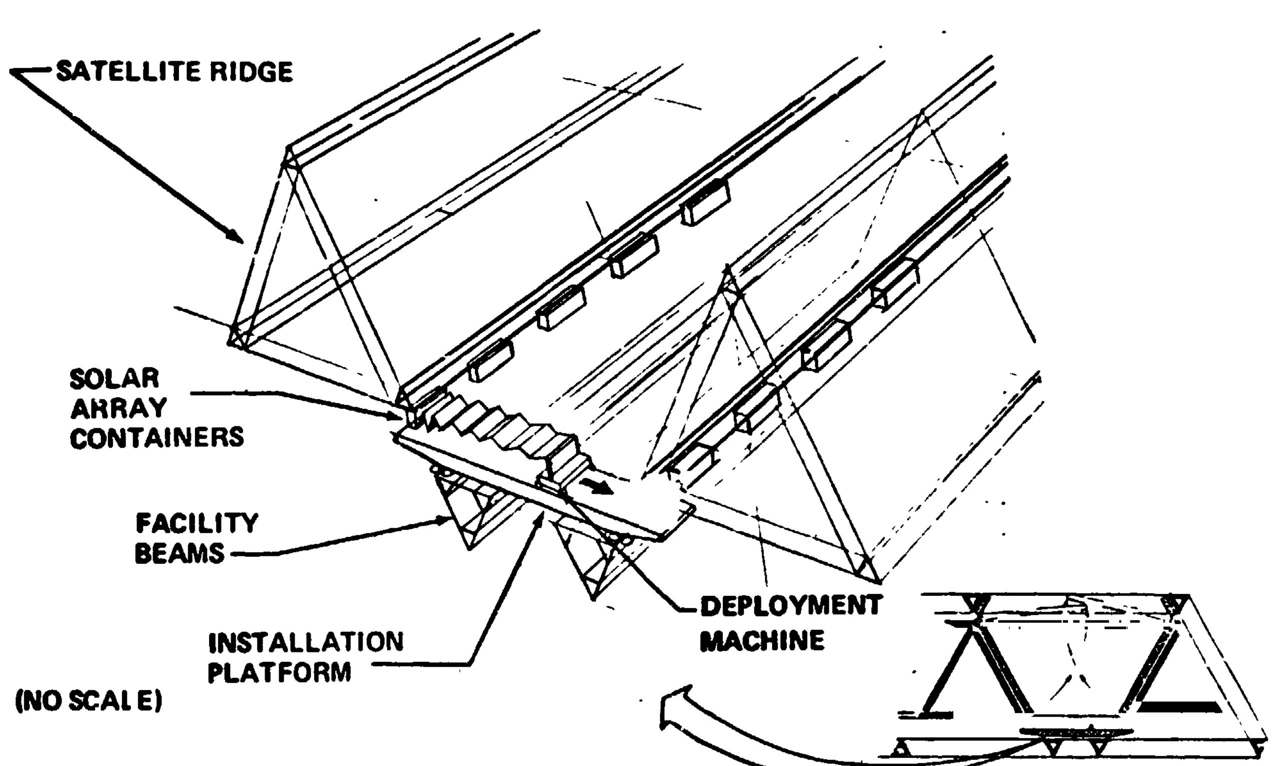

Figure 3.4-33 Solar Array Installation

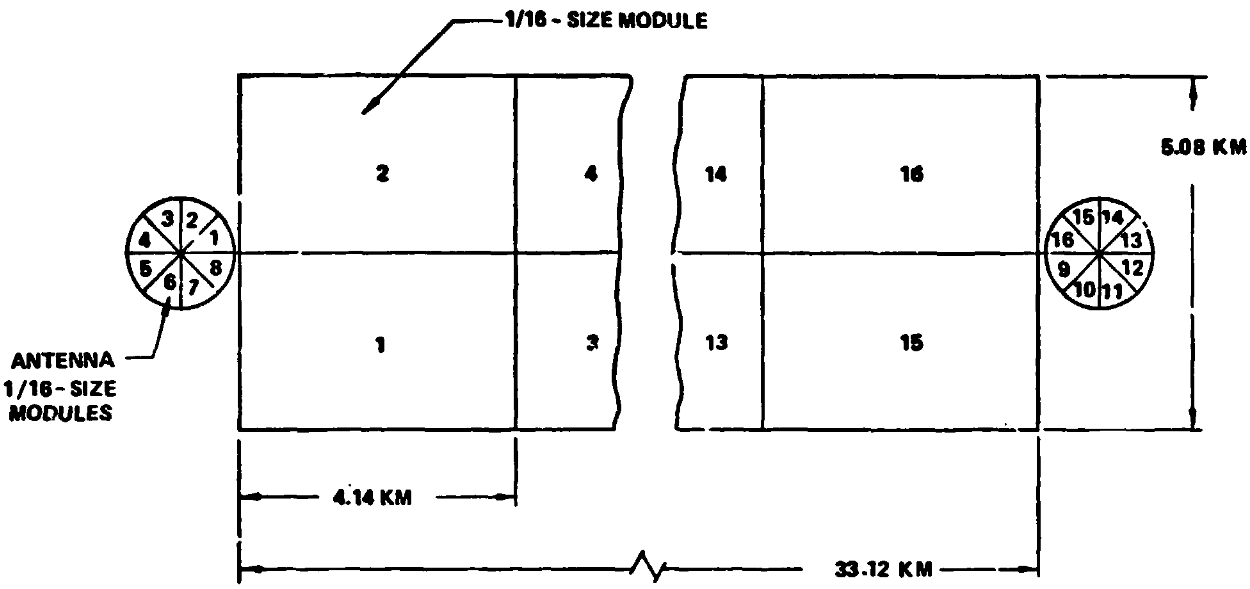

Figure 3.4-34 CR = 2.0 Photovoltaic Satellite Configuration

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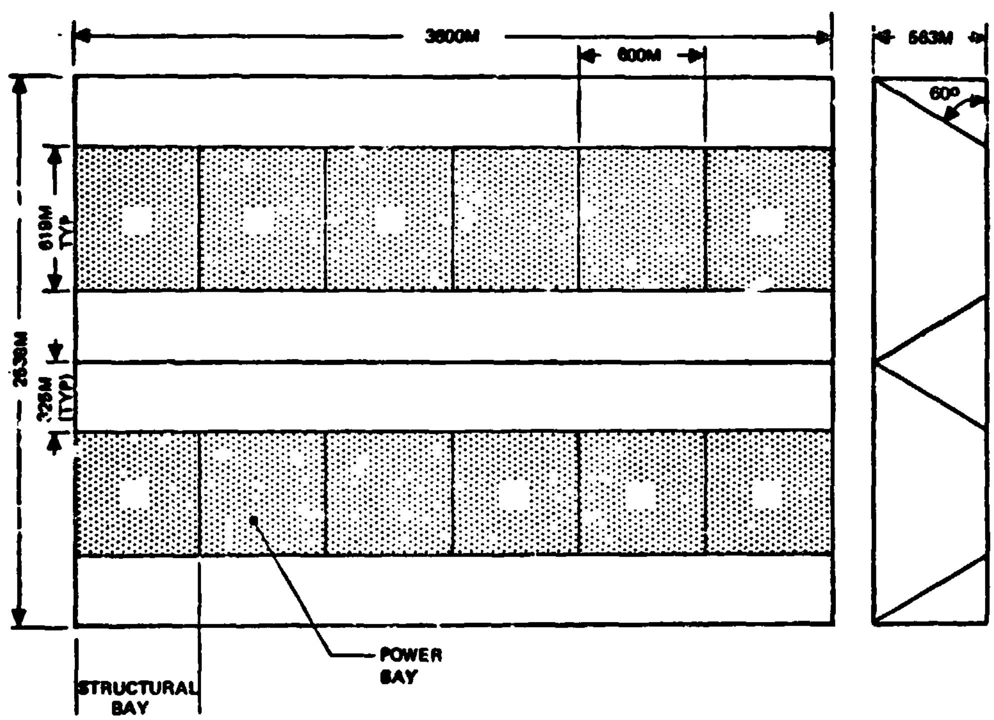

Figure 3.4-35 CR = 2.0 Photovoltaic Satellite 1/16 - Size Module Configuration

Top Level Timeline Analysis

Assumption #1 - Total construction time is 365 days at both LEO and GEO.

Assumption #2 - Allow 30 days for final integration and checkout at GEO

Assumption #3 - Allow 20 days to fabricate end structures and miscellaneous components.

345 days are available at LEO to build 16 modules, or 21.56 days per module.

Assumption #4 - Allow 1.56 days per module to absorb delays, final module checkout, removal from facility, etc.

This leaves 20 days per module actual construction time available.

A total of 335 days are available at GEO for mating operation. This averages 20.94 days per module.

Figure 3.4-36 shows the top level timeline that takes these assumptions and derived times into account. As Figure 3.4-36 shows, even though 20.94 days have been allotted for GEO assembly, for Module #16, there are only about 13 days available due to having allotted 30 days for final test and checkout.

3.4.1.3.2.3 Satellite Module Construction

LEO Construction Timeline Analysis

Each module consists of six bays 655m long. In the previous analysis it took 7 days to make each of the 655m long bays. Hence, it would take (6 bays) (7 days/bay) = 42 days to make the 6 bays using the same number of machines/section and the machine rates derived in the previous analysis.

As stated above only 20 days are available to construct 6 bays. Thus, there is only half the time per bay available as in the GEO case. To get the assembly work done in half the time, there are three alternatives:

1. Use more machines

2. Operate the machines at faster rates

3. Combination of the above

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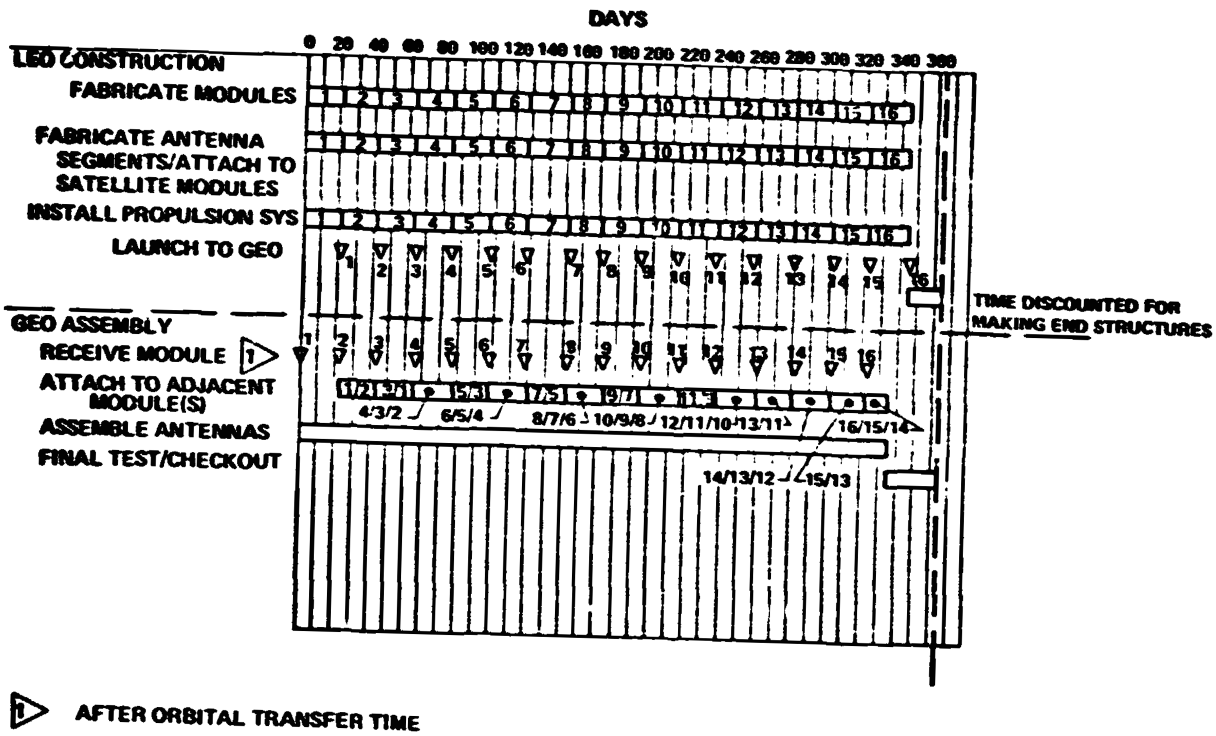

Figure 3.4-36 Photovoltaic Satellite Construction Top Level Timeline

In order to make a valid comparison to the previous analysis, it was chosen to keep the machine rates constant. Therefore, alternative number 2, increase the number of machines, was the choice.

In Bay A, the frame fabrication area, the time critical assembly operation is the beam fabrication. There is plenty of time available for the joint assembly operations. To make the structure in each bay section twice as fast as before, it will be necessary to use dedicated beam machines for each frame instead of moving beam machines to different locations to make 2 or 3 different frames. It will take 2 days to make the frame for each bay; 12 days total to make the frame for a satellite module.

To make the 2-trough wide satellite module, 41 beam machines, 7 joint assembly machines and 48 operators/shift are required.

In Bay B, the reflector/solar cell/busbar deployment bay, the time critical operations are (1) the reflector strip deployment by Gantry A, (2) the solar cell deployment by the solar cell deployment machine and (3) the bus installation. The deployment of solar cell and reflector components by Gantry B can be easily accomplished in enough time to accommodate a higher production rate (it only takes 33 hours/bay to distribute the components).

In order to get the critical Bay B operations performed at twice the rate as before, it will be necessary to increase the number of machines operating in each section. It will require Reflector/Solar Cell Parts Deployment Machines (Gantry B), 8 Reflector Deployment Machines, 4 Solar Cell Deployment Machines and 8 Bus/Switch Gear Installation Machines.

The timeline for construction of a staellite module is shown in Figure 3.4-37.

The machine and operator requirements are summarized in Table 3.4-5.

The LEO construction facility required to make the satellite module is shown in Figure 3.4-38.

GEO Assembly Timeline Analysis

The satellite modules arrive at 20 day intervals.

Assumption #5 - Allow 1 day to dock incoming module to previously attached modules.

Assumption #6 - Allow 1 day to attach end frames to adjacent module after docking.

Figure 3.4-39 shows a facility that will be flown into position on the satellite module. This facility spans the 2 linear trough sections and is one 655m bay in length.

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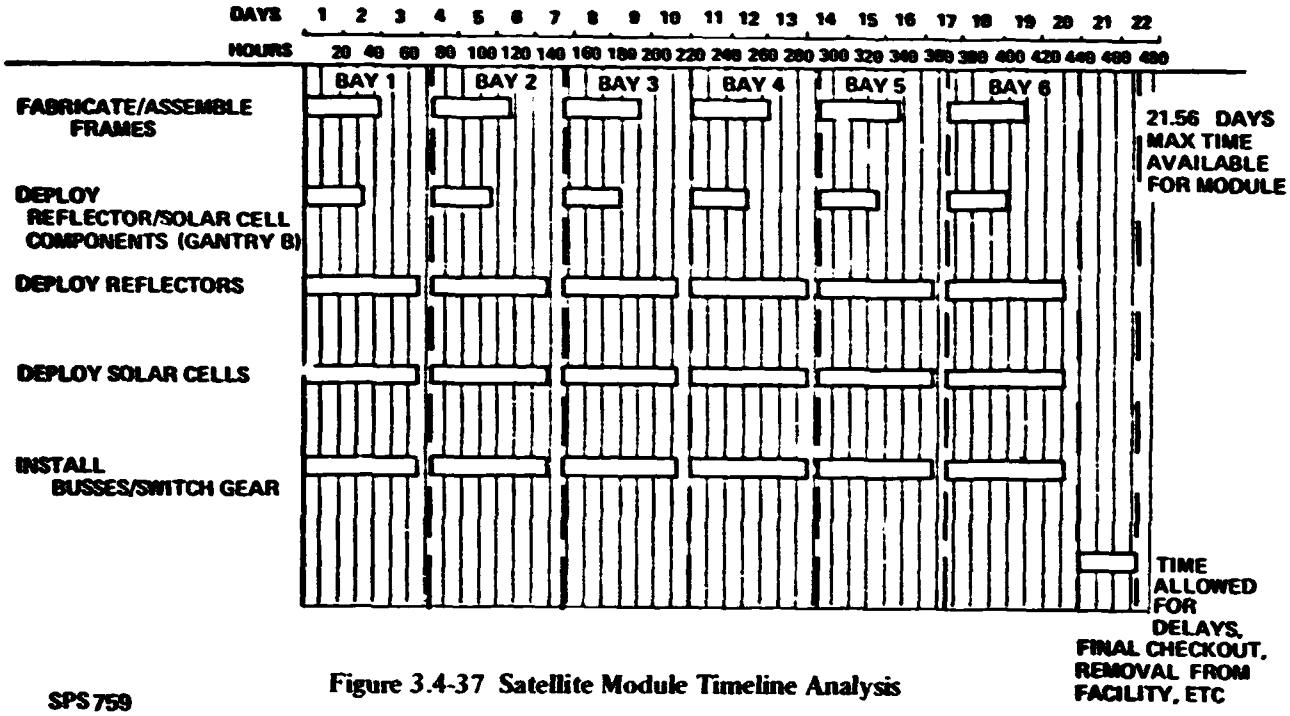

Figure 3.4-37 Satellite Module Timeline Analysis

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Table 3.4-5 LEO Construction Facility Construction Machinery Summary

BEAM MACHINES

- 41 MACHINES REQUIRED
- 41 OPERATORS/SHIFT
- .25 M/MIN FAB RATE x (1.33) = .33 M/MIN

JOINT ASSEMBLY MACHINES

- 7 MACHINES REQUIRED
- 7 OPERATORS/SHIFT (LOCATED ON THE MACHINE)
- 1 HR/JOINT RATE x (1.33) = 1.33 HR/JOINT

REFLECTOR/SOLAR CELL PARTS DEPLOYMENT MACHINES (GANTRY B)

- 4 MACHINES REQUIRED
- 8 OPERATORS/SHIFT (LOCATED REMOTE FROM MACHINE ?)
- 1 HR/STRIP RATE x (1.33) = 1.33 HRS/STRIP RATE

REFLECTOR DEPLOYMENT MACHINE (GANTRY A)

- 8 MACHINES REQUIRED
- 8 OPERATORS/SHIFT (LOCATED REMOTE FROM MACHINE ?)
- 7.2 M/MIN DEPLOYMENT RATE x (1.33) = 9.57 M/MIN
- 5.6 M/MIN EDGE ATTACH RATE x (1.33) = 7.5 M/MIN

SOLAR CELL DEPLOYMENT MACHINE

- 4 MACHINES REQUIRED
- 4 OPERATORS/SHIFT (LOCATED REMOTE FROM MACHINE ?)
- 7.2 M/MIN DEPLOYMENT RATE x (1.33) = 9.57 M/MIN
- 5.6 M/MIN EDGE ATTACH RATE x (1.33) = 7.5 M/MIN

BUS/SWITCH GEAR INSTALLATION MACHINE

- 8 MACHINES REQUIRED
- 8 OPERATORS/SHIFT (LOCATED ON THE MACHINE)
- .1 M/MIN RATE x (1.33) = 0.133 M/MIN RATE

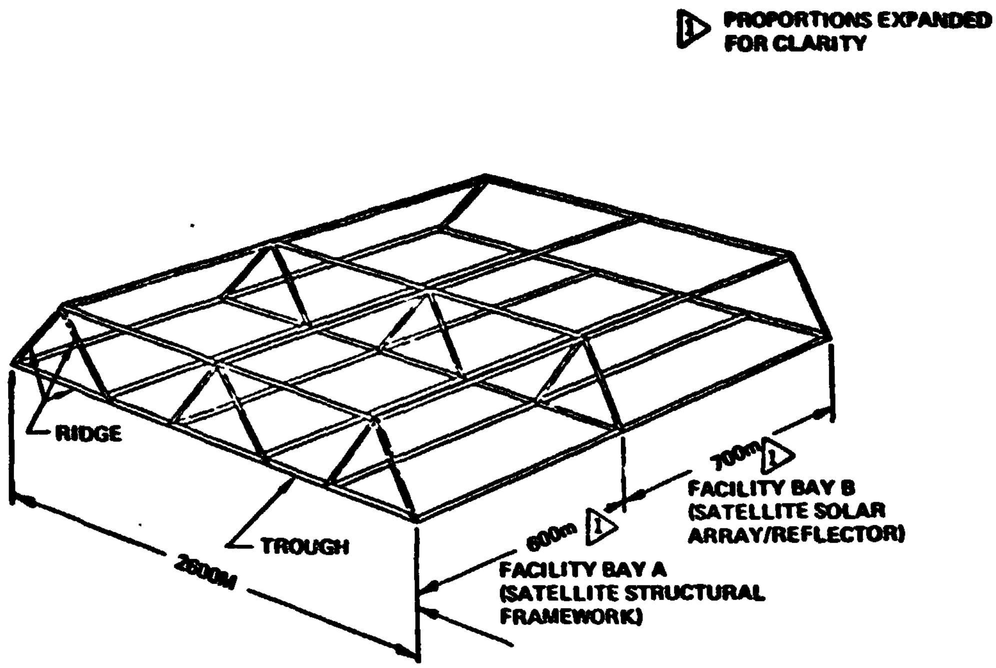

Figure 3.4-38 CR = 2.0 Photovoltaic Satellite LEO Base Construction Facility

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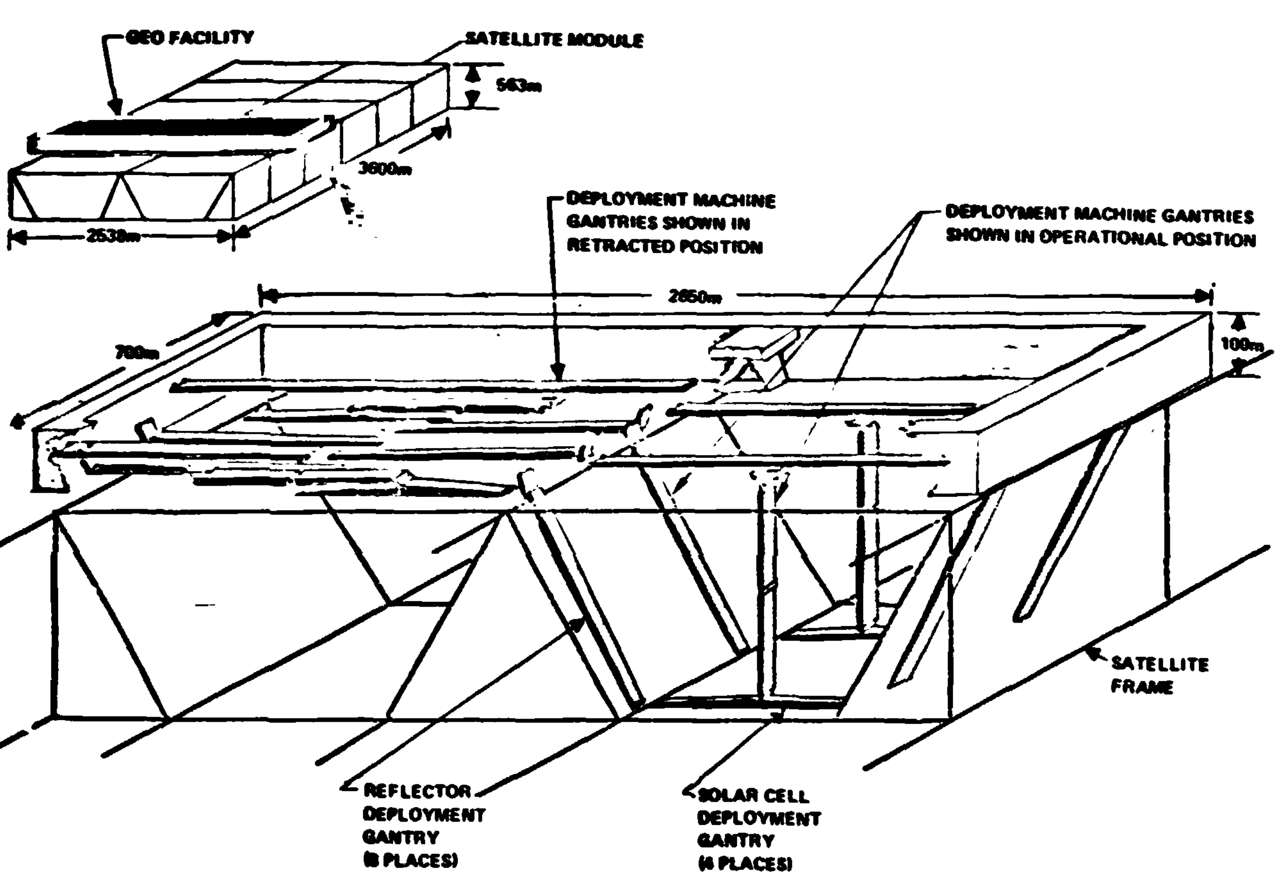

Figure 3.4-39 GEO Deployment Facility/LEO Construction Photovoltaic Satellite (CR = 2)

Assumption #7 - Allow 1/2 day to move facility into place.

17.5 days are therefore to deploy solar cells/reflectors and to index the next bay. (385 hrs.).

Figure 3.4-40 shows the locations where the reflectors and solar cells require deployment. 3 indexing moves of the facility are indicated.

Assumption #8 - Allow 2.5 hrs to move the facility to next bay.

377.5 hrs are then available to deploy reflectors/solar cells, 94 hrs/bay.

In the LEO facility, it was estimated that it takes 6 days to deploy a bay's reflectors and solar cells. To deploy the same quantity of reflectors and solar cells using the same machine rates derived previously, two deploying machines must operate simultaneously on each working surface. Therefore, the facility shown will require the following machines:

8 Reflector deployment machines
4 Solar cell deployment machines

To attach the frames, it will require 4 joint assembly machines.

32 operators are needed to run the machines (2 shifts).

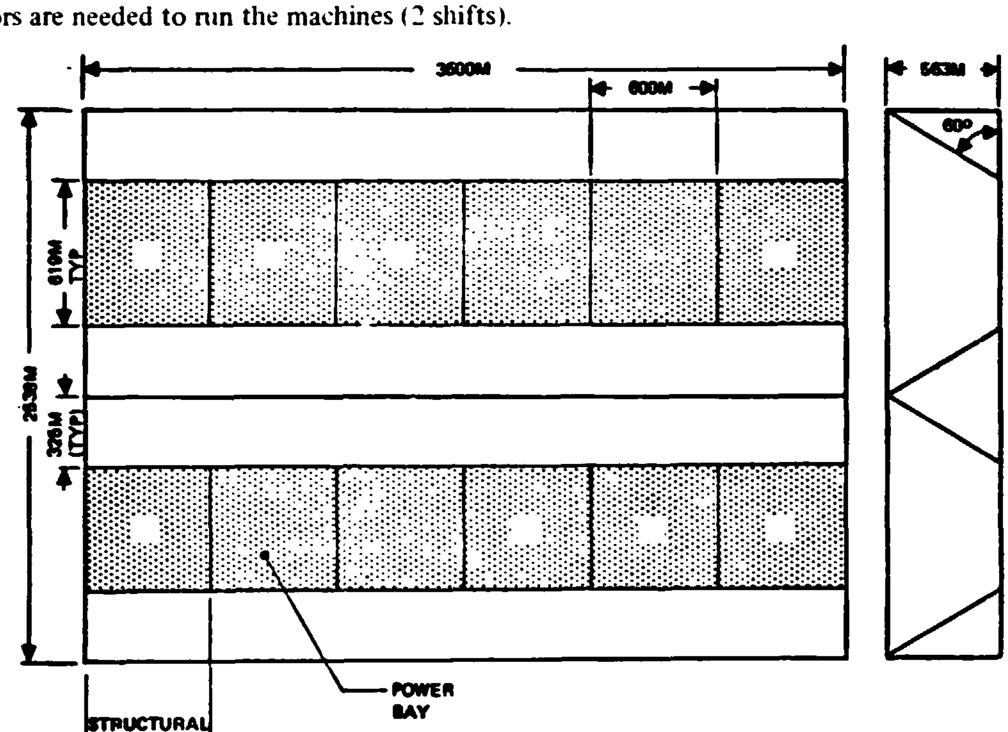

Figure 3.4-40 CR = 2.0 Photovoltaic Satellite 1/16 - Size Module Configuration

3.4.1.3.2.4 Comparison of LEO vs GEO Construction

Table 3.4-6 summarizes the major differences between the LEO and GEO photovoltaic satellite construction. Table 3.4-7 summarizes the total manpower requirements for both the LEO and GEO construction base alternatives.

3.4.1.3.3 Thermal Engine Satellite GEO Base Construction Analysis

3.4.1.3.3.1 Overview

In this section, the reference satellite configuration is described and the top level construction time-line is derived.

Table 3.4-6

LEO vs GEO Base Construction of the CR=2 Photovoltaic Satellite

Construction Differences Summary

GEO Construction Base	LEO Construction Base
● 5 X 2.9 KM size	● 5 X 4.1 KM size
● One construction facility required	● Two construction facilities required
● Made as one contiguous assembly	● Made up of 16 modules that are assembled at GEO
● Machinery required	● Machinery required
● 57 Beam machines	● 41 Beam machines
● 30 operators/2 shift	● 41 operators/shift
● 19 Joint assembly machines	● 11 Joint assembly machines
● 19 operators/shift	● 11 operators/shift
● 8 parts deployment machines	● 4 parts deployment machines
● 16 operators/shift	● 8 operators/shift
● 8 reflector deployment machines	● 16 reflector deployment machines
● 8 operators/shift	● 16 operators/shift
● 4 solar cell deployment machines	● 8 solar cell deployment machines
● 4 operators/shift	● operators/shift
● 8 bus/switch gear installation machines	● 8 bus/switch gear installation machines
● 8 operators/shift	● 8 operators/shift
● Indexing Speed 1.1 m/min	● Indexing Speed 1.5 m/min

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Table 3.4-7 Manpower Summary CR = 20 Photovoltaic Satellite

	LEO construction		GEO construction	
	LEO base	GEO base	LEO base	GEO base
Base management	(10)	(5)	(5)	(10)
Satellite construction	(302)	(135)	(0)	(414)
Management	72	22	—	86
Frame	100	8	—	98
Power generation	52	24	—	72
Subsystems	12	15	—	24
Maintenance	28	28	—	56
Test and checkout	38	38	—	78
Antenna construction	(84)	(54)	—	(84)
Base operations	(138)	(68)	(82)	(124)
Management	12	8	8	12
Data Processing	6	4	4	6
Base maintenance	42	19	19	42
Transportation	24	10	24	10
Materials handling	46	19	19	46
Communications	8	8	8	8
Base support	(64)	(37)	(23)	(64)
Management	7	5	5	7
Utilities	14	8	2	14
Hotel/food service	24	12	4	24
Medical/dental	13	6	6	13
Safety	2	2	2	2
Chaplain	2	2	2	2
control	2	2	2	2
Totals	598	299	110	696
Total	897		806	

Parabolic Dish Concept

The original thermal engine satellite configuration analyzed for construction was the 4-module parabolic dish configuration shown in Figure 3.4-41.

A construction approach was developed to a significant level of detail. Several alternative construction facility concepts were evolved but no straight forward construction method was found.

Due to complexities encountered, the satellite configuration was reevaluated taking constructability into account, resulting in the configuration shown in Figures 3.4-42 and 3.4-43. Slight performance penalties were incurred (see section 3.3.1).

Reference Satellite Configuration

The configuration of the thermal engine satellite selected for construction analysis is shown in Figure 3.4-42 and 3.4-43. This satellite is composed of 16 modules. Each module has 4 thermal engines. The satellite module was described in more detail in section 3.3.

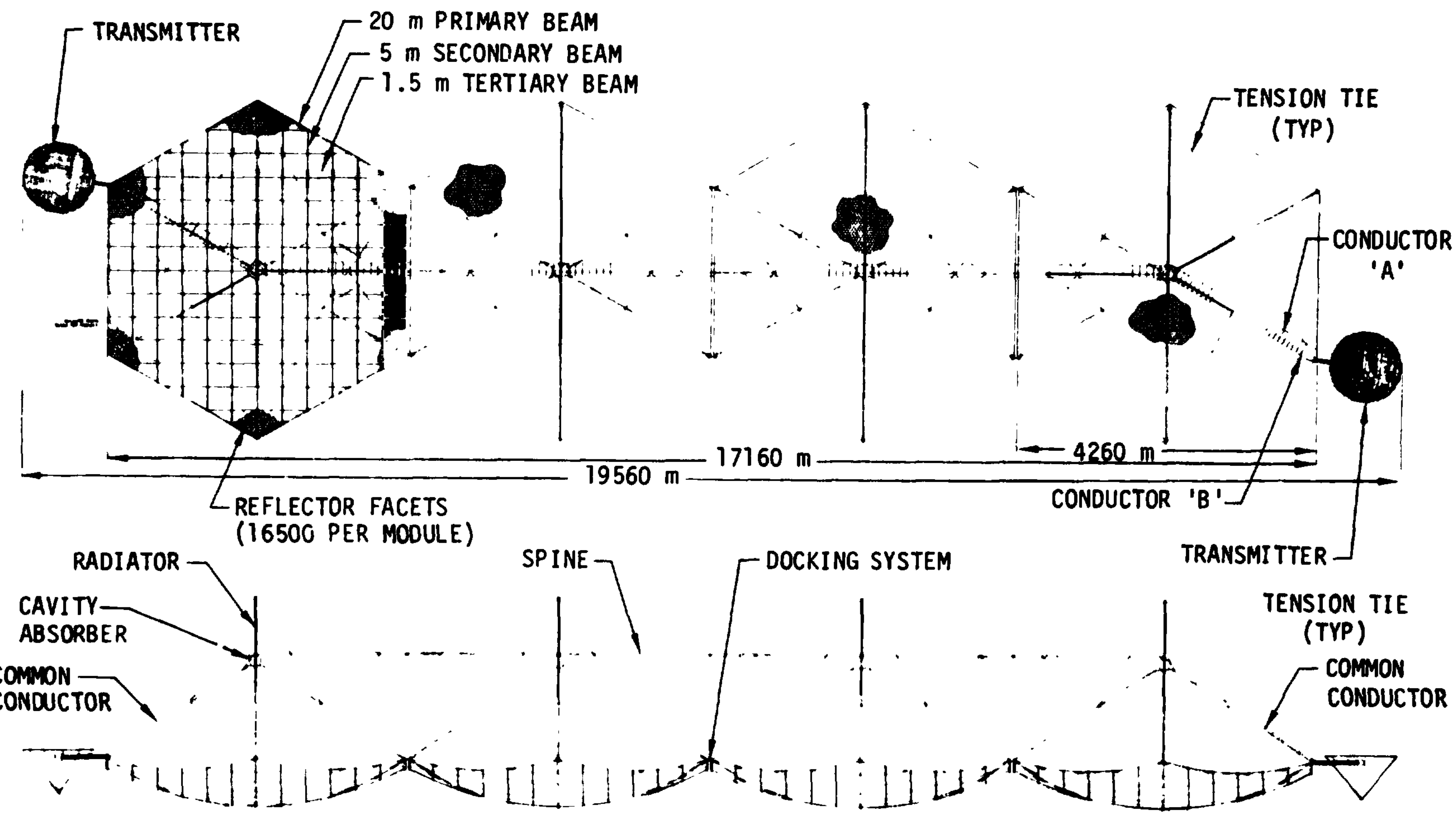

Figure 3.4-41 4- Module Brayton SPS

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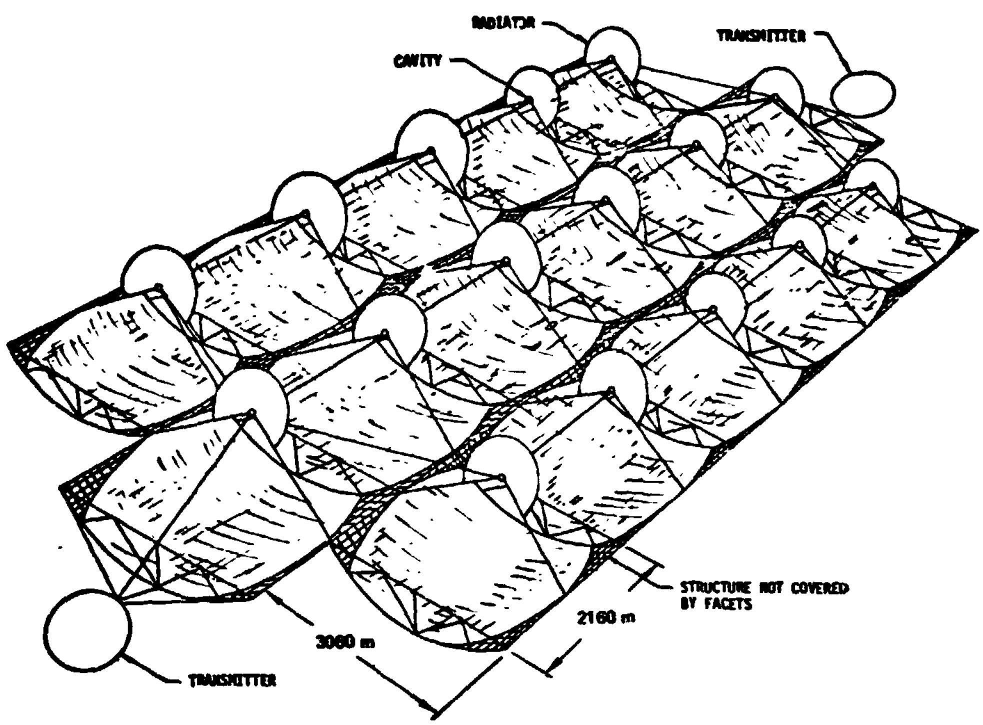

Figure 3.4-42 16 Module Thermal Engine Satellite

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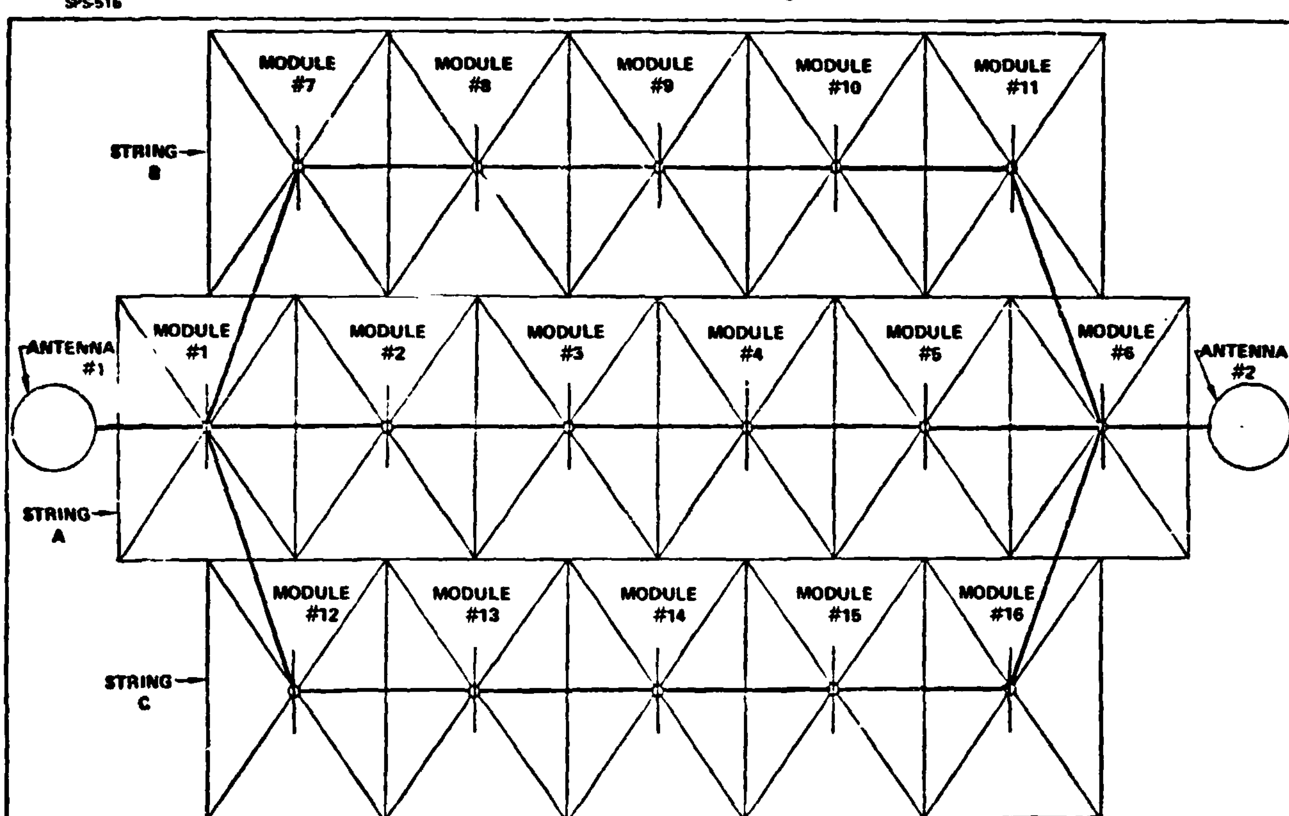


Figure 3.4-43 Thermal Engine Satellite Configuration

Top Level Timeline

Assumption #1 - 365 days total construction time/total satellite

Assumption #2 - Allow 30 days for final integration/checkout

Assumption #3 - Allow 5 days to attach last module to others, interconnect power busses, etc.

Assumption #4 - Antenna installation will be done concurrent with fabrication of satellite modules

Assumption #5 - Although 1 day/week is allotted as an off-duty day for each crewmember; the work phasing can be organized such that a common shutdown day is not required.

Assumption #6 - Allow 10 days for attachment of last satellite string set (modules 12-16) to other module sets.

These assumptions parallel those used for the photovoltaic satellite where applicable.

320 days to fabricate modules, or 20 days/module

Use 20 days as basic satellite module construction time allotment

The top level timeline is shown in Figure 3.4-44.

In each 20 day time period, the following major operations have to be completed:

- Fabricate reflector frame
- Fabricate radiator assembly
- Fabricate cavity assembly
- Attach radiator and cavity assemblies
- Attach radiator/cavity assembly to reflector assembly
- Index module out of facility

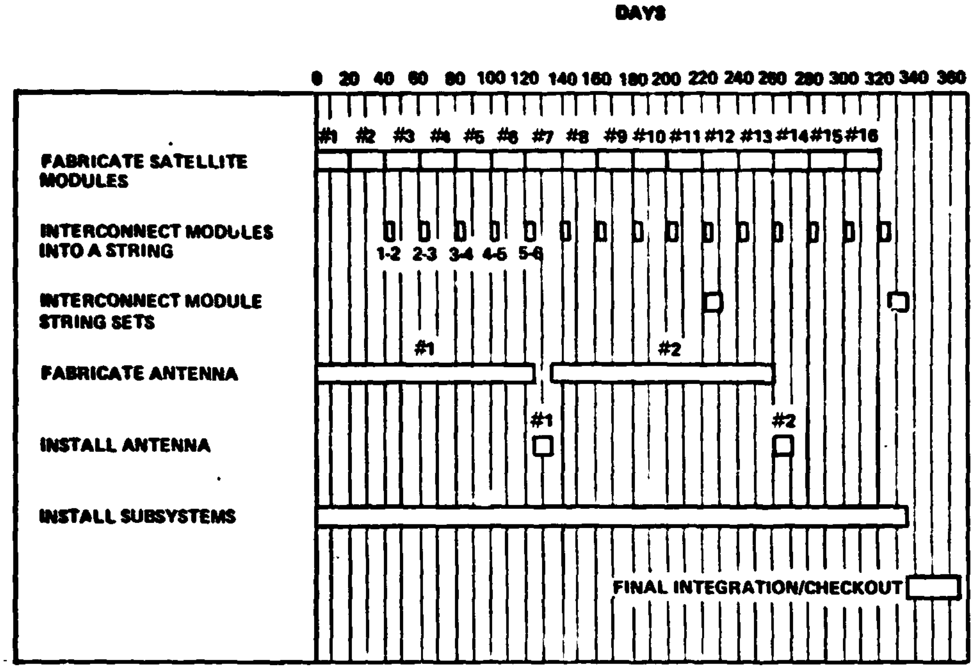


Figure 3.4-44 Thermal Engine Satellite GEO Base Construction Top-Level Timeline

Assumption #7 - Satellite indexing rate 1.1 m/min (same as assumed for photovoltaic satellite)

Assumption #8 - Joint assembly time = 1 hour (same as derived for the photovoltaic satellite)

Using these assumptions and the frame dimensions shown in Figure 3.4-43, the following data can be derived:

Time required to index satellite to halfway point = 16.36 hrs

Time to join radiator legs to reflector = 1 hour (assuming that both legs are joined simultaneously)

Total time from completion of end frames to start of next module is 37.72 hrs

The time available for fabrication of the major elements is computed as follows:

(20 days) (22 hrs/day) = 440 hrs total time per module
-38 hrs to index
402 hrs to complete subassemblies
= 18.3 days

The top level timeline for each satellite module is shown in Figure 3.4-45.

3.4.1.3.3.2 Reflector Construction Concept

Configuration

The satellite module reflector configuration and nomenclature is shown in Figure 3.4-46. As can be seen, the reflector is a trough-shaped dish composed of two major elements: (1) the reflector frame, and (2) the reflector facet assemblies.

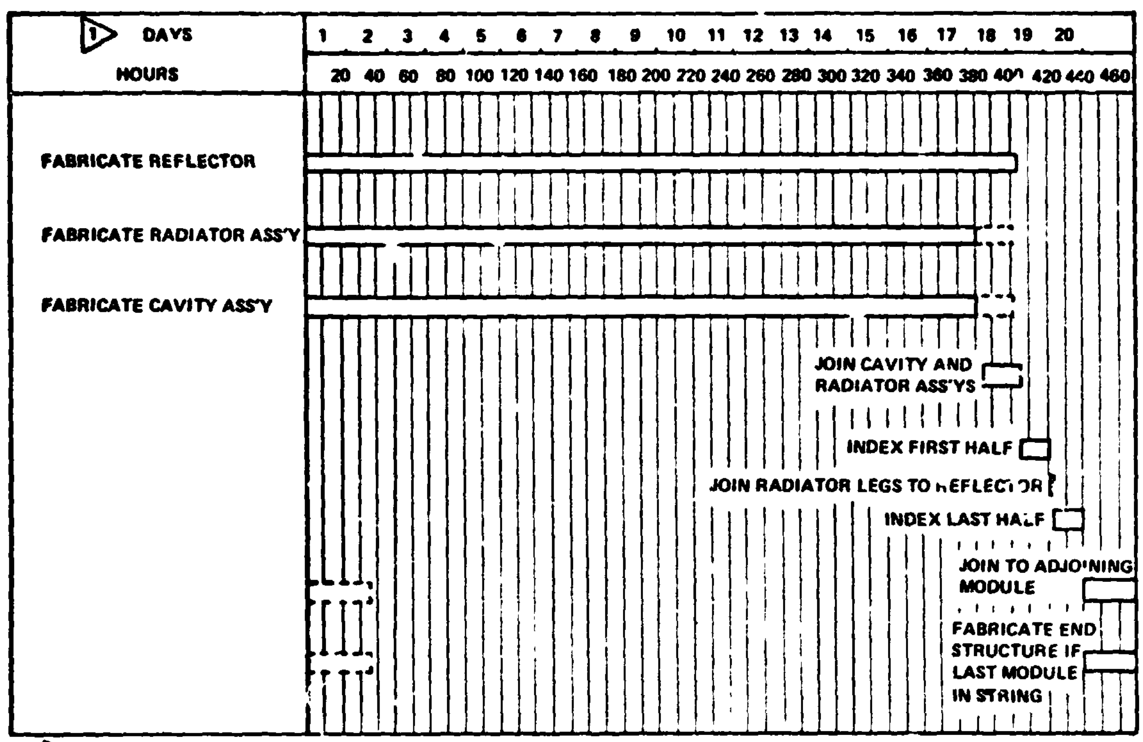
Reflector Construction Timeline

The timeline for constructing the reflector assembly is shown in Figure 3.4-47.

Reflector Construction Machinery Requirements

Beam Machines—Based on the construction philosophy, construction stragety, frame details, and timeline analysis described in the preceding sections, requirements for the beam machines are as described in Table 3.4-8.

The A and B beam machines could also be used as the power units to push the completed satellite module out of the facility.



1 22 HRS WORK TIME/DAY

Figure 3.4-45 Thermal Engine Satellite Module Top-Level Timeline

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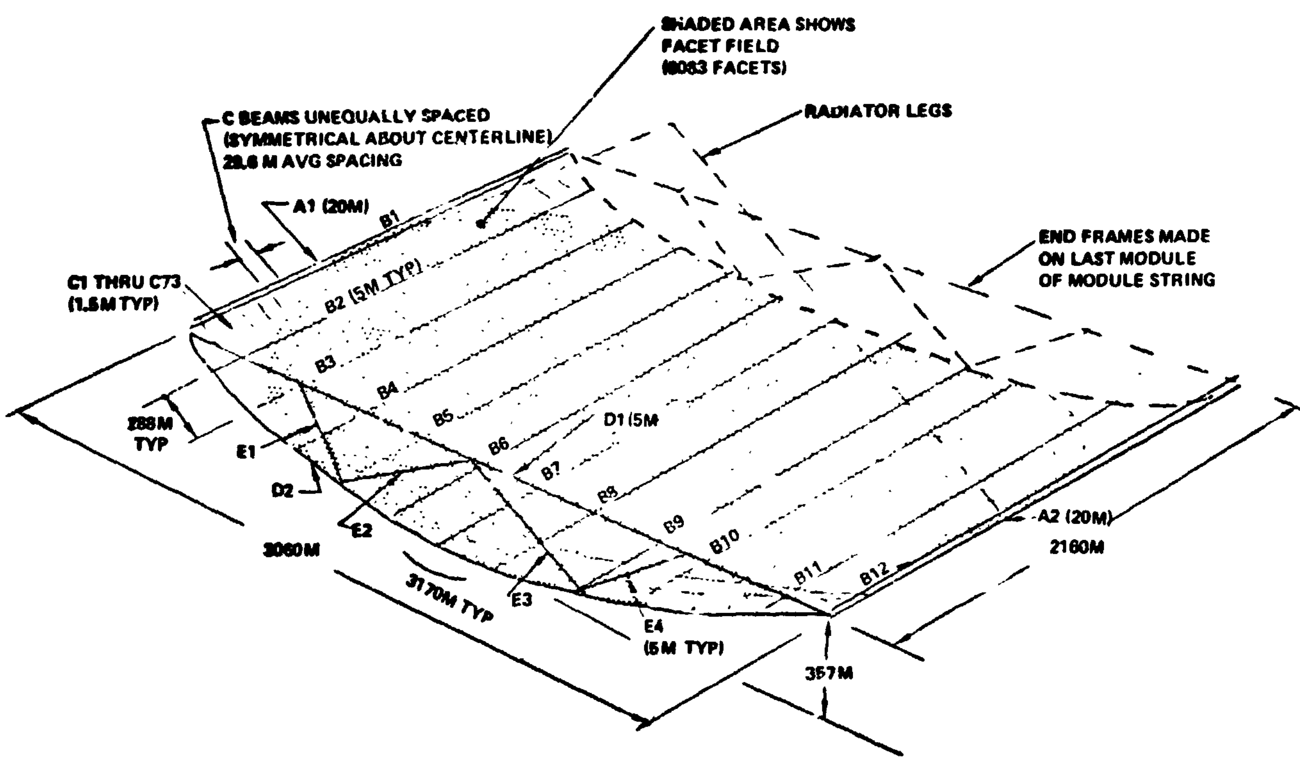

Figure 3.4-46 Reflector Frame Configuration

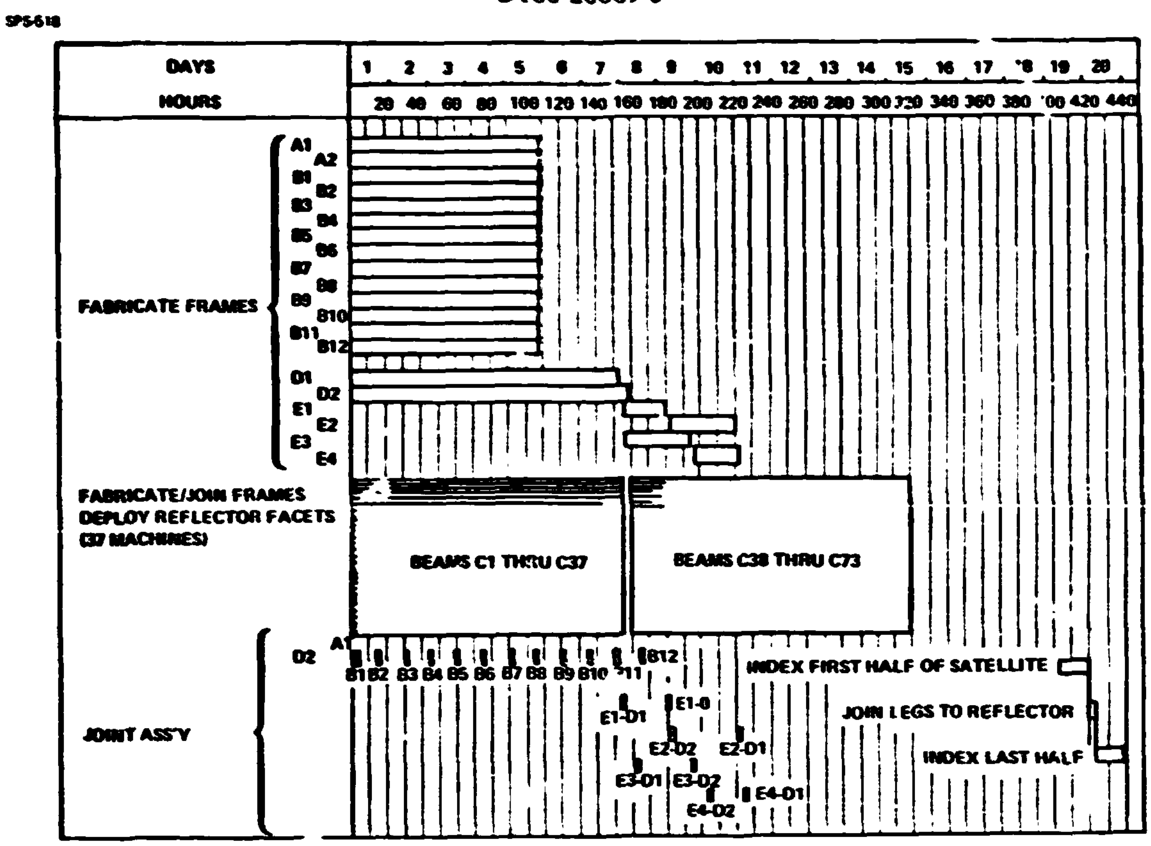

Figure 3.4-47 Reflector Assembly Detailed Timeline

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Table 3.4-8

Beam Machine Requirements

Beam Size	No. Required	Beam Type: Straight	Beam Type: Curved	Mobility: Fixed	Mobility: Movable	Makes Beam No.	Remarks
20m	2	X			X	A1, A2	
5m	12	X			X	B1 thru B12	
5m	1	X	X	X		D2, E3, E4	Required a beam end holding machine when making D2 (see section 3.3.3)
5m	1	X			X	D1, E1, E2	
1.5m	37		X		X	C1-C73	Included in facet deployment machine (section 3.3.5)

Beam Supports--As the A, B, D and E beams are fabricated, it will be necessary to periodically support them. In the case of the D and E beams, these supports can be removed as soon as these beams are joined to the A and B beams. In the case of the B beams, the beam supports will have to be removed to make clearance for the facet deployment. The supports used for the A beams will have to support the entire satellite module while the module is under construction and as the module is indexed out of the facility.

The concept for the various beam supports are shown in Figure 3.4-48.

Assuming that a beam support is required every 200m the following number of beam supports will be required (for the reflector only):

20m beam supports	22 required
5m beam supports	173 required

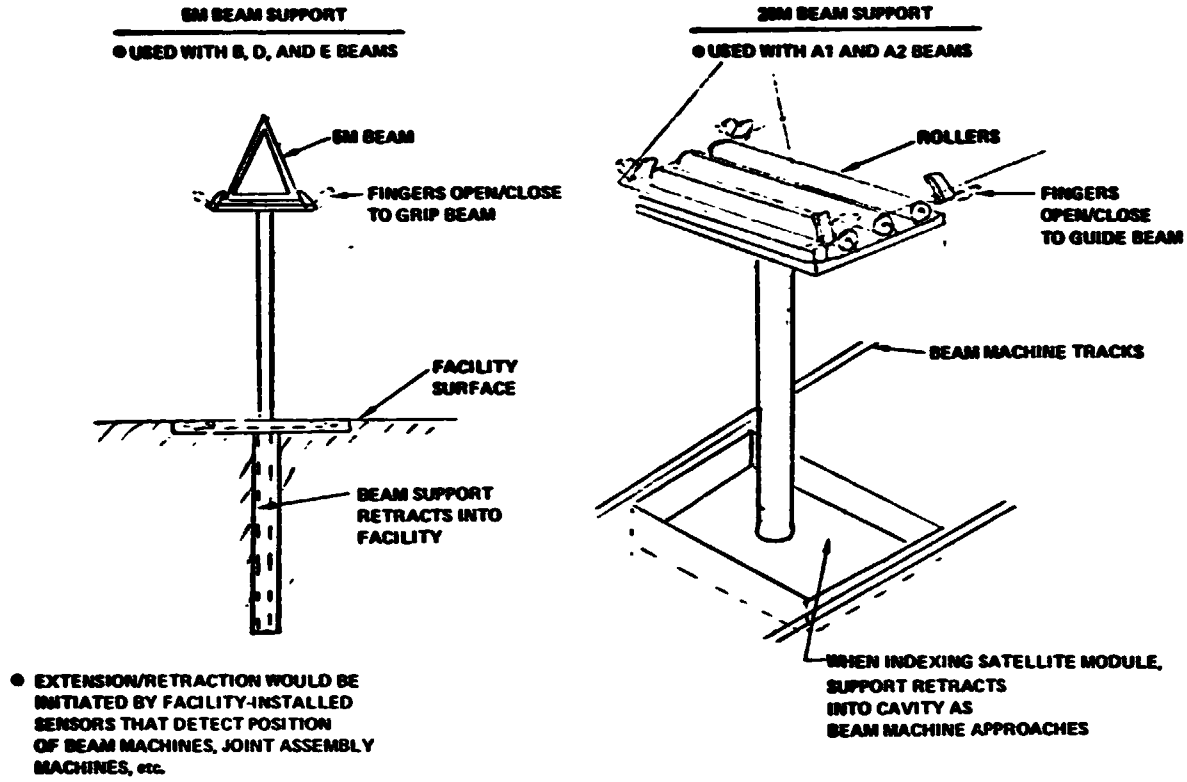

Figure 3.4-48 Beam Support Assemblies

Beam End Holding Machine Whenever a beam must pass by existing transverse beams as it is fabricated (such as when the D2 beam is made), it is not possible to move the beam machine. The beam machine will not clear the transverse beams. It is, therefore, necessary to fix the beam machine and to use a beam end holder that will guide and support the extruded end of the beam as the beam is made. This beam end holder could either be self-propelled or it could be pushed along by the beam as it is being fabricated. A concept for a beam end holder machine is shown in Figure 3.4-46. Only one of these machines is required.

Joint Assembly Machine-At every beam joint, it will be necessary to make some type of beam-to-beam connection. Figure 3.4-50 shows the various types of joints that have to be made.

> NOTE: The type of joint fastening is unknown at this point in time. The joint could be mechanically attached (by rivets of sleeves), welded, or taped and glued.

Figure 3.4-51 shows a concept for a joint assembly machine. A single machine will be able to keep up with all of the A-D, B-D, and E-D joints.

The E-B joints will be made by joint assembly devices attached to the facet deployment machine (see next paragraph).

Facet Deployment Machine-The facet deployment machine concept shown in Figure 3.4-52 is a multipurpose machine that performs 3 basic operations: (1) it fabricates the 1.5m C beams, (2) it attaches the C beams to the B beams, and (3) it deploys the facets. Thirty-six of these machines will be required.

Facet Assembly Machine-In order to prepare the heliostats by unpacking and folding into the shape shown in Figure 3.4-52 it will be necessary to have a heliostat subassembly area located at the main facility. This subassembly area will employ a heliostat assembly machine that assembles the support structure, attaches the reflector sheet assembly, folds the assembled heliostats and loads them into cassettes to be placed on the heliostat deployment machine. Figure 3.4-53 shows a concept for this machine.

Reflector Construction Facility Requirements

Throughout the preceding sections, various facility requirements have been mentioned or implied. These requirements have been collected and summarized in Table 3.4-9. A facility concept that satisfies these requirements is shown in Figure 3.4-54 and 3.4-55.

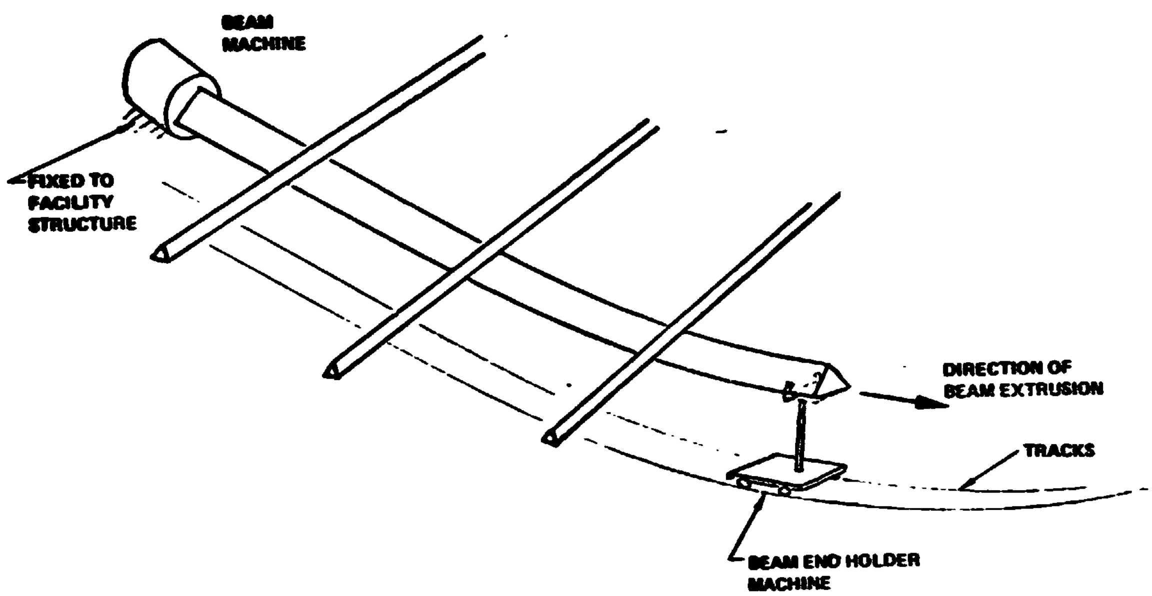

Figure 3.4-49 Beam End Holder Machine

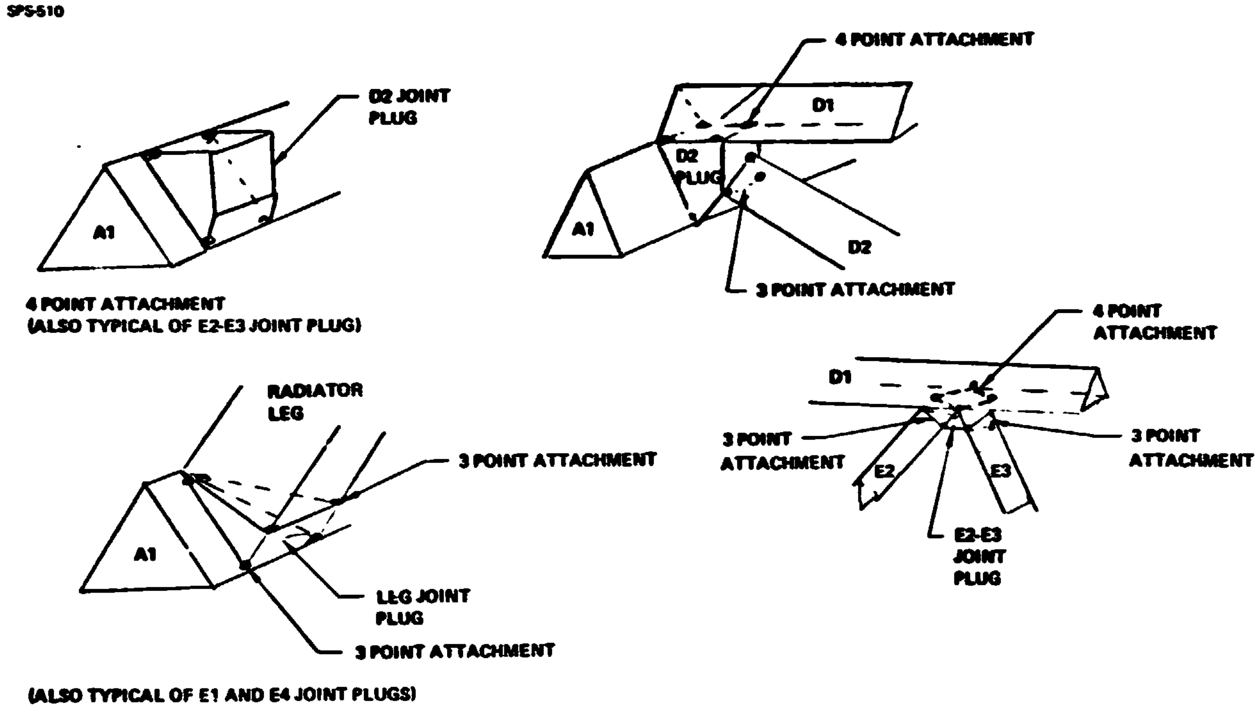

Figure 3.4-50 Typical Joint Types

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REQUIREMENTS

- MOVES ALONG LONGITUDINAL MEMBERS
- MUST BE ABLE TO MOVE BETWEEN RETRACTED BEAM MACHINES
- MUST BE ABLE TO RELOCATE TO JOINT LOCATION IN 15 MINUTES
- MUST BE CAPABLE OF MANIPULATING AND HOLDING FRAMES PRIOR TO JOINING
- MUST BE ABLE TO ALIGN FRAMES
- MUST BE ABLE TO PRECISELY LOCATE FASTING DEVICE AT JOINT POINTS
- MUST BE ABLE TO FASTEN FRAMES
- MUST BE ABLE TO RELOCATE TO NEXT FASTENING POINT IN 5 MINUTES
- MUST BE ABLE TO INSPECT FASTENED JOINT
- MUST BE ABLE TO WORK AROUND STRUCTURE WITHOUT DAMAGING FRAMES
- MUST BE A MANNED MANIPULATOR
- TRANSPORT PLUG STRUCTURES

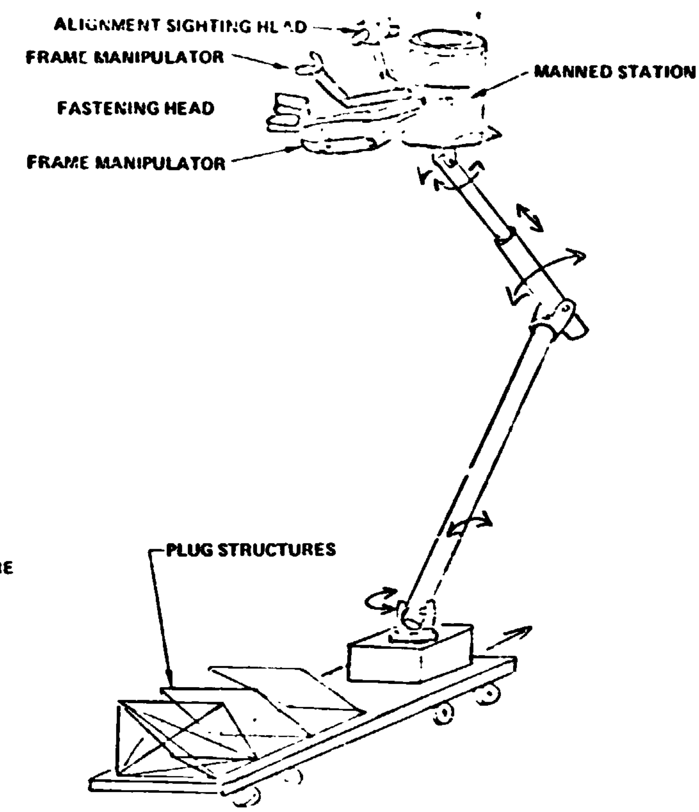

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Figure 3.4-51 Joint Assembly Machine

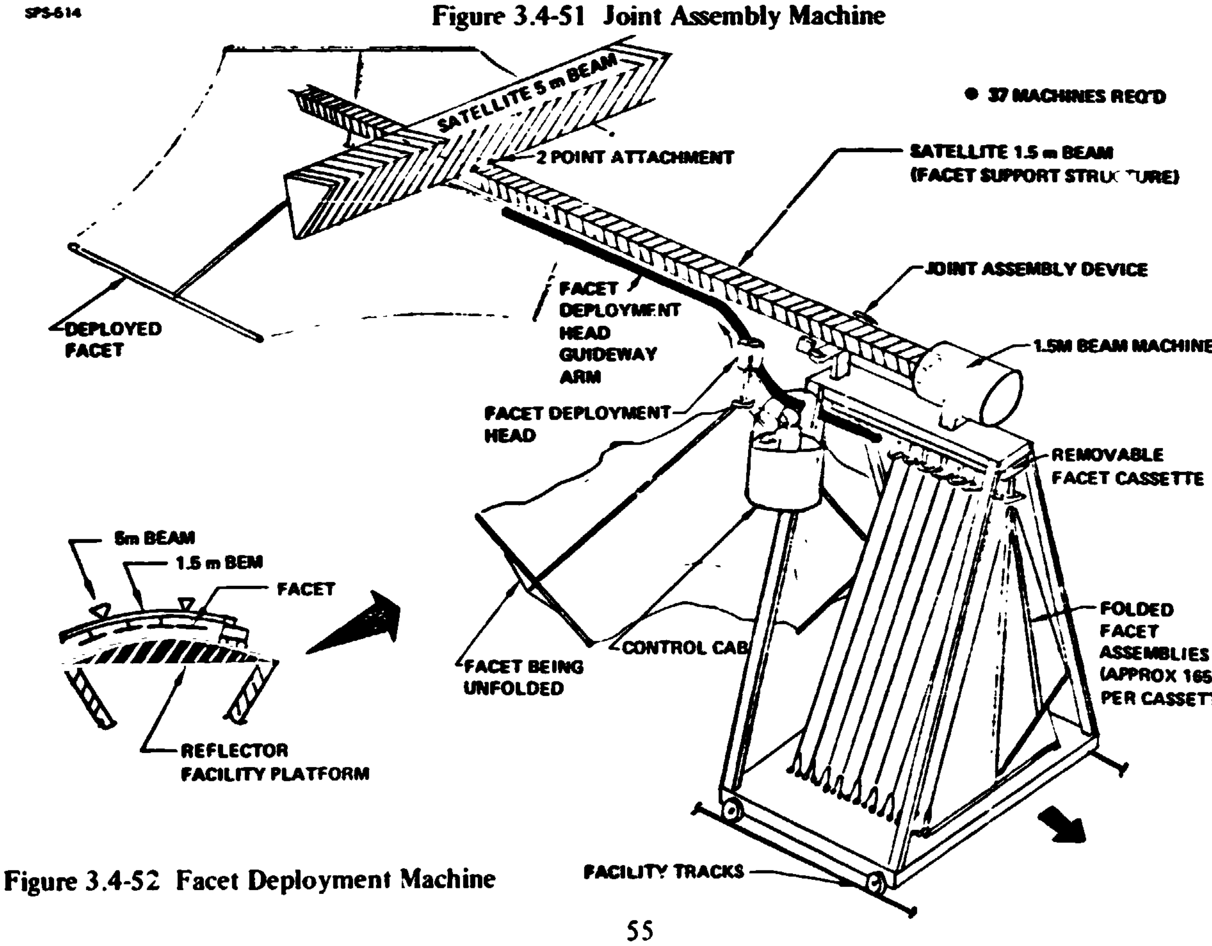

Figure 3.4-52 Facet Deployment Machine

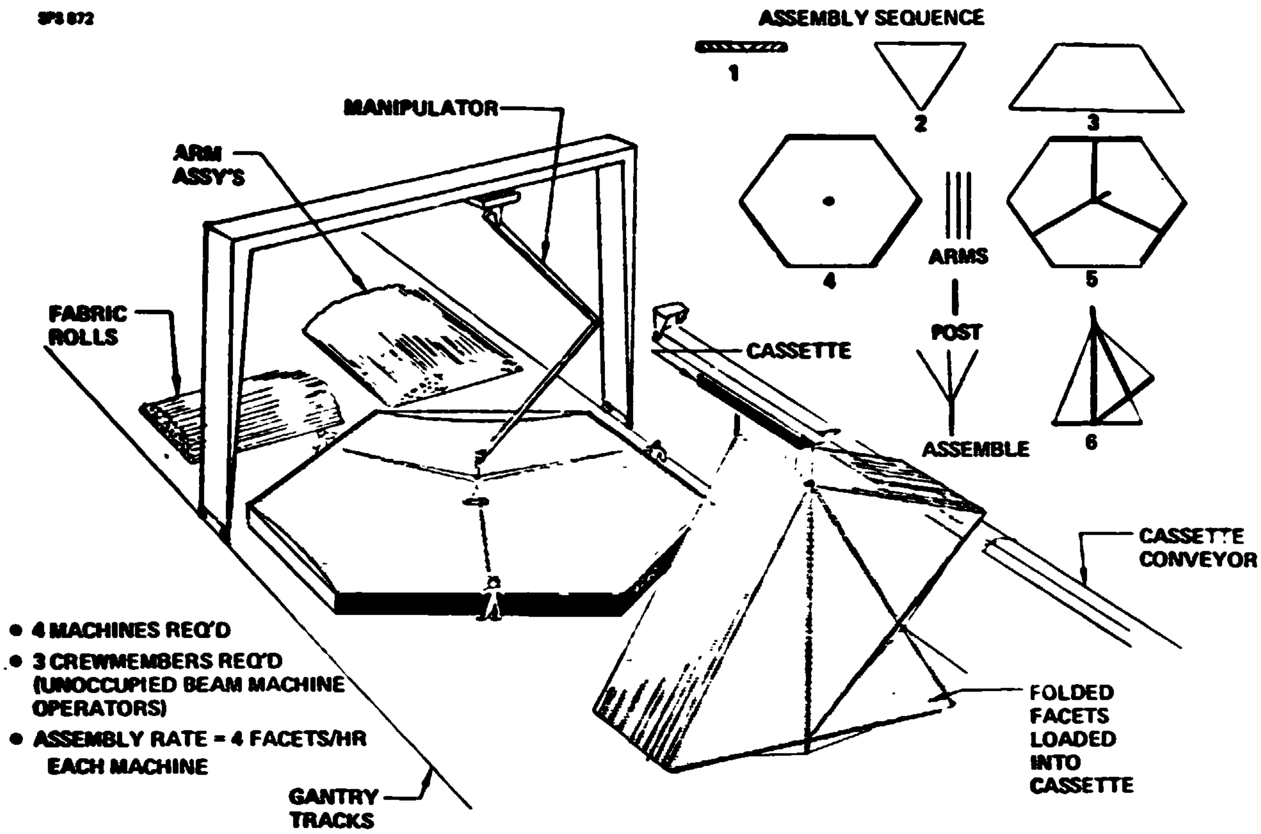

Figure 3.4-53 Facet Assembly Machine

Reflector Construction Manpower Requirements

The manpower required to fabricate the reflector assembly is summarized in Table 3.4-10. This summary does not include supervisory or support personnel.

Table 3.4-9

Reflector Construction Facility Requirements

- o Used to fabricate one satellite module reflector
- o Must be integral with antenna, radiator, and cavity construction facilities
- o Must include a reflector facet subassembly area
- o Fabrication should take place on one side of the reflector if possible
- o Must provide tracks for the various construction modules
- o Must include retractable beam support systems
- o Must allow simultaneous frame fabrication and reflector deployment
- o Must provide means to relocate C beam machines to new C beam location
- o Must allow satellite module to be indexed out of facility
- o Must provide means for constructing beams in defined relative positions
- o Must provide means to clear machines out of way so that satellite module can be indexed
- o Must provide means to move components from receiving area to the user machines
- o Must provide means to transport personnel to manned machines

3.4.1.3.3.3 **Radiator/Cavity/Spine Construction Concept**

Radiator/Cavity/Spine Configuration The overall configuration of the radiator, cavity, and spine assemblies is shown in Figure 3.4.56.

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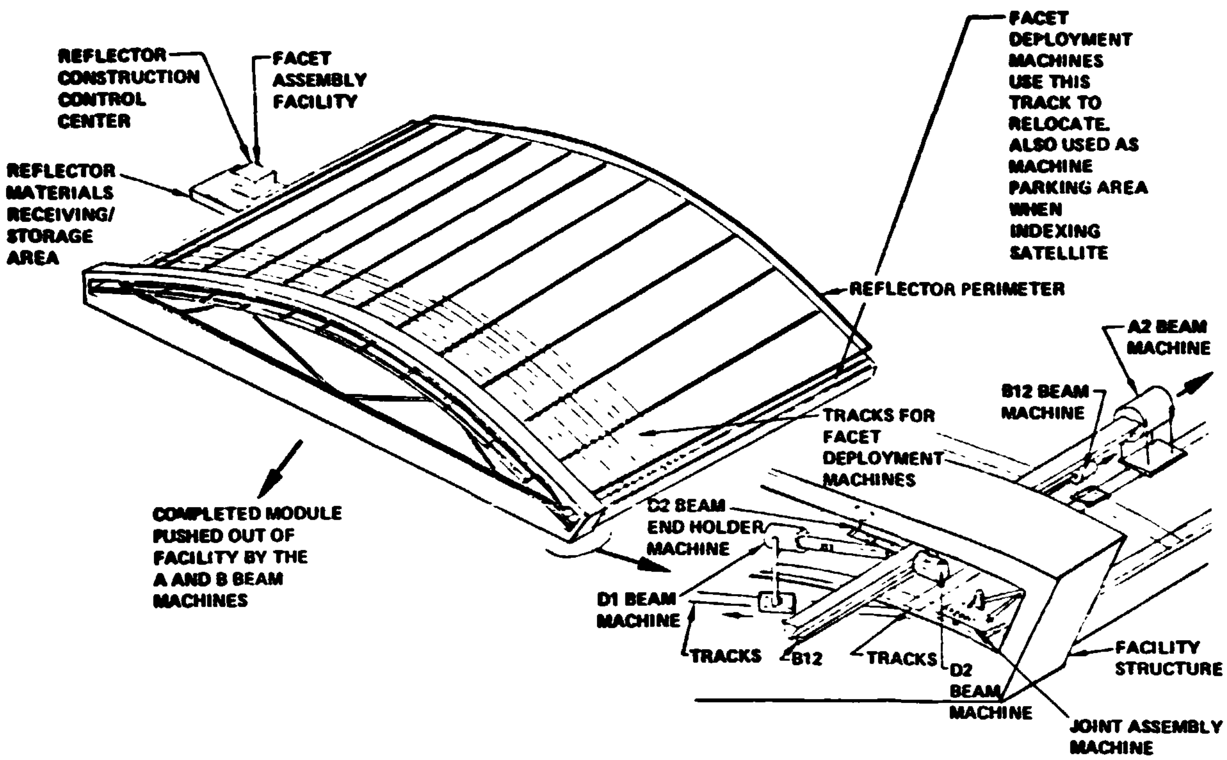

Figure 3.4-54 Reflector Construction Facility

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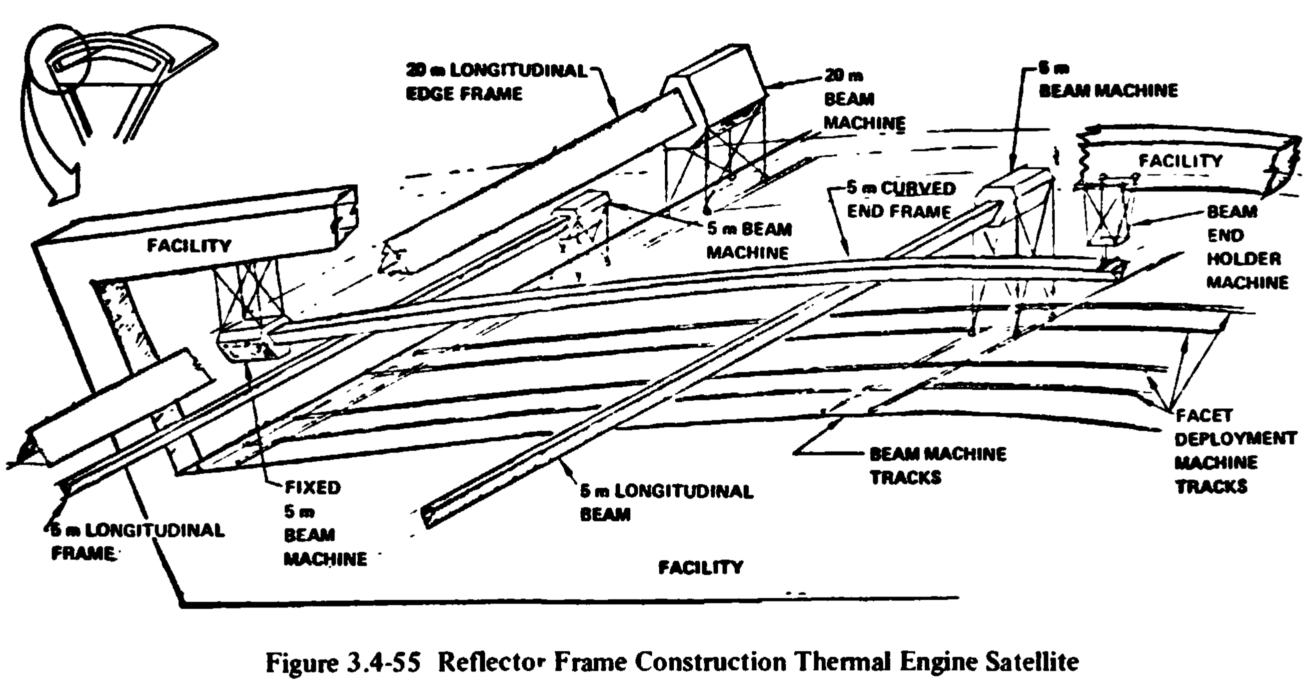

Figure 3.4-55 Reflector Frame Construction Thermal Engine Satellite

Table 3.4-10 Reflector Construction Manpower Requirements

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JOB TITLE	JOB DESCRIPTION	NO. REQ'D SHIFT	TOTAL REQ'D BASE	WHERE LOCATED
BEAM MACHINE OPERATOR	• CONTROLS LOADING OF BEAM COMPONENTS INTO BEAM MACHINE • INITIATES BEAM FABRICATION • MONITORS BEAM FABRICATION • ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL • CONTROLS INTERFACING WITH BEAM END HOLDING MACHINE IF REQ'D • MONITORS BEAM SUPPORT PLACEMENT/RETRACTION	16 [1]	32 [1]	REFLECTOR CONSTRUCTION CONTROL CENTER
JOINT ASSEMBLY MACHINE OPERATOR	• CONTROLS LOADING OF JOINT ASSEMBLY COMPONENTS ONTO MACHINE CARRIAGE • CONTROLS MOVEMENT OF MACHINE ON TRACKS • CONTROLS MOVEMENT OF MANIPULATOR CAB • CONTROLS MANIPULATORS • CONTROLS JOINT FASTENING • ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL	1	2	ON THE MACHINE
FACET DEPLOYMENT MACHINE OPERATOR	• CONTROLS LOADING OF COMPONENTS ONTO MACHINE • INITIATES/MONITORS MACHINE FUNCTIONS • ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL • DRIVES MACHINE WHEN RELOCATING	37	74	ON THE MACHINE
FACET ASSEMBLY MACHINE OPERATOR	• INITIATES/CONTROLS LOADING OF COMPONENTS INTO MACHINE • INITIATES/MONITORS MACHINE FUNCTIONS • ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL	SEE NOTE [1]	SEE NOTE [1]	FACET ASSEMBLY FACILITY

[1] WHEN BEAM FABRICATION IS COMPLETED, THESE OPERATORS GO AND OPERATE THE FACET ASSEMBLY MACHINES.

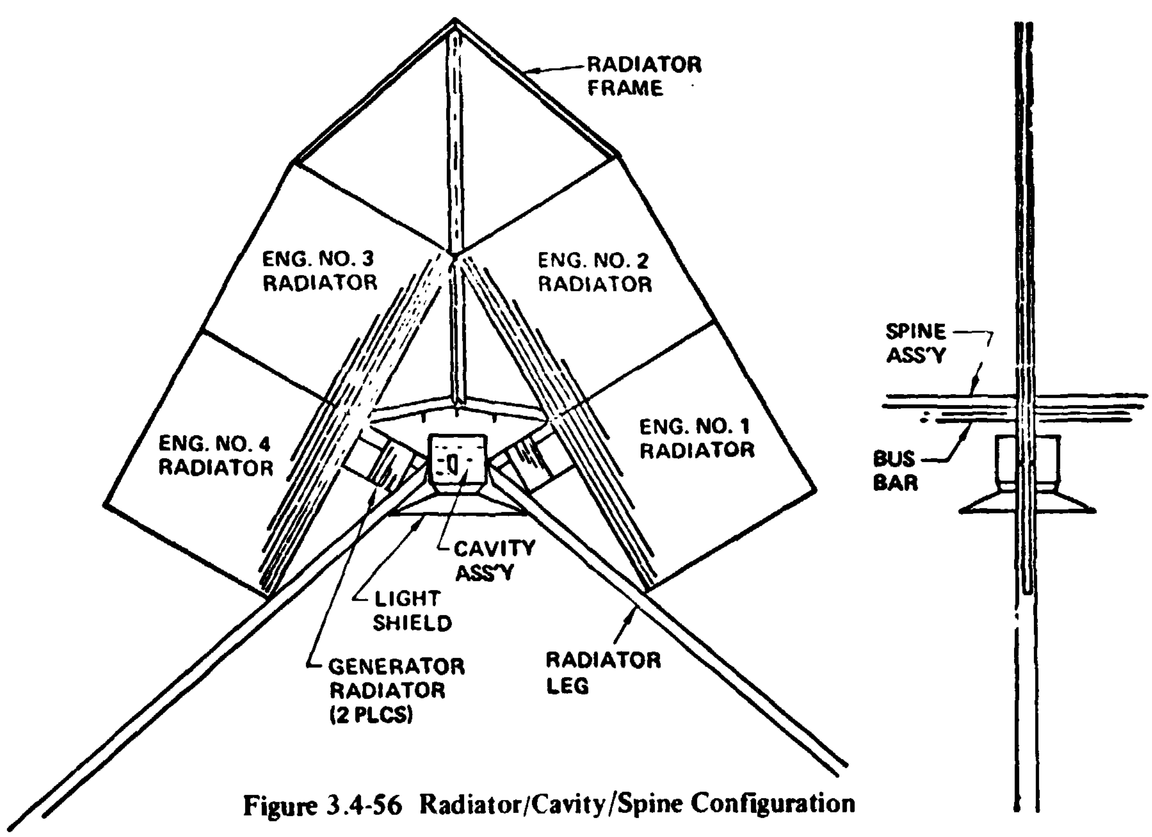

Figure 3.4-56 Radiator/Cavity/Spine Configuration

Radiator/Cavity/Spine Constuction Timeline Analysis–In section 3.4.1.3.3.1 it was found that there are approximately 402 hours (18.3 days) available to complete the radiator/cavity/spine assemblies. In order to get all of these major assembly operations completed within this time frame, it will be necessary to fabricate the radiator, cavity and spine assemblies simultaneously. The construction timelines for these three major assembly operations are shown in Figures 3.4-57, -58, and -59.

Radiator/Cavity/Spine Construction Machinery Requirements

Radiator Construction Machinery Requirements–Review of the radiator construction requirements has identified five types of construction machinery. (1) beam machines, (2) beam supports, (3) joint assembly machines, (4) radiator assembly machines, and (5) manifold assembly machines.

Beam Machines–Two beam machines were identified: (1) a 20m beam machine and (2) a 10m beam machine. These both would have to be traveling beam machines. These machines must wrap the fabricated beams with a thermal protection wrapping.

Beam Supports–It will be necessary to provide facility-mounted retractable beam supports that will be used with the 20m, 10m, and 5m beams. The beam support types shown in Section 3.3.2 illustrate what is needed. The 20m beam supports required for the radiator frames do not need the roller assembly shown before.

The following number of each type of beam support will be required if it is assumed that one is required every 200m or at the end of each beam shorter than 200m:

20m Beam Support - 50 Required
10m Beam Support - 47 Required
5m Beam Support - 7 Required

Joint Assembly Machines–A single joint assembly machine such as was shown before will satisfy this requirement.

Radiator Assembly Machines–It was found that eight radiator assembly machines will be required. Table 3.4-11 lists the functional requirements for this machine.

Figures 3.4-60 and -61 show a concept for a radiator assembly machine that satisfies most of these requirements. One potential problem is that it will not be possible for these machines to operate immediately adjacent to one another.

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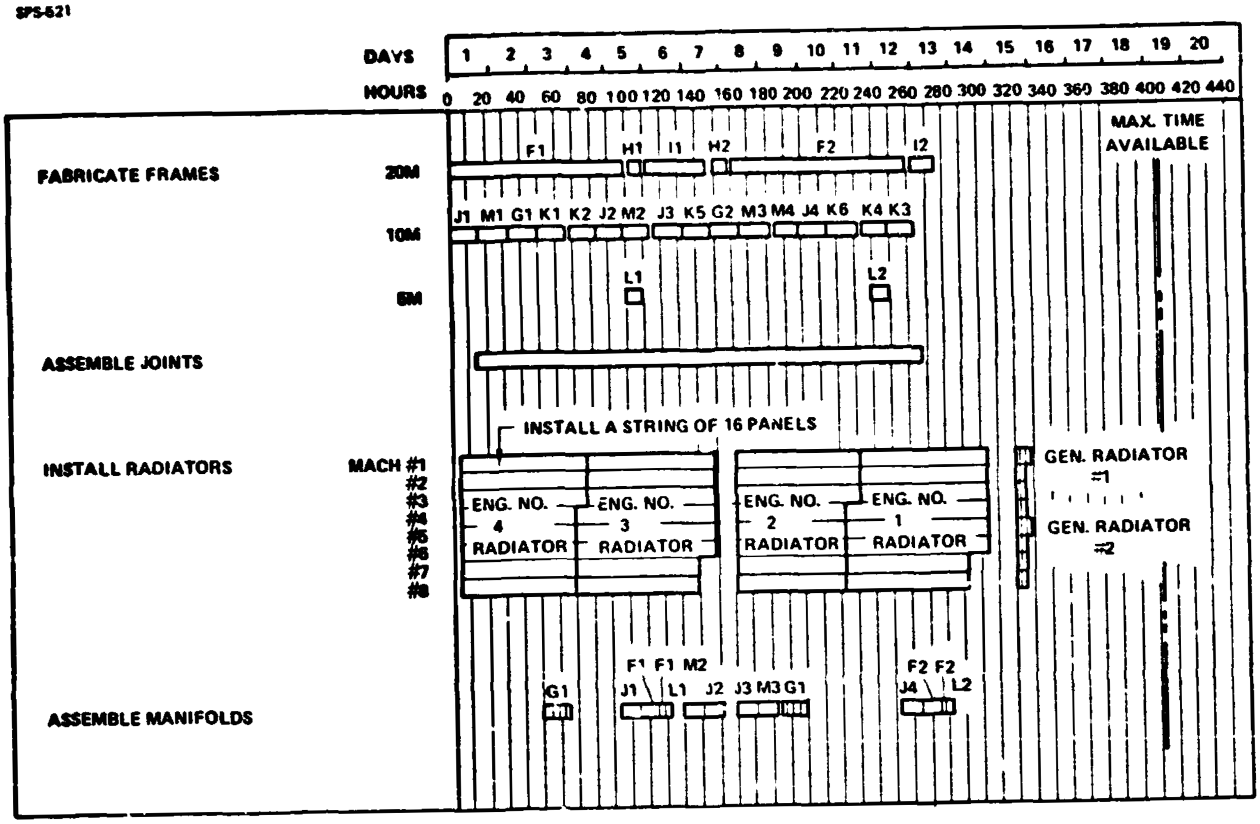

Figure 3.4-57 Integrated Radiator Assembly Timeline

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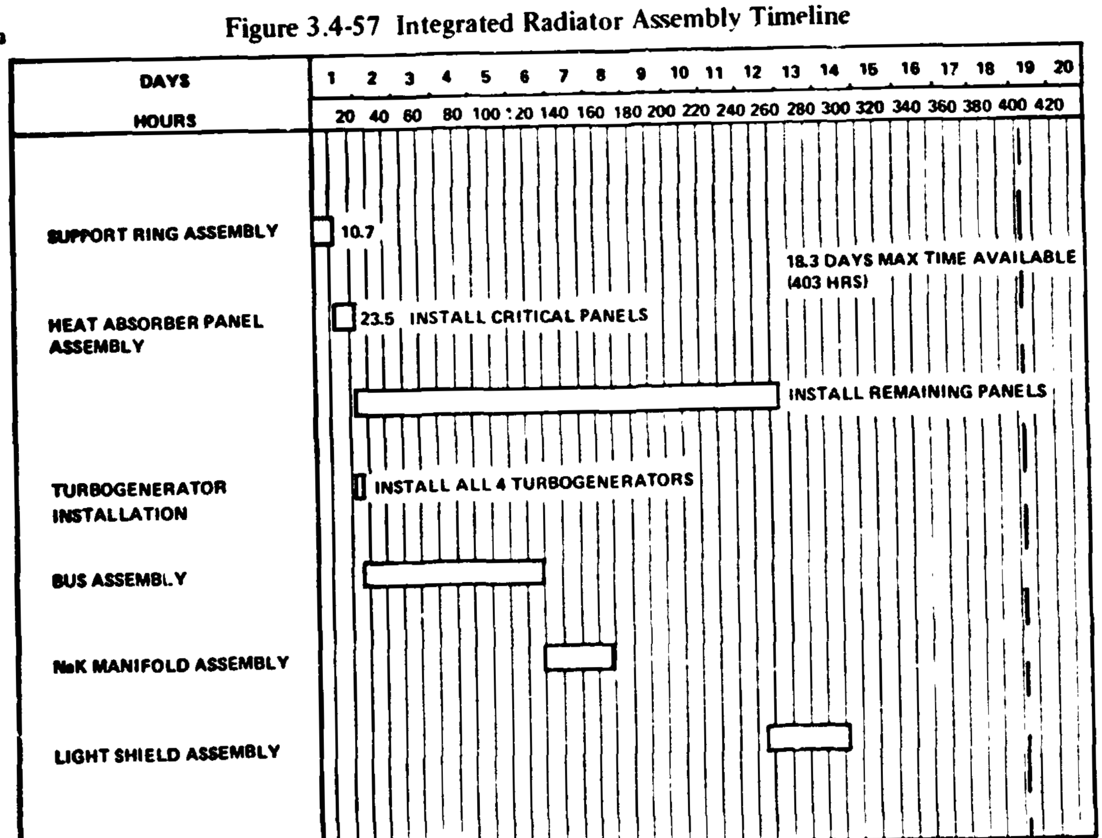

Figure 3.4-58 Cavity Assembly Master Timeline

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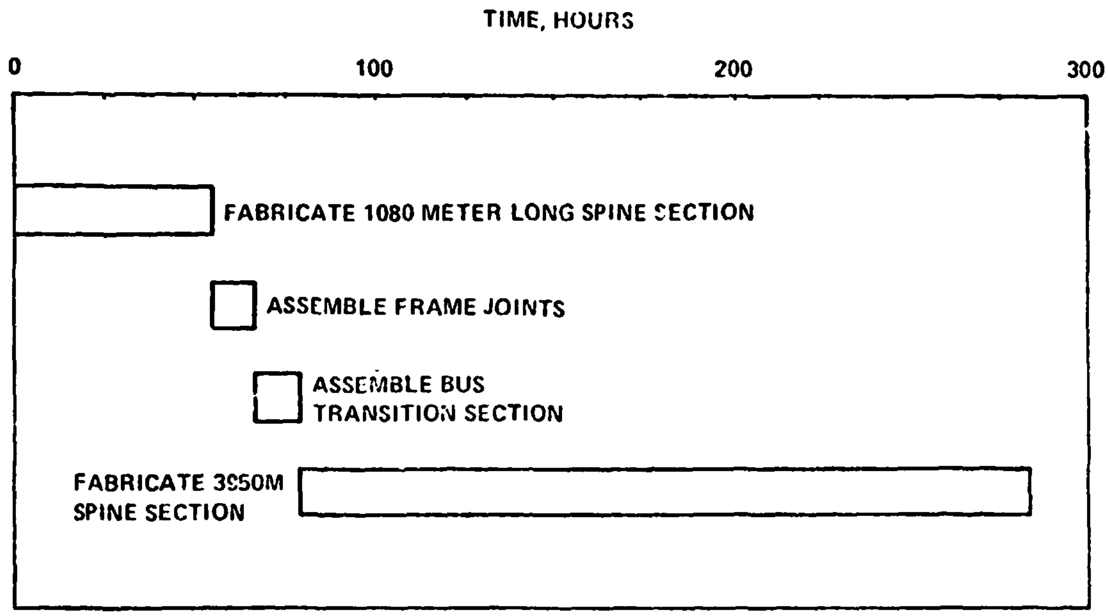

NOTE: THIS TIMELINE REPRESENTS THE WORST CASE.
ALL OTHER SPINE CONFIGURATIONS REQUIRE LESS TIME.

Figure 3.4-59 Spine/Bus Assembly Timeline (Type 2 Spine Configuration)

TABLE 3.4-11
Radiator Assembly Machine Requirements

- o Transport 31 sets of panels (enough for 351 strings)
  - Type A radiator panels (65 pcs)
  - Type B radiator panels (434 pcs)
- o Transport 62 pipe support assemblies
- o Pickup and position panel for welding
- o Pickup and position pipe support assembly
- o Weld header pipe joints (variable diameters)
- o Weld three pipes (same diameter all panels)
- o Attach pipe support to frame
- o Attach pipe support to header pipe
- o Inspect welds
- o Self propelled
- o Support free end of panel string until pipe support installed
- o Can work adjacent to another radiator assembly machine
- o Must be able to move to new string location along tracks

Another requirement that was not satisfied, due to the large panel area required, was being able to carry all 31 sets of panels at one time. Instead, the concept shown has the machine carrying 7 string sets of panels, the supply required for each machine to build its share of an engine radiator. The machines will be reloaded before going to the next radiator.

Manifold Assembly Machine -It was found that a single NaK manifold assembly machine would be required. A concept for this machine is shown in Figure 3.4-62.

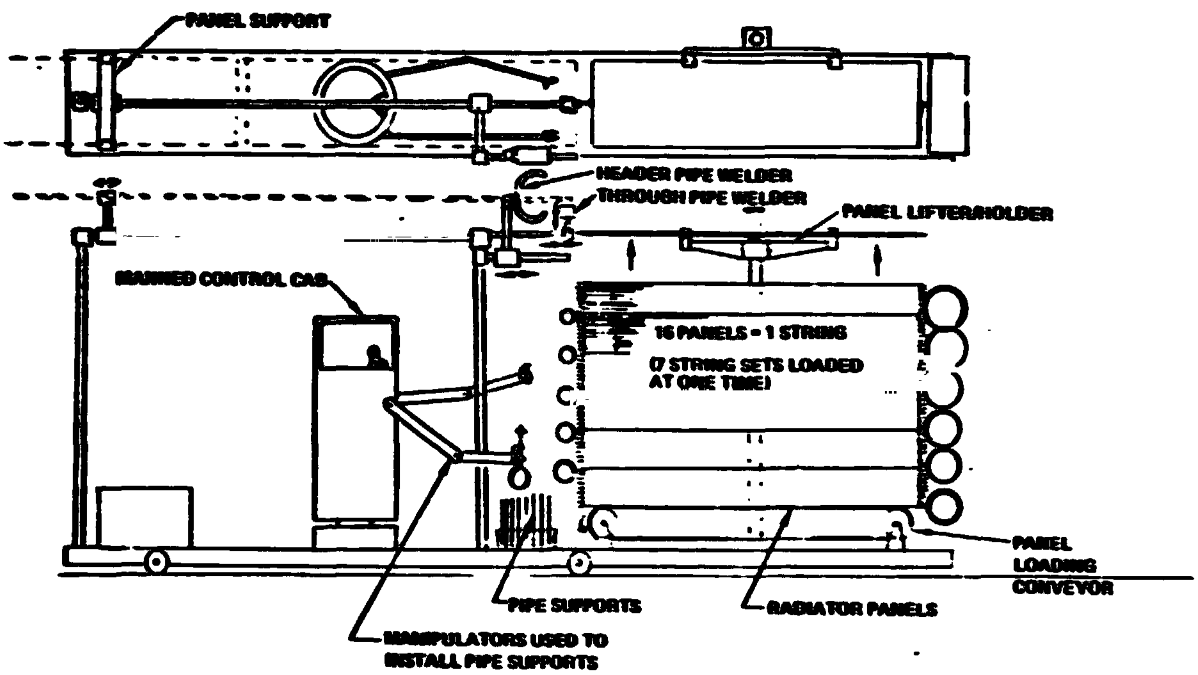

Figure 3.4-60 Radiator Assembly Machine

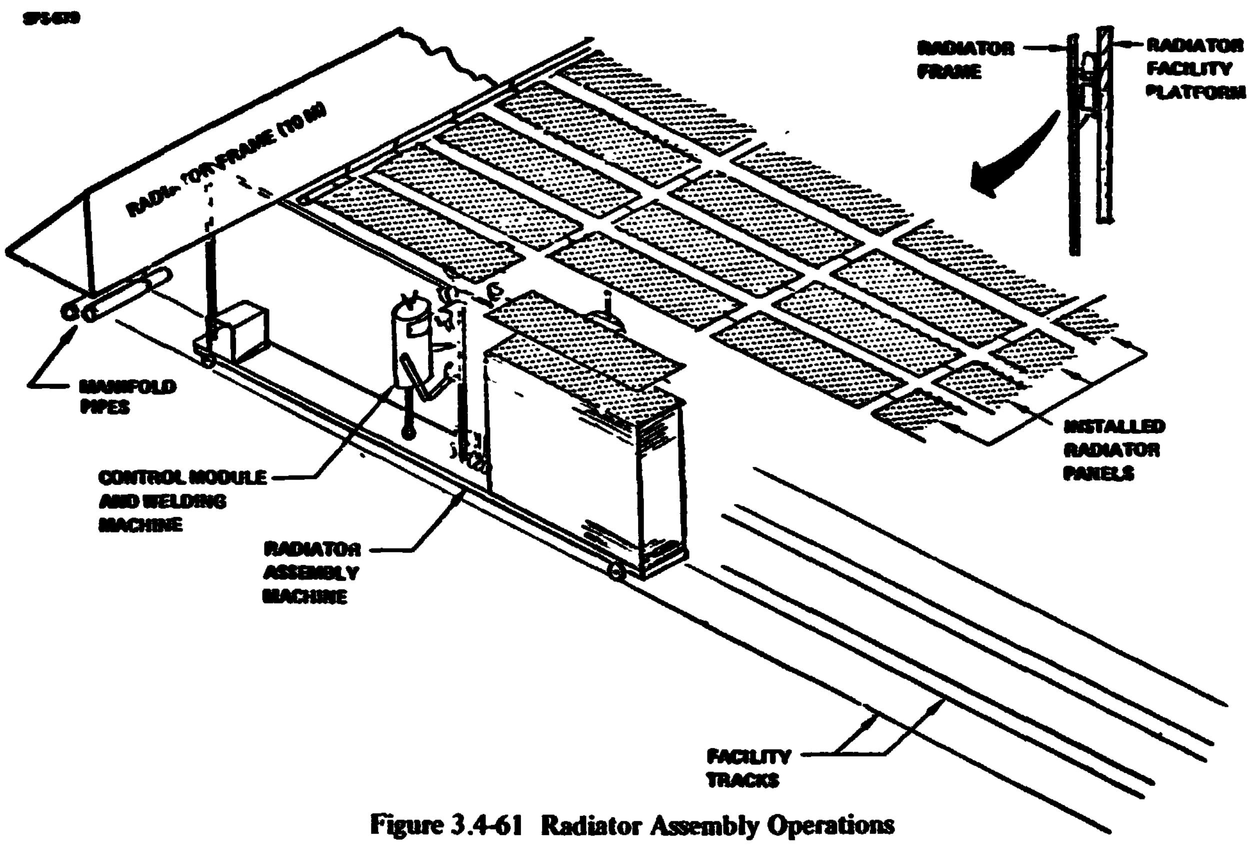

Figure 3.4-61 Radiator Assembly Operations

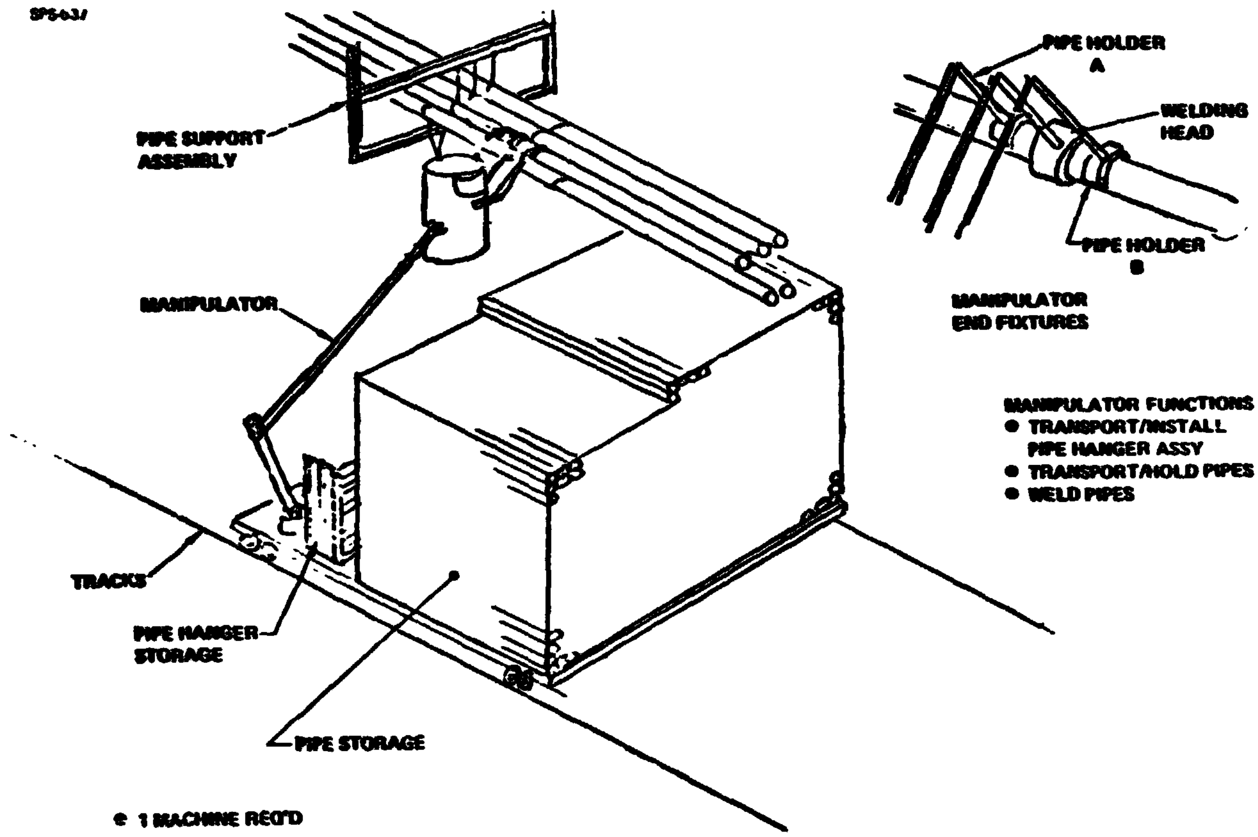

Figure 3.4-62 Manifold Installation Machine Concept

Cavity Construction Machine Requirements–In the timeline analysis of the cavity assembly, requirements for construction machines were either explicitly stated or were implied. These requirements have been collected and summarized in Table 3.4-12. From the timeline summary, it is seen that there are two sets of parallel operations implying that there are two types of construction machines that operate simultaneously:

Type A Assembly Machine -	used to install support ring, heat absorber panels. light shield and turbogenerator
Type B Assembly Machine -	used to install bus bars and NaK manifolds.

Type A Assembly Machine–The requirements for this machine are listed in Table 3.4-12 under the Support Ring, Turbogenerator, Heat Absorber Panel, Light Shield and general requirements. Collating these requirements, it is seen that it will be necessary to provide the following functions:

TABLE 3.4-12
Cavity Assembly Machine Requirements

Support Ring Requirements

- Transport cavity support ring component (17 20m arc-shaped pieces).
- Pick up and place cavity support ring segment into position for fastening.
- Fasten support ring segment into position for fastening.
- Fasten support ring segment to adjacent segment on to radiator leg frame.

Heat Absorber Panel Requirements

- Transport heat absorber panels (some have preattached brackets and/or manifold piping).
- Pick up and place panels into position.
- Mechanically attach panels to one another.
- Place welding head assembly over helium header pipes.
- Weld header pipes.
- Place welding head assembly over manifold pipes.
- Weld manifold pipes.
- Inspect welds.

Turbogenerator Requirements

- Transport turbogenerator set.
- Pick up and place turbogenerator into position.
- Attach turbogenerator to support ring.

Bus Bar Requirements

- Transport bus bar segments
- Pick up and place bus bar segment into position.
- Mechanically attach bus bar segment to heat absorber panel.
- Weld bus bar segments together.

NaK Manifold Requirements

- Transport pipe segments.
- Pick up and position pipe segments
- Weld pipe segments.

Light Shield Requirements

- Transport roll of ring rod material.
- Pick up and position ring rod.
- Form rod into 250 m ϕ ring.
- Transport rolls of light shield sheet material
- Pick up and position roll.
- Attach end of roll to ring.
- Attach end of roll to heat absorber panel.
- Deploy roll between cavity bottom and ring.

General Requirements

- Must be able to travel 360° around the perimeter of the cavity.
- Must have capability to reach throughout the working volume required to install all of the cavity assembly component.
- Must be able to be moved out of the way to allow completed assembly to be moved out of assembly facility.
- Must be able to relocate machine between assembly locations within 5 to 10 minutes.
- Must be able to pick up component from transporter and move it into assembly position within 10 minutes.

  - Component transport.
  - Component manipulator used to lift components from transport carriage up to assembly location.
  - Component alignment manipulator.

- Component Assembly Devices

  - Support ring joint maker
  - Heat absorber panel mechanical interlocker (could be integral with alignment device)
  - Helium header pipe gang welder.
  - Helium manifold pipe welder
  - Turbogenerator interface joint maker
  - Light shield ring joint maker
  - Light shield support ring attacher
  - Light shield roll deployer

- Inspection Devices

  - Helium header pipe weld inspector
  - Helium manifold pipe weld inspector

Consideration of the component transport requirements leads one to elect to use a transporter that can be reloaded prior to each major assembly operation (component cassetts or pallets).

The component alignment, assembly and inspection devices will have to be single purpose end-effectors that can be interchanged on a separate manipulator area. The Type A Assembly Machine designed to satisfy these requirements is shown in Figure 3.4-63 and 3.4-64.

Type B Assembly Machine–The requirements for this machine are given in Table 3.4-12 under the Bus Assembly. NaK Manifold Assembly and General Requirements. Collating these requirements, it is seen that it will be necessary to provide the following functions:

  - Component transport
  - Component manipulator used to lift components from transporter carriage up to the assembly location
  - Component alignment manipulator

- Component Assembly Devices

  - bus bar mechanical attachment tool
  - bus bar welder
  - bus bar jumper attachment tool
  - pipe welder

- Inspection Devices

  - bus weld inspector
  - manifold weld inspector

These components can be met by basically the same type of machine as the Type A Assembly Machine by using dedicated end effectors, component racks and component manipulators (see Figure 3.4-63).

Spine Construction Machine Requirements–Spine construction machine requirements are summarized in Table 3.4-13. The three machine types (beam machine, joint assembly machine and bus deployment machine) can be combined into a single machine as illustrated in Figure 3.4-65.

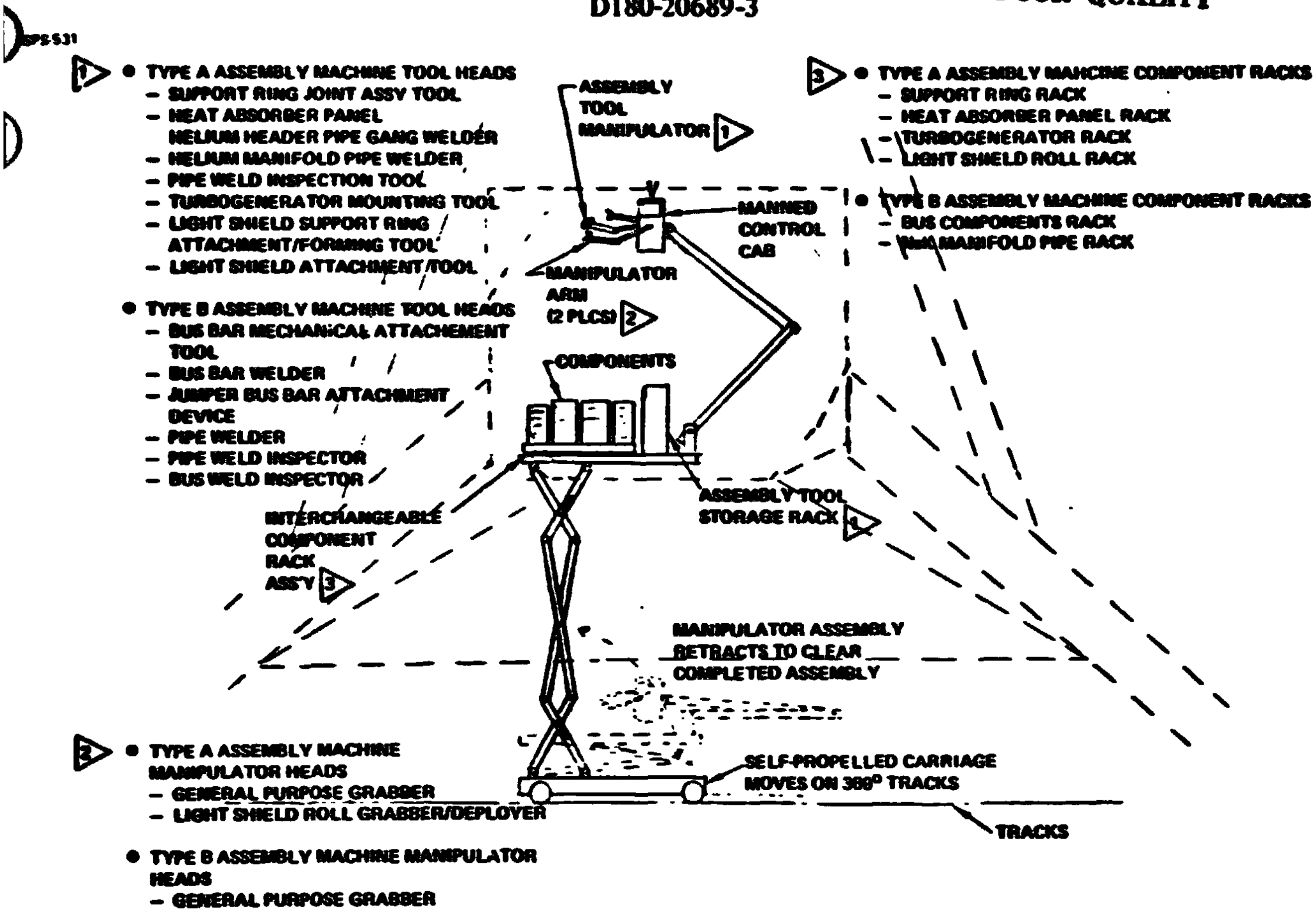

Figure 3.4-63 Cavity Assembly Machine

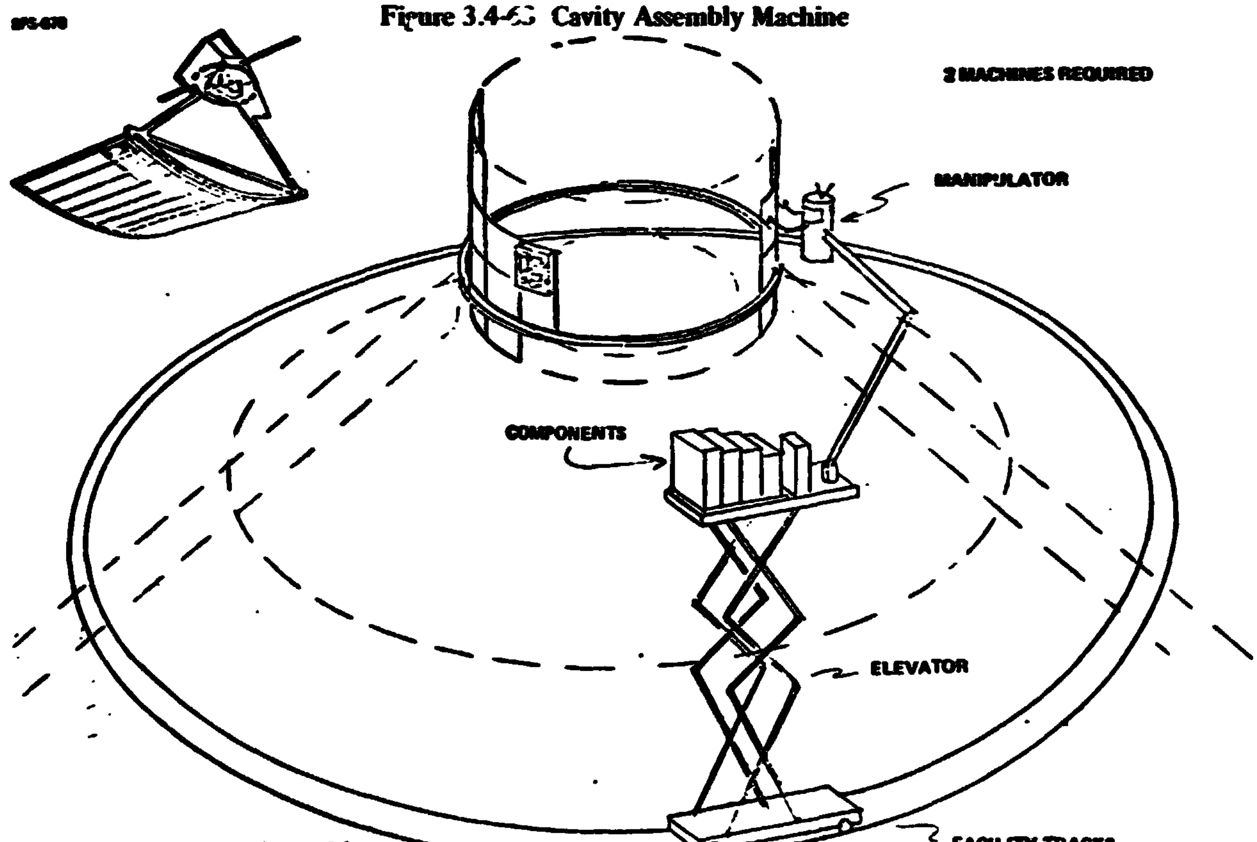

Figure 3.4-64 Cavity Assembly

TABLE 3.4-13
Spine Assembly Machine Requirements

Beam Machine

- 20 beam machine
- Moving machine
- Wrap beam with thermal protection wrapping

Joint Assembly Machine

- Transport joint plugs to joint location
- Pick up joint plug and move to joint location
- Align joint plug
- Attach joint plug to beam

Bus Deploymemt Machine

- Transports rolls of bus sheet metal shock and insulated standoffs
- Forms sheet metal into structural shape (2 or 3 busses simultaneously)
- Attach standoff to bus
- Attach standoff to frame
- Weld busses at joints

Beam supports will also be required every 200m. For the type 2 spine, the worst case, 25 of the 20m beam supports will be required.

Radiator/Cavity/Spine Construction Machinery Summary—Table 3.4-14 summarizes the types and quantities of construction machinery required to make the radiators, cavity and spine subassemblies.

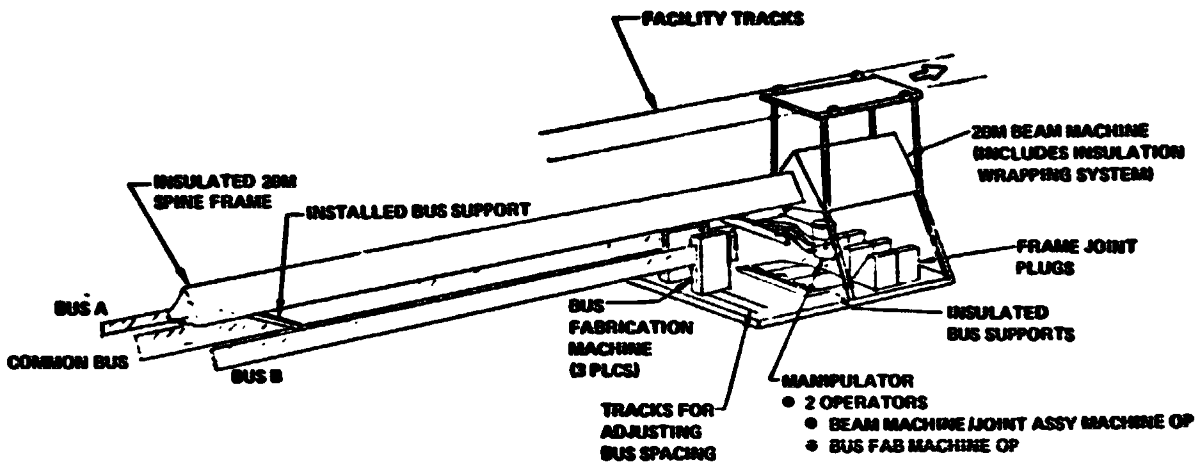

Figure 3.4-65 Spine Assembly Machine

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Table 3.4-14 Radiator/Cavity/Spine Construction Machinery Summary

RADIATOR CONSTRUCTION MACHINERY	NO. REQ'D.
• 20M STRAIGHT BEAM, INSULATION WRAPPING, MOVING BEAM MACHINE	1
• 10M STRAIGHT BEAM, INSULATION WRAPPING, MOVING BEAM MACHINE	1
• 20M BEAM SUPPORT	60
• 10M BEAM SUPPORT	47
• 5M BEAM SUPPORT	4
• JOINT ASSEMBLY MACHINE	1
• RADIATOR ASSEMBLY MACHINE	2
• MANIFOLD ASSEMBLY MACHINE	1
CAVITY CONSTRUCTION MACHINERY	
• CAVITY ASSEMBLY MACHINE – TYPE A	1
• CAVITY ASSEMBLY MACHINE – TYPE B	1
SPINE CONSTRUCTION MACHINERY	
• SPINE ASSEMBLY MACHINE	1
• 20M BEAM SUPPORT	25

Radiation/Cavity/Spine Construction Facility Requirement—Throughout the preceding sections, various facility requirements have been mentioned or implied. These requirements have been summarized in Table 3.4-15. A facility that satisfies these requirements is shown in Figure 3.4-66.

Radiator/Cavity/Spine Construction Manpower Requirements—The manpower required to fabricate the radiator/cavity/spine assembly is summarized in Table 3.4-16. This summary does not include supervisory or support personnel.

TABLE 3.4-15

Radiator/Cavity/Spine Assemblies Facility Requirements

General Requirements

- Radiator, cavity and spine assembly operations must be totally independent of one another so that they can proceed simultaneously.
- Frames must be supported every 200m
- Subassemblies must be supported.
- Facility must be contiguous with reflector facility.
- Fabricate from one side, if possible.
- Must be able to clear construction machines out of way to facilitate indexing of completed satellite module.
- Must provide means to transport personnel to man occupied machines.
- Must provide means to move components/materials from receiving area to use machines.

Radiator Construction Requirements

- Provide for simultaneous operation of all of the radiator construction machinery.
- Provide for installation of the beam supports.
- Provide radiator panel set reload area near each radiator.

Cavity Construction Requirements

- Must be able to operate 2 assembly machines simultaneously.
- Must provide a 360° track located 40 in below lowest point of light shield.
- Provide component rack storage area adjacent to track.

Spine Construction Requirements

- Must be able to operate a spine assembly machine.
- Must provide tracks for spine assembly machine for all spine configurations.
- Provide base installation of beam supports.

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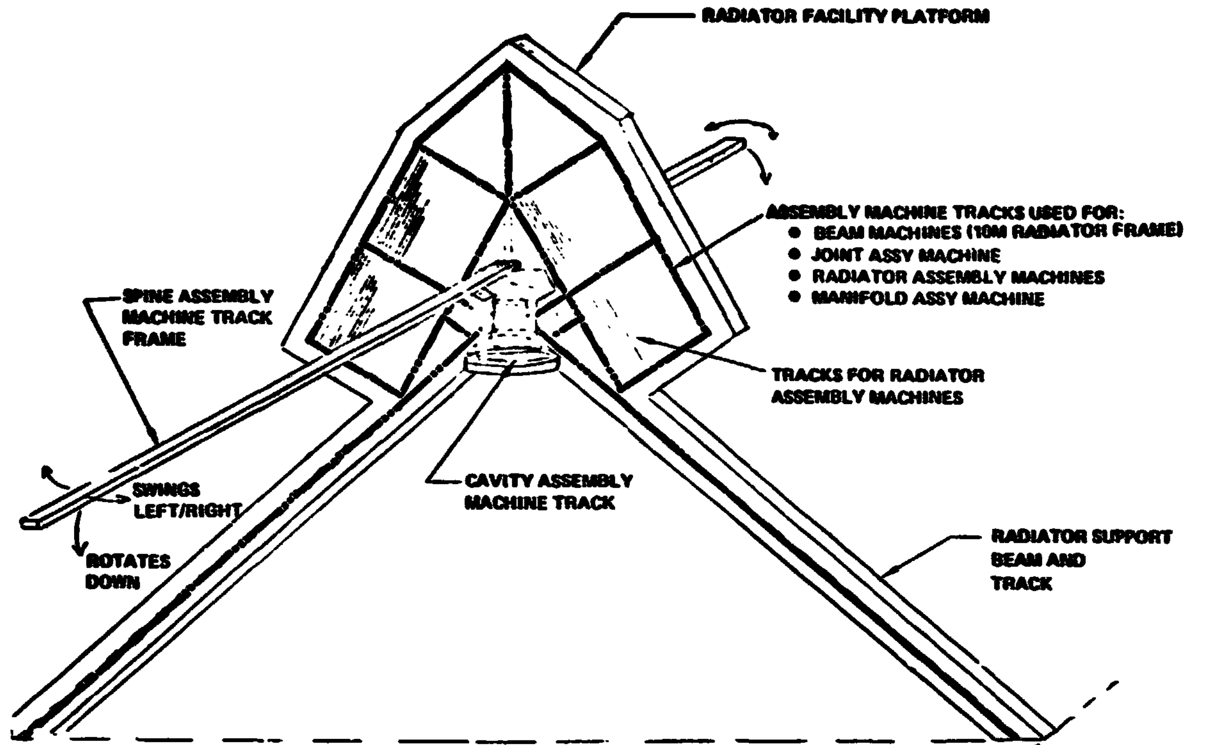

Figure 3.4-66 Radiator/Cavity/Spine Construction Facility

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Table 3.4-16 Radiator/Cavity/Spine Construction Manpower Requirements

JOB TITLE	JOB DESCRIPTION	NO. REQ'D SHIFT	TOTAL REQ'D BASE	WHERE LOCATED
BEAM MACHINE OPERATOR	• CONTROLS LOADING OF BEAM COMPONENTS INTO BEAM MACHINE • INITIATES BEAM FABRICATION • MONITORS BEAM FABRICATION • ISOLATES FAULT CONDITIONS AND ADVISED MAINTENANCE PERSONNEL • CONTROLS INTERFACING WITH BEAM END HOLDING MACHINE IF REQ'D • MONITORS BEAM SUPPORT PLACEMENT/RETRACTION	3	6	• 2 AT CONTROL CENTER • 1 ON SPINE ASSEMBLY MACHINE
JOINT ASSEMBLY MACHINE OPERATOR	• CONTROLS LOADING OF JOINT ASSEMBLY COMPONENTS ONTO MACHINE CARRIAGE • CONTROLS MOVEMENT OF MACHINE ON TRACKS • CONTROLS MOVEMENT OF MANIPULATOR CAB • CONTROLS MANIPULATORS • CONTROLS JOINT FASTENING • ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL	2	4	• 1 ON RADIATOR FACILITY • 1 ON SPINE ASSEMBLY MACHINE
RADIATOR ASSEMBLY MACHINE OPERATOR	• CONTROLS LOADING OF COMPONENTS ONTO MACHINE • INITIATES/MONITORS MACHINE OPERATIONS • ISOLATES FAULT CONDITIONS AND NOTIFIES MAINTENANCE PERSONNEL	8	16	ON RADIATOR ASSEMBLY MACHINE
MANIFOLD ASSEMBLY MACHINE OPERATOR	• CONTROLS LOADING OF COMPONENTS ONTO MACHINE • INITIATES/MONITORS MACHINE OPERATIONS • ISOLATES FAULT CONDITIONS AND NOTIFIES MAINTENANCE PERSONNEL	1	2	ON MANIFOLD ASSEMBLY MACHINE
CAVITY ASSEMBLY MACHINE OPERATOR	• CONTROLS LOADING OF COMPONENTS ONTO MACHINE • INITIATES/MONITORS MACHINE OPERATIONS • ISOLATES FAULT CONDITIONS AND NOTIFIES MAINTENANCE PERSONNEL	2	4	ON CAVITY ASSEMBLY MACHINE

3.4.1.3.3.4 Geo Base Construction Summary—The analysis in this section has indicated how the 16 module thermal engine satellite can be constructed in GEO within a one year time period. The satellite is constructed in 16 satellite module porticns. These 16 modules are fabricated in 5 or 6 module strpings which are then the strings all joined side-by-side to make up the total satellite.

To make the satellite, the construction machinery designed in Table 3.4-17 will be required. The construction machinery operating rates are summarized in Figure 3.4-67.

It will require the personnel designated in Table 3.4-18 to operate the equipment. To this operating personnel quantity will be added the base support, base operations and management personel designated in Table 3.4-19.

The satellite modules are fabricated in the integrated construction facility shown in Figure 3.4-68. Figure 3.4-69 shows the sequence of how the satellites are removed from the facility. Figure 3.4-70 shows how the strings of satellite modules could be assembled into the final configuration.

Table 3.4-17 Thermal Engine Satellite Construction Machinery Summary

SPS-625

ITEM	NO. REQ'D.
20M BEAM MACHINE – STRAIGHT BEAMS, MOVABLE	2
20M BEAM MACHINE – STRAIGHT BEAMS, INSULATION WRAPPED, MOVABLE	1
20M BEAM MACHINE – STRAIGHT BEAMS, INSULATION WRAPPED, MOVABLE	[1]
10M BEAM MACHINE – STRAIGHT BEAMS, INSULATION WRAPPED	1
5M BEAM MACHINE – STRAIGHT BEAMS, MOVABLE	13
5M BEAM MACHINE – STRAIGHT AND CURVED BEAMS, MOVABLE	1
1.5M BEAM MACHINE – CURVED BEAMS, MOVABLE	[2]
20M BEAM SUPPORT	97
10M BEAM SUPPORT	47
5M BEAM SUPPORT	177
BEAM END HOLDING MACHINE	1
JOINT ASSEMBLY MACHINE	2 [3]
FACET DEPLOYMENT MACHINE	37
FACET ASSEMBLY MACHINE	12
RADIATOR ASSEMBLY MACHINE	8
MANIFOLD ASSEMBLY MACHINE	1
CAVITY ASSEMBLY MACHINE – TYPE A	1
CAVITY ASSEMBLY MACHINE – Type B	1
SPINE ASSEMBLY MACHINE	1

[1] INCLUDED ON THE SPINE ASSY MACHINE

[2] INCLUDED ON THE FACET DEPLOYMENT MACHINE

[3] ONE MORE INCLUDED ON THE SPINE ASSEMBLY MACHINE

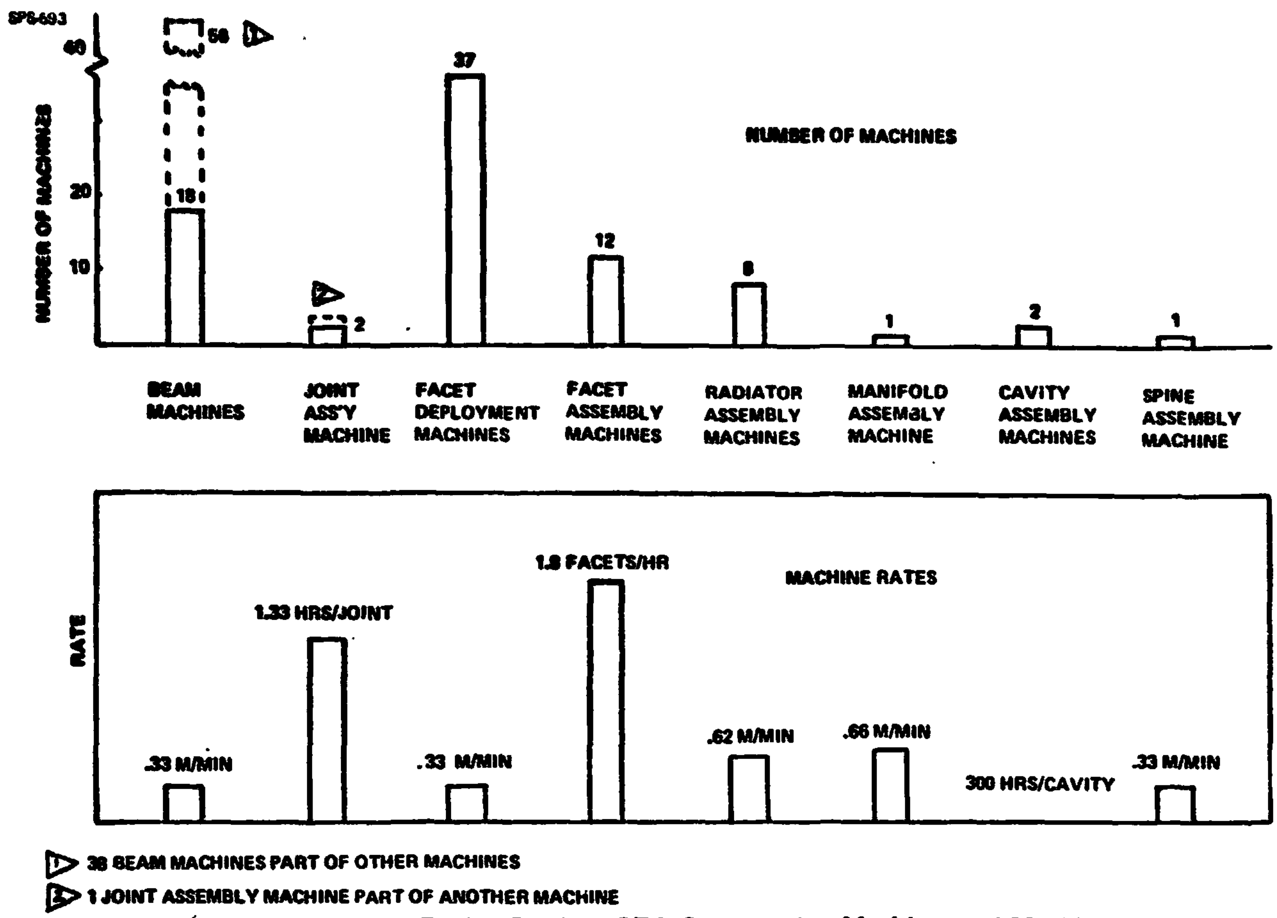

Figure 3.4-67 Thermal Engine Satellite GEO Construction Machines and Machine Rates

SPS-629

Table 3.4-18 Thermal Engine Satellite Machine Operator Summary

OPERATOR	NO. REQ'D. 1
BEAM MACHINE OPERATOR	38
JOINT ASSEMBLY MACHINE OPERATOR	6
FACET DEPLOYMENT MACHINE OPERATOR	74
RADIATOR ASSEMBLY MACHINE OPERATOR	16
MANIFOLD ASSEMBLY MACHINE OPERATOR	2
CAVITY ASSEMBLY MACHINE OPERATOR	4
TOTAL OPERATING PERSONNEL	140

1 NO. REQ'D FOR 2 SHIFT OPERATIONS

Table 3.4-19 Thermal Engine Satellite GEO Base Construction Manpower Requirements

SPS-626

	NO. REQ'D.	
	LEO BASE	GEO BASE
BASE MANAGEMENT	(5)	(7)
SATELLITE CONSTRUCTION		(331)
MANAGEMENT	–	21
MACHINE OPERATORS	–	140
SUBSYSTEMS	–	30
MAINTENANCE	–	68
TEST AND CHECKOUT	–	72
ANTENNA CONSTRUCTION		(84)
BASE OPERATIONS	(82)	(124)
MANANAG		
MANAGEMENT	8	12
DATA PROCESSING	4	6
BASE MAINTENANCE	19	42
TRANSPORTATION	24	10
MATERIALS HANDLING	19	46
COMMUNICATIONS	8	8
BASE SUPPORT	(23)	(64)
MANAGEMENT	5	7
UTILITIES	2	14
HOTEL/FOOD SERVICE	4	24
MEDICAL DENTAL	6	13
SAFETY	2	2
CHAPLAIN	2	2
BASE FLIGHT CONTROL	2	2
BASE SUBTOTAL	110	610
GRAND TOTAL	720	

SPS-668

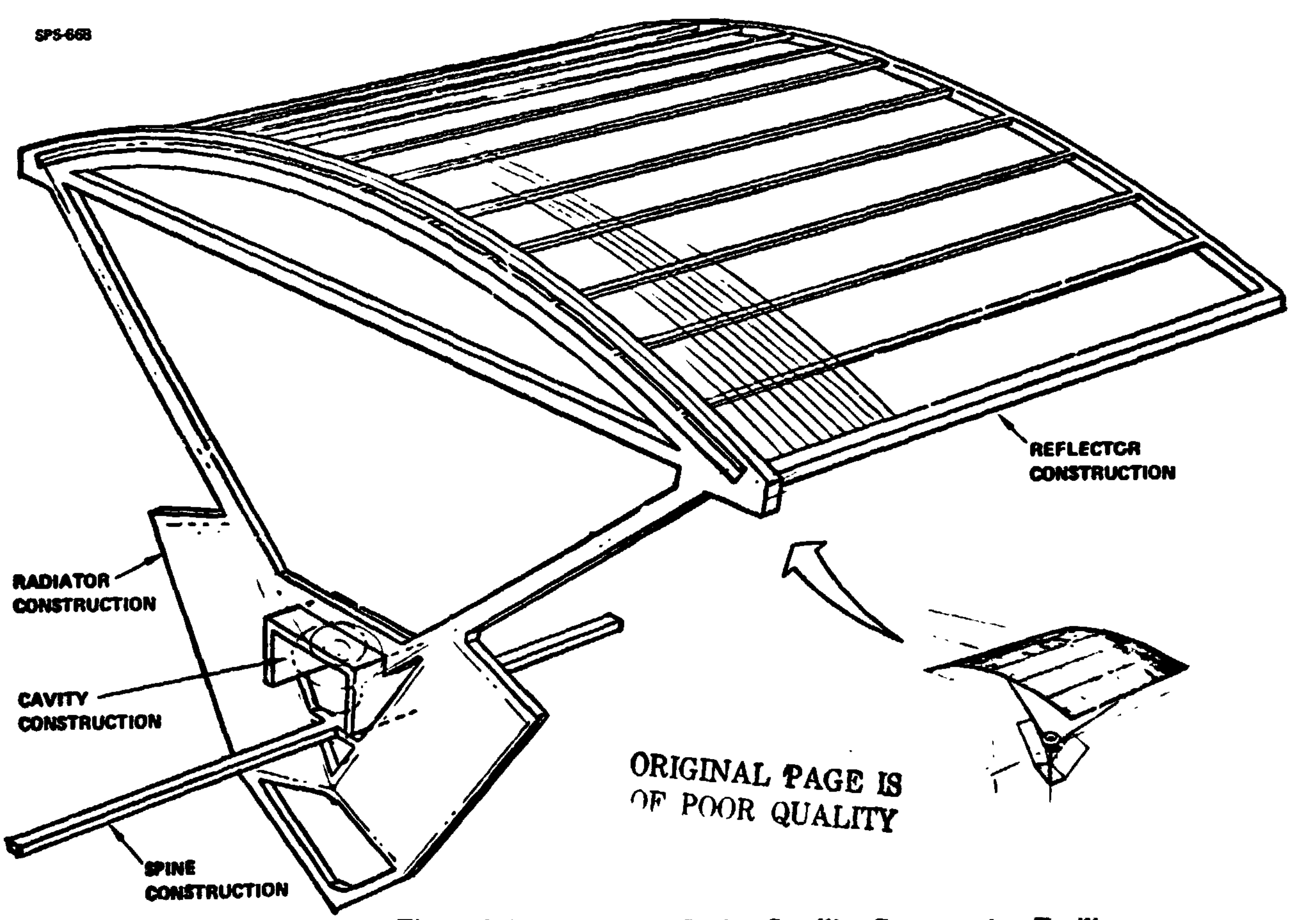

Figure 3.4-68 Thermal Engine Satellite Construction Facility

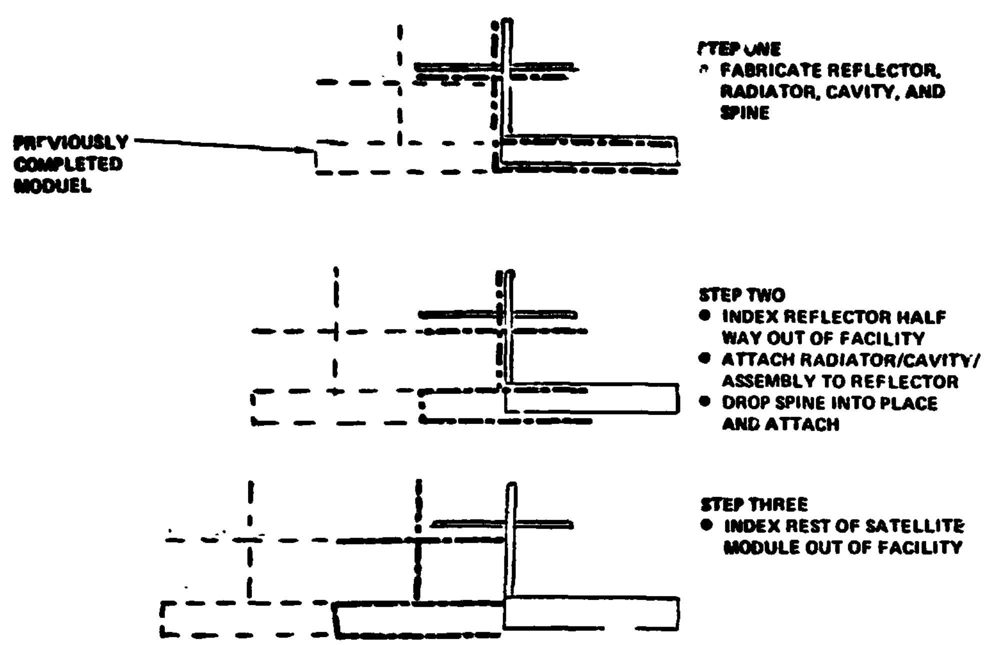

Figure 3.4-69 Satellite Module Construction Sequence

SPS 718

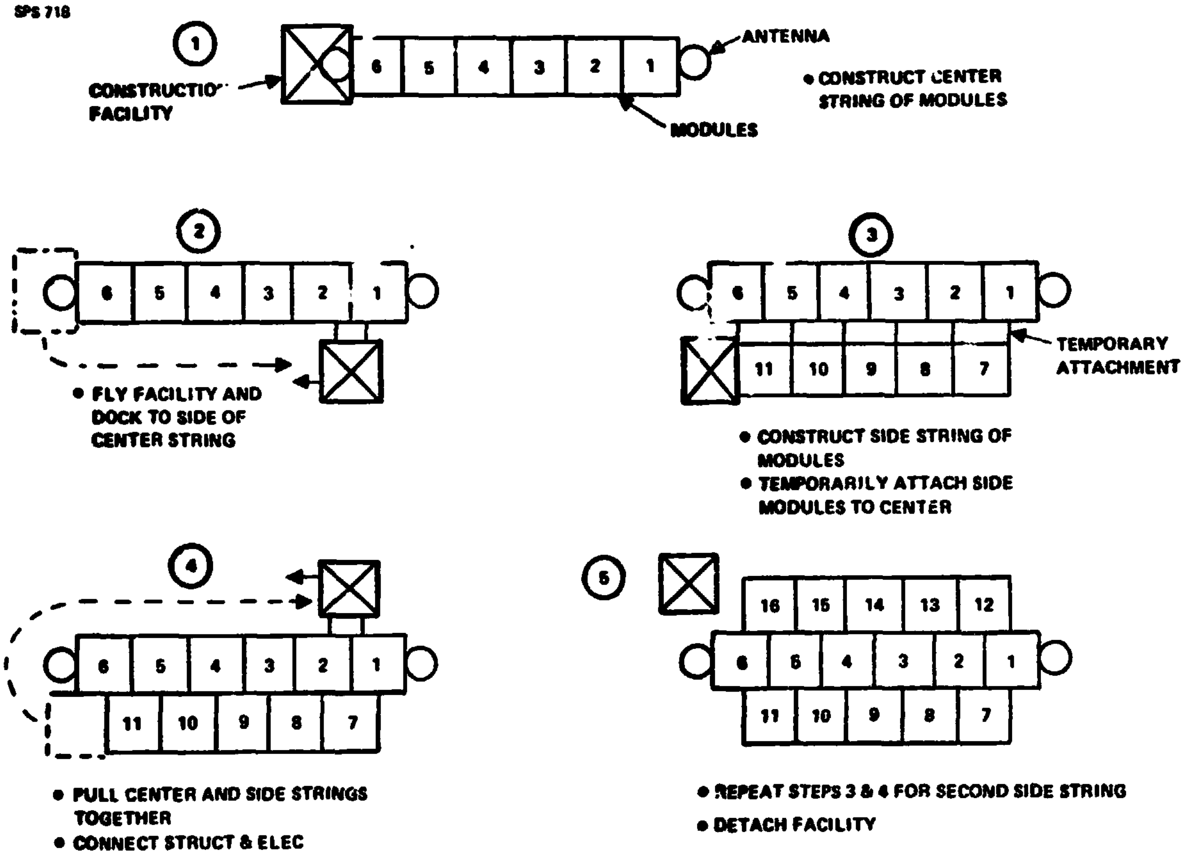

Figure 3.4-70 Assembly Sequence Thermal Engine Satellite

3.4.1.3.4 Thermal Engine Satellite LEO Base Construction Analysis–The significant configuration difference between satellite modules fabricated in LEO vs those fabricated in GEO are the following:

- End frames have to attached on both ends of each module. This adds 16 end frames that do not have to be built when the satellite is constructed in GEO.
- Guy wires have to be strung to support the spine ends and to support the radiator assembly.

If the same facility concept is used, the consequences of the above differences would be that 7.2 days would have to be added to the construction time for each module. This is the amount of time it takes to make the D2 frame (the longest time for any of the end frames), because one set of end frames would have to be added after the module is pushed out of the facility. The other end frames could be made concurrently with D2 by adding one more 20 in beam machine.

An alternative way of fabricating the module using essentially the same facility concept would be to fabricate all of the reflector assembly and then dropping the reflector away from the facility. The completed reflectors would then be free flown to a position where the radiator assembly could be mated with it from a separate radiator facility. This concept would require two additional 20m beam machines, 1 more joint assembly machine, 1 more beam end holder, and 46 more 20m beam supports. To keep within the 20 day/module time allotment, this is the preferred concept.

Table 3.4-20 summarizes the differences and Figure 3.4-71 shows the revised facility concept. Figure 3.4-72 shows the construction sequence.

Table 3.4-21 summarizes the other personnel required at both the LEO and GEO bases and compares the manning to the GEO assembly concept.

3.4.1.3.5 **CR=1 Photovoltaic Satellite GEO Base Construction Analysis**

3.4.1.3.5.1 **Overview**–In this section, the satellite configuration used for analysis is described and the top-level construction timeline is derived. The construction philosophy previously described was used.

Reference Satellite Configuration–The configuration of the CR=1.0, annealed silicon solar cell, photovoltaic satellite is shown in Figure 3.4-73. The framework is composed of an array of 600 m x 600 m structural bays upon which the solar cell blankets are deployed.

Table 3.4-20 Differences Between LEO and GEO Base Construction of the Thermal Engine Satellite

SPS-638

- REVISED FACILITY CONCEPT (SEE FIG. 6-1)
- REQUIRES REFLECTOR ASSEMBLY TO BE FREE FLOWN INTO MATING POSITION TO ADD RADIATOR ASSEMBLY
- ADD
  - 2 20M BEAM MACHINES
  - 1 JOINT ASSEMBLY MACHINE
  - 1 BEAM END HOLDING MACHINE
  - 46 20M BEAM SUPPORTS
- ADD 6 MACHINE OPERATORS

SPS-636

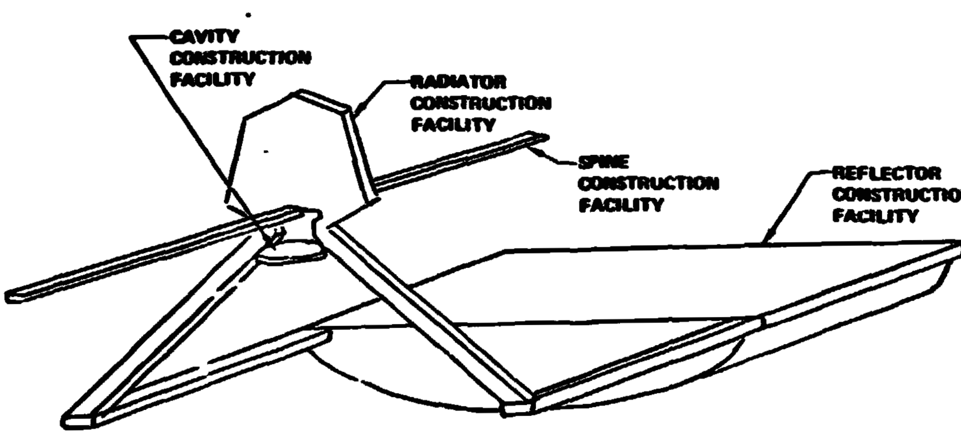

Figure 3.4-71 Thermal Engine Satellite LEO Base Construction Facility

SPS-634

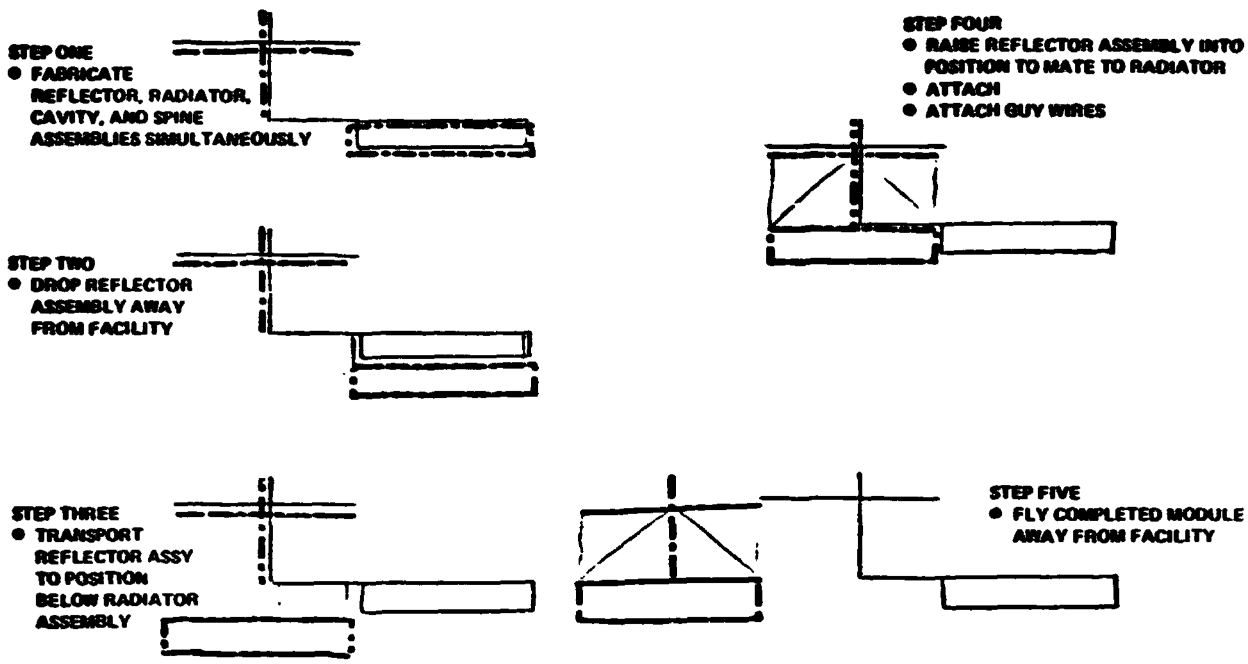

Figure 3.4-72 LEO Construction Satellite Module Major Assembly Sequence

SPS-627

Table 3.4-21 Thermal Engine Satellite Manpower Requirements Summary

	LEO CONSTRUCTION		GEO CONSTRUCTION	
	LEO BASE	GEO BASE	LEO BASE	GEO BASE
BASE MANAGEMENT	(7)	(6)	(5)	(7)
SATELLITE CONSTRUCTION	(327)	(119)		(331)
MANAGEMENT	21	14	–	21
MACHINE OPERATORS	146	28	–	140
SUBSYSTEMS	38	30	–	36
MAINTENANCE	68	30	–	68
TEST & CHECKOUT	72	25	–	72
ANTENNA CONSTRUCTION	(84)	(64)	–	(84)
BASE OPERATIONS	(138)	(68)	(82)	(124)
MANAGEMENT	12	8	8	12
DATA PROCESSING	6	4	4	6
BASE MAINTENANCE	42	19	19	42
TRANSPORTATION	24	10	24	10
MATERIALS HANDLING	46	19	19	46
COMMUNICATIONS	8	8	8	8
BASE SUPPORT	(64)	(37)	(23)	(64)
MANAGEMENT	7	5	5	7
UTILITIES	14	8	2	14
HOTEL/FOOD SERVICE	24	12	4	24
MEDICAL/DENTAL	13	6	6	13
SAFETY	2	2	2	2
CHAPLAIN	2	2	2	2
BASE FLIGHT CONT	2	2	2	2
BASE SUBTOTAL	630	283	110	610
TOTAL	913		720	

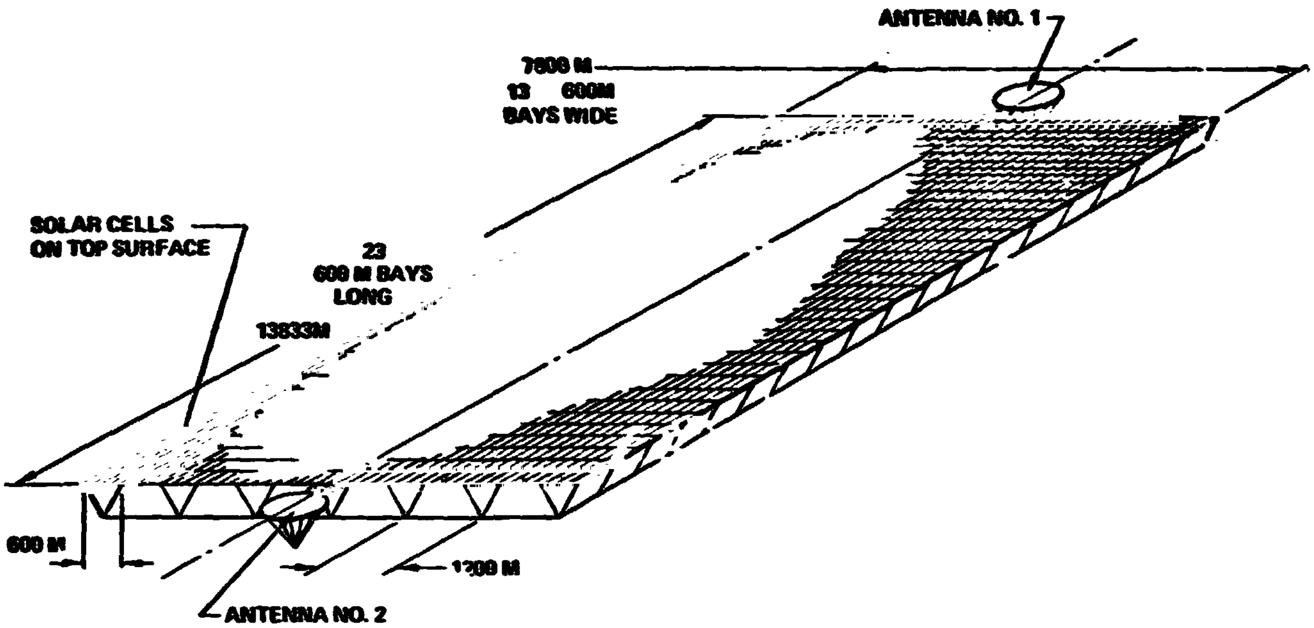

Figure 3.4-73 CR = 1.0, Annealed, Silicon Photovoltaic Satellite Configuration

Top Level Timeline Analysis

Assumption #1 - 365 days total construction time.

Assumption #2 - Allow 30 days for final integration and checkout.

Assumption #3 - Allow 10 days to attach antenna #2 to last end of satellite.

Assumption #4 - Allow 20 days (10 days each end) to fabricate the unique end structures and to deploy bus assemblies.

Assumption #5 - Although 1 day/week is allotted as an off duty day for each crew member, work will be phased to avoid having a common shutdown day each week.

365 days total time; 30 days final integration&checkout; 10 days to install antenna #2; 20 days to fabricate and structures; leaving 305 days total time to construct satellite bays

There are 23 bays, thus allowing 13.26 days per bay.

Nineteen days/bay was used as the basic construction time allotment, leaving 5.98 days unallocated.

The top level timeline is shown in Figure 3.4-74.

In each 13 day time period, the following major operations have to be completed:

- Fabricate frame
  Deploy solar cell blankets
  Index completed portion of satellite 600m

Assumption #6 - Use the satellite indexing rate of 1.1 meter/min.

9.09 hrs are required to index satellite 600m, allowing 276.9 hrs to complete assembly of each bay assuming 22 work hours/day as before.

Assumption #7 - Allow 1 day/bay for machine delays, coordination for indexing, etc., leaving 254.9 hrs/bay for actual assembly work.

255 hrs/bay was used as assembly time allotment

SPS 781

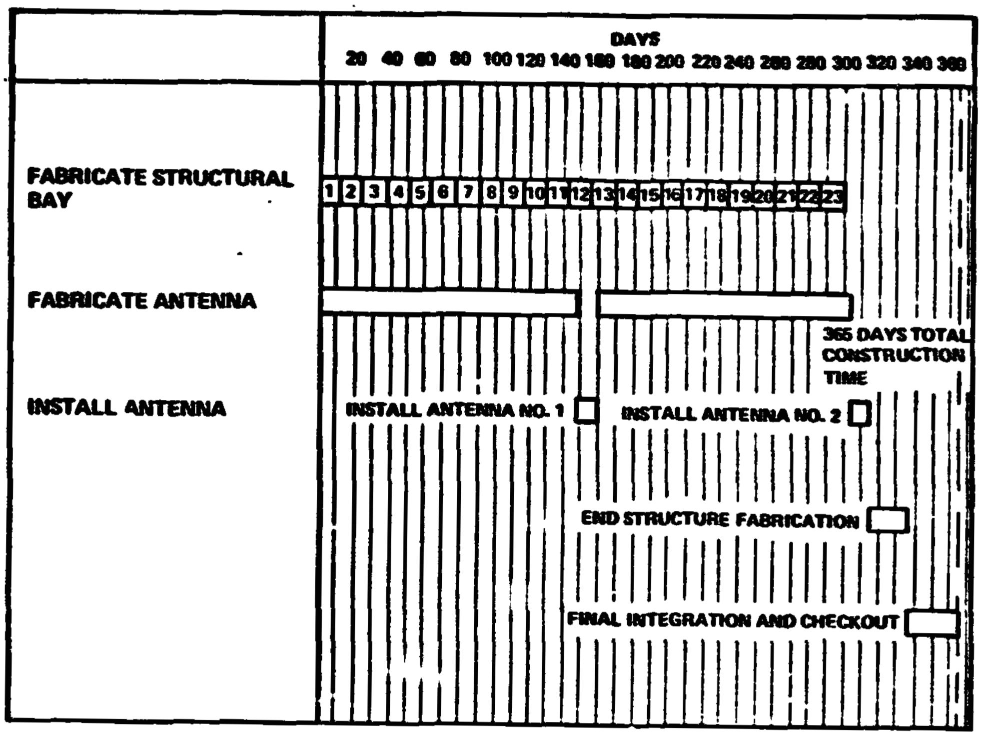

Figure 3.4-74 CR = 1.0, Photovoltaic Satellite GEO Base Top Level Timeline Analysis

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3.4.1.3.5.2 Frame Construction Analysis

Configuration–The satellite frame is shown in detail in Figure 3.4-75. These frame lengths and nomenclatures will be used throughout this report.

All frames are 20 m triangular trusses. The longitudinal members, the D and E frames are continuous, i.e., there are no longitudinal joints. The diagonal (C frames) are lap joined to the longitudinals. The horizontal B frames are made as continuous beams and lap joined to the E frames. The A frames are made outside of their final location, dropped into place and butt joined to the D frames using joint plugs.

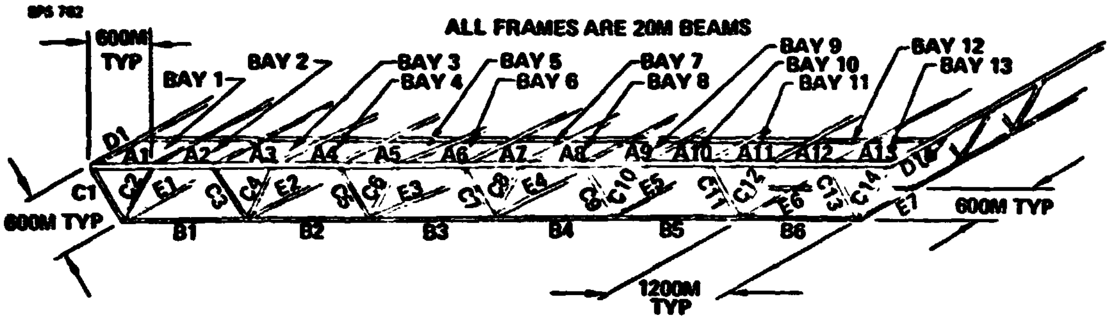
Frame Construction Timeline Analysis–In Section 3.4.1.3.5.1 it was found that 255 hours were needed per longitudinal bay to complete all assembly operations. Table 3.4-22 lists the various frames, their nominal lengths and their fabrication time computed at the .33 in/min beam machine rate used in all previous analyses.

By fabricating the top longitudinal D frames and the A frames first, the solar cell deployment can start as soon as the beam machines have cleared an area large enough for the solar cell deployment machinery to operate. The other frames can then be fabricated while solar cell deployment continues.

In order to start solar cell deployment reasonably early in the 13 day period, it was elected to use 12 beam machine operators so that the A and D frames can be completed within 60 hours.

It would be reasonable to employ the beam machine operators for up to 150 hours per longitudinal bay. Table 3.4-23 shows how 12 operators could be utilized to give each one 150 hours work per day.

Two joint assembly machines, one operating along the A frames and the other operating along the B frames, will be able to keep up with all of the joint assembly work (using the 1 hr/joint assembly time used in previous analyses).



LONGITUDINAL MEMBERS ARE CONTINUOUS

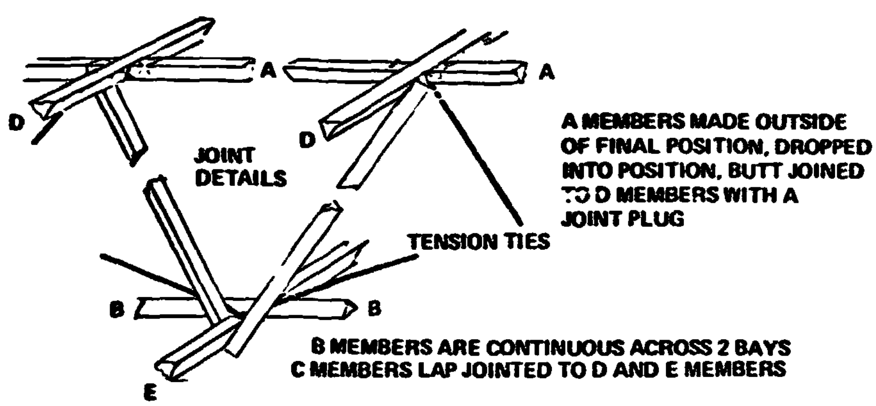


Figure 3.4-75 Satellite Frame Details

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TABLE 3.4-22
Satellite Frame Fabrication Analysis

Frame	Length, m	Fabrication Time Hours
A1	600	30
A2	600	30
A3	600	30
A4	600	30
A5	600	30
A6	600	30
A7	600	30
A8	600	30
A9	600	30
A10	600	30
A11	600	30
A12	600	30
A13	600	30
B1	1200	60
B2	1200	60
B3	1200	60
B4	1200	60
B5	1200	60
B6	1200	60
C1	600	30
C2	600	30
C3	600	30
C4	600	30
C5	600	30
C6	600	30
C8	600	30
C8	600	30
C9	600	30
C10	600	30
C11	600	30
C12	600	30
C13	600	[illegible]
C14	600	30

TABLE 3.4-22 (CONT.)
Satellite Frame Fabrication Analysis

Frame	Length, m	Fabrication Time Hours
D1	600	30
D2	600	30
D3	600	30
D4	600	30
D5	600	30
D6	600	30
D7	600	30
D8	600	30
D9	600	30
D10	600	30
D11	600	30
D12	600	30
D13	600	30
D14	600	30
E1	600	30
E2	600	30
E3	600	30
E4	600	30
E5	600	30
E6	600	30
E7	600	30

TABLE 3.4-23
Beam Machine Operator Schedule

Operator	Frame	Time	Frame	Time	Frame	Time	Frame	Time	Frame	Time	Total Time
1	D1	30	D13	30	C1	30	C2	30	E3	30	150
2	D2	30	D14	30	A8	30	C7	30	C8	30	150
3	D3	30	A1	30	A9	30	C9	30	C10	30	150
4	D4	30	A2	30	C3	30	C4	30	E4	30	150
5	D5	30	A3	30	A10	30	C11	30	C12	30	150
6	D6	30	A4	30	A11	30	C13	30	C14	30	150
7	D7	30	A5	30	C5	30	C6	30	E5	30	150
8	D8	30	A6	30	A12	30	E1	30	E6	30	150
9	D9	30	A7	30	A13	30	E2	30	E7	30	150
10	D10	30	B1/B2	120	-	-	-	-	-	-	150
11	D11	30	B3/B4	120	-	-	-	-	-	-	150
12	D12	30	B5/B6	120	-	-		-	-	-	150

A tension tie cable deployment machine will attach and deploy the cables to the D and E longitudinal members. After the diagonals and horizontal frames are in place, this machine will tension the cables as required.

The integrated timeline for the frame fabrication and assembly will be shown later.

Frame Construction Machinery Requirements

Beam Machines–It was noted that all frames are 20m beams. It was also noted that the longitudinal members are all continuous.

A beam machine will be dedicated to each longitudinal member (21 machines). These beam machines will be movable along facility tracks. These machines will be employed to push the completed satellite bay out of the facility.

The pair of diagonal C frames will be fabricated by a single beam machine located on the facility below the E frame. These beam machines will be rotated so that they can make both diagonal members. Seven of these beam machines are required. The B frames are made as continuous 2 bay-wide lengths from 3 beam machines.

The top horizontal members, the A frames, will be made from facility-mounted beam machines that will make these beams outside of their final, attached locations. Six of these beam machines will be required. They must be able to relocate to different A frame fabrication positions.

Joint Assembly Machines–Two joint assembly machines are required. They operate along the A and B frame paths.

Cable Deployment Machine–This machine must provide the following functional capabilities:

- Transport cable rolls
- Swage end fittings and tensioning devices cable
- Deploy cables
- Cut cables
- Attach end fittings to frame
- Adjust tensioning device

A machine that provides these functions is shown in Figure 3.4-76.

Beam End Holder Machines–The A, B and C beams will be fabricated by stationary beam machines. It will, therefore, be necessary to provide beam end holder machines to support the free end of the beam as the beam is extruded from the beam machines. This machine has been described in previous construction analyses. It will require 12 of these machines.

Beam Support Devices–The beams will be supported every 200m (same as was done in previous construction analyses) by facility mounted, retractable beam supports. These devices have been described previously. It will be necessary to provide 244 20m beam supports.

Frame Construction Machinery Summary–The machinery required to fabricate the frame is summarized in Table 3.4-24.

Frame Construction Facility Requirements–The requirements imposed upon the construction facility to accommodate the frame fabrication work is summarized in Table 3.4-25. As the frame fabrication facility is to be integrated with the solar cell deployment facility, the total facility will be described later.

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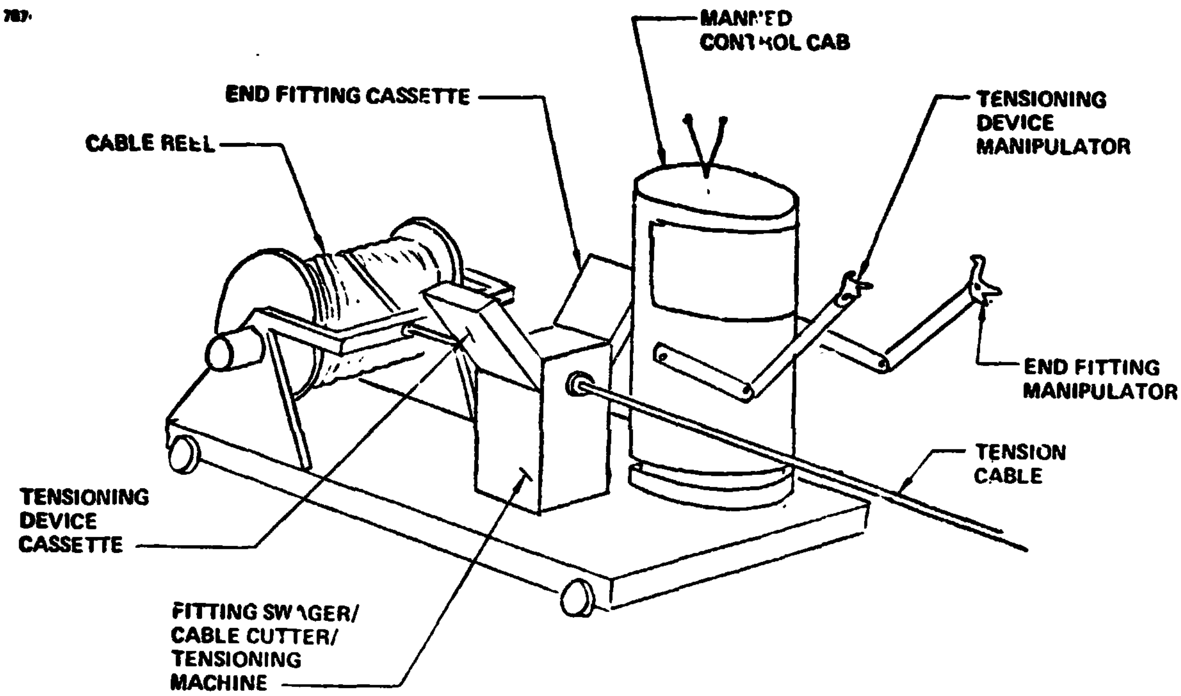

Figure 3.4-76 Cable Deployment Machine

SPS 763

Table 3.4-24 Frame Construction Machinery Summary

- 20M BEAM MACHINES
  - D & E FRAME BEAM MACHINES — 21 REQUIRED
    - MOVE ALONG FACILITY TRACKS
    - USE IN UNISON TO PUSH COMPLETED BAY OUT OF FACILITY
  - C FRAME BEAM MACHINES — 7 REQUIRED
    - ROTATE TO MAKE 2 FRAMES
    - FIXED TO FACILITY
- B FRAME BEAM MACHINES — 3 REQUIRED
  - FIXED TO FACILITY
- A FRAME BEAM MACHINES — 6 REQUIRED
  - FIXED TO FACILITY WHILE MAKING BEAMS
  - MOVE MACHINES TO DIFFERENT A FRAME LOCATIONS

TOTAL BEAM MACHINES — 37 REQUIRED

- JOINT ASSEMBLY MACHINES — 2 REQUIRED
- CABLE DEPLOYMENT MACHINES — 1 REQUIRED
- BEAM END HOLDER MACHINE — 12 REQUIRED
- BEAM SUPPORTS — 244 REQUIRED

TABLE 3.4-25
Frame Fabrication Facility Requirements

- Provide tracks for D and E beam machines.
- Provide tracks for A beam machines to relocate.
- Provide support points for B beam machines.
- Provide rotating support fixture for C beam machines.
- Provide tracks along A and B frames for joint assembly machine access.
- Provide tracks for cable deployment machines.
- Provide beam support device attachment locations.
- Provide access to all machines for parts reloading.

Frame Construction Manpower Requirements—Table 3.4-26 summarizes the crew members required to operate the frame construction machinery. The supervisory and support personnel will be identified later.

3.4.1.3.5.3 Power Collection System Construction Analysis

Configuration—The configuration of the power collection system is shown in Figure 3.4-77. There are three subassemblies: (1) the main collector bus assembly on each end of the satellite, (2) the jumper bus assembly located midway down the length of the satellite, and (3) the solar cell blanket assemblies.

Power Collection System Construction Timeline Analysis

Main Collector Bus Assembly—In the top level timeline analysis, 10 days were allocated to assemble the satellite end frames and the collector bus system on each end of the satellite (20 days total). The end frames can be completed within 120 hours (5.45 days), the time required to make the end B frames.

SPS 764

Table 3.4-26 Frame Construction Manpower Requirements

JOB TITLE	JOB DESCRIPTION	NO. REQ'D SHIFT	TOTAL REQ'D BASE	WHERE LOCATED
BEAM MACHINE OPERATOR	• CONTROLS LOADING OF BEAM COMPONENTS INTO BEAM MACHINE • INITIATES BEAM FABRICATION • MONITORS BEAM FABRICATION • ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL • CONTROLS INTERFACING WITH BEAM END HOLDING MACHINE IF REQUIRED • MONITORS BEAM SUPPORT PLACEMENT/RETRACTION	12	24	FRAME CONSTRUCTION CONTROL CENTER
JOINT ASSEMBLY MACHINE OPERATOR	• CONTROLS LOADING OF JOINT ASSEMBLY COMPONENTS ONTO MACHINE CARRIAGE • CONTROLS MOVEMENT OF MACHINE ON TRACKS • CONTROLS MOVEMENT OF MANIPULATOR CAB • CONTROLS MANIPULATORS • CONTROLS JOINT FASTENING • ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL	2	4	ON THE MACHINE
CABLE DEPLOYMENT MACHINE OPERATOR	• CONTROLS LOADING OF COMPONENTS ONTO MACHINE • INITIATES/MONITORS MACHINE FUNCTIONS • ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL • DRIVES MACHINE WHEN RELOCATING	1	2	ON THE MACHINE

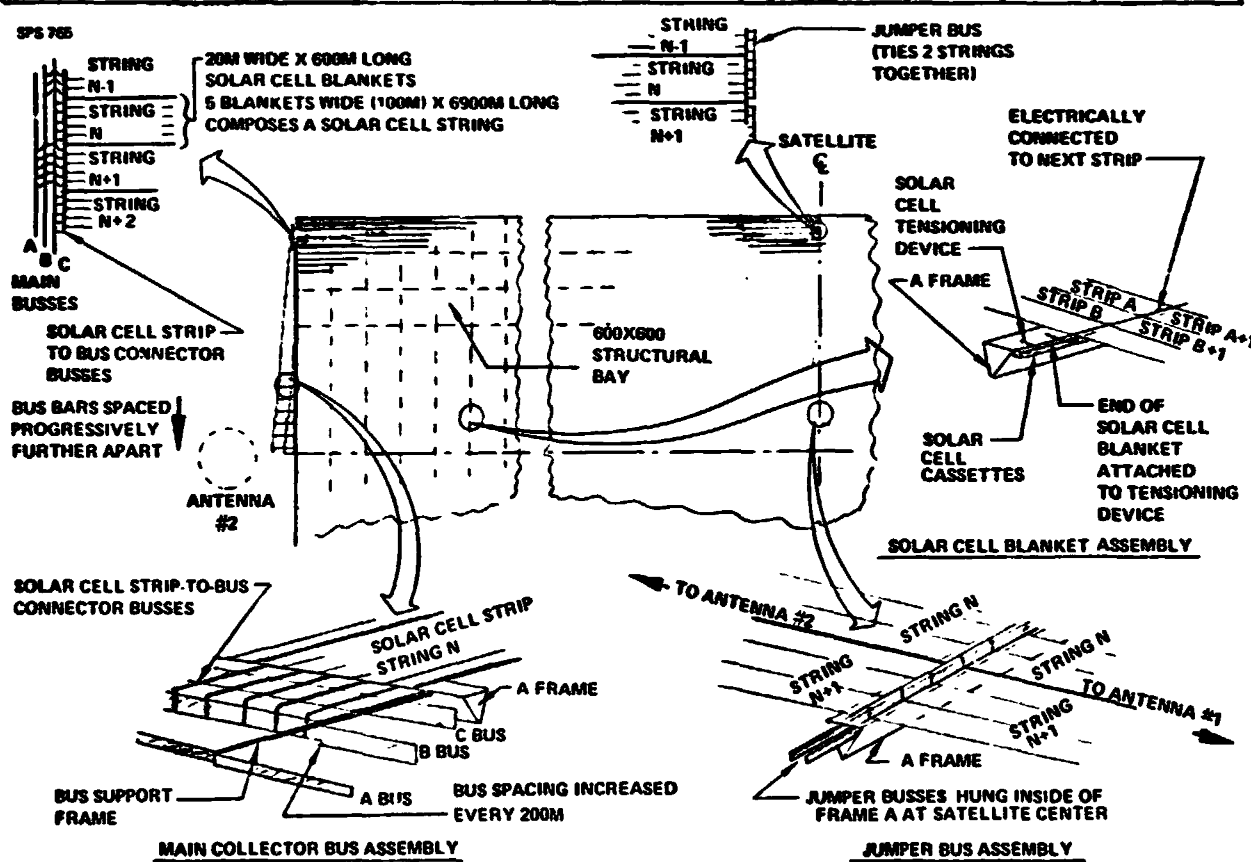

Figure 3.4-77 Power Collection System Configuration

The bus installation can begin simultaneously with the frame fabrication If two bus assembly machines are used, one starting at each outside edge and working toward the center, they must complete deploying/attaching busses within about 8 days in order to allow time for connecting to the slip ring assembly. This requires a bus assembly machine generating rate of 0.[illegible] m/min.

Jumper Bus Assembly—There was an unallocated 5.98 day period that could be used to install the jumper bus assemblies when the satellite is half completed. The joint assembly machine operating on the A frame surface could be used to piece together these jumper busses. This one machine must cover the entire 7900 m width within 6 days; therefore requiring a 1 m/min jumper bus assembly rate.

Solar Cell Deployment—Solar cell blanket deployment is the primary, repetitive operation. To deploy the solar cell blankets, the following operational functions must be accomplished:

- Deploy cannisters containing the 20 m wall solar cell blankets.
- Attach these cannisters to the top of the A frames.
- Deploy solar cell blanket stretchers.
- Attach these stretchers to the top of the A frames.
- Deploy the solar cell blankets 600 m between the corresponding A frames.
- Attach the deployed end of the solar cell blanket to the stretching device.
- Attach the edges of the solar cell blanket to the edge of the adjoining solar cell blanket.
- Make the electrical connection between the end of the deployed solar cell blanket and the electrical connection on the cannister of the solar cell blanket previously deployed in the preceding bay.

Two different types of machines will be used. The first machine, a component deployer, will be used to ferry the cannisters of solar cell blankets and the stretchers out to their intended location where they will then be attached to the frame. The second type of machine, the solar cell deployment machine, will then unroll or unfold the solar cell blanket and stretch it across the bay, and perform the other required functions.

In previous photovoltaic satellite construction analyses, it was found that the component deployment machine was allowed 60 minutes to install a solar cell blanket cannister and a stretcher (includes time required to reposition the machine). There are a total of 390 strips, thus requiring 390 hours.

The first A frames will not be completed until approximately 65 hours into the timeline. This time has to be subtracted from the 255 hours of assembly time available, leaving 190 hours time available to deploy parts and to deploy blankets.

Some time also has to be allowed after the last cannisters/stretchers are attached to the frame so that the solar cell deployment can be accomplished. This leaves an estimated 180 hours time available to install components.

If the machine rate is increased slightly, 2 of the component deployment machines could get the 390 sets of parts attached to the frames within 185 hours.

The solar cell blanket unfolding and stretching across the frames is to be done at the 7.2 m/min deployment rate and 5.65 m/min edge attachment rate previously used in other photovoltaic satellite construction analyses (includes operator productivity factor). Times required are 83.3 minutes to deploy the blanket and 106 minutes to attach edges. Allowing an additional 100 hours to accommodate machine repositioning results in 1335 hours total machine time for the 3% strips.

Solar cell deployment cannot start until after the first set of solar cell cannisters/stretchers are deployed, which will not occur until approximately 61 hours into the timeline (this leaves 194 hours of machine time available), which computes to 6.88 machines required. Therefore, 7 solar cell deployment machines are allocated.

The integrated assembly timeline is shown later.

Power Collection System Construction Machinery Requirements

Bus Assembly Machines–It was determined that two bus assembly machines would be required. These machines have to provide the following functional capabilities:

- Transport rolls of bus material.
- Transport insulated support arms assemblies.
- Roll form the bus material into required deployed shape (3 busses simultaneously).
- Attach busses to support arms.
- Electrically connect solar cell strip-to-bus connector busses.

Component Deployment Machine–It was determined that 2 machines would be required that would provide the following functional capabilities:

- Transport cannisters of solar cell blankets.
- Transport solar cell blanket stretcher devices.
- Attach solar cell blanket cannister to frame.
- Attach solar cell blanket stretcher to frame.
- Electrically connect solar cell blanket to previously deployed blanket.
- Self propelled.
- 1 hr per set installation rate.

Solar Cell Deployment Machine–It was determined that 7 machines would be required that would provide the following functional capabilities:

- Extract solar cell blanket from cannister.
- Deploy the blanket across the 600 in bay.
- Attach end of blanket to stretcher device.
- Attach edge of solar cell blanket to adjoining blanket.
- Operate in close proximity to other machines.
- Self-propelled.

Machine concepts to satisfy these requirements were not developed.

Power Collection System Construction Machinery Summary–The construction machinery items identified in this section are summarized in Table 3.4-27.

Power Collection System Facility Requirements–The requirements imposed on the facility to facilitate the construction operations identified in this section have been summarized in Table 3.4-28.

Table 3.4-27 Power Collection System Construction Machinery Summary

• BUS ASSEMBLY MACHINE	2 REQUIRED
• COMPONENT DEPLOYMENT MACHINE	2 REQUIRED
• SOLAR CELL DEPLOYMENT MACHINE	7 REQUIRED

TABLE 3.4-28
Power Collection System Facility Requirements

- Provide tracks for supporting the bus assembly machines which must operate on both ends of the satellite.
- Provide tracks for supporting the component deployment machines.
- Provide tracks for supporting the solar cell deployment machines.
- Provide means to move the various machines around each other.
- Provide means to perform the power collection system construction operations simultaneously with the frame fabrication.
- Provide access to machines by supply vehicles.
- These facilities must be contiguous with the frame construction facilities.

The facility that satisfies these requirements is described in Section 3.4.1.3.5.4.

Power Collection System Manpower Requirements–The operators required to perform the power collection system assembly operations are described in Table 3.4-29.

3.4.1.3.5.4 Summary

The analysis in this report has described how the CR = 1.0 photovoltaic satellite can be constructed at a GEO base within a 365 day time period. Figure 3.4-78 shows the integrated construction timeline for eacy bay.

To construct this satellite, the construction machinery summarized in Table 3.4-30 will be required. The personnel identified in Table 3.4-31 will be needed or to operate these machines. To these operating personnel will be added the management, support and operations personnel summarized in Table 3.4-32. The satellite will be constructed in the facility shown in Figure 3.4-79, which integrates all of the facility requirements previously identified.

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Table 3.4-29 Power Collection System Construction Manpower Requirements

JOB TITLE	JOB DESCRIPTION	NO. REQ'D SHIFT	TOTAL REQ'D BASE	WHERE LOCATED
BUS ASSEMBLY MACHINE OPERATOR	CONTROLS LOADING OF COMPONENTS ONTO MACHINE CONTROLS MOVEMENT OF MACHINE ON TRACKS CONTROLS MANIPULATORS INITIATES/MONITORS DEPLOY MACHINES ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL	2	4	ON THE MACHINE
COMPONENT DEPLOYER MACHINE OPERATOR	CONTROLS LOADING OF COMPONENTS ONTO MACHINE CONTROLS MOVEMENT OF MACHINE ON TRACKS CONTROLS MANIPULATORS INITIATES/MONITORS DEPLOY MACHINES ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL	2	4	ON THE MACHINE
SOLAR CELL DEPLOYMENT MACHINE OPERATOR	CONTROLS MOVEMENT OF MACHINE ON TRACKS CONTROLS MANIPULATORS INITIATES/MONITORS DEPLOY MACHINES ISOLATES FAULT CONDITIONS AND ADVISES MAINTENANCE PERSONNEL	7	14	AT CONSTRUCTION CONTROL CENTER

SPS 769

Table 3.4-30 CR = 1 Photovoltaic Satellite GEO Base Construction Machinery Summary

- 20M BEAM MACHINES — 37 REQUIRED
- JOINT ASSEMBLY MACHINES — 2 REQUIRED
- CABLE DEPLOYMENT MACHINE — 1 REQUIRED
- BEAM END HOLDER MACHINE — 12 REQUIRED
- BEAM SUPPORT DEVICES — 244 REQUIRED
- BUS ASSEMBLY MACHINES — 2 REQUIRED
- COMPONENT DEPLOYMENT MACHINES — 2 REQUIRED
- SOLAR CELL DEPLOYMENT MACHINES — 7 REQUIRED

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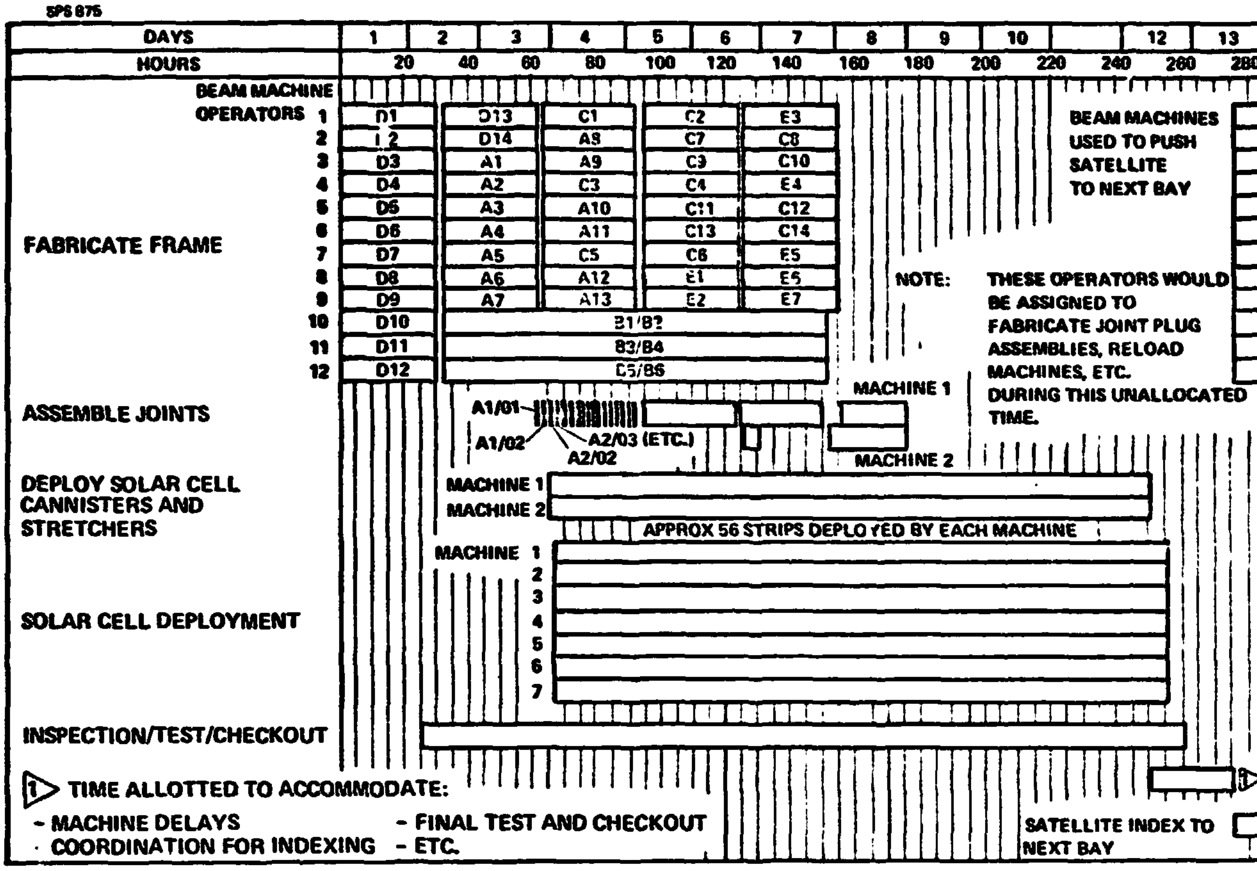

Figure 3.4-78 CR = 1 Photovoltaic Satellite GEO Base Construction Timeline for One Bay

Table 3.4-31 CR = 1 Photovoltaic Satellite GEO Base Machine Operator Summary

SPS 770

BEAM MACHINE OPERATORS	24 REQUIRED
JOINT ASSEMBLY MACHINE OPERATORS	4 REQUIRED
CABLE DEPLOYMENT MACHINE OPERATORS	2 REQUIRED
BUS ASSEMBLY MACHINE OPERATORS	4 REQUIRED
COMPONENT DEPLOYER MACHINE OPERATORS	4 REQUIRED
SOLAR CELL DEPLOYMENT MACHINE OPERATORS	14 REQUIRED
TOTAL	52

Table 3.4-32 CR = 1 Photovoltaic Satellite GEO Base Construction Manpower Summary

SPS 777

	LEO base	GEO base
Base management	(5)	(10)
Satellite construction	—	(220)
Management	—	42
Machine operators	—	52
Subsystems	—	24
Maintenance	—	48
Test and checkout	—	54
Antenna construction	—	(84)
Base operations	(82)	(124)
Management	8	12
Data processing	4	6
Base maintenance	19	42
Transportation	24	18
Materials handling	19	48
Communications	8	8
Base support	(23)	(64)
Management	5	7
Utilities	2	14
Hotel/food service	4	24
Medical/dental	6	13
Safety	2	2
Chaplain	2	2
control	2	2
Totals	110	502
Total	612	

SPS 771

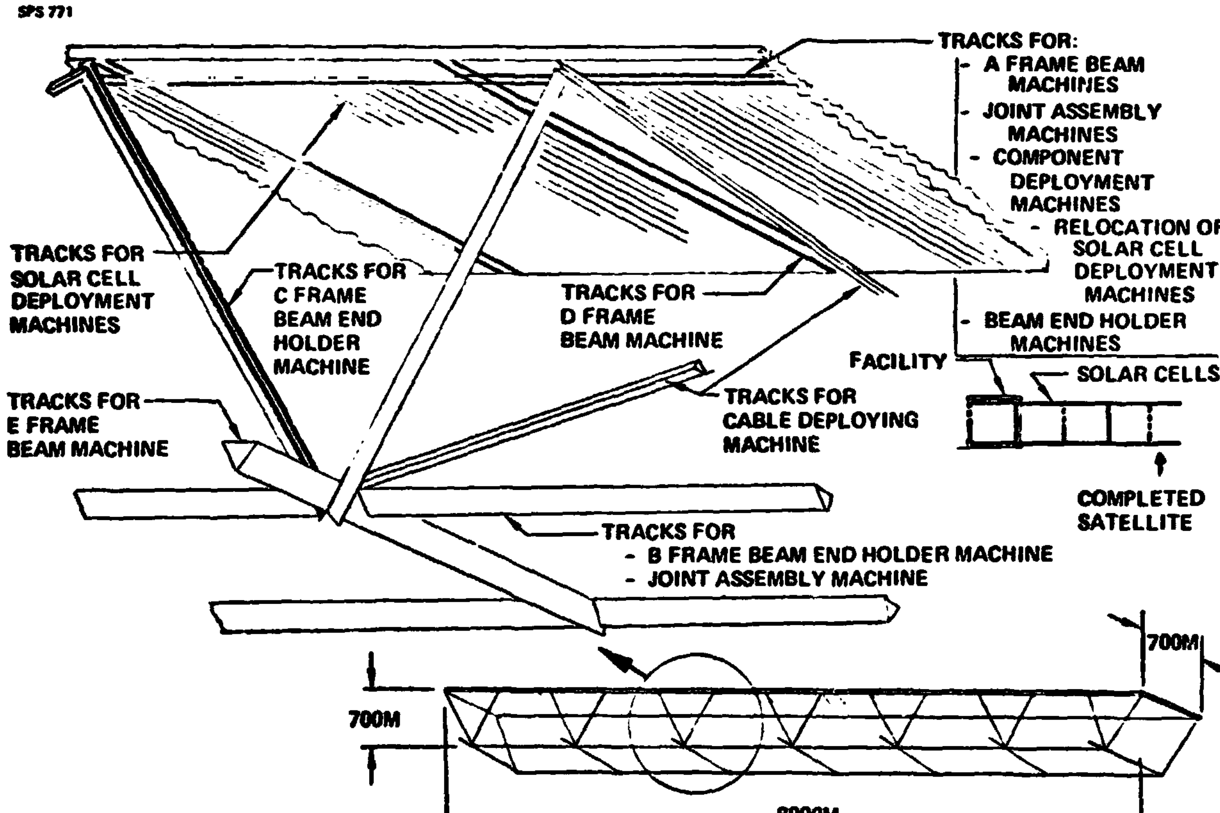

Figure 3.4-79 CR = 1.0 Photovoltaic Satellite GEO Base Construction Facility

3.4.1.3.6 CR=1 Photovoltaic Satellite LEO Base Construction Analysis

3.4.1.3.6.1 Configuration

For LEO base construction, the satellite will be constructed in 1/16-size modules (see Figure 3.4-80), which will then be self-propelled from LEO to GEO.

The 1/16-size module is 1/2 the width of the satellite by 3 bays long (Figure 3.4-81). Approximately 30% of the solar cells will be deployed to provide self-power capability during orbital transfer.

The significant configuration differences between the LEO-constructed modules and the GEO-constructed satellite are summarized in Table 3.4-33.

3.4.1.3.6.2 LEO Construction Timeline Analysis

The same assumptions and top-level timeline that were designated in the CR = 2.0 photovoltaic satellite LEO construction analysis apply here. The net result is that there are 20 days available at LEO to fabricate each module and 20 days at GBO to assemble the modules and deploy the remaining solar cells.

There were approximately 13 days allocated to complete one full-width 600 meter long segment of the satellite. For LEO construction, we have 20 days to build half the width but 3 bays long.

Figure 3.4-82 shows a timeline that depicts how the 3-bay module can be constructed within the 20 day period.

Table 3.4-34 summarizes the number of machines required; Table 3.4-35 summarizes the number of machine operators required. Figure 3.4-83 shows the LEO base facility.

3.4.1.3.6.3 GEO Assembly Timeline Analysis

A satellite module will arrive at the GEO base every 20 days. In a previous section it was noted that there were 17.5 days available to finish deployment of the solar cells after the module had arrived, docked and attached to adjoining modules, and the GEO facility moved into place.

At GEO, after the facility is in position, it will be necessary to perform two major operations: (1) Deploy undeployed solar cell blankets, and (2) anneal the previously deployed solar cells.

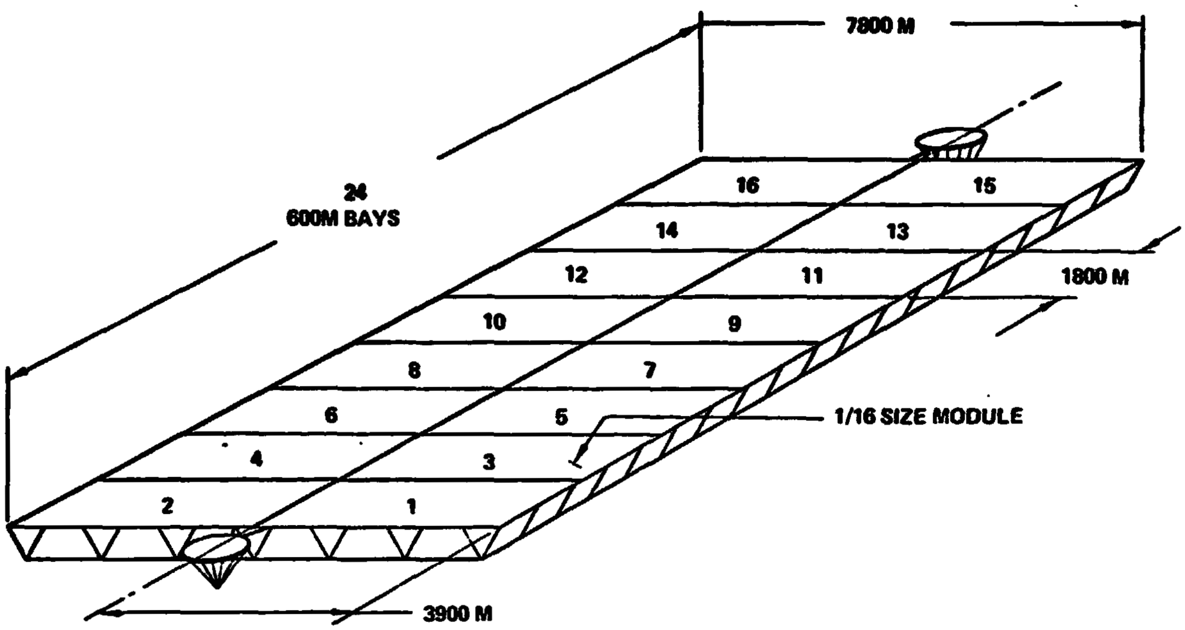

Figure 3.4-80 CR = 1.0, Annealed. Silicon Photovoltaic Satellite Configuration

SPS 790

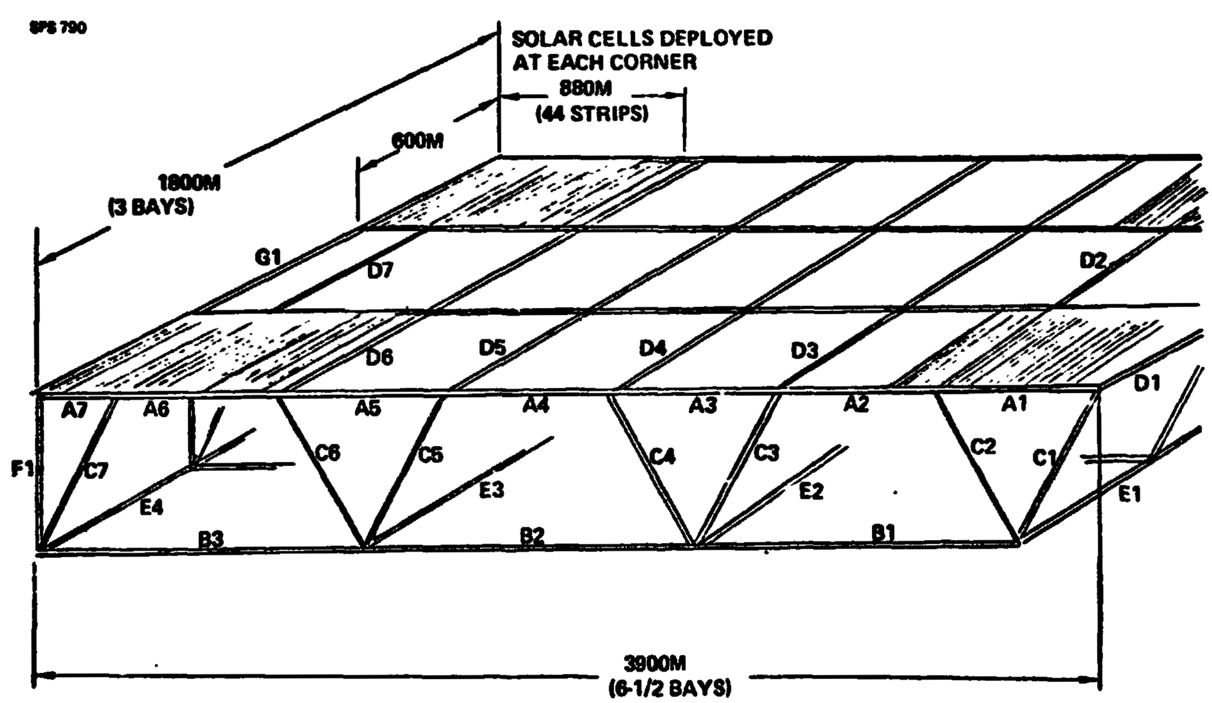

Figure 3.4-81 Satellite Module Configuration

Table 3.4-33 Differences Between LEO and GEO Constructed Satellites

SPS 773

- THE LEO-CONSTRUCTED SATELLITE HAS THESE DIFFERENCES:
  - CONSTRUCTED IN 1/16 - SIZE MODULES
  - 4 VERTICAL BEAMS (F1) AND 3 LONGITUDINAL BEAMS (G1) HAVE TO BE ADDED.
  - 30% OF SOLAR CELLS DEPLOYED AT LEO, THE REST HAVE TO BE DEPLOYED AT GEO.

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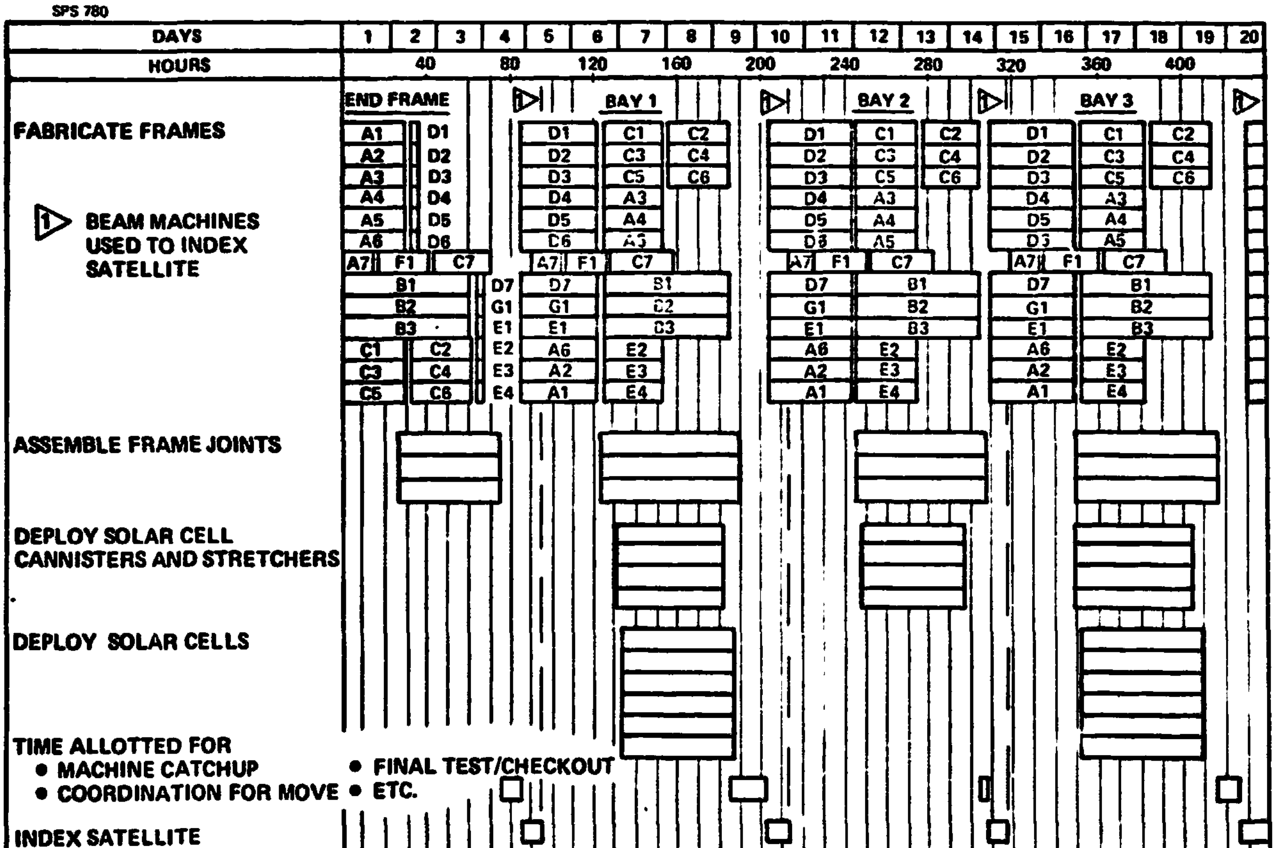

Figure 3.4-82 CR = 1 Silicon Photovoltaic Satellite Module LEO Base Construction Timeline

Table 3.4-34 CR = 1 Photovoltaic Satellite LEO Base Construction Machinery Summary

SPS 774

o	20M BEAM MACHINES	23 REQUIRED
o	JOINT ASSEMBLY MACHINES	3 REQUIRED
o	COMPONENT DEPLOYMENT MACHINES	4 REQUIRED
o	SOLAR CELL DEPLOYMENT MACHINES	6 REQUIRED
o	BEAM END HOLDER MACHINES	13 REQUIRED
o	BEAM SUPPORT DEVICES	150 REQUIRED

Table 3.4-35 CR = 1 Photovoltaic Satellite LEO Base Machine Operator Summary

SPS 775

		NO REQUIRED FOR 2 SHIFTS
o	BEAM MACHINE OPERATOR	46
o	JOINT ASSEMBLY MACHINE OPERATOR	6
o	BUS ASSEMBLY MACHINE OPERATOR	4
o	COMPONENT DEPLOYMENT OPERATOR	8
o	CABLE DEPLOYMENT OPERATOR	2
o	SOLAR CELL MACHINE OPERATOR	12
		78

SPS 791

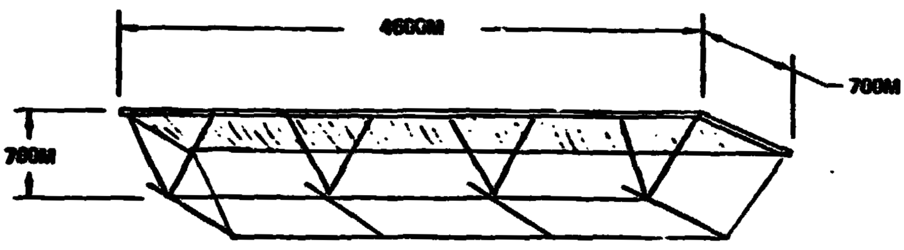

Figure 3.4-83 CR = 1 Photovoltaic Satellite LEO Construction Base Facility

Annealing analyses (Section 3.2.5) indicated a requirement for 10 annealing machines that will operate at a rate of 200 m^2/hr. (This requirement was based on the need to anneal the entire satellite solar cell array within a one year time period). Therefore, it was presumed that these 10 macines will be available to anneal the previously deployed solar cells. The area deployed for transfer is 2.1 X 10^6 m^2, requiring 1056 machine hours to anneal the solar cells.

If the facility has to be indexed two times at 9.1 hrs/index, 332 hours are estimated available to anneal and to deploy solar cells; also 3.18 annealing machines are required.

Therefore, there is plenty of time available to anneal the deployed solar cells using the 10 annealing machines that will be available.

The solar cell deployment will require 190 min/strip.

In Bay 1 and Bay 3, there are 107 strips to be deployed, 339 machine hours are required per bay, requiring 2 solar cell deployment machines.

Attach of the frames of the module to the adjacent modules within the 1 day allotted, requires 4 joint assembly machines.

The GEO base construction machinery is summarized in Table 3.4-36.

The GEO base machine operators required are summarized in Table 3.4-37.

The GEO facility required to support these assembly operations is shown in Figure 3.4-84.

3.4.1.3.6.4 Summary

The total numbers and types of construction machinery required to construct the CR = 1 satellite at LEO and GEO are summarized in Table 3.4-38 and these are compared to the requirements for GEO base construction. The number and types of machine operators required are summarized in Table 3.4-39 and these are compared to the personnel required for GEO base construction.

Table 3.4-36 CR = 1 Photovoltaic Satellite GEO Base Construction Machinery Summary

SPS 776

o	ANNEALING MACHINES	4 REQUIRED (10 AVAILABLE)
o	SOLAR CELL DEPLOYMENT MACHINES	2 REQUIRED
o	JOINT ASSEMBLY MACHINES	4 REQUIRED

Table 3.4-37 CR = 1 Photovoltaic Satellite GEO Base Machine Operator Summary

SPS 768

	NO REQUIRED FOR 2 SHIFTS
ANNEALING MACHINE OPERATOR	8
SOLAR CELL DEPLOYMENT MACHINE UP	4
JOINT ASSEMBLY MACHINE UP	8

SPS 782

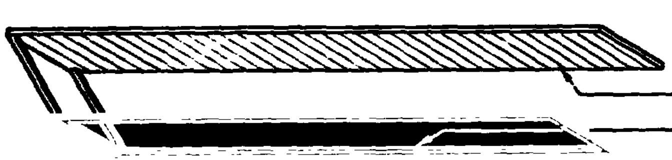

Figure 3.4-84 CR = 1 Photovoltaic Satellite GEO Base Facility

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Table 3.4-38 CR = 1 Photovoltaic Satellite Construction Machinery Summary and Comparison

SPS 778

MACHINE	LEO CONSTRUCTION LEO BASE	LEO CONSTRUCTION GEO BASE	GEO CONSTRUCTION GEO BASE
o 20M BEAM MACHINE	23		37
o JOINT ASSEMBLY MACHINE	3	4	2
o CABLE DEPLOYMENT MACHINE	1		1
o COMPONENT DEPLOYMENT MACHINE	4		2
o SOLAR CELL DEPLOYMENT MACHINE	6	2	7
o BEAM END HOLDER MACHINE	13		12
o BEAM SUPPORTS	150		244
o ANNEALING MACHINE		4	

Table 3.4-39 CR = 1 Photovoltaic Satellite Manpower Summary

SPS 781

	LEO construction		GEO construction	
	LEO base	GEO base	LEO base	GEO base
Base management	(10)	(5)	(5)	(10)
Satellite construction	(186)	(95)	—	(220)
Management	46	22	—	42
Machine operators	78	20	—	52
Subsystems	12	15	—	24
Maintenance	28	16	—	48
Test and checkout	22	22	—	54
Antenna construction	(84)	(54)	—	(84)
Base operations	(133)	(68)	(82)	(124)
Management	12	8	8	12
Data processing	6	4	4	6
Base maintenance	42	19	19	42
Transportation	24	10	24	10
Materials handling	46	19	19	46
Communications	8	8	8	8
Base support	(64)	(37)	(23)	(64)
Management	7	5	5	7
Utilities	14	8	2	14
Hotel/food service	24	12	4	24
Medical/dental	13	6	6	13
Safety	2	2	2	2
Chaplain	2	2	2	2
control	2	2	2	2
Totals	477	259	110	502
Total	736		612	

3.4.2 Crew Scheduling Concept

3.4.2.1 Introduction

Crew scheduling includes consideration of in-orbit staytime, weekly schedule (work days/off duty days), daily schedules (hours of work/day), work/rest cycles (work hours/rest hours), and the number of work shifts per day. These considerations are all insensitive to satellite type of construction location.

Section 3.4.2.2 defines the specific problems to be addressed in this report. Section 3.4.2.3 describes the reference data that was used. Section 3.4.2.4 is an analysis of the data. Section 3.4.2.5 describes the recommended crew scheduling concept.

3.4.2.2 Problem Statement

On-Orbit Stay Time--Derive a recommended on-orbit stay time.

Weekly Work Schedule--Derive a recommended nominal "weekly" work schedule, i.e., how many consecutive work and rest days.

Daily Work Schedule--Derive a recommended daily work/rest schedule.

Number of Shifts--Derive a recommended number of work shifts.

3.4.2.3 Resource Data Review

A survey was conducted to accumulate data that pertain to the problem areas:

Space Flight Experience Data

a. Karpox & Bodrov report that the results of both experimental and actual space flight experience enables one to recommend the following distribution of the daily time budget:

8 hours work (4 hour shift max.)
8 hours uninterrupted sleep
2 hours 15 min. for eating
- 1st breakfast 30 min.
- 2nd breakfast 30 min.
- lunch 30 min.
- supper 45 min.

45 min. for personal hygiene
2 hours 30 min. for personal time and active rest

The interval between the 1st and 2nd breakfasts should not exceed 3 to 4 hours, between second breakfast and lunch - 3 to 4 hours, between lunch and dinner - 4.5 to 6 hours, between dinner and first breakfast 10 to 10.5 hours.

b. Johnson, et al. reported on the medical findings of Skylab. The longest duration spaceflight was 84 days. Medical evidence established the fact that man is fully qualified for in-orbit missions of this duration. Appropriate nutrition, programmed adequate sleep, work, exercise, and recreation periods, and suitable work areas must be provided.

On-Earth Experience Data

The Alaskan oil pipeline construction project offers a potential source of pertinent data due to the long-term, isolated, harsh environment, construction program that it entailed. The following data were obtained (see Kreshak):

	Height of Construction	Now
Daily Work Schedule	-10 hours/day -up to 14 hours/day for high priority short-term projects	-9 hours/day
Weekly Work Schedule	-7 days/week	-6 days on/1 day off
On-site duty time	-8 weeks on/2 weeks off (majority use this schedule)	-8 weeks on/ -2 weeks off
Number of shifts	-2 shifts/day	-2 shifts/day

Experimental Studies

a. Chiles, et al, reported on 13 investigations carried out as a part of an 8-year program of research on the performance effects of various work/rest cycles during confinement in a simulated aerospace vehicle crew compartment.

It was found that a man can work 12 hours per day on a 4-hours work/4-hours rest schedule for periods of at least 30 days. Subjects reported that for a given number of hours per day of duty, duty periods longer than 4 hours would have resulted in performance degradation, especially if the total duration of the mission were extended. Subjects working 12 hours per day on a 4/4 (work/rest) schedule can maintain generally higher levels of performance than subjects working 16 hours per day on a 4/2 schedule. When subjects are highly motivated, performance over a period of 30 days on a 4/4 schedule is indistinguishable from the levels maintained by subjects following a 4/4/4/12 schedule. 16 hours per day on a 4/2 schedule appears to be the maximum feasible number of hours per day a man can work for extended periods of time. The 4/4 work/rest schedule with confinement was no more demanding than a normal 8-hour split-shift work day without confinement.

b. Nicholson has shown that improved performance has not been demonstrated in persons carrying out complex tasks in which absence of circardian activity is associated with a fully adopted sleep pattern. It is considered unreasonable, in the light of present knowledge, to insist on forcing space crews to terrestial rythmns.

Other Related Studies

a. Shields (Boeing IR&D) performed a preliminary assessment of how a solar power satellite be constructed. The study was based on an assumed 8 hour work shift, maximum of 3 shifts per day, and a 6-day work week. No rationale for these assumptions was presented.

b. JSC-11568 (Chapter V.C. Construction Operations) presented a candidate staffing plan and offered an initial method of employing the proposed cadre of personnel by developing the logic leading toward a work/cost cycle for the construction crews. Certain assumptions were made that pertain to the scheduling topic at hand:

- Nominal construction activity will continue on a 3-shift 24 hr/day basis
- Crew stay time limited to 180 days
- Sufficient personnel are required to staff 4 shifts

Based on these assumptions an 8 day rotation cycle (consisting of 6 working days followed by a housekeeping day and then an off-day) was proposed for the 6-month stay time. No rationale is presented for the basic assumptions.

c. VonTiesenhausen analyzed the functional organizational structure of a 50-man space station. He recommends a 3-shift operation to provide the necessary safety and functional readiness of the base. The tour of duty would depend upon the mission. He recommends a common day of rest for all with only critical systems being monitored.

d. Shumate indicated that 90 day in-orbit stay time can be committed to now but that 6 months is probably feasible. The constraint is a bone decalcification problem associated with prolonged weightlessness.

e. Nelson reviewed the literature on sleep loss, work-rest schedules, and performance. Amongst the findings were many of the results reported above. One additional statement was that permanently assigned day-night shift workers generally perform more effectively than do workers who rotate day and night shifts frequently.

f. Gardner, et al, investigated earth-based analogs to the space station environment. They reported the following:

  - Tektite II (an aquanaut habitat) mission of 20 days was not long enough to impact crew performance.

  - The optimum tour of duty on nuclear submarines appears to be 60-70 days, with a high percentage of volunteers for repeat missions. Some missions have approached 90 days duration.

  - Antarctic tours of duty, using a select volunteer crew, typically can maintain good performance throughout the 1 year mission, but with a low percentage of volunteers for repeat missions.

  - Arctic radar sites, using non-volunteers who are not subjected to screening, typically have difficulty in maintaining good performance by all personnel over the one year mission. The percentage of volunteers for repeat missions is very low.

3.4.2.4 Analysis

The data described in the preceding section has been summarized and collated in Table 3.4-40.

Table 3.4-40 Crew Scheduling Reference Data Summary

PROBLEM STATEMENT	DATA	REFERENCE
2.1 On-Orbit Stay Time	- Skylab missions of 84 days show that man is fully qualified for missions of this duration.	Johnson, et al
	- Alyeska pipeline experience has shown that 8 weeks on/2 weeks off is acceptable.	Kreshok
	- 30 day experiments show no degradation of performance if 4/4 schedule used	Chiles, et al
	- Assumed in-orbit stay time of 180 days	JSC
	- 90 day stay time can be committed to, but a 6 month stay time is probably feasible.	Shumate
	- Tektite II mission shows 20 days does not effect performance	Gardner
	- Nuclear submarine optimistic stay time is 60 - 70 days but some missions have approached 90 days.	"
	- Antartic stay time of 1 year can maintain good performance.	"
	- Artic radar site missions of 1 year have shown degradation of performance	"
2.2 Weekly Work Schedule	- Alyeska pipeline construction was conducted on 7 day/week basis for 8 weeks. They have backed off to 6 day/week now that major construction finished.	Kreshok
	- 30 day experiments had no days off	Chiles, et al
	- 6 day work week assumed for preliminary powersat construction analysis	Shields
	- An 8 day schedule was derived (6 days work, 1 day housekeeping, 1 day off duty)	JSC
	- Common day of rest recommended for 50-man space-crew.	Von Tiesenhausen

Table 3.4-40 (continued)

PROBLEM STATEMENT	DATA	REFERENCE
2.3 Daily Work Schedule	- Recommend this schedule 8 hours work (max 4 hour shift) 8 hours uninterrupted sleep 2 hrs. 15 min. eating 45 min. personal hygiene 2 hrs 30 min. personal time and rest	Karpov & Bodrov
	- Alyeska pipeline construction used 10 hours/day except for occassional 12 hours/day during height of construction. Now using 9 hours/day.	Kreshok
	- Results of 8 years of study -12 hours/day using 4/4 schedule for at least 30 days results in no significant performance degradation. -Subject prefer 4 hours shifts. -Based on a 30 day study, a 4/4 schedule is indistinguishable from a 4/4/4/12 schedule. -16 hours/day on a 4/2 schedule maximum feasible for extended periods. -4/4 schedule in confinement no more demanding than 8 hours without confinement.	Chiles, et al
	- If fully adapted sleep pattern is established it is not necessary to maintain a terrestrial accordian activity schedule	Nicholson
	- Assumed 8 hours shift	Shields
	- Assumed 8 hours shift	JSC
	- Recommend 8 hours shift	Von Tiesenhausen
	- Workers more effective if kept on a single shift	Nelson
2.4 Number of Shifts	- Alyeska uses 2 shifts/day	Kreshok
	- Assumed 3 shifts	Shields
	- Assumed 3 shifts	JSC
	- Recommends 3 shifts	Von Tiesenhausen

3.4.2.4.1 Derivation of Min/Max Schedule Constraints

On-Orbit Stay Time

The Alyeska experience has shown that most personnel were anxious to take the 2 weeks R&R after an 8 week continuous stay. This should be a clue that there may be psychological problems associated with longer stays in an isolated environment. The nuclear submarine experience confirms this. Studies of the psychological and psychiatric problems associated with long-term confinement (see Fraser, Romanov, and Leonov) has shown that very little is known about these effects when the mission duration exceeds 70 days (the length of the longest experiement). The 84-day Skylab experience has shown that at least a small crew can work effectively for this length of time. All researchers agree that the crew members will have to be psychologically/psychiatrically screened and that working units will have to be composed of compatible personnel for long-term missions.

Based on the available data, it seems reasonable to establish 90 days as the minimum in-orbit stay time (this is medically and psychologically/psychiatrically practical). Experiments should be conducted to see if 180 days can be feasible as a maximum.

Weekly Work Schedule

The available data shows that a continuous 7 day/week schedule would be tne maximum if the total time before R&R did not exceed 8 weeks. A 6 day week, with either one or two days off, would be the minimum weekly work schedule.

Daily Work Schedule

The available data shows that a maximum of a 4 hour work duty period is the best choice. There are several options of the total work time per day: 8, 10, and 12 hour total work periods are feasible per day. The workers should maintain a regular work schedule. Each crewmember must be allotted a total of 8 hours sleep. This sleep period could be continuous or be in two 4-hour sleep periods. A 24 hour day is not necessarily a requirement.

Number of Shifts

The available data do not dictate a choice of the number of shifts. This can be derived from consideration of the daily work schedules.

3.4.2.4.2 Alternative Work Schedules

Figure 3.4-85 shows 3 alternative 8-hour daily schedules. Figure 3.4-86 shows 3 alternative 10-hour schedules (note that one of the schedules involves a 20-hour "day"). Figure 3.4-87 shows 2 alternative 12-hour daily schedules.

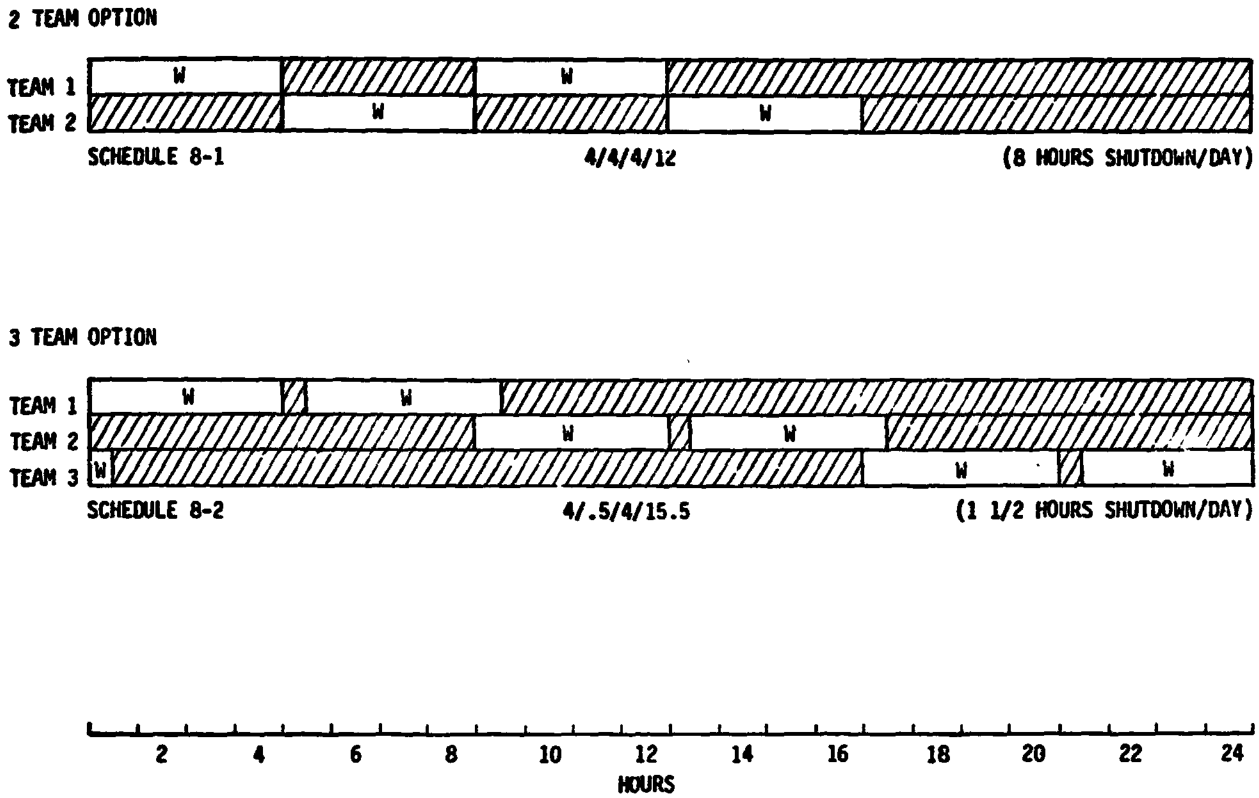

Figure 3.4-85 8-Hour Work/Day Schedule Options

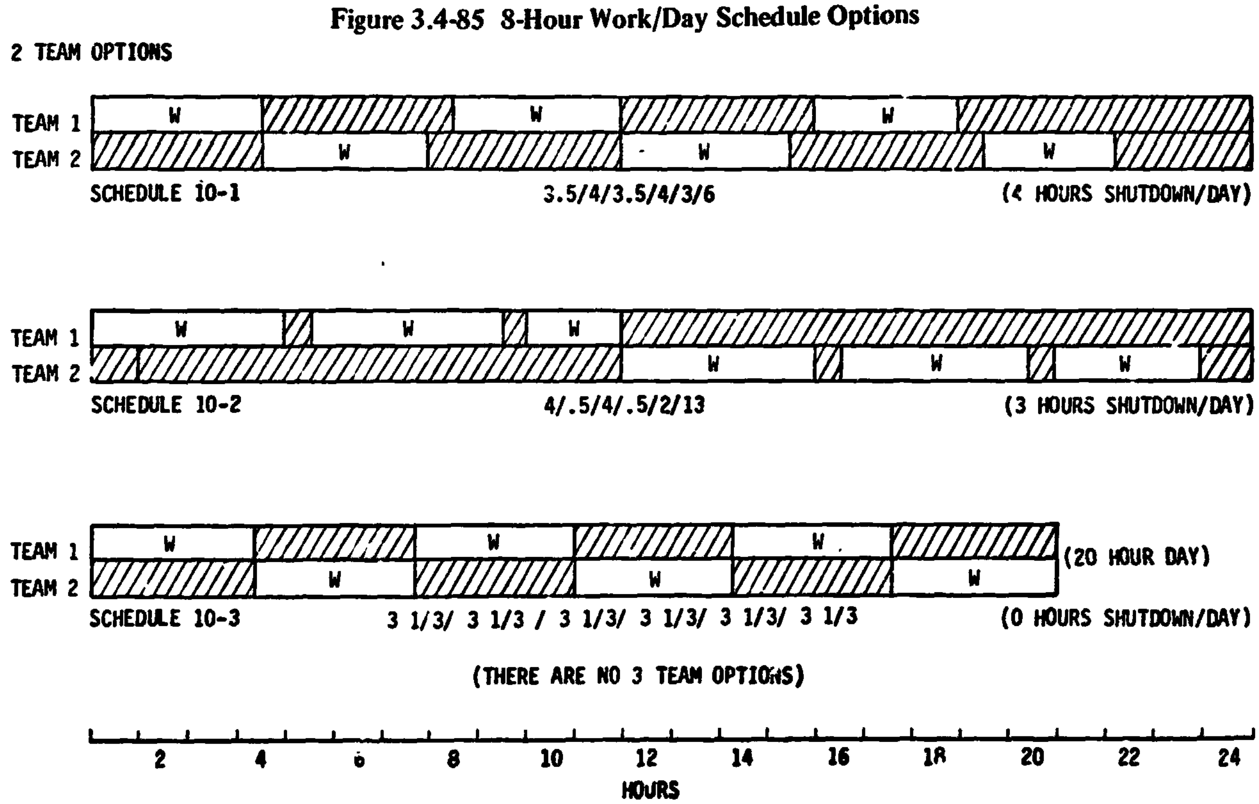

Figure 3.4-86 10-Hour Work/Day Schedule Options

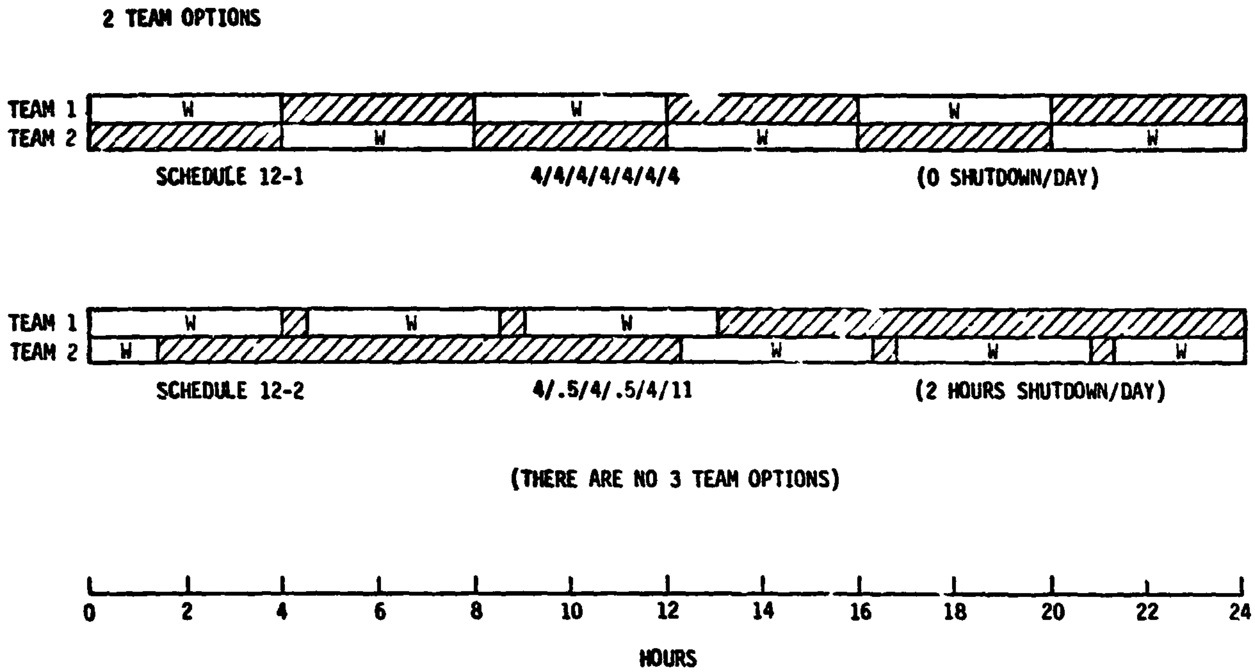

Figure 3.4-87 12 Hours Work/Day Schedule Options

Table 3.4-41 shows 7-day schedules for 2 and 3 shift operations. Table 3.4-42 shows 2 and 3 shift weekly schedules for a 6 day on/1 off schedule. Table 3.4-43 shows the 2 and 3 shift "weekly" schedules for a 6 on/2 off schedule.

Taking into account all of the allowable daily/weekly/stay-time alternatives, there are 40 options. Each of these alternatives are allowable based upon the available data. However, some of these options can be ruled out as impractical. For instance, it would not make sense on a 45-day stay-time to employ any scheduling option that results in any down-time.

Table 3.4-44 was constructed to provide a means of showing all of the realistic options and to evaluate the relative costs associated with the ptions. It is shown that there are 18 options that merit consideration.

3.4.2.4.3 Cost Analysis

To attempt to arrive at a way of comparing the relative merits of the 18 selected scheduling options, a preliminary cost analysis was performed. Figure 3.4-88 shows that there are 5 major cost centers that contribute to total crew operations cost: (1) crew transportation, (2) crew supplies, (3) crew facilities, (4) crew salaries, and (5) down-time cost. This figure shows how various crew-related factors influence these cost centers.

The preliminary cost analysis is focused on the transportation and down-time costs.

3.4.2.4.3.1 Down-time Costs

The various schedules result in two types of down-time: (1) down-time incurred every day due to the various work/rest cycles and number of shifts (see Figures 3.4-85, -86, and -89 and Table 3.4-44 and (2) down-time incurred when the weekly schedule dictates a whole day off (see Table 3.4-41, -42, -43, and -44). The totals of these two down-time contributions are figured over a year's time. Each day of down-time is estimated as \$1.06 million in cost. The total cost of down-time for each option was shown in Table 3.4-44.

3.4.2.4.3.2 Crew Transportation Cost

To compute the crew transportation costs, it was necessary to make the following assumptions:

For LEO Construction (Photovoltaic Satellite)

- There are a total of approximately 500 jobs that need to be staffed uring each shift.

Table 3.4-41 7 Day/Week Schedule Options

NOTE - 7 DAY/WEEK SCHEDULE ADVISABLE ONLY FOR 45 DAYS OR LESS STAYTIME

2 SHIFTS/DAY

DAYS	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
TEAM 1	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W
TEAM 2	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W

(0 DAYS SHUTDOWN/WEEK)

3 SHIFTS/DAY

DAYS	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
TEAM 1	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W
TEAM 2	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W
TEAM 3	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W	W

(0 DAYS SHUTDOWN/WEEK)

WEEKS: 1 2 3

DAYS: 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23

W = WORK DAY
OD = OFF DUTY DAY

Table 3.4-42 6 On/1 Off Weekly Schedule Options

3 SHIFTS/DAY

DAYS	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21
TEAM 1	W	W	W	W	W	W	OD	W	W	W	W	W	W	OD	W	W	W	W	W	W	OD
TEAM 2	W	W	W	W	W	W	OD	W	W	W	W	W	W	OD	W	W	W	W	W	W	OD
TEAM 3	W	W	W	W	W	W	OD	W	W	W	W	W	W	OD	W	W	W	W	W	W	OD

(1 DAY SHUTDOWN/WEEK)

(THERE ARE NO PRACTICAL SHIFTS/TEAMS COMBINATIONS THAT RESULT IN ZERO SHUTDOWN DAYS)

WEEKS: 1 2 3

DAYS: 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21

W = WORK DAY
OD = OFF DUTY DAY

Table 3.4-43 6 On/2 Off Weekly Schedule Options

2 SHIFTS/DAY

DAYS	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
TEAM 1	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD
TEAM 2	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD

(2 DAYS SHUTDOWN/WEEK)

(THERE ARE NO 2 SHIFT/DAY - TEAM COMBINATIONS THAT RESULT IN ZERO OR ONE SHUTDOWN DAYS)
(THIS OPTION WILL NOT BE USED)

3 SHIFTS/DAY

DAYS	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
TEAM 1	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD
TEAM 2	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD	W	W
TEAM 3	W	W	OD	OD	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD	W	W	W	W
TEAM 4	OD	OD	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W	OD	OD	W	W	W	W	W	W

(0 DAY SHUTDOWN/WEEK)

WEEKS: 1 (days 1–8), 2 (days 9–16), 3 (days 17–24)

W = WORK DAY
OD = OFF DUTY DAY

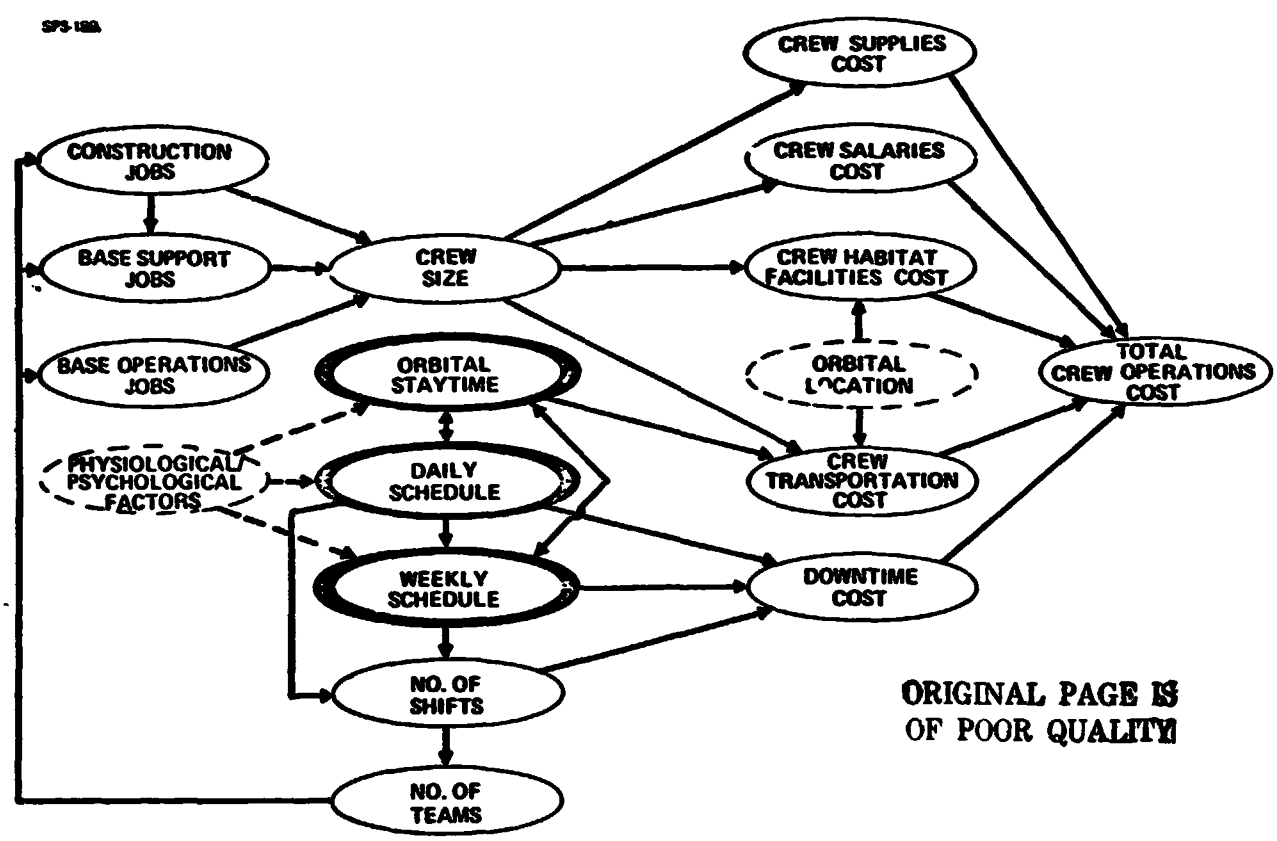

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Figure 3.4-88 Crew Operations Cost Model

Table 3.4-44 Crew Scheduling Options Cost Analysis

OPTION	IN-ORBIT STAYTIME	WEEKLY SCHEDULE	DAILY SCHEDULE: WORK HRS DAY	DAILY SCHEDULE: WORK/REST CYCLE	NO. OF SHIFTS	NO. OF TEAMS	HOURS DOWN PER DAY	DAYS DOWN PER WEEK	NO. TEAM ROTATIONS YEAR
1	45 DAYS	7 DAYS/WEEK	10	3 1/3 / 3 1/3 / 3 1/3 ..	2	2	0	0	16
2			12	4/4/4/..	2	2	0	0	16
3	90 DAYS	6 ON/1 OFF	8	4/4/4/12	2	2	8	1	8
4				4/.5/4/15.5	3	3	1 1/2	1	12
5			10	3.5/4/3.5/4/3/6	2	2	4	1	8
6				4/.5/4/.5/2/13	2	2	3	1	8
7				3 1/3 / 3 1/3 / 3 1/3 ..	2	2	0	1	8
8			12	4/4/4..	2	2	0	1	8
9				4/.5/4/.5/4/11	2	2	2	1	8
10		6 ON/2 OFF	8	4/.5/4/15.5	3	4	1 1/2	0	16
11	180 DAYS	6 ON/1 OFF	8	4/4/4/12	2	2	8	1	4
12				4/.5/4/15.5	3	3	1 1/2	1	6
13			10	3.5/4/3.5/4/3/6	2	2	4	1	4
14				4/.5/4/.5/2/13	2	2	3	1	4
15				3 1/3 / 3 1/3 / 3 1/3 ..	2	2	0	1	4
16			12	4/4/4..	2	2	0	1	4
17				4/.5/4/.5/4/11	2	2	2	1	4
18		6 ON/2 OFF	8	4/.5/4/15.5	3	4	1 1/2	0	8

OPTION	CREW TRANSPORTATION COSTS: LEO BASE 65.5 MILLION/ ROTATION	CREW TRANSPORTATION COSTS: GEO BASE @ $74 MILLION/ ROTATION	DOWNTIME COSTS: TOTAL DAYS HOURLY DOWNTIME YEAR	DOWNTIME COSTS: TOTAL FULL DAYS DOWNTIME YR	DOWNTIME COSTS: TOTAL DOWNTIME YR	DOWNTIME COSTS: COST @ $1.06 MILLION/DAY	LEO COST	GEO COST	Δ
1	1048	1184	0	0	0	0	1048	1184	136
2	1048	1184	0	0	0	0	1048	1184	136
3	524	592	120	52	172	183	707	775	68
4	786	888	22.5	52	74.5	79	865	967	102
5	524	592	60	52	112	119	643	711	68
6	524	592	45	52	97	103	627	695	68
7	524	592	0	52	52	55	579	647	68
8	524	592	0	52	52	55	579	647	68
9	524	592	30	52	82	87	611	679	68
10	1048	1184	22.5	0	22.5	24	1072	1208	136
11	262	296	120	52	172	183	445	479	34
12	393	444	22.5	52	74.5	79	472	523	51
13	262	296	60	52	112	119	381	415	34
14	262	296	45	52	97	103	365	399	34
15	262	296	0	52	52	55	317	351	34
16	262	296	0	52	52	55	317	351	34
17	262	296	30	52	82	87	349	383	34
18	524	592	22.5	0	22.5	24	548	616	68

- 350 of these jobs would be at the major LEO construction sites.
- 150 of these jobs would be at the final LEO assembly location.

For GEO Construction (Photovoltaic Satellite)

- There are a total of approximately 400 jobs that need to be staffed during each shift.
- 50 of the jobs are at the LEO staging depot.
- 350 of the jobs are at the GEO construction site.

For Both LEO and GEO Construction

- 100 people can be transported at one time in either an earth-to-LEO shuttle or in an OTV.
- It will cost \$11 mission/100 people to get to LEO.
- It will cost \$18 million/100 people to get from earth to GEO. (The \$11 million earth-to-LEO cost plus \$7 million LEO-to-GEO cost).

Based on these assumptions, it is possible to estimate the cost of crew transportation as follows:

LEO Construction Site/GEO Final Assembly

To get 350 people (one team) to LEO

$$A = \frac{\text{LEO Transp. Cost}}{\text{Year}} = (350\text{ people})\left(\frac{1}{100\text{ people/shuttle}}\right)\left(\frac{\$11\text{ million}}{\text{shuttle}}\right)\left(N\ \frac{\text{rotations}}{\text{year}}\right)$$
$$= (\$38.5\text{ million})(N)$$

To get 150 people (one team) to GEO

$$B = \frac{\text{GEO Transp. Cost}}{\text{Year}} = (150\text{ people})\left(\frac{1}{100\text{ people/OTV}}\right)\left(\frac{\$18\text{ million}}{\text{OTV/shuttle}}\right)\left(N\ \frac{\text{rotations}}{\text{year}}\right)$$
$$= (\$27\text{ million})(N)$$

Total Transportation Cost = A + B

= (\$38.5 million x N) + \$27 million x N)

$$= (\$65.5\text{ million})\left(N\frac{\text{rotations}}{\text{year}}\right)$$

LEO Staging Depot/GEO Construction

To get 50 people (one team) to LEO

$$A = \frac{\text{LEO Transp. Cost}}{\text{Year}} = (50\ \text{people}) \left(\frac{1}{50\ \text{people/shuttle}}\right) \left(\frac{\$11\ \text{million}}{\text{shuttle}}\right) \left(N\ \frac{\text{rotations}}{\text{year}}\right)$$

$$= (\$11\ \text{million}) \times (N)$$

To get 300 people to GEO

$$B = \frac{\text{GEO Transp. Cost}}{\text{Year}} = (350\ \text{people}) \left(\frac{1}{100\ \text{people/shuttle}}\right) \left(\frac{\$18\ \text{million}}{\text{shuttle}}\right) \left(N\ \frac{\text{rotations}}{\text{year}}\right)$$

$$= (\$63\ \text{million}) \times (N)$$

Total Transportation Cost = A + B

$$= (\$11\ \text{million} \times N) + (\$63\ \text{million} \times N)$$

$$= (\$74\ \text{million}) \left(N\ \frac{\text{rotation}}{\text{year}}\right)$$

These transportation costs for each option were shown in Table 3.4-44.

3.4.2.4.3.3 Sub-Total Cost

The sub-total cost for each option are computed by adding the transportation cost and down-time cost. These sub-totals were given in Table 3.4-44.

A graphical comparison of these sub-total costs is presented in Figure 3.4-89.

3.4.2.4.3.4 Analysis of Results

The following observations came from inspection of Table 3.4-44 and Figure 3.4-89 and -90:

- The least expensive options are for the 100 day stay-time, 6 days on/1 day off schedule.
- The 10 and 12 hours/day schedules using 2 crews are the lowest cost options.
- The 4-team scheduling option is very expensive when compared to the alternatives. It is much more cost effective to tolerate a common shutdown day, than to use 4 teams to avoid a shutdown. Those 2 extra crews create the need for twice as many crew types that cost much more than downtime.

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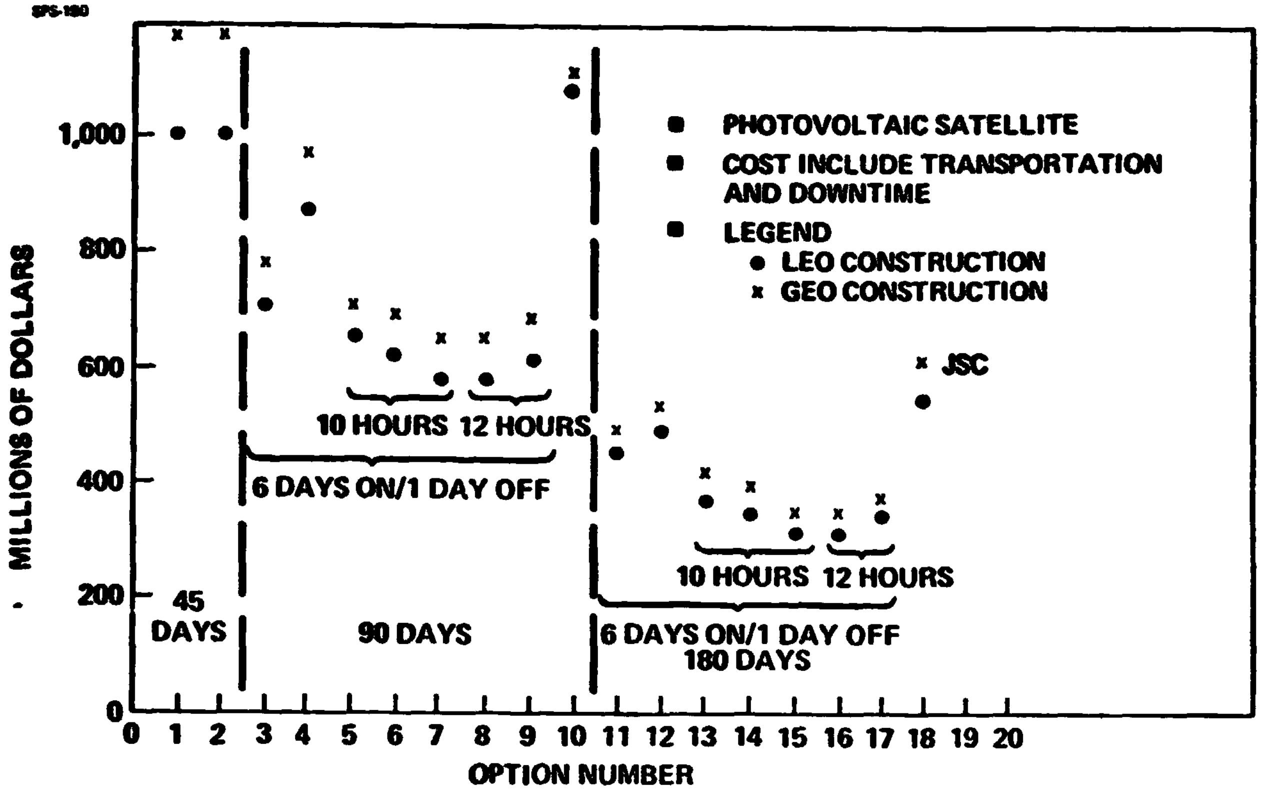

Figure 3.4-89 Crew Scheduling Cost Comparisons

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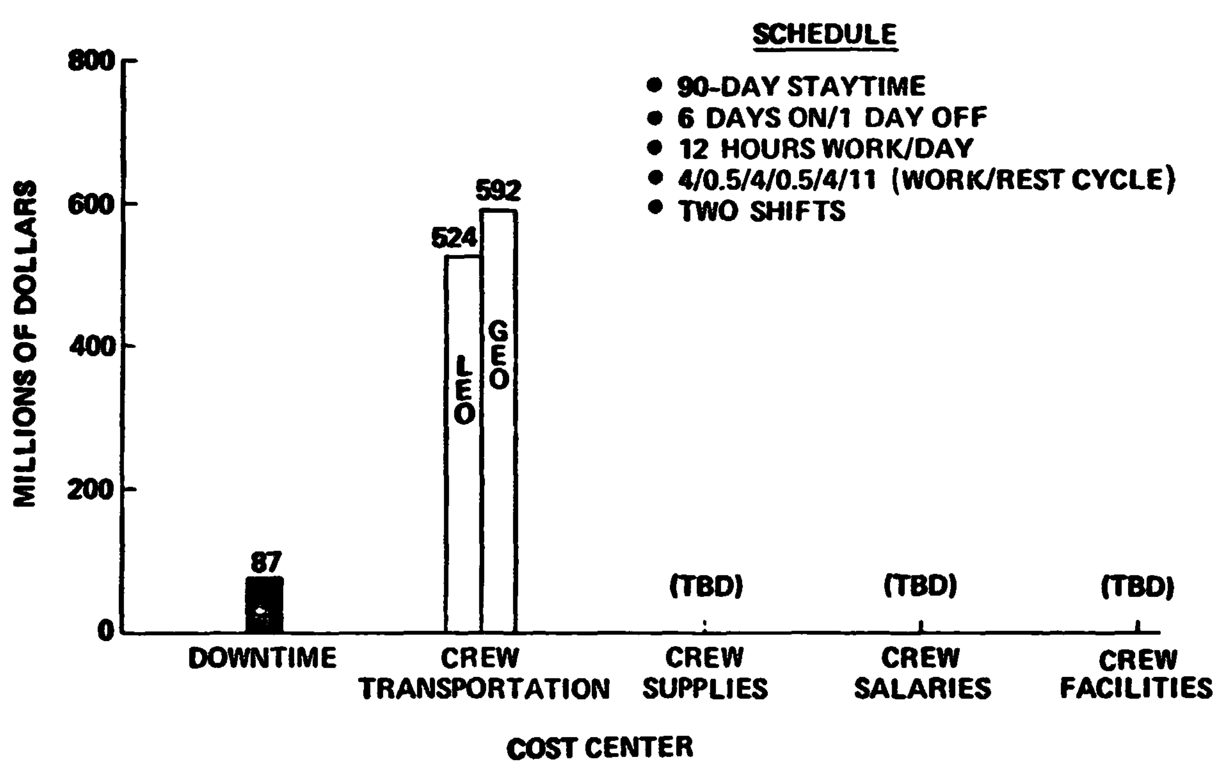

Figure 3.4-90 Crew Operations Cost Distribution

- Of the three daily schedule options, the 12 hour 4/.5/4.5/4/11 schedule is preferred even though it is slightly more expensive:

    - The 10 hour schedules are discarded because they would unnecessarily create the need for forcing the crew onto a synthetic 20 hour "day".

    - The 12 hour 4/4/4...schedule would create some operational problems as well as creating the need for an unnatural work/rest cycle.

    180 day schedules are less expensive than 90 day options.

    - The 45 day schedules offer no economic advantages.

3.4.2.5 Recommendations

The following schedule is recommended based on the economic factors considered:

- 90 day staytime

  6 days on/1 day off per week

  12 hours per day work using a 4/.5/4/.5/4/11 work/rest cycle.

  2 shifts (2 teams)

Even though the 180 day staytime is the most economical, it is not recommended due to the absence of any experience data to support it as operationally practical.

Subsequent to the 12-hour shift recommendation, the JSC Crew System group recommended that a 10 hour schedule be used (based on Skylab experience data). A 5/1/5/13 work/rest cycle was therefore selected for crew size determinations.

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3.4.3 Crew Jobs and Organization

3.4.3.1 Study Approach

The approach used in this study is summarized in Figure 3.4-91.

The base operations and base support jobs were identified by performing a functional analysis wherein various categories of base support and operations functions were postulated. These functional categories were identified in such a way that they were insensitive to satellite type or, in most cases, orbital location. Each of the functional categories were developed to one or two lower levels of detail to identify the jobs required to be staffed in order to carry out the functions. At these lower levels of detail, it was necessary to apply the number of shifts (2). In some cases, the number of habitats and the orbital location had to be taken into account. Information from prior studies (references 1 and 2) was incorporated or was compared to make sure major items were not overlooked.

In order to identify construction jobs, it was necessary to select a satellite configuration concept as a model for analysis. The photovaltaic satellite was selected. To simplify the analysis, GEO construction was selected as a basis. The satellite construction concept was developed to sufficient detail to determine the major construction tasks (fabricate frame, deploy solar cells, etc.). In lieu of detailed trade studies that would identify whether the task should be automated or should be performed by man it was assumed that there would be one man assigned to each of the tasks. It was necessary to determine at how many places the task would be performed simultaneously. Two shifts were assumed. By multiplying these factors together, it was possible to make an estimate of the number of jobs.

To collate the results of the analysis of the construction jobs and base operations support jobs and to identify the management personnel, organization charts were developed.

After the organization charts and manning requirements for the GEO construction base for the photovoltaic satellite were identified the results were adjusted to determine how many jobs would be at LEO and to determine the LEO and GEO jobs for a LEO construction concept.

3.4.3.2 Results

3.4.3.2.1 Base Support/Operations

After several iterations the base support and operations were grouped into the eight functional categories shown in Figure 3.4-92.

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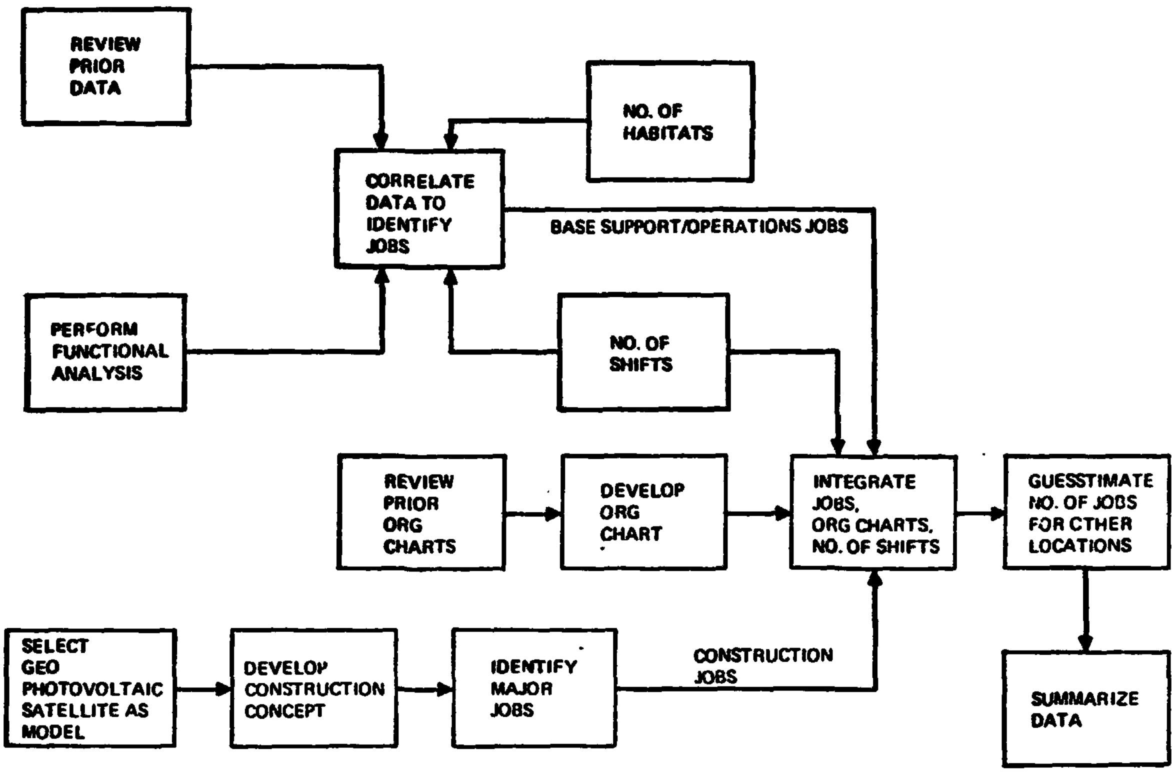

Figure 3.4-91 Study Approach

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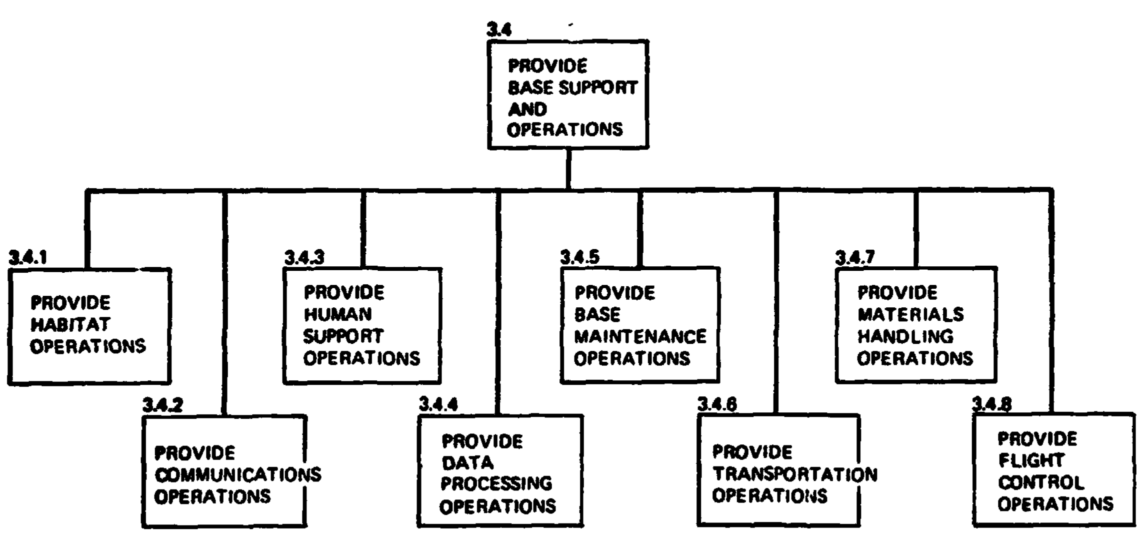


Figure 3.4-92 Base Support/Operations Functional Categories

The habitat operations are identified to lower levels of detail in Figure 3.4-93. To identify the number of personnel associated with these habitat operations it was necessary to determine the total number of people at each base and to assume a habitat population for each habitat. The numbers of personnel shown have been adjusted to reflect the total number of personnel.

The communications function and its associated personnel are shown in Figure 3.4-94.

A functional category called Human Support Operations was created by encompassing the support functions that did not seem to fit into other categories; see Figure 3.4-95.

The data processing functions and associated personnel are shown in Figure 3.4-96. It was assumed that the majority of data processing would be performed on Earth. The operational personnel listed would provide in-orbit, special purpose data processing.

A large base maintenance organization was identified, Figure 3.4-97. These personnel would be concerned with maintaining the habitat, command/control, communications, transportation, etc. equipment. A maintenance team was also assigned to the construction equipment, but is counted as part of the construction crew.

The materials handling function is shown in Figure 3.4-98. The number of personnel shown is probably quite conservative. The materials handling system concept will have to be developed in order to establish a better guess.

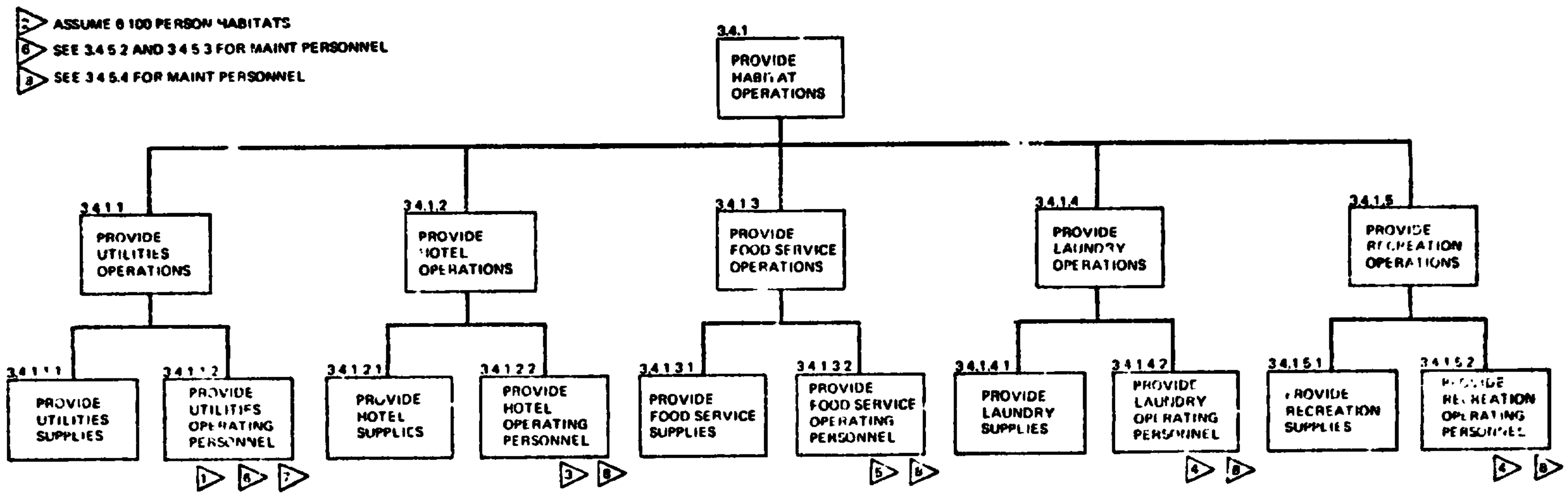
The base flight control function is shown in Figure 3.4-99.

The transportation support function is highly dependent upon where the major construction site is (LEO or GEO):

LEO Construction/LEO Base - See Figure 3.4-100
LEO Construction/GEO Base - See Figure 3.4-101
GEO Construction/GEO Base - See Figure 3.4-102
GEO Construction/LEO Base - See Figure 3.4-103

3.4.3.2.2 Construction

The construction personnel were identified for the photovoltaic satellite only. The construction operations were sorted into satellite construction and antenna construction groups.



PERSONNEL	NO REQ'D/SHIFT	TOTAL REQ'D/BASE
1 UTILITIES OPERATOR – MONITORS/CONTROLS HABITAT UTILITIES (ELECTRICITY, WATER ATMOSPHERE, WASTE TREATMENT, ETC.) – REORDERS, CONSUMABLES/PARTS/ETC. – SCHEDULES AND APPROVES MAINTENANCE	6 2	12
3 HOTEL CLERK – ORDERS SUPPLIES – ASSIGNS ROOMS – SCHEDULES MAINTENANCE 4 – OVERSEES LAUNDRY/RECREATIONAL FACILITIES (THESE FACILITIES TO BE SELF-SERVICE)	6 2	12
5 FOOD SERVICE PERSON – DRAWS FOOD FROM STORES – LOADS PREPARATION/DISPENSING EQUIPMENT – DISPOSALS OF FOOD SERVICE WASTES – KITCHEN HOUSEKEEPING – PREPARES FOOD FOR SHIPMENT TO REMOTE WORK STATIONS	6 2	12
7 BASE ELECTRICAL SYST OPERATION – CONTROLS/MONITORS BASE ELECTRICAL POWER DISTRIBUTION SYSTEM	1	2

Figure 3.4-93 Habitat Operations Functions

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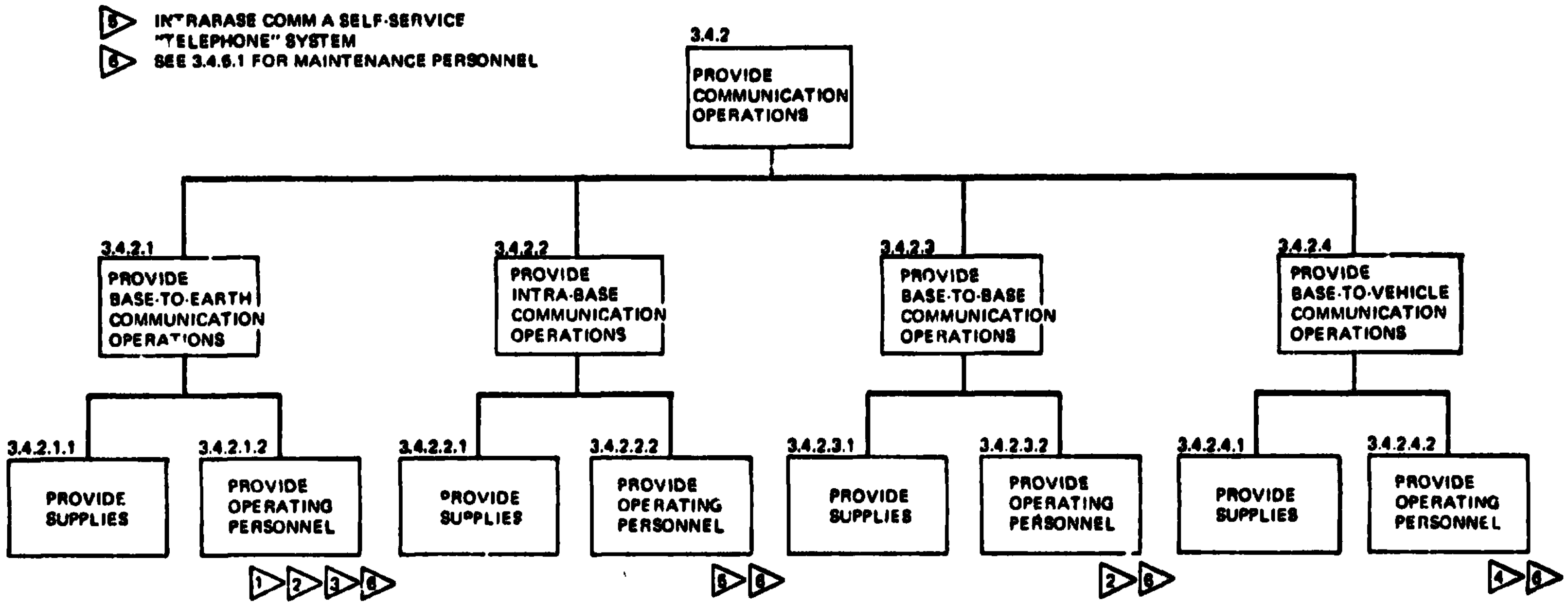

PERSONNEL	NO. REQ'D/SHIFT	TOTAL REQ'D/BASE
1 VOICE COMM OPERATOR – MONITORS/CONTROLS VOICE COMM SYSTEMS	1	2
2 BASE COMMUNICATOR – PROVIDES ALL ROUTINE VOICE COMMUNICATIONS WITH EARTH AND OTHER BASES	1	2
3 DATA COMM OPERATOR – MONITORS/CONTROLS ALL DATA LINK SYSTEMS	1	2
4 TRAFFIC CONTROLLER – PROVIDES ALL VOICE INTERFACE WITH INCOMING/OUTGOING TRAFFIC – PROVIDES TRAFFIC CONTROL DATA TO VEHICLES	1	2

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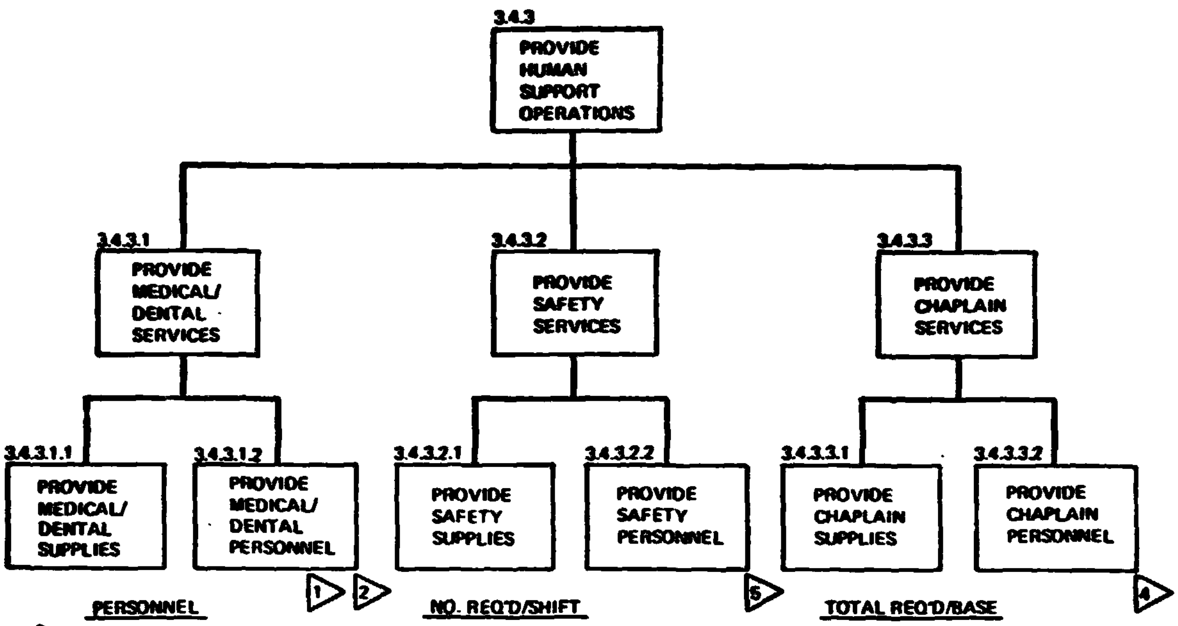
Figure 3.4-94 Communications Functions



PERSONNEL	NO. REQ'D/SHIFT	TOTAL REQ'D/BASE
1 PHYSICIAN	1	2
2 PARAMEDICS	6 3	12
4 CHAPLAIN	1	2
5 SAFETY INSPECTOR	1	2

3 ONE PER HABITAT

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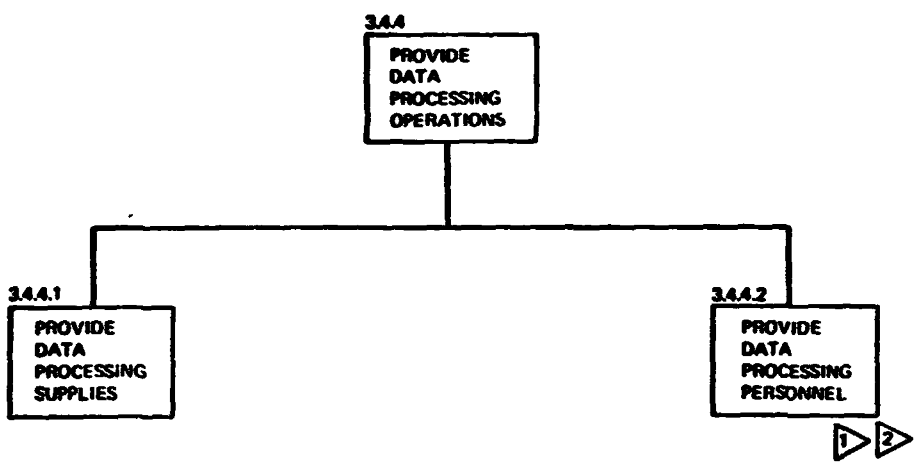
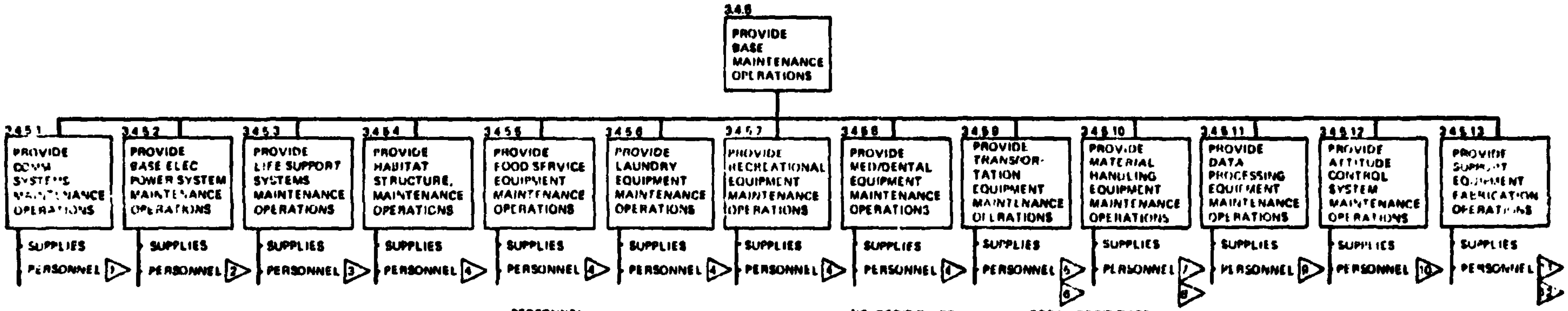
Figure 3.4-95 Human Support Functions



1 PERSONNEL	NO. REQ'D/SHIFT	TOTAL REQ'D/BASE
COMPUTER OPERATOR	1	2
CONSULTANT	1	2
TERMINAL OPERATOR	1	2

2 SEE 3.4.6.11 FOR MAINTENANCE PERSONNEL

Figure 3.4-96 Data Processing Functions



PERSONNEL	NO REQ'D/SHIFT	TOTAL REQ'D/BASE
1 COMM SYSTEMS MAINT TECH – REPAIRS/CALIBRATES COMMUNICATION EQUIPMENT – EVA RATED	2	4
2 ELECTRICAL POWER SYSTEMS MAINTENANCE TECH – REPAIRS/CALIBRATES ELECTRICAL POWER SYS EQUIP – EVA RATED	2	4
3 LIFE SUPPORT EQUIPMENT MAINTENANCE TECH – REPAIRS/CALIBRATES EVA, WASTE TREATMENT, ATMOSPHERE, WATER SUPPLY, ETC. EQUIPMENT	2	4
4 HABITAT EQUIPMENT MAINT TECH – REPAIRS HABITAT STRUCTURES, FOOD SERVICE, LAUNDRY, RECREATIONAL, ETC. EQUIPMENT	2	4
5 TRANSPORTATION VEHICLE MAINT TECH-MECH – REPAIRS VEHICLE ELECTRICAL/ELECTRONIC – EVA RATED	2	4
6 TRANSPORTATION VEHICLE MAINT TECH - ELEC – REPAIRS VEHICLE ELECTRICAL/ELECTRONIC EQUIPMENT – EVA RATED	2	4
7 MATERIALS HANDLING EQUIP MAINT TECH MECH – REPAIRS MECHANICAL MATERIALS HANDLING EQUIP – EVA RATED	2	4
8 MATERIALS HANDLING EQUIPMENT MAINT TECH ELEC – REPAIRS ELECTRICAL/ELECTRONIC MAT'LS HANDLING EQUIP – EVA RATED	2	4
9 DATA PROCESSING EQUIP MAINT TECH – REPAIRS DATA PROCESSING EQUIPMENT	1	2
10 ATTITUDE CONTROL SYSTEM MAINT TECH – REPAIRS ATTITUDE CONTROL SYSTEM – EVA RATED	2	4
11 MACHINIST – FABRICATES SPECIAL STRUCTURAL/MECHANICAL EQUIPMENT		1
12 ELECTRICAL/ELECTRONIC TECH – FABRICATES SPECIAL ELECTRICAL/ELECTRONIC EQUIPMENT		1
MAINTENANCE DISPATCHER	1	2

Figure 3.4-97 Base Maintenance Functions

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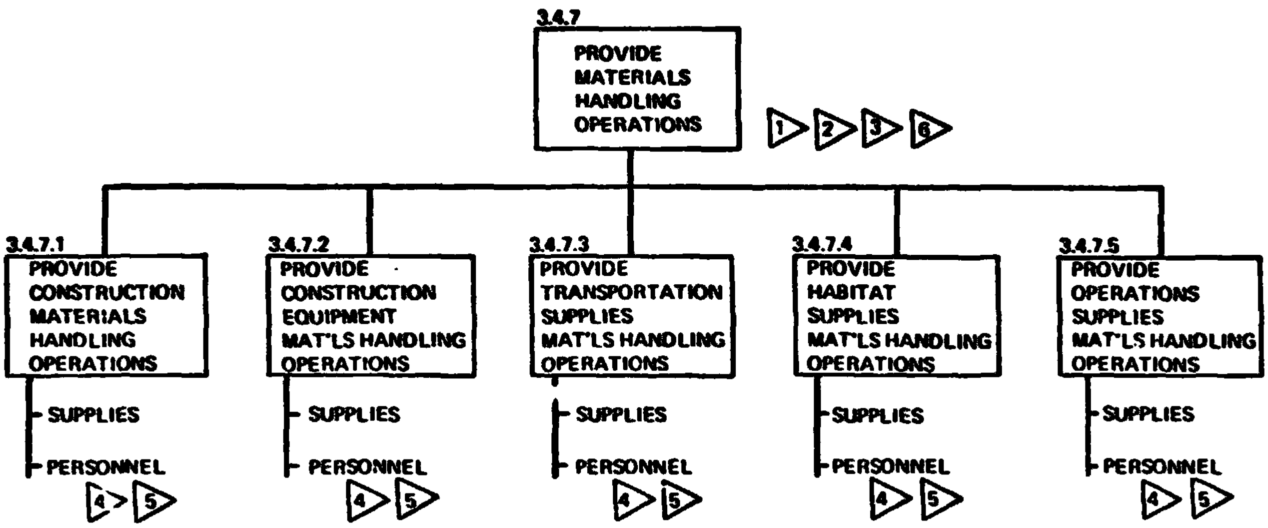


	PERSONNEL	NO. REQ'D/SHIFT	TOTAL REQ'D/BASE
1	• CARGO MASTER • SUPERVISES LOADING/UNLOADING OF VEHICLES	1	2
2	• CRANE OPERATORS • OPERATES MANIPULATORS • EVA-RATED	EST. 5	10
3	• CARGO DISBURSEMENT CONTROLLER • ACTUATES CARGO CONVEYOR SYSTEM • MONITORS/CONTROLS MOVEMENT OF PALLETS/MODULES TO USER SITES	2	4
4	• WAREHOUSEMEN • LOADS/UNLOADS CARGO PALLETS/MODULES AT USER SITES • SORTS INTO/DRAWS GOODS FROM STORAGE	8	16
5	• EVA FREIGHT HANDLER • MANIPULATES CARGO IN VEHICLE CARGO BAYS • SORTS INTO/DRAWS GOODS FROM STORAGE	2	4
6	• SUPPLY CLERK • INVENTORY CONTROL • CHECKS OUT SUPPLIES/PARTS/ETC	5	10

Figure 3.4-98 Materials Handling Functions

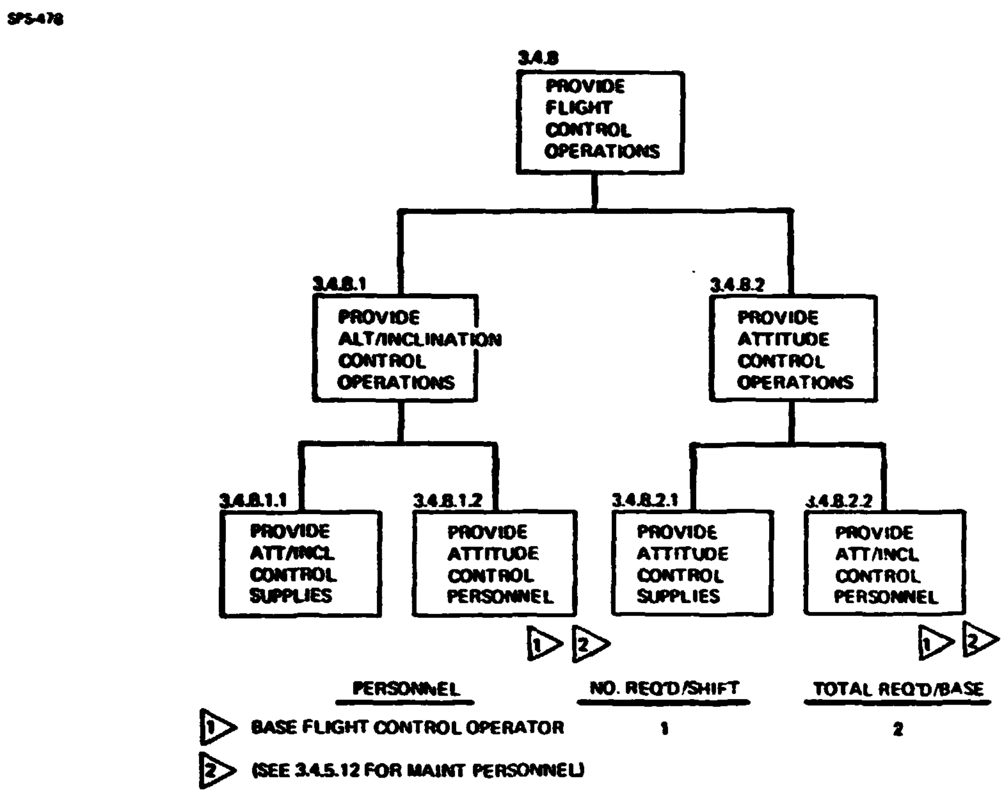
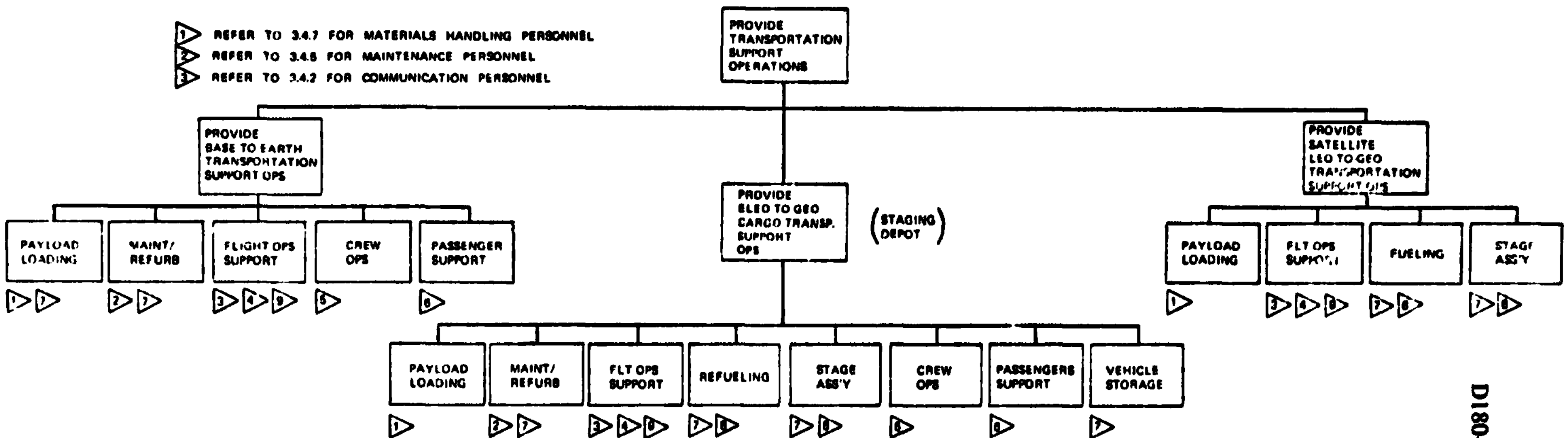


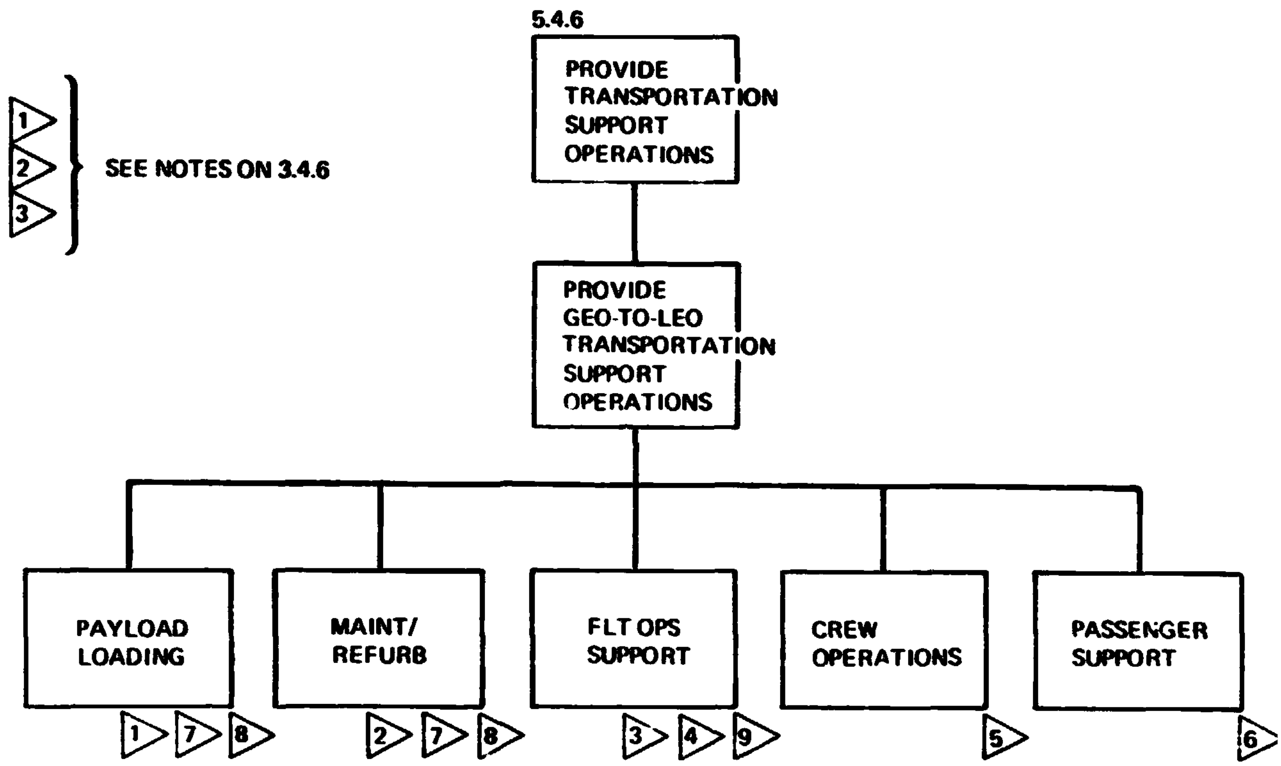
Figure 3.4-99 Flight Control Functions

1. REFER TO 3.4.7 FOR MATERIALS HANDLING PERSONNEL
2. REFER TO 3.4.6 FOR MAINTENANCE PERSONNEL
3. REFER TO 3.4.2 FOR COMMUNICATION PERSONNEL



	NO REQ'D/SHIFT	TOTAL REQ'D/BASE
4. FLIGHT OPERATIONS SUPPORT COORDINATOR – ONE ASSIGNED FOR EACH FLIGHT – COORDINATES IN-ORBIT SUPPORT	2	4
5. FLIGHT CREW SUPPORT COORDINATOR	1	2
6. PASSENGER SERVICE COORDINATOR – COORDINATES PASSENGER SUPPORT OPERATIONS – ASSIGNS SEATS – BRIEFS PASSENGERS – COORDINATES TEMPORARY HOUSING FOR TRANSIENTS	1	2
7. VEHICLE COORDINATOR – COORDINATES/DIRECTS STAGING/REFUELING/REPAIR/REFURBISHMENT	2	4
8. VEHICLE SUPPORT CREW – PERFORMS FUELING OPERATIONS – CONNECTS STAGES – INSPECT VEHICLE – MONITOR SYSTEM STATUS	4	8
9. FLIGHT DATA COORDINATOR – COORDINATES FLIGHT DATA WITH EARTH BASE – INPUTS FLIGHT DATA TO VEHICLE COMPUTER	2	4

Figure 3.4-100 Transportation Functions (LEO Construction/LEO Base)



		PER SHIFT	TOTAL
4	FLIGHT OPERATIONS SUPPORT MANAGER	1	2
5	FLIGHT CREW SUPPORT MANAGER	0	0
6	PASSENGER SERVICE COORDINATOR	–	1
7	VEHICLE MANAGER	1	2
8	VEHICLE SUPPORT CREW	2	4
9	FLIGHT DATA COORDINATOR	–	1

Figure 3.4-101 Transportation Functions (LEO Construction/GEO Base)

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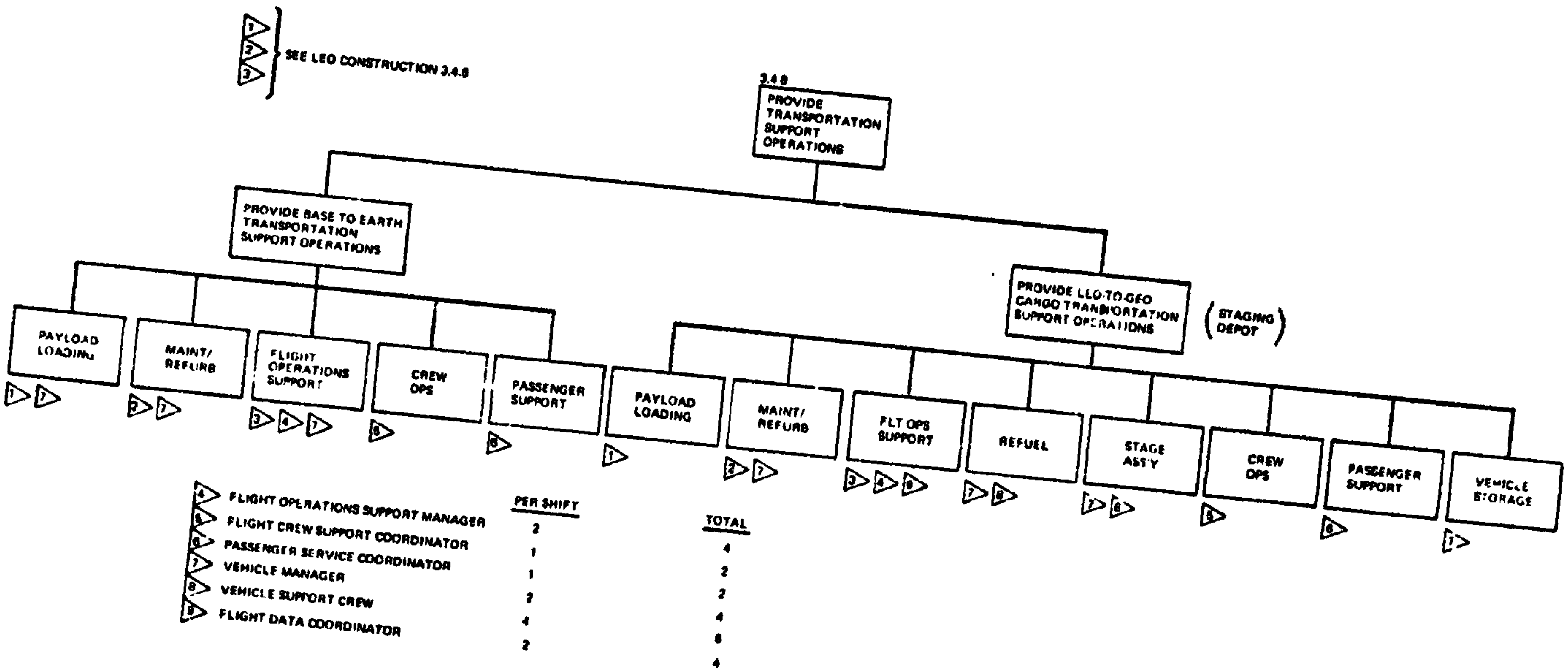

Figure 3.4-102 Transportation Functions (GEO Construction/GEO Base)

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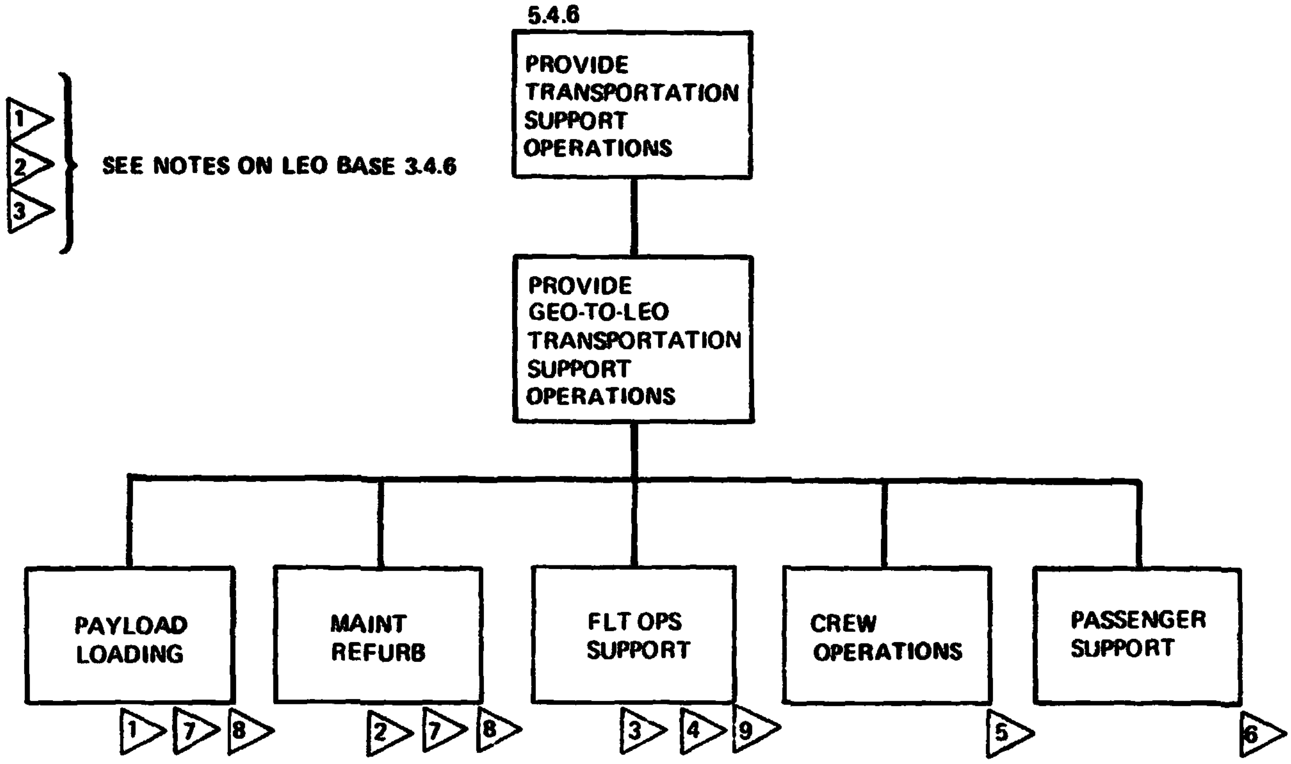

		PER SHIFT	TOTAL
4	FLIGHT OPERATIONS SUPPORT MANAGER	1	2
5	FLIGHT CREW SUPPORT MANAGER	0	0
6	PASSENGER SERVICE COORDINATOR	–	1
7	VEHICLE MANAGER	1	2
8	VEHICLE SUPPORT CREW	2	4
9	FLIGHT DATA COORDINATOR	–	1

Figure 3.4-103 Transportation Functions (GEO Construction/LEO Base)

3.4.3.2.2.1 Satellite Construction

Fabrication/assembly of the satellite frame would be performed by the personnel identified in Figure 3.4-104. (The Section 1, 2, 3, 4, and 5 correspond to the longitudinal ridge sections shown in Figure 3.4-105.) For this iteration, one man operator was associated with each beam machine. At this point, it is undetermined whether the operators would be located with the beam machines or at some remote location. Further analysis may show that one operator could control several beam machines or that a single beam machine could be used to fabricate several of the different beams.

Assembly of the power generation system would be performed by the personnel identified in Figure 3.4-106. The Sections 1, 2, 3, and 4 correspond to the longitudinal through sections shown in Figure 3.4-107. The personnel identified operate the machines which deploy the power generation components. At this point, it has not been determined whether the operators are located at the deploying machines or are in some remote location.

The personnel associated with assembly of the satellite subsystem are shown in Figure 3.4-108. A team of test and checkout personnel are identified in Figure 3.4-109. A team of construction equipment maintenance personnel are identified in Figure 3.4-110.

3.4.3.2.2.2 Antenna Construction

The antenna construction personnel are organized similar to the satellite construction personnel. At this point, the antenna construction operations have not been identified in as much detail as the satellite construction. The antenna construction team is shown in Figure 3.4-111.

SPS-481

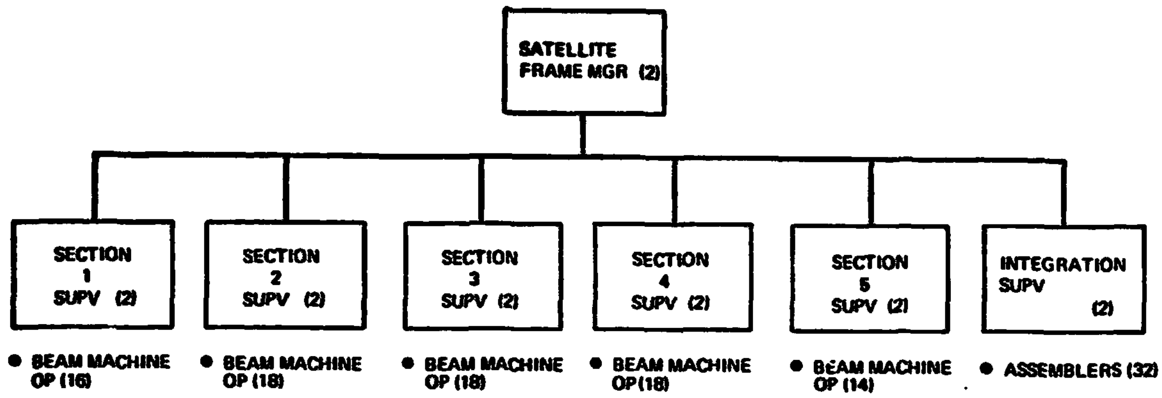

Figure 3.4-104 Satellite Frame Construction Organization

SPS-482

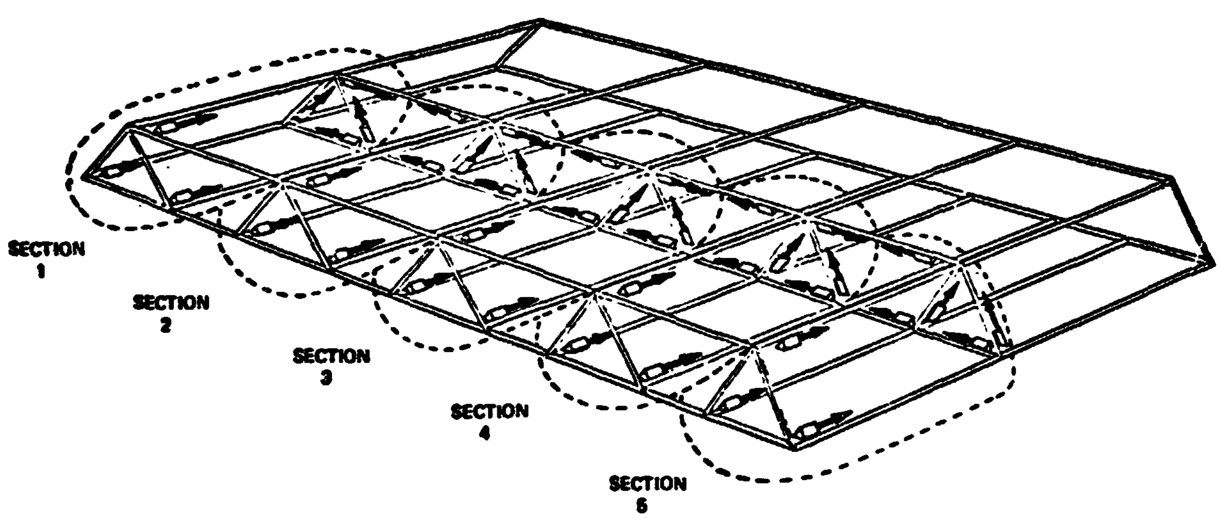

Figure 3.4-105 Satellite Bay 1 Structural Framework Construction

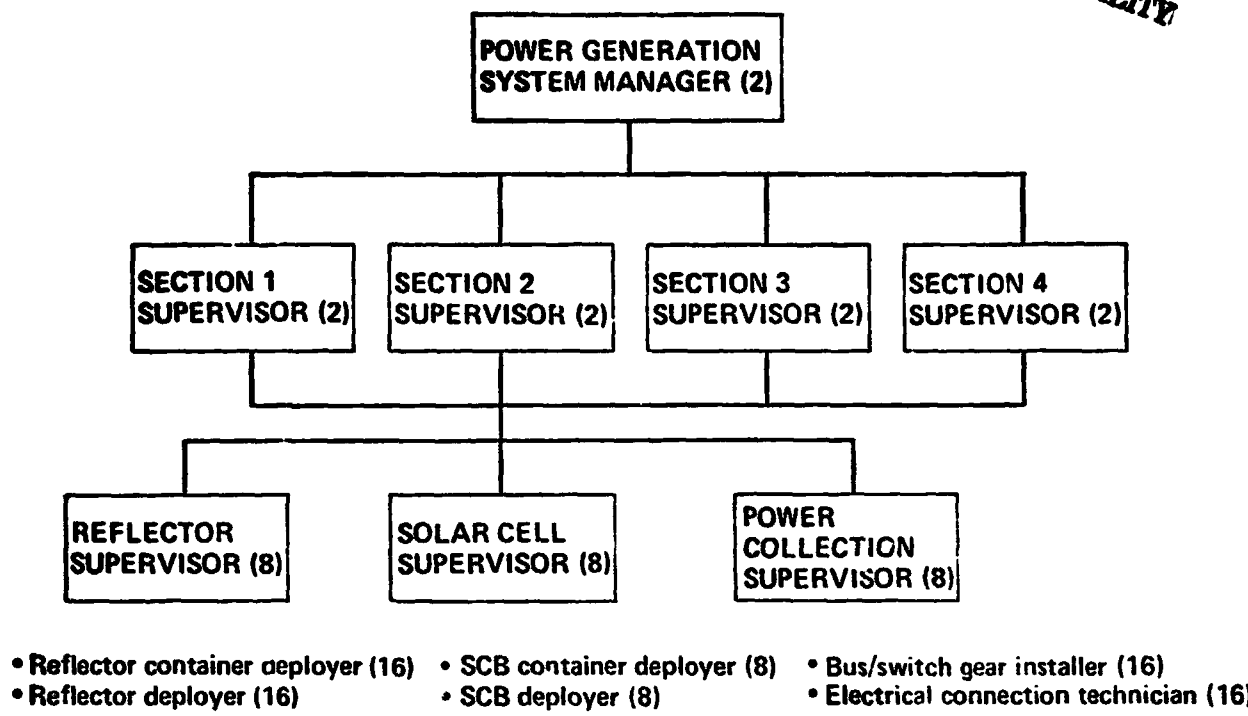

Figure 3.4-106 Construction Organization Power Generation System

SPS-1789

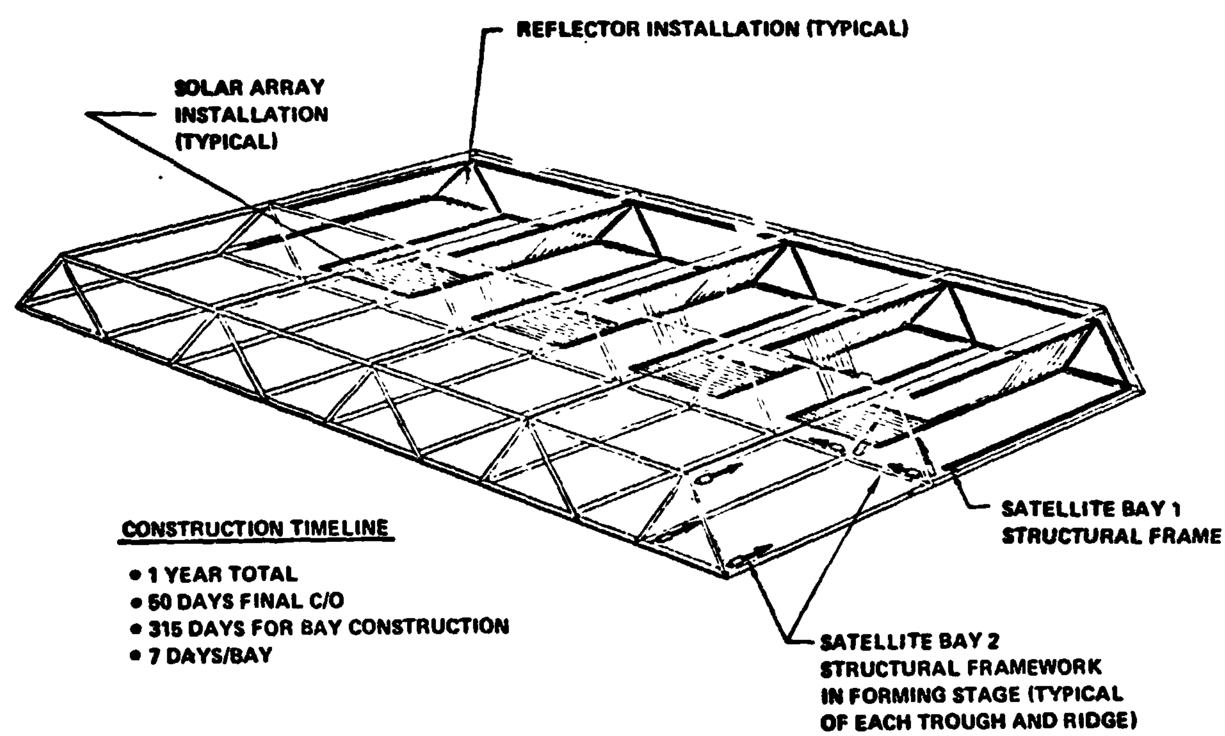

Figure 3.4-107 Facility Bay B Operations

SPS-483

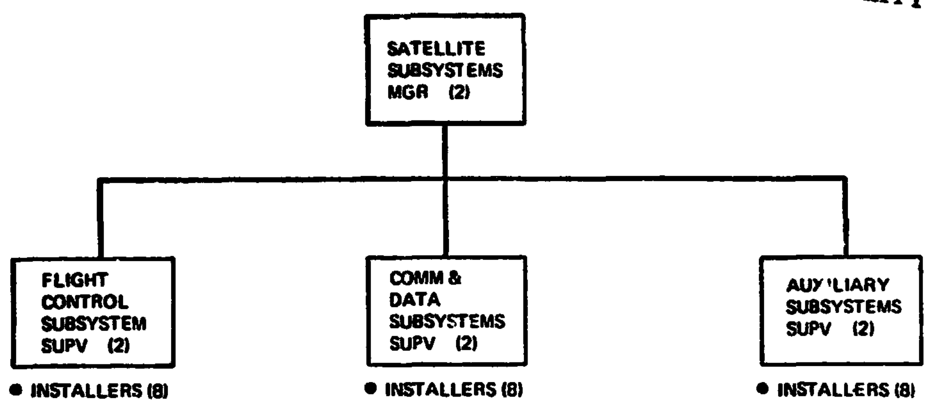

Figure 3.4-108 Satellite Subsystems Construction Organization

SPS-484

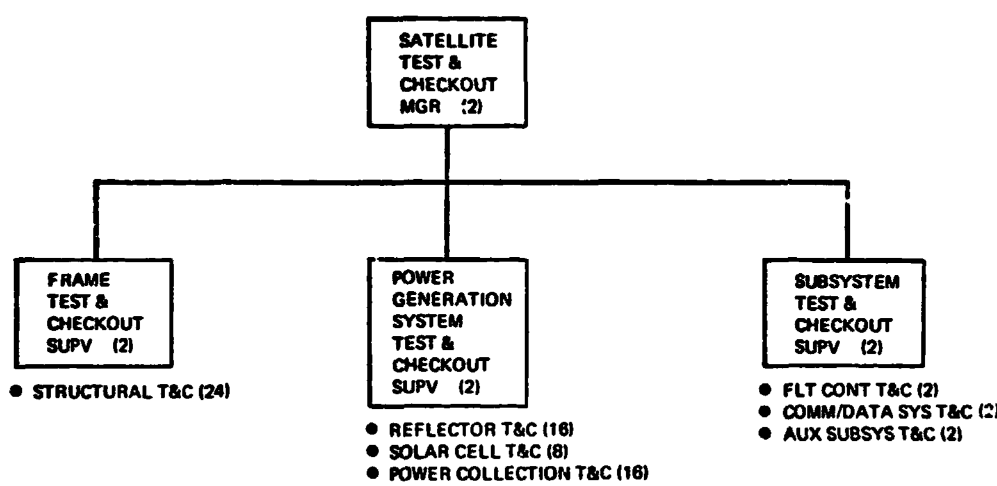

Figure 3.4-109 Satellite Test and Checkout Organization

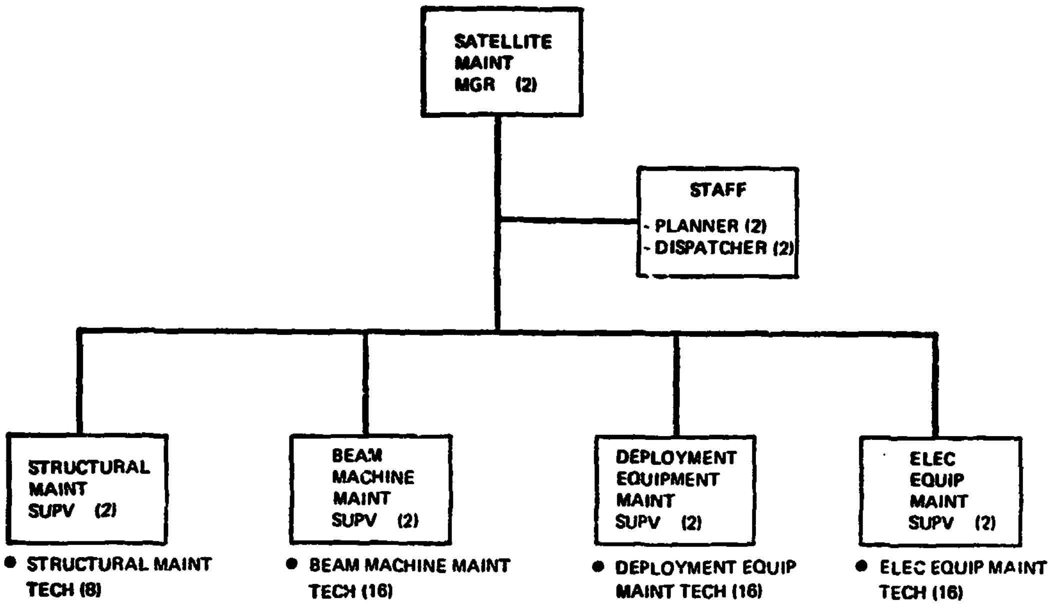

Figure 3.4-110 Satellite Construction Maintenance Organization

SPS-486

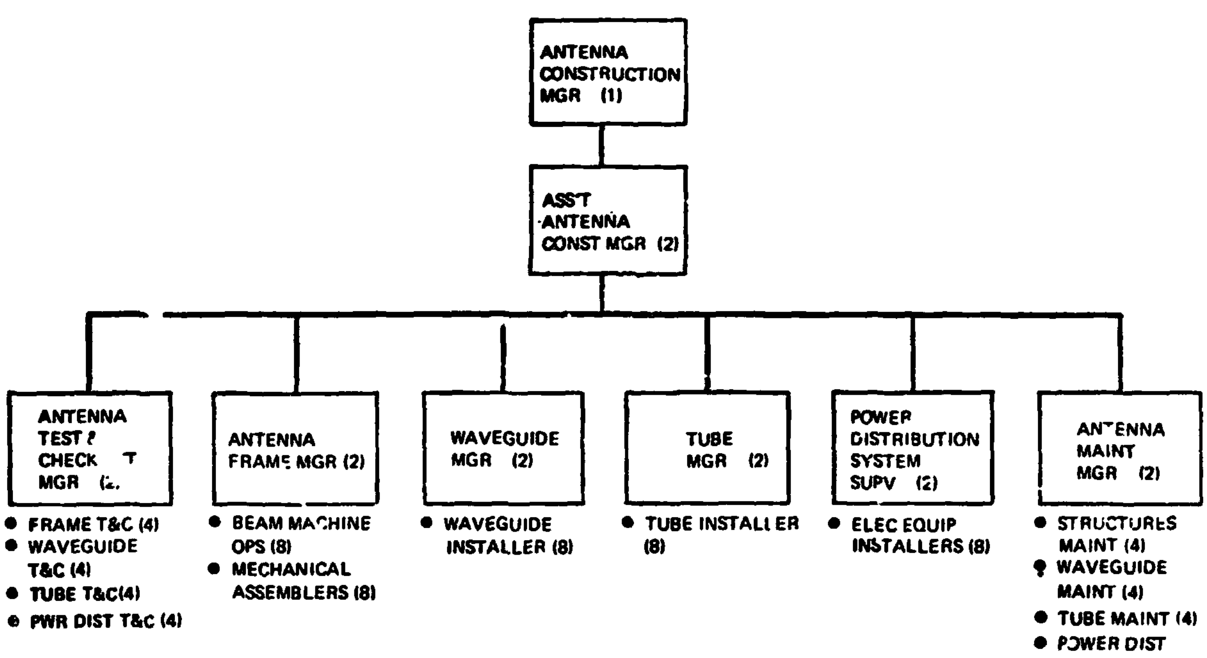

Figure 3.4-111 Antenna Construction Organization

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3.4.3.3 ORGANIZATIONAL CONCEPT

The personnel identified in the preceding sections were organized into a base personnel organization. The base organization concept defined in Reference 1 was used as a starting point. The top level organization is shown in Figure 3.4-112. The construction personnel were organized as shown in Figure 3.4-113. The base support personnel were organized as shown in Figure 3.4-114. The base operations personnel were organized as shown in Figure 3.4-115.

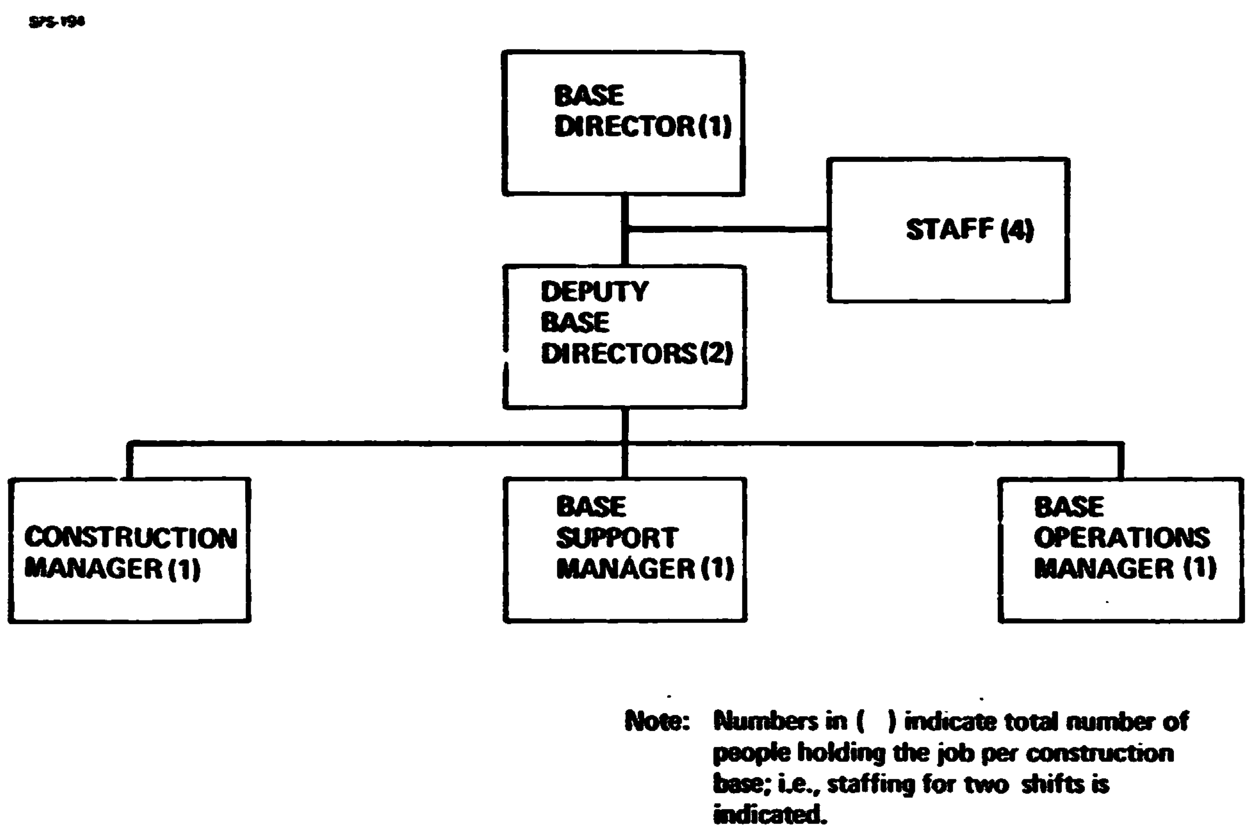


Figure 3.4-112 Construction Base Organization

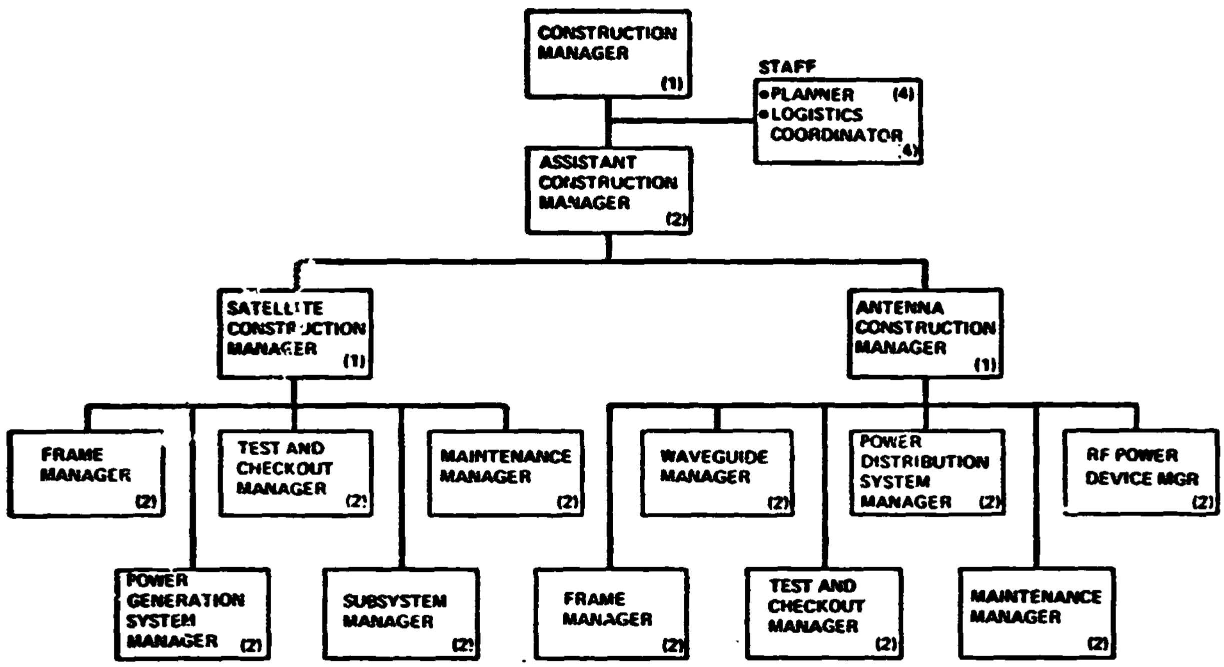
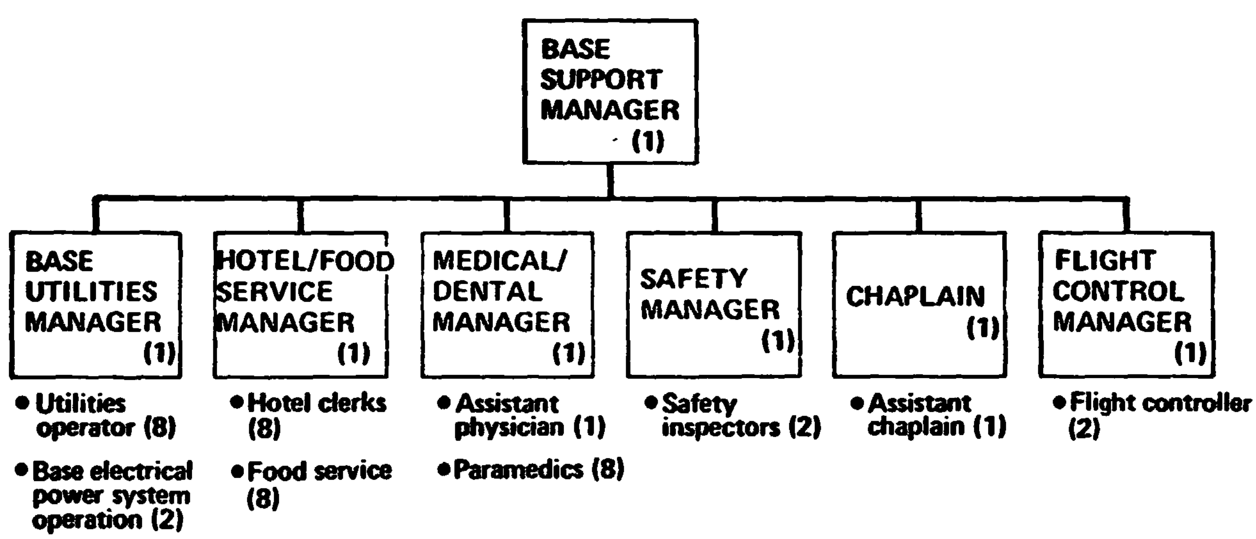


Figure 3.4-113 Construction O.ganization

SPS-195



Numbers in ( ) indicate staffing for two shifts.

Figure 3.4-114 Base Support Organization LEO Construction Base

SPS-347

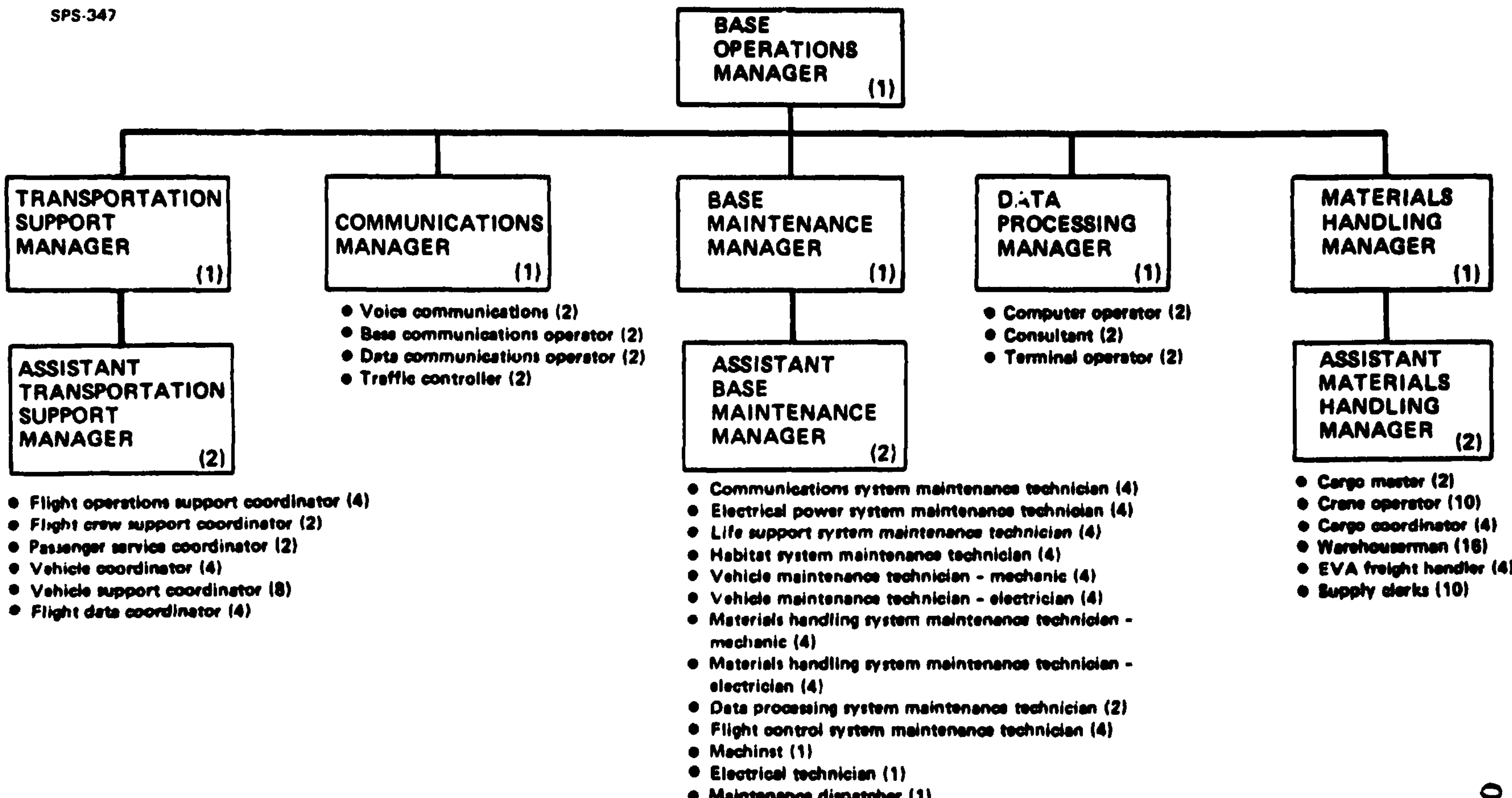


Figure 3.4-115 Base Operations Organization LEO Construction Base

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3.4.3.4 MANPOWER SUMMARY

The personnel identified in the previous sections have been summarized in Table 3.4-45. Note that for the photovoltaic satellite that there are approximately 200 more people required to construct the satellite in LEO than as required for GEO. Inspection of the numbers will show that the difference is due to the need for a substantial construction crew at GEO to complete the construction.

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Table 3.4-45 Manpower Summary

SPS-348

	Photovoltaic satellite				Thermal engine satellite			
	LEO construction		GEO construction		LEO construction		GEO construction	
	LEO base	GEO base	LEO base	GEO base	LEO base	GEO base	LEO base	GEO base
Base management	(7)	(5)	(5)	(7)	(7)	(5)	(5)	(7)
Satellite construction	(433)	(171)	(0)	(446)				
Management	27	19	—	27				
Frame	128	34	—	128	TBD	TBD	TBD	TBD
Power generation	108	48	—	116				
Subsystems	30	30	—	30				
Maintenance	68	30	—	68				
Test and checkout	72	25	—	76				
Antenna construction	(84)	(54)	—	(84)	(84)	TBD		(84)
Base operations	(138)	(68)	(82)	(124)	(138)	(68)	(54)	(124)
Management	12	8	8	12	12	8	8	12
Data processing	6	4	4	6	6	4	4	6
Base maintenance	42	19	19	42	42	19	19	42
Transportation	24	10	24	10	24	10	24	10
Materials handling	46	19	19	46	46	19	19	46
Communications	8	8	8	8	8	8	8	8
Base support	(64)	(37)	(23)	(64)	(64)	(37)	(23)	(64)
Management	7	5	5	7	7	5	5	7
Utilities	14	8	2	14	14	8	2	14
Hotel/food service	24	12	4	24	24	12	4	24
Medical/dental	13	6	6	13	13	6	6	13
Safety	2	2	2	2	2	2	2	2
Chaplain	2	2	2	2	2	2	2	2
Base flight control	2	2	2	2	2	2	2	2
Subtotals	726	335	110	724	TBD	TBD	TBD	TBD
Total	1,061		834		TBD		TBD	

3.4.4 Operator Productivity

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

Bill Faler, BAC Central Industrial Engineering, provided the information shown in Figures 3.4-116 and 3.4-117.

For purposes of computation of machine rates, a operation productivity of 75% over the 12-hour shift will be assumed.

1.33 Machine Rate based on 100% productivity = Adjust machine rate requirement

3.4.5 Constructability Rating Analysis

This section contains the analysis that was used to derive the "constructability rating" given to each concept that was summarized in Figure 3.4-118 in section 3.4.1.2.4.

Evaluation Criteria

There were seven evaluation criteria that were used:

1. Number of operators

2. Number of construction machines

3. Complexity of the construction machines

4. Size of the ma·· ·r facility

5. Size of the secondary facility

6. Complexity of major facility

7. Satellite assembly complexity

Table 3.4-46 shows how these criteria were converted into a 0 to 10 scoring system against which each of the six concepts were evaluated.

SPS-471

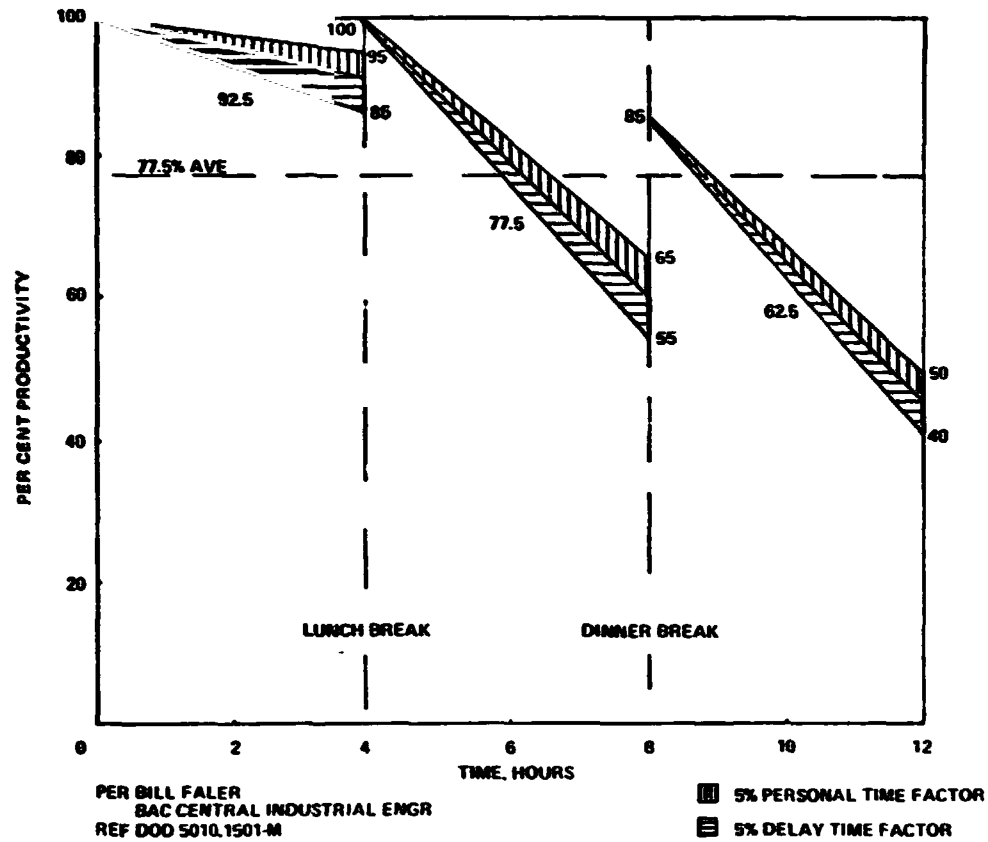

Figure 3.4-116 Productivity Factors

SPS-472

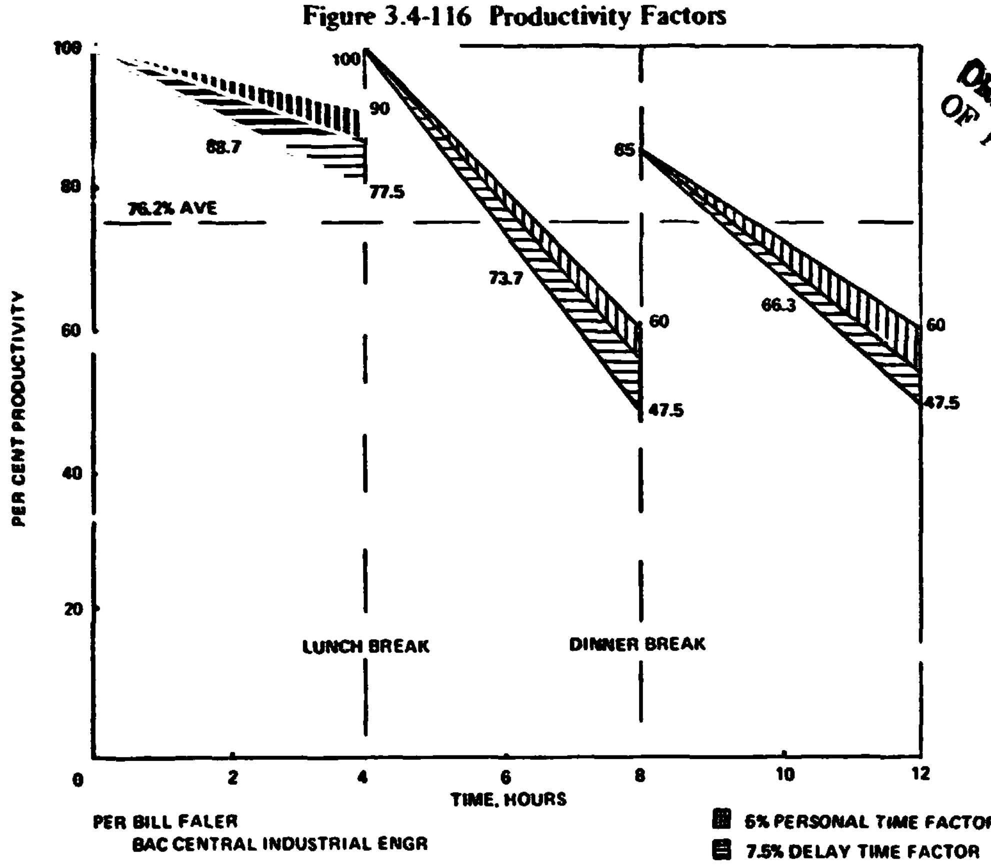

Figure 3.4-117 Productivity Factors

SPS 728

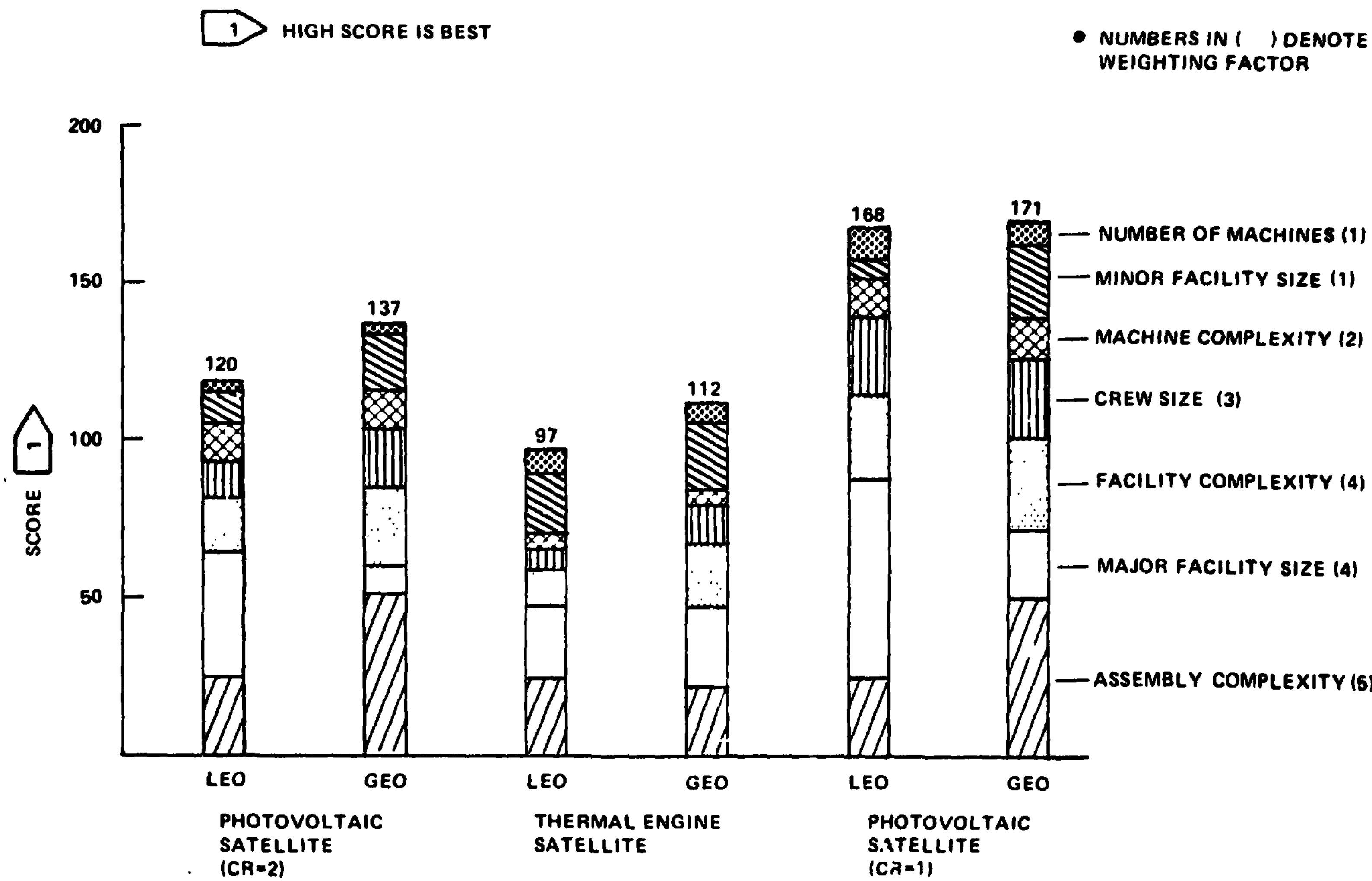

Figure 3.4-118 Preliminary Relative Constructability Rating

Table 3.4-46 Constructability Criteria Scoring Evaluation

SPS-932

		CONCEPT					
		PHOTOVOLTAIC CR=2.0		TURBINE ENGINE		PHOTOVOLTAIC CR=1.0	
		GEO	LEO	GEO	LEO	GEO	LEO
(1) NO. OF OPERATORS	SCORE						
600	10						
700	8			8		8	8
800	6	6					
900	4		4		4		
1000	2						
1000+	0						
(2) NO. OF MACHINES							
25	10						
50	9					9	9
75	8			8	8		
100	7		7				
125	6						
150	5						
175	4						
200	3	3					
250	2						
300	1						
(3) COMPLEXITY OF MAJOR MACHINES	SCORE						
SIMPLE	10						
	8					8	8
	6	6	6				
	4						
	2			2	2		
VERY COMPLEX	0						
(4) SIZE OF MAJOR FACILITY							
SMALLEST	10						10
	8		8				
	6						
	4			4	4	4	
LARGEST	2	2					
(5) SIZE OF MINOR FACILITY							
SMALLEST	10	10		10	10	10	
	8						
	6						
	4		4				
LARGEST	2						2

SPS-934

Table 3.4-46 (con't)

		CONCEPT					
		PHOTOVOLTAIC CR=2.0		TURBINE ENGINE		PHOTOVOLTAIC CR=1.0	
		GEO	LEO	GEO	LEO	GEO	LEO
(6) COMPLEXITY OF FACILITY	SCORE						
LEAST	10						10
	8					8	
	6	6					
	4		4	4			
MOST	2				2		
(7) SATELLITE ASSEMBLY COMPLEXITY							
CONTIGUOUS–NO DOCKING	10	10				10	
SMALL MODULES–DOCKING	5		5		5		5
LARGE STRINGS of MODULES TO BF DOCKED	4			4			

Evaluation Criteria Weighting

It should be obvious that these seven evaluation criteria do not have equal weight. Table Y shows how each of the criteria were compared to the others. This table was constructed by asking the question "Is criteria A a more important than criteria B or vice versa?" The most important criteria was noted. The total number of "votes" for each criteria was then added. The number of votes became the weighting factor for the criteria.

Constructability Score

The data from the preceding tables were summarized in Table 3.4-47. The scores from Table 3.4-46 were multiplied by the weights from Table 3.4-47 to obtain a "Product". All of the "Products" for a given concept were summed to obtain a "Total Score" which became the "constructability Rating" for the concept.

SPS-931

Table 3.4-47 Evaluation Criteria Weighting

EVALUATION CRITERIA	MOST IMPORTANT						
	1	2	3	4	5	6	7
1. NUMBER OF OPERATORS	–	1	1	4	1	6	7
2. NUMBER OF MACHINES		–	3	4	5	6	7
3. COMPLEXITY OF MACHINES			–	4	3	6	7
4. SIZE OF MAJOR FACILITY				–	4	4	6
5. SIZE OF SECONDARY FACILITY					–	5	7
6. COMPLEXITY OF FACILITY						–	7
7. SATELLITE ASSEMBLY COMPLEXITY							–

CRITERIA		NO. OF VOTES = WEIGHT
1	–	3
2	–	1*
3	–	2
4	–	5
5	–	2
6	–	4

*CRITERION NO. 2 WAS GIVEN 1 VOTE IN ORDER TO AVOID A ZERO MULTIPLYING FACTOR.

Table 3.4-48 Constructability Analysis

SPS-927

	CR=2.0 PHOTOVOLTAIC						THERMAL ENGINE						CR=1.0 PHOTOVOLTAIC					
	LEO CONSTRUCT			GEO CONSTRUCT			LEO CONSTRUCT			GEO CONSTRUCT			LEO CONSTRUCT			GEO CONSTRUCT		
EVALUATION FACTOR	WT	SCORE	PROD	WT	SCORE	PROD	WT	SCORE	PROD	WT	SCORE	PROD	WT	SCORE	PROD	WT	SCORE	PROD
NUMBER OF OPERATORS	3	4	12	3	6	18	3	4	12	3	8	24	3	8	24	3	8	24
NUMBER OF MACHINES	1	7	7	1	3	3	1	8	8	1	8	8	1	9	9	1	9	9
COMPLEXITY OF MACHINES	2	6	12	2	6	12	2	2	4	2	2	4	2	8	16	2	8	16
MAJOR FACILITY SIZE	5	8	40	5	2	10	5	4	20	5	4	20	5	10	50	5	4	20
MINOR FACILITY SIZE	2	4	8	2	10	10	2	10	20	2	10	20	2	2	4	1	2	20
COMPLEXITY OF FACILITY	4	4	16	4	6	24	4	2	8	4	4	16	4	10	40	4	8	32
SAT. ASS'Y COMPLEXITY	5	5	25	5	10	50	5	5	25	5	4	20	5	5	25	5	10	50
SCORE			120			137			97			112			168			171

3.5 TRANSPORTATION SYSTEMS

A definition effort was conducted to extend and refine the results of earlier studies of SPS transportation systems. Emphasis areas were:

- Definition of transportation element design requirements.
- Cargo Launch Vehicles, i.e., HLLV's.
- Refueling options for chemical orbit transfer vehicles.
- Personnel transport vehicles.
- Electric orbit transfer propulsion systems and their interfaces with SPS power modules
- Transportation costs.

Results of this effort are being separately documented and are briefly summarized in this report.

3.5.1 Requirements Summary

Cargo Launch Vehicles (Heavy Lift Launch Vehicles, HLLV's):

These vehicles have the primary function of delivering heavy cargo to low earth orbit. Most of this cargo will be SPS hardware and orbit transfer propellant. Low cost per unit payload mass delivered to low earth orbit is an overriding requirement. The following general vehicle requirements were identified.

- Recurring cost should be minimized. Accordingly, the vehicles should be completely reusable, with a design life of at least 300 flights, capable of fast recycle after use, employ low-cost propellants and minimize propellant energy consumption.

- A large payload volume capability should be provided; a payload density of 75 kg m^3 is representative.

- Large payload mass is desirable. Vehicles in the range 100 to 400 (metric) tons payload capability were studied. The high end of this range is desirable for a mature program; the smaller vehicles may be adequate in a developmental or early commercial phase.

- Vehicles and their launch facilities should be capable of sustaining high launch rates, reaching about 10 flights per day after several years' operations, and should allow salvo launches of two to five vehicles at roughly 1-minute intervals.

- The upper stage of the vehicle (or the entire vehicle, if a single-stage system) should be capable of flying to an operation base in low earth orbit to deliver its payload. Payloads will be palletized payload configuration to tanker configuration should be possible at the launch site without major disruption of launch processing operations.

- The design reference launch site is KSC. The design reference orbit is 478 km altitude at 31° inclination.

- The vehicle should be designed for minimum environmental impact. This includes (1) selection of propellants, engine cycles, and flight profiles that minimize atmosphere pollution, and (2) remote launch and recovery operations to the degree necessary to control noise

- In the event of an abort recovery of the vehicle given priority over recovery of the payload.

- The vehicles should have a return payload capability of roughly 10% of delivery capability to allow for return of empty tankers and payload pallets.

- The vehicles should minimize use and/or consumption of critical materials.

- The vehicles and their operational characteristics shall minimize safety of operations, flight crews, and the public. The vehicles should not be manned unless this is found to be necessary to meet one or more of the other system requirements.

Personnel Launch Vehicles:

The personnel launch vehicle was assumed to be an uprated shuttle with the payload bay converted to be capable of carrying 50 passengers. A liquid booster was assumed to replace the solids to reduce cost per flight and atmosphere pollution.

Orbit Transfer Vehicles:

Orbit transfer vehicles (OTV's) serve to transfer crews and cargo between low earth orbit and geosynchronous orbit. Orbit transfer vehicle requirements are summarized as follows:

- Low cost is paramount. Accordingly, the orbit transfer vehicle should use liquid oxygen and liquid hydrogen as propellants, should be completely reusable, should be staged to improve efficiency, should permit fast turnaround, and should be capable of at least 50 reuses.

- Space-basing is desirable. The vehicle should be designed for efficient on-orbit propellant transfer from tankers. Services such as propellant transfer pumping may be provided by an operations base.

- Mission duration capability should be a minimum of 7 days.

- It should be a design goal to eliminate all fluids requirements except LO_2 and LH_2, in order to simplify on-orbit servicing.

- The OTV should be matched to the cargo launch vehicle in the sense of having the capability to deliver an entire cargo payload to GEO without repacking at the LEO base. No cargo return payload is required.

- The OTV shall be capable of autonomous operation except for terminal rendezvous and docking, for which it shall be remote-piloted. The OTV GN&C system shall be capable of interfacing with avionics in a payload crew module for controls and displays and in that mode the OTV shall be controllable from the crew module.

- The OTV will provide no payload services except structural attachments and the above control interface.

- The OTV shall be designed for crew safety. The OTV flight profile shall avoid, even as a transient condition, state vectors that do not represent a stable earth orbit from which a rescue can be accomplished.

Electric Propulsion Orbit Transfer System:

The study indicated that minimum SPS system cost could be realized if SPS modules are constructed in low earth orbit and transferred to GEO under their own power using electric propulsion. Electric propulsion hardware must be fitted to the modules for this purpose. General requirements are as follows:

- Low cost is paramount. Therefore the electric propulsion hardware should be efficient (to minimize power consumption and resultant design scar on the SPS modules). It may be desirable to avoid the necessity for return of the electric propulsion hardware to low earth orbit for reuse. Therefore, this hardware should be designed for low production cost and minimum consumption of critical materials.

- The propellant should be plentiful and non-polluting, e.g. argon.

- The thrusting system should be capable of large gimbal angles as required by flight control.

- The system shall provide power processing as necessary to minimize total cost, including design/mass scars on the SPS modules.

- The system shall provide chemical thrust capability (total impulse and thrust level TBD) as necessary to control SPS module attitude when module power is not available. Up to 90 minutes chemical thrust operation shall be possible without module power.

- The system Isp shall be selected for minimum overall SPS cost. Depending on SPS characteristics, Isp's in the range 2500 sec to 7500 sec may be desired.

- The system shall be capable of at least 5000 hours operation without entering the wearout regime of failures.

- The system shall provide its own services, e.g. thermal control, drawing only unprocessed power and possibly control signals from the SPS module.

- The system shall be capable of at least 1000 electric thrust stops and restarts. Restart shall occur within 10 minutes after power is available from the SPS module.

3.5.2 Earth Launch Vehicles

Earth to low earth orbit transportation accounts for a significant portion of the SPS installation cost as indicated by Figure 3.5-1.

The following material in this section will address vehicle design requirements used; the candidate vehicle types including their characteristics, performance and operational characteristics; the results of the costing effort; and concludes with a summary of conclusions based on the results.

3.5.2.1 Cargo Launch Vehicles

The nominal cargo launch vehicle requirements are as follows:

Ground rules/requirements/assumptions

• Equivalent JSC scenario "B"	4 satellites/year for 28 years
• Delivery orbit	477.5 km (circular) at 31° inclination
• KSC launch	28.5° N. latitude
• Delivered payload	≈ 400,000 kg (net)
• Cargo packaging density	< 150 kg/m^3
• Nominal satellite mass	100 x 10^6 Kg
• Annual number of flights	
LEO assembly	1875
GEO assembly	3125

- Assume 5-day, 52-week, three-shift launch operations
- Design goal: eliminate expendable hardware

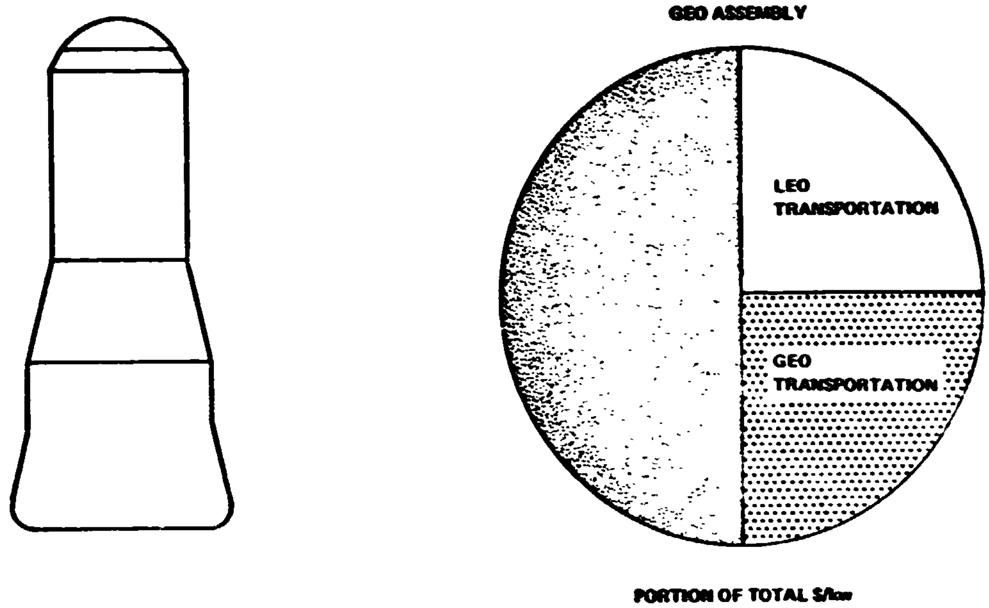

Figure 3.5-1 LEO Transportation

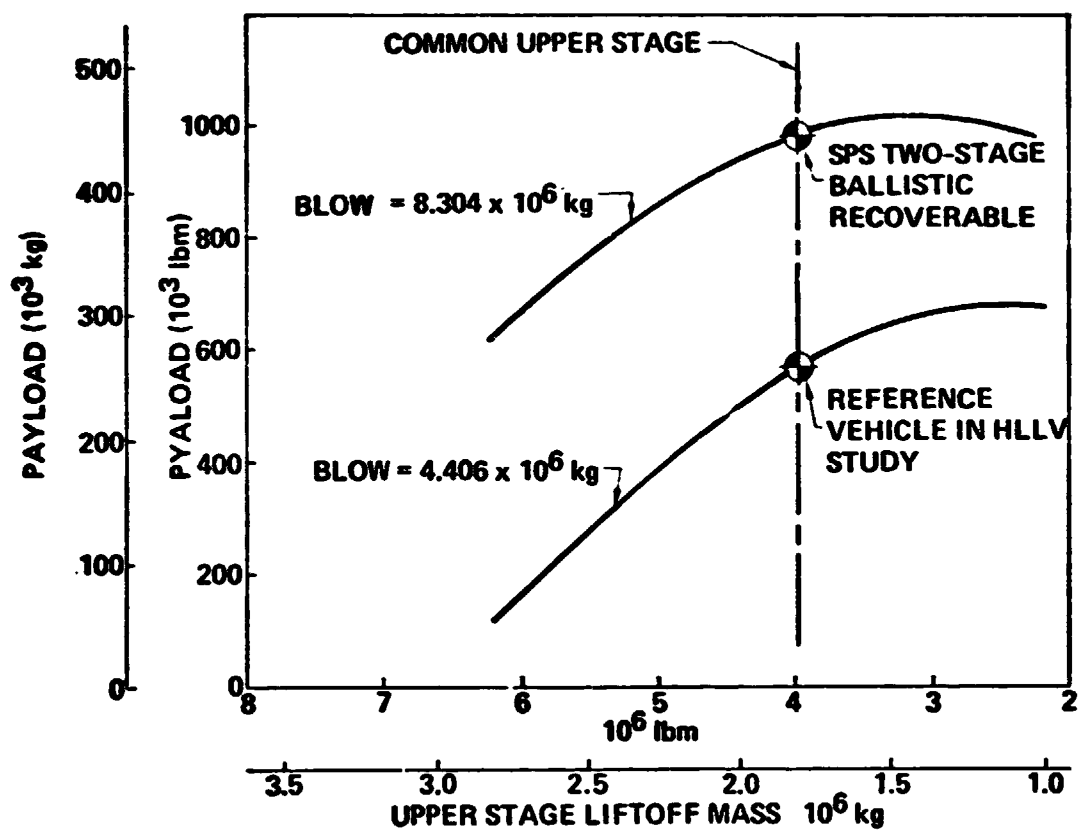

Figure 3.5-2 Vehicle Performance Trends

The equivalent of the 112 satellite JSC scenario 'B' which required a variable rate between 1 and 7 satellites per year was established for purposes of the transportation analysis at 4 satellites per year. The selected delivery orbit at a 31° inclination allows two launch opportunities a day approximately 3-1/3 hours apart. A reference number of flight as shown for the LEO and GEO assembly options was assumed so that parallel activities could be conducted on the study. The actual required flight rates are identified in the GEO Transportation Section.

Prior to developing the new configurations, vehicle sizing trends were investigated to determine the optimum first and second stage combination for a ballistic recoverable vehicle. The lower curve in Figure 3.5-2 is the trend for the point-of-departure cargo launch vehicle (heavy lift launch vehicle: HLLV) with variable upper stage characteristics. As noted the design point results in approximately 20% less payload than optimum. This is due primarily to the requirement for a 20 kg/m^3 payload density shroud which drove the vehicle to a larger diameter and therefore stage size. The upper curve represents the payload impact of a larger first stage and the design point was selected at the same upper stage mass as the reference HLLV vehicle.

The reference vehicle was a 2-stage series burn ballistic recoverable device which uses an expendable payload shroud. Two shroud sizes to satisfy 20 and 100 kg/m^3 payload densities are shown on the configuration sketch of Figure 3.5-3. The LO_2/RP-1 first stage uses 9 gas generator cycle engines at an ϵ=42.5 for boost. LH_2 is used for engine cooling and subsequently injected into the main chamber. The upper stage is powered by 7 standard SSME's at an expansion ratio of 77.5:1. The LH_2 and LO_2 propellants are contained in individual tanks. The payload shroud is jettisoned 60 seconds into the second stage burn.

The ballistic 2-stage vehicle shown in Figure 3.5-4 resulted from the configuration effort and is one of the candidate vehicles for the cargo vehicle. For GEO assembly, a version incorporating the same booster and upper stage but replacing the cargo payload section with a tanker section is also used. Sixteen new LO_2/RP-1/LH_2 engines of greater than 9 x 10^6N thrust each power the booster. The upper stage main propulsion is provided by 8 standard SSME's (E=77.5). A unique feature of the cargo version is the retractable payload shroud. The shroud is totally extended on the ground prior to payload installation and provides for a 75 kg/m^3 packaging density. Once on-orbit and after the payload has been deployed, the shroud is mechanically retracted for the entry configuration as shown on the upper left.

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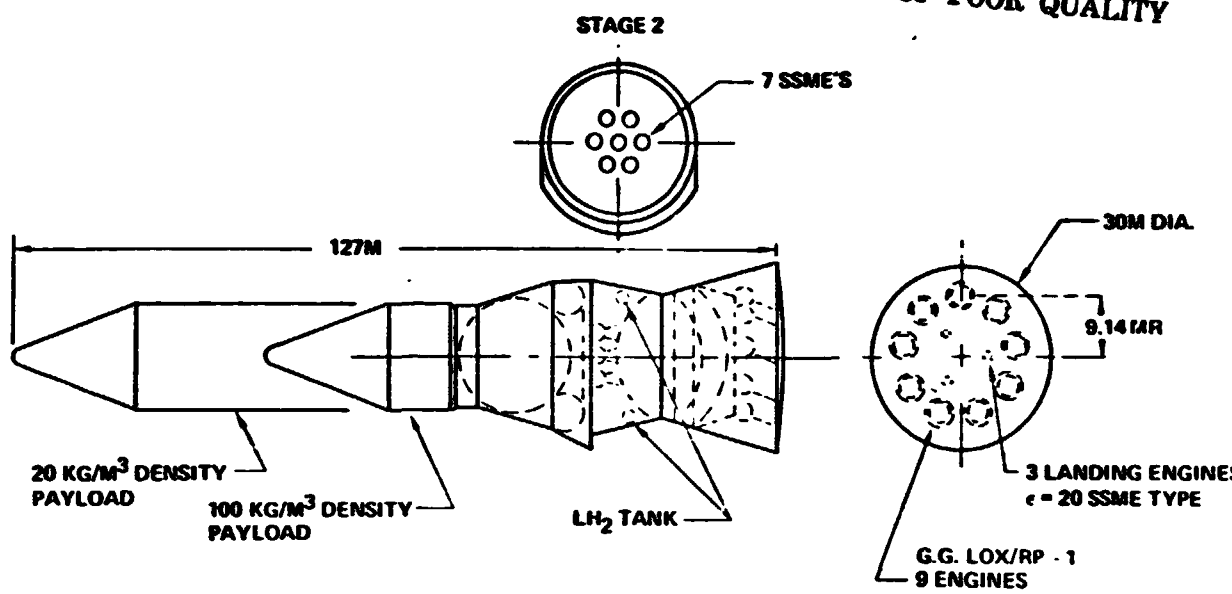
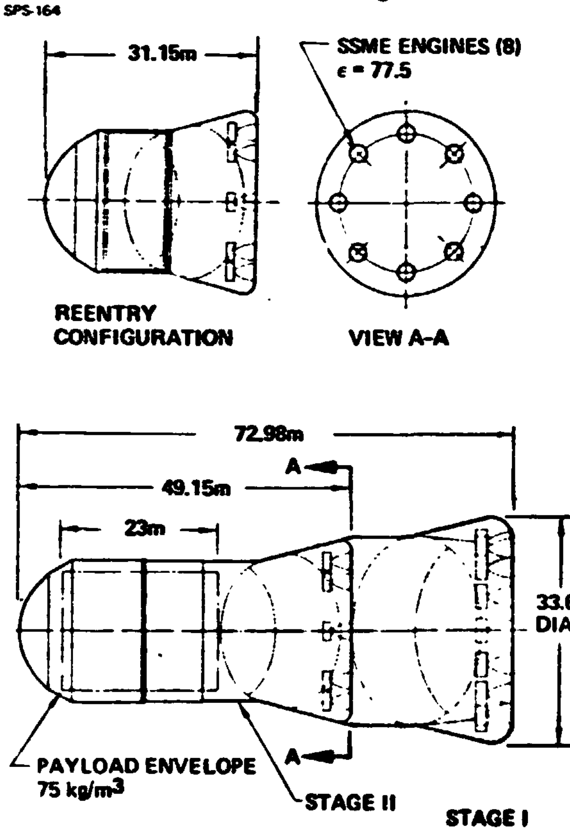


T = 8,780,000 N (S.L.) EACH (1,970,000 LB)

	10^6 KG	10^6 LB
GLOW	6.485	14.300
BLOW	4.405	9.714
ULOW	1.808	3.987
PAYLOAD	0.272	0.599

Figure 3.5-3 HLLV Two-Stage Configuration

SPS-164



GLOW	10.472×10^6 kg	(23.087×10^6 lbm)
BLOW	8.243×10^6 kg	(18.173×10^6 lbm)
W_{P_1}	7.456×10^6 kg	(16.437×10^6 lbm)
ULOW	1.838×10^6 kg	(4.051×10^6 lbm)
W_{P_2}	1.479×10^6 kg	(3.261×10^6 lbm)
PAYLOAD	0.391×10^6 kg	(0.863×10^6 lbm)

T/W AT LIFTOFF 1.30

MAIN PROPULSION

STAGE	ε	NUMBER TYPE	THRUST ENGINE (VACUUM)		EXHAUST VELOCITY		Isp - SEC
			10^6 N	10^6 lb	MPS	FPS	
1ST	42.5	16-LO_2/RP-1	9.060	2.037	3441	11290	[illegible]
2ND	77.5	8/STD SSME	2.091	0.470	[illegible]	[illegible]	[illegible]

VEHICLE CHARACTERISTICS

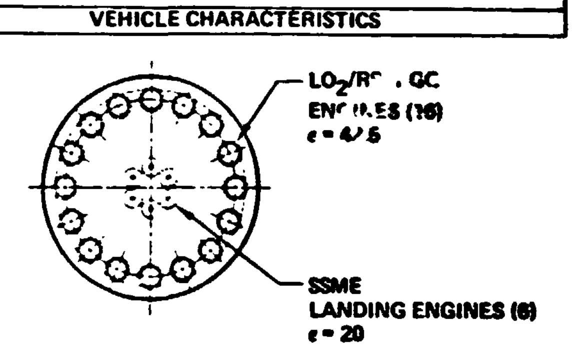


Figure 3.5-4 SPS Launch Vehicle Cargo Version

The booster stage mass statement of the 2-stage ballistic recoverable vehicle is shown in Table 3.5-1. The dry mass is 78% of the inert mass. The large inventory of fluids on board includes the water for base cooling during ascent and entry and also the landing propellant for terminal deceleration. The structure and main propulsion system account for 75% of the dry mass and the largest two single mass elements. A 10% mass growth allowance on the dry mass has been included. The resulting booster mass fraction is 0.904.

The second stage dry mass and mass history is shown in Table 3.5-2. Structure, main propulsion and the cargo shroud account for 84% of the dry mass. The mass growth includes 10% on all new developments, 5% on modifications of existing hardware, and 0% on use of existing hardware such as the SSME's. The second stage mass at main engine cut off (MECO) includes the stage and payload masses. The net payload delivered is 391 metric tons. Stage propellant loading is 1479 x 10^3kg. The overall stage mass fraction is 0.81.

An additional launch vehicle candidate for the SPS cargo flights is the 2-stage winged vehicle. A modified version of the JSC inhouse concept is shown in Figure 3.5-5. The vehicle incorporates 16-LO_2/RP-1/LH_2 engines on the first stage and 14-standard SSME's on the upper stage. An internal payload density of 135 kg/m^3 is available in the nose of the upper stage. Ascent control during boost is provided by 12 gimballing engines with 4 engines in the center fixed. The retractable booster nose cap eliminates the requirement for expendable interstage.

The booster stage mass statement is shown in Table 3.5-3. The booster staging velocity is about 1950 m/sec which allows a "heat sink" thermal protection system. Structure and Ascent Propulsion are the major subsystems and they account for 83% of the dry mass. A growth allowance of 10% is included on all dry mass items.

The second stage mass statement shown in Table 3.5-4 includes a breakdown of the dry mass and the stage mass history during the orbital maneuvers. The three major subsystems from a mass standpoint are structure, ascent propulsion, and thermal protection which account for 55%, 14%, and 14% of the dry mass respectively. The second stage sequence is noted on the right hand portion of the table. A ΔV reserve of 0.85% of the ideal ΔV was installed in the upper stage. A net deployed payload of slightly greater than 381 M tons resulted. Mass growth of 10% on all new hardware developments was included and no growth allowance was applied to the SSME mass data.

The estimated DDT&E cost of $7.6B and $9.1B for the 2 stage ballistic and winged vehicles respectively, include flight vehicle development, major system test, tooling and other program elements. Systems test includes 2-1/2 ground test units and 2 flight test units which are eventually recycled into the fleet. The 2 stage winged vehicle's estimated development cost is approximately 20% greater than the ballistic's. Costs are compared in Figure 3.5-6.

Table 3.5-1 2-Stage Ballistic Vehicle First Stage Mass Statement

SPS-284

Stage element	10^3 kg	10^3 lbm
Structure	283.65	625.34
Thermal protection system	44.47	98.04
Main propulsion	177.75	391.88
Auxiliary propulsion, RCS	1.49	3.28
Landing and auxiliary system	30.48	67.19
Prime power	0.74	1.62
Electric conversion and distribution	3.32	7.31
Hydraulic conversion and distribution	9.87	21.77
Avionics	2.43	5.36
Environmental control system	5.22	11.51
Mass growth (10%)	55.94	123.33
Dry mass (including H_2O for TPS)	615.36	1,356.63
Residual and unusable propellant	117.81	259.72
Reserve retro propellant	6.97	15.37
Usable RCS propellant	3.15	6.94
Usable retro propellant	44.40	97.87
Total inert	787.69	1,736.53
Ascent propellant	7 455.70	16,436.84
BLOW	8,243.39	18,173.37

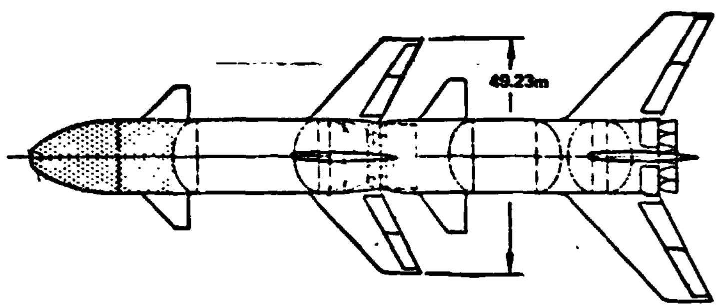
SPS-663

Table 3.5-2 2-Stage Ballistic Vehicle Second Stage Mass Statement

DRY MASS	
STAGE ELEMENT	10^3 kg
STRUCTURE	155.43
THERMAL PROTECTION SYSTEM	3.30
MAIN PROPULSION	29.85
AUXILIARY PROPULSION	6.15
PRIME POWER	0.48
ELECTRIC CONVERSION AND DISTRIBUTION	0.68
HYDRAULIC CONVERSION AND DISTRIBUTION	3.59
AVIONICS	1.59
ENVIRONMENTAL CONTROL SYSTEM	2.07
CARGO SHROUD	33.01
PAYLOAD SUPPORT SYSTEM	1.27
GROWTH	22.40
DRY MASS	258.82

SECOND STAGE SEQUENCE	
EVENT	MASS AFTER EVENT 10^3 kg
STAGE AT MECO	749.58
ΔV RESERVES	736.63
APOGEE CIRCULARIZATION (OMS BURN)	719.11
RCS TRIM BURN	714.76
OMS TRIM BURN	713.05
DEPLOY PAYLOAD (MASS = 391,460 kg)	321.60
DEORBIT ΔV	313.14
H_2O EXPENDED DURING ENTRY	301.12
LANDING RETRO	279.85
MASS AT LANDING	279.85
RESIDUALS AND UNUSABLES	14.28
RESERVES, LANDING PROPELLANT AND H_2O	6.75
DRY MASS	258.82

SPS-660



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

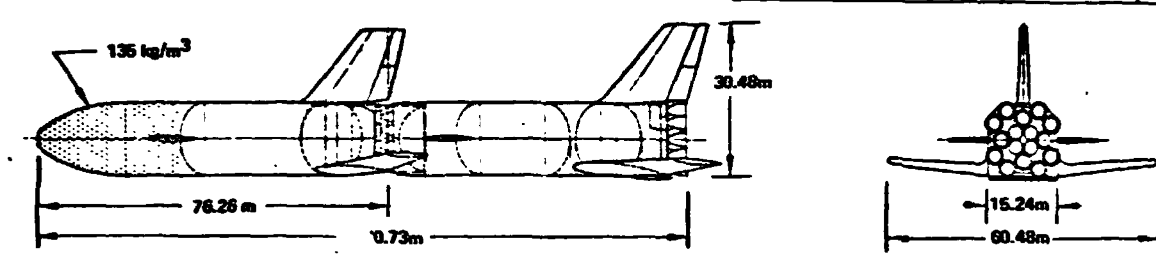


Figure 3.5-5 SPS Launch Vehicle (2 Stage Wing/Wing)

Table 3.5-3 SPS 2-Stage Wing/Wing Freighter Booster Stage Mass Statement

SPS-667

STAGE ELEMENT	10^3kg	
STRUCTURE	366.07	
BODY		(275.74)
AEROSURFACES		(90.33)
TPS	2.40	
LANDING GEAR	25.04	
ASCENT PROPULSION	167.52	
AUXILIARY PROPULSION	0.74	
PRIME POWER	3.04	
ELECTRIC CONVERSION & DISTRIBUTION	0.91	
HYDRAULIC CONVERSION & DISTRIBUTION	7.58	
AEROSURFACE CONTROLS	5.71	
AVIONICS	1.81	
ECS	2.61	
GROWTH	58.34	
DRY MASS	641.77	
RESIDUALS & UNUSABLES	90.00	
USABLE RCS & RESERVES	6.35	
INERT MASS	738.12	

	10^3kg
ASCENT PROPELLANT	5696.4
INERT MASS	738.1
BLOW	6434.5

SPS-656

Table 3.5-4 SPS 2-Stage Wing Freighter Second Stage Mass Statement

DRY MASS	
STAGE ELEMENT	10^3kg
STRUCTURE	199.47
BODY (140.93)	
AEROSURFACES (58.54)	
THERMAL PROTECTION SYSTEM	48.78
LANDING GEAR	13.55
ASCENT PROPULSION	52.23
AUXILIARY PROPULSION	3.27
PRIME POWER	1.52
ELECTRIC CONVERSION AND DISTRIBUTION	0.91
HYDRAULIC CONVERSION AND DISTRIBUTION	4.84
AEROSURFACE CONTROLS	4.25
AVIONICS	1.81
ECS	1.13
GROWTH	29.12
DRY MASS	360.88

SECOND STAGE SEQUENCE	
EVENT	MASS AFTER EVENT 10^3kg
STAGE @ MECO	813.67
ΔV RESERVE	790.66
APOGEE CIRCULARIZATION (OMS BURN)	780.64
RCS TRIM BURN	775.92
OMS TRIM BURN	774.05
DEPLOY PAYLOAD (MASS=381 120 kg)	392.93
DEORBIT ΔV	382.60
MASS AT LANDING	382.60
RESIDUALS AND UNUSABLES	11.49
RESERVES	10.23
DRY MASS	360.88

SPS 833

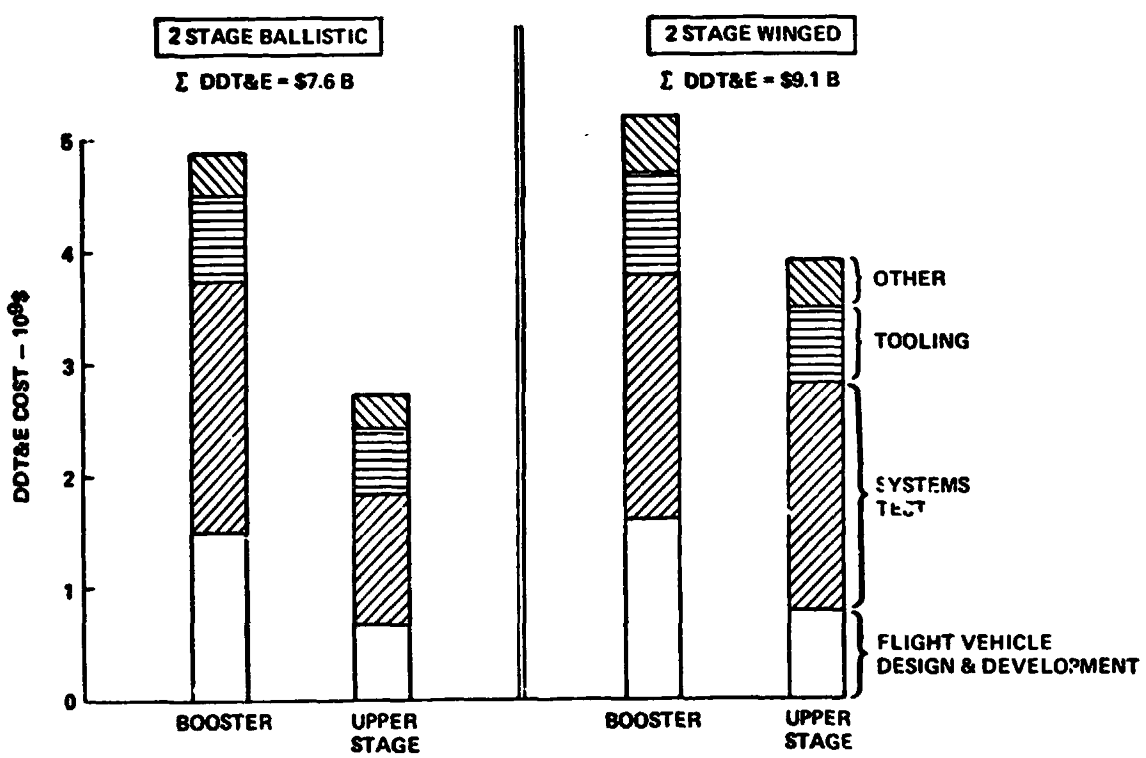


Figure 3.5-6 DDT&E Cost Comparison 2 Stage Ballistic vs Winged

The first production unit cost comparison for the two types of launch vehicles are shown in Figure 3.5-7. Structure, main propulsion and avionics are the major flight vehicle production cost elements. Included in the unit cost is a single shipset of Ground Support Equipment (GSE). A program management factor of 10% has been included for administration type functions The ballistic vehicle is about 11% less expensive than the winged version.

The typical weekly vehicle flow for the 2 stage ballistic vehicle is shown in Figure 3.5-8. The four staging base orbits compatible with the operation of four satellite construction bases are noted by the symbols on the chart. At the first opportunity to orbit (Northerly) we launch a cargo and tanker flight within 15 minutes of each other. On the second opportunity (Southerly) a tanker is launched to the same orbit. Similarly, the launches to the remaining three orbits occur as the launch opportunities occur. The vehicle turnaround times are noted on the bars of the chart. As noted in Figures, 36 and 45 first and second stages are required in the turnaround. Five (5) spare first stages and six (6) second stages are required and therefore an initial buy of 41 first and 51 second stages results.

Operations options for the ballistic/ballistic and winged/winged two stage HLLV's are shown in Figure 3.5-9. The ballistic vehicles are sea-recovered in order that entry sonic over-pressures will not occur over populated land areas. The winged vehicles land horizontally on a runway. The horizontal landing requirement may be met by uprange ship launch or by launch and recovery over an unpopulated land corridor.

3.5.2.2 Personnel Launch Vehicle

An updated version of the current SPS system for the crew rotation flights is shown in Figure 3.5-10. This series burn version incorporates a tandem mounted booster and smaller External Tank. Four propane engines of slightly greater than 8.5×10^6 new tons thrust power the booster. A reduced external tank propellant load, about 77% of the current SPS load, results in a smaller and less expensive expendable item in the system. Using a crew capacity of 50 men, 256 flights of this vehicle are required annually to support the four satellite/year construction rate.

The vehicle mass statement is shown in Table 3.5-5. The external tank dry mass includes growth which accounts for 5% on deleted items and 10% on new items. A potential payload of 73M tons is available excluding orbiter modifications required for the greater payloads.

3.5.3 Cost-Per-Flight Analyses

The cost per flight work breakdown structure (WBS) is shown in Table 3.5-6 The WBS is very similar to the Shuttle User Charge WBS but includes production cost on reusable hardware and tooling costs associated with the tooling shipsets required to support rate production. The following discussion describes the methodology in developing the element costs for the major items of the 2 stage wing/wing launch vehicle.

SPS 834

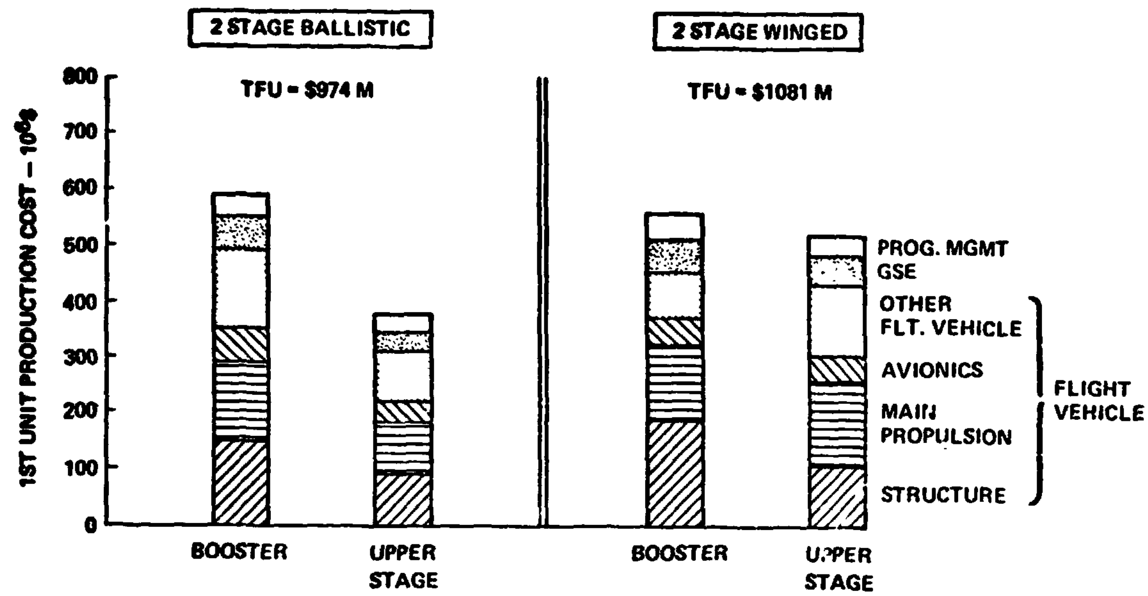


Figure 3.5-7 First Production Unit Cost Comparison

SPS 169

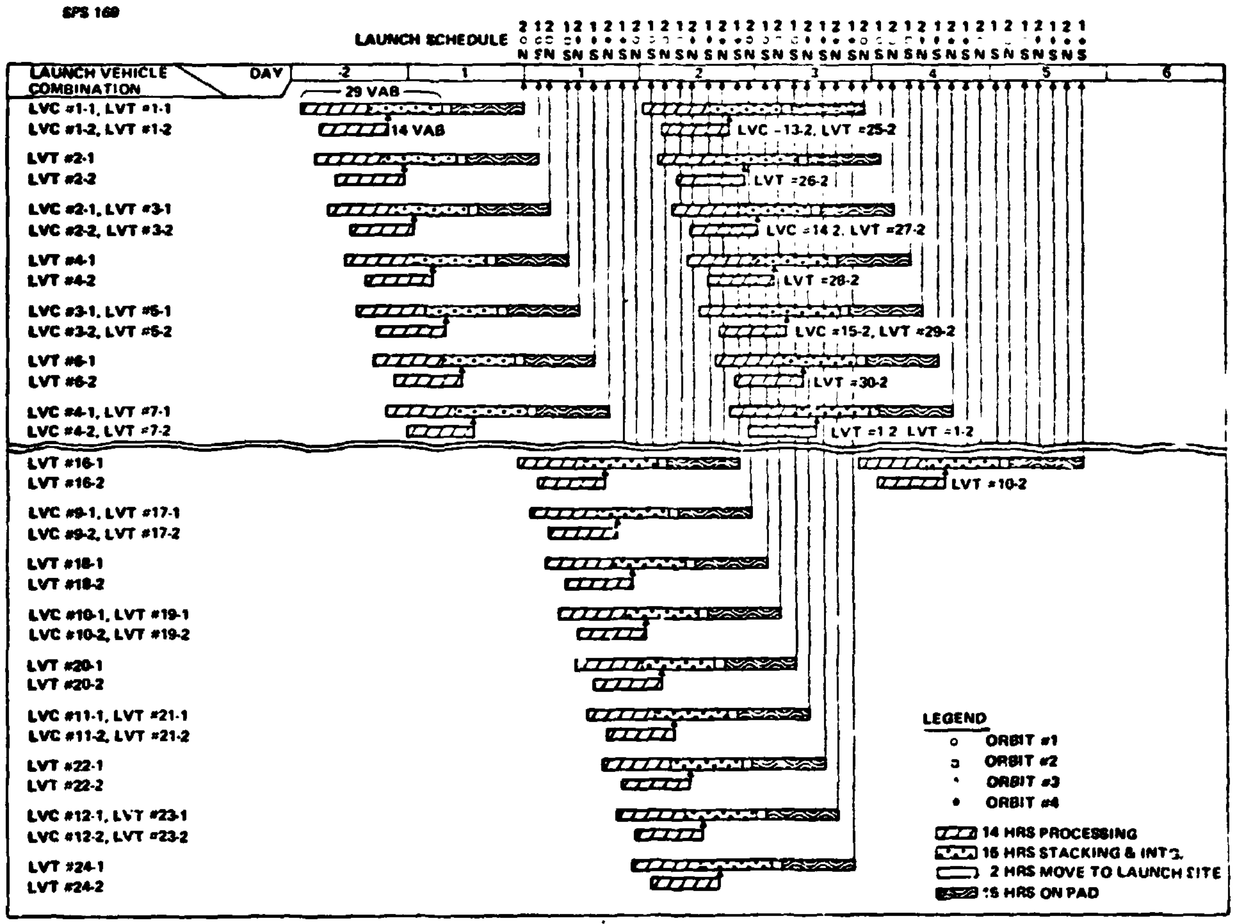


Figure 3.5-8 GEO Assembly Weekly Launch Activity Flow Diagram

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SPS 731

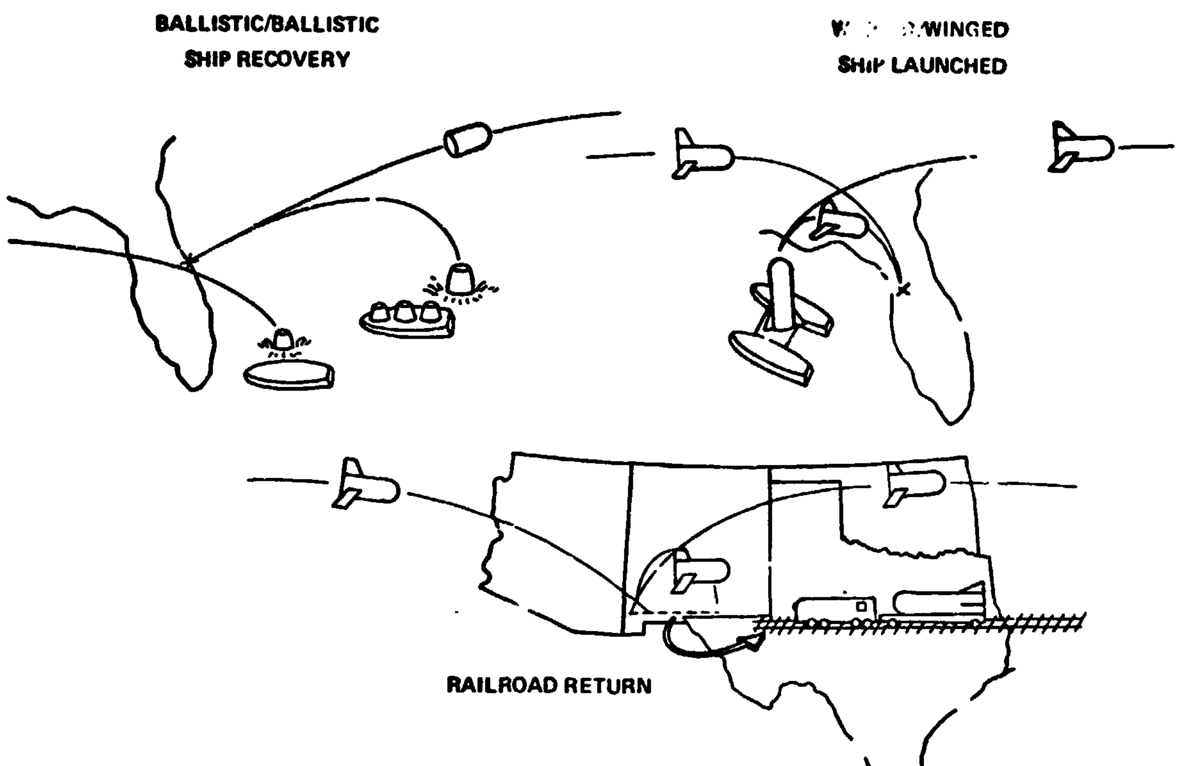
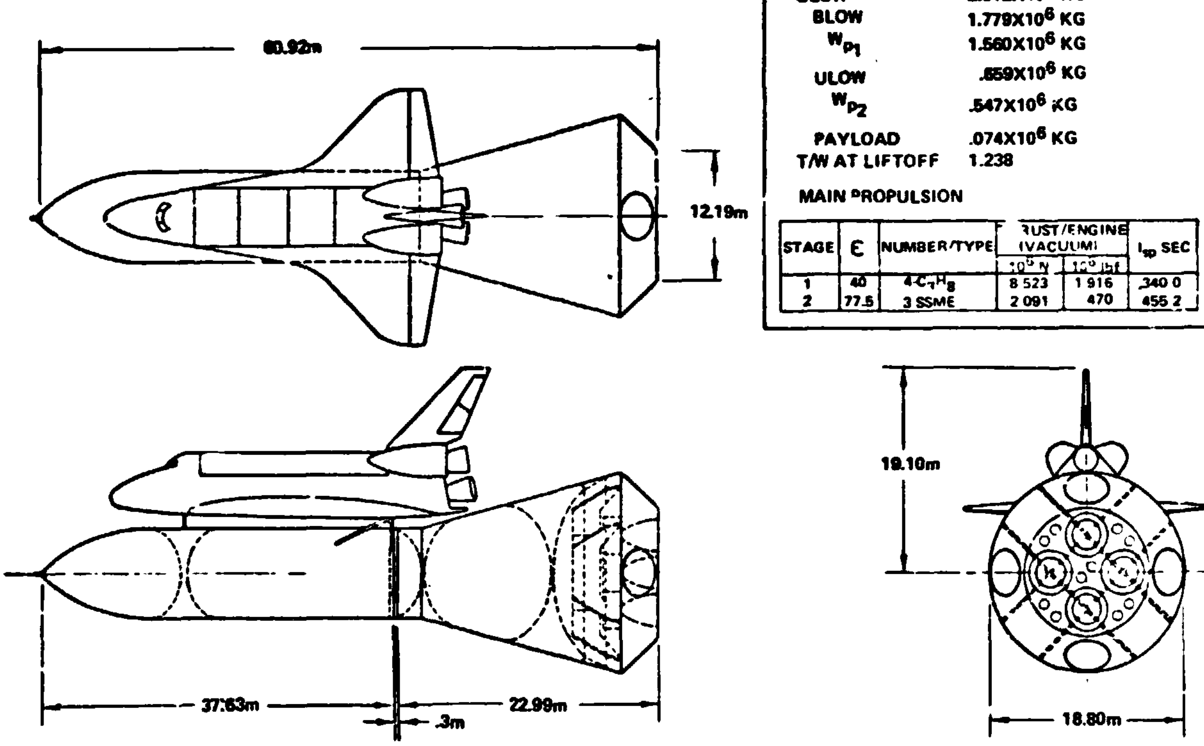


Figure 3.5-9 Operations Scenario

SPS-581



GLOW	2.512×10^6 KG
BLOW	1.779×10^6 KG
W_{P_1}	1.560×10^6 KG
ULOW	$.659 \times 10^6$ KG
W_{P_2}	$.547 \times 10^6$ KG
PAYLOAD	$.074 \times 10^6$ KG
T/W AT LIFTOFF	1.238

MAIN PROPULSION

STAGE	ϵ	NUMBER/TYPE	THRUST/ENGINE (VACUUM)		I_{sp} SEC
			10^3 N	10^3 lbf	
1	40	$4\text{-}C_3H_8$	8 523	1 916	340 0
2	77.5	3 SSME	2 091	470	455 2

Figure 3.5-10 Personnel Launch Vehicle

SPS 661

Table 3.5-5 Personnel Launch Vehicle 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	183.93
DROP ET	155.72
PERIGEE BURN	154.17
APOGEE CIRCULARIZATION	143.94
RCS TRIM	148.05
OMS TRIM	147.54
DEPLOY PAYLOAD (P/L = 73 550 kg)	73.99
DEORBIT ΔV	71.21

Table 3.5-6 Cost/Flight WBS

SPS-690

WBS ELEMENT
OPERATIONS COST
PROGRAM DIRECT
PROGRAM SUPPORT
PRODUCTION AND SPARES
STAGE 1
AIRFRAME
ENGINES
STAGE 2
AIRFRAME
ENGINES
TOOLING
STAGE 1
STAGE 2
GROUND OPS/SYS
GROUND OPS
GROUND SYS
GSE SUSTAINING ENGR
GSE SPARES
PROPELLANT
OTHER
DIRECT MANPOWER
CIVIL SERVICE
SUPPORT CONTRACTOR
INDIRECT MANPOWER
CIVIL SERVICE
SUPPORT CONTRACTOR

The production quantity of equivalent units for 14 years of operations shown in Table 3.5-7 includes: (1) the initial buy required to satisfy turnaround, (2) the additional vehicles required for life (using a 300 flight limit on service time), (3) refurbishment units resulting from a 30% replacement each 100 and 50 flights respectively for airframe and engines, and (4) replenishment spares purchased and installed at a rate of 0.18% and 0.50% per flight respectively for the airframe and engines. The 1st unit costs are noted and improvement curves of 85% and 90% on airframe and engines respectively, were used to develop the total program cost. The cost per flight was developed by averaging the total program cost over the 43750 flights which occur in the 14 years of operations.

The portion of cost per flight associated with rate tooling is shown in Table 3.5-8. The number of shipsets and the respective first unit cost are shown in the two columns on the left. The tool production costs results from using an 85% improvement curve for the units required. Tool sustaining was estimated at 10% per year of the production costs for the 14 years of operations.

Fourteen ground operations tasks were identified and manloaded as summarized in Table 3.5-9. The "hands-on" personnel were estimated for each operations task and the manpower associated with maintenance and corrective fixes also estimated. The tabular annual headcount for each task is noted and a total of nearly 24,000 people are involved. Since 36 vehicles are in the turnaround at any time, this averages 660 men per vehicle and a resulting cost per flight of $355,000. This is in addition to the stage refurbishment and repair activities included in the Production & Spares entry.

Estimates of the major NASA center and contractor manpower for program support are shown in Table 3.5-10. The average annual rates are estimated by extrapolating the Shuttle User Charge Data to 1977 dollars. The resultant headcount per vehicle is 4100. This is between one and two orders of magnitude greater headcount per vehicle than employed by a commercial airline such as United.

Propellant costs are shown in Table 3.5-11. Burden factors account for transfer losses. The energy value of $\underline{LH_2}$ and $\underline{LO_2}$ is sufficient that boiloff will be collected and reliquefied to the greatest extent practicable.

The total average cost per flight is $7.934M for the two stage winged vehicle when we include some of the other minor elements. Approximately 435,000 people would be involved in this total activity. A cost per flight summary is presented in Table 3.5-12.

The 2-stage ballistic recoverable vehicle cost per flight was developed in a similar manner as for the winged vehicle. The resulting total cost per flight was estimated at $7.615M, as summarized in Table 3.5-13.

SPS-605

Table 3.5-7 GEO Assembly Wing/Wing Vehicle Flight Hardware Cost/Flight Elements

PRODUCTION AND SPARES	UNIT QUANTITIES: INITIAL BUY TO SATISFY TURNAROUND	UNIT QUANTITIES: Δ BUY FOR LIFE	UNIT QUANTITIES: REFURBISHMENT	UNIT QUANTITIES: REPLENISHMENT SPARES	EQUIVALENT VEHICLE UNITS	THEORETICAL FIRST UNIT COST \$M	LEARNING CURVE λ(%)	TOTAL PROGRAM COST – \$M	COST/FLIGHT = TOTAL PROGRAM COST/43750 FLTS – \$M
STAGE 1									
AIRFRAME	41	105 (300 FLT LIFE)	88 (30% EACH 100 FLTS)	79 (.18% EACH FLT)	(313)	\$413.7	85	43700	0.999
ENGINES	656	N/A (INDEFINITE LIFE)	4004 (30% EACH 50 FLTS)	3500 (.50% EACH FLT)	(8160)	\$10.3	90	25193	0.576
STAGE 2									
AIRFRAME	51	95	88 (30% EACH 100 FLTS)	79 (.18% EACH FLT)	(313)	\$374.0	85	39503	0.903
ENGINES	714	N/A (INDEFINITE LIFE)	3461 (30% EACH 50 FLTS)	3063 (.50% EACH FLT)	(7238)	\$15.07	90	33304	0.761

- 1977 DOLLARS
- 14 YEAR PROGRAM

SPS-604

Table 3.5-8 GEO Assembly Wing/Wing Vehicle Tooling Cost/Flight Elements

	NUMBER OF SHIPSETS FOR RATE	TOOL FIRST UNIT COST \$M	LEARNING %	TOOL PRODUCTION COST \$M	TOOL SUSTAINING COST \$M [1]	COST/FLT \$M	
STAGE 1 AIRFRAME	10	\$408.9	85	\$2874	\$4024	\$.158	\$.259M
STAGE 1 ENGINES	54	\$67.9	85	\$1839	\$2575	\$.101	
STAGE 2 AIRFRAME	10	\$301.9	85	\$2149	\$3008	\$.118	\$.162
STAGE 2 ENGINES	47	\$33	85	\$802	\$1123	\$.044	

[1] 10% PER YEAR FOR 14 YEARS

- 1977 DOLLARS
- 14 YEAR PROGRAM

Table 3.5-9 GEO Assembly Wing/Wing Vehicle Ground Operations Cost/Flight Elements

SPS-601

	ANNUAL OPERATIONS HEADCOUNT			
	OPERATIONS		MAINTENANCE	
FIRST STAGE PROCESSING	978	- VEHICLE INSPECTIONS	1888	INSPECTION PICKUP & MAINT
SECOND STAGE PROCESSING	1442	–VEHICLE INSPECTIONS	2710	INSPECTION PICKUP & MAINT
MOBILE LAUNCHER ACTIVITIES	1075		4865	EQUIPMENT MAINTENANCE
FIRST & SECOND STAGE INSTALLATION ON MOBILE LAUNCHER	403			
VEHICLE INTEGRATION TESTING	161			
PAYLOAD INSTALLATION & CHECKOUT	161			
SUPPORT FOR MOVE TO LAUNCH SITE	242			
FIRST STAGE RECOVERY OPERATIONS	1913		344	EQUIPMENT MAINTENANCE
VAB TEST STATION	1566		576	EQUIPMENT MAINTENANCE
SECOND STAGE RECOVERY OPERATIONS	604		96	EQUIPMENT MAINTENANCE
LAUNCH CONTROL CENTER	1206		144	EQUIPMENT MAINTENANCE
LAUNCH SITE INSTALLATION & CHECKOUT	645		336	EQUIPMENT MAINTENANCE
PROPELLANT SYSTEM	1276		706	EQUIPMENT MAINTENANCE
GAS STORAGE & DISTRIBUTION	288		144	EQUIPMENT
Σ	= 11960		= 11807	

- 36 VEHICLES IN THE TURNAROUND AT ANYTIME

PERSONNEL/VEHICLE = 660

TOTAL COST = $1108M
COST/FLT = $0.355M

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Table 3.5-10 GEO Assembly Wing/Wing Vehicle Major Manpower Cost/Flight Elements

	ANNUAL HEADCOUNT	AVERAGE YEARLY RATE	ANNUAL COST $M	COST/FLIGHT
PROGRAM SUPPORT	23166	$38,000	$87	$.281
DIRECT MANPOWER				
CIVIL SERVICE	29400	$38,000	$1116	$.357
SUPPORT CONTRACTOR	30800	$33,000	$1016	$.325
INDIRECT MANPOWER				
CIVIL SERVICE	32900	$38,000	$1250	$.400
SUPPORT CONTRACTOR	31700	$33,000	$1047	$.335
Σ	= 147900			

HEADCOUNT/VEHICLE = 147900/36 = 4100

- UNITED AIRLINES HAS
  - TOTAL HEADCOUNT/AIRCRAFT = 125
  - MAINTENANCE HEADCOUNT/AIRCRAFT = 22

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Table 3.5-11 GEO Assembly Wing/Wing Vehicle Propellant Cost/Flight Element

	LOADED MASS (kg)	BURDEN FACTOR	PROPELLANT COST ($/kg)	COST/FLIGHT
FIRST STAGE				
LO_2	4 190 720	1.05	.095	417130
RP-1	1 444 560	1.02	.214	316090
LH_2	60 950	1.05	2.623	167900
SECOND STAGE				
LO_2	1969900	1.05	.095	196080
LH_2	328320	1.05	2.623	904400
			TOTAL PROPELLANT COST/FLIGHT	$ 2,000,600

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Table 3.5-12 2 Stage Wing/Wing Vehicle 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				0.356	
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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Table 3.5-13 Ballistic/Ballistic Vehicle Average Operating Cost/Flight

WBS ELEMENT	COST BY WBS LEVEL – $M				
	1	2	3	4	5
OPERATIONS COST	7.615				
PROGRAM DIRECT		6.198			
PROGRAM SUPPORT			0.281		
PRODUCTION AND SPARES			2.986		
STAGE 1				1.835	
AIRFRAME					0.943
ENGINES					0.892
STAGE 2				0.990	
AIRFRAME					0.617
ENGINES					0.473
PAYLOAD SHROUD				0.161	
TOOLING			0.383		
STAGE 1				0.258	
STAGE 2				0.107	
PAYLOAD SHROUD				0.018	
GROUND OPS/SYS			2.548		
GROUND OPS				0.379	
GROUND SYS				0.050	
GSE SUSTAINING ENGR				0.047	
GSE SPARES				0.091	
PROPELLANT				1.964	
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 3.5-11 illustrates both the average cost/flight and the payload transportation cost for the major options investigated. In addition, using the ballistic vehicle as reference, the *LEO Assembly* cost per flight is about 10% higher than for *GEO Assembly* due to the difference in the 1875 and 3125 flight rates respectively. The delivery costs range between $8.80 and $9.60 per pound of payload delivered.

Cost per flight results for the growth Shuttle personnel carrier are summarized in Figure 3.5-12. These data reflect a launch rate of 256 flights per year for 14 years. Additional orbiter productions is required to support this rate and is included in the cost figures.

A "rough order of magnitude" facility cost estimate was developed to identify the differences between LEO and GEO assembly shown in Figure 3.5-13. The 2-stage ballistic vehicle was selected as the reference. The number of facility units is tabulated on the chart and the estimated cost is shown on the bar graph. A $5.2B facility cost advantage for LEO assembly resulted from this preliminary analysis.

LEO Transportation summary based on the reference annual flight rate of 3125 and 1875 flights, for GEO and LEO assembly respectively, a $2.1B per satellite advantage results for LEO assembly. The primary difference is the number of freighter flights required.

Both ballistic and winged recoverable vehicles appear to be viable candidates and provide LEO transport costs of between $9 and $10 per pound of payload. Each as a number of specific concerns as noted below, but both candidates appear to be viable.

Ballistic vs Winged Launch Vehicle

- Performance and Cost/Flight are about equal ≈ $10/LBM
- Each type has unique concerns

Ballistic	Winged
Sea Recovery	Payload Density
Salt water compatibility	Higher DDT&E
Launch Siting	Launch & Booster Recovery Siting

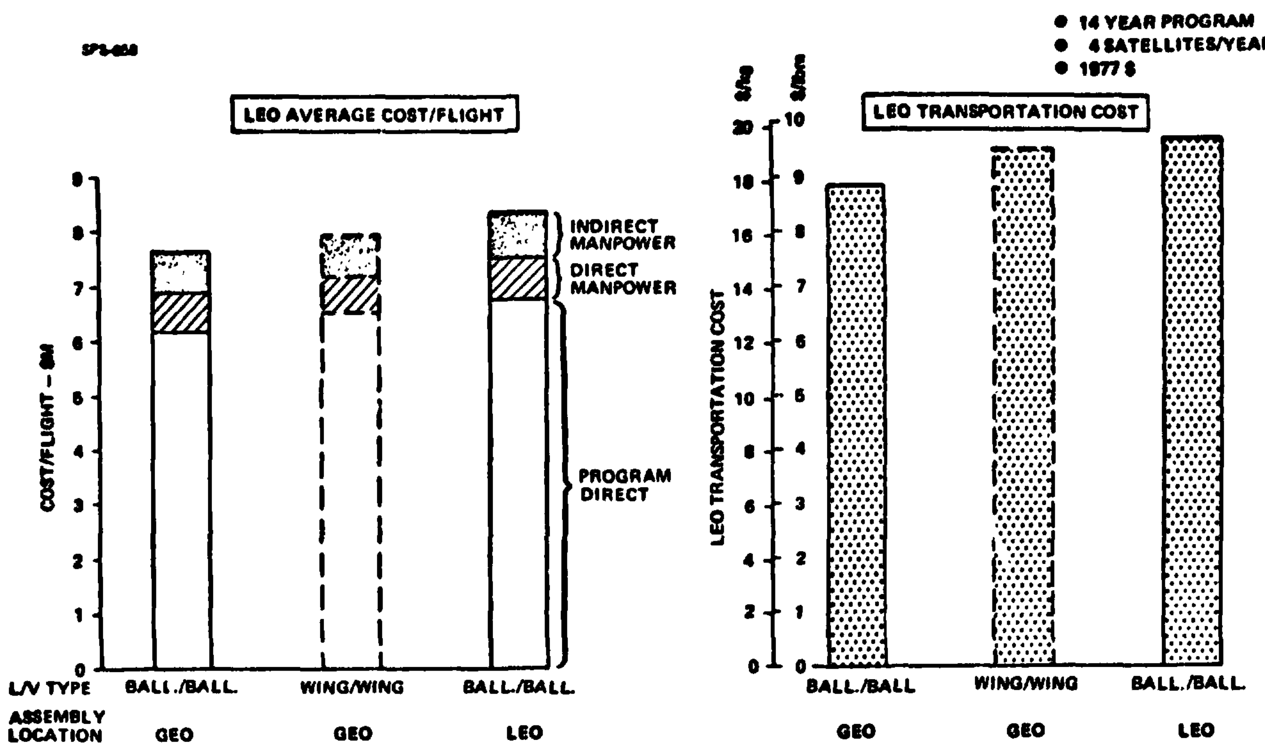

Figure 3.5-11 Comparison of LEO Transportation Costs

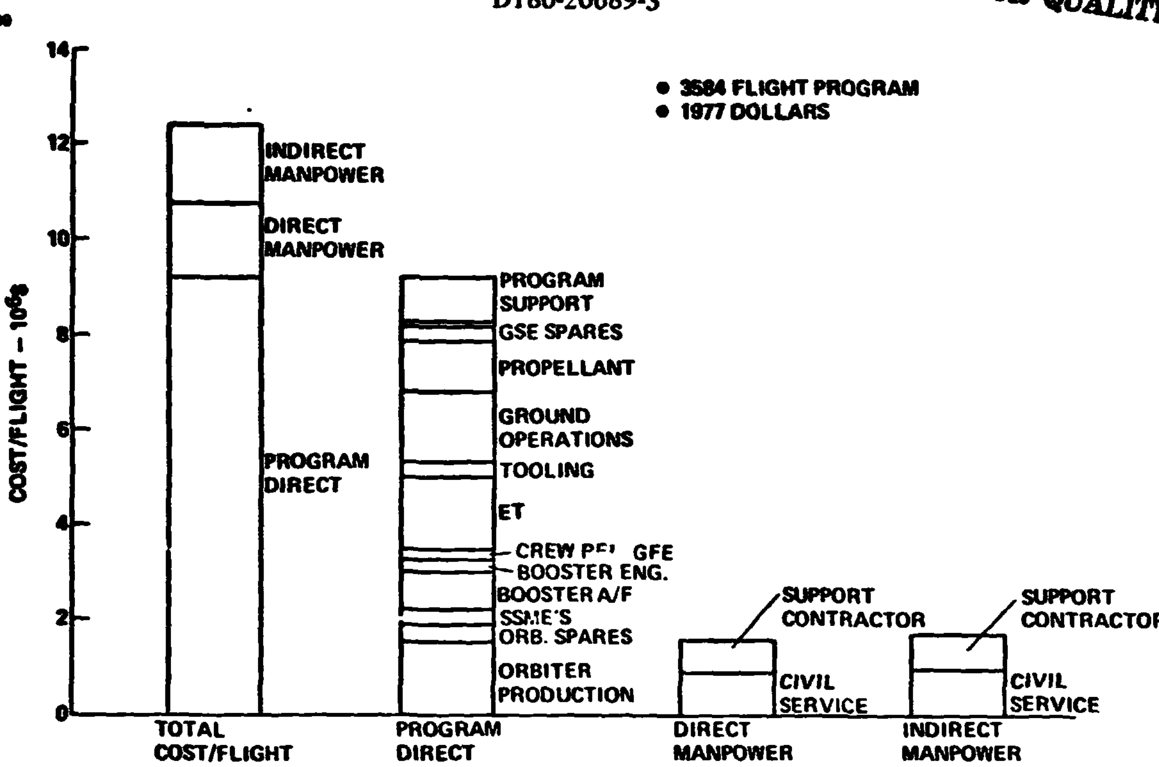

Figure 3.5-12 Personnel Carrier Average Cost/Flight

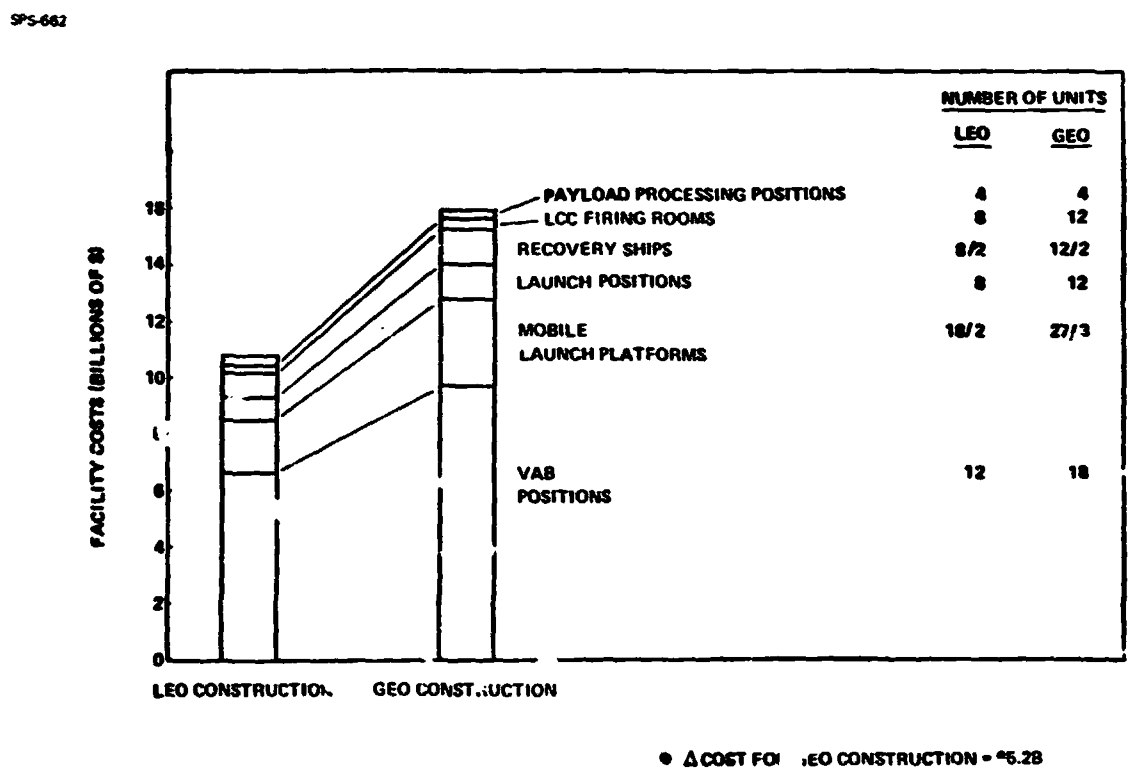

Figure 3.5-13 Launch Site Differentials Estimated Facility Costs

3.5.4 Orbit-to-Orbit Transportation

The orbit-to-orbit transportation discussion includes systems required for crew rotation and resupply, delivery of power satellites from LEO to GEO and a comparison of the transportation options in terms of cost and their cost sensitivities.

3.5.4.1 GEO Construction

The major operations associated with the use of a chemical orbit transfer vehicle in the GEO construction option are illustrated in Figure 3.5-14. The initial operations include the use of a space freighter to bring payload from Earth to a low Earth orbit (LEO) staging depot. The space freighter also brings propellant for orbit transfer vehicles based at the LEO staging depot. Payloads are transferred to the orbit transfer vehicle which in turn delivers the payloads to GEO where the components are then constructed into a power satellite. Following delivery of the components to GEO, the orbit transfer vehicle returns to the LEO staging depot for subsequent reuse.

Three types of chemical OTV systems were investigated in Part I as illustrated in Figure 3.5-15. The basic difference between these options is in the method of propellant handling. All options make use of the LEO staging depot. The first option is the space-based version. A two-staged vehicle is used with both stages identical in propellant capacity. Propellant for this system is brought to LEO by a launch vehicle and a tanker with propellant transfer occurring between the tanker and each of the OTV stages. The second option, identified as a mission tanker, again makes use of the ground based tanker. However, in this case, the tanker continues throughout the whole mission. Its propulsion systems and avionics are provided in a separate space-based module. Consequently, assembly of the tanker with the propulsion module is required for each stage; however, no propellant transfer is required. The third option, identified as a tanker OTV, is actually a ground-based orbit transfer vehicle. Again, a tanker is used, but in this case the engines and avionics are integrated directly into the tanker system and no propellant transfer or assembly of the stage is required. Preliminary analysis indicated the mission tanker has considerably more operational complexity than the tanker OTV. Consequently, the mission tanker was not included in performance and cost comparisons.

Comparisons of the space based and tanker OTV options for performance, the number of Earth launches required, and resulting satellite transportation costs are shown in Figure 3.5-16. The tanker OTV option required approximately 100,000 kilograms additional vehicle startburn mass, primarily as a result of the additional inert mass of structure and thermal control systems required for that vehicle due to launch loads and entry heating. This additional mass, in turn, translates into additional Earth launches required as indicated by the middle bar graph. When expressed as transportation costs for one satellite including both the launch vehicle and the orbit transfer operations, the tanker OTV results in about a 10% penalty over the space-based OTV. Consequently, the space based OTV was selected as the reference LO_2/LH_2 system

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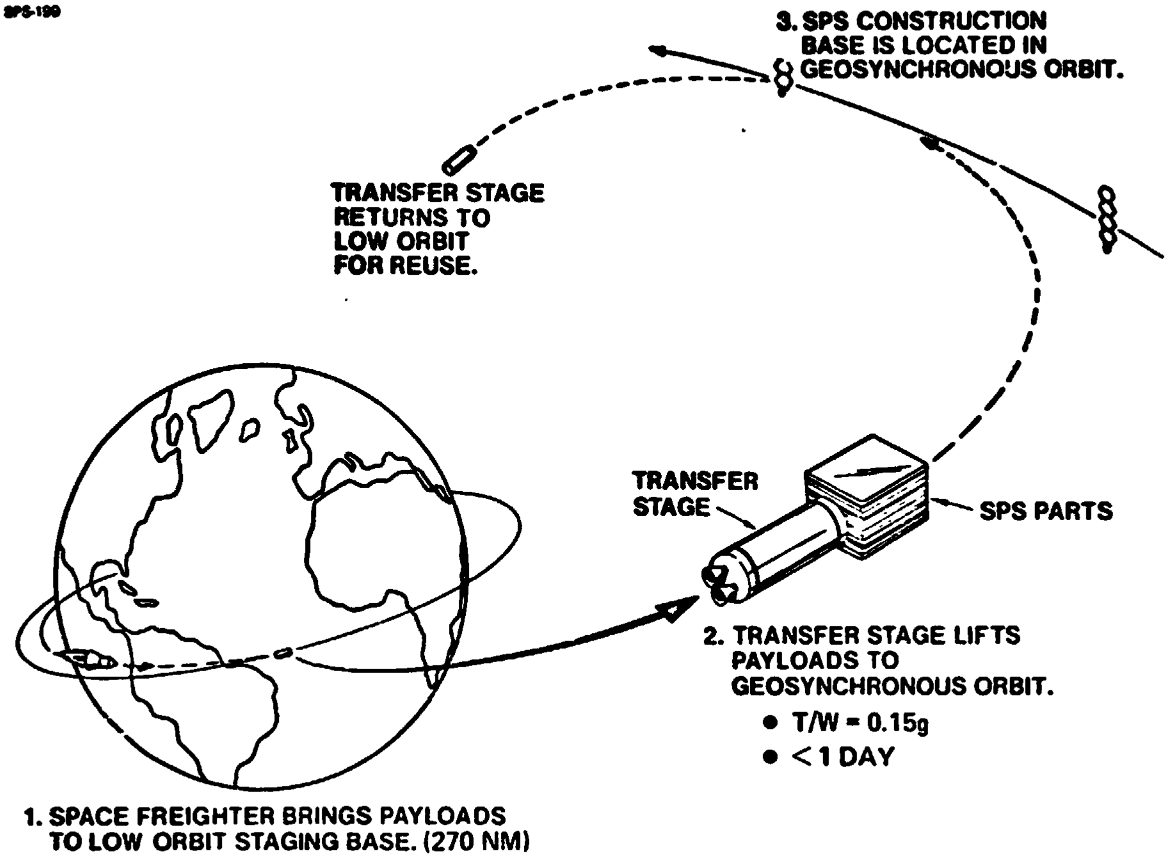

SPS 714 Figure 3.5-14 Chemical Orbit Transfer Operations GEO Construction

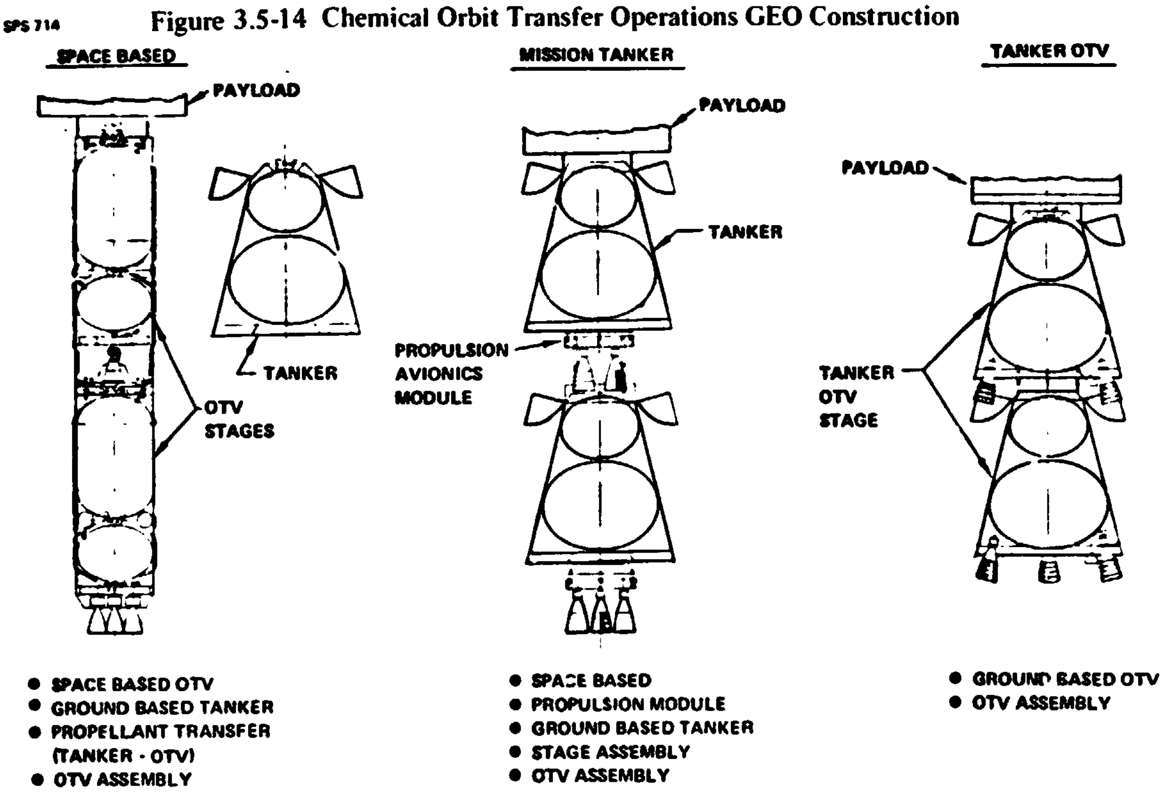

Figure 3.5-15 LO_2/LH_2 OTV Options

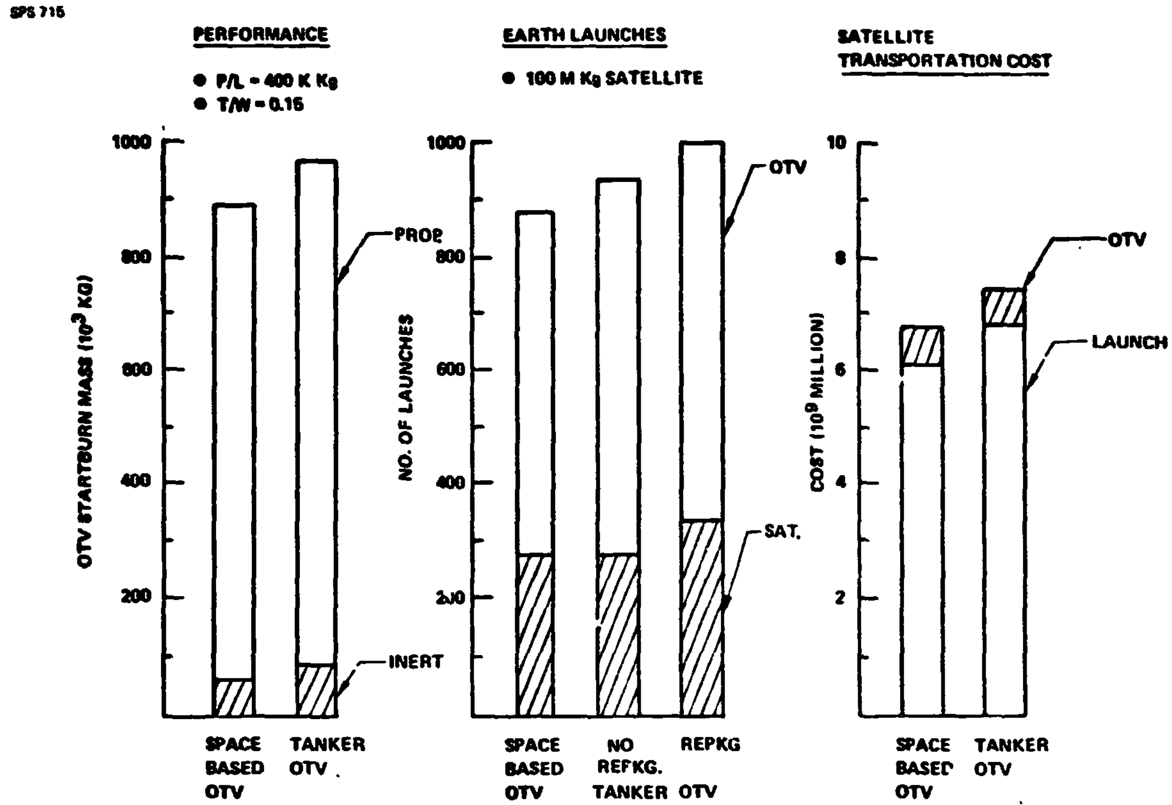

Figure 3.5-16 LO_2/LH_2 OTV Comparison

The space-based common stage OTV is a two-stage system with both stages having identical propellant capacity as shown in Figure 3.5-17. The first stage provides approximately 2/3 of the delta V requirement for boost out of low Earth orbit at which point it is separated 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 stage shown 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.

The requirements and implementation methods for crew rotation/resupply are shown in Figure 3.5-18. The primary requirements are the support of 100 men at LEO staging depot and 700 men at the GEO construction facility. Crew stay times are 90 days. Delivery of the crew to the LEO staging depot uses the shuttle growth launch vehicle with the delivery of 50 men per flight.

Delivery of the crew between LEO and GEO makes use of one stage of the two-stage orbit transfer vehicle previously described. It requires 28 flights per year. Propellant for the orbit transfer vehicle is delivered by the SPS HLLV. Crew and facility supplies will be delivered to the LEO staging depot, also used in the SPS HLLV. The majority of these supplies will in turn be delivered to the GEO construction facility using the two-stage SPS OTV; six flights per year are required for the delivery to GEO. Again, propellant for the orbit transfer vehicle will be delivered to the LEO staging depot using the SPS HLLV.

The ground rules used to establish the cost per flight of the chemical orbit transfer vehicle and the resulting cost per flight are as follows:

- 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
- 14 Year Program Life

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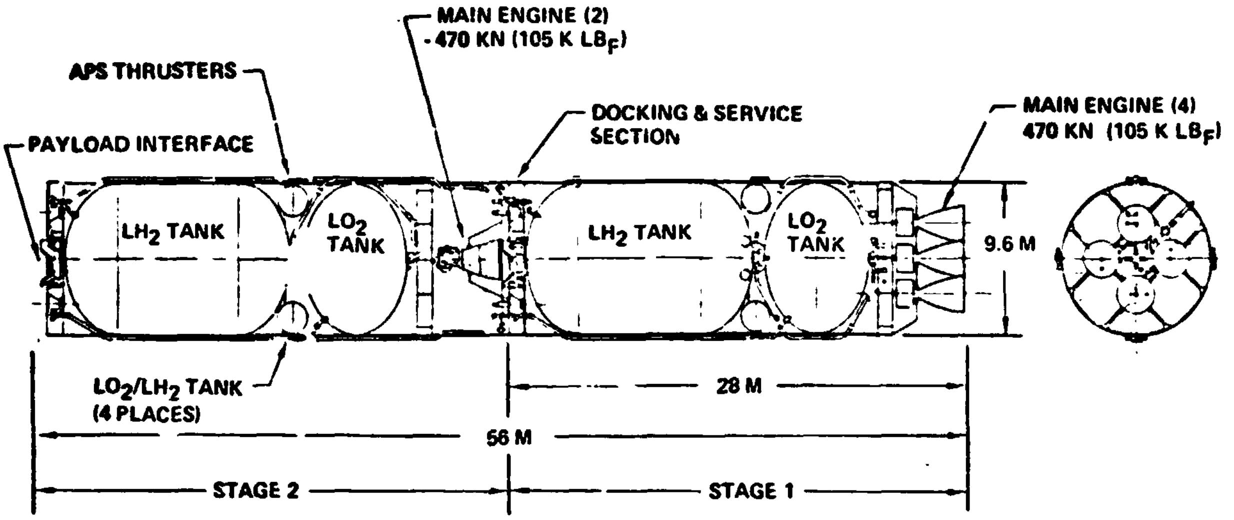

- PAYLOAD CAPABILITY = 400,000 KG
- OTV STARTBURN MASS = 890,000 KG
- STAGE CHARACTERISTICS (EACH
  - PROPELLANT = 415,000 KG
  - INERTS = 29,000 KG
    (INCLUDING NONIMPULSE PROPELLANT)
- 280 OTV FLIGHTS PER SATELLITE

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Figure 3.5-17 Space Based Common Stage OTV GEO Construction

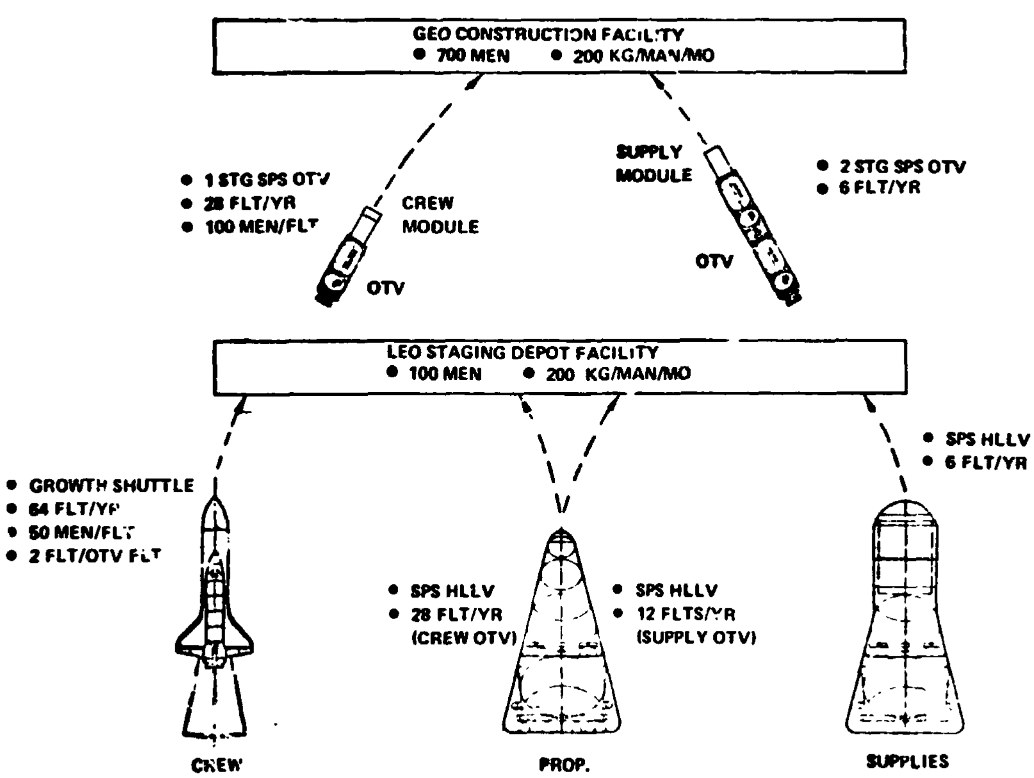

Figure 3.5-18 Crew Rotation/Resupply GEO Construction/Photovoltaic Satellite

- 50 Flight Design Life
- Stage Learning Factor of 0.88
- Spares Equal 50% of Operational Units

Cost Per Flight

• Operational Units		$1.24M
• Propellant		$0.40M
• Spares		$0.62M
	Total	$2.26M

The majority of these ground rules are self-explanatory. However, several merit further explanation. Stage theoretical first unit (TFU) costs are based on data developed during the FSTA study and updated to 1977 dollars. 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 spaced 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. Using these ground rules, the resulting cost per flifht is 2.26 million, including a total of 640 operational stages.

The transportation cost for the placement of one satellite at GEO using chemical orbit transfer vehicles and the crew rotation/resupply associated with the construction phase is estimated at 7.8 billion dollars for the reference photovoltaic system (10 GW BOL CR2). The transportation elements involved in this cost include the SPS HLLV which contributes 80% of the cost, a chemical orbit transfer vehicle at 10%, and the growth shuttle vehicle used to deliver crew to low Earth orbit also contributing 10%. These estimates are shown in piechart fashion in Figure 3.5-19.

3.5.4.2 Orbit-To-Orbit Transportation for LEO Construction

The major operations associated with use of an electric propulsion system in the transfer of satellite modules from LEO to GEO are indicated in Figure 3.5-20. Again, space freighters bring satellite components to LEO. However, in the LEO construction option, the components are assembled into satellite modules at LEO. The modules will have the capability to generate electric power which can

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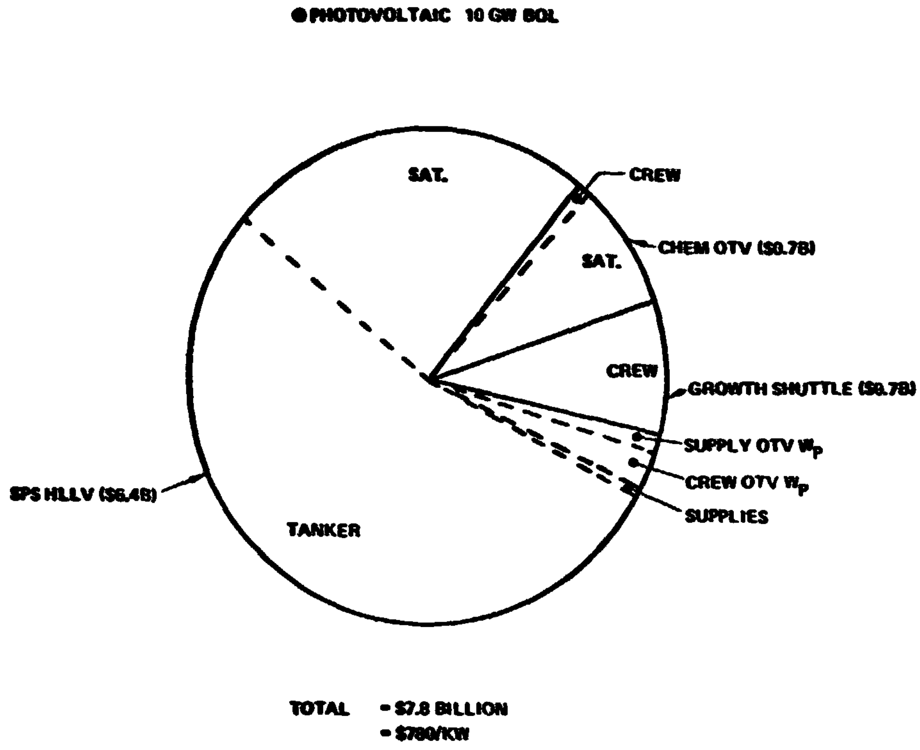

Figure 3.5-19 Transportation Cost To GEO Chemical OTS/GEO Construction

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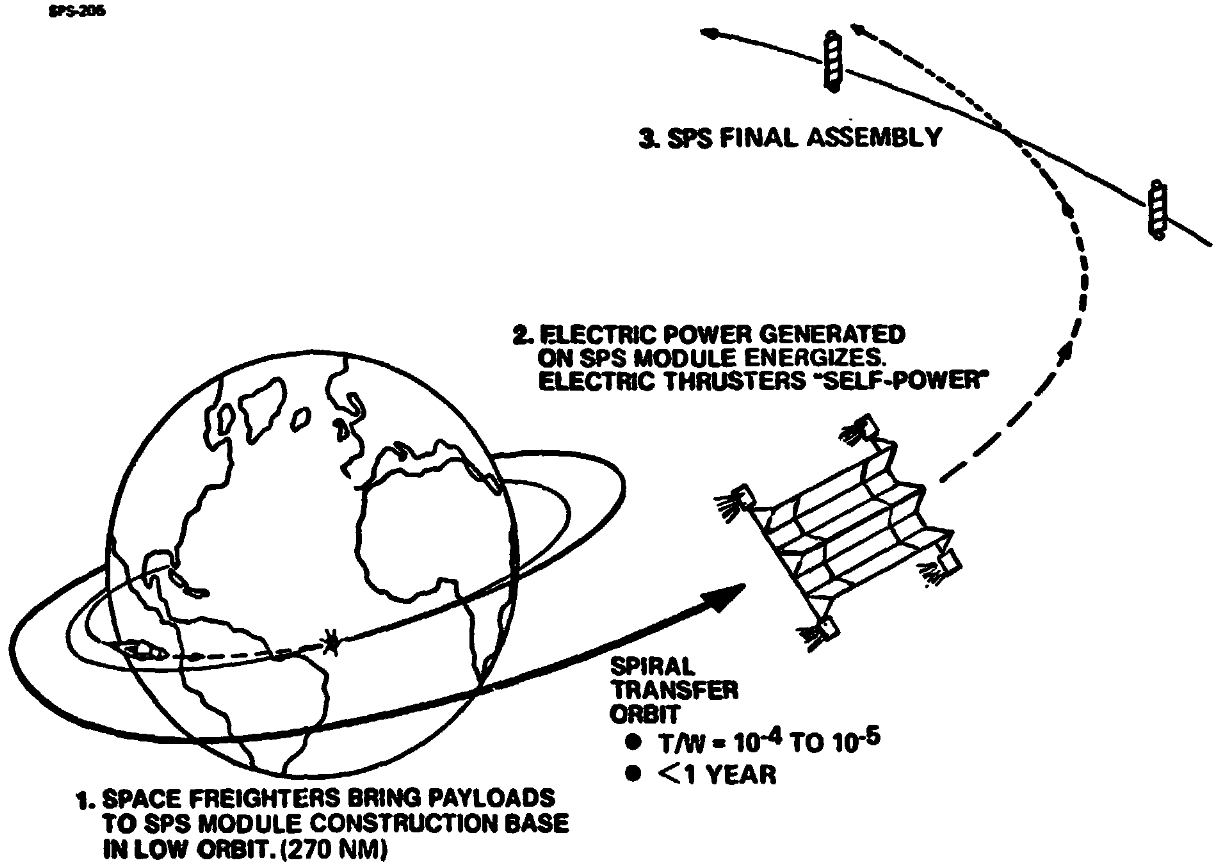

Figure 3.5-20 Electric Propulsion Orbit Transfer Operations LEO Construction

be used to drive electric thrusters that provide the thrust to move the satellite module from LEO to GEO. Transfer in this case, however, will be done at acceleration levels of 10^{-4} to 10^{-5} g's and result in trip times as long as six months to one year. After modules arrive at GEO, they then must be assembled into the final satellite configuration.

Seven major system elements make up the electric propulsion system as shown in Figure 3.5-21. 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, and the thrusters themselves which may be either ion or MPD devices. (Propellant for either ion or MPD thrusters is argon.) 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 factors and incorporated into an optimization model.

One of the principal variables in selecting a design point for the electric propulsion system is the thruster specific impulse. The principal ion thruster performance characteristics as a function of specific impulse are shown in Figure 3.5-22. Example influences of each of these parameters is as follows: Beam voltage will have an impact on the I^2R losses and the amount of plasma losses involved in the power distribution system; efficiency influences the amount of power required for the operation; thrust level will establish the number of engines required; and finally, the input power will determine the amount of solar array which must be deployed for the transfer operation. These characteristics along with trip time options were incorporated into the performance/cost optimization model.

The principal estimating factors used in costing a self-power system were as follows:

- Orbit Transfer System
  - Ion Thrusters (120 CM Dia-Argon) — \$2700 EA
  - Power Processing Unit (DC-DC Converter and Switch Gear) — \$50/KW_e
  - Radiator (Low Temp: 370°C) — \$50/Kg
  - Propellant Tanks (Cryogenic) — \$100/Kg
  - Installation Structure — \$100/Kg
  - Propellant (Argon) — \$0.10/kg
    (LO_2/LH_2) — \$0.40/kg (Bulk)

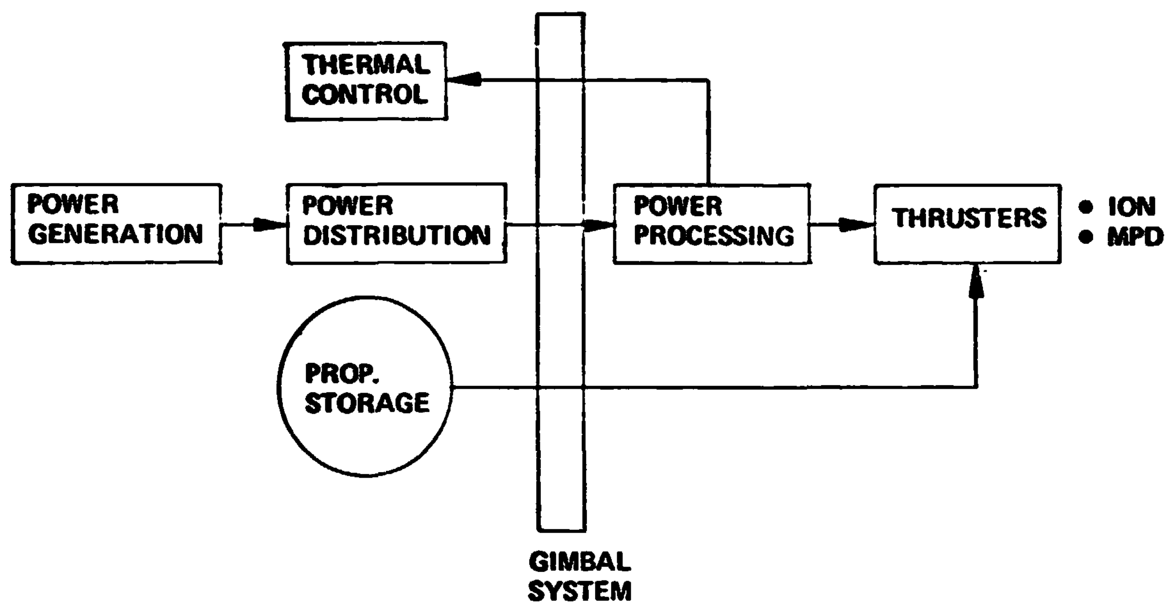

Figure 3.5-21 Electric Propulsion System Elements

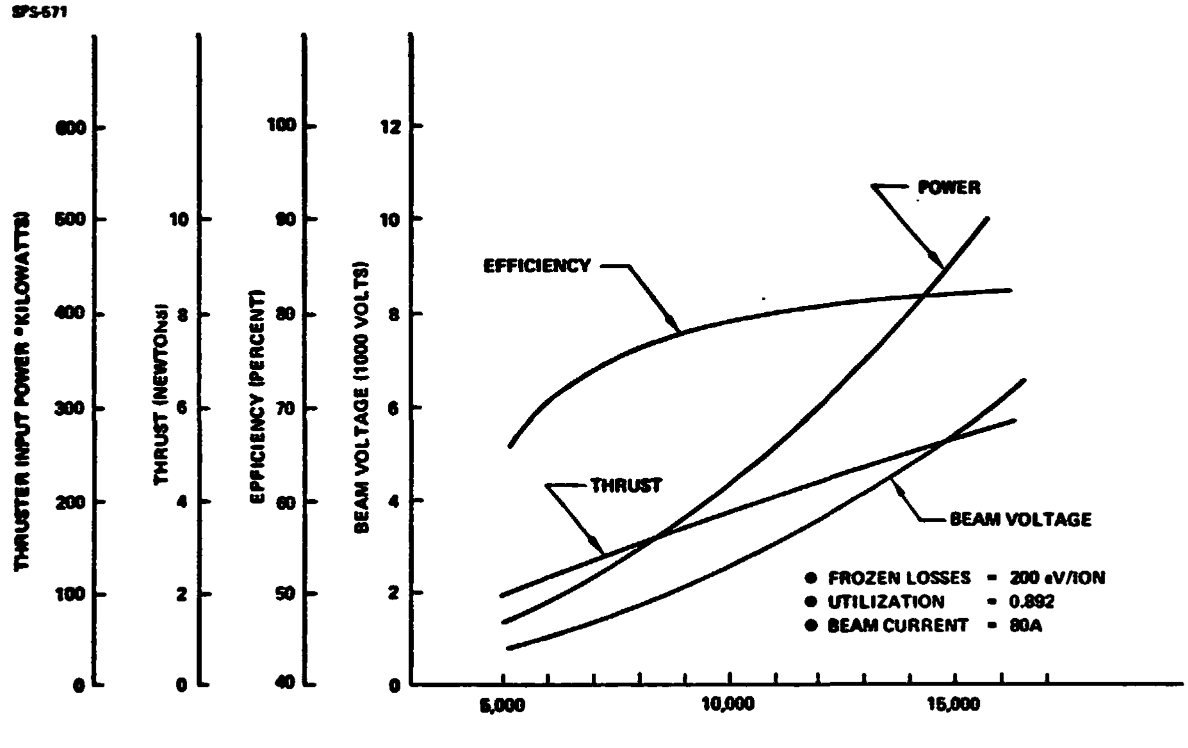

Figure 3.5-22 120-CM Argon Ion Thruster Performance

- Satellite Related
  - Satellite (Excl Mpts) — $5 Billion
  - Power Distribution — $20/Kg
  - Include Mass Growth Allowance — 25%
  - Launch System — $7.5 M/Flt
- Programmatic
  - Trip Delay & Other Interest — 7.5%

The effect of I_{SP} and trip time on transportation costs GEO for a 89 million kilogram satellite is shown in Figure 3.5-23. This particular case presumed a non-anneanable satellite so that the radiation damage incurred during the orbit transfer was permanent. Transportation cost reduces with lower I_{SP}, primarily because less power is required, resulting in less radiation degradation of the satellite. (Radiation degradation is compensated in the model by oversizing the satellite and the resultant cost is reflected as a part of the transportation cost. Only the solar cells actually used for the transfer are degraded, as the remainder need not be deployed.) Transportation cost also is reduced with trip times as long as 350 days. A constraint occurs on the trip time in the form of an attitude control limit. With transfer times beyond 200 days, the thrust levels available are so small that gravity gradient torque cannot be overcome. Consequently, for a satellite to be transferred with full attitude control capability, the transfer must be done less than 200 days.

The configuration arrangement of the system elements required to transfer each of satellite modules is shown in Figure 3.5-24. The module itself requires oversizing due to the radiation degradation of the solar blankets used for the transfer. (Approximately 22% of the solar arrays and reflectors must be deployed to provide the required power for the electric thrusters.) Thrusters and power processing systems are located at four corners of the satellite module. Each thruster power processing panel is connected to a gimbal system to enable required pointing. Propellant tanks for the thrusters have been located along the center line of the vehicle 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. The mass associated with the electric propulsion system consists of approximately one million kilograms for the oversizing and power distribution, while the orbit transfer system has a dry mass of approximately one million kilograms with approximately 2 million kilograms total of argon propellant for the electric thrusters and LO_2/LH_2 propellant (for attitude control during the occultation periods).

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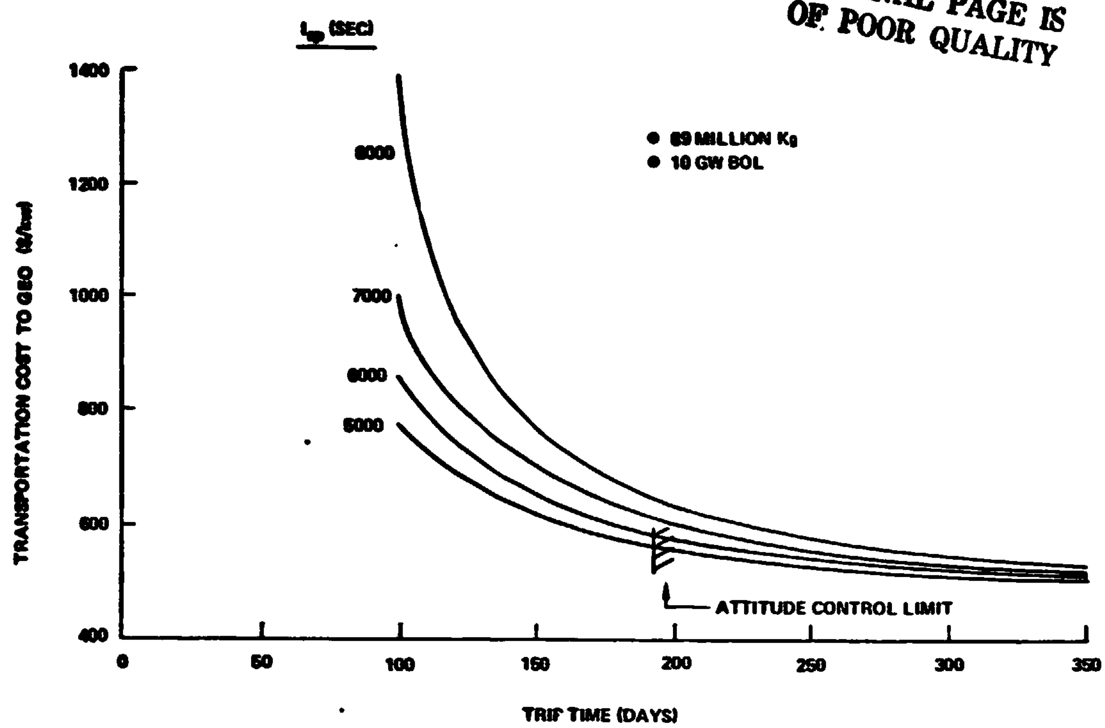

Figure 3.5-23 I_{sp} And Trip Time optimization Reference Photovoltaic Satellite

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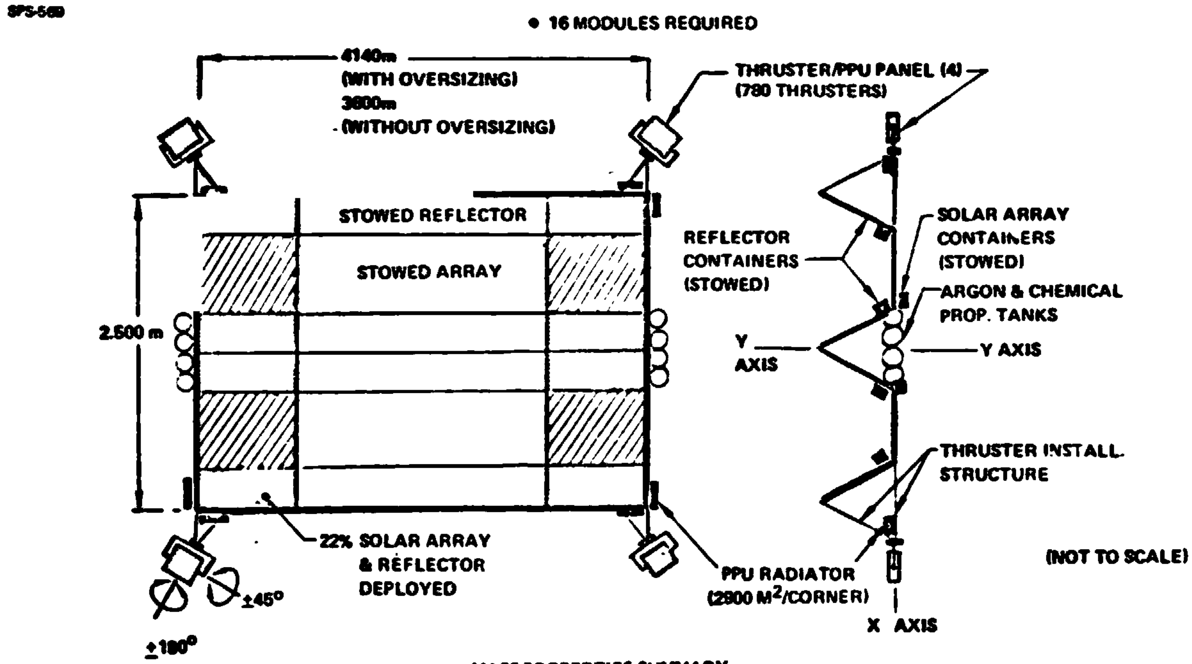

MASS PROPERTIES SUMMARY

SATELLITE MODULE	(10^6 Kg)	ORBIT TRANSFER SYSTEM	(10^6 Kg)
● BASIC	5.56	● DRY	0.71
● SELF POWER MODIFICATIONS	1.0	● ARGON PROP.	1.52
		● LO_2/LH_2 PROP.	0.38

● $I_{yy}-I_{xx} = 3.78 \times 10^{12}$ KgM

Figure 3.5-24 Typical Ion Electric Propulsion Configuration Photovoltaic Satellite (10 GW BOL)

For a satellite module being transferred from LEO to GEO using electric propulsion, the reference flight attitude maintains the solar arrays always aimed at the sun as indicated in Figure 3.5-25. The thruster thrust levels and the pointing angles shown in the figure are necessary to control gravity gradient torque at each of the clock positions around the Earth and to provide the required transfer acceleration capability. During the shadow period, chemical thrusters must be used to control the attitude. Should control not be employed during the shadow periods, the satellite would accelerate to a 0.1 degree per second rotation and as it reenters the sunlight will have rotated nearly 180° with solar arrays facing away from the sun.

The requirements for LEO construction crew rotation/resupply are different from these for GEO construction option primarily as a result of the difference in distribution of the personnel (rather than the quantity). Three hundred crew are required at GEO rather than 700. The method of implementing crew rotation/resupply is the same, as illustrated in Figure 3.5-26, but the number of SPS HLLV launches is only 40% that for the GEO construction option.

Transportation cost to place one satellite in GEO, for the reference photovoltaic (10GW BOL CR2) configuration, using electric self-power propulsion, and to support the crew rotation/resupply operation during construction, is estimated at 6.5 billion dollars or $650 per delivered kilowatt. SPS HLLV flight contribute 50% of this cost. The self-power orbit transfer system, including satellite modifications, contributes 20%, the shuttle growth vehicles used to deliver crewmen to LEO add 10%, chemical orbit transfer vehicles used to transfer crewmen from LEO to GEO add approximately 20% of the total cost. The largest contributors for the orbit transfer system are the thrusters and power processing units. In the area of satellite modification, the oversizing is the largest contributor. Cost estimates are displayed in pie-chart fashion in Figure 3.5-27.

Thermal Engine SPS Self-Power

The effects of I_{SP} and trip time for the thermal engine satellite on transportation costs to GEO, are shown in Figure 3.5-28. For this satellite, optimum trip time is considerably shorter and the I_{SP} higher than for the photovoltaic satellite. This situation is brought about because the higher power requirement for both conditions can be obtained without significant oversizing, the thermal engine SPS is less sensitive to radiation degradation. (Similar results were obtained for annealable photovoltaics). The selected I_{SP} is 7,000 seconds and the trip time is 160 days.

The thermal engine satellite module to be built in LEO and transferred to GEO is approximately 3 by 2 kilometers in size with a basic mass of approximately 6.25 million kilograms as shown in Figure 3.5-29. Power to drive the electric thrusters requires approximately 37% of the heliostats to be deployed, but in order to simplify the GEO construction operations, 100% of heliostats are deployed in LEO. Flight control and transfer acceleration requirements for this configuration can be accommodated through three thruster installation locations with approximately 700 thrusters at

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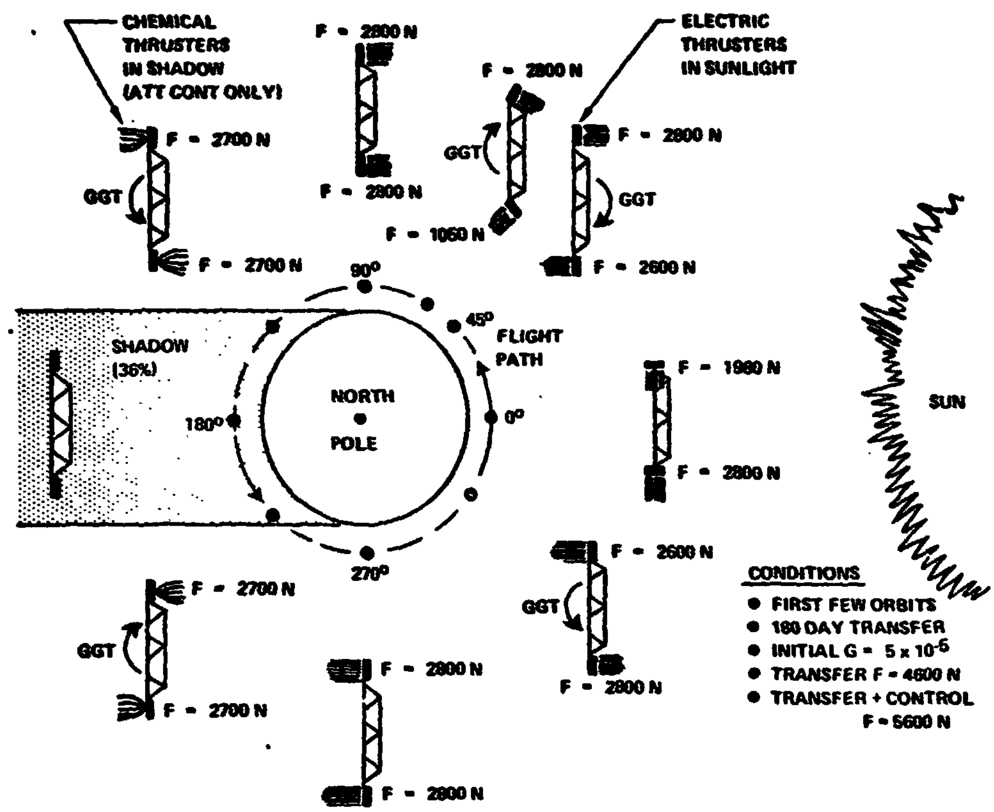

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Figure 3.5-25 Orbit Transfer Thruster Utilization Photovoltaic Satellite

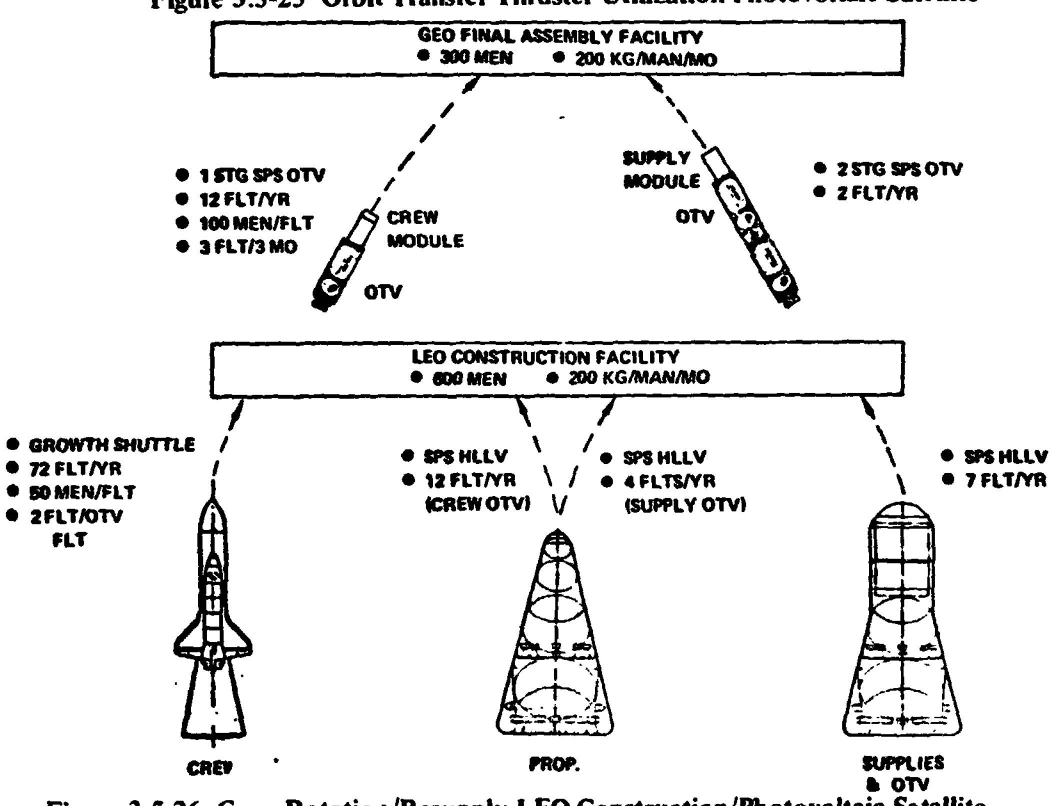
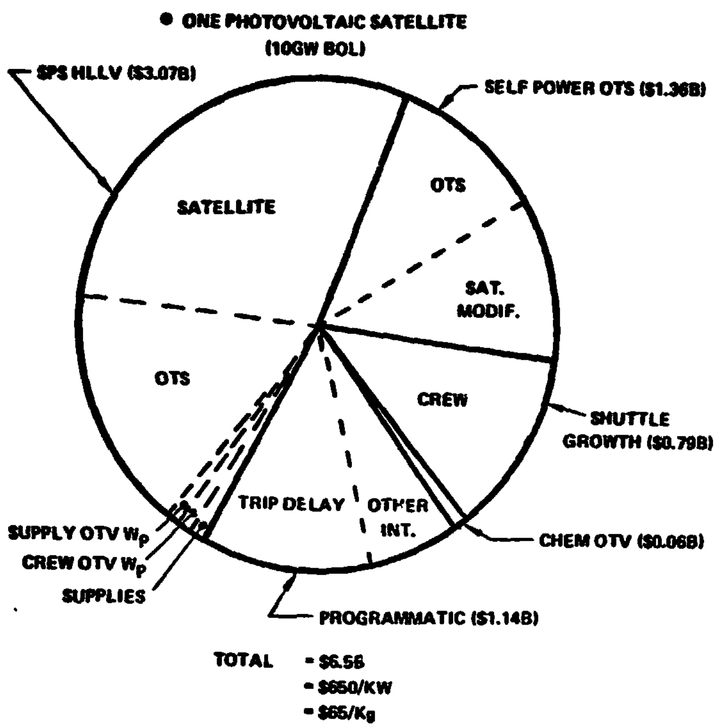


Figure 3.5-26 Crew Rotation/Resupply LEO Construction/Photovoltaic Satellite



SELF POWER OTS ($10⁹)	
• OTS	(0.64)
• THRUSTERS	0.14
• PPU	0.23
• TANKS	0.04
• STRUCTURE	0.14
• RADIATOR	0.09
• PROP	NIL
• CHEM ENG	NIL
• SAT. MODIF	(0.71)
• PWR DIST.	0.07
• OVERSIZING	0.64

SPS-688 Figure 3.5-27 Transportation Cost to GEO Electric OTS/LEO Construction

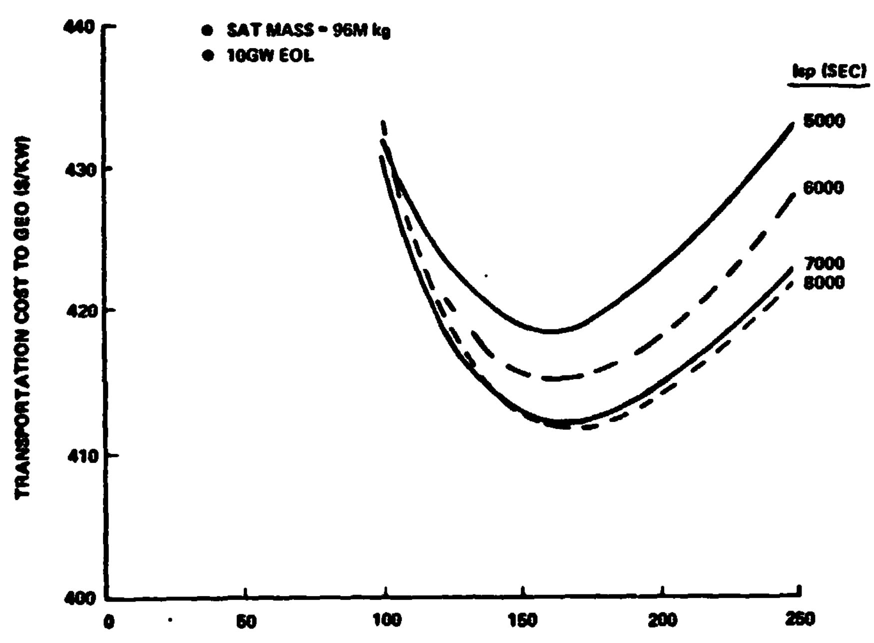


Figure 3.5-28 I_{sp} and Trip Time Optimization Thermal Engine - Brayton

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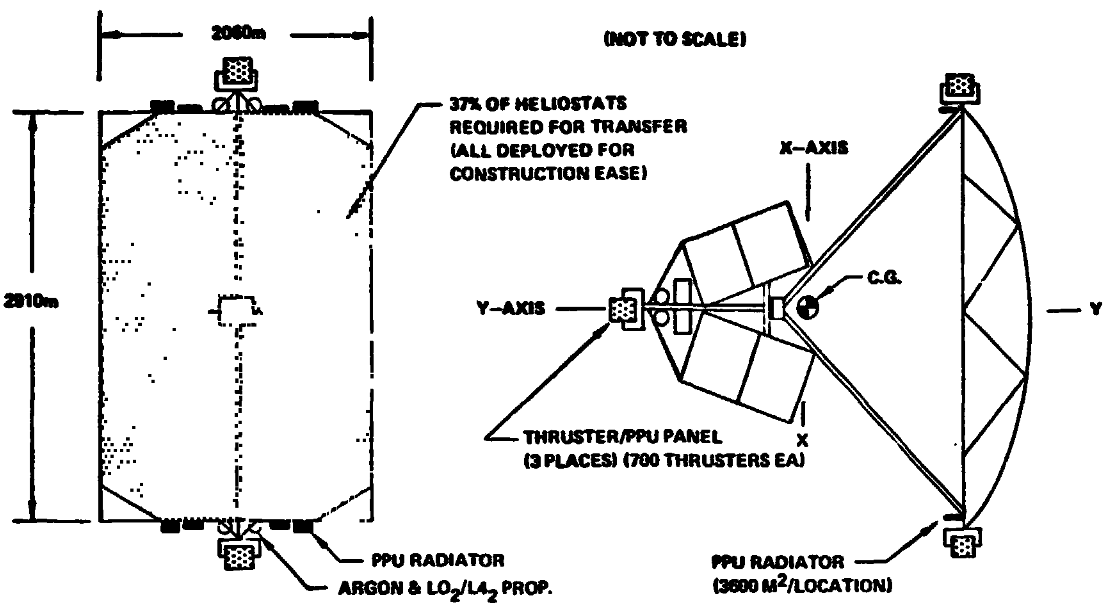

MASS PROPERTIES SUMMARY

SATELLITE MODULE (10^6 kg)		ORBIT TRANSFER SYSTEM (10^6 kg)	
• BASIC	6.25	• DRY	0.6
• SELF POWER MODIFICATIONS	0.10	• ARGON	1.28
		• LO_2LH	0.27

• $I_{xx} - I_{yy} = 0.54 \times 10^{12}$ KgM

Figure 3.5-29 Typical Ion Electric Propulsion Installation Thermal Engine Satellite (10 GW EOL)

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each location. Satellite modification to provide self power requires a small amount of oversizing and minimal power distribution modifications in terms of mass and orbit transfer penalty. The orbit transfer system dry mass is approximately 0.6 million kilograms and requires 1.5 million kilograms of propellant. The component affecting gravity gradient torque for the thermal engine satellite module is approximately 1/7 that of the equivalent photovoltaic satellite module.

Thruster utilization in terms of panels utilized, thrust level and pointing angle is illustrated in Figure 3.5-30 for the first few revolutions of the transfer of a thermal engine satellite module. Maximum thrust of a given panel is 2,000 newtons. Chemical thrusters are used during the shadow periods of the orbit as for the photovoltaic module. However, in this case the thrust is considerably less than for the photovoltaic satellite module due to the lesser inertia. Without control during thrust is shadow periods, the module would be off sun-aiming by approximately 20°.

3.5.5 Transportation Options Comparison and Sensitivities

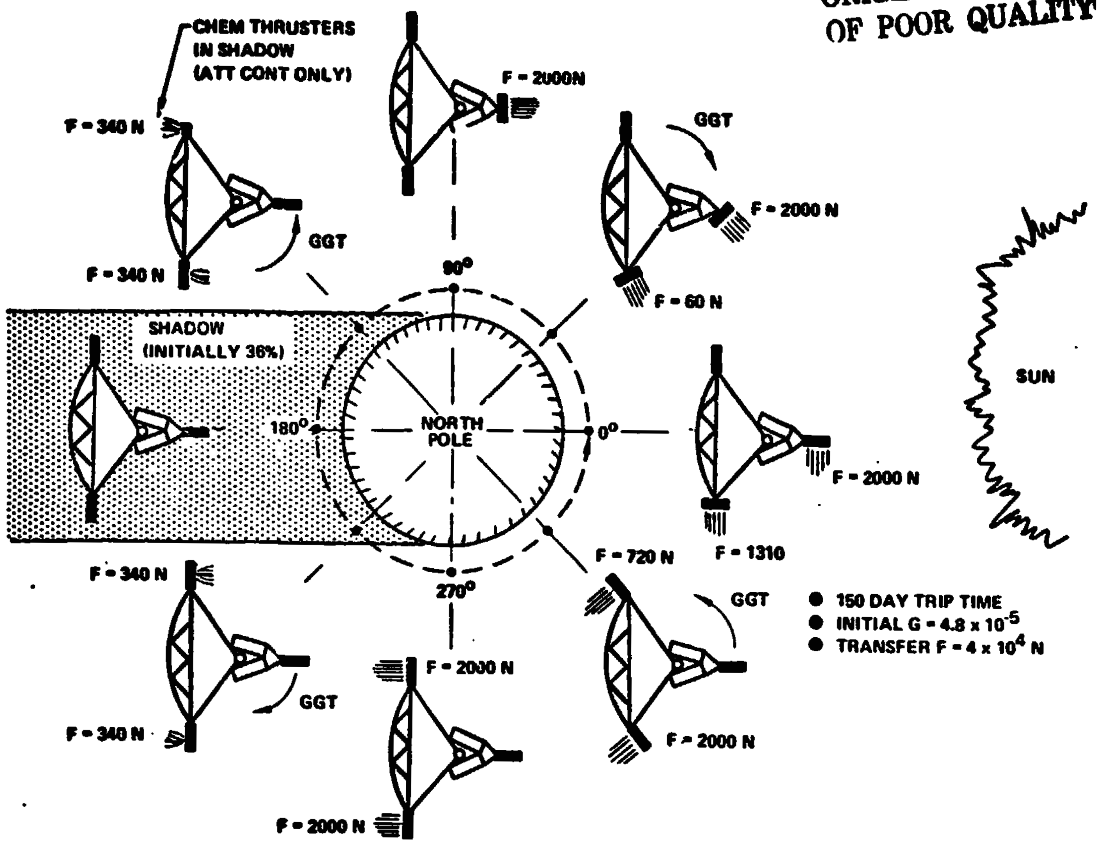
Although the impact of the number of launches was included in the cost comparison, it is important to recognize the difference in quantity between two construction location options shown in Figure 3.5-31. In general, the GEO construction option using chemical orbit transfer vehicles will require about twice as many launches per day as the LEO option using electric propulsion. The significance of this difference, in addition to cost, may include such factors as propellant production rates, environmental impacts in terms of noise pollution, and launch operations scheduling.

Transportation cost to GEO is compared in Figure 3.5-32 for five different satellite options. For the photovoltaic (beginning of life) and the array addition options, the LEO option provides a cost savings of approximately 15%. For photovoltaic satellites assuming annealing capability or for the thermal engine satellites, all less sensitive to radiation, transportation cost savings of 25 to 30% or 2.5 billion dollars per satellite are available through the LEO construction option. This comparison includes estimated cost penalties for the satellite modifications necessary to enable self-powered LEO-GEO transportation.

A transportation cost breakout is presented in Table 3.5-14 for the photovoltaic CR=2 annealable satellite. The most significant cost difference between the options is SPS HLLV utilization; more launches incur a greater cost penalty. It may be that the programmatic costs could be treated as a life cycle cost item. In addition, recovery of electric thrusters and power processing systems may prove cost effective. These factors could combine to reduce the cost of the LEO option by an additional 0.5 to 0.75 billion dollars.

Transportation costs to GEO for the two construction options can also be compared in terms of sensitivity to various program elements. Satellite mass sensitivity is shown in Figure 3.5-33. The sensitivity of the GEO construction option is approximately 75% greater to satellite mass than that of the LEO construction/electric orbit transfer vehicle option, for either the photovoltaic or the thermal engine satellite.

SPS 720



SPS 760 **Figure 3.5-30 Orbit Transfer Thruster Utilization Thermal Engine Satellite**

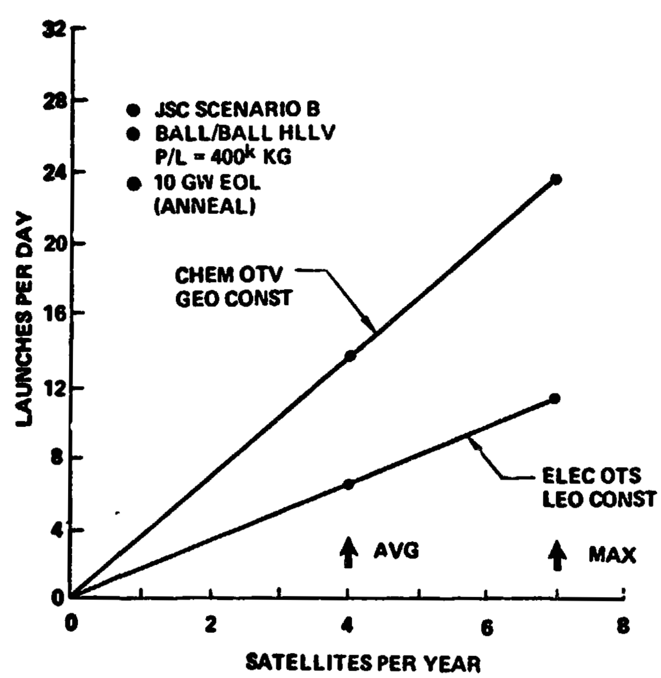


Figure 3.5-31 Number of HLLV Launches

SPS-661

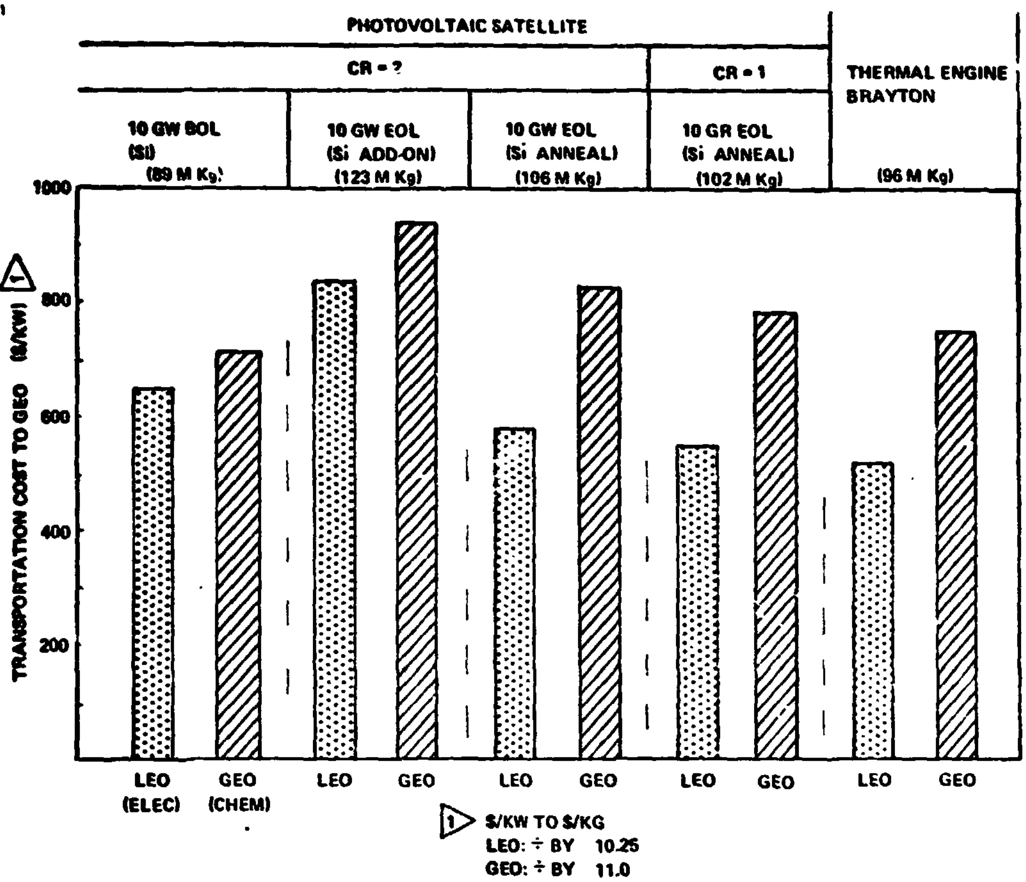


Figure 3.5-32 Transportation Cost Comparison Per Satellite

SPS 719

Table 3.5-14 GEO vs LEO Transportation Cost Photovoltaic Satellite (Annealed)

• SATELLITE COST IN BILLIONS

SYSTEM ELEMENT	GEO CONSTRUCTION	LEO CONSTRUCTION
• SPS HLLV	(6.77)	(3.40)
• SATELLITE	2.03	2.23
• ORBIT TRANSFER/ TANKER	4.43	1.01
• CREW ROTATION/ RESUPPLY SUPPORT	0.31	0.16
• ORBIT TRANSFER (RECUR)	(0.72)	(0.80)
• CREW	0.06	0.04
• SATELLITE	0.63	0.76
• SATELLITE MODIFICATION	-	(0.10)
• PROGRAMMATICS	(0.24)	(0.78)
• TRIP DELAY	-	0.55
• HLLV INTEREST	0.24	0.12
• OTHER INTEREST	-	0.11
• GROWTH SHUTTLE (CREW)	(0.70)	(0.79)
TOTAL	8.43	5.89
COST DIFFERENCE	$2.56B	

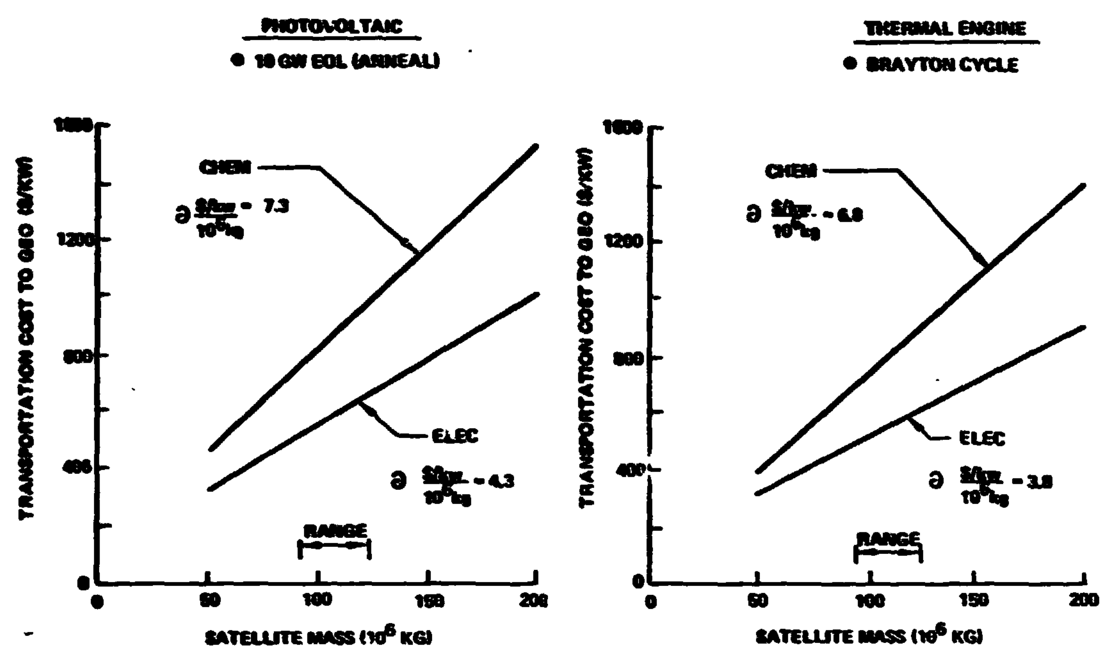


Figure 3.5-33 OTV Cost Sensitivity To Satellite Mass

The reference LEO delivery cost is approximately $17 per kilogram. The total cost sensitivity to LEO delivering cost for a chemical orbit transfer vehicle, in terms of transportation costs to GEO, is approximately 90% greater than that of the electric orbit transfer system, as shown in Figure 3.5-34.

3.5.6 Orbit-To-Orbit Transportation Summary

The transportation of a satellite to GEO using a self-power electric propulsion system after modular construction at LEO, appears to offer cost advantage of over 25%. In addition, it is less sensitive to changes in LEO delivery costs and to satellite mass. Self-power of the thermal engine satellite appears to have a slight advantage over that of a photovoltaic satellite, primarily as a result of simplified integration and flight control operations. The LEO construction option requires roughly half as many HLLV launches. Transportation relative to crew rotation/resupply has not been found to be significant since there is only a 10% cost difference between the two construction location options.

SPS-683

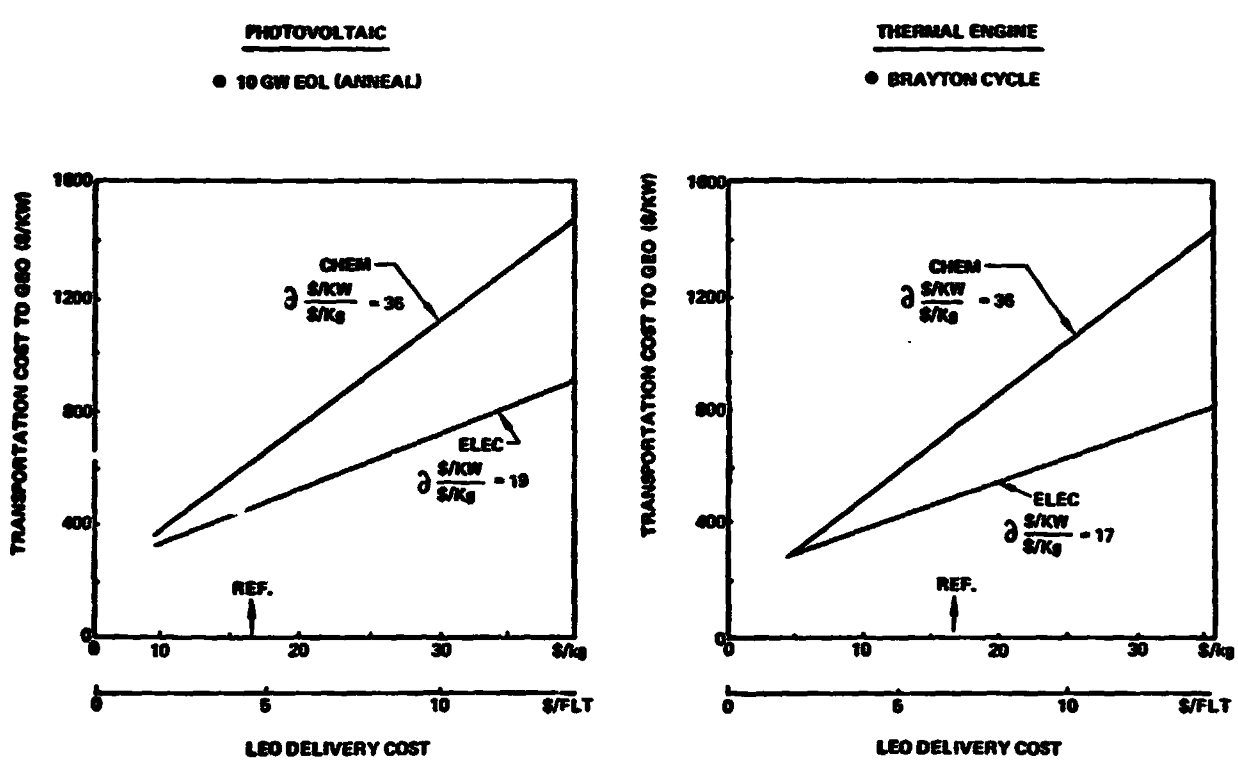


Figure 3.5-34 OTV Cost Sensitivity To LEO Delivery Cost

3.6 COLLISION ANALYSIS

Consideration of space operations with objects as large as an SPS or SPS module raises questions of collision hazards. For historical space systems, even as large as Skylab, the probability of collision with a manmade object is negligible, whereas the probabiliy of collision with meteoritic matter of potentially damaging size is appreciable. Vehicles like Skylab have accordingly been designed with suitable meteoroid protection, generally in the form of a "bumper" (impact armor). The flux of manmade objects in near Earth space, although small, is large enough to present a potential hazard to SPS's, and is orders of magnitude greater than the flux of natural objects of comparable relative momentum or kinetic energy. The flux of manmade objects is considerably greater at LEO than at GEO. Therefore, relative collision hazards enter into the selection of LEO or GEO as a construction location.

3.6.1 Flux Model Analysis

The idea that an SPS satellite can collide with another orbiting object is brouht about by the fact that there were over 3700 man made objects in space as of late 1975.

(Satellite Situation Report - GSFC Volume 15, December 31, 1975.)

Most of these objects have apogee, perigee and inclination characteristics which can intersect an SPS satellite during the LEO construction phase and transfer to GEO. In addition, although the volume sweepout in one orbit of an object is quite small, that volume becomes quite large as the orbit of that object regresses, sweeping out a volume bounded by the objects apogee, perigee and inclination characteristics.

The purpose of this subtask was to develop a flux model and estimate the number of collisions between objects and the SPS satellite as a function of its altitude and inclination. A flux model is by nature a first-order statistical approximation to collision probabilities. More accurate models can be constructed, e.g. Monte Carlo simulations, but in view of uncertainties in source data, are probably not worth the added effort required.

The initial step in this analysis was to establish the flux (number) of objects per KM^2-sec that will be encountered by an SPS satellite. Several key assumptions were used in this initial analysis:

1. The distribution of objects in orbit as listed in the December 1975 Goddard Satellite Situation Report is representative of the future distribution;

2. the Flux (objects) of objects in orbit is isotropic (true for low-medium altitudes); and
   KM^2-sec

3. The size of any object in orbit is so small in comparison to and SPS, that the object is considered a point rather than a volume. The flux contribution that each orbiting object makes was calculated as illustrated in Figure 3.6-1 using the following equation:

$$\phi = \frac{(T_F) \times (VEL)}{VOL}$$

where ϕ = Flux $\dfrac{\text{objects}}{\text{KM}^2\text{-sec}}$

T_F = Fraction of an objects orbit time that is spent within a given "toroid" where each toroid is defined by an altitude and inclination band.

VOL = The actual volume of the toroid (KM^3)

VEL = The average velocity of an object within a given toroid (KM/sec)

The toroids considered in this analysis were bounded by the following altitude and inclination bands: Altitude (KM): 400-440, 440-480, 480-520 (LEO), 520-550, 550-600, 600-700, 700-800, 800-1000, 1000-1500, 1500-2000, 2000-3000, 3000-5000, 5000-10000, 10000-20000, 20000-35750, 35750-35890 (GEO); and inclination boundaries of (deg): 0-5, 5-10, 10-15, 15-20, 20-25, 25-30, 30-35.

Summation of the flux made by all objects within a given toroid results in the total flux an SPS satellite will encounter within a given toroid.

A computer program was used to perform the flux calculations for each of the specified toroids. The data were then combined within a typical SPS satellite LEO to GEO transfer trajectory (altitude vs inclination). This results in the plot shown in Figure 3.6-2, which estimates the flux encountered by the satellite. The highest flux is indicated at the 500 to 1000 KM region as would be expected due to the large number of satellites having perigees within this range. The relatively high flux at the GEO location is somewhat misleading, since the isotopic flux assumption becomes invalid, (most of the objects at or passing through this location are traveling at the same velocity and in the same direction as the SPS).

The flux contribution that each orbiting object makes was calculated using the following equation:

$$\phi = \left(\frac{T_F}{VOL}\right) \times (VEL)$$

Where:

ϕ = Flux $\left(\frac{\text{objects}}{\text{km}^2\text{-sec}}\right)$

T_F = fraction of an object's orbit time that is spent within a given "toroid," where each toroid is defined by an altitude and inclination band

VOL = The actual volume of the toroid (km^3)

VEL = the average velocity of an object within a given toroid (km/sec)

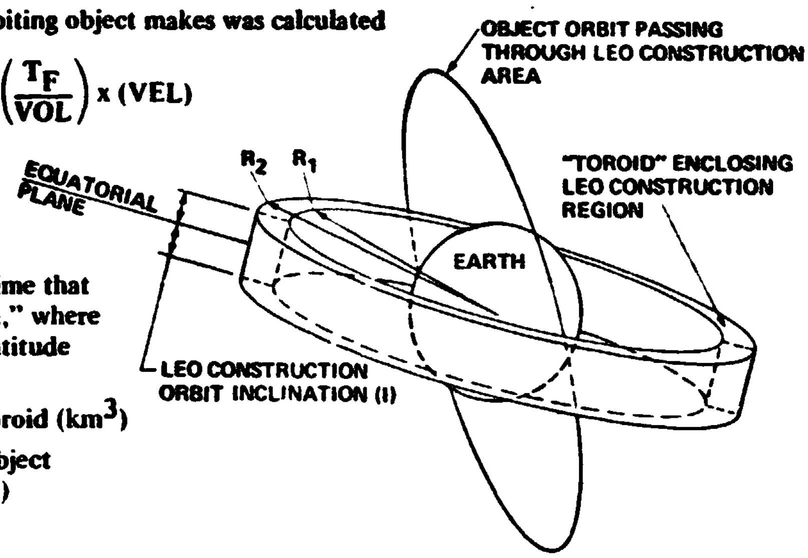


Figure 3.6-1 Orbiting Object Flux

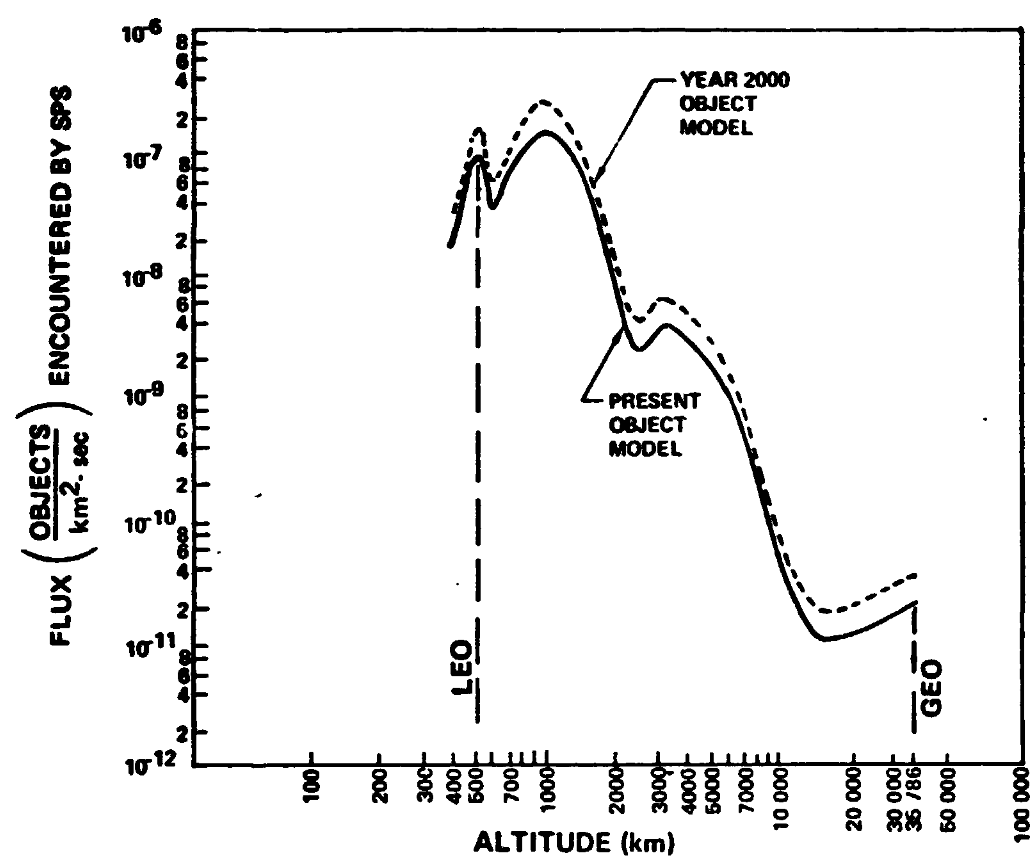


Figure 3.6-2 Object Flux Level

3.6.2 Flux Model Analysis Results

The collision model data reported at midterm were updated to reflect a "growth" object model (assumes the number of objects presently in orbit will increase due to continuing space launches) and modular construction with sixteen modules. Assumptions and expected numbers of collisions are shown in Figure 3.6-3. The 3x3 meter object assumption relates to calculations of collision cross-section for small SPS elements such as structure–the object model included all objects now listed in the Goddard Space Flight Center satellite situation report. In low Earth orbit, objects down to about 10 sq cm can be tracked.

Figure 3.6-4 shows a collision prediction for the thermal engine option similar to the previous figure for the photovoltaic option.

3.6.3 Collision Avoidance Considerations

The flux model analysis presented above assumes no measures are taken to avoid collisions. During the orbit transfer outboard propulsion could be used for evasive action, either in changing the path of the transferring module or in changing its attitude to minimize the collision cross-section. The available propulsive acceleration is expected to be 5×10^{-4} m/sec^2 or greater. This is sufficient to move an SPS module a distance equivalent to its own size in about 1 hour (linear acceleration assumption). In low Earth orbit, during the early part of the transfer up to 2 revs might be required. Ephemerides of objects in LEO are known to roughly 50 meters, so adequate warning should be available for tracked objects. Collisions during the construction phase are somewhat more problem-iatical since the construction facility will presumably be far less maneuverable.

3.6.4 Junk Cleanup Concept

Most of the problem objects are not operable satellites, they are "junk". Conceptual studies of a junk cleanup pursuit vehicle were included in the SEPS study program. This vehicle would propulsively match orbit parameters with target junk objects (one by one), perform a noncooperative rendezvous, acquire the object with some sort of "grabber" and either deorbit it or return it to a controlled disposal area.

SPS 716

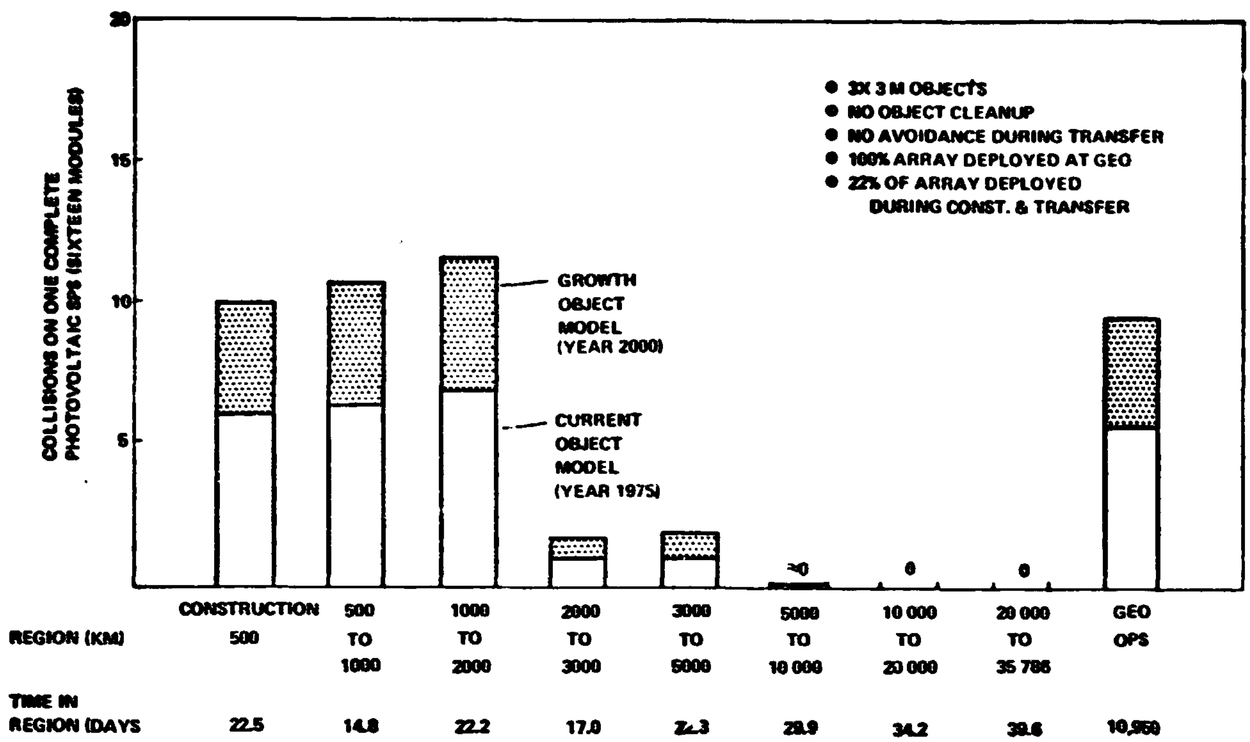
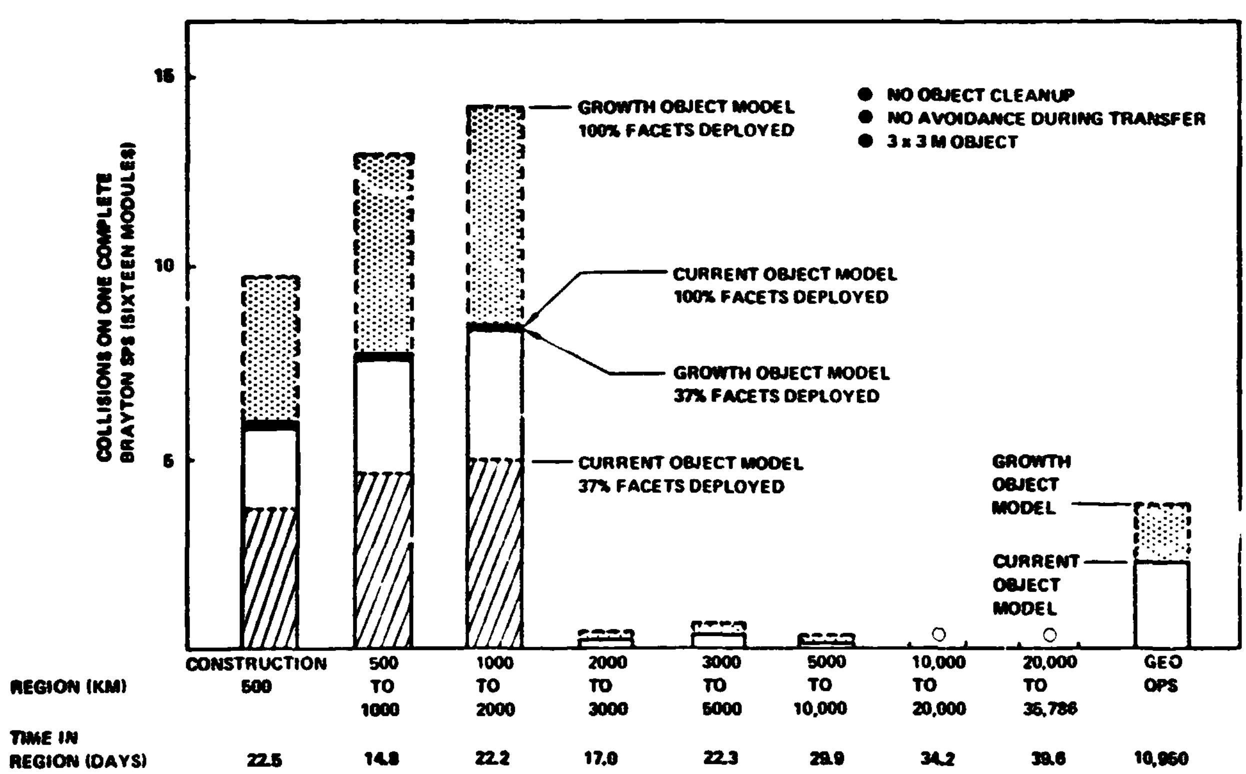


Figure 3.6-3 Photovoltaic Satellite

SPS 711



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During the present activity, an interceptor vehicle was suggested as an alternative. The interceptor would not rendezvous with the target objects, but merely fly into their path, a maneuver requiring far less delta v and propellant. The interceptor would employ a "catcher's mitt" to absorb the target objects by an inelastic collision. Various materials such as old matresses, styrofoam, and water-filled plastic microballoons or tubing mats, have been suggested as catcher's mitt could be separated from the interceptor vehicle such that the collision would deorbit the (in this case expandable) mitt as well as the object. Ephemeris uncertainties would require the interceptor to have an active terminal rendezvous capability.

3.7 COST ANALYSES

The cost analyses began with a review of a prior SPS study cost analysis and of Boeing design-to-cost studies. This background was employed to develop updated SPS cost analyses based on traditional aerospace and mature industry correlations. Particular emphasis was given to solar cell cost projections because of the high leverage solar cell costs exhibit with respect to overall SPS costs and economics.

3.7.1 SPS Data Base Cost Review

A review was conducted on SPS element costs developed for the MSFC space-based power contract (contract NAS8-31628). This was done to understand and asses the current data base as a starting point for cost analyses for the SPS system definition study. Present were: Richard Bock. J. Gauger, D. Gregory. S. Otrosa and G. Woodcock.

Photovoltaic and thermal engine methodologies and data were discussed. Microwave power transmission system costs were not discussed since these were common factors in Part I of the SPS study. Most of the cost estimating employed aerospace estimating relationships and learning curves. (Learning curves do not appear to be appropriate for SPS costing, nor do aerospace estimating relationships in most instances.) Costs described below are FOB launch site.

1. **Photovoltaic**–The silicon photovoltaic systems costs were largely driven by solar blanket costs. Solar blanket costs were based on a theoretical first unit (TFU) of $270,000 per square meter for the first square meter. This figure was derived from solar array estimates developed for Space Telescope; it correlates well with SEPS cost estimates for solar arrays. This TFU was run down an 80% learning curve for the entire 60-satellite program involving a total solar cell buy of roughly 2900 km^2. The average unit/TFU ratio os 0.00133 for an average cost of \$360/M^2, equivalent to \$1650/KWe solar cells at 16% efficiency. (The cells were operated at a concentration ration of 4.5; actual operating efficiency at that concentration ratio was ≈ 10%). Gallium arsenide cells were arbitrarily assigned a TFU per square meter twice that for silicon.

Assessment–This cost is considerably higher than the ERDA goal of $500/KWe, and is believed to be pessimistic.

2. **Structure**–The same methodology was used for photovoltaic and thermal engine structures. The following data are for the thermal engine options.

   Unit - structure for an entire module, mass 2960 tons (6.525 x 10^6 lb)

   Number of Units - 240 (4 modules x 60 satellites)

   Learning Curve - 90%

   Total Cost - $52 billion (240 units)

   Average - $70/Kg ($32/lb)

   Assessment–High for the type of structure (aluminum) assumed. Current efforts are directed to graphite composite structures. Vendor projections indicate raw material (prepreg tape) costs in the range of $6/lb. Parts fabrication costs should not exceed materials cost, leading to an ROM of $12/lb. In addition, the graphite structure will be lower in mass.

3. **Heliostats**–Thermal engine heliostats were priced as follows:

   Unit - One Heliostat

   Mass - 65 Kg (144 lb)

   Learning Curve - Not stated

   Total Cost - $90 billion (4.25 x 10^6 units)

   Average - $21,000 each

   Assessment–Heiostat costs will probably be dominated by plastic film cost at $300 - $350/lb for aluminized Kapton. Materials cost per heliostat is ≈ $12K. Estimate is probably reasonable.

4. **High Temperature Heat Exchanger**- This item was to be fabricated from columbium alloy:

   Mass - 250,000 tons (60 satellites)

   Total Cost - $21 billion

   Average - $38/lb

   Assessment–Materials cost is estimated as $120/lb as tubing. Therefore this cost estimate is low by a factor of 4 to 5. A change in material may be in order.

5. **Cavity Shells and Insulation**–The total cost was $6.8 billion. This does not appear to be a cost swinger.

   Assessment–The reference configuration employed a tantalum multifoil insulation. Materials cost need examination.

6. **Turbomachines and Recuperator Coolers**

   Unit - 300 MWe machine with recuperator-cooler heat exchanger set

   Total Cost - $72 Billion

   Average - $73/KWe of actual on-board output

   Assessment–Industrial experience and estimates for ground-based hardware indicate costs should be in the $100/KWe range.

7. **Radiators**

   Unit - 20 x 20 meter panels and header reactions — roughly 20,000 per SPS

   Total Cost - $107 Billion

   Average - $36/lb.

   Assessment–Radiator panels and pipes will be fabricated from steel alloys and aluminum. Assuming automated fabrication, the radiator cost appears to be high by a factor of 5 to 10.

8. Space construction and transportation costs were separately estimated.

9. **Other**—Additional items contributing to u cost (program totals for 60 satellites).

   Tooling - $64 Billion

   Initial Spares - 10%

   Sustaining Engineering and SE&I - $25 billion

   GSE - $19 Billion

   Program Management - $44 Billion

   Assessment—Tooling, engineering, and management costs appear high for the presumed commercial environment.

10. **Totals**—The total costs attributable to the thermal engine SPS, excluding antennas, space construction, and space transportation, add to slightly less than $600/KWe of useful ground output. Adding the costs of the space-based antenna, increases this to about $700/KWe, very close to a figure derived by the high-level Dix-Riddell correlation. (See "Satellite Power Systems for Large-Scale Power Generation" by G. Woodcock, presented at the 27th IAF, October 12, 1976.)

The plusses and minuses discussed under the assessment headings may roughly cancel. As noted above, photovoltaic estimates appeared pessimistic; all costs were re-estimated in the SPS systems using a generally different methodology.

3.7.2 Design-to-Cost Review

Typical Program Cost History—For a typical program, the cost history is as shown in Figure 3.7.1. Point A is the initial program cost estimate carried out by mid-management and engineering during the conceptual phase of a program.

This is submitted for corporate approval and slides down to Point B, the rationale usually being "we can't win at Point A." The B estimate is given to the customer who says "that's too high-we can't get congressional approval," so the RFP goes out and a cost auction brings the cost to Point C. Now the changes start-the contract is negotiated up to "D" as customer and contractor begin to realize the contractor bid too low.

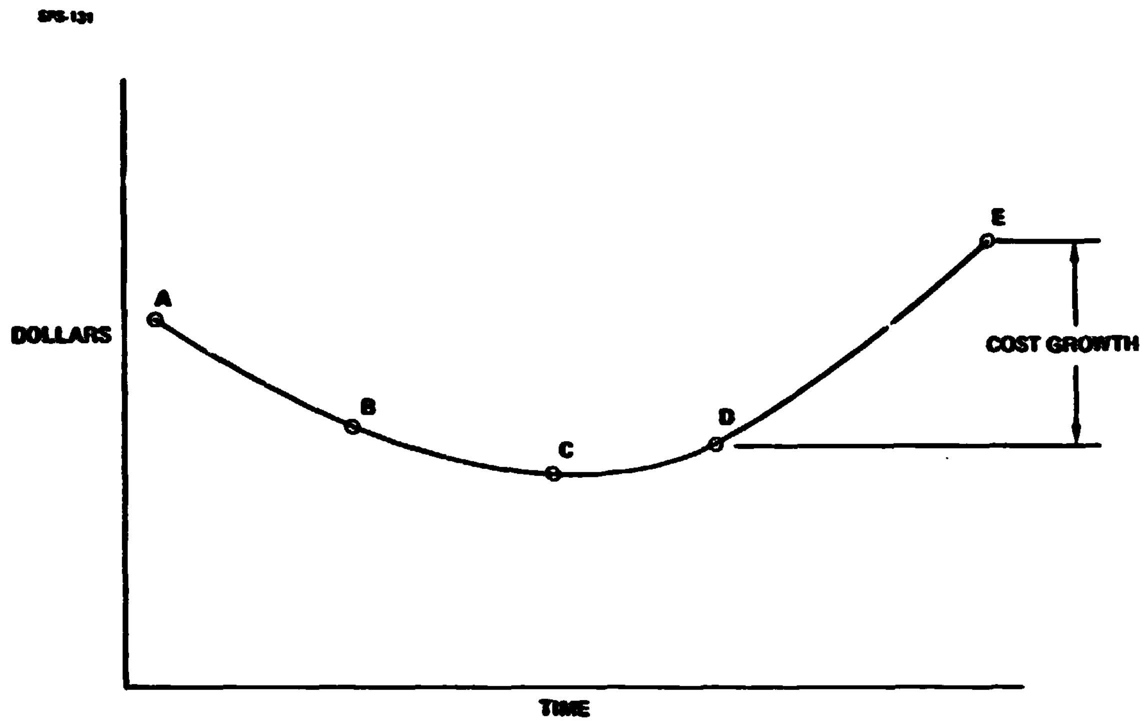

Figure 3.7-1 Typical Program Cost History

By the time the program ends, the costs have risen to E-and the contractor-customer team have done it again.

The SPS program should not allow this to happen and with proper early consideration and control, costs can probably be brought to point D. This will require the development of an accurate cost prediction and control capability.

When is cost injected into a program? Analyses of both commercial and government Aerospace programs indicate that by the time concept definition has been completed (which is DOD programs is the DSARC I decision point), program decisions have been made which will result in approximately 70% of the life cycle costs. By the end of the program validation phase (DSARC II), program decisions have obligated approximately 85% of the life cycle cost (LCC). Essentially no cost leverage is available at the end of the development phase. This is illustrated in Figure 3.7-2.

Next let's take a look at "learning" curves, better titled, "improvement curves."

We have found that if we extrapolate to the 1000th unit, all aircraft programs we studied would have passes through a gate between 0.75 and 1.25 manhours/lb. Three programs have been sketched in as examples in Figure 3.7-3.

Program A is driving from a 60% curve and will pass through the top of inadequate management attention, improper funding, poor production and tooling planning, probably poor skill mixes.

Program A is in deep trouble, a cost delta has been added because of inadequate management attention, improper funding, poor production and tooling planning, probably poor skill mixes.

Program B has been developed as a "good" aerospace program but has a cost burden added by design complication.

Program C is a good aerospace program characterized by simple design and an adequate tool and production plan. The cost for these programs through 1000 units is in ratio 4:1.7:1.

Next we looked at the cost composition for aircraft, and it looks like the data sketched in Figure 3.7-4 are assembly; about 25% are fabrication. By the 1000th unit, they are about 50-50. So assembly costs are the prime cost swinger of airplane production.

A good assembly improvement curve characteristically follows λ=0.80, while fabrication is at λ=0.89, or the net production curve at λ0.83 to 0.85.

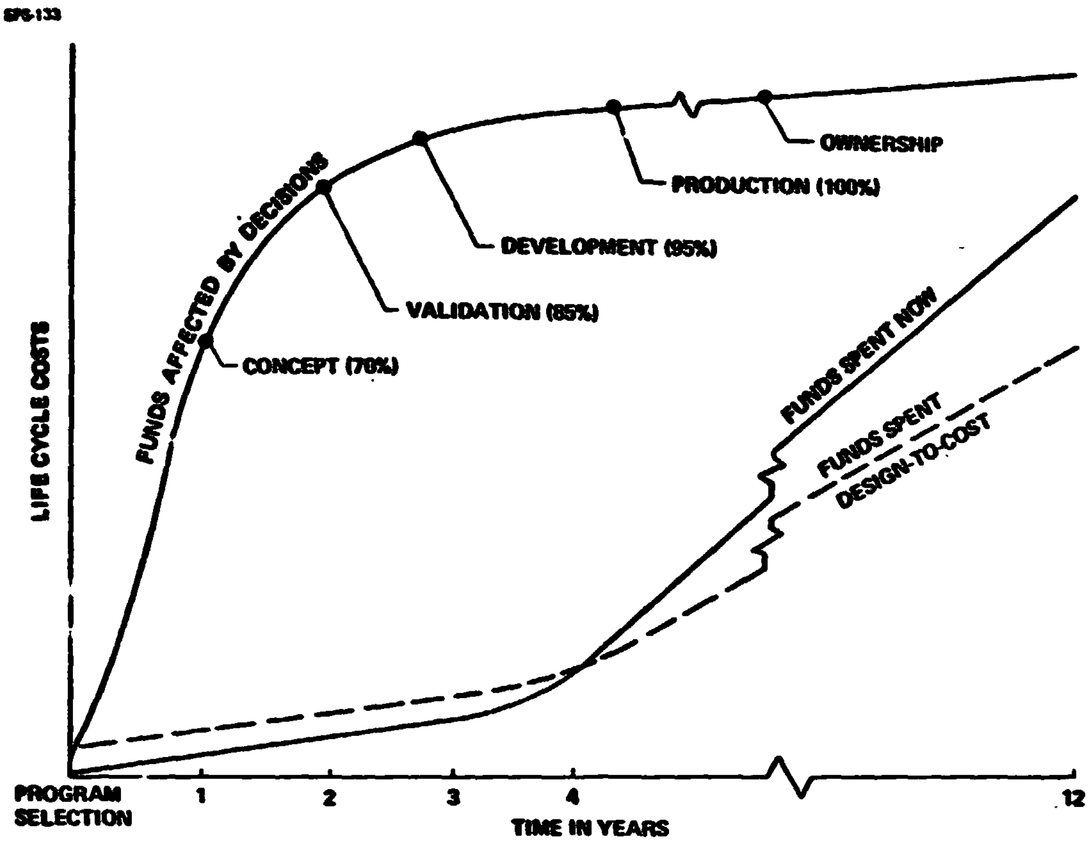

Figure 3.7-2 When?

SPS-136

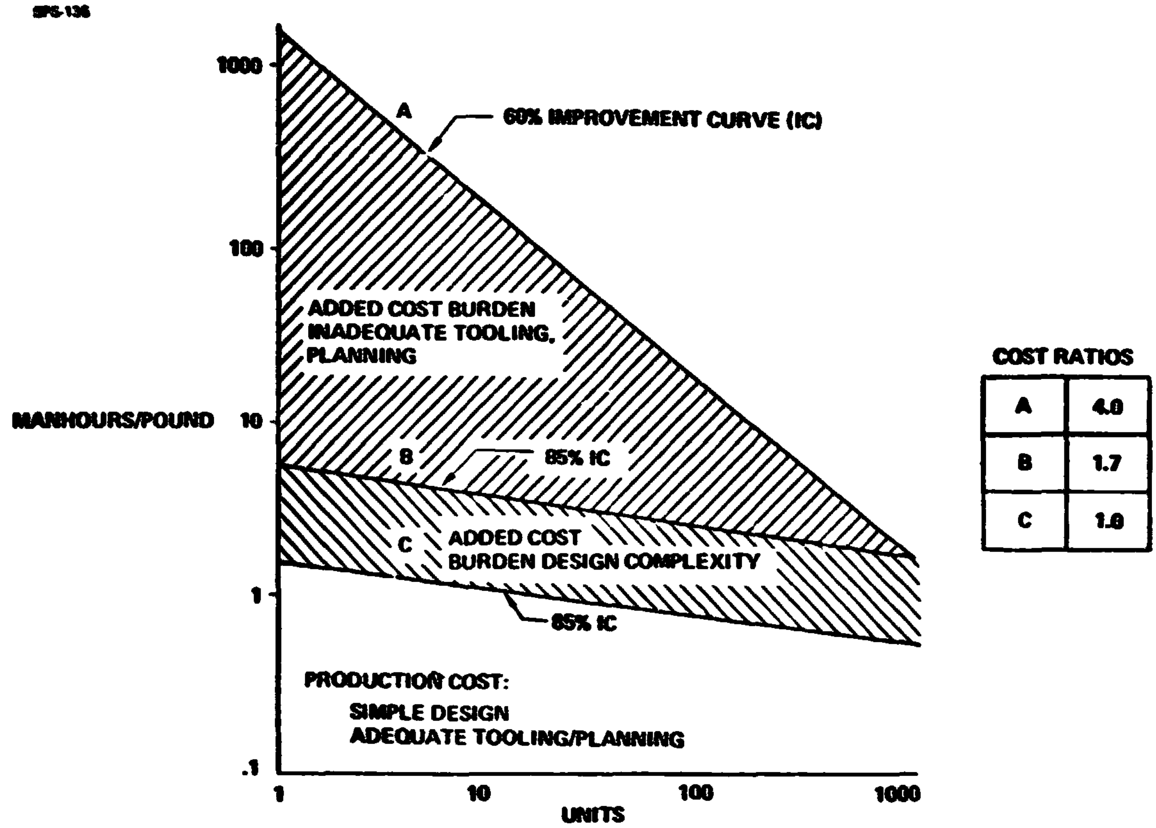

Figure 3.7-3 Production Improvement Curve Comparisons

SPS-137

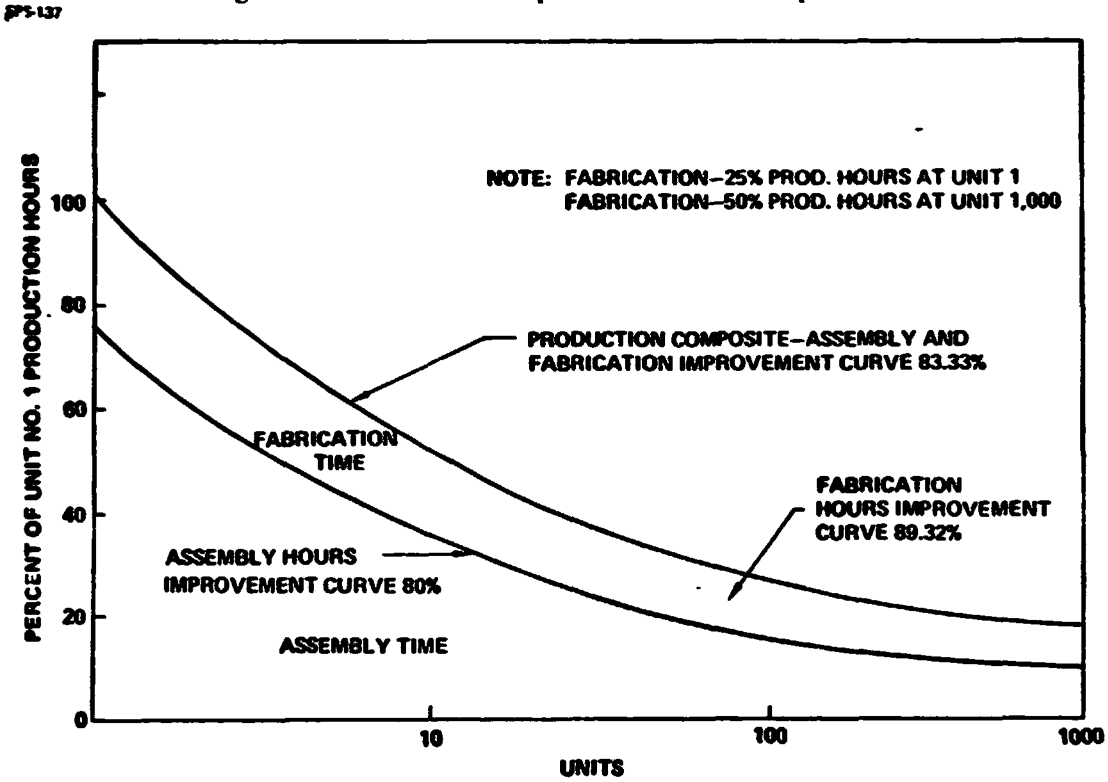

Figure 3.7-4 Cost Composition

A steep curve is an indication of poor planning and control of a production program.

Since 60% of airframe costs through the first 1000 units are assembly costs, we then analyzed the assembly charges in detail. Figure 3.7-5 shows the relative values of productive time.

1. Productive time was 6% of first unit cost and 50% of 1000th unit.

   Job familiarization, "mechanics learning," was ~40% of first unit and drove down a λ=0.5 curve. The mechanic learns quickly. By unit 10, this element has virtually disappeared if no changes in paper, tooling, skill, etc. have been made.

3. Overall, the mechanic loses about 10% of total assembly time for personal reasons.

4. Stacked upon that, 50% of costs are attributable to nonproductive factors over the same 1000 units. That is, time spent by the mechanic in overcoming the deficiencies of the management plan.

The non-productive elements can be directly related to unrealistic schedules, part shortages, etc. If we look at the nonproductive cost data, it is apparent that a program manager must be provided with "should cost" rationale. If a program manager can identify the management changes to implement and determine how they will impact the program nonproductive elements, an improvement curve approaching λ=1.0 can be predicted.

Program Cost/Management Matrix–The previous figure establishes some requirements of a "should-cost" philosophy, but more is needed.

To assist in the definition and control of a program as complex as SPS, the adoption of a 3-dimensional pure hardware/software WBS even at this early stage is essential. An example of the work/cost management matrix is illustrated in Figure 3.7-6, along the

- Z-asis are deliverable hardware/software items
- X-axis time phased program tasks
- Y-axis are functional cost elements

Standard Hours–The "standard hour" methodology can be described thus: For any specific industry, the device to be produced, under the management policies of that industry has a minimum cost which is purely a function of the design/configuration of the item.

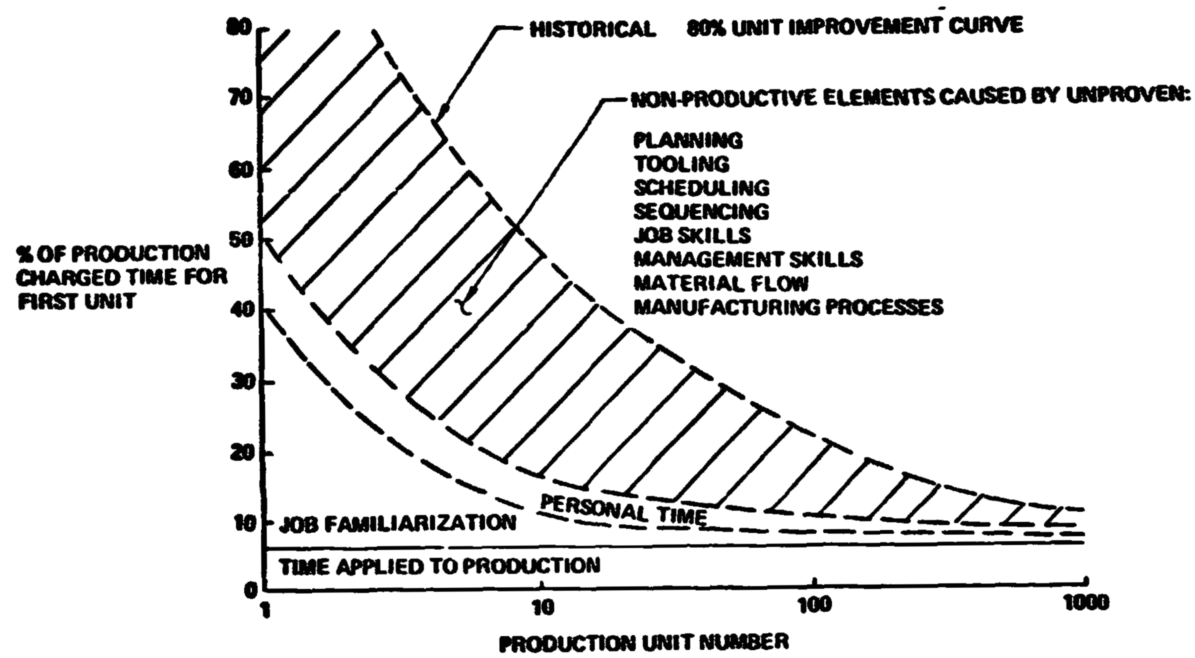

Figure 3.7-5 Assembly Analysis

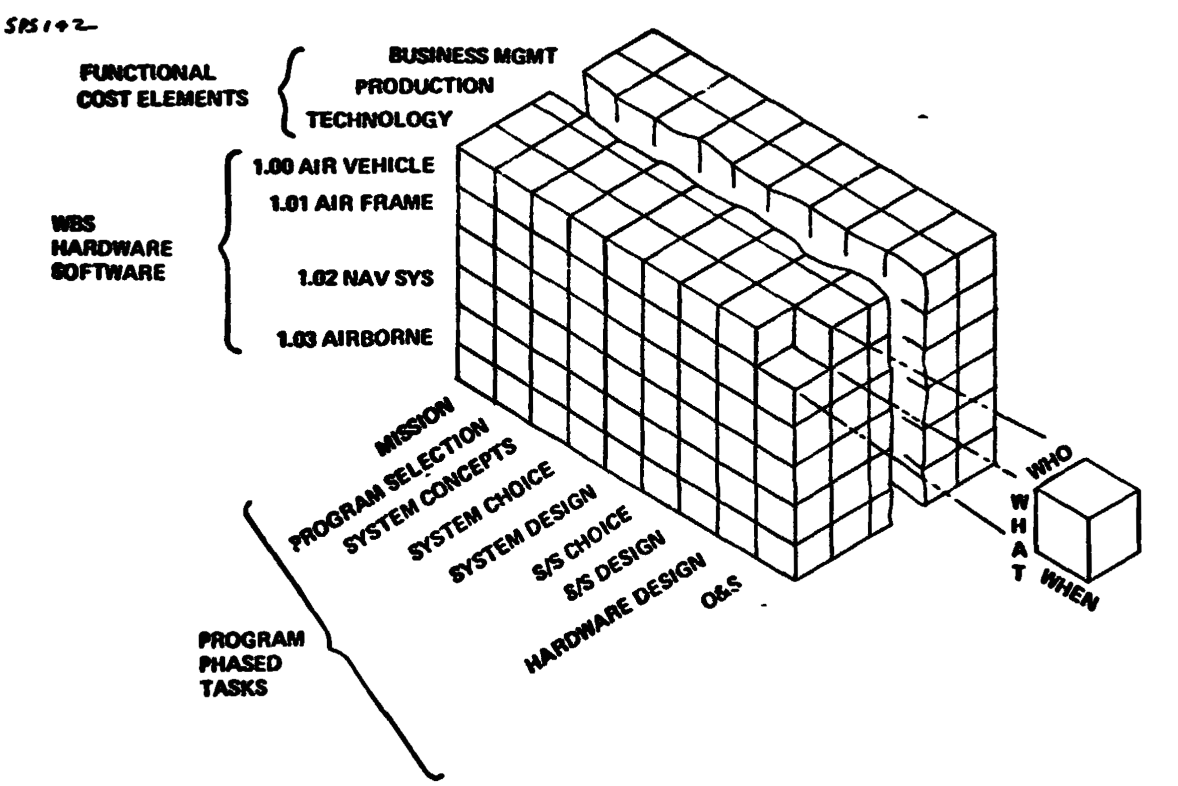

Figure 3.7-6 Program Cost/Management Matrix

The cost generated in producing a single unit are the "standard hours" for that unit. The "standard hours" cost level should be reached by the time the peak production rate has been reached. Figure 3.7-7 illustrates this approach.

Unit costs are then calculated by combining the standard hours for a unit with calculable program variables and the line position along a management imposed improvement curve.

3.7.3 Mature Industry Concept

The "mature industry" analogue requires a comparable analysis which relates the "standard hour" cost of a unit to the production rate in units per year.

Since the "standard hour" value is a function of the optimum resource allocation for a specific industry at a specific yearly production rate, one essentially has a "design" for the industry which is rate sensitive.

This industry design is a function of the volume and the optimum resource allocation and is not the same at all rates. Tooling, facility layouts and size, material flow, process controls, and the ratio of energy expended to manhours expended all vary, as the rate is changed, to produce the optimum standard hour cost for a unit.

Figure 3.7-8 illustrates the functional dependence of the standard hour value of a unit as the production rate varies.

The production rate slope is roughly 0.70. The industry rates, aerospace, computer, electrical appliances all seem to be consistent.

The apparent limit of cost reduction seems to be reached at production rates approaching 10^7 units per year for an industry, where the per unit cost should be about 1.5-2.5 times the basic material cost. The automobile industry reaches this level.

A number of items for SPS may reach this limit, e.g., the individual radiator panels for the thermal engine system.

Mature Industry Source Data: Metal, Ores If we examine the throughput in pounds per year of industries involved in handling large amounts of ores or metals, we can correlate the volume in number/year against the $'s/year for the industry. The result of such a study is in Figure 3.7-9.

For ore the correlation was 0.80.
For metals the correlation was 0.89.

SPS-147

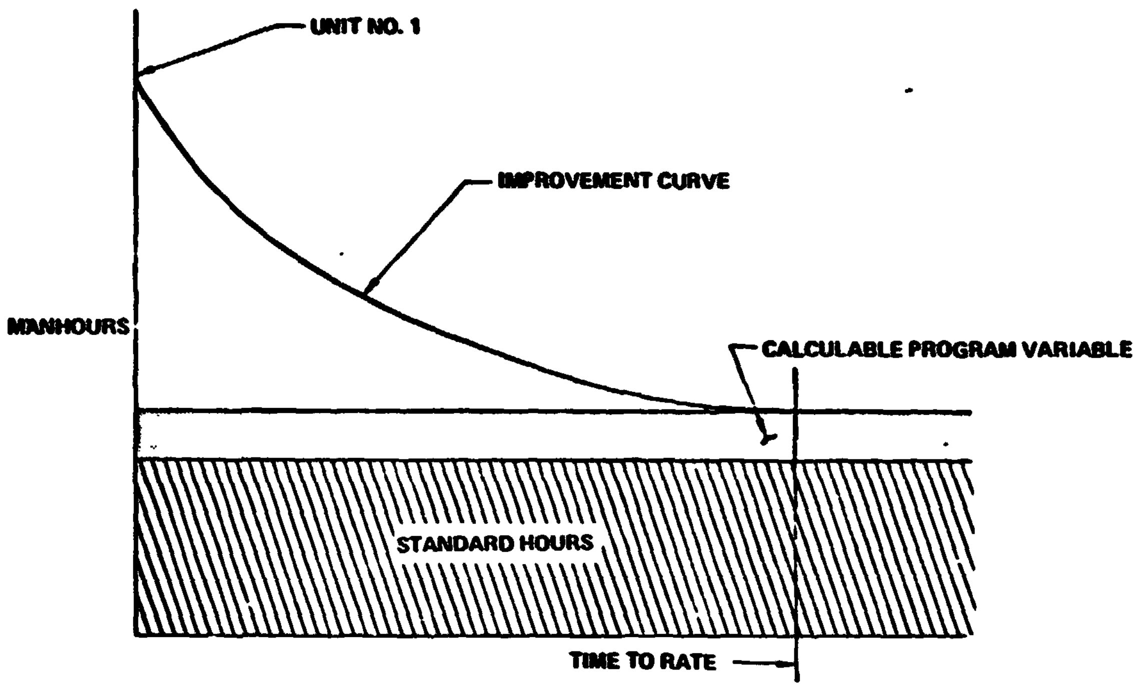

Figure 3.7-7 Program Cost Baseline

SPS-148

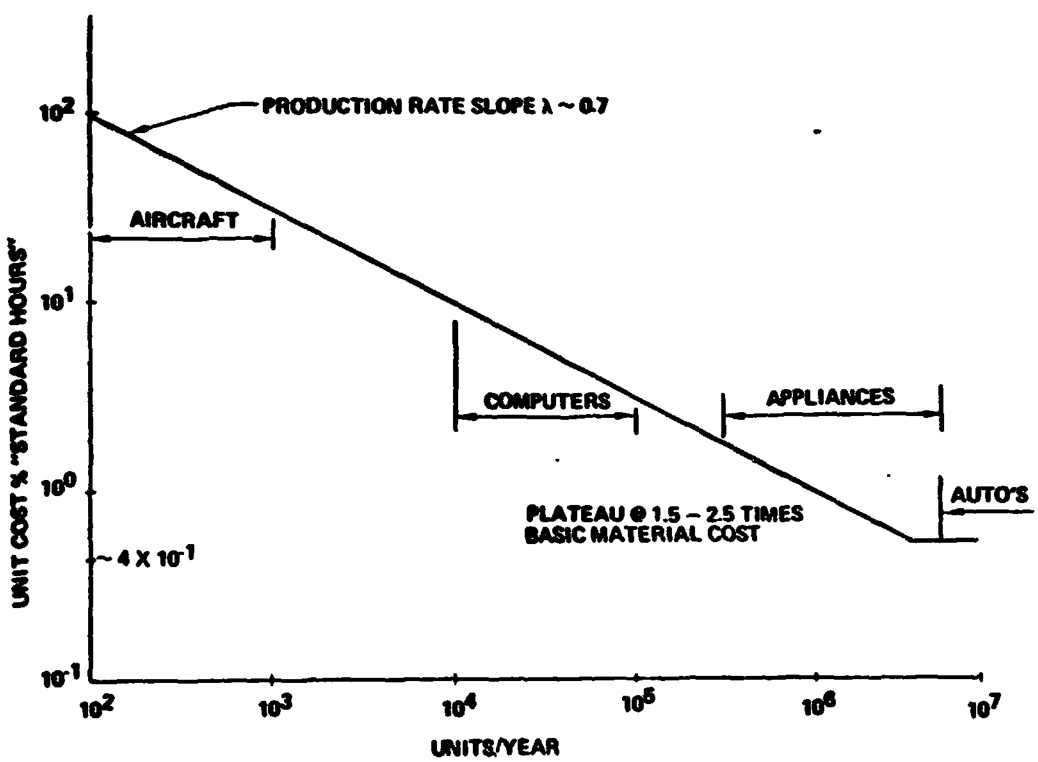

Figure 3.7-8 Mature Industry: Production Rate Curve

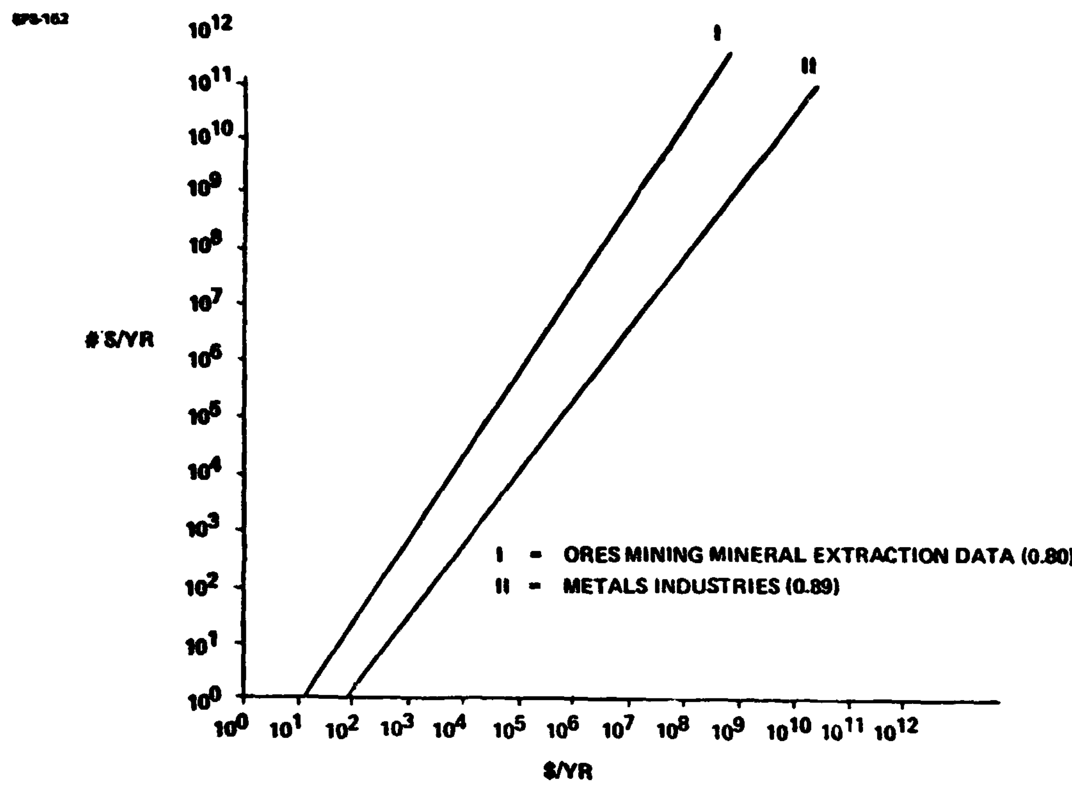

Figure 3.7-9 Mature Industry; Metal, Ores

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The conclusions we can draw from this are significant.

If an industry is to survive the birth pangs and reaches maturity, it must develop technology and manufacturing capabilities which are designed to handle the volume of material needed for a specific purpose.

In the case of solar cells, we are forced to extrapolate to production volumes of 100 km^2/year of cells or more.

We cannot and must not expect that an extrapolation along a pure "learning curve" will be valid.

Example: DUPONT KAPTON QUOTES

Mature industry cost-Kapton film.

1. Dupont, the sole producer of the polyimide film, quotes the cost of Kapton as follows:

Film Thickness	Cost $/#
0.3 mils	$375
0.5	115
1.0	29.75
to	
5.0	29.75

2. The cost is certainly a consequence of production rates. The 0.3 mil Kapton is a special order requiring significant lead time.

3. If we assume a reasonable consistent pricing policy, then the cost elements contributing to the quoted prices should include:

   a. Profit (estimate @ 15%)

   b. A setup time per pound

   c. A run time per pound

   d. Basic material costs

Because of the constant prices at thicknesses from 1 to 5 mil, we assume that, the 1 mil Kapton is being produced at "mature" industry rates and further assume a rate of $>10^6$ #'s/year.

Assuming the 1 mil production rate is the bottom end of the production plateau region at 10^6#'s/year and a production volume parameter of 0.7 then the volume produced/year for the 0.5 mil @ \$115/# is ~10^4#

Kapton case conclusion:

The basic material cost is ~\$10/# and the basic run cost is ~\$20/#.
The prices quoted for 0.3 and 0.5 mil Kapton are clearly a consequence of run size.
The cost of 0.3 mil of Kapton film in quantities of 10^6 or more #'s/year will not exceed ~\$25/#.

Example #2: 120 CM ARGON ION THRUSTER

Shown in Figure 3.7-10 is a sketch of the 120 cm diameter ion thruster assumed in transportation analysis of SPS self-power to GEO. An estimate was made using the hierarchical approach.

1. Prototype to:

2. Standard hour at Aerospace production rates then to:

3. Mature industry costs at 10-20 x 10^4 units per year.

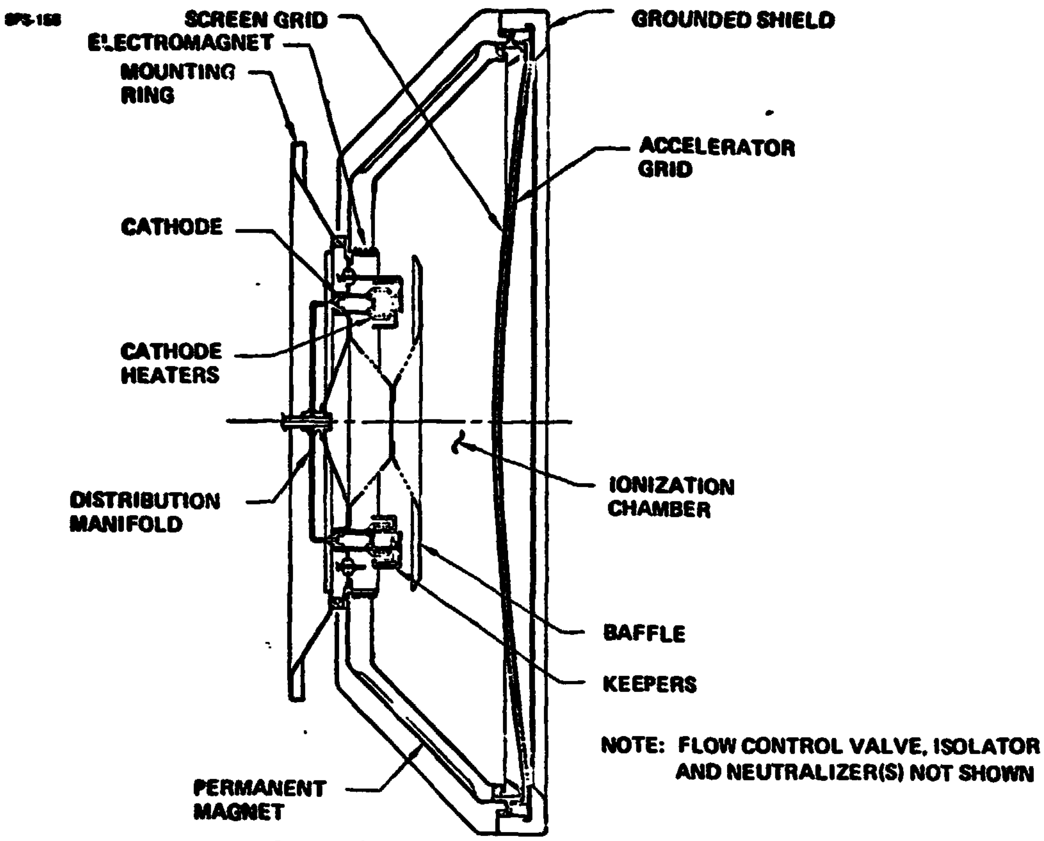

Figure 3.7-10 120 cm Argon Ion Thruster

The analysis proceeds as follows:

1.	30 cm Prototype cost	$70,000/unit
2.	Prototype factor (1.5) first unit	48,000/unit
3.	(a) First unit assembly cost (75%)	36,000
	(b) First unit fabriaction cost (25%)	12,000
4.	(a) 1024th Unit assembly on 80% LC	$ 3,850
	(b) 1024th Unit fabrication = assembly 1024th	$ 3.850
5.	Total cost 1024th unit	$ 7,700
6.	Change + O/S removal (0.40)	$ 3,080
7.	30 cm thruster mass	18kg
8.	120 cm thrust CER	$171/kg
9.	120 cm thrust mass	50kg
10.	120 cm thrust cost (Aerospace production rates)	$ 8,550
11.	Production rate cost factor estimate	÷ 10
12.	Unit cost "mature industry" production rate 10-20,000/year	$ 855

3.7.4 Silicon Solar Cell Costs

Silicon solar cell cost estimates were analyzed in three ways: (1) Mature industry projection; (2) Review of manufacturer's projections; (3) Energy cost check and production methods projection:

Mature Industry Projection—Figure 3.7-9 shows the functional relationship between $/year for the industry and pounds of material processed per year in the industry. Line 1 is for the ores and mineral extraction industry. Line 2 is for the primary metals industry (excluding "precious" metals).

Note that the data for some 20 or so industries, from which line 2 was derived, yielded a correlation coefficient of 0.89. A good fit. The correlation coefficient is a measure of the quality of the fit of data to tne calculated equation.

In using this to predict silicon cell costs we calculated the amount of bulk quartzite needed to produce the required amount of mono crystal silicon required for a specific satellite–judge the ore cost, "metal" cost, and volume, and a value added factor to yield a solar cell blanket.

The results are as follows:

#'s/YEAR BULK = 1.33 X 10^8kg
(2.93 X 10^8#)

#'s/YEAR FINISHED = 1.7 X 10^7kg
(3.7 X 10^7#)

$'s/YEAR BULK ≈ $$10^7$

$'s/YEAR FINISHED $$10^8$

VALUE ADDED ≈ factor of 20

SOLAR CELL COST ≈ -2 X 10^9
(10-20 CENTS/WATT)

For approximately 10 GW delivered power we will need to process 133 million kilograms of bulk quartzite at a cost of 10^7 dollars to obtain 17 million kilograms of semiconductor grade silicon at a cost of $$10^8$ which when finished into the solar cell blanket will cost about $2 X 10^9 for a satellite.

We originally said that the cells would cost 10-20 cents/watts for the power delivered to the on board power distribution system.

Manufacturer Projections–RCA projects 20 cents/watt for a scenario as follows:

- Three people + machines = 4000 cells per hour
- 80% Yield for ION implantation
- Project 20 cents per watt for material and expense
- Semi-annual review meeting silicon technology programs ERDA, Jan., 1977

Motorola's prediction based on the program variables below is 13 cents/watts:

- No new processes, 15% cell efficiency
- Dedicated factory produces 500 MW per year

- Equipment depreciated in 7 years, buildings in 40 years
- Work 22.5 hours per day, 240 days per year, use 2.5 cents/KWH power
- Advanced ION implanter
- Used learning curve from semiconductor experience
- Predict 13 cents per watt
- Semi-Annual Review Meeting Silicon Technology Programs ERDA, Jan., 1977

Texas instruments projects 26 cents/watt reducing to approximately 14 cents/watt with learning:

- Demonstrate by 1982 all processes for 1985 manufacturing plant
- 13.5% cell, textured, 0.2 to 0.3 μM junction depth
- Additive processes instead of etching and grinding
- Only single-crystal silicon
- Avoid costly Ag, Pt, Au for Backside Metalization
- Cost: $0.2559 per watt, 75.3% Yield
  Expected Experience X 0.5 without new inventions
  0.1359 per watt
- Semi-Annual Reviewing Meeting
  Silicon Technology Programs
  ERDA, Jan., 1977

GE Estimate High Voltage Production Silicon Cell Costs–And G.E., is now projecting 17 to 20 cents/watt for LEO manufacture:

- LEO MANUFACTURE
- CELL EFFICIENCY
- MASS - 167 GM/M^2 (100 GM/M^2 IS 25 MICRON COVER GLASS)
- ANNEALING
- COST
    - MATERIALS 6.8 CENTS/W
    - FACILITY (INCLUDES POWER) 9-10 CENTS/W
    - TRANSPORT ($10/lb) 2 CENTS/W

  TOTAL 17-20 CENTS/W

Energy Cost and Production Methods–Solar cells are very energy intensive. Presented in Figure 3.7-11 are energy costs in kilowatt hours per kilogram of cells. The energy payback for solor cells as a function of this energy cost is also shown on two scales. These scales show SPS and ground applications. Pricing the energy at 40 mills per kilowatt hour, the actual cost of the energy is shown on theoutside scale.

The main reason today's cells are so intensive is that yields are very poor. Most of the silicon, in which a great deal of energy is invested, ends up as waste (saw filings and trimming). Continuous processes can probably reach a yield range of 60% to 80%, making the payback very attractive. Energy cost is a basic factor in the cost of solar cells, like materials cost in building hardware. If the energy cost is below 10 cents/watt one might be reasonably confident that cells in the 20 cents/watt range, made by a continuous production process, would be possible.

Figure 3.7-12 compares today's process with a probable mature industry process. The mature industry projection is as follows:

Step 1: **Bulk quartzite - met. grade silicon.** The requirement to handle 100 - 200,000 tons annually is easily met. Mining companies, quarries, construction outfits, routinely handle much greater quantities than this.

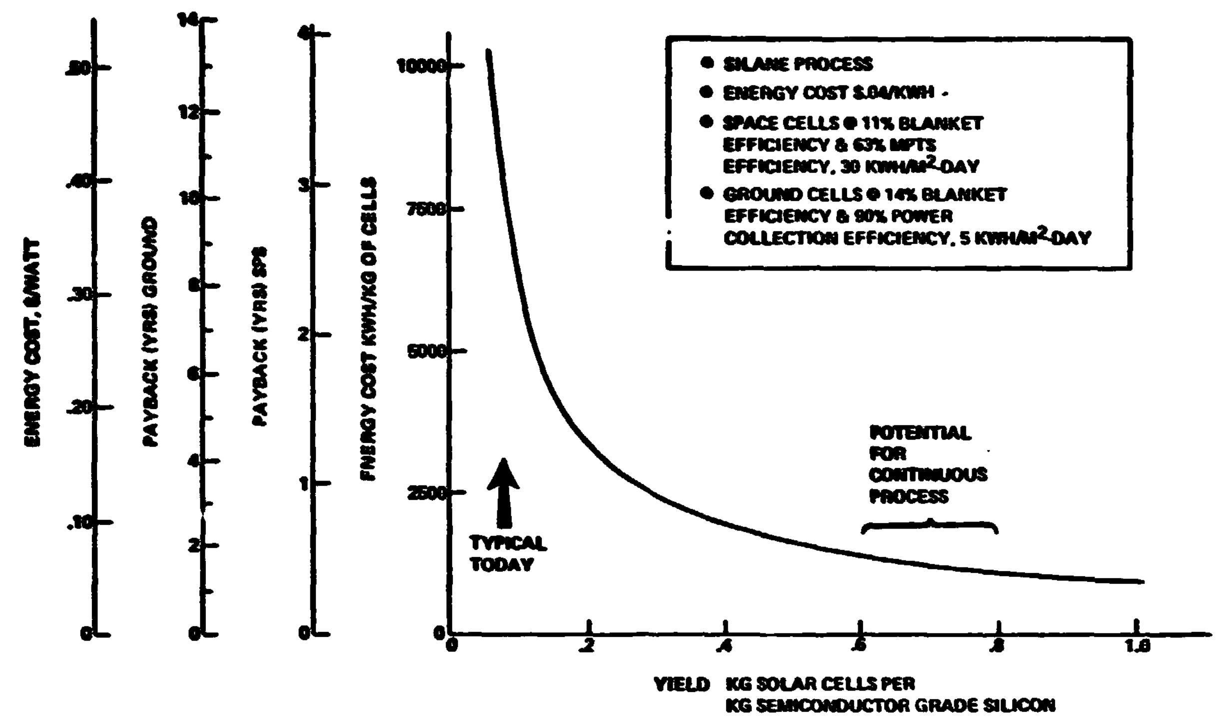

Figure 3.7-11 Energy Costs and Payback for Silicon Solar Cells

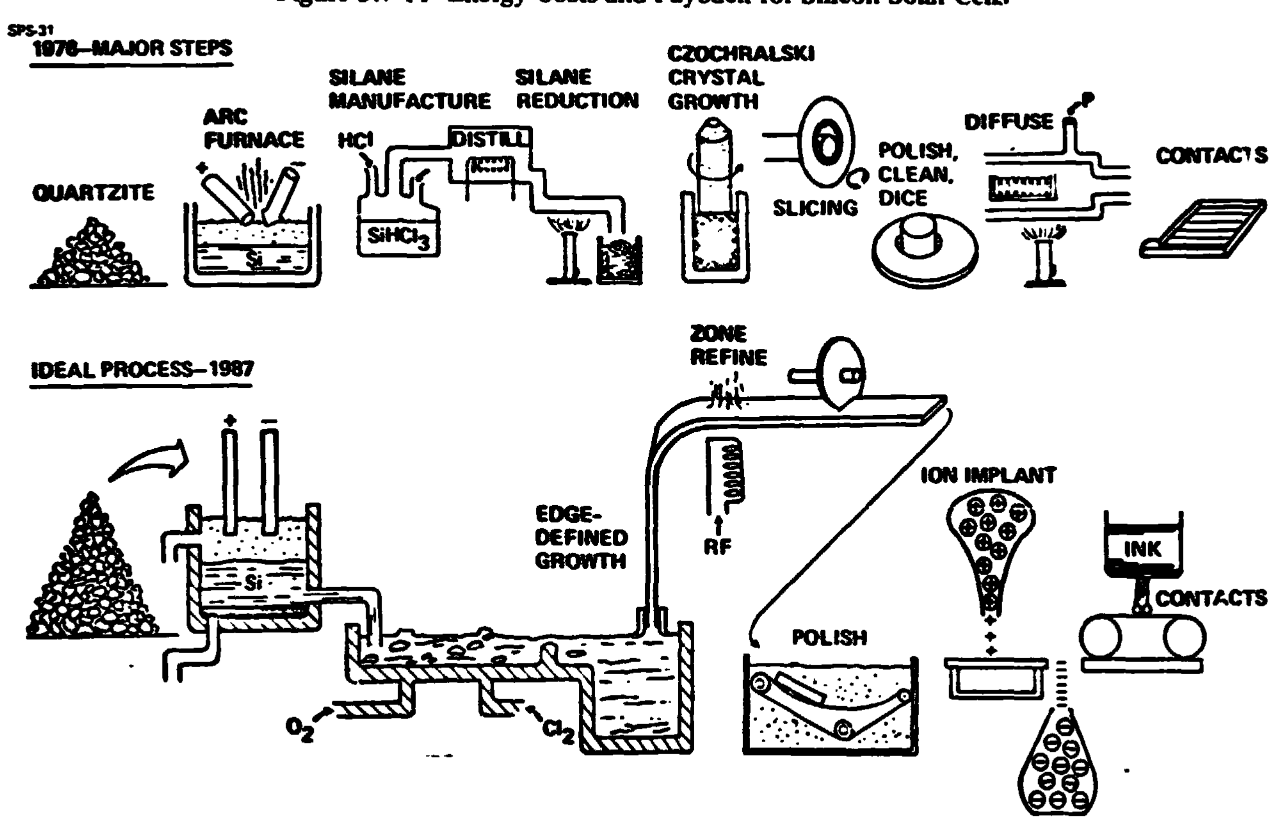

Figure 3.7-12 Silicon Solar Cell Manufacture

Step 2: **Silane production, etc.** This step resembles a petroleum refining process or other closed-cycle chemical and distillation process. It is amenable to and demands a highly automated closed cycle, hands-off process. The costs incurred here, where the silicon is processed to 6- or -7 nines pure, are mainly energy, process, and facility utilization costs.

Step 3: **Silicon monocrystal production**

a. Silicon - Ribbon:

This is a conceptual approach to a totally integrated system. A comment made by Dr. Handy of Motorola is relevant, when asked if he would or could respond to a request to deliver one dollar per square foot silicon cells. His answer was, "Maybe not one dollar, but three to five is possible."

The capillary dies used contribute to the impurity levels in the silicon monocrystal ribbon at present. This is not an insurmountable problem, but will require work, time and money, before this method is usable for mass production.

The maximum size of a unit cell for the Czochralski pull process appears to be about 10 X 10 cm. This size limitation would appear to rule out this method for any reasonable production quantity and probably will exclude this method for production quantities of 100 $(Km)^2$ per year.

Step 4: **Silicon cell, unit fabrication**

The preparation of the silicon monocrystal sheet or other to solar cells will be similar regardless of the previous steps This can be a highly automated system and, for mass production, the principal costs will be incurred because of process demands and facility utilization - not labor.

Step 5: **Solar cell panel assembly and test**

The panel size - cell size optimization will be a function of several things; unit size, panel size, end use (space or terrestrial), power levels, etc., all will contribute to the cost.

Test costs will be a functional quality and process control, end use and automation.

This will be the most intensive labor area.

3.7.5 Aerospace Historical Cost Estimating

Aerospace cost estimating employs correlations between historical physical characteristics of systems and cost experienced on those systems. These correlations are usually developed at the subsystems or component level. The Boeing Parametric Cost Model (PCM) has proven to be a particularly effective model of this type. Its basic estimating units are manhours (rates are used to convert to dollars) and it includes functional correlations to develop results in terms of design hours, development shop, software engineering, basic manufacturing, quality control, etc. Cost estimating algorithms include cost estimating relationships, use of factors, summing of lower level elements, direct input of hours or dollars, hardware off-the-shelf or modified off-the-shelf considerations, complexity, and subelement learning for systems with repetitive subelements.

CER's used in the SPS effort are plotted in Figure 3.7-13. DDT&E and theoretical first unit (TFU) estimates were developed for the reference silicon photovoltaic and Brayton thermal engine systems as updated by the study activity. Results are shown in Tables 3.7-1 and 3.7-2 for the thermal engine and photovoltaic systems. Relatively little effort was devoted to reviewing the DDT&E figures; the primary intent was to develop first unit estimates that could be compared with the mature industry results discussed below. As a guide to reading the tables, note item 9 ("20 Metre") or the first page. This is an element of the structure. The unit costed is 5000 lb mass. Item 9 is sub to, i.e., added into, item 8 "Beams" which is in turn added into item 6 for the total structure estimate. Estimating is by CER, #2 for DDT&E and #36 for TFU. The blend factor =1 indicates that the estimated value is equal to (1 times) the CER prediction. Support hours are derived from CER.s #28 and #54. This element is a new item (0% off-the-shelf, 0% mod), there are 215 of them in the SPS first unit and an 85% learning curve is used.

3.7.6 Mature Industry Estimates

The mature industry estimating concept was discussed in section 3.7.3. For the Part I activity, mature industry estimating used cost factors in dollars per kg representative of current large-scale production of hardware that is comparable to the SPS item in terms of materials used and complexity. Projections were necessary for solar cells and graphite composites where present-day mature industry analogs are not available. The discussion following begins with the photovoltaic system, then describes the thermal engine system, and concludes with some comparisons.

3.7.6.1 Photovoltaic System Cost

For the photovoltaic satellite, the costing was done to level 5 of the hardware WBS. The midterm mass statement was used. Level 3 is the satellite level and level 4 the major subsystems. The WBS is summarized in Table 3.7-3.

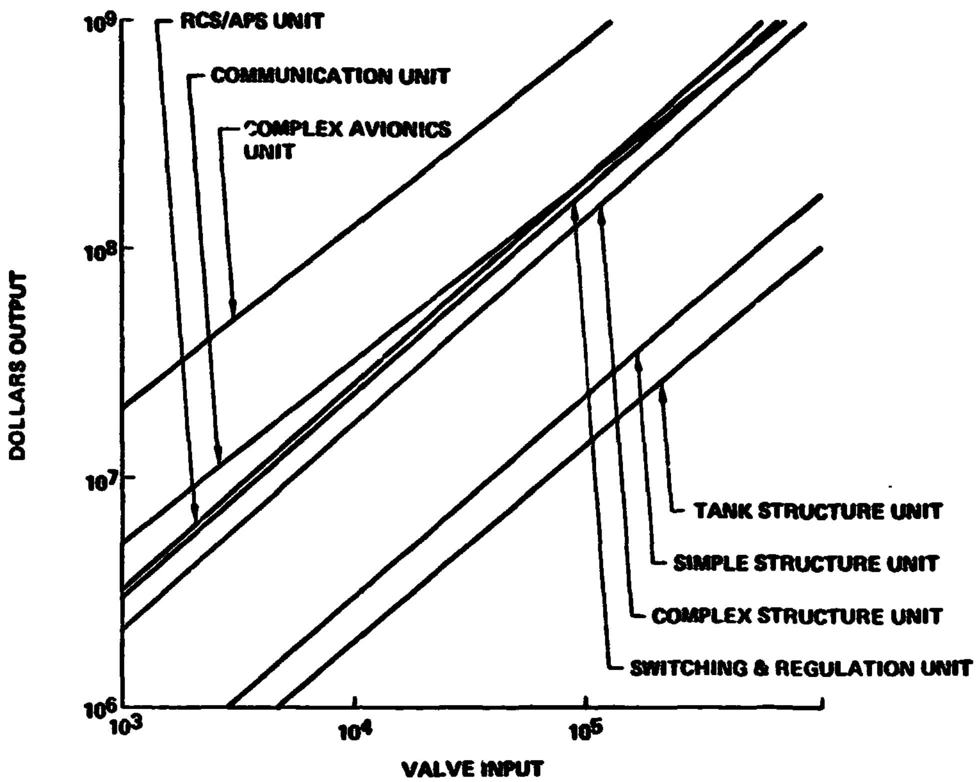

Figure 3.7-13 Sheet 1

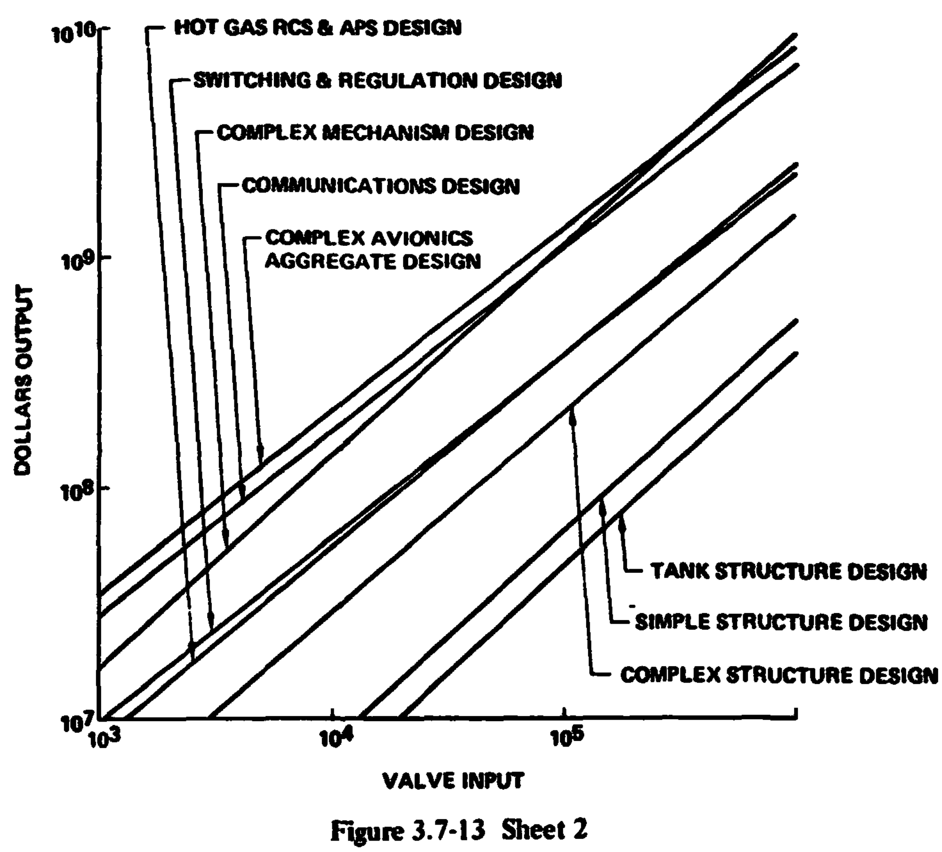

Figure 3.7-13 Sheet 2

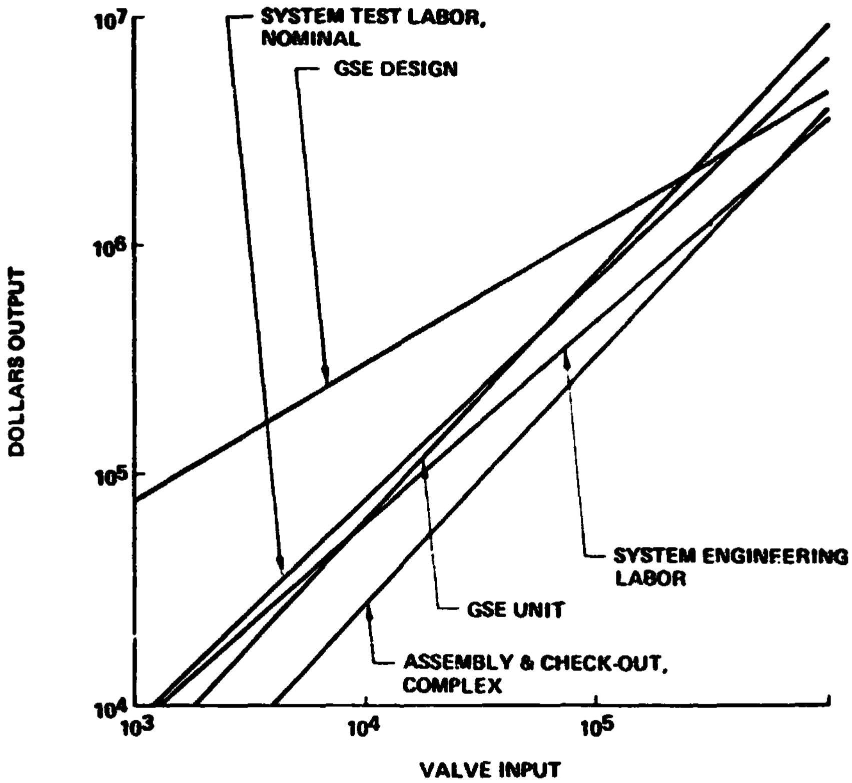
SYSTEM TEST LABOR, NOMINAL
GSE DESIGN
SYSTEM ENGINEERING LABOR
GSE UNIT
ASSEMBLY & CHECK-OUT, COMPLEX
DOLLARS OUTPUT
VALVE INPUT
10^7
10^6
10^5
10^4
10^3
10^4
10^5

Table 3.7-1 Sheet 1

NO	NAME	SUB TO	ELEMENT	METHOD	SOURCES	BLEND FACTORS	SUPT FROM	OTS %	MOD %	MOD CMPLX	NUMBER	LRN %	COST (000)
1	TOTAL PROGRAM	0	DDT&E	SUBS	0	0.00	0	0	0	0.0			27,494,784
			UNIT	SUBS	0	0.00	0				0	0	23,688,944
2	PROG INTER & MANG	1	DDT&E	FACTOR	3	0.10	0	0	0	0.0			1,347,644
			UNIT	FACTOR	3	0.10	0				0	0	79,447
3	THERMAL ENG SAT	1	DDT&E	SUBS	0	0.00	0	0	0	0.0			25,147,136
			UNIT	SUBS	0	0.00	0				0	0	23,809,488
4	SATELLITE D & D	3	DDT&E	SUBS	0	0.00	0	0	0	0.0			7,832,612
			UNIT	SUBS	0	0.00	0				0	0	23,658,320
5	MULTIPLE/COMMON USE	4	DDT&E	SUBS	0	0.00	0	0	0	0.0			877,857
			UNIT	SUBS	0	0.00	0				0	0	4,996,476
6	STRUCTURE	5	DDT&E	SUBS	0	0.00	0	0	0	0.0			53,214
			UNIT	SUBS	0	0.00	0				0	0	3,437,350
7	PLUGS 1808 LBS	6	DDT&E	CER	7	1.00	28	0	0	0.0			14,589
			UNIT	CER	36	1.00	54				32	85	74,182
8	BEAMS	6	DDT&E	SUBS	0	0.00	0	0	0	0.0			38,625
			UNIT	SUBS	0	0.00	0				0	0	3,413,168

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Table 3.7-1 Sheet 2

9	20 METRE 5000	LBS	8	DDTLE	CER	2	1.00	28	0	0	0.0			17,554
				UNIT	CER	36	1.00	54				215	85	814,498
10	10 METRE 2000	LBS	8	DDTLE	CER	2	1.00	28	0	0	0.0			8,211
				UNIT	CER	36	1.00	54				150	85	270,498
11	5 METRE 2000	LBS	8	DDTLE	CER	2	1.00	28	0	0	0.0			8,211
				UNIT	CER	36	1.00	54				445	85	625,405
12	1.5 METRE 1000	LBS	8	DDTLE	CER	2	1.00	28	0	0	0.0			4,648
				UNIT	CER	36	1.00	54				3700	85	1,702,765
13	CONTROL SYS		5	DDTLE	SUBS	0	0.00	0	0	0	0.0			710,295
				UNIT	SUBS	0	0.00	0				0	0	1,238,722
14	APS 60000	LBS	13	DDTLE	CER	7	1.00	28	0	0	0.0			255,679
				UNIT	CER	39	1.00	54				1	90	145,015
15	MDM X-CH 30000	LBS	13	DDTLE	CER	5	1.00	28	0	0	0.0			390,301
				UNIT	CER	36	1.00	54				16	90	621,722
16	AVIONICS 2000	LBS	13	DDTLE	CER	17	1.00	28	0	0	0.0			64,314
				UNIT	CER	48	1.00	54				16	90	471,983

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Table 3.7-1 Sheet 3

17 COMMUN/DATA 5500 LBS	5	DDT&E	CER	11	1.00	28	0	0	0.0			114,347
		UNIT	CER	44	1.00	54				16	90	270,406
18 ENERGY CELL SYS	4	DDT&E	SUBS	0	0.00	0	0	0	0.0			2,562
		UNIT	SUBS	0	0.00	0				0	0	2,130,605
19 FACETS 105 LBS	18	DDT&E	CER	5	1.00	28	0	0	0.0			2,562
		UNIT	CER	36	5.24	54				9999	85	2,130,805
20 ENERGY CONV SYS	4	DDT&E	SUBS	0	0.00	0	0	0	0.0			2,798,562
		UNIT	SUBS	0	0.00	0				0	0	11,894,980
21 CAVITY ABSORBER	20	DDT&E	SUBS	0	0.00	0	0	0	0.0			257,149
		UNIT	SUBS	0	0.00	0				0	0	5,133,328
22 STRUCTURE 79000 LBS	21	DDT&E	CER	3	1.00	28	0	0	0.0			55,266
		UNIT	CER	37	1.00	54				16	85	214,928
23 PANELS 2600 LBS	21	DDT&E	CER	2	1.00	28	0	0	0.0			10,198
		UNIT	CER	36	1.00	54				900	85	1,360,410
24 HEAT ABSORBER 312000 LBS	21	DDT&E	CER	3	1.00	28	0	0	0.0			131,684
		UNIT	CER	37	1.00	54				64	85	2,157,991

Table 3.7-1 Sheet 4

25 HEAT ABSORB MATL	21	DDT&E	N/A	0	0.00	0	0	0	0.0			0
		UNIT	$	0	0.00	0				1	85	1,400,000
26 THERMAL ENGINE SYS	20	DDT&E	SUBS	0	0.00	0	0	0	0.0			2,464,934
		UNIT	SUBS	0	0.00	0				0	0	3,856,513
27 TURBOMACHINES 216000 LBS	26	DDT&E	CER	5	1.00	28	0	0	0.0			2,355,665
		UNIT	CER	36	0.17	54				64	90	2,458,896
28 RECUP/COOLER 167000 LBS	26	DDT&E	CER	3	1.00	28	0	0	0.0			109,269
		UNIT	CER	37	1.00	54				64	85	1,247,617
29 RECUP MATL	26	DDT&E	N/A	0	0.00	0	0	0	0.0			0
		UNIT	$	0	0.00	0				1	84	150,000
30 HEAT REJECT SYS	20	DDT&E	SUBS	0	0.00	0	0	0	0.0			76,478
		UNIT	SUBS	0	0.00	0				0	0	2,905,144
31 DUCTING 25000 LBS	30	DDT&E	CER	2 3	0.50 0.50	28	0	0	0.0			47,919
		UNIT	CER	36 37	0.50 0.50	54				64	85	855,117
32 PUMPS 1600 LBS	30	DDT&E	CER	5	1.00	28	0	0	0.0			27,876
		UNIT	CER	36	1.00	54				512	90	862,554

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Table 3.7-1 Sheet 5

33 RADIATOR PANELS	30	DDT&E	CER	62	1.00	28	0	0	0.0			692
703 LBS		UNIT	CER	63	4.04	54				9999	85	1,119,472
34 RAK	30	DDT&E	N/A	0	0.00	0	0	0	0.0			0
		UNIT	$	0	0.00	0				1	89	68,000
35 POWER DIST SYS	4	DDT&E	SUBS	0	0.00	0	0	0	0.0			153,632
		UNIT	SUBS	0	0.00	0				0	0	1,673,094
36 POWER BUSSES	35	DDT&E	CER	62	0.50	28	0	0	0.0			104,595
460000 LBS		UNIT	CER	63	0.50	54				16	85	330,597
37 SWITCH GEAR	35	DDT&E	CER	13	1.00	28	0	0	0.0			49,036
6900 LBS		UNIT	CER	46	1.00	54				128	90	1,342,496
38 POWER TRANS	4	DDT&E	$	0	0.00	0	0	0	0.0			4,000,000
		UNIT	$	0	0.00	0				1	89	2,962,979
39 DESIGN INTEGRATION	3	DDT&E	FACTOR	4	1.00	0	0	0	0.0			3,795,870
		UNIT	N/A	0	0.00	0				0	0	0
40 ASSY & C/O	3	DDT&E	N/A	0	0.00	0	0	0	0.0			0
		UNIT	CER*	4 60	0.00 0.00	0				0	0	44,929

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Table 3.7-1 Sheet 6

41 TOOLING	3	DDT&E	FACTOR	4	0.50	0	0	0	0.0			3,566,955
		UNIT	N/A	0	0.00	0				0	0	0
42 SYSTEM TEST	3	DDT&E	SUBS	0	0.00	0	0	0	0.0			10,256,352
		UNIT	N/A	0	0.00	0				0	0	0
43 SYS TEST LABOR	42	DDT&E	CER*	4	0.00	0	0	0	0.0			731,119
				30	0.00							
		UNIT	N/A	0	0.00	0				0	0	0
44 GRD TEST HDWE	42	DDT&E	FAC UN	4	0.20	0	0	0	0.0			4,731,617
		UNIT	N/A	0	0.00	0				0	0	0
45 FLT TEST HDWE	42	DDT&E	FAC UN	4	0.20	0	0	0	0.0			4,731,617
		UNIT	N/A	0	0.00	0				0	0	0
46 SE & I	3	DDT&E	CER*	4	0.00	0	0	0	0.0			275,413
				29	0.00							
		UNIT	N/A	0	0.00	0				0	0	0
47 SATELLITE DDCT	0	DDT&E	FACTOR	4	1.00	0	0	0	0.0			0
				46	1.00							
				43	1.00							
		UNIT	N/A	0	0.00	0				0	0	0
48 SOFTWARE ENGR	3	DDT&E	CER*	47	0.00	0	0	0	0.0			331,499
				33	0.00							
		UNIT	N/A	0	0.00	0				0	0	0

Table 3.7-1 Sheet 7

49 GSE	3	DDTLF CER*	4 56	0.00 0.00	0	0	0	0.0			90,459
		UNIT CER*	4 57	0.00 0.00	0				0	0	106,247
50	0	DDTLE SUBS	.0	0.00	0	0	0	0.0			0
		UNIT SUBS	0	0.00	0				0	0	0
51	0	DDTLE SUBS	0	0.00	0	0	0	0.0			0
		UNIT SUBS	0	0.00	0				0	0	0
52	0	DDTLE SUBS	0	0.00	0	0	0	0.0			0
		UNIT SUBS	0	0.00	0				0	0	0
53	0	DDTLE SUBS	0	0.00	0	0	0	0.0			0
		UNIT SUBS	0	0.00	0				0	0	0
54	0	DDTLE SUBS	0	0.00	0	0	0	0.0			0
		UNIT SUBS	0	0.00	0				0	0	0
55	0	DDTLE SUBS	0	0.00	0	0	0	0.0			0
		UNIT SUBS	0	0.00	0				0	0	0
56	0	DDTLE SUBS	0	0.00	0	0	0	0.0			0
		UNIT SUBS	0	0.00	0				0	0	0

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Table 3.7-2 Sheet 1

NO	NAME	SUP TO	ELEMENT	METHOD	SOURCES	BLEND FACTORS	SUPT FROM	CTS %	MOD %	MOD CMPLX	NUMBER	LRN %	COST (000)
1	TOTAL PROGRAM	0	DDTCE	SUBS	0	0.00	0	0	0	0.0			19,858,032
			UNIT	SUBS	0	0.00	0				0	0	22,048,752
2	PROG INTER & MANG	1	DDTCE	FACTOR	3	0.10	0	0	0	0.0			341,910
			UNIT	FACTOR	3	0.10	0				0	0	41,811
3	PHOTOVOLTAIC ~~THERMAL~~ ENG SAT	1	DDTCE	SUBS	0	0.00	0	0	0	0.0			18,516,112
			UNIT	SUBS	0	0.00	0				0	0	22,006,944
4	SATELLITE D & D	3	DDTCE	SUBS	0	0.00	0	0	0	0.0			7,482,441
			UNIT	SUBS	0	0.00	0				0	0	21,930,528
5	MULTIPLE/COMMON USE	4	DDTCE	SUBS	0	0.00	0	0	0	0.0			842,397
			UNIT	SUBS	0	0.00	0				0	0	5,562,769
6	PRIMARY STRUC	5	DDTCE	SUBS	0	0.00	0	0	0	0.0			17,754
			UNIT	SUBS	0	0.00	0				0	0	4,053,643
7	JOINTS ETC LES	6	DDTCE	CER	7	1.00	28	0	0	0.0			6,155
			UNIT	CER	36	1.00	54				2598	85	839,767
8	BEAMS	6	DDTCE	SUBS	0	0.00	0	0	0	0.0			11,598
			UNIT	SUBS	0	0.00	0				0	0	3,213,876

Table 3.7-2 Sheet 2

9	20 METER CHORDS 1478 LBS	8	DDTLE CER	2	1.00	28	0	0	0.0			6,401
			UNIT CER	36	1.00	54				2598	85	1,845,494
10	20 METER BATTENS 647 LBS	8	DDTLE CER	2	1.00	28	0	0	0.0			3,261
			UNIT CER	36	1.00	54				2598	85	871,693
11	20 METER DIAG 339 LBS	8	DDTLE CER	2	1.00	28	0	0	0.0			1,935
			UNIT CER	36	1.00	54				2598	85	490,688
12	CONTROL SYS	5	DDTLE SUBS	0	0.00	0	0	0	0.0			710,295
			UNIT SUBS	0	0.00	0				0	0	1,238,722
13	APS 60000 LBS	12	DDTLE CER	.7	1.00	28	0	0	0.0			255,679
			UNIT CER	39	1.00	54				1	90	145,015
14	MOM X-CH 30000 LBS	12	DDTLE CER	5	1.00	26	0	0	0.0			390,301
			UNIT CER	36	1.00	54				16	90	621,722
15	AVIONICS 2000 LBS	12	DDTLE CER	17	1.00	28	0	0	0.0			64,314
			UNIT CER	48	1.00	54				16	90	471,983
16	COMMUN/DATA 5500 LBS	5	DDTLE CER	11	1.00	26	0	0	0.0			114,347
			UNIT CER	44	1.00	54				16	90	270,406

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Table 3.7-2 Sheet 3

17	ENERGY COLL SYS	4	DDT&E	SUBS	0	0.00	0	0	0	0.0			22,470
			UNIT	SUBS	0	0.00	0				0	0	649,960
18	SUPPORT STRUC 84.6 LBS	17	DDT&E	CER	5	1.00	28	0	0	0.0			2,129
			UNIT	CER	36	1.11	54				9999	85	435,959
19	KAPTON SHEETS	17	DDT&E	$	0	0.00	0	0	0	0.0			20,000
			UNIT	$	0	0.00	0				1	90	153,610
20	TENSION TIES 9.4 LBS	17	DDT&E	CER	5	1.00	28	0	0	0.0			340
			UNIT	CER	36	1.11	54				9999	85	60,390
21	ENERGY CONV SYS	4	DDT&E	SUBS	0	0.00	0	0	0	0.0			2,500,000
			UNIT	SUBS	0	0.00	0				0	0	12,156,145
22	SOLAR CELL BLANKET	21	DDT&E	$	0	0.00	0	0	0	0.0			2,500,000
			UNIT	$	0	0.00	0				17	80	12,156,145
23	POWER DIST SYS	4	DDT&E	SUBS	0	0.00	0	0	0	0.0			117,574
			UNIT	SUBS	0	0.00	0				0	0	578,695
24	POWER BUSSES 477932 LBS	23	DDT&E	CER	62	0.50	28	0	0	0.0			108,246
			UNIT	CER	63	0.50	54				16	85	341,588

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Table 3.7-2 Sheet 4

25 SWITCH GEAR 794 LBS	23	DDT&E CER	13	1.00	28	0	0	0.0			9,327
		UNIT CER	46	1.00	54				176	90	257,107
26 POWER TRANS	4	DDT&E $	0	0.00	0	0	0	0.0			4,000,000
		UNIT $	0	0.00	0				1	89	2,962,979
27 DESIGN INTEGRATION	3	DDT&E FACTOR	4	1.00	0	0	0	0.0			948,622
		UNIT N/A	0	0.00	0				0	0	0
28 ASSY & C/O	3	DDT&E N/A	0	0.00	0	0	0	0.0			0
		UNIT CER*	4 60	0.00 0.00	0				0	0	22,716
29 TOOLING	3	DDT&E FACTOR	4	0.50	0	0	0	0.0			891,602
		UNIT N/A	0	0.00	0				0	0	0
30 SYSTEM TEST	3	DDT&E SUBS	0	0.00	0	0	0	0.0			8,975,671
		UNIT N/A	0	0.00	0				0	0	0
31 SYS TEST LABOR	30	DDT&E CER*	4 30	0.00 0.00	0	0	0	0.0			203,548
		UNIT N/A	0	0.00	0				0	0	0
32 GRD TEST HDWE	30	DDT&E FAC UN	4	0.20	0	0	0	0.0			4,386,062
		UNIT N/A	0	0.00	0				0	0	0

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Table 3.7-2 Sheet 5

33 FLT TEST HDWE	30	DDTCE	FAC UV	4	0.20	0	0	0	0.0			4,386,062
		UNIT	N/A	0	0.00	0				0	0	0
34 SE & I	3	DDTCE	CER*	4	0.00	0	0	0	0.0			79,904
				29	0.00							
		UNIT	N/A	0	0.00	0				0	0	0
35 SATELLITE DDCT	0	DDTCE	FACTOR	4	1.00	0	0	0	0.0			0
				34	1.00							
				31	1.00							
		UNIT	N/A	0	0.00	0				0	0	0
36 SOFTWARE ENGR	3	DDTCE	CER*	35	0.00	0	0	0	0.0			98,567
				33	0.00							
		UNIT	N/A	0	0.00	0				0	0	0
37 GSE	3	DDTCE	CER*	4	0.00	0	0	0	0.0			39,102
				56	0.00							
		UNIT	CER*	4	0.00	0				0	0	53,702
				57	0.00							

SPS-650

Table 3.7-3 A1.01.01 Photovoltaic Satellite, CR = 2.0, Si

WBS	NAME
A1.01.01	PHOTOVOLTAIC SATELLITE
A1.01.01.00	MULTIPLE/COMMON USE EQUIP
A1.01.01.00.00	PRIMARY STRUCTURE
A1.01.01.00.01	CONTROL SYSTEMS
A1.01.01.00.02	COMMUNICATION/DATA SYSTEMS
A1.01.01.00.03	MECHANICAL SYSTEMS
A1.01.01.01	ENERGY COLLECTION SYSTEMS
A1.01.01.01.00	SUPPORT STRUCTURE
A1.01.01.01.01	KAPTON REFLECTOR SHEETS
A1.01.01.01.02	SHEET TENSION TIES
A1.01.01.02	ENERGY CONVERSION SYSTEM
A1.01.01.02.00	SUPPORT STRUCTURE
A1.01.01.02.01	SOLAR CELL BLANKET
A1.01.01.02.02	ANNEALING MACHINES
A1.01.01.03	POWER DISTRIBUTION SYSTEM
A1.01.01.03.00	SUPPORT STRUCTURE
A1.01.01.03.01	POWER BUSES
A1.01.01.03.02	DISCONNECTS
A1.01.01.03.03	IN-LINE SWITCH GEAR
A1.01.01.03.04	ROTARY JOINT
A1.01.01.04	POWER TRANSMISSION SYSTEM

SPS-664

Table 3.7-4 A10101 Photovoltaic Satellite

WBS	NAME	MASS (MT)	DOLLARS (MILLIONS)
A101.01	PHOTOVOLTAIC	74874	6422.7
A1010100	MULTIPLE/COMMON	15354	983.5
A1010101	ENERGY COLLECTION	3197	175.8
A1010102	ENERGY CONVERSION	37962	2114.6
A1010103	POWER DISTRIBUTION	3361	185.8
A1010104	POWER TRANSMISSION	15000	2963.0

Silicon photovoltaic cost results are summarized in Table 3.7-4 and Figure 3.7-14. Note that the largest cost driver appears to be the MPTS for which we used the JSC baseline cost of 2.963 billion. The principal costs are associated with the solar cell blanket and the next largest cost driver is the multiple/common use systems which includes the primary structure.

The next set of tables and figures (Tables 3.7-5 through 3.7-8 and Figures 3.7-15 through 3.7-18) provide cost breakouts of the major systems. First, A1010100, multiple common equipment, (Table 3.7-5 and Figure 3.7-15) which includes:

Primary structure
Satellite control systems
Satellite communication/data systems and
Mechanical systems

The cost estimating relations are shown in millions of dollars per metric ton.

The primary structure is the major item (80+%) and control systems next at about 16%.

The energy collection system is summarized in Table 3.7-6 and Figure 3.7-16. It is apparently the lowest cost system in the set. Kapton reflector sheets at $55 Kg dominate the cost (90+%).

The energy conversion system (Table 3.7-7 and Figure 3.7-17) has really only one entry the solar cell blanket. The support structure entry is a token entry only. The cell blanket costs were calculated using the "mature industry" data shown earlier where the volume of mono crystal silicon needed per satellite was taken as the production volume. This is a net cost of about $56 Kg for the blanket.

The on board power distribution system accepts 17.3 GW to delivery 15.9 GW to the Microwave Power Transmission system. The largest cost driver here appears to be the in-line switch gear. The next largest driver is the power buses. Results are summarized in Table 3.7-8 and Figure 3.7-18.

The "should-cost" numbers for the microwave power transmission system used the JSC baseline cost number of last summer (JSC-11568, Aug. 76). Since the system is common to both the Brayton and Photovoltaic system, no new "should-cost" numbers were generated for this part of the study.

3.7.6.2 Thermal Engine System Cost

The next set of data relates the "should-cost" story for the Brayton cycle system. As seen in Table 3.7-9, we were forced lower in the WBS (to level 6) in order to track the major cost drivers. The system costed here was based on a 16-module satellite. The top level costs are presented in Table 3.7-10 and Figure 3.7-19.

SPS-659

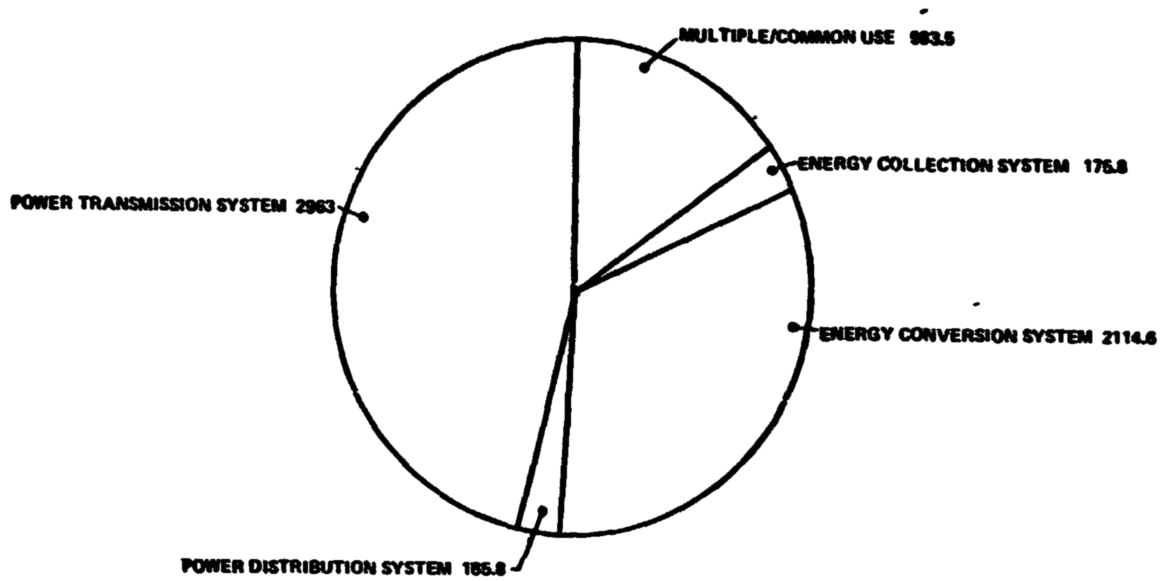

Figure 3.7-14 A1.01.01 Photovoltaic Satellite

Table 3.7-5 A1.01.01 Photovoltaic Satellite

SPS-674

WBS	NAME	CER	MASS (MT)	DOLLARS (MILLIONS)
A1.01.01.00	MULTIPLE/COMMON USE EQUIP	X	15354	983.5
A1.01.01.00.00	PRIMARY STRUCTURE	.055	14970	823.4
A1.01.01.00.01	CONTROL SYSTEMS	.440	340	149.6
A1.01.01.00.02	COMMUNICATION/DATA SYSTEMS	.440	4	1.8
A1.01.01.00.03	MECHANICAL SYSTEMS	.220	40	8.8

SPS-884

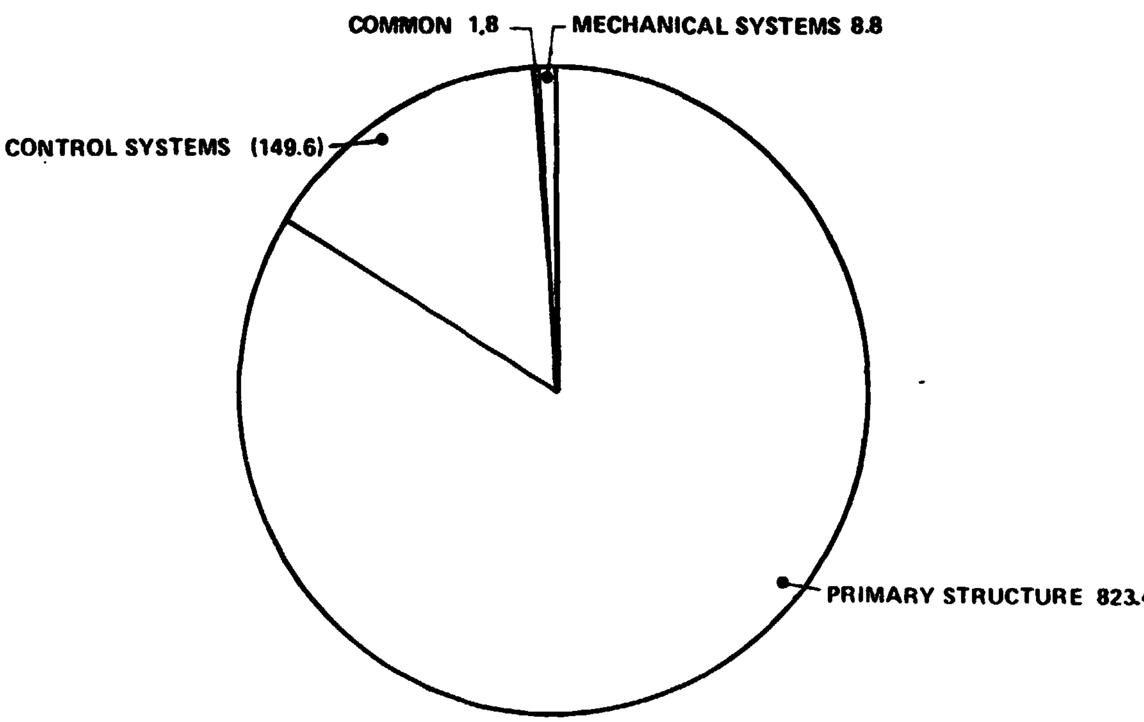

Figure 3.7-15 A1.01.01 Photovoltaic Satellite A1.01.01.00 Multiple/Common Use Equipment 983.5

SPS-886

Table 3.7-6 A1.01.01 Photovoltaic Satellite

WBS	NAME	CER	MASS (MT)	DOLLARS (MILLIONS)
A1.01.01.01	ENERGY COLLECTION SYSTEMS	X	3197	175.8
A1.01.01.01.00	SUPPORT STRUCTURE	.055	209	11.5
A1.01.01.01.01	KAPTON REFLECTOR SHEETS	.055	2978	163.8
A1.01.01.01.02	SHEET TENSION TIES	.055	10	0.6

SPS-883

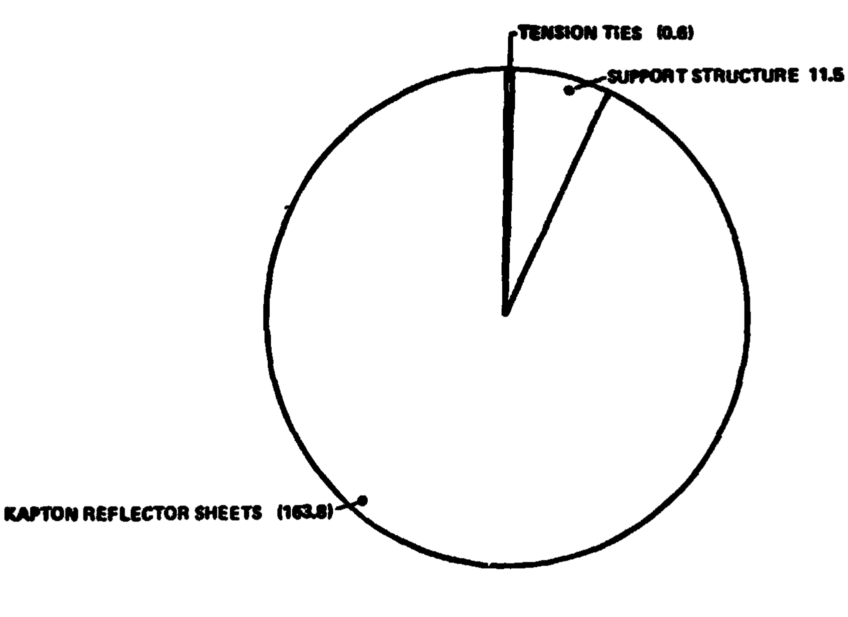

Figure 3.7-16 A1.01.01 Photovoltaic Satellite A1.01.01.01 Energy Collection System 175.8

SPS-680

Table 3.7-7 A1.01.01 Photovoltaic Satellite

WBS	NAME	CER	MASS (MT)	DOLLARS (MILLIONS)
A1.01.01.02	ENERGY CONVERSION SYSTEM	X	37962	2114.6
A1.01.01.02.00	SUPPORT STRUCTURE	.055	10	.6
A1.01.01.02.01	SOLAR CELL BLANKET	CALC.	37952	2114.1
A1.01.01.02.02	ANNEALING MACHINES	–	–	–

SPS-682

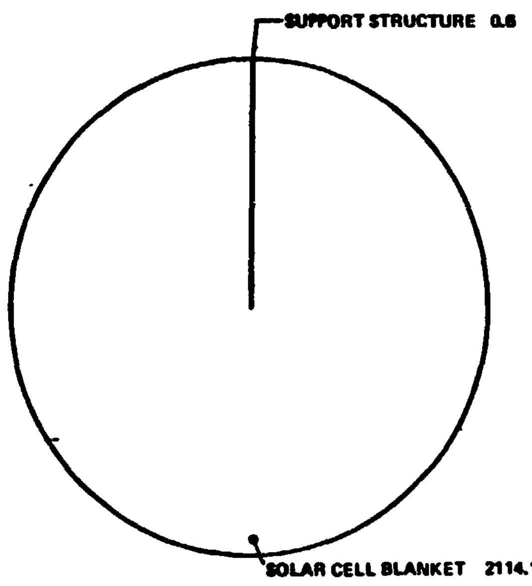

Figure 3.7-17 A1.01.01 Photovoltaic Satellite A1.01.01.01 Energy Conversion System 2114.6

Table 3.7-8 A10101 Photovoltaic Satellite

WBS	NAME	CER	MASS (MT)	DOLLARS (MILLIONS)
A1.01.01.03	POWER DISTRIBUTION SYSTEM	X	3361	185.8
A1.01.01.03.00	SUPPORT STRUCTURE	.055	10	0.6
A1.01.01.03.01	POWER BUSES	.020	3000	60.0
A1.01.01.03.02	DISCONNECTS	.020	50	1.0
A1.01.01.03.03	IN-LINE SWITCH GEAR	.414	300	124.2
A1.01.01.03.04	ROTARY JOINT	X	X	X

SPS-681

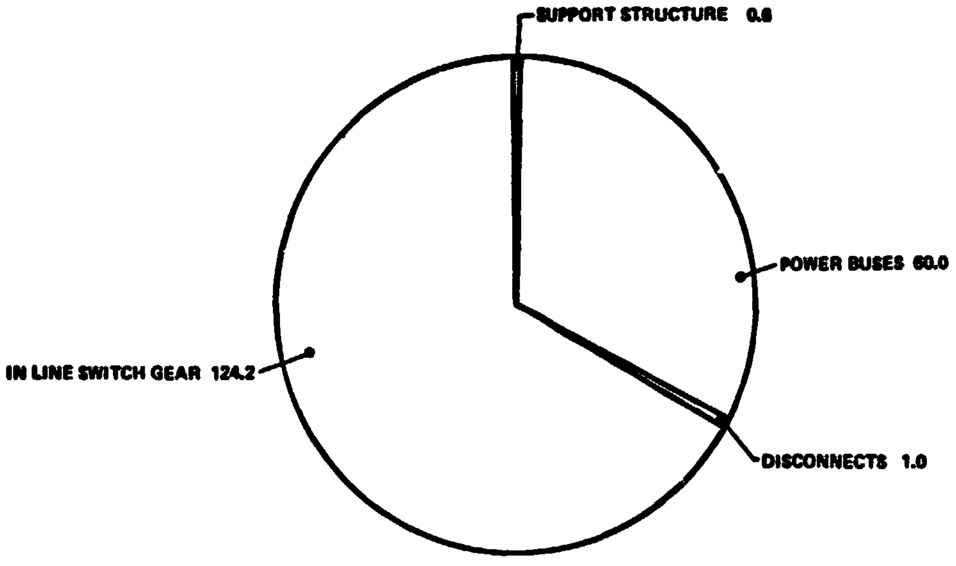

Figure 3.7-18 A1.01.01 Photovoltaic Satellite A1.01.01.03 Power Distribution System 185.8

Table 3.7-9 WBS Thermal Engine Satellite

SPS-679

B1.01.01	THERMAL ENGINE SATELLITE
B1.01.01.00	MULTIPLE/COMMON USE
B1.01.01.00.00	PRIMARY STRUCTURE
B1.01.01.00.01	CONTROL SYSTEMS
B1.01.01.00.02	COMMUNICATION/DATA SYSTEMS
B1.01.01.00.03	MECHANICAL SYSTEMS
B1.01.01.01	ENERGY COLLECTION SYSTEMS
B1.01.01.01.00	SUPPORT STRUCTURE
B1.01.01.01.01	FACETS
B1.01.01.01.01.00	STRUCTURE
B1.01.01.01.01.01	KAPTON REFLECTORS
B1.01.01.01.01.02	TENSION TIES
B1.01.01.01.02	FACET AIMING AND CONTROL
B1.01.01.01.02.00	STRUCTURE
B1.01.01.01.02.01	SOLARCELLS & BRIDGE
B1.01.01.01.02.02	LOGIC CKT
B1.01.01.01.02.03	BI-METALLIC DRIVE
B1.01.01.02	ENERGY CONVERSION SYSTEM
B1.01.01.02.00	SUPPORT STRUCTURE
B1.01.01.02.01	CAVITY ABSORBER
B1.01.01.02.01.00	STRUCTURE
B1.01.01.02.01.01	PANELS
B1.01.01.02.01.02	HEAT ABSORBER
B1.01.01.02.02	THERMAL ENGINE SYSTEM
B1.01.01.02.02.00	SUPPORT STRUCTURE
B1.01.01.02.02.01	TURBO-MACHINES, ET AL
B1.01.01.02.02.02	RECUPERATOR-COOLER
B1.01.01.02.03	HEAT REJECTION SYSTEM
B1.01.01.02.03.00	STRUCTURE
B1.01.01.02.03.01	DUCTING
B1.01.01.02.03.02	PUMPS
B1.01.01.02.03.03	RADIATOR PANELS
B1.01.01.02.03.04	NAK
B1.01.01.03	POWER DISTRIBUTION SYSTEM
B1.01.01.03.00	SUPPORT STRUCTURE
B1.01.01.03.01	POWER BUSES
B1.01.01.03.03	IN-LINE SWITCH GEAR
B1.01.01.03.04	ROTARY JOINT
B1.01.01.04	POWER TRANSMISSION SYSTEM

SPS-677

Table 3.7-10 B10101 Thermal Engine Satellite

WBS	NAME	MASS (MT)	DOLLARS (MILLIONS)
B10101	THERMAL ENGINE SATELLITE	79481	6047.0
B1010100	MULTIPLE/COMMON	3110	317.4
B1010101	ENERGY COLLECTION	4205	278.2
B1010102	ENERGY CONVERSION	53385	2372.6
B1010103	POWER DISTRIBUTION	3781	115.8
B1010104	POWER TRANSMISSION	15000	2963.

SPS-800

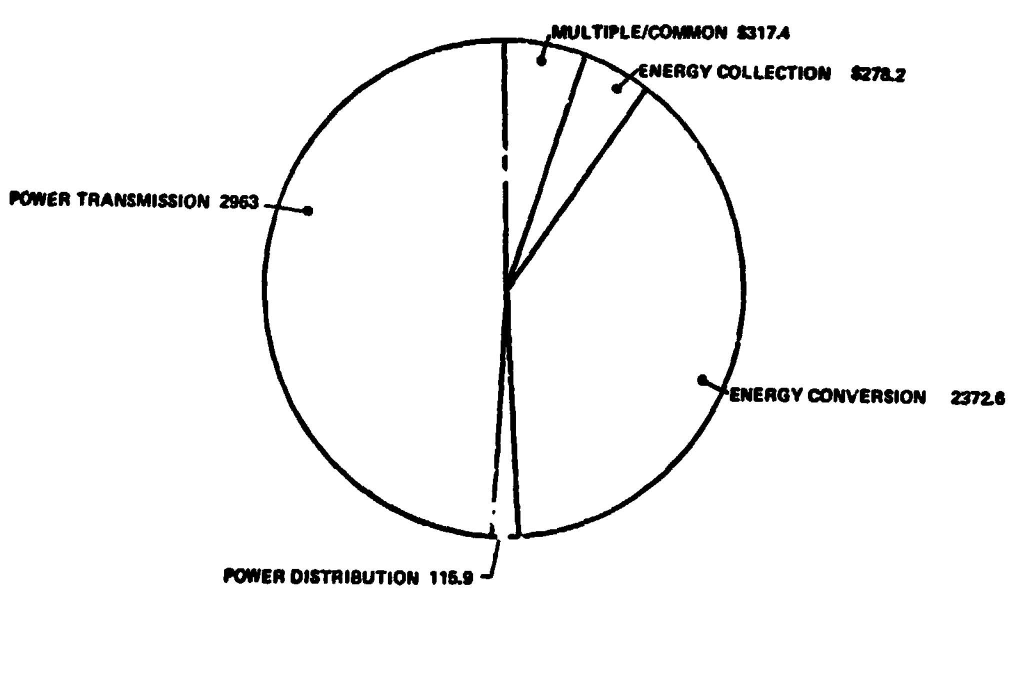

Figure 3.7-19 B1.01.01 Thermal Engine Satellite

Again the MPTS dominates, with the energy conversion system second. The masses of the system are shown. This system was heavier (79 481 MT vs. 74 874 MT) than the photovoltaic, but less costly (6047 vs. 6423) million for the production cost.

Table 3.7-11 and Figure 3.7-20 summarize results for the multiple/common system. The primary structure mass was 3110 MT and the cost at .055 million dollars per metric ton was 150 2 million. Again the control, comm/data, and mechanical systems were those used for the photovoltaic system.

For the energy collection system of thermal satellite, the largest cost drivers are the facets within which the structure cost at 154 million is the largest, as summarized in Table 3.7-12 and Figure 3.7-21.

The principal cost accumulating area is the energy conversion system within which the cavity absorbers at 1.126 billion and heat rejection system at 863.3 million dominate.

Note that the cost of 64 power generation units has been estimated at just over 4 million each. Energy conversion system results are summarized in Table 3.7-13 and Figure 3.7-22.

For the power distribution system (Table 3.7-14 and Figure 3.2-23) the cost of in-line switch gear is somewhat smaller, while the power bus cost has increased slightly with respect to the photovoltaic system.

The "should-cost" for the power distribution system places it at the bottom of the cost drivers so far as priority is concerned.

For the MPTS system the JSC values were also used in estimating thermal engine SPS costs. This is an area where considerable effort will be spent during Part II of the study.

3.7.6.3 Comparison and Summary

Summarizing the preceding sets of data, the reference photovoltaic and the Brayton system acquisition costs including DDTE, Production, and Installation are compared in Figure 3.7-24. The DDTE costs should be comparable on a per satellite basis.

The production costs discussed in the mature industry section show a small delta in favor of Brayton while the installation costs (which include estimates of construction base, LEO and GEO assembly, and transport costs) bring the totals for LEO assembly to 11,409 billion for the Brayton and 11.512 for the reference silicon system. The cost delta is insignificant and overriden by probable error for the should-cost numbers.

SPS-678

Table 3.7-11 B1.01.01 Thermal Engine Satellite

WBS	NAME	CER	MASS (MT)	DOLLARS (MILLIONS)
B1.01.01.00	MULTIPLE/COMMON USE	X	3110	317.4
B1.01.01.00.00	PRIMARY STRUCTURE	.055	2730	150.2
B1.01.01.00.01	CONTROL SYSTEMS	.440	340	149.6
B1.01.01.00.02	COMMUNICATION/DATA SYSTEMS	.440	40	17.6
B1.01.01.00.03	MECHANICAL SYSTEMS	.220	0	0

SPS-680

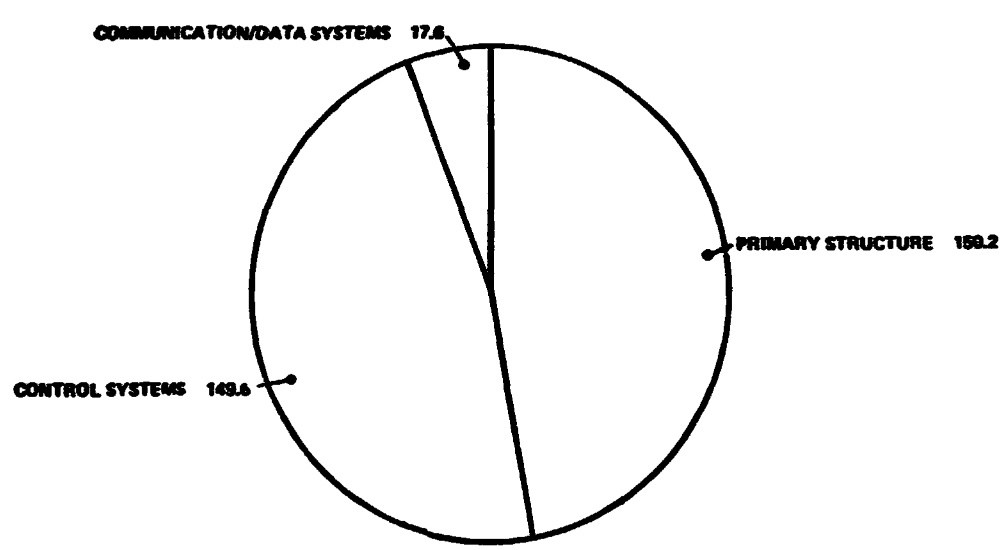

Figure 3.7-20 B1.01.01 Thermal Engine Satellite B1.01.01.00 Multiple/Common Use Equipment 317.4

Table 3.7-12 B1.01.01 Thermal Engine Satellite

WBS	NAME	CER	MASS (MT)	DOLLARS (MILLIONS)
B1.01.01.01	ENERGY COLLECTION SYSTEMS	X	4206	278.2
B1.01.01.01.00	SUPPORT STRUCTURE	.055	44	2
B1.01.01.01.01	FACETS	X	3952	217
B1.01.01.01.01.00	STRUCTURE	.055	2791	154
B1.01.01.01.01.01	KAPTON REFLECTORS	.055	1061	58
B1.01.01.01.01.02	TENSION TIES	.055	100	5
B1.01.01.01.02	FACET AIMING AND CONTROL	X	209	58
B1.01.01.01.02.00	STRUCTURE	.055	87	5
B1.01.01.01.02.01	SOLAR CELL AND BRIDGE	.440	17	7
B1.01.01.01.02.02	LOGIC CKT	.440	18	8
B1.01.01.01.02.03	BI-METALLIC DRIVE	.440	87	38

SPS-696

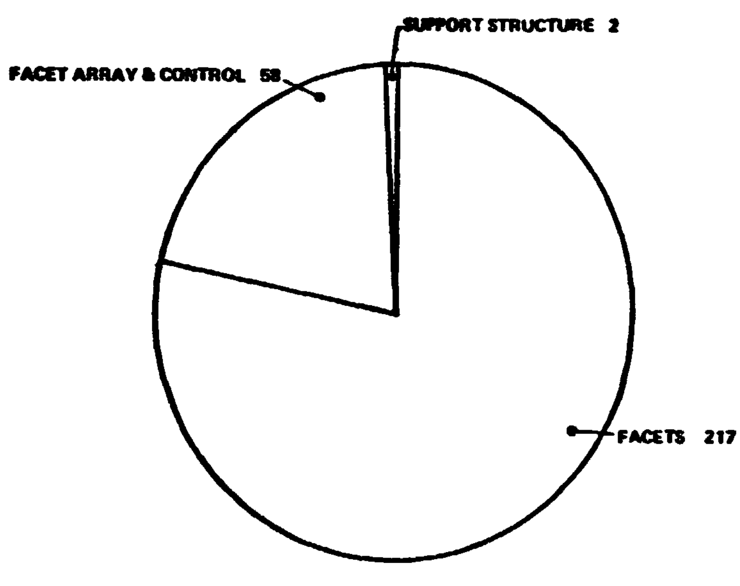

Figure 3.7-21 B1.01.01 Thermal Engine Satellite B1.01.01.01 Energy Collection System 278.2

SPS-673

Table 3.7-13 B1.01.01 Thermal Engine Satellite

WBS	NAME	CER	MASS (MT)	DOLLARS (MILLIONS)
B1.01.01.02	ENERGY CONVERSION SYSTEM	X	53385	2372.6
B1.01.01.02.00	SUPPORT STRUCTURE	.055	0	0
B1.01.01.02.01	CAVITY ABSORBER	X	10694	1126.5
B1.01.01.02.01.00	STRUCTURE	.022	566	12.45
B1.01.01.02.01.01	PANELS	.110	1058	116.38
B1.01.01.02.01.02	HEAT ABSORBER	.110	9070	997.70
B1.01.01.02.02	THERMAL ENGINE SYSTEM	X	11130	382.8
B1.01.01.02.02.00	SUPPORT STRUCTURE	0.55	0	0
B1.01.01.02.02.01	TURBO-MACHINES, ET AL.	.044	6270	275.9
B1.01.01.02.02.02	RECUPERATOR-COOLER	.022	4860	106.9
B1.01.01.02.03	HEAT REJECTION SYSTEM	X	31561	863.3
B1.01.01.02.03.00	STRUCTURE	.055	459	25.2
B1.01.01.02.03.01	DUCTING	.022	724	15.9
B1.01.01.02.03.02	PUMPS	.440	368	161.9
B1.01.01.02.03.03	RADIATOR PANELS	.022	19763	432.8
B1.01.01.02.03.04	NAK	.022	10337	227.4

SPS-668

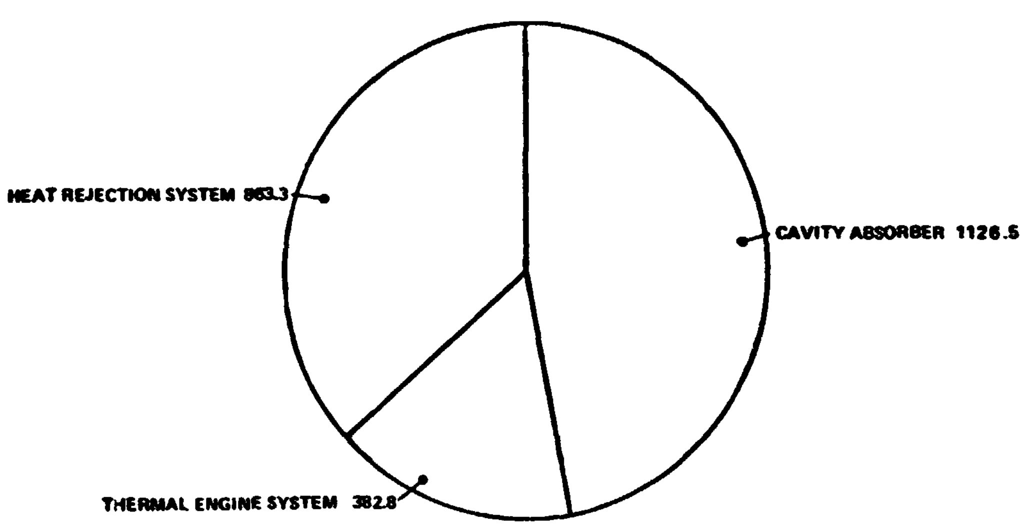

Figure 3.7-22 B1.01.01 Thermal Engine Satellite B1.01.01.02 Energy Conversion System 2372.6

SPS-658

Table 3.7-14 B1.01.01 Thermal Engine Satellite

WBS	NAME	CER	MASS (MT)	DOLLARS (MILLIONS)
B1.01.01.03	POWER DISTRIBUTION SYSTEM	X	3781	115.8
B1.01.01.03.00	SUPPORT STRUCTURE	.055	10	0.6
B1.01.01.03.01	POWER BUSES	.020	3370	67.40
B1.01.01.03.02	DISCONNECTS	.020	300	6.00
B1.01.01.03.03	IN-LINE SWITCH GEAR	.414	100	41.4
B1.01.01.03.04	ROTARY JOINT	X	X	X

SPS-685

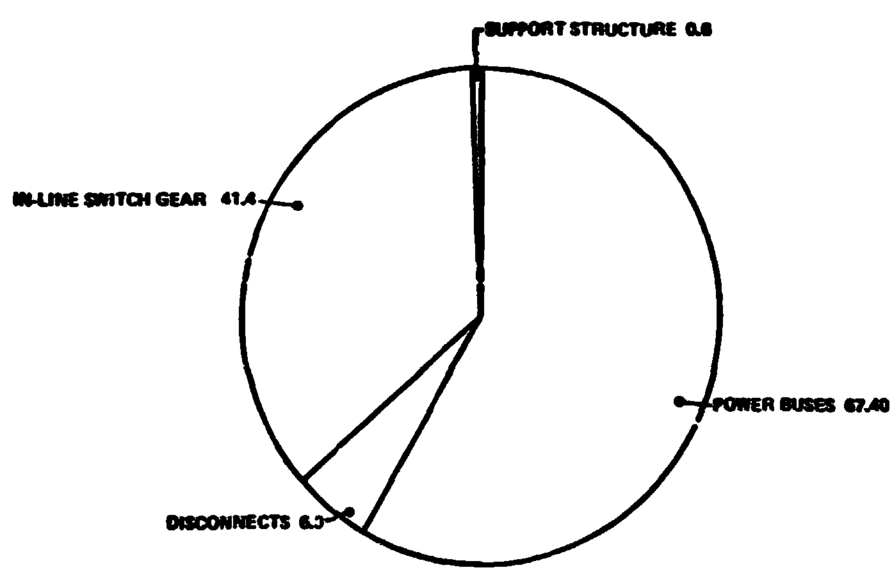

Figure 3.7-23 B1.01.01 Thermal Engine Satellite B1.01.01.03 Power Distribution System 115.8

SPS 727

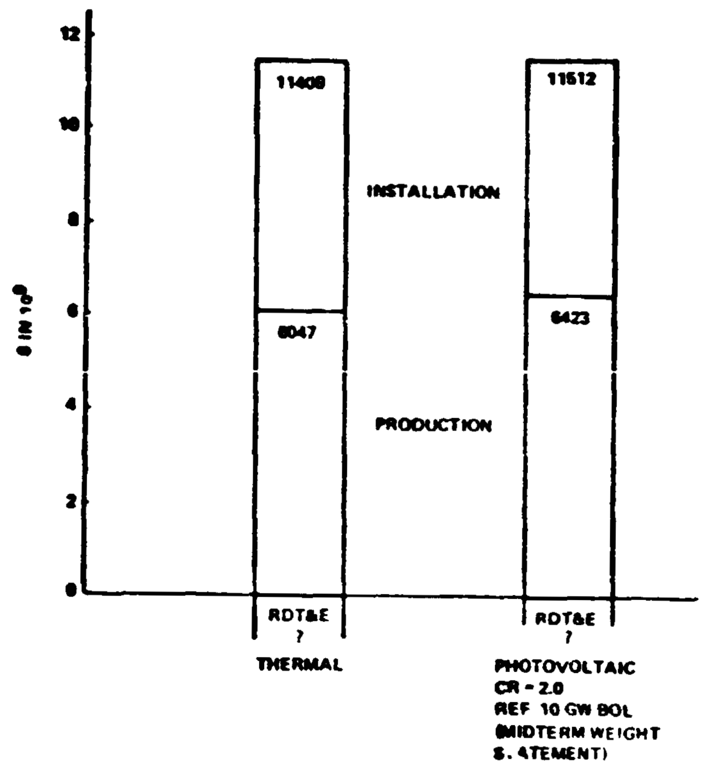

Figure 3.7-24 Acquisition Cost, LEO Assembly

In considering life cycle costs over a 30-year lifetime, the acquisition cost will probably be overshadowed by operations and support (O&S) costs, as suggested by Figure 3.7-25. The O&S costs for power systems have historically been 50% of gross revenue with the majority of these costs incurred in the distribution system. The staff hierachy built up in support of the system will probably dominate the O&S costs. To what extent historical relationships can be extrapolated to SPS is questionable. The operational costs are certainly functionally dependent on the availability required, MTBF, mean time to repair and mean logistics delay time.

The "will-cost" (aerospace PCM) and "should-cost" (mature-industry) estimates are compared in the next four figures. First, the Brayton system in Figure 3.7-26.

The "will-cost" numbers on the left presume business as usual without consideration of design-to-cost or its equivalent. The first unit cost, under standard aerospace management and estimating, will be 20.605 billion. The average cost using an 85% improvement curve will be 8.788 billion.

The should cost number is 3.034 billion. Both numbers exclude the MPTS value. The significant item here is that the relative distribution of costs is the same by major system and the should cost number is ~ 40% of the average cost.

The relatively greater share of cost occupied by multiple/common in the PCM is in part due to a difference in handling of structure. The PCM analysis considered primary and secondary structure as multiple/common. The mature industry analysis included secondary structure in the related subsystems.

Similarly, for the photovoltaic satellite a comparison of "aerospace costing" and mature industry (Figure 3.7-27) shows that (on the left) TFU (typical aerospace) will be 18.97 billion, with an average (over 112 units on 85% IC) 8.09 billion while the "should cost" mature industry number is 3.46 billion-again ~ 40% of the aerospace value.

Figures 3.7-28 and 3.7-29 show the mature industry estimates as percent of aerospace TFU for the total SPS and for the level 4 major subsystems.

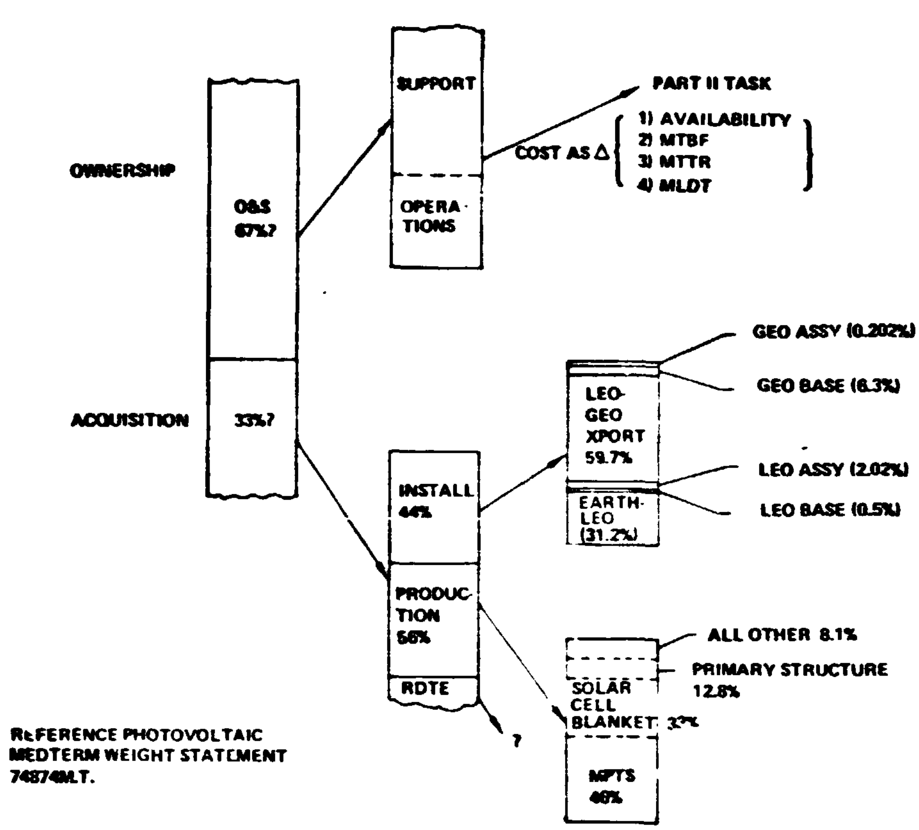

Figure 3.7-25 ? Where Does The Money Go?

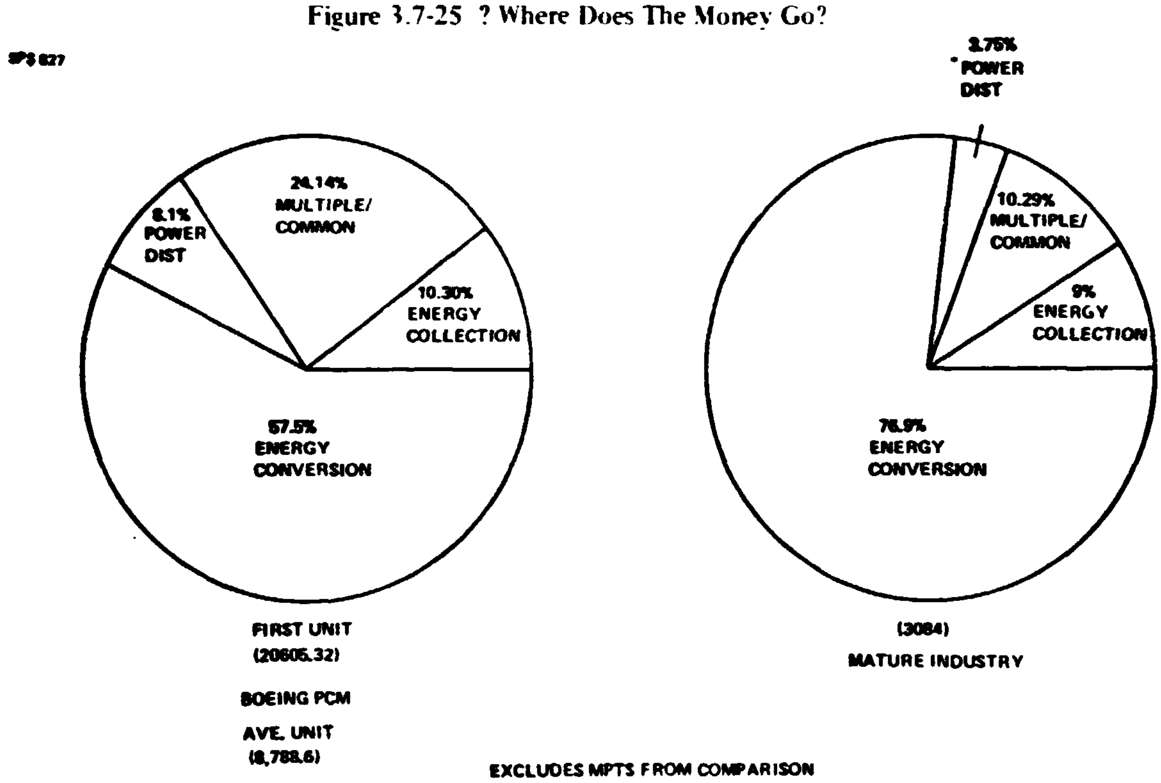

Figure 3.7-26 Costing Methodology Comparison Thermal Engine Satellite

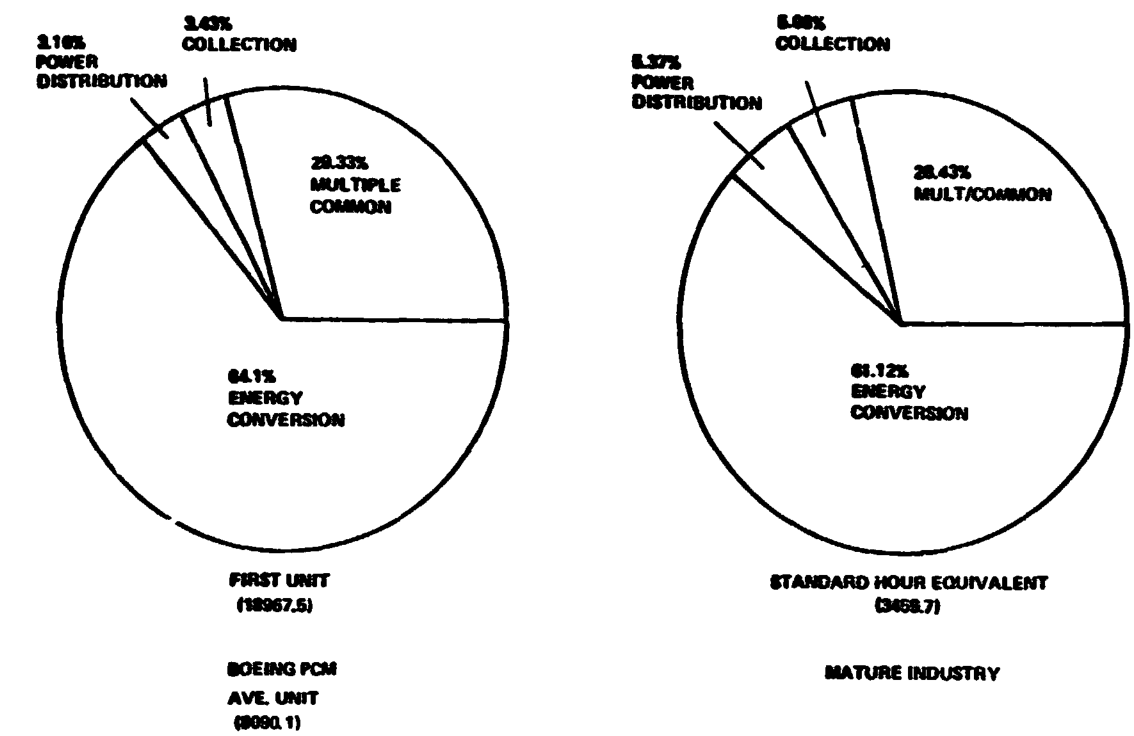

Figure 3.7-27 Costing Methodology Comparison Photovoltaic (Excluding MPTS)

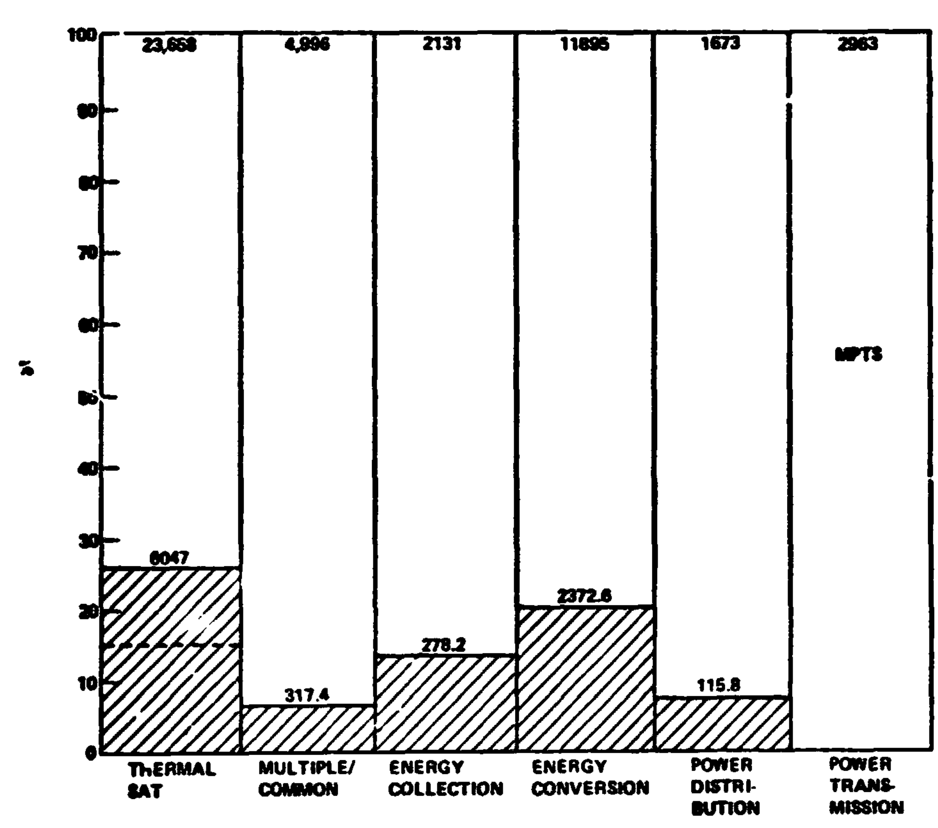

Figure 3.7-28 Aerospace vs Mature Industry Costing Thermal Satellite Production Cost

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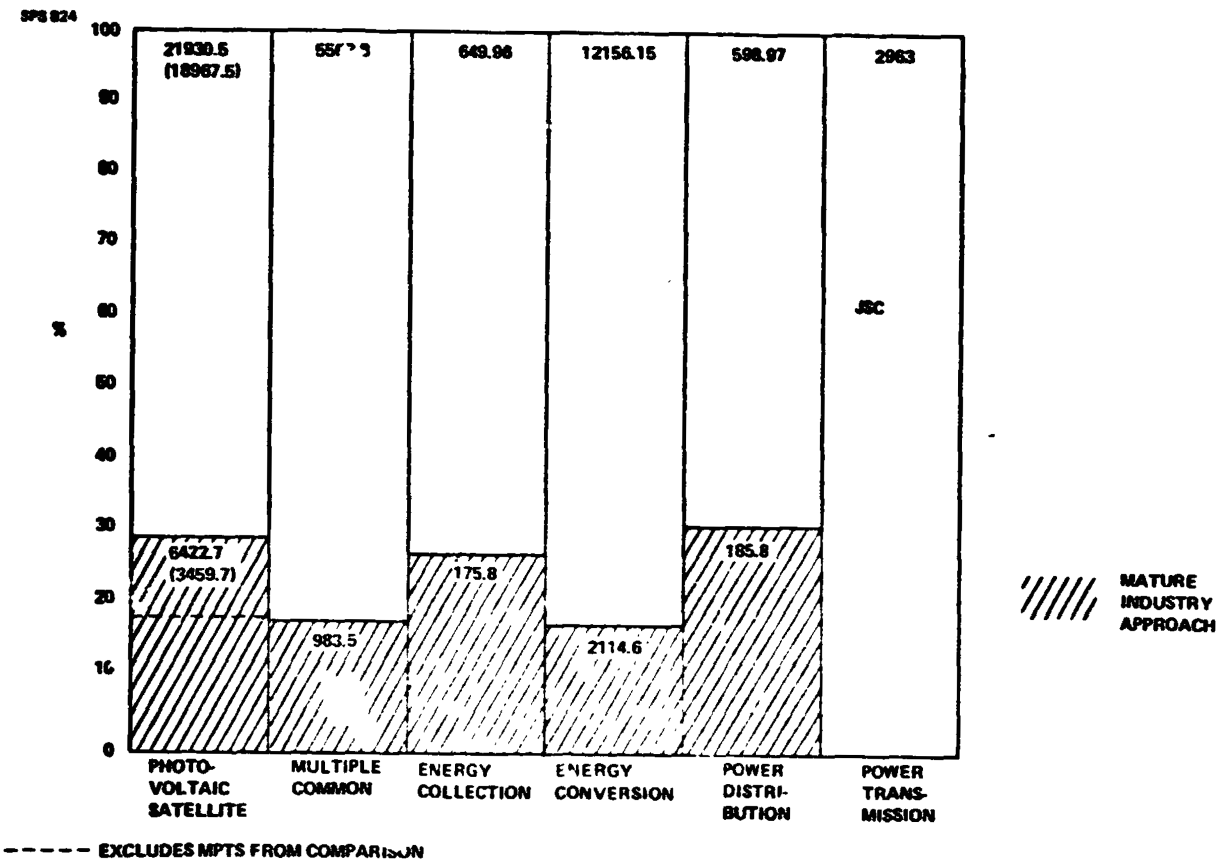

Figure 3.7-29 Aerospace vs Mature Industry Costing Photovoltaic Satellite Production Cost

3.8 RADIATION ENVIRONMENT AND EFFECTS STUDIES

Radiation analyses were conducted to support analyses of SPS degradation in the operational orbit and during low-thrust orbit transfers. A composite crew dose estimate for GEO was also developed.

3.8.1 Environment Analyses

Transfer Orbit Radiation Environment

During transfer from low earth orbit (LEO) to geosynchronous orbit (GEO), the SPS will be exposed to the trapped electron and proton fluxes in the most intense regions of the earth's radiation belts. The equatorial flux of trapped protons, as a function of energy and radius, is shown in Figure 3.8-1 taken from the AP-8 trapped proton environment. The relative intensity of high energy proton decreases with increasing altitude. The trapped electron flux, taken from the AE-4 and AE-5 electron flux maps, is shown in Figure 3.8-2. Although only the electron flux above 0.5 MeV is shown, the electron energy spectrum also becomes softer (in high energy particles) with increasing altitude. The fluxes at 30° inclination are typically 2.3 times lower than the 0° inclination values. Both proton and electron flux maps are provided by the National Space Science Data Center by J. Vette and co-workers.

Transfer Orbit Proton & Electron Dose

The trapped proton and electron dose resulting from a 180 day transfer orbit from 30° inclination LEO to 0° inclination GEO is shown in Figure 3.8-3 and 3.8-4. Spherical aluminum shielding is assumed in Figure 3.8-3 while Figure 3.8-4 shows both spherical and slab shielding values for the trapped proton. Bremsstrahlung from electrons is not included. The total doses are sufficient to preclude manned operation during the low altitude portion of the transfer. Electronic component radiation hardening requirements are implied at doses above 10^3 rads.

Solar & Transfer Orbit Proton Incident Fluence

The integral proton spectrum incident on the SPS during a 180 day transfer from 30° inclination, LEO to 0° inclination GEO is shown in Figure 3.8-5. The solar proton integral proton spectrum typical of an average year near solar maximum during solar cycle 21 is also given. Using the solar proton model of J. King of NSSDC, the 30 year solar proton fluence estimate that has a 90% probability of not being exceeded is obtained by increasing the yearly fluence by a factor of 75 In the region of concern for solar cell degradation, the 180 day transfer orbit proton fluence exceeds the 30 year 90% solar proton fluence by an order of magnitude.

SPS 1133

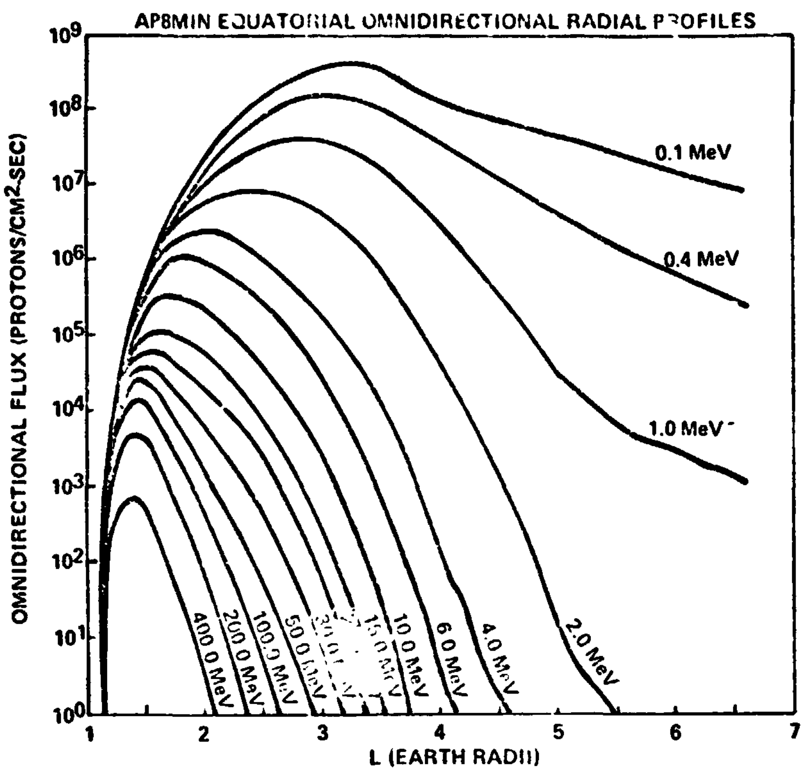

Figure 3.8-1 AP8MIN Equatorial Radial Profiles with Energies Between 0.1 and 400 MeV

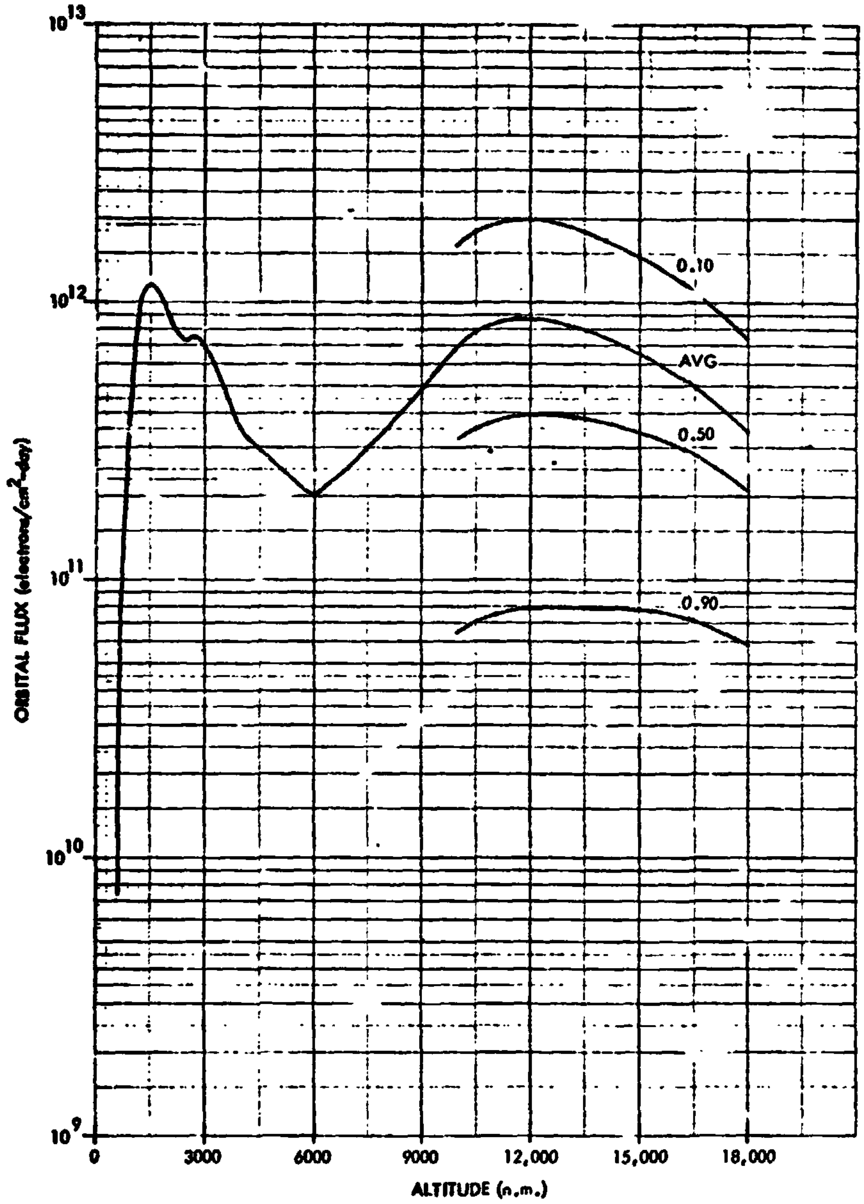

Figure 3.8-2 Transfer Orbit Radiatior Environment

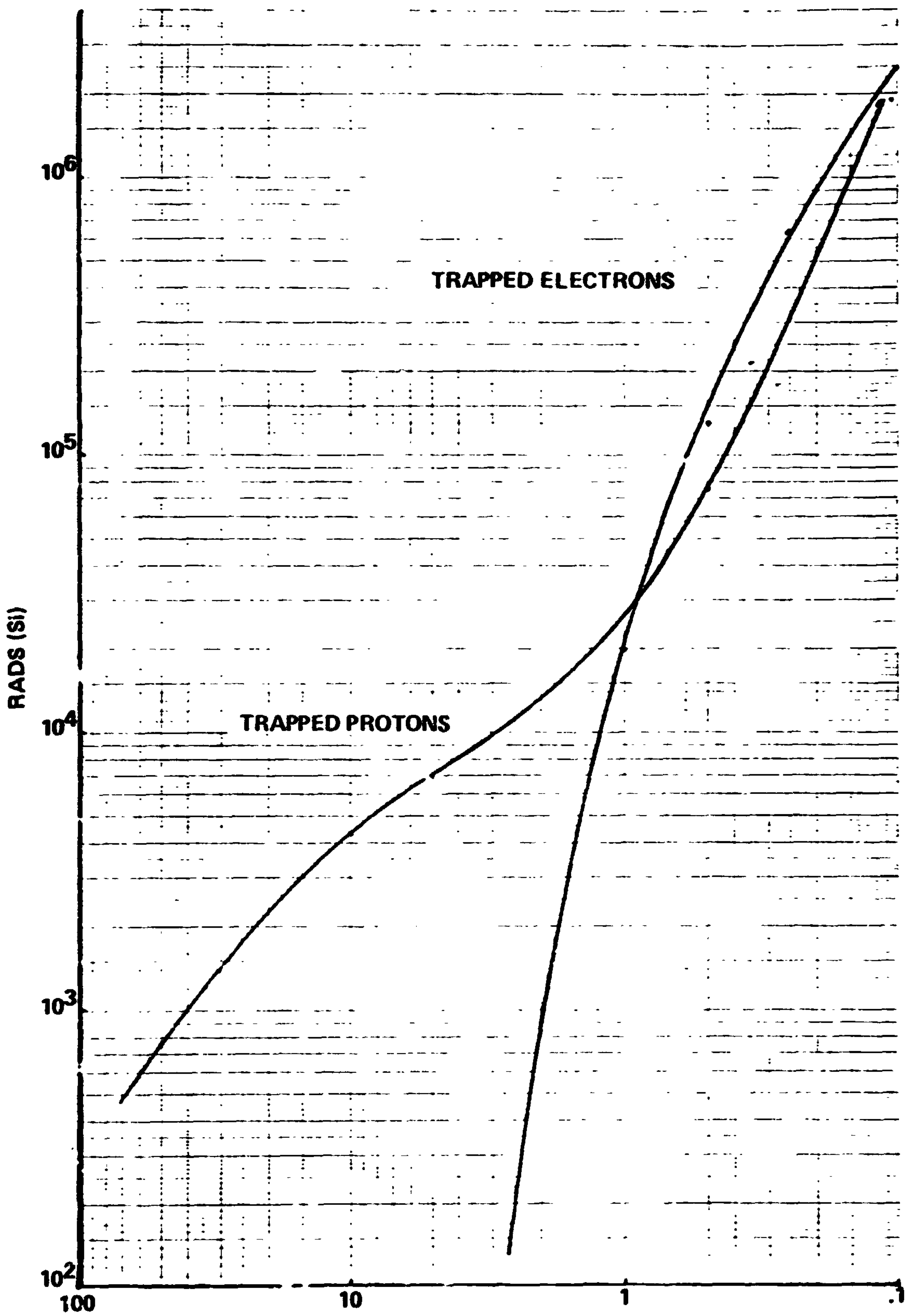

Figure 3.8-3 Transfer Orbit Proton and Execution Pool

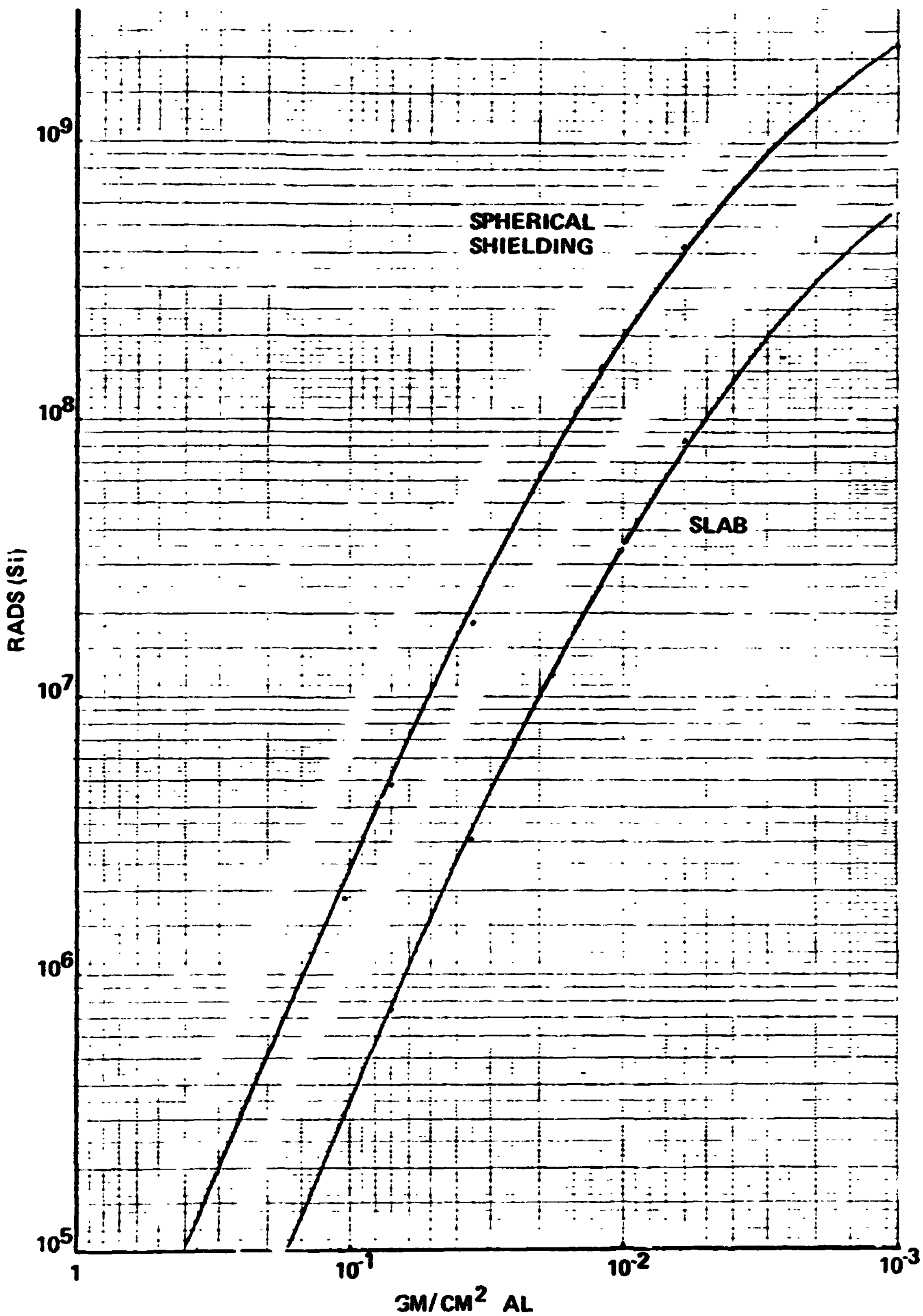

Figure 3.8-4 Transfer Orbit Trapped Proton Pose

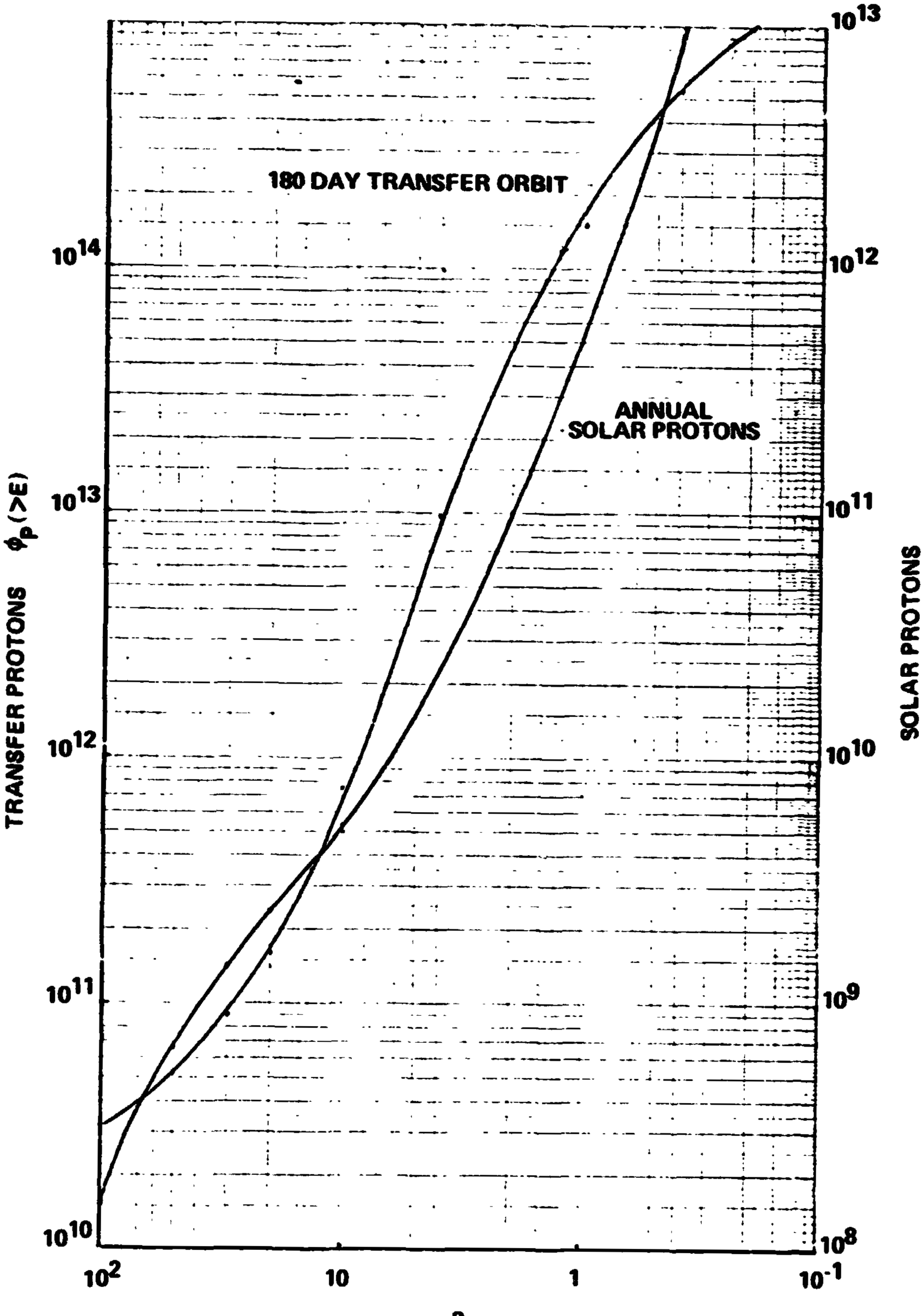

Figure 3.8-5 180 Day Transfer and Annual Solar Proton

Geosynchronous Environment

The composite total dose at synchronous altitude shown in Figure 3.8-6 is the result of galactic cosmic rays, solar protons and trapped electrons. For a solar proton intensity corresponding to three solar cycles similar to cycle 21, and with the 90% model of J. King, the solar proton dose is shown to dominate over long time periods between 1 and 10 g/cm^2 of aluminum. The electron dose will dominate below 1 g/cm^2, with the electron Bremsstrahlung important above 10 g/cm^2. The galactic proton dose provides a penetrating low level radiation background. The high Z cosmic rays present a separate radiation problem, one not described by the concept of absorbed dose.

3.8.2 **Degradation Analyses**

Reflector Degradation

The reflector degradation during the transfer orbit is calculated from Project ABLE data and the low energy proton fluence versus time dependence during ascent. The degradation as a function of time is shown in Figure 3.8-7 for three transfer orbits, 180, 90 and 75 days.

GEO Solar Proton Degradation

The solar proton models developed by J. King are based on data taken at 10 MeV and greater. The extrapolation of this model to the lower energies of importance to SPS can be based either on an exponential rigidity model, as was done by JPL for the Halleys Comet mission or on a power law expression as found by W. R. Webber and used for INTELSAT. These two methods lead to importance differences at the cover slip thicknesses of interest. We have used the conservative degradation values, but the importance of this assumption on solar cell degradation is obvious, as is shown in Figure 3.8-8.

Transfer Orbit Damage Gradients

The displacement damage gradient in silicon resulting from the transfer proton spectrum increases rapidly with decreasing cover slip thickness as is shown in Figure 3.8-9. As the solar cell degradation expressed in terms of 1-MeV equivalent electrons CM^{-2}, follow this damage gradient closely, it is apparent that the abundant low energy protons encountered in the transfer orbit make cover slip thickness a very parameter in determining solar cell degradation for conventional criterion solar cells.

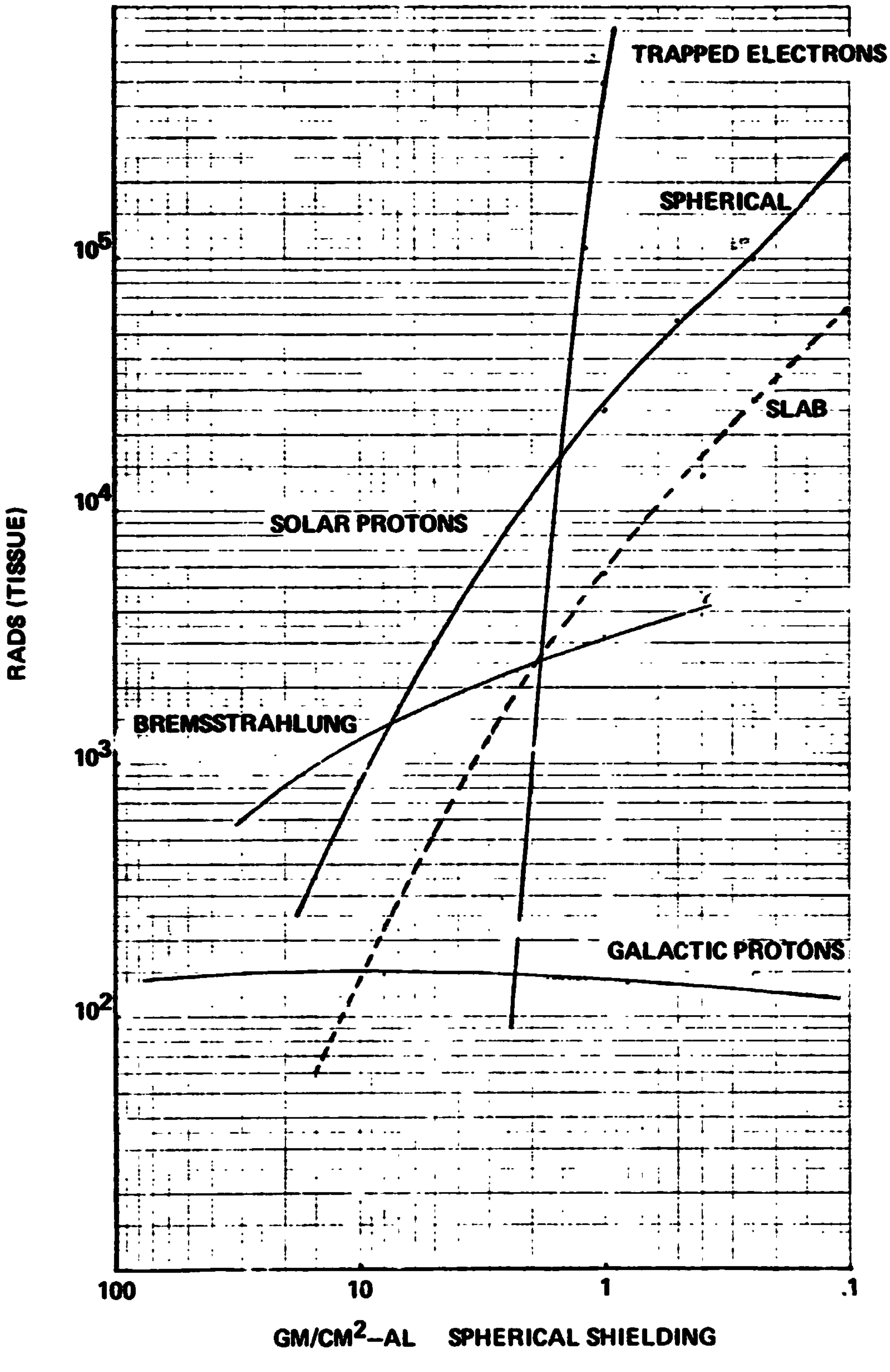

Figure 3.8-6 30 Year GEO Total Dose Solar Protons, Galactic Protons, Trapped Electrons

SPS-1134

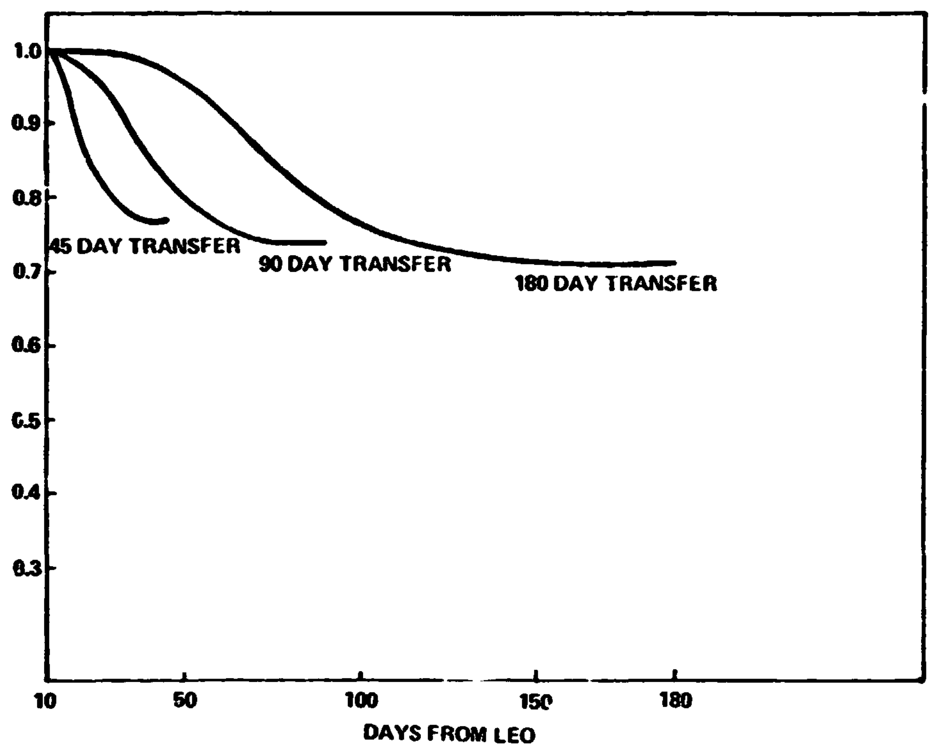

Figure 3.8-7 Transfer Reflector Degrapation

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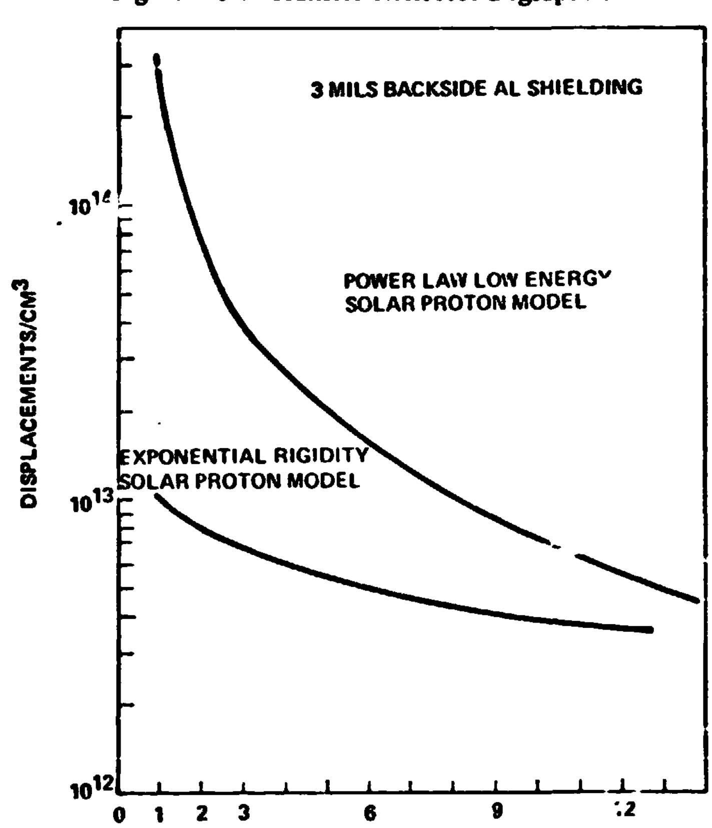

Figure 3.8-8 Solar Proton Degradation Gradient

SPS-1136

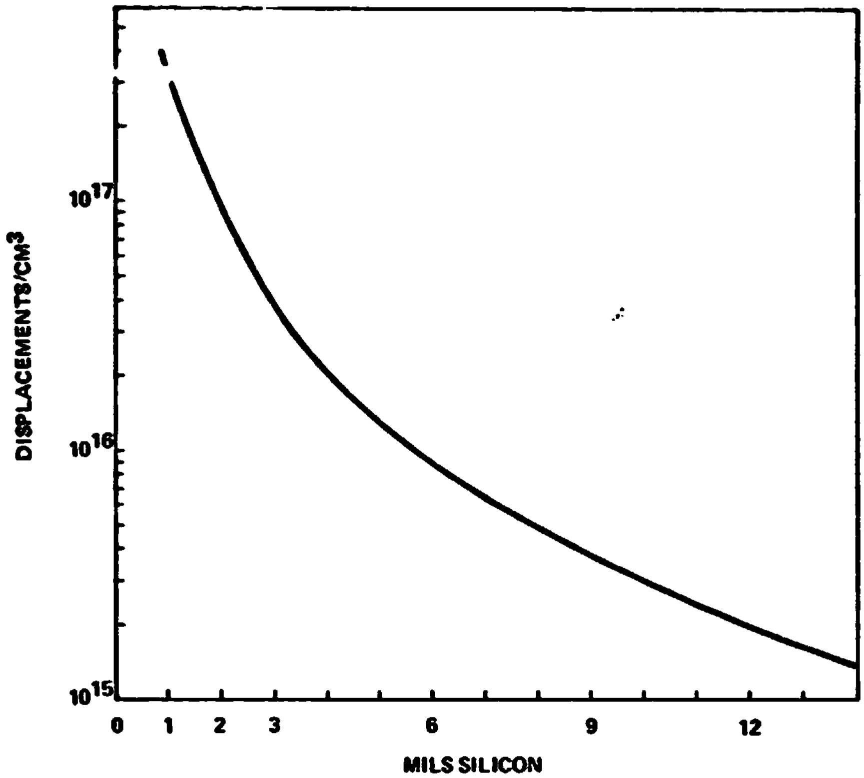

Figure 3.8-9 Transfer Proton Displacement Damage Gradient

Transfer Orbit Solar Cell Degradation

Parametric values of solar cell degradation are shown in Figure 3.8-10 for transfer orbit time and cover slip thickness assuming 6 mil n/p 10 ohm-cm cells. The conversion of displacement damage to 1-MeV electron fluence was based on the results of NASA and AF funded studies of solar cell degradation which produced a reasonable calculational method for low energy proton degradation. Due to the importance of low energy protons damage for these cover slips, the degradation estimates depend critically on the low energy proton damage evaluation method used. Experimental verification would be desirable for this environment.

SPS-1137

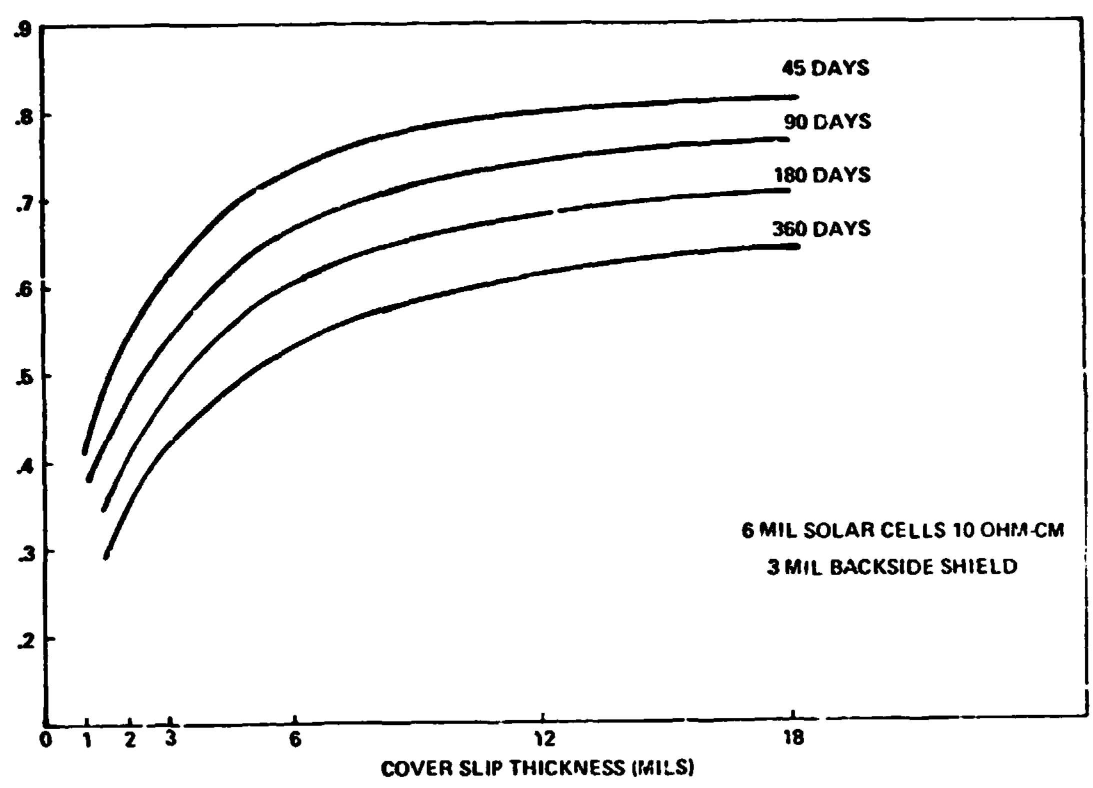

Figure 3.8-10 End of Transfer