GAC-NSS-P-75-001

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A31640

Space-Based Solar Power Conversion & Delivery Systems (Study)

Engineering Analysis

NAS8-31308
GAC No. NSS-P-75-001

Space–Based Solar Power Conversion & Delivery Systems (Study)

Engineering Analysis

Prepared by:
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For:
ECON, Inc

Aug. 6, 1975

Space-Based Solar Power Conversion and Delivery Systems

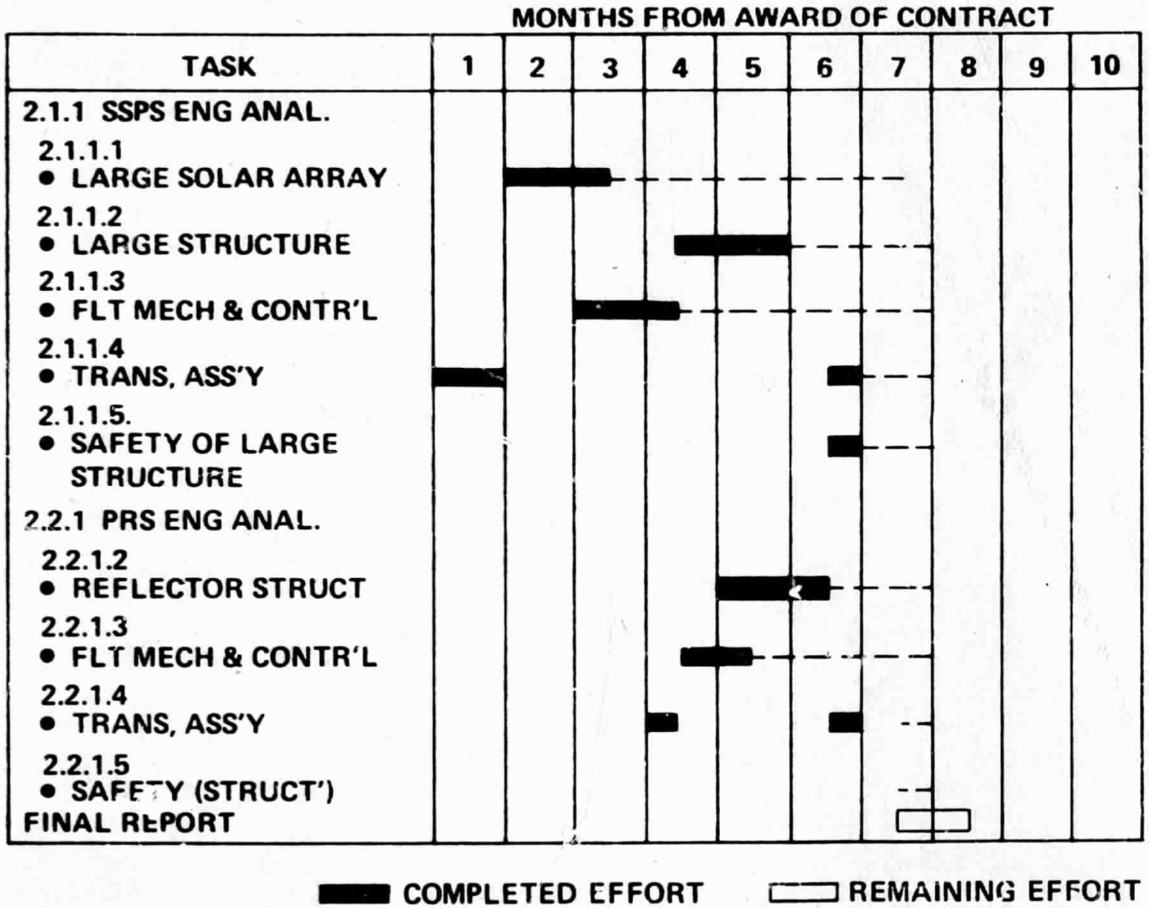
GAC Study Status

The objective of this study is an in-depth systems analysis of synchronous, orbit-based power generation and relay systems that could be operational in the 1990's and a comparison with earth-based systems to be operational in the same time frame.

Grumman's effort represents approximately 20 percent of the study and is meant to concentrate on the Engineering Analysis of special requirements for both the SSPS and PRS. Grumman's objectives are to: identify operational and economic requirements for the orbiting systems; and to define near-term research activities which will be required to assure feasibility, development, launch and operational capabilities of such systems in the post-1990 time frame.

The facing page is a status of task completion. We have completed all assigned tasks under Engineering Analysis of Special Requirements and are currently supporting ECON's cost/trade studies. The remaining task to be completed is the compilation of the final report.

SPACE BASED SOLAR POWER CONVERSION & DELIVERY SYSTEMS – GRUMMAN STUDY STATUS



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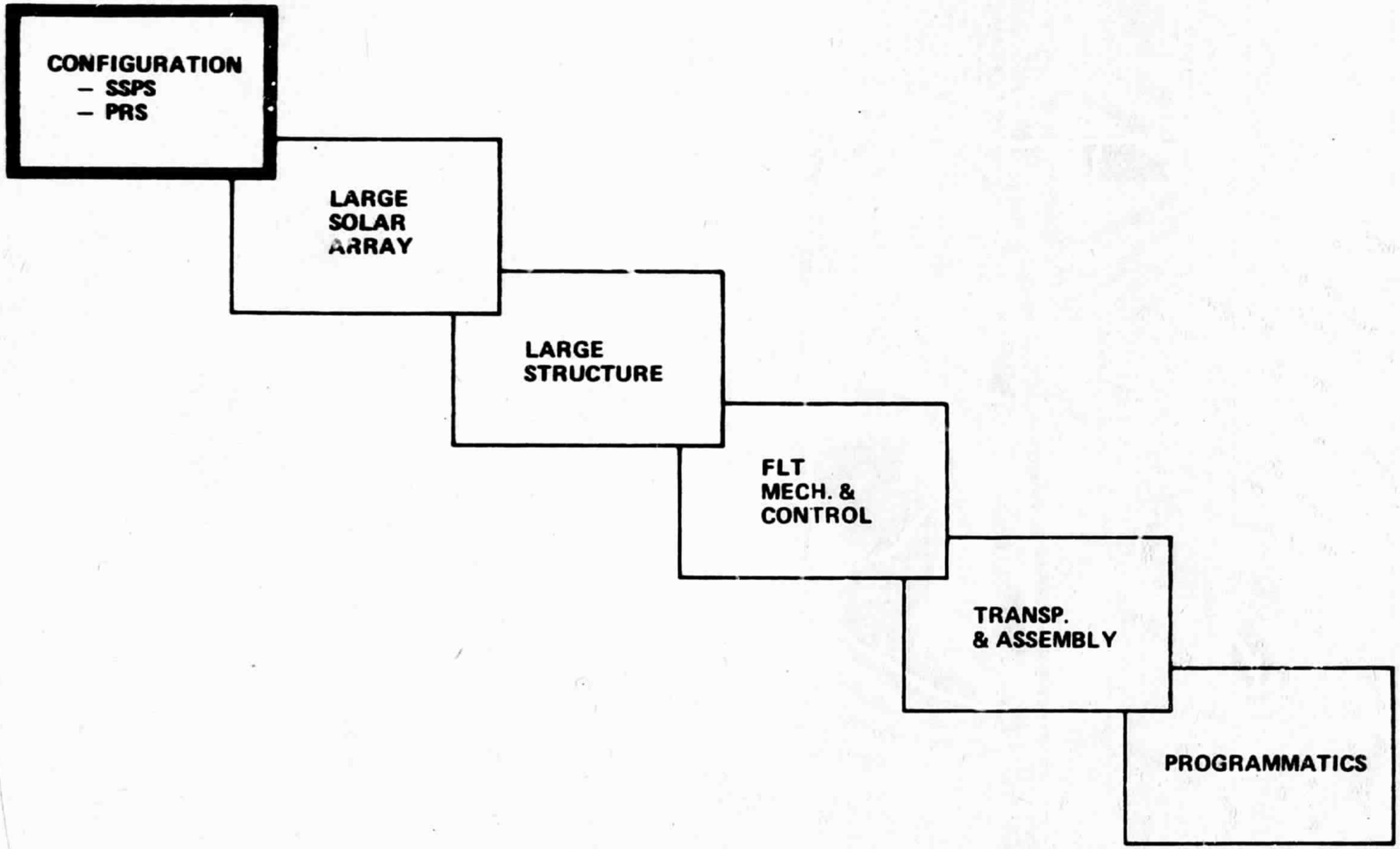
CONFIGURATION
– SSPS
– PRS
LARGE SOLAR ARRAY
LARGE STRUCTURE
FLT MECH. & CONTROL
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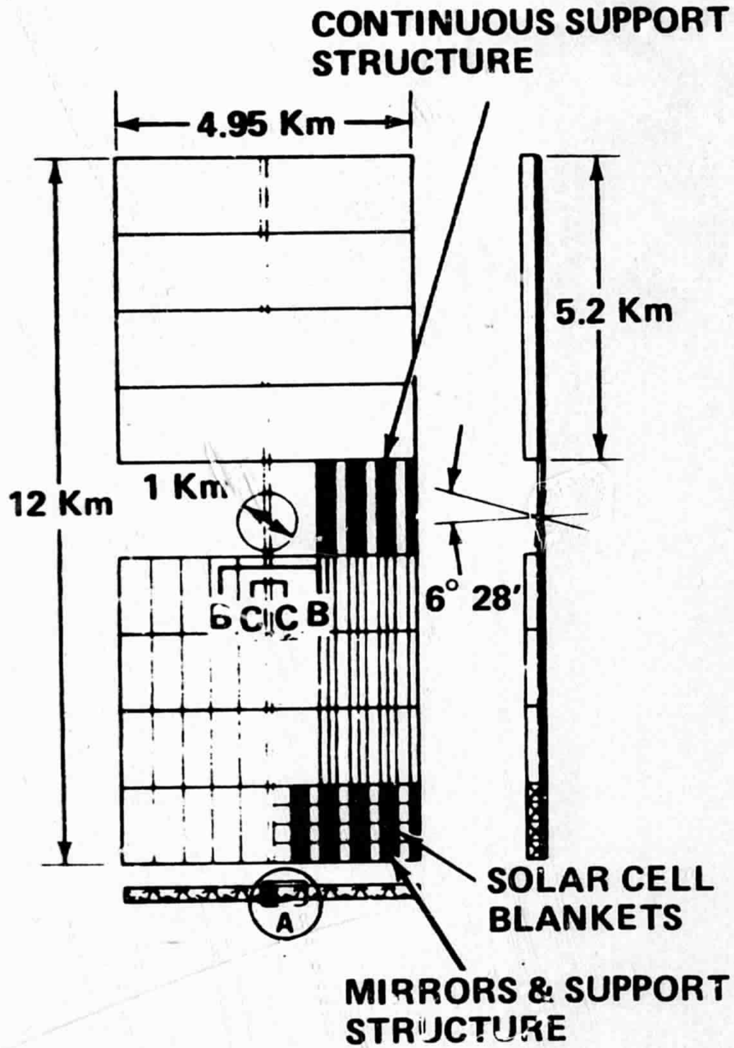
SSPS Configuration

The facing page shows the basic spacecraft design used in the assessment of transportation and assembly options. Light-weight photovoltaic arrays are enhanced by concentrators to reflect sunlight onto the solar cells. Solar power is directly converted to high voltage d-c electricity at the two symmetrically arranged arrays. Bus bars feed the 1-km transmitting antenna, located between the two large solar arrays, where generators convert the energy for power transmission to earth.

This system is sized for 5000 MW rectified power on the ground. The solar cell blankets are layed out between channel concentrators, consisting of thin, reflective plastic films stretched over a supporting frame. The backbone of the structural framework is a large-diameter coaxial mast which runs the length of the entire assembly. Eight transverse structural beams also serve as d-c power buses to carry electrical current to the central mast. The array structure is stiffened using a series of transverse non conducting elements and large open trusses to support the concentrators.

SSPS CONFIGURATION



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SSPS Mass Properties

The facing page presents the SSPS mass properties. The solar array represents 67% of the system weight with the solar blankets being the major contributor at 7.83×10^6 kg. The solar cell blankets are of advanced design with an efficiency of 13.7% at a concentration ratio of 2. These array blankets weigh .282 Kg/M^2 and operate at 20 kv.

The transmitting antenna represents 32% of the total weight. These weights are consistent with the latest antenna weights baselined by Raytheon in the Microwave Power Transmission System (Studies) - NAS3-17835. The major weight contributors are the microwave conversion tubes and the transmission system (waveguides).

This satellite was sized assuming a microwave system efficiency of 57% (antenna input to rectified power on the ground). A 10% reduction in array collection efficiency was assumed to take into account variations in the sun's normal component to the solar blanket throughout the year.

SSPS MASS PROPERTIES

SUBSYS/COMP	WEIGHT			
	Kg X 10^6		LBM X 10^6	
SOLAR ARRAY		(11.90)		(26.21)
• BLANKETS	7.83		17.25	
• CONCENTRATORS	1.23		2.71	
• NONCONDUCTING STRUCT	1.98		4.36	
• BUSSES, SWITCHES	0.24		0.53	
• MAST	0.62		1.37	
MW ANTENNA		(5.70)		(12.56)
• MW TUBES	2.34		5.15	
• POWER DISTRIBUTION	0.52		1.15	
• COMMAND ELECT.	0.13		0.29	
• TRANSMISSION	2.32		5.11	
• STRUCTURE	0.257		0.57	
• CONTOUR CONTROL	0.13		0.29	
ROTARY JOINT		(0.20)		(0.44)
• MECHANISM	0.08		0.18	
• STRUCTURE	0.12		0.26	
CONTROL SYSTEM		(0.036)		(0.08)
• ACTUATORS	0.012		0.03	
• PROPELLANT/YR	0.024		0.05	
TOTAL SYSTEM	17.84		39.29	

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PRS Mass Properties

The facing page is a compilation of PRS orbiting system weights. The PRS configuration shown consists of a primary structure with 25-meter deep truss girders spaced at 108 meters. Each 108-meter module is spanned by an 18-meter grid of 5-meter depth. At the corners of the 18-meter modules are electrically driven screw jacks to which are mounted the reflector surface.

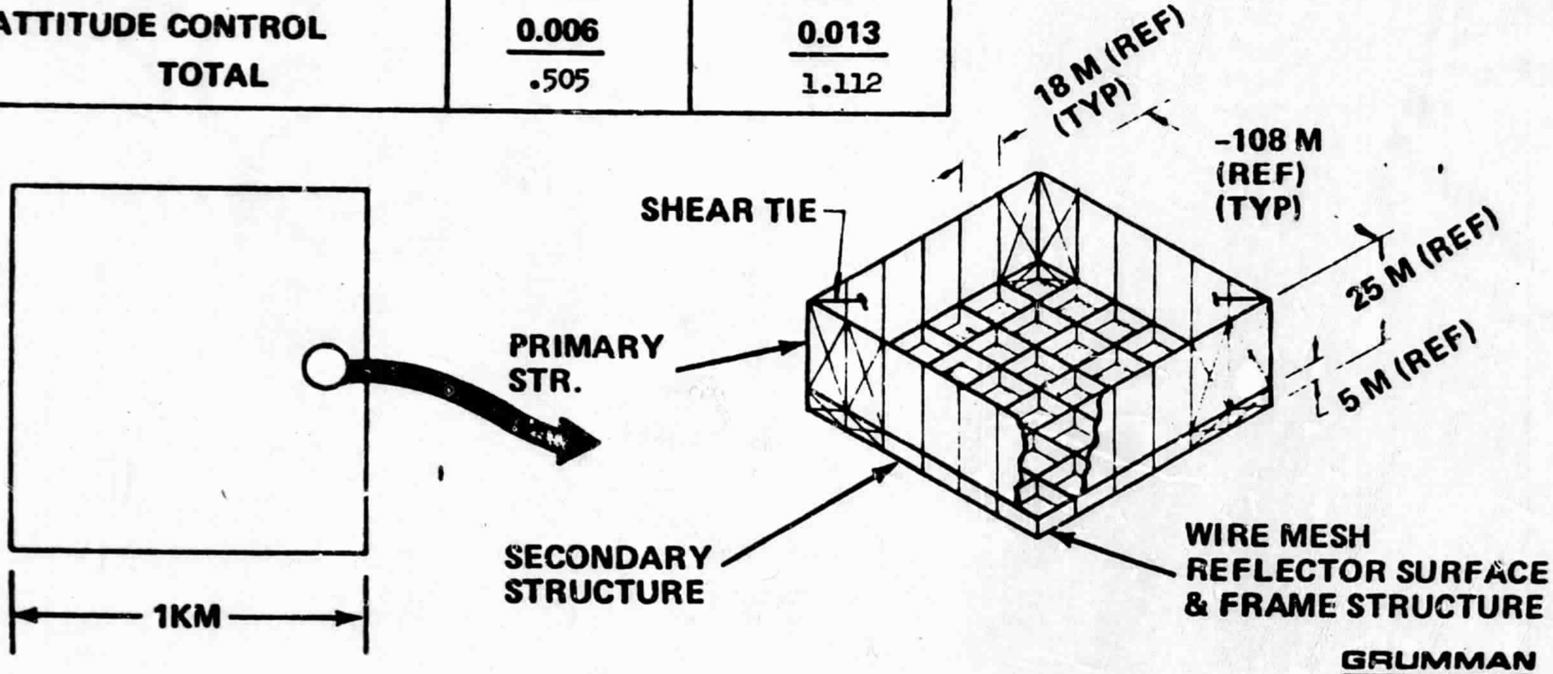
The primary structure is built up of 108m x 108m x 20m deep bays; the upper cap consists of a triangular truss girder 108m long by 3m deep. The material used is is a graphite composite. The secondary structure, which forms the lower cap of a primary bending structure, is 5 m deep in 18m x 18m square bays. The secondary structure is also fabricated of a hybrid composite graphite/epoxy boron epoxy.

The aluminum wire mesh reflector surface is mounted to the supporting frame with pretensioned springs. The tension magnitude was selected to maintain surface smoothness to 1/20 of a wavelength through wide variations in thermal conditions (+200°F to -250°F) under a 5.65 kp/km^2 microwave pressure.

The mechanical contour control system positions the 18 x 18m wire mesh subarrays to achieve beam focus.

PRS MASS PROPERTIES

SUBSYS/COMP.	WEIGHT	
	Kg X 10^6	LBM X 10^6
• PRIMARY STRUCT	0.119	0.262
• SECONDARY STRUCT	0.038	0.084
• COATINGS & INSULATION	0.028	0.062
• FRAME STRUCTURE	0.101	0.222
• WIRE MESH	.058	.127
• CONTOUR CONTROL	0.155	0.341
• ATTITUDE CONTROL	0.006	0.013
TOTAL	.505	1.112



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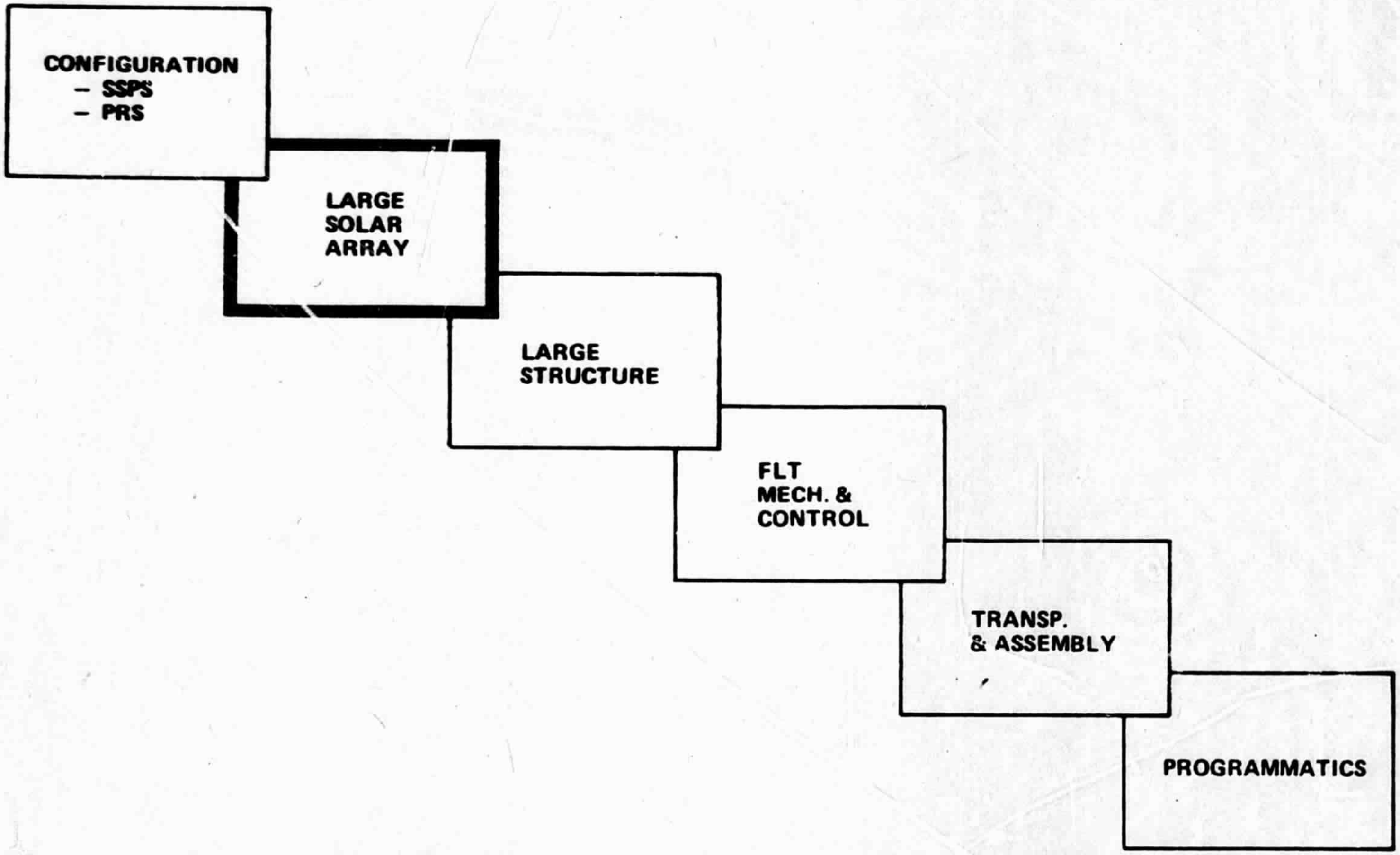
CONFIGURATION
– SSPS
– PRS
LARGE
SOLAR
ARRAY
LARGE
STRUCTURE
FLT
MECH. &
CONTROL
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& ASSEMBLY
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Selection of Concentration Ratio

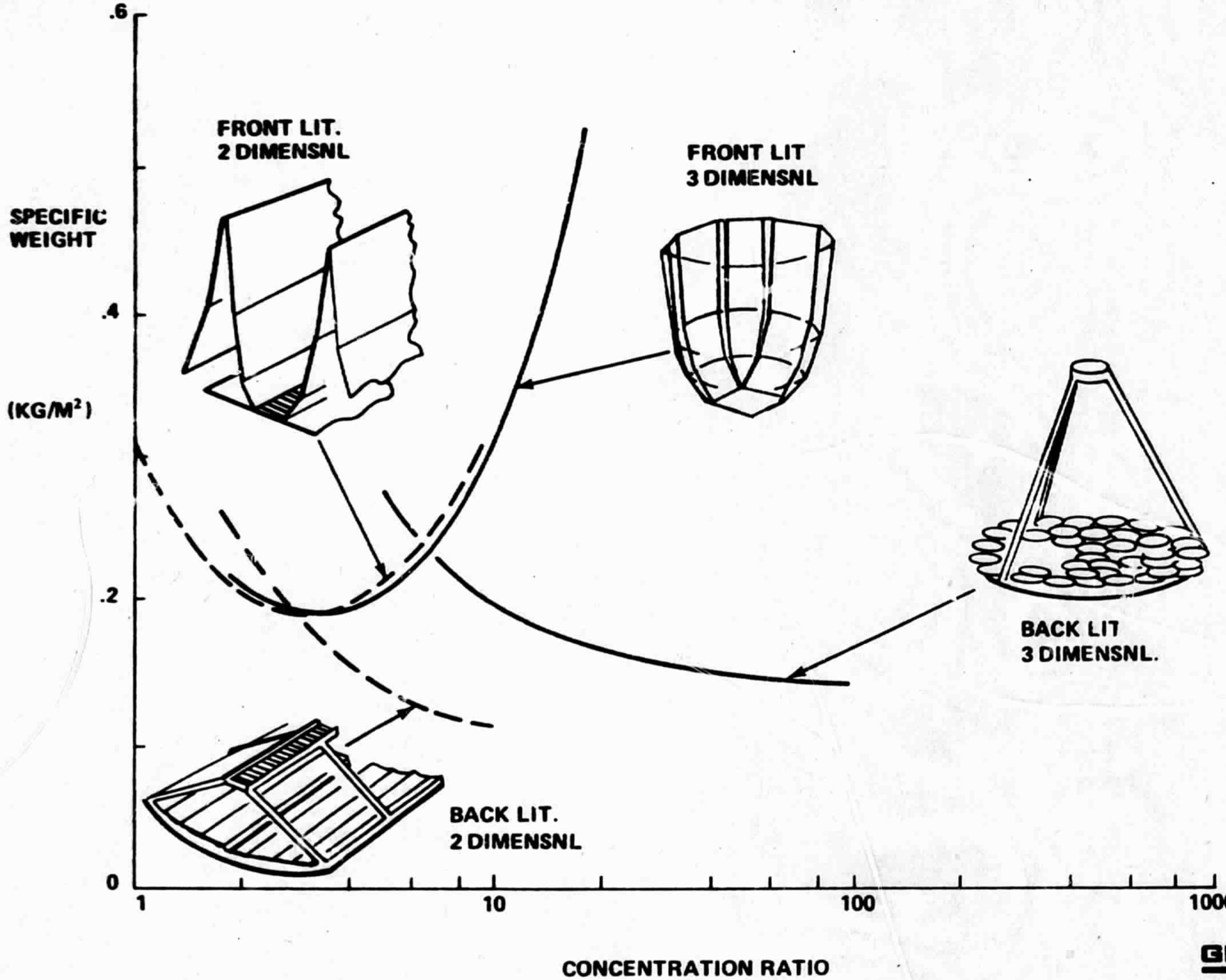
A key systems issue to be addressed prior to selection of the solar array configuration is to delineate the preliminary analysis of concentration ratio shown on the facing page. This preliminary analysis was performed to determine weight variations due to structural arrangement. It assumes that solar cell efficiency does not vary with increased concentration and does not consider the thermal control system weight to achieve constant efficiency.

Though the model used in this analysis is simplified, the results do indicate a trend. Back lighted arrangements result in lower specific weight for concentration ratios above 2. A two-dimensional back-lighted design is lighter than a three-dimensional design, though the pointing requirements may prevent achieving concentration ratios above 10. The parabolic back-lit design is the most attractive for high concentration, though if the degradation in cell efficiency and increased weight for thermal control were added to the parametrics, the resulting design may not be lower in weight and cost than the SSPS baseline.

An across-the-board design analysis is needed to fully consider:

- Concentration Ratio
- Cell efficiency with increased temperature
- Thermal Control
- Pointing Control
- Transportation and Assembly Cost.

SELECTION OF CONCENTRATION RATIO



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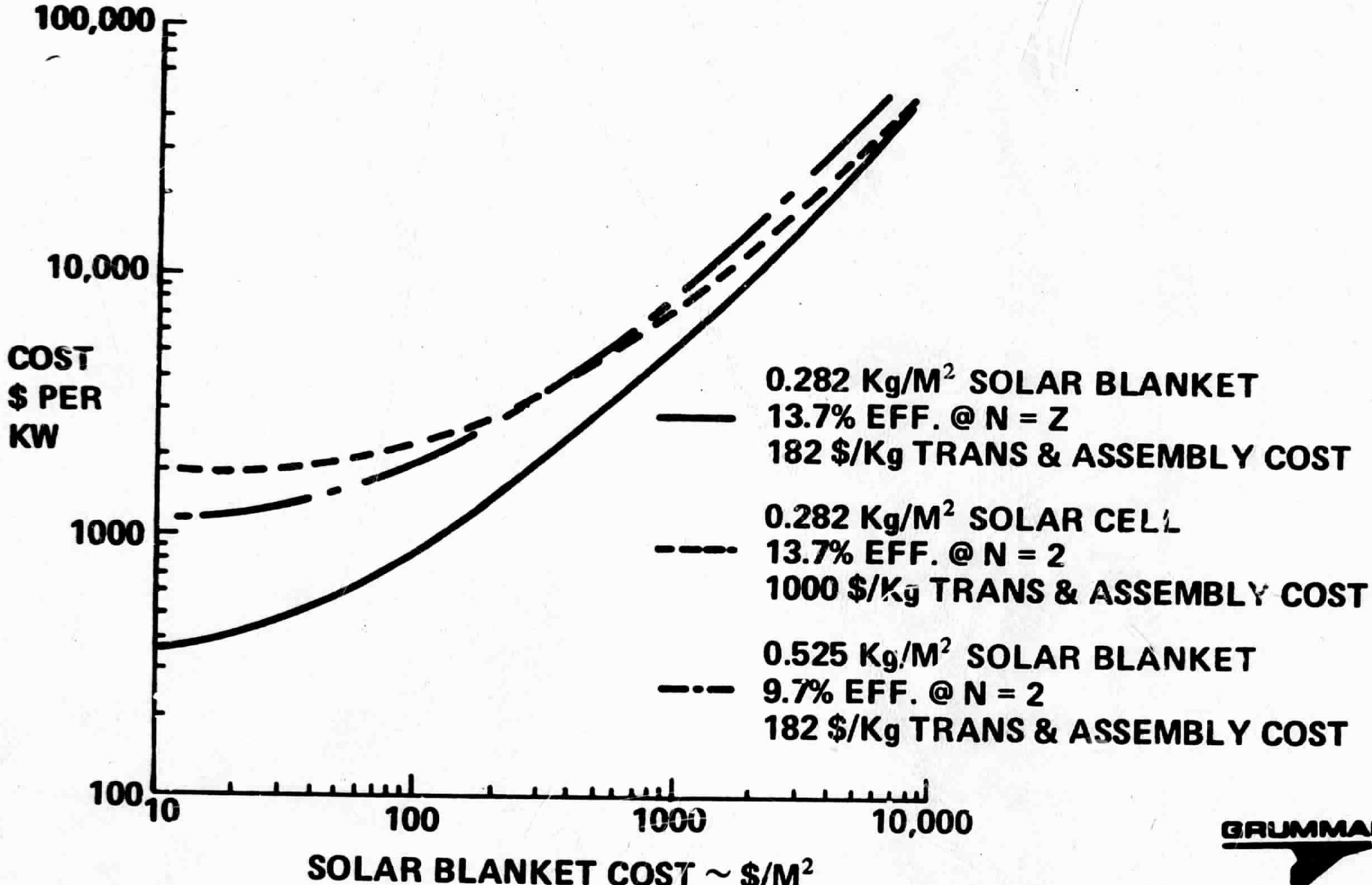
Solar Array Design Cost Trade

The facing page presents the trends in a major solar array design/cost trade. Solar Array Costs ($/KW) are plotted against variations in solar blanket costs, solar blanket weight/efficiency and transportation/assembly costs. The solid line represents the SSPS goal for efficiency (13.7% @ N = 2), weight (0.282 Kg/M^2) and transportation/assembly cost (182 $/Kg). The dashed line shows the effect of an increase in transport/assembly costs to 1000 $/Kg; while the dashed/dot line represents near-term technology solar blanket weight (0.525 Kg/M^2) and efficiency of 9.7% @ N = 2 at a transport cost of 182 $/Kg.

Significant improvements in solar cell performance and design as well as low-cost transportation are required to achieve a cost effective blanket in the region of $55/$m^2$, the national goal for solar blanket costs. A more in-depth assessment of these trade parameters should be undertaken including evaluation of the development costs required to achieve the desired goal. More dollars spent on low cost, ultra light, space qualified solar arrays may be a better investment than development of new transportation systems. If the solar blanket technology programs cannot feasibly achieve the performance goals, development of low-cost assembly and transport systems would be the better investment.

SOLAR ARRAY DESIGN COST TRADE



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Key Issues - Large Solar Array

Investigations into meth ds to improve cell efficiency are important to meeting SSPS goals. The efficiency must increase from about 14% (N = 1) to 18% (N = 1) while reducing thickness of the device from 200µm to 50µm. These investigations should include experimental development of new conversion devices such as the heterojunction Ga Al As/GaAs cell.

The need to reduce cell fabrication cost is critical to SSPS. Large quantity production will naturally help reduce cost but additional cost savings can be achieved with new crystal growth processes and new cell fabrication techniques that are automated.

Large solar arrays that can be effectively handled is key to SSPS. Presently, solar arrays are made much like an art mosaic, where individual cells are fitted, interconnected, and bonded to substrate. Improvements can be achieved by developing light-weight blanket attachment techniques, light-weight structural weaving techniques, new thermal control coatings, improved radiation-resistant materials and better automated techniques for integrating and testing the blanket. Automated blanket fabrication techniques are needed to reduce cost.

Solar concentration is shown to reduce SSPS cost. Light-weight mirror design concepts and their implementation are needed. New filter designs for concentrations will help improve solar cell life and performance. If high concentration is used, techniques for fabricating light weight structure and contour control are needed.

The SSPS will generate high voltage power in a relatively stable thermal environment but must maintain performance during a 30 yr exposure to ultraviolet radiation as well as articulate radiation. The objective is 6% degradation over 5 years.

Improvements in environment resistance can be achieved by improved material, radiation spectral tailoring, high-voltage plasma protection, meteroite hardening and improved annealing techniques.

Multi-megawatt solar power generation requires switching protection at high voltage and current. Development of high voltage switches and blocking devices are needed. Circuit design must consider induced magnetic moments to reduce effects on the overall spacecraft control. The high voltage also could lead to corona formation that could reduce component life. The power distribution system designer must address long transmission distances on SSPS. A key trade is to determine the extent to which the conducting busses can also be used as structure. A trade-off between ease of assembly, cost, weight, reliability, and electrical efficiency should be addressed.

KEY ISSUES – LARGE SOLAR ARRAY

- SOLAR CELL PERFORMANCE IMPROVEMENT
- SOLAR CELL COST REDUCTION
- SOLAR ARRAY BLANKET IMPROVEMENT
- SOLAR ARRAY BLANKET COST REDUCTION
- SOLAR CELL CONCENTRATION TECHNIQUE
- LONG LIFE OF SOLAR ARRAY IN SPACE
- HIGH-VOLTAGE, HIGH POWER SWITCHING
- HIGH-VOLTAGE CIRCUIT DESIGN & HIGH LEVEL DC POWER DISTRIBUTION
- SOLAR BLANKET ASSEMBLY IN-ORBIT.

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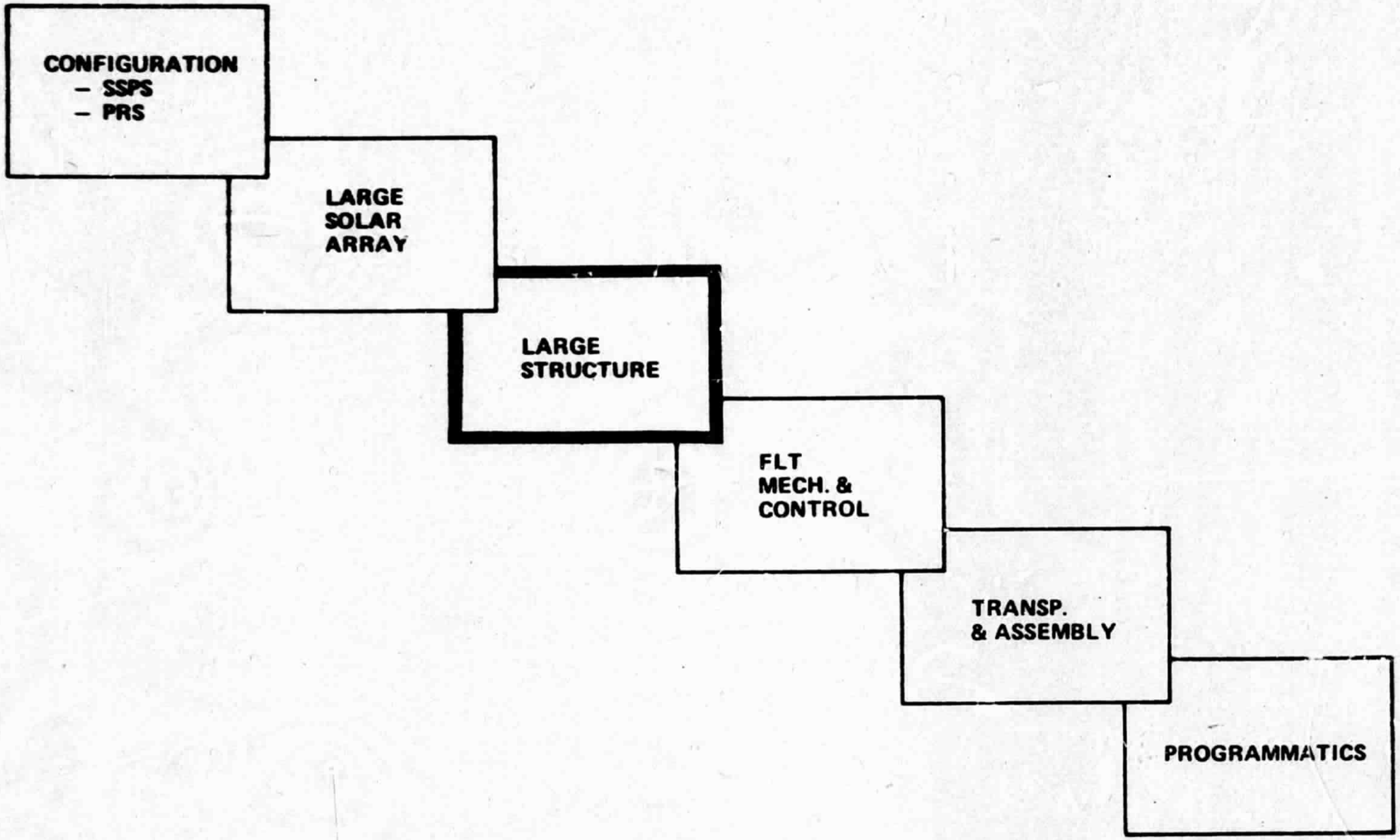
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Solar Array Non-Conducting Structure

The facing page summarizes the solar array structural arrangement and weights. The primary structural element is a truss girder built-up from roll formed modified vee hat sections with bent up stabilizing angles at the outstanding legs. The basic structural member was designed as a 1.5-meter deep truss girder. This member is used to build either a 223-meter or 433-meter girder.

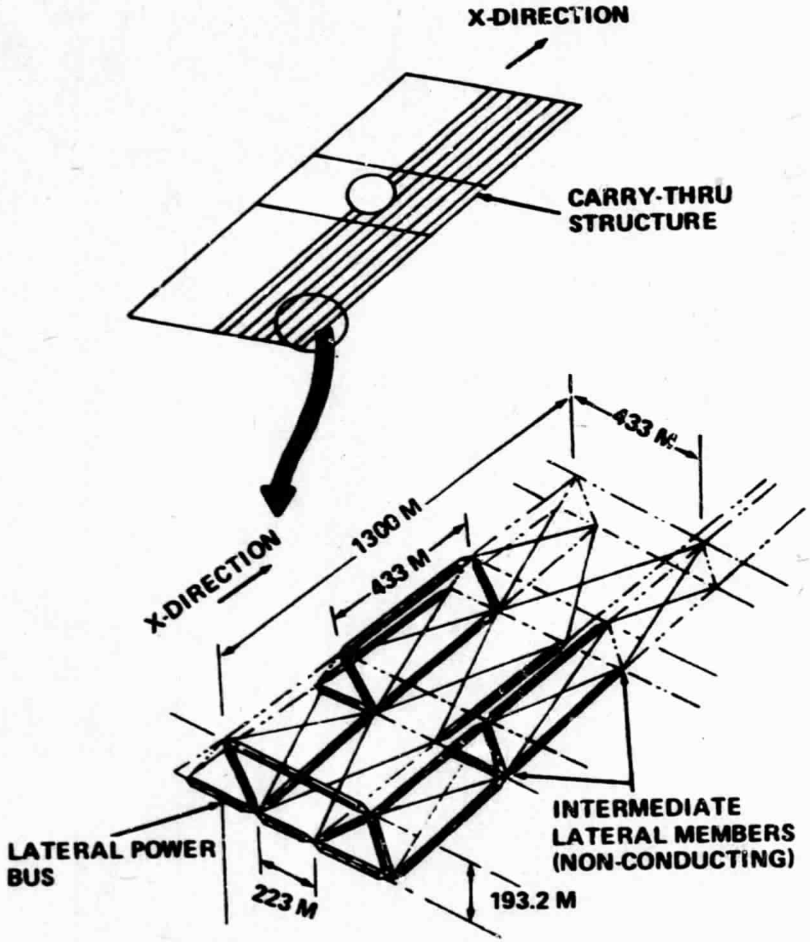
The structural members are designed for a limit control force at each array tip, of 670 lbs times a factor of safety of 1.50. A peak 136 lb ultimate compression load was used to size the aluminum cross-section.

Pretension forces in the mirrors and solar blankets were combined with the axial compression loads to assess the beam column strength of the 433-meter longitudinals. The total weight of all non-conducting structure was calculated at 4.29×10^6 LB, including 10% non-optimum and contingency factors.

The major uncertainty in designing the array non-conducting structure is definition of the design load. The indicated 670 LB control force used to size the structure is arbitrary. This force is an ROM estimate of the thruster size required for finite burn stationkeeping maneuvers. An across-the-board assessment of all loading conditions is required, including:

- Ground handling loads
- Launch loads
- Assembly loads
- Synchronous Orbit Transport loads
- Operational loads
- Thermal induced loads.

SOLAR ARRAY NON-CONDUCTING STRUCTURE



ARRAY

CHORDWISE MEMBERS (ALUMINUM) (1.499×10^6 LB)

MEMBER	NUMBER	WGT/MEMBER LB	WGT LB X 10^6
223 M	520	1360	0.71
433 M	260	2643	0.687
193 M	52	1231	0.064
153 M	39	975	0.038
LONGITUDINAL MEMBERS (ALUMINUM) (2.093×10^6 LB)			
433 M	792	2643	2.093
CARRY THROUGH STRUCTURE (GLASS) (0.31×10^6 LB)			
CHORDWISE MEMBERS			
223 M	24	1360	0.033
433 M	12	2643	0.032
193.2 M	8	1231	0.010
LONGITUDINALS			
433 M	72	2643	0.190
BRACING			0.043
SUBTOTAL			3.900
10% NON OPTIMUM FACTOR			0.39
TOTAL			4.290

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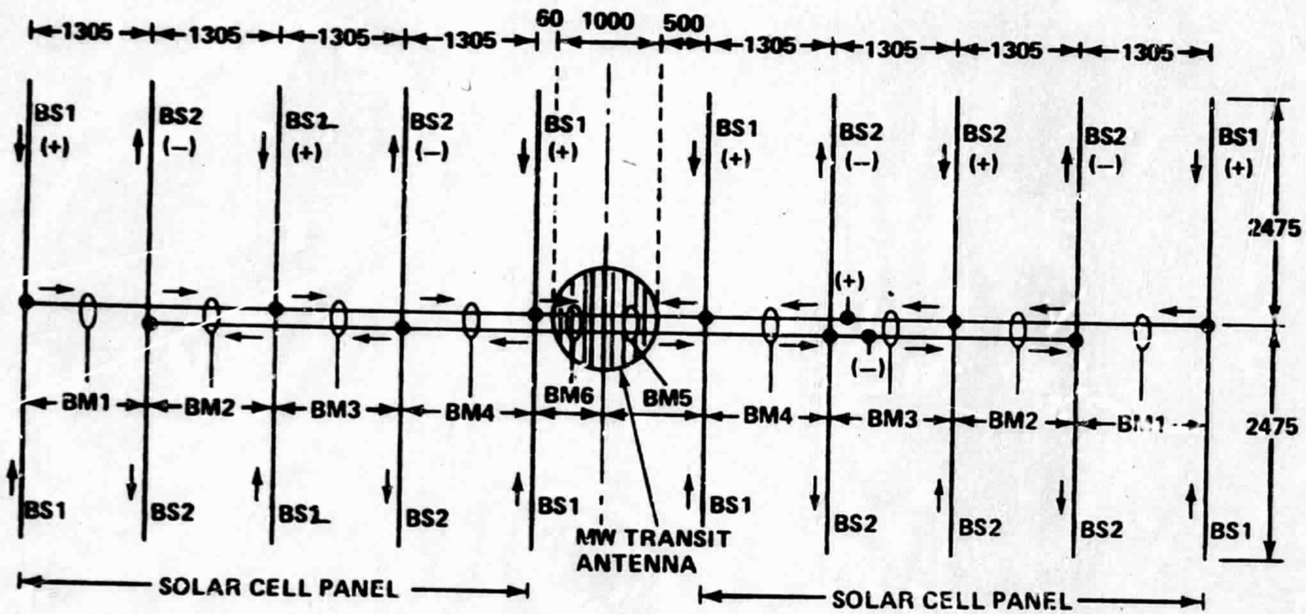
SSPS CONDUCTING STRUCTURE – 1
ELECTRICAL BUSES GRID CONFIGURATION



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SSPS Conducting Structure

Because of the large amount of conducting material required to collect the electrical power generated by the solar blankets and transmit it to the microwave antenna, the bus material has been integrated into the structure. Forces are generated in the bus/structure by the electric currents along the conductors. An effort was undertaken to size the power distribution system taking into account transmission efficiency, induced electromagnetic forces and system weight.

A weight optimization computer program was used to determine the preferred transmission efficiencies throughout the power distribution system. The facing page summarizes the results of this task for a 40 KV system. The optimum efficiency is 92% at an operating temperature of 38°C dropping to 91% at a temperature of 149°C. Electromagnetic forces between parallel Mast bus elements are low due to the wide separation between members; the maximum force is less than 0.03 Newton.

The major design issues requiring further study include:

- Selection of the optimum operating voltage
- Design of structural/power transfer joints
- Failure modes and effects analysis to design switching and protection system
- Thermal analysis of integrated system.

SSPS CONDUCTING STRUCTURE-2 SOLAR ARRAY AND MAST TRANSMISSION ELECTRIC BUSES DIMENSIONS ELECTRIC CURRENT FLOW

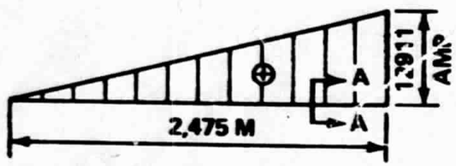
0.24 x 10⁶ Kg

LATERAL BUS WEIGHT = [redacted]

MAST WEIGHT = 0.62×10^6 Kg

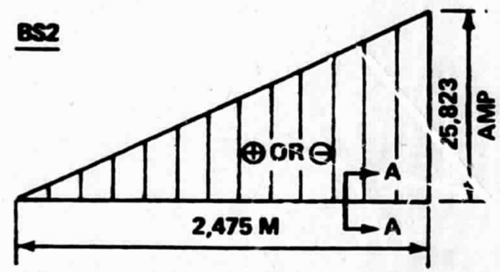
STRUCTURE CROSS-SECTION AREAS DESIGNED FOR OPERATING TEMPERATURE OF 38°C, DISTRIBUTION SYSTEM EFFICIENCY OF 0.92

BS1

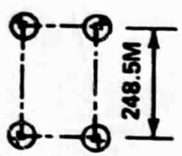


ELECTRIC CURRENT FLOW DIAGRAM

BS2

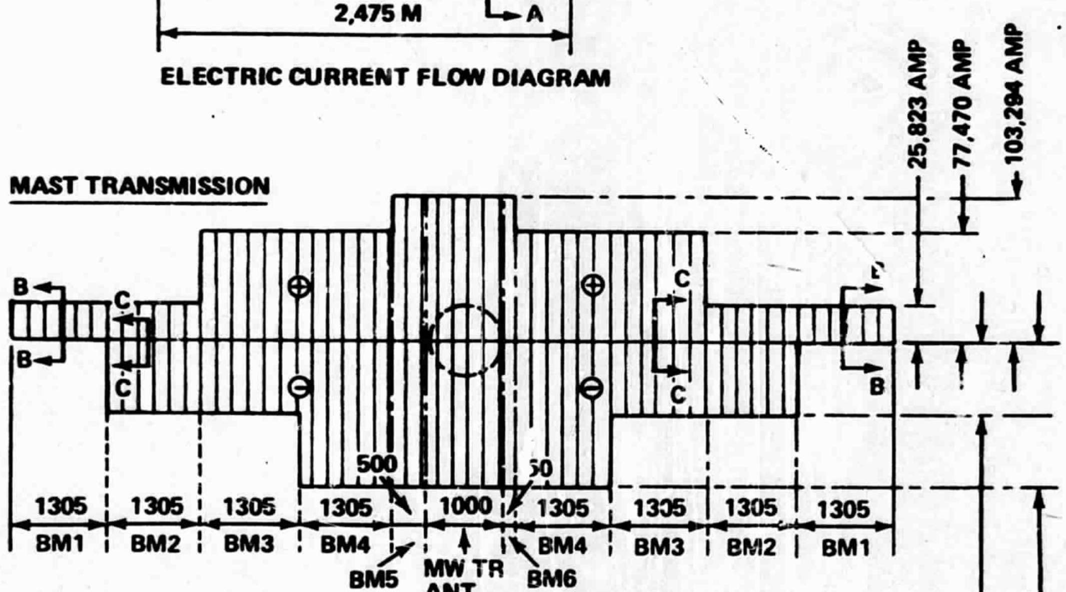


ELECTRIC CURRENT FLOW DIAGRAM

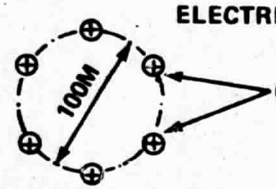


BUS STRUCTURE SECTION, A-A

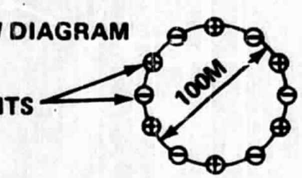
MAST TRANSMISSION



ELECTRIC CURRENT FLOW DIAGRAM



SECTION B-B (MB1 SECTION ONLY)



SECTION C-C (ALL SECTIONS EXCEPT BMI)

Antenna Structural Arrangement

The facing page summarizes the antenna structural arrangement (see MPTS Studies - NAS 3-17835, Report #MPTS-R-002). The arrangement shown is for a 1 KM diameter antenna 40 meters deep. The antenna is assembled in two rectangular grid structural layers to save weight. The primary structure is built-up in 108 x 108 x 35 meter bays using triangular girder compression members. The secondary structure is used as support points for the waveguide subarrays.

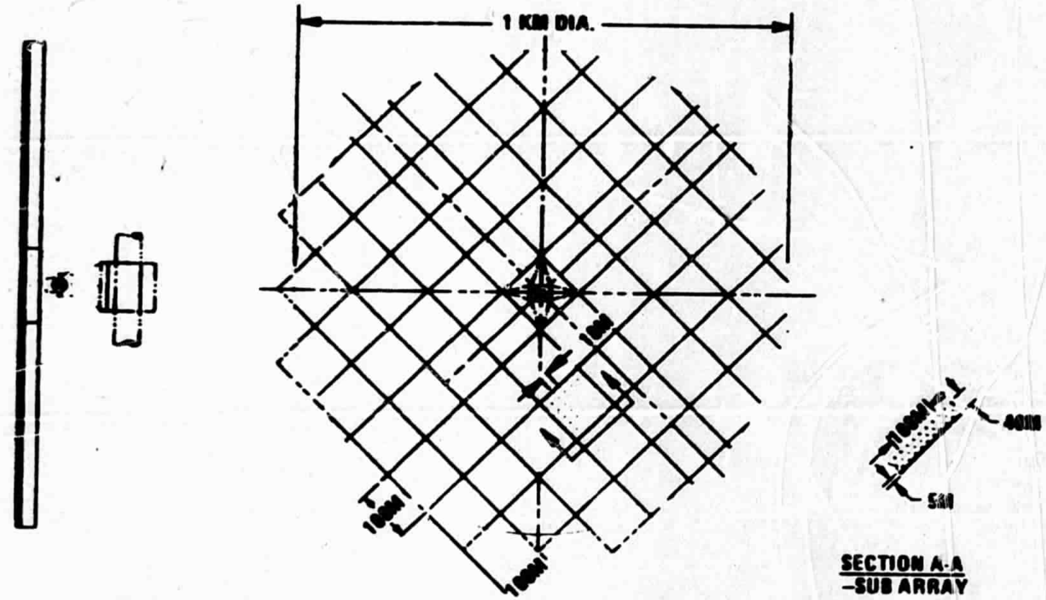
The following summarizes the structural analysis of this arrangement.

- The principal applied load is that induced by the response of the antenna pointing control system during breakaway from the 1×10^6 N·m slip-ring torque. The second design load is caused by gravity gradients

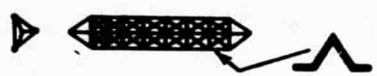
- The selection of the structural material and cross-section is driven by the thermal waste heat from the microwave converters. A triangular hat section made of graphite/polyimide was found to be the preferred design, resulting in the lowest operating temperature and internal thermal induced loads. Steel or titanium could result in a lower cost design

- Analysis of thermally induced deflections led to the 40m depth of the antenna. Selection of a graphite composite would allow a reduction in this depth and would result in a lower weight design.

ANTENNA STRUCTURAL ARRANGEMENT



SECTION A-A
–SUB ARRAY



	ALUMINUM (2024-t6)		COMPOSITES (POLYIMIDE)	
• TEMP, ^{o}K	450		560	
• MODULUS OF ELASTICITY, Psi	9×10^6		6×10^6	
• DENSITY, LB/IN^3	0.101		0.055	
• THICKNESS RANGE, IN.	0.015 TO 040		0.020 TO 0.055	
• WEIGHT	LB (10^3)	KG	LB (10^3)	KG
SUBARRAY PRI STRUCT	300	137	207	94
SUBARRAY SEC STRUCT	103	47	65	30
ANT. SUPPORT STRUCT	233	106	157	71
YOKE & MECHANISMS	146	66	122	56
COATINGS	46	21	49	22
AMPLITRON SUPPORT				
CONTOUR CONTROL ACTUATORS	268	122	268	122
AMPLITRON ATTACH STR	51	23	36	16
TOTAL	1147	522	904	411

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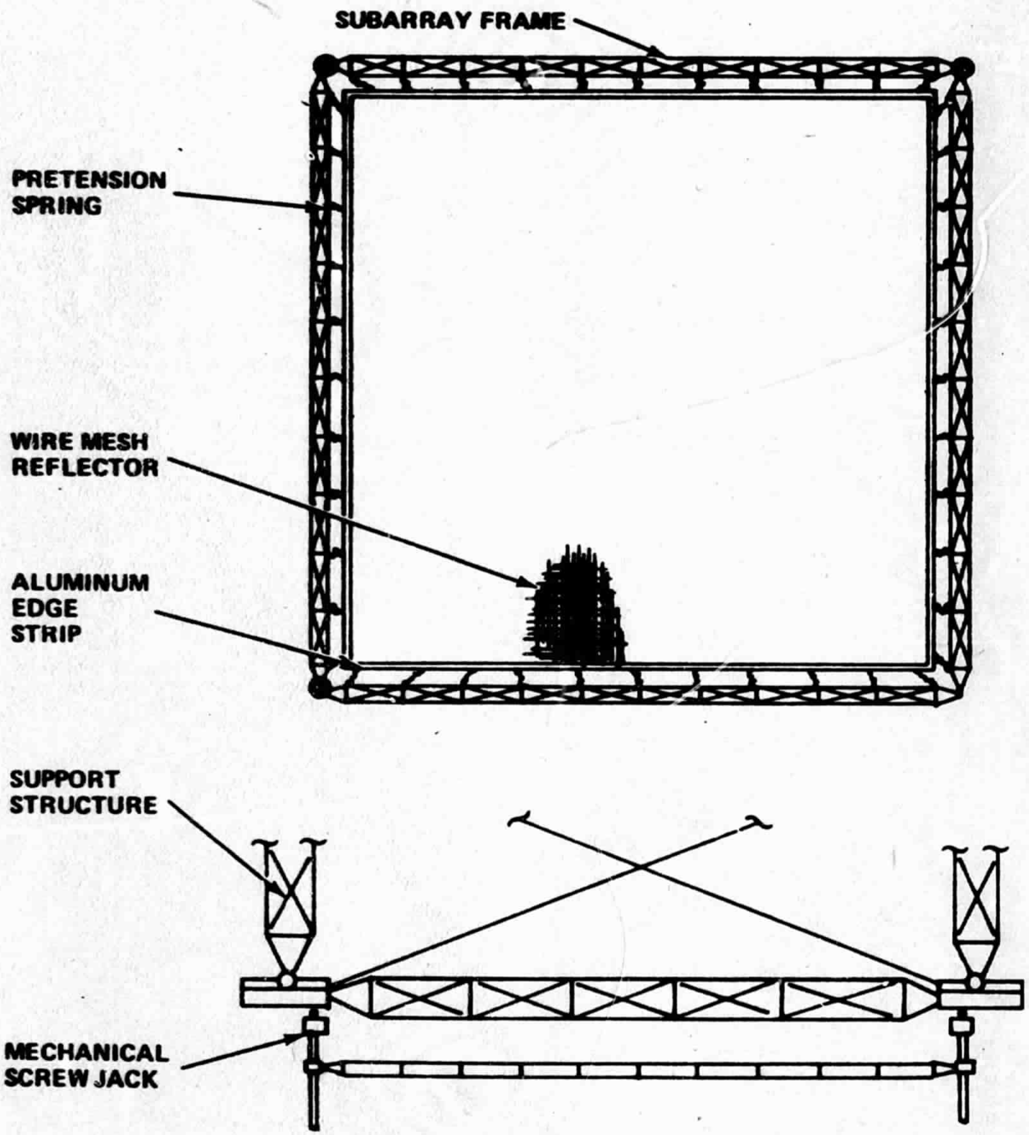
PRS Subarray Concept

The facing page presents the design concept used in the assessment of the Power Relay Satellite to meet the microwave reflector requirements. The following preliminary conclusions were reached:

- Reflector surface roughness can be maintained to within 1/20 of a wavelength for a satellite with 18m x 18m reflector surface subarrays
- A mechanical screw jack system mounted at the corners of the 18 x 18m subarray desensitizes reflector flatness to the distortions of the supporting structure
- The wire mesh reflector surface can tolerate sudden temperature variations by utilizing pretensioned springs between the mesh and subarray frame
- The subarray frame's thermal distortions can be maintained within limits with the use of low thermal coefficient of expansion graphite composites.

PRS SUBARRAY CONCEPT



DESIGN REQUIREMENT:

- MAINTAIN WIRE MESH SUBARRAY SURFACE FLATNESS TO LESS THAN $\lambda/20$ (5MM)
- THIRTY YR. LIFE (WITH PRETENSION SPRING)
- 1.2×10^{-8} PSI MIRCROWAVE PRESSURE ON SUBARRAY MESH
- TEMPERATURES
  - MESH 200°F TO −250°F
  - STRUCTURE 200°F TO −100°F

DESIGN ANALYSIS

- WIRE STRESS = 55.2 PSI
- SUBARRAY FRAME LOAD
  - BENDING = 1206 IN-LB
  - END LOAD = 6.81 LB
- FRAME THERMAL BENDING WILL STAY WITHIN 5MM LIMIT IF ΔT BETWEEN CAPS IS LESS THAN 16°F

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Key Issues - Large Structure

The design load environments during launch, manufacture-in-space, assembly, and orbit transfer as well as orbital operations should be determined before a meaningful structural design is made. A dynamic simulation of these loads is needed to determine the response of the large, flexible lightweight structure at various points in the spacecraft life cycle. Load/stress time histories and thermal gradient/stress time histories are required to determine: stress/strength integrity, fatigue life, cumulative creep/rupture properties, creep fatigue, deflections, and fracture properties. Significant contribution to the thermal stress/distortions results from the eclipse of the SSPS during 45 days in the spring and fall. The effect of the repeated thermal variation needs further assessment.

A three-phase analysis is needed to fully assess the SSPS structure. A temperature time-history could be predicted using NASTRAN or Grumman's Integrated Thermal Analysis Procedure. NASTRAN or Grumman's ASTRAL-COMP computer program system could be used to evaluate the deformations due to external loads. An expanded integrated program is felt necessary to fully evaluate the structure, including cross coupling between modes.

Materials and processes studies on technique for fabricating structure in-orbit is required. The studies to date emphasized the use of 5052 aluminum alloy 0 condition in sheet strip rolls formed into structure by rolling mills. Other materials and processes should be evaluated.

Methods for verifying the SSPS structural integrity should be evolved. The application of ground test techniques do not necessarily apply. Ground test techniques using scale models with similar structural and dynamic characteristics should be developed. A flight test of an instrumented structural model would be needed.

KEY ISSUES – LARGE STRUCTURE

- STATIC & DYNAMIC STRUCTURAL RESPONSES TO THERMAL AND LOAD ENVIRONMENTS
- STRUCTURAL ANALYSIS TOOLS & TECHNIQUES
- MATERIALS & PROCESSES
- ON-ORBIT MANUFACTURE & ASSEMBLY TECHNIQUE IMPACT ON STRUCTURAL DESIGN
- DEVELOPMENT OF STRUCTURAL VERIFICATION TECHNIQUES

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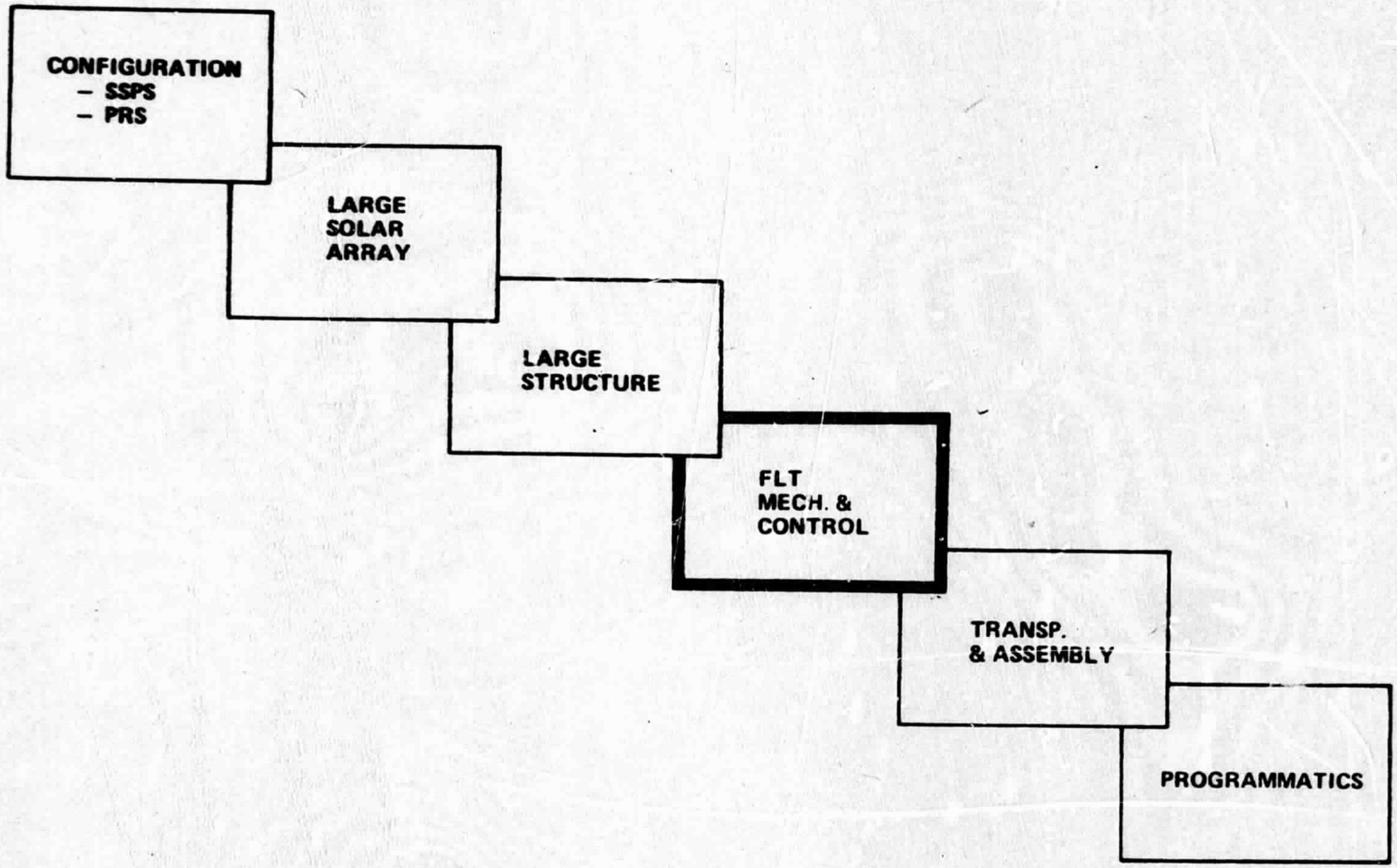
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Orbit Perturbations/Attitude Disturbance

There are four major influences on the SSPS and PRS causing them to drift from its nominal orbital location. These are:

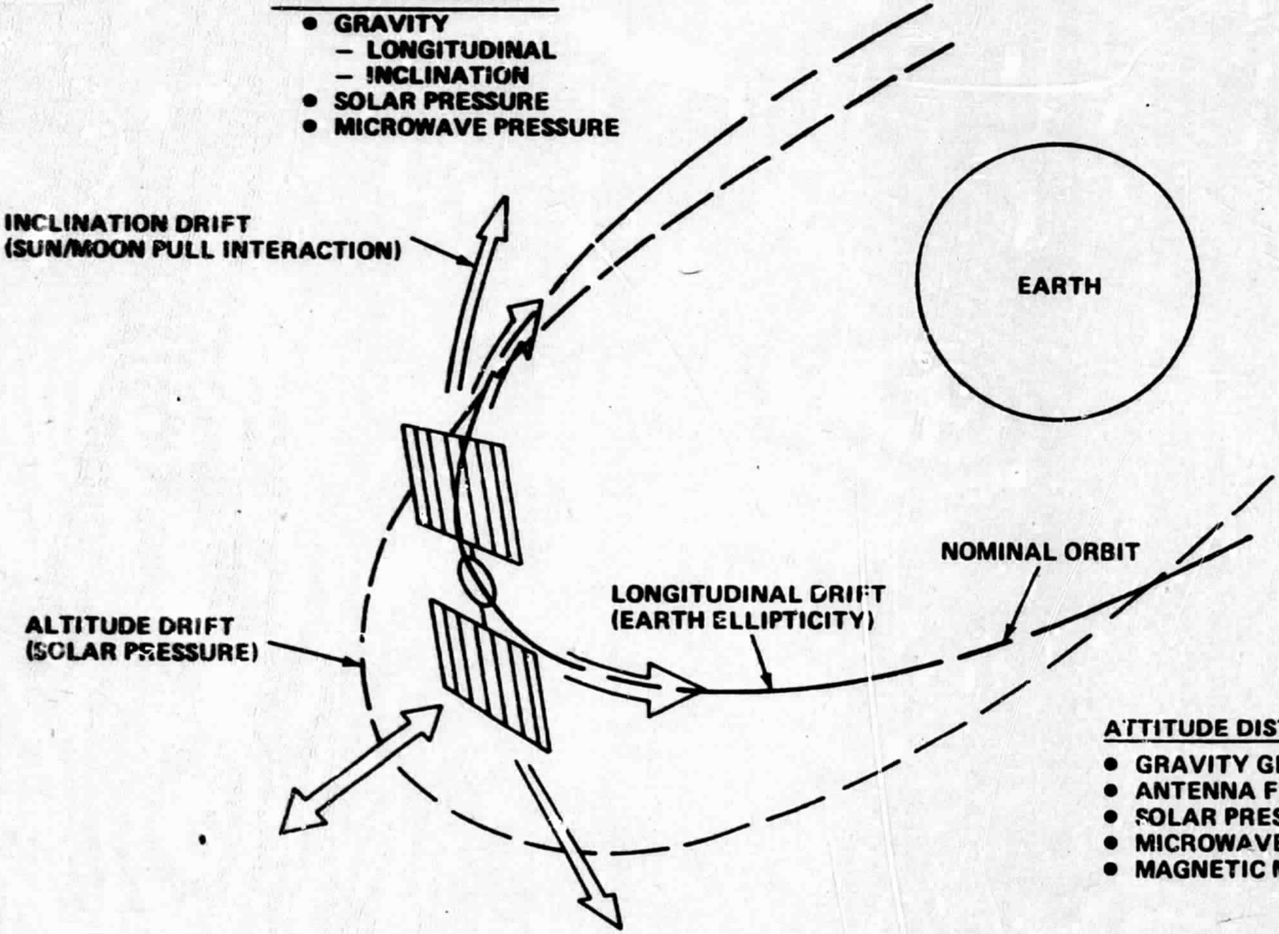
- Longitudinal drift - The ellipticity of the earth causes the satellite to seek out the earth's minor axes
- Inclination Drift - The interaction of the sun and moon's gravitation causes the orbit to regress, so that its inclination changes with respect to the equator
- Altitude and Eccentricity Drift - Solar pressure distorts the orbit from circular to elliptical and back again over a one year period
- Microwave Pressure - The electromagnetic fields at the aperture of the slotted array causes a "rebound" pressure of the SSPS antenna. The reflected pressure of the PRS causes a rebound.

The disturbance torques on the satellite result from gravity gradient solar pressure, magnetic, microwave pressure and Rotary Joint Friction Torque (SSPS only). At an altitude of 35.8×10^6m, the atmospheric density is too small to cause any significant disturbance to the attitude control system.

ORBIT PERTURBATIONS/ATTITUDE DISTURBANCES

ORBIT PERTURBATIONS

- GRAVITY
  - LONGITUDINAL
  - INCLINATION
- SOLAR PRESSURE
- MICROWAVE PRESSURE



ATTITUDE DISTURBANCES

- GRAVITY GRADIENT
- ANTENNA FRICTION TORQUE
- SOLAR PRESSURE
- MICROWAVE PRESSURE
- MAGNETIC MOMENTS

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SSPS Propellant Requirements

The facing page summarizes SSPS propellant requirements for stationkeeping and attitude control (see also monthly report NSS-R-003) propellant requirements.

The stationkeeping requirements are approximately 9700 kg/year while attitude control requires 14,400 kg/year. Solar pressure is the dominant orbit perturbation, which is most economically handled by continuously controlling orbit period and not correcting eccentricity drift. However, if the SSPS satellite density goes above 15 units over the continental United States, the eccentricity drift should be corrected. This would add 14,883 kg/yr to the propellant requirement.

Transients from the antenna rotary joint control system used for antenna pointing, sizes the array thrusters (40 Newton mounted at the extremes of the array). Gravity gradient disturbances require the most control system propellant consumption, 13,804 kg/yr. CMG's were evaluated for the control function and were found to be excessive in size (20×10^6 kg for roll axis control alone).

The total propellant consumption using electric propulsion actuators (I_{sp} = 8000 sec) is 24, 149 kg/yr without eccentricity drift control and 39,032 kg/yr with correction of eccentricity drift.

SSPS PROPELLANT REQUIREMENTS

(I_{SP} = 8000 SEC, SSPS WGT = 25 MLB)

	LBM/YR		Kg/YR	
STATION KEEPING				
LONGITUDE DRIFT	1,600		726	
INCLINATION DRIFT	14,700		6,673	
SOLAR PRESSURE				
– ALTITUDE DRIFT	5,100		2,315	
– ELLIPTICITY DRIFT	0	(32,784)*	0	(14,883)*
MICROWAVE PRESSURE	68		31	
SUBTOTAL	21,470	(54,252)*	9,745	(24,628)*
ATTITUDE CONTROL				
GRAVITY GRADIENT	30,408		13,804	
ANTENNA CONTROL	162		74	
SOLAR PRESSURE	870		394	
MICROWAVE PRESSURE	292		132	
SUBTOTAL	31,732		14,404	
TOTAL	53,202	(85,986)*	24,149	(39,032)*

*REQUIREMENT AFTER 15 SSPS ARE PLACED IN ORBIT TO SERVICE THE UNITED STATES.

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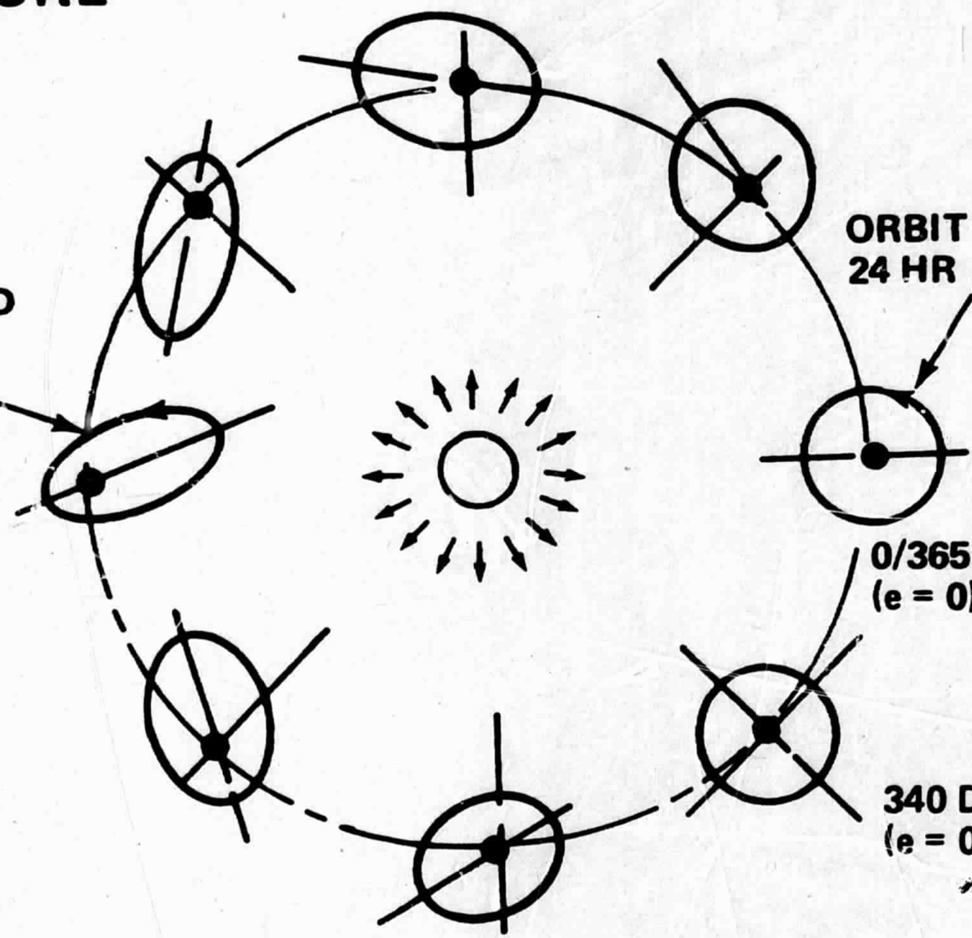
Altitude and Eccentricity Drift Characteristics - Solar Pressure

Solar pressure has a considerable effect on the large area SSPS solar array. The facing page illustrates this effect on the SSPS orbit. Over a period of 170 days the circular orbit distorts to an ellipse with an eccentricity of 0.96. In addition, the orbit period increases from 24 hrs to 24 hrs, 14 minutes. Both the orbit shape and period return to nominal after 340 days.

If the change in period goes unchecked, the SSPS will precess around the equator at a rate of approximately 3 1/2 deg per day. A propellant expenditure of 5100 lb/yr would be required to offset this highly undesirable motion. The propellant required to correct the ellipticity has been calculated at 3.5×10^5 lb/yr. This assumes that an opposing force of 50 to 70 LBS is continuously applied to offset the solar pressure. The effect of ellipticity on overall system performance, however, is not significant. Ellipticity causes an apparent longitudinal drift of only 3.5 deg to an observer on the ground. This magnitude of motion has little effect on ground rectenna size or conversion efficiency. Therefore, it is concluded that this motion go "unchecked."

Uncorrected eccentricity drift is acceptable if more than 2200 N.M. of orbit arc length can be assigned to the SSPS free of other SSPS or satellites. This condition suggests that this drift condition be checked. An alternate approach to continuous thrusting to null solar pressure, precluding the 3.5×10^5 lb/yr propellant consumption can be used. Periodic posigrade/retrograde maneuvers performed at apogee and perigee of the eccentric orbit would economically maintain the orbit. The yearly propellant using this technique is 33,000 lb.

ALTITUDE & ECCENTRICITY DRIFT CHARACTERISTICS – SOLAR PRESSURE



ASSUMPTIONS

- SSPS AREA = 50 Km^2
- M = 1×10^6 SLUGS
- $F_{SOL\ PRESS} \approx 50$ LB
- ACCELERATION $\approx 5 \times 10^{-5}$ FT/SEC^2

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PRS Propellant Requirements

The facing page summarizes the PRS stationkeeping and attitude control propellant requirements for two levels of propulsion system performance. The most severe perturbation is due to microwave pressure. The 34 LB pressure requires 387 fps/yr delta-V for daily correction with an apogee/perigee maneuver. If a continuous opposing thrust were applied to correct this perturbation, an excess of 134,000 LB/YR would be required. It is clear that an orbitkeeping technique using periodic impulsive maneuvers is significantly lower in propellant consumption.

The need for impulsive maneuvers for orbit keeping, rather than continuous low thrust techniques causes problems for both the SSPS and PRS. The high performance electric propulsion unit is almost mandatory for SSPS. The issue to be addressed is to determine if these low-thrust devices can be utilized in an impulsive orbit-keeping algorithm. The PRS, on the other hand, could use chemical propulsion for this function without causing excessive propellant consumption.

PRS PROPELLANT REQUIREMENTS ~ LB

STATION KEEPING	I_{SP} = 8000 SEC	I_{SP} = 200 SEC
LONGITUDINAL DRIFT	45.3	1808
INCLINATION DRIFT	442.3	17490
SOLAR PRESSURE		
— ALTITUDE	2.5	99
— ECCENTRICITY q	171.4	683
MICRO WAVE PRESSURE	7127	43818
SUBTOTAL	1788.5	63,898
ATTITUDE CONTROL		
GRAVITY GRADIENT	161.4	6457.6
TOTAL	1949.9	70,355.6

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Key Issues - Flight Mechanics and Control

The analysis of SSPS structure/control system interactions to date have restricted simulation to uncoupled modes. Future studies should be expanded to include torsional vibration modes and cross coupling between modes.

Finite thrust stationkeeping algorithms using low-thrust electric propulsion should be studied. The high levels of propellant needed to thrust continuously (in the case of SSPS to null solar pressure and in the case of PRS to null microwave pressure) suggests a more economical horizontal maneuver. The suitability of performing this function with electric engines must be evaluated.

Stationkeeping accuracy requirements are dictated by the effect of orbit position changes on ground collection efficiency. The major causes for power loss are cosine loss, rectenna spill over and ray loss between rectenna panels. All are effected by the relative position of the spacecraft and rectenna.

The studies to date (NAS 3-17835) on the rotary joint control system excluded the compliance of the central mast. This is a destabilizing effect and could change the design requirements on the system.

Ion propulsion is a natural SSPS control device because of the readily available power. Little or no flight experience is available on these devices in the control role.

If a system of SSPS's are located in-orbit, occulation of one satellite by another will occur during the vernal and autominal equinox. This anomoly complicates mission operations and could cause problems in the design of the array power distribution system.

KEY ISSUES – FLT MECHANICS AND CONTROL

- FLEXIBLE STRUCTURE/CONTROL INTERACTION
- DEFINITION OF STATIONKEEPING ALGORITHM USING LOW THRUST ACTUATORS
- DEFINITION OF STATIONKEEPING ACCURACY REQUIREMENTS
- ROTARY JOINT CONTROL SYSTEM DEFINITION
- ELECTRIC PROPULSION PERFORMANCE AND RELIABILITY IMPROVEMENT
- EFFECT OF SATELLITE TO SATELLITE OCCULTATION ON SYSTEM PERFORMANCE & DESIGN

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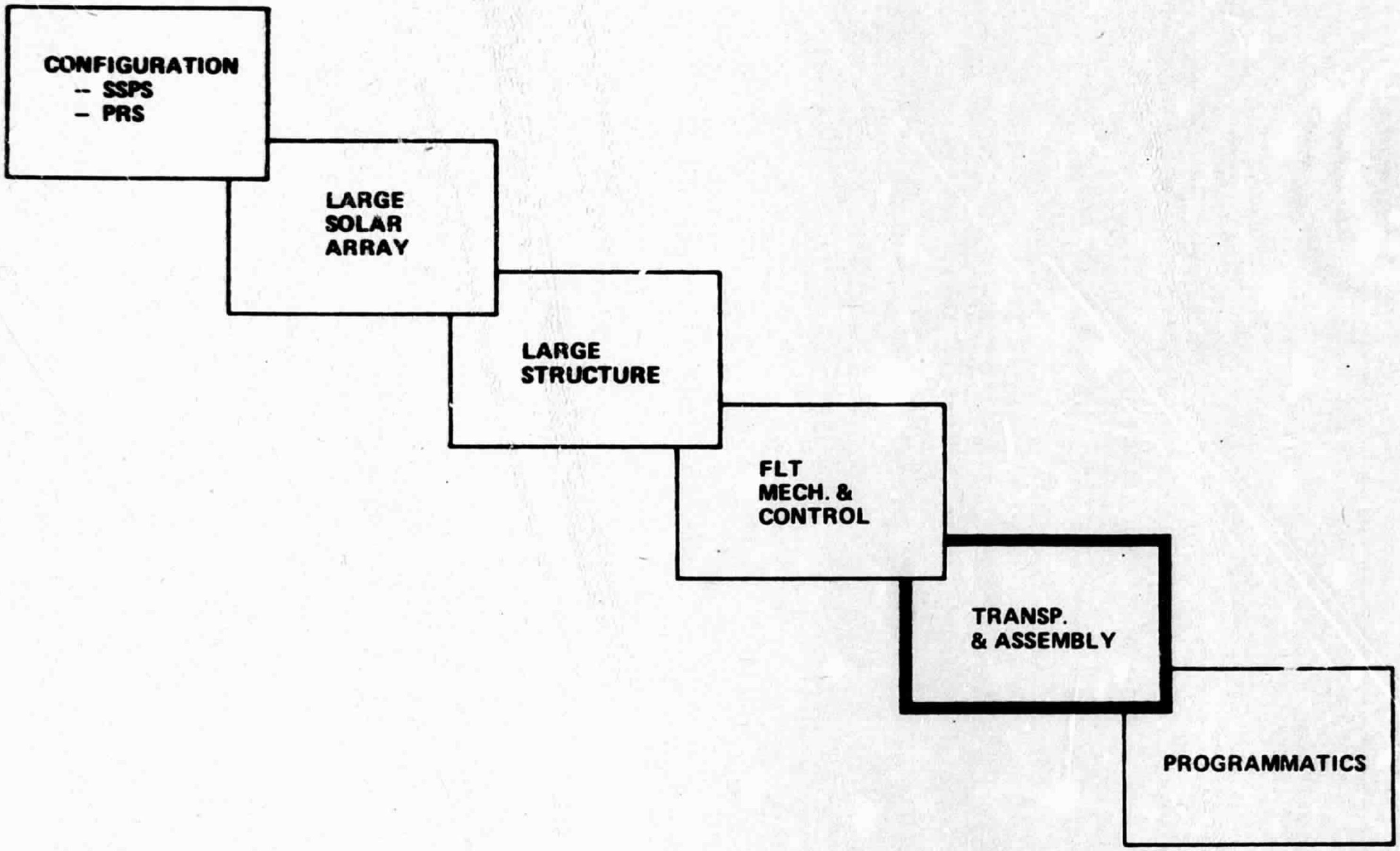
CONFIGURATION
-- SSPS
-- PRS
LARGE SOLAR ARRAY
LARGE STRUCTURE
FLT MECH. & CONTROL
TRANSP. & ASSEMBLY
PROGRAMMATICS

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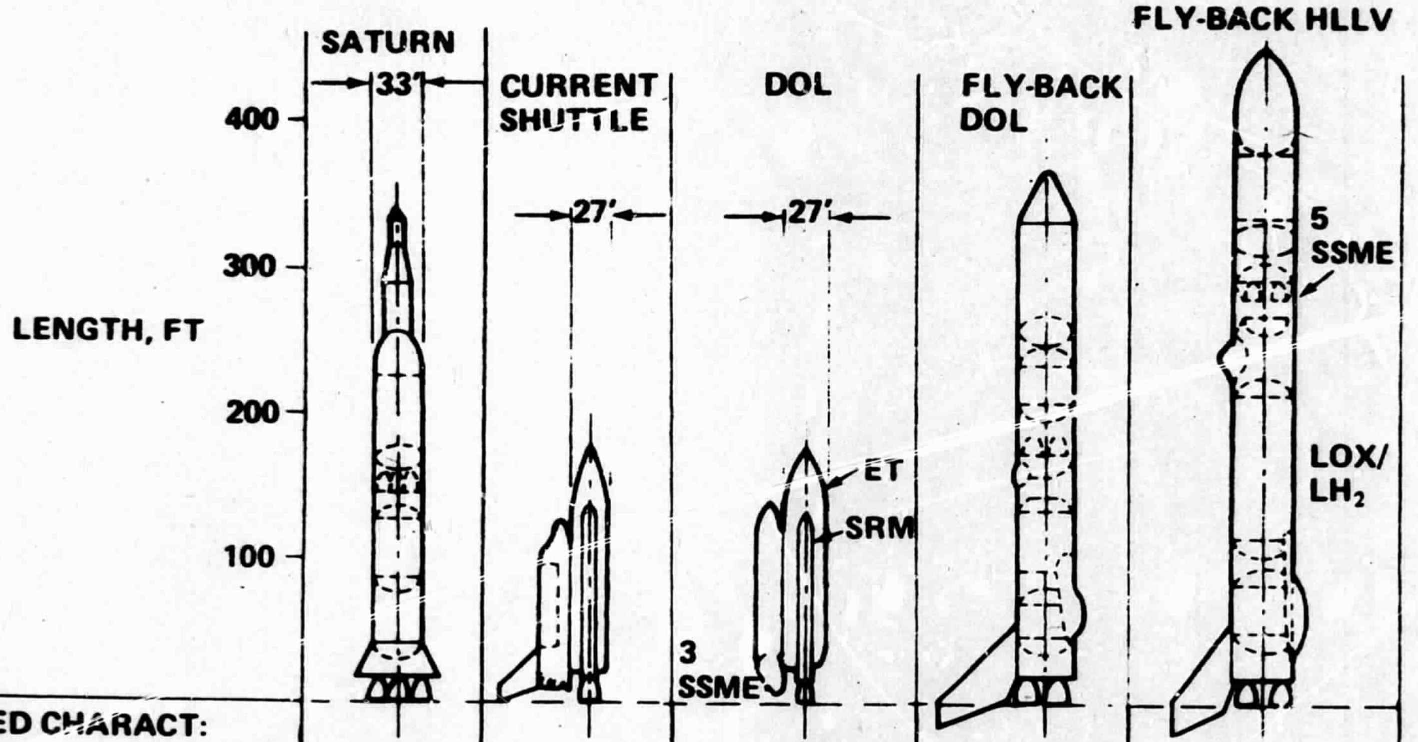
Typical Candidate Launch Systems

A selected matrix of potential launch systems are given on the facing page. These launch systems span a range of design approaches from the use of the current Shuttle to the development of a fully reusable Heavy Lift Launch Vehicle (HLLV) with 400,000 LB (186,000 Kg) payload to Low Earth Orbit (LEO). Two intermediate design approaches, which are derivatives of Shuttle, are included to assess the potential of vehicles with moderate performance.

Operating costs for the system to the right of the Shuttle vary between $8M and $35M, depending on the mode of recovery of the second stage engines and avionics. If the second stage components could be recovered with low components refurbishment, the lower operating cost could be achieved. The upper cost per flight value assumes that 70% of the upper stage is recovered with minimum refurbishment. These costs were established by using the subsystem costs established for Shuttle in the 1972 Phase A/B studies and scaling factors for structure.

TYPICAL CANDIDATE L/S's



	SATURN	CURRENT SHUTTLE	DOL	FLY-BACK DOL	FLY-BACK HLLV
• ASSUMED CHARACT:					
– GROSS L/O WT, KLB	6367	4207	4207	6530	7012
– PAYLOAD TO LEO, KLB	270	65	160	195	400
• COST SPAN					
– UNIT, $M	N/A	200 – 300	150 – 250	350 – 450	600 – 900
– COST/FLIGHT, $M	300	10.5	13 – 35	8 – 30	9 – 35
– $/LB (EXCLUDING FLEET COSTS & DDT&E)		161	82 – 219	46 – 154	23 – 88

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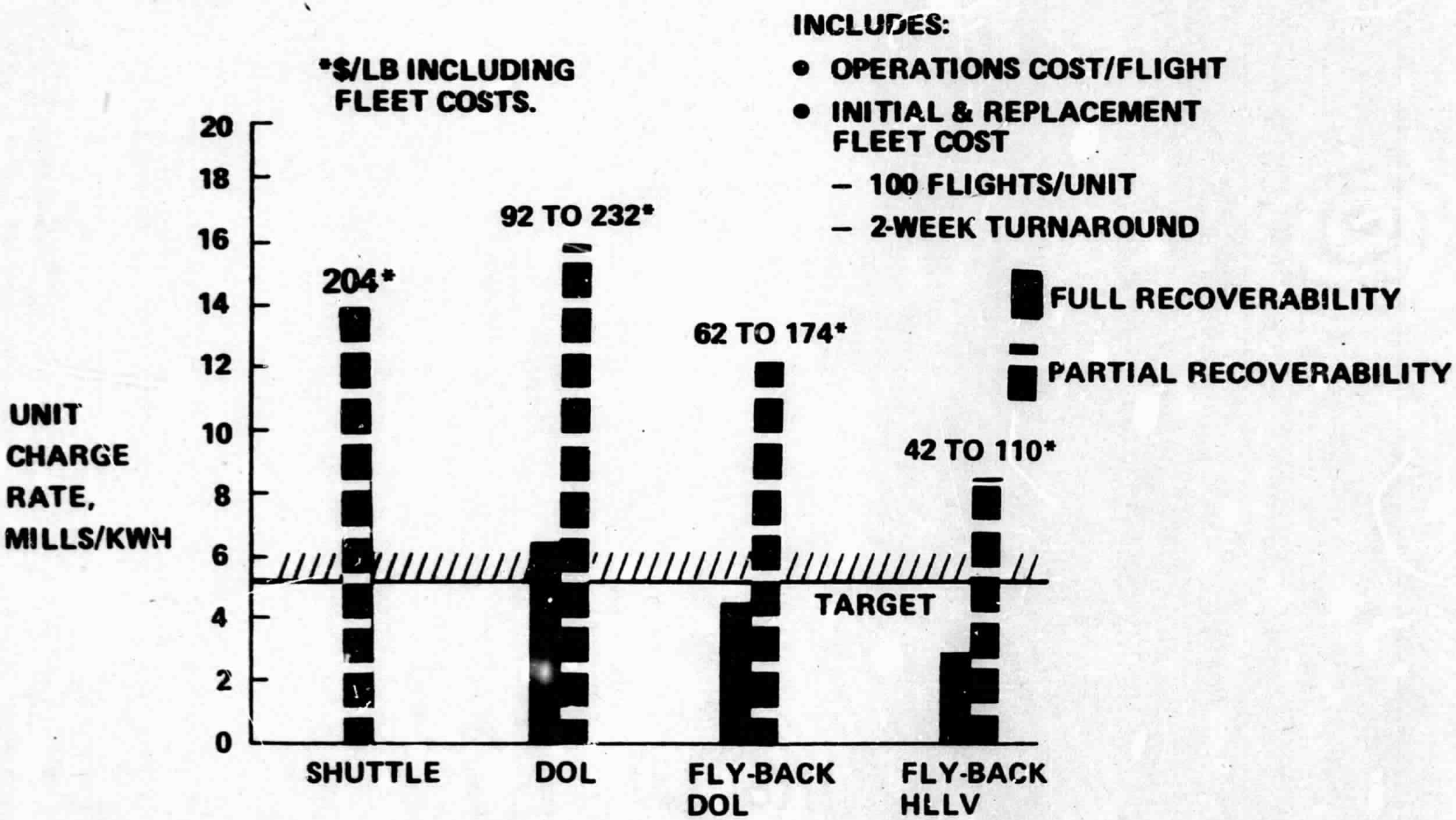
Launch System Comparisons

The facing page compares the contribution each launch system makes to the "unit charge rate," or annuity to the entire system using the following assumptions:

- Discount Rate = 8.6%
- Time period = 1990 to 2020
- Mission Model 1
- 100 Flights life per vehicle
- 2 week ground turnaround

A target of 5 MILLS/KWH "unit charge rate" has been established for transport to low earth orbit for purposes of study. With this target for reference, it can be seen that recoverability of the launch system and heavy lift capability are both essential. An HLLV with 400,000 lb payload capability and full recoverability could achieve a cost of 42 $/lb (93 $/Kg). Other work being performed on future launch systems suggest 1 million pound payload performance which would reduce launch costs to between 20 and 36 $/lb., depending upon the extent of recoverability.

L/S COMPARISON



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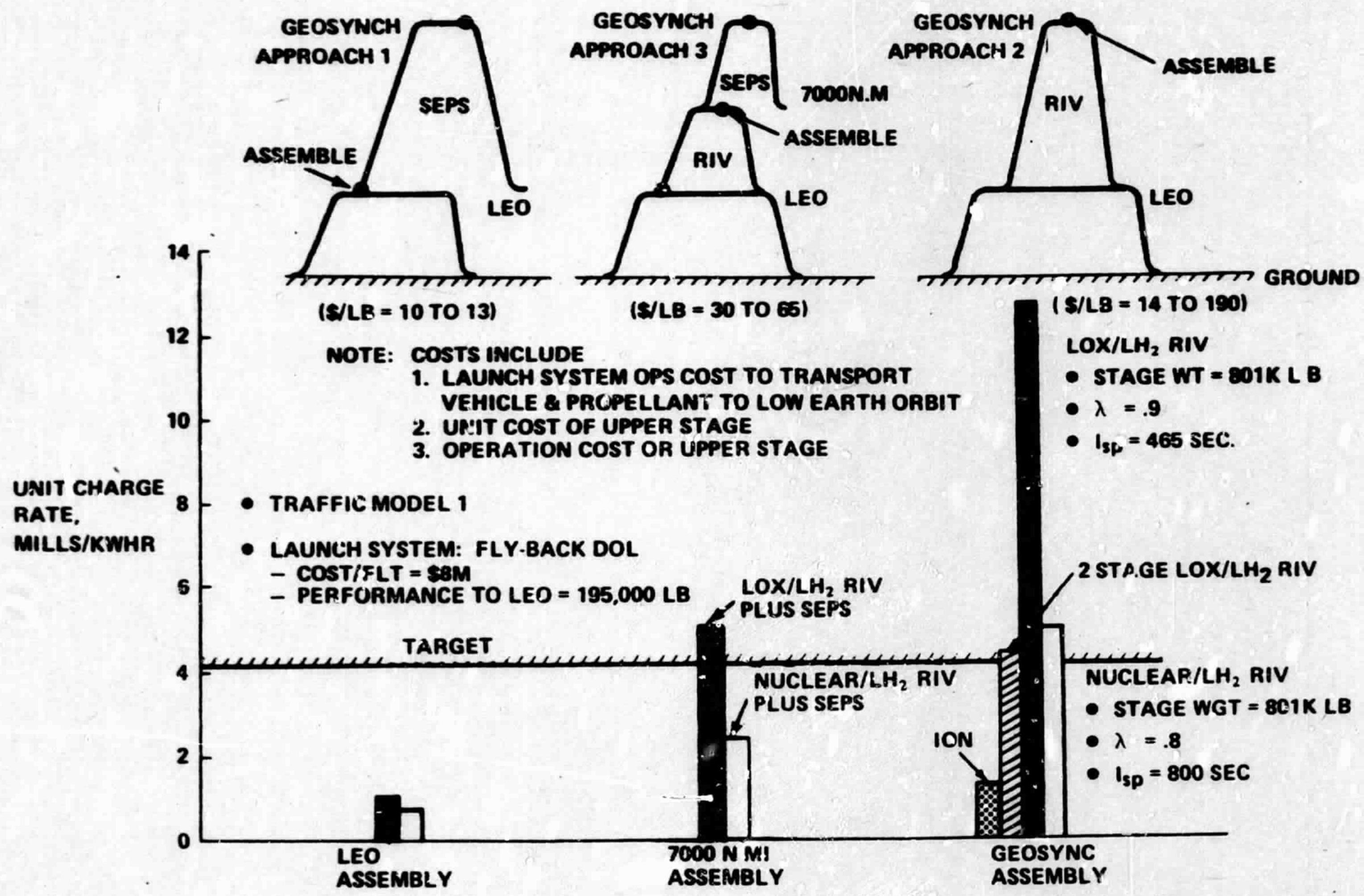
Orbit-To-Orbit Transport Costs

The facing page summarizes a comparison of orbit-to-orbit transport costs contribution to the unit charge rate. Electric propulsion clearly lowers cost for the case in which final assembly is at LEO or at geosynchronous orbit. The option to assemble at 7000 N.M. circumvents the problems of maneuvering through the Van Allen Belts with SEP, but does not appear to provide the type of cost benefits needed. The delta-V to the 7000 N.M. location is 70% of the total trip, requiring significant numbers of large chemical or nuclear Tug flights before the advantage of high performance ion stages can be introduced. A two stage large chemical Tug can just as effectively perform the mission with assembly at geosynchronous orbit as the 7000 nm assembly case.

ORBIT-TO-ORBIT TRANSPORTATION COSTS



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Assembly Cost-Manned Control From Orbit

The facing page relates the major cost driver for a space based manned controlled assembly operations to the contribution final assembly makes to the unit charge rate (MILLS/KWH)-and cost per pound for assembly. The major drivers are assembly rate in terms of Lb/man-hour, the cost of space stations (assumes 10 year life) and the cost to recycle crews. No assumptions are made as to the level of automation achieved in the operation.

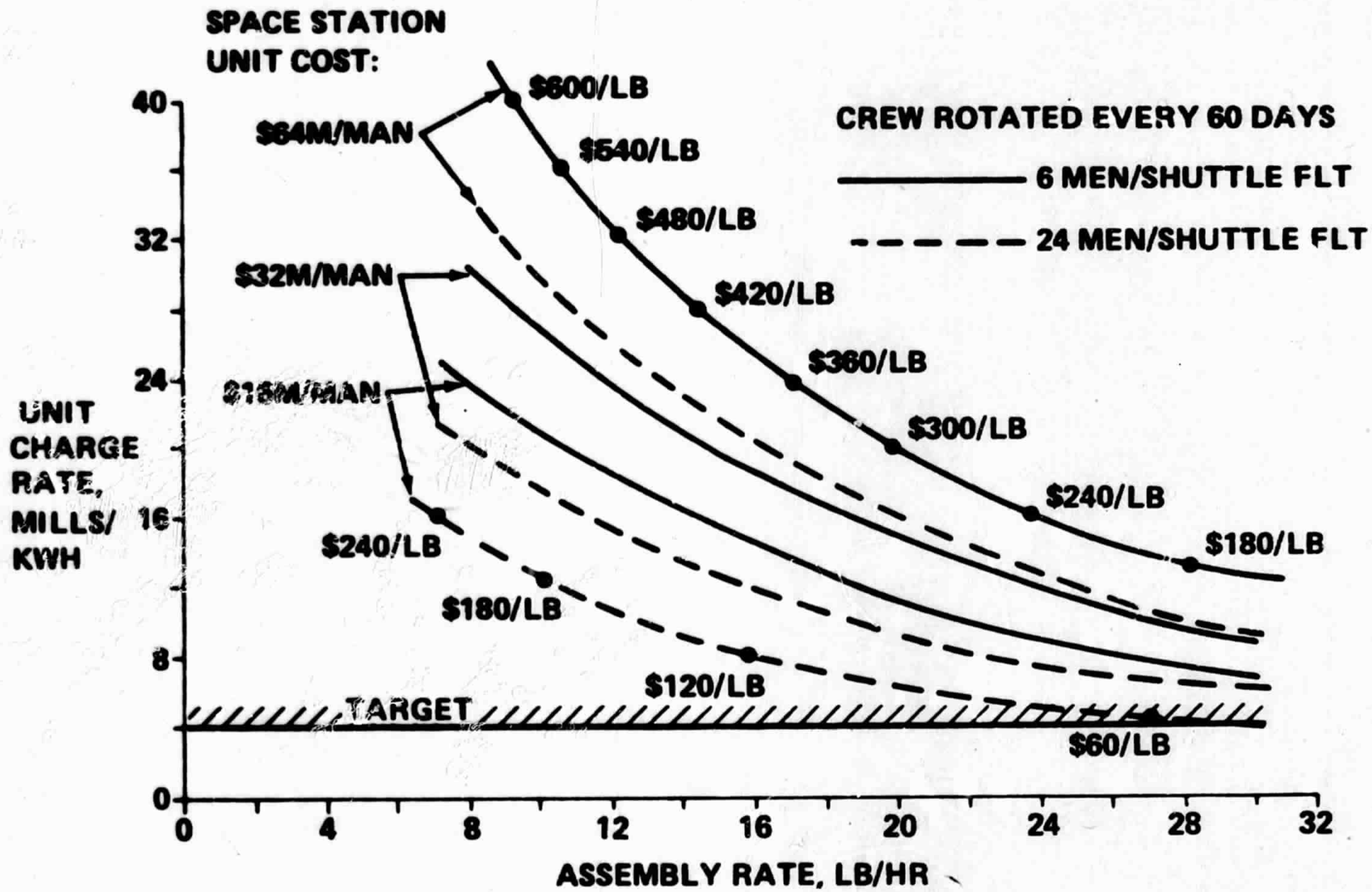
To achieve reasonable cost levels, i.e., 4 MILLS/KWH, production rates in excess of 25 Lb/Hr. are required along with low cost space stations ($16M/Man) and transport modes that can recycle large numbers of crew members in one flight.

As a means of comparison, a 12-man modular space station has been estimated to cost $760M or $64M/Man. Automobile assembly is performed at the rate of approximately 25 Lb/man-hour.

The major assembly issues to be addressed include:

- Determination of the extent of man's involvement. It has been shown that the target unit cost rates can be achieved at lower production rates using ground controlled teleoperators
- Simulations to determine production rates for assembly approaches ranging from EVA operations to remote controlled free flyer teleoperators
- Systems analysis and prototype development of automated on-orbit fabrication equipments for structure, waveguides and electronics installation.

ASS'Y COST – MANNED CONTROL FROM ORBIT (LOW ALT ASS'Y SITE)



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Key Issues - Transportation and Assembly

The preliminary transportation and assembly trade studies have indicated that recovery and low cost reuse of the launch system, in addition to heavy lift launch capability, is essential to competitive SSPS user costs. The issue involves a trade-off between large launch vehicle (and associated development cost) vs the cost to develop full recoverability. The impact of high density launch rate on launch facility operations and costs should be addressed before selection of launch vehicle size is made.

Ion propulsion offers the lowest cost approach for orbit-to-orbit transport of materials. Development of large diameter thrusters, selection of a power source and choice of propellant are key issues that should be addressed in an overall systems study of upper stage concepts.

Top level assessment of assembly requirements has led to some general conclusions. The first is that manned participation in the assembly must be minimized to keep costs of supporting equipments down. Remote control of the assembly operation offers cost advantages, though more technology in the form of simulation is required.

A key trade-off between prefabricating deployable structure on the ground vs on-orbit fabrication is needed. The prefabricated structure would result in poor packaging density and therefore poor launch system load factor, but could offer high on-orbit assembly rate. On-orbit fabrication would fully use the capability of the launch system.

KEY ISSUES – TRANSPORTATION & ASSEMBLY

- DEVELOPMENT OF FULLY RECOVERABLE HEAVY LIFT VEHICLE
- DEVELOPMENT OF ION PROPULSION
- FEASIBILITY OF IN-ORBIT FABRICATION AT HIGH PRODUCTIVITY
- LOW COST SPACE STATIONS
- FEASIBILITY OF REMOTE CONTROLLED ASSEMBLY
- DETERMINATION OF EXTENT OF MAN'S INVOLVEMENT
- FEASIBILITY OF ON-ORBIT LOGISTICS OPERATIONS
  - PROPELLANT TRANSFER
  - UPPER STAGE MATTING TO PAYLOAD
  - RENDEZVOUS OF HLLV

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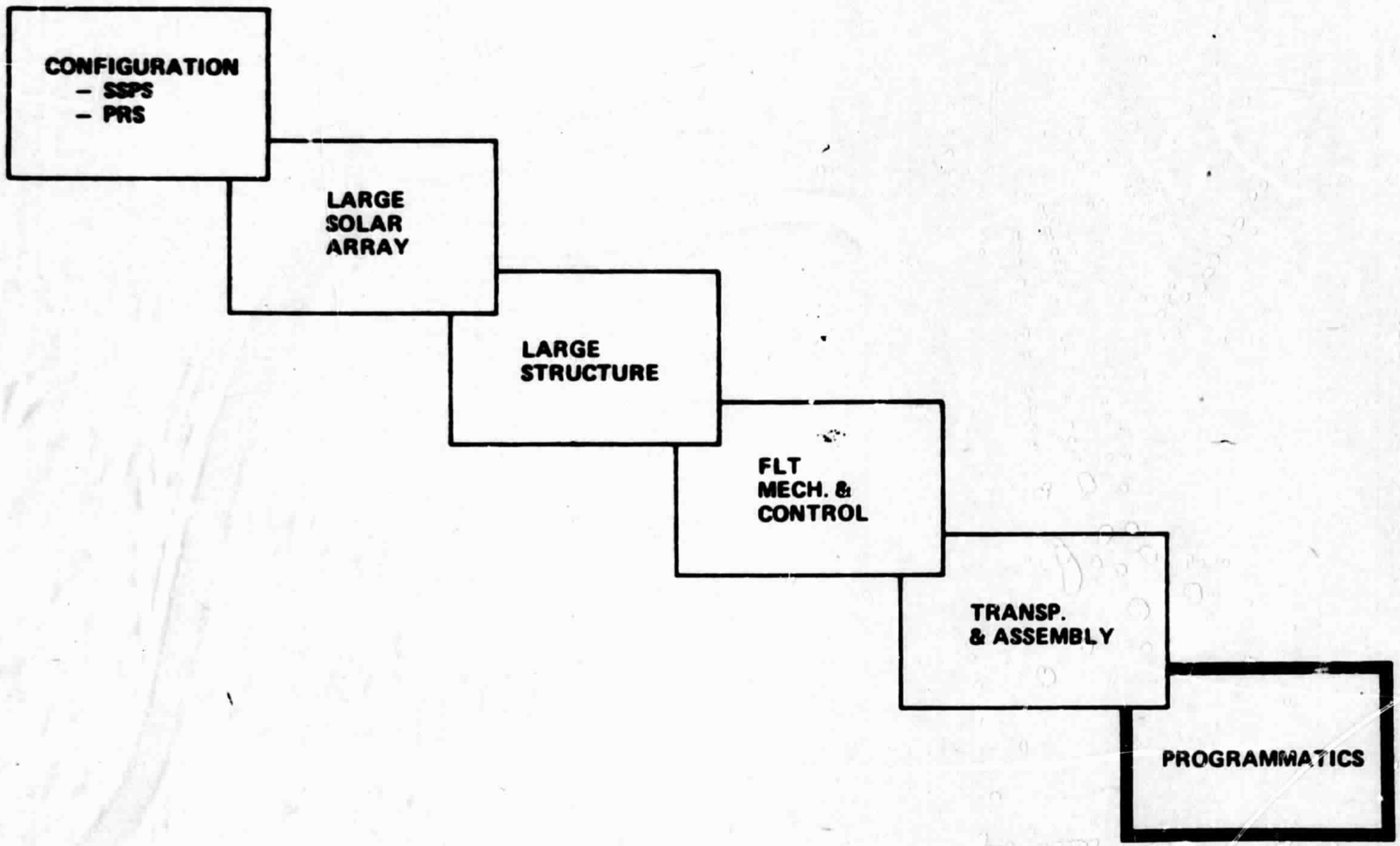
CONFIGURATION
– SSPS
– PRS
LARGE SOLAR ARRAY
LARGE STRUCTURE
FLT MECH. & CONTROL
TRANSP. & ASSEMBLY
PROGRAMMATICS

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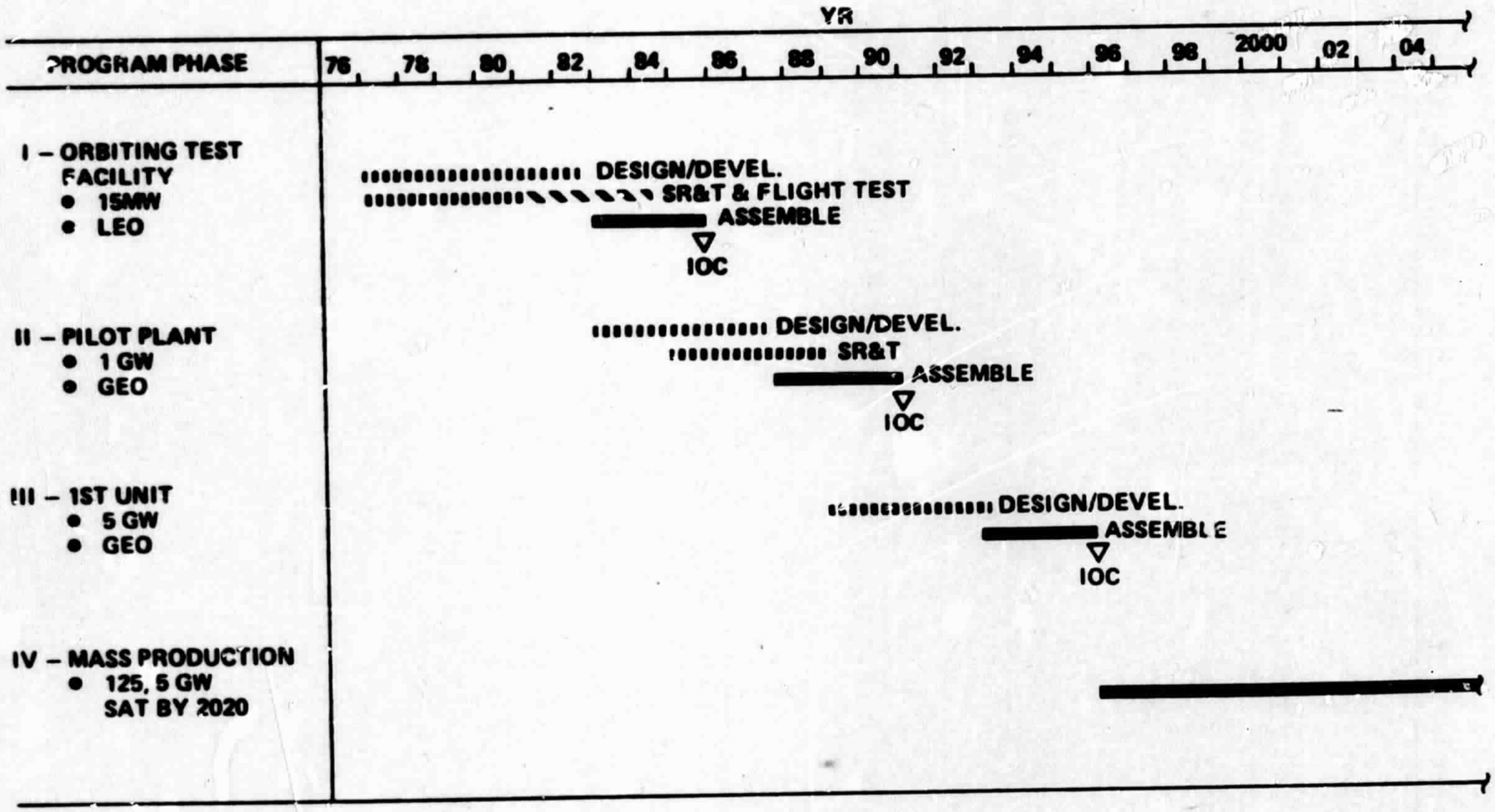
Program Schedule Baselined for Preliminary SSPS Costing

The facing page is an SSPS development schedule used in a preliminary assessment of costs. A three-step program is utilized in which a small LEO Orbiting Demonstration and Test facility is deployed in 1985. A geosynchronous stationed 1GW pilot plant is scheduled for 1990. A full capability plant (5GW) is scheduled for 1995.

PROGRAM SCHEDULE BASELINED FOR PRELIMINARY SSPS COSTING



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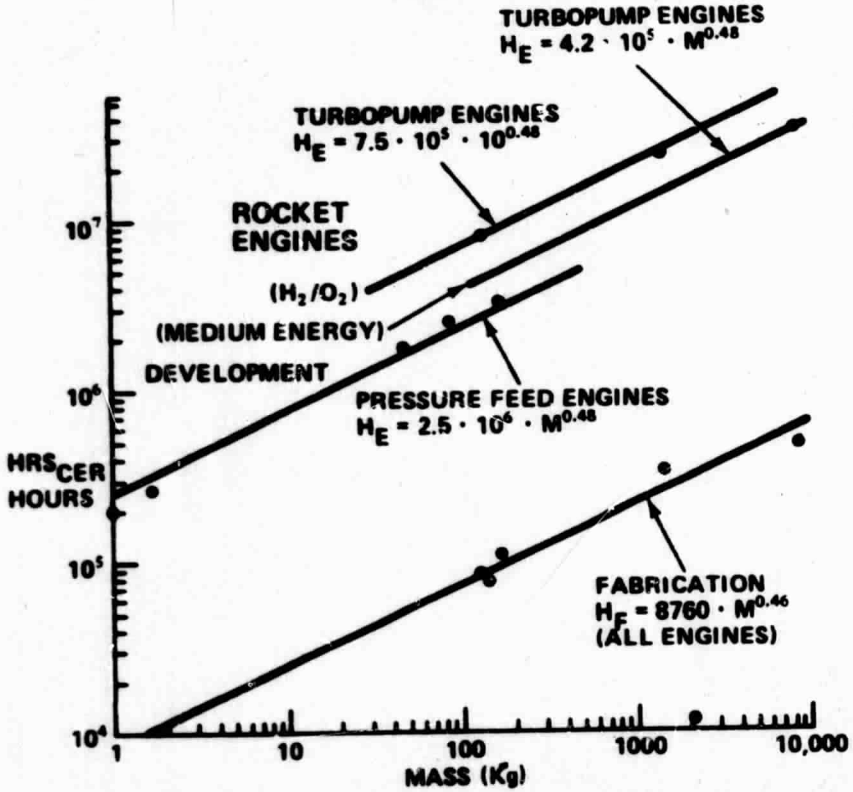
Cost Estimating Relationships

The facing page presents the cost estimating relationships (CER) used to establish ROM development and fabrication costs for the transportation systems. This data was presented at the "2nd Symposium on Cost Reduction in Space Operations" at the International Academy of Astronautics, 14 Oct. 1972.

COST ESTIMATING RELATIONSHIPS

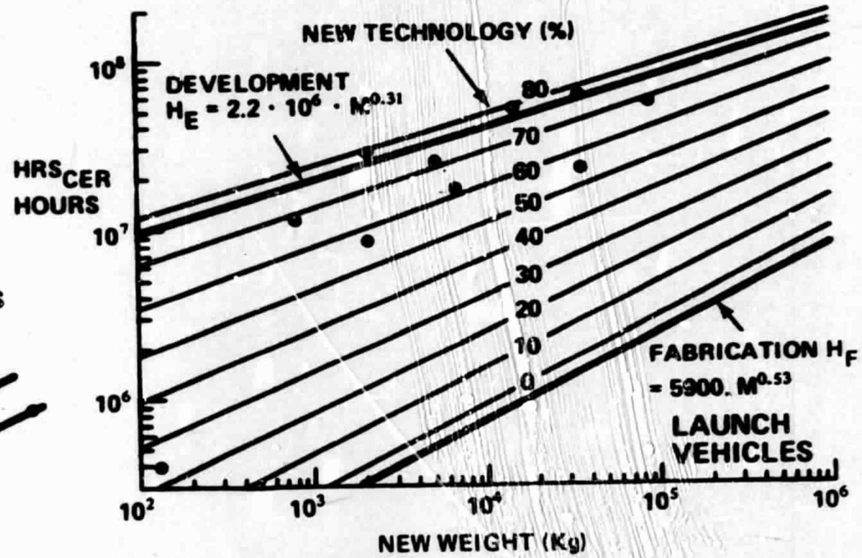
$\$ = 25\ \$/HR \times HRS_{CER}$

CER #1



DEVELOPMENT AND FABRICATION EFFORT FOR LIQUID PROPELLANT ROCKET ENGINES

CER #2



DEVELOPMENT AND FABRICATION EFFORT FOR LAUNCH VEHICLE STAGES VS. NET WEIGHT WITHOUT PROPELLANTS (EXCLUDING ENGINES; INCLUDING ADAPTERS)

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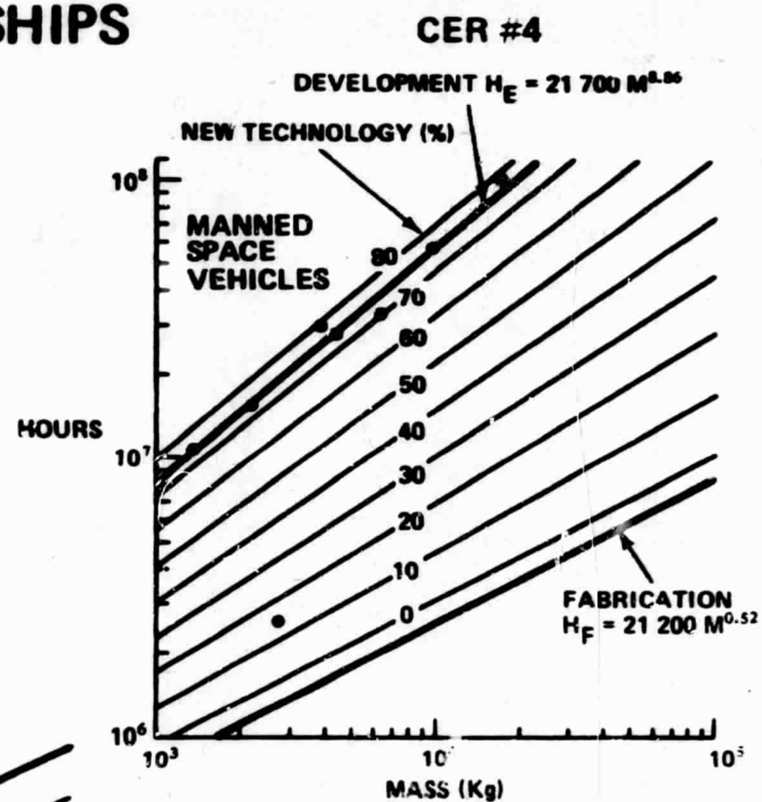
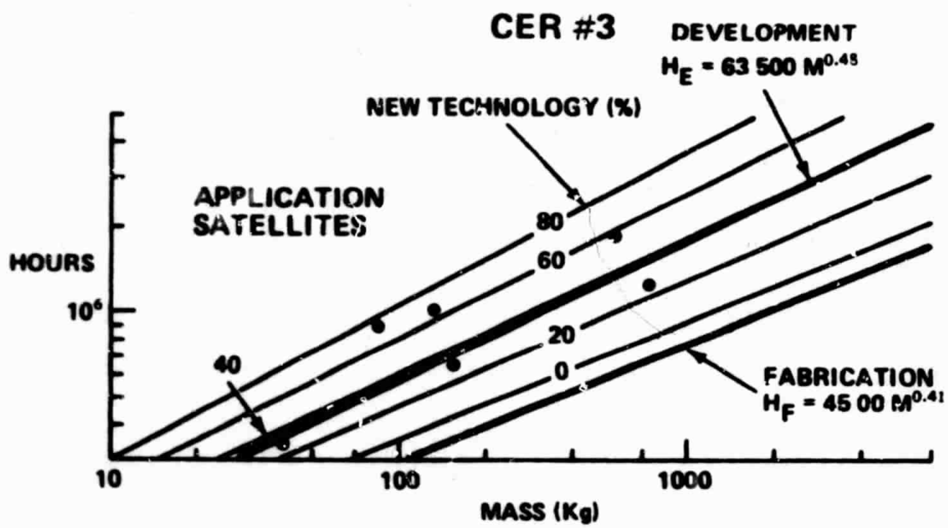
Cost Estimating Relationships

The facing page presents the CER's used to estimate the ROM development and fabrication costs for manned and unmanned assembly support equipments, and for the SSPS subsystems.

COST ESTIMATING RELATIONSHIPS



DEVELOPMENT AND FABRICATION EFFORT FOR MANNED SPACE VEHICLES VS. MASS (EXCLUDING PROPELLANTS)



DEVELOPMENT AND FABRICATION EFFORT FOR APPLICATION SATELLITES

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ROM Costs

Transportation/Assembly/Maintenance Equipment

The facing page lists the ROM costs for those support equipments that are directly related to SSPS and those system elements that are indirectly associated with SSPS. The indirect costs signifies that the identified system has a high potential of being developed independent of the SSPS program.

Included on the chart is the CER number used, the weight used in the estimate, the percent new technology assumed and the number of units used to establish production costs on an 85% learning curve.

ROM COSTS – TRANSPORTATION/ASSEMBLY/ MAINTENANCE EQUIPMENT

ELEMENT	CER # USED	WGT USED IN CER Kg	% NEW TECH	ROM DDT&E $M	UNIT PRODUCTION COST $M	IOC	COMMENTS	NO. OF PROD UNITS ON 85% LEARNING CURVE
DIRECTLY CHARGEABLE TO SSPS:								
• ASSEMBLY EQUIP								
– TELEOPERATORS	3	180	35	19	2.5	1985		300
– MAN. MANIPULATORS	4	1,940	75	365	11	1985		50
– EVA EQUIPMENT	4	90	75	26	1.5	1985		300
• LOGISTICS EQUIP.	3	–	0	44	–	1985	MODS TO ASSEMBLY EQ'P	–
• MAINTENANCE EQUIP.	3	–	0	44	–	1990	MODS TO ASSEMBLY EQ'D	–
• FABRICATION MOD.	3	4,540	50	271	12	1985	3 TYPES DEVEL'D	100
SUBTOTAL				769				
INDIRECT CHARGES:								
• LEO TRANSPORT								
– SHUTTLE	–			N/A	200	1980		
– DEPLOY ONLY LAUNCHER	2	286,000	30	380	150	1990	NO ENGINE DEVEL'MT	–
– HLLV	1&2	477,000	75	6540	400	1995	ENG WGT = 63,600 Kg/ENG (H_2/O_2 = TURBOPUMP)	–
• SO TRANSPORT								
– LARGE CRYO TUG	2	36,000	30	166	15	1990	DERIVATIVE OF ET/SSME	–
– ADVANCED ION	1&2	726,000	75	3847		1995		–
– PROPELLANT DEPOT	3	30,000	30	223	27	1990		100
– TUG FOR DEPOT	2	1,300	30	215	2.6	1990		300
• SO CREW TRN MODULE	4	11,640	20	190	23	1990		100
• LEO SPACE ST	4	76,450	50	2225	62	1990		50
• SO SPACE ST	4	76,450	0	224	62	1990		–
SUBTOTAL				14,010				

ROM Costs

SSPS System

The CER #3 was used to establish ROM development costs for the SSPS subsystems. The production unit costs used the costs predicted for the 1995 operational spacecraft as a goal. An 85% learning curve is used to establish cost goals for the 1985 Demonstration Satellite and the 1990 Pilot plant. If these cost targets for the early systems cannot be achieved, the feasibility of economic power generation from space using photovoltaic systems is questionable. For example, a 55$/M^2 target is established for the 1995 solar blanket. The 1985 cost target which is on an 85% learning curve is 233$/M^2. The development dollars spent between now and 1985 must be sufficient to establish high production, low cost solar blanket techniques in the 233$/M^2 range if confidence in the operational system cost is to be achieved.

ROM COSTS – SSPS SYSTEM

ELEMENT	CER # USED	WGT USED IN CER Kg	% NEW TECH	ROM DDT&E $M	UNIT PRODUCTION COST $M	IOC	COMMENT
• SOLAR BLANKET	3	9,892	80	340	233 $/M²	1985	• CER WGT IS ONE SOLAR BLANKET PANEL
		88,125	35	375	76 $/M²	1990	• PRODUCTION COST ON 85% LEARNING CURVE
		97,900	0	134	55 $/M²	1995	
• CONCENTRATOR		377	80	53	4.8 $/M²	1985	• CER WGT IS ONE MIRROR BLANKET
		4,843	35	93	1.5 $/M²	1990	• 85% LEARNING CURVE 1985–1995
		7,687	0	47	1.1 $/M²	1995	
• NON-CONDUCTING STRUCTURE		4,173	80	205	247 $/Kg	1985	• CER WGT REPRESENTS ¼ OF STRUCTURE PLANFORM
		152,500	35	488	107 $/Kg	1990	
		495,000	0	260	81 $/Kg	1995	
• CONDUCTING STRUCT		4,831	80	96	281 $/Kg	1985	• CER WGT REPRESENTS ¼ OF STRUCTURE PLANFORM
		112,500	35	422	94 $/Kg	1990	
		215,000	0	185	81 $/Kg	1995	
• REACTION CONTROL		200	80	240	$3.6M/UNIT	1985	• CER WGT REPRESENTS 1 FIVE LBF ION THRUSTER
		200	35	45	$2.8M/UNIT	1990	
		200	0	45	$2.1M/UNIT	1995	
• ROTARY JOINT		12,670	80	383	$40M/UNIT	1985	• CER WGT REPRESENT TOTAL WGT OF JOINT
		120,000	35	446	$105M/UNIT	1990	
		120,000	0	149	$90M/UNIT	1995	
• MN ANTENNA				SEE RAYTHEON EST			
• RECTENNA				SEE RAYTHEON EST			
SUBTOTAL				4006			

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Assembly Operations - 1995 Operational SSPS

The facing page lists the equipments, equipment costs and transportation flights needed to assemble a 17.93×10^6 Kg SSPS. The Heavy Lift Launch Vehicle is used as the ground to LEO transport system and an Advanced Ion Stage for transport from LEO to geosynchronous. Eighty percent of the assembly is assumed to be performed remotely, while the remaining 20% requires man-tended functions. A need for a Synchronous Orbit space station is assumed for purposes of final assembly, check out and maintenance.

The total transportation and assembly cost is estimated at $3264M or an equivalent 182$/Kg.

ASSEMBLY OPERATIONS – OPERATIONAL SSPS

- SSPS WGT = 17.92×10^6 Kg
- 2 YEAR ASSEMBLY PERIOD
- 80% REMOTE ASSEMBLY; 20% MANNED ASSEMBLY

ELEMENT	# UNITS	EQUIP COSR $M	WGT TO LEO Kg X 10^6	# HLLV FLTS	# SHUTTLE FLTS	COST $M
EQUIPMENT:						
• LEO SPACE STATION	12[5]	149[1]	0.92	5		158[1]
• SO SPACE STATION	1	62	0.076	1		63
• ASSEMBLY EQUIPMENT						
– MANNED MANIPULATORS	12	26[1]	0.023			
– TELEOPERATORS	218[6]	96[1]	0.039		3	191[1]
– EVA EQUIPMENT	204	61[1]	0.018			
• FABRICATION MODULES	3	10[1]	0.016			
• LARGE CRYO TUG	2	8[1]	0.072	1		7.8[1]
• SUPPORT TUGS	10	5.2[1]	0.013	1		7.0[1]
• CREW MODULE	1	5[1]	0.012		1	7.6[1]
• PROPELLANT STORAGE TANKS	26	83[1]	0.780	5		92[1]
• ORBIT MAINTENANCE MODULE	1	3.2	0.002			3.2
• ADVANCED ION	1	38[1]	0.726	4		45[1]
SUPPLY						
• CRYO PROPELLANTS			0.981[2]	5		45
• ION PROPELLANTS			0.772	4		36
• S/S & EQUIP. RESUPPLY			0.772	4		36
• CREW ROTATION					72[3]	936
MATERIAL TRANSPORT			17.92	99		891
• SUBTOTAL		544	23.142	129	76	2519
• PERSONNEL	858					77[4]
• AMORTIZE L/V COST				516	152	668
TOTAL						3264
TRANSP & ASSEMBLY COST $/Kg						182

ASSUMPTIONS

(1) COST AMORTIZED OVER 5 SSPS
(2) THREE TRIPS TO SO
(3) CREW CYCLED EVERY 90 DAYS
(4) • 45,000/YR/MAN
  • 204 ORBIT OPS MEN
  • 654 GRND TELEOPERATORS
(5) 9 Kg/HR ASSEMBLY RATE OF 20% OF SSPS; 56 HR WORK WEEK
(6) 4.5 Kg/HR ASSEMBLY RATE OF 80% OF SSPSS; 20 HRS PER DAY UP TIME

Maintenance Cost - SSPS Spacecraft

The facing page summarizes the analysis of maintenance requirements for the Spacecraft. The following results were determined:

- A trade-off between cost of repair versus the loss of revenue if no repair is performed, indicated that an LRU should not be replaced before power degrades more than 5.6%
- The subsystem requiring the most repair is the control system
- Proper series/parallel lay-out of the solar blanket circuitry and microwave tube feed system could result in a near maintenance-free design.

MAINTENANCE COST – SSPS SPACECRAFT

ELEMENT	LRU DESCRIPTION	LRU WGT Kg	LRU FAIL OVER 30 YRS	AVE $/YR $M	COMMENTS (LRU REPLACED AFTER 5.6% POWER REDUCTION DUE TO FAILURE)
SOLAR ARRAY					
• BLANKET	80–1670 X 207 M MODULES	97,900	1	1.4	2.6 X 10^{-4} /YR OPEN CIRCUIT FAILURE RATE (OAO)
• CONCENTRATOR	160–1670 X 207 M MODULES	7,687	1	0.01	MIRROR FAILURE LESS LIKELY THAN BLANKET
• NON COND STRUCT	TO DESIGN	–	–	–	STRUCTURE ASSUMED NOT TO FAIL
• COND STRUCT					
– BUSSES	400 M LONG	26,000	1	0.28	10^{-9} F/YR (OAO)
– SWITCHES	59 BLOCK DIODES/BLANKET LRU	97,484	1	1.40	10^{-7} F/YR (OAO)
• MAST	6 (+), 6(–) BUSSES/PANEL	85,000	1	0.9	SAME AS CONDUCTING STRUCTURE
MW ANTENNA					
• TUBES	1670–18 X 18 M SUBARRAYS	3,017 Kg	4	0.19	MTBF = 1.14 X 10^{-6} HRS PROJECTED
• POWER DIST	18 X 18 M SUBARRAY	3,017 Kg	1	0.05	HIGHLY REDUNDANT SYSTEM
• ELECT	1670 UNITS	467 Kg	3%	0.69	HIGHLY REDUNDANT SYSTEM
• TRANS ANTENNA	1670 UNITS	3,017 Kg	1	0.05	WAVEGUIDE CONSIDERED STRUCT- NO FAILURES
• STRUCTURE	TO DESIGN	–	–	–	ASSUMED NOT TO FAIL
• CONTOUR CONT	6680 UNITS	220 Kg	1404	0.01	0.8F/10^{-6} (1% DUTY FACTOR)
ROTARY JOINT					
• SLIP RING	4 UNITS	10 Kg	72	0.01	MTBF = 10 YRS (SPACE STATION STUDIES)
• BRUSH	24 UNITS	63 Kg	12	0.01	MTBF = 10 YRS (SPACE STATION STUDIES)
• DRIVE	8 BRUSHLESS DC MOTORS	1,367 Kg	24	0.37	MTBF = 10 YRS (SPACE STATION STUDIES)
REACTION CONTROL	64 ELECTRIC ENGINES	203 Kg	640	33.0	3800F/10^{6} HR (ORDER MAGNITUDE IMPROVEMENT + 10% DUTY FACTOR)
PROPELLANT		24,000 Kg		5.7	YEARLY CONSUMPTION

• SUBTOTAL	44.1
• SO CREW ROTATION	73.0
• MISSION CONTROL	14.0
TOTAL	131.1 (3 MILLS/KWH)

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SSPS Unit Cost

The facing page lists the costs of the Operational SSPS (1995) by subsystem. The total cost is $6.21B or an equivalent 1242$/KW.

SSPS UNIT COST

ELEMENT	WGT X 10^6 Kg	DESIGN VARIABLE	UNIT COST	T&A*** COST	COST $B
SOLAR ARRAY					
• BLANKETS	7.83	27.8 KM^2	55 $/$M^2$	182 $/Kg	2.95
• CONCENTRATORS	1.23	61.1 KM^2	1.1 $M^2		0.29
• STRUCTURE	2.22	2.22 X 10^6 Kg	81 $/Kg		0.58
• MAST	0.62	0.62 X 10^6 Kg	81 $/Kg		0.16
MICROWAVE ANTENNA**					
• POWER DIST	0.52	5 GW	9.5 $/KW		0.14
• TRANS ANTENNA	2.32	5 GW	13 $/KW		0.49
• TUBES	2.34	5 GW	11.5 $/KW		0.48
• MECHANICAL SYS*	0.59	5 GW	12.6 $/KW		0.17
• COMMAND & CONTROL	0.78	5 GW	0.9 $/KW		0.14
RECTENNA**	—	5 GW	161 $/KW		0.81

TOTAL	6.21
$/KW	1242

*INCLUDES ROTARY JOINT
**SEE RAYTHEON MPTS STUDIES
***T&A = TRANSPORTATION & ASSEMBLY COST

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SSPS Satellite Cost Summary

The facing page summarizes the SSPS program costs through the first operational unit. The \$20.7B excludes development costs for the microwave systems but includes microwave costs in the "Unit Cost." The directly SSPS chargeable support equipments, such as fabrication modules and teleoperators, are included. The \$14B required for development of transportation system and space stations are excluded because these elements are of general service to many programs.

This preliminary plan indicates that the intermediate 1990 Pilot Plant is the costly element in the development. The high unit cost is the result of not introducing the Heavy Lift Vehicle earlier. A programmatic sensitivity study is required to determine the cost effective time to introduce technology. It is recommended that a computer simulation of the SSPS program be developed. This tool would significantly reduce the effort required to perform these programmatic trade studies.

SSPS SATELLITE COST SUMMARY

	1985 ORBITING TEST FACILITY	1990 PILOT PLANT	1995 OPERATIONAL PLANT
POWER LEVEL	15 MW (GENERATED)	1 GW (GROUND)	5 GW (GROUND)
WGT	228,000 Kg	8.3×10^6 Kg	17.9×10^6 Kg
DDT&E	\$2,000 M	\$1900 M	\$720 M
UNIT COST	\$840 M	\$9050 M	\$6210 M
MAINTENANCE/YR	—	—	\$131 M

* 1) DDT&E COSTS, EXCLUDES MICROWAVE SUBSYSTEM; 2) INCLUDES DIRECTLY CHARGEABLE SUPPORTING EQUIPMENTS FOR ASSEMBLY; 3) EXCLUDES INDIRECT DDT&E CHARGES.

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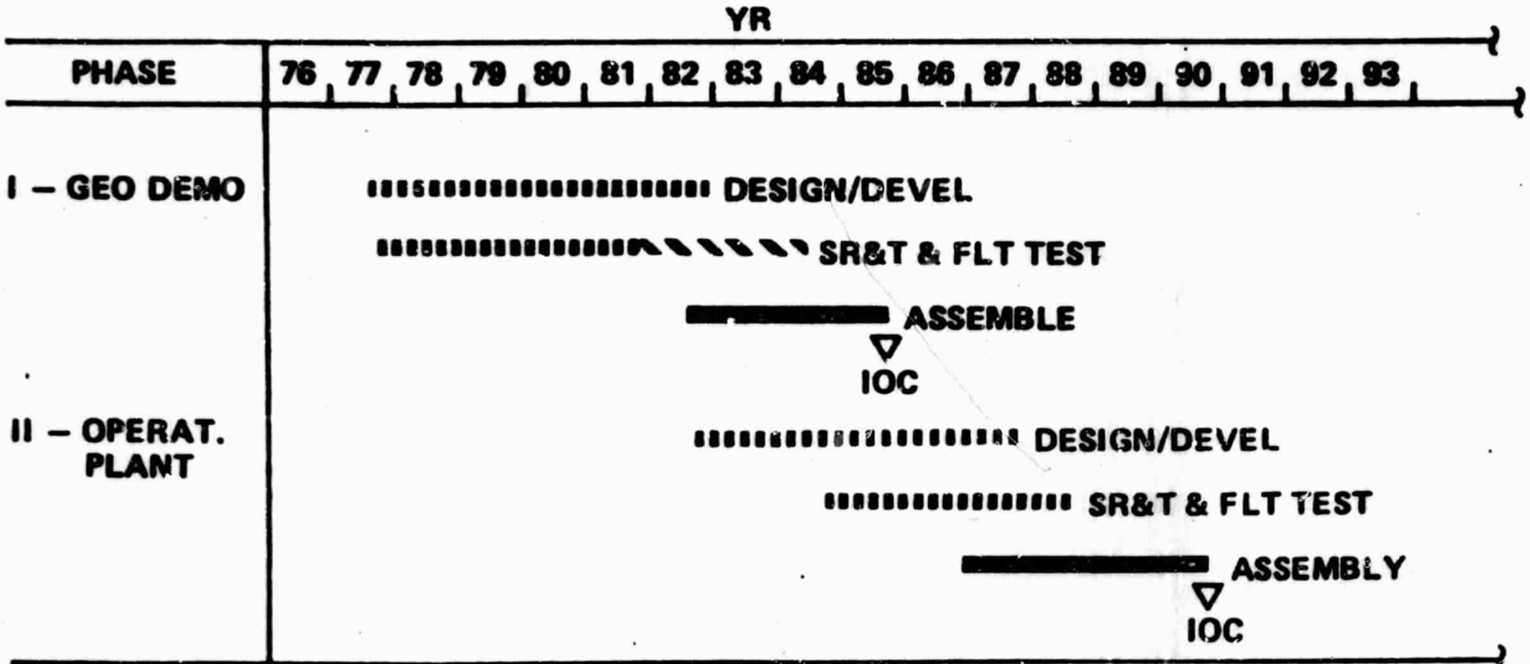
PRS Orbital System Program Schedule and Cost

The facing page is a preliminary schedule and cost estimate for one PRS program option. A 1Km demonstration satellite is scheduled for geosynchronous deployment in 1985 and the operational satellite placed in 1990.

The early deployment of such a large structure in geosynchronous orbit using only Shuttle and a Large Tug does not appear to provide an attractive program. Delay of the program to at least wait for the development of a Shuttle derivative DOL would reduce cost significantly; or deployment of a small LEO satellite for demonstration in 1985 would also be more attractive.

PRS ORBITAL SYSTEM PROGRAM SCHEDULE AND COST



	1985 SYSTEM	1990 SYSTEM
WGT	.581 X 10^6kg	0.505 X 10^6kg
DDT & E	$1696M	$264M
UNIT COST	$2491M	$567M
MAINTENANCE	–	$90 M/YR

TOTAL PROGRAM THRU 1ST OPER. UNIT = $5.1B

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