BOEING

GENERAL ELECTRIC

CR-151673

D180-24071-3

NAS9-15196
DRL T-1346
DRD MA-665T
LINE ITEM 4

Part III
Final Briefing
March 7, 1978

Solar Power Satellite

SYSTEM DEFINITION STUDY PART III

D180-24071-3

Solar Power Satellite

SYSTEM DEFINITION STUDY PART III

Part III
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Approved:

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Solar Power Satellite
Systems Definition Study
Part III

Final Briefing

AGENDA

INTRODUCTION AND HIGHLIGHTS	G. WOODCOCK
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DEVELOPMENT REQUIREMENTS FOR INITIAL SPS COMMERCIALIZATION	D. GREGORY

PART 3 HIGHLIGHTS

The principal design change made in the reference system was to modify the transmitter design for maintainability and crew access.

At the completion of Part II, the comparison between ballistic and winged launch vehicles indicated difficulties with the winged vehicle: inadequate payload bay volume and requirement for downrange recovery of the booster. A winged vehicle has now been configured with larger payload volume and with a flyback booster. Its cost per flight is competitive with the ballistic sea landing vehicle.

A preliminary maintenance analysis was completed. The estimated cost for manned SPS maintenance (every six months) is about 3% of the power cost attributable to initial system capital cost: approximately 1 mill per kilowatt hour.

The far sidelobes study for SPS transmitters showed that the grating lobes are widely separated points rather than rings. This minimizes concern with overlap of grating lobes. The levels of grating lobes can be reduced by improved mechanical pointing. The dominant contributor to grating lobe magnitude is the saw-toothing of the wave front from the subarray that results from mechanical aiming errors.

Requirements for SPS demonstration are not clear at the present time. It is possible that development might include a technical demonstration of economic viability. This would be likely in scenarios including a commercial funding contribution to the development program. Technical demonstration options were evaluated and a preferred demonstration approach selected.

Part III Highlights

SPS-1986

- TRANSMITTER DESIGN MODIFIED FOR MAINTAINABILITY
- TWO-STAGE WINGED LAUNCH VEHICLE COMPETITIVE WITH BALLISTIC OPTION
- MAINTENANCE COSTS ADD ≈3% TO POWER COST
- TRANSMITTER GRATING LOBES ARE WIDELY SEPARATED POINTS —LEVELS CAN BE REDUCED BY IMPROVED MECHANICAL POINTING
- SPS DEMONSTRATION OPTIONS EVALUATED

SYSTEM SELECTION RATIONALE

A summary of the rationale for system selection is presented on the facing page. This rationale has been aimed at maximizing credibility of the SPS concept and at improving confidence in mass cost and technology estimates. If the resulting system had been too massive or too costly then it would have been necessary to step forward to more advanced technology as a reference design. However, the results we have found indicate that the system selected is adequate in terms of mass, performance, and cost.

System Selection Rationale

- CREDIBILITY OF THE SPS SYSTEM CONCEPT IS MAXIMIZED BY MINIMIZING TECHNOLOGY EXTRAPOLATIONS
- SYSTEM DEFINITION UNCERTAINTY IS MINIMIZED BY MINIMIZING TECHNOLOGY EXTRAPOLATION
- THE RESULT IS AN ADEQUATE SYSTEM
- MORE ADVANCED TECHNOLOGY WILL EVENTUALLY YIELD BETTER SYSTEMS.

SELECTION EXAMPLE

Three examples of the selections among various options are indicated here. Single crystal silicon was selected over thin film gallium arsenide because of its greater technology and production base and better overall understanding. The transmitter selection of klystrons over amplitrons was comparatively arbitrary and principally motivated in order to develop design detail on the klystron option. The launch system was a comparatively conservative technology two-stage reusable rocket similar in many respects to the fully reusable shuttle concepts examined in 1970 and 1971 except for its larger size.

Selection Examples

- SOLAR CELL OPTIONS:
  - SINGLE-CRYSTAL SILICON
  - THIN-FILM GALLIUM ARSENIDE
  - OTHER THIN FILMS

- TRANSMITTER OPTIONS:
  - KLYSTRON
  - AMPLITRON
  - SOLID-STATE

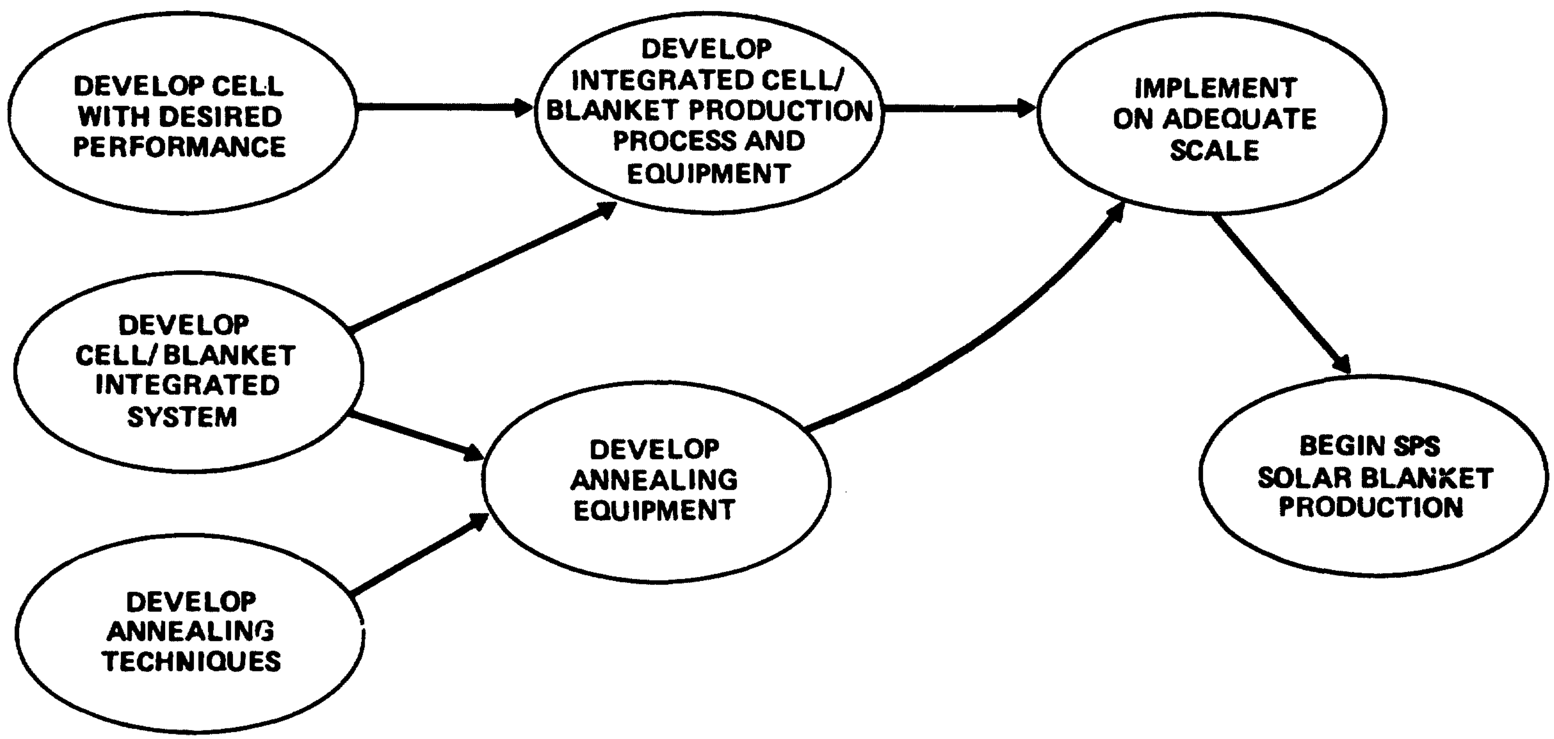
- LAUNCH SYSTEM OPTIONS:
  - TWO-STAGE ROCKET
  - SSTO ROCKET
  - AIRBREATHER/ROCKET MIXES
  - LASER AND OTHER ADVANCED PROPULSION

SINGLE CRYSTAL SILICON

Although single crystal silicon has been regarded as a low technology selection, a serious examination of technology advancement requirements indicate that several difficult tasks are involved in attaining the needed performance and production levels. It is quite likely that these tasks represent the schedule tent pole for an SPS program.

Single-Crystal Silicon

SINGLE-CRYSTAL SILICON IS OFTEN VIEWED AS A "LOW TECHNOLOGY" APPROACH TO SPS . . . . .

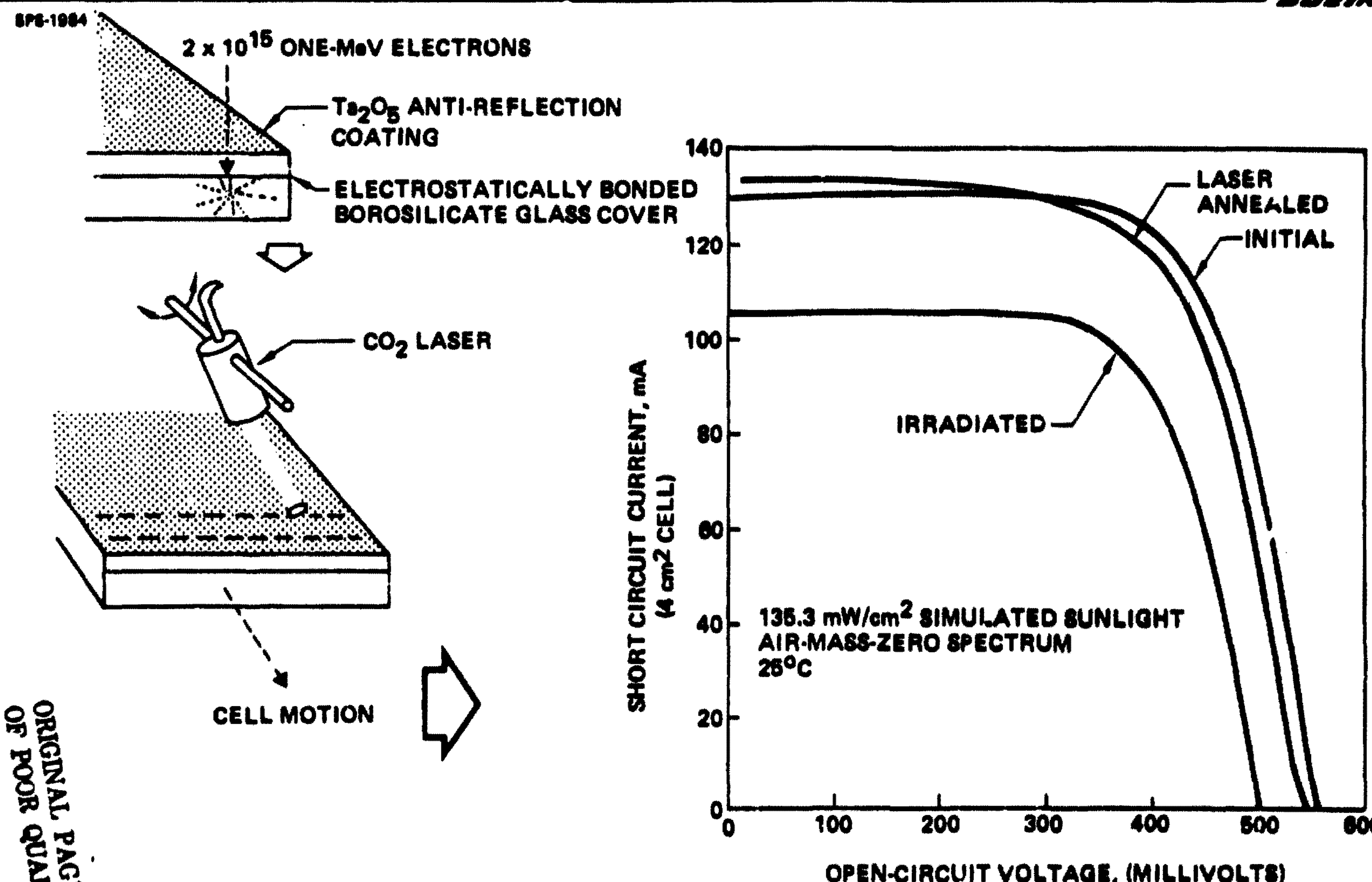


BUT THIS APPEARS TO BE THE SCHEDULE TENT POLE FOR AN SPS PROGRAM.

THERMALLY ANNEALED SILICON SOLAR CELLS

The directed energy annealing effort was continued into Part III and the principal result was a successful anneal of solar cell using a laser. Illustration of the technique and the performance achieved are shown on the chart.

Thermally Annealed Silicon Solar Cells



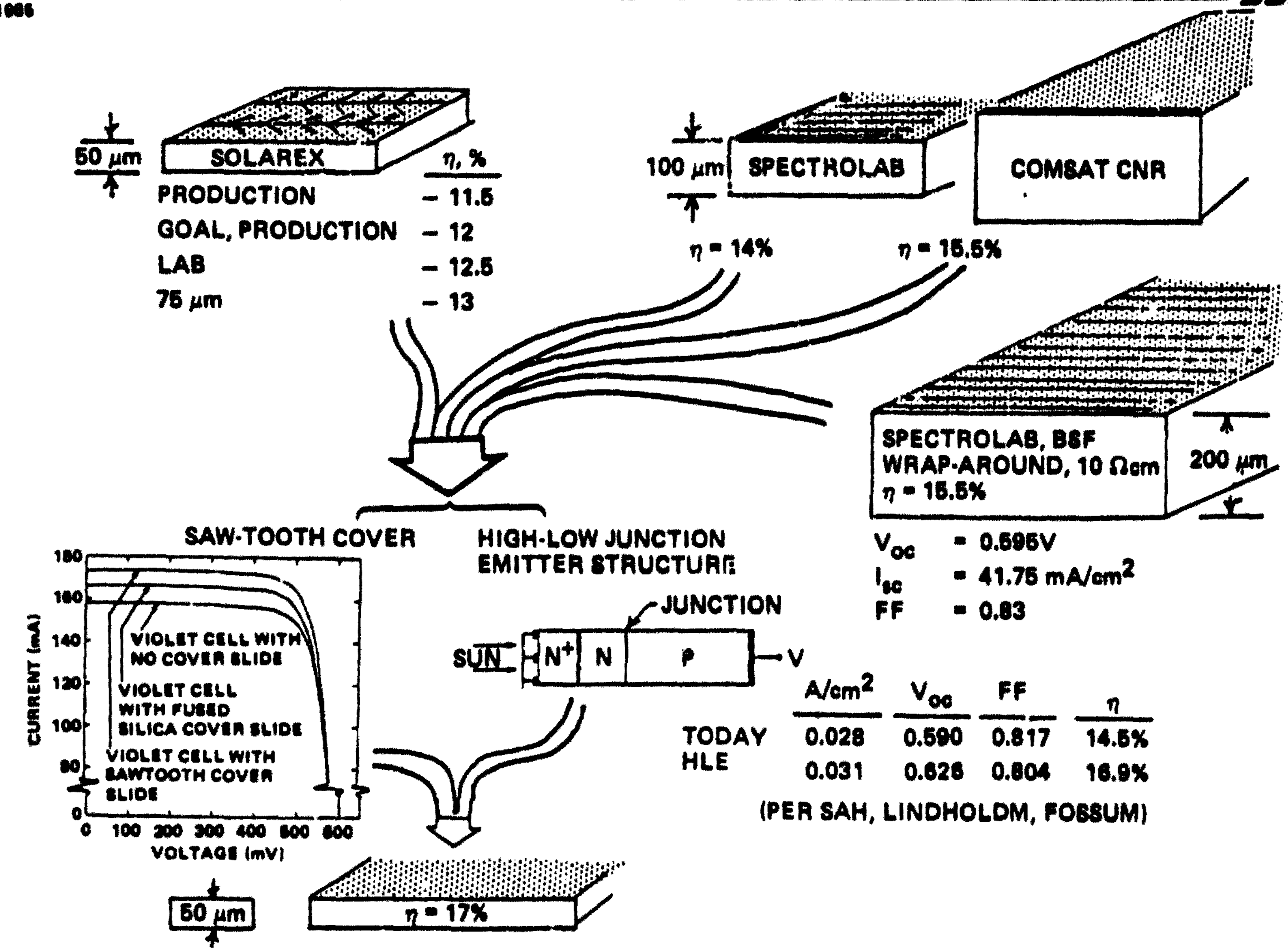
ROUTES TO HIGH EFFICIENCY IN SILICON SOLAR CELLS

An approach to achieving the desired cell performance is reasonably clear although the elements of the technology have never been combined in a single device, and one of the elements has not been demonstrated experimentally. Current 50 micrometer single crystal silicon cells are nearing 12% efficiency. Thicker cells are in the 14-15½ range. A combination of the COMSAT non-reflective sculpturing technique with the hi-lo junction emitter technique (and possibly back surface field), combined with diffractive saw-tooth glass covers, is predicted to yield a 17% efficient 50 micrometer solar cell.

SPS

Routes to High Efficiency in Silicon Solar Cells

SPS-1985 BOEING



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SPS WORK BREAKDOWN STRUCTURE

The SPS system description follows this work breakdown structure. The structure is basically similar to that used in Part 2 with the exception of adding maintenance items under the SPS Space Construction block. This briefing shows only significant changes to the system description included in the final briefing and final documentation of Part 2. The system description document, to be provided as a part of the Part 3 documentation, will present a formal system description in its entirety.

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SPS Work Breakdown Structure

SPS-1996

SPS PROGRAM

- **SOLAR POWER SATELLITE**
  - SUPPORT SUBSYSTEMS
  - ENERGY COLLECTION
  - ENERGY CONVERSION
  - POWER DISTRIBUTION
  - MICROWAVE POWER TRANSMISSION
- **GROUND RECEIVING STATION**
  - REAL ESTATE
  - CONTROL AND COMMUNICATION
  - PRIMARY STRUCTURE
  - ENERGY COLLECTION
  - POWER DISTRIBUTION AND PROCESSING
- **SPS SPACE CONSTRUCTION AND MAINTENANCE**
  - LOW ORBIT BASES
  - GEOSYNCHRONOUS BASES
  - MOBILE MAINTENANCE BASES
  - SATELLITE-BASED MAINTENANCE EQUIPMENT
  - OPERATIONS AND SUPPORT
- **SPACE TRANSPORTATION**
  - CARGO LAUNCH VEHICLE
  - PERSONNEL LAUNCH VEHICLE
  - ORBIT TRANSFER VEHICLE
  - SPS-INSTALLED ORBIT TRANSFER SYSTEMS

REFERENCE SYSTEM POWER BUDGET AND SIZING CRITERIA

These are the factors used in calculating the solar array power output. We start with solar cells having 15.75 percent efficiency. To this we add a 10 percent improvement, which could be achieved by any one of several means. For example, A. Meulenberg of COMSAT Laboratories estimates that the sawtooth cover that he invented will improve the efficiency of solar cells by 8 to 12 percent.

The blanket factors of 0.9453 account for the power losses shown. The individual elements of the blanket factors will change, but the product will probably remain around 0.9453.

The summer solstice loss accounts for the 23.5 degrees mis-orientation with respect to the Sun's rays. This loss could be avoided by having the satellite oriented perpendicular to the ecliptic plane, but the cost in thrusters and propellants required for attitude control in that mode shows to no real advantage.

The aphelion intensity factor accounts for the reduced solar intensity when the Earth is at its aphelion, around the first part of July.

The temperature losses result from the solar cells operating between 36.5°C and 46°C, rather than at the 25°C at which cell efficiency is commonly tested.

The output is further reduced by 3 percent to account for radiation damage that cannot be removed by thermal annealing. In past tests, 95 percent of the radiation damage in solar cells has been annealed out, even though the cells had not been designed for thermal annealing. There is no theoretical reason why all of the radiation damage in solar cells cannot be annealed out, annealing temperatures of around 500°C being well below the 800°C region where diffusion of impurities starts. On the other hand, the operating plan for the solar power satellite involves repeated annealings, which have not been attempted by anyone, as far as we know.

Those sections of the solar array blanket that are used for the orbit transfer power supply, are subjected to a significantly higher radiation degradation than the stowed solar array. If, after annealing, the highly degraded portions of solar array can only be restored to 95 percent of their initial output, a penalty results. This is compensated for by an orbit transfer power compensation factor and by increasing the solar cell string lengths.

The array power requirement of 18.31×10^9 watts is based on providing a ground output of 10.0×10^9 watts using the current efficiency chain from the array to the grid interface.

The array power requirement and the effective blanket output determine the solar cell area requirement. The increase shown for the array area compensates for the lost areas in the solar array blanket. Another area increase, to compensate for non-array lost area, is necessary to establish the total projected satellite area.

Reference System Power Budget and Sizing Criteria

BOEING

SPS-1847

- EFFECTIVE BLANKET OUTPUT—180.6 w/m^2 (E.O.L.)
  - BASIC CELL PERFORMANCE (0.1575 @ AMO-25°C) – 213.1 w/m^2
  - 10% IMPROVED PERFORMANCE—DUE TO TEXTURED COVERS – 234.4 w/m^2
  - BLANKET FACTORS—STRING I^2R, UV LOSSES, & MISMATCH (0.9453) – 221.6 w/m^2
  - TEMPERATURE LOSSES—36.5°C @ SUMMER SOLSTICE (0.9540) – 211.4 w/m^2
  - SUMMER SOLSTICE COSINE, LOSSES (0.9190) – 194.3 w/m^2
  - APHELION INTENSITY FACTOR (0.9675) – 188 w/m^2
  - 30-YEAR NON-ANNEALABLE RADIATION DEGRADATION (0.970) – 182.3 w/m^2
  - ORBIT TRANSFER COMPENSATION (.9906) – 180.6 w/m^2

- ARRAY POWER REQUIREMENT—18.31 $(10)^9$ WATTS
  - GROUND OUTPUT – 10.0 $(10)^9$ WATTS
  - SLIP RING TO GROUND OUTPUT EFFICIENCY LINK (1.693) – 16.93 $(10)^9$ WATTS
  - SATELLITE BUS I^2R LOSSES (1.071) – 18.13 $(10)^9$ WATTS
  - OVERSIZE—REGULATION, AUX. PWR., ANNEALING (1.01) – 18.31 $(10)^9$ WATTS

- SOLAR CELL AREA REQUIREMENT—101.4 km^2
- ARRAY AREA REQUIREMENT (INCLUDES LOST AREAS ON ARRAY)—110.2 km^2
- TOTAL SATELLITE AREA (EXCLUDING ANTENNAS)—114.5 km^2

REFERENCE PHOTOVOLTAIC SYSTEM DESCRIPTION

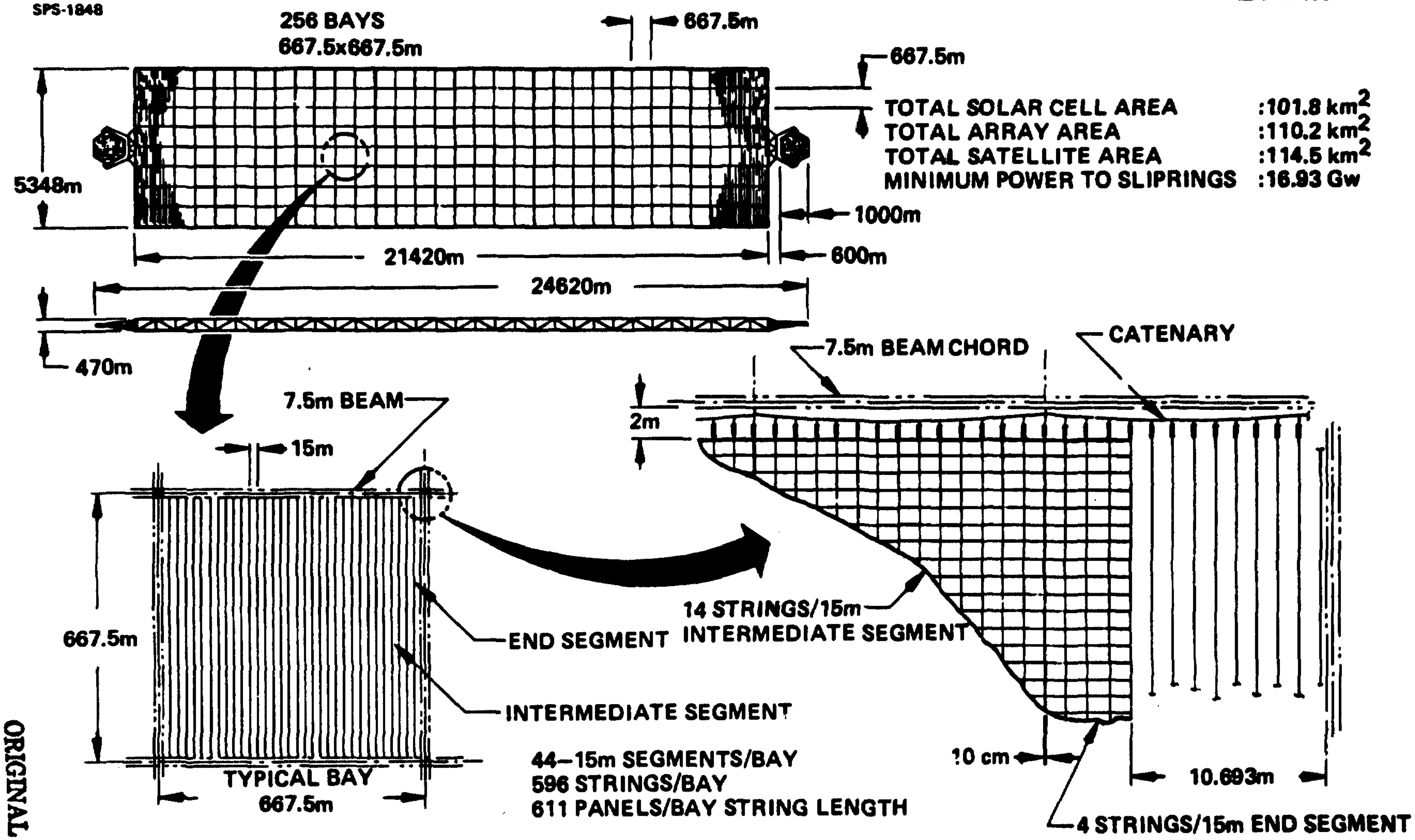
The Part II silicon photovoltaic system provided an output of 4650 megawatts per antenna. To normalize this output to 5000 megawatts it was necessary to increase the satellite bay size to 667.5 meters which was more than adequate to satisfy the increased area requirement.

Shown here is the final reference system size and configuration. Details are shown of a typical bay and the array support within the bay.

The array segment width was changed to 14.9 meters. This change provided better packaging for transport but made it necessary to provide 15 meter catenary attachment points on the structural beams. A 10 cm spacing was provided between array segments for clearance during array deployment.

Reference Photovoltaic System Description

SPS-1848



REFERENCE PHOTOVOLTAIC SYSTEM DESCRIPTION

This is the basic panel adopted for design studies. It has a matrix of 224 solar cells, each 6.55 by 7.44 cm in size, connected in groups of 14 cells in parallel by 16 cells in series. The cells are electrostatically bonded between two sheets of borosilicate glass. Spacing between cell and edge spacings are as shown. Tabs are brought out at two edges of the panel for electrically connecting panels in series. Cells within the panel are interconnected by conducting elements printed on the glass substrate.

Important panel requirements were these:

- o The panel components and processes should be compatible with thermal annealing at 500°C.
- o Presence of charge-exchange plasma during ion-engine operation may necessitate insulating the electrical conductors on the panel.
- o The panel design should be appropriate for the high-speed automatic assembly required for making the some 93 million panels required for each satellite.
- o Low weight and low cost are important.

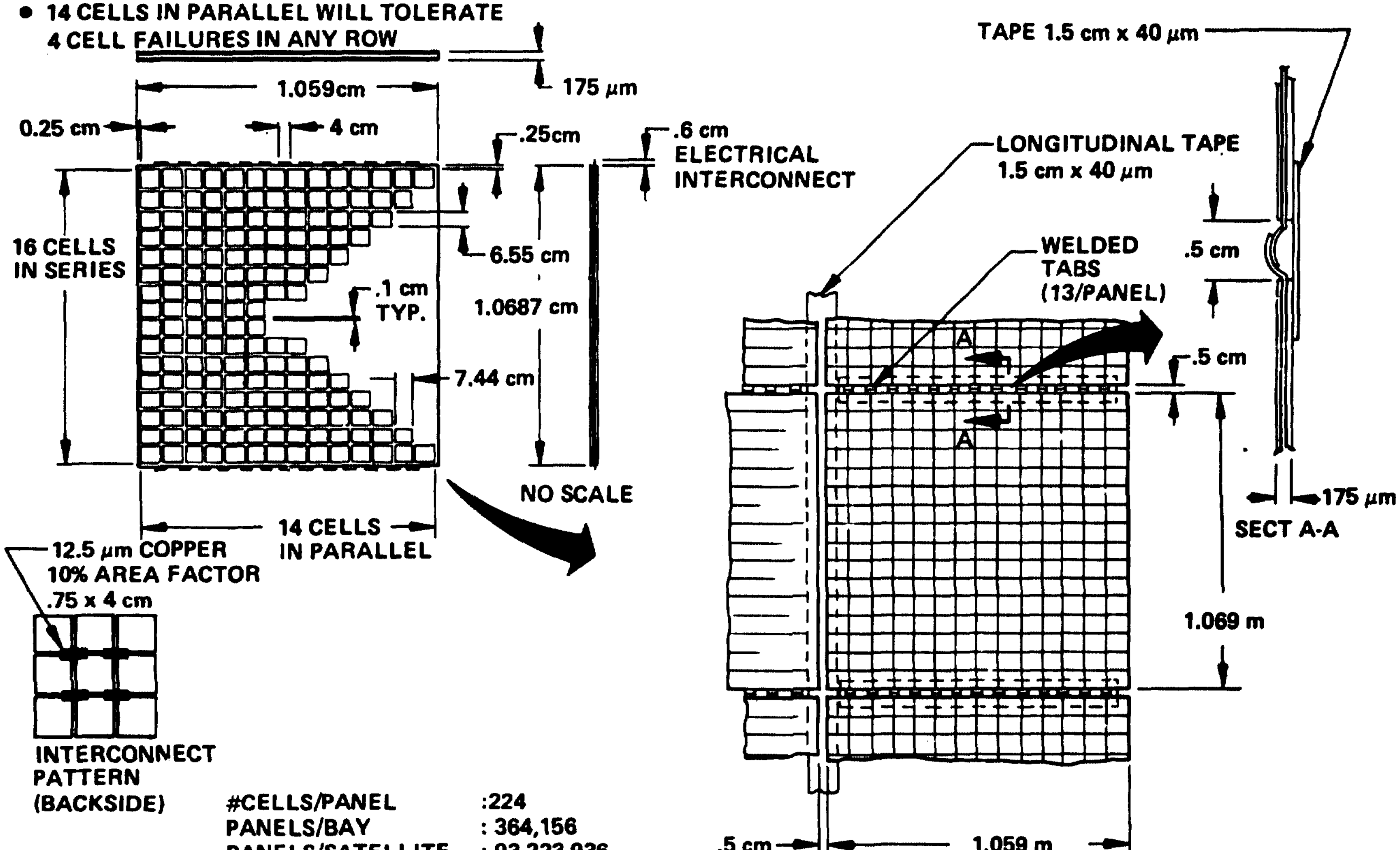
Also shown here is the way panels would be assembled to form larger elements of the solar array. The interconnecting tabs of one panel are welded to the tabs of the next panel in the string, and then the interconnections are covered with a tape that also carries structural tension between panels. After joining, the panels are accordion-folded into a compact package for transport to the low-Earth-orbit assembly station.

The 0.5 cm spacing between panels provides room for the welding electrodes, and also permits reasonable tolerances in the large sheet of 75 μm glass that covers the cells and the 50 μm sheets of substrate glass.

Reference Photovoltaic System Description

SPS-1849

- 14 CELLS IN PARALLEL WILL TOLERATE 4 CELL FAILURES IN ANY ROW



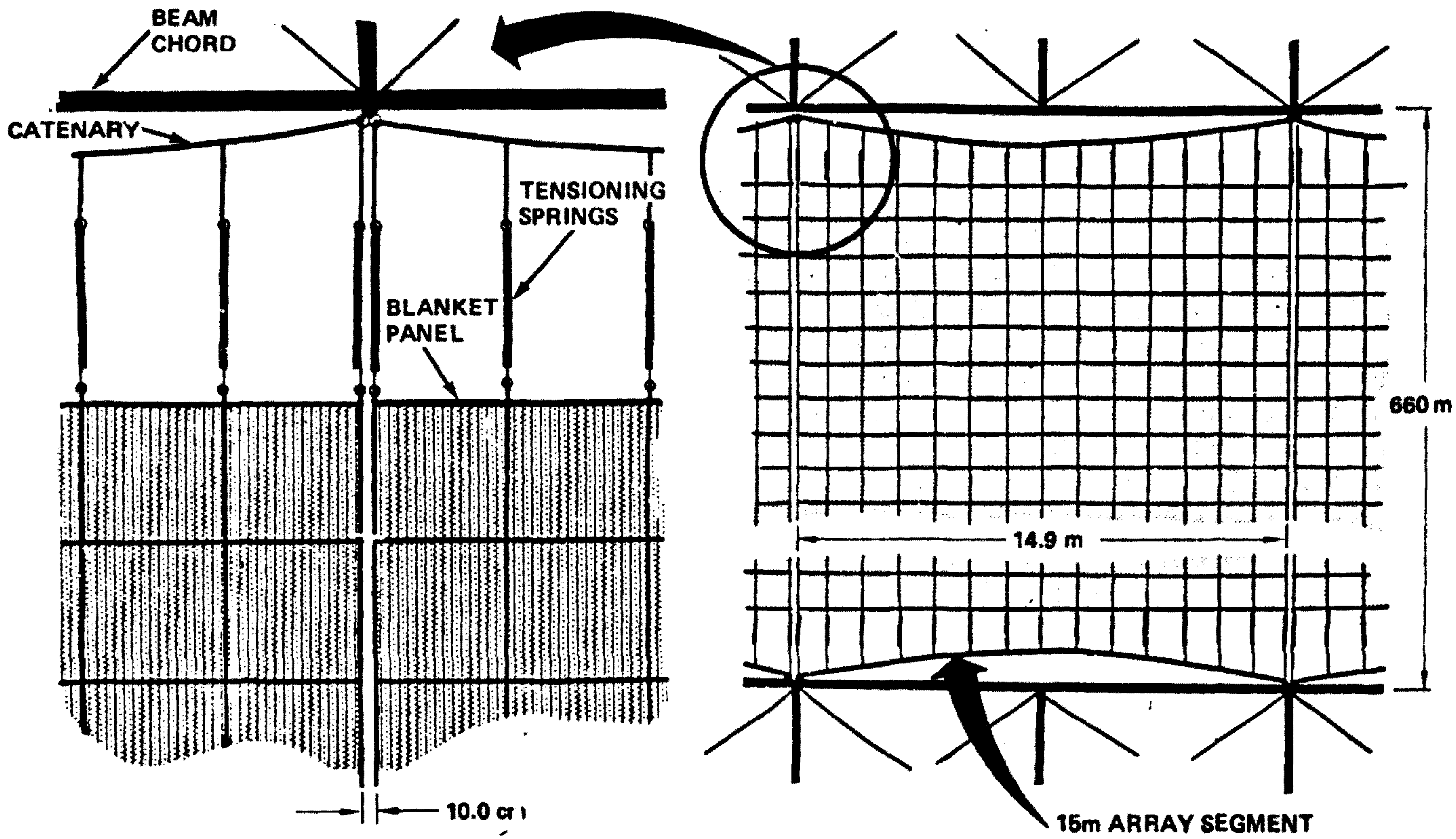
#CELLS/PANEL	:224
PANELS/BAY	: 364,156
PANELS/SATELLITE	: 93,223,936

REFERENCE ARRAY BLANKET SUPPORT

This illustration shows the method of providing tension to the solar array blanket segments. This method of support will provide a uniform tension to the end of each array segment by the use of constant-force compression springs at each blanket support tape.

A uniaxial blanket support was selected over the biaxial support shown in Part II of this study. This change was the result of analysis of construction techniques and associated blanket uniformity problems. It will be necessary to provide batten tapes between blanket segments, at a few intervals along the segment length, to provide correct segment-segment orientation.

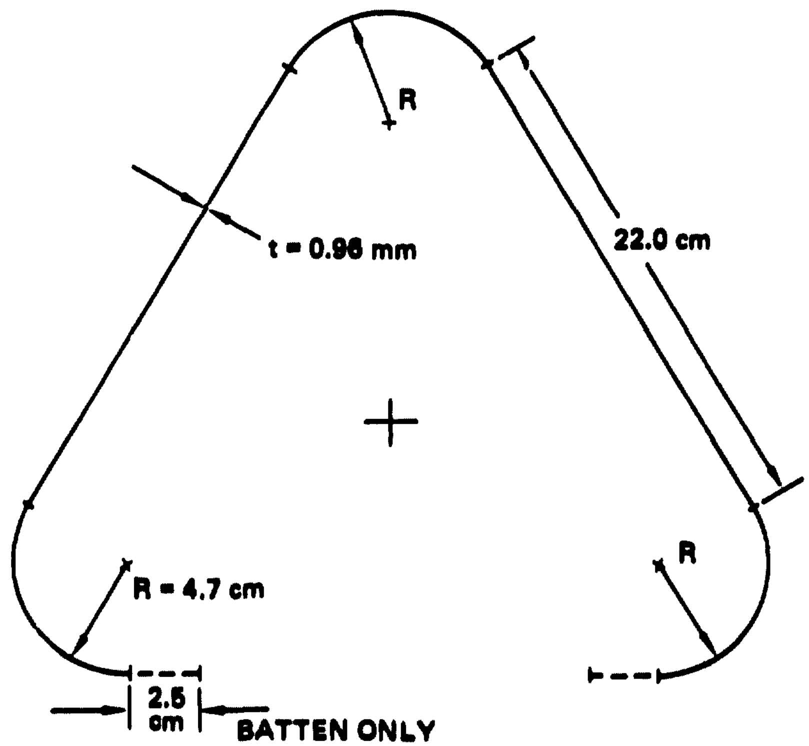
Reference Array Blanket Support



CONTINUOUS CHORD/BATTEN CONFIGURATION

Shown here is the basic configuration and dimensions of the continuous chord beam elements. The load carrying capability of the continuous chord beam is very sensitive to EI_x of both the chord and batten. The analysis suggested that the chord and batten configuration should be the same for optimum performance. This is also beneficial from the fabrication standpoint since the same type of equipment can be used to make the chord and batten. The major differences between the chords and battens are that the battens have an additional 2.5 cm of material on each side of the open face for bonding to the chord and the battens are terminated at beam widths.

Continuous Chord/Batten Configuration



MATERIAL: P-1700 GRAPHITE (POLYSULFONE IMPREG)
E-181 GLASS COVER

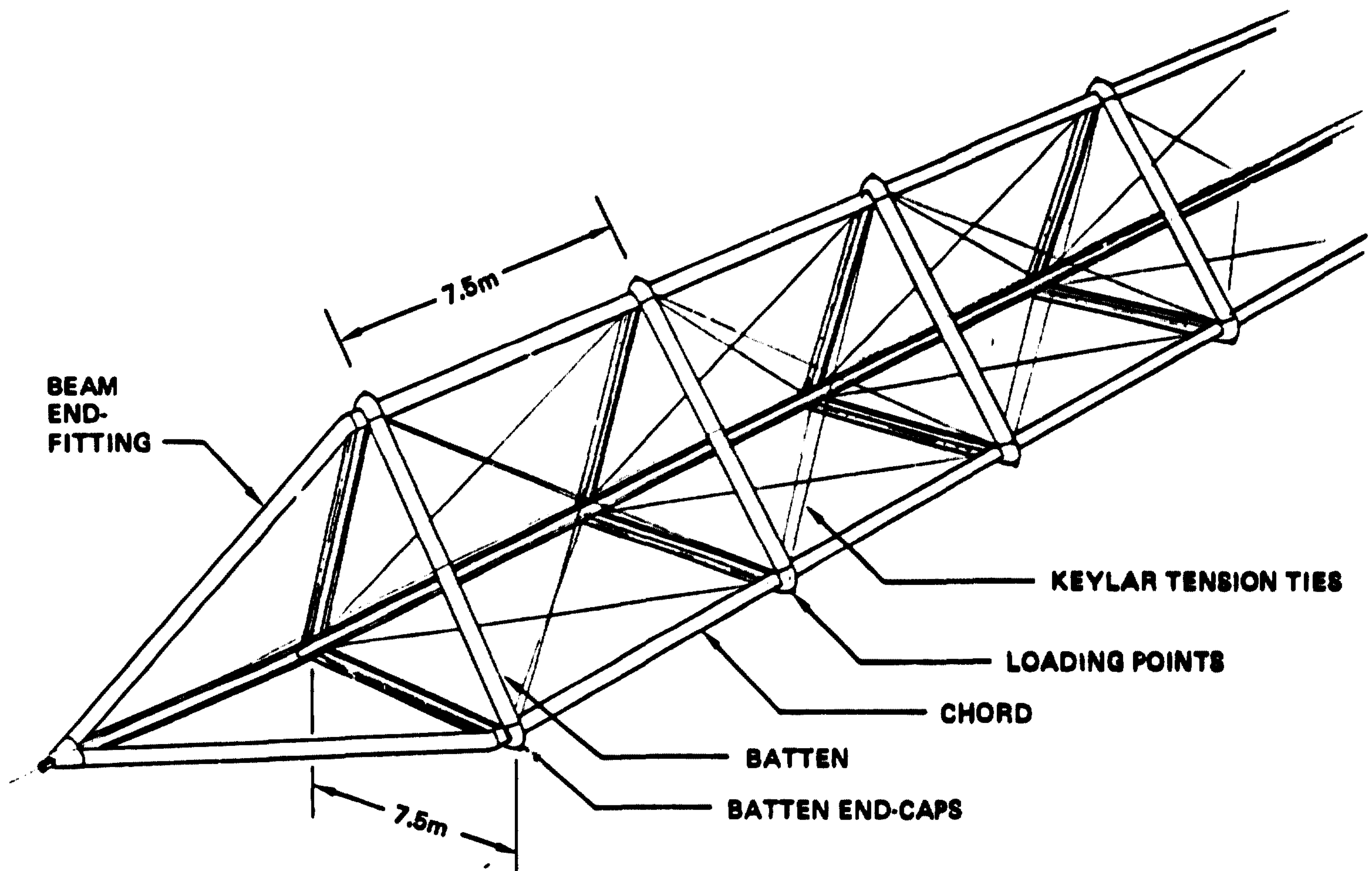
BEAM: WIDTH–7.5m
BATTEN SPACING–7.5m
MASS/LENGTH–5.54 kg/m

CONTINUOUS CHORD BEAM APPROACH

A comparison was made between the continuous chord beam approach and the tapered tube beam used in the Part II reference system. Solutions were found that met the load requirements of the tapered tube beam and resulted in a relatively small mass increase for the overall system.

Shown here is a continuous chord beam approach that satisfies load/stiffness requirements for the reference photovoltaic system. This approach has loading points that are consistent with the current solar blanket geometry and also provides for centroidal beam-to-beam load transmittal.

Continuous Chord Beam Approach



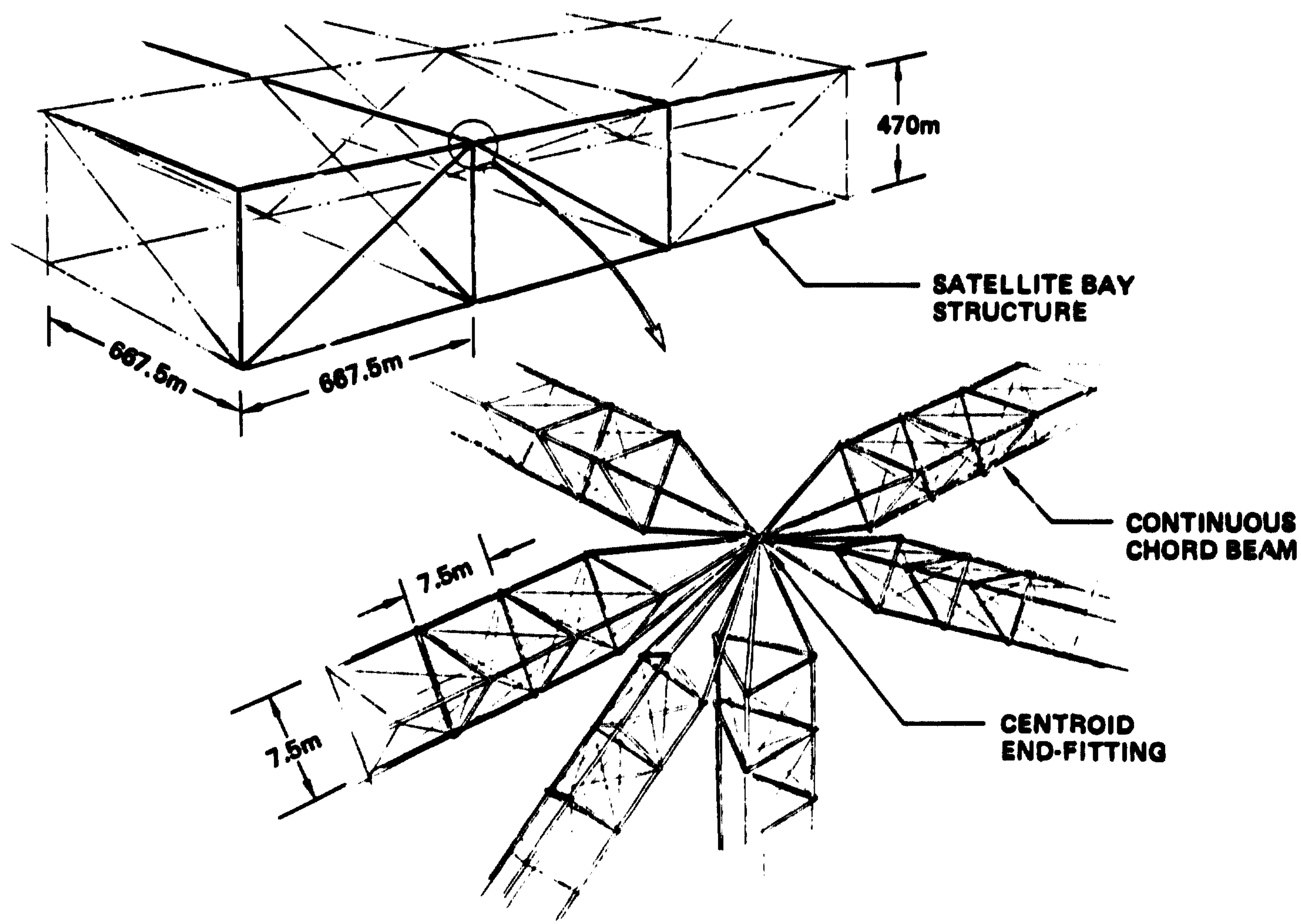
CONTINUOUS CHORD BEAM-BEAM INTERSECTION

Shown here is a typical satellite module edge joint. This type of joint permits centroidal beam-to-beam load transmittal.

This structural approach, with centroidal end-fittings, is consistent with current construction techniques and construction facility sizing.

Continuous Chord Beam-Beam Intersections

SPS-1844



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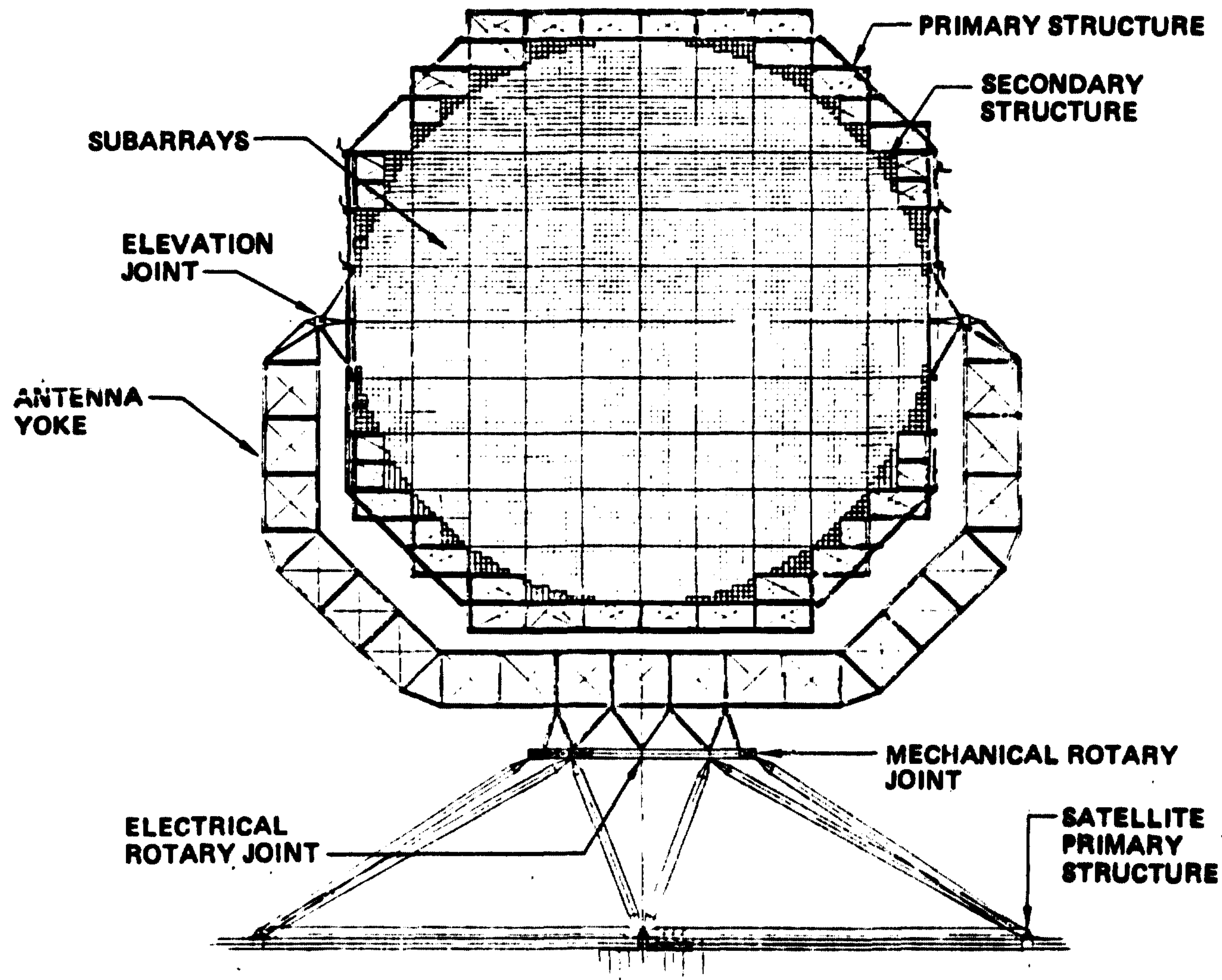
REFERENCE MPTS STRUCTURAL CONCEPT

Shown here is an illustration of the new orthogonal approach used for the MPTS reference system. This change was the result of construction/maintenance trades and involves the incorporation of a klystron module as the system LRU.

The mechanical and electrical rotary joints are the same as that shown at the Part II final review. However, the interface between the satellite primary structure and the mechanical rotary joint has been changed to provide for better load transmittal.

Other changes have occurred that are reflected in the antenna geometry and in the yoke on which it is mounted. These changes are reflected down to the subarray level where a square matrix (10.43 m on a side) was used to provide the three point subarray support.

Reference MPTS Structural Approach

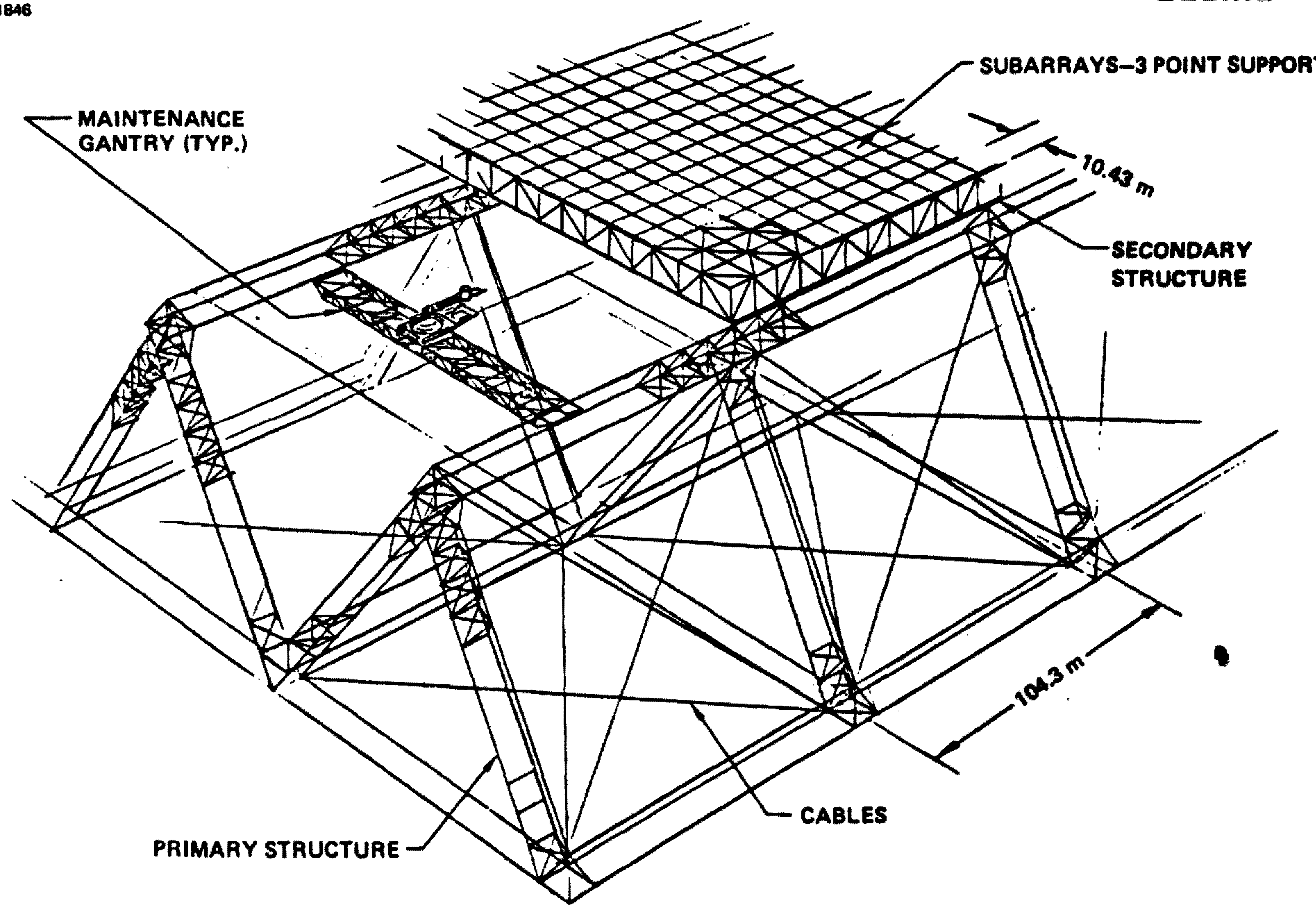


REFERENCE MPTS STRUCTURE INTERFACES

The relative size and configuration of the MPTS system is shown in perspective. The primary structure gives the depth necessary for platform stiffness and provides support points for the cubic secondary structure. The secondary acts as an interface between the subarray and the primary structure. Each subarray is provided with three support points on the secondary structure to allow the necessary adjustments for array flatness and pointing ability.

Power converters will be located on the back of the primary structure. The electrical busing will run along the primary beams and be distributed at the secondary level to provide power to the subarrays/klystrons. Power converter thermal control equipment will also be located on the back of the primary structure.

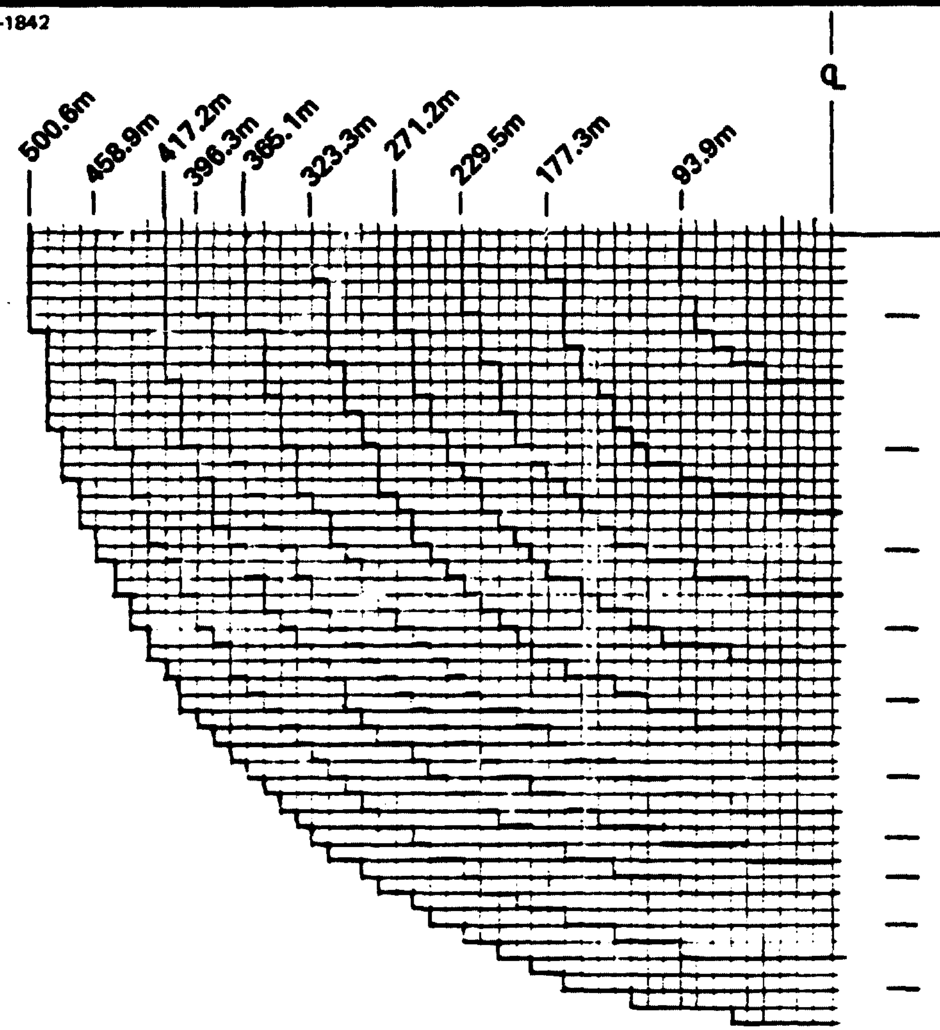
Reference MPTS Structure Interfaces



POWER TAPER INTEGRATION

The new MPTS baseline uses a square subarray instead of the rectangular subarray shown in Part II. It was necessary to iterate this into the integration of the power taper on the MPTS array. The actual integration of power density rings is illustrated on this view of one-fourth of the radiating face of the antenna. The integration simulates a gaussian power taper of 9.5 dB using the quantized power levels available. Note the change in numbers of subarrays and klystrons over the Part II reference system.

MPTS Reference Power Taper Integration



STEP	NUMBER SUBARRAYS	NUMBER KLYSTRONS/ SUBARRAYS	NUMBER KLYSTRONS
1	276	36	9,936
2	632	30	18,960
3	644	24	15,456
4	628	20	12,560
5	784	16	12,544
6	900	12	10,800
7	664	9	5,976
8	612	8	4,896
9	1,052	6	6,312
10	1,028	4	4,112
Totals	7,220		101,552

INTEGRATED SUBARRAY

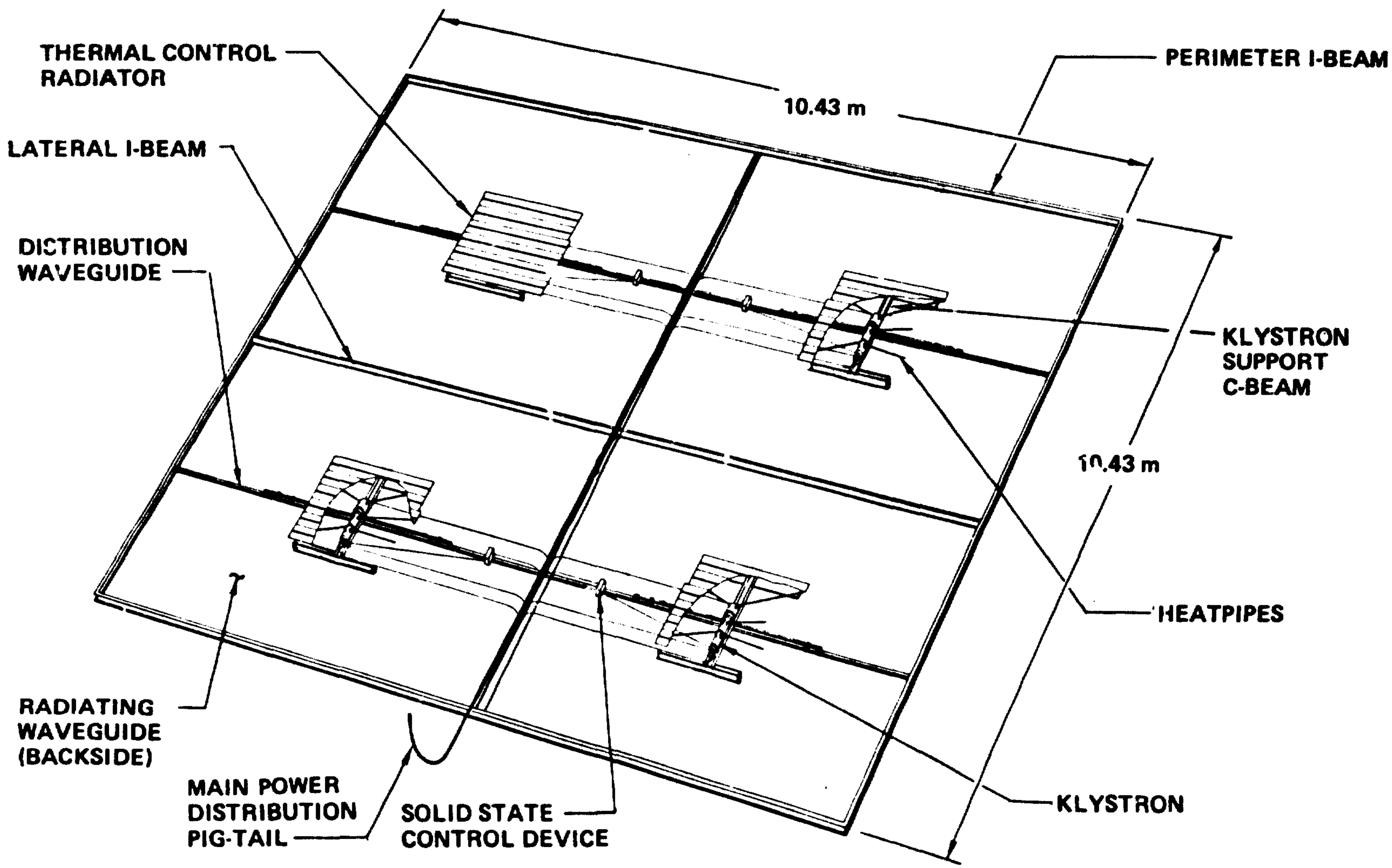
The same basic layout of subarray components, used in the Part II final review, is used in the current reference. The only changes that have occurred are in the basic subarray geometry (square instead of rectangular) and in the method of klystron support within the module.

The square subarray was the result of going to an orthogonal support structure for an improved maintenance approach over the triangular support shown previously.

The new method of klystron support within the subarray reflects the new module LRU approach. To facilitate klystron module removal (excluding the radiating waveguide) it was necessary to shorten the klystron support C-beam and provide a support block for load transmittal into the waveguide. In this way the klystron, output waveguide, thermal control, control circuitry, and support mechanism can be removed as an integral unit. This provides for an improved maintenance scheme over that shown previously.

Reference MPTS–Integrated Subarray

SPS-1217



4-MODULE SUBARRAY

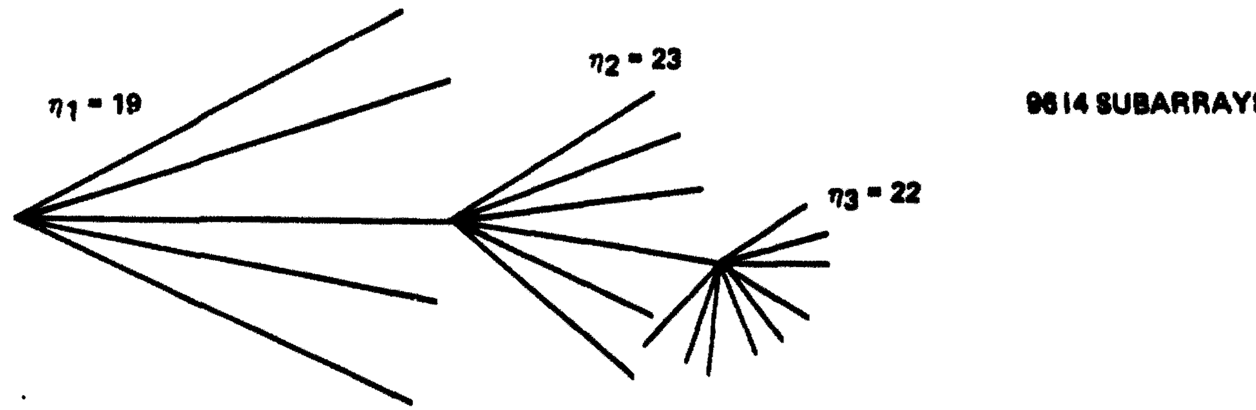
FEATURES OF BASELINE RETRODIRECTIVE PHASE CONTROL

The phase control system utilizes a 3 node phase distribution network similar in some respects to the distribution network earlier suggested by Dickinson of JPL. To minimize phase error buildup, only three nodes are used. This network feeds phase control to a total of 9614 subarrays.

At the lower left is shown the frequency plan for the pilot beam system. The pilot beam is a double sideband, suppressed carrier, AM modulated system. The suppressed carrier is slightly offset from the power beam, and the pilot beam sidebands are either side of the power beam. This frequency plan avoids certain types of errors that would arise with only a single frequency pilot beam. At the lower right is shown the 3 pilot transmitter system located at the rectenna. The use of a 3 pilot transmitter allows the creation of a virtual phase center, at the center of the rectenna, that can be moved to correct for pointing errors. This adds another degree of flexibility to the phase control system that may be necessary to compensate for inosphere disturbances.

Features of Baseline Retrodirective Phase Control

SPS-1865



THREE NODE PHASE DISTRIBUTION SYSTEM FOR ERROR CONTROL

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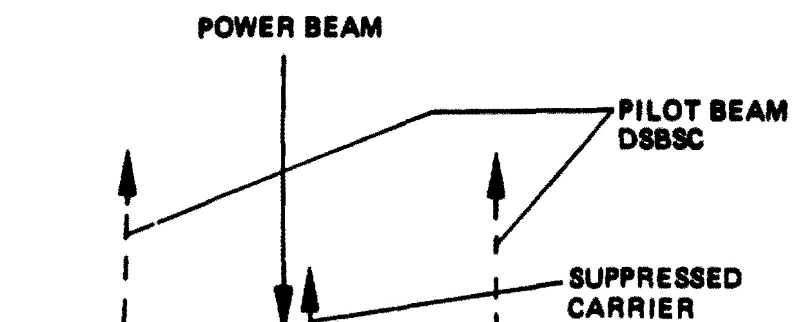
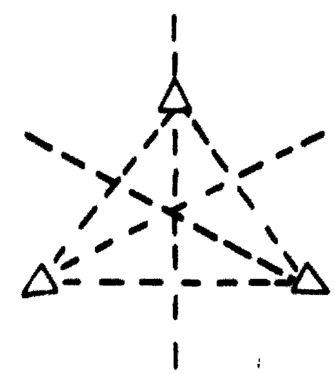


DOUBLE SIDEBAND SUPPRESSED CARRIER
AM MODULATED PILOT BEAM



3-PILOT ANTENNA TRANSMITTER

PHOTOVOLTAIC REFERENCE CONFIGURATION NOMINAL MASS SUMMARY

The structural mass difference from previous analysis reflects a change in the structural beam concept and integration of new sizing criteria to normalize the E.O.L. ground output to 10.0 GW. The majority of the structural mass increase is due to the lower load-to-mass capability of continuous chord elements. A small increase can also be attributed to the slight increase in satellite bay size to accommodate the increase in solar array necessary for normalizing power.

An increase in solar array area, to normalize the power to 10.0 GW, is reflected in the increase of solar cell blanket mass.

The small increase in power distribution mass can be attributed to the increased length of main buses caused by increased bay size.

MPTS mass increased to reflect the increased inventory of klystrons and power conversion equipment to normalize ground output power to 10.0 GW, using the current efficiency chain.

Approximately two thirds of the subtotal mass increase from Part II can be attributed to normalizing the power output to 10.0 GW. The remainder of the mass increase was caused by reflecting the continuous chord beam approach. It is interesting to note that the new structural approach did not significantly change the overall system mass.

SPS

Photovoltaic Reference Configuration Nominal Mass Summary Weight in Metric Tons

BOEING

SPS-1996

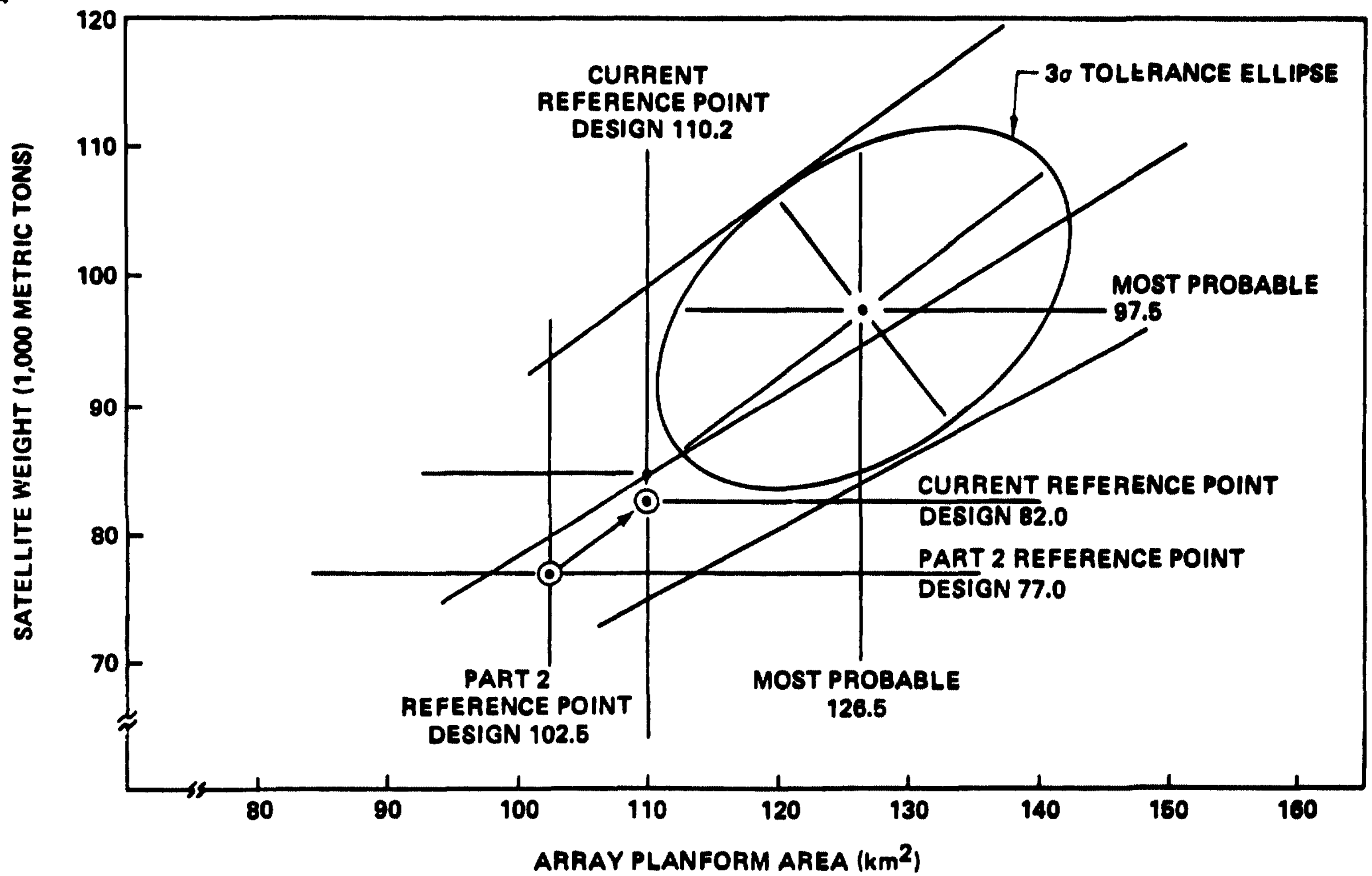
COMPONENT	PART II FINAL	CURRENT	REMARKS
1.0 SOLAR ENERGY COLLECTION SYSTEM	(51,782)	(55,602)	
1.1 PRIMARY STRUCTURE	5,385	7,155	CONTINUOUS CHORD BEAMS AND NORMALIZING POWER
1.2 SECONDARY STRUCTURE	—	—	
1.3 MECHANICAL SYSTEMS	67	67	NO CHANGE
1.4 MAINTENANCE STATION	—	—	
1.5 CONTROL	178	178	NO CHANGE
1.6 INSTRUMENTATION/ COMMUNICATIONS	4	4	NO CHANGE
1.7 SOLAR-CELL BLANKETS	43,750	45,773	INCREASED ARRAY AREA TO NORMALIZE POWER TO 10 GW
1.8 SOLAR CONCENTRATORS	—	—	
1.9 POWER DISTRIBUTION	2,398	2,425	SLIGHT INCREASE IN TRANSMISSION LENGTH
2.0 MPTS	25,212	26,379	NORMALIZED POWER AND SQUARE SUBARRAY
SUBTOTAL	76,994	81,998	
GROWTH	20,400	17,590	NORMALIZED POWER ON GROWTH CURVE
TOTAL	97,474	99,588	

MASS/SIZE UNCERTAINTY UPDATE

Shown here is the current and Part II reference point designs compared to the three sigma mass/size uncertainty elipse. The increase in size and mass of the current reference point design is attributed only to normalizing the system output to 10.0 GW.

Mass/Size Uncertainty Update

SPS-1994



COST UPDATE

The Part II capital cost estimate was updated to reflect Part III changes. The principal new items were the addition of a grid interface system and an allotment of initial spares. Also, the power capability was renormalized to 10 gigawatts. Because the power renormalization was less expensive than predicted by the uncertainty analysis, the Part III capital cost/kwe changed little from Part II.

Capital Cost Update Summary: 1 SPS Per/Year (In Millions of 1977 $)

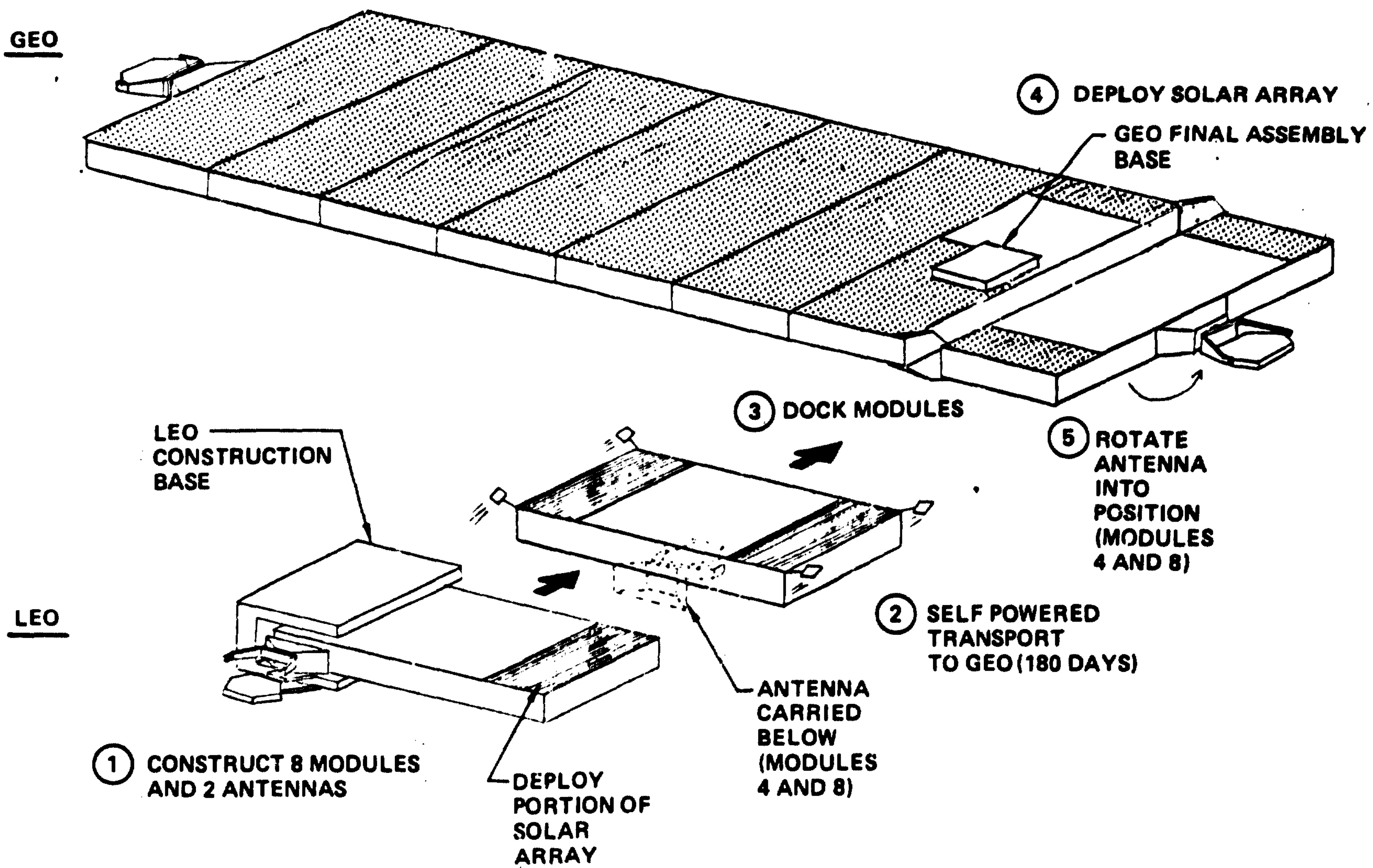
ITEM	PART II FINAL (9.3 GW)	PART III UPDATE (10 GW)	REASON FOR CHANGE
• SUPPORT SUBSYSTEMS	697	637	
• ENERGY CONVERSION (SOLAR BLANKETS)	3,750	4,023	LARGER ARRAY FOR 10 GW
• POWER DISTRIBUTION	133	142	HIGHER POWER
• MICROWAVE POWER TRANSMISSION	2,622	2,724	HIGHER POWER
• GROUND RECEIVING STATION (2)	4,442	4,520	RE-ESTIMATE
• GRID INTERFACE	—	1,348	NOT INCLUDED IN PART II
• CONSTRUCTION & SPACE SUPPORT	1,109	1,109	—
• SPACE TRANSPORTATION	6,445	6,387	INCREASED EARTH LAUNCH COST BUT SAVINGS BY ORBIT TRANSFER SYSTEM RECOVERY
• INITIAL SPARES	—	240	NEGLECTED IN PART I
• PACKAGING & OTHER	314	602	INCREASED TO 5% OF APPLICABLE ITEMS
• INTEREST DURING CONSTRUCTION	1,864	2,082	HIGHER BASE COST
• GROWTH	3,450	3,115	SOME OF PART II GROWTH INCLUDED POWER DEFICIENCY
TOTAL	24,766 ($2,663/kWe)	26,929 ($2,693/kWe)	

LEO CONSTRUCTION CONCEPT
PHOTOVOLTAIC SATELLITE

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

LEO Construction Concept
Photovoltaic Satellite

SPS-1383

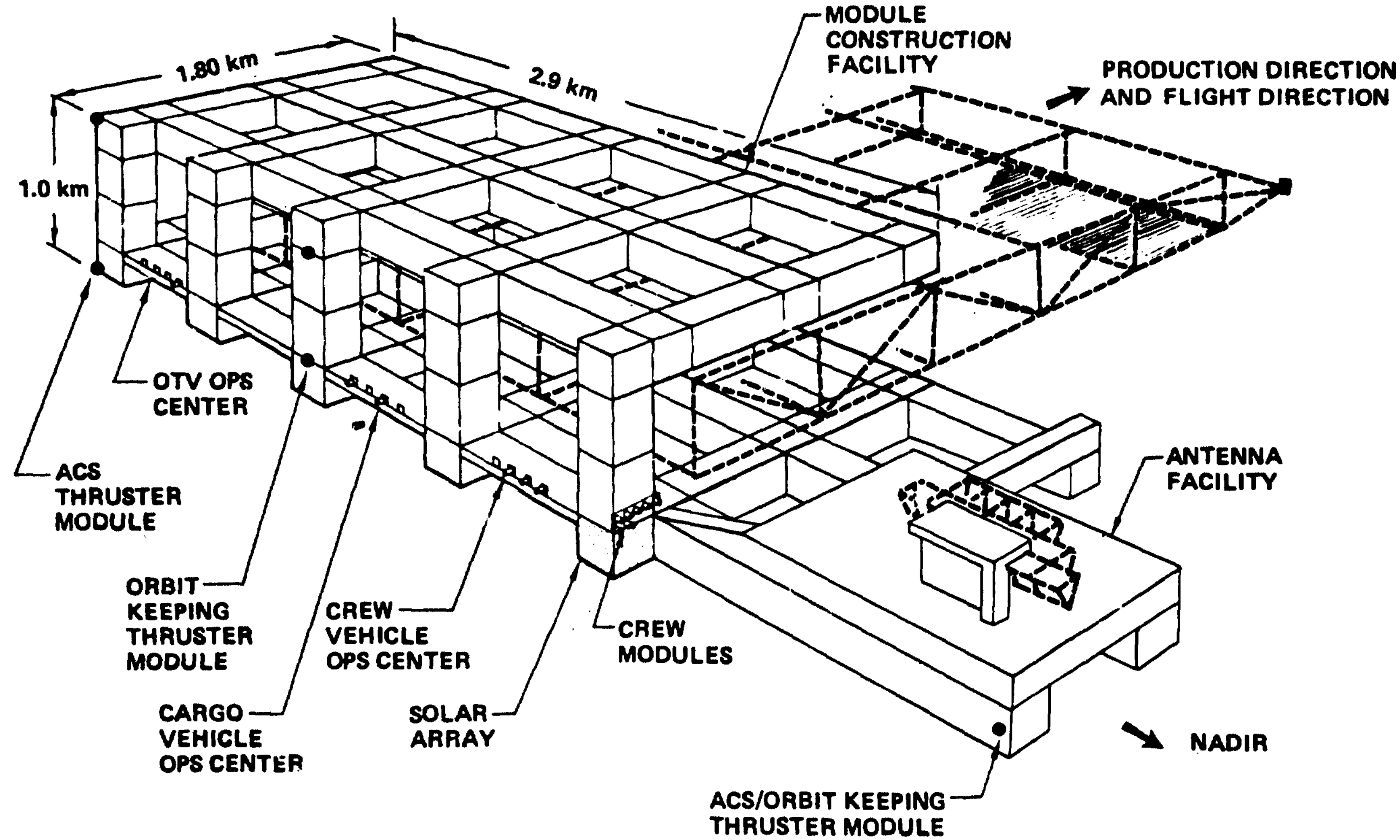


LEO CONSTRUCTION BASE
PHOTOVOLTAIC SATELLITE

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

The antenna facility is configured to enclose five bays of antenna in width and one row of bays in length. The antenna facility shown reflects the new antenna configuration. The upper surface of the facility is used to support beam machines, joint assembly machines, support indexing machines and bus deployment equipment. The lower surface is used to support beam machines, joint assembly machines and a deployment platform that is used to deploy the secondary structures and antenna subarrays.

LEO Construction Base Photovoltaic Satellite



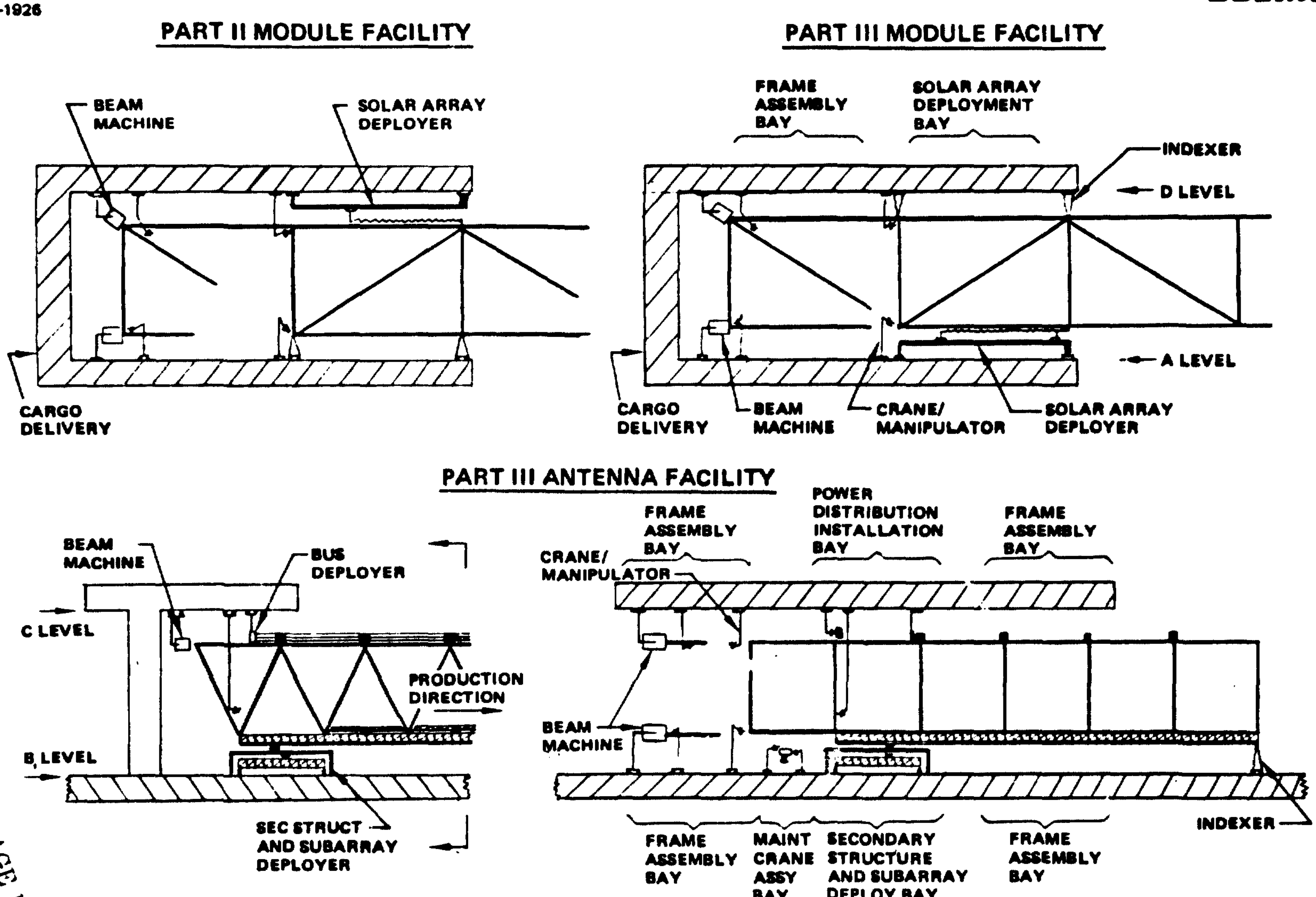
CONSTRUCTION BASE EQUIPMENT AND OPERATION

One of the principal changes in the construction base since Part II was to "invert" the satellite module with respect to the construction base to bring the solar blanket installation area closer to the payload arrival area. Most of the tonnage moved by the logistics network is solar blankets. It was felt important to minimize the distance over which these blankets are transported.

The other significant change was to modify the antenna construction facility to minimize the indexing and movement problems associated with installing the antenna in its yoke, also providing better crew access to the joint area where the antenna connects to the yoke.

Construction Base Equipment/Operations

SPS-1926

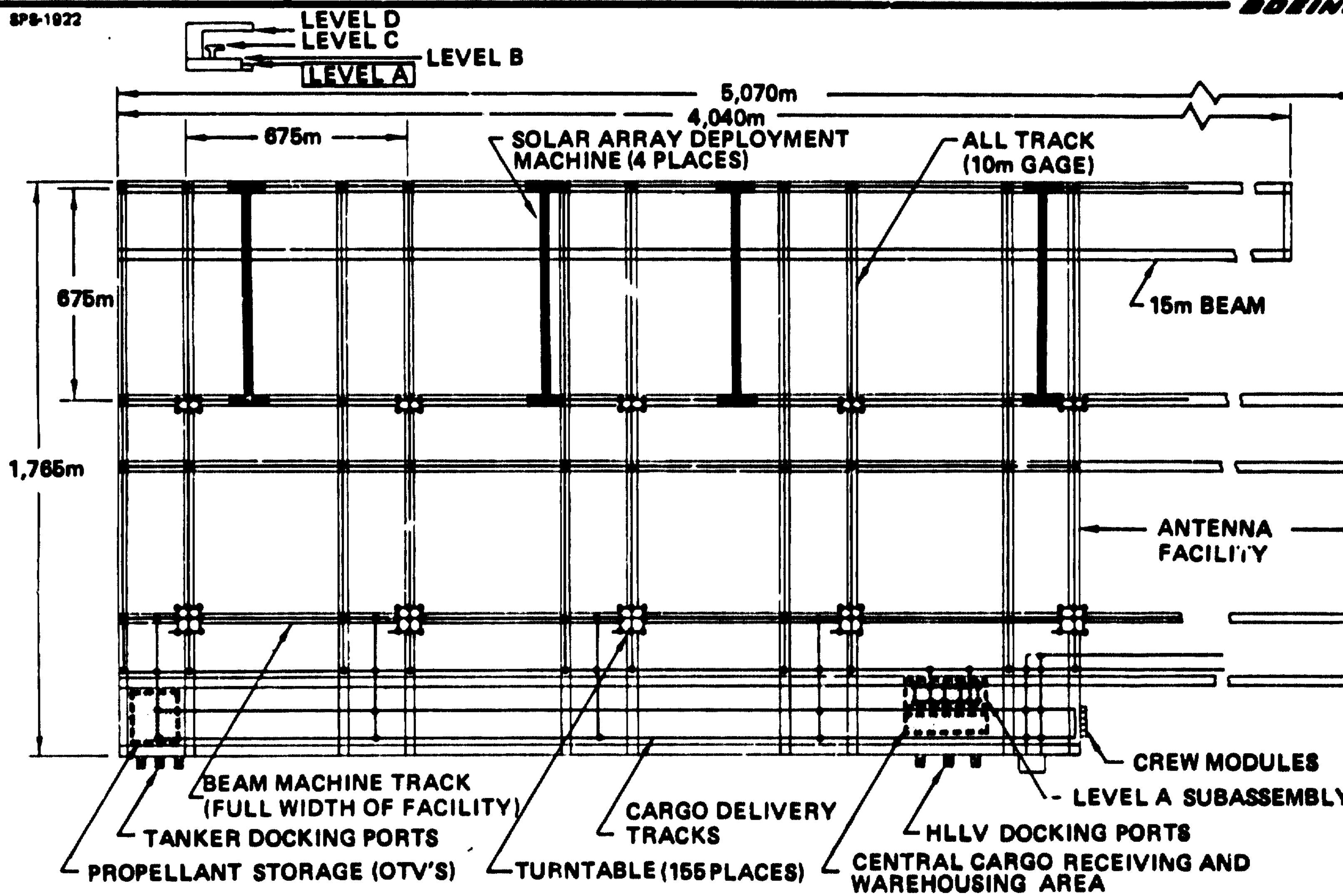


LOGISTICS NETWORK LEVEL AT LEO CONSTRUCTION BASE

One of the significant requirements identified during Part II but not characterized was the need for a logistics network to move SPS hardware and construction equipment and crews around on the facility to bring them to the location where the work is actually being done. During Part III a logistic network was characterized. A part of the network is shown here. The tracks allow movement of construction equipment, crews, and satellite parts, and also provide for movement of the satellite module under construction by moving the indexing fixtures that tie the module to the facility.

Logistic Network Level A
LEO Construction Base

SPS-1922



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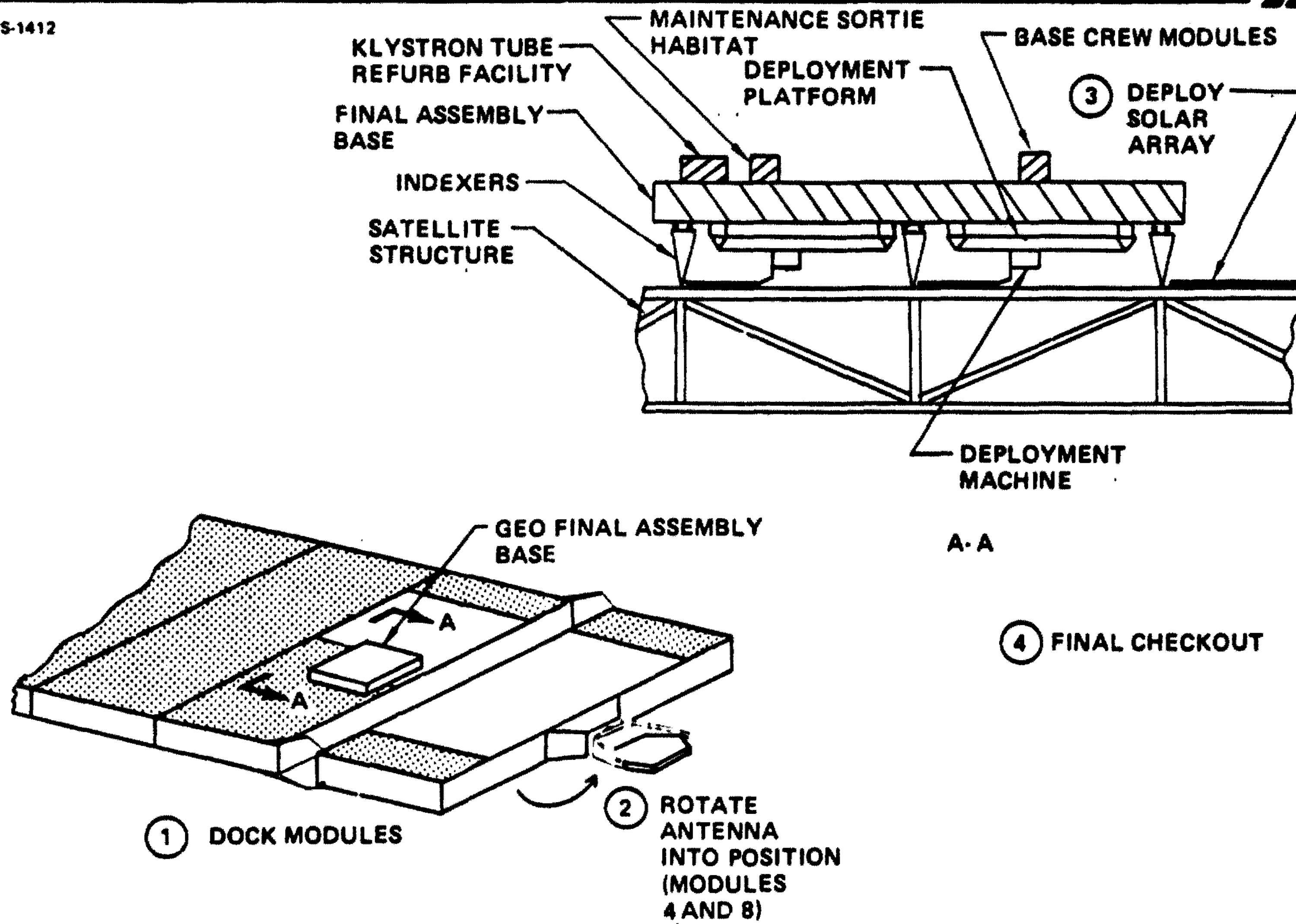
GEO FINAL ASSEMBLY BASE AND OPERATIONS

The maintenance analysis indicated a significant need for crews in geosynchronous orbit to carry out maintenance operations. The most straightforward way of providing for these crews appears to be to include their provision in the geosynchronous final assembly base and make this base also an operations base. The additions required include a klystron tube refurbishment facility, a docking location for the mobile maintenance habitat, and crew modules for the maintenance crews in addition to those for the final assembly base crews.

GEO Final Assembly Base/Operations

SPS-1412



CONSTRUCTION SYSTEMS CHARACTERISTICS

Highlights of the construction base are given on this table.

Construction System Characteristics

	LEO CONSTRUCTION BASE	GEO FINAL ASSEMBLY BASE
•CREW [1]	(480)	(65)
•CONSTRUCTION	200	25
•CONSTRUCTION SUPPORT	140	10
•OPERATIONS	140	30
•MASS (MILLIONS OF Kg)	(5.6)	(0.9)
•FACILITY	5.2	0.7
•CONSTRUCTION EQUIPMENT	0.4	0.2
•DDT&E COST	( )	( )
•FACILITY		
•CONSTRUCTION EQUIPMENT		
•UNIT COST (BILLIONS)	(4.8)	(1.2)
•FACILITY	3.5	0.4
•CONSTRUCTION EQUIPMENT	1.3	0.4
•WRAP-AROUND	2.2	0.4

[1] SATELLITE MAINTENANCE CREW NOT INCLUDED.

TWO-STAGE WINGED SPS LAUNCH VEHICLE (FULLY REUSABLE CARGO CARRIER)

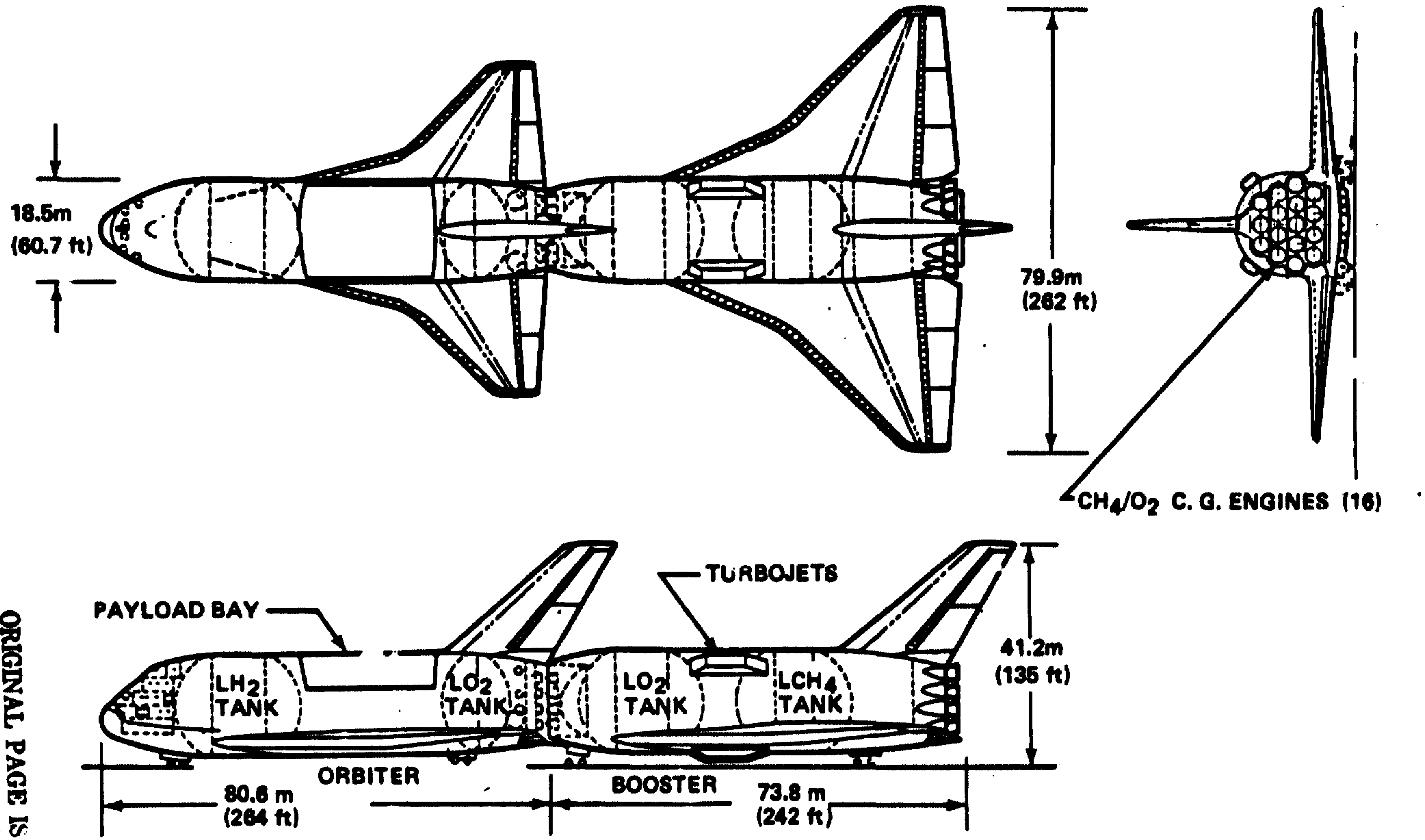
The launch configuration of the SPS cargo vehicle is shown on the adjacent chart with the overall geometry noted. This series burn concept uses 16 LCH_4/LO_2 engines on the booster and 14 standard SSME's on the orbiter. The LCH_4/LO_2 booster engines are a gas generator cycle providing a vacuum thrust of 9.79×10^6 newtons each. The SSME's on the orbiter provide a vacuum thrust of 2.09×10^6 newtons (100% power level). The nominal 100% power level for the SSME's was selected based on engine life considerations which indicated about a 3 factor reduction in life if the 109% power level is used.

An airbreather propulsion system (12 installations of an SST type engine) has been provided on the booster for flyback capability to simplify the booster operational mode. The reference wing for both stages is

$$S_W \text{ (Orbiter)} = 1446 \text{ m}^2 \quad (15{,}560 \text{ ft}^2)$$
$$S_W \text{ (Booster)} = 2330 \text{ m}^2 \quad (25{,}080 \text{ ft}^2)$$

Heat sink thermal protection system is provided on the booster and the Shuttle's Reusable Surface Insulation (RSI) is used on the orbiter.

Two-Stage Winged SPS Launch Vehicle
(Fully Reusable Cargo Carrier)



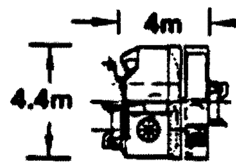
CREW ROTATION AND RESUPPLY SYSTEM LEO/GEO APPLICATION

Shown here is the oxygen/hydrogen orbit transfer vehicle and the payload module. This system provides the capability to transfer 75 crew, up and return, per flight with their supplies. The orbit transfer vehicle is a two stage, oxygen/hydrogen fueled, conventional rocket vehicle. The total propellant loading for the two stages is 460,000 kilograms.

Crew Rotation/Resupply System
LEO/GEO Application

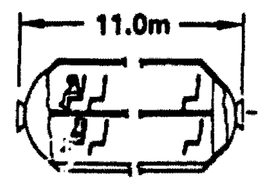
SPS-1939

- **FLIGHT CONTROL MODULE**



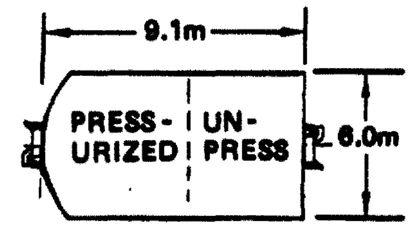
CREW = 2
MASS = 4,000 kg

- **GEO PASSENGER MODULE**

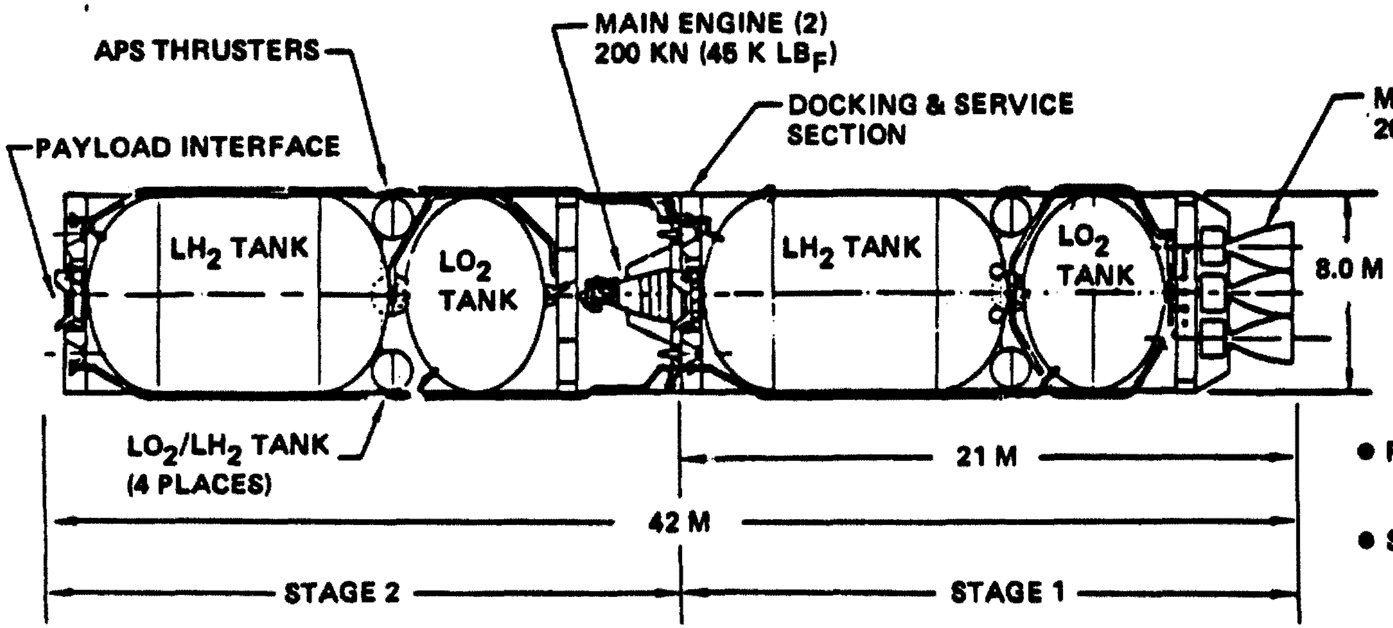


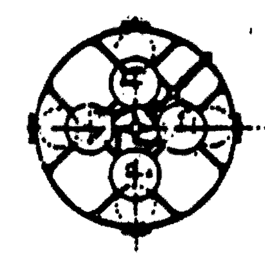
CREW = 75
MASS = 20,000 kg

- **SUPPLY MODULE**



CARGO = 300 MAN MO.
30,000 kg
MODULE = 10,000 kg





- **PAYLOAD**
  - UP 65,000 kg
  - DOWN 40,000 kg
- **STAGE**
  - PROP 230,000 kg
  - INERT 15,000 kg

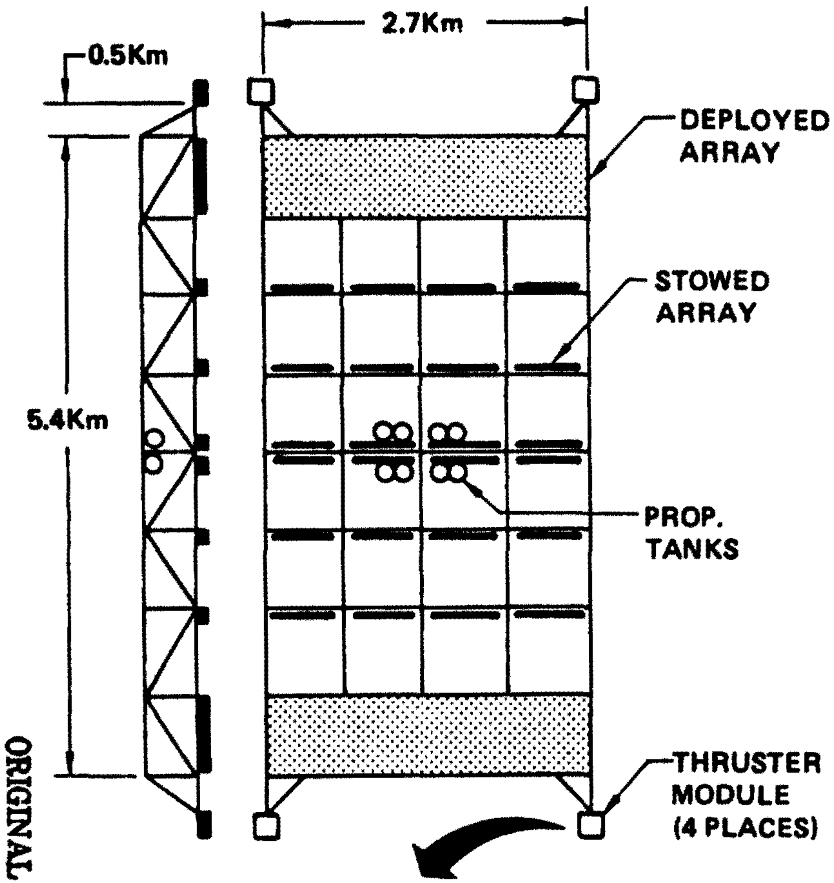
SELF-POWER CONFIGURATION
PHOTOVOLTAIC SATELLITE

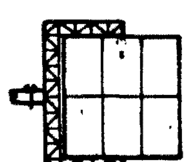
The transfer of the satellite modules from LEO to GEO involves the use of electric propulsion using power provided by the module (thus the name self-power). The characteristics associated with self-power of a photovoltaic module are shown for both those modules transferring antennas and those that do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

Self Power Configuration Photovoltaic Satellite

BOEING

SPS-1619





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

GENERAL CHARACTERISTICS

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

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

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RECOVERY CONCEPT FOR ELECTRIC OTS

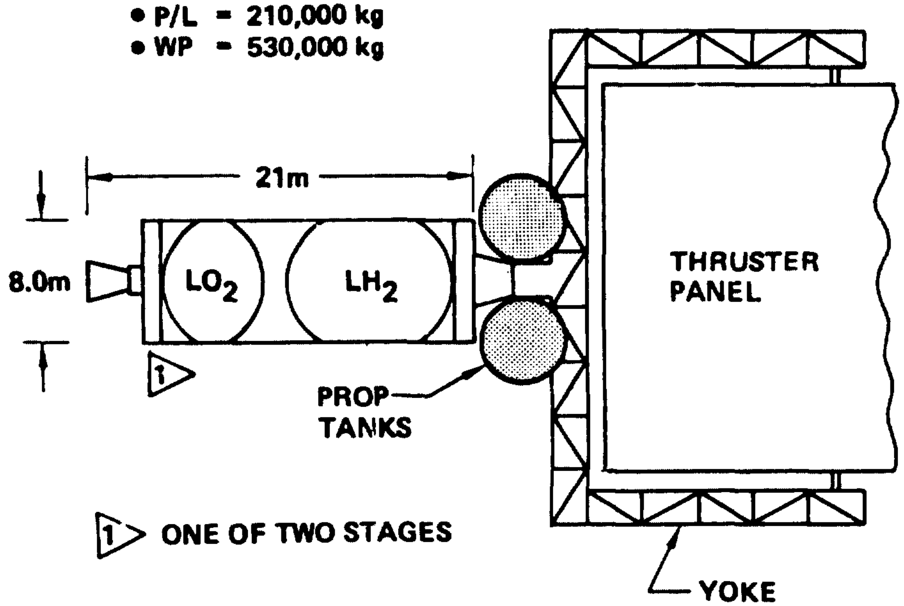
The system analyzed for the return of the electric components to LEO is a single-stage LO_2/LH_2 OTV. Return of 210,000 kilograms of payload requires a propellant loading of approximately 530,000 kilograms. This size of stage is slightly larger than that used for the crew rotation in the LEO construction option. When combined with a second stage, the combination vehicle serves to provide propulsion capability for resupply flights to GEO. Delivery of the LO_2/LH_2 stage to GEO involves mounting the stages below the satellite module. The resulting impact on the electric propulsion system of transporting an additional 1.6 million kilograms of LO_2/LH_2 stages is relatively minor.

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

Electric OTS Recovery

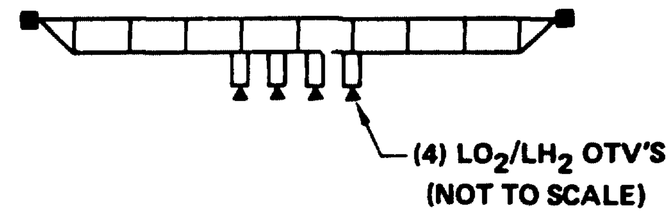
RECOVERY SYSTEM

- P/L = 210,000 kg
- WP = 530,000 kg



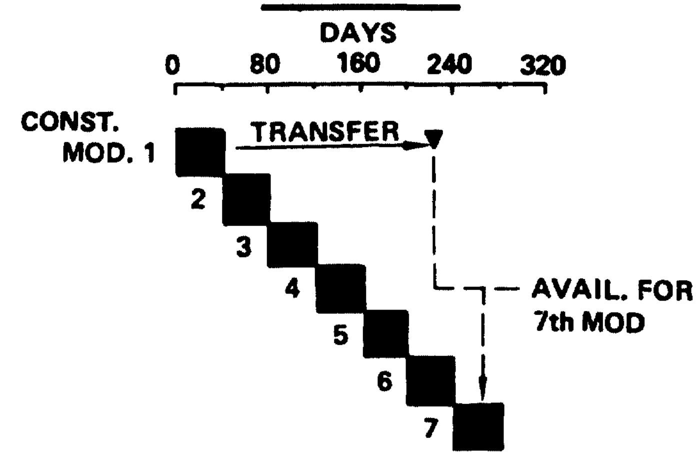
1 ONE OF TWO STAGES

DELIVERY SYSTEM



- PIGGYBACK ON SATELLITE
- Δ P/L = 1.6 MILLION KG
- ELEC OTS MODIF
  Δ POWER ~ 2%
  Δ W_p ~ 0.4m Kg

OPERATIONS



COST SUMMARY

	1 SAT/YR	4 SAT/YR
• RECOV VALUE	$1,280M	$690M
• RECOV COST	$ 720M	$535M
SAVINGS PER SAT	$ 560M	$165M

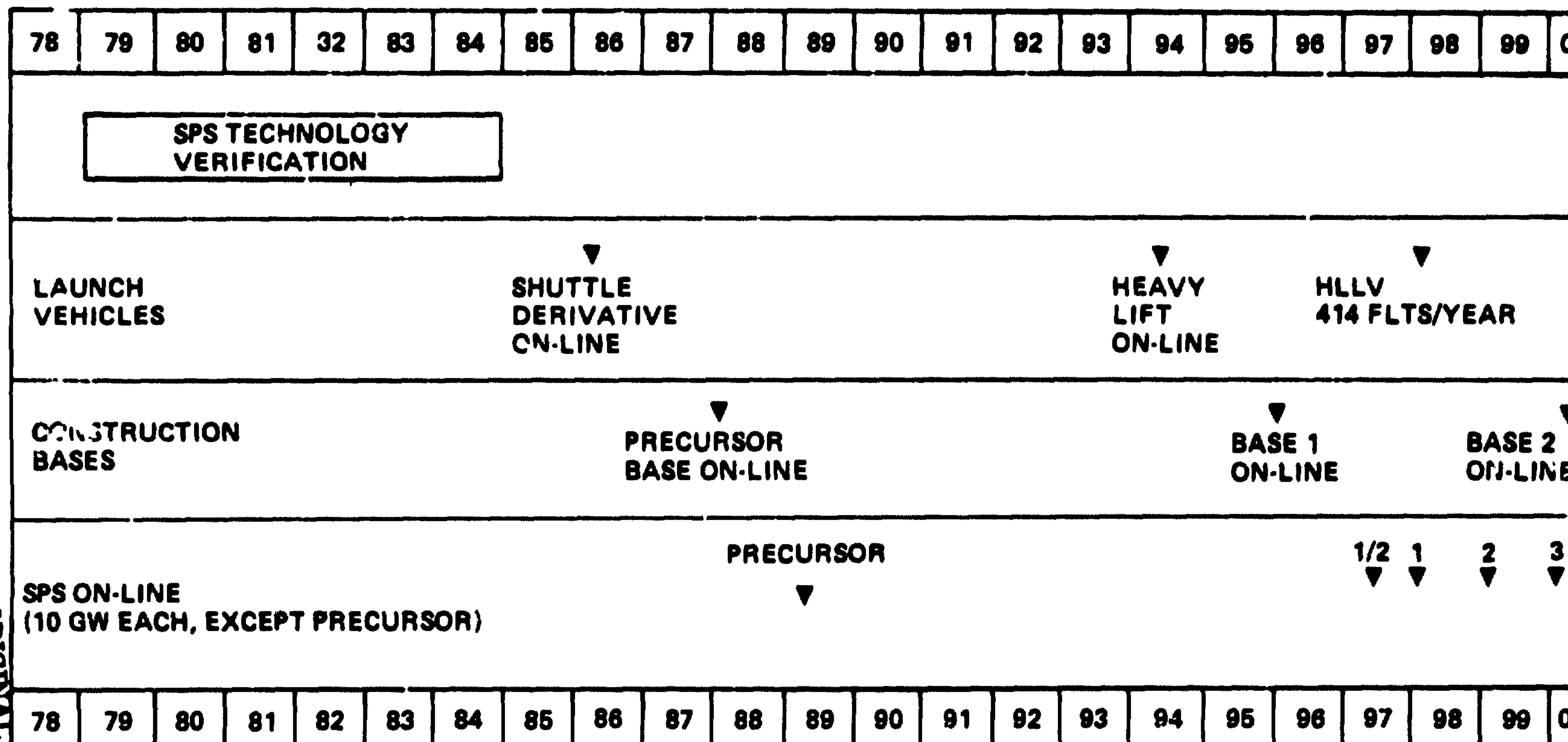
A "CONSERVATIVE PACE" SPS SCENARIO

This scenario represents a low-risk approach to SPS implementation. Major funding for the full size SPS system is not committed until successful on-orbit operation of a precursor satellite has been accomplished. This major funding includes not only that for the first full SPS but also that for development and implementation of the heavy lift launch vehicle fleet. Derivation of this scenario included allowances for the time period inherent in new vehicle development (five years for new stages, six years for new engines) and the time necessary for certain in-space operations. For example, the precursor program requires four and one half years from the first launch of equipment from which its construction base is assembled until the end of the test operations in geosynchronous orbit. SPS construction bases require two years of assembly and checkout operations prior to beginning SPS assembly. The self-power transfer of SPS modules to geosynchronous orbit takes 180 days.

A "Conservative Pace" SPS Scenario

SPS 1983

BOEING



NOTE: 1) THE PRECURSOR SPS SERIVES AS A "COMMITMENT PRODUCER"

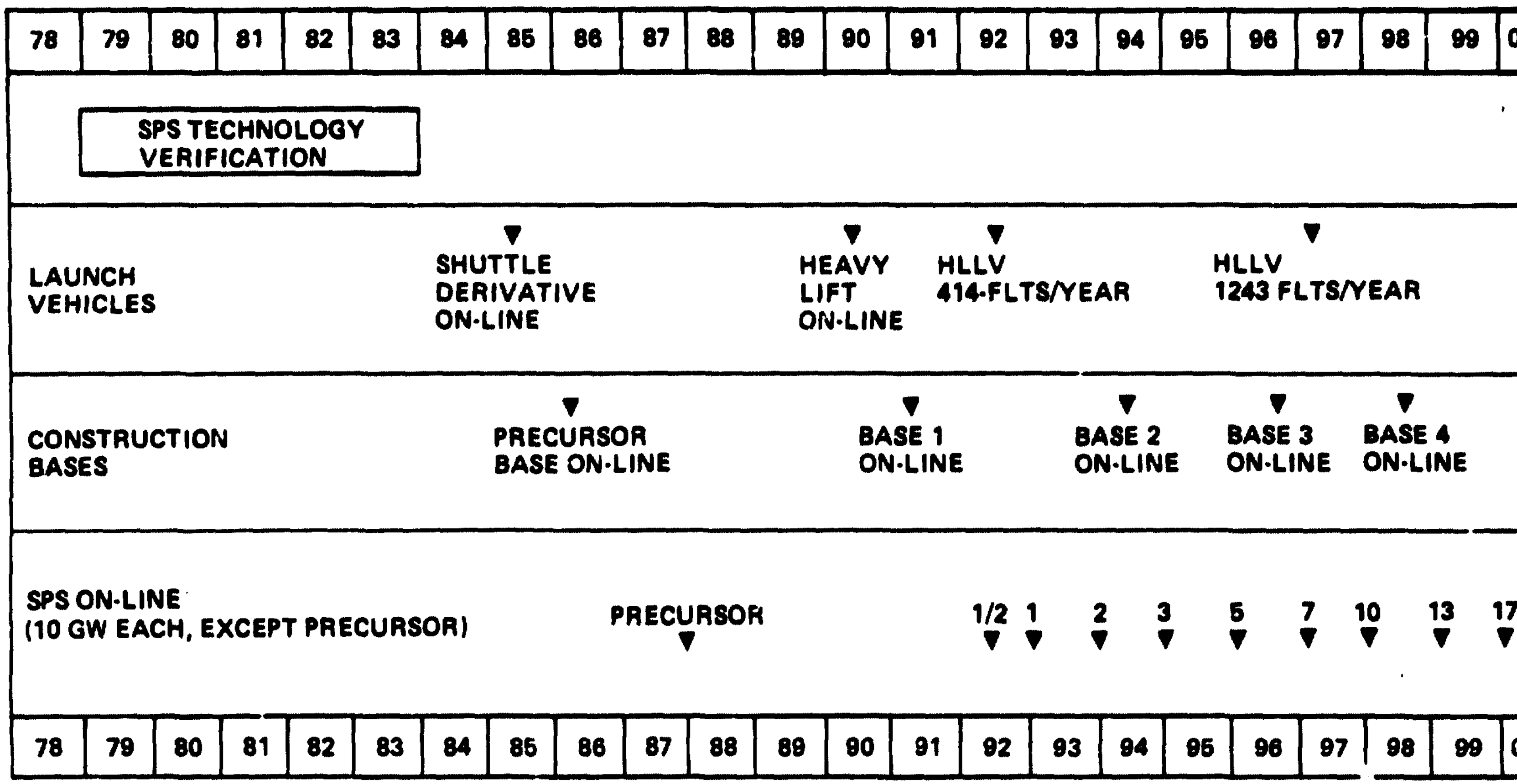
2) BY 2000, THE INSTALLED UNITS ARE PRODUCING ABOUT 11% OF THE ELECTRICAL ENERGY CONSUMED IN THE U.S. IN 1976.

OF THE ELECTRICAL ENERGY CONSUMED IN THE U.S. IN 1976.

AN "APOLLO PACE" SPS SCENARIO

The program represents a pace intermediate between that of the "conservative" scenario given on the previous chart and a "crash, Manhattan project". In this scenario development of the shuttle-derivative launch vehicle and the precursor SPS construction base are begun in 1979. Development of the heavy lift launch vehicle and other elements involved with the full size SPS starts before completion of the test operations with the precursor SPS.

An "Apollo Pace" SPS Scenario



NOTE: 1) THE PRECURSOR SPS SERVES AS A DEVELOPMENTAL TOOL RATHER THAN AS A "COMMITMENT PRODUCER."

2) BY 2000, THE INSTALLED UNITS CAN BE PRODUCING APPROXIMATELY 60% OF THE ELECTRICAL ENERGY CONSUMED IN THE U.S. IN 1976.

GROUND BASED TECHNOLOGY ADVANCEMENT PLAN

Shown here are the principal technology advancement areas and a preliminary estimate of the annual funding required to pursue these areas. No significant differences in the technology advancement plan have been identified since the completion of Part II. The technology advancement plan is described in additional detail in Volume 2 of the Part II Final Report.

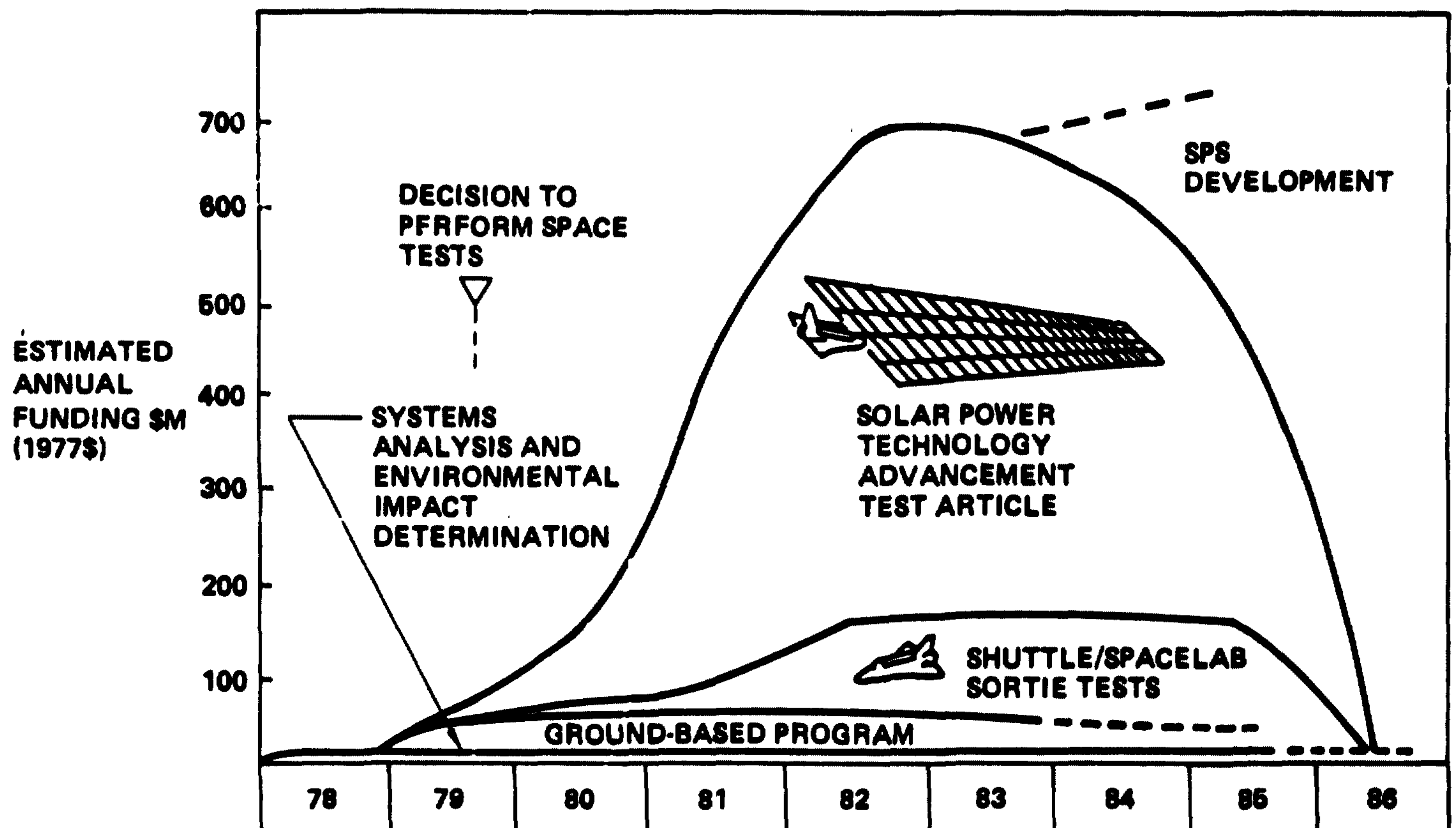
Ground-Based Technology Advancement Plan

TECHNOLOGY AREA	YEARS 1	2	3	4	5	TOTAL
SOLAR CELLS	2.5	3.6	3.85	3.7	2.6	16.25
THERMAL ENGINES & THERMAL SYSTEMS	2.5	3.5	3.75	3.5	2.5	15.75
MICROWAVE POWER TRANSMISSION SYSTEM	6.0	7.5	8.75	8.5	6.5	37.25
SPACE STRUCTURES	0.75	2.0	2.2	2.0	1.8	8.75
MATERIALS	1.5	2.0	2.5	2.0	2.0	10.0
FLIGHT CONTROL SYSTEMS	0.5	0.8	1.0	0.9	0.8	4.0
CONSTRUCTION SYSTEMS	3.0	4.0	4.5	5.5	5.5	22.5
TRANSPORTATION SYSTEMS	4.5	7.5	8.75	7.5	7.5	35.75
POWER DISTRIBUTION AND CONTROLS	1.5	2.0	2.5	3.5	2.5	12.0
SPACE ENVIRONMENT FACTORS	2.0	2.6	2.45	2.2	2.0	11.25
TOTALS	24.75	35.50	40.25	39.3	33.7	173.5

OVERALL TECHNOLOGY ADVANCEMENT PROGRAM

The overall technology advancement program includes, in addition to the ground based program, shuttle spacelab sortie tests and a solar power technology advancement test article. The shuttle spacelab sortie test would include tests of beam fabricator machine and RF equipment and also tests of prototype manipulator systems and other construction aids. The solar power technology advancement test article would be a solar array in the 100 - 500 kilowatt range constructed and supported using the shuttle as an operating base. This test article would use solar arrays similar in nature to those planned for SPS. It would also use similar structural techniques and would provide a test base for testing of higher power RF equipment, electric thrusters and tests operating the solar array, at high generating voltages, to develop information on plasma interactions.

Overall Technology Advancement Program



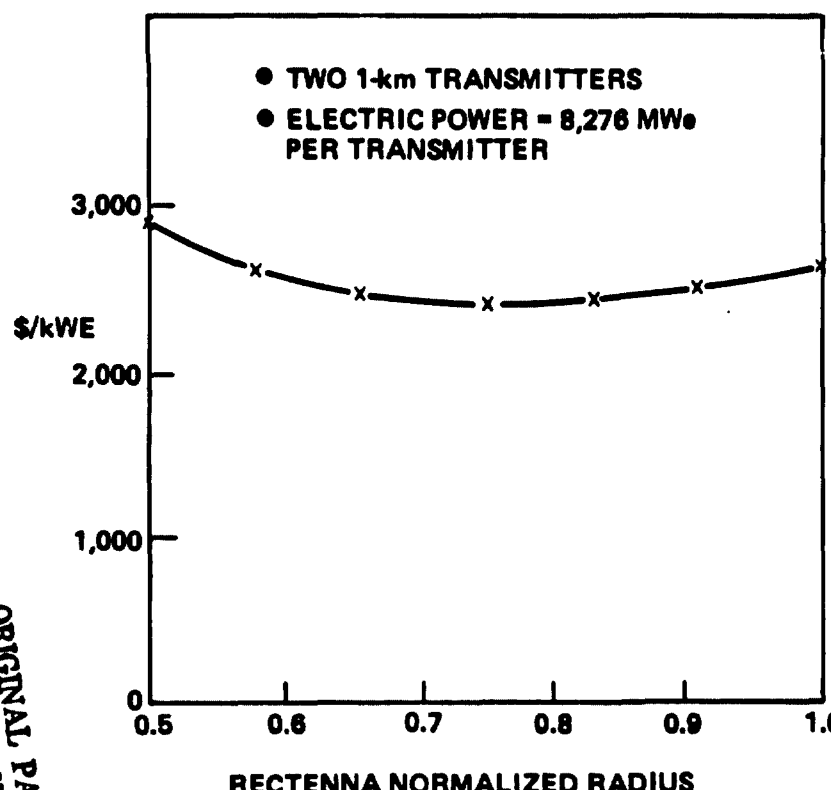
SIZE SENSITIVITY RESULTS POWER TRANSMISSION OPTIMIZATION

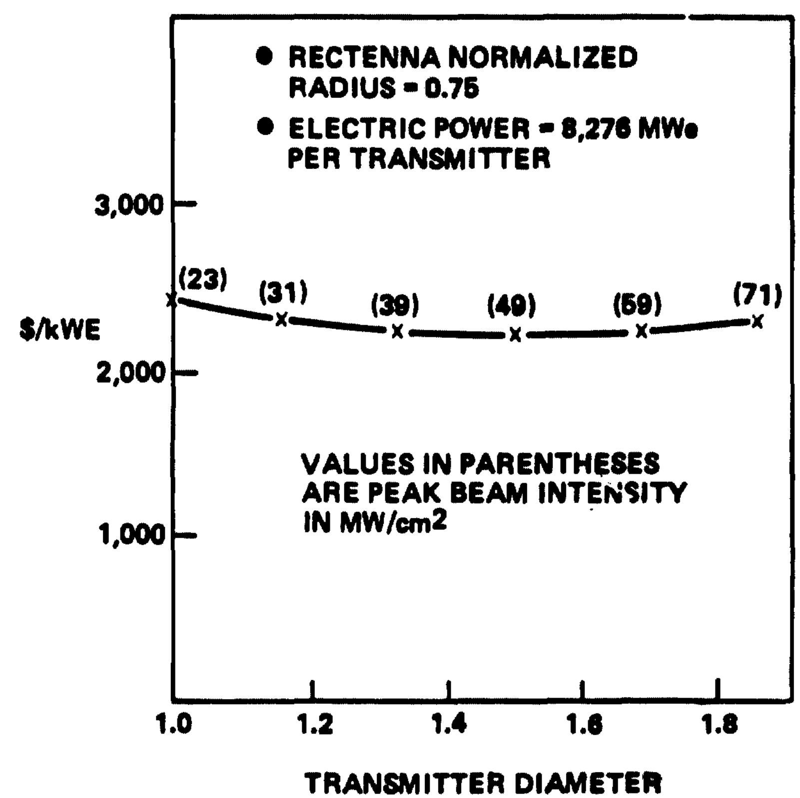
A size sensitivity design model was constructed and exercised. The first run of the model optimized power transmitter and rectenna sizes at the nominal power level of approximately 5,000 megawatts per link. The new results, although executed in somewhat more detail than earlier results, confirmed the earlier estimates that the optimum rectenna size is 3/4 of the transmitted beam diameter and that the optimum transmitter size is in the vicinity of 1.4 kilometers. However, transmitter sizes larger than one kilometer violate the peak beam intensity limit of 23 millowatts per centimeter squared. Therefore the best system uses a 1 kilometer transmitter and a rectenna diameter 3/4 of the beam diameter.

Size Sensitivity Results Power Transmission Optimization

SPS-1990

SPS-1980





SIZE SENSITIVITY ANALYSIS MODELING DETAIL

The size sensitivity model was implemented on the ISAIAH modeling system. The model consisted of 37 designer selected variables and 95 computed variables. The values generated by the model for the nominal design point of a 1 kilometer diameter transmitter are shown on the chart. A complete design point was generated for each sensitivity point analyzed.

Size Sensitivity Model Details

NOMINAL VALUES, INDEPENDENT VARIABLES

1	SUMMER SOLSTICE FACTOR	=	9.763E-01	
2	COSINE LOSS	=	9.190E-01	
3	SOLAR CELL EFFICIENCY	=	1.730E-01	
4	RADIATION DEGRADATION	=	9.700E-01	
5	TEMPERATURE DEGRADATION	=	9.540E-01	
6	COVER UV DEGRADATION	=	9.560E-01	
7	CELL-TO-CELL MISMATCH	=	9.900E-01	
8	CELL & PANEL LOST AREA	=	9.610E-01	
9	STRING I-SQ-R	=	9.980E-01	
10	PANEL/BLANKET LOST-AREA	=	9.800E-01	
11	DC-RF CONVERSION	=	8.500E-01	
12	WAVEGUIDE I-SQ-R	=	9.850E-01	
13	INTER-SUBARRAY ERRORS	=	9.460E-01	
14	INTRA-SUBARRAY ERRORS	=	9.810E-01	
15	ATMOS. ABSORPTION	=	9.800E-01	
16	GRID INTERFACE	=	9.700E-01	
17	POWER ACROSS ROTARY JOIN	=	8.280E+03	MEGAWATT
18	SIDELOBE LIMIT	=	1.000E-01	MW/CM2
19	ANTENNA DIAMETER	=	1.000E+00	KM
20	RECTENNA NORMALIZED RADI	=	7.500E-01	
21	NOMINAL BAY SIZE	=	6.600E+02	METERS
22	SPS WIDTH	=	8.000E+00	BAYS
23	CLEARANCE IN BAY	=	5.000E+00	METERS
24	NO. OF ANTENNAS	=	2.000E+00	
25	BEAM ELEMENT SIZE	=	1.500E+01	METERS
26	POWER PER KLYSTRON	=	7.200E+01	KWRF
27	SUBARRAY SIZE	=	1.040E+01	METERS
28	ATTITUDE CONTROL ISP	=	2.000E+04	SEC
29	MASS GROWTH	=	2.600E+01	PCT
30	TRANSP & CONSTR TIME	=	1.600E+00	YEARS
31	INTEREST RATE	=	7.500E+00	PCT
32	PLANT FACTOR	=	9.200E+01	PCT
33	KLYSTRON "A" VOLTAGE	=	4.000E+04	VOLTS
34	KLYSTRON "B" VOLTAGE	=	3.800E+04	VOLTS
35	RECTENNA LATITUDE	=	3.500E+01	DEGREES
36	"A" BUS PWR FRAC	=	6.300E-01	
37	"B" BUS PWR FRAC	=	3.700E-01	

SOLUTION RESULTS

1	LIGHT INPUT-EFFICIENCY	=	8.579E-01	
2	NET CELL EFFICIENCY	=	1.601E-01	
3	BASIC CONVERSION EFFY	=	1.360E-01	
4	BLANKET FACTORS	=	9.399E-01	
5	BUS I-SQ-R	=	9.415E-01	
6	NET ENERGY CONV EFFY	=	1.203E-01	
7	AREAWISE EFFICIENCY	=	9.370E-01	
8	ANTENNA POWER DISTR EFFY	=	9.701E-01	
9	NET DC-RF EFFICIENCY	=	8.123E-01	
10	IDEAL BEAM EFFICIENCY	=	9.650E-01	
11	NET BEAM EFFICIENCY	=	8.955E-01	
12	INTERCEPT EFFICIENCY	=	9.512E-01	
13	RECTENNA RF-DC EFFICIENC	=	8.670E-01	
14	NET RF LINK EFFY	=	8.348E-01	
15	DC-TO-DC EFFICIENCY	=	5.879E-01	
16	DC-TO-GRID EFFICIENCY	=	5.702E-01	
17	OVERALL PHYSICAL EFFY	=	6.861E-02	
18	AREA EFFECTIVE EFFY	=	6.428E-02	
19	BLANKET AREA	=	1.017E+08	M2
20	ANTENNA DIA	=	1.000E+00	KM
21	REQUIRED SIDELOBE SUPPR	=	2.363E+01	DB
22	TAPER REQUIRED FOR SL SU	=	9.555E+00	DB
23	TRANSMITTER POWER TAPER	=	1.000E+01	DB
24	RECEIVER AVG/PEAK RATIO	=	2.061E-01	
25	XMTR AVG/PEAK RATIO	=	3.909E-01	
26	BEAM SPREAD FACTOR	=	1.450E+00	
27	RADIATED RF POWER	=	6.725E+03	MEGAWATT
28	BEAM DIAMETER	=	1.270E+01	KM
29	BEAM AREA	=	1.267E+08	M2
30	AVERAGE BEAM POWER DENS	=	4.753E+00	MW/CM2
31	PEAK BEAM INTENSITY	=	2.306E+01	MW/CM2
32	POWER IN MAIN BEAM	=	6.023E+03	MEGAWATT
33	SATELLITE LENGTH	=	3.104E+01	BAYS
34	NUMBER OF BAYS	=	2.484E+02	BAYS
35	XMTR PWR DISTR LOSS	=	2.985E-02	
36	ADJ BAY USEFUL AREA	=	4.096E+05	M2
37	BAY SIZE	=	6.600E+02	METERS
38	SPS AREA	=	1.086E+12	KM2
39	MEAN SOLAR INSOLATION	=	1.469E+02	GW
40	SOLAR CELL OUTPUT	=	1.759E+01	GW
41	ROTARY JOINT CURRENT "A"	=	1.344E+05	AMPS
42	ROTARY JOINT CURRENT "B"	=	7.895E+04	AMPS
43	TOTAL PROCESSED POWER	=	2.484E+03	MEGAWATT
44	TOTAL KLYSTRON INPUT	=	1.607E+04	MEGAWATT
45	TOTAL KLYSTRON OUTPUT	=	1.366E+04	MEGAWATT
46	NUMBER OF KLYSTRONS	=	1.897E+05	
47	MAX KLYSTRON PACKING DEN	=	3.291E+01	PER SUB
48	MAX RF POWER DENSITY	=	2.191E+01	KW/M2
49	NUMBER OF SUBARRAYS	=	7.261E+03	PER ANT
50	RECTENNA AREA	=	7.671E+07	M2

51	PEAK ANT THERMAL PWR	=	4.730E+00	KW/M2
52	DC OUTPUT POWER	=	4.868E+00	GW/LINK
53	GRID POWER	=	9.443E+00	GW TOTAL
54	LAND AREA PER RECT	=	1.798E+08	M2
55	"Y" MOM OF INERTIA	=	1.161E+14	KG-M2
56	THRUST PER CORNER	=	8.848E+01	NEWTONS
57	NUMBER OF THRUSTERS	=	8.848E+01	PER INST
58	CONTROL POWER	=	1.111E+02	MEGAWATT
59	ANNUAL PROPELLANT	=	3.625E+01	TONS
60	STRUCTURE MASS	=	5.700E+03	TONS
61	CONTROL SYS MASS	=	1.976E+02	TONS
62	SOLAR BLANKET MASS	=	4.344E+04	TONS
63	POWER DISTR MASS	=	2.431E+03	TONS
64	MECH & ELEC R/J MASS	=	1.022E+02	TONS
65	ANT STRUC MASS	=	5.000E+02	TONS
66	ANT WAVEGUIDE MASS	=	4.314E+03	TONS
67	ANT KLYSTRON MASS	=	1.318E+04	TONS
68	ANT CONTROL CKTS MASS	=	1.026E+03	TONS
69	ANT PWR DISTR MASS	=	9.471E+02	TONS
70	ANT PWR PROC&TC MASS	=	4.881E+03	TONS
71	ANT MASS	=	2.485E+04	TONS
72	STRUCTURE COST	=	2.850E-01	BILLION
73	CONTROL SYS COST	=	8.893E-02	BILLION
74	SOLAR BLANKET COST	=	3.560E+00	BILLION
75	POWER DISTR COST	=	6.321E-02	BILLION
76	MECH&ELEC R/J COST	=	2.146E-02	BILLION
77	ANT STRUC COST	=	3.485E-01	BILLION
78	ANT WAVEGUIDE COST	=	2.588E-01	BILLION
79	ANT KLYSTRON COST	=	5.997E-01	BILLION
80	ANT CONTROL CKTS COST	=	2.261E-01	BILLION
81	ANT PWR DISTR COST	=	1.025E-01	BILLION
82	ANT PWR PROC&TC COST	=	3.368E-01	BILLION
83	ANT COST	=	1.872E+00	BILLION
84	NO OF FREIGHT FLIGHTS	=	3.383E+02	
85	CREW SERVICE NO OF FLTS	=	1.933E+01	
86	OTS COST	=	9.077E-01	BILLION
87	TOTAL TRANSP COST	=	6.709E+00	BILLION
88	RECTENNA COST	=	4.749E+00	BILLION
89	CONSTRUCTION COST	=	1.160E+00	BILLION
90	INTEREST DURING CONSTR	=	1.417E+00	BILLION
91	LATITUDE AREA FACTOR	=	1.419E+00	
92	TOTAL MASS	=	9.666E+04	TONS
93	TOTAL COST	=	2.268E+01	BILLION
94	COST/KWE	=	2.402E+03	$
95	COST/KWH	=	4.464E+01	MILLS

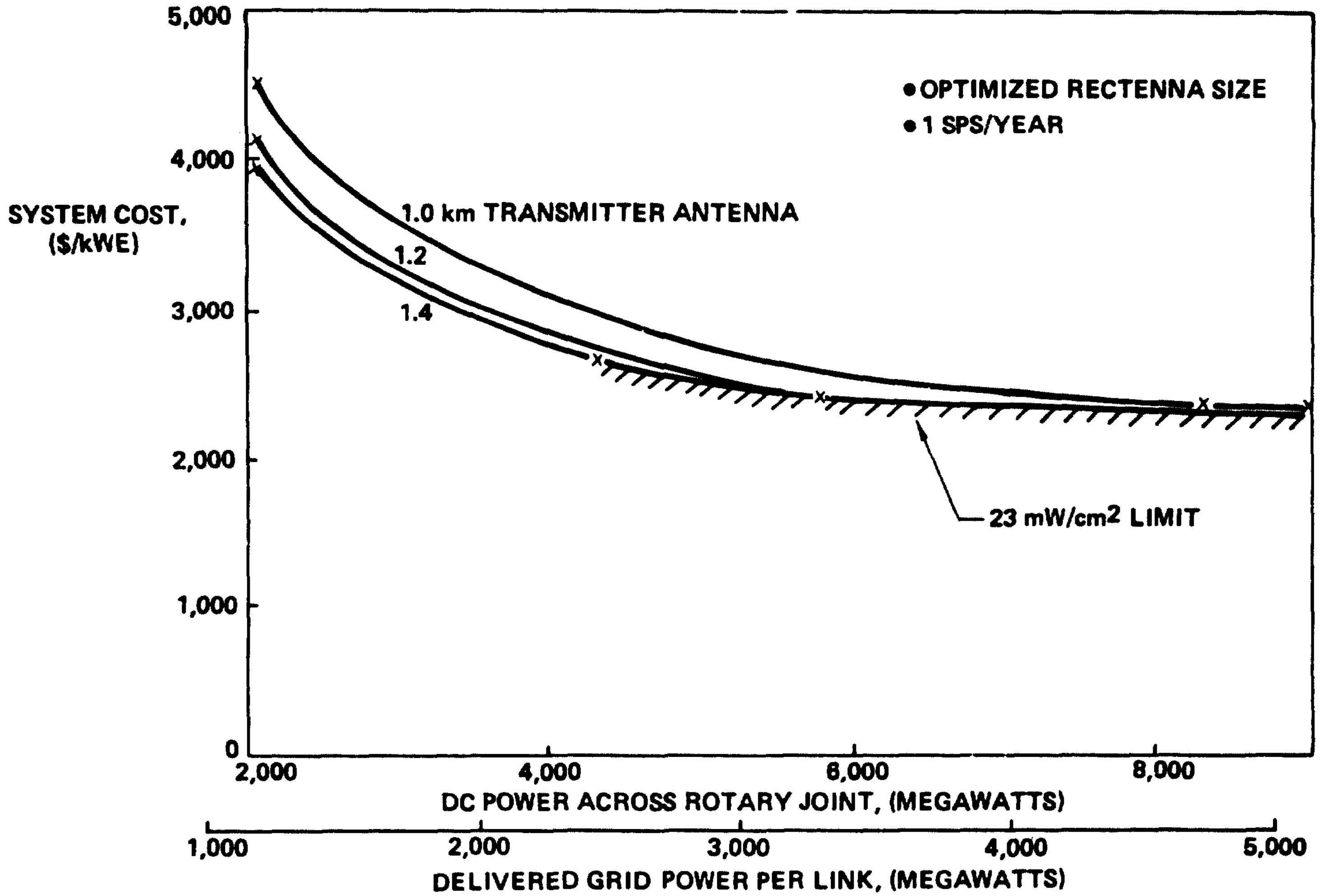
SIZE SENSITIVITY ANALYSIS POWER LEVEL AND TRANSMITTER DIAMETER

This chart shows a joint optimization of transmitter diameter and power level holding the rectenna size constant at the optimum value. As the system power level is reduced it is possible to employ somewhat larger transmitting antennas without violating the 23 mw/cm^2 limit. Transmitter diameters larger than 1.4 kilometers do not pay off; the minimum system cost in dollars per kilowatt follows along the 23 mw/cm^2 limit to about 2500 megawatts and then follows up the 1.4 kilometer diameter transmitter curve. Note that comparatively little cost penalty is incurred going down as low as 3000 megawatts of grid power. Below 3,000 megawatts the system cost in dollars per kilowatt begins to turn up rapidly.

SPS-1991

Size Sensitivity Analysis Power Level and Transmitter Diameter

BOEING



LFO TRANSPORTATION

The Earth to LEO transportation effort, during this portion of the study, concentrated on definition of 1) a 2-stage fully reusable winged SPS cargo vehicle and 2) a 2-stage reusable ballistic recoverable concept. A number of significant changes have been incorporated into the SPS cargo vehicle since the completion of the Part II study effort. These changes on the cargo vehicle include:

- o A methane/liquid oxygen fueled booster with flyback capability
- o Delta winged stages with a crew manning the orbiter and also the capability in the orbiter to transport personnel
- o Incorporation of a mid-body cargo bay in the orbiter capable of handling a payload density of 75 kg/m^3

The 2-stage ballistic recoverable vehicle has a payload in the 90 metric ton class that could possibly support the SPS precursor program.

Leo Transportation

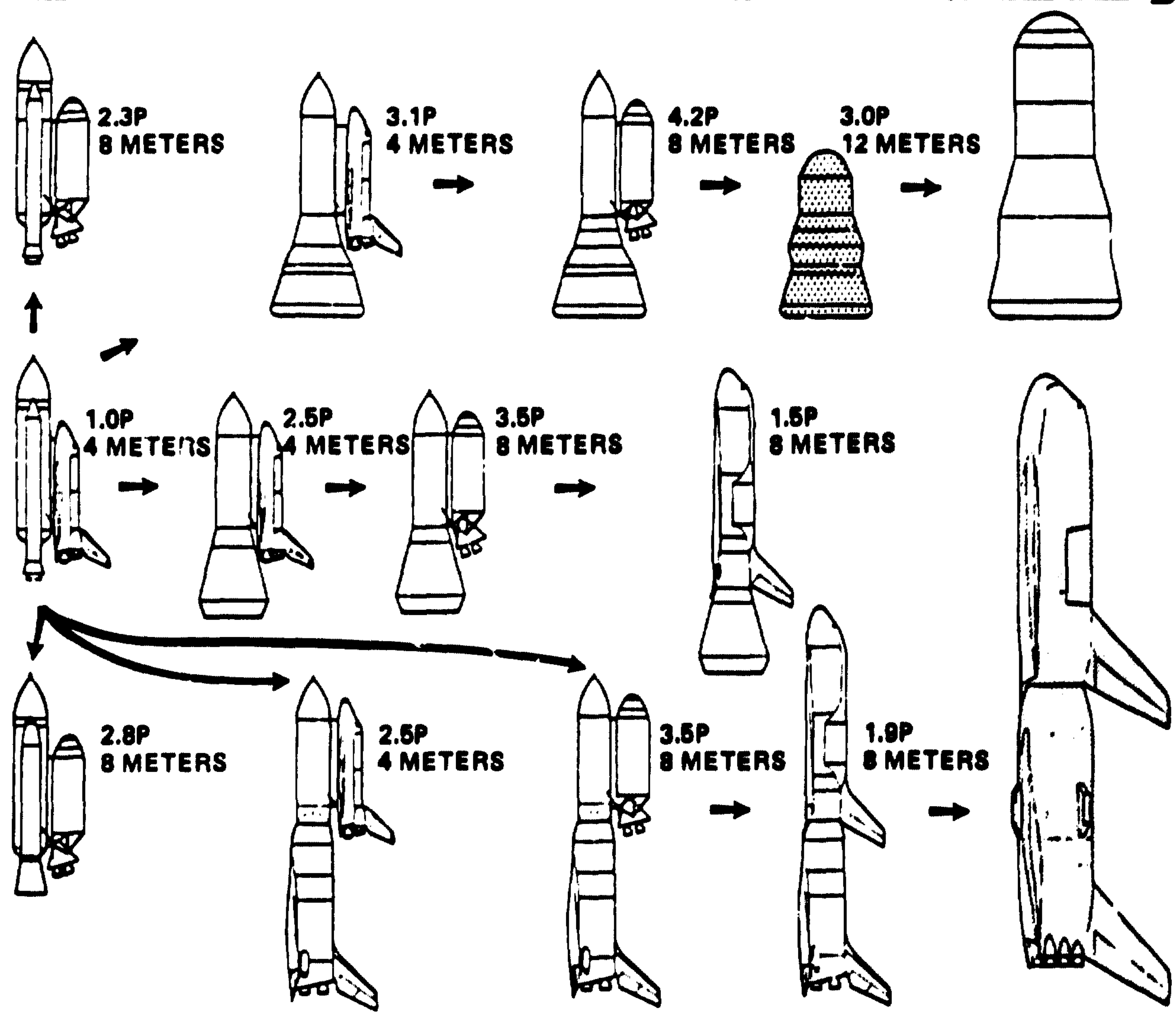
- TWO-STAGE FULLY REUSABLE WINGED FREIGHTER
- TWO-STAGE BALLISTIC RECOVERABLE CANDIDATE CONCEPT FOR THE PRECURSOR PROGRAM
- SUMMARY

POTENTIAL LAUNCH VEHICLE

The two vehicle concepts studied, identified by the shading on the opposite chart, are highlighted in the potential launch vehicle candidate family. The net payload capability to low earth orbit is

- ≈ 400 000 kg for the 2-stage winged SPS cargo vehicle
- ≈ 90 000 kg for the 2-stage ballistic recoverable vehicle

Potential Launch Vehicles



TWO-STAGE WINGED SPS LAUNCH VEHICLE (FULLY REUSABLE CARGO CARRIER)

The launch configuration of the SPS cargo vehicle is shown on the adjacent chart with the overall geometry noted. This series burn concept uses 16 LCH_4/LO_2 engines on the booster and 14 standard SSME's on the orbiter. The LCH_4/LO_2 booster engines are a gas generator cycle providing a vacuum thrust of 9.79×10^6 newtons each. The SSME's on the orbiter provide a vacuum thrust of 2.09×10^6 newtons (100% power level). The nominal 100% power level for the SSME's was selected based on engine life considerations which indicated about a 3 factor reduction in life if the 109% power level is used.

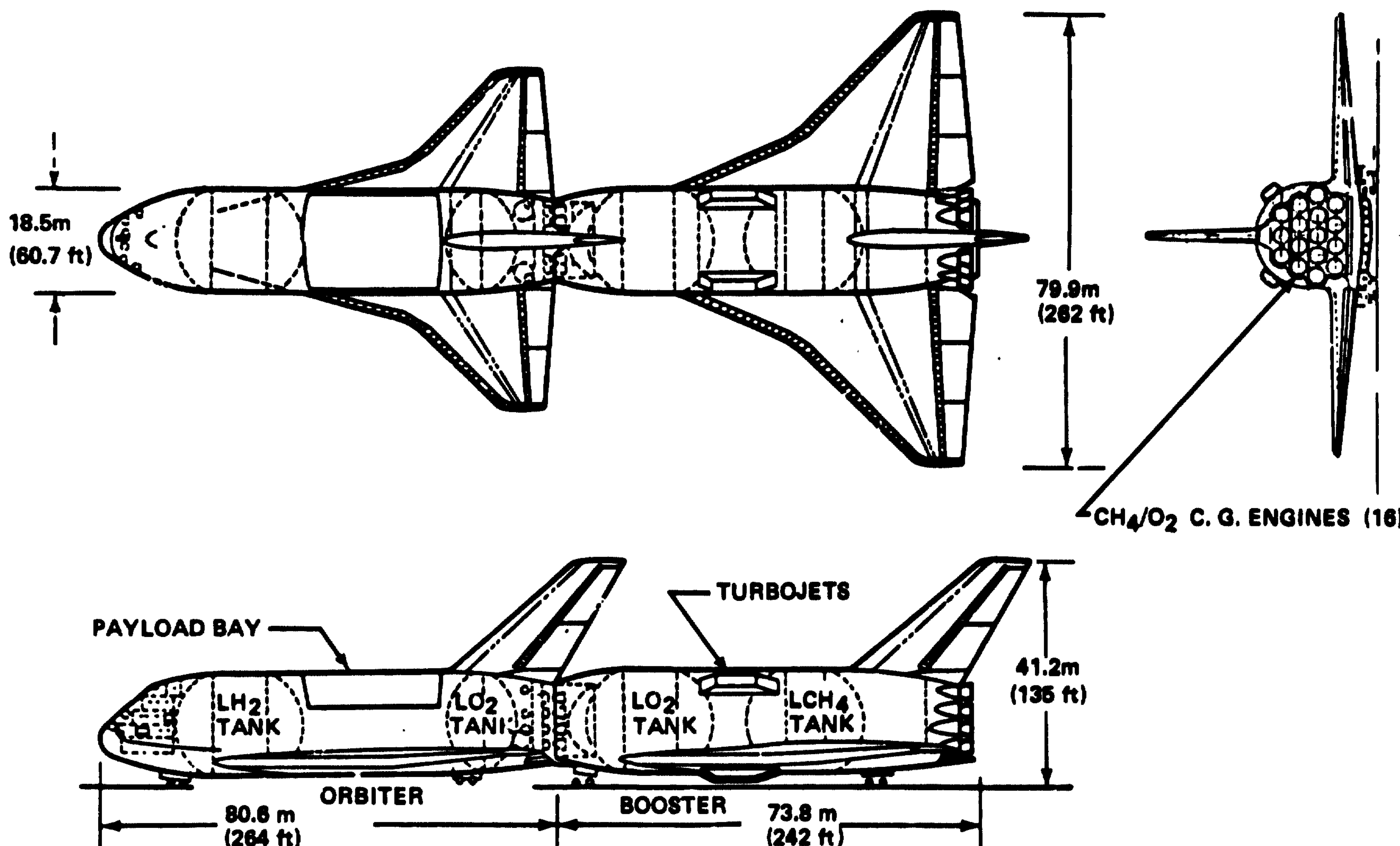
An airbreather propulsion system (12 installations of an SST type engine) has been provided on the booster for flyback capability to simplify the booster operational mode. The reference wing for both stages is

$$S_W \text{ (Orbiter)} = 1446 \text{ m}^2 \quad (15,560 \text{ ft}^2)$$
$$S_W \text{ (Booster)} = 2330 \text{ m}^2 \quad (25,080 \text{ ft}^2)$$

Heat sink thermal protection system is provided on the booster and the Shuttle's Reusable Surface Insulation (RSI) is used on the orbiter.

Two-Stage Winged SPS Launch Vehicle
(Fully Reusable Cargo Carrier)

SPS-1953



2-S AGE WINGED VEHICLE DESIGN CHARACTERISTICS

The vehicle characteristics are noted on the adjacent chart. The net delivered payload is 424 000 kg. A return payload of 15% (63 500 kg) of the delivered payload was assumed for the orbiter entry and landing conditions. The resulting mass fraction is 0.875 for the booster and 0.841 for the orbiter.

SPS

Two-Stage Winged Vehicle Design Characteristics

BOEING

SPS-1962

	ORBITER		BOOSTER
GLOW		10,978,400	
BLOW	—		7,813,700
BOOSTER FUEL (LCH_4)	—		1,708,900
BOOSTER OXIDIZER (LO_2)	—		5,126,700
BOOSTER INERTS	—		978,100
OLOW–LESS PAYLOAD	2,740,700		—
ORBITIER FUEL (LH_2)	329,400 *		—
ORIBITER OXIDIZER (LO_2)	1,976,200 *		—
ORBITER INERTS	435,100		—
ASCENT PAYLOAD	424,000		—
RETURN PAYLOAD ~ 15%	63,500		—
MASS FRACTION	0.841		0.875
ENTRY WEIGHT–NO PAYLOAD	335,200		936.600
–WITH RETURN P/L	456,000		—
START CRUISE WEIGHT–NO P/L	—		932,900
–WITH RETURN P/L	—		—
LANDING WEIGHT–NO PAYLOAD	391,800		846,700
–WITH RETURN P/L	452,600		—

(ALL MASS DATA IN kg)

*MAINSTAGE + FLIGHT PERFORMANCE RESERVE

ASCENT PERFORMANCE CHARACTERISTICS

The SP5 launch vehicle ascent characteristics are noted on the adjacent chart. A '3g' maximum acceleration thrust profile was used due to the manned capability and also to minimize the load conditions on the orbiter. The booster staging velocity of 2170 m/sec is well within the "heat sink" capability of the aluminum/titanium airframe.

Ascent Performance Characteristics

FIRST STAGE

T/W AT IGNITION	=	1.30	
MAXIMUM DYNAMIC PRESSURE	=	35.91 kPa	(750 psf)
MAXIMUM ACCELERATION	=	3.0g	
STAGE BURN TIME	=	155.24 sec	
RELATIVE STAGING VELOCITY	=	2170 m/sec	(7,120 fps)
DYNAMIC PRESSURE AT STAGING	=	1.16 kPa	(24 psf)

SECOND STAGE

INITIAL T/W	=	0.94
MAXIMUM ACCELERATION	=	3.0 g
STAGE BURN TIME	=	350.24 sec

SPS WINGED VEHICLE
REENTRY CHARACTERISTICS

The reentry characteristics for the booster and orbiter are noted on the opposite chart. The maximum deceleration for the booster is 4.27 g's and the subsonic transition altitude is 17.86 km. The orbiter reentry has been limited to a normal load factor of 1.41 g's until the subsonic transition which occurs at an altitude of 13.62 km.

SPS Winged Vehicle Reentry Characteristics

SPS-1954

BOOSTER

APOGEE CONDITIONS

$h = 80.82$ km

$V_{rel} = 1955$ m/sec

MAXIMUM DECELERATION CONDITION

$q = 10.77$ kPa

$h = 32.61$ km

$V_{rel} = 1327$ m/sec

NORMAL LOAD FACTOR = 4.27 g's

MAXIMUM DYNAMIC PRESSURE CONDITION

$q = 13.29$ kPa

$h = 22.96$ km

$V_{rel} = 686$ m/sec

NORMAL LOAD FACTOR = 1.49 g's

SUBSONIC TRANSITION CONDITION

$h = 17.86$ km

$\alpha = 15$ deg

ORBITER

MAXIMUM DYNAMIC PRESSURE CONDITION

$q = 13.17$ kpa

$h = 15.55$ km

$V_{rel} = 361$ m/sec

NORMAL LOAD FACTOR = 1.41

SUBSONIC TRANSITION CONDITION

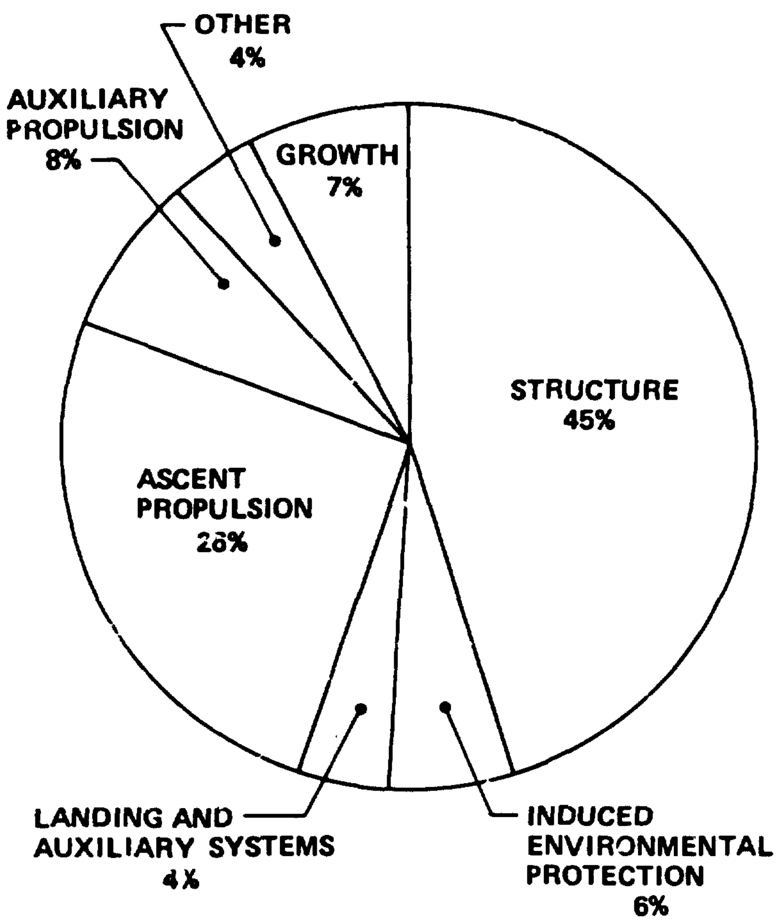
$h = 13.62$ km

$\alpha = 6.4$ deg

SPS BOOSTER MASS STATEMENT

The flyback booster mass characteristics are shown on the opposite chart. The structure, induced environment protection, ascent and auxiliary propulsion, and landing subsystems account for 89% of the dry mass. The induced environmental protection subsystem mass includes the additional structural thickness required for "heat sink capability" and the base heat shield.

Booster Mass Statement



DRY MASS BREAKDOWN

	MASS (kg)
STRUCTURE	360 800
INDUCED ENVIRONMENTAL PROTECTION	46 400
LANDING AND AUXILIARY SYSTEMS	34 500
ASCENT PROPULSION	204 600
AUXILIARY PROPULSION	60 600
PRIME POWER	4 300
ELECTRICAL CONVERSION AND DISTRIBUTION	4 200
HYDRAULIC CONVERSION AND DISTRIBUTION	10 900
SURFACE CONTROLS	10 300
AVIONICS	1 500
ENVIRONMENTAL CONTROL	200
GROWTH	58 600
DRY MASS =	796 900
RESIDUALS AND RESERVES	49 800
LANDING MASS =	846 700
LOSSES DURING FLYBACK	86 200
START FLYBACK MASS =	932 900
ENTRY IN-FLIGHT LOSSES	3 700
START ENTRY MASS =	936 600
IN-FLIGHT LOSSES PRIOR TO ENTRY	27 000
STAGING MASS =	963 600
THRUST DECAY PROPELLANT	14 500
INERT MASS =	978 100

SPS ORBITER MASS STATEMENT

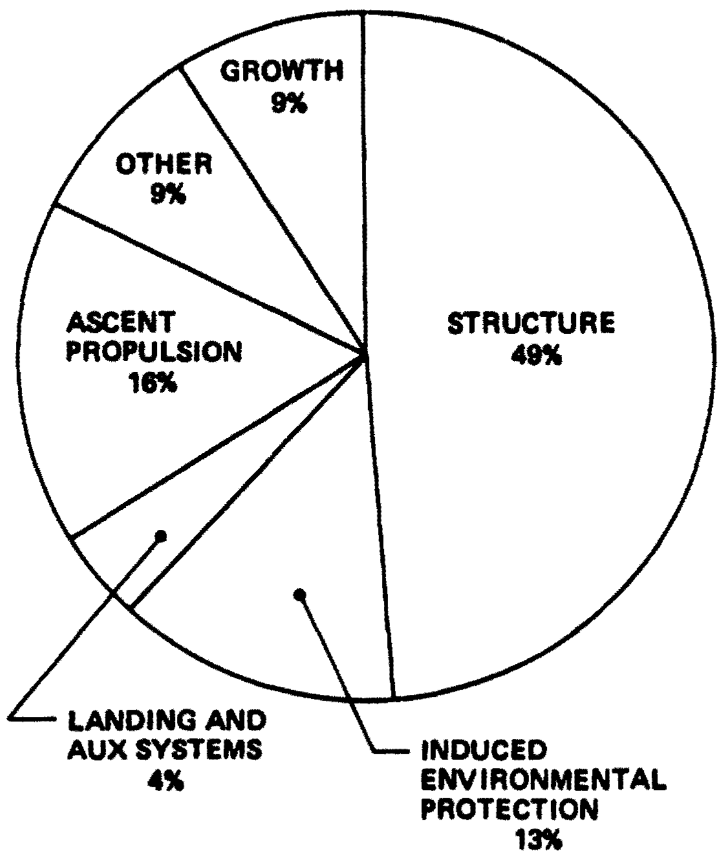
The orbiter mass characteristics are shown on the opposite chart. Structure accounts for approximately 50% of the stage dry mass. The ascent propulsion and thermal protection subsystems are an additional 29% of the dry mass. The dry mass is 86% of the inert mass with the remainder including residuals and reserves, personnel and payload accommodations, and inflight losses.

Orbiter Mass Statement

SPS

BOEING

SPS-1955



DRY MASS BREAKDOWN

	MASS (kg)
STRUCTURE	182 900
INDUCED ENVIRONMENTAL PROTECTION	48 300
LANDING AND AUX SYSTEMS	15 800
ASCENT PROPULSION	60 800
AUXILIARY PROPULSION	9 500
PRIME POWER	2 500
ELECTRICAL CONVERSION AND DISTRIBUTION	4 800
HYDRAULIC CONVERSION AND DISTRIBUTION	3 600
SURFACE CONTROLS	6 800
AVIONICS	2 400
ECLSS AND PERSONNEL PROV	2 900
GROWTH	32 900
DRY MASS =	373 200
PERSONNEL AND PAYLOAD ACCOMMODATIONS	4 100
RESIDUAL AND RESERVES	14 500
LANDING MASS =	391 800
ENTRY IN-FLIGHT LOSSES	3 400
START ENTRY MASS =	395 200
IN-FLIGHT LOSSES PRIOR TO ENTRY	39 900
INERT MASS =	435 100

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C-2

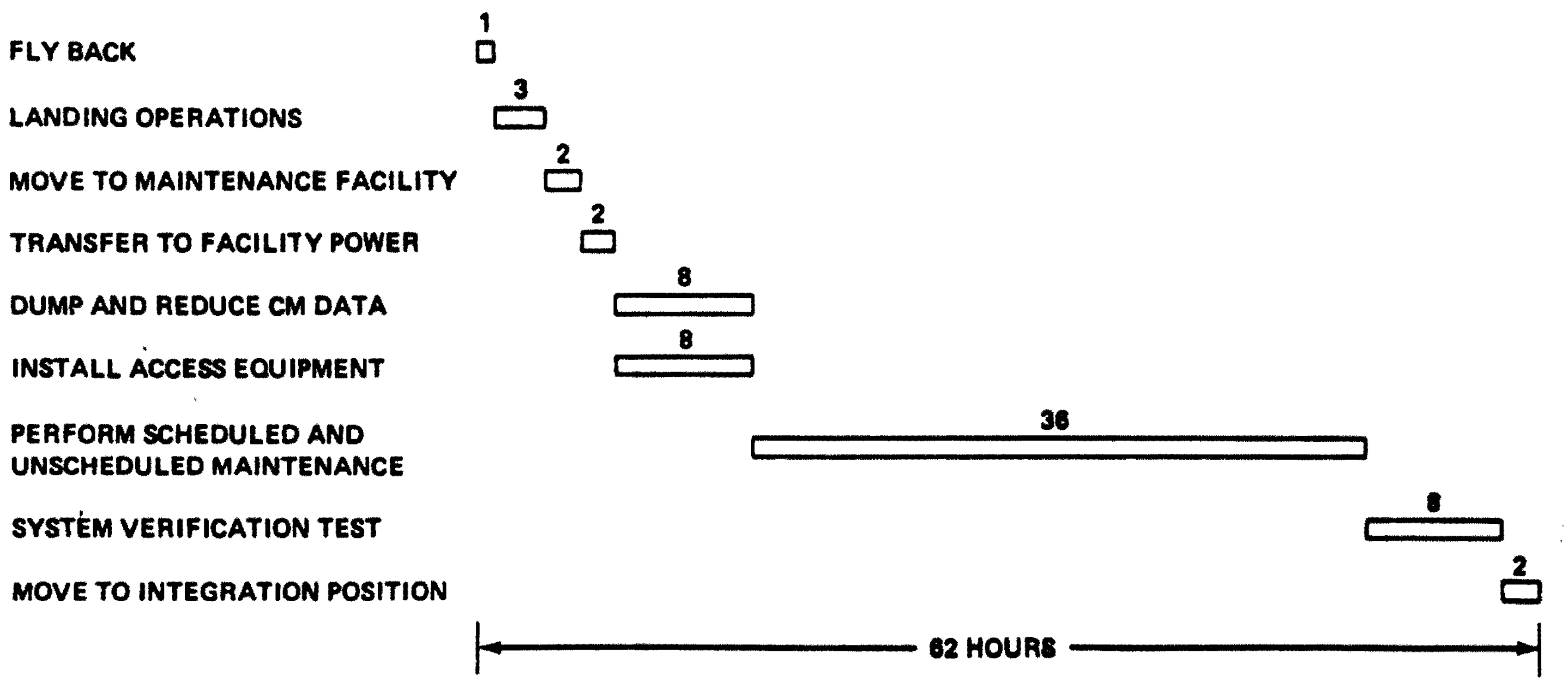
BOOSTER PROCESSING TIMELINES

The booster timeline from launch to its move to the integration position is shown on the chart. These timelines reflect the average turnaround times for the mature system. A total of 62 hours is estimated for this portion of the turnaround with the scheduled and unscheduled maintenance activity requiring 36 hours. On-board condition monitoring equipment will enhance the operations by

1) Providing performance monitoring of the subsystems
2) Aiding in fault isolation and detection

Rocket engine maintenance is anticipated to be the major portion of the booster operations.

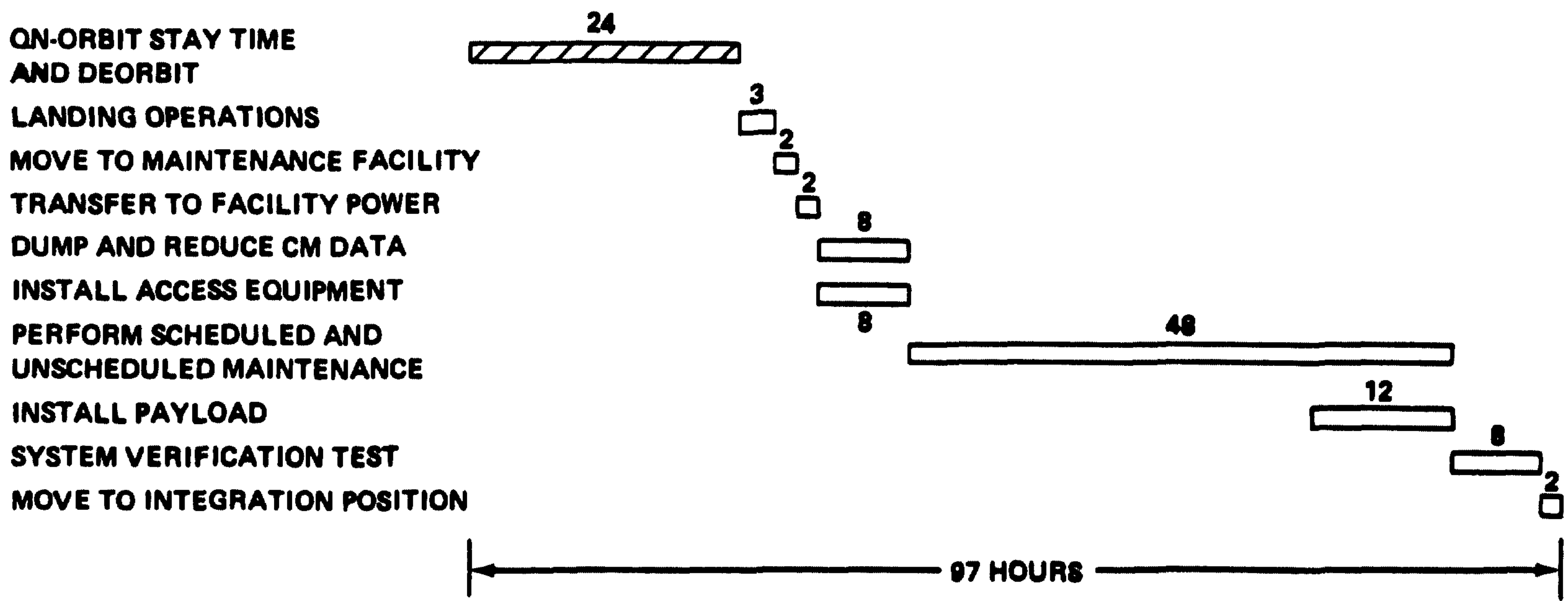
Booster Processing Timelines



ORBITER PROCESSING TIMELINES

The orbiter timeline from launch to its move to the integration position is shown on the chart. A total time of 97 hours for orbiter processing including the 24 hour-on-orbit staytime is estimated for the mature system. The maintenance activity is estimated to be 48 hours, due to the thermal protection system and the additional systems/equipment required for the manned stage. A total of 12 hours has been allocated for payload installation in a parallel operation with the orbiter maintenance.

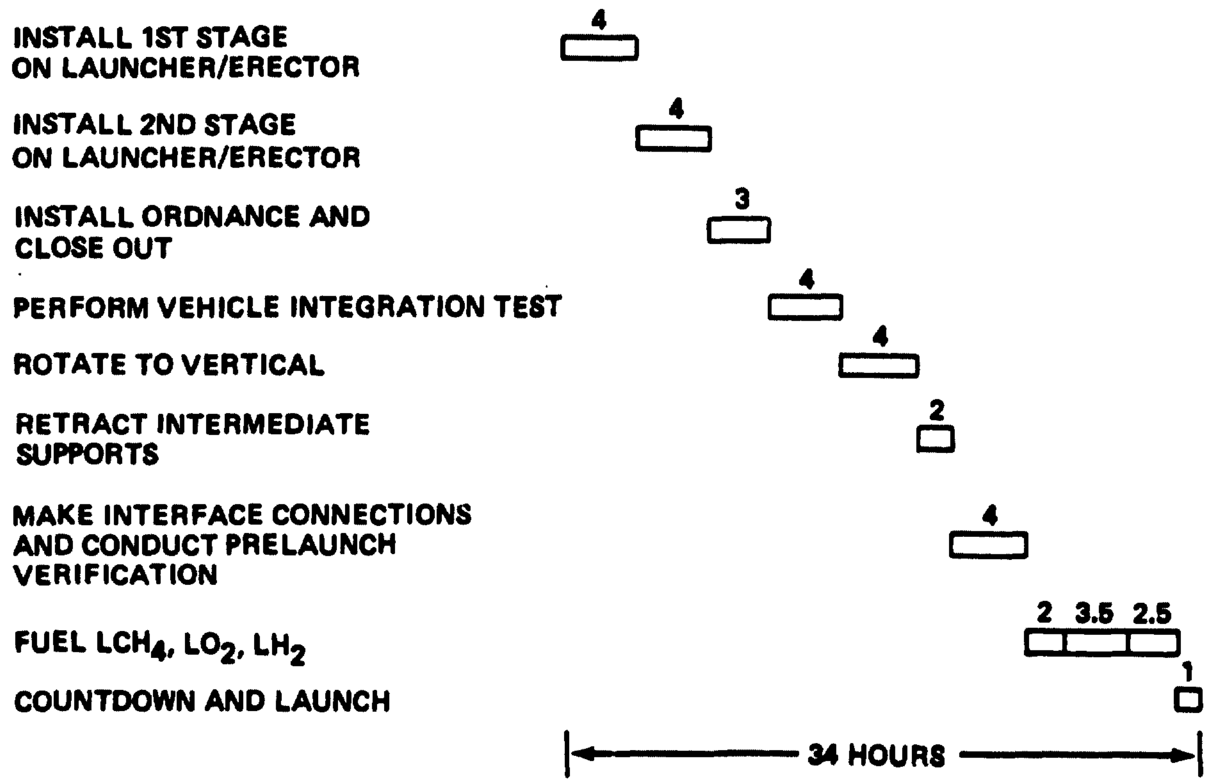
Orbiter Processing Timelines



INTERGRATED VEHICLE OPERATIONS TIMELINES

The integrated vehicle timelines for operations beginning with booster positioning through launch are shown on the adjacent chart. This portion of the launch operations requires 34 hours for the booster and 30 hours for the orbiter.

Integrated Vehicle Operations Timelines



VEHICLE TURNAROUND ANALYSIS SUMMARY

The total turnaround times for the SPS winged launch vehicle are shown on the chart. The total booster turnaround time is 97 hours and the corresponding orbiter time is 127 hours. The estimated 2-stage ballistic recoverable turnaround times are shown for reference. The flyback capability on the booster, with its inherent return to launch site ability, provides for a minimum turnaround time.

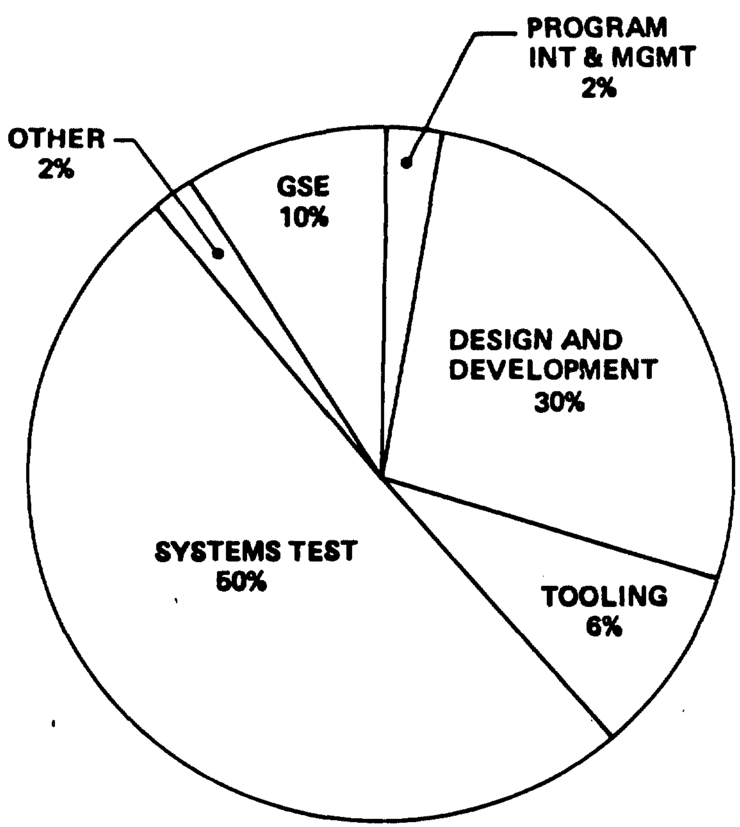
Vehicle Turnaround Analysis Summary

VEHICLE CONCEPT	STAGE OPS ONLY	INTEGRATION AND LAUNCH OPS	TOTAL TURNAROUND
WING/WING			
BOOSTER	63 HOURS	34 HOURS	97 HOURS
ORBITER	97 HOURS	30 HOURS	127 HOURS
BALLISTIC/BALLISTIC			
BOOSTER	93 HOURS	34 HOURS	127 HOURS
UPPER STAGE	102 HOURS	30 HOURS	132 HOURS

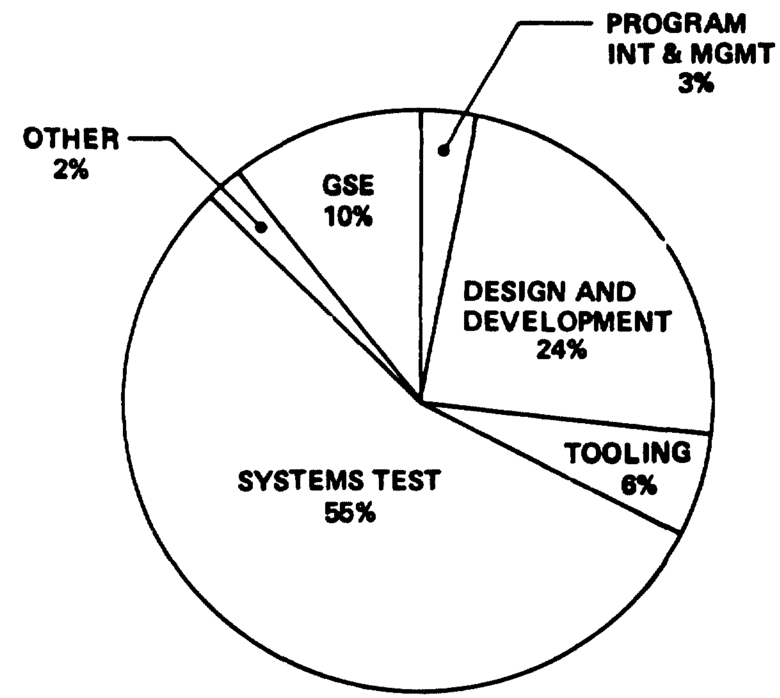
SPS LAUNCH VEHICLE DDT&E COST

The DDT&E cost per the flight hardware and its associated ground support equipment is shown on the adjacent chart for both the booster and orbiter stages. The total development cost for both stages is $11.2B. Systems test, which includes all the ground and flight test hardware in addition to the test labor, accounts for in excess of 50% of the total development cost. The booster DDT&E cost includes a new rocket engine and airbreather engine development. The orbiter DDT&E reflects use of the Space Shuttle's SSME's and some of the other subsystems which were modified rather than new developments.

SPS Vehicle DDT&E Cost



BOOSTER DDT&E = $6,528M



ORBITER DDT&E = $4,674M

- TOTAL VEHICLE DDT&E = $11.2B (LESS FACILITIES)

SPS LAUNCH VEHICLE PRODUCTION COST

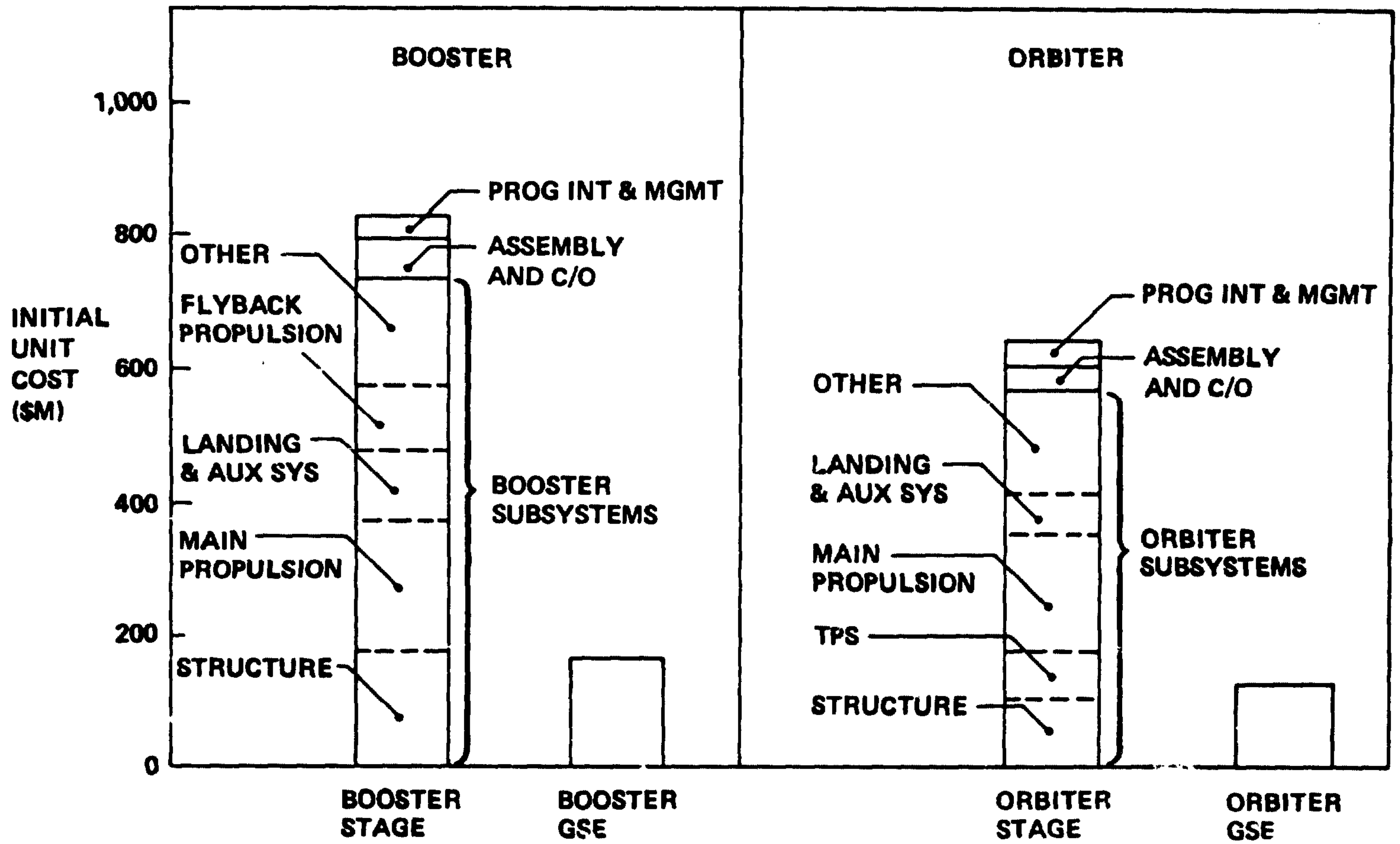
The initial unit production cost for both the SPS cargo vehicle booster and orbiter is shown on the adjacent chart. The theoretical first unit cost (TFU) for the booster of $821.4M and $638.5M for the orbiter were developed using the Boeing Parametric Cost Model (PCM). The following is a breakdown of the TFU cost by major subsystem:

SUBSYSTEM	BOOSTER	ORBITER
Structure	21%	16%
TPS	N/A	10%
Main Propulsion	24%	28%
Landing and Aux. Sys.	13%	9%
Flyback Propulsion	11%	N/A
Other Subsystems	19%	26%

The ground support equipment TFU cost is estimated to be $162.8M and $126.0M for the booster and orbiter respectively.

SPS Launch Vehicle Production Cost

SPS-1958



SPS LAUNCH VEHICLE AVERAGE COST/FLIGHT (1 SATELLITE/YEAR)

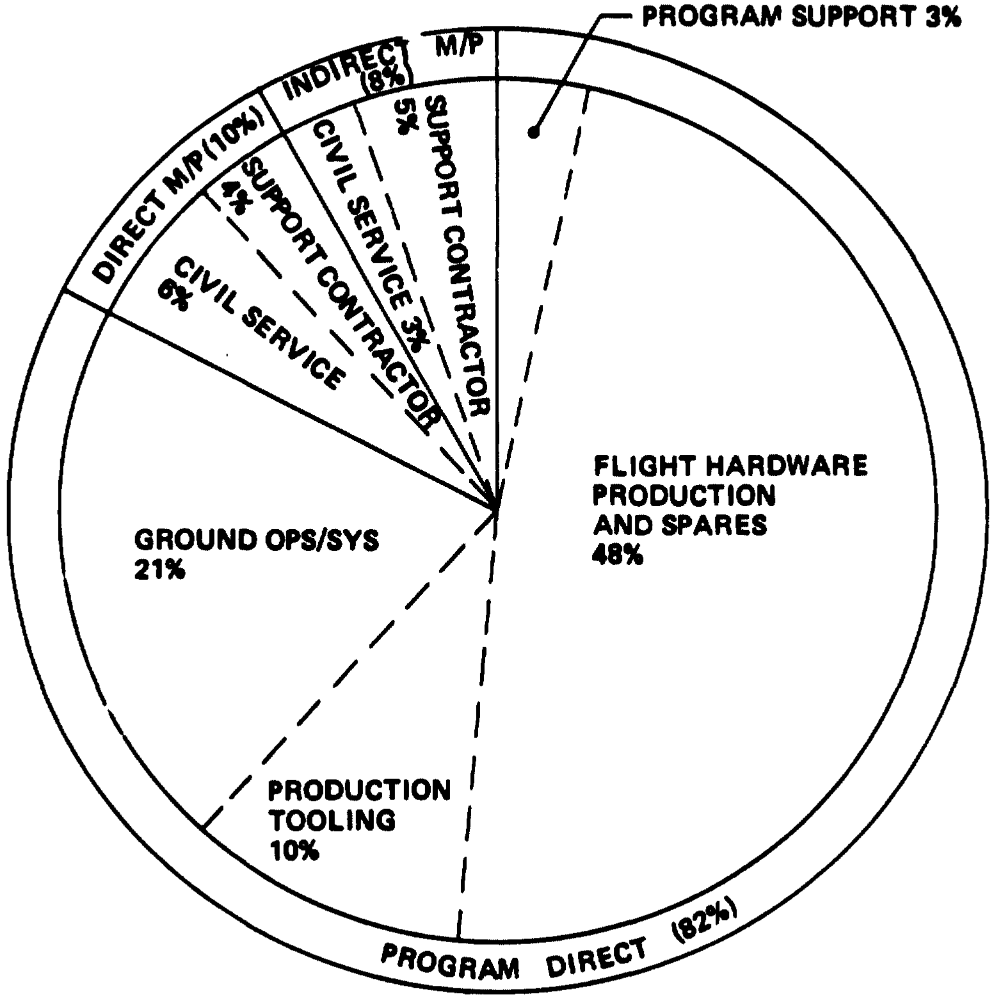
The cost/flight breakdown shown on the opposite page is the average for the 400 per year launch rate and 14 years of operation. The cost/flight items follow the Shuttle User Charge Policy guidelines with the following additions

1) Amortization of the fleet production costs
2) Inclusion of the rate tooling cost due to the hardware quantities required.

Flight Hardware production and spares is the largest single item with the booster and orbiter accounting for 55% and 45%, respectively. Propellant cost amounts to 12% of the total per flight cost.

SPS Launch Vehicle Average Cost/Flight (One Satellite/Year)

SPS-1963



- TWO-STAGE WINGED VEHICLE
- PLACEMENT OF 1 SATELLITE PER YEAR (400 FLIGHTS)
- 14 YEAR PROGRAM
- NO ATTRITION

AVERAGE COST/FLIGHT = $13.447M

EFFECT OF LAUNCH RATE ON COST PER FLIGHT

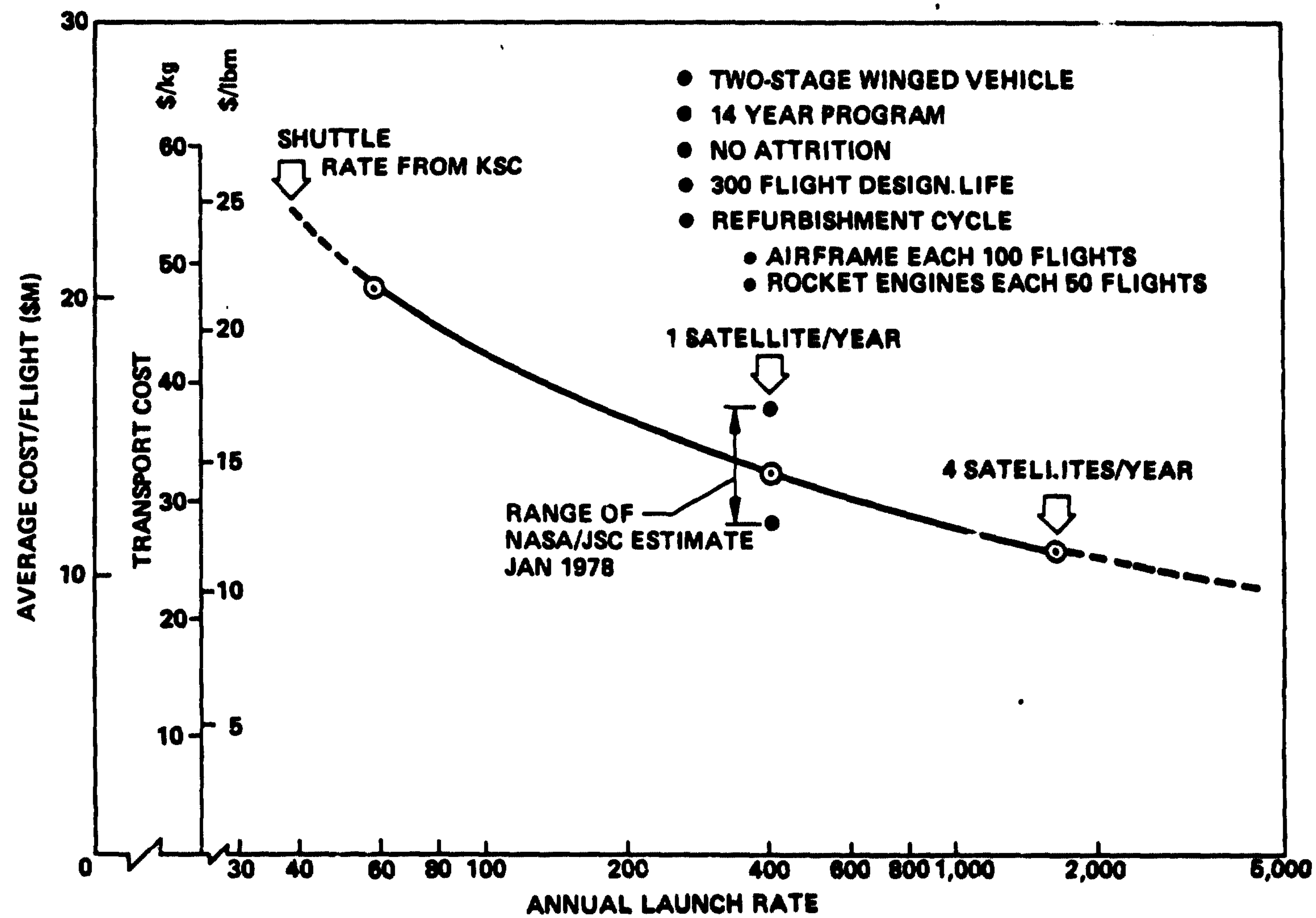
The chart on the opposite page illustrates the effect of launch rate on the average cost/flight and the transport cost to low Earth orbit for the SPS cargo vehicle. The required launch rate of approximately 400 flights per satellites results in the following:

Annual Launch Rate	Cost/Flight	Transport Cost $/kg	($/lbm)
400 Flights	$13.447M	31.71	(14.38)
1600 Flights	$10.754M	25.36	(11.50)

A 40 launch per year rate, comparable to the planned rate for Shuttle from KSC, would result in an average cost of $23M per flight for the SPS cargo launch vehicle. Also noted on the chart, are the NASA/JSC in-house cost estimates as of January 1978.

Effect of Launch Rate on Cost Per Flight

SPS-1964

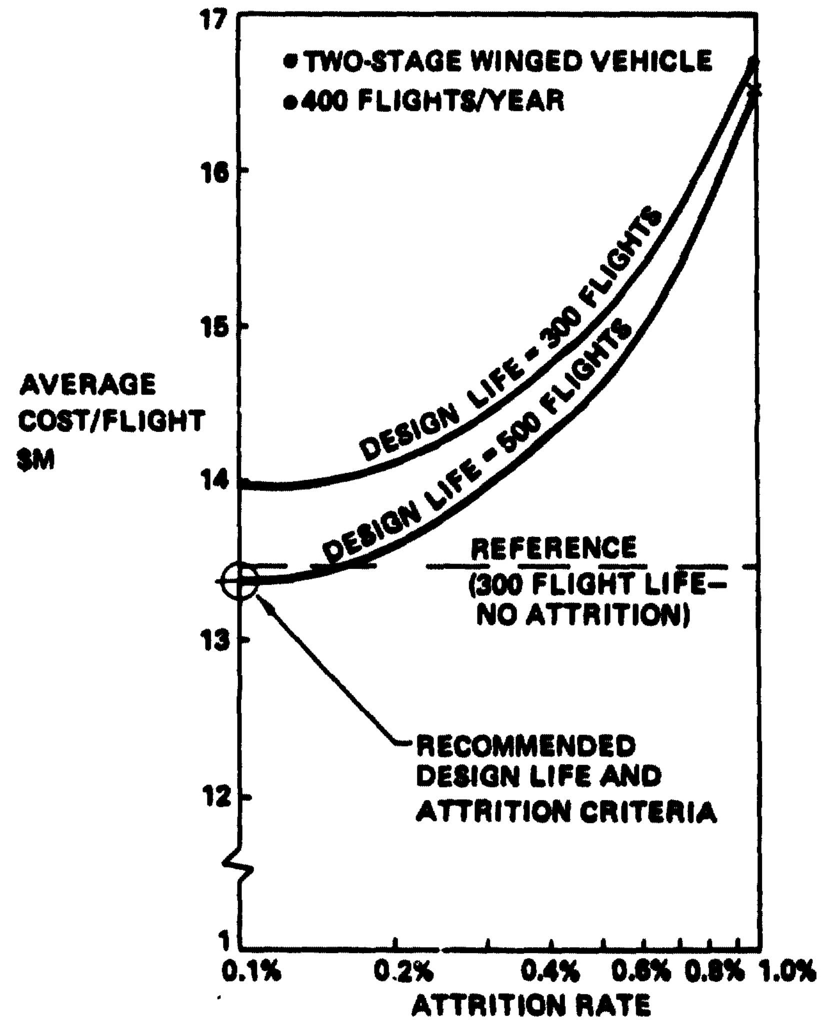


EFFECT OF DESIGN LIFE AND ATTRITION RATE

A sensitivity analysis was conducted to determine the impact of various design life and attrition rate criteria. The results of this analysis are shown on the adjacent chart. Attrition rates of between 0.1% and 1% were evaluated along with a design life criteria of 300 and 500 flights. The attrition rate influence for the range of values investigated resulted in a 32% variation in the average cost per flight. Design life has a decreasing influence as the attrition rate increases. A recommended criteria for the 2-stage winged vehicle is a 500 flight design life and 0.1% attrition rate which should be achievable within the time span available.

Effect of Design Life and Attrition Rate

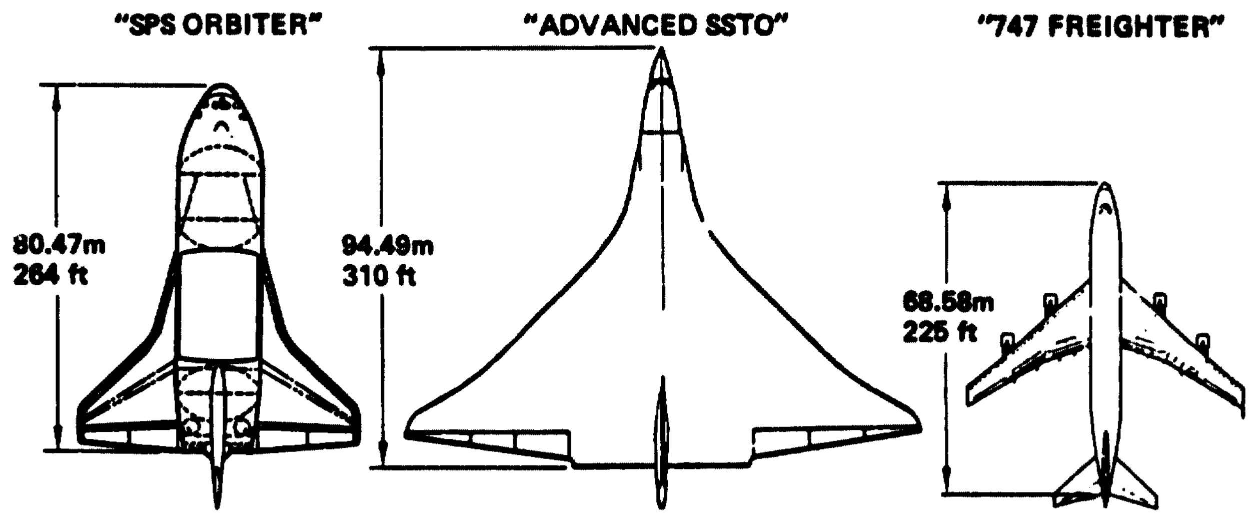
SPS-1966



VEHICLE SIZE COMPARISON

A comparison of the physical size (planform area) for the SPS launch vehicle orbiter (2nd stage only), a single stage to orbit (SSTO) vehicle, and a 747 Freighter are shown on the opposite chart. For the two space transportation concepts, the gross liftoff weight (GLOW) to payload ratio are noted. The two-stage SPS vehicle has a liftoff weight to payload ratio of 25.9 which uses the benefits of staging, whereas the SSTO has a ratio of 19.1. The inert and dry mass data are noted on the chart. The "Advanced SSTO" mass data reflects the maximum to achieve the desired performance level. The required maximum SSTO mass is about 33% greater than the dry weight of a 747 Freighter or 47% of the SPS Orbiter (2nd Stage) inert mass.

Vehicle Size Comparison



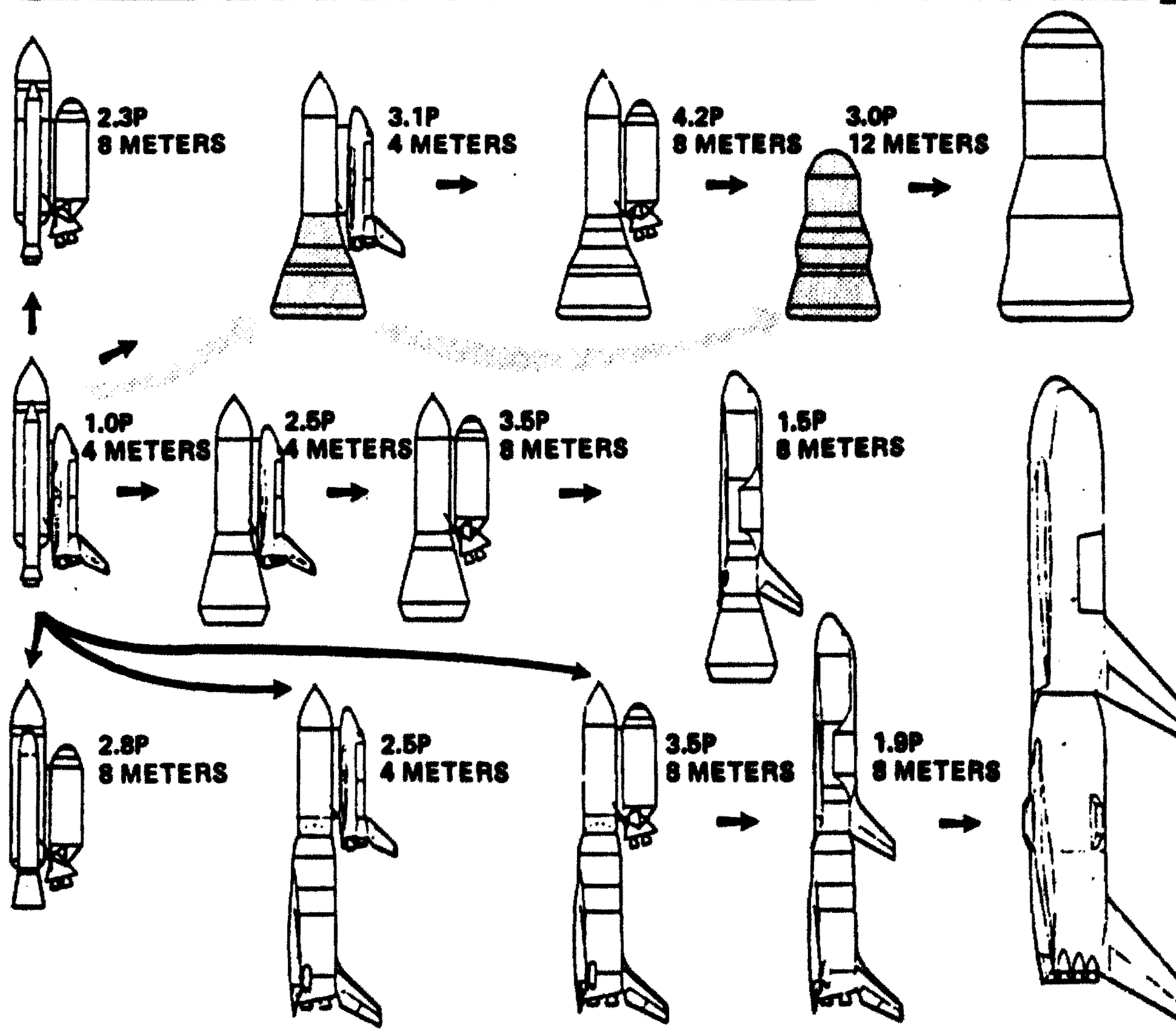
	"SPS ORBITER"	"ADVANCED SSTO"	"747 FREIGHTER"
PAYLOAD	424 000 kg	90 700 kg	90,700 kg
GLOW/PAYLOAD	25.9	19.1	4.0
MASS DATA	W_{INERT} = 435 100 kg	W_{INERT} = 204 110 kg	
	W_{DRY} = 373 200 kg	W_{DRY} = 181 450 kg	W_{DRY} = 152 860 kg
	$W_{LIFTOFF}$ = 3 164 700 kg	$W_{TAKEOFF}$ = 1 732 740 kg	W_{GROSS} = 365 150 kg
ASCENT PROPULSION SYSTEM	14-SSME'S	?? AIRBREATHERS/ RAMJETS + 3 SSME'S	4 JT-9D'S OR CF-6'S

POTENTIAL LAUNCH VEHICLES

The other vehicle concept investigated in this portion of the SPS study is the 2-stage ballistic recoverable vehicle in the 90 metric ton payload class. The potential evolution of this vehicle is noted on the adjacent chart of launch vehicle family concepts. The vehicle evaluation would begin with a booster (including a new gas generator engine) to support an increased performance Space Shuttle, and then proceed to a fully reusable upper stage for cargo missions. The following charts will describe the 2-stage reusable ballistic concept.

D180-24071-3

Potential Launch Vehicles

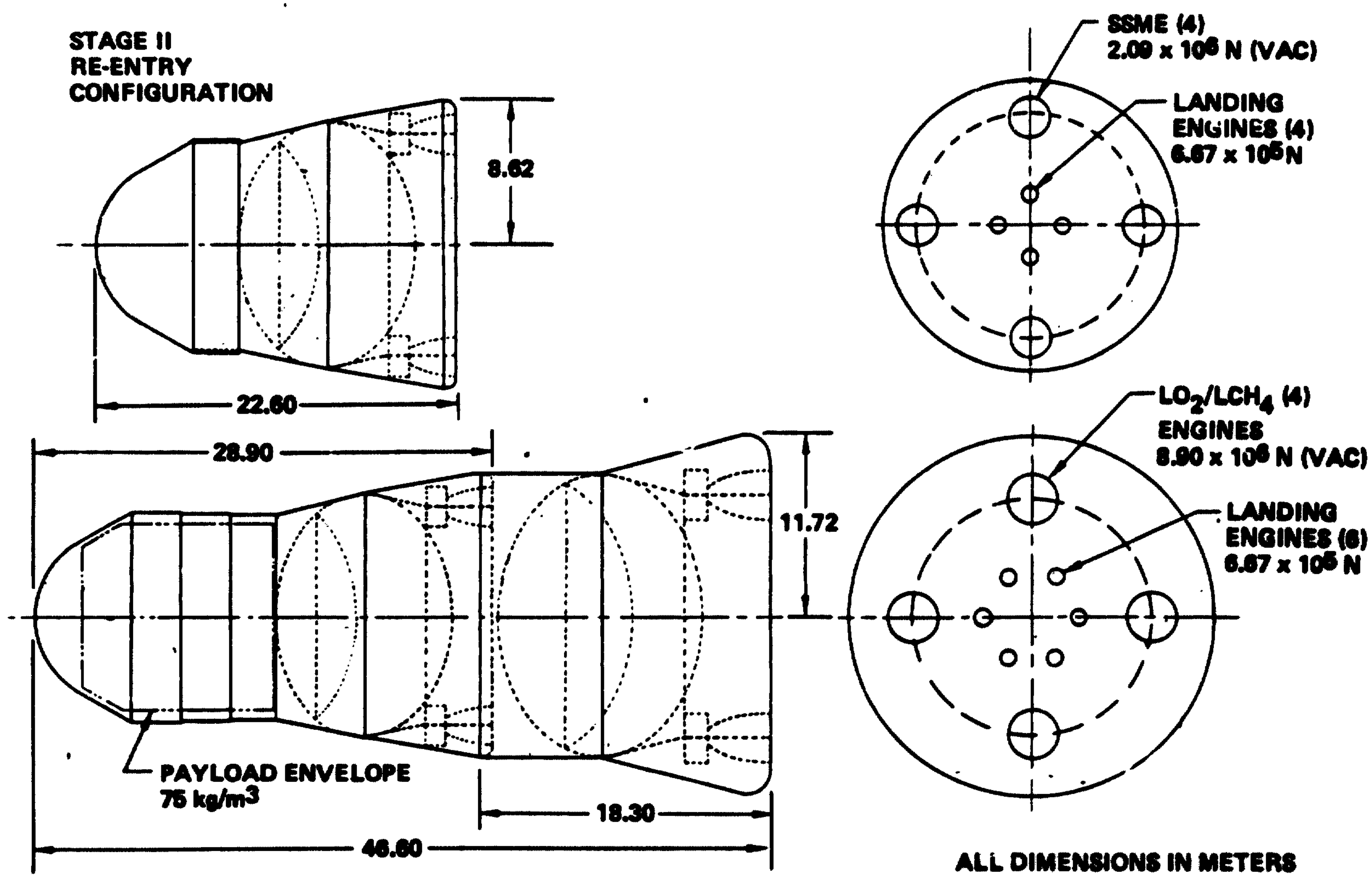


TWO-STAGE BALLISTIC SPS LAUNCH VEHICLE (FULLY REUSABLE)

The ballistic launch vehicle configuration is shown on the adjacent chart. This is a series burn configuration in which both stages re-enter ballistically and soft-land on the water. Ascent propulsion for the booster consists of 4 LCH_4/LO_2 gas generator cycle engines providing 8.90×10^6 newtons vacuum thrust each. Four standard SSME's operating at 100% power level (2.09×10^6 newtons vacuum thrust each) provide ascent propulsion for the second stage.

Both stages are equipped with pressure fed landing engines to provide terminal deceleration after ballistic re-entry. Thermal protection for both ascent and re-entry is accomplished by water cooled base heat shields in both stages.

Two-Stage Ballistic Vehicle Concept (Precursor)



TWO-STAGED BALLISTIC LAUNCH VEHICLE
DESIGN CHARACTERISTICS

The two-stage ballistic launch vehicle characteristics are specified on the adjacent chart. The vehicle delivers a net payload of 93,700 kg. The booster and orbiter stages have mass fractions of .903 and .812 respectively.

2 Stage Ballistic Vehicle (Precursor) Design Characteristics

	ORBITER	BOOSTER
GLOW	2,589,480	
BLOW	—	1,921,810
BOOSTER FUEL (LCH_4)	—	434,000
BOOSTER OXIDIZER (LO_2)	—	1,301,940
BOOSTER INERTS	—	185,870
OLOW–LESS PAYLOAD	573,970 *	—
ORBITER FUEL (LH_2)	66,560 *	—
ORBITER OXIDIZER (LO_2)	399,390	—
ORBITER INERTS	108,020	—
ASCENT PAYLOAD	93,700	—
MASS FRACTION	.812	)3
ENTRY WEIGHT	99,500	180,190
LANDING WEIGHT	89,870	163,180

*MAINSTAGE + FLIGHT PERFORMANCE RESERVE

(ALL MASS DATA IN kg)

ASCENT PERFORMANCE CHARACTERISTICS

The ascent performance characteristics of the two-stage ballistic vehicle are noted on the adjacent chart. The booster staging velocity is 2477 m/sec (8125 fps) and the maximum acceleration experienced is 4.23 g's at booster burnout.

SPS Ballistic Vehicle (Precursor) Ascent Performance Characteristics

FIRST STAGE		
T/W @ IGNITION	=	1.28
MAXIMUM DYNAMIC PRESSURE	=	29.35 kPa (613 PSF)
MAXIMUM ACCELERATION	=	4.23 g's
STAGE BURN TIME	=	168.39 SEC
RELATIVE STAGING VELOCITY	=	2477 M/SEC (8125 FPS)
DYNAMIC PRESSURE AT STAGING	=	1.48 kPa (31 PSF)
SECOND STAGE		
INITIAL T/W	=	1.28
MAXIMUM ACCELERATION	=	4.15 g's
STAGE BURN TIME	=	246.84 SEC

SPS BALLISTIC LAUNCH VEHICLE
MASS STATEMENT

The mass characteristics of the ballistic launch vehicle are shown on the adjacent chart. Structure accounts for 46% and 59% of the dry masses of the booster and orbiter respectively. Ascent propulsion is the other major fraction of booster and orbiter masses accounting for 39% and 21% respectively.

SPS Ballistic Launch Vehicle (Precursor) Mass Statement

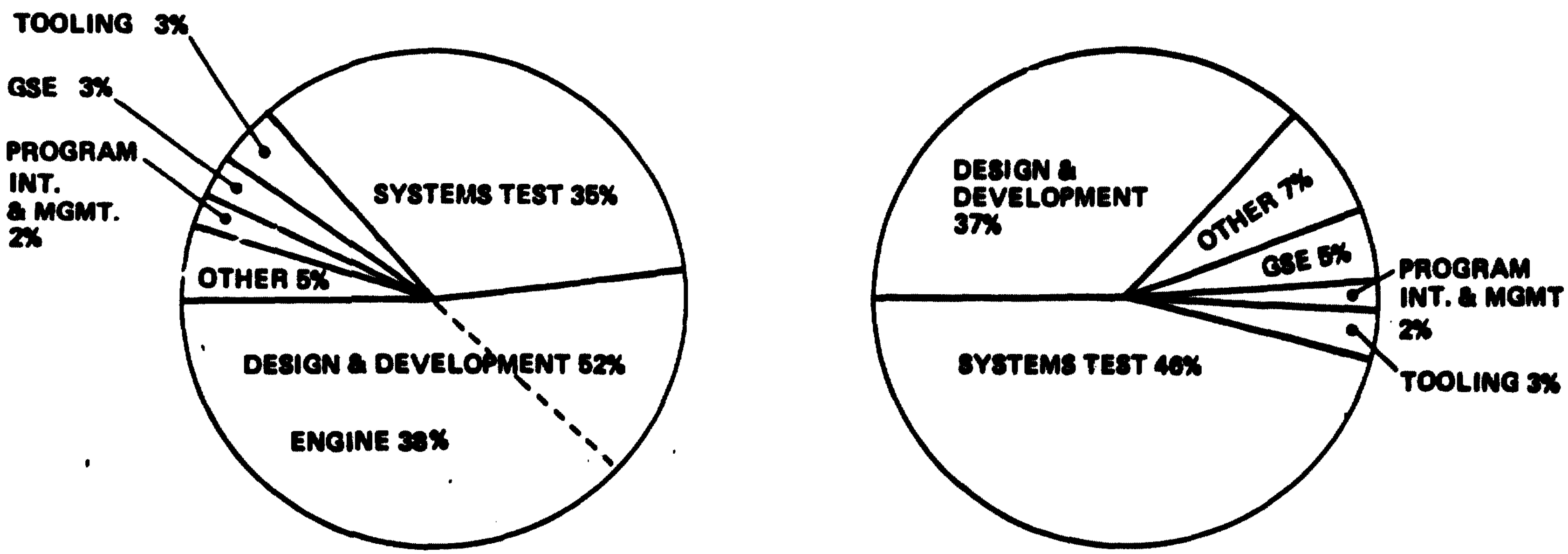
	BOOSTER (10^3 KG)	ORBITER (10^3 KG)
STRUCTURE	60.36	49.80
LANDING & AUX. SYSTEMS	4.73	2.70
ASCENT PROPULSION	50.69	17.93
AUX. PROPULSION	1.12	2.60
PRIME POWER	1.00	.81
HYD. CONV./DIST.	.95	.73
ELECTRIC CONV./DIST.	1.36	1.00
AVIONICS	.88	.97
ECS	.94	.94
GROWTH	8.70	6.48
DRY MASS	130.73	83.95
RESERVES & RESIDUALS	32.45	5.92
LANDING MASS	163.18	89.87
INFLIGHT LOSSES	22.70	18.15
INERT MASS	185.88	108.02

SPS BALLISTIC LAUNCH VEHICLE DDT&E COST

The DDT&E cost for flight hardware and associated ground support equipment for both stages is shown on the opposite chart. The total vehicle DDT&E cost is $3.81 B. The booster DDT&E includes the cost of a new engine development. The costs of both stages reflect the use of some modified space shuttle subsystems.

SPS Ballistic Launch Vehicle (Precursor) DDT&E Cost

SPS-1970



BOOSTER DDT&E = $1,985M

ORBITER DDT&E = $1,828M

TOTAL VEHICLE DDT&E = $3.81B

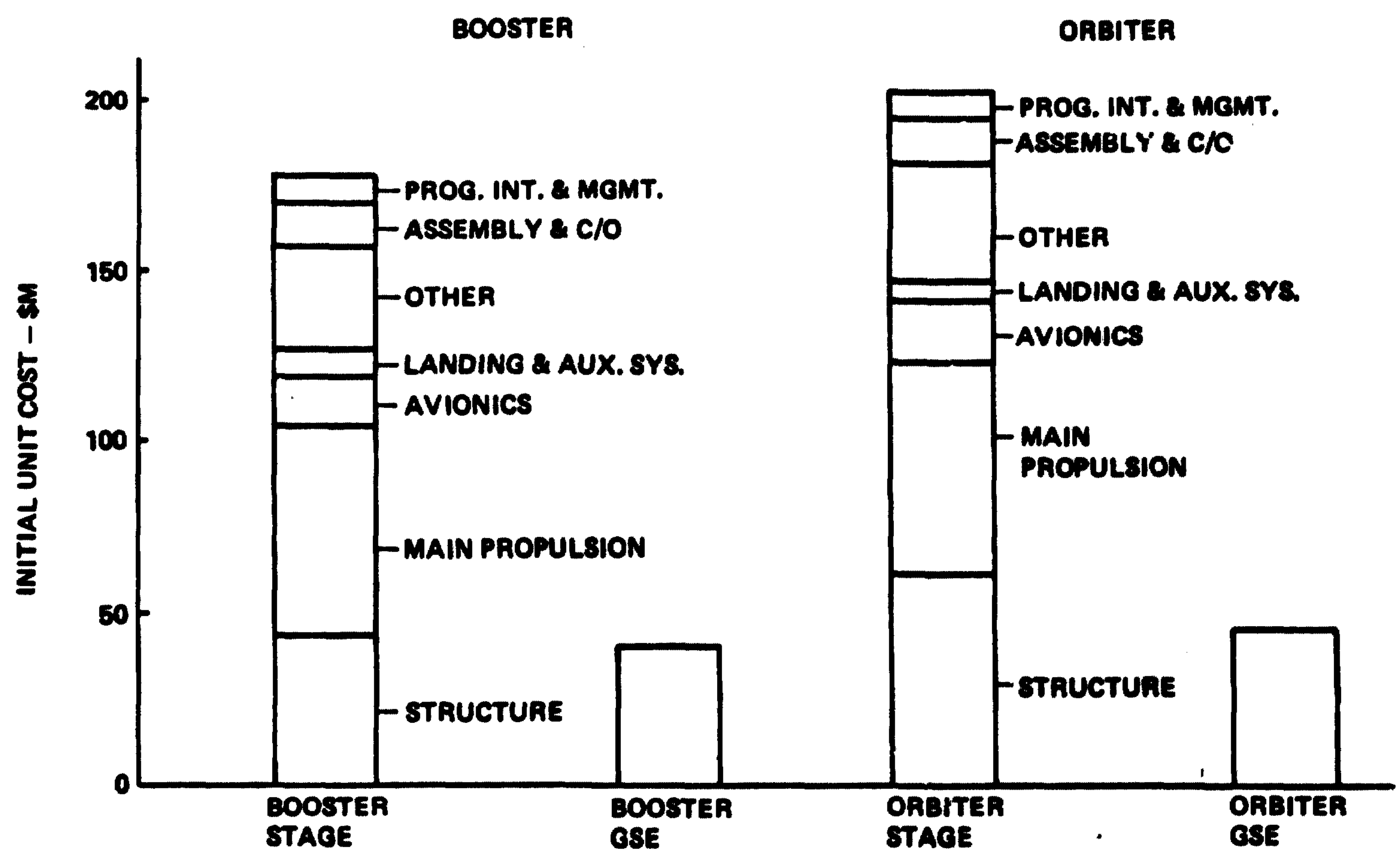
SPS BALLISTIC LAUNCH VEHICLE PRODUCTION COST

The theoretical first unit costs (TFU) of the booster and orbiter are shown on the adjacent chart. These were developed using the Boeing Parametric Cost Model (PCM). The booster and orbiter TFU costs are $176.7M and $202.7M respectively. The breakdowns by subsystem are:

SUBSYSTEM	BOOSTER	ORBITER
Structure	25%	30%
Main Propulsion	34%	30%
Avionics	9%	9%
Landing & Aux. Systems	4%	2%
Other	28%	29%

The estimated ground support equipment TFU costs are $41.6M and $44.8M for the booster and orbiter respectively.

SPS Ballistic Launch Vehicle (Precursor) Production Cost

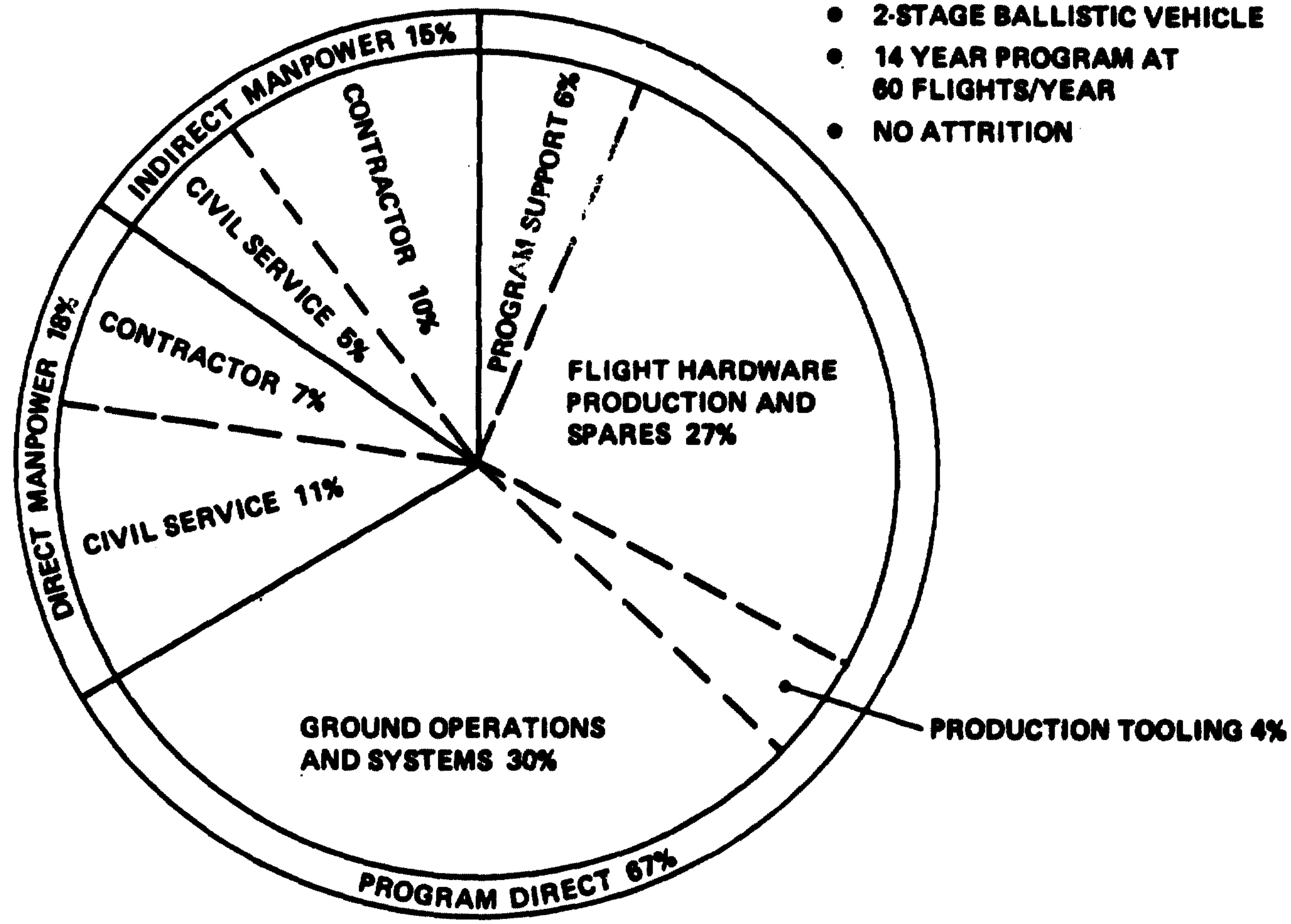


SPS BALLISTIC LAUNCH VEHICLE
AVERAGE COST/FLIGHT

The cost/flight breakdown shown on the opposite chart is the average for a 14 year program at 60 flights per year. The major single elements are the flight hardware and ground operations costs accounting for 27% and 30% respectively.

Average Cost/Flight
(Precursor 2-Stage Ballistic Recoverable)

- 2-STAGE BALLISTIC VEHICLE
- 14 YEAR PROGRAM AT 60 FLIGHTS/YEAR
- NO ATTRITION



- AVERAGE COST/FLIGHT = $9.986M

LEO TRANSPORTATION SUMMARY

The highlights of the results from the LEO transportation activity are noted on the adjacent chart. The revised SPS two-stage winged vehicle concept has incorporated a number of desirable features such as booster flyback propulsion to enhance its operational characteristics. The economics of the winged vehicle appear attractive, and the additional benefit of a manned orbiter capability provides a single vehicle for both the cargo and crew transportation requirements.

The precursor 2-stage ballistic vehicle illustrates the results of one potential vehicle evolution path that begins with the SPACE SHUTTLE and progresses to a fully reusable vehicle in the 90,000 kg payload range. A key element in the development for any of these proposed 2-stage vehicles is the booster engine. Traditionally, a new engine development requires about eight years of development time compared to about five years for the airframe. As a result, a new booster engine will be the long lead time development item.

LEO Transportation Summary

- 2-STAGE WINGED VEHICLE OFFERS:
  - *AN AVERAGE COST/FLIGHT OF BETWEEN $13.45M AND $10.75M* DEPENDENT ON SATELLITE INSTALLATION RATE (4/YEAR VS. 1/YEAR)
  - A 75 kg/m^3 PAYLOAD DENSITY
  - MINIMUM TURNAROUND TIME
  - MANNED CAPABILITY FOR CREW ROTATION/RESUPPLY
- PRECURSOR 2-STAGE BALLISTIC VEHICLE OFFERS
  - AN AVERAGE COST/FLIGHT FOR CARGO DELIVERY OF $10M FOR 60 FLIGHTS PER YEAR
  - A MODEST DDT&E INVESTMENT OF $3.8B
  - A BOOSTER STAGE FOR A SHUTTLE GROWTH CONCEPT (IMPROVED PERFORMANCE AND LOWER COST/FLIGHT)
- A NUMBER OF OPTIONS EXIST FOR LAUNCH VEHICLE EVOLUTION
  - LARGE THRUST BOOSTER ENGINE DEVELOPMENT PERIOD OF 6-8 YEARS WILL BE THE LONG LEAD ITEM

MAINTENANCE AND OPERATIONS ANALYSIS

The topics to be covered in the maintenance and operations analysis are indicated. Initially, those items resulting in the greatest satellite power output loss will be identified. These items will be analyzed to determine the most desirable level of replacement at the satellite. The actual method of making the replacement will then be analyzed including the design impact on the antenna. Several aspects of the maintenance schedule will then be considered. The selected method will then be utilized in defining the maintenance mission characteristics. The selected approach in each of the above areas will then be incorporated into an overall maintenance system description followed by a summary of the maintenance operations identifying plant factor and annual maintenance cost per satellite.

Maintenance and Operations Analysis

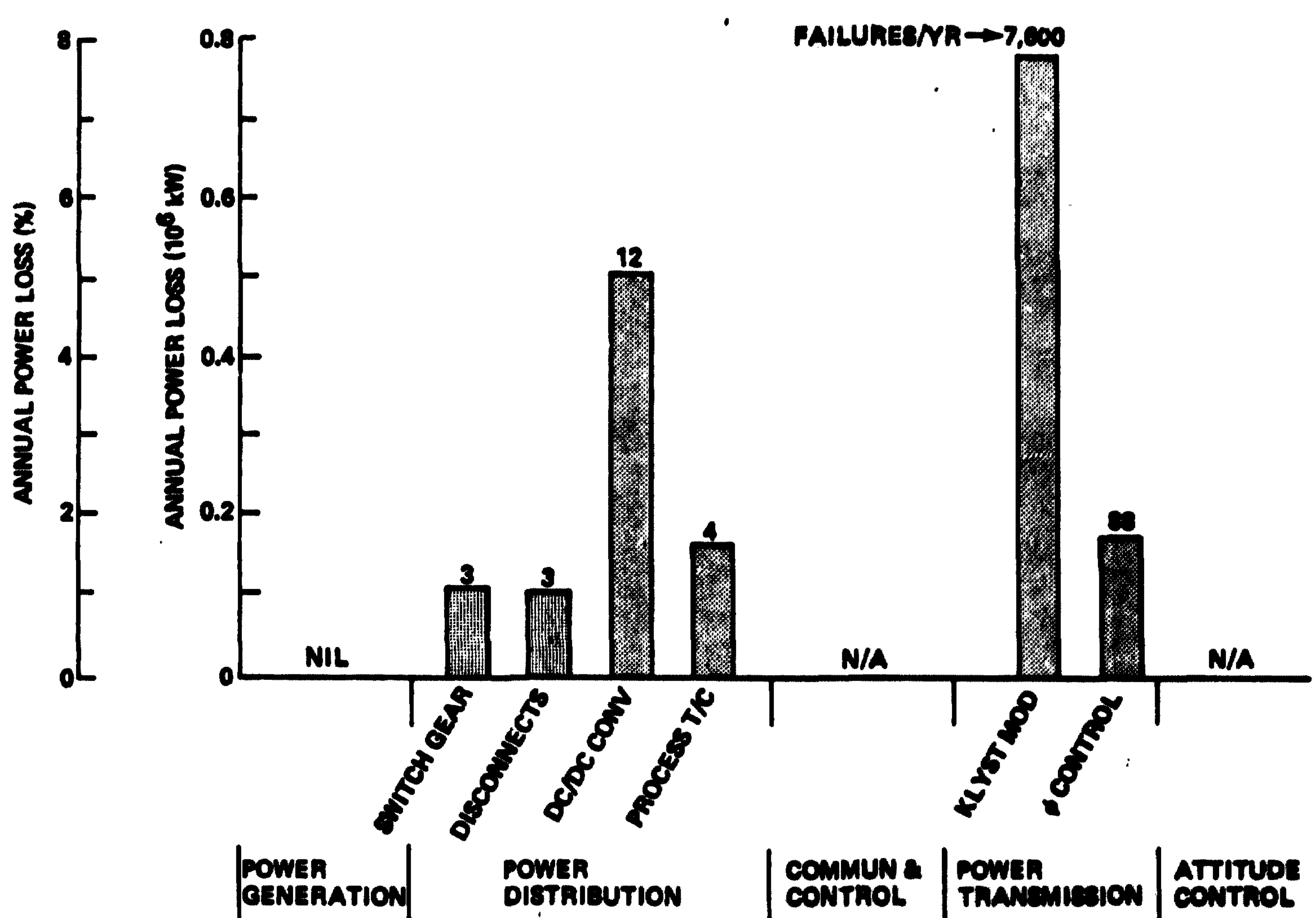
- ITEMS REQUIRING MAINTENANCE
- LEVEL OF REPLACEMENT
- REPLACEMENT CONCEPT
  - EQUIP REQ'D
  - ANTENNA DESIGN IMPACT
- MAINTENANCE SCHEDULE
  - HOW OFTEN
  - HOW RAPIDLY
  - WHEN
- MAINTENANCE MISSION
  - HABITAT LOCATION
  - REFURB LOCATION
  - TRANSPORTATION
- REFERENCE SYSTEM DESCRIPTION
- SATELLITE MAINTENANCE OPERATIONS SUMMARY

ITEMS REQUIRING MAINTENANCE

A number of major components of the satellite have been analyzed for their nature of failures, mean time between failure, power loss per failure, and finally the power loss per year. The results of this analysis in terms of the power loss is presented.

As indicated, the component having the greatest impact in terms of power loss and in the time required to fix the failures is the klystron tube modules. DC/DC converters present a significant power loss although the number of failures is quite low and, consequently, require less repair time. The remainder of this analysis therefore will focus on the repair of the klystron tube modules.

Items Requiring Maintenance

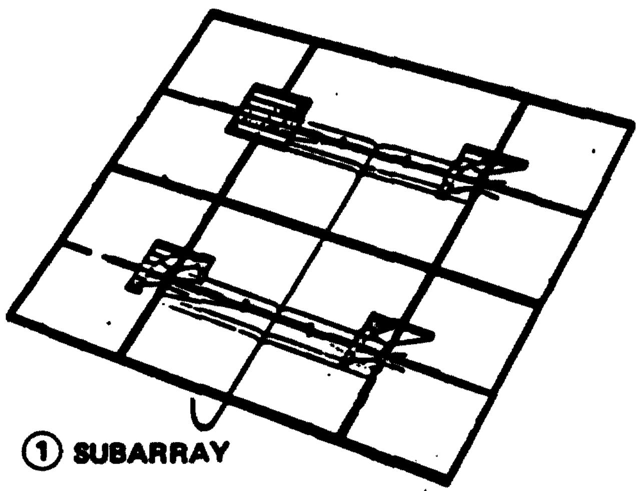


LEVEL OF REPLACEMENT OPTIONS AT SATELLITE

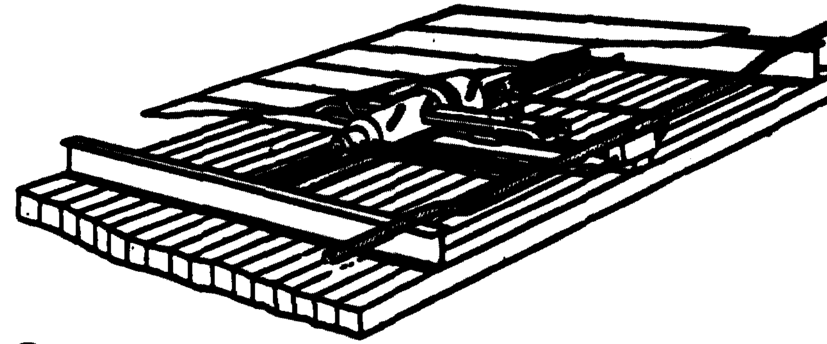
Several options exist for the replacement of a klystron tube module. The first of these options involves replacement of a complete subarray which may contain as many as 36 klystron tubes. The next level of replacement considered is that of a complete klystron tube module including the wave guide section associated with the module. Potential difficulty in segmenting the wave guide resulted in considering the third option which is the removal of the tube module and its thermal control system. The final option deals with removal of individual components which required the radiator to be sectionalized and have gimbaled panels. These options are compared in the following charts.

Level of Replacement Options at Satellite

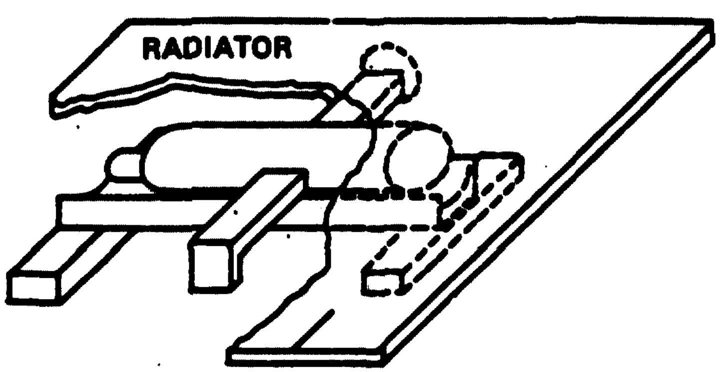
SPS-1911



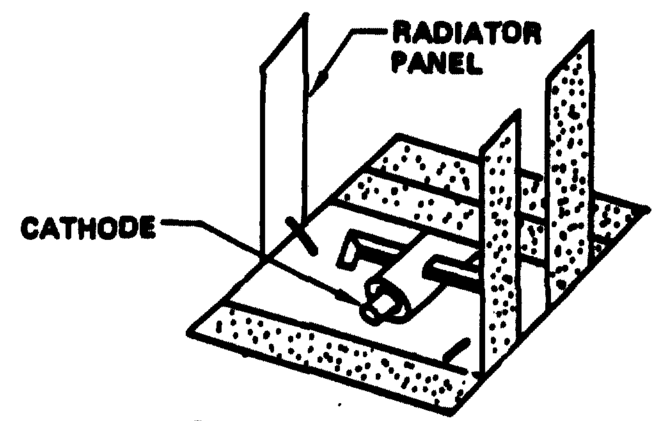
① SUBARRAY



② COMPLETE KLYSTRON MODULE INCLUDING WAVEGUIDE



③ KLYSTRON TUBE PLUS THERMAL CONTROL



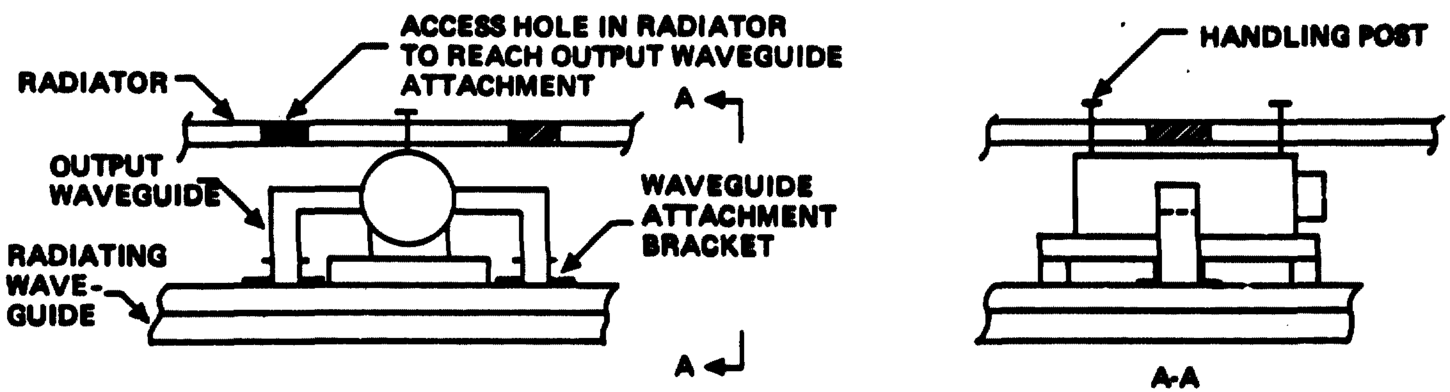
④ COMPONENTS

LEVEL OF REPLACEMENT SELECTION

The level of replacement selected is that of the klystron tube module plus its thermal control system. The rationale for selection of this option is indicated. Actual removal of the tube module involves access through holes in the radiator to reach the distribution wave guide attachment bracket which secures the module to the distribution wave guide. Once this attachment is released the module is free to be removed.

Level of Replacement Selection

- SELECTION: TUBE PLUS RADIATOR



- RATIONALE
  - SUBARRAY – REQUIRES REMOVAL OF 30% (2400) OF UNITS FOR 4% (7600) OF TUBES
    – MAINTENANCE TIME AND MASS PENALTY
  - COMPLETE MODULE – MASS PENALTY AND INCREASED ALIGNMENT DIFFICULTY
  - COMPONENTS – RADIATOR DESIGN COMPLEXITY, TEST VERIFICATION PROBLEM

TUBE MODULE REPLACEMENT OPTIONS

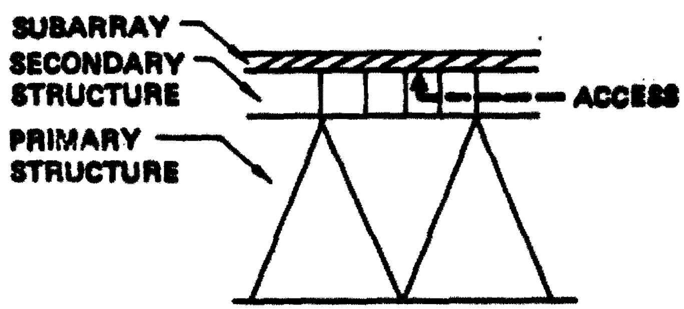
Using the klystron tube module as the level of replacement, it is now possible to analyze various options relative to the method of replacing the module. In all cases, it is assumed the satellite is shut down while maintenance is performed.

Two options involve servicing the antenna from the back side (non-radiating). One of these has the repair vehicle moving horizontally through the secondary structure until the failed tube is reached. Another option has the repair vehicle moving horizontally through the primary structure and then vertically through the secondary structure to reach the failed tube. The last option has the servicing done from the front side through use of moveable overhead platform.

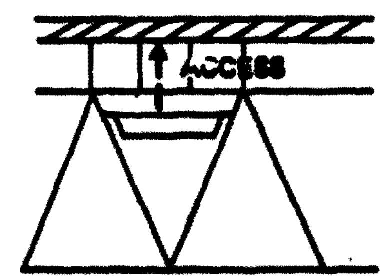
Each of these options are discussed in more detail in subsequent charts.

Tube Module Replacement

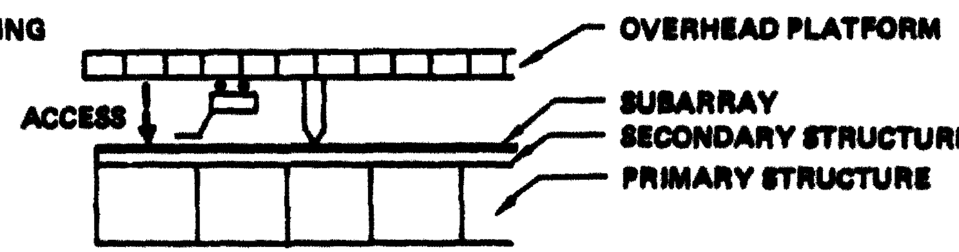
- OPTIONS
  - BACKSIDE SERVICING
    - HORIZONTAL ACCESS THROUGH SECONDARY STRUCTURE



    - VERTICAL ACCESS THROUGH SECONDARY STRUCTURE



  - FRONTSIDE SERVICING



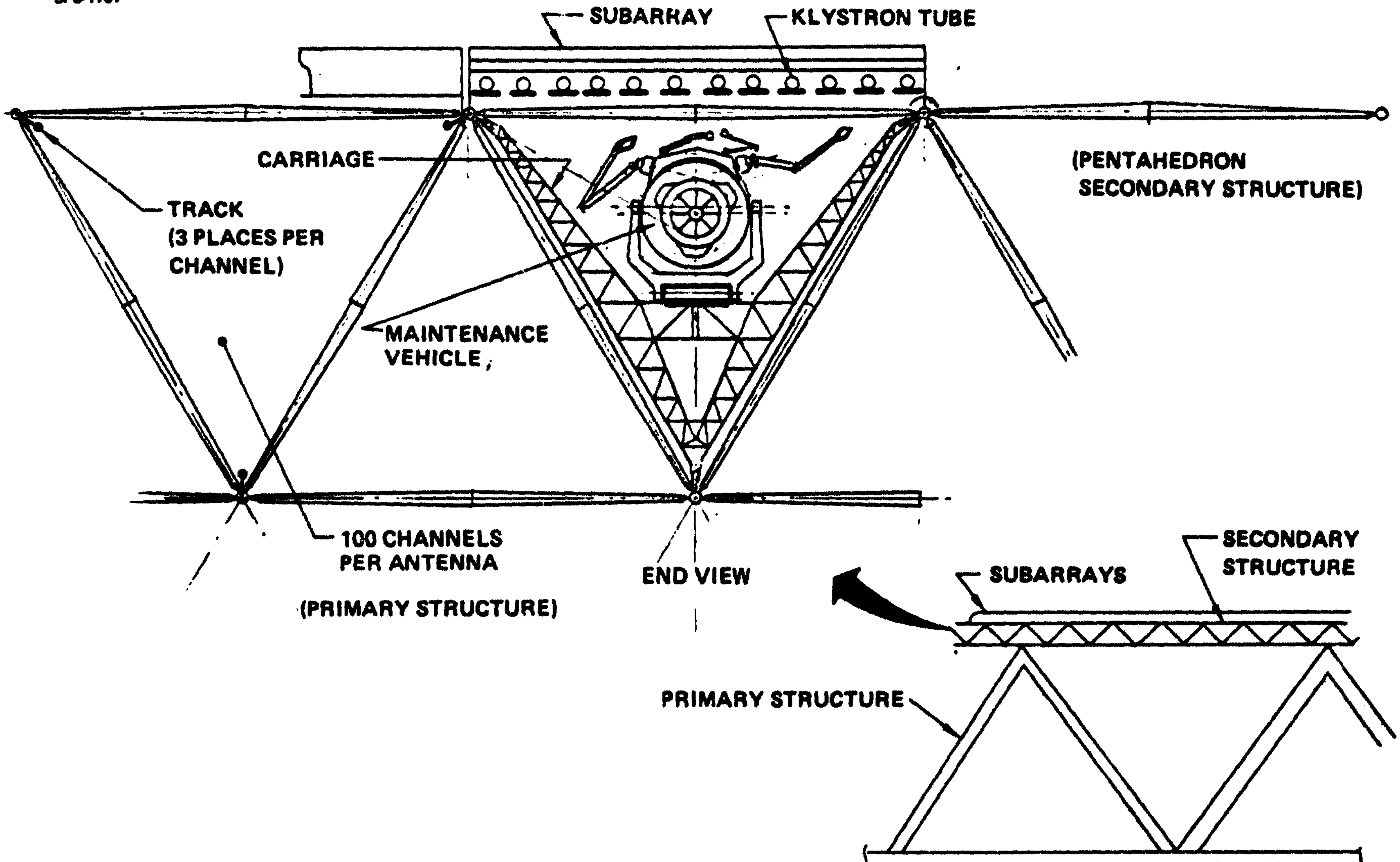
- KEY CONSIDERATIONS
  - ACCESS
  - EQUIPMENT REQUIRED
  - ANTENNA DESIGN IMPACT

HORIZONTAL ACCESS FOR TUBE MAINTENANCE

The next three charts deal with reaching the failed klystron tube module by horizontal travel through the secondary structure. In this concept a two-man maintenance vehicle is mounted to a carriage that in turn is supported from three tracks within a channel formed by the secondary structure. Across the antenna a total of 100 channels exist, each requiring a built-in track system. Early estimates indicate as many as 10 maintenance vehicles to be required within the 100 channels.

Horizontal Access For Tube Maintenance

SPS-1787

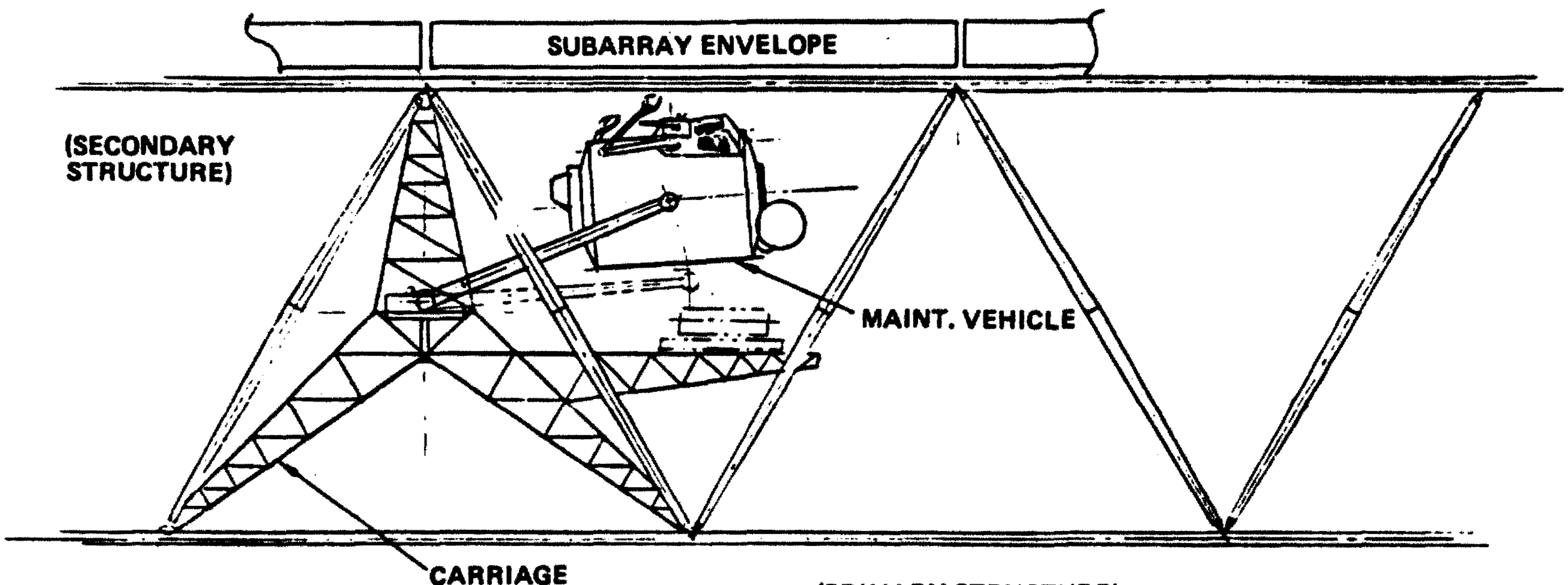


HORIZONTAL ACCESS FOR TUBE MAINTENANCE – SIDE VIEW

A side view of the maintenance vehicle in the horizontal access concept is shown. A gimbal system between the carriage and the maintenance vehicle allows positioning of the maintenance vehicle relative to the tube above. The platform also provides an area for storing klystron tube modules.

Horizontal Access For Tube Maintenance

SPS-1788

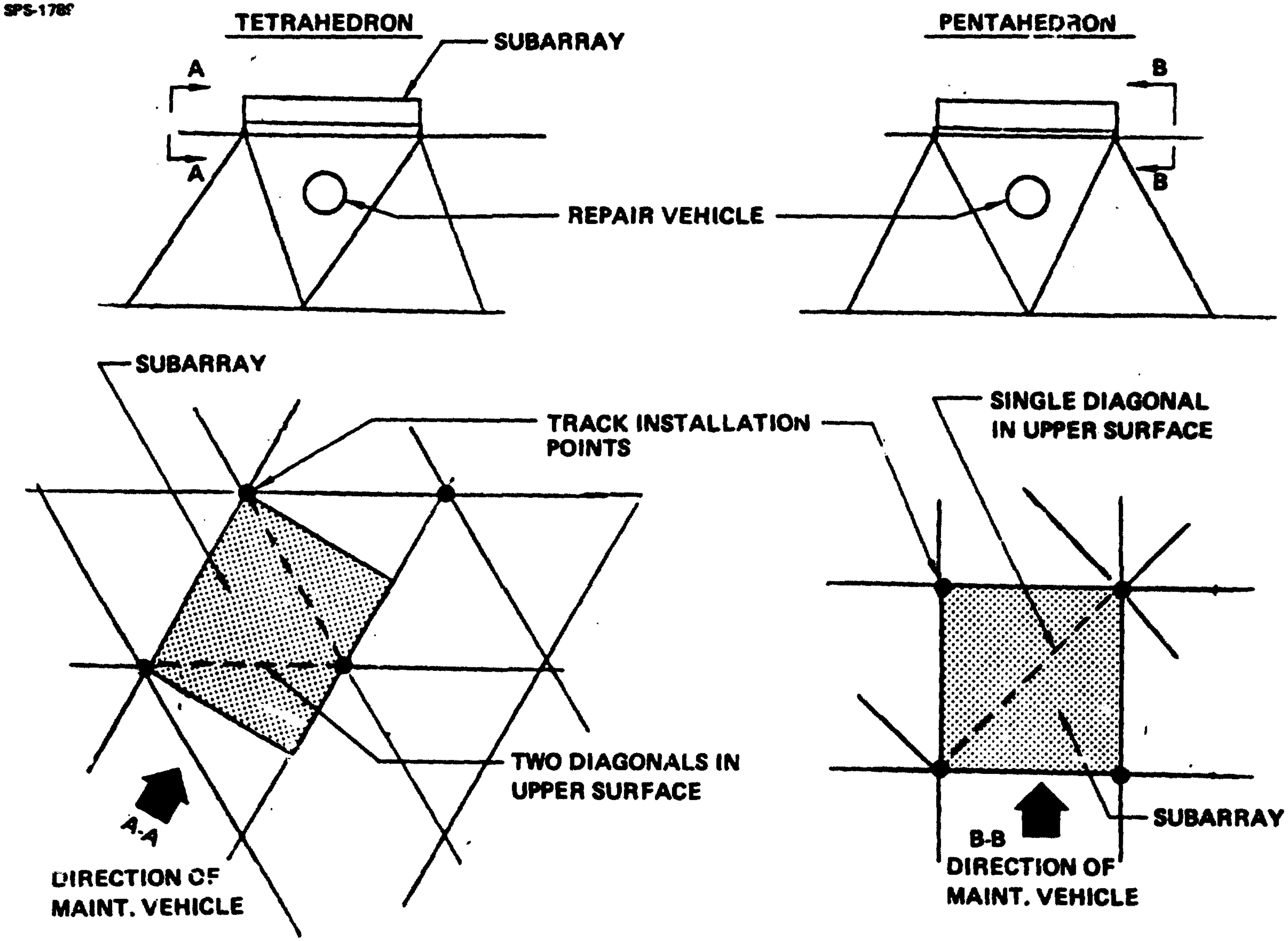


SIDE VIEW

HORIZONTAL ACCESS – SECONDARY STRUCTURE OPTIONS

Several secondary structure options were analyzed for the horizontal access option. The tetrahedron structure was used througout part 2. Part 3 of the study considered a pentahedron type structure. Both structures are deployable. Comparison of the structures show that the pentahedron only has one diagonal between the maintenance vehicle and the klystron tube. The tetrahedron structure would have 2 diagonals, potentially resulting in a longer repair time due to diagonal removal and replacement. Other factors favoring the pentahedron structure was that a little more space is available for the maintenance equipment within the channel and the track installation would be somewhat easier.

Horizontal Access Secondary Structure Options



VERTICAL ACCESS FOR TUBE MAINTENANCE

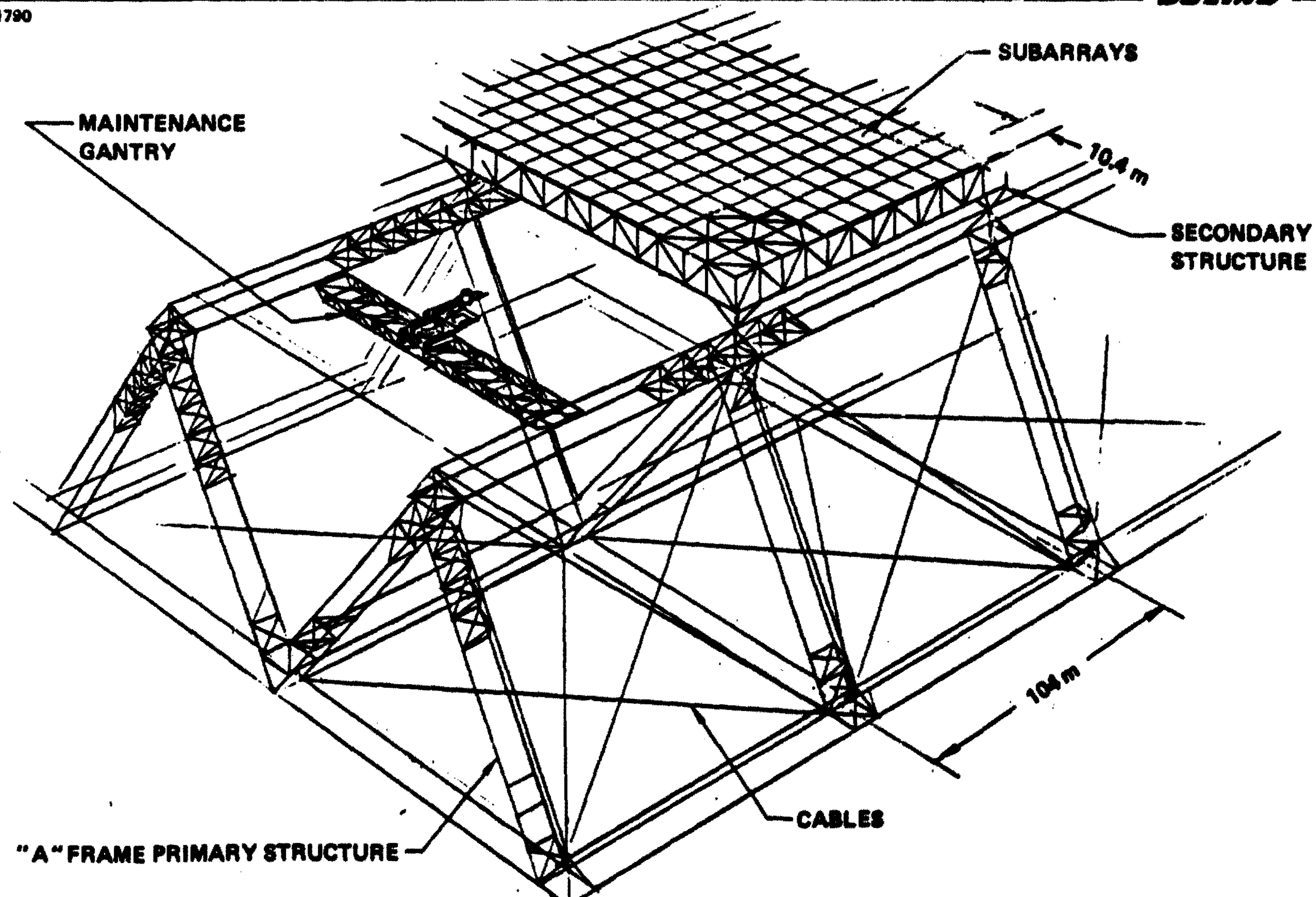
For the vertical access approach, the antenna has a total of 10 channels in which maintenance gantries can be mounted. Attached to each of the gantries is the maintenance vehicle which reaches up through the secondary structure to reach the failed klystron tubes. In this option, a cubic (hexahedron) structure has been found to offer the greatest clearance for the maintenance vehicle.

SPS

Vertical Access For Tube Maintenance

BOEING

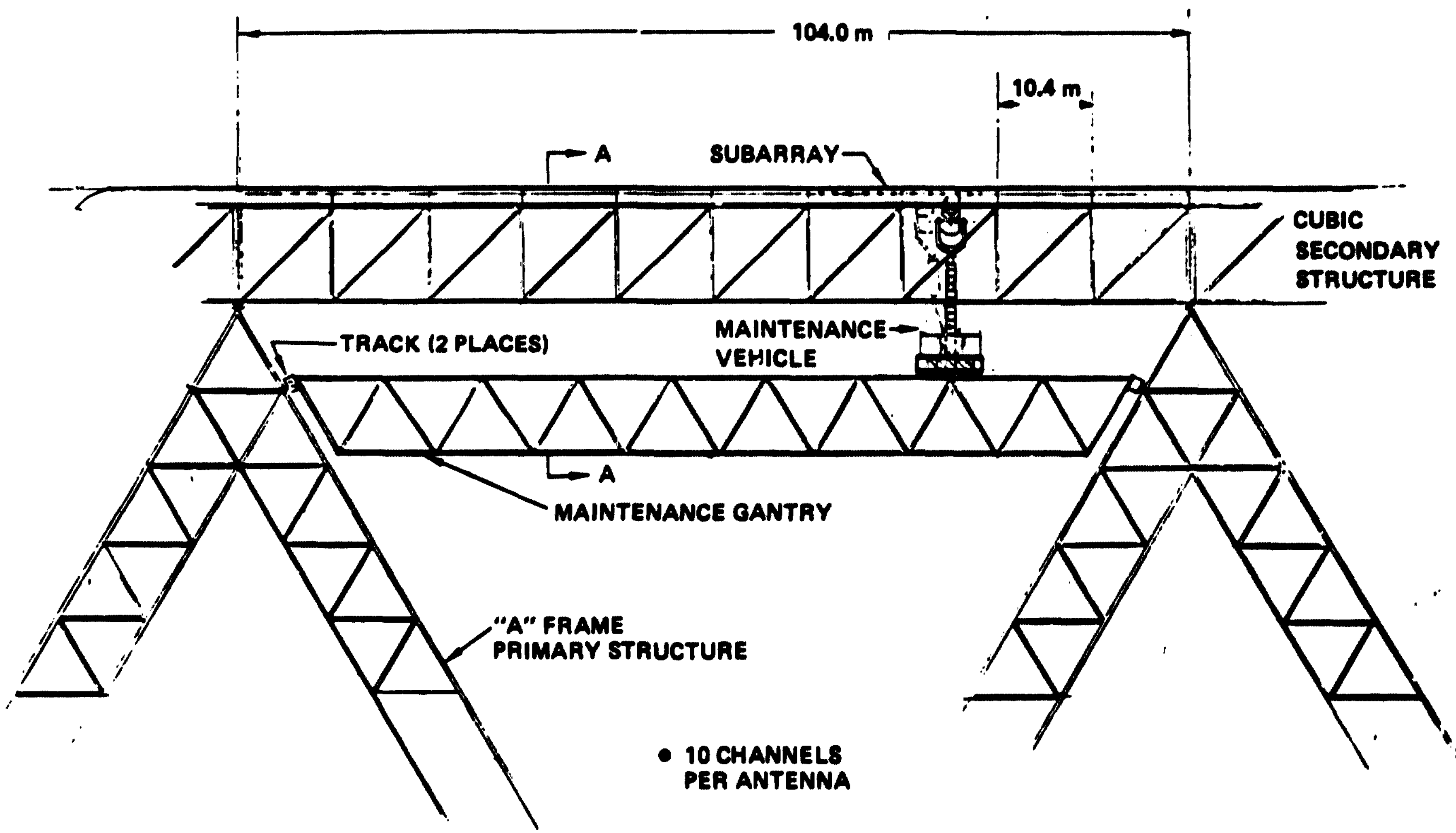
SPS-1790



VERTICAL ACCESS FOR TUBE MAINTENANCE

The next three charts deal with the back side servicing concept involving vertical access for klystron tube maintenance. The overall concept is illustrated in this chart. The primary structure is a frame design forming ridges that allows free unobstructed movement of the maintenance gantry moving horizontally across the antenna. Stability of the primary structure in one direction is provided by the cubic secondary structure above the primary structure while stability in the opposite direction is provided through guy-wire cabling.

Vertical Access For Tube Maintenance



- 10 CHANNELS PER ANTENNA

VERTICAL ACCESS MAINTENANCE VEHICLE

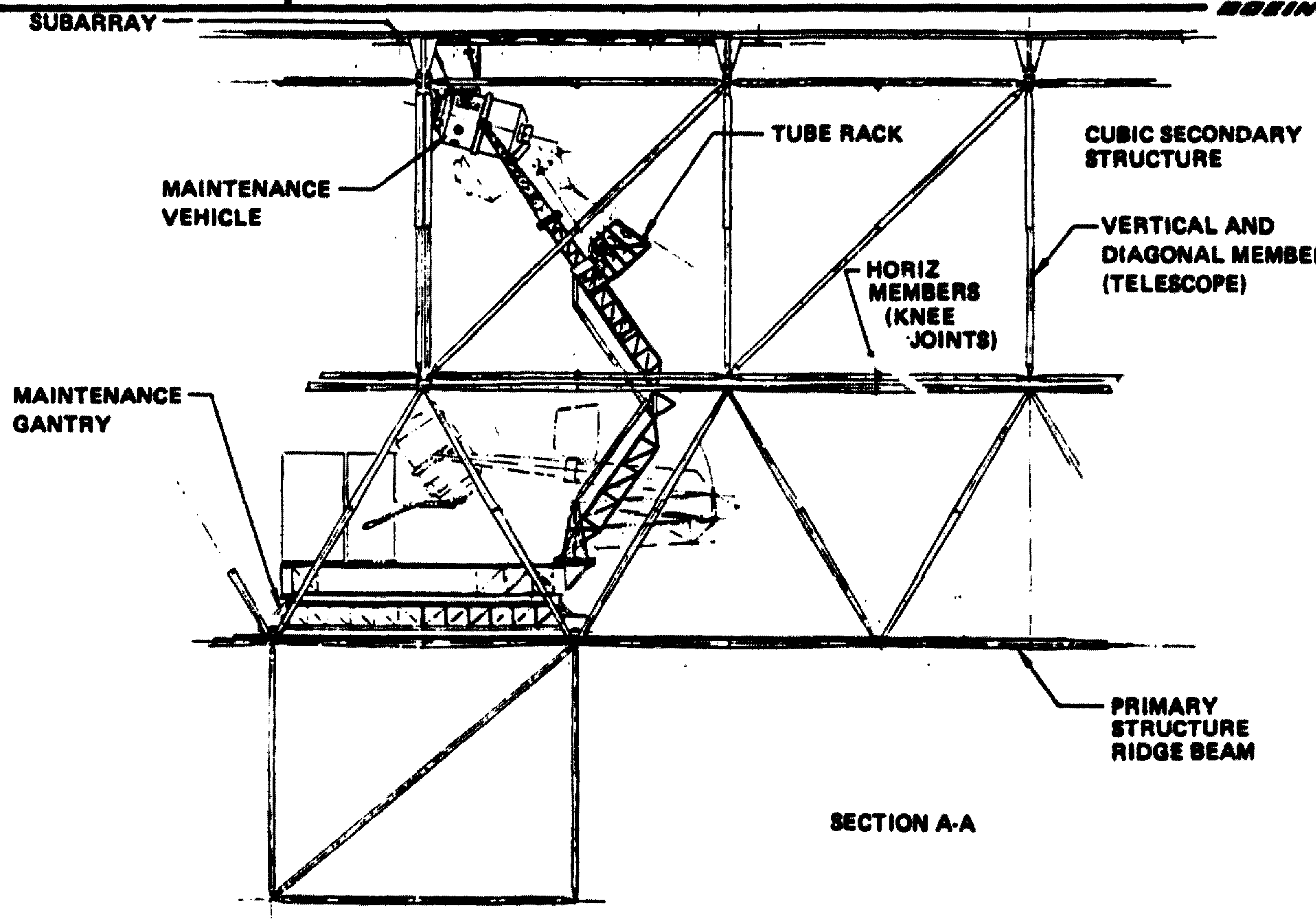
Additional detail of the cubic secondary structure and the maintenance vehicle is presented with the maintenance gantry shown moving along in the direction of the channel. The gantry itself will be designed to transport all of the spare klystron tubes necessary for a given shift. The maintenance vehicle consists of a hinged boom and a 2-man crew cabin with manipulators. A small klystron rack is also attached to the boom to eliminate the need for the manipulators to reach back down to the gantry for a new tube. In the case of a 36 tube subarray, as many as 3 tubes may require replacement.

The cubic secondary structure is deployable and to satisfy packaging constraints has its vertical and diagonal members telescoping approximately 25%, while horizontal members have knee joints. A packaging density comparable to the part 2 tetrahedron structure is anticipated and is estimated at 70 kilograms a cubic meter.

Using this concept, a tube replacement time of 45 min. is expected, which includes removal and replacement of two diagonals (one in lower and upper surface of secondary structure), removal and replacement of one klystron tube module, and movement to the next failed klystron tube estimated at a distance of 2 subarrays away or 20 meters.

Vertical Access Maintenance Vehicle

SPS-1792



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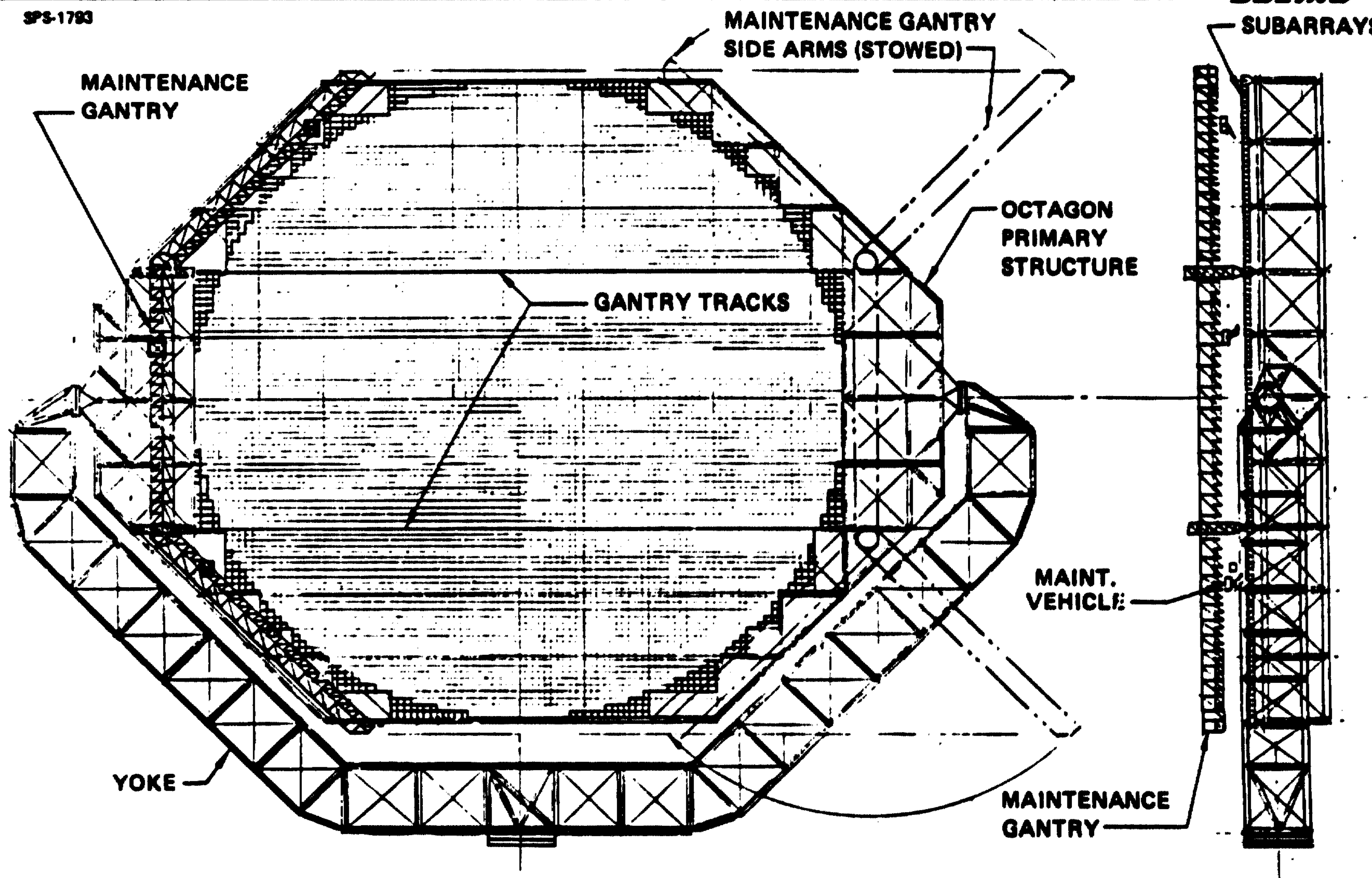
ANTENNA FRONT SIDE MAINTENANCE CONCEPT

The next two charts deal with the front side servicing concept. This concept consists of an overhead maintenance gantry that is supported from two tracks running across the antenna. Supported from the platforms are the actual repair vehicles. The antenna design for this option has been altered to form an octagon shape. The actual primary structure for the antenna is oversized to allow the maintenance gantry to be moved out of the way when the antenna is in the operating condition.

Antenna Frontside Maintenance Concept

SPS-1793

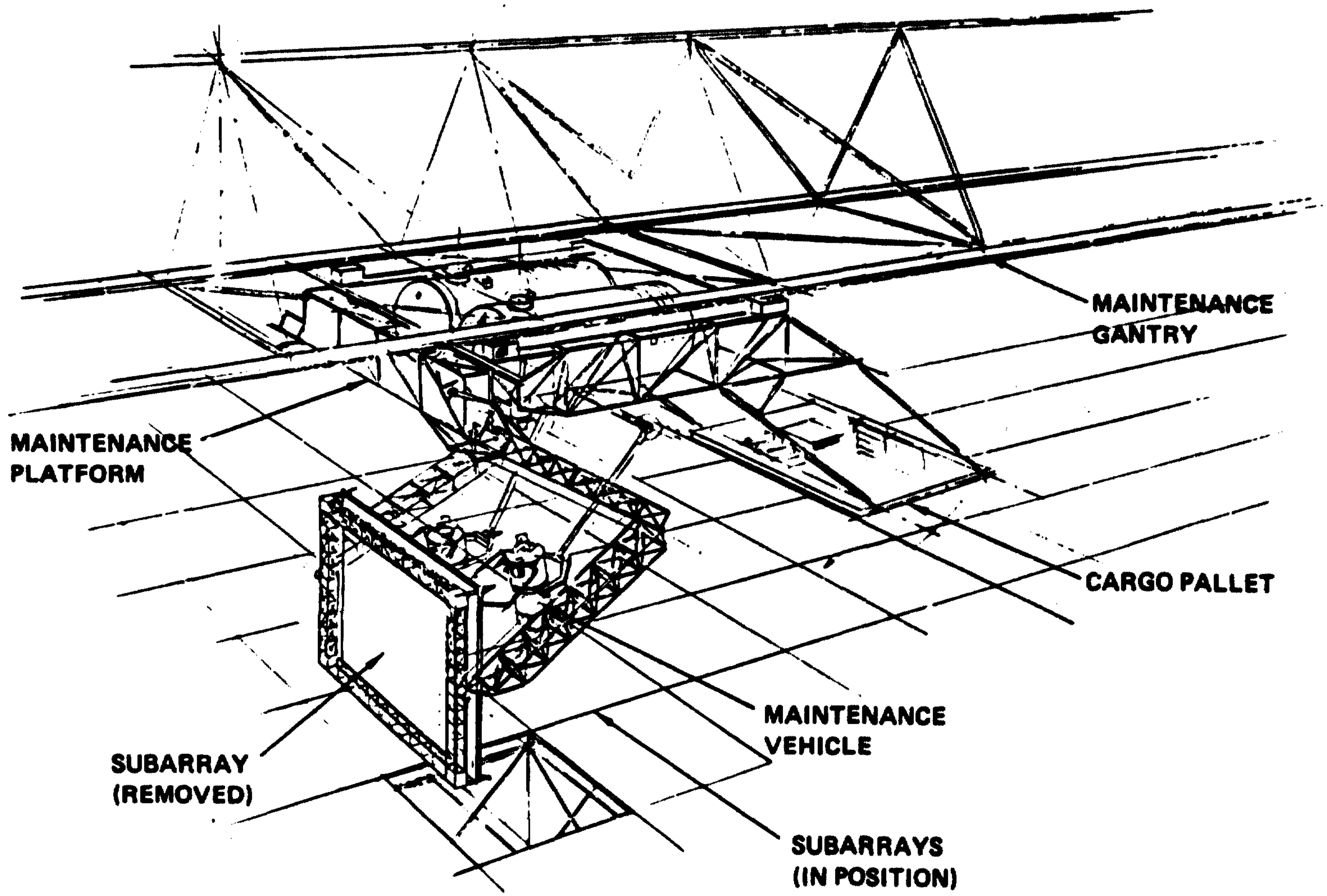


MAINTENANCE VEHICLE CONCEPT–FRONT SIDE ACCESS

The actual replacement of the klystron tubes in this option requires the subarrays to be removed as shown. Once removed, the maintenance vehicles can then remove the failed klystron tubes. When all failed tubes of a given subarray are repaired the subarray is replaced.

Maintenance Vehicle Concept
Frontside Access

SPS-1786

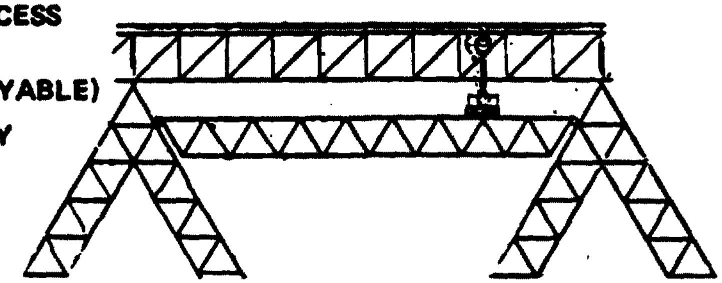


REPLACEMENT CONCEPT SELECTION

The selected klystron tube module replacement concept uses vertical access through the cubic secondary structure which is attached to the "A"-frame primary structure. The advantages of this option over the other options are indicated. The chief advantage over the horizontal access option was that only 10 track systems were required while less repair time and less mass were the chief advantages over the front side repair concept.

Replacement Concept Selection

- SELECTION: VERTICAL ACCESS
  - CUBIC SECONDARY
  - STRUCTURE (DEPLOYABLE)
  - "A" FRAME PRIMARY STRUCT



- RATIONALE: ADVANTAGES OVER
  - HORIZONTAL ACCESS
    - 10 TRACK SYSTEMS VERSUS 100
    - EASE IN INSTALLING TRACK SYSTEM
    - ABILITY TO POSITION REPAIR VEHICLE DIRECTLY UNDER ALL TUBES
    - EASIER REMOVAL OF COMPLETE SUBARRAY IF NECESSARY
  - FRONTSIDE ACCESS
    - LESS DOWNTIME (LOST REVENUE)
    - REPAIR VEHICLES ARE DECOUPLED
    - SMALLER ANTENNA PRIMARY STRUCTURE AND YOKE
    - NO ANTENNA PERFORMANCE PENALTY

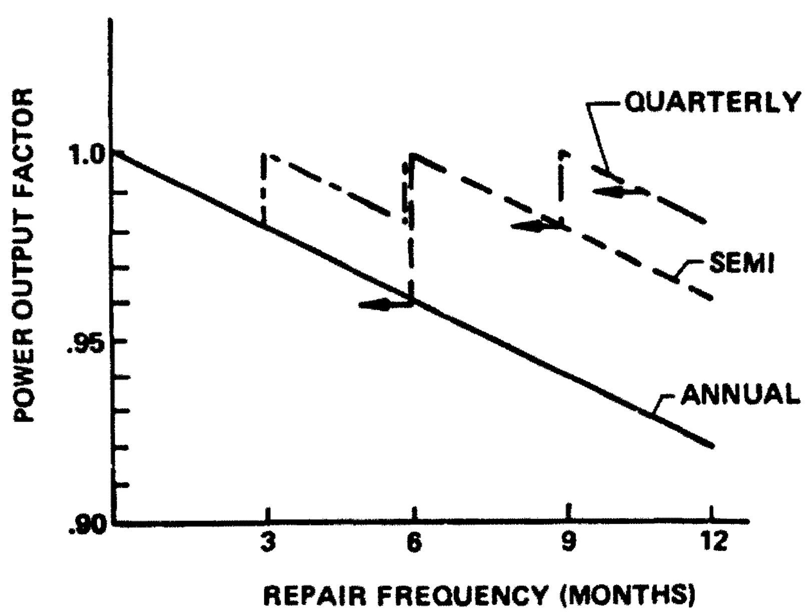
MAINTENANCE SCHEDULE–HOW OFTEN TO REPAIR

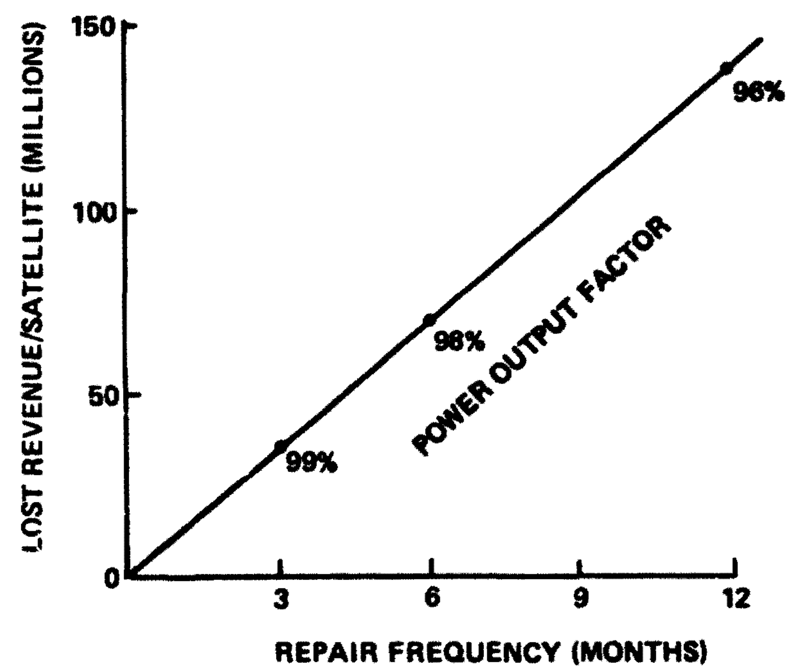
The maintenance schedule employed for servicing the antenna deals with (1) how often to repair, (2) how rapidly to repair once the antenna is reached, and (3) when (time of day and year) the antenna should be repaired.

In this chart the effect of how often illustrates the average power output factor and the associated lost revenue. As the antenna is repaired more frequently, the average power output is higher and accordingly the lost revenue is less. It should be noted, however, that for frequencies more often than 6 months the benefit gets proportionately less. Further maintenance analysis will utilize the 6 month repair frequency concept.

Maintenance Schedule
How Often To Repair

- KEY PARAMETER: POWER OUTPUT FACTOR
- 1% TUBE FAILURE GIVES 2% POWER LOSS
- 4% TUBE FAILURE PER YEAR
- ANNUAL REVENUE ~ $3.5 BILLION @ 100% OUTPUT FACTOR





MAINTENANCE SCHEDULE—HOW RAPIDLY TO REPAIR

Once the satellite is reached, a key issue then becomes how quickly the repair should be performed. The key factor in this issue then is how many repair crews and associated equipment are provided. The factors to be considered in this assessment are indicated and are discussed in tne subsequent charts.

Maintenance Schedule
How Rapidly To Repair

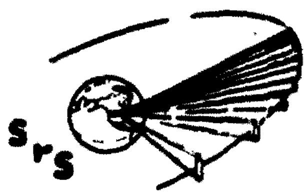
KEY PARAMETER: CREW SIZE

FACTORS:

- DOWNTIME—LOST REVENUE
- CREW LOGISTICS COST
- MAINTENANCE PROVISIONS COST
  - HABITATS
  - REPAIR EQUIP
- PRACTICALITY

DOWN TIME COST

Down time cost deals with the actual time the satellite is shut down and not transmitting power. Using the semi-annual maintenance approach a total 3800 tubes per satellite involving 2850 hours of work are involved. Characteristics for each repair crew are indicated and result in a down time of 14 days per semi-annual repair when using one repair crew per satellite. Larger numbers of crews reduce the down time and the resulting down time cost.

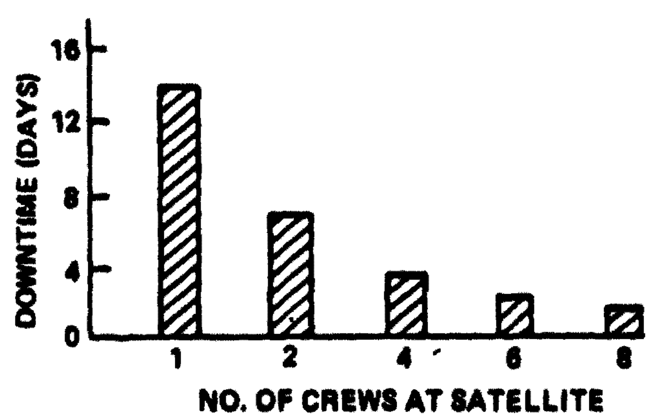


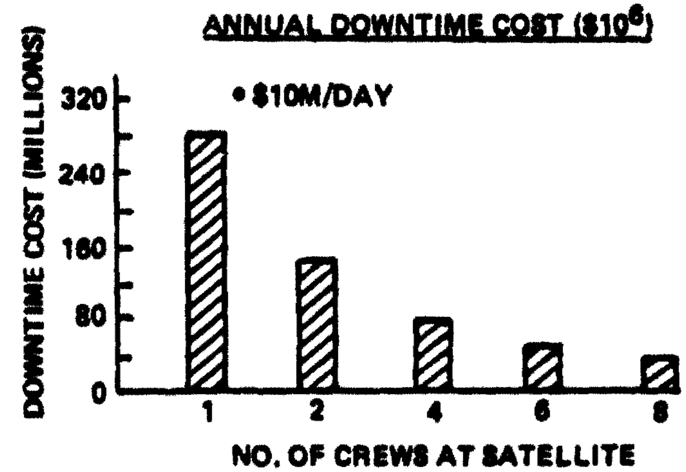
SPS-1906

Downtime Cost

- SINGLE TUBE REPLACEMENT TIME: 45 MIN

  TOTAL TUBE REPLACEMENT TIME: 2850 HRS
  - SEMI-ANNUAL REPAIR
  - 3800 TUBES
- REPAIR SYSTEMS AND OPERATIONS PER CREW
  - 10 REPAIR VEHICLES/ANTENNA (FIXED EQUIP)
  - 2 SHIFTS (TOTAL OF 60 PEOPLE)
  - 10 HOURS/SHIFT
- RESULTS PER SATELLITE





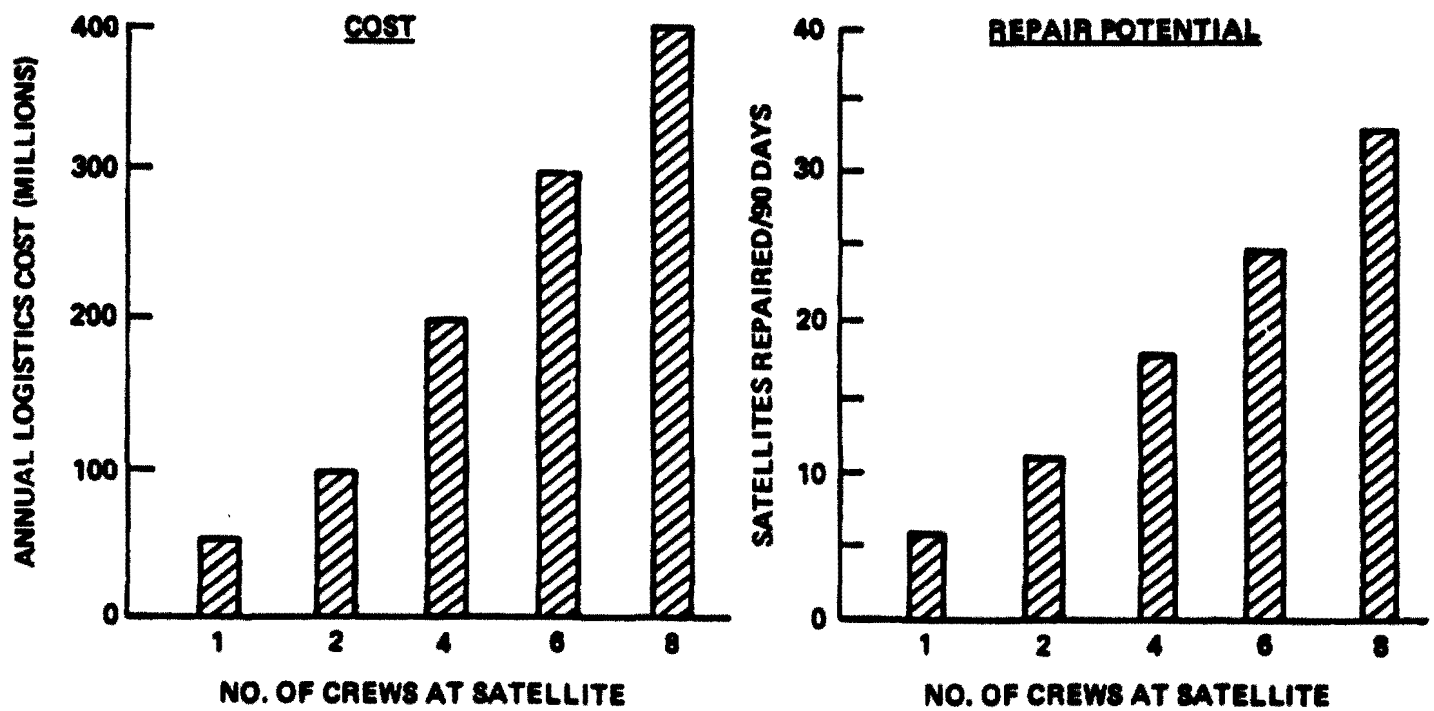
LOGISTICS COST

Logistics cost, meaning the transportation of crew and their supplies, goes up with the number of crews as indicated. It must be noted however that although the higher number of crews means more cost, that complement of people also has a greater potential in terms of how many satellites can be repaired in a given increment of time. Performing an equal amount of work (repair of satellites) therefore is necessary in order to have a legitimate comparison.

Logistics Cost

SPS-1907

- ONE SATELLITE IN ORBIT
- A CREW IS 60 PEOPLE
- 2 REPAIR SORTIES/SATELLITE/YEAR
- SHUTTLE GROWTH TO LEO
- CHEM OTV TO GEO



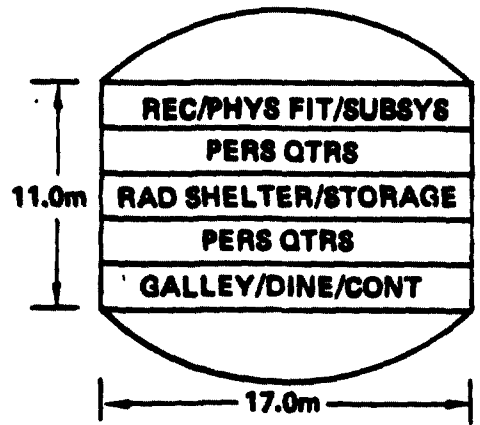
MAINTENANCE PROVISIONS

The third key factor in the issue of repair speed is the amount of maintenance provisions. The two key provisions in this case are the movable crew habitats and the actual repair equipment (maintenance gantry and vehicle) that is fixed to the satellite. Major characteristics of each of these systems are provided including mass and cost estimates. A 15% capital charge factor (same as used in costing the satellite) is used for the write-off.

Maintenance Provisions

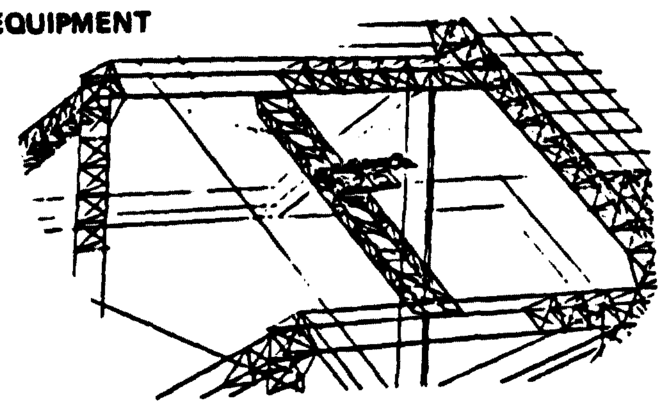
SPS-1909

- CREW HABITAT



- ONE HABITAT PER 60 PEOPLE
- MODIFIED CREW QUARTERS MODULE
- 240,000 kg
- $240 MILLION INVESTMENT
- 15% CAPITAL CHARGE

- REPAIR EQUIPMENT



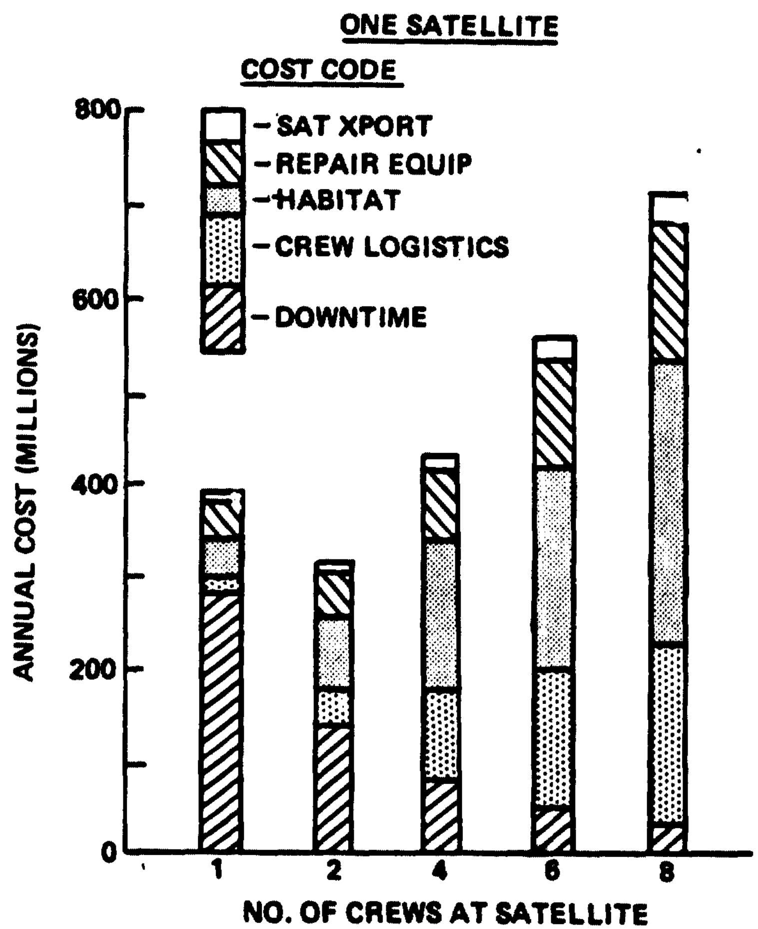
- GANTRY/REPAIR VEHICLE (EA)
  - 5,000 kg
  - $20 MILLION TFU
  - 15% CAPITAL CHARGE
- TRANSPORTATION COST
  EARTH TO GEO SELF POWER = $45/kg

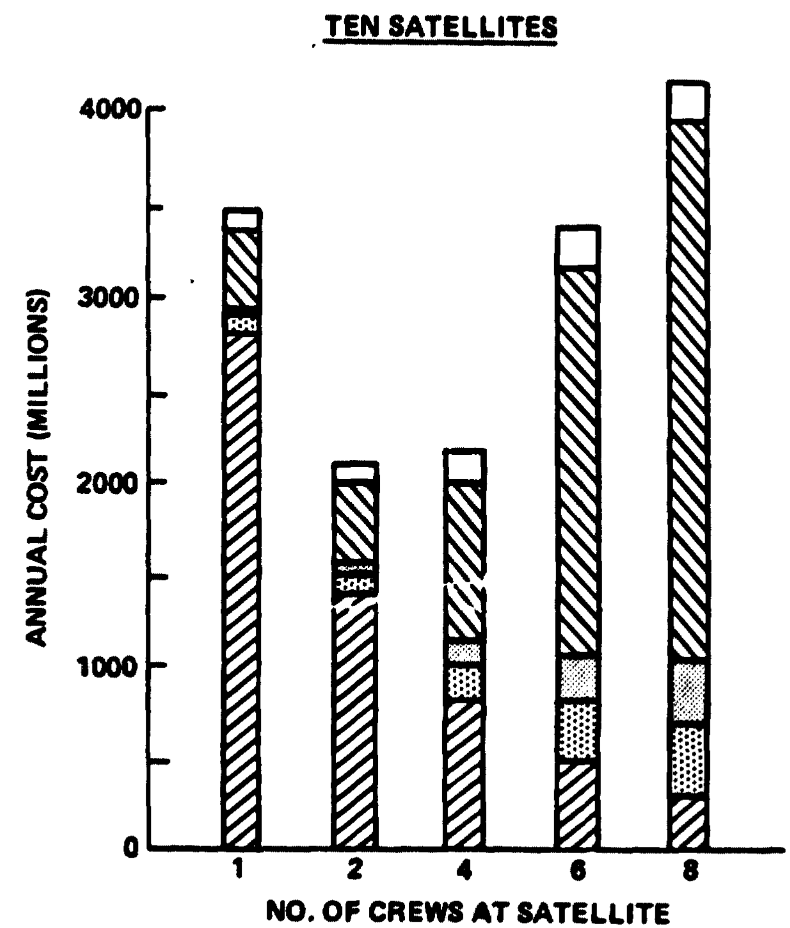
CREW SIZE COST

Combining the factors of down time cost, logistics cost, and maintenance provision cost allows the selection of the crew size in terms of annual cost. For the case of only one satellite in orbit, two crews per satellite gives the least cost. For the 10 satellites in orbit case, two and four crews give about the same annual cost. It may be noted that in the 10 satellite case the down time and the permanently installed repair equipment at the satellite have a much greater impact on total cost. From this trend it would appear that with a greater number of satellites in orbit, a larger number of crews may be more cost optimum. Selection of the number of crews at the satellite, however, requires a final consideration in terms of the practicality discussed on the next chart.

Crew Size Cost

SPS-1908





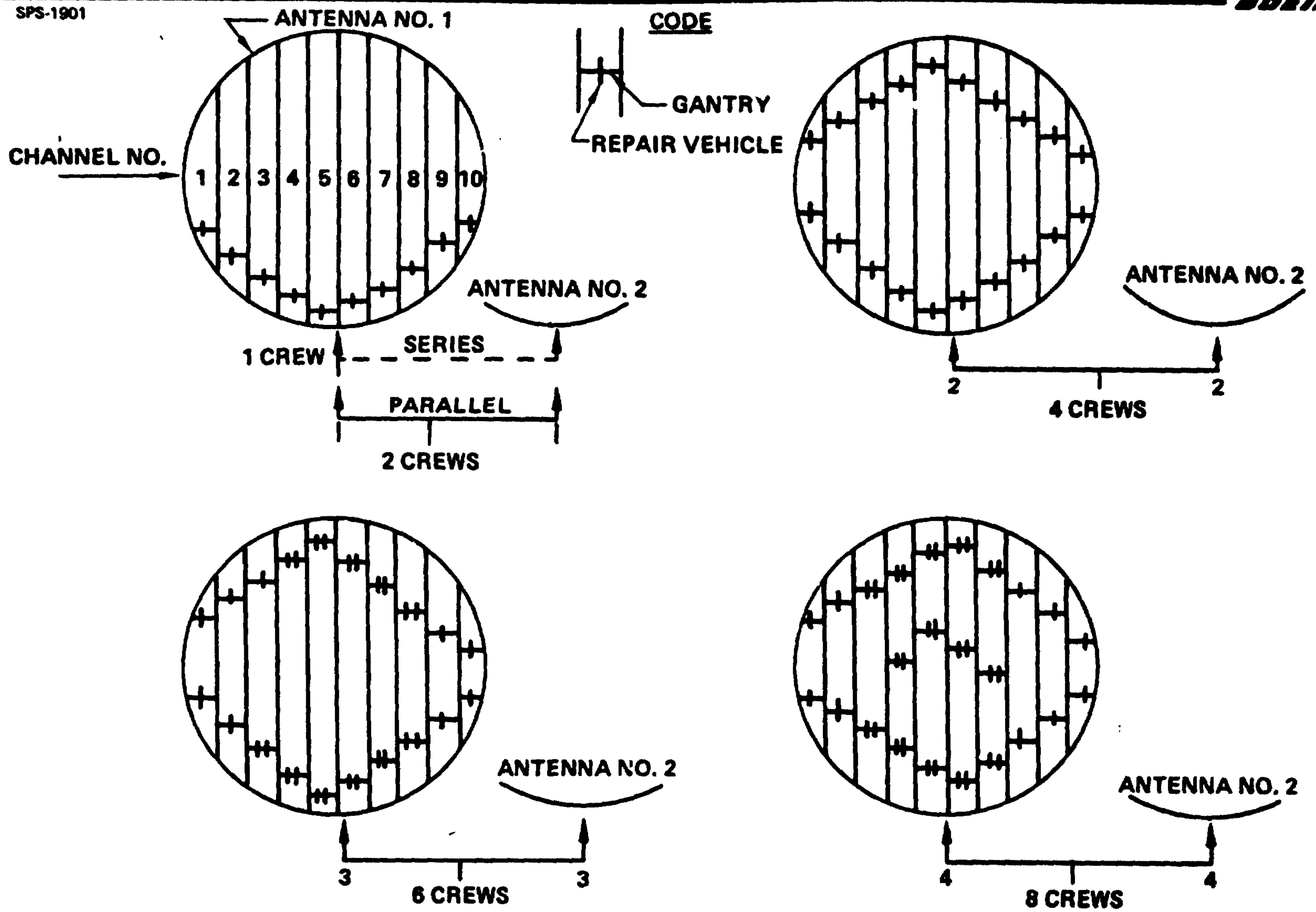
CREW/EQUIPMENT UTILIZATION–PRACTICALITY

This chart attempts to illustrate the relative complexity of actually integrating into an antenna various numbers of repair equipment. It may be remembered that each crew utilizes 10 repair vehicles, 20 hours a day (two shifts). For the case of one crew, 10 repair vehicles are installed in each antenna during the antenna construction at the LEO construction base. Once the crew completes the repair of antenna number 1 they move to antenna number 2 for its repair.

Use of two crews at the satellite allows one crew to work on each antenna in parallel. Consideration of four crews requires additional repair vehicles to be installed as indicated. When 6 or more crews are considered several repair vehicles must be put on each gantry which results in potential dynamics problems and/or multiple gantries installed per channel.

From the standpoint of cost, four crews appear reasonable, and from a crew equipment utilization viewpoint the same number of crews may be the practical limit. Four crews per satellite are consequently selected for the remainder of the maintenance analysis.

Crew/Equipment Utilization Practicality



MAINTENANCE SCHEDULE—WHEN TO REPAIR

A third key issue regarding the maintenance schedule is when the repair is to be performed. Two subissues are involved in this primary issue. The first of these deals with the time of day maintenance is to be performed, while the other concerns itself with the time of year. Both will be discussed in subsequent charts. Key considerations in these issues are similar to past issues with the additional consideration of the impact on the ground power grid.

Maintenance Schedule
When to Repair

- **KEY ISSUES:**
  - TIME OF DAY—DOES IT PAY TO LIMIT REPAIR TO A FEW HOURS AROUND THE DAILY OCCULTATION DURING EQUINOX?
  - TIME OF YEAR—DOES IT PAY TO REPAIR JUST DURING THE EQUINOXES?
- **KEY CONSIDERATIONS:**
  - DOWNTIME COST
  - LOGISTICS COST
  - GROUND POWER GRID IMPACT
- **KEY VARIABLE:**
  - NUMBER OF SATELLITES

WHEN TO REPAIR--ORBIT GEOMETRY

A key factor in analyzing the "when to repair" issue is the orbit geometry involved since down time is a major factor. The time of day issue deals with utilizing the daily occultation time occurring during the equinox when an average of 50 minutes per day of mandatory satellite down time occurs.

The time of year issue deals with confining the repair to the equinox which lasts 44 days. In the extreme case, one can also consider the case of doing the repair work beginning at the beginning of one equinox and lasting until the beginning of the second equinox which means 180 days available to repair the satellites.

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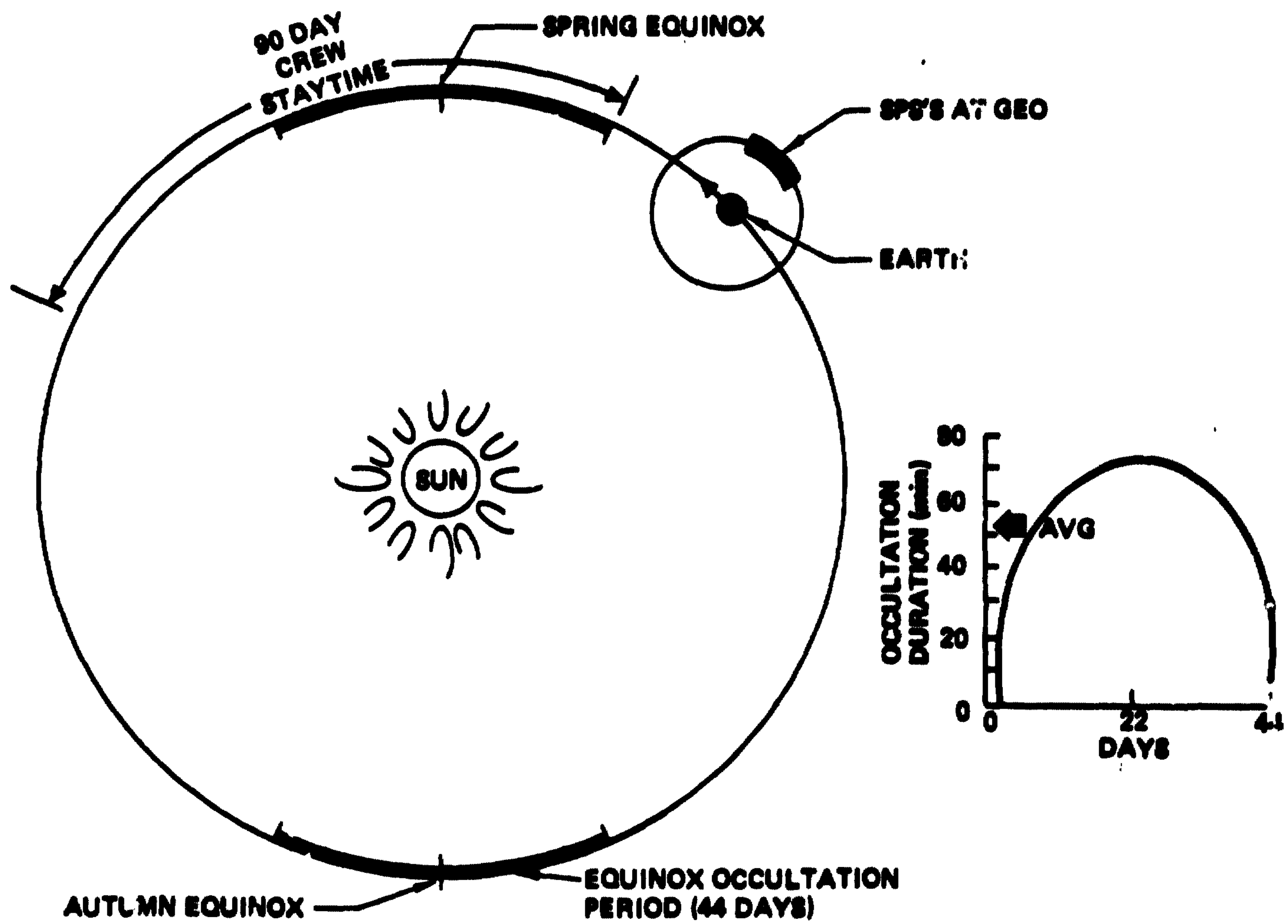
When to Repair Orbit Geometry



DAILY REPAIR SCHEDULE OPTIONS

Two primary options were considered relative to the daily repair schedule. The first of these called "Continuous" involves shutting down the satellite and then working on it until finished using four crews per satellite. Using four crews per satellite results in a repair time of 3½ days. A transportation time of ½ day to the next satellite and ½ day to activate the repair equipment at the satellite results in this one group of four crews repairing 10 satellites during an equinox period.

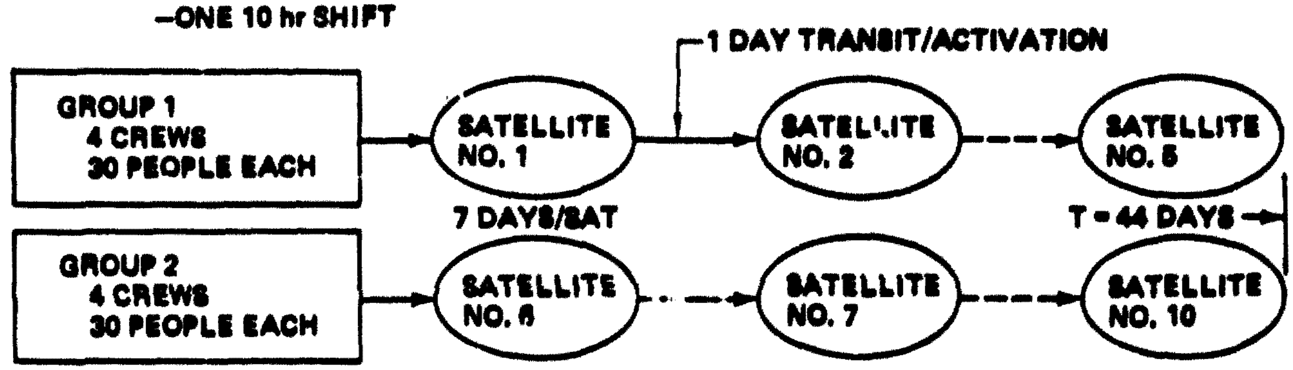
An alternate daily repair schedule would be an intermittent approach which centers the repair around the 50 minute occultation period per day. In the case investigated, about 4.5 hours are added on either side of the occultation period making it a 10 hour work day as has been used throughout the study. Since only one 10 hour shift is worked, each crew consists of only 30 people. This combination results in requiring 7 days to repair each satellite and consequently a group of 4 crews can only repair 5 satellites during an equinox period. Therefore, in order to repair 10 satellites as was accomplished in option one, another group of 4 crews is required for the repair of the other 5 satellites.

Daily Repair Schedule Options

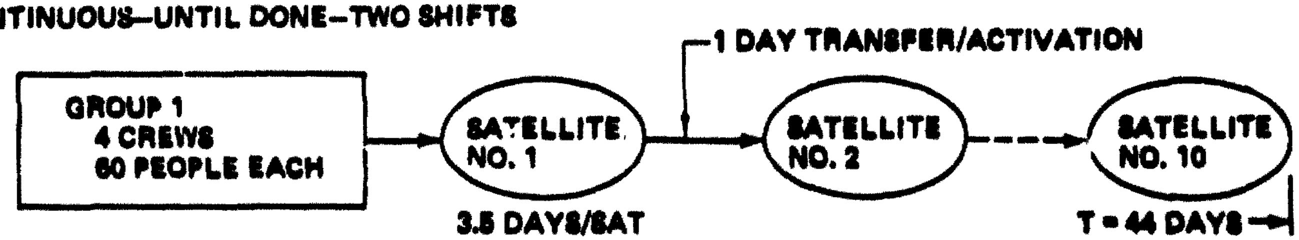
SPS-1916

BOEING

- CONDITIONS
  - 2,850 hr OF REPAIR PER SATELLITE
  - MAXIMUM OF 4 CREWS PER SHIFT
  - 10 REPAIR VEHICLES PER CREW
- OPTIONS
  - INTERMITTANT—CENTER AROUND OCCULTATION
    —ONE 10 hr SHIFT



  - CONTINUOUS—UNTIL DONE—TWO SHIFTS



DAILY REPAIR SCHEDULE COMPARISON

Comparison of the two daily repair schedule options with 10 satellites in orbit indicates no appreciable difference in cost. The ground grid impact, however, shows that the "continuous" repair case only has one satellite shutdown at a time while the "intermittent" approach requiring two groups working in parallel has two satellites shutdown per unit of time. The effect of the satellite shutdown during a low power period of the day versus continuously shut down during a 3½ day period has not been analyzed at this time.

Further analysis therefore will make use of the "continuous" repair approach.

Daily Repair Schedule Selection

SPS

SPS-1917

BOEING

- 10 SATELLITES
- ONE EQUINOX

	CONTINUOUS	INTERMITTANT
● DOWNTIME	24 hr/DAY 2.5 DAY/SAT ($350M)	11 hr/DAY 7 DAY/SAT ($330M)
● LOGISTICS	4 FLIGHTS ($100M)	4 FLIGHTS ($100M)
● GROUND POWER GRID SATELLITES DOWN PER UNIT OF TIME	ONE	TWO

SELECTION: CONTINUOUS UNTIL DONE

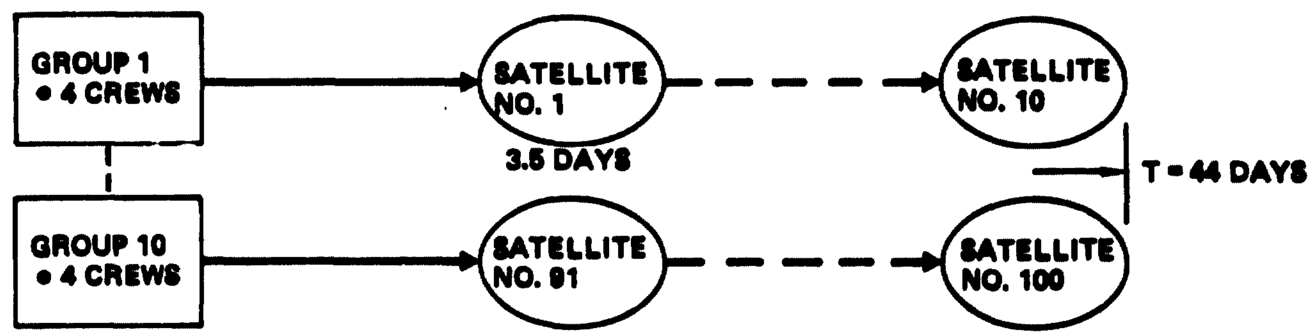
TIME OF YEAR REPAIR OPTIONS

The time of year comparison is made at a time well into the SPS program where 100 satellites are in orbt. The first of the options called "only during equinox" means all 100 satellites are repaired during the 44-day equinox period. This of course also occurs during the second equinox period when using a semi-annual maintenance concept. As indicated in the time of day analysis, one group of 4 crews can repair 10 datellites in 44 days. Therefore to repair 100 satellites, 10 groups of crews are required to repair the satellites in the designated time.

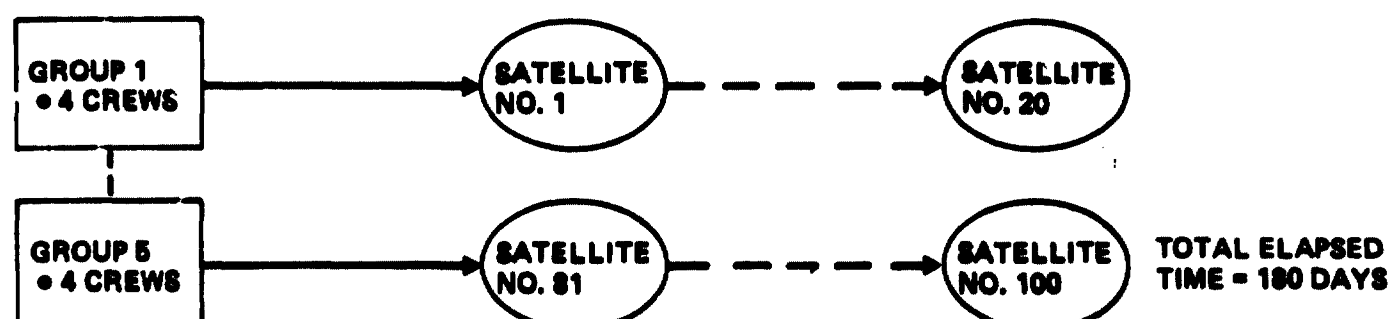
The second repair option called "equinox to equinox" has the crew beginning to work on the satellite at the start of one equinox and continuing until the start of the next equinox, at which time the cycle is repeated again to satisfy the semi-annual maintenance contract. Working in this manner, a total of 180 days is available to repair the satellites. If each group can repair 20 satellites, a total of 5 groups are required to service the 100 satellites.

Time of Year Repair Options

- CONDITIONS
  - 100 SATELLITES IN ORBIT
  - 4 CREWS/2 SHIFTS
  - TRANSFER TIME BETWEEN SATELLITES AND EQUIPMENT ACTIVATION = 1 DAY
- OPTIONS
  - ONLY DURING EQUINOXES



  - EQUINOX TO EQUINOX



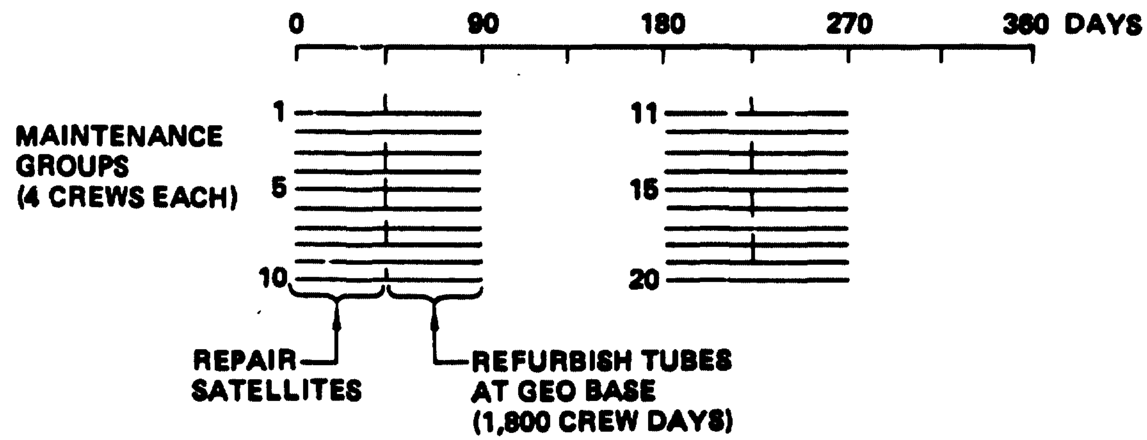
TIME OF YEAR MAINTENANCE SCHEDULE

The schedule used by the crews in these two options is shown. For the "only during equinox" options, the 10 groups are working in parallel on 10 different satellites and by definition finish the repair work in 44 days. When crew orbital stay times of 90 days are used, the remaining 46 days are spent back at the GEO final assembly base doing klystron tube refurbishment. The second set of 10 groups repeats this operation between days 180 and 270.

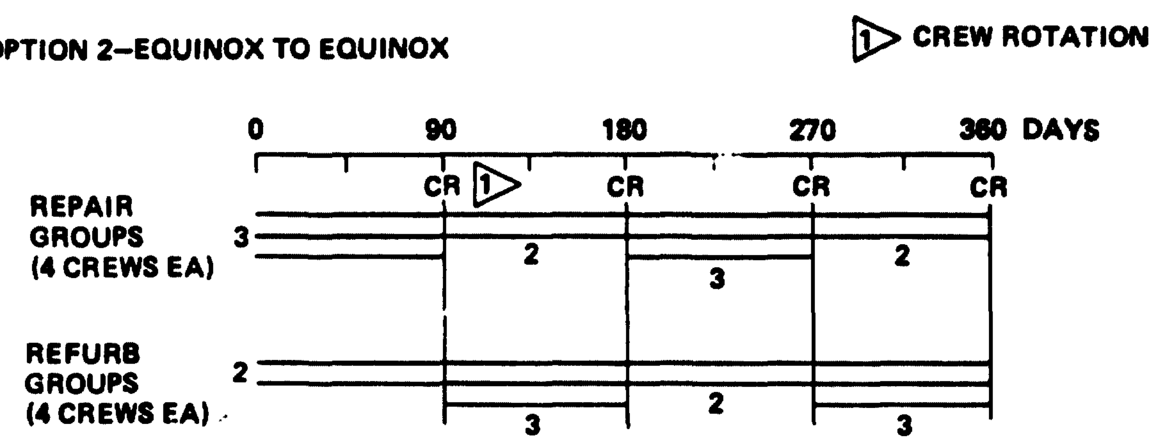
The "equinox to equinox" option involves five groups of crews to accomplish the required work. The implementation of this option has three groups repairing satellites the first 90 days in conjunction with two groups stationed at the GEO final assembly base doing klystron tube refurbishment. The second 90 day period has two groups repairing satellites (for a total of five groups in 180 days) along with three groups refurbishing klystron tubes.

Time of Year Maintenance Schedule

- OPTION 1—ONLY DURING EQUINOXES



- OPTION 2—EQUINOX TO EQUINOX



TIME OF YEAR MAINTENANCE SCHEDULE SELECTION

Comparison of the two maintenance schedules indicates a small cost advantage for the "only during equinox" option for those items have a direct annual cost. Consideration of amortized items, however, give a considerable advantage to the "equinox-to-equinox" option due to fewer crews being in orbit at a given time. This results in a total cost saving of $1230 million (11%) for the "equinox-to-equinox" option. Another factor highly in favor of the "equinox-to-equinox" maintenance option is the number of satellites shut down per unit of time. Again this advantage is the result of spreading the repair activities out over a longer period of time. The recommendation at this time concerning time of year is that "equinox-to-equinox" maintenance should be utilized.

SPS

SPS-1920

Time of Year Maintenance Schedule Selection

BOEING

	ONLY DURING EACH EQUINOX	EQUINOX TO EQUINOX
• DOWNTIME (ANNUAL	• 3.36 DAYS/100 SAT • $6720M	• 3.38 DAY/30 SAT • 3.5 DAY/70 SAT • $6,900M
• LOGISTICS (ANNUAL)	• 40 CREWS • 2 TIMES/YEAR • $2,000M	• 20 CREWS • 4 TIMES/YEAR • $2,000M
• MAINTENANCE HABITATS (AMORTIZED)	• 40 UNITS • $1,440M	• 20 UNITS • $720M
• REFURB FACILITY AT GEO BASE (AMORTIZED)	• 20 UNITS • $960M	• 6 UNITS • $270M
• TOTAL COST DELTA	$11120M	$9890M (11% SAVINGS)
• GROUND GRID IMPACT (SATELLITES DOWN PER UNIT OF TIME)	10%	3%

SELECTION: EQUINOX TO EQUINOX

MAINTENANCE MISSION OPTIONS

Several options exist relative to the orbital location of the maintenance system elements and the associated transportation operations. A simplified version of each of these options is presented in this chart, with a transportation comparison following on the next chart.

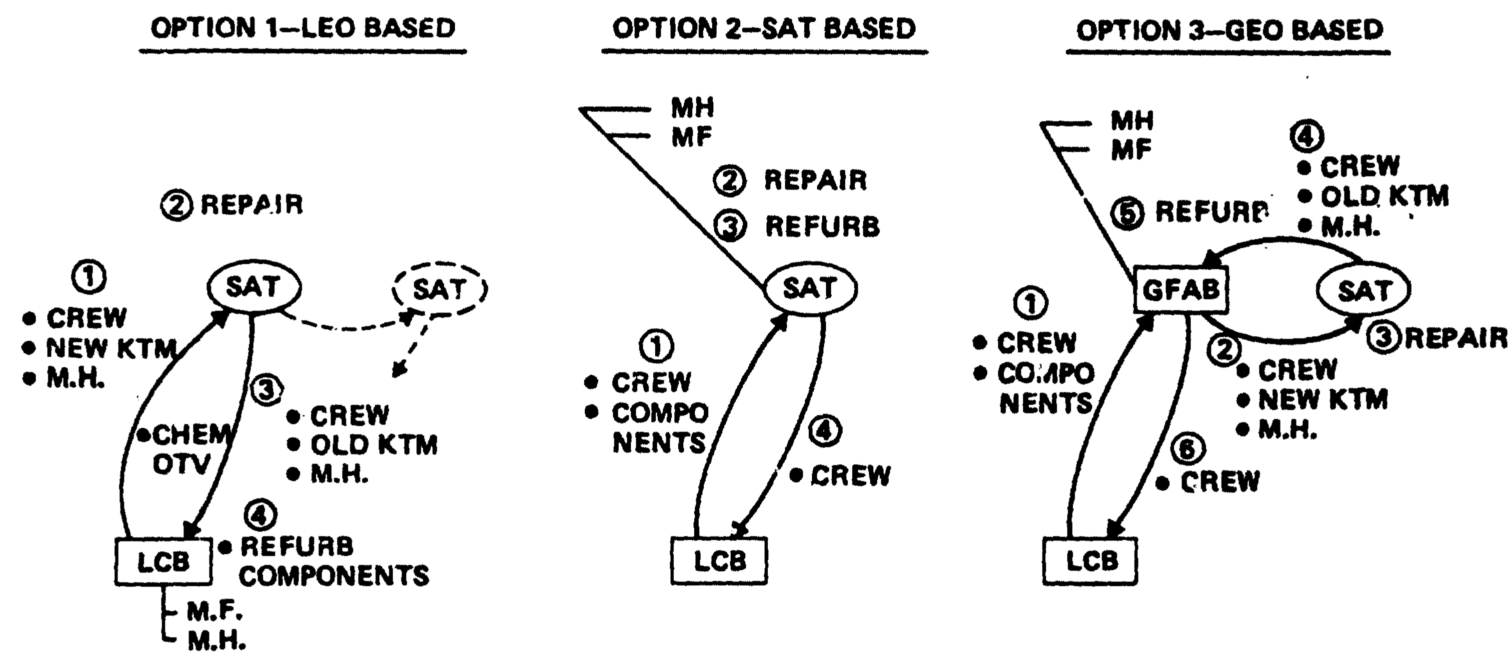
Option 1 is primarily considered for the case where there are only a few satellites in orbit and consequently a mission approach consisting of maintenance sorties to a satellite similar from a LEO base may be advantageous. In this option, all maintenance elements are located at the LEO construction base except when a repair sortie is performed. Each sortie to GEO has a payload consisting of maintenance habitat plus completely refurbished klystron tube modules.

Option 2 consists of installing as permanent equipment at each satellite the required maintenance habitat and refurbishment facility and equipment. Flights from a LEO base include crews and components to repair the klystron tube modules rather than ready to go klystron tube modules.

Option 3 has the maintenance elements all based at the GEO final assembly base. All maintenance crews and klystron tube module components come to the GEO final assembly base. Maintenance sorties to the satellite are made from the GEO final assembly base and include crew habitats and completely refurbished klystron tube modules. After the satellite is repaired, the sortie returns to the GEO final assembly base with the failed klystron tubes where they will be refurbished with components that have previously been brought up.

Maintenance Mission Options

- KEY ISSUES
  - MAINTENANCE HABITAT LOCATION
  - MAINTENANCE FACILITY (REFURB) LOCATION
  - TRANSPORTATION



- LCB = LEO CONST BASE
- GFAB = GEO FINAL ASSEMBLY BASE

- M.H. = MAINTENANCE HABITAT
- M.F. = MAINTENANCE FACILITY
- KTM = KLYSTRON TUBE MODULES

TRANSPORTATION REQUIREMENTS—MISSION OPTIONS

C-3

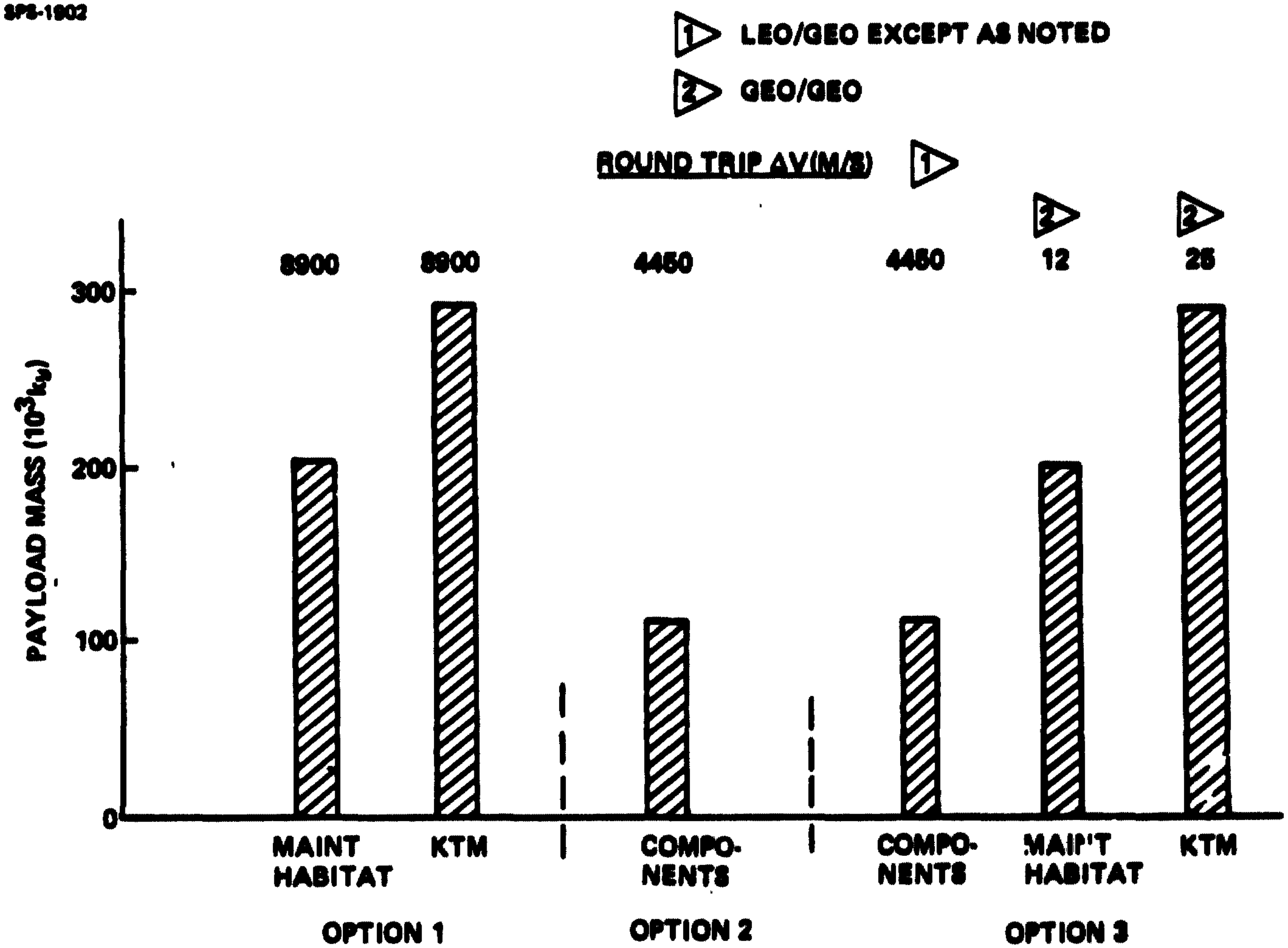
Comparison of the mission options for their transportation requirements in terms of payload mass and delta V indicates Option 3 using GEO based systems the most desirable.

Option 1 with the maintenance system based at LEO has both high mass since habitats and complete klystron modules must be transported and high delta V since these two elements must be brought back to the LEO base.

Option 2 with the maintenance system permanently attached to satellites only requires components which have a mass 38% as much as complete klystron tube modules to be delivered to GEO. Although this option is good from the transportation standpoint, installation of the habitats and refurbishment facilities at each satellite presents an unacceptable investment cost.

Option 3 also only delivers klystron components to GEO (after the habitats and refurb facilities are initially delivered) and then transfers the habitat and refurbished klystron tube modules from one satellite to another which involves very small delta V requirements.

Transportation Requirements
Mission Options



KTM = KLYSTRON TUBE MODULE

SELECTED MAINTENANCE MISSION CONCEPTS

The final operations associated with a typical 90-day period are illustrated. After the 20th satellite has been repaired, the crew and habitat return to the GEO final assembly base where the habitat is left for the next repair crew. The initial crew then returns back to the LEO construction base and eventually back to Earth. The refurbishment crew has also completed their 90 day stay time and also returns back to Earth.

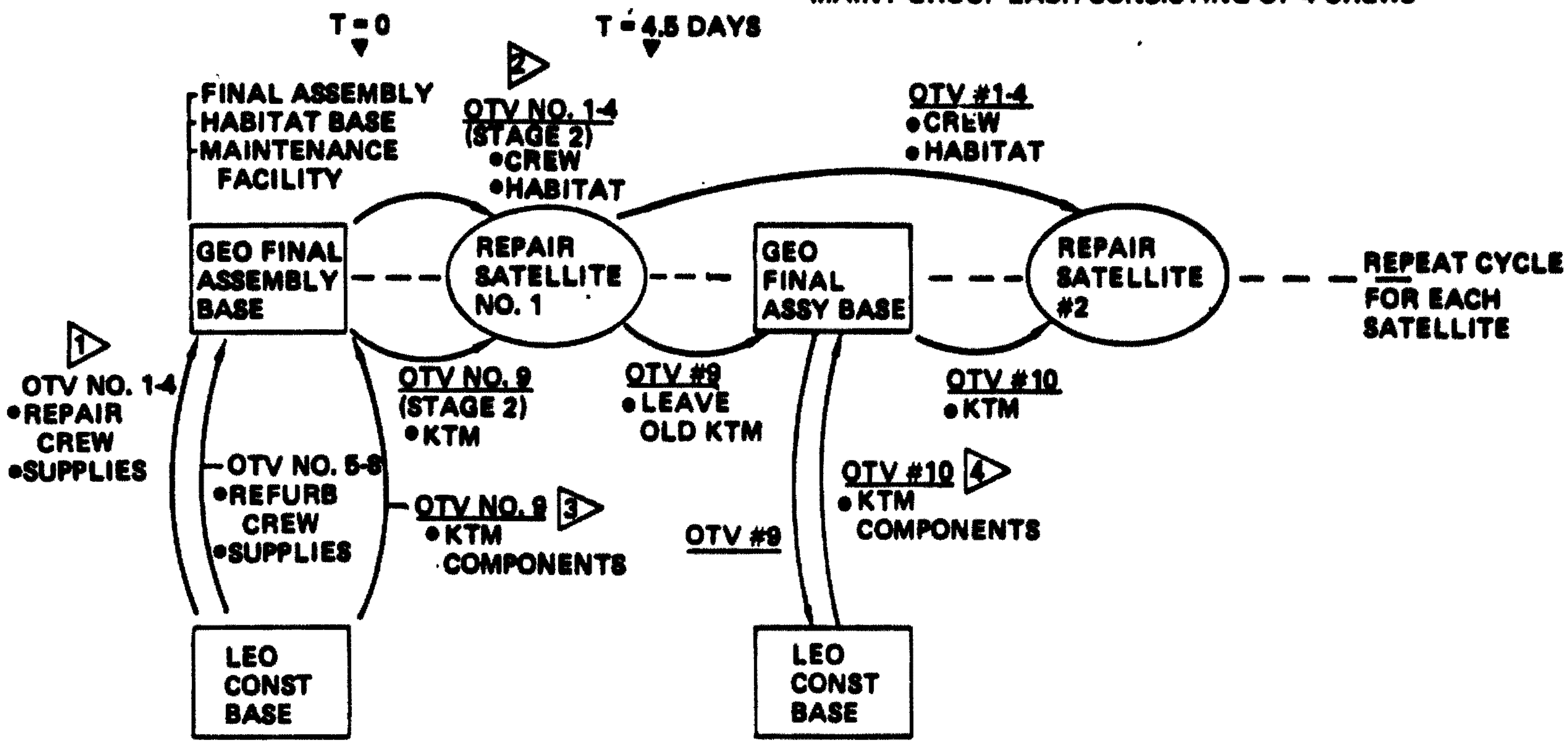
Four new repair crews and four new refurbishment crews are then transferred to the GEO final assembly base. The complete cycle is repeated again. The crew size at the GEO final assembly base will have a maximum operating size of 310 (240 associated with refurbishment and 70 with satellite assembly) and at the time the four repair crews return at the end of their tour of duty the crew size will be 550.

The annual number of orbit transfer vehicles and launch vehicles flights which occur in the maintenance of 100 satellites are also indicated. During this time period, maintenance operations will completely dominate the GEO transportation operations rather than assembly of satellites at GEO.

Selected Maintenance Mission Concept

SPS-1935

• FLIGHT OPERATIONS FOR ONE REPAIR AND ONE REFURB MAINT GROUP EACH CONSISTING OF 4 CREWS



KTM = KLYSTRON TUBE MODULE

1. ALL OTV'S HAVE W_p = 460[k] kg. NO PROP TRANSFER REQUIRED AT GEO
2. 1 deg SEPARATION BETWEEN SATELLITES
3. COMPONENTS COULD BE DELIVERED VIA SELF POWER MODULE
4. CAN BE DELIVERED BEFORE OTV NO. 9 RETURNS

SELECTED MAINTENANCE MISSION CONCEPT

The selected maintenance mission concept is now presented in greater detail in the next two charts. The operations discussed in these charts will be those associated with one GEO final assembly base and the operations associated with one repair group and one refurbishment group. Other final assembly bases would have comparable operations.

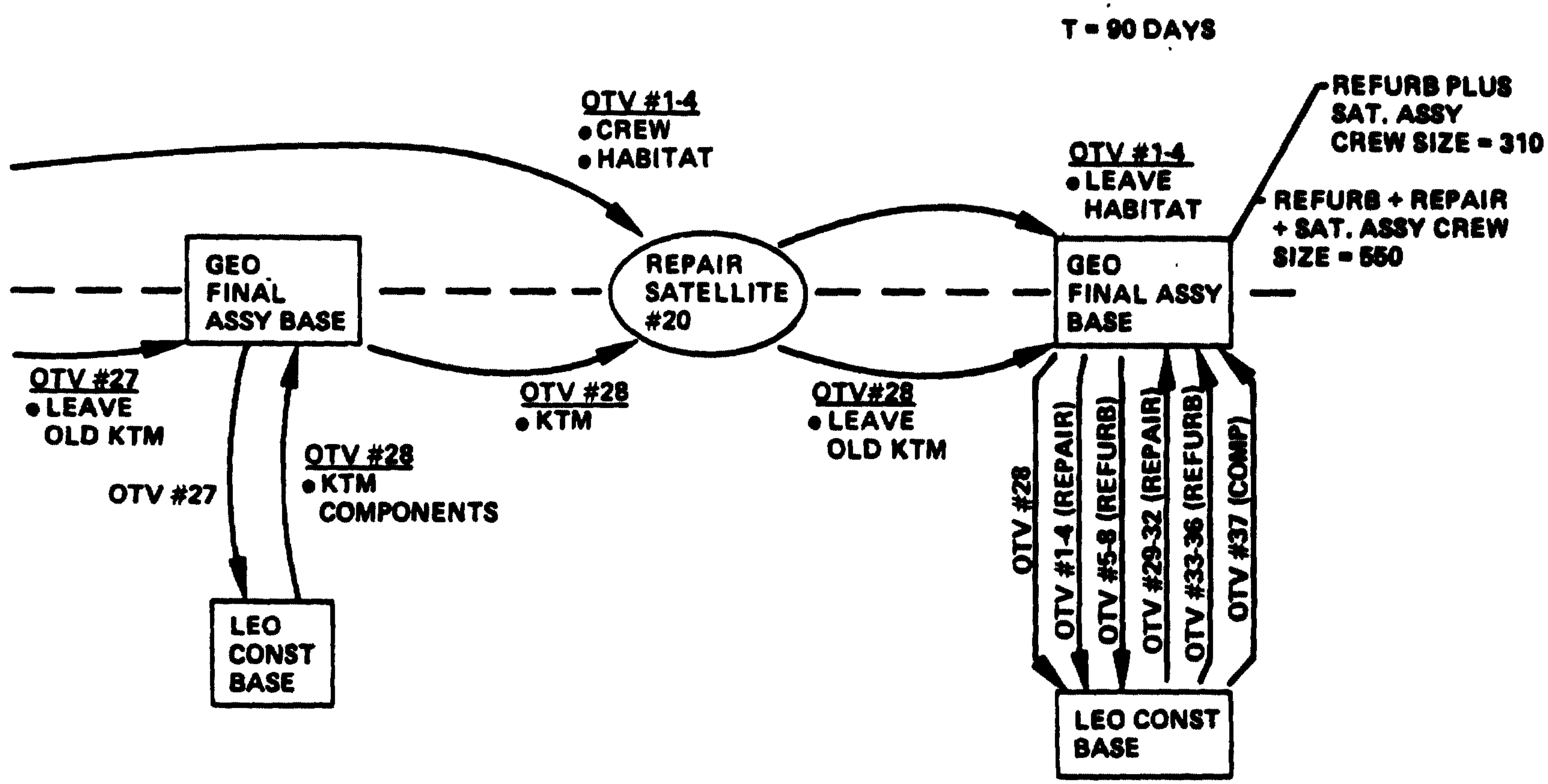
Once maintenance operations are begun, the GEO construction base serves as a major staging depot for the maintenance crews and their hardware in addition to its role of constructing the satellites. Typically, the following operations occur. Four repair crews and four refurb crews are transported to the GEO final assembly base. Each crew is provided with its own orbit transfer vehicle. At approximately the same time another orbit transfer vehicle delivers klystron tube module components to be used in the refurbishment of failed tubes.

Refurbishment crews remain at the GEO final assembly base, repairing the failed klystron tube modules that have previously been delivered by other repair crews. Repair crews transfer to the satellite designated for repair taking with them their habitat. The second stage of the orbit transfer vehicle which brought the crew to GEO is used for the transfer to the satellite. The second stage of the orbit transfer vehicle used to deliver the klystron tube components to the GEO final assembly base is then loaded with refurbished klystron tube modules and transferred to the first satellite to be repaired.

At the completion of repairs on the first satellite, the crew and habitat transfer to the next satellite to be repaired. The other orbit transfer vehicle transport the failed klystron tube modules back to the GEO final assembly base where they will be refurbished. The OTV then returns back to the LEO construction base. Prior to this time, however, another orbit transfer vehicle has come from the LEO construction base to the GEO final assembly base delivering additional klystron tube components and is then dispatched with completely refurbished klystron tube modulesto the second satellite that is to be repaired.

This cycle is repeated for each satellite to be repaired.

Selected Maintenance Mission Concept



ANNUAL FLT SUMMARY (100 SATELLITES)

- 280 OTV FLTS TO GEO
- 400 OTV FLTS GEO TO GEO
- 350 HLLV FLTS TO LEO
- 80 SHUTTLE GROWTH FLTS TO LEO

SATELLITE MAINTENANCE SYSTEMS

The satellite maintenance systems and their location on an antenna are shown on the next two charts. The systems are shown as they relate to one side of one antenna. Since four crews work on each satellite, those same systems are present on both sides of both antennas.

To enable the docking of the various maintenance system elements and to transfer cargo around the antenna, the antenna structure has been designed to incorporate a cargo distribution system and has structural additions to allow maintenance gantries to be positioned so they can be maintained and supplied with new klystron tube modules.

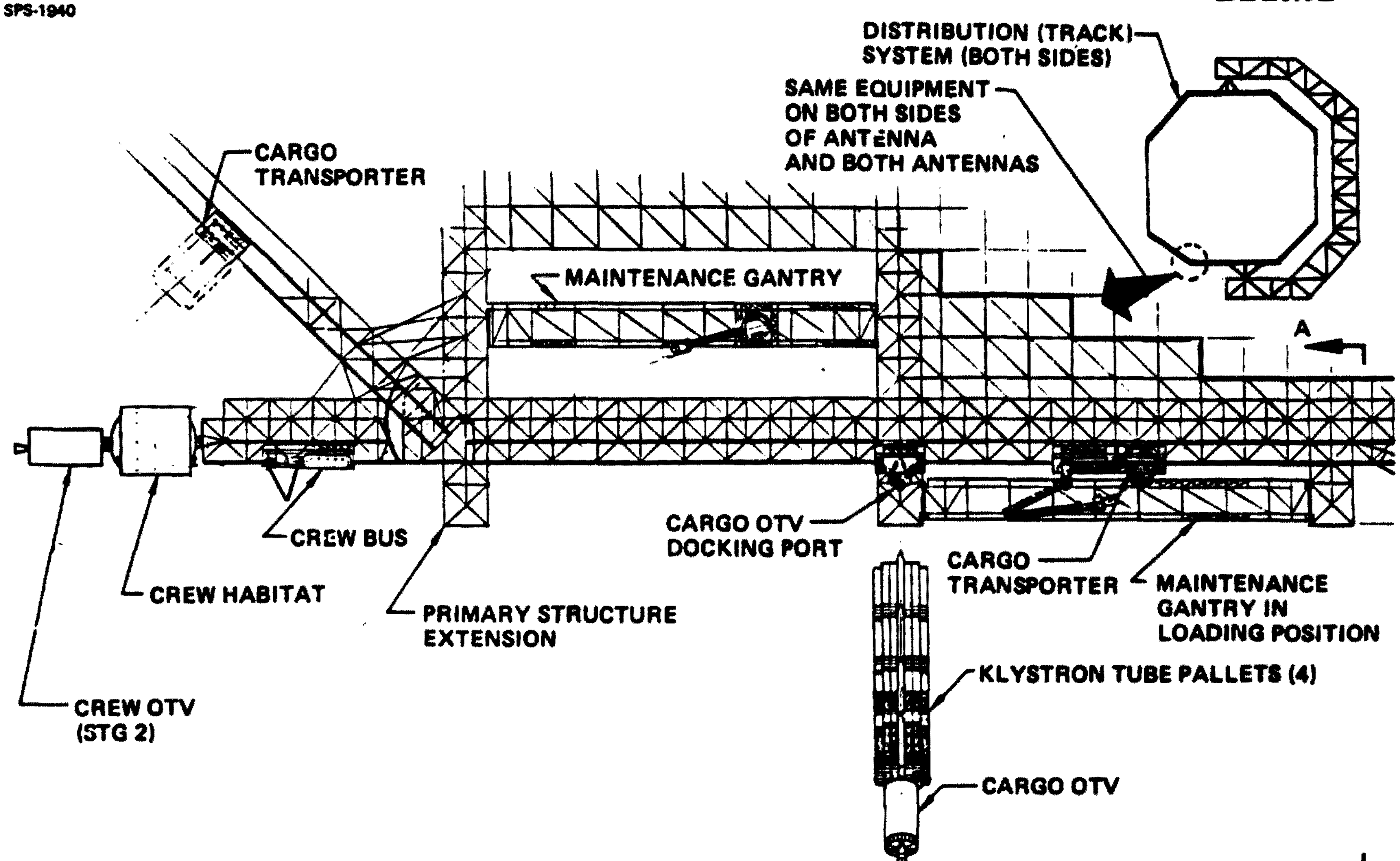
The 60 person crew is delivered to the satellite in the crew habitat using the second stage of the OTV that initially brought the crew from the LEO construction base to the GEO final assembly base. Once at the satellite (antenna), a crew bus is used to transfer persons between the habitat and the maintenance repair vehicles.

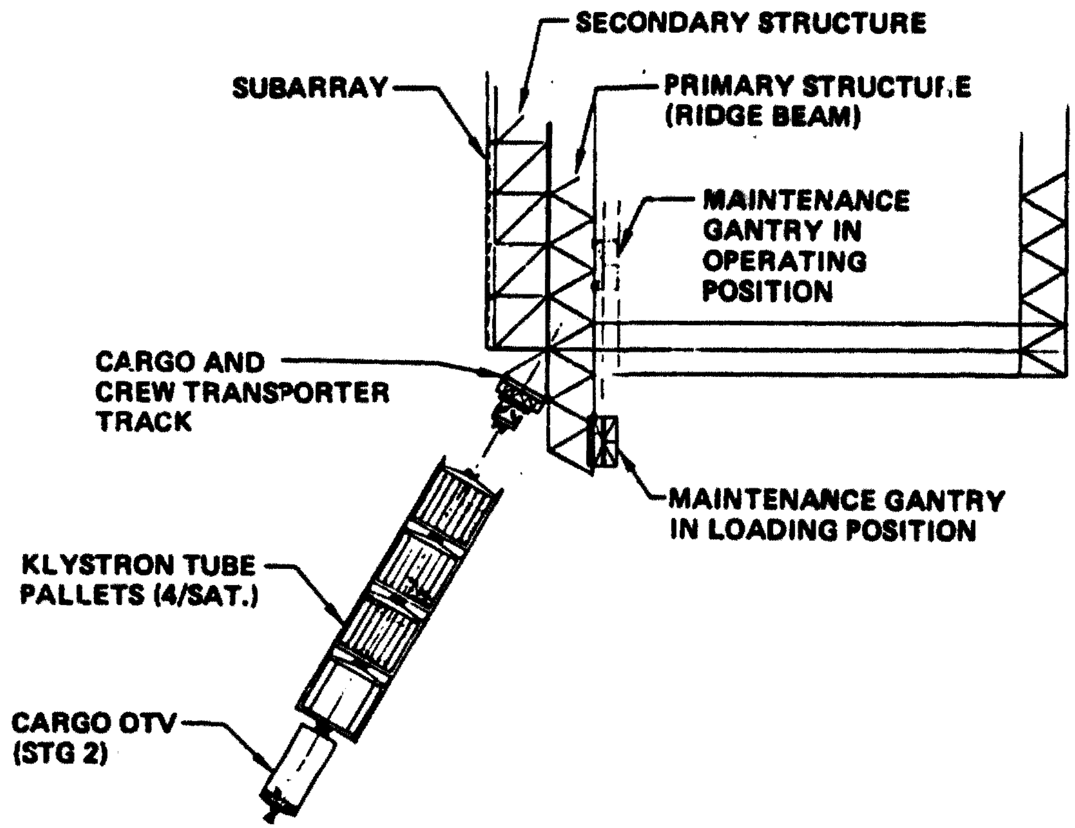
Cargo, primarily in the form of klystron tube modules is also delivered to the satellite using a dedicated OTV (stage 2) that had initially brought klystron components to the GEO final assembly base for refurbishment of "failed" klystron tubes. The operations associated with an OTV include docking and release of one klystron tube pallet on one side of the antenna and then free-flying to the other side of the antenna leaving another pallet followed by flying to the other antenna and leaving two pallets in a similar manner. At the completion of the repair operation, the pallets are loaded with "failed" klystron tubes. The OTV then moves to the four docking locations collecting the pallets and then returns them back to the GEO final assembly base where they will be refurbished. Following the release of the pallets, the OTV returns to the LEO construction base where it is made ready to deliver another load of klystron components.

The actual distribution of the cargo around the antenna is accomplished through use of cargo transporters operating on the track system on two sides of each antenna. The cargo transporter system consists of three separate units attached together to form a "train". The middle unit is a control unit that has a crew cabin, power systems and crane/manipulator that moves the cargo between the train and the maintenance gantries. Units on either side of the control unit are essentially trailers that carry either new klystron tube modules or those that have failed and have been removed. The train system moves down to each gantry and delivers to it the number of klystron tubes required in that particular antenna channel during one shift or one day of operation depending on the channel.

Satellite Maintenance Systems

SPS-1940




SECONDARY STRUCTURE
SUBARRAY
PRIMARY STRUCTURE
(RIDGE BEAM)
MAINTENANCE
GANTRY IN
OPERATING
POSITION
CARGO AND
CREW TRANSPORTER
TRACK
MAINTENANCE GANTRY
IN LOADING POSITION
KLYSTRON TUBE
PALLETS (4/SAT.)
CARGO OTV
(STG 2)

A-A

ANTENNA MAINTENANCE SYSTEM INSTALLATION

The installation of the maintenance equipment on an antenna being repaired by two crews is shown.

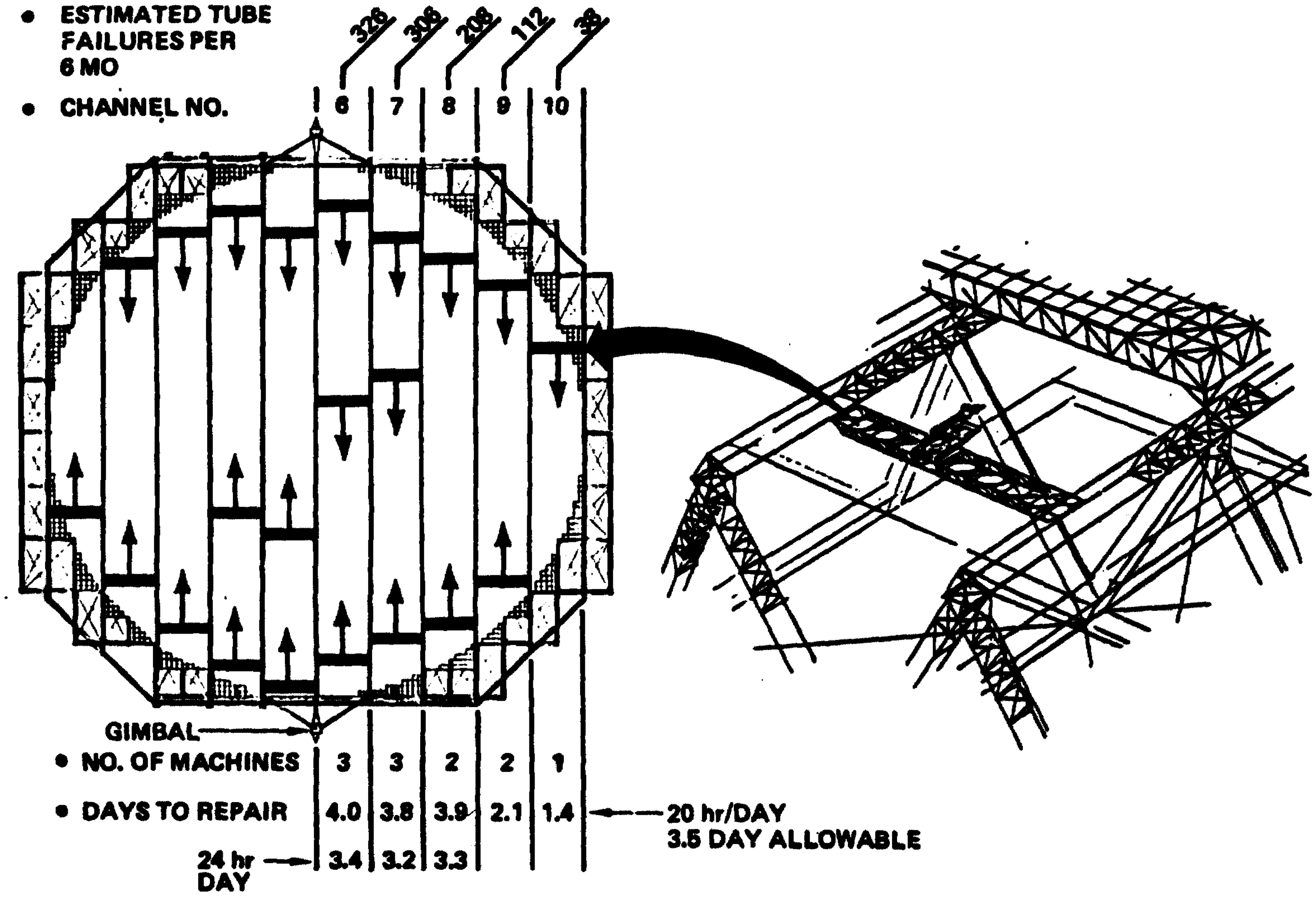
The number of maintenance vehicles (machines) installed in each channel of the antenna is a function of the estimated number of tube failures. This value is larger in the middle channels of the antenna since the center of the antenna has subarrays containing 36 and 30 klystron tube modules while near the edge of the antenna, the subarrays have 4 or 6 tubes per subarray. Consequently it will be noted that the middle channel has three maintenance systems consisting of a gantry and repair vehicle.

With this equipment distribution and working 20 hours per day, the middle channels require slightly more time than previously identified for repair—3 1/2 days per satellite. The addition of 1/2 day to the schedule, however, will not appreciably alter the prior analysis.

It should also be noted, the outside channels require far less time to repair and less equipment due to fewer failed tubes. Consequently when the crews assigned to this particular equipment are finished, they can then be used to repair other components on the satellite such as the dc-dc converters mentioned earlier in the discussion.

Antenna Maintenance System Installation

SPS-1937



ANNUAL OPERATING CHARACTERISTICS

The plant factor resulting from the selected maintenance approach is 0.92. The occultation loss cannot be prevented. The output degradation loss is the average which results from using a semi-annual maintenance approach. Downtime loss is the 3-1/2 to 4 day repair time per satellite occuring twice a year. A 3% allowance for such things as unexpected failures and unknowns is also included.

The total cost to maintain 100 satellites per year is over $11 billion or approximately $118 million per satellite. It should be noted that some of these costs are annual costs such as transportation (delivery) while others are amortization of the maintenance equipment. Although the maintenance cost may appear high it still is less than 3% of the yearly revenue per satellite and can be expressed as $0.0015 per kilowatt hour.

Annual Operating Characteristics

PLANT FACTOR LOSSES	
• OCCULTATION	1%
• OUTPUT DEGRADATION DUE TO FAILURES	2%
• PLANNED MAINTENANCE DOWNTIME	2%
• ALLOWANCE FOR OTHER FACTORS	3%
ESTIMATED FACTOR	0.92

MAINTENANCE COST/YEAR	(100 SATELLITES)
• CREW DELIVERY	\$2,000M
• KTM DELIVERY	\$3,385M
• KTM COST	\$1,520M
• MAINTENANCE HABITAT [1]	\$720M
• REFURB FACILITY [1]	\$270M
• REPAIR EQUIPMENT [1]	\$2,400M
• REPAIR EQUIPMENT DELIVERY	\$1,490M
TOTAL	\$11,785M
AVG/SAT	\$118M
	(3% OF YEARLY REVENUE)

• MAINTENANCE COST (\$/kWh)

$$= \frac{\text{ANNUAL MAINT COST/SAT}}{8{,}766 \text{ HR} \times \text{PLT FACTOR} \times 10^7 \text{ kW}}$$

$$= \frac{\$118 \times 10^6}{8{,}766 \times 0.92 \times 10^7} = 0.0015$$

[1] AMORTIZED (15% CAPITAL CHARGE)

MAINTENANCE AND OPERATIONS SUMMARY

An overall summary of the selected maintenance approach for each of the key issues mentioned at the beginning of the analysis is presented. Although these selections should not be considered as the most optimum they do present a reasonable approach from the standpoint of both cost and practicality. In summary, maintaining solar power satellites does not present an overwhelming obstacle as some had initially envisioned.

Maintenance and Operations Summary

SPS-1923

• KEY CONCERN	KLYSTRON TUBES
• LEVEL OF REPLACEMENT	TUBE MODULE
• REPAIR CONCEPT	VERTICAL ACCESS
	CUBIC SECONDARY STRUCTURE
	"A" FRAME PRIMARY STRUCTURE
• FREQUENCY OF MAINTENANCE	SEMI-ANNUAL
• MAINTENANCE SCHEDULE	EQUINOX TO EQUINOX
	DAILY UNTIL DONE
• CREWS/SATELLITE	FOUR
• MAINTENANCE HABITAT	BASE AT GFAB
	MOVE TO SATELLITE
• REFUBRISH LOCATION	AT GFAB
• LEO-GEO TRANSPORTATION	CREW → CHEM OTV
	...COMPONENTS → CHEM OR SAT MODULE
• GEO-GEO TRANSPORTATION	ALL ITEMS VIA CHEM OTV
• COST/SATELLITE	LESS THAN $120 MILLION
	...LESS THAN 2 MILL/KWHR

D180-24071-3

MPTS Studies

BOEING

- ARRAY ERROR ANALYSIS
- ALTERNATE FREQUENCY STUDIES
- PHASE CONTROL
- RECTENNA

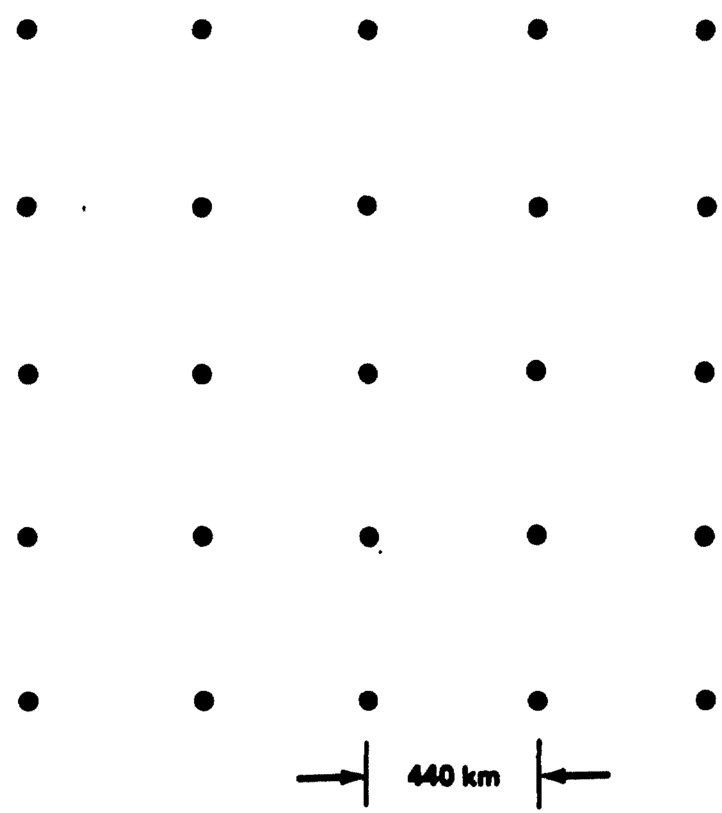
E. Nalos

ARRAY ERROR ANALYSIS

GRATING LOBE STUDIES

The grating lobe positions along the principal planes are determined by the following equation: $X = R \tan(\sin^{-1} \frac{n\lambda}{L+gap})$ where R is the orthoginal range, n is the order of the grating lobe, L is the subarray length, and gap is the spacing between subarrays. In this study only the lobes along the principal planes have been studied. The computer program is presently being modified to study grating lobe behavior at any ϕ cut desired. Systematic tilt has the major effect on the grating lobe levels (over random subarray tilt, power taper, quantization, subarray spacing and other errors) due to the shifting of the grating lobe peaks out of the nulls of the subarray pattern.

Grating Lobe Positions

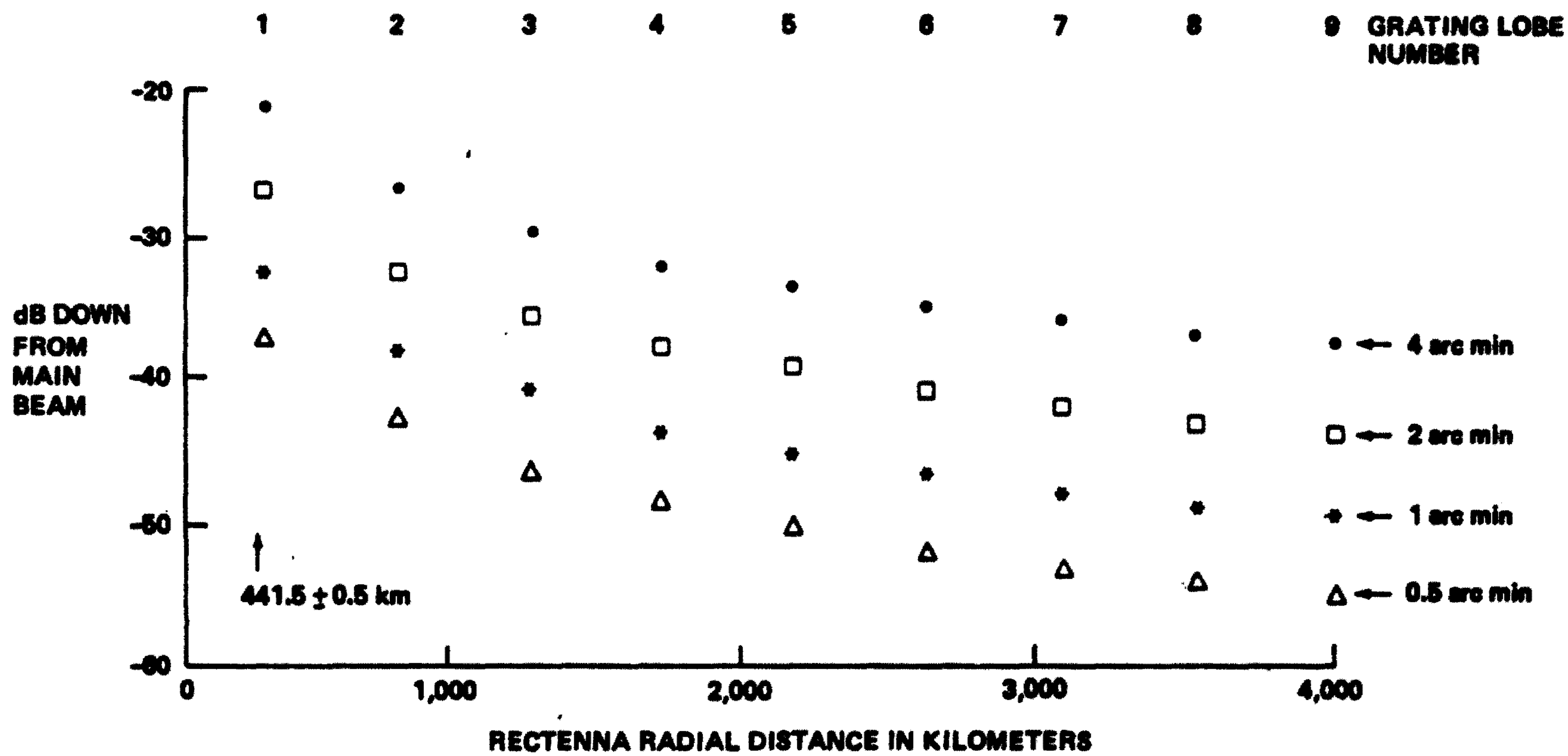


- 10 x 10m SUBARRAYS HAVE GRATING LOBES SEPARATED BY 440 km AT 2.45 GHz
- WIDTH OF UNSPLIT GRATING LOBES CORRESPOND TO ≈ 13 km, SAME AS MAIN BEAM.
- SYSTEMATIC TILT HAS MAJOR EFFECT ON GRATING LOBE LEVELS.

GRATING LOBE PEAKS PRODUCED BY SYSTEMATIC SPACETENNA TILT

Nine grating lobe peaks are shown for each of four cases of systematic spacetenna tilt. As a simple rule-of-thumb for each doubling of systematic tilt the grating lobe amplitudes increase approximately six decibels. It is important to note that the grating lobe levels may be larger than the first sidelobe (≈23.5 dB down) as shown for the first grating lobe for the four arc-minute tilt case. The grating lobes roll off at approximately 22 dB per decade.

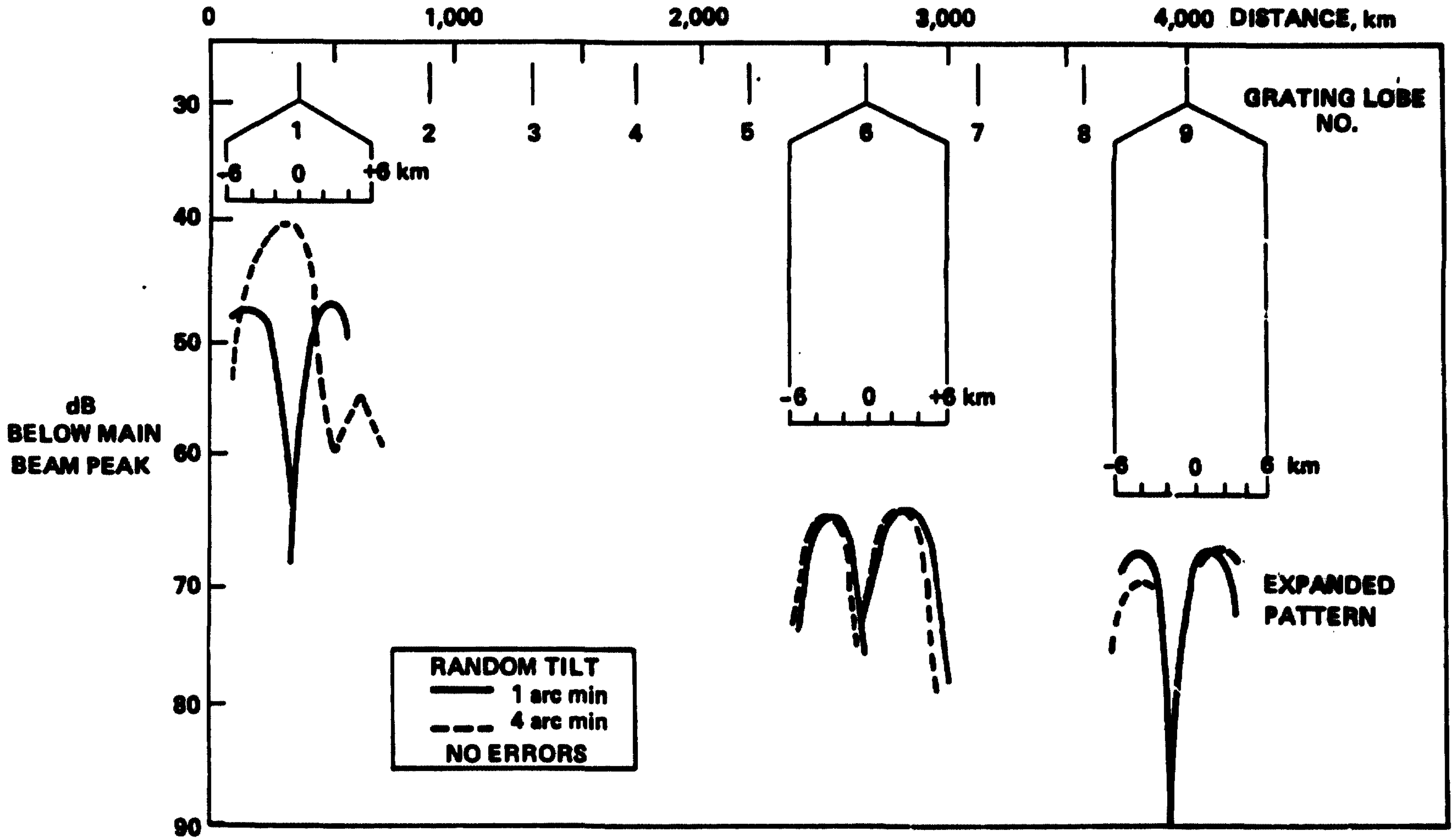
Grating Lobe Peaks Produced by Systematic Spacetenna Tilt



EFFECT OF RANDOM TILT ON GRATING LOBE LEVEL

Two cases of random tilt are shown, one and four arc minutes, for three of the nine grating lobes studied. The random tilt is incorporated in the phased array computer program by adding a random angle to theta, utilizing the Gaussian random number generator and the specified standard deviations of one or four arc minutes.

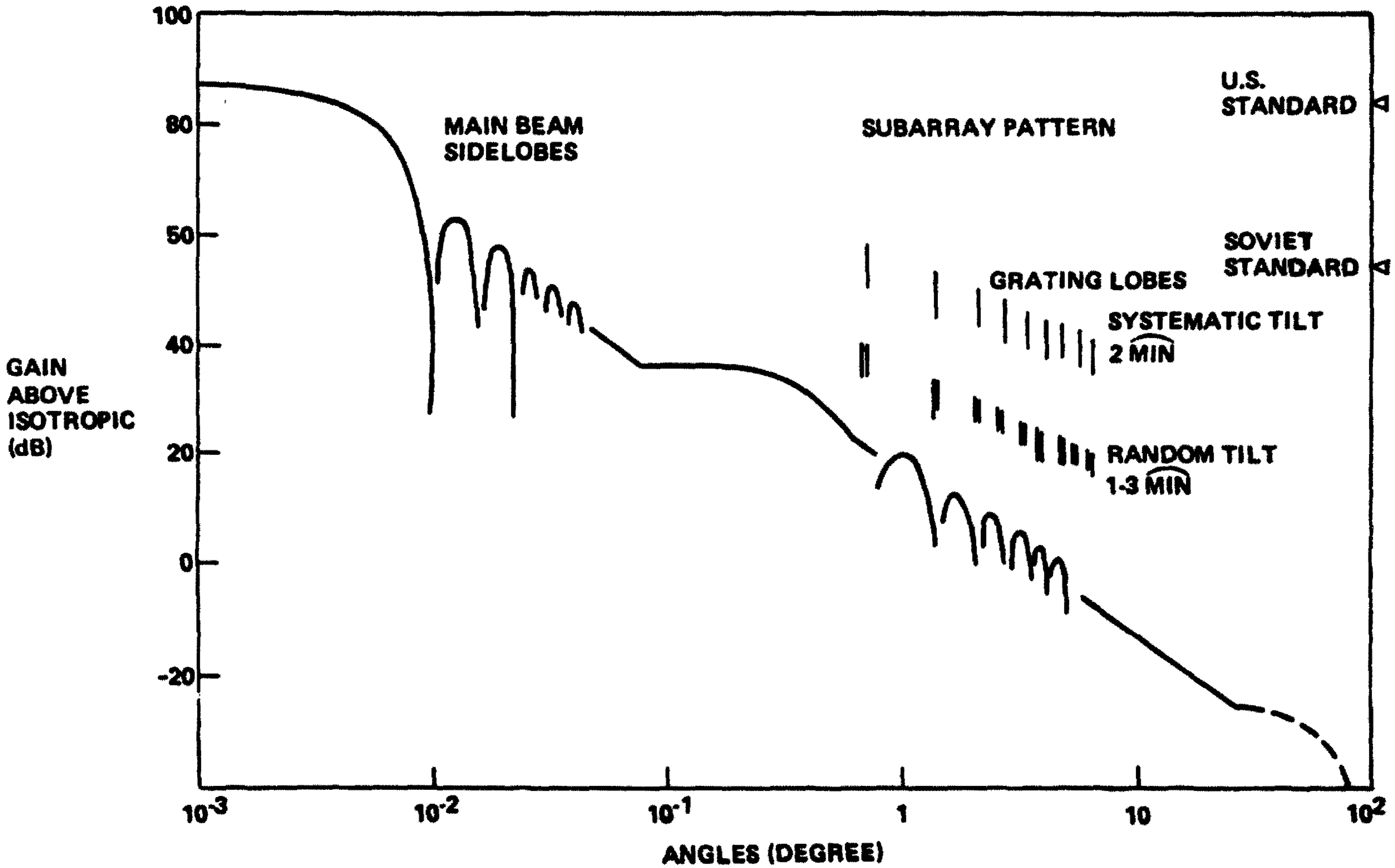
Effect of Random Tilt on Grating Lobe Level



SPACETENNA PATTERN ROLL-OFF CHARACTERISTICS

The grating lobe levels are shown for both systematic and random tilt. For the case of systematic tilt, the grating lobe peaks are moved out of the nulls of the subarray pattern. For random tilt, the grating lobes are split because statistically the peaks lie in the nulls of the subarray pattern.

Array Pattern Roll-Off Characteristics



Grating Lobe Effects

SUBARRAY TILT	SYSTEMATIC TILT IS THE MAJOR FACTOR AND NEEDS TO BE MINIMIZED. RANDOM TILT HAS LITTLE EFFECT FOR NOMINAL VALUES.
QUANTIZATION	THERE IS NO DIFFERENCE ON THE GRATING LOBE AMPLITUDES BETWEEN THE CONTINUOUS GAUSSIAN DISTRIBUTION AND THE 10 STEP QUANTIZED DISTRIBUTION.
ILLUMINATION TAPER	AS TAPER DOUBLES FROM 10 TO 20 DB THE GRATING LOBE AMPLITUDES INCREASE BY 0.1 DB
SUBARRAY SPACING	FOR A UNIFORM OR RANDOM ¼" SUBARRAY GAP SPACING (1.65 ARCSEC), THERE IS NO SIGNIFICANT CHANGE IN GRATING LOBE LEVEL.
OTHER ERRORS	INCORPORATING THE NOMINAL ERRORS OF: 10° RANDOM PHASE ERROR, 1DB RANDOM AMPLITUDE ERROR, AND 2% FAILURES PRODUCED NO CHANGE IN GRATING LOBE AMPLITUDES. INTRA SUBARRAY ERRORS NEED FURTHER STUDY.

KLYSTRON DESIGN FOR 5.8 GHZ

The klystron scaling to higher frequencies is relatively straightforward; however, a conservative power level for 30 year life needs to be further assessed. The assumptions made are: the same rf power level of 70 kw, same voltage limit of 42 kv, 5 segment depressed collector, output gap thermal stress of .5 kw/cm^2, solenoid focusing and estimated collector recovery of 54%. The various beam and efficiency parameters are listed in the following table.

The scaling relationships are illustrated in the attached figure and reflect a frequency factor of 2.37. The length of the rf section is reduced by 2.37 for the same transit time, the solenoid power increases by 2.37 to a 3.5 kw design for 300°C operation, cavity Q's are reduced by $1/\sqrt{2.37}$, resulting in an estimated rf efficiency reduction of 1.62% with a total efficiency reduction of 3.59% including the increased solenoid power.

The mass reduction, indicated in the following table, is estimated at 27%, primarily due to the shorter, lighter solenoid. If an equal weight were selected, the solenoid power could be reduced and the efficiency reduction would drop from 3.59% to 2.61%.

Alternate Frequency Assessment

- KLYSTRON EFFICIENCY ANALYSIS
- PARAMETRIC ACQUISITION COST ANALYSIS
- FREQUENCY OPTIMIZATION

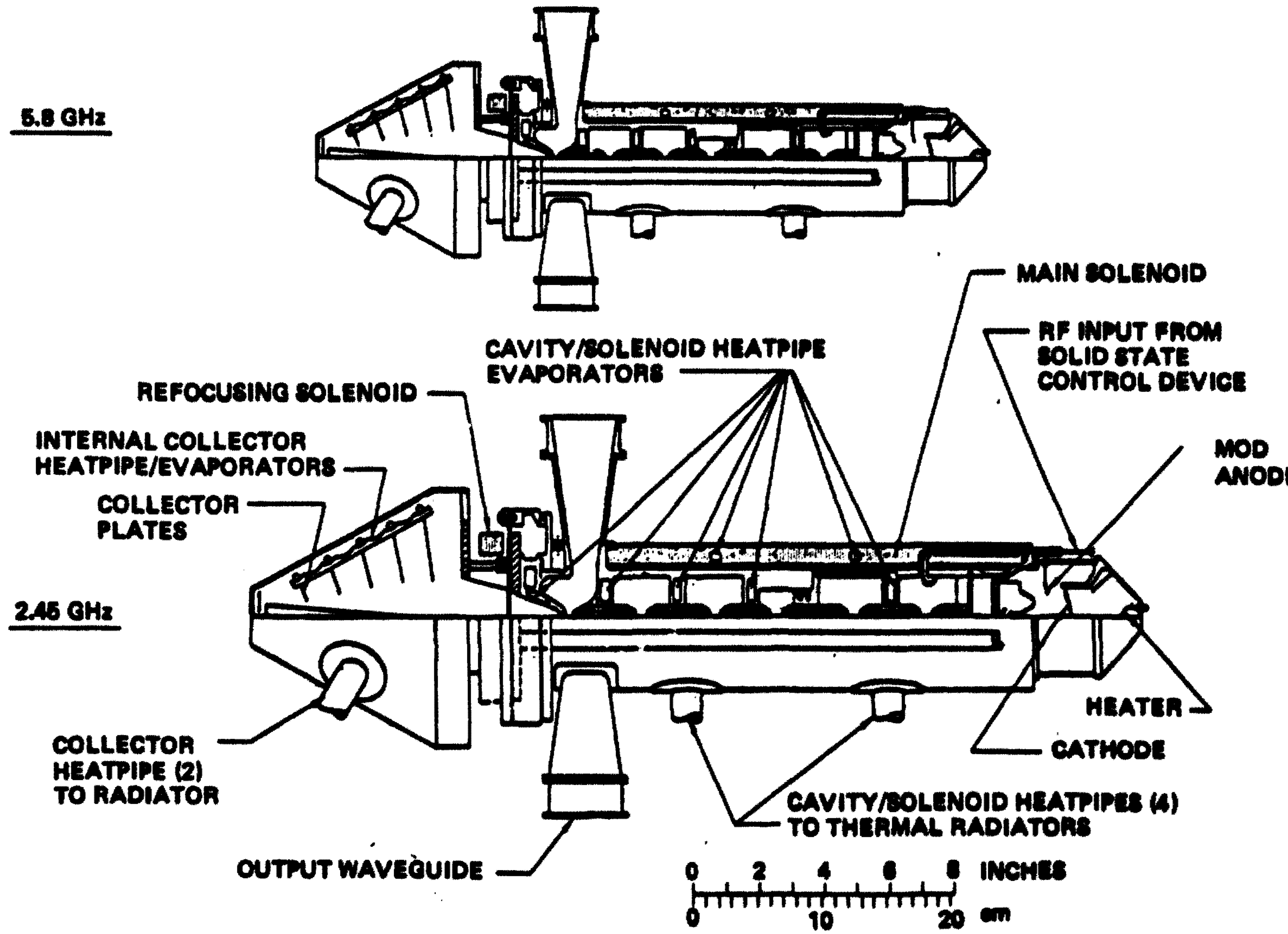
Klystron Design Parameters Versus Frequency

EFFICIENCY PARAMETER		2.45 GHz	5.8 GHz
CIRCUIT	η_{ckt}	0.967	0.949
ELECTRONIC	η_e	0.769	0.768
COLLECTOR	η_c	0.54	0.54
TOTAL		.850	.834
UNDEPRESSED	η	0.745	0.728

	2.45 GHz	5.8 GHz
RF POWER	70 kW	70 kW
BEAM POWER	93.96 kW	96.15 kW
VOLTAGE	42 kV	42 kV
CURRENT	2.24 AMPS	2.29 AMPS
PERVEANCE x 10^{-6}	0.260	0.266
COLLECTOR RECOVERY	11.6 kW	12.20 kW
NET BEAM POWER	82.3 kW	83.95 kW
RF EFFICIENCY	85.0%	83.38%
SOLENOID POWER	1.4 kW	3.5 kW
NET EFFICIENCY	83.63%	80.04%

Approximate Scaling Considerations for High Power SPS Klystron

SPS-1881



Klystron Mass Estimate

ITEM	2.45 GHz (REF. 2)		5.8 GHz		
	WEIGHT (kg)	POWER (kW)	SCALE FACTOR	WEIGHT (kg)	POWER (kW)
SOLENOID (300°C)					
1000 GAUSS	20	1.4-1.6			
3-in ID, 4-½ in OD					
16.5 in LONG					
2,370 GAUSS			1.2	8.34	
1-3/4 in ID, 3-½ in OD			1.1		3.5
6.96 in LONG					
BODY					
16.5-in LONG	10				
6.96-in LONG				6.5	
COLLECTOR	7			5.4	
RADIATOR					
BODY AT 300°C	9.5		1.2	11.4	
COLLECTOR AT 500°C	4.9		1.2	5.9	
TOTAL	51.4 kg			37.5 kg	

END-END EFFICIENCY DEGRADATION FACTORS

To arrive at a dollar revenue from the grid, it is necessary to obtain a parametric expression with frequency of the various efficiency degradation factors. The attached table shows this variation, combining in the best aluminum all the frequency independent factors as well. These are listed in detail in the rectenna optimization section.

A cursory analysis was also made of the difference of integrated attenuation factors for 2 different sites within U.S., Seattle and El Paso. With unverified assumptions of how often an SPS beam would actually be in a cloud or rain path on a cloudy and rainy day, it was surprising to see the relatively small difference between the 2 sites; even at 5.8 GHz it was only 1.7%. These calculations require further refinement.

SPS

Frequency Variation of End-End Efficiency Factors

SPS-1896

FREQUENCY	KLYSTRON EFFICIENCY η_K	ATMOSPHERIC LOSS FACTOR η_A	RF-DC RECTIFICATION FACTOR R	ANTENNA BEAM EFFICIENCY η_B	AVERAGED RECTENNA η_R	P_{DC} OUTPUT TO GRID
2.0	85.5%	0.985	100%	94.7%	0.8925	0.6187 P_p
2.45	85.0%	0.98	100%	95.0	0.892	0.6135 P_p
5.8	82.4%	0.97	97.5%	96.1	0.891	0.5800 P_p

MPTS ACQUISITION COST

The parametric analysis is based on the assumption that about 5 GW of useful power output is desired, that ionospheric heating effects scale as $1/f^2$, that a 10 dB taper will provide adequately low sidelobes, and that the thermal dissipation at the spacetenna center will remain the same. This means that the spacetenna remains at a 1 km diameter and the rectenna area is reduced by $1/f^2$, and beamwidth by $1/f$. The number of subarrays for equivalent dimensional errors must scale with f^2. The rf power level, to first order, remains constant.

The spacetenna costs, based on modified NASA-JSC numbers, with \$60 kg transportation cost, are derived as in the attached table.

The rectenna costs are based on a fixed number of dipoles per wavelength and the optimized rectenna area derived previously. Dipole and diode costs are taken @ 3¢ independent of frequency.

The results indicate that, although total acquisition cost is minimum at just below 4 GHz, the net return based on the stated assumptions optimizes between 2 and 2.5 GHz. The main valu(of the more intense (130 mw/cm^2 on axis density) 5.8 GHz design is a reduced rectenna size, at an increased technical risk, and greater environmental impact.

MPTS Acquisition Cost Parameters

SPS-1891

- RECTENNA COST AND DIMENSIONAL PARAMETERS

FREQUENCY GHz	DIPOLES PER M2	OPTIMIZED RECTENNA AREA x 10^6	% OF AREA TO 1ST NULL	DIPOLE AND DIODE COST @ 0.03, $ M	LAND AND SUPPORT @ $11.07 PER M2	TOTAL COST $ B	COST PER M2, $	RECTENNA RADIUS, km
2.0	124	114	60	424	1,334 M	1.758	14.74	5.534
2.45	185	80	62	444	936	1.380	16.62	5.082
5.8	1,036	16.2	69	503	190	0.693	42.15	2.271

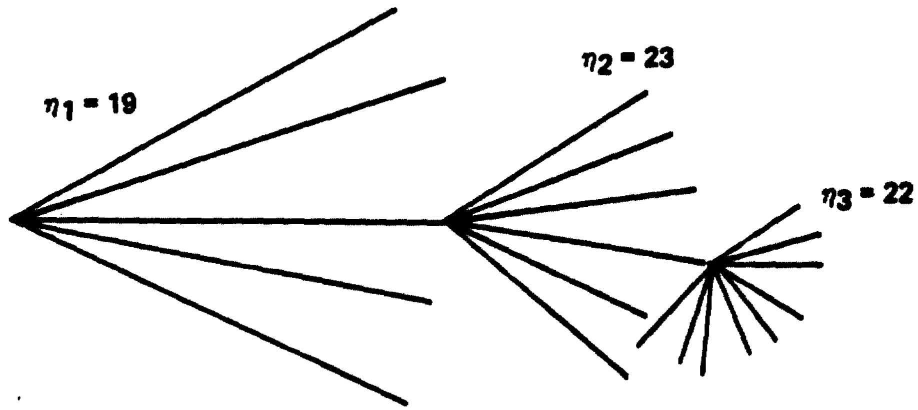
- SPACETENNA COST FACTORS

FREQUENCY GHz	KLYSTRON COST $ M	ARRAY COST M^2 ($)	KLYSTRON POWER kW	KLYSTRON EFFICIENCY (%)	NO. OF SUB-ARRAYS	SPACETENNA COST $ B
2.0	530	1303	100	85.5	4,440	1.245
2.45	525	1506	70	86	6,938	1.312
5.8	515	2078	35	81.5-82.5	38,883	2.174

RETRODIRECTIVE PHASE CONTROL SYSTEM

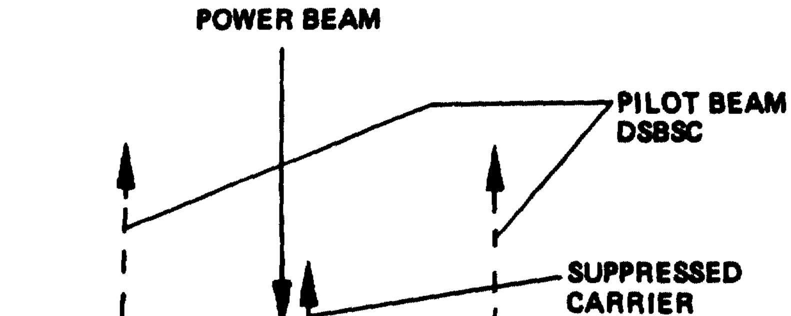
The basic central phasing system shown coherently receives the uplink and derives from it a signal to drive the transmitters in the central subarray (A_O) shown at left. This same phase reference signal (ϕ_O) is used to conjugate (phase for retrodirection) the signal which has been received at the K^{th} subarray and sent down the connecting transmission line. The conjugated signal returns over the same line resulting in cancellation of line length effects. By the use of an N-node reference proliferation system, it is possible to avoid having to run all the transmission lines back to the central array. It is necessary however to regenerate the phase reference ϕ_O. A number of different circuits have been identified that can perform the receiving, conjugating, and phase reference regeneration functions. The preferred concept is to use 2 pilot frequencies separated by 200 MHz which avoids phase ambiguity. Three pilot antennas are proposed to permit the shifting of the apparent pilot phase center or position and hence compensate for beam motion due to propagation effects.

Features of Baseline Retrodirective Phase Control

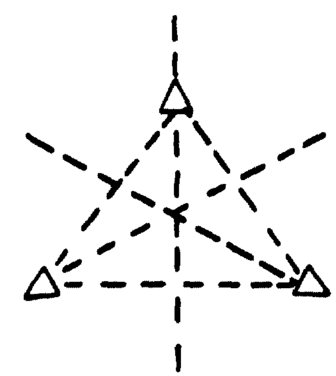


9614 SUBARRAYS

THREE NODE PHASE DISTRIBUTION SYSTEM FOR ERROR CONTROL



DOUBLE SIDEBAND SUPPRESSED CARRIER
AM MODULATED PILOT BEAM

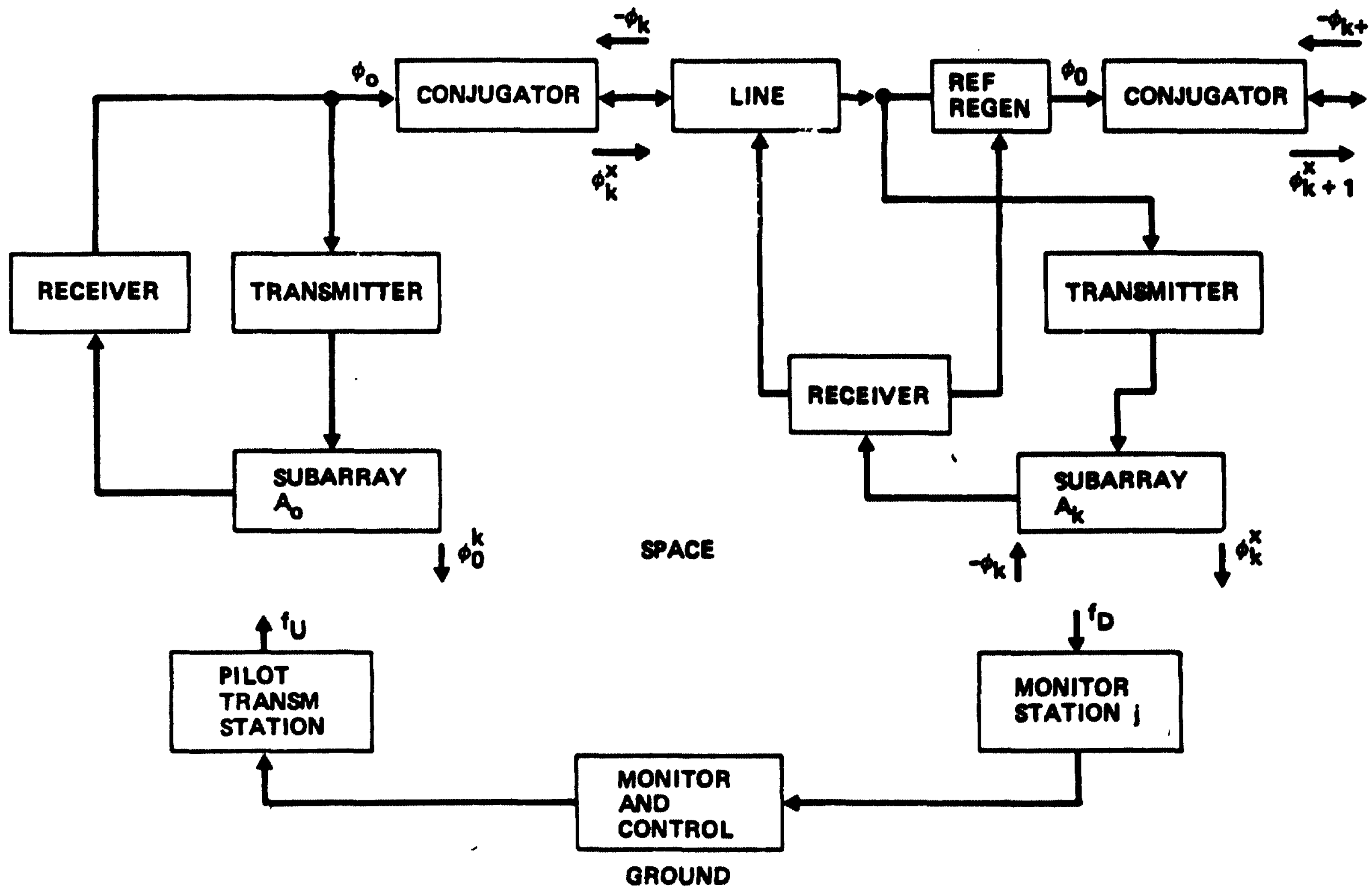


3-PILOT ANTENNA TRANSMITTER

Retrodirective Phase Control System

BOEING

SPS-1874



SPS Phase System Comparisons

	PILOT BEAM	PHASE CONJUGATION	REF. DISTRIBUTION	FEATURES
JPL	DUAL DSBSC	EXACT @ IF DIFFERENCE FREQ.	SERIES TREE CENTRAL PAHSING	PLL MINIMUM CABLE
LIN COM	SINGLE	APPROX. @ RF CORRECTED BY PHASE SHIFTER	SERIES STRING CENTRAL PHASING	MONOPULSE COMPUTER
BOEING/ G.E.	DUAL DSBSC (3-PILOT)	EXACT @ IF DIFFERENCE FREQ.	3 LAYER TREE CENTRAL PHASING	SYSTEMATIC PROPAGATION ERRORS TAKEN OUT

Phase Control Studies

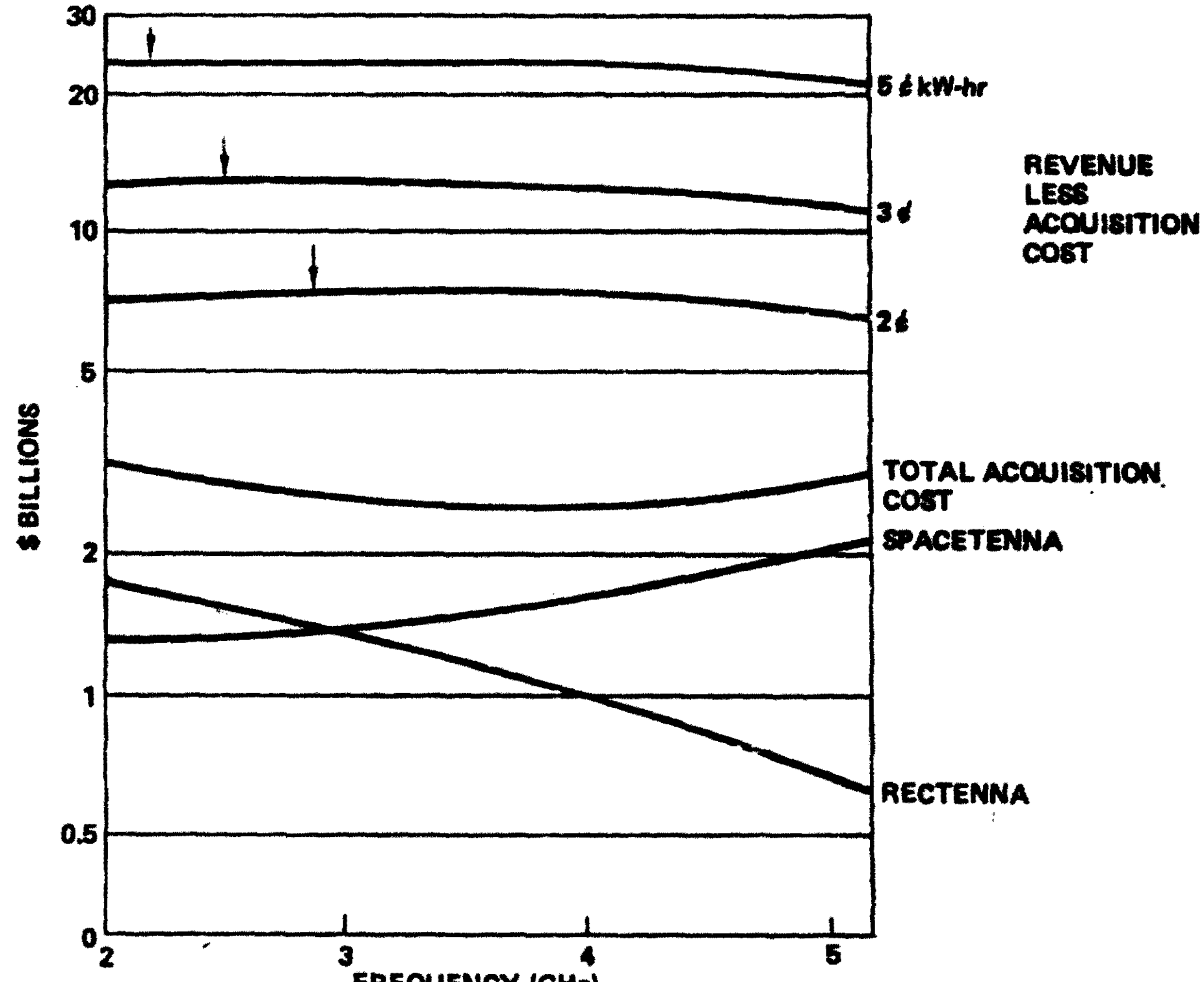
- ALTERNATE APPROACHES
  - RETRODIRECTIVE SYSTEM
  - GROUND COMMANDED SYSTEM

- ERROR BUDGET ESTIMATION
  - COMPONENT EFFECTS
  - PROPAGATION MEDIUM
  - ERROR MINIMIZATION

Atmospheric Attenuation Factors

	GHz	SEATTLE	EL PASO
INTEGRATED	2.45	0.985	0.985
ATMOSPHERIC FACTOR	5.8	0.959	0.976

Acquisition Cost and Revenue Variation With Frequency



5 GW NOMINAL DC 1 KM SPACETENNA 10 dB GAUSSIAN TAPER

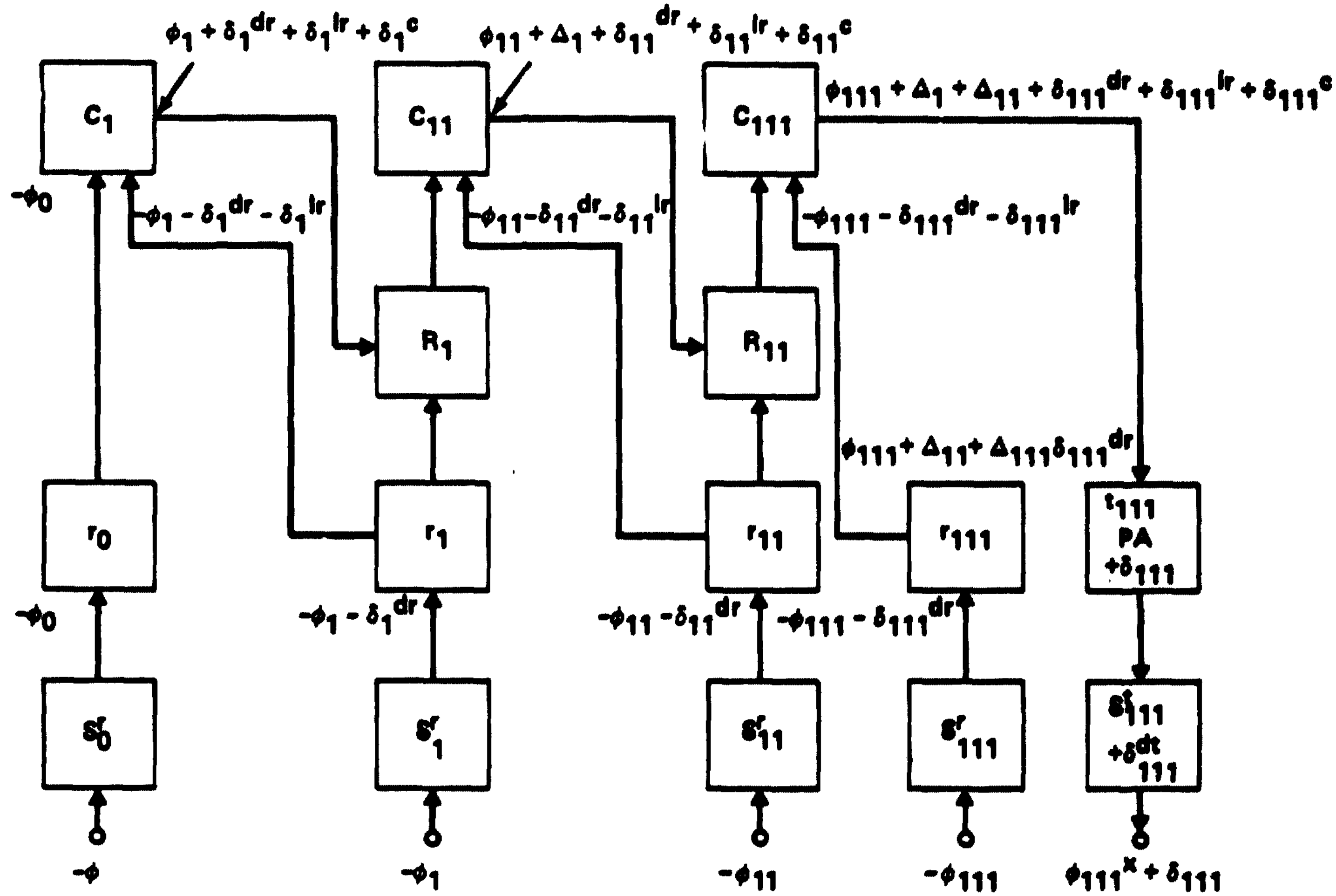
PHASE ERROR BUILDUP IN THREE LAYER REFERENCE DISTRIBUTION SYSTEM

The proposed phase control system utilizes three layers as illustrated. 95% of the subarrays are at the third level. Errors due to conjugators, transmission line reflections, diplexer and transmitter add in the distribution system as indicated by the RSS formula. Using the following estimated values of component phase errors

Conjugator ($g_c = 0.6^{\circ}$)
Transmission Line ($g_l = 2.5^{\circ}$)
Diplexer ($g_d = 1.8^{\circ}$)
Transmitter ($g_p = 1.6^{\circ}$)

The cumulative RSS total is 7.0°

Phase Error Buildup in Three Layer Reference Distribution System



$$\text{RSS ERROR} = \sqrt{3\delta_c^2 + 6\,\delta_f^2 + 2\,\delta_d^2 + \delta_p^2}$$

REDUCTION OF POINTING ERRORS BY THREE PILOT STATION SYSTEM

The pointing errors listed are not correctable by the phase control system. They can produce power beam displacement from the rectenna. A ground system of 3 pilot stations is proposed to correct for these beam position errors should they develop to be of sufficient magnitude. This system utilizes monitor antennas placed about the rectenna. Displacement of the beam results in a corrective counter displacement of the pilot phase center (or effective position). The latter is accomplished by relative amplitude and phase control of the radiation from the 3 separated pilot antennas.

Reduction of Pointing Errors by Three Pilot Station System

POINTING ERRORS (deg)			
SOURCE		1 PILOT STATION	3 PILOT STATION
DOPPLER ($\dot{i} \doteq 2.2$ deg, $\dot{r}^j_m = 13.6$ m/s, $2\,\dot{f}_{Dop} = 112$ Hz)		1.43×10^{-6}	7.15×10^{-8}
ABERRATION ($Z^j_m = 100$ m/s)		19.3×10^{-6}	9.65×10^{-8}
IONOSPHERIC DIFFERENTIAL (0.1 deg 1 WAY REFRACTION)		2.35×10^{-3}	1.17×10^{-4}
ATMOSPHERIC DIFFERENTIAL (0.3 deg 1 WAY REFRACTION, 2% IRREGULARITY)		6.00×10^{-3}	3.00×10^{-4}
POINTING ERROR (deg)	PEAK	8.35×10^{-3}	4.175×10^{-4}
	RSS	6.44×10^{-3}	3.221×10^{-4}

Random Errors for 3 Layers Phase Distribution

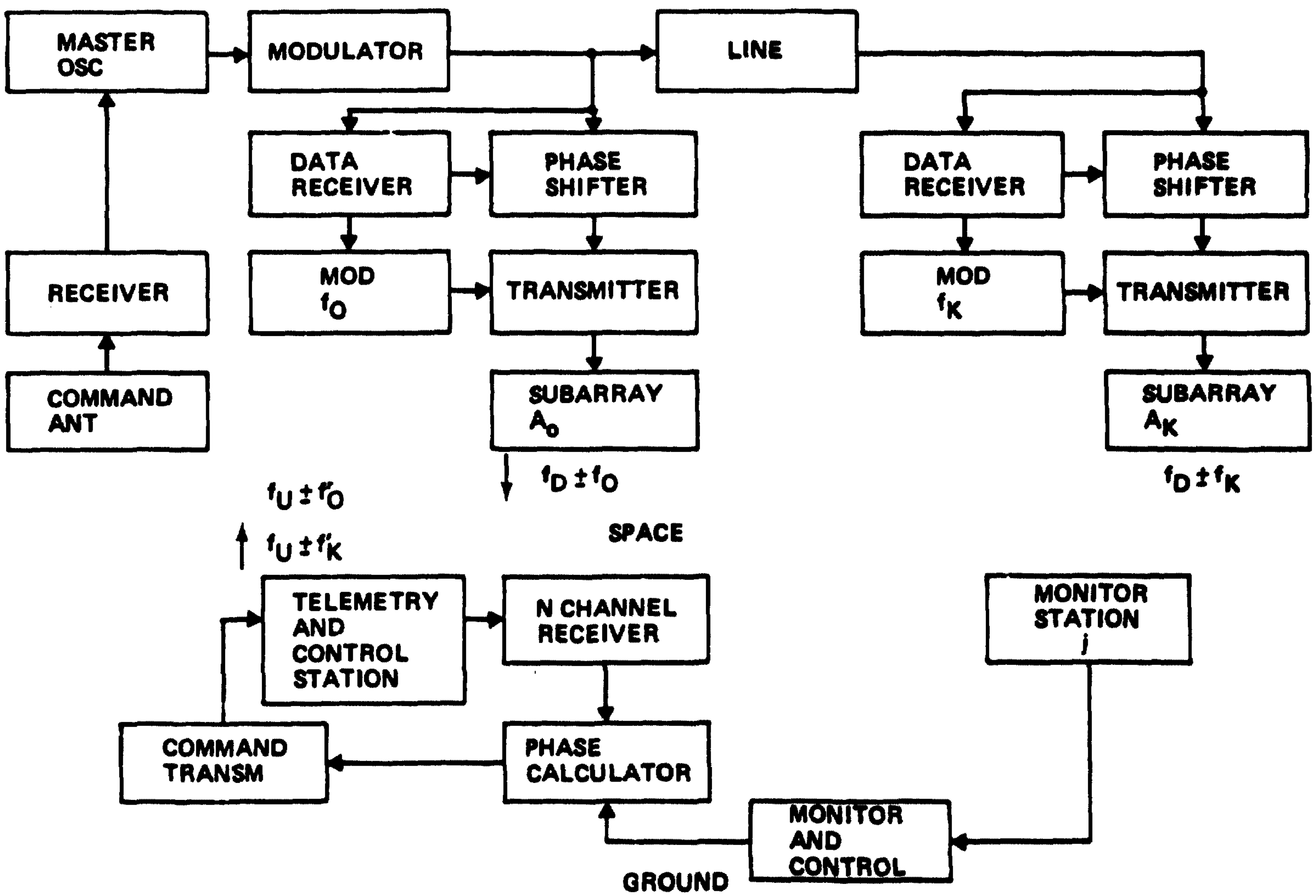
PHASE ERRORS (deg)	
SOURCE	DEG
PHASE JITTER	1.13
TRANSMITTER NOISE (c/n = 30 dB)	0.36
CONJUGATORS (δ_c = 0.6 deg)	1.04
LINES (δ_1 = 2.54 deg)	6.22
DIPLEXERS (δ_d = 1.81 deg)	2.56
TRANSMITTERS (δ_p = 1.6 deg)	1.60
DIFFERENTIAL DOPPLER (V_d = 6.25 m/s)	0.18
	PEAK:13.09 RSS: 7.09

PHASE ERROR CAUSED LOSS: 1.53%

GROUND COMMANDED PHASE CONTROL SYSTEM

It is possible to transfer much of the phase control system complexity to the ground. The ground commanded alternate system shown here utilizes a unique identifying tone that is modulated on the output of each subarray. From this low level signal, the relative phase of a subarray's contribution to the total field is derived. Phase shifter adjustments of the radiated carrier phase are then commanded from the ground. To hold the number of channels required to a reasonable value a 100 slot time division multiplex scheme is proposed. With under 100 tones, the sampling period of this system would be ≈ 25 sec unless an interlaced technique were used which could possibly reduce it to 0.25 sec. Convergence of this type of phase control system needs to be studied with an appropriate structural dynamic model.

Ground Commanded Phase Control System



PROJECTED CONVERSION EFFICIENCY OF OPTIMIZED RECTENNA ELEMENT

The data for modeling the rf-dc conversion efficiency is based on experimental work by Raytheon and JPL with projections of improved diode and circuit performance made by NASA-JSC. The resulting equation is:

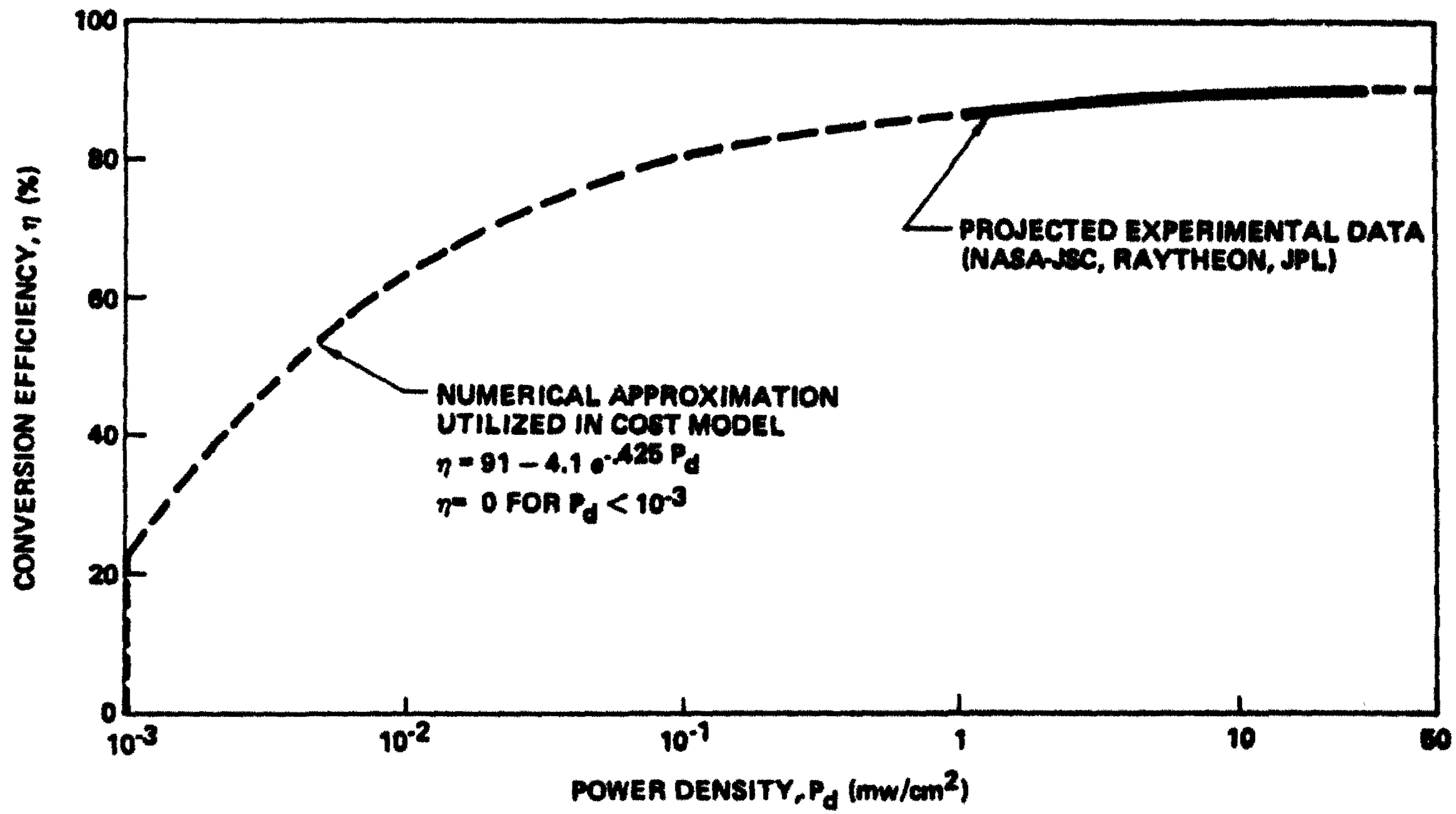
$$\eta = 91 - 4.1e^{-.415\, P_d}, \text{ and}$$
$$\eta = 0 \text{ for } P_d < 10^{-3}, \text{ where}$$

η is in percent and P_d is incident power density in mW/cm^2.

The conversion efficiency equation was incorporated into the modified "Bigmain" program in order to calculate the average rectenna conversion efficiency as a function of rectenna area.

Projected Conversion Efficiency of Optimized Rectenna Element

SPS-1893



Rectenna Analysis Tasks

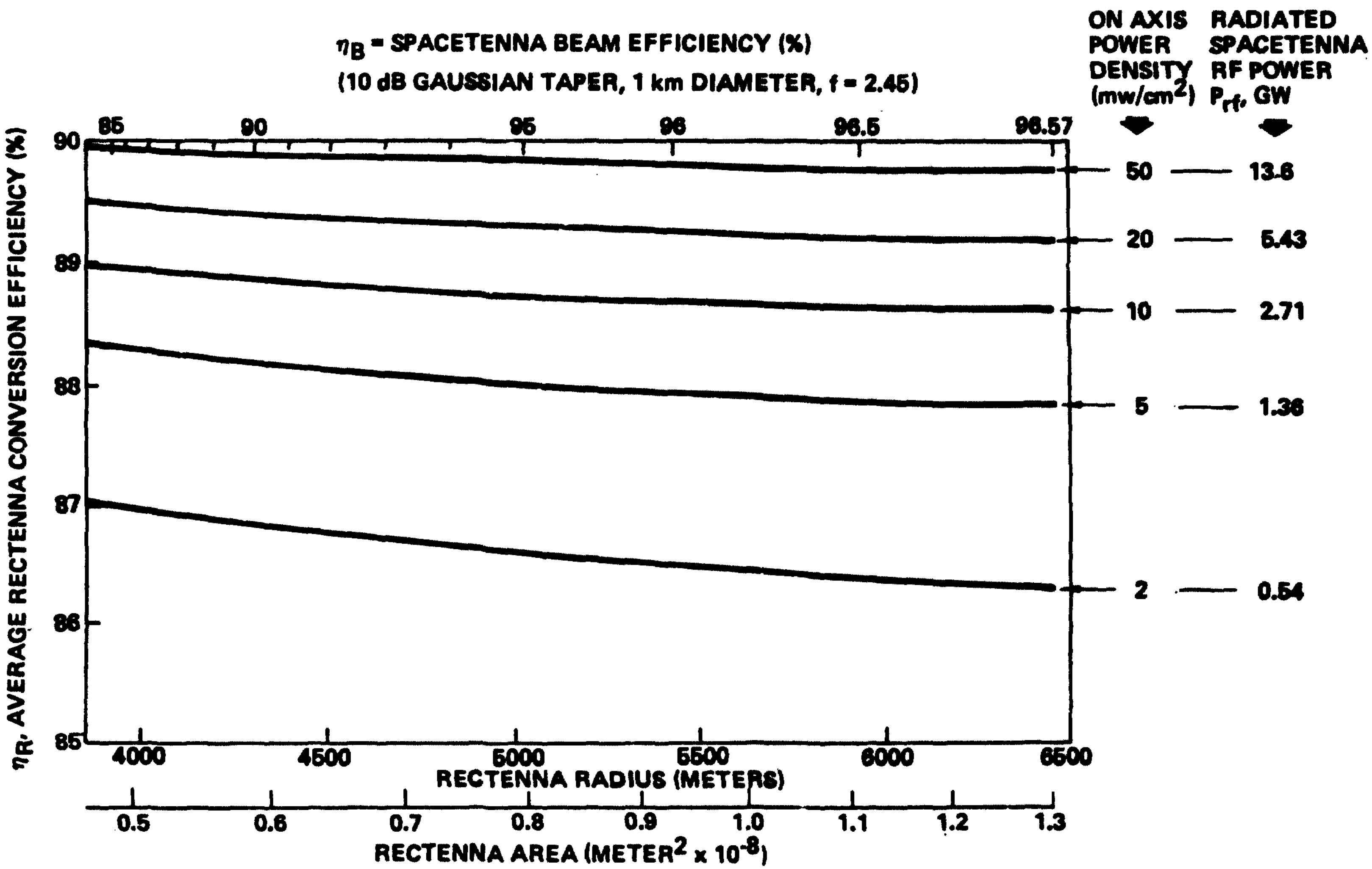
- IMPROVED RECTENNA EFFICIENCY MODEL
- RECTENNA SIZE OPTIMIZATION
- EDGE DIFFRACTION EFFECTS
- HARMONIC GENERATION
- MECHANICAL DESIGN
- ALTERNATE APPROACHES

PROJECTED AVERAGE RECTENNA CONVERSION EFFICIENCY AS A FUNCTION OF RECTENNA AREA AND ON AXIS POWER DENSITY

The family of curves shows the average conversion efficiency as a function of rectenna area for the five different values of on axis power densities and their associated spacetenna radiated power. For the baseline 10 dB gaussian taper, the spacetenna beam efficiency has a one-to-one correspondence to the rectenna area as is indicated by the top scale. The averaging process, carried over 50 nominal ring sections of the rectenna, derives an average conversion efficiency

$$\eta_R = \frac{\sum_{o}^{R} \eta P_d A}{\sum_{o}^{R} P_d A}$$ for any desired rectenna radius R.

Projected Average Rectenna Conversion Efficiency Versus Rectenna Area



RECTENNA OPTIMIZATION

The optimum rectenna size is selected on the basis of matching the present value of 30 year revenue per spare meter with the acquisition cost per square meter. This process is shown for 2.45 GHz and 5.8 GHz. for various values of revenue. For the selected parameters, the rectenna area is only 62% of the area out to the first null.

The values of the various efficiency degradation factors contributing to the net dc power to the ground network are treated parametrically with frequency as follows:

$$P_{dc} = [\eta_d\, \eta_i\, \eta_g\, \eta_w][\eta_k\, \eta_B\, \eta_R\, \eta_A] P = .8692\ K(f) P_{dc}$$

space photovoltaic power

distribution — atmospheric

inter & intra subarray effic — averaged rectification

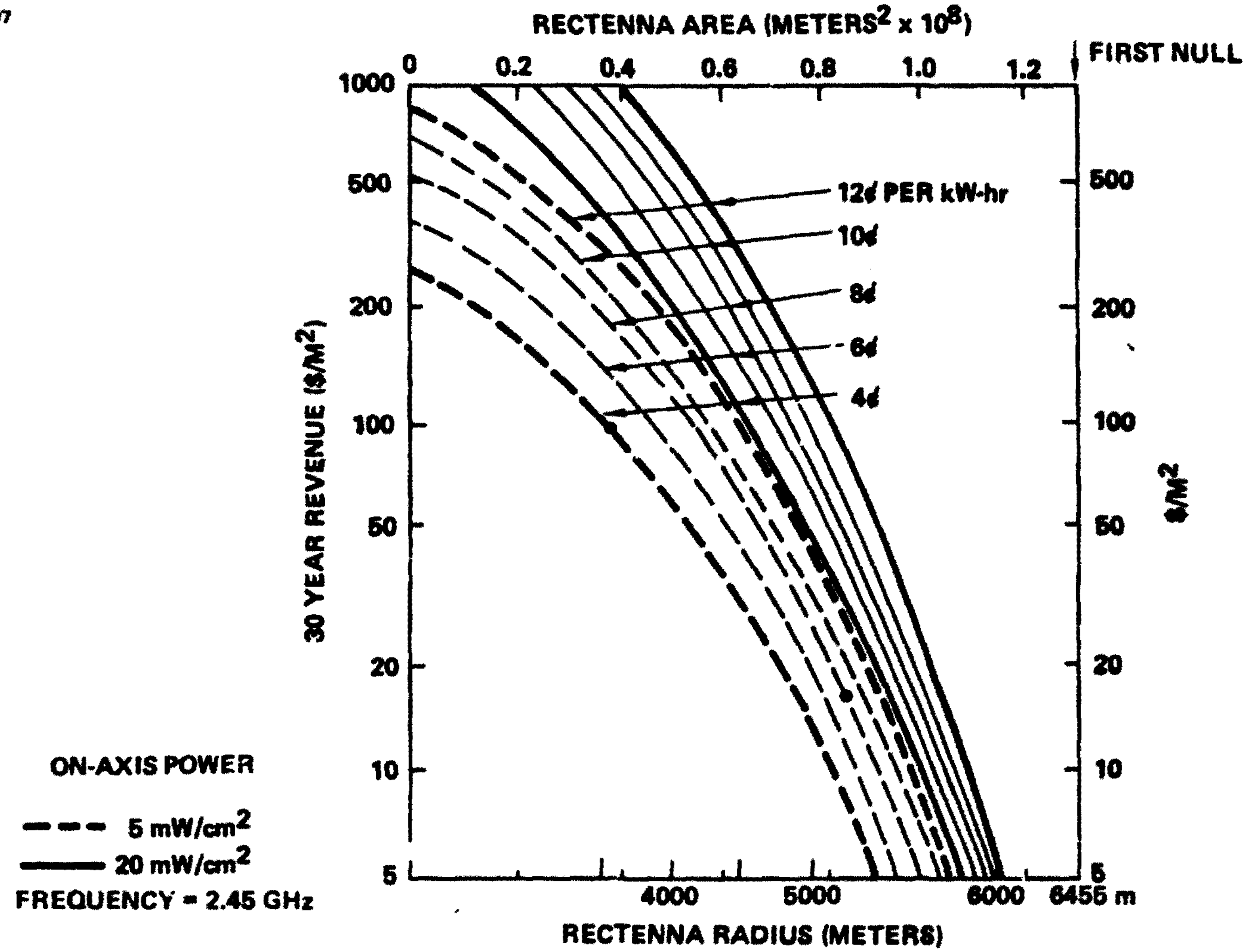
grid distribution — antenna-beam efficiency

waveguide loss — Klystron efficiency

The frequency varying portion is given in the attached table, from which the useful dc power output can be plotted as a function of rectenna radius. From this plot, the revenue per m^2 can be obtained and equated to the rectenna cost per square meter.

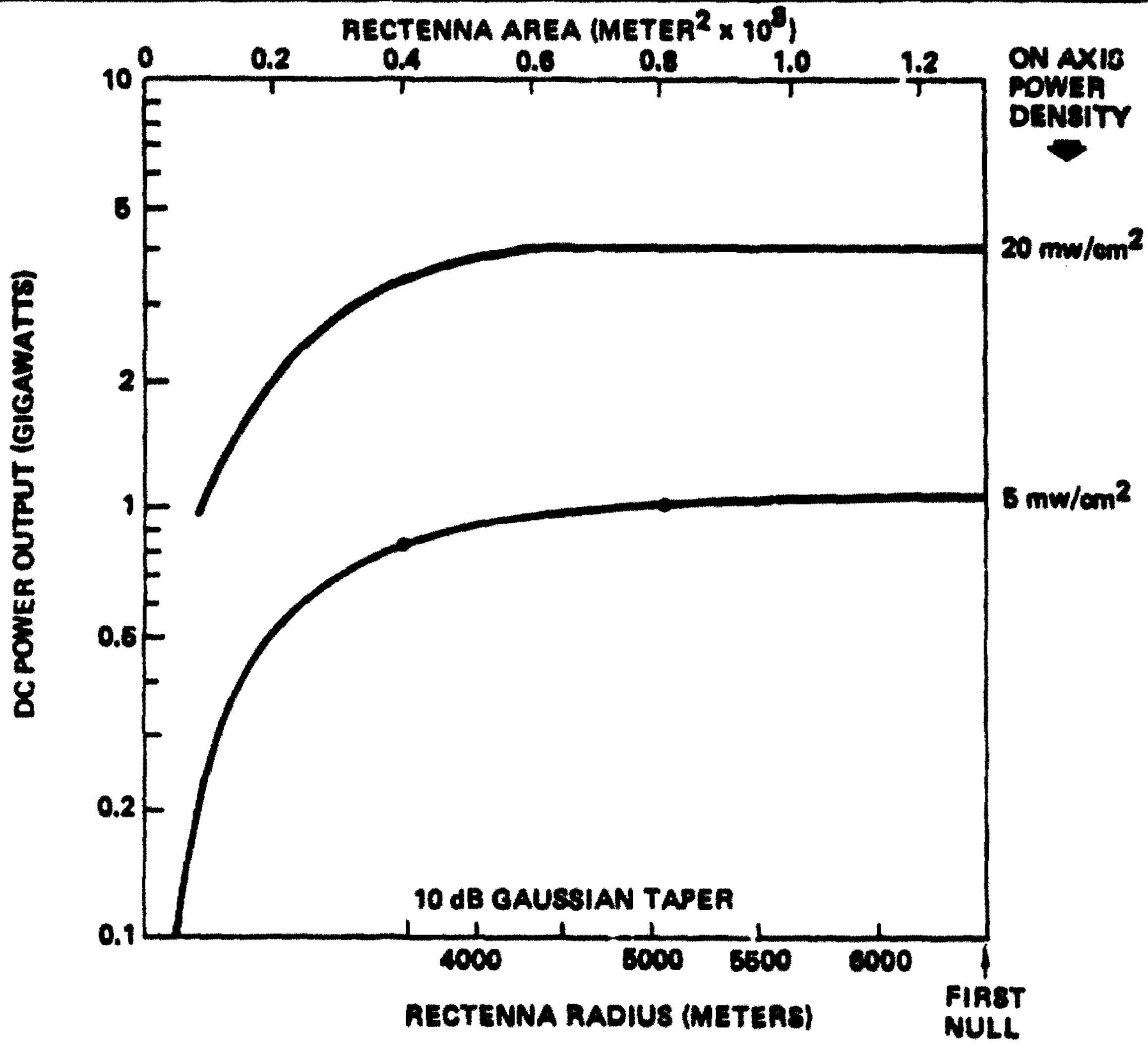
Optimum Rectenna Dimension for Given Rectenna Cost

SPS-1897



Power Output Versus Rectenna Size

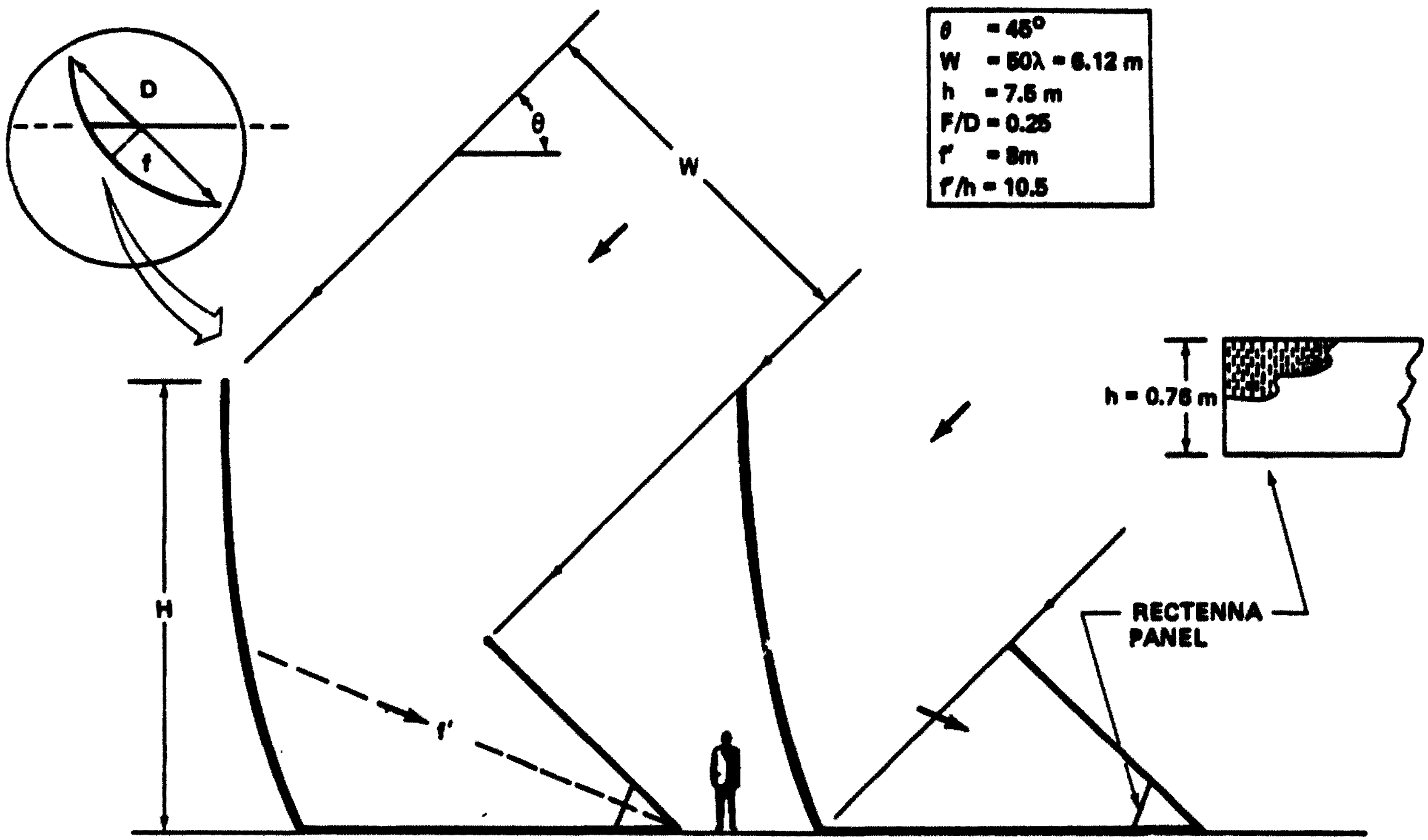
SPS-1896



HOGLINE RECTENNA CONFIGURATION

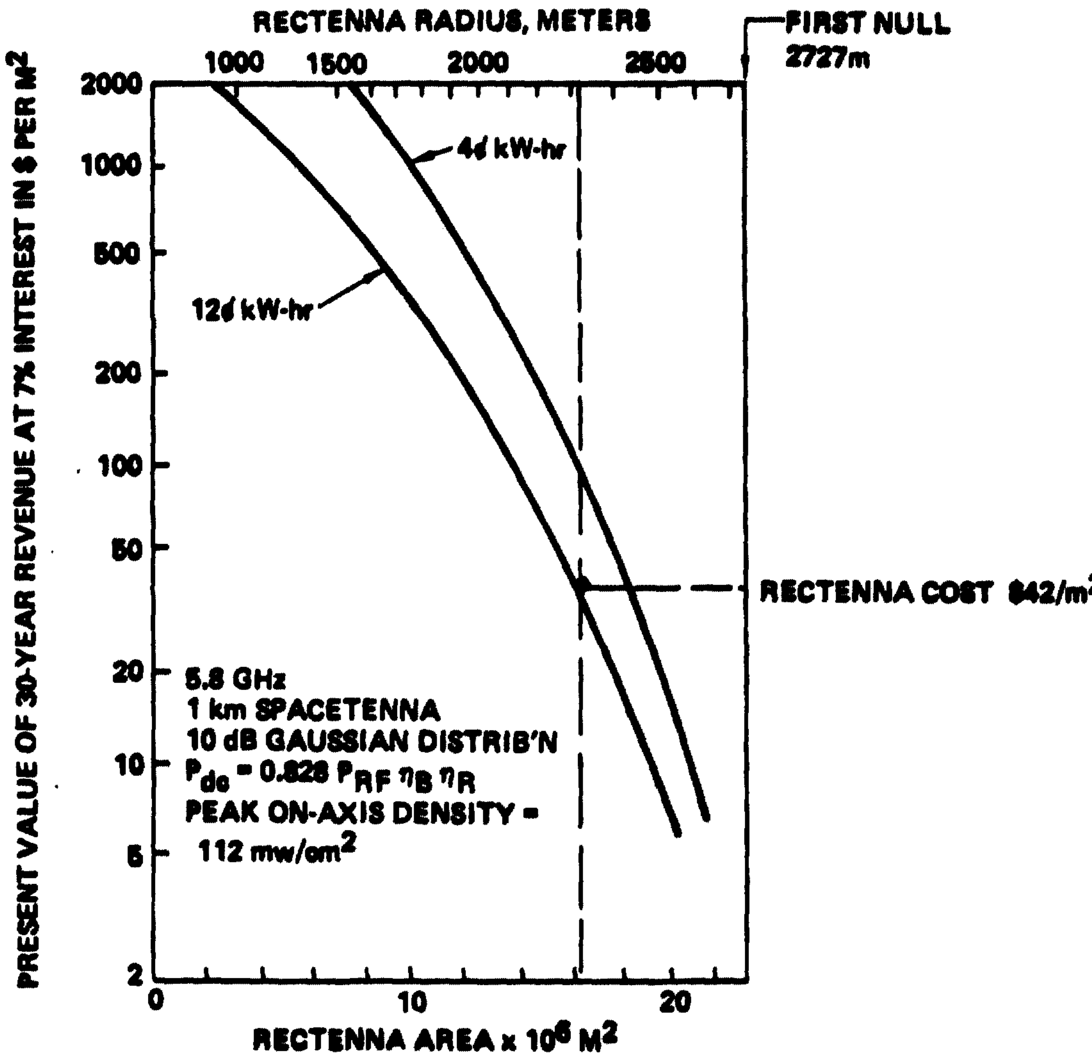
A suggested hogline rectenna configuration is shown. An offset cylindrical parabola is fed by a trough or line horn. A dipole/diode panel of 12 rows is located in this example where the incident power density has been increased by a factor of 8.7. Spillage, blockage and taper losses can all be made negligible with this configuration. The greater power density permits higher rectification efficiency, or lower power demonstrations at the same efficiency. The dipole and diodes are protected from weather and EMP and there are significantly fewer of them. Size is a compromise between the degree of concentration desired, edge diffraction losses and the latitude stability of the SPS satellite.

Hogline Rectenna Configuration for SPS



Rectenna Size Optimization

SPS-1808

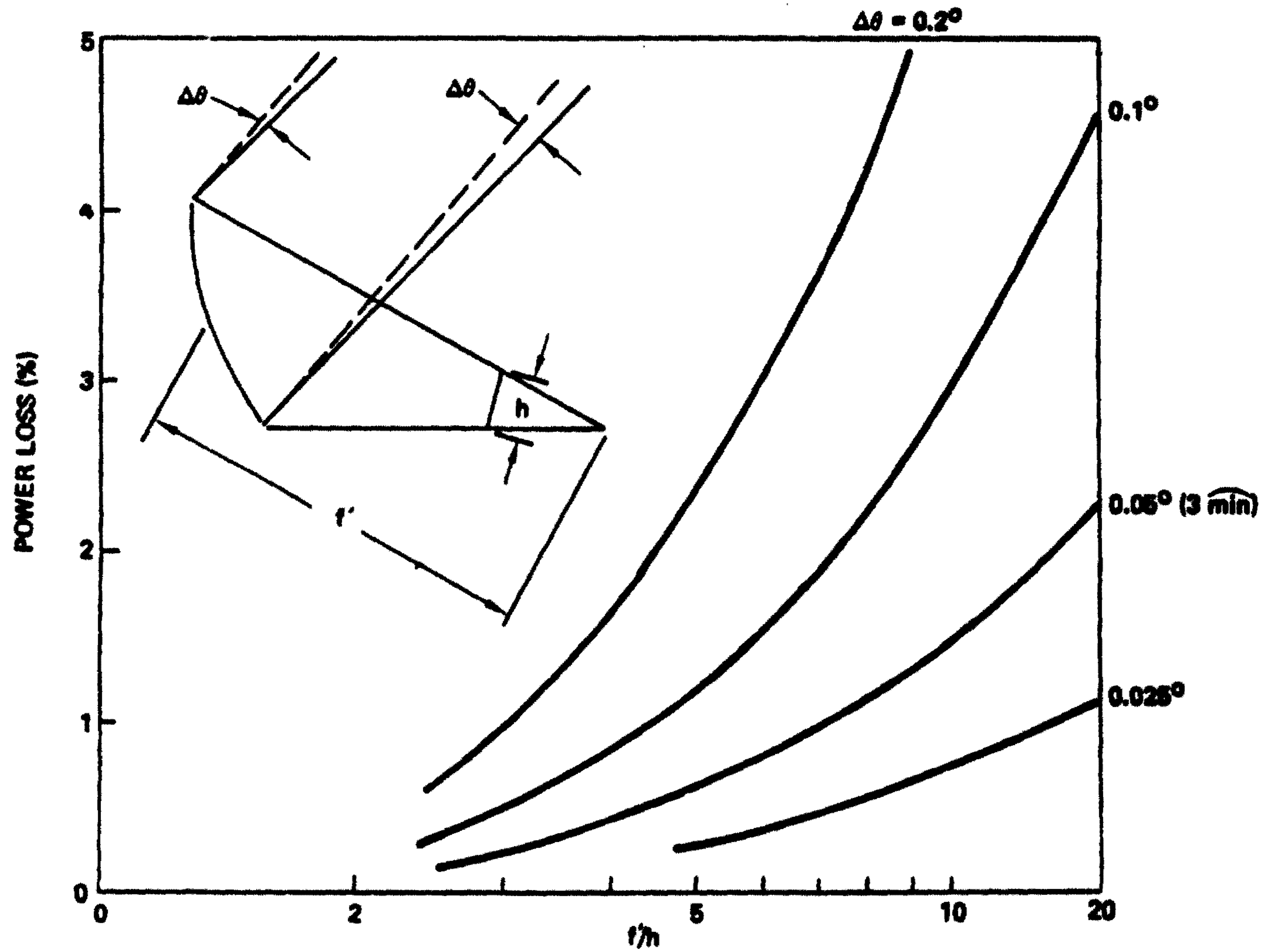


HOGLINE ANTENNA POINTING CONSIDERATIONS

Due to the vertical plane pattern directivity, variations in satellite latitude will induce pointing losses. These losses are estimated by considering the fraction of the converging beam that misses the diode/dipole panel. The losses are a function of the relative panel size (f/h). The geometry shown in the previous figure would require a stability of 0.05° to hold losses to 1.5%.

Looked at from the simple pattern of the 50λ aperture loss point of view, it is estimated that an error of 0.08° could be tolerated for a 1.5% power loss.

Pointing Power Loss Versus Rectenna Panel Size



MPTS Follow-On Topics

SPS-1868

- ARRAY ANALYSIS
  - PHASE CONTROL SYSTEM DEFINITION
  - SPS PRECURSOR SPACETENNA CONFIGURATION AND PATTERN ANALYSIS
  - GRATING LOBE MAGNITUDE ASSESSMENT
  - DYNAMICS OF GROUND CONTROLLED PHASE COMPENSATION
  - SPACETENNA BACK RADIATION ANALYSIS
  - FAR SIDELOBE ROLL-OFF
- COMPUTER PROGRAM EXTENSION
  - MODEL ARRAY AT KLYSTRON LEVEL
  - EFFECT OF SUBARRAY OFFSET AND INTRA-SUBARRAY ERRORS
  - SYSTEMATIC AMPLITUDE AND PHASE ERROR EFFECTS
  - EXTEND CAPABILITY TO OTHER THAN PRINCIPAL PLANES
- RF TRANSMITTER
  - KLYSTRON FAILURE MODE ANALYSIS AND PROTECTION
  - DETAILED TUBE SPECIFICATION TO MANUFACTURER
  - TEST PLAN FOR GROUND AND SPS PRECURSOR TESTS
  - HARMONIC SUPPRESSION FEASIBILITY
- COMPONENT TEST PROGRAM
  - COMPOSITE WAVEGUIDE MICROWAVE POWER TEST
  - CROSSGUIDE DESIGN AND ANALYSIS, DIPLEXER DESIGN
- RECTENNA COLLECTION EFFICIENCY OPTIMIZATION
  - MULTIPLE DIPOLE PER DIODE CONFIGURATION
  - HOGLINE ANTENNA
  - ELEMENT IMPEDANCE GAIN VERSUS SPACING
- SYSTEM STUDIES
  - AC GENERATION AND DITHER FEASIBILITY
  - TURN ON/OFF PROCEDURES
  - FLOPPY REFLECTOR LEO TEST

SPS-1850

Development Requirements for Initial SPS Commercialization

D. L. Gregory

PAST AND FUTURE ELECTRIC POWER ALTERNATIVES USE PILOT PLANTS

Today's nuclear power reactors are the culmination of a series of developmental reactors which preceded them. In the ground solar power program the thermal engine "tower top" systems are to be preceded by the five megawatt thermal test facility at Alburquerque, New Mexico and by a ten magwatt (electric) output pilot plant at Barstow, California. These will lead to a commercial unit to be rated at 100 megawatts. Thus the pilot plant is 1/10 of the size of the eventual commercial unit. Ground solar power plants of the tower top type have been investigated as large as 1,000 megawatts. The future of the liquid metal fast breeder reactor is uncertain, however, for that program a series of developmental systems is envisioned. The total fusion power development program is as yet undefined, however, a series of experimental reactors and a demonstration reactor are envisioned.

Past and Future Electric Power Alternatives Use "Pilot" Plants

SPS-1859

(1976 DATA)

PAST:	PROGRESSION TO THE COMMERCIAL LIGHT WATER REACTOR	
	• EXPERIMENTAL REACTORS,	
	• DEMO REACTOR,	SHIPPINGPORT, PA
	• PROTOTYPE PLANT,	OYSTER CREEK, NJ
FUTURE:	GROUND SOLAR POWER (TOWER TOP TYPE)	
	• TEST FACILITY	(5 MW_t) ALBUQUERQUE, NM
	• PILOT PLANT	(10 MW_e) BARSTOW, CA
	• COMMERCIAL DEMONSTRATOR	(100 MW_e)
	BREEDER REACTOR (LIQUID METAL FAST BREEDER)	
	• FAST FLUX TEST FACILITY	400 MW_t
	• CLINCH RIVER BREEDER	380 MW_e
	• PROTOTYPE COMMERCIAL BREEDER	1,200 MW_e
	FUSION (MAGNETIC)	
	• EXPERIMENTAL POWER REACTOR 1 (20-50 MW_e)	
	• EXPERIMENTAL POWER REACTOR 2 (> 100 MW_e)	
	• DEMONSTRATION REACTOR (> 500 MW_e)	

PRECURSOR SOLAR POWER SATELLITE

Many scenarios can be postulated for the development sequence leading to full size solar power satellites. The most agressive of these would advance directly from small scale tests with the shuttle to a full size unit in a very rapid development program. At the other extreme is a "long chain" development beginning with shuttle and ground tests, then small scale power modules, then pilot plants, then commercial demonstrators and finally full size power satellites. Some development scenarios for SPS include participation in SPS funding by utility firms. In order to achieve this participation it may be necessary to demonstrate commercial viability of SPS to these utility firms. To do so a precursor satellite may be required. A precursor satellite would be a subscale unit put on line in geosyncrhonous orbit before the full size satellite. It would operate in the fashion of the full size satellite, that is direct an energy beam to earth to a ground rectenna. The demonstration/operational period might be six months to 2 years and during this time the majority of the major elements of a full sized solar power satellite would be exercized. From this operation, analyses could be performed which would indicate eventual full commercial viability for the SPS. It is also possible to construct scenarios in which the full size satellite is preceded by a precursor which serves only as part of the development leading to the full sized unit. Other scenarios go directly to the full size SPS without the use of the precursor.

Precursor Solar Power Satellite

SPS-1851

- MANY SPS DEVELOPMENT SCENARIOS CAN BE IDENTIFIED.
- SOME OF THESE INCLUDE FUNDING PARTICIPATION BY UTILITY FIRMS.
- FOR THESE SCENARIOS, A PRECURSOR SPS COULD BE USED TO ESTABLISH EVENTUAL SPS COMMERCIAL VIABILITY.
- IN OTHER SCENARIOS, A PRECURSOR SPS WOULD SERVE PRIMARILY AS A DEVELOPMENT ITEM.
- SOME SPS SCENARIOS WOULD NOT INVOLVE A PRECURSOR.

PRECURSOR SPS LIMITATIONS

Due to its small size and the scaling effects related to microwave transmission a precursor or subscale satellite cannot develop more than perhaps 1 or 2 milliwatts per square centimeter in its central beam. This is well below the quantity proably required to excite ionospheric heating. Also because of its small size and the associated development costs it is probable that a precursor unit cannot be procurred for less than approximately $100,000 per milowatt; as a consequence the electric energy produced during the short term of operation would be quite expensive.

Precursor SPS Limitations

THE PRECURSOR SPS CANNOT:

- PRODUCT CENTRAL BEAM STRENGTHS WHICH EXCITE IONOSPHERIC HEATING
- PRODUCE "ECONOMIC" ELECTRIC ENERGY (FORTUNATE TO OBTAIN $100,000/kW)

STUDY ASSUMPTION: PRECURSOR SPS

For this study we have assumed that a demonstration of eventual commercial viability will be the goal of the precursor solar power satellite. In addition to a demonstration of eventual commercial viability the operation of a subscale unit will accomplish the secondary goals identified here.

Study Assumptions: Precursor SPS

- <u>A DEMONSTRATION OF EVENTUAL COMMERCIAL VIABILITY IS THE PRIMARY GOAL OF A PRECURSOR SPS (PSPS)</u>

- SUCH A DEMONSTRATION WILL ACCOMPLISH SECONDARY GOALS:
  - SHOW THAT THE SPS CONCEPT "WORKS"
  - VERIFY DESIGN ASSUMPTIONS
  - REFINE DATA ON ENVIRONMENTAL EFFECTS ON SPS
    - IONOSPHERIC EFFECTS ON BEAM
    - PLASMA PHENOMENA
    - SPACECRAFT CHARGING
  - ESTABLISH OPERATIONAL EXPERIENCE
    - HIGH LAUNCH RATES
    - ON-ORBIT CONSTRUCTION (HIGH "THROUGHPUT")
    - OPERATION AND MAINTENANCE
    - COLLISION AVOIDANCE

COMMERCIAL VIABILITY OF SPS

In order to develop the concept for a precursor solar power satellite which will demonstrate eventual commercial viability for an SPS program it is necessary to first define commercial viability. The definition here is literally that the product of the SPS be in sufficient demand. In order to be in sufficient demand its price, expressed as dollars per kilowatt hours, must be equal or below some criterion. Generally for SPS this criterion may be viewed as approximately 10 cent per kilowatt hour. However, our studies have identified satellite programs wherein the cost is probably no more than 5 cent per kilowatt hour. The price per kilowatt hour of energy produced by any power plant is made up of the annual cost for that plant divided by the kilowatt hours produced by that plant in a year. Hence, good availability, that is the capability of producing a large fraction of the total plant capacity in kilowatt hours per year, is as important as the cost smortization per year.

Commercial Viability of SPS

- COMMERCIAL VIABILITY: A COMMODITY, IN DEMAND AT OR BELOW A GIVEN PRICE, IS PRODUCIBLE AT OR BELOW THAT PRICE
- COMMODITY PRICE FOR ELECTRICAL ENERGY: EXPRESSED AS DOLLARS PER KILOWATT HOUR OF DELIVERED ENERGY = \$/kWh
- PRICE = COST, PLUS FACTORS SUCH AS PROFITS
- COSTS FOR SPS ARE PRIMARILY INITIAL OUTLAYS: SATELLITE + RECTENNA + LAUNCH + CONSTRUCTION
- THE ANNUAL AMOUNT PAID IS (FOR ELECTRIC UTILITIES) EXPRESSED AS INITIAL COSTS TIMES A "CAPITAL CHARGE FACTOR"
- $\$/\text{kWh} = \dfrac{\text{(INITIAL COSTS)} \times \text{(CAPITAL CHARGE FACTOR)}}{\text{kWh PRODUCED IN YEAR}}$
- THUS AVAILABILITY IS AS IMPORTANT AS INITIAL COSTS IN DETERMINING COMMODITY PRICE

$$\text{AVAILABILITY} = \frac{\text{kWh PRODUCED IN YEAR}}{\text{(PLANT CAPACITY)} \times \text{(8,766 HOURS)}}$$

- HENCE A DEMONSTRATION OF THE ACHIEVEMENT OF REQUIRED AVAILABILITY IS AS IMPORTANT AS THE DEMONSTRATION OF ACHIEVEMENT OF REQUIRED COSTS

SPS COST FACTORS

The projections of cost from a precursor satellite to a full sized unit are relatively straightforward. Learning factors can be used to relate initial unit costs to the cost of hardware and mass production.

SPS Cost Factors

SPS-1836

BOEING

ACQUISITION COSTS

• RECTENNA	SITE, SITE PREP., MATERIALS FABRICATION INSTALLATION
• SATELLITE HARDWARE	MATERIAL, FABRICATION
• ORBITAL PLACEMENT	SITES, FACILITIES, FLEET, PROPELLANTS, OPERATIONS
• ORBITAL CONSTRUCTION	CONSTRUCTION BASE, OPERATIONS

OPERATIONS COSTS

• OPERATIONS AND MAINTENANCE	LOGISTICS, SPARES, OPERATIONS

DEVELOPMENT

- SIGNIFICANT IN INVERSE PROPORTION TO THE PROGRAM SIZE

DEMONSTRATION OF COMMERCIAL VIABILITY

In order to show that full size solar power satellites can achieve a high plant factor, that is high percentage of potential use, it will be necessary to demonstrate good availability for the precursor unit. To do this the precursor unit can be operated for a period of time and the availability can be recorded. From this demonstrated availability statistical analyses can be used to compute the probable availability of a full size unit Whenever availability or reliability are calculated it is necessary to express an associated level of confidence. If a statement is made that the expected reliability is 0.95, for example, and the confidence in this factor is 0.9, what we are saying is that there is only one chance in ten that the eventual reliability will be less than 0.95. Even a relatively small precursor satellite will have sufficient components to allow a test of perhaps of one year duration to have sufficent statistical sig nificance. Even with extremely good reliability for such components as klystrom transmitter tubes a SPS program would not have commercial viability unless effective maintenance method can be developed. Maintenance of the precursor satellite should therefore be demonstrated.

AVAILABILITY

- DEMONSTRATE NECESSARY AVAILABILITY (RELIABILITY)

  1) OPERATE A PRECURSOR SPS UNIT FOR A PERIOD AND MEASURE THE AVAILABILITY

  2) USE STATISTICAL ANALYSES TO COMPUTE THE PROBABLE AVAILABILITY OF A FULL SIZE UNIT

- THE LEVEL OF CONFIDENCE IN THIS AVAILABILITY IS INFLUENCED BY:

  - DURATION

    HAS STATISTICAL SIGNIFICANCE: ALSO PROVIDES OPPORTUNITY FOR UNEXPECTED EVENTS

  - SIZE OF UNIT

    THE NUMBER OF COMPONENTS HAS STATISTICAL SIGNIFICANCE: A UNIT WITH 100 KLYSTRONS WOULD REQUIRE A LONGER TEST THAN ONE WITH 1,000 KLYSTRONS (FOR THE SAME CONFIDENCE)

- DEMONSTRATION OF MAINTAINABILITY IS OF OBVIOUS NECESSITY IN SHOWING AVAILABILITY

  MAINTAINABILITY INVOLVES: FAULT DETECTION, SPARES, LOGISTICS, REPLACEMENT, CHECKOUT, ETC.

PERFORMANCE ASSURANCE: SATELLITE POWER GENERATION

Critical parameters for the power generation portion of the SPS are listed here, along with the demonstrations which is probably required for those parameters. For example, solar array efficiency can probably be completely verified on small solar blanket panels. The exact geosynchronous radiation environment, including electrons and protons with broad energy spectra, is probably quite difficult to quantify and even more difficult to exactly simulate. Hence, long term exposure of solar array panels to this environment is probably required in order to have full assurance of eventual SPS operational factors. If a self power transfer from a low assembly orbit to geosynchronous orbit is baselined, and it currently appears that this is the most economic approach, then the precursor demonstration should include exposure of solar array panels in a self power transit. If annealling is to be part of the operational satellite program then it should also be demonstrated in the precursor program using solar array panels which had been degraded as a result of exposure to the actual self power transfer and geosynchronous environment.

Performance Assurance
Satellite Power Generation

PERFORMANCE	DEMONSTRATION
1. SOLAR ARRAY EFFICIENCY ——————%	MEASUREMENTS ON SMALL PANELS $(M)^2$.
2. SOLAR ARRAY RADIATION RESISTANCE —————— —————— ——————	EXPOSURE TO THE GEOSYNCHRONOUS (AND ORBIT TRANSFER, IF LEO ASSEMBLY) ENVIRONMENT FOR AN APPROPRIATE PERIOD (TO INCLUDE SUB-STORMS AND FLARES). POSSIBLE ADDITIONAL CHAMBER TESTS. THE FULL ENVIRONMENT IS DIFFICULT TO MEASURE/ SIMULATE.
3. SOLAR ARRAY ANNEALABILITY	MEASURE ARRAY PERFORMANCE BEFORE AND AFTER ANNEALING (BY THE "OPERATIONAL" METHOD). DAMAGE TO HAVE BEEN CARUSED BY ACTUAL GEO-SYNCHRONOUS ENVIRONMENT (PLUS TRANSFER FROM LEO IF SELF POWER IS TO BE USED)
4. POWER DISTRIBUTION: CELL STRINGS —————— I^2R —————— DAMAGE SUSCEPTABILITY	FULL HIGH VOLTAGE SERIES–PARALLEL STRINGS–PLASMA INTERACTIONS, CHARGING

PERFORMANCE ASSURANCE: SATELLITE POWER GENERATION (CONTINUED)

This chart continues the list of critical power generation parameters along with the probable required demonstration for each. Satellite dynamics are probably most critical as regards pointing of the microwave power transmitter. A preliminary pointing tolerance for this transmitter is one minute of arc, mechanical. The eventual full size transmitting antenna will probably incorporate its own attitude control system including control moment gyros and yoke drives. These critical elements should be demonstrated on the precursor unit, since this relates not only to commercial viability of the solar power system but also to environmental aspects related to the pointing of the microwave beam.

Performance Assurance Satellite Power Generation (Contd)

PERFORMANCE	DEMONSTRATION
5. POWER DISTRIBUTION: BUSSES I^2R LOSS < TBD%	DESIGN ANALYSIS ELEMENT TEST PLASMA INTERACTIONS CHARGING
6. ROTARY JOINT SLIP RING WEAR	ELEMENT TEST
7. SYSTEM DYNAMICS (INCLUDING POINTING, SMOOTHNESS OF MOTION, INTERACTION WITH REST OF SPS)	OPERATION OF SUB-SCALE SPS SYSTEM. SYSTEM TO INCLUDE POSITION SENSORS ON BOTH TRANSMITTER AND POWER GENERATION SYSTEM, YOKE DRIVES, THRUSTERS ON POWER GEN SYSTEM, CMGS (AND/OR OTHER CONTROLLERS) ON ANTENNA, CONTROL COMPUTER, ETC.. MAINTAIN REQUIRED POINT (±1 MINUTE, PRELIM) OVER A PERIOD OF TIME WHICH ALLOWS SIGNIFICANT ACCUMULATION OF METEOROID INDUCED MOMENTUM.
8. STRUCTURAL PROPERTIES	ACCELERATED ENVIRONMENTAL EXPOSURE PROPERTIES TEST

PERFORMANCE ASSURANCE: TRANSMITTER

The transmitter waveguides require high dimensional accuracy and stability. The waveguides for the precursor unit should be produced by the methods expected for the full size satellite and should be transported and assembled by similar methods, to demonstrate that the eventual mechanical flatness of the antenna can be achieved. Continuous high accuracy operation of the microwave power transmission phase loop control system should also be accomplished. Critical elements here are the successful conjugation of the reference frequency and the transmitter frequency by the antenna electronics, distribution of the reference frequency from the central generator to all parts of the antenna with compensation for changes in length of this distribution path, the frequency offset of the pilot beam from the power beam, the reception of the pilot beam by the distributed receivers of the spacetenna, the reliability of these receivers and the ground transmitter and phase control system. A long term demonstration of successful operation should include various weather conditions including rain, Faraday rotation of the ionosphere and other ionospheric variations. A critical reliability element for the transmitter is of course the klystron tubes. Current klystron reliability would be inadequate for the SPS. During the SPS development program this reliability must be increased. The demonstrator programs would operate a sufficient quantity of tubes for a sample period long enough to get through their "infant mortality" period and provide, with the required confidence, a test of the eventual reliability. Of course full reliability may not have been achieved by the time of precursor program. However, the achieved reliability could be correlated with historical trends of increase in reliability to indicate eventual operation with the required reliability in the full size program.

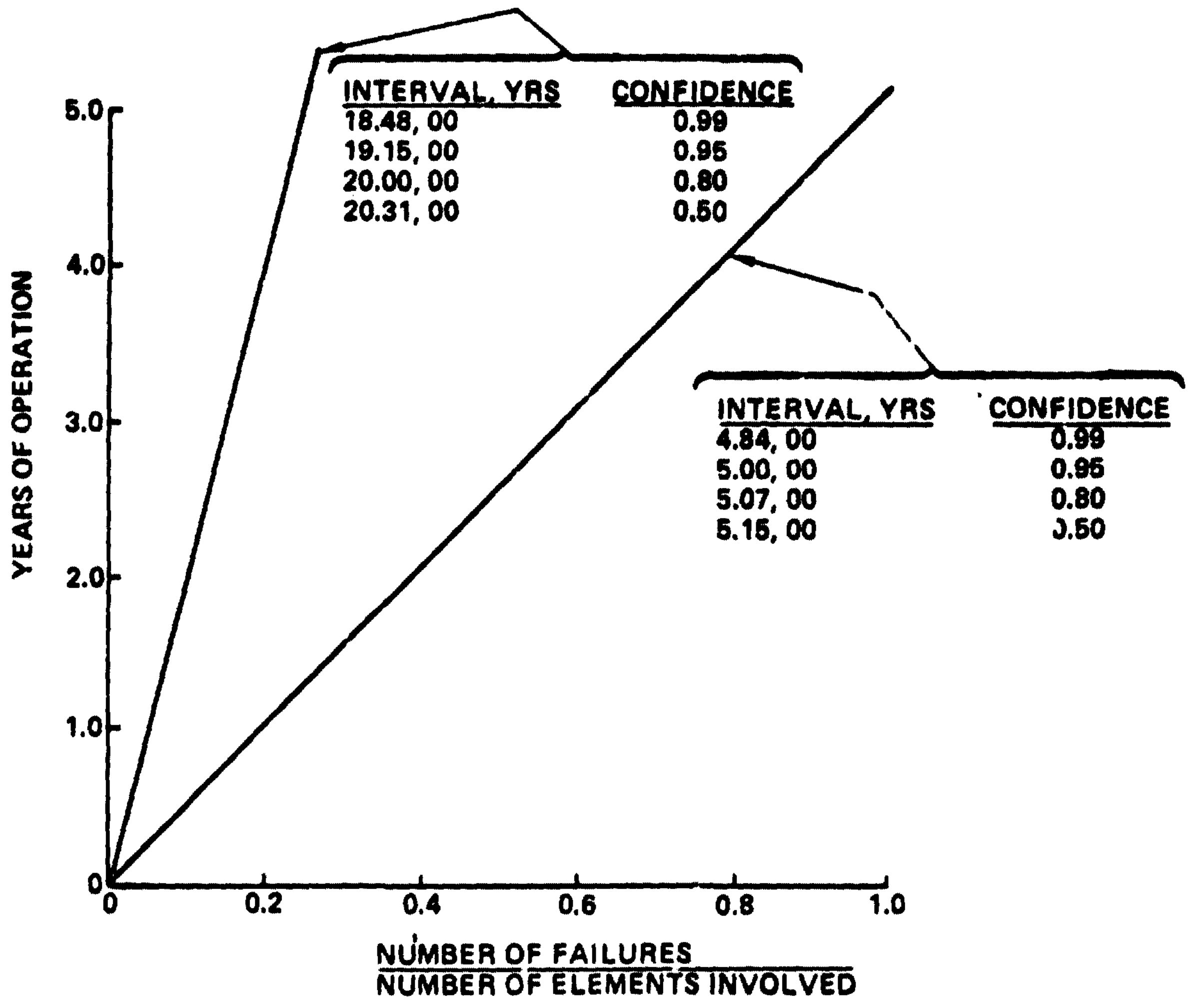
Performance Assurance Transmitter

1. WAVEGUIDE	DIMENSIONAL ACCURACY AND STABILITY OF SIGNIFICANT SAMPLES OF WAVEGUIDE PANELS, MANUFACTURED AND ASSEMBLED IN THE "OPERATIONAL" METHOD.
2. STRUCTURAL STABILITY	SUBARRAY DEPLOYMENT PRECISION AND STABILITY WITH FULL THERMAL LOAD
3. PHASE LOOP OPERATION	CONJUGATION, DISTRIBUTION PILOT BEAM OFFSET–RECEIVE PILOT VIA SPACETENNA RECEIVER RELIABILITY, GROUND TRANSMITTER LONG TERM DEMONSTRATION, VARIOUS WEATHER CONDITIONS, IONOSPHERIC CONDITIONS (BEAM "WIDTH EFFECTS"?)
4. THERMAL INTEGRITY	DEMONSTRATION OF MULTIPLE PANEL SYSTEM AT VARIOUS POWER LEVELS
5. KLYSTRON PERFORMANCE INCLUDING RELIABILITY	1. OPERATE A SAMPLE FOR A PERIOD OF TIME SUFFICIENTLY LONG TO: a. GET THROUGH INFANT MORTALITY b. PROVIDE DESIRED CONFIDENCE 2. CORRELATE WITH PRODUCTION RATE AND INDUSTRY MATURITY AND HISTORICAL IMPROVEMENT TRENDS 3. MEASURE EFFICIENCY–SMALL SAMPLE

ESTABLISHING A MTBF INTERVAL

This chart shows how various levels of meantime between failure intervals can be demonstrated based on a variable period of operation and the observed failure rate. For example, if a meantime between failure of 20 years or more is required the upper of the two criterion lines applies. If a plot of test results, that is fractional failures vs time, falls above this line it can be said that the meantime between failures is either 18.48 years with a confidence level of 0.99 or, for example, 20 years with a confidence level of 0.8. The lower of the two lines indicates lower MTBF intervals. For example if the actual test results fall along this line a MTBF of only 5.07 years with a confidence level of 0.8 would be indicated. This is probably the lowest MTBF for commercial acceptability of the SPS, since this could lead to plant factors as low as 0.6.

Establishing a MTBF Interval



(FOR SAMPLES ≦ 500)

PERFORMANCE ASSURANCE: RECTENNA

Critical rectenna performance parameters are listed along with the demonstration probably required to assure that the required levels will be present in the full sized system. For example, the conversion of the RF energy to direct current is probably demonstratable on a few production panels (perhaps only several hundred square meters). The 89% figure here given is probably the conversion efficiency achievable at the center of the microwave beam; the 80% number is close to that achievable at approximately 1 milliwatt per square centimeter at the edge of the microwave beam. Attenuation here relates to the value of microwave energy which is present beneath the rectenna panels. This probably is critical if multiple land use is invisioned, for example agriculture performed beneath the rectenna. Full long-term environmental resistance is probably the most difficult of the rectenna characteristics to demonstrate. Rectenna panels could be tested in a large weather chamber, for example, the type available at Eglin Air Force Base. This, coupled with a field test, would probably be sufficient to demonstrate adequate environmental resistance to proceed with at least 1 or 2 full-sized rectennas.

Performance Assurance Rectenna

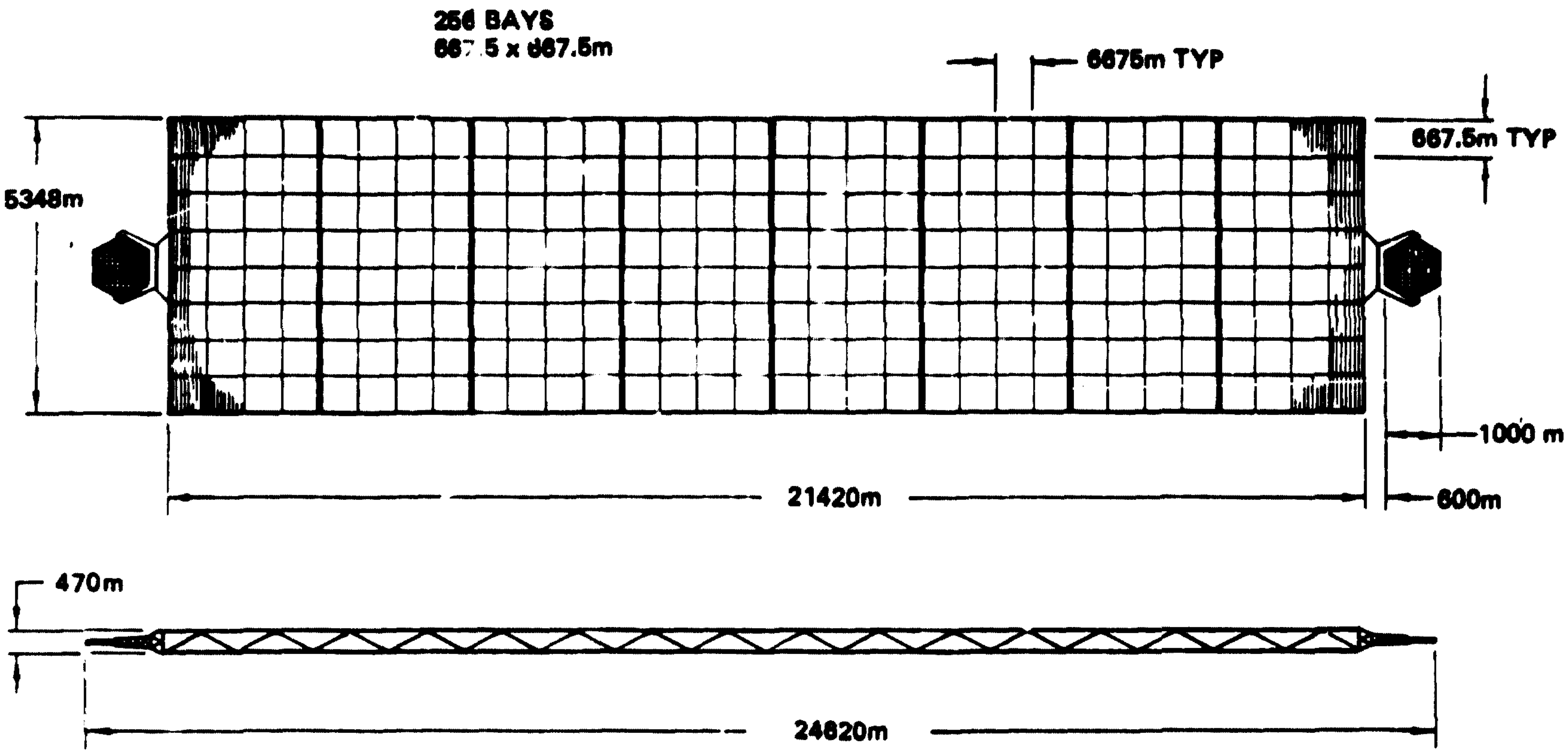
PERFORMANCE	DEMONSTRATION
1. CONVERSION EFFICIENCY 89% AT 23 mW/cm^2 80% AT 1 mW/cm^2	MEASUREMENTS ON PANELS OF APPROXIMATELY 100M^2
2. ATTENUATION (BEAM STRENGTH BELOW ANTENNA) TBD dB	MEASUREMENTS ON PANELS OF APPROXIMATELY 100M^2
3. ENVIRONMENTAL RESISTANCE RAIN ——— SNOW ——— WIND ——— HAIL ——— SEISMIC ———	WEATHER CHAMBER TEST (FIELD TEST?)
4. POWER DISTRIBUTION I^2R LOSS < TBD%	DESIGN ANALYSIS, ELEMENT TEST
5. POWER CONVERSION DC→AC OR DC→DC EFFICIENCY > %	COMPONENT TEST (CONVERTER OR MOTOR GENERATOR)
6. SWITCHING, CROWBAR OPEN CIRCUIT, TRANSIENT PROTECTION	COMPONENT TEST AND TEST WITH RECTENNA PANEL

PHOTOVOLTAIC REFERENCE CONFIGURATION

This chart shows a plan and side view of a full size SPS having a 5,000 megawatt output at each of the two rectennas. The power generation portion of the satellite is divided into 8 bays which are transferred individually from the low assembly orbit to geosynchronous orbit using the self-power method. These 8 modules are each composed of 32 identical bays. It is possible to invision a precurser SPS as being made up of 1, 2 or more bays up to a full 32, that is 1 module. Thus the precurser design would be directly relatable to the full size system. The transmitter would be a subscale unit with a power transmission capability appropriate to the number of bays provided. The transmitter can be made up of full size antenna components. The precurser SPS receiving major attention in this study was one composed of 4 bays. This will be given on subsequent charts. Such a precurser allows the use of full, exact scale, solar array panels which generate the full required voltage over the same path link with the same physical arrangement.

Photovoltaic Reference Configuration (5,000 MW Output Each Transmitter)

SPS-1861



TOTAL SOLAR CELL AREA:	101.8 km^2
TOTAL ARRAY AREA:	110.2 km^2
TOTAL SATELLITE AREA:	114.5 km^2
OUTPUT:	16.93 GW MINIMUM TO SLIPRINGS

SUITABILITY OF A SMALL PRECURSOR

A precursor unit composed of 4 bays will have 1.56% of the area of the full size SPS. Hence, the power generated by such a precursor would be approximately 1.5% of that of the full size satellite. This chart addresses the question of whether or not such a satellite is adequate to demonstrate commercial viability. Some of the critical developmental parameters of the satellite are listed along with their relationship to the full size satellite. For example, the solar cells can be of the same basic type and same thickness, i.e., 50 micrometers, (2 mils) although the cells may not have achieved the full efficiency of those to be used in the eventual satellite. The solar cell blankets can have the same physical parameters as those for the eventual full size SPS as regards cover type, thickness, substrate, cell interconnects, etc. The structural elements of the precursor can be exactly those to be used on the full sized system, if the launch vehicle has the capacity to carry up either the required beam machines, the structural components, or both. Solar cell costs are of course critical to the eventual commercial viability of this system. The number of solar cells required for a 4 bay precursor is sufficiently large to warrant development of the full production capacity to be used in the SPS system. That is, perhaps 1 or more production lines, to achieve a capacity of approximately 20,000 cells per hour, the rate required to make the cells for the precurser in a period of two years For the full size satellite, more of these production lines would be used to obtain the production rate of 2 million cells per hour which goes with a satellite addition rate of 1 per year. For the transmitter the subarrays can be of the type and size to be used in the full sized system. The klystron tubes can be full power tubes, that is, 72 kilowatts, however, in the precursor the tube mass and efficiency may not meet the target goals required for the full size system. Other transmitter elements can closely parallel those to be used in the full size SPS. Rectennas can be smaller area versions of the full size system and use panels, mounting methods, switch gear, etc. of the final design. A self-powered transport to geosynchronous orbit can be used with power processors and thrusters of the type to be employed on the full size SPS. Their efficiency might be slightly lower than that for the eventual system.

Suitability of a "Small" Precursor

DEVELOPMENT	RELATIONSHIP TO FULL SIZE	1.56% SPS ADEQUATE
POWER GENERATION		
CELL	SAME THICKNESS, SOMEWHAT LOWER EFFICIENCY	✓
BLANKET	SAME	✓
VOLTAGE, CURRENT	SAME	✓
STRUCTURE	SAME (15M WIDTH, JOINTS, MATERIAL)	✓
ATTITUDE CONTROL	SAME (PERHAPS LOWER EFFICIENCY)	✓
PRODUCTION EQUIPMENT	SAME (16,000 CELLS PER HOUR, TWO YEARS)	✓
ROTARY JOINT	SMALLER DIAMETER, SAME CONTACT SYSTEM	✓
SWITCHGEAR	SAME (PERHAPS SUBSCALE)	
RELIABILITY DEMONSTRATION	YES, ENOUGH COMPONENTS	
POWER TRANSMISSION		
TUBE	SAME (70KW)(PERHAPS HEAVIER, LOWER MTBF)	✓
WAVEGUIDE	SAME	✓
STRUCTURE	SAME (FULL DEPTH)	✓
PHASE CONTROL	SAME (EXCEPT 1/5 DISTANCE FOR REF. PHASE DIST)	✓
PRODUCTION EQUIPMENT	SAME (NEED ABOUT 5000 TUBES)	✓
SWITCHGEAR	SAME	✓
ATTITUDE CONTROL	SAME (PERHAPS SUBSCALE CME's)	✓
RELIABILITY DEMONSTRATION	YES, ENOUGH COMPONENTS	✓
POWER RECEPTION		
RECTENNA	SAME (AT LOWER EFFICIENCY, PERHAPS 60% NOT 85%)	✓
LEO TRANSPORT	GOOD (WITH FLYBACK BOOSTER)	✓
GSO TRANSPORT		
THRUSTER	SAME OR SUBSCALE	✓
POWER PROCESSOR	SAME OR SUBSCALE	✓

SUITABILITY OF A SMALL PRECURSER CONSTRUCTION BASE

Analyses of potential construction methods for a 1 bay to 32 bay size precurser satellite have indicated that a one-eighth scale construction base would be appropriate. This one-eighth scale construction base would closely approximate a segment of a full size construction base. It would have the capability to produce the full depth bays to be used in the full sized satellite. The chart shows that each of the critical construction base parameters are demonstratable on this on-eighth scale unit. As in the full size construction process, the antenna is built separately and mated to the yoke and the power generation module. The four power generation bays would be built in two groups of two, forming two modules which individually self-power to the operational orbit. One of these modules would carry the transmitter. In the operational orbit the two modules would be berthed together in a process similar to that to be used in the full sized system.

Suitability of "Small" Precursor (Construction Base)

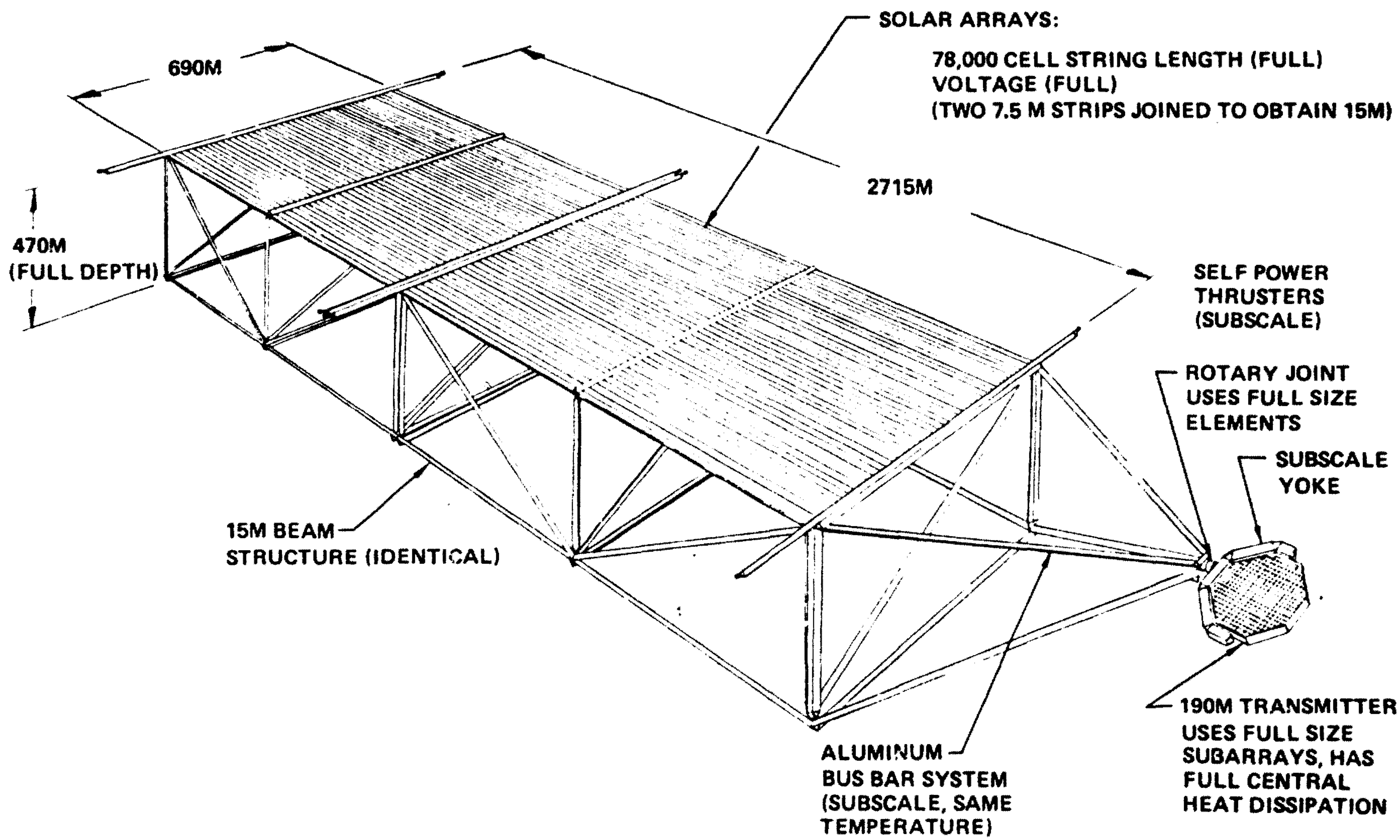
DEVELOPMENT	RELATIONSHIP TO FULL SIZE	WITH "12% BASE"
CONSTRUCTION		
BEAM MACHINE	SAME (EXCEPT TRANSPORT PACKAGING)	√
BEAM INTERCONNECTS	SAME	√
SOLAR ARRAY DEPLOYMENT	2X7.5m=15M ("CANISTER BOOSTER")21X3.75=15M(ORBITER)	√
BUSBAR DEPLOYMENT	SAME, BUT SUBSCALE	√
MODULE "INDEXING"	SAME	√
ASSEMBLE ANT. STRUC.	SAME	√
DEPLOY ANT. SUBARRAYS	SAME	√
MATE ANT. TO YOKE	SAME	√
MODULE BERTHING	SAME	√
ANTENNA XPORT LOCATION	SAME	√

PRECURSOR SPS

This chart shows the four ba· precursor SPS including its transmitter system. Each of the four bays is identical to that to be used in the full SPS. The 470 meter depth of this unit is exactly that, for example, of the full size system. The left-hand two bays form one module which is constructed first and dispatched to geosynchronous orbit, then the right-hand two modules and the transmitter are constructed and dispatched. Each of the two modules mounts self-power transfer electric thrusters at its four corners on extensions of the 15 meter structural beams. The antenna is displaced from the edge of its power module by a structural system which allows the antenna to be rotated to a position beneath the center of that module. This method is used in the full size system, where the transmitter antenna is rotated to a position beneath the center of the module. The 190 meter diameter transmitter is composed of full sized subarrays arranged so as to provide the requisite power taper. In the center of the transmitter, nine subarrays of the full power type are grouped so as to duplicate the maximum thermal environment of the full size transmitter. Flat sheet aluminum busbar conductors route the power from the solar arrays to the rotary joint and yoke system.

PRECURSOR SPS

SPS-1812



POWER TRANSMITTED

If a precurser satellite is to be operated in the later part of the 1980's the solar cells for that satellite must undergo a design freeze in approximately 1983. The chart shows how the natural progression from today's solar cells to the high efficiency solar cells of the full size SPS will result in an efficiency of approximately 16% for the precurser cells. With this efficiency and with the probable transmitter parameters achievable at the time of the precurser satellite, approximately 185 megawatts will be launched from the face of the transmitter. This is approximately 3% of the power of the full size transmitter.

Power Transmitted

- SOLAR ARRAY PERFORMANCE: (50 μM CELLS)

1977	1983	1990
OCLI 11.2 TO 12.3% SPECTROLAB 10.3 TO 11.3% SOLAREX 11.2% ASSUMPTION: 75 μM VEE-GROOVE COVERS WILL ALLOW A 12% CELL TO OBTAIN 13%	ESTIMATES OF HELIOTEK & H. OMAN 14.5% BARE CELL YIELDS 16% WITH TEXTURING & VEE-GROOVES	15.75% CELL 17.33% WITH TEXTURING & VEE-GROOVES

- TRANSMITTER OUTPUT

OUTPUT OF FOUR FULL SIZE SPS "BAYS," WORST ILLUMINATION 289 MW
WITH 1983 CELLS: (16%/17.33% x 289) ——— 267 MW
BUSBAR I^2R, 0.98 (SHORTER THAN FULL SIZE) 261 MW

ANTENNA POWER DISTRIBUTION	0.98	(SHORTER THAN FULL SIZE)
DC → RF CONVERSION	0.82	(0.85, ULTIMATE)
WAVEGUIDE I^2R	0.985	FULL SPS
IDEAL BEAM	0.965	FULL SPS
INTER SUBARRAY	0.946	FULL SPS
INTRA SUBARRAY	0.981	FULL SPS
	0.71	

HENCE 185 MW ARE "LAUNCHED"

FULL SIZE SPS IS 6220 MW LAUNCHED (PER TRANSMITTER) 185/6220 = 0.03

BEAM PATTERN

This chart relates attenuation of the microwave beam to the radial distance from the center of that beam. The central beam strength of the full size power satellite is approximately 23 milliwatts or 23,000 microwatts. The chart shows a simple relationship for determining the central beam strength developable by the precursor unit. The 4-bay precurser with a 190 meter diameter transmitter achieves a central beam strength of slightly over 23 microwatts per square centimeter or 1/1,000 of the beam strength of the full size system. At a radius of approximately 10 miles, the beam strength diminishes to approximately 10 microwatts per square centimeter which is the Soviet continuous exposure standard. Thus, the beam diameter of 20 miles is relatable to the tightest exposure standard in the world.

C-4

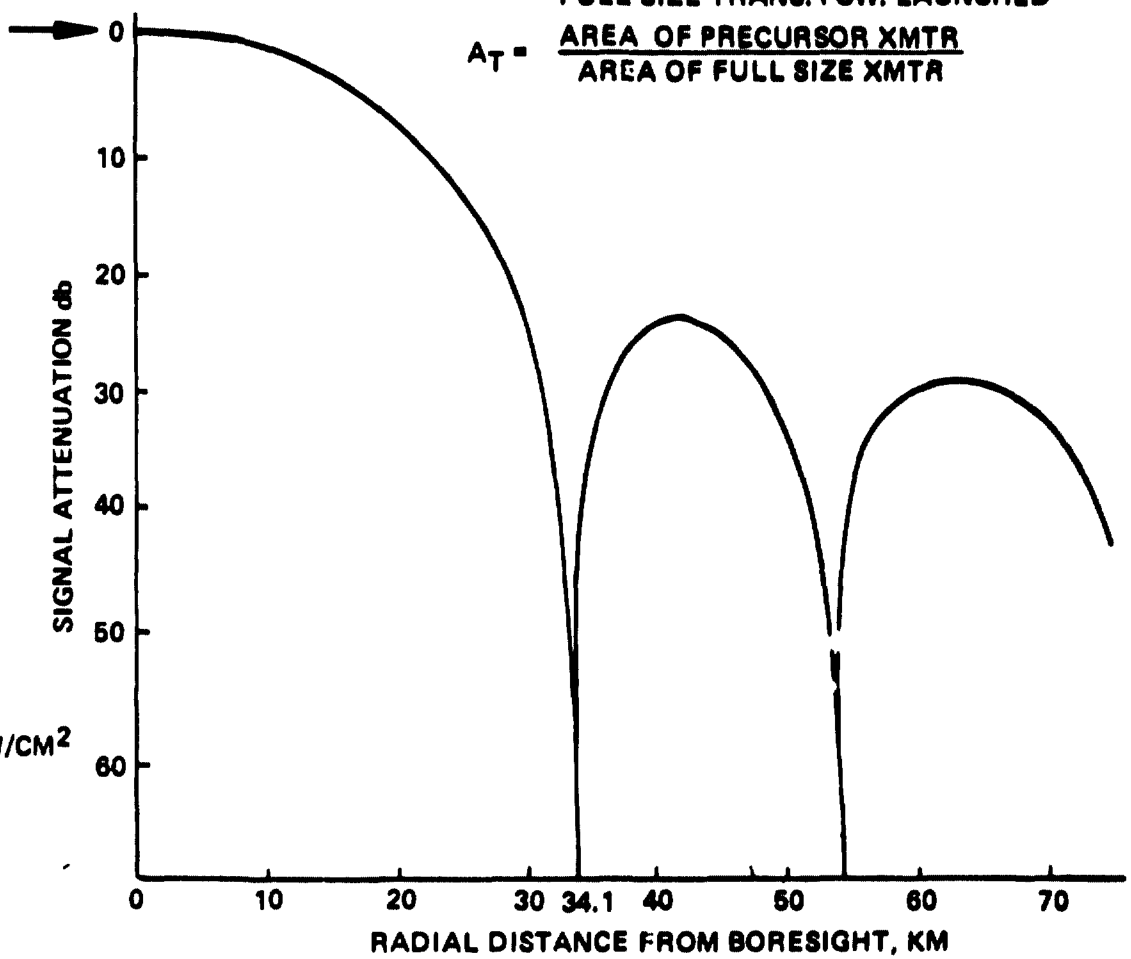
Beam Pattern

P_{MAX} = CENTRAL BEAM STRENGTH

$P_T = \dfrac{\text{POWER LAUNCHED BY TRANS}}{\text{FULL SIZE TRANS. POW. LAUNCHED}}$

$A_T = \dfrac{\text{AREA OF PRECURSOR XMTR}}{\text{AREA OF FULL SIZE XMTR}}$

$P_{MAX} = 23{,}000\ \mu W/CM^2 \times (P_T) \times (A_T)$ →



REFERENCE PRECURSOR

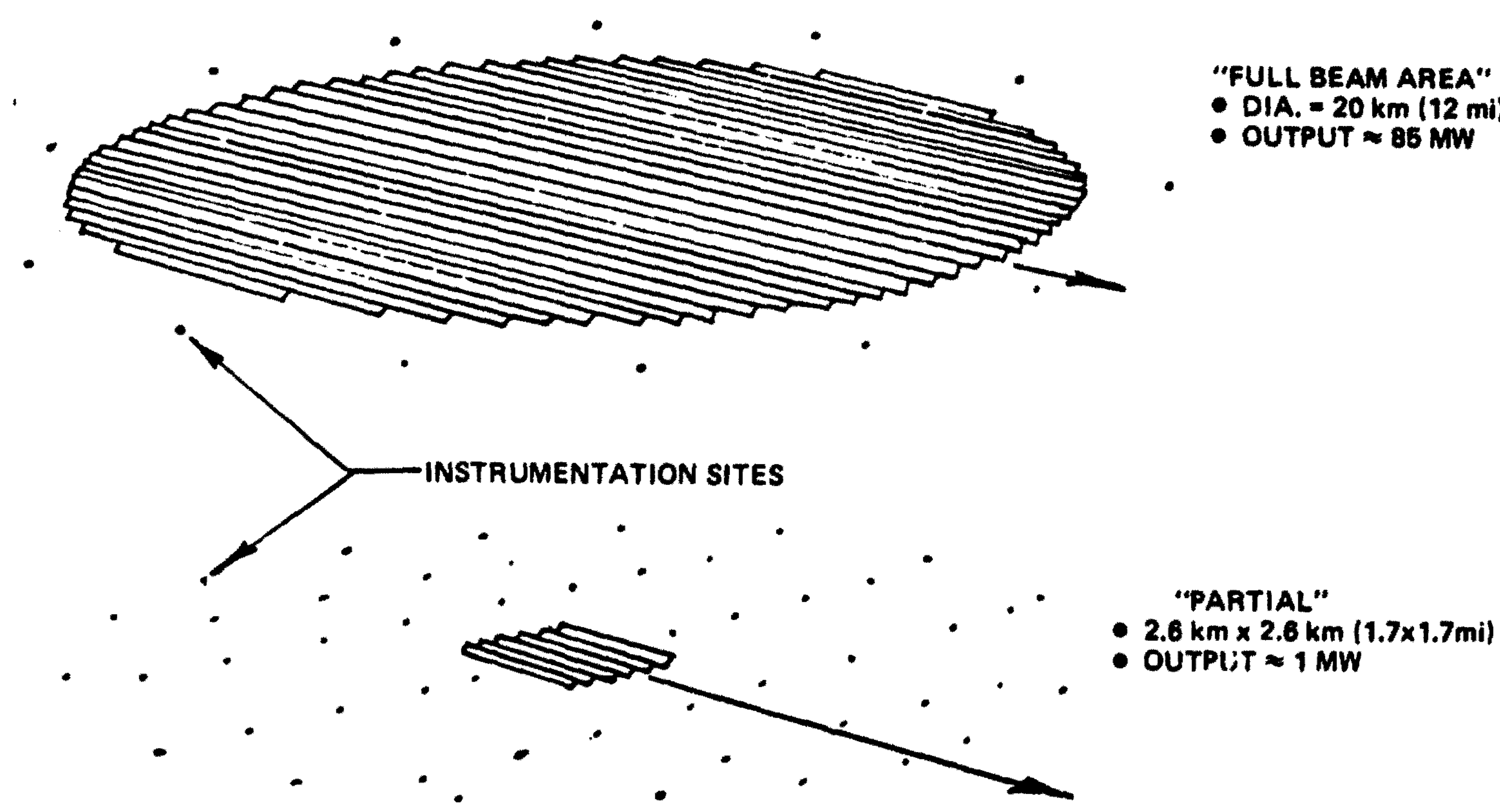
- $P_T = 0.03$ (OF FULL SIZE)
- $A_T = 0.036$ (OF FULL SIZE)

$P_{MAX} = (23{,}000)(0.03)(0.036) = 24.6\ \mu W/CM^2$

RECTENNA OPTIONS

It is probably not appropriate to construct a full area rectenna for the precurser SPS. With the full rectenna area a power output of approximately 85 megawatts would be developed by the 4-bay precurser SPS studied. This power level is of little commercial significance relative to the cost of the precurser program. A partial rectenna, as shown, with an area of 7 square kilometers would probably produce about 1 megawatt with the precurser's central beam strength. This 1 megawatt is probably quite adequate to demonstrate successful integration of space generated power into a commercial utility network. The beam area around the rectenna section would be instrumented in order to determine the characteristics of the transmitted beam.

Rectenna Options
(For "1.56 % PSPS")



RE :TENNA CONSIDERATIONS

It is probably appropriate to direct the microwave beam from the precursor satellite to a government reservation. If the most stringent current microwave exposure standards is used, and such a standard may be levied upon the solar power project before it has demonstrated long term successful microwave beam pointing, then we will be working with a standard of 10 microwatts per square centimeter; at and above this power level the precursor beam diameter is 20 miles. This will fit conveniently within a government reservation such as White Sands Proving Ground, New Mexico and would be conveniently near the power distribution network at Alamagordo, New Mexico. Despite the low central beam strength and the lower efficiency of rectennas at these levels a precursor rectenna of approximately 7 square kilometers, that is, 1.7 miles by 1.7 miles, would be adequate to develop one mega-watt and would cost approximately 5% of the cost associated with the total precursor program.

Rectenna Considerations

- GOVERNMENT RESERVATION DESIRED
  - CONTAIN APPROXIMATELY 20 MILE DIAMETER BEAM SECTION WHICH IS ABOVE SOVIET STANDARD (10 μW/cm^2)
  - SIGNAL LEVEL (~ 23 μW/cm^2) MAXIMUM WOULD PERMIT PERSONNEL EXPOSURE
- LOCATE NEAR EXISTING TRANSMISSION LINE
  - INSERT ONE MEGAWATT
- RECTENNA COULD USE "FINAL SPS" COMPONENTS
- EVEN AT ONLY 60% EFFICIENCY, 7 km^2 (1.7 MILE x 1.7 MILE) PRODUCES 1 MW
- AT \$100/m^2, RECTENNA FOR 1 MW IS "ONLY" \$0.7B

PRECURSOR SPS CONSTRUCTION BASE

This chart shows the configuration of the precursor SPS LEO construction base. Comparison of this facility to that shown for the full-size SPS LEO base reveals that the precursor facility is a one-bay corner of the full size facility.

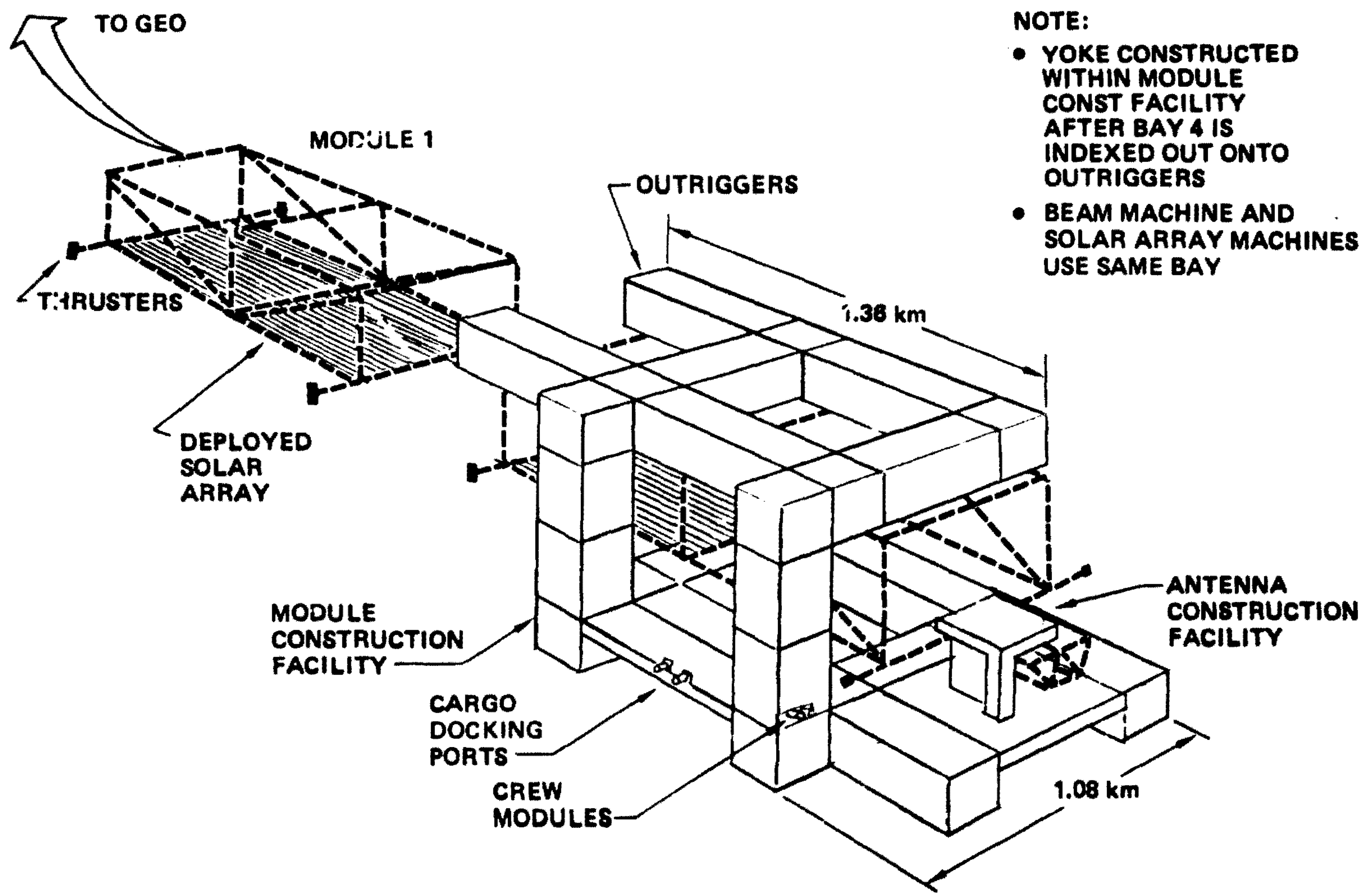
The modules and the yoke would be constructed in the larger part of the base using the full-size construction equipment that would be used to construct the full size SPS modules. The significant operational difference would be that after the frame for one of the module bays is constructed the frame assembly equipment would be moved out of the way so that the solar array deployment machine could be moved into the same construction bay. After the array is deployed, the completely assembled bay is indexed out onto the outriggers so that the second bay could be assembled.

The yoke and thruster systems would also be constructed in this bay.

The antenna would be constructed in the antenna construction facility which is located in such a way that the yoke and antenna could be mated without any vertical movement of the antenna.

Precursor SPS Construction Base

SPS-1925



PRECURSOR SATELLITE
LEO BASE CREW SIZE/CONSTRUCTION TIME

This chart shows the results of a preliminary comparison of three concepts for constructing the precursor satellite.

Alternative A is a very optimiztic approach wherein it was assumed that the operational rates designed into the baseline full-size SPS construction concept could be achieved when making the precursor. The other major assumption was that the modules and antennas are constructed simultaneously. This results in a large crew but achieves a very short construction time.

Alternative B is the same as Alternative A except that a machine rate 25% as high as the operational satellite was assumed since in the early days of construction a learning process will be in effect in addition to more machine down time.

Alternative C is the concept selected as the reference approach. In this concept, it was assumed that it would be necessary to minimize the crew size. This is achieved by having a smaller number of people perform all of the construction tasks in a series approach; 1) assemble subassemblies, 2) assemble the two modules, 3) assemble the yoke, and then 4) assemble the antenna. Again, 25% of the designed operational rates were assumed. This approach will obviously take longer than the other concepts but will result in a lower capital investment cost.

Precursor Satellite
LEO Base Crew Size/Construction Time

SPS-1810

BOEING

CONCEPT	BASELINE SYSTEM	Ⓐ • ANT AND MODULE CONSTRUCTED IN PARALLEL • 100% OPERATIONAL MACHINE RATES	Ⓑ • ANT AND MODULE CONSTRUCTED IN PARALLEL • 25% OPERATIONAL MACHINE RATES	Ⓒ • ANT AND MODULE CONSTRUCTED IN SERIES • 25% OPERATIONAL MACHINE RATES
CREW SIZE				
BASE MGMT	10	10	10	6
CONSTRUCTION				
MGMT	22	16	16	10
MODULE CONST	68	40	40	40
ANT CONST	82	76	76	0
SUBASSY	49	16	16	0
MAINT	49	12	12	12
LOGISTICS	42	4	4	4
TEST/QL	40	10	10	5
BASE OPS	39	12	12	5
BASE SUPPORT	77	17	17	17
TOTAL	478	213	213	99
CONSTRUCTION TIME	340 DAYS	59 DAYS	242 DAYS	404 DAYS

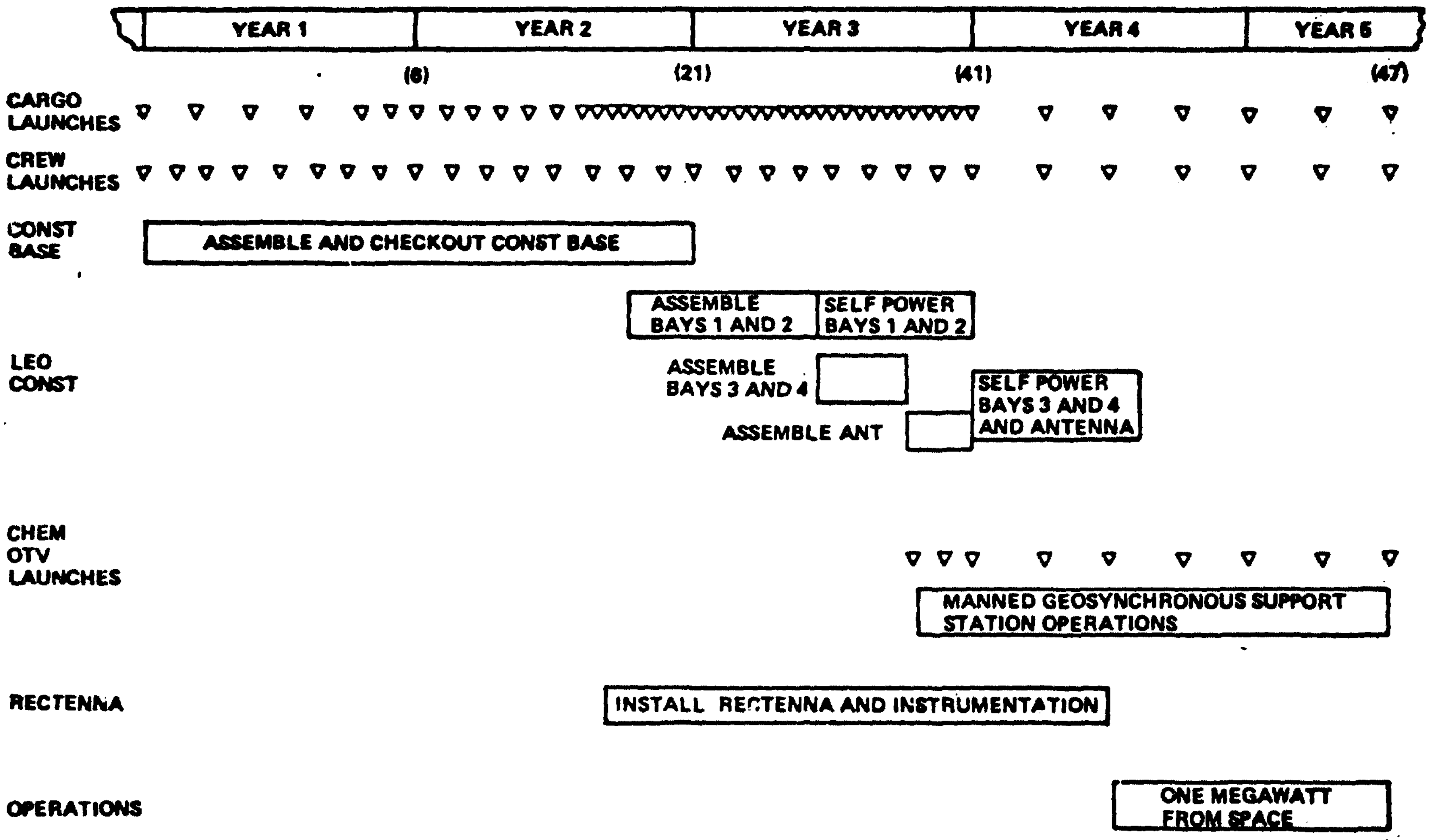
- ALL OF THE NEW CONCEPTS UTILIZE IDENTICAL FACILITY

PRECURSOR SPS OPERATIONS SCENARIO

The precursor program invisioned requires approximately 4-1/2 years of in-space operations. The in-space operations begin with the cargo and crew launches associated with the placement, assembly, and checkout of the construction base. These launches continue for approximately 2 years at which point additional launches begin to bring up the mass of the precursor satellite itself. The first and second bays of the satellite are assembled into one module which is dispatched, using the self-power method, to geosynchronous orbit. At the time of dispatch, assembly and checkout of the third and fourth bays are started in order to form the second module. This module carries the antenna with it to geosynchronous orbit. The self-power transfer time is 180 days. Before the first module arrives in geosynchronous orbit a manned geosynchronous orbit support station is made operable. This support station is the base of operations for the berthing operation (which joins the two modules together), final checkout and make-operable operations, and for the one year operational period baselined for this precursor satellite. It is from this support station that maintenance operations will be accomplished. Prior to the beginning of power transmission from space the test rectenna and its associated instrumentation made operable.

Precursor SPS Operations Scenario

SPS-1858



PRECURSOR SPS MASS

The mass is given for a precursor SPS which is 1.56% of the full system size. Because the full size SPS development will not be complete at the time of the precursor, "novelty" multiplying factors are used, with the largest factor applied to the transmitter elements (such as the klystron). To the mass of the precursor SPS in geosynchronous orbit must be added the mass of the systems necessary to accomplish self-power transfer to geosynchronous orbit, including the argon propellant, electric thrusters, power processors, etc. The total mass to be placed in low orbit for accomplishment of the precursor program is thus approximately 3600 metric tons. If a geosynchronous orbit assembly site was selected, with transfer of the satellite equipment to that orbit by chemical orbit transfer vehicles, the initial mass in low orbit would be approximately 6200 metric tons.

Precursor SPS Mass (Metric Tons)

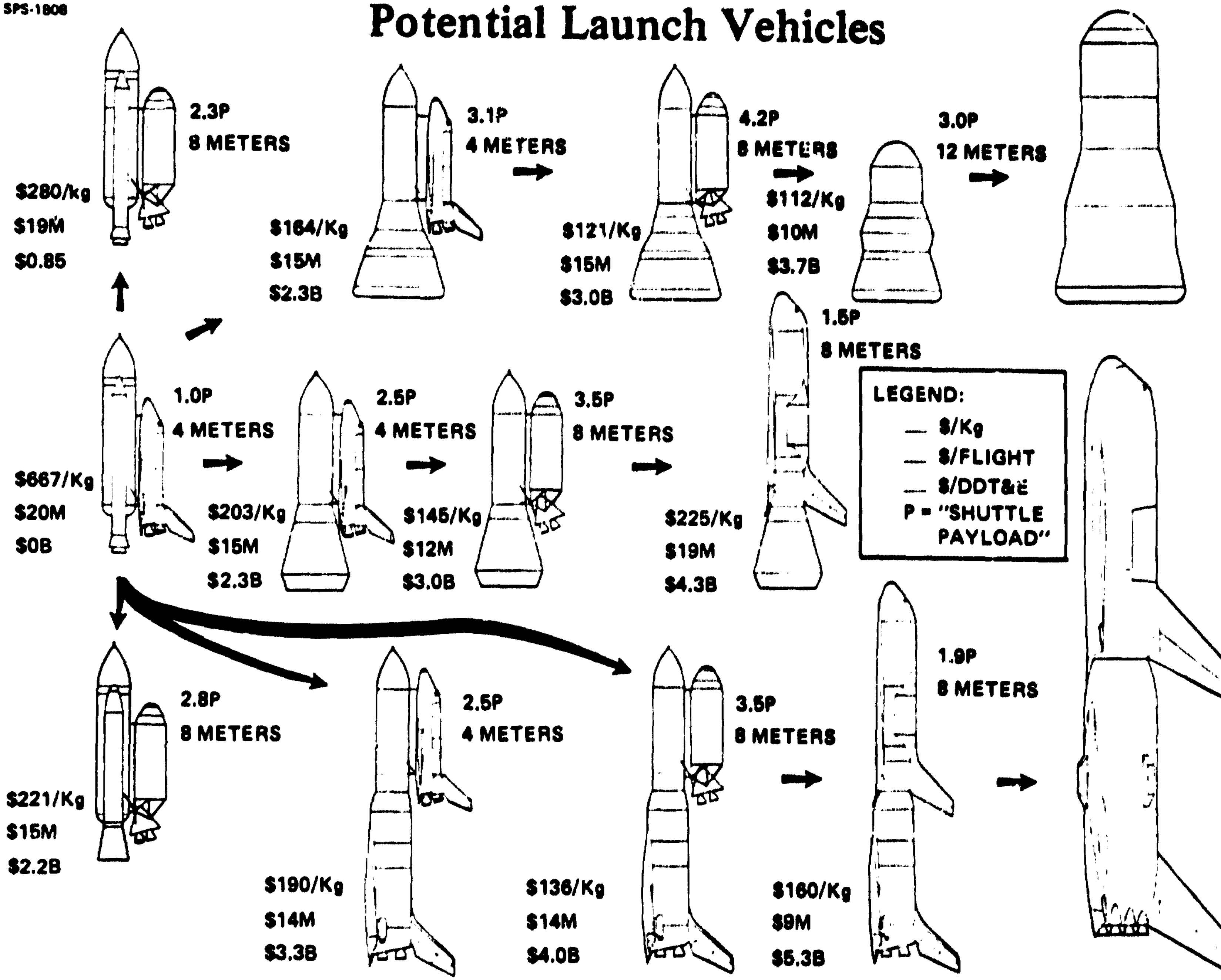
ELEMENT	FULL SPS MASS (W/GROWTH)	FRACTION OF FULL ELEMENT	"NOVELTY/ SUBSCALE" FACTOR	PRECURSOR MASS
POWER GENERATION	66,000	0.0156	1.10	1,130
POWER TRANSMISSION (ONE TRANSMITTER)	12,600	0.0312	1.25	490
PRECURSOR SPS IN GEO ORBIT				1,620
INITIAL MASS IN LOW ORBIT FOR SELF POWER WITH 5% SPARES (1.40 FACTOR)				2,520
ALLOWANCE FOR PALLETS, ETC. IN LOW ORBIT LAUNCHER (1.10) INCLUDED				2,77[illegible]
LOW ORBIT CONSTRUCTION BASE				850
TOTAL MASS LAUNCHED IN PRECURSOR SPS PROGRAM				3,620

POTENTIAL LAUNCH VEHICLES

On the center left is shown the space shuttle which is nearing the end of its development phase. On the right are shown two heavy lift launch vehicle concepts which have been developed for achievement of a solar power satellite program. The upper of the two is a two-stage ballistic vehicle employing ocean landing for each of the stages. The lower is a two-staged winged vehicle which uses runway landings for recovery. Both of these have payloads in excess of 400 metric tons. They are larger than would be required for a precurser program and it is probably inappropriate to baseline their use in a precurser program. In fact one of the major advantages of a precurser as a demonstrater is that it allows successful accomplishment of a small size "power from space program" before commitment to maximum funding for either of these vehicles. The other vehicle configurations shown on the chart are derivatives of the space shuttle. Above the space shuttle is shown a vehicle employing an 8 meter diameter payload shroud with a SSME recovery capsule. This vehicle uses the standard external tank and solid rocket boosters. The payload is approximately 2.3 times that of the standard shuttle. Below it is a similar vehicle which instead of the solid rocket boosters employs liquid rocket stages which are recovered at sea. The space shuttle orbiter and the 8-meter diameter payload shroud/SSME combination are also shown with two types of ballistically recovered liquid rocket stages, and with a winged turbojet flyback booster. The larger of the ocean-recovery ballistic rocket stages is also shown with a ballistic orbiter; the smaller is shown with an internally tanked liquid hydrogen/liquid oxygen orbiter vehicle employing SSME's. This internally tanked orbiter is also shown in combination with the winged booster. For each of the shuttle derivative vehicles the multiple of the shuttle payload, the diameter of the payload, the cost per kilogram to low orbit, the cost per flight, and the total development of that vehicle are given. Each of these development costs assumes that successful completion of the entire shuttle development program has occurred, and that the full SSME, reusable insulation, etc. is available.

SPS-1808

Potential Launch Vehicles

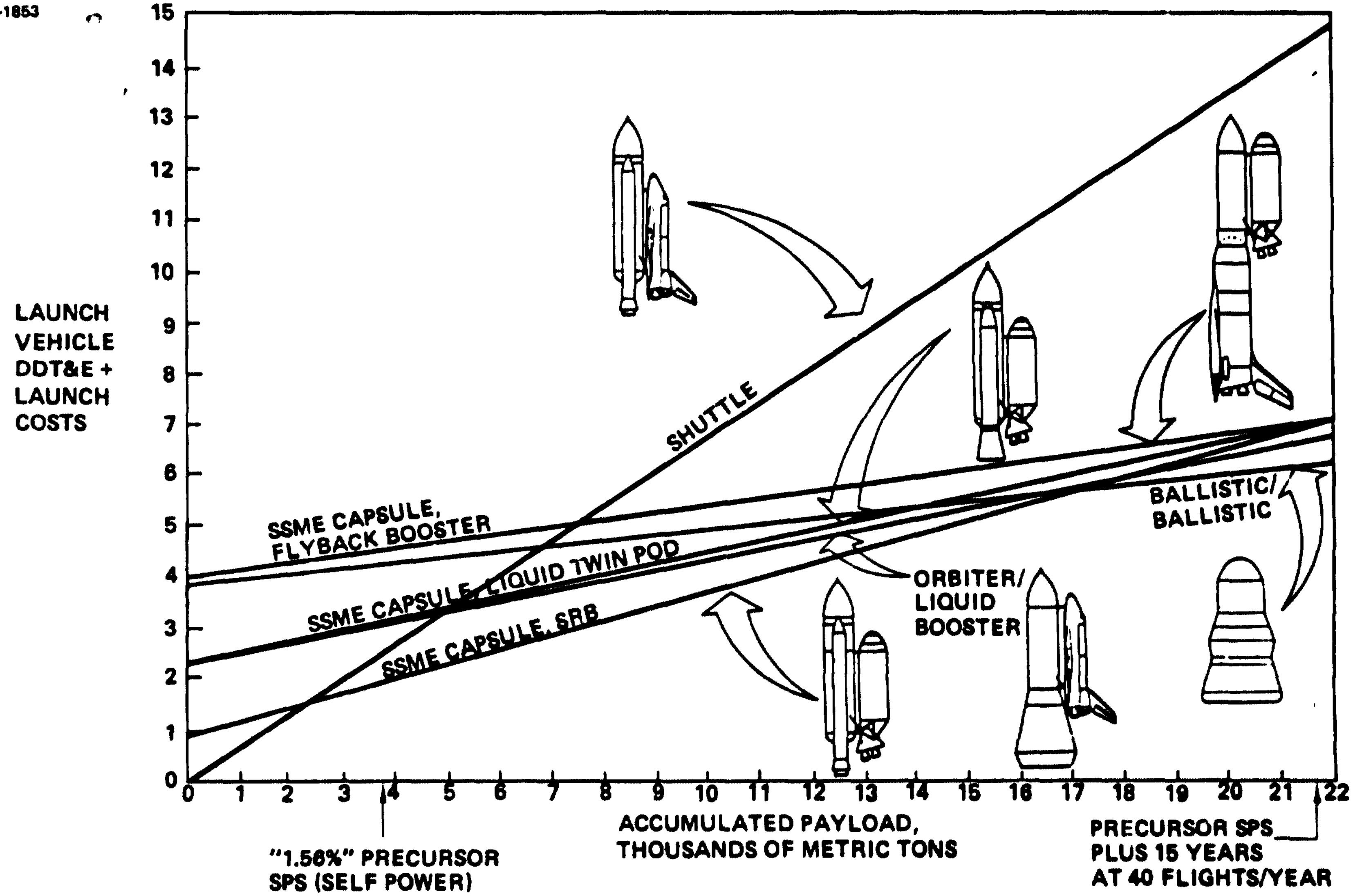


LOW ORBIT PLACEMENT COSTS

This chart shows the total cost required to place an accumulated mass figure into low orbit. The costs include not only the per flight cost but also the developmental cost association with that vehicle. The mass required to accomplish the 1.56% size precursor program is shown along with a mass figure for that precursor and the mass accumulated by 15 years of vehicle operation at a launch rate of 40 flights per year. The line associate , with the space shuttle begins at 0 since it is assumed here that its development has been accomplished. However, its relatively high per-flight cost causes it to be above the ther systems before eight years of operation have taken place. The other shuttle derivatives have higher DDT&E cost but lower cost per flight. The lowest cost vehicle for placement of the precursor SPS shown here is the shuttle derivative employing the 8 meter diameter shroud and a SSME recovery capsule with the standard external tank and solid rocket boosters.

Low Orbit Placement Costs
(No Discounting)

SPS-1853



THE SRB AND STRATOSPHERIC OZONE

This sentence is an excerpt from the draft environmental impact statement for the space shuttle orbiter and is indicative of the environmental concerns associated with large scale operations of the solid rocket boosters for an extended period of time.

The SRB and Stratospheric Ozone

SPS-1825

BOEING

"THE DECAY TIME OF THE SPACE SHUTTLE EXHAUST EFFECT IS CORRESPONDINGLY SHORT, SO THAT AFTER REPLACEMENT OF THE CURRENT BOOSTER BY A NONCHLORINE BOOSTER, THE OZONE LAYER WOULD RETURN TO NORMAL IN A FEW YEARS."

ENVIRONMENTAL IMPACT STATEMENT, SPACE SHUTTLE PROGRAM

(DRAFT) JULY, 1977

LOW VERSUS HIGH ORBIT ASSEMBLY FOR PRECURSOR SPS

In this chart it is assumed that 1/4 of the development cost associated with shuttle derivative launch vehicles is chargeable to the precursor SPS program. Costs for the transportation elements of low and high orbit assembly operations are given for the shuttle and two shuttle derivative vehicles. With the shuttle, low orbit assembly is far lower in cost than with high orbit. With one derivative, low orbit assembly is slightly more expensive. We may conclude that self-power assembly for the precurser program will approximately pay for itself and that therefore it should be assumed to be part of the program, since it demonstrates and develops the eventual self-power transfer system.

Low Versus High Orbit Assembly for Precursor SPS

BOEING

SPS-1856

(WITH ¼ OF LAUNCH VEHICLE DDT&E) ($B, 1977)

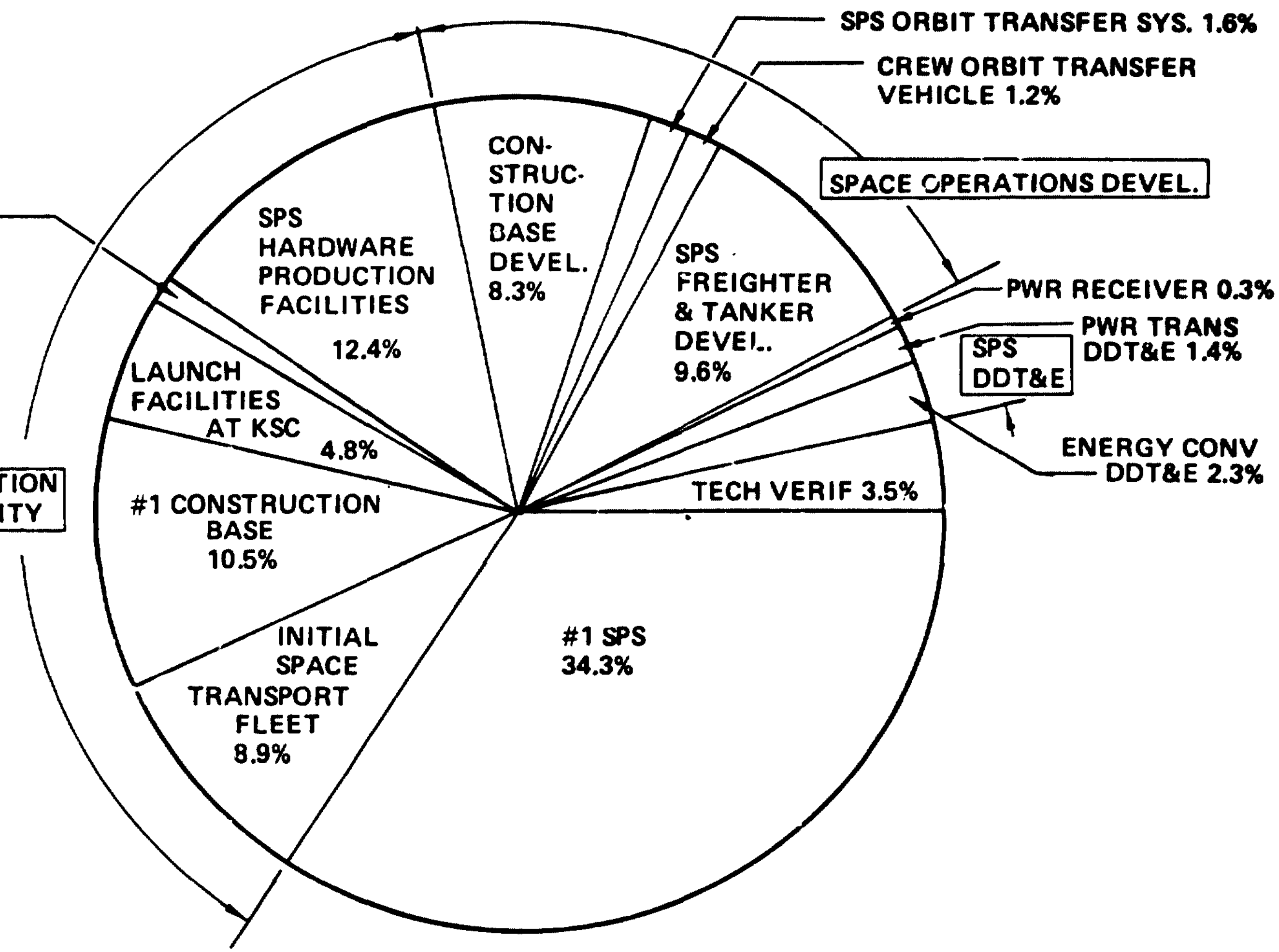
	SHUTTLE	SRB/ET SSME CAPSULE	FLYBACK/ET SSME CAPSULE
LOW ORBIT (SELF POWER)			
LAUNCH 3,790 MT	2.52	1.27	1.52
DEVELOP SELF POWER	0.60	0.60	0.60
SELF POWER EQUIPMENT	0.25	0.25	0.25
DEV 40 MT OTV	0.20	0.20	0.20
NINE OTV FLIGHTS	0.04	0.04	0.04
	3.61	2.36	2.61
HIGH ORBIT			
LAUNCH 6,240 MT	4.16	1.96	1.66
DEV 40 MT OTV	0.20	0.20	0.20
51 OTV FLIGHTS	0.21	0.21	0.21
	4.57	2.37	2.26

CONCLUSION: SELF POWER ABOUT PAYS FOR ITSELF AND DEMONSTRATES/DEVELOPS FINAL SYSTEM.

TOTAL COST THROUGH NUMBER 1 SPS PHOTOVOLTAIC SYSTEM

This chart shows development, facilitization and unit costs for the SPS program through the first full size unit. Note that the development cost for the SPS itself are relatively small in comparison with the total. This indicates that a good precurser program may accomplish nearly all of the direct development associated with the SPS. This will be shown on the next chart.

Total Costs Through #1 SPS Photovoltaic System



TOTAL = 83.6 BILLION

PRECURSOR SPS COST ESTIMATE

A preliminary cost estimate is made for each of the major elements of the precursor program, along with a 15% allowance for miscellaneous items, operational costs, etc. which have not been identified. Note that not all the cost associated with the shuttle derivative launch vehicle, its facilities, the chemical OTV, etc. are directly charged to this program, since it is quite probable that all of these vehicles will have other uses. In fact it is probable that one would not imbark upon a shuttle derivative launch vehicle unless it had significant other use than for a precursor SPS program. Many of the cost elements have direct applicability to the full size SPS so that development funds expended in the precursor program directly reduces the funding necessary to accomplish the eventual SPS. The total value of this reduction is approximately $6 billion for the precursor program.

Precursor SPS Cost Estimate ("1.56%")

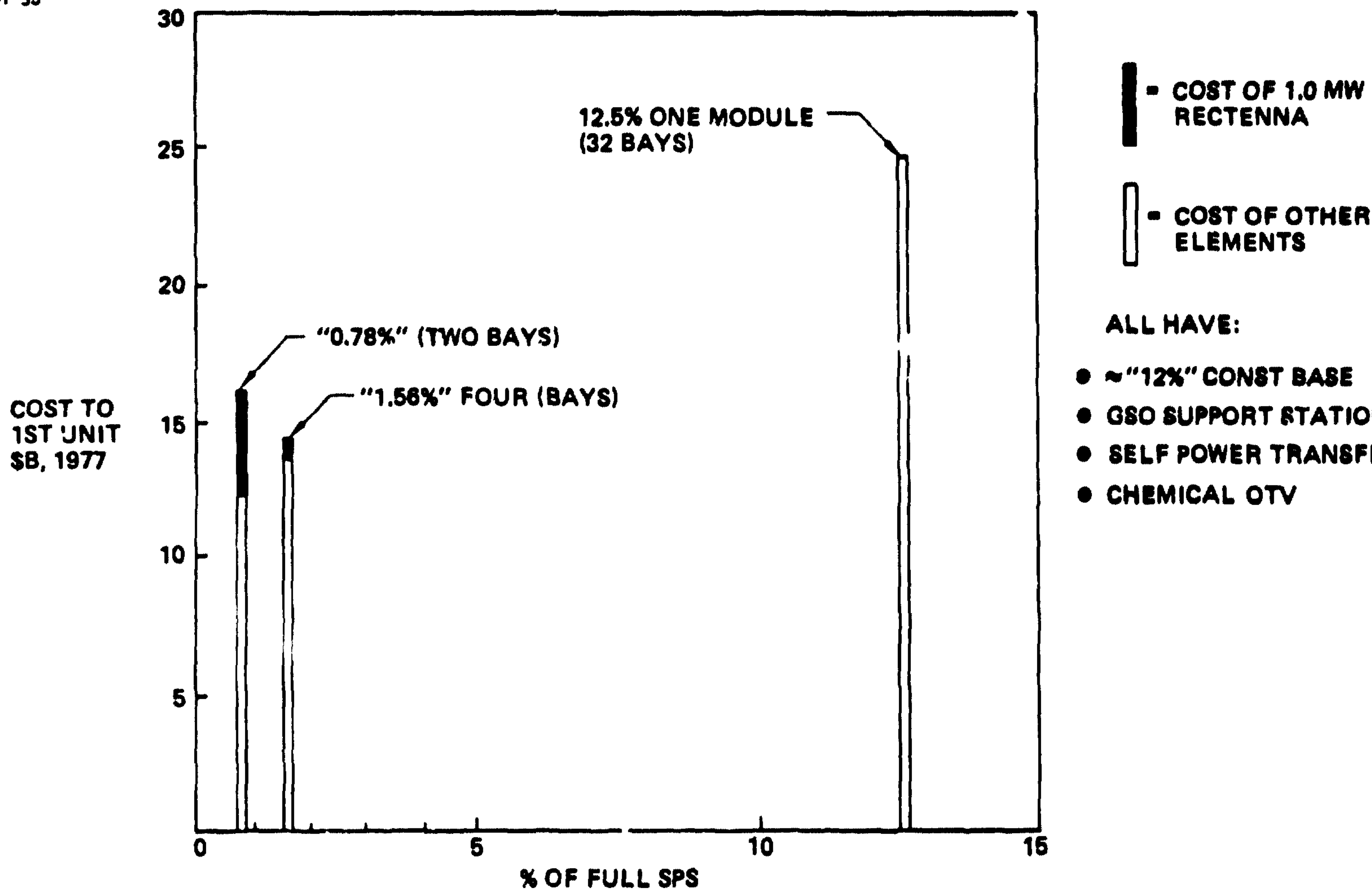
SPS-1854

ELEMENT		$B (1977)	APPROXIMATE CONTRIBUTION TO SPS DDT&E $B (1977)
CONSTRUCTION BASE (WITH $3.0B DDT&E)		5.30	3.1
SPS DDT&E:	POWER GENERATION	0.96	0.8
	POWER TRANSMISSION	0.59	0.4
	POWER RECEPTION	0.12	0.1
SPS HARDWARE:	POWER GENERATION	0.35	
	POWER TRANSMISSION	0.25	
	STRUCTURE, MISCELLANEOUS	0.20	
SELF POWER TRANSFER (WITH DDT&E)		0.85	0.6
GSO SUPPORT STATION (WITH DDT&E)		1.20	0.4
LEO TRANSPORT (FLYBACK BOOSTER/ET/8M SHROUD/SSME CAPSULE)			
	¼ OF DDT&E	1.00	0.2
	47 FLIGHTS (9 SUPPORT GSO STATION)	0.66	
	FLEET (½ BOOSTER, ½ SSME CAPSULE)	0.80	
FACILITIES (¼ PAD, PAYLOAD HANDLING, ETC.)		0.40	
CHEMICAL OTV (40 MT CLASS, ¼ DDT&E)		0.40	0.3
CREW ROTATION (75 PERSON CARRIER)			
	DDT&E	0.15	0.15
	25 SHUTTLE LAUNCHES (OVER 3 YEARS)	0.50	
RECTENNA (ONE MEGAWATT OUT)		0.70	
SUBTOTAL		14.43	6.05
WITH 15% FOR OPERATIONS, MICSCELLANEOUS		16.57	

SIZE EFFECT ON PRECURSOR PROGRAM COST

Three sizes for precursor SPS were investigated in this study. The 2-bay unit is 0.78% of the full size SPS. The 4-bay unit baselined is 1.56% of the full size SPS, and a 32-bay (one module) unit which is 12.5% of the full size SPS. Because it is considered fundamental that one of the major functions of the precursor unit is to develop and demonstrate the construction base "throughput" required for commercial viability of the SPS, each of these precursor unit sizes has associated with it a 1/8 size construction base segment which is literally a segment of the full size construction base. Because of this and other scaling effects there is relatively little cost difference between the 2-bay and 4-bay unit. If it is further baselined that 1 megawatt of useful power is to be produced by the test rectenna the smaller unit actually has a higher total program cost. Because of the lower central beam strength of the smaller precursor (approximately 5 microwatts per square centimeter) it requires a much larger test rectenna to generate the 1 megawatt. The one module large precursor SPS requires only a very small rectenna to generate one megawatt.

Size Effect on Precursor Program Cost



ALL HAVE:

- ≈"12%" CONST BASE
- GSO SUPPORT STATION
- SELF POWER TRANSFER
- CHEMICAL OTV

OTHER PRECURSOR SIZE OPTIONS

This chart shows the major effects of selecting the smaller and larger precursor units investigated. The 12.5% unit is one complete SPS module and would allow duplication of this large SPS assembly, certainly a major contributer to demonstration of the full satellite construction capability. However, its larger size and mass will require a much more ambitious associated launch program. For example, if the shuttle were to be used exclusively, over 800 shuttle flights would be required, and even with the higher payload capability of the shuttle derivative launch vehicle, over 200 flights are required. Total precursor program cost is estimated at approximately $28 billion. This larger size unit allows a central beam strength of over 1 milliwatt per square centimeter, much stronger than that producible by the 4-bay unit. Consequently, a very small rectenna would be sufficient to produce the 1 megawatt output baselined. Conversely, the 0.78%, 2-bay, precursor would allow a much more modest launch program and might be accomplished by the shuttle alone. However, the full high voltage output of the solar arrays would be achievable only with some cell string configuration other than that baselined for the full size SPS. The lower central beam strength requires a much larger rectenna if the full 1 megawatt is to be produced.

Other Precursor Size Options

- "12.5%": ONE SPS MODULE (32 BAYS)
  - EXACT DUPLICATION OF EVENTUAL SPS CONSTRUCTION ITEM.
  - 810 SHUTTLE FLIGHTS OR 225 FLIGHTS OF LIQUID BOOSTER/ CAPSULE SSME SHUTTLE DERIVATIVE
  - APPROXIMATELY $28B PROGRAM.
  - 1.37 mW/cm^2 (1,370 $\mu W/cm^2$) CENTRAL BEAM STRENGTH.
  - $50M FOR A 1.0 mW ANTENNA
- 0.78%: TWO SPS BAYS
  - CONSTRUCT ALL SPS "ELEMENTS".
  - FULL VOLTAGE ONLY BY EITHER SUBSCALE CELLS OR RE-ROUTED STRINGS. (NOT DUPLICATION OF PLASMA SUSCEPTIBILITY).
  - 95 SHUTTLE FLIGHTS OR 31 FLIGHTS OF LIQUID BOOSTER/ CAPSULE SSME SHUTTLE DERIVATIVE.
  - 1.40 $\mu W/cm^2$ CENTRAL BEAM STRENGTH
  - $3.8B FOR A 1.0 mW ANTENNA.

PRECURSER GOALS

The goals that are thus emerging from the study may be summarized as follows. The precurser will operate in geosynchronous orbit and develop for a test period (probably one year) approximately 1 megawatt of useful power; while doing so, an environmentally acceptable microwave beam will be continuously and accurately pointed into the required receiving area. By accomplishing this test, all major elements of the full size SPS will be demonstrated, including the high required construction "throughput" necessary to assemble satellites on orbit at the required rate. The precurser satellite is relatively small compared to a full size SPS and, consequently, is capable of being launched along with its construction base by the space shuttle or a derivative of the space shuttle.

Precursor Goals

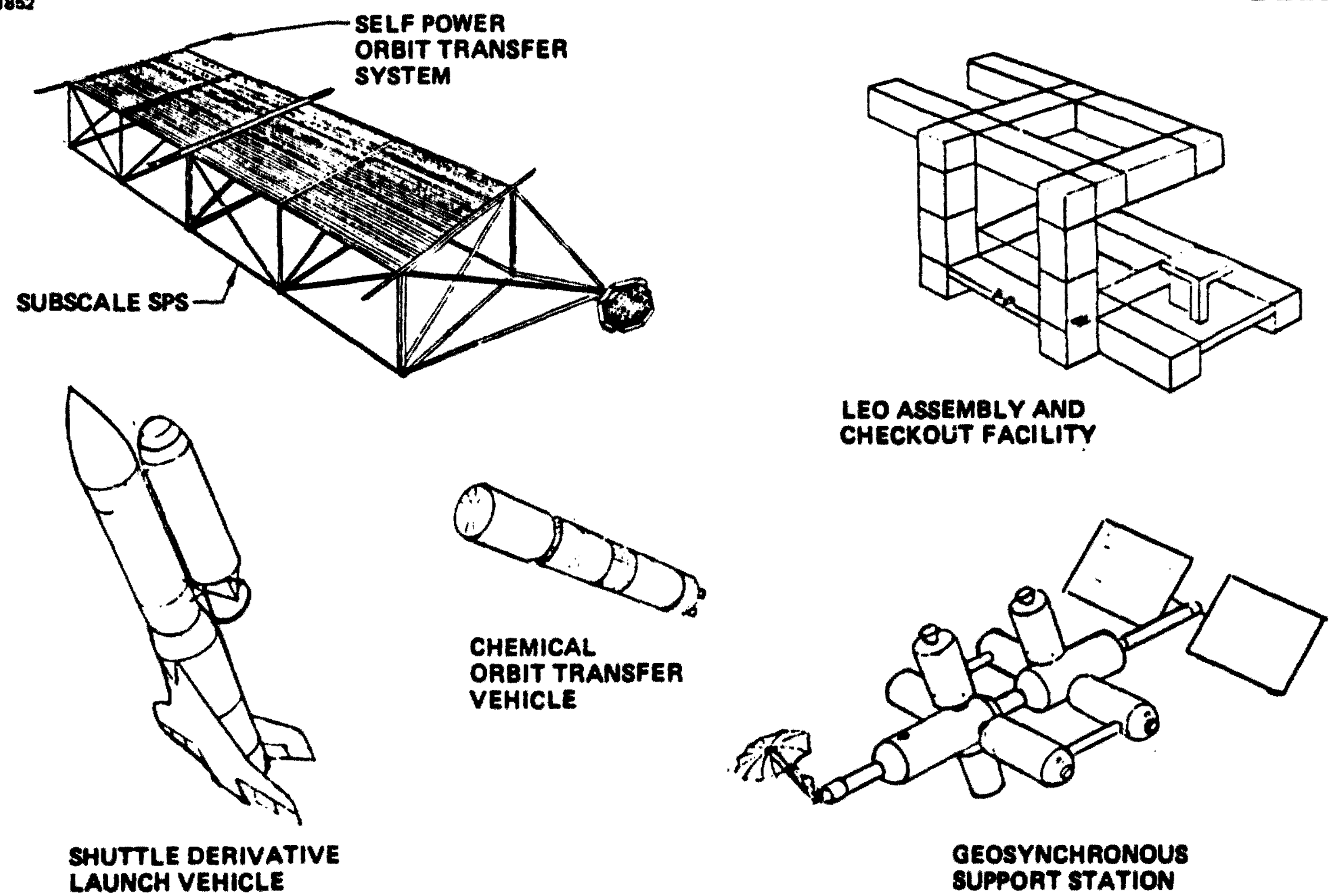
- ALLOWS LONG-TERM (APPROXIMATELY 1 YEAR) DEMONSTRATION THAT MICROWAVE BEAM CAN BE CONTINUOUSLY POINTED INTO REQUIRED AREA
- BEAM TO FIT INTO GOVERNMENT RESERVATION (e.g., WHITE SANDS PROVING GROUND); SIGNAL STRENGTH AT EDGE TO BE BELOW SOVIET EXPOSURE STANDARD (10 $\mu W/cm^2$)
- CENTRAL BEAM STRENGTH TO ALLOW WORKERS EXPOSURE (UNDER 1/10 CURRENT U.S. STANDARD)
- SIGNIFICANT POWER OUTPUT ($\gg$ 1 MW)
- TO REPRESENT ELEMENTS OF FULL-SIZE SPS
  - POWER GENERATION
  - TRANSMITTER
  - ROTARY JOINT
  - TRANSFER TO HIGH ORBIT
  - CONSTRUCTION "THROUGHPUT"
  - POWER RECEPTION
  - MAINTENANCE
- SHUTTLE DERIVATIVE ADEQUATE FOR ALL LAUNCHES

MAJOR ELEMENTS OF THE PRECURSOR PROGRAM

These are the larger elements of the 1.56% precursor program baselined in this study. In addition to the precursor satellite itself with its transmitter, the self-power orbit transfer system is required. A low earth orbit assembly and checkout facility (construction base) is also a major element of the program. Some shuttle derivative launch vehicle, preferably with an eight meter diameter cargo capability, should be provided in order to reduce launch cost not only for the precursor program but also subsequent shuttle type operations. A chemical orbit transfer vehicle is required to place the manned geosynchronous orbit support station which is used during the operational test period of the precursor satellite.

Major Elements of the Precursor Program

SPS-1852



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PHASE III FINAL REVIEW
MARCH 1, 2, 1978

SOLAR POWER SATELLITE

GENERAL ELECTRIC
SPACE DIVISION

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MICROWAVE POWER TRANSMISSION

SCOPE OF GE'S PHASE CONTROL SYSTEM STUDY

GENERAL ELECTRIC COMPANY'S STUDY EFFORT IN THE PHASE CONTROL CIRCUIT AREA WAS CONCENTRATED ON THE DEVELOPMENT OF SYSTEM ARCHITECTURE, DEFINITION OF MAIN REQUIREMENTS AND ANALYSIS OF THE POWER TRANSFER EFFICIENCY OF THE SYSTEM.

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SCOPE OF GE'S PHASE CONTROL SYSTEM STUDY

- SURVEY AND INTEGRATE PRIOR EFFORTS
- COMPARE ALTERNATE APPROACHES
- DEFINE MAIN SYSTEM PARAMETERS ON THE BASIS OF TRADE-OFF ANALYSIS
- ANALYZE OPERATION OF SYSTEM AND DETERMINE ERRORS CAUSING POWER TRANSFER INEFFICIENCY.
- PROVIDE DATA FOR COST MODEL.

BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

REALISTIC AND RELATIVELY EASILY MAINTAINABLE ORBIT PARAMETERS WERE ASSUMED. ORBIT INCLINATION ANGLE LIMIT NECESSITATES A DAILY MECHANICAL OR ELECTRICAL TILT ALIGNMENT OF SUBARRAY PATTERN.

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BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

ORBIT

EXCENTRICITY	4×10^{-4}
INCLINATION	2.2°
LONG TERM POSITION DRIFT	$\pm$ 10 km

BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

THE GOVERNING CHARACTERISTICS FOR THE SPACE SEGMENT OF THE PHASE CONTROL SYSTEM WERE SELECTED ON A CONSERVATIVE BASIS. WITH EVOLVING CONSTRUCTION TECHNOLOGY OF THE ANTENNA STRUCTURE THE NUMBER OF SUBARRAYS MAY BE REDUCED RESULTING IN A SIGNIFICANT REDUCTION OF PHASE CONTROL SYSTEM COMPLEXITY.

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BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

SPACE SEGMENT

NOMINAL SPACECRAFT ANTENNA DIAMETER	1 km
NUMBER OF SUBARRAYS	10000
NOMINAL SUBARRAY AREA	
TRANSMIT	10^2 m^2
RECEIVE	5.33^2 m^2 (max)
ARRAY MISALIGNMENT	$.15^\circ$ (max)
POLARIZATION	LINEAR
NOMINAL TRANSMIT POWER OF ARRAY	10^{10} w
MAXIMUM TRANSMIT POWER PER TUBE	125 kw
MAXIMUM NOISE DENSITY FOR 125 KW TUBE 90 MHz FROM CENTER OF BAND	-107.8 dBw/Hz
RECEIVER IF BANDWIDTH	1 MHz
PHASE DISTRIBUTION NETWORK ARCHITECTURE	3 LAYERS

BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

A TWO TONE PILOT SIGNAL TRANSMITTED BY A THREE ELEMENT EARTH STATION ANTENNA ARRAY WAS SELECTED ALLOWING A SIMPLIFICATION OF THE SPACE SEGMENT AND COMPENSATION OF SYSTEMATIC POINTING ERRORS IN THE POWER BEAM.

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BASIC ASSUMPTIONS FOR SYSTEM DESIGN OF PHASE CONTROL NETWORK

EARTH SEGMENT

EARTH STATION SITE	WITHIN CONTINENTAL US
UPLINK FREQUENCY (f_U)	2460 MHz
DOWNLINK FREQUENCY (f_D)	2450 MHz
UPLINK MODULATION FREQUENCY (f_1)	100 MHz
NUMBER OF PILOT ANTENNAS	3
NOMINAL RECTENNA DIAMETER	10 km
NUMBER OF MONITORING ANTENNAS	4 (MIN)

FUNCTIONS IN PHASE CONTROL CIRCUIT FOR RETRODIRECTIVE SPS ANTENNA

THE NECESSARY FUNCTIONS REQUIRED FOR A RETRODIRECTIVE PHASE CONTROL SYSTEM WERE IDENTIFIED. THESE FUNCTIONS ARE INDEPENDENT OF THE ACTUAL CIRCUIT IMPLEMENTATION.

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FUNCTIONS IN PHASE CONTROL CIRCUIT FOR RETRODIRECTIVE SPS ANTENNA

- PILOT SIGNAL GENERATION, CONTROL AND TRANSMISSION AT f_U FROM EARTH STATION.
- PILOT PHASE RECEPTION AND DOWN CONVERSION AT SPACECRAFT SUB-SUBARRAYS.
- REFERENCE PHASE GENERATION, DISTRIBUTION AND REGENERATION.
- PILOT PHASE TRANSMISSION FOR PHASE REGENERATION AND CONJUGATION.
- CONJUGATED PHASE TRANSMISSION AND UP CONVERSION TO f_D FOR PA SYSTEM.
- PHASE CONTROLLED POWER AMPLIFICATION AND TRANSMISSION BY SPACECRAFT SUBARRAYS.
- MONITORING OF RECEIVED SIGNAL ON GROUND.

RETRODIRECTIVE SPS PHASE CONTROL SYSTEM

THE RETRODIRECTIVE PHASE CONTROL SYSTEM IS BASED ON THE USE OF A GROUND GENERATED REFERENCE PHASE AGAINST WHICH THE PHASE OF THE RECEIVED SIGNALS OF THE SUBARRAYS ARE CONJUGATED FOR THE TRANSMITTED SIGNALS. THE REFERENCE PHASE IS DISTRIBUTED ON A RETURNABLE TIME BASIS, WHICH IS INDEPENDENT ON THE VARIATIONS OF ELECTRICAL LENGTH IN THE PHASE DISTRIBUTION TRANSMISSION LINES. THE PRACTICAL IMPLEMENTATION OF THIS PRINCIPLE REQUIRES THAT THE CONJUGATOR FOR A GIVEN SUBARRAY IS LOCATED AT THE RECEIVER OF THE NEXT HIGHER LEVEL SUBARRAY IN THE PHASE DISTRIBUTION TREE.

GENERAL ELECTRIC

RETRODIRECTIVE SPS PHASE CONTROL SYSTEM

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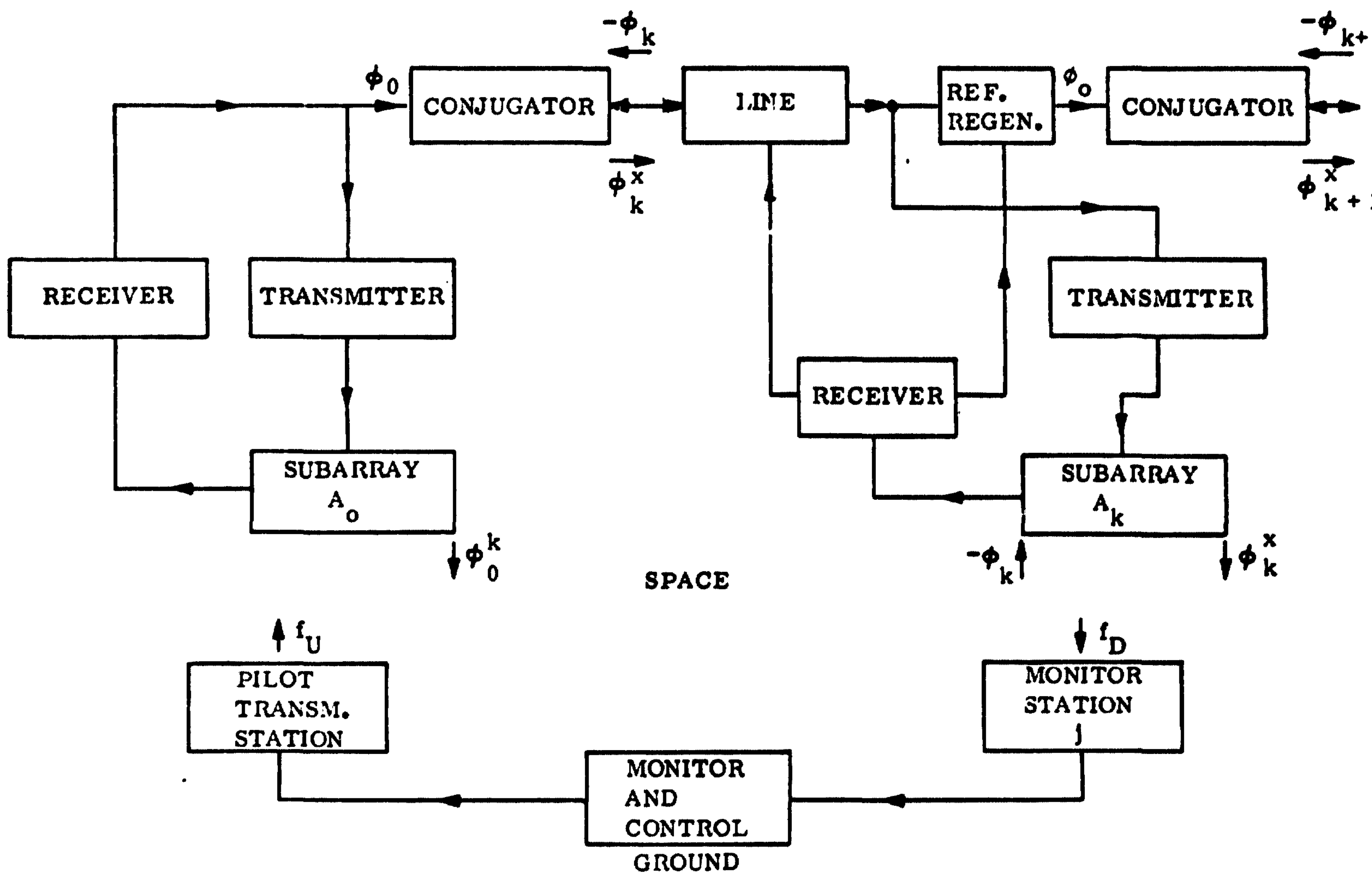

THREE PILOT ANTENNA CONTROL SYSTEM

ON THE GROUND THE POSITION OF THE RECEIVED BEAM IS MONITORED AND THE EFFECTIVE PHASE CENTER OF THE TRIANGULARLY CONFIGURED PILOT ANTENNA ARRAY IS VARIED IN SUCH A WAY THAT THE BEAM CENTER IS KEPT AT THE CENTER OF THE RECTENNA.

GENERAL ELECTRIC

THREE PILOT ANTENNA CONTROL SYSTEM

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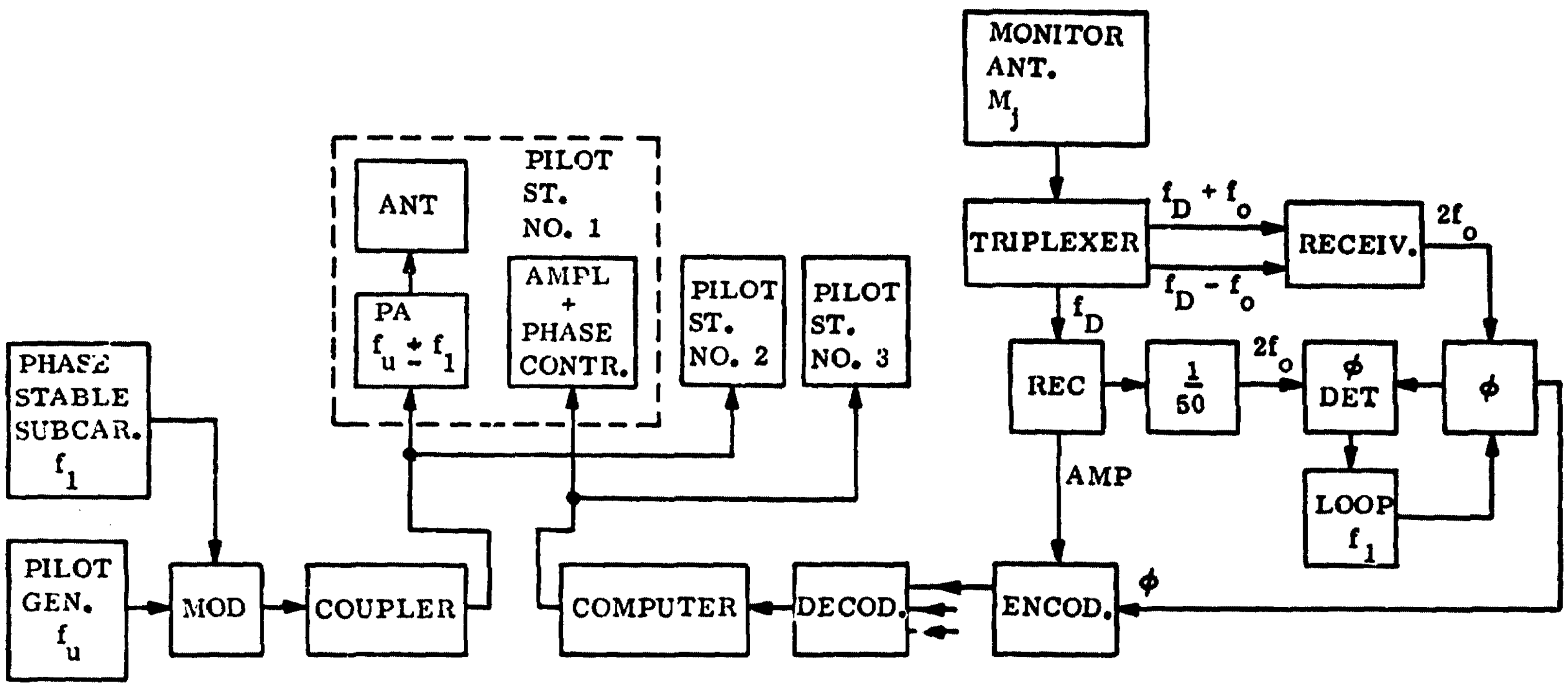

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PHASE COHERENT RECEIVER ALTERNATIVE

SEVERAL PHASE COHERENT RECEIVER TYPES ARE USABLE, AMONG WHICH THE TWO TONE RECEIVER WITH A FIXED LO IS THE SIMPLEST AND PRODUCES THE RECEIVED PHASE AT THE SUBARRAY AT A CONVENIENTLY LOW INTERMEDIATE FREQUENCY.

GENERAL ELECTRIC

PHASE COHERENT RECEIVER ALTERNATIVE

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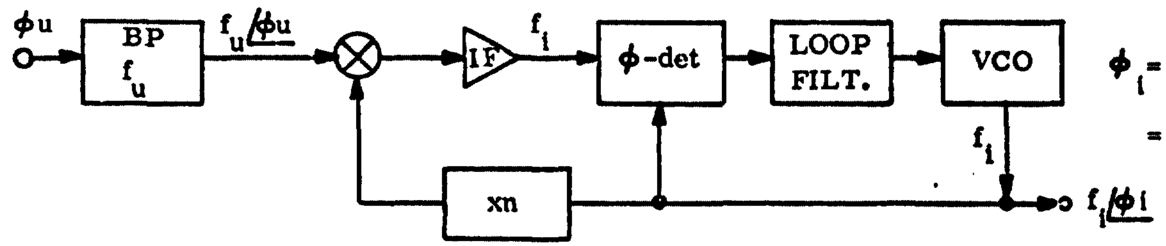
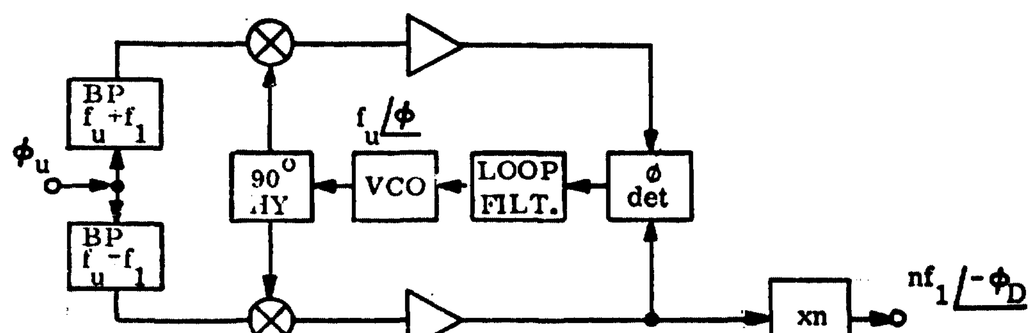


$$\phi_i = \frac{\phi_u}{1+n} = \frac{1}{1+n}\,\frac{-w_u r}{c} = \frac{1}{1+n}(-\phi_u)$$

SINGLE TONE PHASE LOCKED LOOP (if)



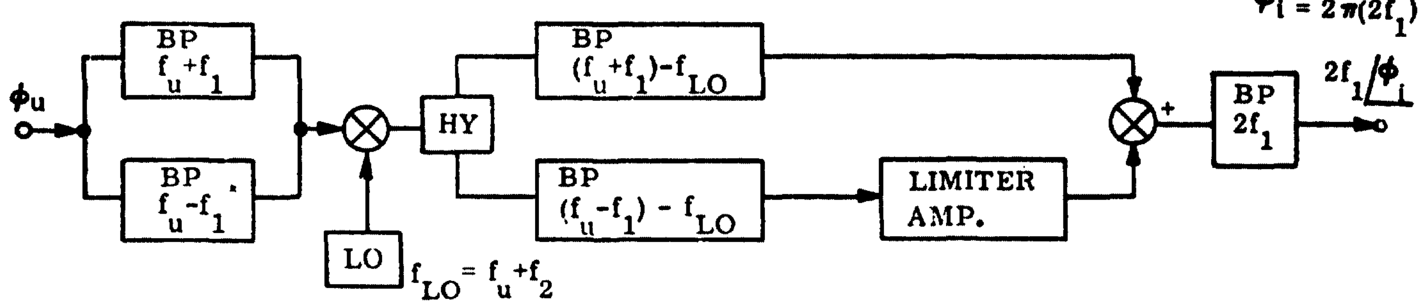
$$\phi = -\pi\left[(f_u+f_1)t_+ + (f_u-f_1)t_-\right] - \pi k$$

$$f_D = nf_1$$

$$\phi_D = n\phi$$

TWO TONE PHASE LOCKED LOOP (rf) AND PHASE CONJUGATOR

$$\phi_i = 2\pi(2f_1)\frac{r}{c}$$



TWO TONE RECEIVER WITH FIXED LO.

PHASE DISTRIBUTION NETWORK LAYOUT

A THREE LAYER i-f REFERENCE PHASE DISTRIBUTION SYSTEM WAS SELECTED TO MINIMIZE CIRCUIT LOSSES, WEIGHT AND PRACTICAL ERRORS ASSOCIATED WITH THE TERMINATING IMPEDANCE VARIATIONS. THIS SYSTEM REQUIRES ONLY $n_1(n_2 + 1)$ PHASE REGENERATORS ($n_1 = 19$, $n_2 = 23$) AND AN i-f DIPLEXER AT THE END OF EACH TRANSMISSION LINES, WHICH ARE USED AT $2f_1$ AND $4f_1$ FREQUENCIES FOR THE BACK AND FORTH TIME RETURNING OF THE REFERENCE SIGNALS.

GENERAL ELECTRIC

PHASE DISTRIBUTION NETWORK LAYOUT

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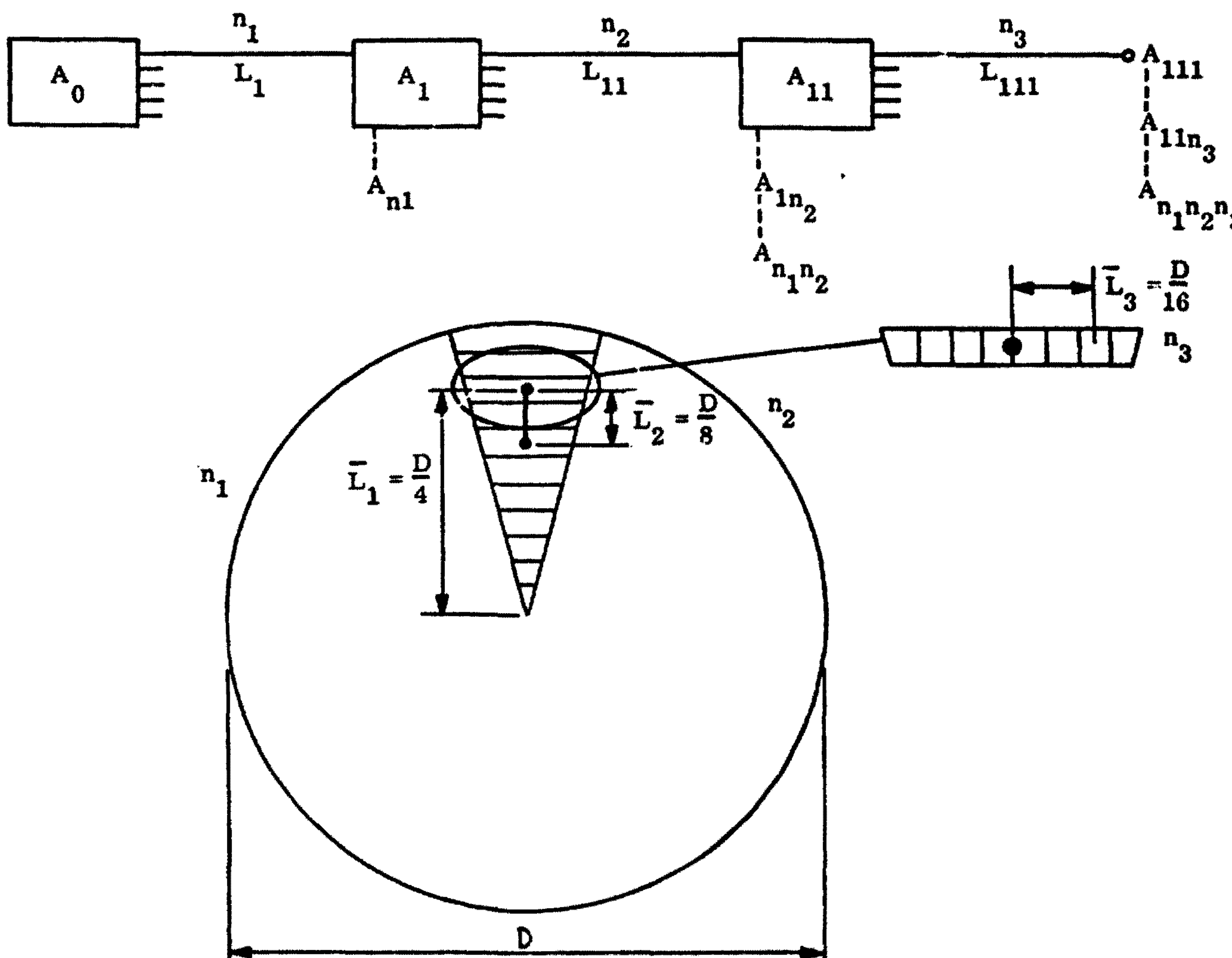

ALTERNATIVE PHASE REGENERATORS

IN THE REFERENCE PHASE DISTRIBUTION PROCESS A REGENERATION OF THE ORIGINAL REFERENCE PHASE IS REQUIRED AT EVERY NODE FROM THE RECEIVED SUBARRAY SIGNAL AND FROM THE CONJUGATED SUBARRAY SIGNAL. THIS "REGENEATOR" REQUIRES A NUMBER OF MIXERS AND FREQUENCY DIVIDERS. WHEN THE DIVIDERS ARE USED IN A LARGER OVERALL SYSTEM CARE MUST BE TAKEN BY THEIR PROPER SYNCHRONIZATION.

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ALTERNATIVE PHASE REGENERATORS

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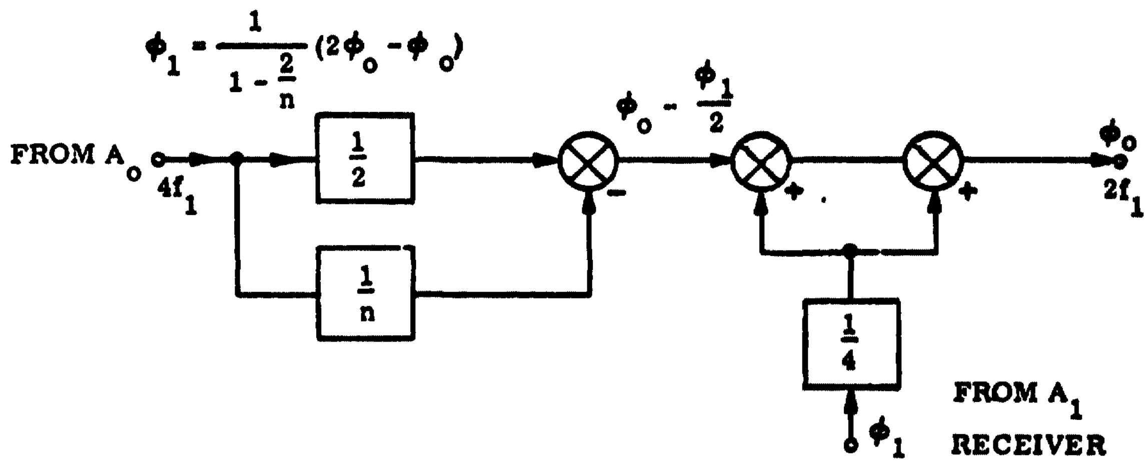
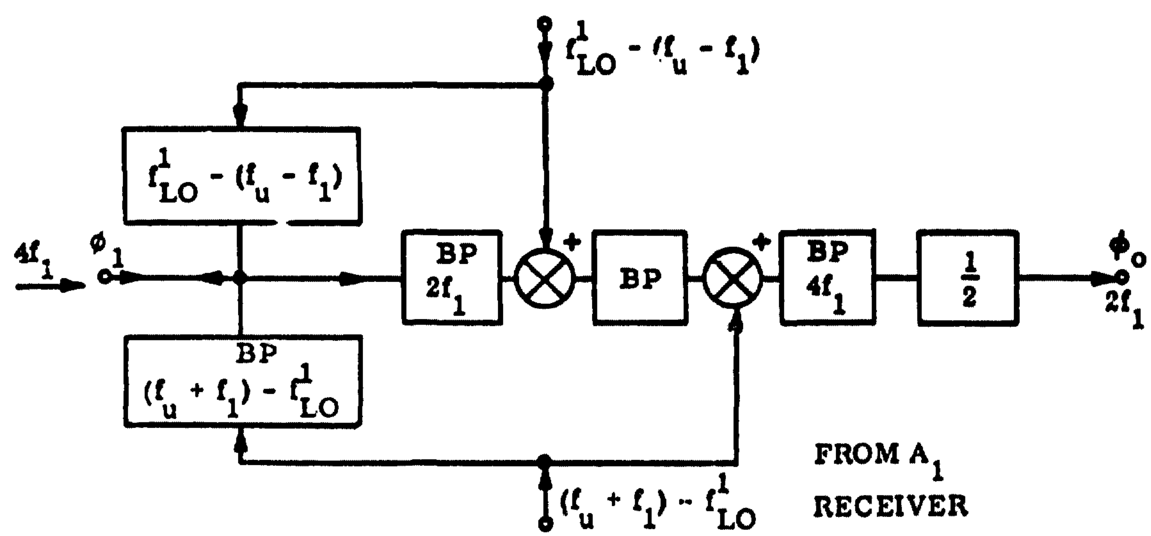


SINGLE TONE PHASE REGENERATOR

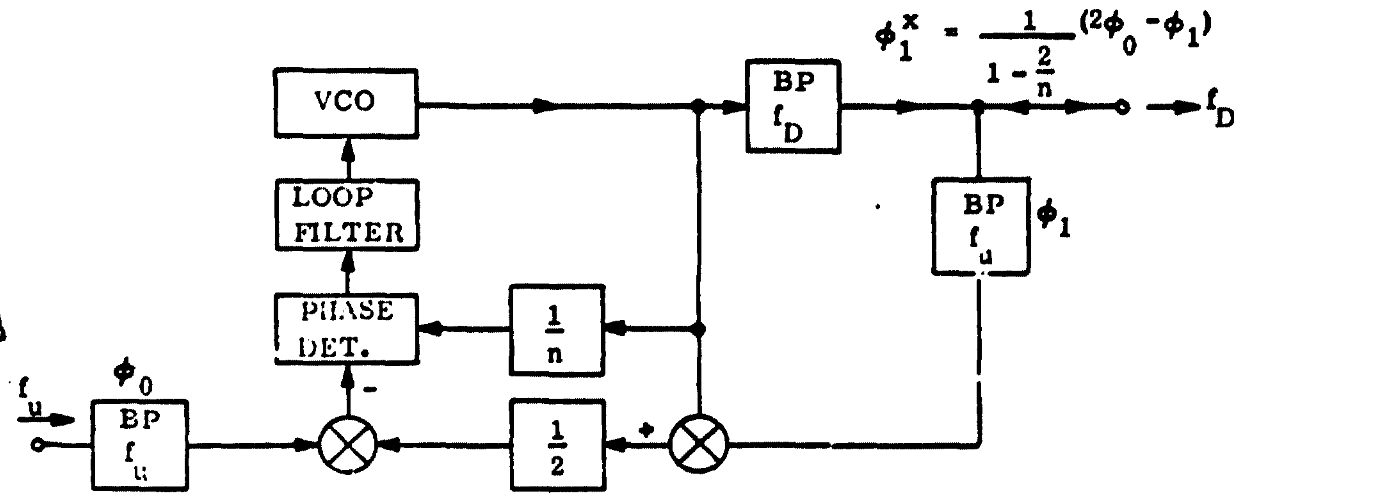


TWO TONE PHASE REGENERATOR

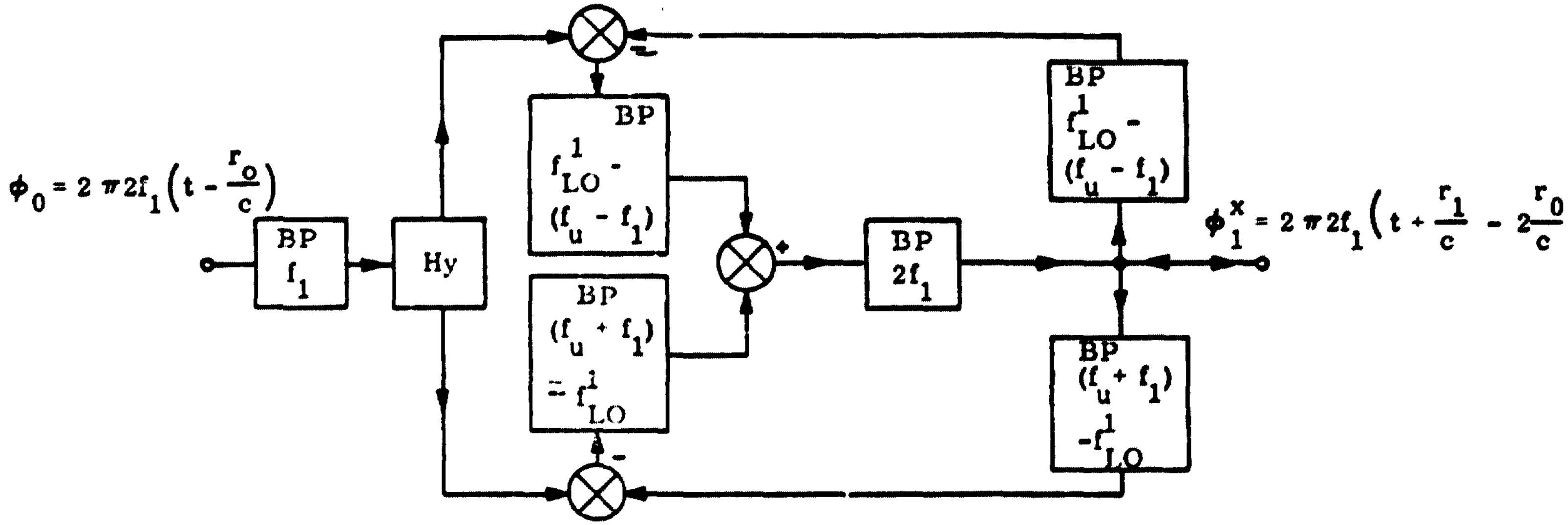
ALTERNATIVE PHASE CONJUGATORS

"EXACT" AND "APPROXIMATE" PHASE CONJUGATORS CAN BE USED FOR THE PRESENT PURPOSE. THE FIGURE SHOWS TWO OF THE "EXACT" TYPES OF CONJUGATORS, WHEN THE FREQUENCY TRANSLATION IS ACHIEVED WITHOUT AN ERROR IN THE CONJUGATION. CIRCUITS WHICH USE FREQUENCY DIVIDERS MUST BE SYNCHRONIZED IN ORDER TO AVOID π PHASE AMBIGUITY.

ALTERNATIVE PHASE CONJUGATORS



PHASE LOCKED LOOP TYPE PHASE CONJUATOR (rf)

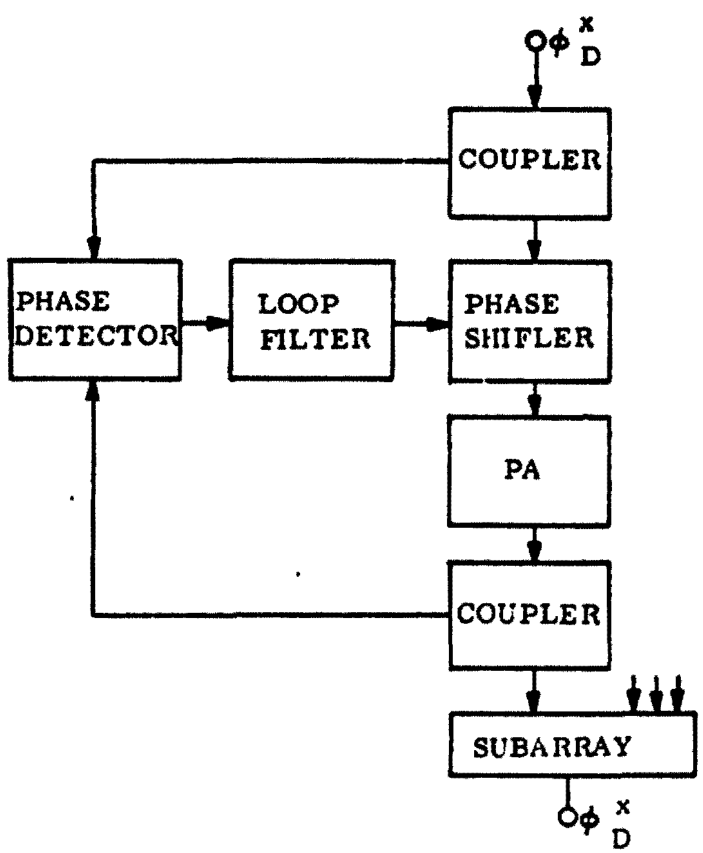


TWO TONE TYPE PHASE CONJUATOR (if)

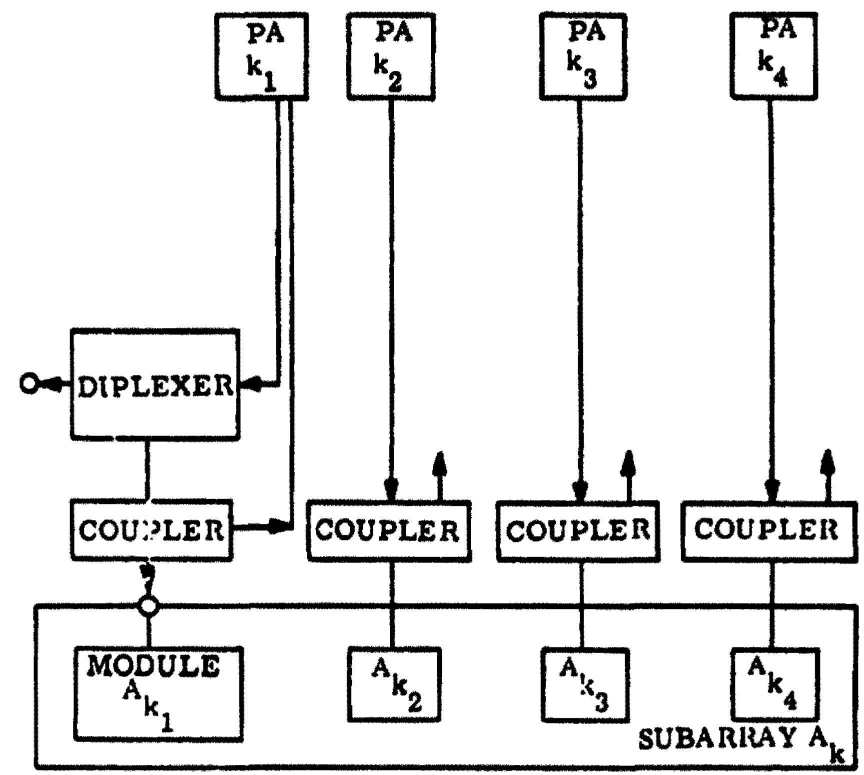
TRANSMITTER BLOCK DIAGRAM

THE TRANSMITTERS OPERATING INTO A SUBARRAY ARE INDEPENDENT FROM EACH OTHER, BUT RECEIVE THEIR INPUT PHASE FROM A COMMON PHASE CONJUGATOR. ADDITIONALLY ONE TRANSMITTER MODULE SHARES A PART OF THE SUBARRAY SURFACE WITH THE RECEIVER ESTABLISHING THE PHASE FOR THE ENTIRE SUBARRAY. WHEN ELECTRONIC STEERING OF THE SUBARRAY PATTERN IS DESIRABLE FOR INCLINATION ANGLE COMPENSATION THE NORTH-SOUTH WIDTH OF A TRANSMIT ARRAY MODULE HAS TO BE RESTRICTED TO ABOUT .7 m AND THE INPUT PHASE TO THE PHASE DETECTORS OF THE TRANSMIT PHASING CIRCUIT HAS TO BE MODULATED BY A SMALL, CALCULATED ERROR SIGNAL WITH 24 HOURS PERIODICITY.

TRANSMITTER BLOCK DIAGRAM



PHASING CIRCUIT FOR ONE HIGH POWER TRANSMITTER MODULE



OVERALL TRANSMITTER SYSTEM.

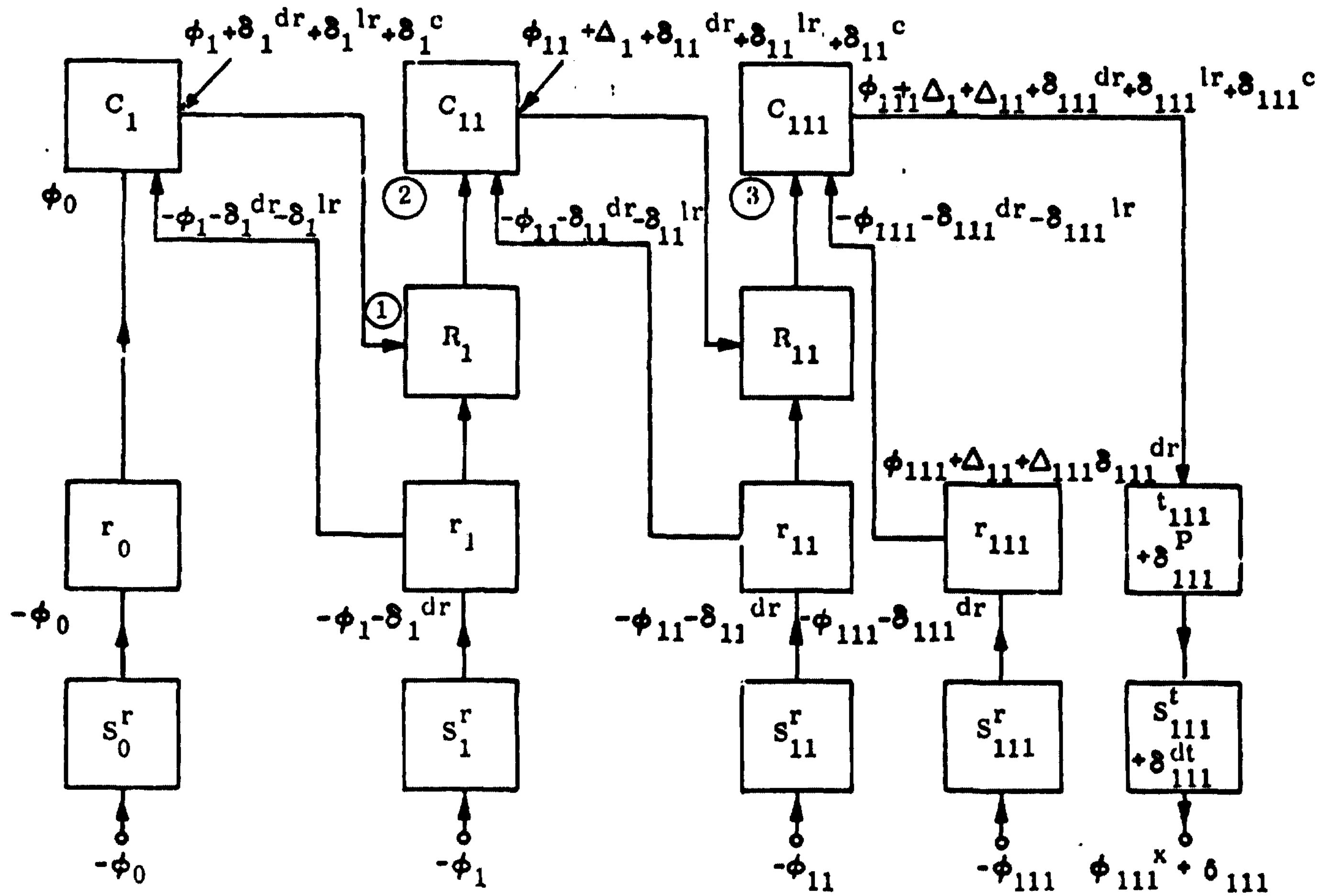
THREE LAYERS PHASE DISTRIBUTION NETWORK PHASE ERROR BUILD UP

RANDOM AMPLITUDE AND PHASE, AND SYSTEMATIC POINTING AND AMPLITUDE ERRORS ARE AFFECTING THE RESULTANT POWER TRANSFER EFFICIENCY OF ANTENNA. EFFICIENCY IS CONSIDERED BETWEEN THE OUTPUT PLANE OF THE SPACE ANTENNA AND THE INPUT PLANE OF THE RECTENNA.

GENERAL ELECTRIC

THREE LAYERS PHASE DISTRIBUTION NETWORK PHASE ERROR BUILD UP

space division



RSS PHASE ERROR $\delta=\left[3\delta_c^{\,2}+6\delta_1^{\,2}+2\delta_d^{\,2}+\delta_p^{\,2}\right]^{1/2}$

(1) $\phi_1+\delta_1^{dr}+\delta_1^{lr}+\delta_1^{c}+\delta_1^{lt}$

(2) $\phi_0+\delta_1^{dr}+\delta_1^{lr}+\delta_1^{c}+\delta_1^{lt}+\delta_1^{dr}$
$=\phi_0+\Delta_1$

(3) $\phi_0+\Delta_1+\delta_{11}^{lr}+\delta_{11}^{c}+\delta_{11}^{lt}$
$=\phi_0+\Delta_1+\Delta_{11}$

ILLUMINATION ERRORS AFFECTING ANTENNA EFFICIENCY

FOR THE SELECTED THREE LAYER PHASE DISTRIBUTION NETWORK APPROXIMATELY 95% OF ALL SUBARRAYS BELONG TO THE THIRD LAYER, THUS THEIR PHASE CONJUGATION ERRORS WILL DOMINATE THE RESULTANT PHASE CONJUGATION ERROR. THE FIGURE SHOWS THE ERROR BUILD UP FOR SUCH AN ELEMENT. THE ERROR IS A FUNCTION OF ERRORS ASSOCIATED WITH THE CONJUGATORS, CONNECTING LINE MISMATCHES, r-f DIPLEXER DIFFERENTIALS AND ERRORS ASSOCIATED WITH THE TRANSMITTER PHASING CIRCUIT OF ALL THE ELEMENTS WHICH ARE ASSOCIATED WITH THE PARTICULAR TRANSMIT SUBARRAY.

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ILLUMINATION ERRORS AFFECTING ANTENNA EFFICIENCY

RANDOM		SYSTEMATIC	
PHASE	AMPLITUDE	POINTING	AMPLITUDE
PHASE JITTER (f_u, f_i)	TRANSMIT POWER SUBARRAY ROTATION	DOPPLER FREQUENCY SHIFT	ILLUMINATION QUANTIZATION
TRANSMITTER NOISE			POLARIZATION ROTATION
CONJUGATOR (δ_c)		ABERRATION	
LINE MATCH DIFFERENTIALS (δ_d)		IONOSPHERIC DIFFERENTIAL	
DIPLEXER MATCH DIFFERENTIALS (δ_d)		ATMOSPHERIC DIFFERENTIAL	
TRANSMITTER PHASING (δ_p)			
DIFFERENTIAL DOPPLER			

RANDOM ERRORS FOR 3 LAYERS PHASE DISTRIBUTION

THE RANDOM PHASE ERRORS ARE DOMINATED BY THE i-f DIPLEXER IMPERFECTIONS. THE AMPLITUDE ERRORS ARE DOMINATED BY ARRAY TILT.

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RANDOM ERRORS FOR 3 LAYERS PHASE DISTRIBUTION

PHASE ERRORS (deg)

SOURCE	DEG.
PHASE JITTER	1.13
TRANSMITTER NOISE (c/n = 30 db)	.36
CONJUGATORS (δ_c = .6°)	1.04
LINES (δ_l = 2.54°)	6.22
DIPLEXERS (δ_d = 1.81°)	2.56
TRANSMITTERS (δ_p = 1.6°)	1.60
DIFFERENTIAL DOPPLER (V_d = 6.25 m/s)	.18

PEAK: 13.09 RSS: 7.09

PHASE ERROR CAUSED LOSS: 1.53%

AMPLITUDE ERRORS (%)

SOURCE	PEAK	RMS
TRANSMIT POWER FLUCTUATION (1 db, rms)	10.64	2.38
ARRAY ROTATION ($L_s \leq 10$m, $\Delta\Theta_s$ = .15°)	13.50	1.41
PEAK:	24.14	RSS: 2.51

AMPLITUDE ERROR CAUSED LOSS:

For $\Delta\theta_s$ = .15° 2.51%

$\Delta\theta_s$ = .05° 1.34%

SYSTEMATIC ERRORS FOR 3 .YERS PHASE DISTRIBUTION

THE SYSTEMATIC ERRORS ARE DOMINATED BY PROPAGATION ERRORS. ALTHOUGH THESE MAY SHOW UP IN A SMALL PERCENTAGE OF TIME ONLY THEIR VALUE CAN BE SIGNIFICANT IF ONLY ONE PILOT ANTENNA IS USED.

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SYSTEMATIC ERRORS FOR 3 LAYERS PHASE DISTRIBUTION

POINTING ERRORS (deg)

SOURCE		1 PILOT STATION	3 PILOT STATION
DOPPLER ($i = 2.2^o$, $r^i_m = 13.6$ m/s, $2 f^i_{Dop} = 112$ Hz)		1.43×10^{-6}	7.15×10^{-8}
ABERRATION ($\dot{Z}^i_m = 100$ m/s)		19.3×10^{-6}	9.65×10^{-8}
IONOSPHERIC DIFFERENTIAL (.1° way refraction)		2.35×10^{-3}	1.17×10^{-4}
ATMOSPHERIC DIFFERENTIAL (.3° 1 way refraction, 2% iregularity)		6.00×10^{-3}	3.00×10^{-4}
Pointing error (deg)	PEAK	8.35×10^{-3}	4.175×10^{-4}
	RSS	6.44×10^{-3}	3.221×10^{-4}
Pointing loss (%)	PEAK	1.19	.60
	RSS	.92	.46

AMPLITUDE ERRORS (%)

QUANTIZATION	
16 LEVEL DISTRIBUTION	.078
8 LEVEL DISTRIBUTION	.312
FARADAY ROTATION (WORST YEAR)	.48

SUMMARY OF LOSSES

NET LOSS IN THE ANALYZED SECTION OF THE OVERALL SYSTEM CAN BE KEPT A 3.65% RMS LEVEL. FARADAY ROTATION LOSS, ALTHOUGH IT IS SMALL, CAN BE FURTHER REDUCED BY ABOUT A FACTOR OF TWO ON THE AVERAGE, IF THE POLARIZATION ANGLE OF THE SPACE ANTENNNA IS CORRECTED ONCE A YEAR.

SPACE DIVISION

SUMMARY OF LOSSES

SOURCE	LOSS (%)
RANDOM PHASE	1.53
RANDOM AMPLITUDE	1.34
SYSTEMATIC POINTING (3 PILOT STATION)	.46
SYSTEMATIC AMPLITUDE (8 LEVELS)	.32

RESULTANT LOSS ASSOCIATED TO SPACECRAFT ARRAY	3.65, RMS
FARADAY ROTATION (HOUSTON, WORST YEAR)	.48%, "AVERAGE" PEAK.

MULTI TONE PHASE COMPUTING SPS PHASE CONTROL SYSTEM

THE MULTITONE PHASE COMPUTING SYSTEM IS BASED ON THE MEASUREMENT OF THE PHASE OF EACH SUBARRAY ON THE GROUND BY THE USE OF A SIGNATURE TONE ASSOCIATED WITH THE SUBARRAY. THE REQUIRED PHASE CORRECTION FOR THE SUBARRAY TRANSMITTER IS RETURNED BY THE USE OF AN UPLINK CONTROL CHANNEL. BOTH FREQUENCY AND TIME DIVISION IS USED TO REDUCE COMPLEXITY AND REQUIRED FREQUENCY BAND FOR THE TONES. APPROXIMATELY 25 MHz BANDWIDTH IS NEEDED FOR A 1 SEC PHASE UPDATING PERIOD.

GENERAL ELECTRIC

MULTI TONE PHASE COMPUTING SPS PHASE CONTROL SYSTEM

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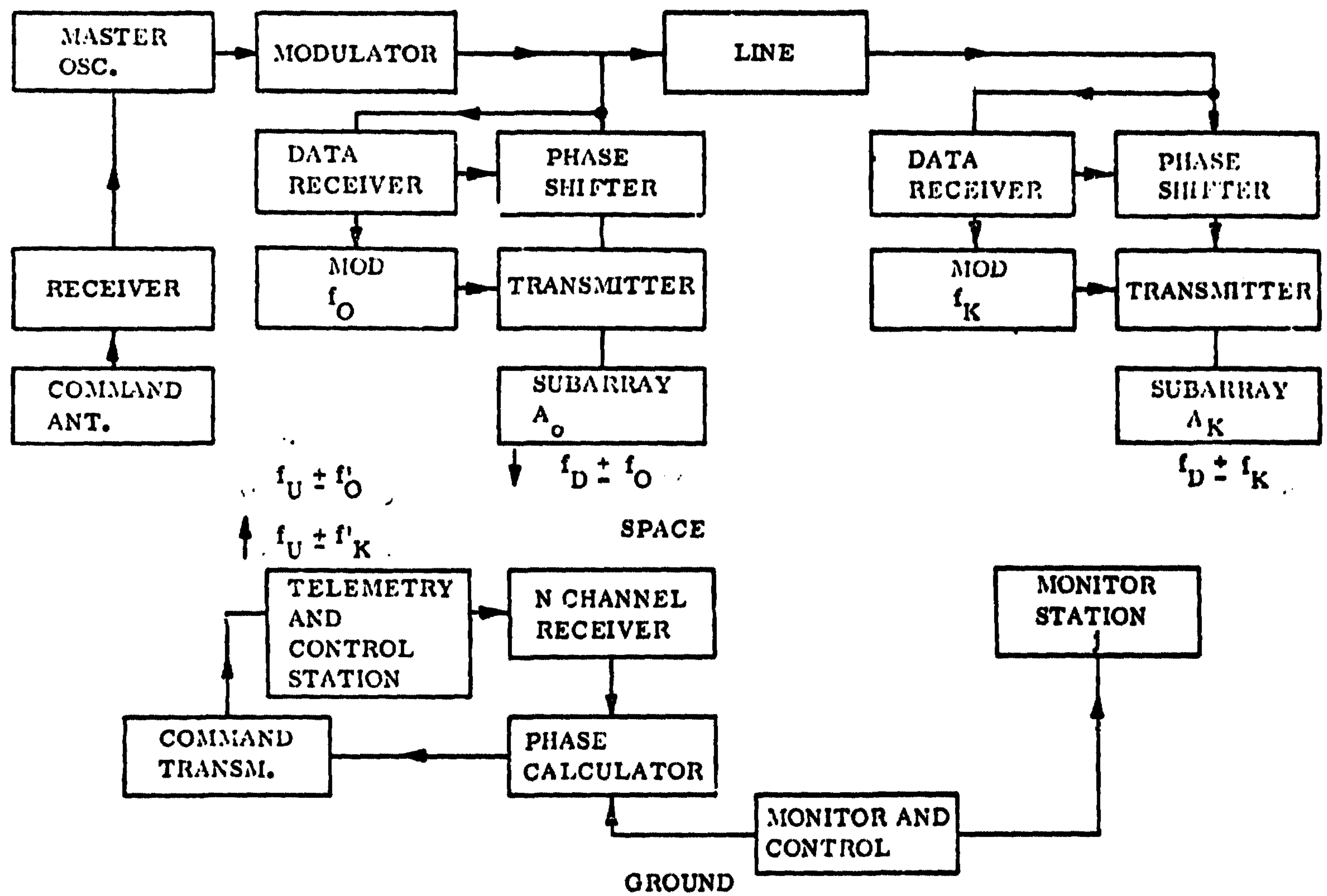

SYSTEM BLOCK DIAGRAM OF SPS PHASE CONTROL CIRCUIT IN PHASE COMPUTING MODE OF OPERATION (N TONE DOWNLINK)

THE IMPLEMENTATION OF THE N TONE PHASE COMPUTING SYSTEM REQUIRES A SINGLE TELEMETRY-CONTROL ANTENNA ON THE GROUND CLOSE TO THE MIDDLE OF THE RECTENNA AND ON N TONE RECEIVER SYSTEM. THE POWER BEAM AT THE SPACECRAFT TRANSMITTER IS MODULATED BY A LOW LEVEL TONE. THESE TONES ARE DETECTED ON THE GROUND AND THEIR PHASE IS COMPARED AGAINST THE PHASE OF AN ARBITRARILY SELECTABLE REFERENCE TONE. A TYPICAL SYSTEM MAY USE 100 TONES, 100 TIME DIVISION CHANNELS AND 1 SEC FOR THE COMPLETE PHASE UPDATING OF THE TRANSMITTERS.

GENERAL ELECTRIC

SYSTEM BLOCK DIAGRAM OF SPS PHASE CONTROL CIRCUIT IN PHASE COMPUTING MODE OF OPERATION (N TONE DOWN LINK)

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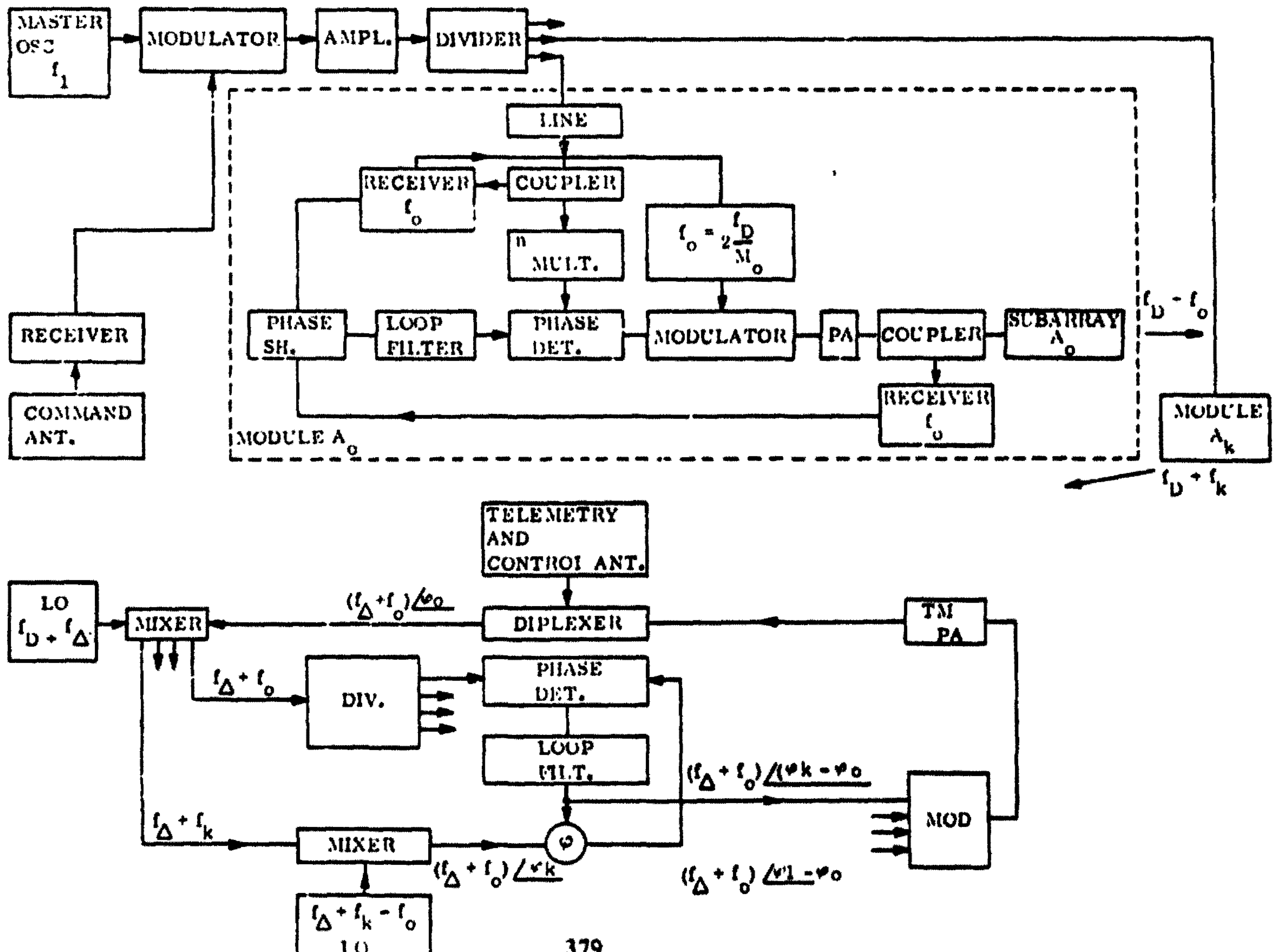

SUMMARY

A SUITABLE PHASE CONTROL SYSTEM CAN BE ESTABLISHED EITHER ON THE BASIS OF GROUND OR SPACE ORIGINATED PILOT SIGNALS. THE RESULTANT POWER TRANSFER EFFICIENCY IS A FUNCTION OF A LARGE NUMBER OF PARAMETERS, BUT A REALISTIC SET OF SYSTEM PARAMETERS RESULTS IN A TYPICAL 3.65% EFFICIENCY ASSOCIATED WITH THE PHASE CONTROL SYSTEM.

SPACE DIVISION

SUMMARY

- SEVERAL FEASIBLE SYSTEMS ARE ESTABLISHED
- RETRODIRECTIVE SYSTEM-COMPLEXITY IN SPACE
- PHASE COMPUTING SYSTEM-COMPLEXITY ON THE GROUND
- RETRODIRECTIVE SYSTEM HAS FAST RESPONSE, ACCEPTABLE ACCURACY BUT MAY BE MORE COSTLY AND LESS RELIABLE.
- MORE WORK IS NEEDED ON PHASE COMPUTING SYSTEM BEFORE FAIR COMPARISON CAN BE MADE.
- TRANSMISSION LOSS IN RETRODIRECTIVE SYSTEM CAN BE LESS THAN 3.65% WITHOUT FARADAY ROTATION LOSS.
- MAJOR LOSS FACTORS: TRANSMIT POWER FLUCTUATION, MISMATCHES, PROPAGATION ERRORS.
- FARADAY LOSS COULD REACH .48% DAILY PEAK IN WORST YEAR FOR STANDARD IONOSPHERE.
- EXPERIMENTAL WORK IS RECOMMENDED TO IMPROVE PHASE ERROR PREDICTION OF SUBSYSTEMS.
- UNIT COUNT IS OVER 10^6 IN RETRO SYSTEM. (TYPICAL UNITS: INTEGRATED CIRCUITS, PA TUBE, SUBARRAY PANEL, ETC.)
- CAPITAL INVESTMENT PER WATT CAPACITY VARIES SLOWLY FOR 6-8 Gw OUTPUT POWER RANGE.

GEOMETRY OF SPS PROPAGATION PATH

THE PENETRATION PATH THROUGH THE ATMOSPHERE AND THROUGH THE IONOSPHERE IS SMALL RELATIVE TO THE TOTAL PATH LENGTH. THESE YIELD MAXIMUM BENDING EFFECT, DUE TO THE ERROR ANGLE δ, OF APPROXIMATELY 0.02° (WHICH CORRESPONDS TO BEAM DEVIATION OF LESS THAN 10 METERS FROM THE CENTER OF THE RECTENNA).

GENERAL ELECTRIC

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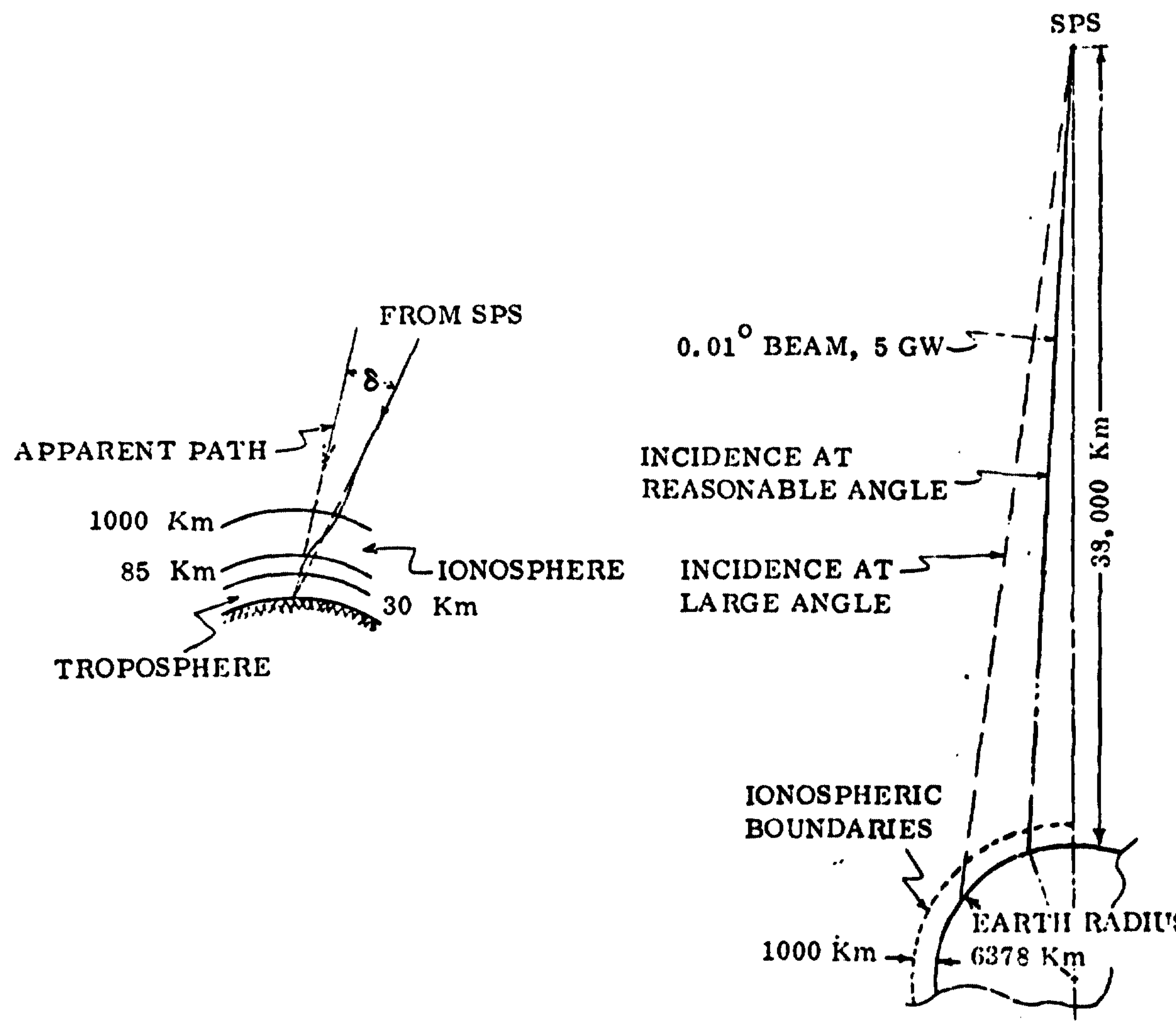
GEOMETRY OF SPS PROPAGATION PATH



SUMMARY OF ATMOSPHERIC EFFECTS ON SPS RADIATION PATTERN

- ATMOSPHERIC REFRACTION EFFECTS ARE NEGLIGIBLE.
- ATMOSPHERIC ABSORPTION IS APPROXIMATELY TWO PERCENT.
- 2.45 GHz IS PREFERABLE THAN 5.8 GHz BECAUSE OF DRASTIC RAIN ATTENUATION AT 5.8 GHz.

GENERAL ELECTRIC

SUMMARY OF ATMOSPHERIC EFFECTS ON SPS RADIATION PATTERN

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Cause	Effect		
Absorption	< 0.02%		
Accumulated Phase Front Perturbations		2.45 GHz	5.8 GHz
	Elevation 90° (Zenith)	106°	45°
	Elevation 20°	162°	69°
Scintillation Attenuation Effects	Elevation 90° (Zenith)	0.785 dB	0.35 dB
	Elevation 20°	1.172 dB	0.528 dB
Scattering	-27 dB Sidelobe level relative to the peak of radiation		

SUMMARY OF IONOSPHERIC EFFECTS ON SPS RADIATION PATTERN

IONOSPHERIC ABSORPTION IS NEGLIGIBLY SMALL.

MAXIMUM PHASE FRONT ERROR DUE TO PERTURBED IONOSPHERE IS 162° AT ELEVATION OF 20° AT A RATE OF CHANGE OF APPROXIMATELY 18°/SECOND.

GENERAL ELECTRIC

SUMMARY OF IONOSPHERIC EFFECTS ON SPS RADIATION PATTERN

space division

Cause	Effect			
Refraction, Unperturbed Atmosphere	Elevation 20°	Elevation error angle of 0.03° ± 0.006° on cloudy days Elevation error angle of 0.02° ± 0.002° on clear days		
	Elevation 90° (Zenith)	Elevation error angle of ± 0.006° on cloudy days Elevation error angle of + 0.002° on clear days		
Refraction, Perturbed Atmosphere	± 0.0036° peak angle of arrival fluctuations ± 10° peak phase error in incoming phase front			
Absorption, Without Rain		Humidity	2.45 GHz	5.8 GHz
	Elevation 90°	0%	1.2%	1.3%
		100%	1.2%	1.6%
	Elevation 20°	0%	1.4%	1.5%
		100%	1.4%	2%
30 mm/hr. Rain Absorption	Elevation 90°	100%	.42%	11.2%
	Elevation 20°	100%	1.25%	33.6%

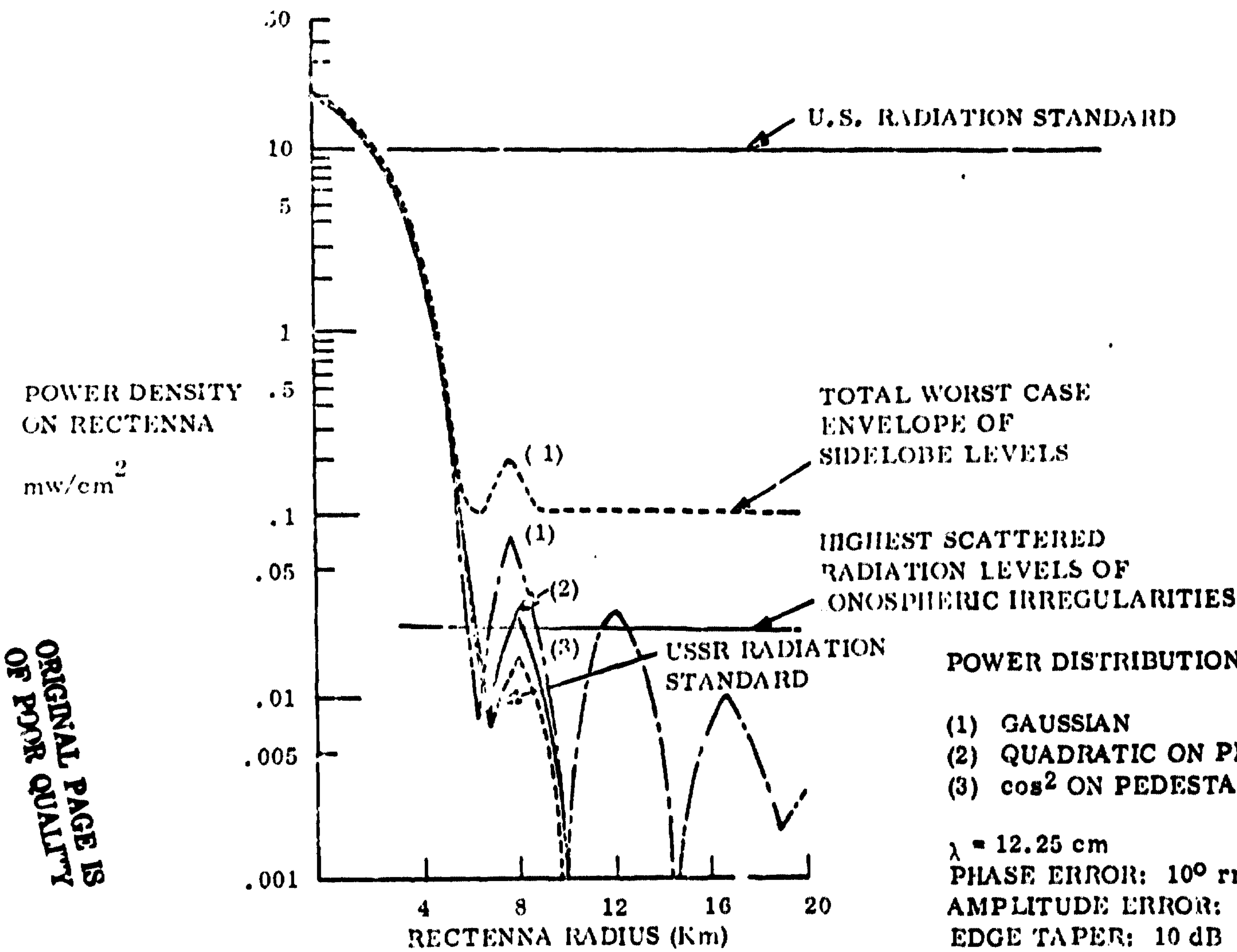
EFFECT OF PROPAGATION MEDIA ON SIDELOBE LEVEL
(VARIOUS POWER DISTRIBUTIONS ON SPACE ANTENNA)

MAXIMUM SCATTERED RADIATION LEVELS NEAR THE MAIN BEAM OCCUR WHEN THE PHASE ERRORS OF THE WAVEFRONT REACH THE VALUE OF ONE RADIAN. THE ENVELOPE OF THESE PEAK LEVELS IS SHOWN AT APPROXIMATELY 27 DB BELOW THE PEAK LEVEL OF RADIATION. WHEN THESE LEVELS INTERACT WITH THE WORST SIDELOBE LEVELS OF CURVE (1) (GAUSSIAN TRANSMIT BEAM WITH 10 DB TAPER) THE WORST CASE OF INTERACTION ENVELOPE (1) APPEARS AS SHOWN BY DOTTED LINES.

GENERAL ELECTRIC

space division

EFFECT ON PROPAGATION MEDIA ON SIDELOBE LEVEL



POWER DISTRIBUTION ON SPACE ANTENNA:

(1) GAUSSIAN
(2) QUADRATIC ON PEDESTAL
(3) $\cos^2$ ON PEDESTAL

λ = 12.25 cm
PHASE ERROR: 10° rms
AMPLITUDE ERROR: ±1 dB
EDGE TAPER: 10 dB
TOTAL POWER OUTPUT AT RECTENNA: 5 GW

POLARIZATION ROTATION VS. FREQUENCY DUE TO IONOSPHERE
(TYPICAL VALUE FOR CONTINENTAL US IN WORST YEAR)

LOSS OF POWER DUE TO THE FARADAY ROTATION OF 17.2° AT 2.45 GHz (WITH RECTENNA POLARIZATION ADJUSTED FOR MID RANGE OF ROTATION) IS 1.1 PERCENT DURING THE DAY TIME. (THE IONOSPHERIC MAGNETIC FIELD IS ASSUMED TO BE 0.62 GAUSS FOR THESE COMPUTATIONS).

GENERAL ELECTRIC

POLARIZATION ROTATION VS FREQUENCY
DUE TO IONOSPHERE

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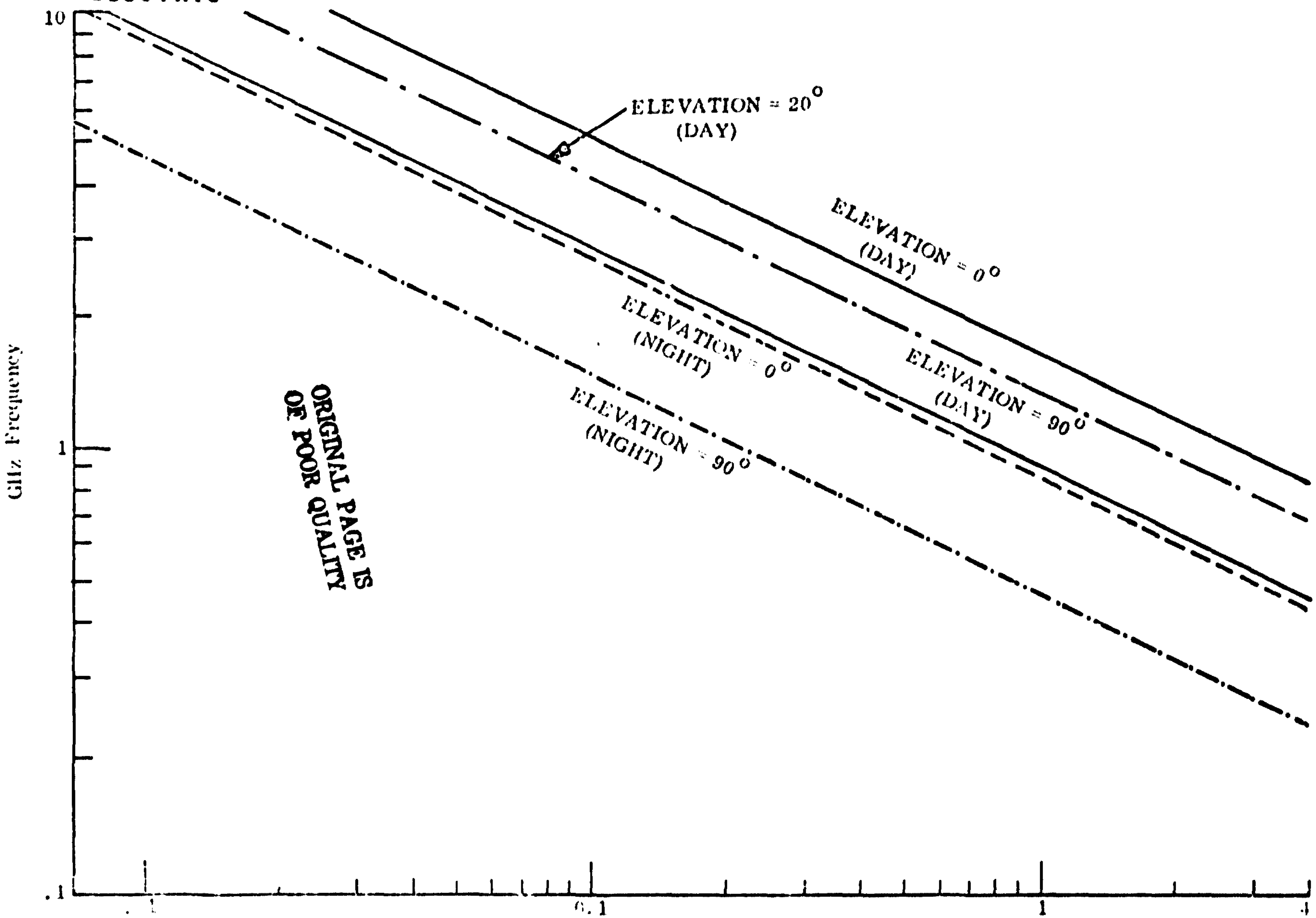


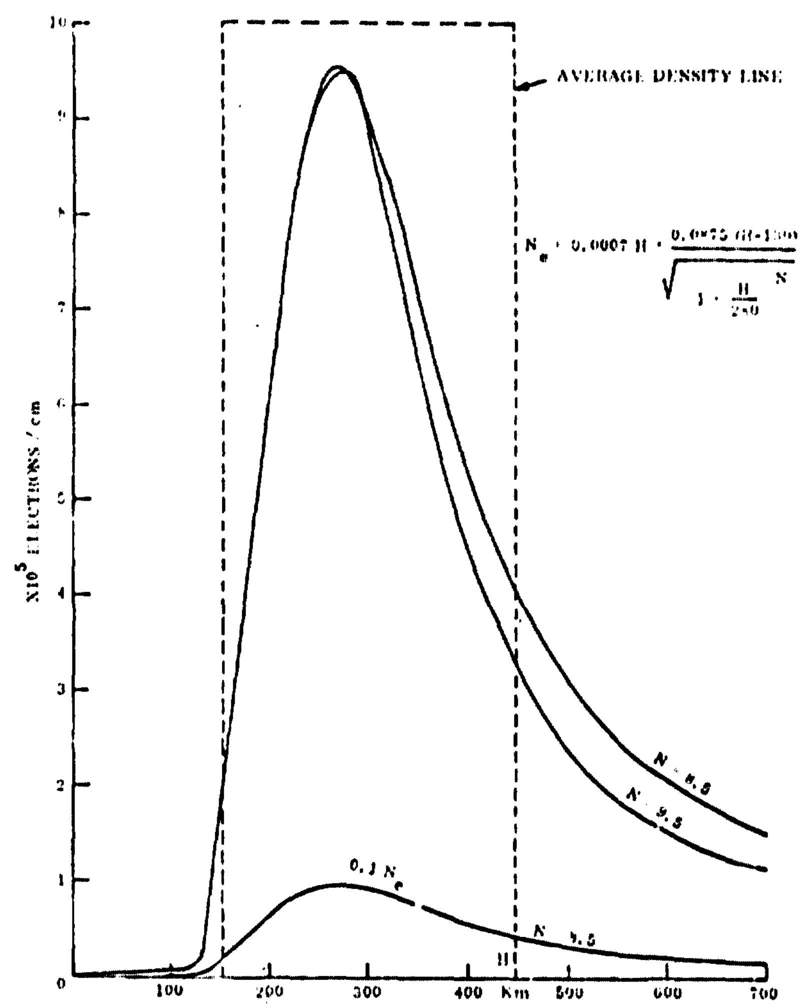
VARIATION OF ELECTRON DENSITY IN THE IONOSPHERE WITH HEIGHT

GENERAL ELECTRIC MATHEMATICAL MODEL OF THE IONOSPHERIC ELECTRON DENSITY AS A FUNCTION OF HEIGHT.

GENERAL ELECTRIC

space division

VARIATION OF ELECTRON DENSITY IN THE IONOSPHERE WITH HEIGHT



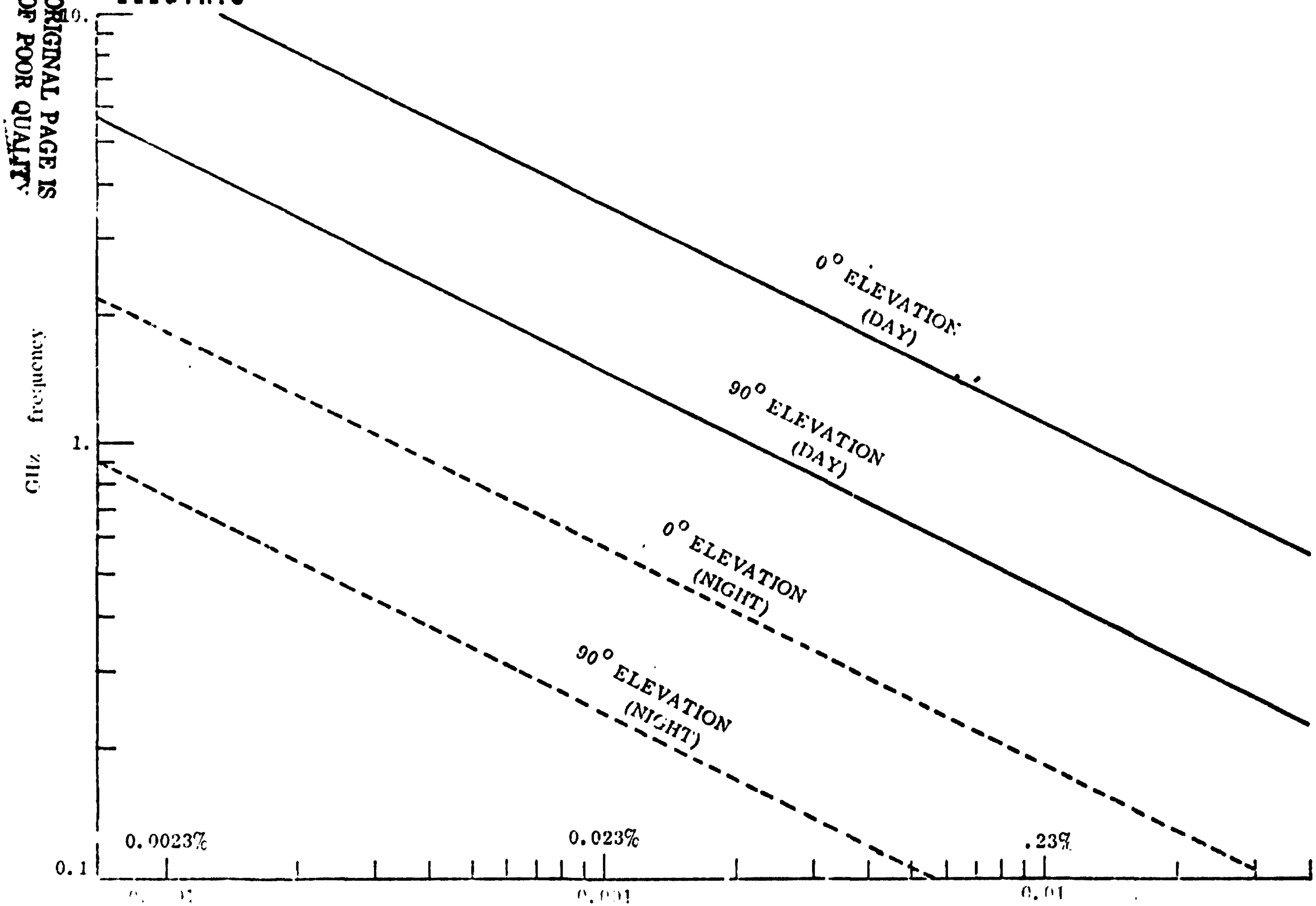
ATTENUAT ON VS. FREQUENCY IN THE IONOSPHERE

IONOSPHERIC ATTENUATION IS INVERSELY PROPORTION TO THE SQUARE OF THE FREQUENCY.

THIS AMOUNTS TO LESS THAN 0.02% AT ALL FREQUENCIES OF INTEREST AND AT WORST CONDITIONS OF PROPAGATION.

SCATTERED RADIATION DUE TO THE CROSS SECTION OF THE ELECTRONS IS QUITE SMALL RELATED TO THE SIDELOBE LEVELS OF THE RECTENNA. THESE LEVELS MIGHT, HOWEVER, BE HARMFUL FOR INTERFERENCE WITH OTHER COMMUNICATION SYSTEMS.

ATTENUATION VS FREQUENCY IN THE IONOSPHERE



RECTENNA/GROUND POWER COLLECTION & TRANSMISSION

RECTENNA TECHNOLOGY BASIC ASSUMPTIONS
(5 GW SYSTEM)

THE IMPORTANT DESIGN CONSIDERATIONS OF THE RECTENNA ARE DETERMINED FROM:

1. THE CAPABILITY OF THE RF/DC CONVERTERS TO WORK IN HARMONY AT THE PROJECTED LEVELS OF EFFICIENCY AT DIFFERENT LEVELS OF RF DENSITIES OF THE RECTENNA STRUCTURE.

2. THE STRUCTURAL CAPABILITY TO MINIMIZE THE DIFFRACTION SHADOWING FACTOR K, AS FUNCTION OF THE LATITUDE LOCATION, IN ORDER TO MINIMIZE THE FIELD LEVEL VARIATIONS ACROSS THE COLLECTING PANELS.

GENERAL ELECTRIC

RECTENNA TECHNOLOGY BASIC ASSUMPTIONS

space division

• DETECTION EFFICIENCY (RF/DC)	85 - 92 %
• RANGE OF POWER DENSITIES	1 - 40 MW/CM^2
• QUALITY PERFECTED DIODES	¢ 1 / DIODE
• PRIMARY UNIT CAPACITY	1 MW
• PRIMARY UNIT D.C. VOLTAGE	(5000 UNITS) $\pm$ 2 KV
• CONVERTER UNIT CAPACITY	(2 x 20) 40 MW (125 UNITS)
• PANEL WIDTH (W) IS A FUNCTION OF LATITUDE LOCATION AS DETERMINED BY DIFRACTION SHADOWING FACTOR (K) AND PANEL SEPARATION (H)	$W^2 \geq \frac{H \lambda}{K^2}$
• SUPPRESSION OF HARMONIC RADIATION TO ACCEPTABLE LEVELS	

CHARACTERISTICS OF POWER FLOW

- RADIAL FLOW OF POWER AMONG THE PRIMARY UNITS MINIMIZES DISSIPATION LOSSES, IN CONTRAST TO THE CIRCULATORY FLOW OF POWER. THE FORMER NEEDS DESIGN CARE FOR THE EQUAL PRIMARY UNITS AT DIFFERENT FIELD LEVELS.
- SLIGHTLY DIFFERENT GROUPING OF ELEMENTS OR SLIGHTLY DIFFERENT ELEMENTS ARE NEEDED ANY WAY FOR PROPER POWER FLOW IN THE RECTENNA.

GENERAL ELECTRIC

CHARACTERISTICS OF POWER FLOW

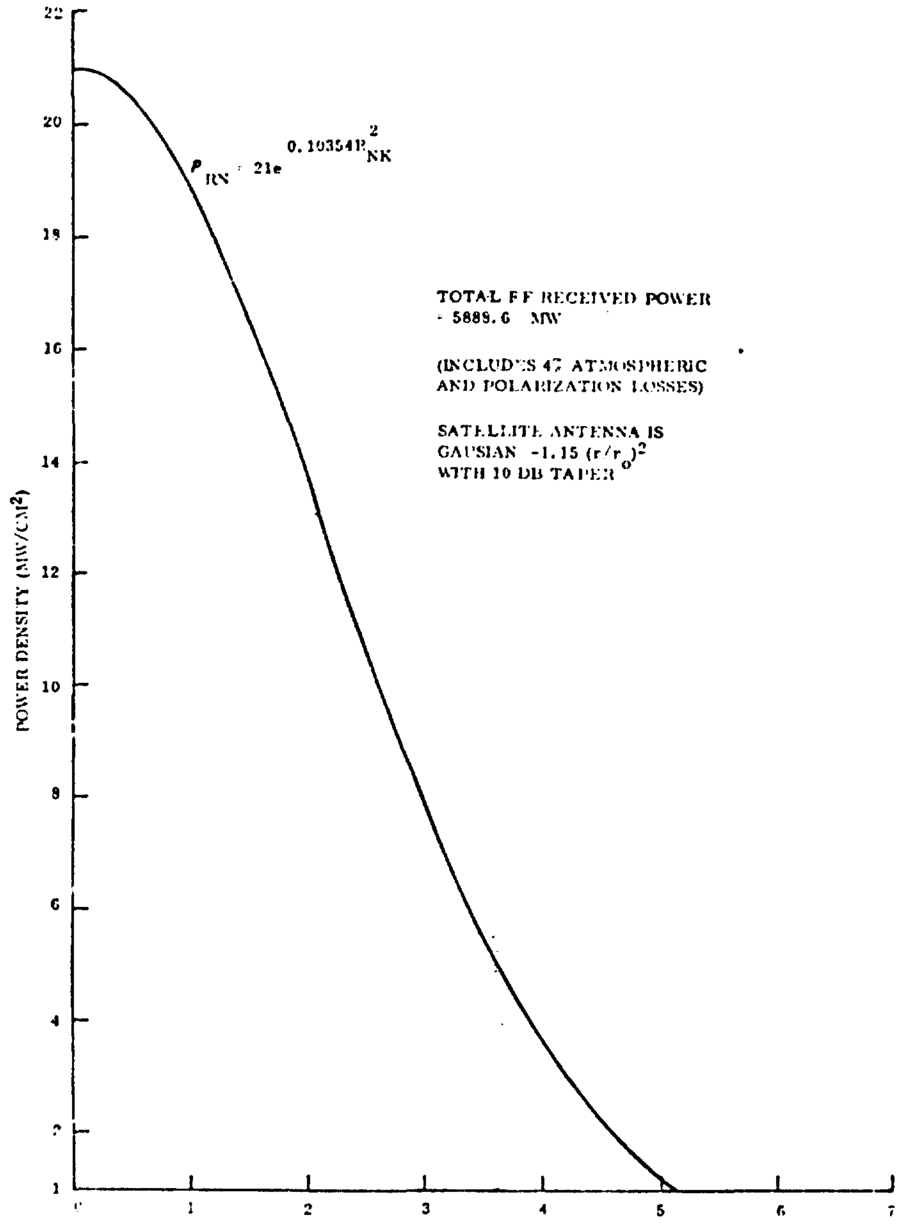
space division

- EQUAL POWER LEVELS ARE CONCENTRIC ELLIPTIC RINGS
- THE AREA OF COLLECTING RINGS FOR A SPECIFIC AMOUNT OF POWER INCREASES EXPONENTIALLY TO THE EDGE OF THE RECTENNA
- POWER FLOW IS RADIAL FROM OR TO THE CENTER OF THE RECTENNA
- IONOSPHERIC DISTURBANCES INDUCE SLOW OSCILLATORY FORM TO THE POWER FLOW. THIS EFFECT IS RECTIFIED BY RADIAL FLOW OF POWER
- CROWBARS AND D.C. BREAKERS ARE USED FOR EACH PRIMARY UNIT OF 1 MW
- FAULTY SECTIONS MAY BE ELECTRICALLY BYPASSED BY LARGE CAPACITY DIODES PRIOR TO D.C. BREAKER ISOLATION

RF POWER DISTRIBUTION OF RECTENNA

THIS CHART SHOWS THE r-f POWER DISTRIBUTION OF THE RECTENNA FOR A 5 GW DC OUTPUT POWER. A GAUSSIAN POWER DISTRIBUTION WITH 10 DB EDGE TAPER IS ASSUMED ON SPACE ANTENNA. ALSO 4% LOSS DUE TO MEDIA IS ASSUMED.

RF POWER DISTRIBUTION OF RECTENNA



PERCENTAGE OF ACCUMULATED POWER ON 10 KM DIAMETER RECTENNA

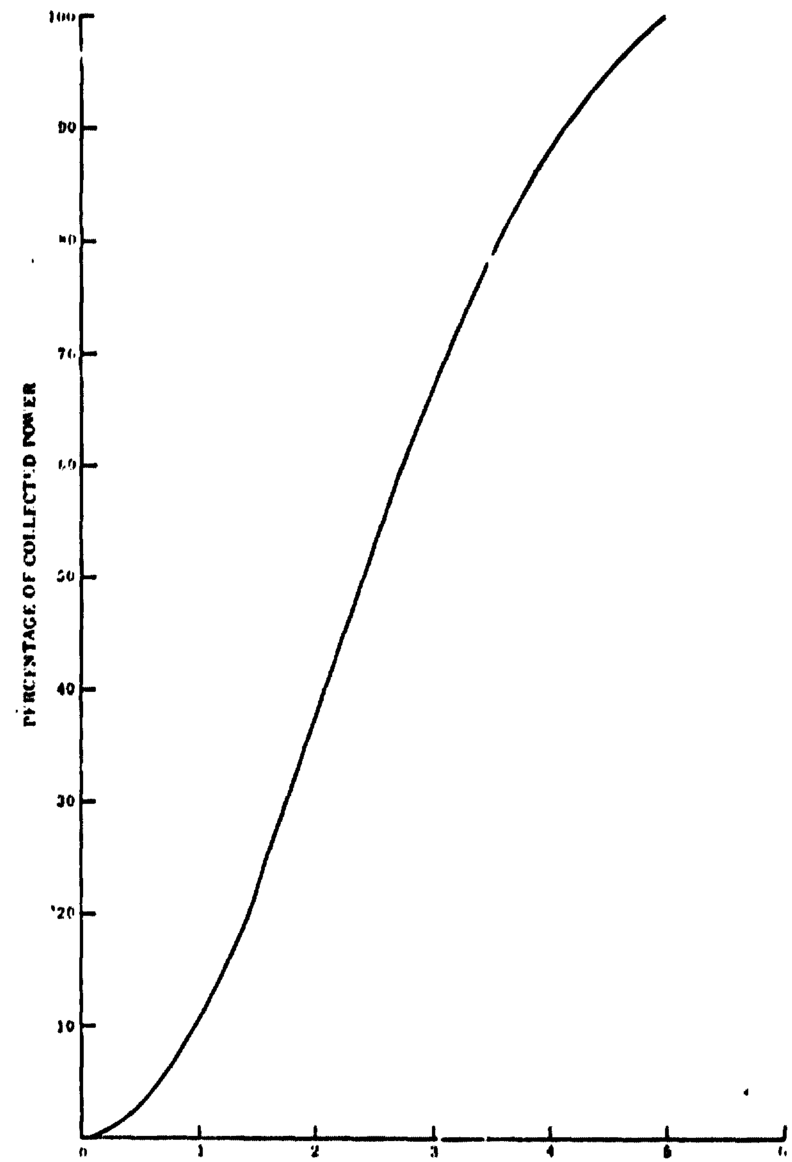
THIS FIGURE SHOWS THE TOTAL COLLECTED POWER FOR A GIVEN DOWNLINK BEAM (1 KM DIAMETER SPACE ANTENNA, GAUSSIAN DISTRIBUTION, 10 DB TAPER) AS A FUNCTION OF THE RADIUS OF RECTENNA.

C-5

GENERAL ELECTRIC

space division

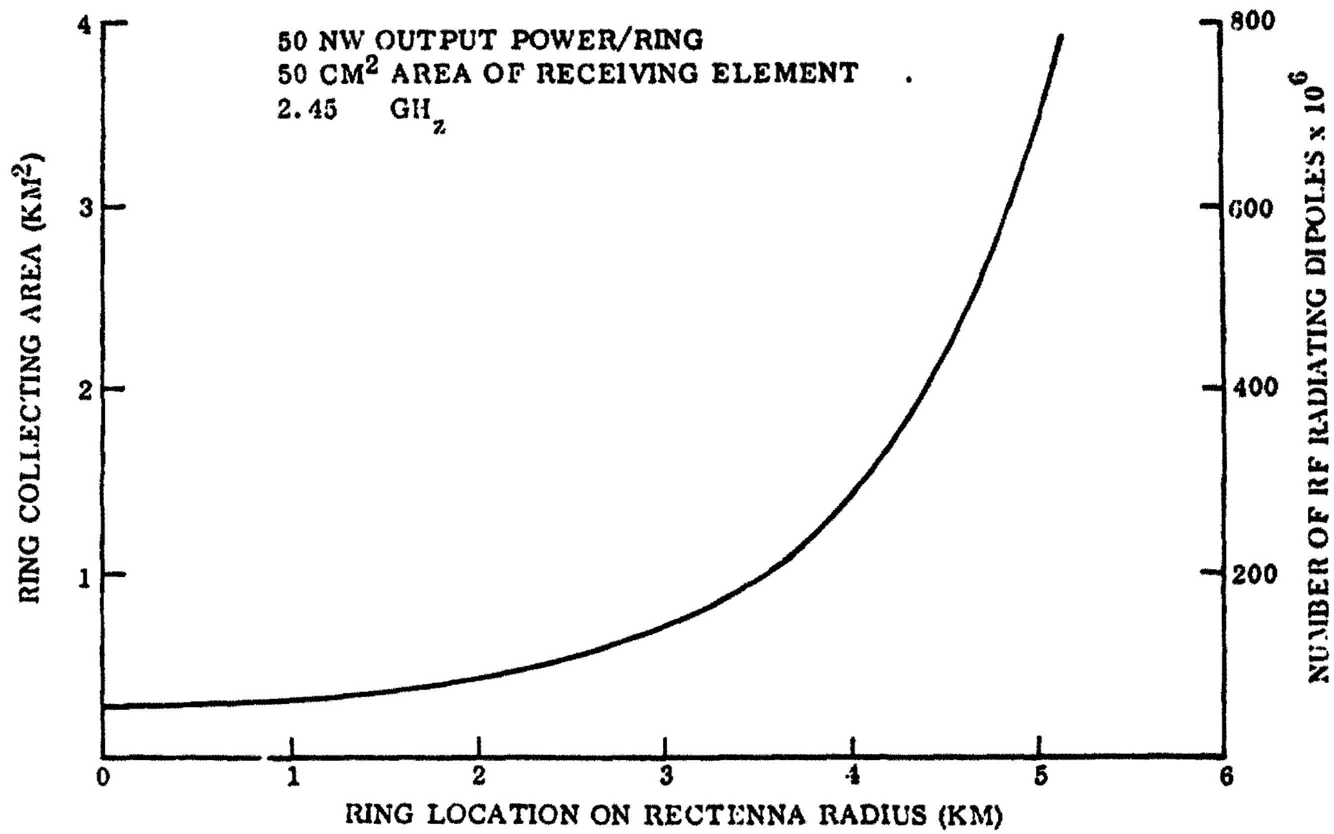
PERCENTAGE OF ACCUMULATED POWER ON 10KM DIAMETER RECTENNA



RECTENNA COLLECTING RING AREA AND NUMBER OF RADIATING ELEMENTS AS A FUNCTION OF RADIUS

THE POWER OF THE RECTENNA IS COLLECTED IN SEGMENTS CORRESPONDING TO CONCENTRIC RINGS. THE AREA OF THESE RINGS INCREASES WITH THE RING RADIUS IN ORDER TO KEEP THE POWER OUTPUT FROM THE SEGMENTS CONSTANT.

RECTENNA COLLECTING RING AREA AND NUMBER OF RADIATING ELEMENTS AS FUNCTION OF RADIUS



ELEVATION ANGLES FOR SPS AT DIFFERENT LATITUDES

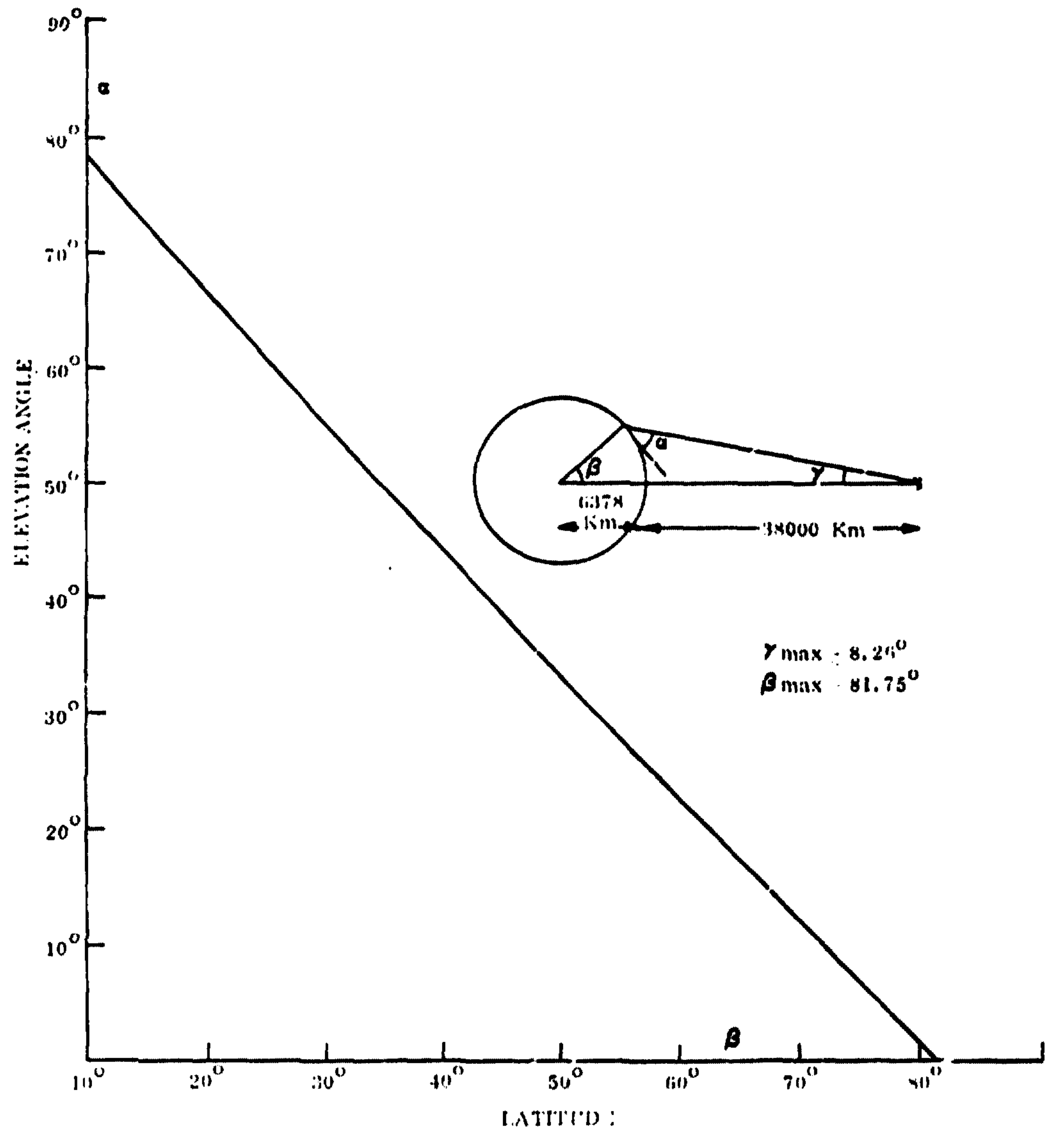
IT IS NOT RECOMMENDED USING ELEVATION ANGLES BELOW 20°, WHICH CORRESPOND TO LATITUDE LOCATIONS ABOVE 62°.

LOW ELEVATION ANGLES ARE SUSCEPTIBLE TO DUCT EFFECTS AND MORE DEGRADATION FOR THE MAIN BEAM AND THE SIDELOBE ENVELOPE.

GENERAL ELECTRIC

ELEVATION ANGLES FOR SPS AT DIFFERENT LATITUDES

space division



POWER DENSITY OVER RECTENNA

THE FORMULA SHOWS RELATIONSHIPS BETWEEN GEOMETRY AND POWER DISTRIBUTION

GENERAL ELECTRIC

POWER DENSITY OVER RECTENNA

space division

- RADIAL DISTANCE RM (IN KM) AT ANGLE Φ

$$RM = \frac{B}{\sqrt{SIN^2 \Phi + COS^2 \Phi \quad SIN^2 \alpha}} = \frac{B}{SK}$$

α = ELEVATION BEAM ANGLE
B = SEMI MINOR AXIS DISTANCE

- POWER DENSITY

$$\rho_{RN} = \rho_o \, e^{-k \times SK^2 \times RN^2}$$

$\rho_o = 21 \quad mw/cm^2$

k = 0.10364

RN = ANY RADIAL DISTANCE OF RECTENNA

RECTENNA LAYOUT

THE INCLINATION ANGLE OF THE MAJOR AXIS OF THE RECTENNA IS DETERMINED BY THE LONGITUDE LOCATION OF THE RECEIVING SITE.

ALL THE PANELS OF THE RECTENNA, OF ELLIPTIC RINGS, OF EQUAL POWER LEVELS, ARE ARRAYED IN PARALLEL.

PRESENT DIVISION OF POWER USES 100 RINGS, EACH DELIVERS 50 MW OF D.C. POWER. EACH RING IS DIVIDED INTO 50 1 MW PRIMARY UNITS.

THE NUMBER OF RADIATION DETECTION ELEMENTS IS 15.7 BILLION.

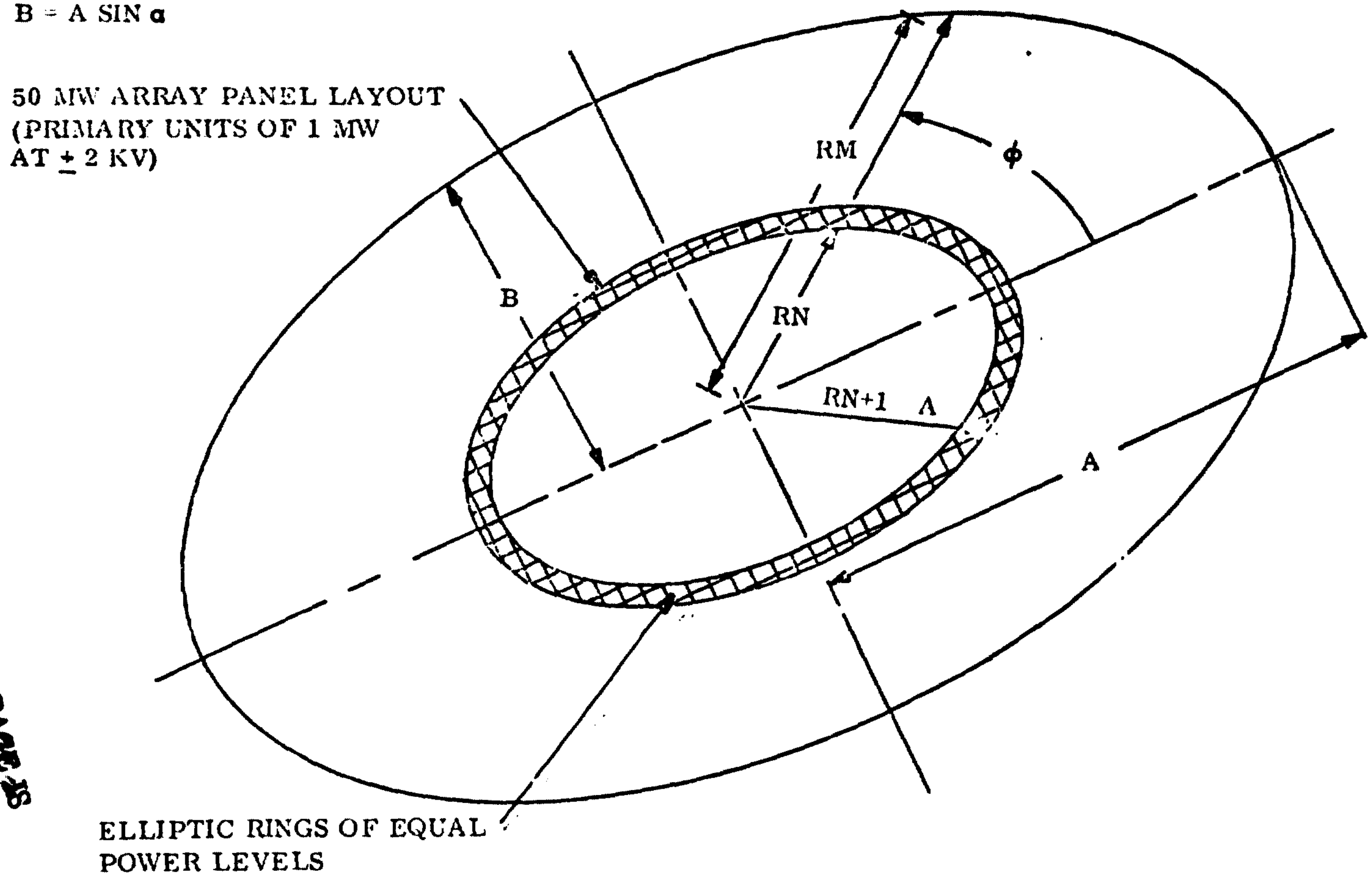
POWER FLOW IS RADIAL FROM THE CENTER OF THE RECTENNA.

GENERAL ELECTRIC

RECTENNA LAYOUT

space division

GROUND POWER OUTPUT = 5GW
MINOR AXIS (2B) = 10 KM
NUMBER OF RINGS (M) = 100
B = A SIN α



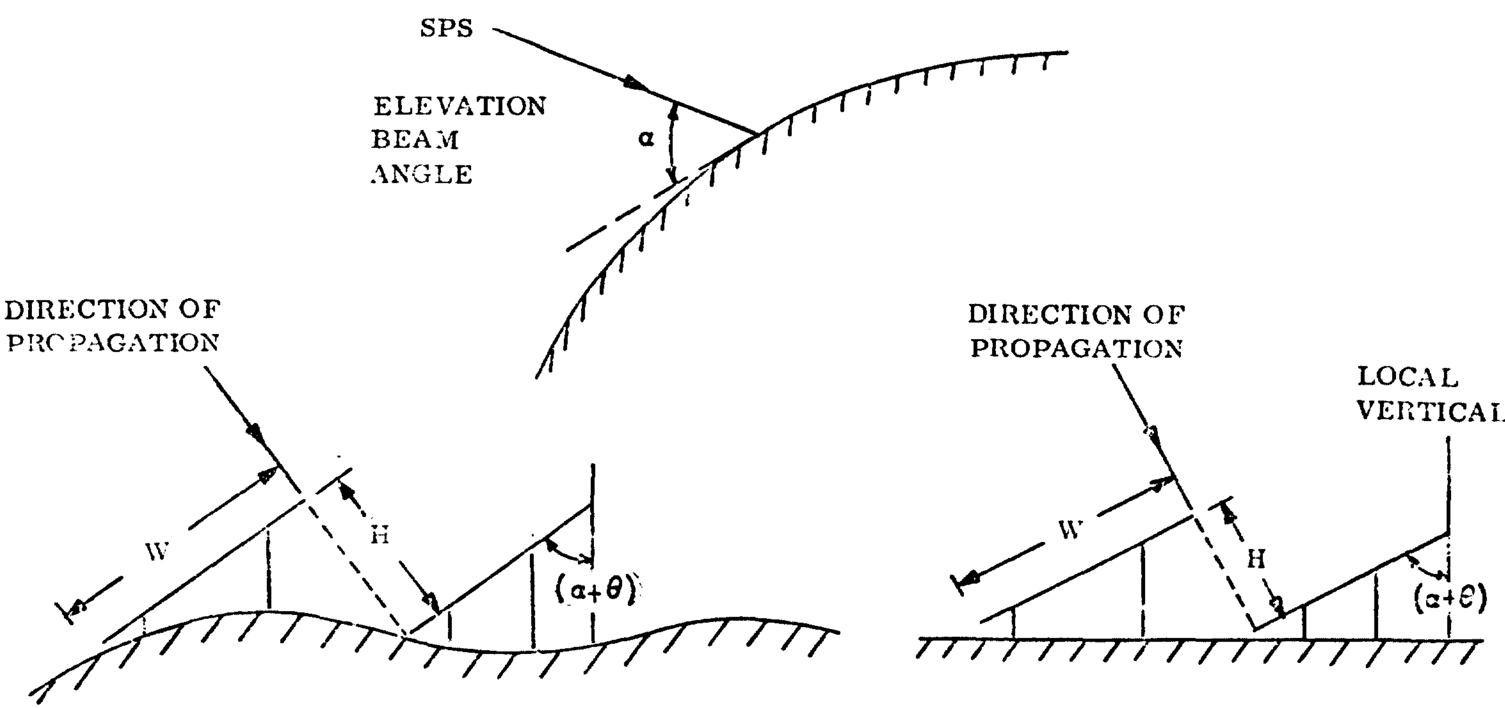
LAYOUT OF RECTENNA PANELS

- THE RECTENNA STRUCTURE IS DETERMINED BY THE TERRAIN CONFIGURATION, THE DIFFRACTION LIMITATIONS OF THE PANELS AND THE LATITUDE-LONGITUDE LOCATION.
- THE CHOICE OF THE NUMBER OF RADIATING ELEMENTS ON EACH PANEL IS A FUNCTION OF THE DIRECTIVITY OF THE RADIATING ELEMENTS, THE ADDITIONAL ANGLE ϕ WHICH IS NEEDED TO YIELD A REASONABLE VALUE FOR K AND THE LOSSES OF POWER CARRYING WIRES BETWEEN LARGE SEPARATED ELEMENTS AT HIGH LATITUDES OF RECTENNA LOCATIONS.
- THE RELATION BETWEEN THE ELEVATION BEAM ANGLE AND THE LATITUDE OF RECTENNA LOCATION (OF SPS LONGITUDE) IS SHOWN IN THE FOLLOWING CHART.

GENERAL ELECTRIC

LAYOUT OF RECTENNA PANELS

space division



UNLEVELED RECTENNA

LEVELED RECTENNA

$H < K^2 N^2 \lambda$

$W = N \lambda$

λ = WAVE LENGTH
κ = SHADOWING FACTOR
(SMALL FRACTION)

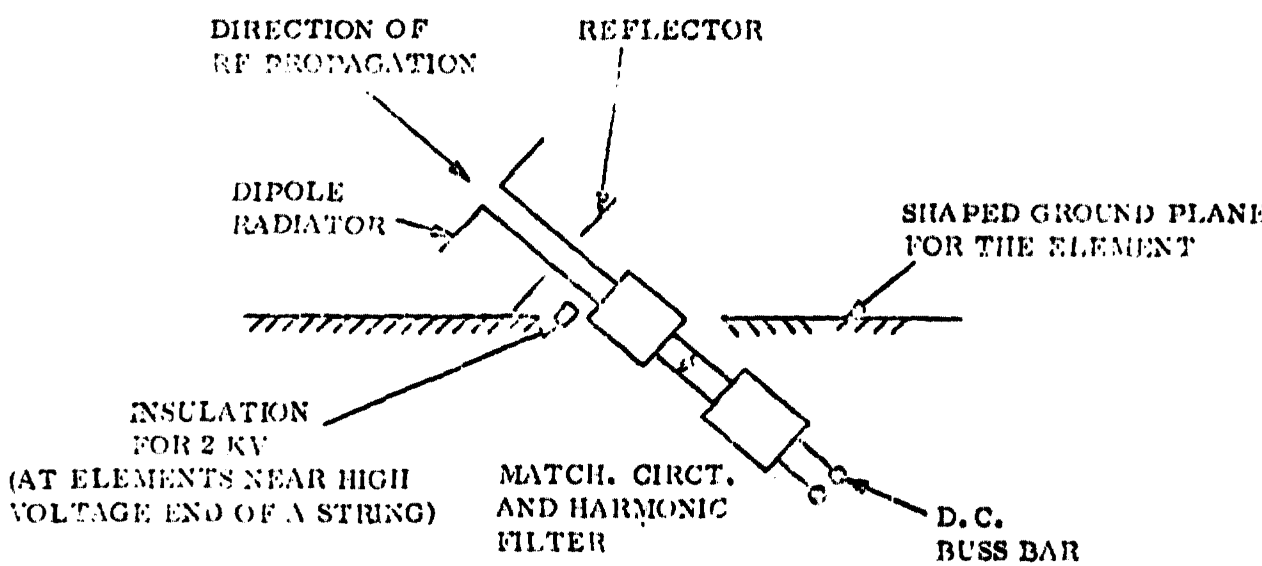
PREFERRED POLARIZATION ATTITUDE

THE DIFFRACTION LIMITATIONS OF THE RECTENNA PANELS PROHIBIT USING A GROUND PLANE ORTHOGONAL TO THE WAVE FRONT AT HIGH LATITUDES. THIS NECESSITATES USING ADDITIONAL REFLECTOR TO THE DIPOLE RADIATORS. THIS FUNCTIONS PROPERLY WHEN THE POLARIZATION IS PARALLEL TO THE MAJOR AXIS OF THE RECTENNA.

GENERAL ELECTRIC

PREFFERED POLARIZATION ATTITUDE

space division



PREFERRED POLARIZATION IS PARALLEL TO THE MAJOR AXIS OF ELLIPTIC SHAPED RECTENNA

RECTENNA EFFICIENCY

POWER COLLECTION EFFICIENCY INCREASES BY INCREASED ELEVATION DUE TO IMPROVEMENTS IN SEQUENTIAL CHARGING, INCREASED CURRENTS IN PARALLEL ARRAYS OF ELEMENTS AND LARGER PRIMARY UNITS.

D.C. TRANSMISSION EFFICIENCY FROM PRIMARY UNITS ELEVATION DECREASES WITH LATITUDE AND INCREASED ANGLE.

GENERAL ELECTRIC

space division

RECTENNA EFFICIENCY (RF/DC)
(5.889 GW/5 GW)

ORIGINAL PAGE IS OF POOR QUALITY

D.C. CONVERSION EFFICIENCY OF ELEMENTS	89%
POWER COLLECTION EFFICIENCY	98%
D.C. TRANSMISSION EFFICIENCY (99.5% AT CENTER & 94% AT PERIPHERY)	97%
TOTAL D.C. CONVERSION EFFICIENCY (D.C. POWER OUTPUT = 5 GW)	85%

RECTENNA COST

COST OF DIPOLE ASSEMBLY IS BASED ON ASSUMPTION THAT \$0.1 WOULD BE REASONABLE FOR MASS PRODUCTION OF 15,710 x 10^6 ELEMENTS NEEDED FOR ONE RECTENNA. PROJECTED COST OF SCHOTTKY BARRIER DIODES 1¢/ DIODE. SUPPORT STRUCTURE AND LAND IS ESTIMATED CN THE BASIS OF SYSTEM #7 OF BOVAY INGRS., INC. REPORT OF JUNE 20, 1977. (GROUND PLANE @ \$.36/LB.)

BUS BAR COST IS ESTIMATED ON THE BASIS OF ROCKWELL'S EVALUATION.

GENERAL ELECTRIC

space division

RECTENNA COST

DIPOLE ASSEMBLY	$\$ 1,571 \times 10^6$
IMPROVED S.B. DIODES	$\$ 157 \times 10^6$
POWER DISTRIBUTION BUS BARS	$\$ 57 \times 10^6$
SUPPORT STRUCTURE	$\$ 2,051 \times 10^6$
LAND (TYPICAL MIDWEST)	$\underline{\$ 71} \times 10^6$
TOTAL RECTENNA	$\$ 3,907 \times 10^6$

REQUIRED INVESTIGATIONS

IMPROVED RADIATING ELEMENTS WHICH MAY BE MASS PRODUCED ECONOMICALLY (ESPECIALLY IF THE SPS IS STABLE ENOUGH TO ALLOW USING SMALLER NUMBER OF HIGHER GAIN ELEMENTS).

ANALYSIS OF POWER FLOW SCHEMES WHICH ACCURATELY ASSESS THE D.C. COLLECTION EFFECIENCY BASED ON THE MANUFACTURING TOLERANCES OF DETECTION ELEMENTS AND THE SLOW OSCILLATORY IONOSPHERIC AND ATMOSPHERIC EFFECTS.

EVALUATION OF RE-RADIATION OF CROW-BARED RECTENNA OF THE RANDOM PHASE FRONT CONCERNING RADIATION HAZARD.

EVALUATION OF GRATING LOBES OF SCATTERED RADIATION OF PERIODIC RECTENNA STRUCTURES.

EFFECT OF HEATED IONOSPHERE ON SCINTILLATION LEVELS OF THE RECTENNA AND ON SCATTERED RADIATION LEVELS.

GENERAL ELECTRIC

space division

ORIGINAL PAGE IS OF POOR QUALITY

REQUIRED INVESTIGATIONS

- IMPROVED RADIATING ELEMENTS WHICH MAY BE ECONOMICALLY MASS-PRODUCED (ESPECIALLY IF THE SPS IS STABLE ENOUGH TO ALLOW USING SMALLER NUMBER OF HIGHER GAIN ELEMENTS)

- ANALYSIS OF POWER FLOW SCHEMES TO ACCURATELY ASSESS THE D.C. COLLECTION EFFICIENCY BASED ON THE MANUFACTURING TOLERANCES OF DETECTION ELEMENTS AND THE SLOW OSCILLATORY IONOSPHERIC AND ATMOSPHERIC EFFECTS

- EVALUATION OF RERADIATION OF CROWBARRED RECTENNA OF THE RANDOM PHASE FRONT IN CONCERN OF RADIATION HAZARD

- EVALUATION OF GRATING LOBES OF SCATTERED RADIATION OF PERIODIC RECTENNA STRUCTURES

- EFFECT OF HEATED IONOSPHERE ON SCINTILLATION LEVELS OF THE RECTENNA AND ON SCATTERED RADIATION LEVELS

GROUND POWER COLLECTION

AND

TRANSMISSION SYSTEM

KEY CHARACTERISTICS OF SPS-UTILITY INTERFACE

THE CHARACTERISTICS OF THE SPS WITH THE MOST IMPACT ON THE UTILITY SYSTEM IN AN OPERATIONAL SENSE IS THE LARGE BLOCK OF POWER CONCENTRATED IN ONE UNIT, AND THE RELATIVE INFLEXIBLE NATURE OF A CONSTANT ENERGY SOURCE.

THE SPS CHARACTERISTICS IN LIGHT OF THE ELECTRIC UTILITY SERVICE CRITERIA OF ADEQUATE SYSTEM RELIABILITY AND MINIMIZING COST TO THEIR CUSTOMERS REQUIRES SPECIAL ATTENTION WHEN DESIGNING THE GROUND POWER COLLECTION AND TRANSMISSION SYSTEM BETWEEN THE REFERENCE PRIMARY UNITS AND THE CONNECTION/INSERTION POINT OF THE UTILITY BULK POWER GRID.

THE UTILITY LOAD CHARACTERISTICS HAVE PARTICULAR IMPACT, SINCE THE SPS GROUND SYSTEM HAS NO AUTOMATIC LOAD FOLLOWING CAPABILITY AND OTHER GENERATION CAPACITY WILL BE NEEDED AT ALL TIMES TO PROVIDE FOR UTILITY SYSTEM FREQUENCY CONTROL AND SPINNING RESERVE NEEDS.

KEY CHARACTERISTICS OF SPS-UTILITY INTERFACE

SPS:

- LARGE UNIT SIZE
- CONSTANT ENERGY SOURCE

UTILITY:

- SERVICE QUALITY CRITERIA
  - RELIABILITY
  - MINIMIZING COST TO CUSTOMERS
- VARIABLE LOAD DEMAND
  - BY HOUR
  - BY SEASON
  - BY YEAR

DESIGN CONSIDERATIONS FOR RECTENNA GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

TO ASSURE RELIABILITY IT WAS AN OBVIOUS NEED TO USE A MODULAR CONCEPT, INCORPORATING EQUIPMENT OF KNOWN TECHNOLOGY AND PERFORMANCE. THE DESIGN PRINCIPLES USED FOLLOWS CONVENTIONAL UTILITY DESIGN PRACTICE FOR CONTROL AND PROTECTION.

TO MINIMIZE COST THE PLANT LAYOUT FOLLOWS A SYMMETRICAL CONCEPT AND EQUIPMENT RATINGS LIKE CONVERTER STATION POWER LEVELS AND SYSTEM VOLTAGE LEVELS ARE CHOSEN TO BE COMPATIBLE WITH RECTENNA PRIMARY UNITS, AND MINIMIZE COST WHILE PROVIDING MAINTAINABILITY.

DESIGN CONSIDERATIONS FOR RECTENNA GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

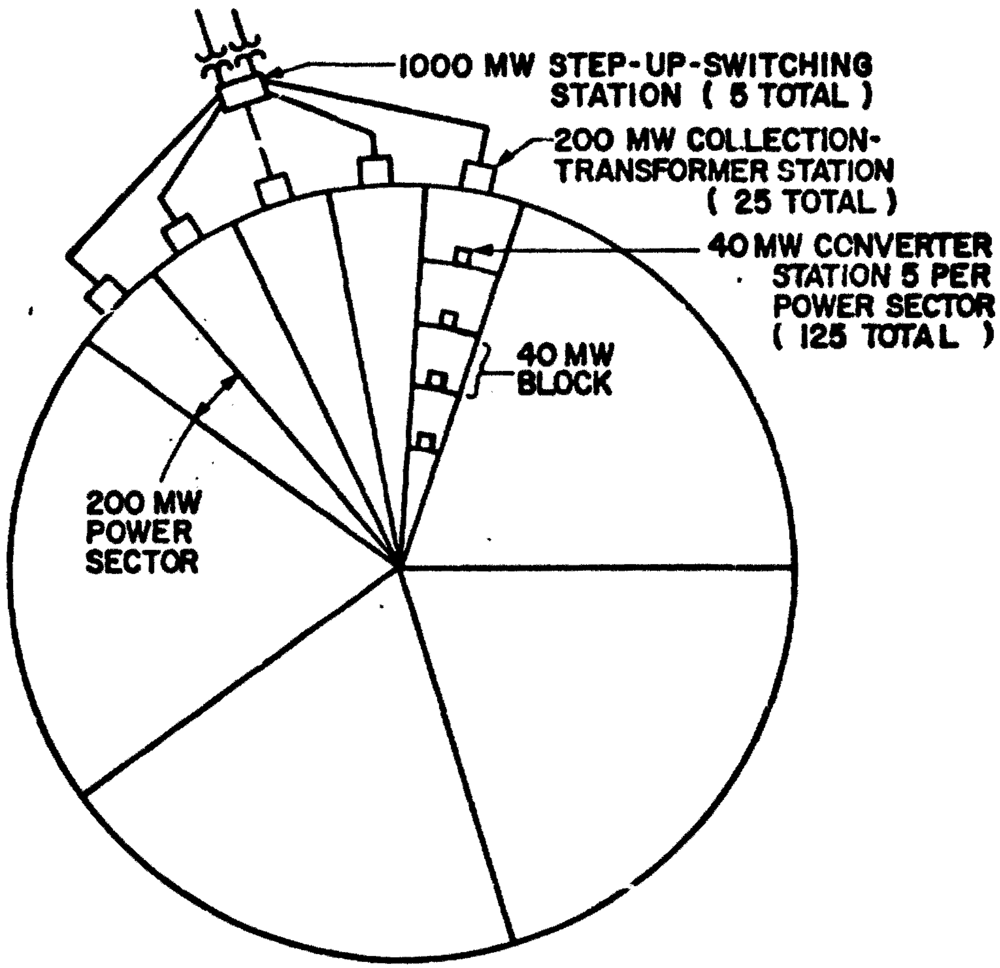
- RELIABILITY
  - MODULAR CHARACTER
  - EQUIPMENT PERFORMANCE
  - CONTROL AND PROTECTION
- COST
  - PLANT LAYOUT
  - EQUIPMENT, STRUCTURES AND MATERIAL
  - OPERATION AND MAINTENANCE

GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

GENERAL LAYOUT

THE PLOT PLAN IS ASSUMED TO BE CIRCULAR WITH A TOTAL NET OUTPUT OF 5000 MW. THE RECTENNA AREA IS DIVIDED INTO 5 EQUAL AREAS EACH FEEDING ONE 1000 MW STEP-UP-SWITCHING STATION. EACH STEP-UP-SWITCHING STATION IS IN TURN FED BY FIVE 200 MW POWER SECTORS. EACH POWER SECTOR CONTAINS FIVE 40 MW BLOCKS. EACH CONVERTER STATION COLLECTS 40 MW DC POWER FROM PRIMARY RECTENNA UNITS AND INVERTS DC TO AC POWER.

RECTENNA POWER COLLECTION AND TRANSMISSION SYSTEM



GROUND POWER COLLECTION AND CONVERSION SYSTEM

EACH 40 MW POWER BLOCK CONSISTS OF FORTY 1 MW PRIMARY UNITS WITH OUTPUT VOLTAGE OF ± 2 kV. THE END OF EACH PRIMARY UNIT IS CONNECTED THROUGH DC CIRCUIT BREAKERS TO 2 kV DC CABLES RUNNING RADIALLY AS SHOWN IN THE DIAGRAM TO THE CONVERTER STATION. DC SMOOTHING REACTORS REDUCE THE RIPPLE CURRENTS.

SINCE THE RECTENNAS ARE CONSTANT POWER DEVICES AND THE DC/AC CONVERTER CAN IN NO WAY AFFECT POWER FLOW, THE CONTROL OF POWER BE APPLIED ON THE DC SIDE. THIS MEANS THAT EITHER THE RF LEVEL MUST BE CONTROLLED AT IT'S SOURCE OR THE NUMBER OF RECTENNAS CONNECTED IN PARALLEL MUST BE VARIED. CIRCUIT BREAKERS PROVIDED FOR RECTENNA PROTECTION CAN ALSO BE USED TO ADD OR REMOVE UNITS IN ORDER TO CONTROL POWER, BUT NOT ON A CONTINUOUS BASIS.

IT IS RECOGNIZED THAT THE SPS SYSTEM WILL OPERATE AT CONSTANT POWER BUT POWER VARIATION IS NEEDED TO GET ON LINE AND TO GET OFF LINE FOR SUCH THINGS AS MODIFICATION OR MAINTENANCE. IT MAY BE THAT THE 20 MW POWER GROUPS ARE SMALL ENOUGH THAT THEY CAN BE PICKED UP OR DROPPED AS THE MINIMUM SIZE INCREMENT.

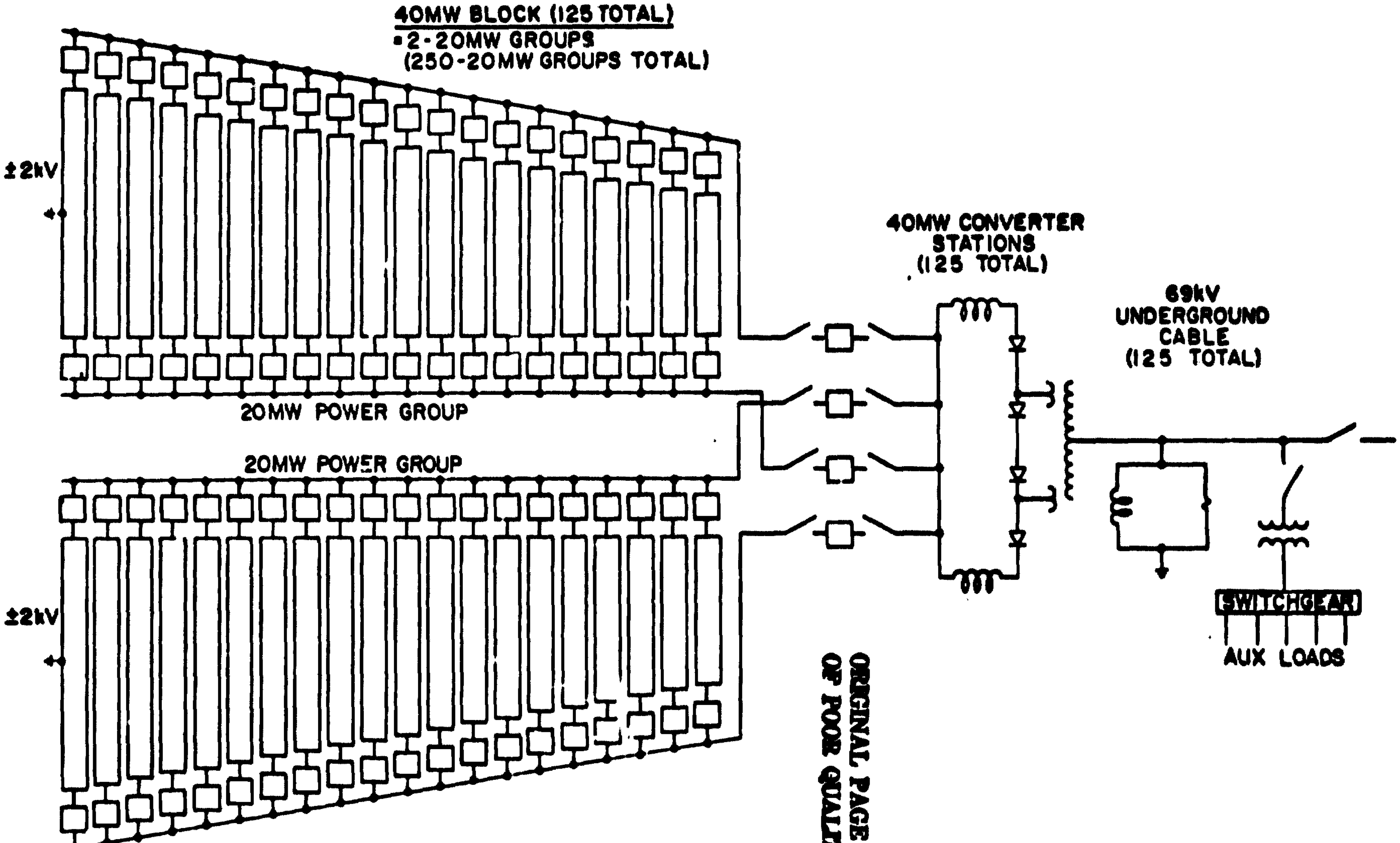
THE CONVERTER THYRISTOR BRIDGE CIRCUIT FEEDS ALTERNATING CURRENT TO THE CONVERTER TRANSFORMER WHICH STEPS THE VOLTAGE UP TO 69 kV AT 60 HZ.

FILTERS CONNECTED THE AC BUS ABSORB CURRENT HARMONICS GENERATED IN THE CONVERTER. THE AC WAVE SHAPE IS THEREBY KEPT WITHIN ACCEPTABLE HARMONIC CONTENT LIMITS FOR THE UTILITY GRID AND ASSOCIATED PLANT EQUIPMENT.

THE CONVERTER STATION OUTPUT, AT 69 kV AND A MAXIMUM CURRENT OF 400 AMPERES IS TRANSMITTED BY UNDERGROUND CABLE TO THE TRANSFORMER STATION.

THE CONVERTER STATION, ONCE COMMISSIONED, OPERATES AUTOMATICALLY. ALL SWITCHING, STARTUP AND SHUTDOWN ARE DIRECTED AND MONITORED BY A SMALL COMPUTER SYSTEM IN CONJUNCTION WITH OTHER CONVERTER AND STATION CONTROL EQUIPMENT.

GROUND POWER COLLECTION AND CONVERSION SYSTEM



CONVERTER CHARACTERISTICS AND CONTROL BLOCK DIAGRAM

1. CHARACTERISTIC CURVE OF THE RECTENNA - THE VOLTAGE-CURRENT CHARACTERISTIC OF THE RECTENNA, OVER THE RANGE TO BE CONSIDERED, CAN BE DESCRIBED AS A CONSTANT POWER RECTANGULAR HYPERBOLA. AT HIGH VOLTAGE AND LOW CURRENT, AN AUTOMATIC SHORT CIRCUITING DEVICE OR "CROWBAR" WILL BE PROVIDED AS AN INTEGRAL PART OF THE RECTENNA. LIKEWISE AT THE LOW VOLTAGE, HIGH CURRENT END, THE CHARACTERISTICS WILL PROBABLY BE TERMINATED BY A SHORT CIRCUIT.

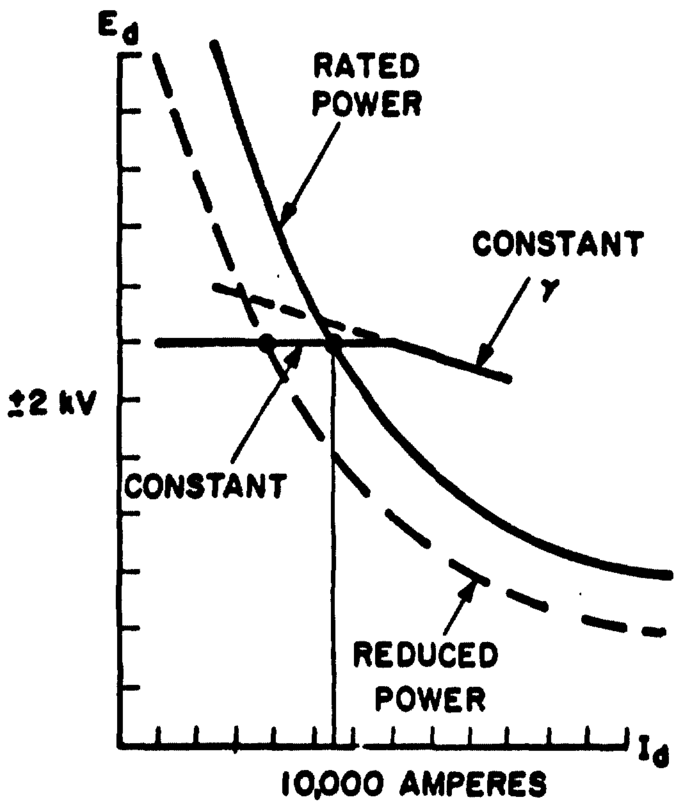
2. CHARACTERISTIC CURVE OF THE POWER CONDITIONING SYSTEM - THE INVERTER OF THE POWER CONDITIONING SYSTEM IS AN ELECTRONIC DEVICE CAPABLE OF SEVERAL MODES OF CONTROL. THE LINE-COMMUTATED INVERTER IS CHOSEN FOR THE SPS SYSTEM AND THE INVERTER IS OPERATED IN A CONSTANT VOLTAGE MODE.

3. CONTROL BLOCK DIAGRAM - THE "REGULATOR" IS SEEN FEEDING A TRANSDUCER T WHICH CONVERTS A VOLTAGE INTO AN ANGLE FOR THE FIRING PULSES TO THE THYRISTORS OF THE CONVERTER BRIDGE CIRCUIT.

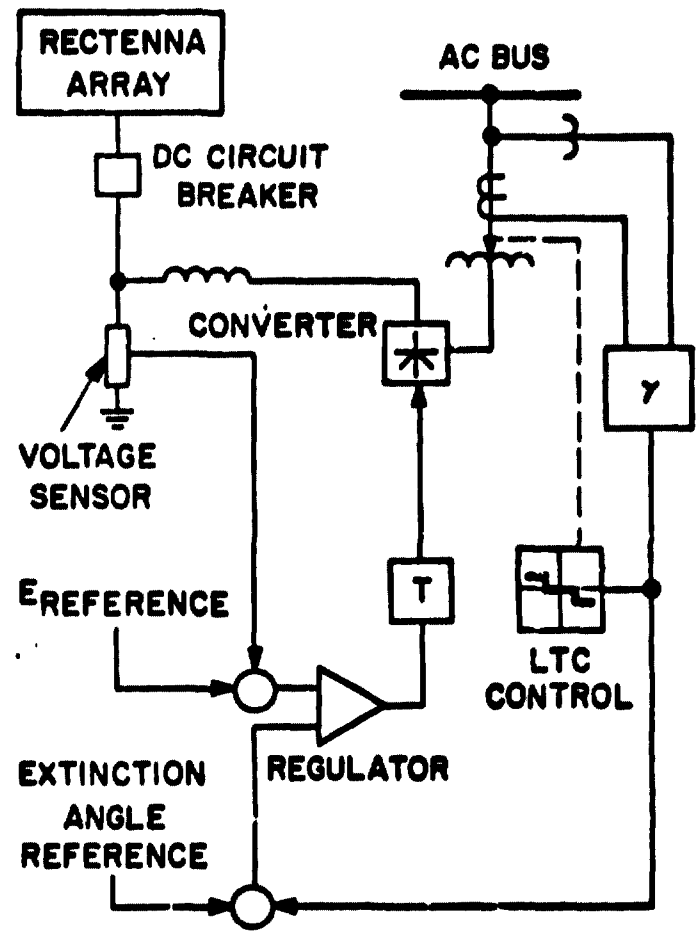
   A DC VOLTAGE SENSOR PROVIDES A MEASURE OF THE ACTUAL VOLTAGE WHICH PROVIDES THE ERROR SIGNAL FOR THE REGULATOR. THIS AUTOMATIC VOLTAGE CONTROL LOOP IS THE PRIMARY CONTROL OF THE CONVERTER.

   AC LINE CURRENT AND AC LINE VOLTAGE ARE MEASURED (AS SHOWN) AND A RELATIVE TIMING SIGNAL IS DEVELOPED WHICH RESULTS IN MEASUREMENT OF EXTINCTION ANGLE γ. THIS QUANTITY IS COMPARED TO THE EXTINCTION ANGLE REFERENCE AND THE RESULTING ERROR SIGNAL PASSES TO THE REGULATOR AND HOLDS THE CONSTANT γ CURVE WHEN CONDITIONS ARE SUCH THAT CONSTANT VOLTAGE CANNOT BE HELD.

CONVERTER CHARACTERISTIC



CONVERTER CONTROL BLOCK DIAGRAM



GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

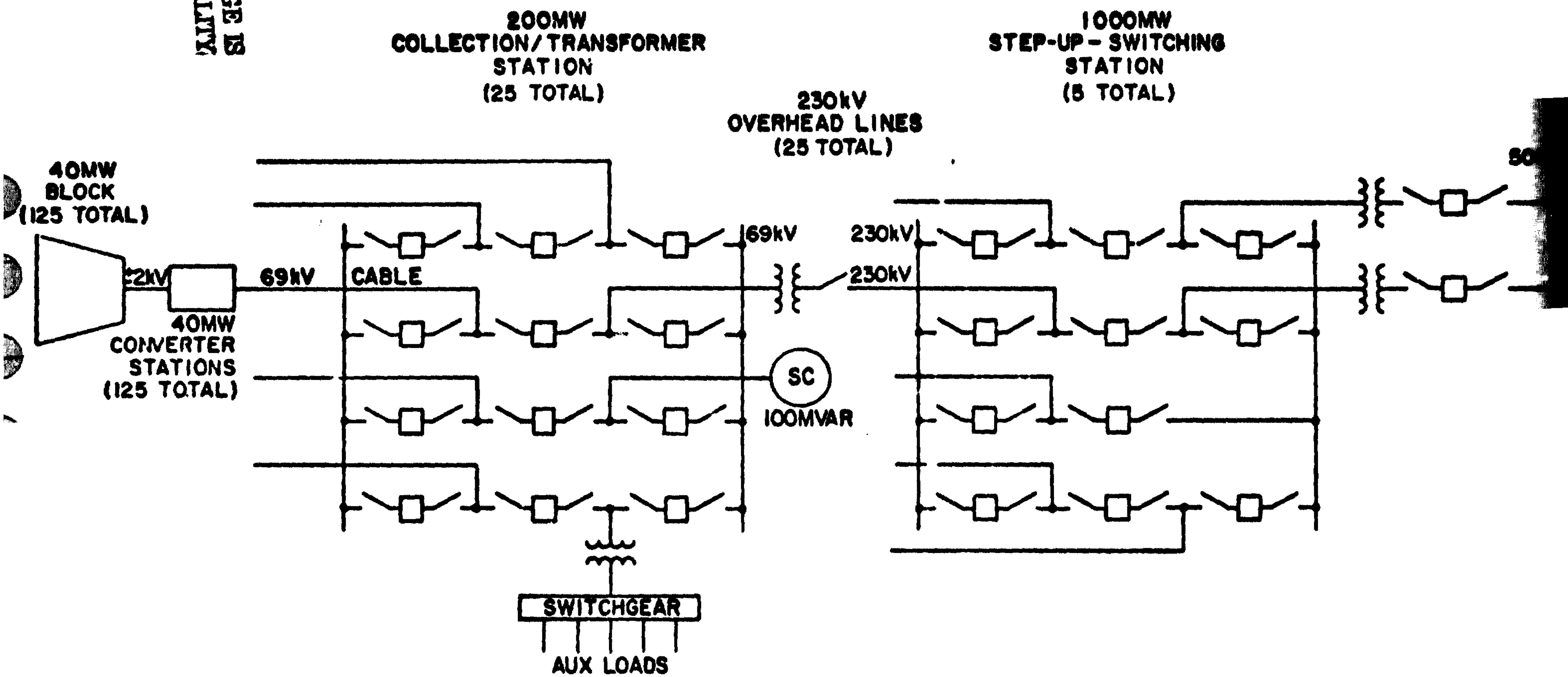
ONE LINE DIAGRAM

THE COLLECTION/TRANSFORMER STATION GATHERS THE POWER OUTPUT OF 5 CONVERTER STATIONS, CONNECTS THESE CIRCUITS INTO A RELIABLE SWITCHING ARRANGEMENT, AND TRANSFORMS THE AC POWER FROM 69 kV UP TO 230 kV. THIS IS DONE BY PHYSICALLY AND ELECTRICALLY ARRANGING AND CONNECTING STANDARD ELECTRICAL EQUIPMENT INTO THE DESIRED CONFIGURATION. THE ELECTRICAL CONFIGURATION PROVIDES RELIABILITY BY A "BREAKER AND A HALF" SCHEME 69 kV SWITCHYARD. A SINGLE CONTINGENCY OUTAGE CAN BE SUSTAINED IN THE 69 kV SWITCHYARD WITHOUT LOSS OF POWER OUTPUT CAPABILITY. TO PROVIDE COMPENSATION FOR THE INHERENT LAGGING POWER FACTOR CHARACTERISTICS OF THE CONVERTER VALVE AND TRANSFORMER EQUIPMENT ONE 100 MVAR SYNCHRONOUS CONDENSER IS CONNECTED TO THE 69 kV BUS. THE SYNCHRONOUS CONDENSER RATING IS CHOSEN TO ALLOW SYNCHRONOUS CONDENSER MAINTENANCE ON ADJACENT COLLECTION/TRANSFORMER STATIONS WITHOUT CURTAILING POWER OUTPUT.

THE STEP-UP SWITCHING STATION RECEIVES THE OUTPUT FROM FIVE COLLECTION/TRANSFORMER STATIONS AT 230 kV AND TRANSFORMS THE VOLTAGE TO 500 kV. THE "BREAKER AND A HALF" SCHEME EMPLOYED CAN SUSTAIN ANY SINGLE CONTINGENCY 500 kV SWITCHYARD FAULT WITHOUT REDUCTION IN STATION OUTPUT. THE SELECTION OF THE VOLTAGE LEVEL FOR THE ULTIMATE BULK POWER TRANSMISSION INTERFACE WITH THE UTILITY GRID AS WELL AS THE POSSIBILITY OF INTERCONNECTING TWO OR MORE OF THE 1000 MW SWITCHING STATIONS TOGETHER SHOULD BE OPTIMIZED BASED ON DETAILED INFORMATION ABOUT THE CONNECTING UTILITY SYSTEM. THE SOLUTION SHOWN IS ONE OF SEVERAL POSSIBLE.

ORIGINAL PAGE IS
OF POOR QUALITY

GROUND POWER COLLECTION AND TRANSMISSION SYSTEM



GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

COST ESTIMATES

THE COST ESTIMATES ARE BASED ON THE CONCEPTUAL DESIGN SHOWN ON THE ONE-LINE DIAGRAM. THE COST INCLUDES MAJOR EQUIPMENT, BUILDINGS, MATERIAL AND DIRECT LABOR.

NOT INCLUDED ARE:

- INDIRECT LABOR
- A/E FEES
- CONTINGENCY
- INTEREST DURING CONSTRUCTION
- INSURANCE
- TAXES
- TRANSPORTATION OF EQUIPMENT AND MATERIALS TO SITE

ALL COSTS ARE IN 1978 U.S. DOLLARS.

GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

SUMMARY OF COST ESTIMATES OF MAJOR EQUIPMENT, BUILDINGS, MATERIAL & DIRECT LABOR

QUANTITY	ITEM	TOTAL COST (M$) (1976 DOLLARS)
1	PLANT CONTROL CENTER	2.9
5	STEP-UP-SWITCHING STATION	22.6
39 km	230 kV TRANSMISSION LINE	4.2
25	COLLECTION-TRANSFORMER STATION	72.0
25	100 MVAR SYNCHRONOUS CONDENSER	87.5
381 km	69 kV CABLE & CONDUIT	69.5
125	CONVERTER STATION	307.0
5443 km	2 kV DC CABLE/BUS	48.6
10,000	2 kV, 250 A DC CIRCUIT BREAKER	60.0
	TOTAL	674.3

GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

OPERATION AND MAINTENANCE

THE NORMAL MODE OF OPERATION WILL BE AS A CONSTANT ENERGY SOURCE. THE UTILITY SYSTEM HAS NEEDS FOR FLEXIBLE GENERATION FOR LOAD FOLLOWING, SPINNING RESERVE AND SYSTEM FREQUENCY CONTROL. THE SPS SYSTEM WILL NOT BE ABLE TO CONTRIBUTE TO FREQUENCY CONTROL AND SPINNING RESERVE, BUT COULD CONCEIVABLY GIVE NEED FOR ADDED GENERATION FOR SPINNING RESERVE AND FREQUENCY CONTROL. THE UTILITY OPERATIONAL CONSTRAINTS MIGHT WELL BE IMPORTANT CONSIDERATIONS IN DETERMINING THE PRACTICAL LIMITS TO THE PENETRATION OF SPS POWER IN FUTURE UTILITY SYSTEMS.

THE MODULAR DESIGN OF THE GROUND POWER COLLECTION AND TRANSMISSION SYSTEM SHOULD PROVIDE HIGH AVAILABILITY SINCE ROUTINE MAINTENANCE COULD BE PERFORMED ON A PARTIAL SHUTDOWN BASIS. CONSIDERABLE EFFECTIVENESS IN MAINTENANCE ROUTINES SHOULD ALSO BE EXPECTED ALTHOUGH A SIGNIFICANT SPARE PARTS INVENTORY WOULD APPEAR NECESSARY.

GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

OPERATION AND MAINTENANCE

NORMAL OPERATION:

- CONSTANT ENERGY SOURCE
  - NO FREQUENCY CONTROL
  - NO LOAD FOLLOWING CAPABILITY EXCEPT BY SWITCHING MODULES
- MAINTENANCE BY MODULAR SHUTDOWN
  - HIGH AVAILABILITY
  - EFFECTIVE MAINTENANCE

GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

OPERATION AND MAINTENANCE

LOSS OF TOTAL OR PARTIAL POWER FROM THE SATELLITE WOULD OCCUR SEVERAL TIMES A YEAR DUE TO SATELLITE ROUTINE MAINTENANCE, SHADOWING EFFECT FROM THE EARTH AND FROM OTHER SPS SYSTEMS. WHEN THESE OUTAGES ARE PREDICTABLE AND SCHEDULED IT SHOULD HAVE A MINIMAL EFFECT ON THE UTILITY SYSTEMS INTEGRITY AND OPERATIONS SINCE THE TIMING OF SUCH OUTAGES WOULD BE DURING THE LOW LOAD PERIODS.

HOWEVER, ASSUMING A SIGNIFICANT PENETRATION OF SPS POWER SYSTEMS IN THE FUTURE, THE GENERATION RESERVE NEEDED TO MAINTAIN THE UTILITY SERVICE RELIABILITY WOULD BE EXPECTED TO INCREASE TO COVER THE EMERGENCY SHUTDOWNS OF THE SATELLITE. MORE DETAILED UTILITY SYSTEM STUDIES WOULD BE NEEDED TO PREDICT THE IMPACT ON RESERVE LEVELS FROM EMERGENCY SPS POWER SYSTEM OUTAGES.

GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

OPERATION AND MAINTENANCE

LOSS OF POWER FROM SATELLITE:

- SCHEDULED MAINTENANCE
  - LOW IMPACT IN LOW LOAD PERIODS
- EMERGENCY SHUTDOWN
  - POTENTIAL NEED FOR INCREASED GENERATION RESERVE

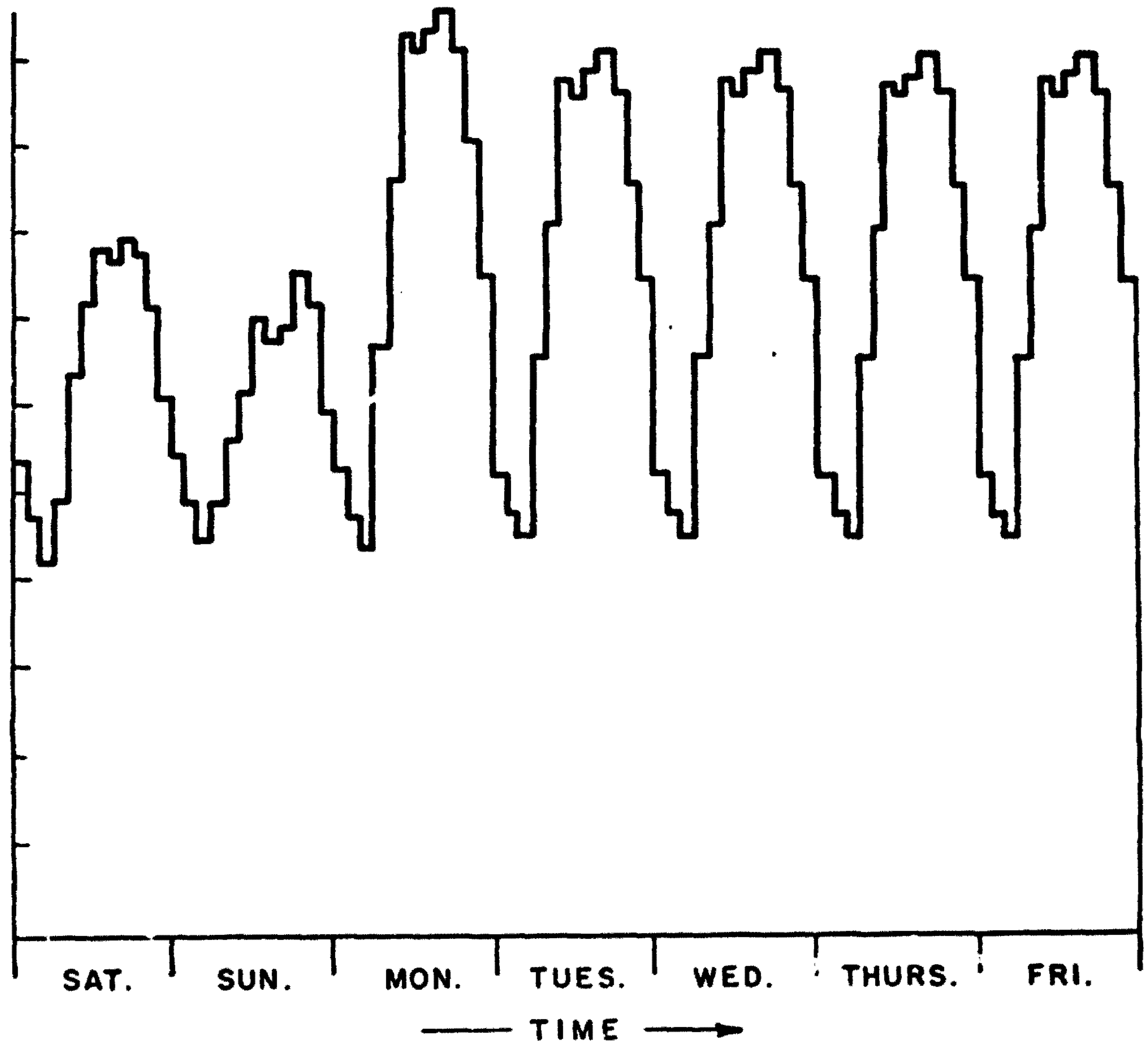
ELECTRIC UTILITY WEEKLY DEMAND PROFILE

THE ELECTRIC UTILITY DEMAND IS CHANGING BY THE SECOND, MINUTE, HOUR, DAY, SEASON AND YEAR. THE FIGURE SHOWS A TYPICAL ELECTRIC UTILITY WEEKLY DEMAND PROFILE.

THE GENERATION CAPACITY APPLIED TO SERVE THIS DEMAND MUST IN TOTAL BE CONTROLLABLE TO ADOPT TO THE DEMAND PROFILE. WITH A SIGNIFICANT AMOUNT OF INFLEXIBLE GENERATION INTRODUCED, THIS WILL CAUSE OPERATIONAL PROBLEMS IN THE LOW LOAD PERIODS. ONE RULE OF THUMB IS THAT AT LEAST 50% OF MINIMUM LOAD SHOULD BE CAPABLE OF CONTRIBUTING TO SYSTEM FREQUENCY CONTROL. THIS ASSUMPTION WOULD BE FOR A TOTALLY INTERCONNECTED SYSTEM WITH SEVERAL UTILITY POOLS CONNECTED TOGETHER. "SPINNING RESERVE" IS OPERATIONAL RESERVE CAPACITY PARTLY CONNECTED TO THE LINE AND PARTLY AVAILABLE WITHIN MINUTES. THIS RESERVE IS DESIGNED TO BE AVAILABLE IMMEDIATELY FOLLOWING EMERGENCY SHUTDOWNS OF OPERATING EQUIPMENT. IF THE UNIT SIZE OF A SPS POWER PLANT IS LARGE AS COMPARED TO OTHER LARGE UNITS ON THE SYSTEM, AN INCREASE IN THE SPINNING RESERVE REQUIREMENTS WOULD BE EXPECTED.

THE UNIT SIZE OF SPS POWER PLANT WOULD ALSO HAVE AN INCREASING EFFECT ON THE SYSTEM OVERALL GENERATION RESERVE FOR RELIABILITY. A FURTHER EVALUATION WOULD BE NEEDED INCLUDING UTILITY SYSTEM PARAMETERS, INDIVIDUAL UNIT RELIABILITY, AND SPS POWER SYSTEM OVERALL RELIABILITY AS INPUT DATA TO ASSESS THE MAGNITUDE OF THIS IMPACT.

ELECTRIC UTILITY WEEKLY DEMAND PROFILE



GROUND POWER COLLECTION AND TRANSMISSION SYSTEM

SUMMARY AND CONCLUSIONS

- DESIGN USING CONVENTIONAL TECHNOLOGY
- CONVENTIONAL UTILITY DESIGN PRACTICE
- HIGH RELIABILITY AND AVAILABILITY
- INSTALLED COST APPROX. 135 $/KW

AREAS FOR FURTHER STUDY:

- SPS-UTILITY SYSTEM OPERATIONAL INTERFACE
  - RELIABILITY AND AVAILABILITY
  - OPERATIONAL ECONOMICS