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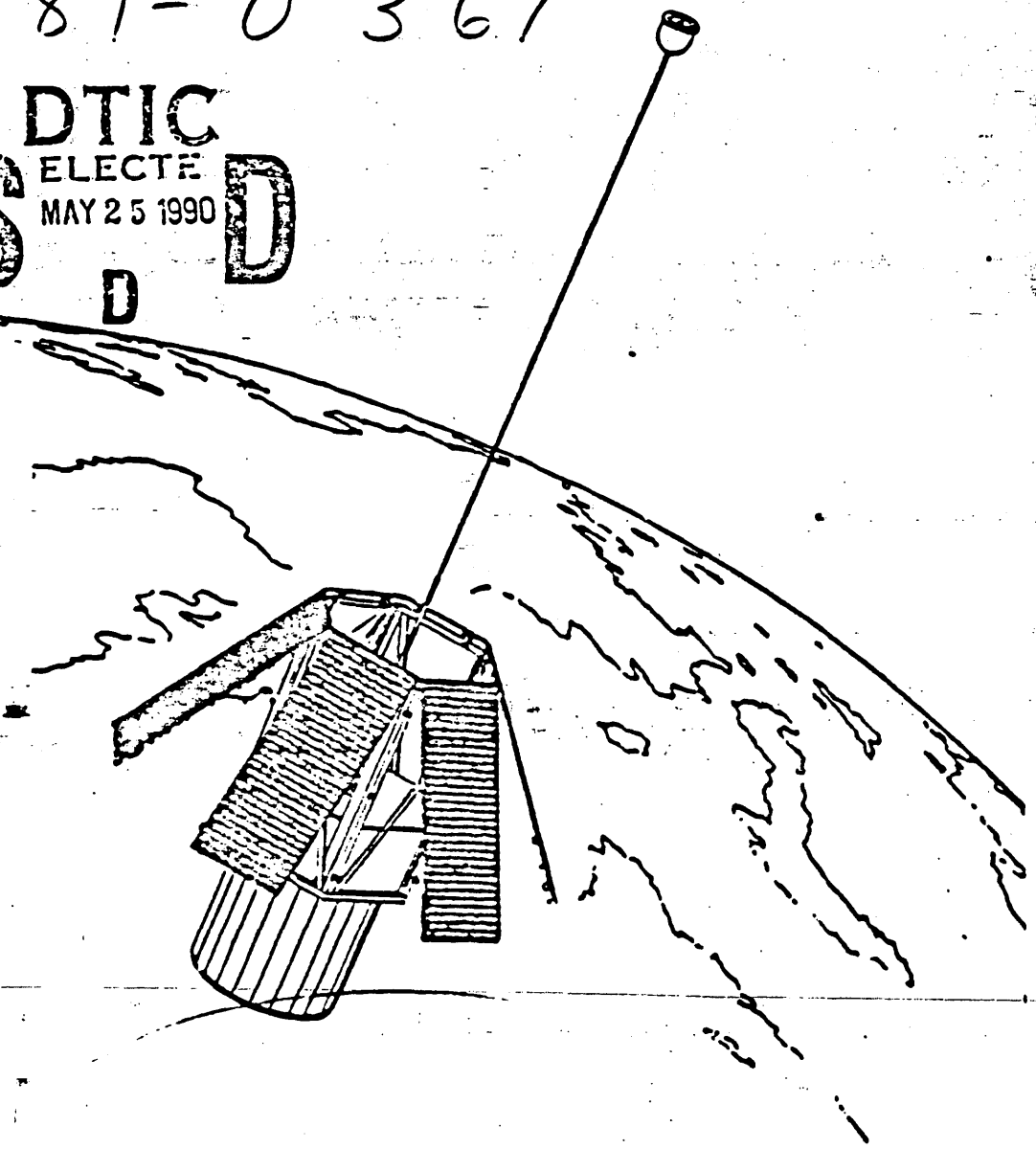

# MULTIPLE SATELLITE SYSTEM PROGRAM

Ball Aerospace Systems Division
SPACE SYSTEMS

BOULDER, COLORADO 80306

# MULTIPLE SATELLITE SYSTEM PHASE I
# SATELLITE INTEGRATOR REPORT

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20 March 1987

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The emphasis of the effort documented in this report was to perform satellite integrator trades to develop the Preliminary Satellite Design Report for the multiple satellite system (MSS). The design trades were for a low cost satellite for a high bandwidth global communication system integrating burst radio, digital processor and electronically steerable antenna designs. This objective was to be met by analysis and trade-offs with the other MSS team members, providing a means of iterating the satellite, radio and antenna designs to reach a cost effective solution.

The data presented in this report identifies the satellite interfaces with the payload - radio/antenna, Flight operations - launch, orbits, altitudes, etc., and the Ground user or communications station and relates the impact of these interfaces with the satellite subsystems - power, attitude control, structure, etc., in parametric form, if possible, such that cost/performance trades could be performed. The final resolution of the system trades are being executed by the MSS system engineer in the MSS A specification.


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## SUMMARY

This report describes the analyses and trades performed by BASD during Phase I of the Multiple Satellite System Program. These analyses and trades were performed to develop the Preliminary Satellite Design Report for the multiple satellite system (MSS). The design trades were for a low cost satellite for a high bandwidth global communication system integrating burst radio, digital processor and electronically steerable antenna designs. This objective was to be met by analysis and trade-offs with the other MSS team members, providing a means of iterating the satellite, radio and antenna designs to reach a cost effective solution. Communication was maintained through telephone conversations, meetings between contractors and working groups meetings between all contractors.

The approach described by the analyses and trades in this document was to identify the satellite interfaces and relate the impact of these interfaces in parametric form, if possible, such that cost/performance trades could be made. The main interfaces were Flight operations, payload, and the Ground user or final communication station.

Flight operations were concerned with system parameters such as the satellite orbits, altitude, environment, launch and deployment strategies versus the satellite lifetime and the satellite launch cost. Also to be considered were the impacts of the satellite communication range, earth multipath, and satellite altitude differentials versus the number of satellites required for the system communication mission.

The payload interface concerned the satellite power, attitude control, structure, and telemetry and command necessary for the radio and antenna. Power subsystem analyses presented the cost estimates for satellite power including solar array cost, battery cost and launch cost, analysis of solar array configuration for optimum energy output, and analysis of the required battery size, the satellite charge controller, and the impact of peak power averaging of the transmit power. The attitude control interface was concerned with the cost of various levels of attitude control versus antenna cost and complexity in order to obtain the system pointing of the antenna for communication. Structure and telemetry and command discussions mainly addressed requirements and possible approaches to obtain those requirements. Also, addressed as a payload interface were the concerns of the operation of high RF power components in space (vacuum) and the implications on the system testing.

The Ground user interface discussed the impact of the ground user operating with low earth orbit satellites and suggested possible configurations of Ground user RF power and Antennas and even possible changes to the spacecraft which could lower the total system cost.

The analyses in this report were used to generate a preliminary design plan for the satellite which included recommended satellite specifications and a preferred approach for each specification. Preliminary design plans were also presented by the other team members. The final resolution of the system trades are being executed by the MSS system engineer in the MSS A specification.

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TABLE OF CONTENTS


| | | Page |
|---|---|---|
| 1.0 | INTRODUCTION | 1 |
| 2.0 | MULTIPLE SATELLITE SYSTEM PROGRAM DESCRIPTION | 2 |
| 3.0 | BASD MSSP STUDY OVERVIEW | 5 |
| 4.0 | SATELLITE INTEGRATOR SYSTEM TRADES | 9 |
| 4.1 | SATELLITE ORBITS | 9 |
| 4.1.1. | ORBIT ALTITUDE | 9 |
| 4.1.1.1 | EXOATMOSPHERIC TEMPERATURE AND ORBIT DECAY | 10 |
| 4.1.2 | SPACECRAFT ORBITAL DENSITY | 19 |
| 4.1.2.1 | ORBITAL DENSITY ANALYSIS | 19 |
| 4.1.3 | SPACECRAFT ORBITAL DISPERSION | 26 |
| 4.1.4 | SPACECRAFT LAUNCH STRATEGIES | 34 |
| 4.1.4.1 | MSSP LAUNCH STRATEGY ANALYSIS | 34 |
| 4.1.4.2 | MSSP DERIVED LAUNCH REQUIREMENTS | 36 |
| 4.1.4.3 | CARRIER VEHICLES | 37 |
| 4.1.4.4 | CARRIER FUEL REQUIREMENTS | 41 |
| 4.1.4.5 | LAUNCH STRATEGY CONCLUSIONS | 41 |
| 4.1.5 | SPACECRAFT LAUNCH COST | 41 |
| 4.1.5.1 | LAUNCH VEHICLE COST | 42 |
| 4.1.5.2 | SATELLITE SIZE IMPACT | 42 |
| 4.1.5.3 | ORBIT ACHIEVEMENT COST | 45 |
| 4.1.6 | SPACECRAFT ORBITAL ENVIRONMENT | 46 |
| 4.1.6.1 | RADIATION DOSAGE | 46 |
| 4.1.6.2 | ATOMIC OXYGEN | 50 |
| 4.1.7 | SATELLITE ORBITS CONCLUSION | 51 |
| 4.2 | GROUND USER | 52 |
| 4.2.1 | GROUND USER/SPACECRAFT ANTENNA TRADE | 52 |
| 4.2.2 | GROUND USER ANTENNA CONSIDERATIONS | 53 |
| 4.2.3 | GROUND USER QUALITY FACTOR | 60 |
| 4.2.4 | ANTENNA GAIN/RF POWER TRADE | 64 |
| 4.2.5 | CONCLUSION | 65 |
| 4.3 | MSSP COMMUNICATION SYSTEM | 66 |
| 4.3.1 | SATELLITE RANGE | 66 |
| 4.3.2 | 625/675 ALTITUDE DIFFERENTIAL | 67 |
| 4.3.2.1 | POINTING AND RANGE ERROR | 67 |
| 4.3.2.2 | IMPACT OF ERRORS | 71 |
| 4.3.3 | MAXIMUM CROSS LINK RANGE | 81 |
| 4.3.4 | LINK EQUATION | 86 |
| 4.3.5 | CONCLUSION | 88 |
| 5.0 | SATELLITE INTEGRATOR SUBSYSTEM TRADES | 94 |
| 5.1 | SATELLITE POWER | 94 |
| 5.1.1 | POWER SUBSYSTEM COST ANALYSIS | 96 |
| 5.1.1.1 | SOLAR ARRAY SIZING | 97 |
| 5.1.1.2 | BATTERY SIZING | 97 |
| 5.1.1.3 | POWER SUBSYSTEM COST ESTIMATE | 100 |
| 5.1.1.4 | POWER SUBSYSTEM COST REDUCTION | 103 |
| 5.1.2 | SOLAR ARRAY OPTIMIZATION | 110 |
| 5.1.2.1 | SELECTED ARRAY CONFIGURATIONS | 123 |
| 5.1.2.2 | ARRAY OPTIMIZATION CONCLUSION | 128 |
| 5.1.3 | POWER SUBSYSTEM AND CONTROLLER ANALYSIS | 129 |

TABLE OF CONTENTS (cont.)


| | | |
|---|---|---|
| 5.1.3.1 | BASIC TENETS OF POWER SUBSYSTEM DESIGN | 129 |
| 5.1.3.2 | MSSP SATELLITE REQUIRED POWER | 131 |
| 5.1.3.3 | BATTERY SIZE | 131 |
| 5.1.4 | POWER SUBSYSTEM TRADES | 133 |
| 5.1.4.1 | MSSP EXTRA BATTERY STORAGE | 133 |
| 5.1.4.2 | SOLAR ARRAY TRADE | 136 |
| 5.1.4.2.1 | ARRAY ORIENTATION DETERMINES ARRAY EFFICIENCY | 138 |
| 5.1.4.2.2 | LARGER SPACECRAFT BATTERIES | 144 |
| 5.1.5 | POWER SUBSYSTEM CONCLUSION | 150 |
| 5.2 | SATELLITE ATTITUDE CONTROL | 151 |
| 5.2.1 | GRAVITY GRADIENT POINTING ERROR SOURCES | 151 |
| 5.2.2 | GRAVITY GRADIENT SUBSYSTEM COST ESTIMATES | 157 |
| 5.2.3 | SIMULATION RESULTS | 160 |
| 5.2.4 | CONCLUSION | 162 |
| 5.3 | SPACECRAFT COMMAND AND TELEMETRY | 167 |
| 5.3.1 | SPACECRAFT TELEMETRY | 167 |
| 5.3.2 | SPACECRAFT COMMANDS | 169 |
| 5.4 | SPACECRAFT CARRIER CONFIGURATION | 170 |
| 5.4.1 | SPACECRAFT CARRIER CONFIGURATION | 170 |
| 5.5 | RADIO/ANTENNA INTERFACES | 178 |
| 5.5.1 | ANTENNA POINTING ERRORS AND ATTITUDE CONTROL | 178 |
| 5.5.2 | ANTENNA SIZE | 185 |
| 5.5.3 | MSSP RF POWER CONCERNS | 190 |
| 6.0 | CONCLUSION AND RECOMMENDATION | 197 |


TABLES


| | | |
|---|---|---|
| TABLE 2-1 | MULTIPLE SATELLITE SYSTEM | 3 |
| TABLE 3-1 | MSSP SYSTEM ENGINEERING REPORTS | 6 |
| TABLE 3-2 | MSSP SER AREA OF DESIGN | 8 |
| TABLE 4-1 | RADIATION DOSAGE SPECIFICATION PARAMETERS | 49 |
| TABLE 5-1 | POWER SUBSYSTEM DESIGN REQUIREMENTS | 95 |
| TABLE 5-2 | DEPTH OF DISCHARGE VERSUS BATTERY SIZE | 132 |
| TABLE 5-3 | TRADES OF YAW-AROUND SPACECRAFT | 146 |
| TABLE 5-4 | PERFORMANCE OF BASIC GRAVITY GRADIENT SYSTEMS | 154 |
| TABLE 5-5 | QUANTITATIVE FLIGHT PERFORMANCE OF GEOS-II | 156 |
| TABLE 5-6 | ACDS COMPONENT CHARACTERISTICS | 158 |
| TABLE 5-7 | ACDS SYSTEM DESIGN COMPARISON | 159 |
| TABLE 5-8 | PRELIMINARY TELEMETRY LIST | 168 |
| TABLE 5-9 | SYSTEM POINTING ERRORS VS. ANTENNA GRANDULARITY | 184 |
| TABLE 6-1 | PROMINENT SATELLITE DESIGN ASPECTS | 198 |


Figures


| | | |
|---|---|---|
| Figure 2-1 | MSSP Phase I Structure | 4 |
| Figure 4-1 | Calculated Exoatmospheric Temperature | 12 |
| Figure 4-2 | Orbit Decay (King-Hele) | 12 |
| Figure 4-3 | Orbit Lifetimes | 13 |

TABLE OF CONTENTS (cont.)


| | | |
|---|---|---|
| Figure 4-4 | Average Altitude and Separation Δhl = 10 km T∞ = 1400 deg k | 14 |
| Figure 4-5 | Average Altitude and Separation Δhl = 30 km T∞ = 1400 deg k | 14 |
| Figure 4-6 | Average Altitude and Separation Δhl = 50 km T∞ = 1400 deg k | 15 |
| Figure 4-7 | Average Altitude and Separation Δhl = 10 km T∞ = 1100 deg k | 15 |
| Figure 4-8 | Average Altitude and Separation Δhl = 30 km T∞ = 1100 deg k | 16 |
| Figure 4-9 | Average Altitude and Separation Δhl = 50 km T∞ = 1100 deg k | 16 |
| Figure 4-10 | Average Altitude and Separation Δhl = 10 km T∞ = 900 deg k | 17 |
| Figure 4-11 | Average Altitude and Separation Δhl = 30 km T∞ = 900 deg k | 17 |
| Figure 4-12 | Average Altitude and Separation Δhl = 50 km T∞ = 900 deg k | 18 |
| Figure 4-13 | Ideal Uniform Density & Distribution of Spacecraft | 21 |
| Figure 4-14 | ESL-TM1632 Distribution and Density of Spacecraft | 23 |
| Figure 4-15 | Alternate Orbit Set Distribution and Density of Spacecraft | 25 |
| Figure 4-16 | Satellite Velocity versus Altitude | 28 |
| Figure 4-17 | Satellite In-orbit Dispersion | 28 |
| Figure 4-18 | Nodal Regression versus Altitude (Ai=28.5) | 29 |
| Figure 4-19 | Nodal Regression Separation of Satellites versus Altitude (Ai=57) | 29 |
| Figure 4-20 | Nodal Regression versus Altitude (Ai=57) | 30 |
| Figure 4-21 | Nodal Regression Separation of Satellites versus Altitude (Ai=57) | 30 |
| Figure 4-22 | Satellite Dispersion/Ground Coverage First Launch 0 Months | 31 |
| Figure 4-23 | Satellite Dispersion/Ground Coverage Second Launch 3 Months | 31 |
| Figure 4-24 | Satellite Dispersion/Ground Coverage Third Launch 6 Months | 32 |
| Figure 4-25 | Satellite Dispersion/Ground Coverage 9 Months | 32 |
| Figure 4-26 | Satellite Dispersion/Ground Coverage 12 Months | 33 |
| Figure 4-27 | Minimum Time to First Satellite Lapping vs. Separation Velocity Components (Ai=57 deg 24 satellites) | 39 |
| Figure 4-28 | Minimum Time to First Satellite Lapping vs. Separation Velocity Components (Ai=57 deg 24 satellites) | 39 |

TABLE OF CONTENTS (cont.)


| | | |
|---|---|---|
| Figure 4-29 | Minimum Time to First Satellite Lapping vs Separation Velocity Components (Ai=80 deg 24 satellites | 40 |
| Figure 4-30 | Minimum Time to First Satellite Lapping vs Separation Velocity Components (Ai=80 deg 24 satellites) | 40 |
| Figure 4-31 | Possible MSSP Launch Vehicles | 43 |
| Figure 4-32 | Satellite Per Launch versus Satellite Diameter | 44 |
| Figure 4-33 | Satellite Launch Cost versus Satellite Diameter | 44 |
| Figure 4-34 | Radiation Dosage - 6 yr, 28 deg Inclination vs. Shielding | 47 |
| Figure 4-35 | Radiation Dosage - 6 yr, 57 deg Inclination vs. Shielding | 47 |
| Figure 4-36 | Radiation Dosage - 6 yr, 95 deg Inclination vs. Shielding | 48 |
| Figure 4-37 | Radiation Dosage vs. Duration of Lifetime | 48 |
| Figure 4-38 | Antenna Gain vs. Area | 54 |
| Figure 4-39 | Antenna Beamwidth vs. Area | 54 |
| Figure 4-40 | Multipath | 55 |
| Figure 4-41 | Total Signal Level Variation (Delta Gain) versus Reflected Signal Level | 55 |
| Figure 4-42 | Required Gain vs. Elevation Angle for 20 dB Reflection Level | 57 |
| Figure 4-43 | Antenna Size vs Elevation Angle | 57 |
| Figure 4-44 | Gain Delta versus Elevation Angle Due to Path Loss Delta wrt 10 Deg Elevation | 58 |
| Figure 4-45 | Required Antenna Gain vs. Elevation Angle Required 17 dB Gain at 10 Deg Elevation | 58 |
| Figure 4-46 | Required S/C Antenna Scan Angle vs. Ground User Elevation Angle | 59 |
| Figure 4-47 | Number of Satellites vs. Elevation Angle Assumption: Uniform Satellite | 61 |
| Figure 4-48 | Number of Satellites vs. Elevation Angle Per Cell: Assumption: Uniform Satellite | 62 |
| Figure 4-49 | Gimbal Rate vs. Elevation Angle | 62 |
| Figure 4-50 | Cumulative Time vs. Elevation Angle for Spacecraft Overhead Pass | 63 |
| Figure 4-51 | Number of S/C to Cell Divided by Gimbal Rate versus Elevation Angle | 63 |
| Figure 4-52 | 4 deg and 50 km Separation | 68 |
| Figure 4-53 | Relationship of Antenna Pointing Angle and Satellite Range | 69 |
| Figure 4-54 | Pointing Angle and Range with Altitude Difference | 70 |
| Figure 4-55 | Range Versus Design Angle for 675 km Altitude | 72 |

TABLE OF CONTENTS (cont.)


| | | |
|---|---|---|
| Figure 4-56 | Range versus Design Angle for 675, 625 Altitude.... | 73 |
| Figure 4-57 | Link Real Angle vs. Design Angle Satellite 625/675 km Offset (Design Nagle 0-22)..... | 74 |
| Figure 4-58 | Link Real Angle vs. Design Angle Satellite 625/675 km Offset (Design Angle 1-22)..... | 75 |
| Figure 4-59 | Range, Ant Gain and Total Variation versus Design Angle..... | 76 |
| Figure 4-60 | Range, Ant Gain and Total Variation versus Design Angle..... | 78 |
| Figure 4-61 | Total Link Variation versus Design Angle for 625 and 675 km ..... | 79 |
| Figure 4-62 | Total Link Variation versus Design Angle for 625 and 675 km..... | 80 |
| Figure 4-63 | Number of Spacecraft vs. Range for Uniform Global Distribution ..... | 83 |
| Figure 4-64 | Probability of Link to Other Spacecraft in One Direction (P1) or Two Directions (P2)..... | 85 |
| Figure 4-65 | K Factor versus K Factor for Various Antenna Gains..... | 89 |
| Figure 4-66 | Range (NMi) versus K Factor for Various Antenna Gains ..... | 90 |
| Figure 4-67 | Range (NMi) versus RF Power for 1 Mbit Data Rate... | 91 |
| Figure 4-68 | Range (NMi) versus RF Power for 5 Mbit Data Rate... | 92 |
| Figure 4-69 | Range (NMi) versus RF Power for 12.5 Mbit Data Rate..... | 93 |
| Figure 5-1 | Solar Array Size versus Avg Power Required..... | 98 |
| Figure 5-2 | Solar Array Weight versus Avg Power Required..... | 98 |
| Figure 5-3 | Solar Array Cost versus Avg Power Required..... | 99 |
| Figure 5-4 | Battery Size versus Avg Power Required..... | 99 |
| Figure 5-5 | Battery Weight versus Avg Power Required..... | 101 |
| Figure 5-6 | Battery Dimension versus Avg Power Required..... | 101 |
| Figure 5-7 | Battery Cost versus Avg Power Required..... | 102 |
| Figure 5-8 | Battery Parameters vs. System Average Power..... | 104 |
| Figure 5-9 | Launch Cost versus Avg Power Required..... | 104 |
| Figure 5-10 | Total Power Cost versus Avg Power Required..... | 104 |
| Figure 5-11 | Battery Size versus Avg Power Avg Power Required C/2 Controller..... | 106 |
| Figure 5-12 | Battery Weight versus Avg Power Required C/2 Controller..... | 106 |
| Figure 5-13 | Battery Dimension versus Avg Power Required C/2 Controller..... | 107 |
| Figure 5-14 | Battery Cost versus Avg Power Required C/2 Controller..... | 107 |
| Figure 5-15 | Launch Cost versus Avg Power Required C/2 Controller..... | 108 |

TABLE OF CONTENTS (cont.)


| | | |
|---|---|---|
| Figure 5-16 | Total Lower Cost versus Avg Power Required C/2 Controller | 108 |
| Figure 5-17 | Total Cost versus Avg Power Required C/2 Controller | 109 |
| Figure 5-18 | Possible Solar Array Configurations | 111 |
| Figure 5-19 | Sun Beca Angle for Inclination Angle ai=0 | 113 |
| Figure 5-20 | Beta Angle Equator Satellite | 114 |
| Figure 5-21 | Eclipse Angle versus Sun Angle Beta | 116 |
| Figure 5-22 | Eclipse Time (Percent of Orbit) versus Sun Angle Beta | 116 |
| Figure 5-23 | Eclipse Time versus Sun Angle Beta | 117 |
| Figure 5-24 | Illustration of Array Tilt | 117 |
| Figure 5-25 | Power Output Equations (Orbital Average) | 118 |
| Figure 5-26 | Power Output versus Outer Panel Tilt Angle (Panel Tilted Toward Velocity Direction 0 to 90 Degrees) | 119 |
| Figure 5-27 | Power Output versus Outer Panel Tilt Angle (Panel Tilted Toward Velocity Direction 0 to 55 Degrees) | 120 |
| Figure 5-28 | Power Per Sq Ft of Array versus Beta for Various Tilt Angles (Panel Tilted Toward Orbit Normal 0 to 90 Degrees) | 120 |
| Figure 5-29 | Power Per Sq Ft of Array versus Beta for Various Tilt Angles (Panel Tilted Toward Orbit Normal 0 to 90 Degrees | 122 |
| Figure 5-30 | Power Per Sq Ft of Array versus Beta for Various Tilt Angles (Panel Tilted Toward Orbit Normal, Negative Angle 0 to 45 Degrees) | 122 |
| Figure 5-31 | Power Per Sq Ft of Array versus Beta for Various Tilt Angles (Roof Top Array 0 to 55 Degrees) | 124 |
| Figure 5-32 | Power Output versus Outer Panel Tilt Angle (Power Output of Four Flush Panels) | 124 |
| Figure 5-33 | Power Output versus Outer Panel Tilt Angle (Four Flush Arrays with Top Array) | 125 |
| Figure 5-34 | Power Output versus Outer Panel Tilt Angle (Four Panel Tilted Array 0 to 55 Degrees) | 125 |
| Figure 5-35 | Power Output versus Outer Panel Tilt Angle (Four Panel Tilted Array 0 to 90 Degrees) | 126 |
| Figure 5-36 | Power Output versus Outer Panel Tilt Angle (Five Panel Mansard Array 0 to 55 Degrees) | 126 |
| Figure 5-37 | Power Output versus Outer Panel Tilt Angle (Five Panel Mansard Array 0 to 90 Degrees) | 127 |
| Figure 5-38 | Power Output versus Beta for Various Tilt Angles ("Omni' Array Tilt 0 to 55 Degrees) | 127 |

TABLE OF CONTENTS (cont.)


| | | |
|---|---|---|
| Figure 5-39 | Battery Capacity Needed versus Transmit Duty Factor (Depth of Discharge 20 to 80 Percent; Power System 40W, Transmit 60W Non-transmit 30W)... | 135 |
| Figure 5-40 | Battery Recharge Time in Orbits versus Duty Factor (After 2 Orbits at Duty Factor (40, 80, 30W system))..... | 135 |
| Figure 5-41 | Battery Capacity Needed versus Transmit Duty Factor..... | 137 |
| Figure 5-42 | Battery Recharge Time in Orbits versus Duty Factor (50, 80, 30W System)..... | 137 |
| Figure 5-43 | Sun Beta Angle for Inclination Angle..... | 139 |
| Figure 5-44 | Example of Beta Angle Variation..... | 141 |
| Figure 5-45 | Power of single Array Panel versus Beta for Various Tilt Angles..... | 142 |
| Figure 5-46 | Power of Single Array versus Days (Beta in 10 Deg Steps)..... | 145 |
| Figure 5-47 | Impact of Array Output Rate (Watts/Sq Ft) vs. Array Cost ($7/k Sq Ft)..... | 148 |
| Figure 5-48 | Battery Required vs. Array W/Sq Ft Integrated Average (112 Watts 60% DOD)..... | 149 |
| Figure 5-49 | Attitude Response (Three Rods, 0.1 AM2 Dipole, No Wheel, Solar Torques)..... | 161 |
| Figure 5-50 | Attitude Angles - Without Wheel Eddy Current Damper..... | 163 |
| Figure 5-51 | Three Rods, 0.1 AM2 Dipole, 0.25 NMS Wheel, Solar Torques..... | 164 |
| Figure 5-52 | Attitude Angles - With Wheel Eddy Current Damper... | 165 |
| Figure 5-53 | Satellite Carrier Concept 1..... | 171 |
| Figure 5-54 | Satellite Carrier Concept 2..... | 172 |
| Figure 5-55 | Satellite Carrier Concept 3..... | 173 |
| Figure 5-56 | Satellite Carrier Concept 4..... | 174 |
| Figure 5-57 | Configuration to Accommodate S/C with 28 in Diameter Antenna..... | 175 |
| Figure 5-58 | Satellites per Launch vs. Satellite Diameter..... | 177 |
| Figure 5-59 | Satellite Launch Cost vs. Satellite Diameter..... | 177 |
| Figure 5-60 | Maximum Antenna Gain versus Pointing Error..... | 180 |
| Figure 5-61 | Pointing Loss Gain (Pointing Errors of 6 to 16 Degrees)..... | 181 |
| Figure 5-62 | Illustration of Antenna Parameters..... | 182 |
| Figure 5-63 | Antenna Phase Error..... | 186 |
| Figure 5-64 | Antenna Beam Step: Degrees Per Step..... | 187 |
| Figure 5-65 | Antenna Step Angle vs. Antenna Radius (1/4 Wavelength Step)..... | 188 |

TABLE OF CONTENTS (cont.)


Figure 5-66 Antenna Array Length vs. Antenna Radius (1/4 Wavelength Phase Error) ..... 189
Figure 5-67 Antenna Beamwidth vs. Antenna Radius (1/4 Wavelength Phase Error) ..... 191
Figure 5-68 PA Turn On Illustration ..... 196
Figure 5-69 RF Configuration 1 ..... 196
Figure 5-70 RF Configuration 2 ..... 196
Figure 6-1 MSS Satellite Concept ..... 199

## 1. Introduction

The following report presents Ball Aerospace Systems Division's (BASD) effort on the Phase I study of the Multiple Satellite System Program (MSSP). The final product of this study is the Preliminary Design Report of a satellite bus upon which the MSSP burst radio and antenna are integrated. This final report will describe the various analyses and trades performed by BASD to derive the Preliminary Design report. This document will describe the selected approach, as well as approaches, ideas, and concerns which were analyzed and discarded.

Section 2 provides a description of the MSS program and the objective of the total program and Phase I study. Section 3 presents an overview of the BASD approach during the Phase I study. Sections 4 and 5 describe the trades and analyses made by BASD as the satellite integrator during the Phase I study. The conclusions and recommendations of the BASD MSSP Phase I study will be discussed in Section 6.

## 2. Multiple Satellite System Program Description

This section presents an overview of the Multiple Satellite System (MSS) to justify the trades and analyses made and criteria used for the selection of the preferred approach. This overview is not a complete description of the objectives and requirements of the MSS, but should be sufficient to provide an understanding for this report.

As stated in the Multiple Satellite System Program Management Plan, "The Multiple Satellite System (MSS) is a concept for a proliferated low-altitude system intended to provide a global packet communication network for data and voice. Its primary objective is to provide a highly survivable network that can continue to support a minimum level of communication services in the presence of intentional jamming, loss of a significant fraction of the satellites, and/or loss of the terrestrial control functions. A second objective is to provide efficient, wideband communications under benign conditions."

Previous developmental work has concentrated on space segment configuration, preliminary antenna design studies, and link communications architectures. Applying this work to the design of the satellite system is the next logical step. The basic system would consist of approximately 240 satellites orbiting at a low earth orbit altitude of 350 to 400 nmi servicing a global user community of 200 to 1000 users. The satellites would consist of the satellite structure and subsystems, a burst radio, an antenna for crosslink and up/down communication, and a network processor to perform the system routing and control.

The key issue in a program such as MSS is the cost of the satellite. Because of the quantity of satellites, the individual satellite cost must be reduced. The cost of earlier types of communication satellites can be reduced for two reasons: 1. small distances between satellites reduces satellite performance required and 2. large production build techniques can be used.

The MSS program is a three-phased approach described as follows:

Phase I: Establish a system definition and preliminary design

Phase II: Hardware development and tests

Phase III: Full scale development of the MSS

Designing a cost-effective system by examining the design areas and tradeoffs between hardware and software, or antenna and attitude control, or antenna gain and radio RF power was the goal of Phase I. This was completed with the interaction and technological expertise of several different contractors. The MSSP Phase I team members are listed in Table 2-1. The organizational structure of the Phase I participants is shown in Figure 2-1. MA-COM Linkabit Incorporated was the system engineering contractor. Defense Systems Incorporated was under contract to examine the application of low-cost satellite design technologies to MSSP. The contractors were to help resolve the issues arising in system definition.

TABLE 2-1 MULTIPLE SATELLITE SYSTEM PROGRAM (MSSP) CONTRACTORS

| AREA | CONTRACTORS |
|---|---|
| Burst Radio | Harris Corporation<br>Cincinnati Electronics |
| Antenna Development | Rockwell<br>Harris Corporation |
| Satellite Integration | Ball Aerospace<br>RCA Astro |
| Support Contracts | Qualcomm Incorporated<br>Space Communications Company<br>COMSAT Laboratories |

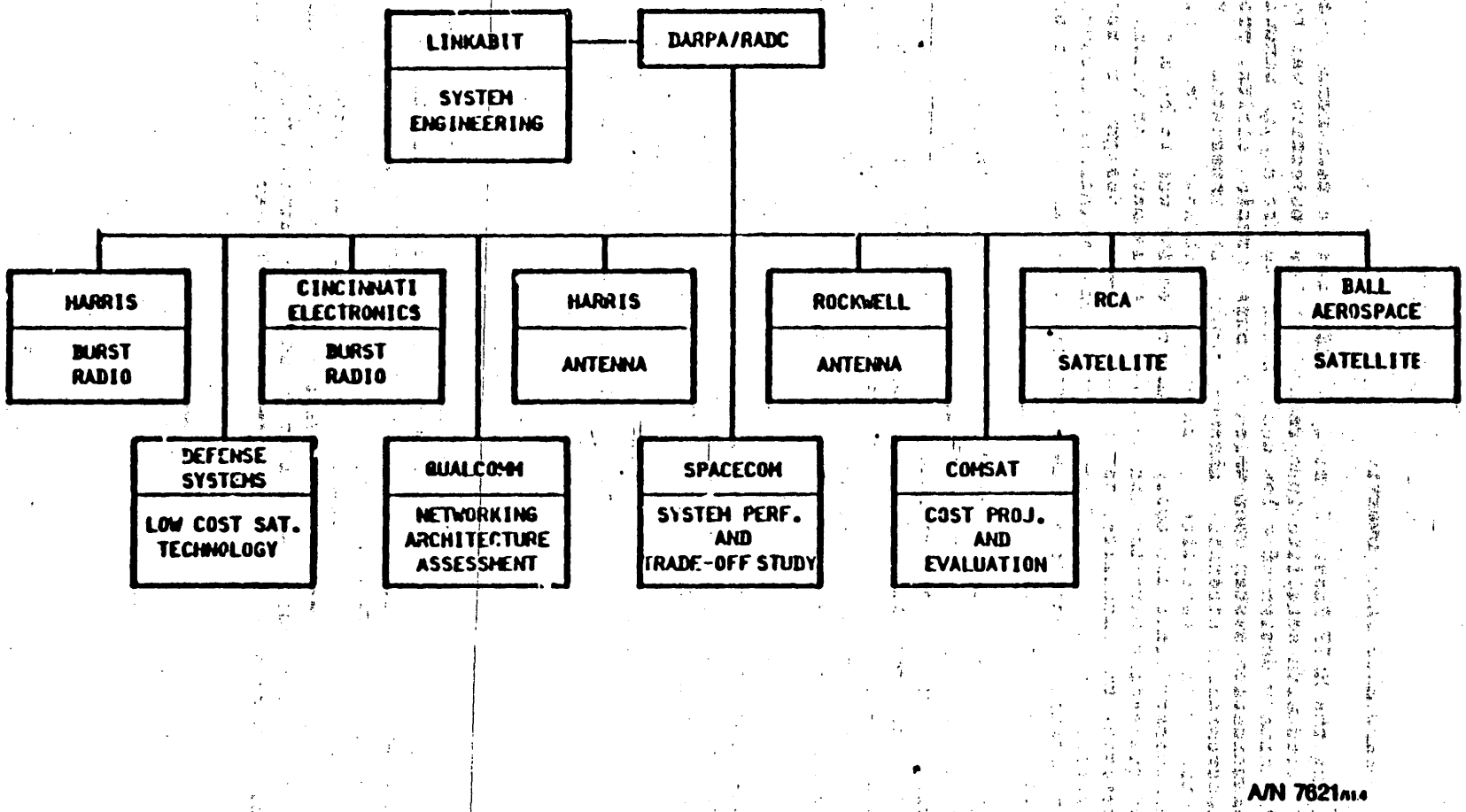


Figure 2-1 MSSP Phase I Structure

## 3. BASD MSSP Study Overview

During the MSSP Phase I study, BASD was a team member (contractor) involved with satellite integration. The objective was to develop a preliminary design of a low cost satellite for a high bandwidth global communication system upon which a burst radio, digital processor and electronically steerable antenna would be integrated. Production, testing, and satellite deployment techniques were also to be incorporated into the design. This objective was to be met by analysis and trade-offs with the other MSSP team members, providing a means of iterating the satellite, radio and antenna designs to reach a cost-effective solution. Communication was maintained through telephone conversations, meetings between contractors, and working group meetings between all contractors.

BASD's approach to the study was to address the issues and concerns identified at the working group meetings, or by BASD, in short memo form or system engineering reports (SERs) and distribute these to the other team members. It was hoped that this approach would maximize team communication and identify errors in any analysis, resulting in a thorough discussion resolving the issues. The SERs, rewritten as synopses, provide the basis of this report.

Table 3-1 lists the titles of the BASD SERs. The SERs, listed in numerical order, are also in chronological order. Letters indicating the area of the MSSP design to which the SER directly applies are located beside the SERs. Table 3-2 lists the area of design concern for each letter. As can be seen in Table 3-1, the application of the SERs varied throughout the Phase 1 study. Several SERs have almost the same titles, such as "MSSP Preliminary Power Analysis" and "Power System Second Cut". This is a reflection of iterations of the analysis as new information was added by the team members and working group meetings. This report discusses all analyses results and SERs to describe what system trades were made.

The sections of this report address either: a satellite subsystem, the satellite programmatic concerns, an important interface such as the antenna or the radio, or basic system concerns such as the satellite orbits. The topics are confined and to BASD's concerns as the satellite integrator, although to the radio and antenna interface many other fine analyses were performed by the other MSSP team members. However, this report will address only the narrow aspects of the design relevant to a satellite integrator and BASD's work on this study.

TABLE 3-1
MSSP SYSTEM ENGINEERING REPORTS

| NUMBER | DATE | TITLE | DESIGN AREA |
|---|---|---|---|
| 01 | 03/04/86 | Attitude Control vs Antenna Pointing | D |
| 02 | 03/04/86 | Max Ant. Gain vs Pointing Error | D |
| 03 | 03/19/86 | Solar Array Proj. Area vs Panel Tilt | A |
| 04 | 03/25/86 | Satellite Dispersion | G |
| 05 | 03/19/86 | Antenna Point Geometry | D |
| 06 | 03/24/86 | Antenna Radius vs Beam Steps | D |
| 07 | 03/24/86 | MSSP Ant. Gain vs Range & Pointing Error | D |
| 08 | 03/25/86 | Supplier Reliability Requirements | C |
| 09 | 03/26/86 | MSSP Preliminary Power Analysis | A |
| 010 | 04/02/86 | Spacecraft Antenna Strawman | D |
| 011 | 04/07/86 | Candidate Gravity Gradient Stab. Sys. | B |
| 012 | 04/08/86 | S/C Range vs Min. Ht. Above Earth Comm. | E |
| 013 | 04/08/86 | MSSP Link Equation | E |
| 014 | 04/02/86 | Battery Filter | A |
| 015 | 04/11/86 | Satellite Lifetime | C |
| 016 | 04/15/86 | Satellite Min. Range vs Altitude Delta | E |
| 017 | 04/17/86 | Considerations on Dyn. of Slow. Spin. Gravity Gradient Stabilized Satellite | B |
| 018 | 04/30/86 | Production Input on Requirements | C |
| 019 | 05/01/86 | MSSP User Concerns | F |
| 020 | 05/12/86 | Spacecraft Orbital Density | G |
| 021 | 07/10/86 | Linearized Equa. of Att. Motion in the Stan. Orbital Reference Frame | B |
| 022 | 06/04/86 | Power Systems Second Cut | A |
| 023 | 06/05/86 | Att. Pointing Perform. of the Basic Gravity Gradient Stabilized S/C | B |
| 024 | 06/06/8, | MSSP Orbital Decay/Separation Study | G |
| 025 | 06/18/86 | Radiation Dosage vs Shielding Depth | G |
| 026 | 06/19/86 | Suggested MSSP Test Levels | C |
| 027 | 06/23/86 | Att. Determ. & Cont. Hdwr ROM's for MSSP | B |
| 028 | 06/25/86 | Poten. Reduc. in Sys. Costs via Selective Increases in Element Costs | E |
| 029 | 07/02/86 | Grd. Station Ant. User Elevation Angle | F |
| 030 | 07/09/86 | MSSP Magnetic Considerations | C |
| 031 | 07/16/86 | A Disc. of Orbit Selection Criteria | G |
| 032 | 07/28/86 | Cal. of Rad. Dosage vs Mission Duration | G |
| 033 | 07/18/86 | Preliminary TLM Listing MSSP | H |
| 034 | 07/24/86 | Maximum Range | E |
| 035 | 08/05/86 | S/C Carrier Config. for Shuttle Launch | H |
| 036 | 08/20/86 | Carrier Configuration to Accommodate S/C with 28 inch Antenna | H |
| 037 | 08/25/86 | MSSP 100 Watt Operations Concerns | I |
| 038 | 09/12/86 | MSSP Impact of Satellite Size | H |

TABLE 3-1 (CONT.)
MSSP SYSTEM ENGINEERING REPORTS

| | | | |
|---|---|---|---|
| 039 | 09/24/86 | Ground User Parameters | F |
| 040 | 10/17/86 | Inv. of Sys. Implic. of Incr. the No. of Common Nodes Between Two Points | E |
| 041 | 10/21/86 | 625/675 Altitude Differential | G |
| 042 | 10/24/86 | Atomic Oxygen. | G |
| 043 | 10/31/86 | MSSP Solar Array Trades | A |
| 044 | 1/06/87 | Simplified Approach Toward Decay Orbit | G |

TABLE 3-2
MSSP SER AREA OF DESIGN

| | |
|---|---|
| A | Power subsystem |
| B | Attitude subsystem |
| C | Programmatic concerns |
| D | Satellite/Antenna interface |
| E | Communication system |
| F | Ground user interface |
| G | Orbit concerns |
| H | Satellite configuration |
| I | Satellite/Radio interface |

## 4.0 SATELLITE INTEGRATOR SYSTEM TRADES

The satellite integrator system trades involve those areas of design which are not necessarily satellite hardware or payload but are system trades which affect the hardware. The primary areas analyzed during the MSSP Phase I study were the required satellite orbits, the impact of the ground user, the communication ranges vs. the number of satellites required, and the programmatic concerns which can drive the cost of the satellite system.

Many of the topics analyzed affect several areas of concern. for example: the satellite's orbit selection will actually affect the ground user, the communication range, and the programmatic concern of the orbit environment (atomic oxygen and radiation). This makes a discussion of a certain topic under a specific area of concern somewhat arbitrary.

### 4.1 SATELLITE ORBITS

The orbits selected for the MSS program will directly affect the design, launch, and life cycle cost of the MSSP satellites. The altitude selected will affect the projected lifetime and the radiation and environmental levels experienced by the satellite and payload.

The number of satellite orbit planes and inclination angles, driven by the desire to obtain complete global communication, will increase the cost of launching the satellites. That cost can be much greater than the cost of the individual satellites. Therefore, a prime concern will be to maximize communication performance while minimizing system cost.

#### 4.1.1. ORBIT ALTITUDE

The satellite altitude for this study was stated to be between 350 and 400 nmi or between 630 and 740 km. This altitude range is not a fixed specification but a starting point for the system analysis and is acceptable for the communication mission. The prime concerns of the initial satellite altitude analysis were:

a) Satellite lifetime: 5 years

b) Satellite altitude differentials less than 50 km

c) Satellite launch cost

Fundamental to this analysis was the necessity of establishing an approach for estimating the exoatmospheric temperature. It is crucial for spacecraft drag and orbital decay estimation.

Implicit in the requirements to minimize satellite launch cost is the requirement to minimize satellite altitude. The cost differential to launch a single satellite to a higher altitude is not prohibitive. However, the cost to increase the orbits of 240 satellites will drive the total system cost.

The satellite's five year lifetime requirement demands that the initial altitude be sufficient to compensate for orbit decay. The requirement for satellite altitude differentials of less than 50 km was derived from the antenna coverage requirement of the MSSP antenna. When the altitude separation between satellites becomes too wide, the minimum range for communication becomes very large. When satellites are launched, altitude differentials will exist due to errors, design offsets for dispersion, or replacement of failed satellites. The initial altitude must be such that after five years, the spread of satellites due to differentials of orbit decay does not exceed 50 km and eliminate valuable resources. During this analysis calculations will be made for a seven year satellite orbit lifetime. The lifetime requirement of the physical spacecraft is five years.

4.1.1.1 EXOATMOSPHERIC TEMPERATURE AND ORBIT DECAY

The purpose of this analysis was to determine the initial average circular orbit altitude and relative altitude spacing between a pair of satellites placed into circular Earth orbit beginning in 1991. Variable solar activity during a seven year mission and its affect on orbit decay rates were also considered.

The dynamics of orbital decay are necessarily related to the dynamics of atmospheric density, which can be represented in the daily observation of the 10.7 cm solar flux (F10.7) and geomagnetic indices (Ap). The prediction of daily variations in F10.7 can not be accomplished with any acceptable level of confidence, but a running 13 month average of these values has shown some historical significance. Long range prediction of other solar disk activity for the purposes of orbit lifetime studies rely heavily on statistical prediction techniques based on these historical trends. Long range information is available from Marshall Space Flight Center's (MSFC) Atmospheric Sciences Division providing a smoothed 13 month prediction of the F10.7 solar activity, and Ap geomagnetic indices. This information is periodically updated, and the information contained in the MSFC report of April 14, 1986 was used as the most recent data for this analysis.

Figure 4-1 shows the calculated average exoatmospheric temperature generated from the above information. This curve represents the diurnal (day/night) average of temperature fluctuations as well as the influence of the geomagnetic index. The method of exoatmospheric temperature calculation follows Jacchia[1] (1977) and illustrates a temperature maximum (cycle 22) taking place in May of 1992, and

decreasing steadily through 1998. The empirically computed percentiles of 2.3, 50, and 97.7 percent exoatmospheric temperature are extrapolated based on the past 20 solar cycles. In addition, the estimates of exoatmospheric temperature are based on an assumed mean sun cycle period of 11 years. The plus/minus two standard deviation of sun cycle period range from 9 to 13 years producing a stretching or compression of the temperature profile of Figure 4-1.

To perform a strict prediction of the orbital decay rate requires the inclusion of a time-dependent relationship between the solar activity, exoatmospheric temperature, and atmospheric density. However within the scope of this study, three "constant" exoatmospheric temperatures (1400, 1100, and 900 deg K) were assumed throughout the mission life. Values were predicted from Figure 4-1 and maximum temperatures obtained during the May 1992 maximum for each of the three confidence intervals. MSFC recommends the use of the 97.7 percent confidence interval temperature (1400 deg K) for the calculation of orbit lifetime assuming that the actual lifetime of an orbiting satellite will meet or exceed the calculated lifetime in 97.7 percent of all cases. On the other hand, by noting the approximate linear decay of the three exoatmospheric temperatures shown in Figure 4-1, it may be assumed that the 50 percent temperature of 1100 deg K represents the approximate average of the worst case exoatmospheric temperature of 1400 deg K over the period of investigation, and can be considered as representative of the most likely median temperature over a mission life of seven years. The higher/lower temperatures are shown throughout the analysis to indicate the sensitivities and impact of solar flux on the analysis.

Figure 4-2 illustrates the orbital decay rates following the method of King-Hele[2] for the three statistical exoatmospheric temperatures mentioned above. Density and scale height values were taken from Jacchia (1977) for each of the altitude constants descriptive of each decay curve. The range in initial altitudes diverges as time from reentry and temperature increase accounting for the increased density present at the higher altitudes. These orbit lifetime curves are plotted again in Figure 4-3 using semilog coordinates. The dashed lines relate to a A/M of 0.1 ($M^2/kg$), and a Cd of 2.2 The orbit lifetime values above - 7 years should not be believed due to the lack of long range solar activity data predictions to support this period.

Figures 4-4 through 4-12 illustrate the average circular altitudes and relative separation decay rates of a pair of satellites placed on orbit in 1991 for the three "constant" suns mentioned above. Three circular orbit separation distances of 10, 30, and 50 km are assumed for each "constant sun" between the two satellites at their initial orbit placement. The appropriate initial average altitude would be selected based on an assumed maximum allowable spacecraft to spacecraft altitude separation, which is a function of inter-satellite antenna pointing requirements, as well as satellite replacement driven primarily by the

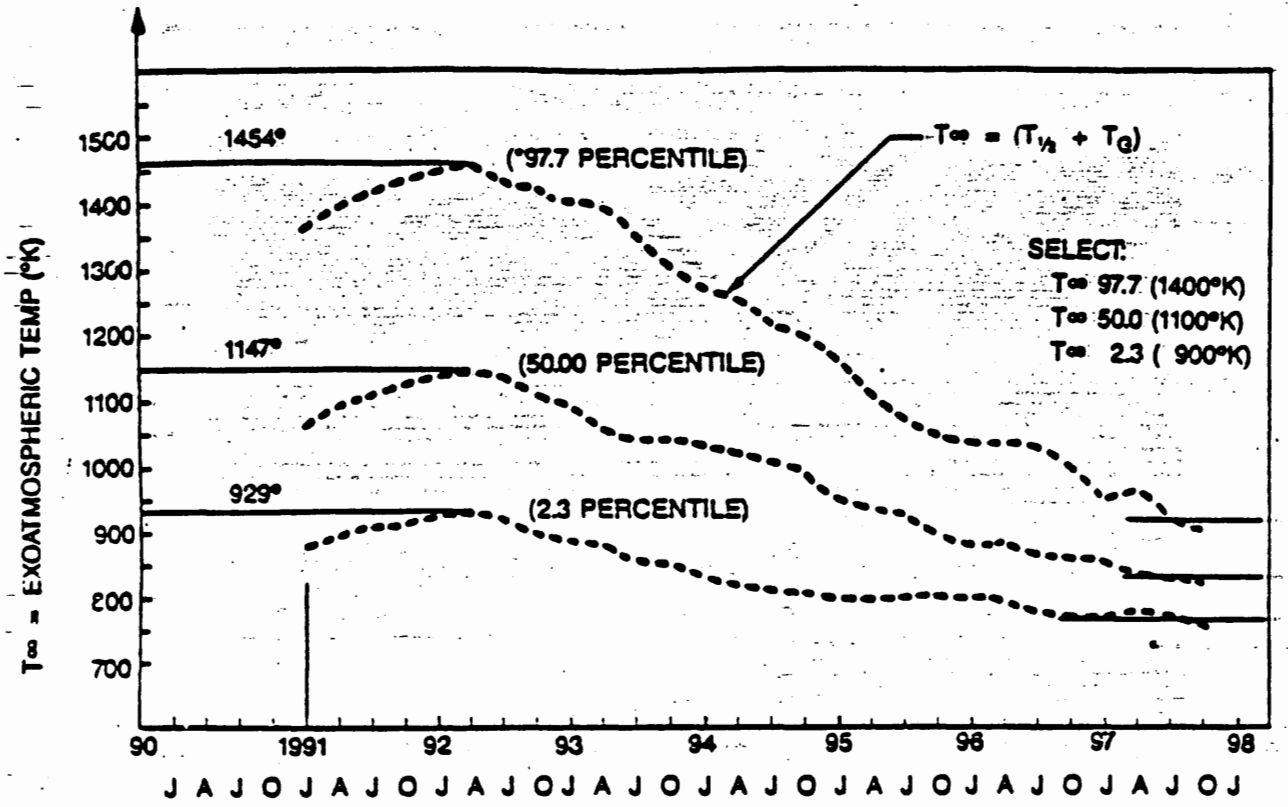


Figure 4-1 Calculated Exoatmospheric Temperature

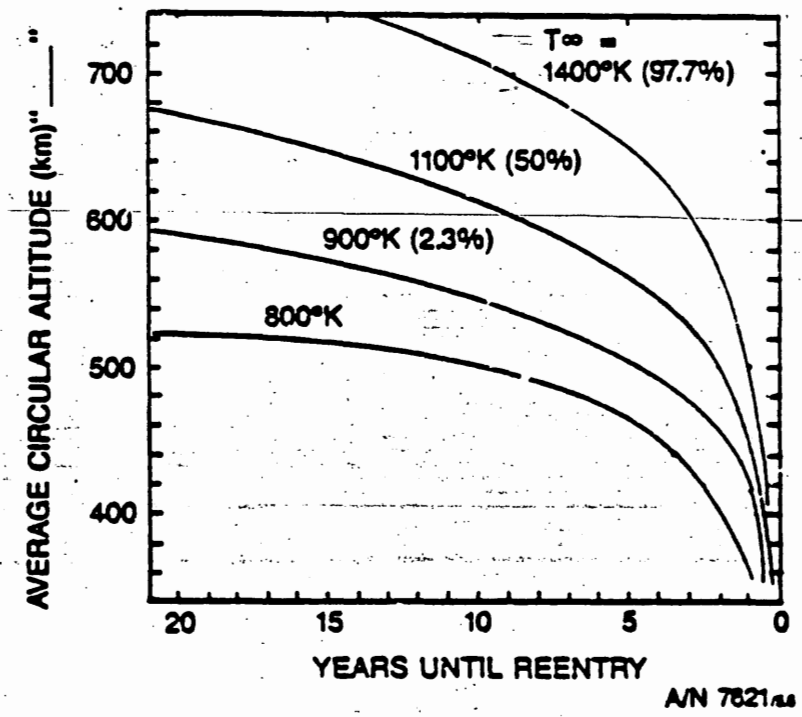


Figure 4-2 Orbital Decay (King-Heley)

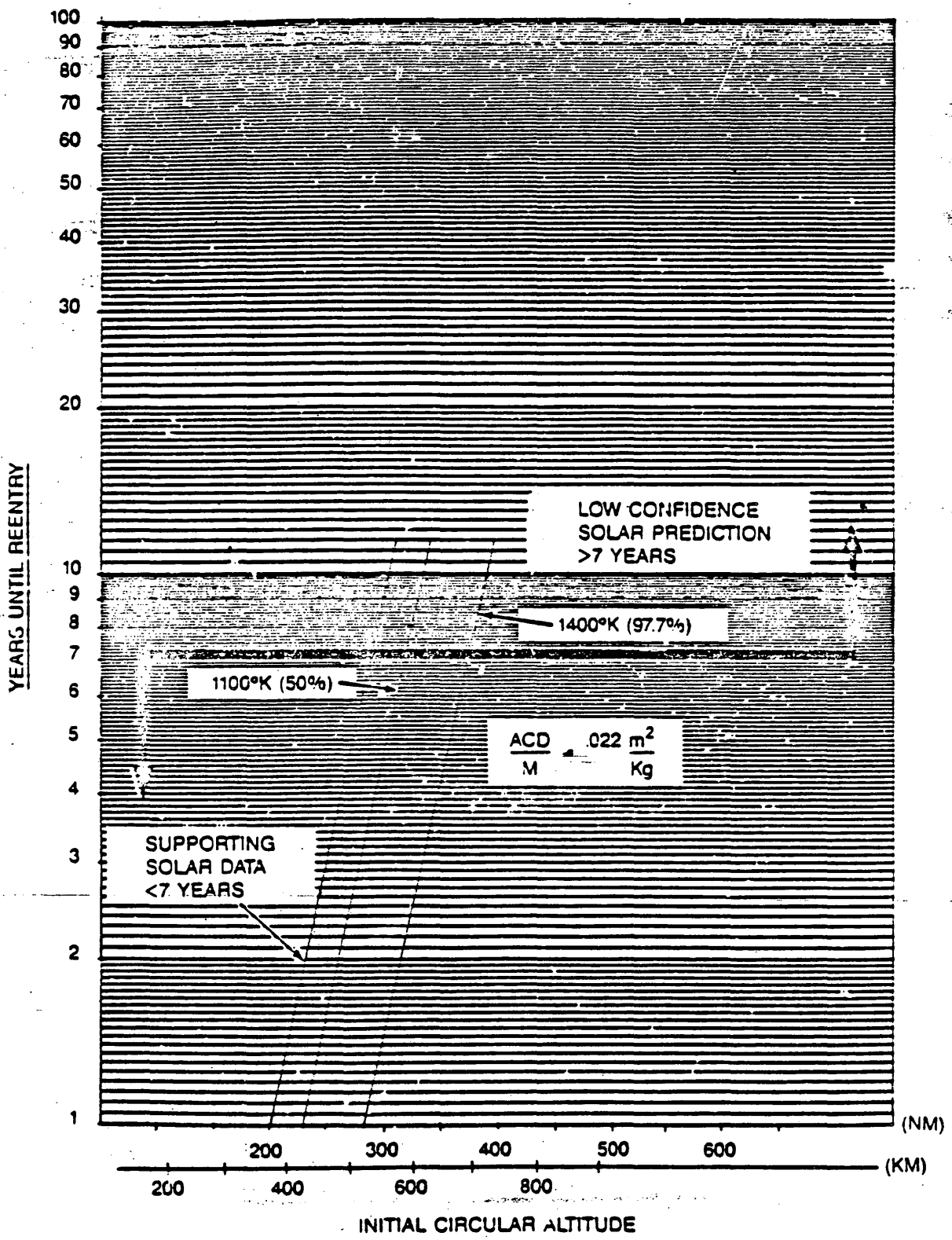


Figure 4-3 Orbit Lifetime

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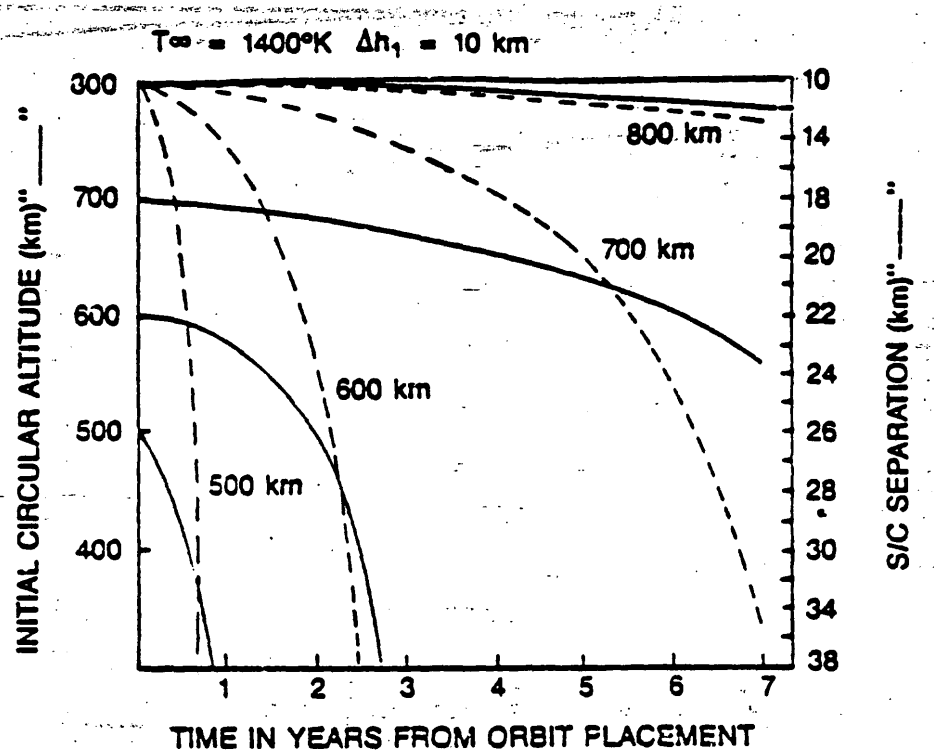



Figure 4-4 Average Altitude and Separation $\Delta h_1$ = 10 km $T_\infty$ = 1400 deg k

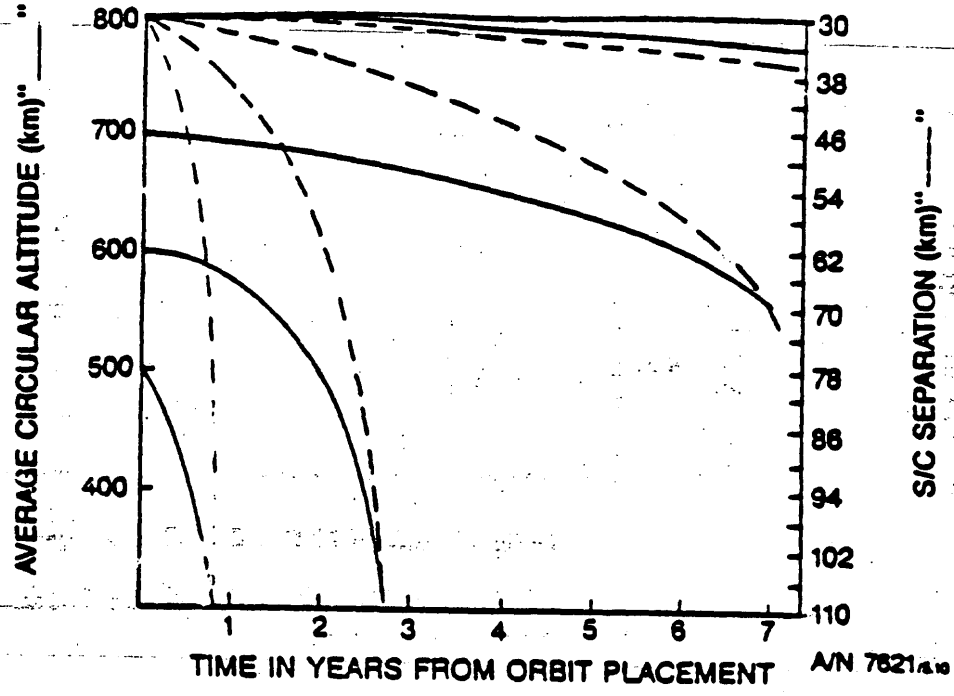



Figure 4-5 Average Altitude and Separation $\Delta h_1$ = 30 km $T_\infty$ = 1400 deg k

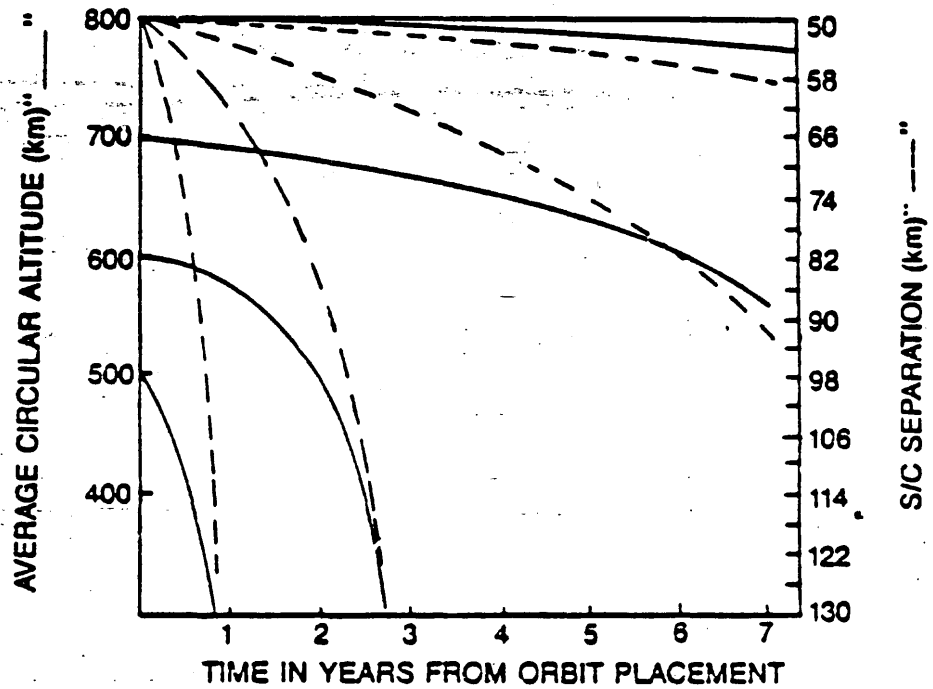



Figure 4-6 Average Altitude and Separation $\Delta h_1$ = 50 km $T_\infty$ = 1400 deg k

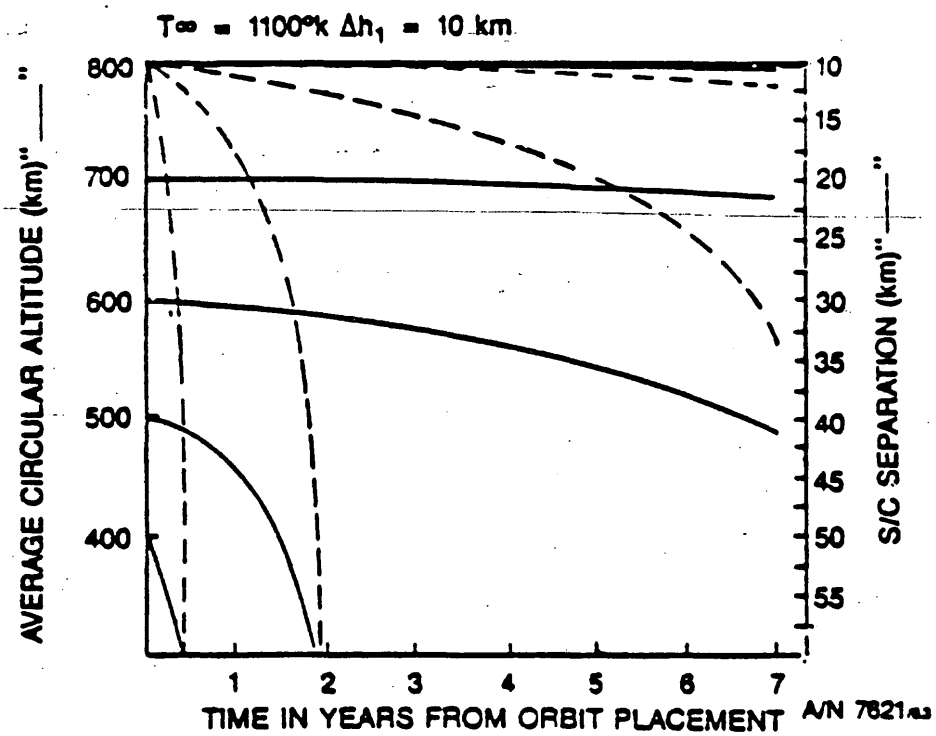



Figure 4-7 Average Altitude and Separation

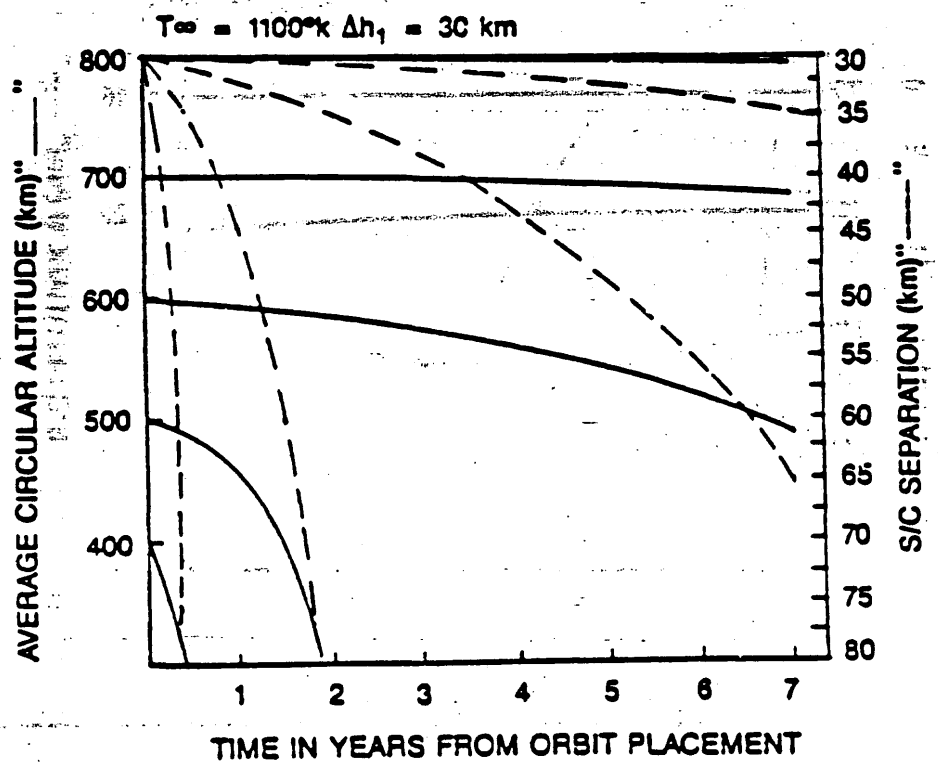



Figure 4-8 Average Altitude and Separation

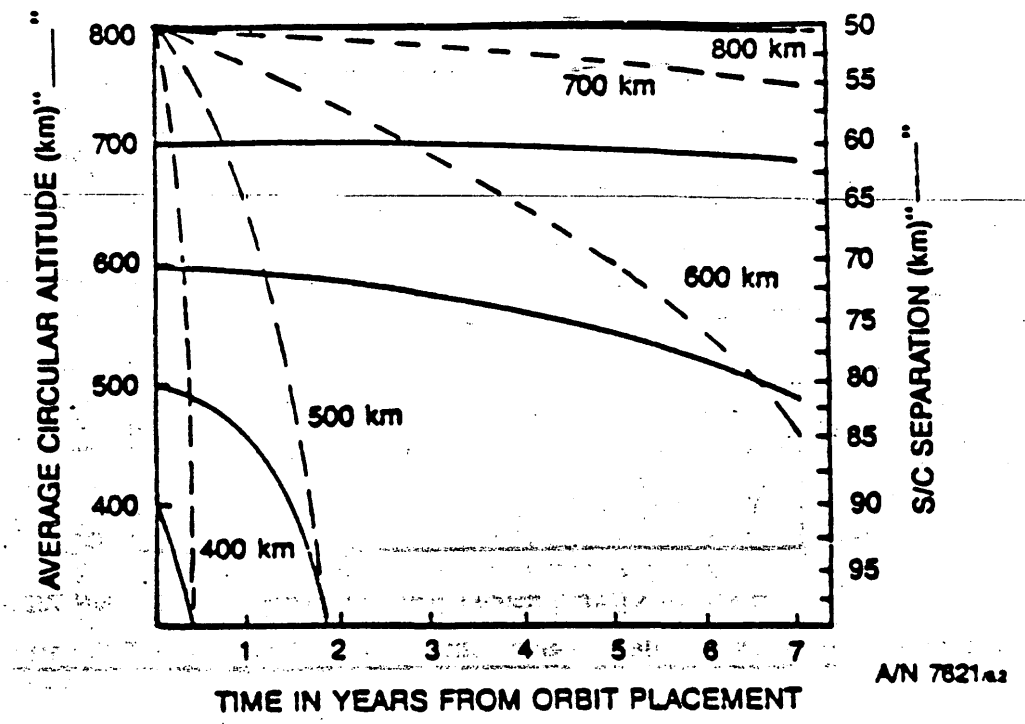



Figure 4-9 Average Altitude and Separation $\Delta h_1$ = 50 km T∞ = 1100 deg k

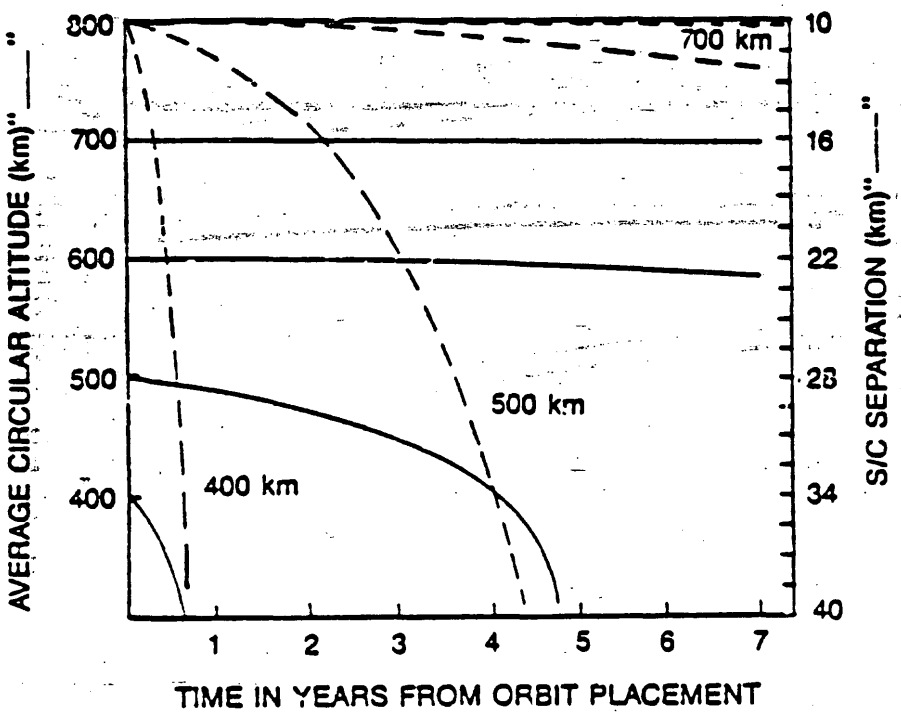



Figure 4-10 Average Altitude and Separation $\Delta h_1$ = 10 km $T_\infty$ = 900 deg k

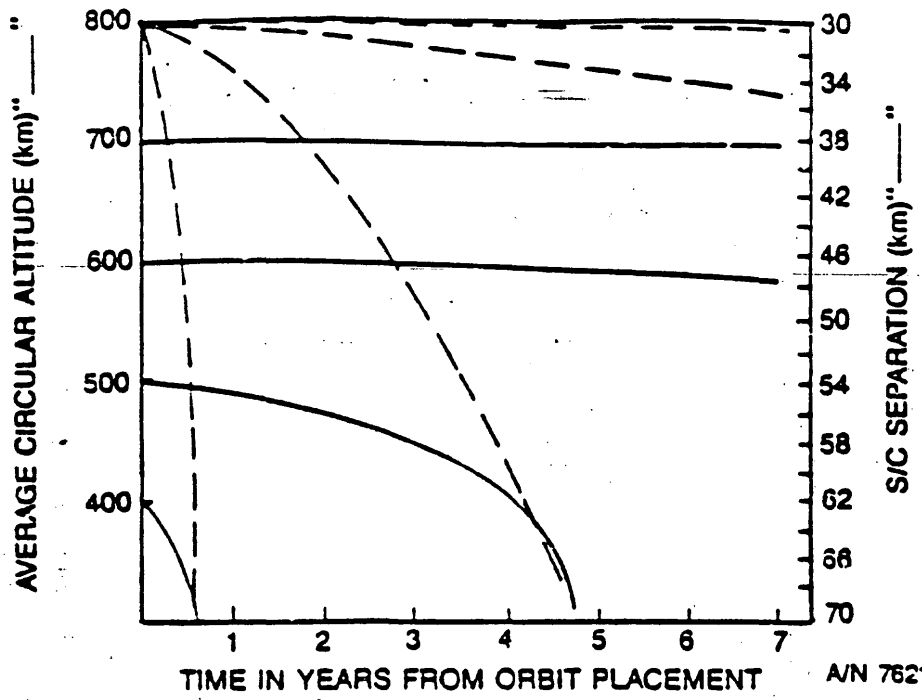



Figure 4-11 Average Altitude and Separation $\Delta h_1$ = 30 km $T_\infty$ = 900 deg k

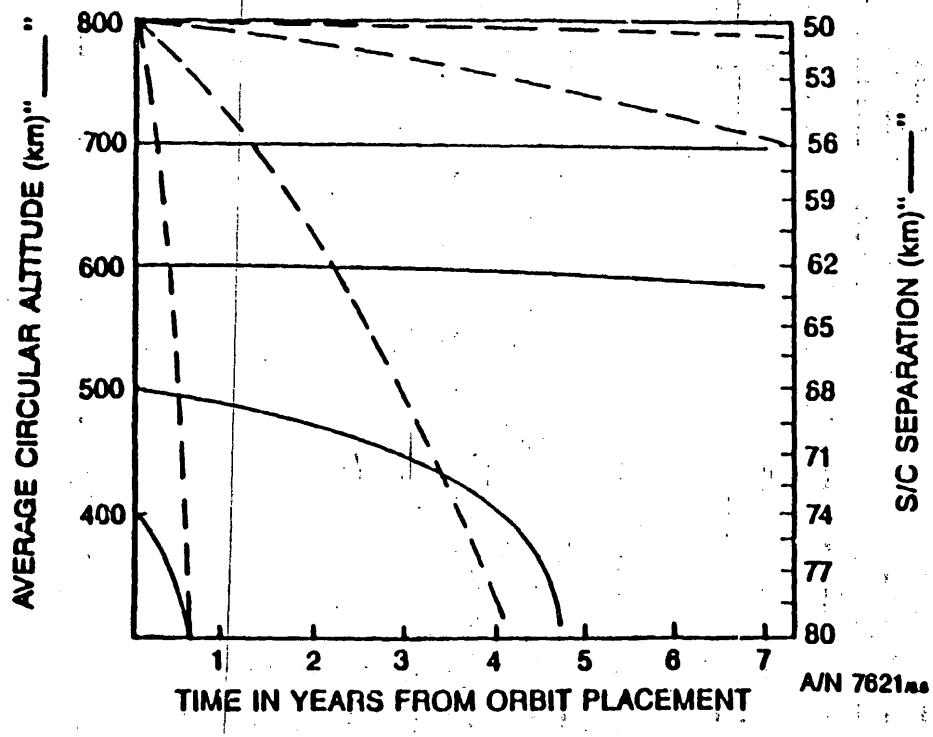



Figure 4-12 Average Altitude and Separation $\Delta h_1$ = 50 km $T_\infty$ = 900 deg k

desired mission lifetime. The maximum allowable inter-satellite separation and initial average circular altitude will be derived based on future analysis, and are unknown at this time. Presuming that these values were known, the orbit selection would follow directly from Figures 4-4 through 4-12. For example, looking at the "hot sun" shown in Figure 4-4 and choosing a mission lifetime requirement of seven years and a maximum allowable separation of 30 km would produce an initial average circular altitude of greater than 700 km.

It can be concluded that with the constant exoatmospheric temperature approach taken here, the most probable orbital decay rates are represented by the 1100 deg K temperature profile for the solar activity period of 1991 to 1998. From a viewpoint of a median analysis, the 1100 deg K sun produces a probable (50 percent) initial starting altitude of between 625 km and 675 km circular altitude for a probable initial inter-satellite altitude difference of 30 km. Since the predictions of orbital decay are so dependent upon exoatmospheric temperatures which are not easily or accurately predicted, this analysis will have to be updated later in the MSS program. However, this analysis is sufficient to establish the initial satellite altitude for the MSSP preliminary design.

### 4.1.2 - SPACECRAFT ORBITAL DENSITY

Once the spacecraft altitude had been selected, the number of orbit planes and the number of satellites per orbit must be analyzed to assure that global communication coverage is possible. This cursory analysis was performed to assure that the suggested distributions would meet the performance requirements and that unforeseen system requirements did not exist. The initial spacecraft distribution assumed from a previous study (ESL-TM1632) called for 90 spacecraft at 27.5 deg inclination, 90 at 57.5 deg, and 60 spacecraft at 90 deg. This distribution and another suggested by BASD were examined for their impact of the communication ranges. The conclusion derived from this analysis was that unforeseen system problems did not exist, but that the communication performance could be enhanced with additional satellite planes at higher inclination angles.

#### 4.1.2.1 ORBITAL DENSITY ANALYSIS

The ideal distribution of spacecraft for total global coverage is given by a density is

$$f_u = \frac{240}{4\pi} = 19.1 \quad \text{SPACECRAFT/STERIDAN}$$

The distribution of spacecraft as a function of latitude ($\lambda$) is given by the area integral of the density function

$$F_u = \int_0^{\lambda} f_u 2\pi \cos\lambda d\lambda = 120 \sin\lambda \text{ SPACECRAFT}$$

The distribution is the total number of spacecraft between the equator and north latitude $\lambda$. The ideal uniform density and distribution are shown in Figure 4-13. The ideal distribution curve decreases in slope with increasing latitude because of the curvature of the Earth. Each increment in latitude near the pole results in a smaller area than a corresponding increment near the equator.

The spacecraft density is fundamental to determining the angular spacing ($\theta\mu$) between spacecraft as measured from the center of the Earth and the intersatellite range (Ru) to the nearest neighbor. It is mathematically impossible to place an arbitrary number of spacecraft around the globe with uniform spacing between them. However, it is possible to place 4, 6, 8, 12, or 20 spacecraft uniformly about a sphere by placing them at the centers of the faces of one of the five regular polygons. A uniform distribution of 240 spacecraft can be approximated by taking a hexagonal pattern, which is the most dense packing pattern that can be placed on a plane. This pattern gives 6 nearest neighbors to each spacecraft, each at an angular separation $\theta\mu$ and 60 deg in azimuth from each other. The angular separation for 240 spacecraft is

$$\theta\mu = \frac{180/\pi}{\sqrt{tu \sin 60}} = 14.1 \text{ deg}$$

The intersatellite range is

$$R_u = 2\ (RE + H) \sin \frac{\theta\mu}{2} = 1724 \text{ km}$$

where the Earth's radius is RE = 6378 km and the spacecraft altitude is H=650 km.

The uniform density described above is an unobtainable ideal for two reasons. First, it is mathematically impossible for the distribution to exist, except in an average sense, since no 240 sided polyhedron exists. Second, even if the ideal density could exist at one point in time, orbit dynamics will constantly be altering the ranges between spacecraft, thereby altering the local densities.

The primary objective in selection of orbits for MSSP is to provide a constellation that averages the time and space dependent variations in

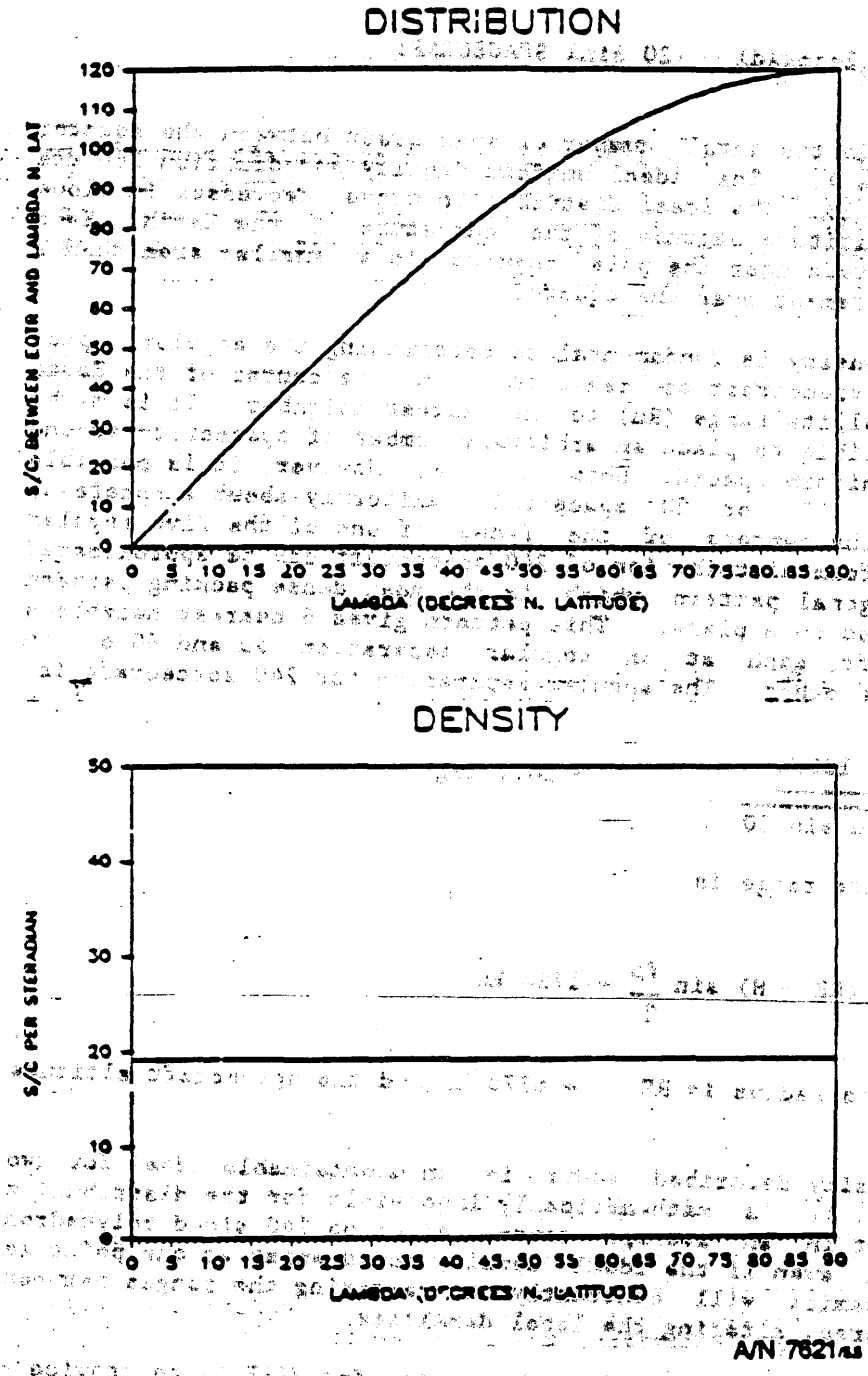



Figure 4-13 Ideal Uniform Density and Distribution of Spacecraft

density. To achieve this objective, the dependence of density on inclination is examined alone. In order to do this, it is assumed that N spacecraft are all at an inclination i and that their longitudes of ascending nodes and true anomalies are uniformly distributed. This results in a spherical shell of spacecraft between north and south latitudes equal to the inclination angle. Continuous analysis is justified in the limit if N is very large or if the time average over many orbital revolutions is considered. The density (f) as a function of latitude ($\lambda$) for this inclinations is

$$f = \frac{Ni}{2\pi^2 \sqrt{\sin^2 i - \sin^2 \lambda}} \qquad \lambda \leq i$$

$$f = o \qquad \lambda > i$$

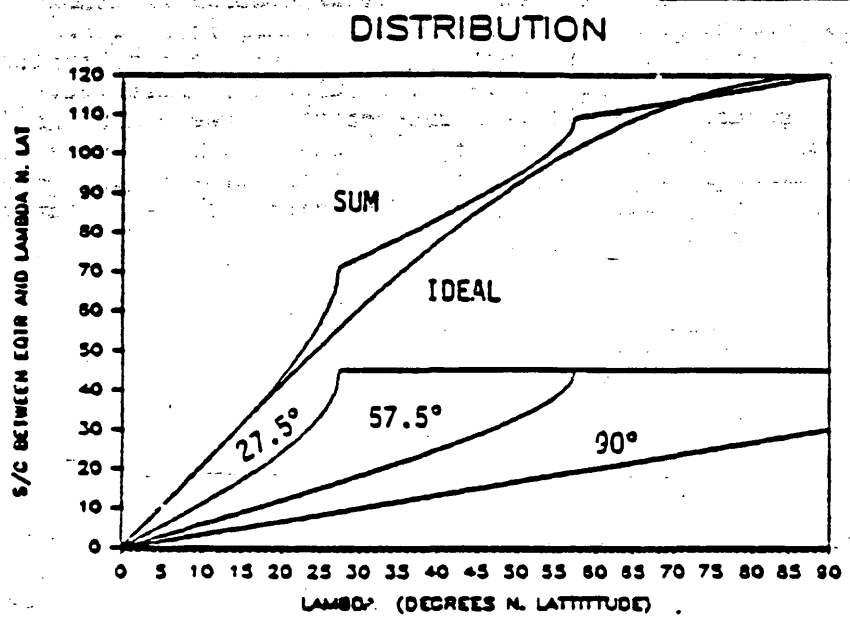
The corresponding distribution is found by integration of the density functions

$$F = \frac{Ni}{\pi} \sin^{-1} \left( \frac{\sin \lambda}{\sin i} \right) \qquad \lambda \leq i$$

$$F = \frac{Ni}{2} \qquad \lambda > i$$

These density and distribution functions are plotted in Figure 4-14 for the deployment inclinations that were proposed in ESL-TM1632. This deployment consisted of 90 spacecraft at 27.5, 90 at 57.5, and 60 at 90. The density at low latitudes is very close to the ideal uniform density and then rises to a spike as the latitude approaches 27.5. This spike reflects the fact that the orbits of spacecraft at the same inclination bunch together at north and south latitudes near their inclination angle and spread apart near the equator. The density function drops sharply to 9.9 spacecraft/steradian as the latitude increases past 27.5, because the 90 spacecraft associated with the lowest inclination orbit are all to the South and can no longer contribute to the total density. The density then slowly increases to a second spike at 57.5 before falling to a global minimum of 5.6 spacecraft/steradian. The intersatellite range just north of 57.5 latitude is of interest. Looking to the south, the density is high where the 57.5 inclination orbits bunch together, whereas looking to the east or west, the density is only 5.7. The angular spacing between spacecraft in the east-west direction is

| i | 27.5° | 57.5° | 90° |
|---|---|---|---|
| N1 | 90 | 90 | 60 |



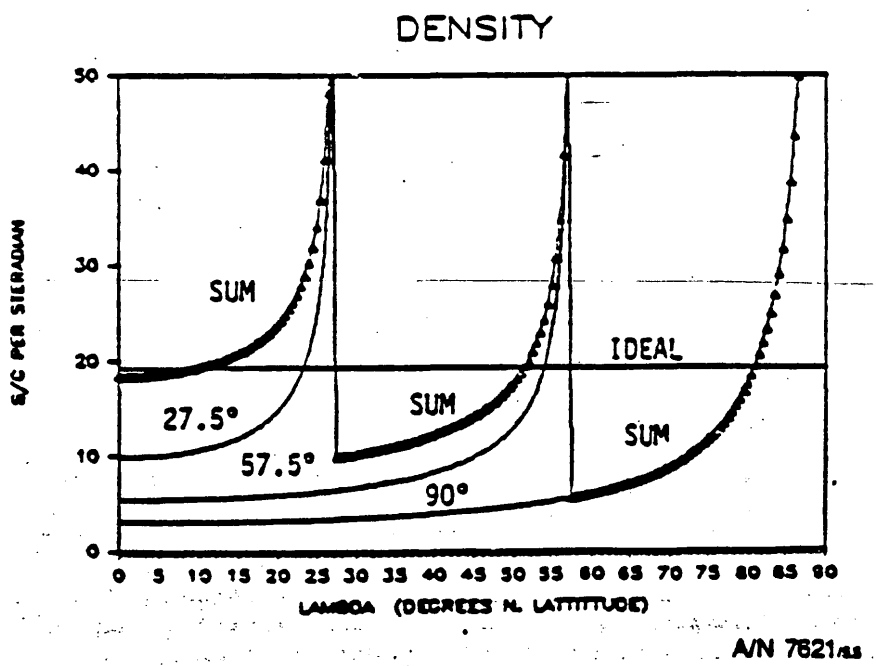




Figure 4-14 ESL-TM1632 Distribution and Density of Spacecraft

$$\theta = \sqrt{f_u/5.7} \qquad \theta_\mu = 25.9 \text{ deg}$$

which corresponds to an intersatellite range of

$$R = 2(R_e + H) \sin \frac{\theta}{2} = 3148$$

for a 650 km altitude orbital shell. Finally, looking to the north, the density is below $f_\mu$ until latitude 80.8 N. The latitude difference is 23.3, which corresponds to a range of 2843 km (1535) nm at an altitude of 650 km (351 nm).

An alternate set of inclination angles was investigated in an attempt to alleviate the range requirement at high latitudes. The alternate constellation uses four inclinations rather than three, and consists of integer multiples of 24 spacecraft in each plane for comparability with a proposed carrier vehicle. The inclinations and number of spacecraft are as follows:

| | | | | |
|---|---|---|---|---|
| i | 28.5 | 57 | 70 | 90 |
| Ni | 48 | 72 | 72 | 48 |

The density and distribution functions are shown in Figure 4-15. A comparison of Figures 4-14 and 4-15 reveals that the alternative deployment strategy results in a distribution curve that more closely follows the ideal sine function, and that the minimum density is 7.1 for the alternate as opposed to 5.7 for the original deployment scenario. Now consider the east-west range from just north of 70 N latitude. The angular spacing between spacecraft in the east-west direction is

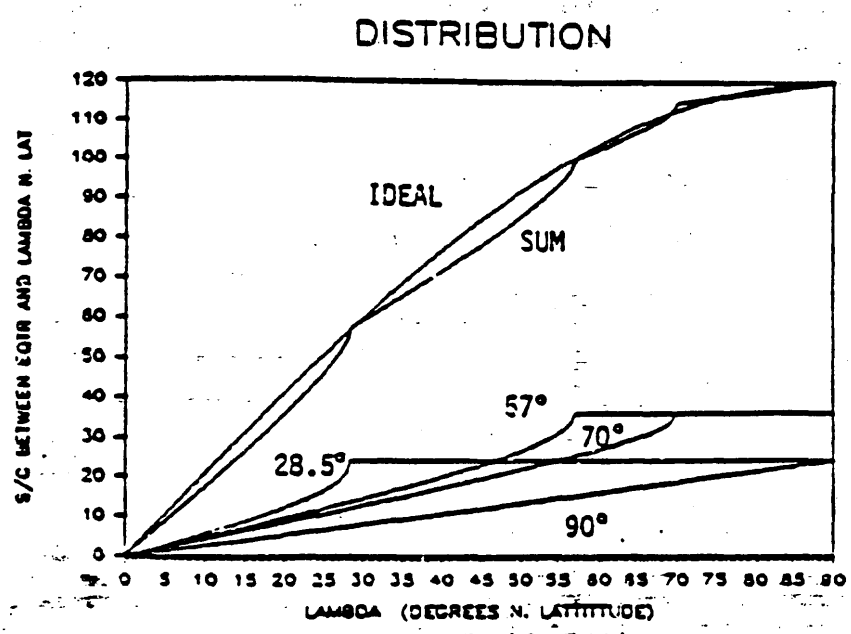
$$\theta = \sqrt{f_u/7.1} \qquad \theta_u = 23.1$$

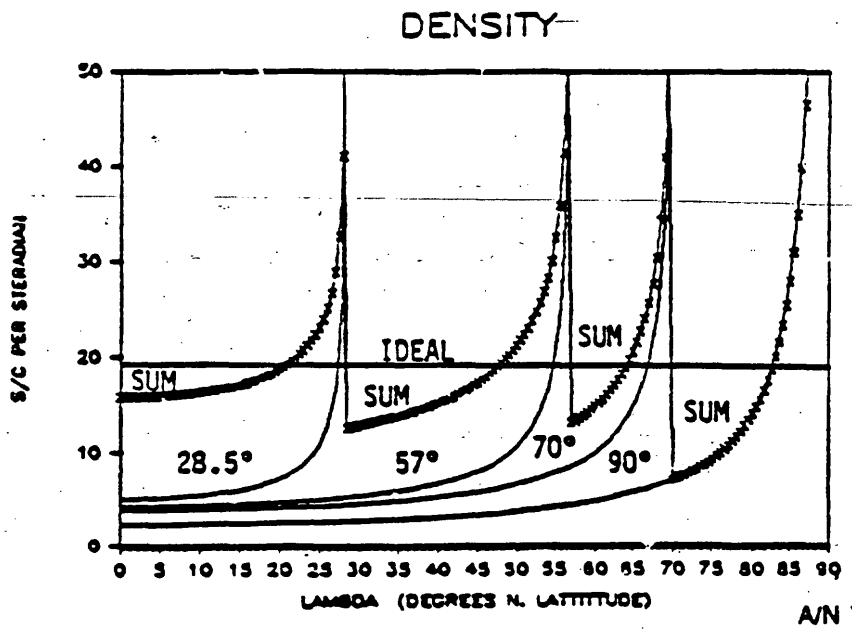
For a 650 km altitude spacecraft, this corresponds to an intersatellite range of

$$R = 2(R_E + H) \sin \frac{\theta}{2} = 2813 \text{ km}$$

which is not much smaller than the East-West range at 57 + latitude for the original case. However, considerable improvement has been achieved

| i | 28.5° | 57° | 70° | 90° |
|---|---|---|---|---|
| N1 | 48 | 72 | 72 | 48 |




A/N 7621/ss

Figure 4-15 Alternate Orbit Set Distribution and Density of Spacecraft

for a Northward looking antenna. The density equals f when the latitude reaches 82.7 N. The latitude difference is 12.7, which corresponds to a range of 1553 Km (838 nm) at an altitude of 650 Km (351 nm). Consequently North-South communication links at high latitudes are facilitated using the alternate set of inclinations. East-West links are not substantially different and will tend to follow the lines of spacecraft bunching at latitudes near the inclination angle of a particular shell.

The conclusion to this analysis are:

a) Both of the possible deployments of satellites would provide adequate communication performance with no major system or satellite concerns.

b) The satellite density will tend to increase at the latitudes equal to the orbit inclination angles.

c) More uniform distribution may be obtained with more satellites at higher inclination angles.

d) An approach has been presented which may be applied to obtain a first cut evaluation of spacecraft density of various deployment schemes or to evaluate deployments which might accentuate communication at particular latitudes.

e) The final resolution will depend upon the cost to obtain uniform density or what level of uniform density is cost effective.

### 4.1.3 SPACECRAFT ORBITAL DISPERSION

The deployment of many satellites with varying orbit planes and varying inclination angles is necessary in order to obtain a global communication system. However, the cost of dispersing 240 satellites into many orbits will be prohibitive for a "low cost " system, if the satellite orbits are obtained by separate launches. One approach for the dispersion of the MSSP satellites is to launch the satellites with reduced cost in a group from either an Expendable Launch Vehicle (ELV), or shuttle and then deploy them at varying altitudes or inclination angles.

The variation of the altitude of satellites will cause the satellites to separate from each other. The altitude difference with different radial velocities will cause the satellites to separate within the orbit plane. The external torques from the earth will cause the orbit planes of satellites at different altitudes to regress at different rates. Inclination angle differences will also cause different regression rates. The latter phenomenon is called nodal regression.

The spacecraft velocity with altitude is:

$$V = Re \times sqr[\ g\ /\ (Re+h)\ ]$$

where: Re = Earth radius
h = satellite altitude
g = acceleration of gravity

The spacecraft orbit regression rate vs. altitude and inclination is:

$$\Omega = -10 \times [\ Re\ /\ (Re+h)\ ]^{(7/2)} \times \cos(\ Ai\ )$$

where: Ai = satellite inclination angle

Figure 4-16 shows the satellite velocity variation vs. altitude. Figure 4-17 shows how satellites would separate in the orbit plane due to velocity differentials (altitude) reference to an altitude of 300 km. Figures 4-18 and 4-21 show the change in nodal regression and the separation in orbit node for satellites at varying altitudes.

Dispersion of satellites can only be obtained at a cost. The above equations and Figures 4-17 through 4-21 state that the cost could be time rather than the dollars to pay for the many launch vehicles. Since the launching of 240 satellite via ELV would require an extended period of time, the trade of time vs. dollars is not a straight forward exchange. Especially if the launch vehicles are not immediately available for trade considerations.

Figures 4-22 through 4-26 show how 60 satellites, launched 20 at a time every 3 months with 0.15 deg of inclination separation between satellites of a launch group, would disperse after a 12 month period. The white areas in the figures show the areas on the ground which have a satellite within view (10 deg ground user elevation angle). After a 12 month period, the only shaded areas are at the earth poles. The illustrations in Figures 4-22 through 4-26 are not meant to show how 60 satellites will provide a global communication, because many other parameters must also be analyzed. However, these figures show how three launches of the small MSSP satellites could provide wide satellite dispersion.

The dispersion of Figures 4-22 through 4-26 was obtained through inclination angle variation. If the satellite nominal inclination angle is low, then altitude variations would be the best method of obtaining dispersion. For high nominal inclination angle orbits, the use of inclination angle variation would be best.

This analysis does not have a final conclusion because it depends upon the final orbits selected for the communication system capability. This analysis was performed to show that an alternate, possibly lower

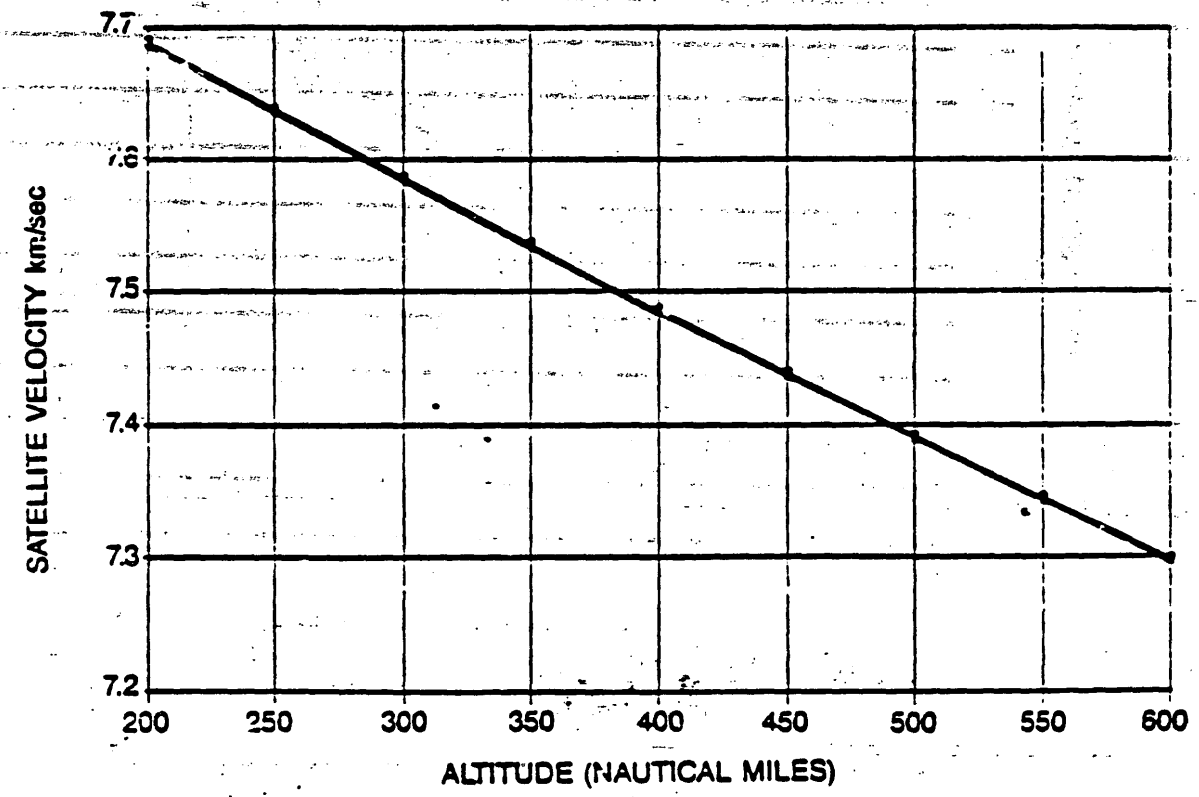



Figure 4-16 Satellite Velocity Versus Altitude

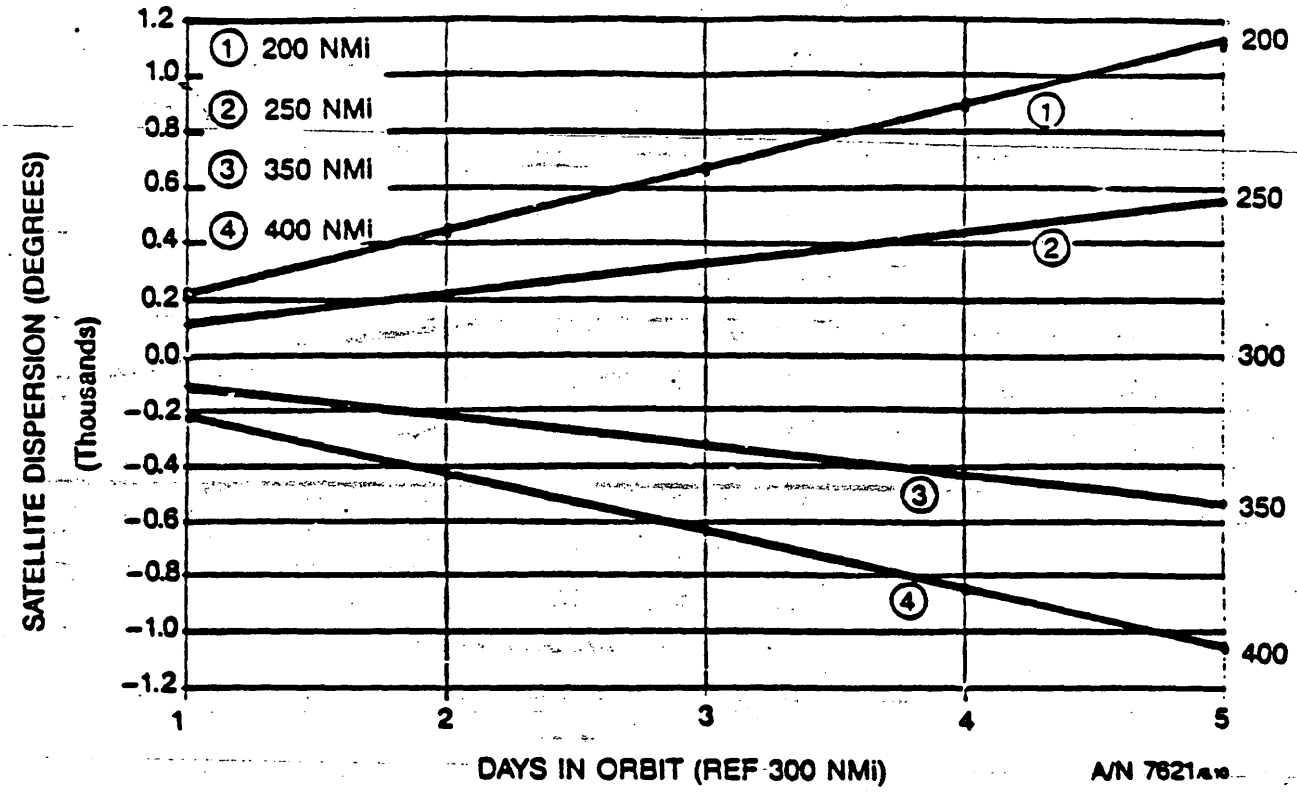



Figure 4-17 Satellite In-Orbit Dispersion

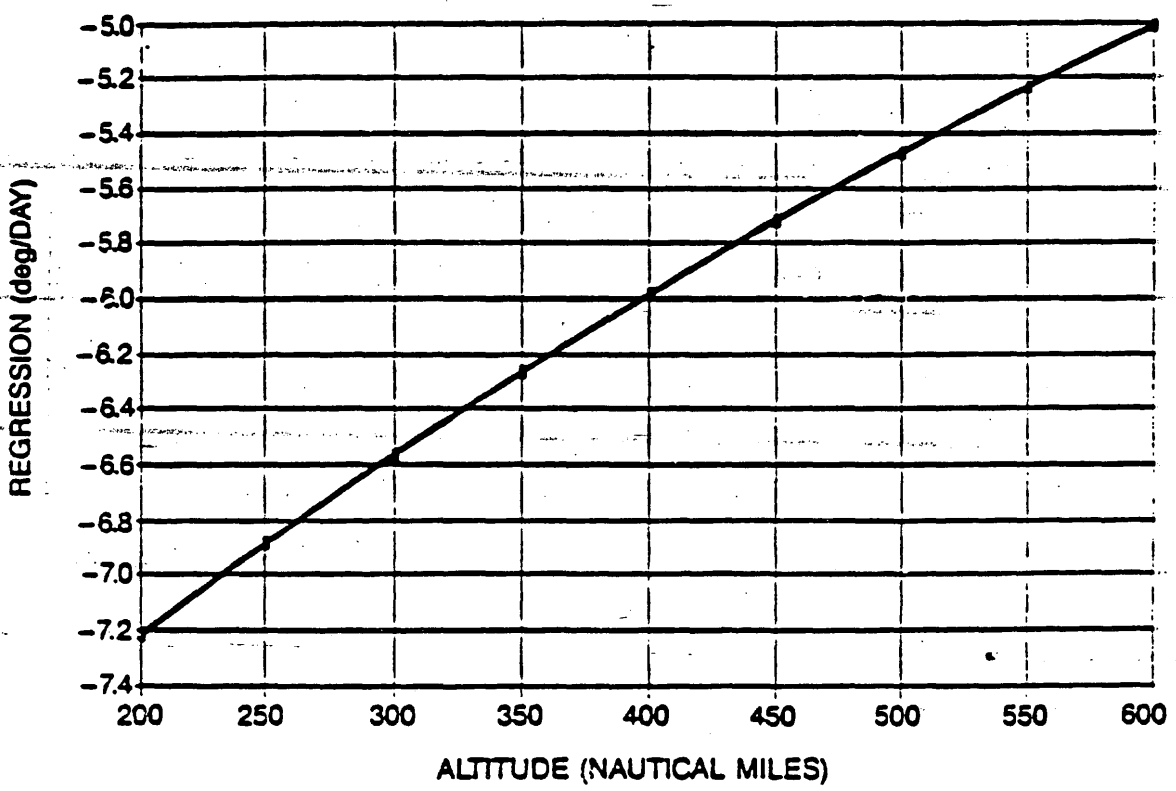



Figure 4-18 Nodal Regression Versus Altitude (Ai = 28.5)

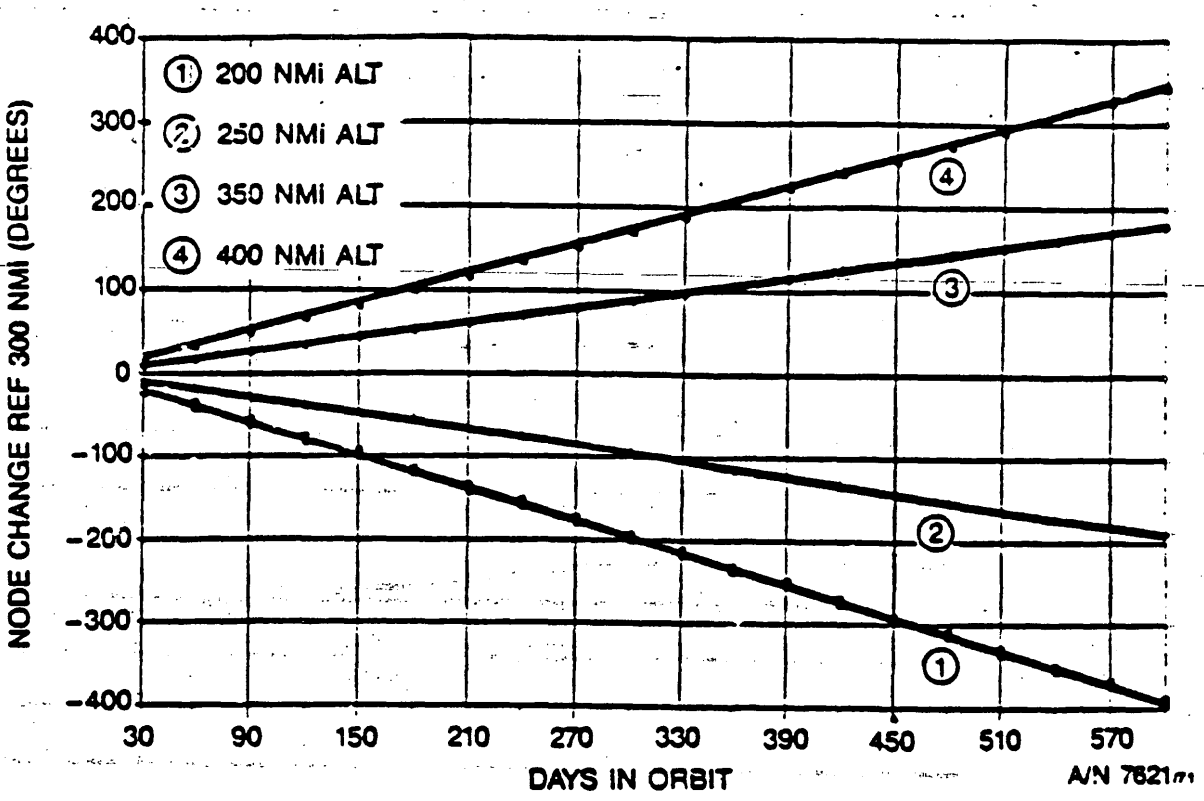



Figure 4-19 Nodal Regression Separation of Satellites Versus Altitude (Ai = 28.5)

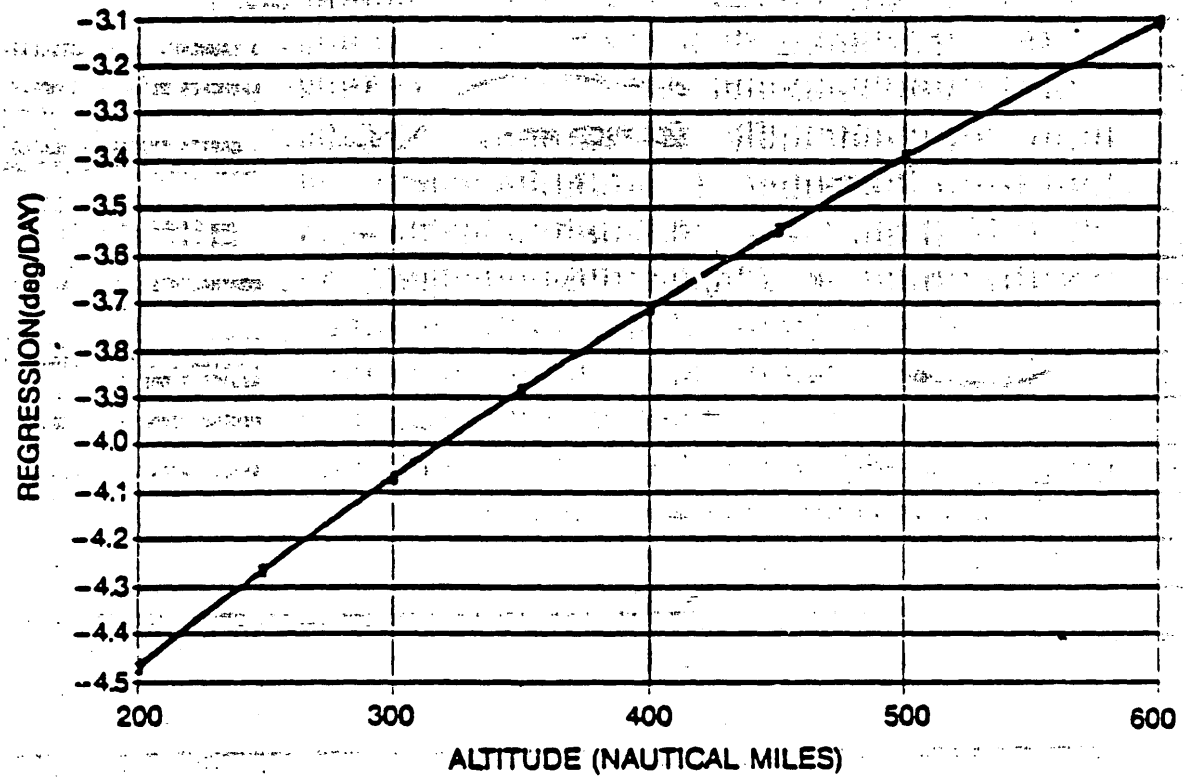



Figure 4-20 Nodal Regression Versus Altitude (AI = 57)

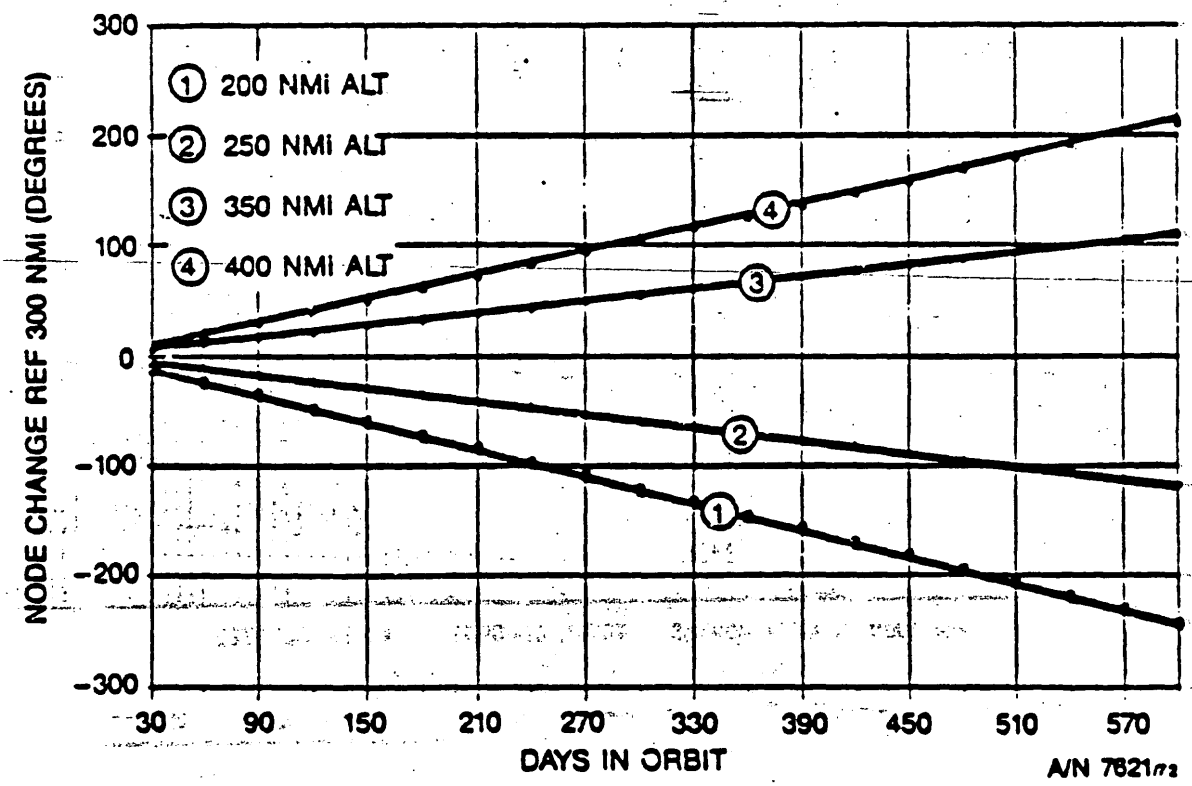
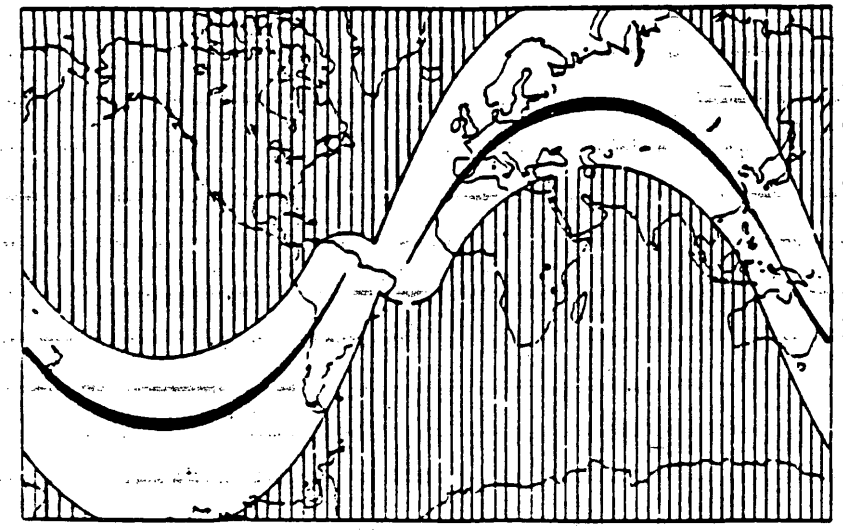



Figure 4-21 Nodal Regression Separation of Satellites Versus Altitude (AI = 57)



BASIC MSSP NETWORK DISPERSION SHOWN ASSUMES:

- 3 LAUNCHES OF 20 SATELLITES EACH
- LAUNCHES OCCUR AT 3 MONTH INTERVALS (EG., 0 MO., 3 MO., and 6MO.).
- 3 ORBITS INITIALLY PLACED WITH RT. ASCENSION OF THE ASCENDING NODE AT:

  1st LAUNCH = 30°W
  2nd LAUNCH = 90°E
  3rd LAUNCH = 150°W

- NOMINAL MSSP ORBIT AT 57° AND 675KM x 675KM.
- SATELLITE DISPERSION VIA 0.15° INCLINATION SEPARATION BETWEEN ADJACENT ORBITS.
- MINIMUM GROUND STATION ELEVATION ANGLE = 10°.
- SHADED AREAS: NO GROUND COVERAGE

1st LAUNCH AT 0 MONTHS. TOTAL ON-ORBIT = 20 SATELLITES

Figure 4-22 Satellite Dispersion/Ground Coverage First Launch to 0 Months

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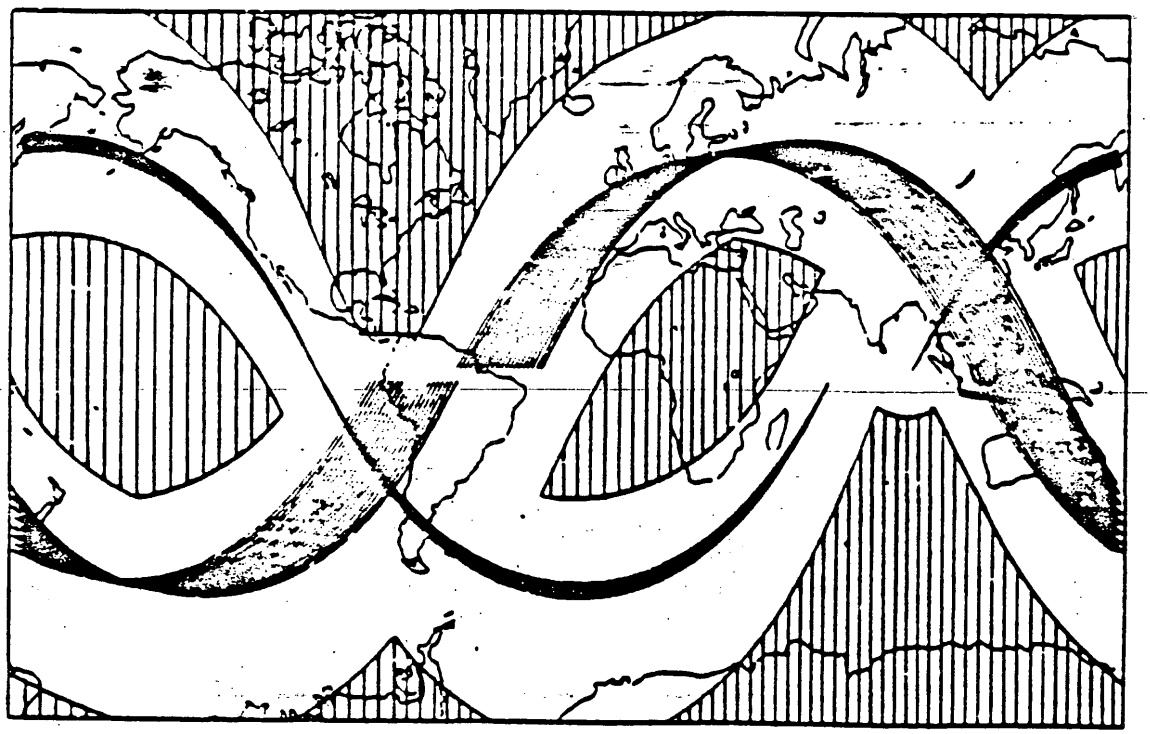
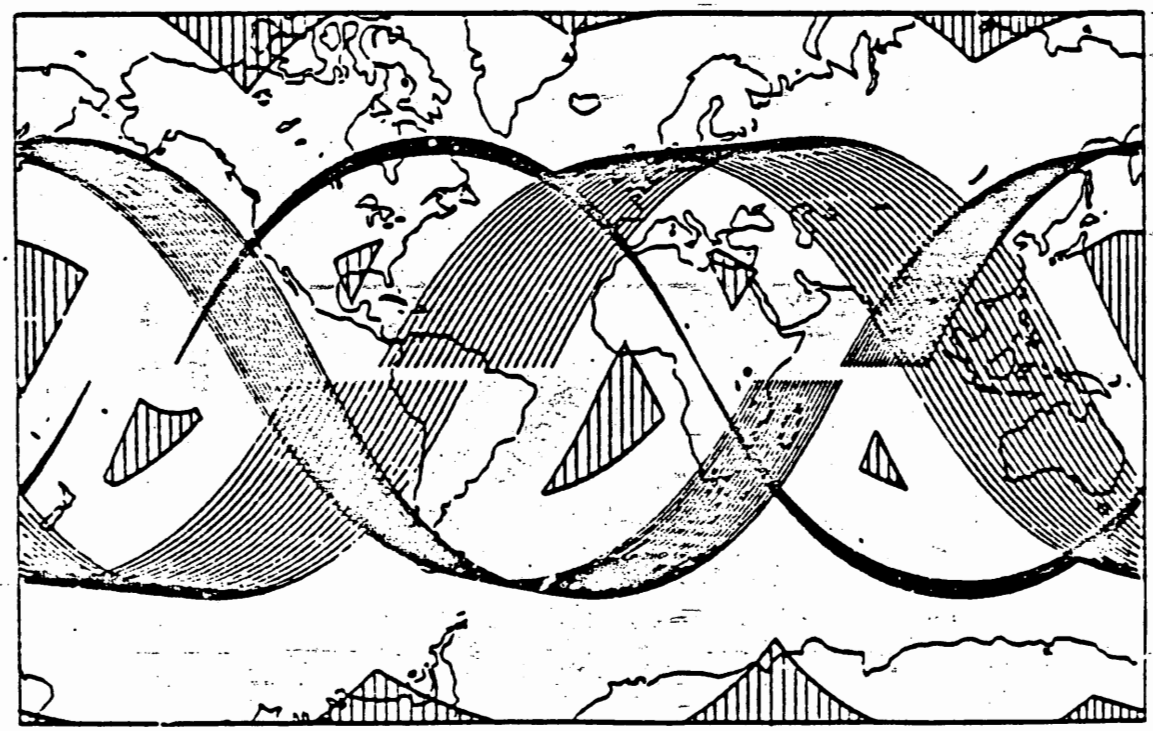


2nd LAUNCH AT 3 MONTHS. TOTAL ON-ORBIT = 40 SATELLITES

Shaded Area: No Ground Coverage

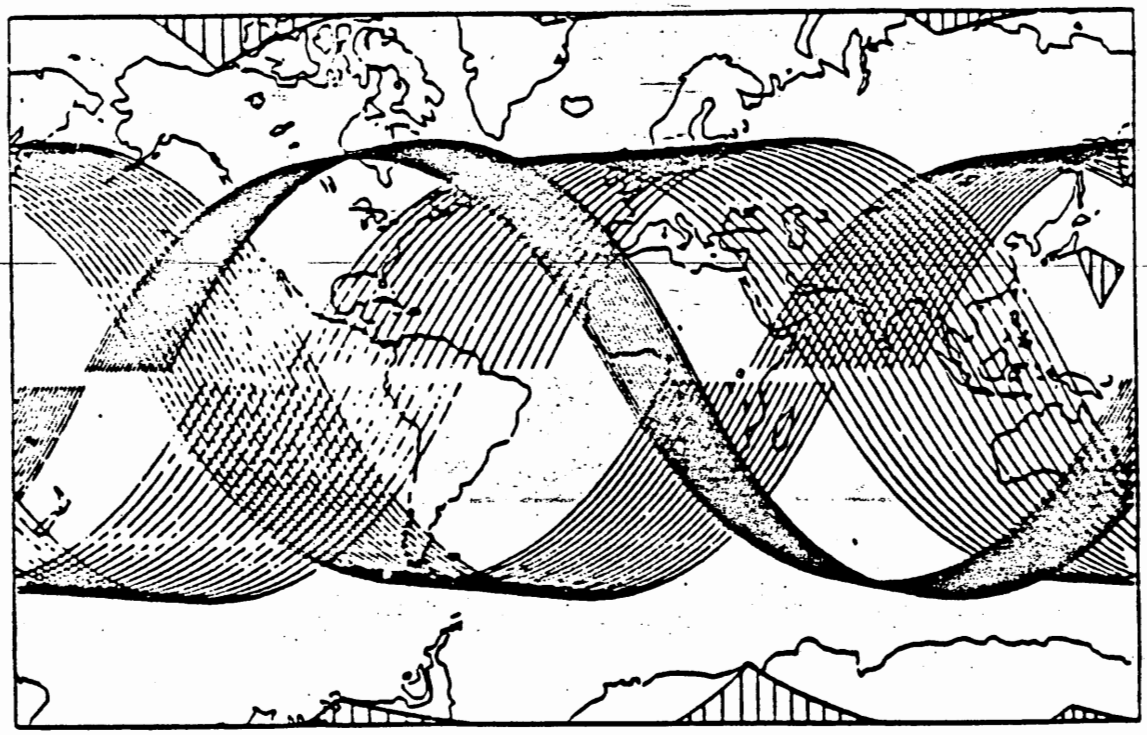
Figure 4-23 Satellite Dispersion/Ground Coverage Second Launch 3 Months



3rd LAUNCH AT 6 MONTHS. TOTAL ON-ORBIT = 60 SATELLITES

Shaded Area: No Ground Coverage

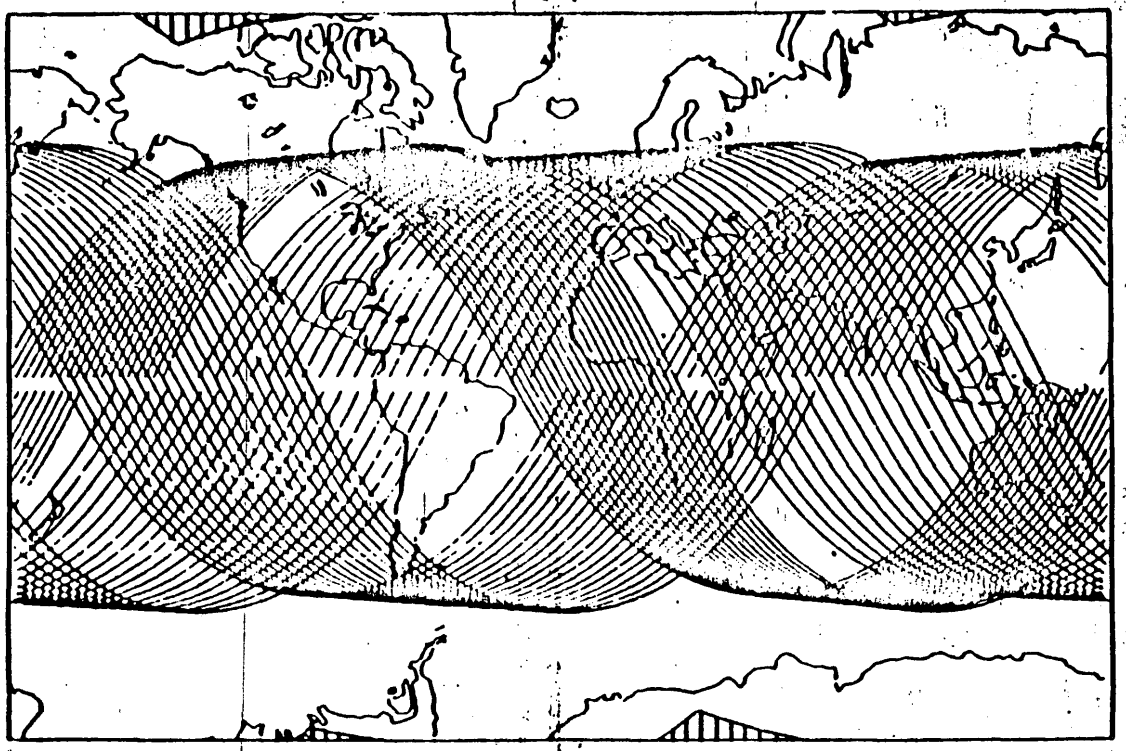
Figure 4-24 Satellite Dispersion/Ground Coverage Third Launch 6 Months



MSSF NETWORK DISPERSION AT 9 MONTHS. TOTAL ON-ORBIT = 60 SATELLITES.

Shaded Area: No Ground Coverage

Figure 4-25 Satellite Dispersion/Ground Coverage 9 Months A/N 7621



MSSP NETWORK DISPERSION AT 12 MONTHS. TOTAL ON-ORBIT = 60 SATELLITES

Shaded Area: No Ground Coverage A/N 7621

Figure 4-26 Satellite Dispersion/Ground Coverage 12 Months

cost, method exists for the deployment of 240 satellites the MSSP. The final resolution of the trade between separate satellite launch and this method of dispersion will depend upon the final analysis of the desired satellite orbits for communication performance and launch costs. However, as will be seen in section 4.1.5, the cost of separate launches will be so prohibitive that a group method of launch is essential.

### 4.1.4 SPACECRAFT LAUNCH STRATEGIES

Section 4.1.3 discussed the technique of satellite nodal regression and how this approach could perform the satellite dispersion necessary to deploy 240 MSSP satellites for global communication. Once this technique was established, a means of implementing the final satellite orbit deployment was investigated. The orbit selection and delivery analysis for the establishment of an MSSP satellite network highlighted the need for a carrier vehicle to perform the final orbit transfer and initial satellite orbit placement maneuvers. This orbit selection criteria yields a preferred method of establishing the orbits by imparting a constant separation of orbit inclination between satellite orbits to produce a dispersion in longitude of ascending node and true anomaly position of each satellite. Using the methods described in this section, the total amount of delta (i) (inclination) required to place N satellites on orbit is a function of the relative nodal regression rates to achieve a select orbit dispersion in a given period of time.

For this analysis, the maximum delta v capability of a reusable carrier vehicle is sized for 20 satellites injected initially into a 300 km X 300 km, 57 dag parking orbit with a desired Earth centered lunge angle of 3.25 deg in approximately 6 weeks. This yields an equatorial cross range separation of 400 km between satellites having a minimum "lapping period" (the time required to produce a 360 deg relative true anomaly rotation between the first and final satellite placed in orbit) of six weeks.

#### 4.1.4.1 MSSP LAUNCH STRATEGY ANALYSIS

The relative movement of a MSSP system of circular satellite orbits can to the first order be described by a set of particular perturbed orbit ephemerides for each of the satellite orbits. The orbit shape and orientation can be defined using classical Keplerian orbital elements. For a circular orbit, these are

1) orbital radius ($r_c$)
2) orbit eccentricity (e)

} orbit shape

3) orbit inclination (i)

} orbit orientation

4) longitude of ascending node (Ω) )

5) true anomaly (M) : satellite position in orbit plane

A variation of circular orbital elements applied to the MSSP system of satellites will produce advantages and disadvantages affecting global coverage and network utilization as well as mission lifetime and program cost. The major network and program influences produced from the perturbed orbital elements are briefly summarized below:

| CHANGE IN CIRCULAR ORBIT ELEMENT | ADVANTAGES | DISADVANTAGES |
|---|---|---|
| 1) Increase eccentricity | Increased nodal regression rate greater global frequency | Aprisidal rotation causes large variation in inter-satellite altitude requiring increased elevation beamwidth |
| 2) Decrease/Increase Inclination | Nodal regression increases/decreases as inclination decrease/increases | Requires multiple launches or substantial on-board propulsive capability (eg SRM) |
| 3) Decrease Orbital altitude | Nodal regression decreases | Multiple launches or on-board propulsion is required. Broader elevation beamwidth required for some altitudes. Shorter mission life expected. |
| 4) Longitude of ascending node | Greater global frequency | Multiple launches or substantial on-board propulsion. |

The production of an eccentric orbit (Item 1) quickly expands the MSSP antenna beamwidth elevation requirements without producing large nodal regression rates. An eccentricity of 0.0028 with a perigee altitude of 675 km will produce a 20 km apogee altitude increase, while yielding only a 6 x 10-5 deg/day relative nodal rate with respect to a circular orbit of 675 km altitude. Similarly, a change in the circular orbit altitude (Item 2) between adjacent satellites (although producing larger rates than an eccentric orbit) will adversely increase the satellite elevation beamwidth with relatively small nodal regression rates. Here a 20 km circular altitude difference will yield only a 0.037 deg/-day nodal rate relative to the reference 675 km circular orbit. This will be seen as insufficient in meeting MSSP mission requirements in a

later discussion. Item 4 can be summarily dismissed because of the staggering launch costs associated in discrete satellite placement for the 240 elements. Therefore, this study will address the establishment of the MSSP network, using a perturbation method involving the separation of satellite orbits via small inclination changes.

4.1.4.2 MSSP DERIVED LAUNCH REQUIREMENTS

Deviations in discrete orbital elements may be produced via the initial orbit placement of a satellite or with perturbation forces applied to the satellites while in orbit (i.e. thrusters rockets, etc). The appropriate selection of required sat. lite orbits will depend on mission objectives and requirements set forth for the MSSP. The following general requirements are assumed for this analysis.

a) two orbit inclinations of 57 and 80
b) network consisting of 240 satellites
c) STS baselined as stage "0"
d) mission orbit lifetime of seven years (satellite lifetime baseline is 5 years.)
e) minimize launch and production cost/satellite
f) maximize global coverage
g) maximize the effective use of network satellites

Item d) drives the selection of the appropriate orbital altitude to achieve the mission life and has been discussed in section 4.1.1. As a result of this analysis, the preliminary determination of the initial circular orbit altitude to achieve a 7 year mission is between 625 km and 675 km, which requires an orbit raising to be performed from the STS operational altitude of 300 km x 300 km. In keeping with item e), the satellites have not been designed to contain on-board propulsion capabilities, therefore an orbit transfer carrier seems appropriate. This carrier may be thought of as reusable - "smart" (i.e. three-axis, restartable engines) or expendable - "dumb" (spinner, SRMs), or a combination thereof. Preliminary packing layouts show a carrier capacity of 24 MSSP satellites for the reusable and 20 for the expendable carrier designs. This high density is necessary to limit the recurring launch costs to establish the network and, provides sufficient numbers of satellites per launch to establish a basic system in a reasonable period of time. At this capacity it would take approximately 10 to 12 launches to establish the network of 240 satellites. A launch rate of 3 to 4 launches per year produces a time-to-completion of the network of approximately 3 to 4 years. The element and program costs for each carrier approach have been given preliminary examination in Section 4.1.5, Space Launch Cost.

The length of time to completion of the network of 240 satellites has t a advantage of selecting the initial right ascension of the ascending node displacement with each launch. Ideally it can be shown that three polar launches with 120 deg separation in ascending node and with

randomly distributed true anomalies among each complement of coplanar satellites will produce equatorial cross ranges which preserve the east-west intersatellite link. However, this basic system is subject to mortality, hostile threat, "clustering" around the poles at the common intersection of the orbital planes, and requires active satellite station keeping to prevent "bunching" of the satellites-a situation created when adjacent satellites have slightly differing orbital periods and collect in similar true anomaly position. To prevent this and lessen the network degradation via threat and mortality, the satellites are positioned in orbits spread in longitude of ascending node, and true anomaly position. The required orbit perturbations to dispense each launch of the 20 to 24 satellites, as well as the orbit-raising energies to establish the initial circular orbit, are relegated to the carrier vehicle and will vary depending on the carrier design capabilities.

4.1.4.3 CARRIER VEHICLES

A) "DUMB" CARRIER

In general, the greater the orbit placement flexibility required of the MSSP, the greater the complexity of the carrier vehicle. A simple "dumb" (e.g. spinner) carrier may contain back to back solids located along the spin-axis of the vehicle. The carrier with its 20 satellites would maintain coarse attitude during the orbit raising via momentum biasing created during an initial spin-up maneuver created using small spin-up rockets. To maintain a low recurring cost, a command and telemetry link is eliminated and an on-board sequencer commands the firing of the perigee solid rocket motor (SRM) with a set time delay for the firing of the apogee circularization SRM. Once the delivery orbit is established, the command sequencer would then initiate the simultaneous release of all 20 satellites.

There are a number of undesirable results of this design. The carrier lacks attitude knowledge and control functions and is therefore incapable of autonomously establishing the correct inertial perigee burn direction and must rely on the shuttle to preselect the carrier burn attitude and minimize RMS deployment hand-off errors (a "frisbee" type of deployment with a more sophisticated cradle may help here). These errors at deployment would translate to apogee/perigee errors whereby the target final orbit altitude may be missed. In addition, a "dumb" carrier can not measure or correct for the three-sigma variations in total impulse of the perigee burn. The net result is the likely delivery onto an initially undesirable orbit; an orbit which may not guarantee mission lifetime and is overly eccentric to reduce network utilization. Finally, with the above scenario, the "dumb" carrier is not able to provide selective orientation or discrete timing for release of

each MSSP satellite necessary to produce efficient dispersion of satellite ascending nodes and true anomalies. Preliminary estimates show that the simultaneous release of the satellites from a spinning carrier will produce eccentric "bunching" of satellites reducing the use of the network and having the potential, at high release spin rates, to produce periodic intersatellite altitude differences which exceed the beamwidth capabilities of the MSSP antennas.

B) AUTONOMOUS CARRIER

An alternate approach involves using a smart carrier vehicle with a restartable liquid propulsion system. Attitude knowledge would be maintained with on-board three-axis capability following STS reference initialization and RMS deployment from the shuttle. The orbit raising and circularization would be performed using a continuous low-thrust maneuver incurring less than 2 percent delta-V penalty (good up to h<2000 km) when compared to the Hohmann impulsive transfer. The MSSP deployment sequence is accomplished using a satellite placement onto discrete orbits which are to be achieved by the carrier vehicle prior to satellite orbit placement.

Each successive satellite would be placed in an orbit separated from the adjacent orbit in inclination (to produce nodal dispersion) and eccentricity (to produce true anomaly separation). The dispersion in true anomaly would produce a "lapping" effect created by the slightly different orbital periods in which adjacent satellites would exhibit large lapping periods. The shortest period would exist between the first and last satellite deployed where the largest relative rate is created. To avoid a loss in resolution of two satellites when viewed from a ground site ( a minimum equatorial cross range of 400 km separation at an orbital altitude of 675 km is required) the relative nodal regression rates between the satellites with the shortest lapping period should be selected to produce the minimum required separation in the minimum lapping period.

For sizing the liquid propulsion tankage, the magnitude of the velocity maneuver to produce adjacent discrete orbits with the appropriate relative nodal regression rates should be kept as small as possible. Figures 4-27 and 4-28 show a preliminary analysis for a "smart" carrier vehicle at 57 deg and 80 deg inclinations carrying 24 satellites. Figures 4-29 and 4-30 demonstrate the effect of a reduction in the number of satellites if 20 are carried to orbit. The analysis shows that a maximum carrier delta-V of approximately 5.0 m/sec (4.9 m/sec normal to and 20 m/sec tangential to the velocity direction) between 24 successive MSSP satellite placements initially at the 57 deg inclined orbit will produce the minimum desired separation between the first and last satellite from a given carrier load in approximately a six week lapping period. The inclination difference

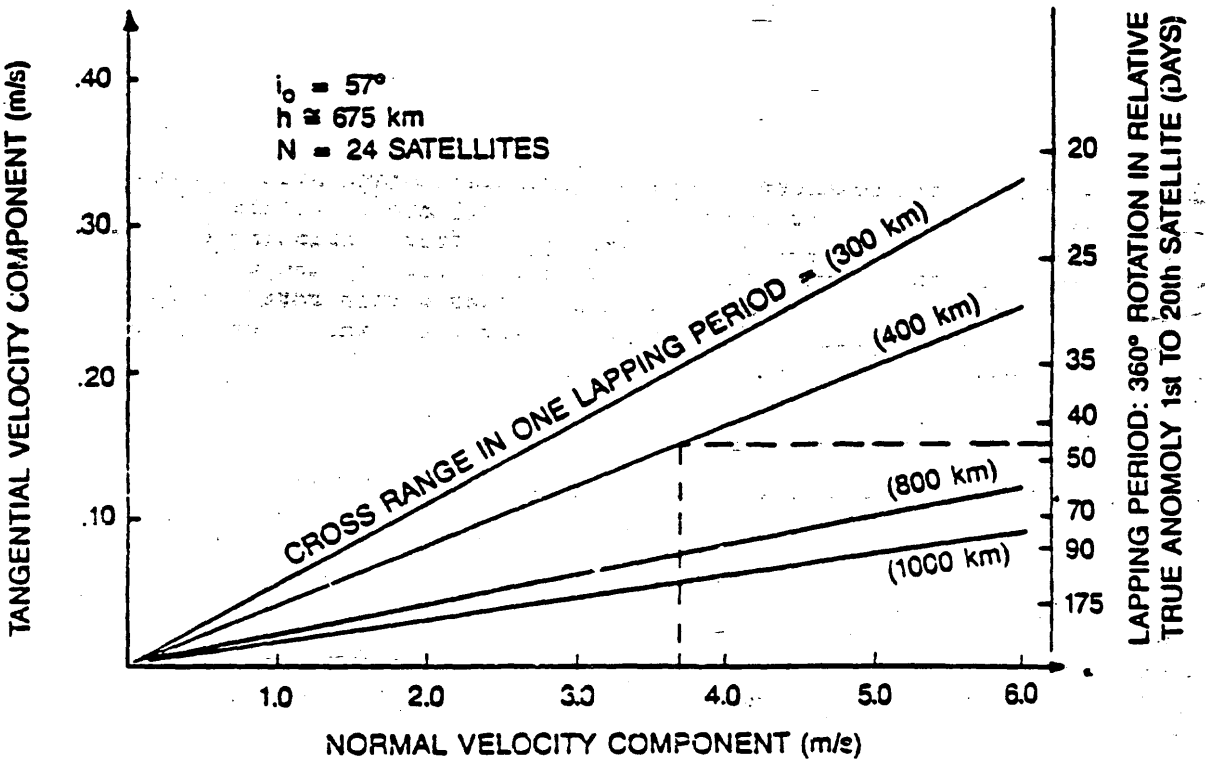



Figure 4-27 Minimum Time to First Satellite Lapping Versus Separation Velocity Components (Ai = 57° 24 Satellites)

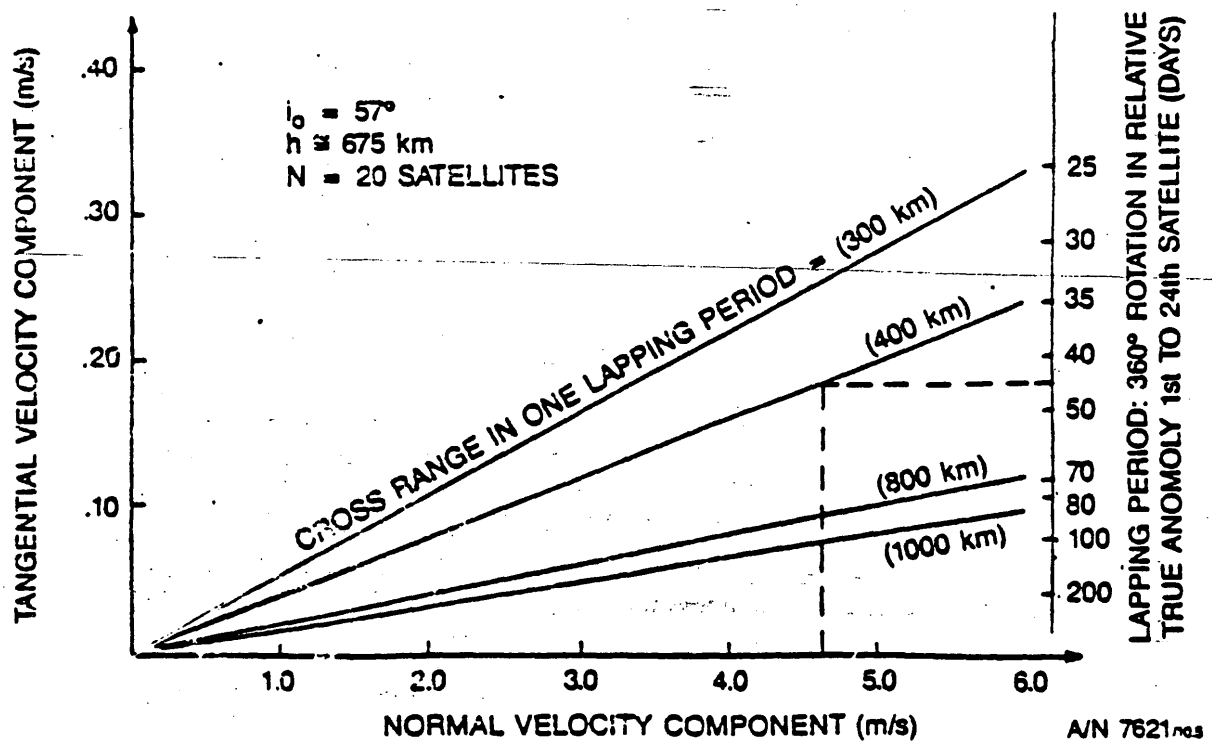



Figure 4-28 Minimum Time to First Satellite Lapping Versus Separation Velocity Components (Ai = 57° 20 Satellites)

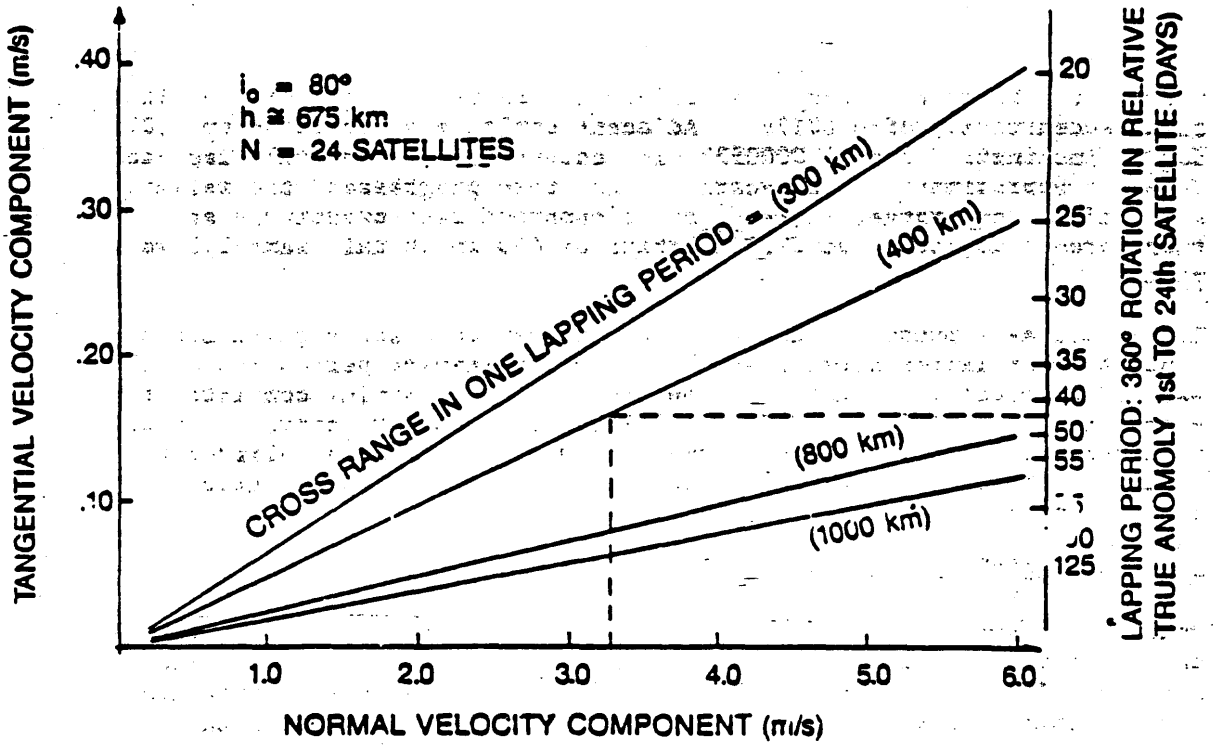



Figure 4-29 Minimum Time to First Satellite Lapping Versus Separation Velocity Components (Ai = 80° 24 Satellites)

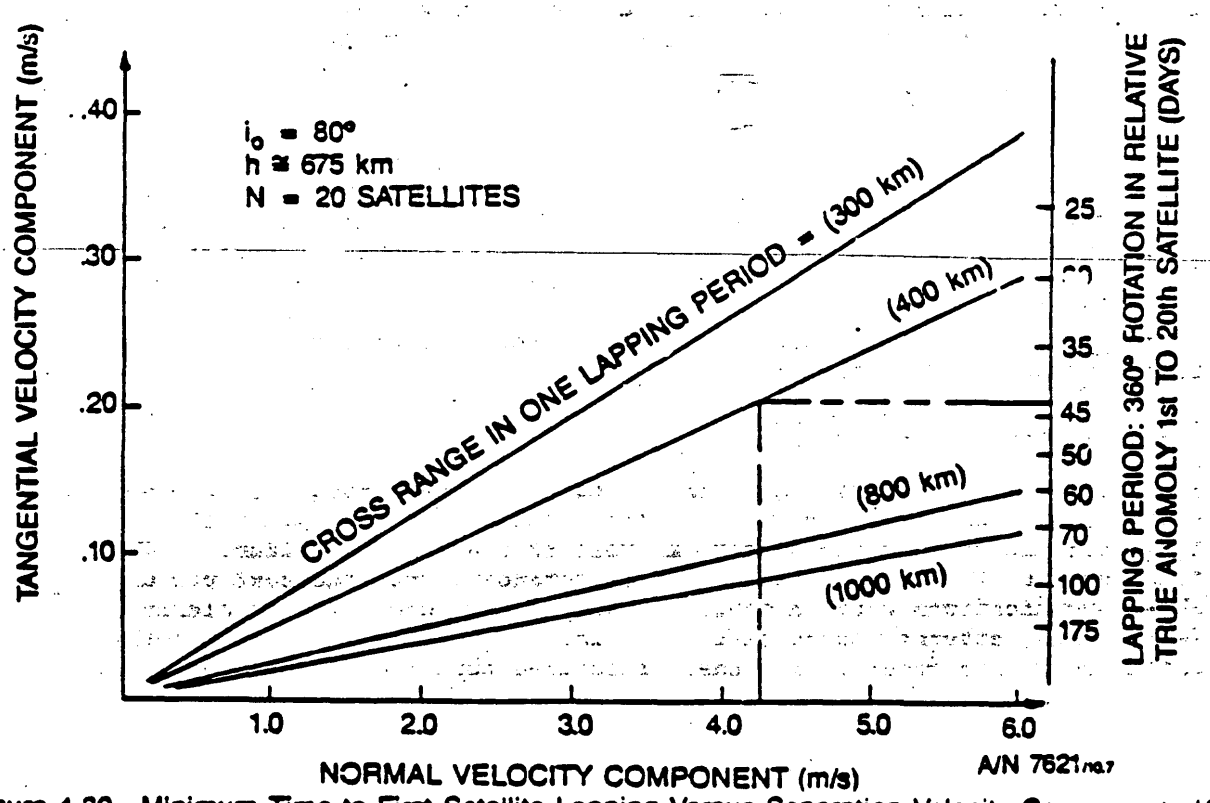



Figure 4-30 Minimum Time to First Satellite Lapping Versus Separation Velocity Components (Ai = 80° 20 Satellites)

between the first and last satellite is about 0.8 deg with a final orbit eccentricity of 0.00128. Adjacent orbits are separated by 0.038 deg in inclination and 0.0000533 in eccentricity and would lap each other in approximately 2.3 years. As time progresses, the relative separation cross ranges increase at a constant rate creating a separation between adjacent satellite orbits of 400 km in this same 2.3 year period.

Larger delta-Vs produce shorter lapping periods for a given minimum separation or larger separations in a given lapping period but at the expense of greater amounts of on-board fuel. Having completed the placement of the final MSSP satellite, the "smart" carrier vehicle would perform a plane change orbit lowering and nodal alignment to return to the STS park orbit for retrieval and return to the ground for refurbishment.

4.1.4.4 CARRIER FUEL REQUIREMENTS

Using a hydrazine liquid system sized to the maximum delta-V capability, the estimated total amount of fuel to perform the above mission (with a 900 kg dry carrier) is 710 kg. Using a bi-propellant system reduces this figure to 500 kg. Roughly two-thirds of the fuel is used in the initial orbit-raising and circularization maneuvers and could be replaced in total or in part with a hybrid design combining solid rocket motors and the liquid system. The Solid Rocket Motors (SRMs) size would be on the order of two Star-24 motors to perform the initial orbit- raising. With the current tankage capability of the preliminary smart carrier design, they are not an advantage.

4.1.4.5 LAUNCH STRATEGY CONCLUSIONS

The requirements of establishing the MSSP network of satellites in an efficient, flexible, and timely manner point to the need for a smart carrier vehicle. This carrier must provide for an orbit- raising circularization maneuver with relative satellite movement control via dispersion of nodal regression rates and randomized true anomaly locations for adjacent satellite orbits. Furthermore, this carrier should be retrievable and refurbished to minimize total program cost for establishment of the MSSP network.

4.1.5 SPACECRAFT LAUNCH COST

The cost of the total system is important to the requirement of a "low-cost" global communication system. The satellite and payload will be a large cost factor for the system, as well as the satellite launch. The total cost of the MSSP system will be dependent upon the cost per usable communications node in orbit. We wish to use low cost elements (e.g. radio, antenna, spacecraft bus, launch vehicle) but only to the extent that this results in a cost- effective number of usable nodes in

orbit. Minimization of the cost of each element may impact the total system cost greatly.

The spacecraft launch cost will depend upon two factors: the launch vehicle and the size of the MSSP satellite. The launch vehicle cost depends upon the type of vehicle and the effort needed to place the satellites in their final orbits. The spacecraft size will impact the number of satellites which can be launched per launch vehicle. If an expensive launch vehicle can launch more satellites at a more effective cost per satellite, or if smaller satellites can be launched in larger numbers per launch, the total system cost goal can be achieved. Also, the means of obtaining the final orbit required for the satellites will impact the cost of the launch system.

4.1.5.1 LAUNCH VEHICLE COST

The first process in the analysis of launch cost was to gather a data base on the cost of the possible satellite launch vehicles and to estimate the number of satellites which could be launched per vehicle. Figure 4-31 shows the possible MSSP launch vehicles with the estimates of satellites per launch and the estimated ROM cost per satellite for several possible launch inclinations. A conclusion from this figure is that shuttle launch configuration would cost about $2M per satellite and ELVs about $4 to $7M. If the cost of the satellite and payload is less than $1 to $2M, then the cost of the system launch will be considerably more than the satellites. The shuttle is the lowest cost launch approach indicated in Figure 4-31.

4.1.5.2 SATELLITE SIZE IMPACT

Size estimates for the MSSP satellite diameter has been from 18 to 28 in. The main consideration for determining the satellite diameter is the antenna size requirement. Antenna size is a trade between the necessity of attitude control and antenna capability. A sophisticated attitude control system would allow the use of a simpler antenna design which would scan the antenna beam in azimuth. A simple attitude control system might require a more complex and larger antenna. Therefore, the satellite launch will have an impact on the antenna/attitude trade. Satellite size will also impact launch cost. A larger, heavier satellite will cost more to launch. A larger satellite will reduce the number of satellites capable of being launched per shuttle or ELV launch. In terms of shuttle launch, Figures 4-32 and 4-33 show a three-point curve based upon an analysis of three possible spacecraft diameters for the number of spacecraft per launch and the projected cost per launch (not including the cost of a an Orbit Maneuvering Vehicle (OMV)or carrier). The conclusion is that satellite launch cost will be substantially impacted by satellite size and the attitude/antenna trade must include the antenna size impact.

| LAUNCH VEHICLES | NO. S/C PER ELV LAUNCH | | | ROM COSTS M$/SPACECRAFT | | | REMARKS |
|---|---|---|---|---|---|---|---|
| | 57° | 63.5° | 80° | 57° | 63.5° | 80° | • INCLINATION |
| • EXPENDABLE LAUNCH VEHICLES | | | | | | | |
| – TITAN 34D | N/A | 21 | 21 | N/A | 5.1 | 5.1 | • STAR 26B USED FOR ORBIT ADJUST TO 675X675 KM ORBIT. AVAILABILITY OF 34D LIMITED IN NEAR TERM. |
| – DELTA 3920 | 10 | N/A | 8 | 4.1 | N/A | 5.1 | • NOT IN CURRENT PRODUCTION. PRODUCTION LINE START-UP ANTICIPATED. |
| – TITAN II | N/A | 6 | 4 | N/A | 4.8 | 7.1 | • REFURBISHMENT OF TITAN ICBM's UNDERWAY BY AIR FORCE. |
| – SCOUT G1 | N/A | N/A | 1 | N/A | N/A | 5.2 | • NOT IN PRODUCTION: 12 IN STORAGE ALL ALLOCATED TO PAYLOADS. |
| • SHUTTLE W/NASA OMV | 21 | N/A | 21 | 1.5 | N/A | 1.8 | • OMV CONTRACT UNDER NEGOTIATION. VAFB STS (80° LAUNCH) QUESTIONABLE. 3M$ OMV LEASING FEE INCLUDED. PORTION OF CARGO BAY USED FOR MSSP. |

LEBSOCK6

A/N 7621mo

Figure 4-31 Possible MSSP Launch Vehicles

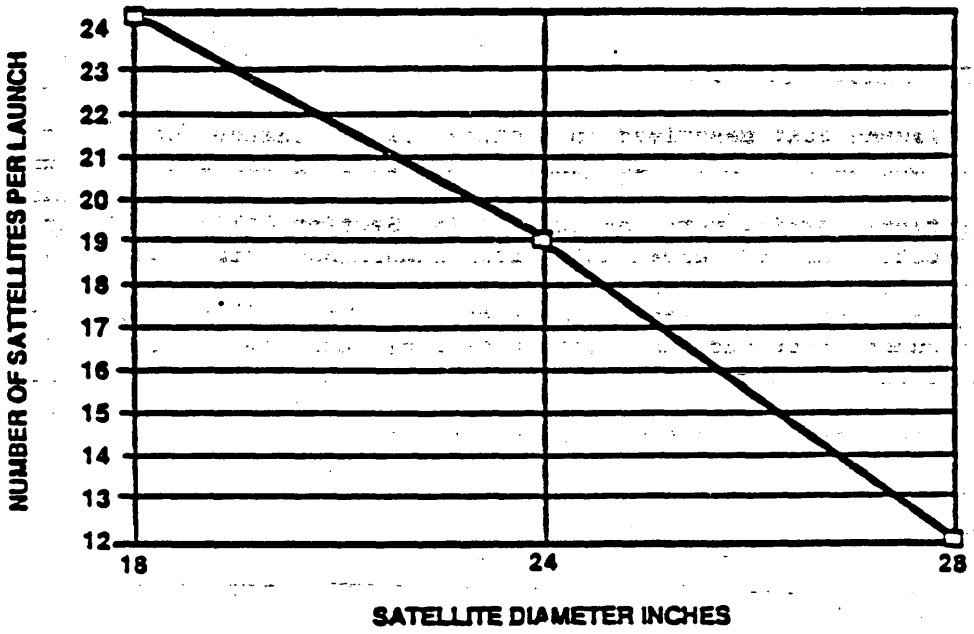



Figure 4-32 Satellites Per Launch Versus Satellite Diameter

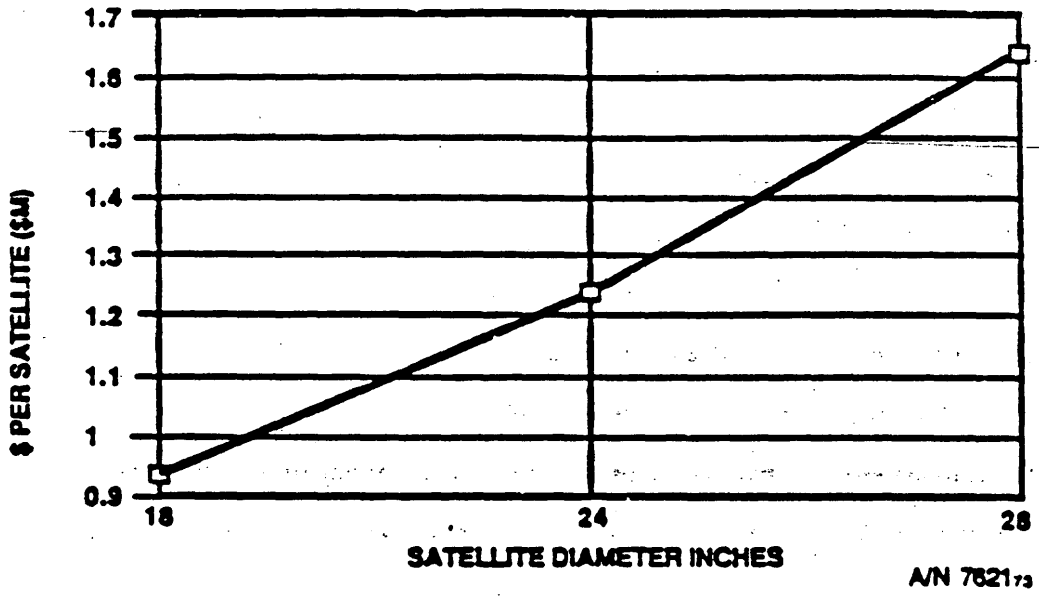



Figure 4-33 Satellite Launch Cost Versus Satellite Diameter

### 4.1.5.3 ORBIT ACHIEVEMENT COST

The shuttle launch cost described in section 4.1.5.1 listed the cost of an OMV which boosts the MSSP satellites to their final orbits. Two methods of boosting the spacecraft from the 300 km shuttle orbit to the 675 km operational orbit were described in Section 4.1.4. The less expensive booster is an expendable spin-stabilized stage using two solid motors. The more expensive booster is a recoverable three-axis stabilized stage using hydrazine thrusters. The following element costs are assumed to compare the cost per usable node in orbit for the two booster designs. The baseline numbers for launch were set at 20 satellites per expendable launch vehicle and 24 per recoverable vehicle. These numbers will vary with spacecraft size.

A) Take the recurring cost of the expendable solid booster including shuttle charges to be $15M.

B) Assume that the recurring cost for the reusable liquid booster is $17.5M and that the non-recurring cost is $45M.

The launch cost per usable node for the expendable booster system is then:

$$\frac{\$15M}{20} = \$0.75M/\text{node}$$

The comparable cost for the reusable system launching 240 spacecraft in 10 launches is

$$\frac{\$45M}{240} + \frac{\$17.5M}{24} = \$0.92M/\text{node}$$

The cost difference per usable node in orbit is approximately $170K for the reusable system compared to the expendable system. This cost does not reflect the hidden programmatic cost such as

A) Cost of two additional launches (12 vs. 10)

B) Cost of providing safety for launch of rockets in shuttle

C) Operational cost of satellite launches

D) Possible reimbursement of carrier after final launch

The orbit achievement hardware cost for an expendable or recoverable will not be much different for either approach. However, the reusable "smart" carrier could provide better performance of the valuable

resources (satellites) with more accurate spacecraft placement in orbit and initial knowledge of position and orientation of the resource. The basic conclusion to be drawn is that the orbit achievement cost will be about $1M per satellite. The total launch cost with the shuttle will be about $2M per satellite.

### 4.1.6 SPACECRAFT ORBITAL ENVIRONMENT

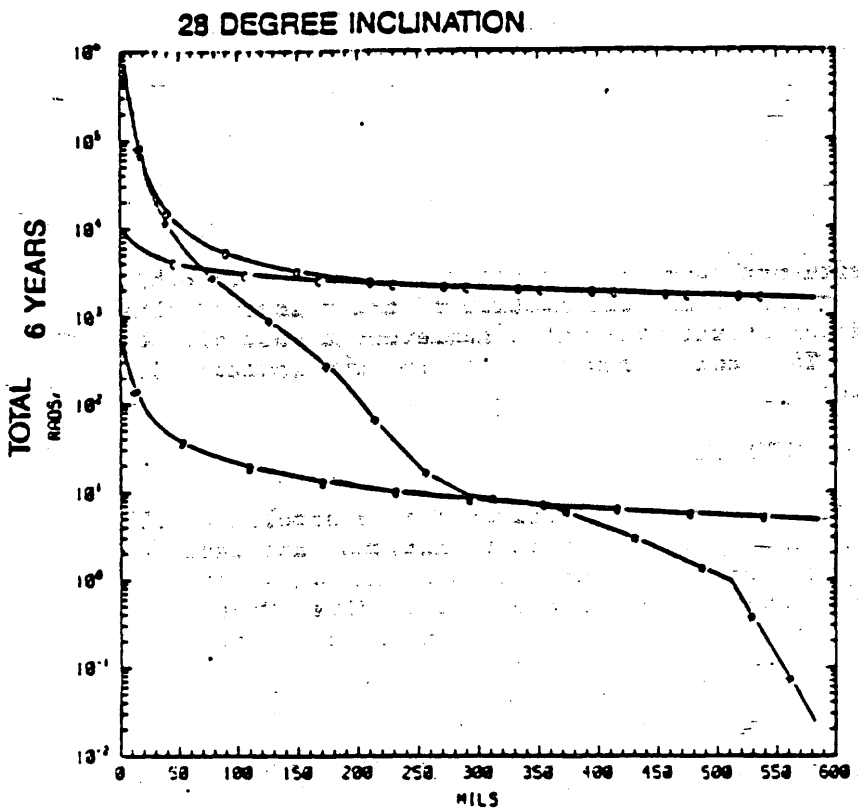
The radiation and environmental levels experienced by a satellite will be dependent upon the selection of the satellite altitude and inclination angle. The primary environmental concerns which vary with the selected altitude is the exposure of the spacecraft to "Free Oxygen" or "Atomic Oxygen" (AO) and the radiation levels to which the spacecraft is exposed. The magnitude of the effects of these two factors - radiation and AO-vary with increasing altitude. Radiation levels increase with altitude. Exposure to AO decreases with altitude. It was desired for the MSSP to define the impact of these two factors and compare the impact with the selected satellite altitude of 675 km (section 4.1.1).

#### 4.1.6.1 RADIATION DOSAGE

The MSSP satellites will be processor-intensive. The purpose of the MSSP satellites will be to provide a stable platform in space for the network processor. The reliability of a microprocessor and therefore the mission of the MSSP, will be dependent upon the total radiation dosage, (specifically, trapped particle radiation) to which the satellite is exposed. It is highly desirable to know the total dosages of radiation the spacecraft will be exposed to, the impact of radiation shielding, and the variation of the radiation exposure with years in orbit and orbit altitude.

The first analysis was performed with the parameters listed in Table 4-1. SOFIP and SHIELDOSE are software programs used to the projected radiation dosage. Figures 4-34, 4-35 and 4-36 show the expected radiation dosage (6-year lifetime) for 28, 57, and 95 deg inclination angles and 740 km spacecraft orbits vs. the depth of aluminum shielding. The 28.5 deg inclination angle has the worst cumulative radiation level and the effect of shielding diminishes for thicknesses beyond 100 to 150 mils (1 mil=0.001 in.). The curve labeled D is the sum total radiation. For shielding thickness beyond 150 mils, the major component of radiation is trapped protons. If the minimum shielding thickness provided by a satellite structure is assumed to be 0.064 in. or 64 mils (=1/16 in.), the maximum benefit from internal shielding will be obtained with a small amount of processor box shielding.

The radiation levels for higher inclination angle orbits is greater for lower shielding mainly because of free electrons. Once the shielding level is 150 to 200 mils thick, the radiation dosage will be less than



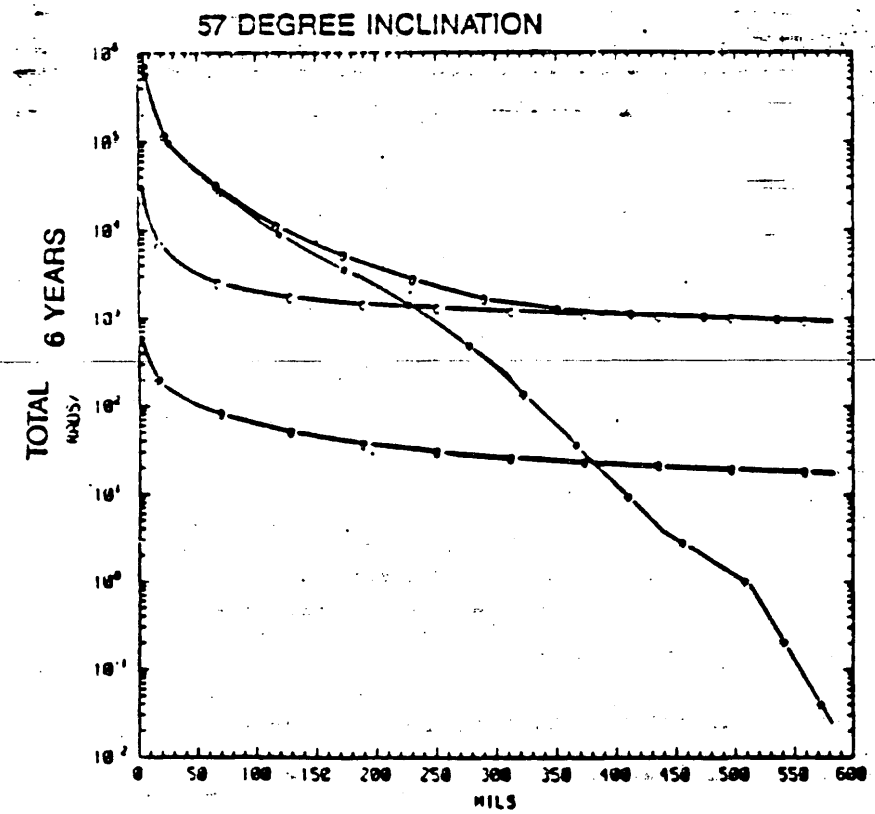

**CASE:**
ORBIT: 740 km x 743 km
INCLINATION: 28°
KEPL. PERIOD: 1.660513 HRS.

**MODEL:**
FIELD: MAGSAT 1980
EPOCH 1992
ELECTRONS: IZ AE8
OZ AE17-LO
THRESH LEVEL 0.5 MeV
PROTONS: AP8-MAX
THRESH LEVEL 5 MeV
CONFIDENCE: 90%
MEDIUM: ALUMINUM
GEOMETRY: SOLID SPHERE

**OTHER:**
MISSION DURATION: 6 YR.
SIMULATION DURATION: 35 ORBITS
58.2 HRS.

**KEY**
A = ELECTRONS
B = BREMSSTRAHLING
C = TRAPPED PROTON
D = TOTAL

Figure 4-34 Radiation Dosage — 6 yr, 28 deg Inclination Versus Shielding




**CASE:**
ORBIT: 740 km x 743 km
INCLINATION: 57°
KEPL. PERIOD: 1.660513 HRS.

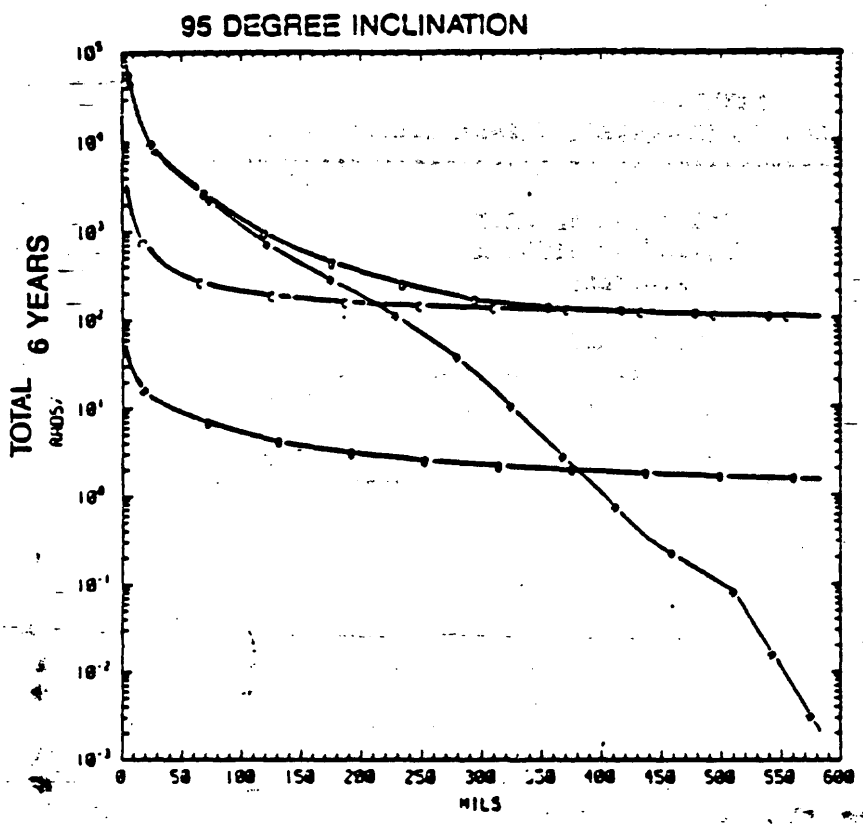
**MODEL:**
FIELD: MAGSAT 1980
EPOCH 1992
ELECTRONS: IZ AE8
OZ AE17-LO
THRESH LEVEL 0.5 MeV
PROTONS: AP8-MAX
THRESH LEVEL 5 MeV
CONFIDENCE: 90%
MEDIUM: ALUMINUM
GEOMETRY: SOLID SPHERE

**OTHER:**
MISSION DURATION: 6 YR.
SIMULATION DURATION: 35 ORBITS
58.2 HRS.

**KEY**
A = ELECTRONS
B = BREMSSTRAHLING
C = TRAPPED PROTON
D = TOTAL

Figure 4-35 Radiation Dosage — 6 yr, 57 deg Inclination Versus Shielding

A/N 7621



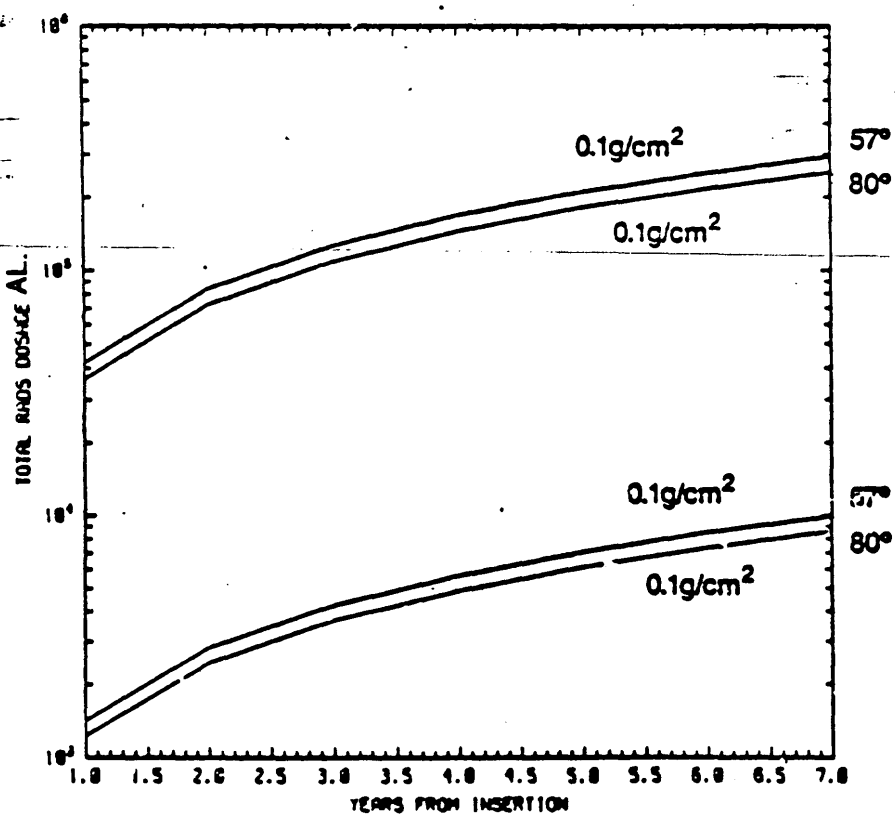

**CASE:**
ORBIT: 740 km x 743 km
INCLINATION: 95°
KEPL. PERIOD: 1.660513 HRS.

**MODEL:**
FIELD: MAGSAT 1980
EPOCH 1992
ELECTRONS: IZ AE6
OZ AE17-LO
THRESH LEVEL 0.5 MeV
PROTONS: AP8-MAX
THRESH LEVEL 5 MeV
CONFIDENCE: 90%
MEDIUM: ALUMINUM
GEOMETRY: SOLID SPHERE

**OTHER:**
MISSION DURATION: 6 YR.
SIMULATION DURATION: 35 ORBITS
58.2 HRS.

**KEY**
A = ELECTRONS
B = BREMSSTRAHLING
C = TRAPPED PROTON
D = TOTAL

Figure 4-36 Radiation Dosage — 6 yr. 95 deg Inclination Versus Shielding




0.1g/cm² = 14.6 mils

1.0g/cm² = 146 mils

A/N 7621ma

Figure 4-37 Radiation Dosage Versus Duration of Lifetime

TABLE 4- 1
RADIATION DOSAGE SPECIFICATION PARAMETERS

| | |
|---|---|
| MISSION SPECIFICATIONS | 740 X 743 km ORBIT<br>PERIOD: 99.6308 min<br>1.66 hrs<br><br>MISSION LENGTH: 6 years<br>MISSION START DATE: 1-1-1992<br>3 INCLINATIONS: 28, 57, AND 95 deg |
| MODEL SPECIFICATIONS | SOFIP used to find incident flux<br>SHIELDOSE used to find depth-dose data<br>Solar maximum proton model used<br>Solar min-lo electron model used<br>Spherical satellite model<br>35 orbit simulation projected over 6 years |

the dosage at 28 deg inclination. Spacecraft flying at higher inclination will fly through the South Atlantic Anomaly and will fly through the poles of the Earth's magnetic dipole.

Figure 4-37 shows the the dosage vs. mission duration for a satellite altitude of 675 km and inclinations of 57 and 80 deg. The radiation dosage builds as the satellite lifetime extends. Also, the factor of the thickness of the shielding is shown to reduce the total dosage, but the curves have the same shape. The radiation dosage for a 675 km altitude, 6 year mission, Figure 4-37, is slightly less than a 740 km altitude, 6 year mission of Figure 4-35.

The radiation levels of approximately 10^4 rad Al (Rad Si = 0.857 Rad Al) should not be a problem for bipolar components or linear ICs. For CMOS parts and components such as A/D convertors, however, it would be desirable to reduce the cumulative radiation levels to around 10^3. These requirements will require that the shielding thickness be increased to the larger values of about 300 mils total around those parts. These parts should be radiation hardened. If possible, a lower satellite altitude or the lower altitude of the altitude range be chosen.

4.1.6.2 ATOMIC OXYGEN

During the course of the MSSP study, concern was expressed regarding the effects of Atomic Oxygen (AO) on the low Earth orbit satellites. BASD analyzed the possible effect of AO on the MSSP satellite and determined that it would be minimal and easily nullified.

AO is the presence of free oxygen molecules within the atmosphere. Free means the oxygen radical (O) rather the normal oxygen (O2). When a spacecraft flies through the atmosphere with AO, the spacecraft exterior surfaces will erode. Simply put, the spacecraft will rust away. The rate of this erosion will depend upon the spacecraft altitude. At the higher altitudes, less AO will exist and the rate of erosion will decrease. Therefore, for less erosion the higher 740 km altitude would be beneficial for the MSSP satellites.

The surfaces which will be impacted by AO are any exterior structure, solar arrays, antenna radome, thermal blankets, and thermal radiators. The amount of impact of AO on any of these surfaces for a satellite with a lifetime of five years will be minimal. For example, the exterior structure will be aluminum which will perform in space similarly to its performance on earth. A thin layer of aluminum oxide will be formed and then the surface will show minimal erosion.

The thermal blankets, which have a Kapton exterior surface, will experience the most erosion at a rate of 0.004 in over the five-year lifetime. The exterior Kapton layer will generally be 0.010 in. thick simply because a thinner layer would increase fabrication cost due to handling.

Also, the erosion of the exterior surface will not affect the thermal resistance required of the thermal blankets. Therefore, the effect of AO on the thermal blankets can be eliminated by simply using a thicker outer layer (or a layer commensurate with normal fabrication procedures). The cost of eliminating the impact is minimal.

### 4.1.7 SATELLITE ORBITS CONCLUSION

The analyses performed suggest that an altitude of 625 to 675 km is sufficient for MSSP lifetime and satellite separation requirements. Altitudes higher than 675 km are acceptable, but would increase the launch cost of the MSSP system. The higher altitudes will also have higher levels of radiation. Altitudes to 740 km will not greatly increase the radiation levels, but since the MSSP satellites will be "network processors in space", the lower altitudes would be optimum. The effects of AO at 625 to 675 km will be minor. The effect must be considered in the satellite final design, but will not a cost driver.

Analysis of the spacecraft density with various launches and inclination angles showed that placing more satellites in the higher inclination angles tended to provide a uniform density distribution. The ultimate uniform distribution could be obtained only with the costly approach of individually launching satellites. However, the dispersion of the satellites can be obtained by varying the satellite altitudes or inclination angle in minor amounts so that satellites dispersion would occur due to orbital regression. The trade is between the cost of satellite launch vs. the time required for the satellites orbits to separate. The process of orbital regression is cost beneficial.

The final conclusion of the analysis is that the size of the satellites are the main cost driver for the MSSP system. The satellite size will reduce the number of satellites which can be inserted into orbit per launch and increase the number of launches required. The size will increase the cost of the orbit achievement vehicle and the number per launch will extend the time required to finally obtain system operational status. Valuable assets with limited lifetimes will be in space waiting for the completion of the system.

## 4.2 GROUND USER

Is the most important concern of the MSS program. The satisfaction of the ground user. Because there may be thousands of ground users and only 240 satellites, the system cost emphasis must be tilted toward the user. The design of the satellite, antenna, and radio must also be compatible with the ground user. For example, if the ground user can transmit very high power with a small antenna to complete the uplink, but cannot hear the satellite downlink because the ground antenna does not have enough gain, then the system does not work. Conversely, if the satellite system has capability beyond what can be used by the normal ground user, then the satellite is not cost effective.

On this basis, BASD performed an analysis of ground-user-oriented system concerns and outlined possible trades with respect to RF power and antenna gain. The objective of the analysis was to identify system concerns and initiate the definition of the user interface thereby aiding satellite design.

### 4.2.1 GROUND USER/SPACECRAFT ANTENNA TRADE

Before preceding with an analysis of the ground user, a baseline specification for the ground user was generated:

BASD GROUND USER SPECIFICATION

| | |
|---|---|
| Transmit power: | 10W |
| Antenna gain: | 17 dB |
| Receive sensitivity: | same as the satellite |
| Elevation angle: | 10 deg |

Assumption S/C 10W, 17dB Gain

Although this table is trivial, it is a start. The main interface requirements between the satellite and ground user and the major cost for the ground station are listed. Each of these items represents a trade between the cost of the satellites and the thousands of ground users. The impact of atmospheric losses of about 0.5dB at low elevation angles has not been included.

The main trade parameters are antenna coverage, antenna gain, and RF transmit power for the ground user and satellite. The purpose is to minimize requirements and cost of the user and satellite but ultimately MSSP system cost. The main equations used to define antenna characteristics in this analysis follow:

antenna gain vs. area:

$$\text{Gain} = \frac{4\ \text{Pi}\ A}{\text{Lamda}^2} * \text{Eff}$$

A: area Lamda: wavelength

Eff: antenna efficiency

antenna gain vs. antenna beamwidth:

Gain = ( 41253 / BW^2 ) * EFF  BW: 3dB beamwidth

antenna pattern which is parabolic:

$$\frac{dB}{3} = \frac{BWx}{BW3} \text{ squared}$$

BW3: 3 dB beamwidth
BWx: beamwidth at x dB

The frequency is assumed to be 3 GHz. The antenna efficiency is assumed to be 55 percent. This efficiency is a conservative number which will not greatly impact the final conclusion. Smaller ground antennas with efficiency of up to 65 percent will only add to the system capability.

### 4.2.2 GROUND USER ANTENNA CONSIDERATIONS

Figure 4-38 shows antenna gain vs. area. The presentation is for antennas 1 x 1 inch to 20 x 20 in. The antenna could have non-symmetrical dimensions or be a dish, but the square is chosen to give a representative idea of the size of the ground antenna. A 17 dB antenna gain would be about 12 in. sq. Figure 4-39 shows the beamwidths for the antennas vs. area for the same dimensions as used for the antenna gains. A 17 dB antenna would have a 19 deg beamwidth.

When an antenna is used on a ground configuration, the user must be concerned with the problem of multipath (Illustrated in Figure 4-40). A direct signal up to the satellite will have an interferometer pattern with a reflected signal from the ground. Even on an airframe, the multipath problem will occur. The "final" antenna gain will be the sum of the direct signal and the reflected signal and will depend upon the height of the antenna above the reflecting surface. This will determine the amplitude and phase of the reflected signal. For MSSP, the height and the physical orientations are not known for the general ground user and therefore, we must assume that the signals will subtract. The signals are "added" voltage-wise not power-wise. Figure 4-41 shows the effect of the total signal variation vs. the amplitude of the reflected level assuming that the two signals are out of phase. A 10 dB

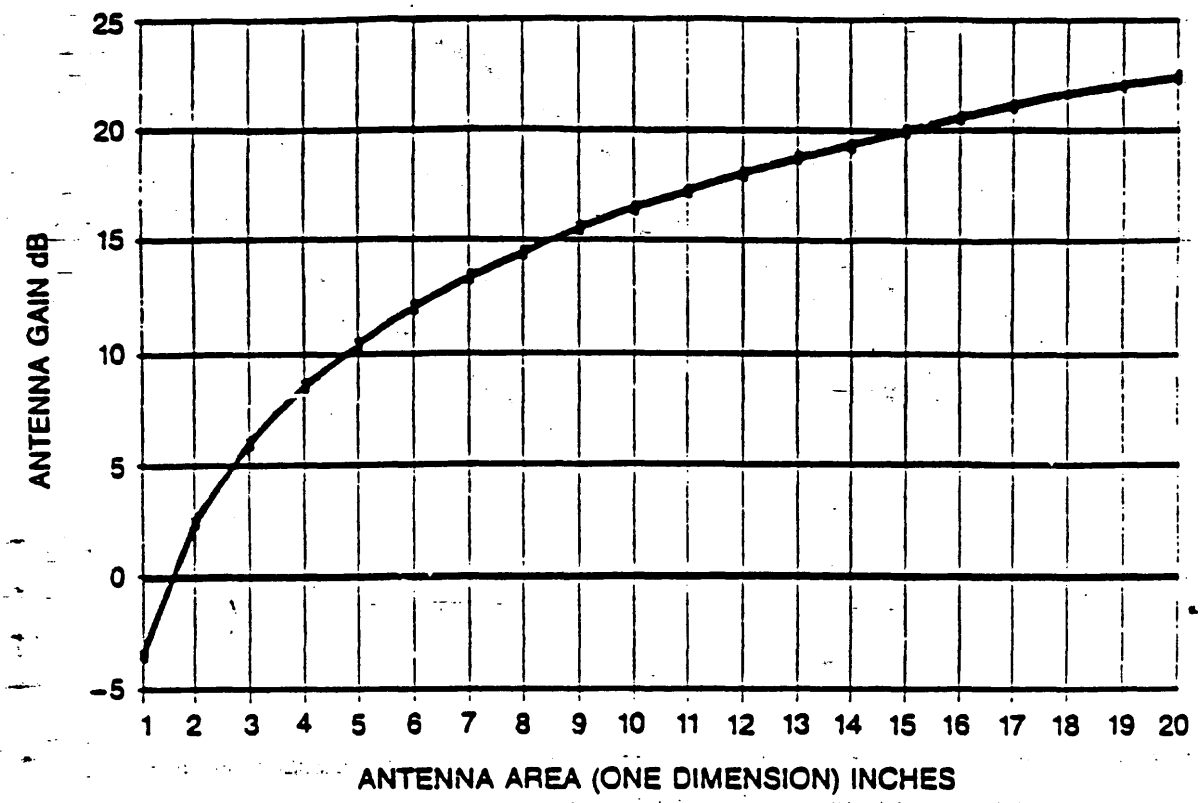



Figure 4-38 Antenna Gain Versus Antenna Area

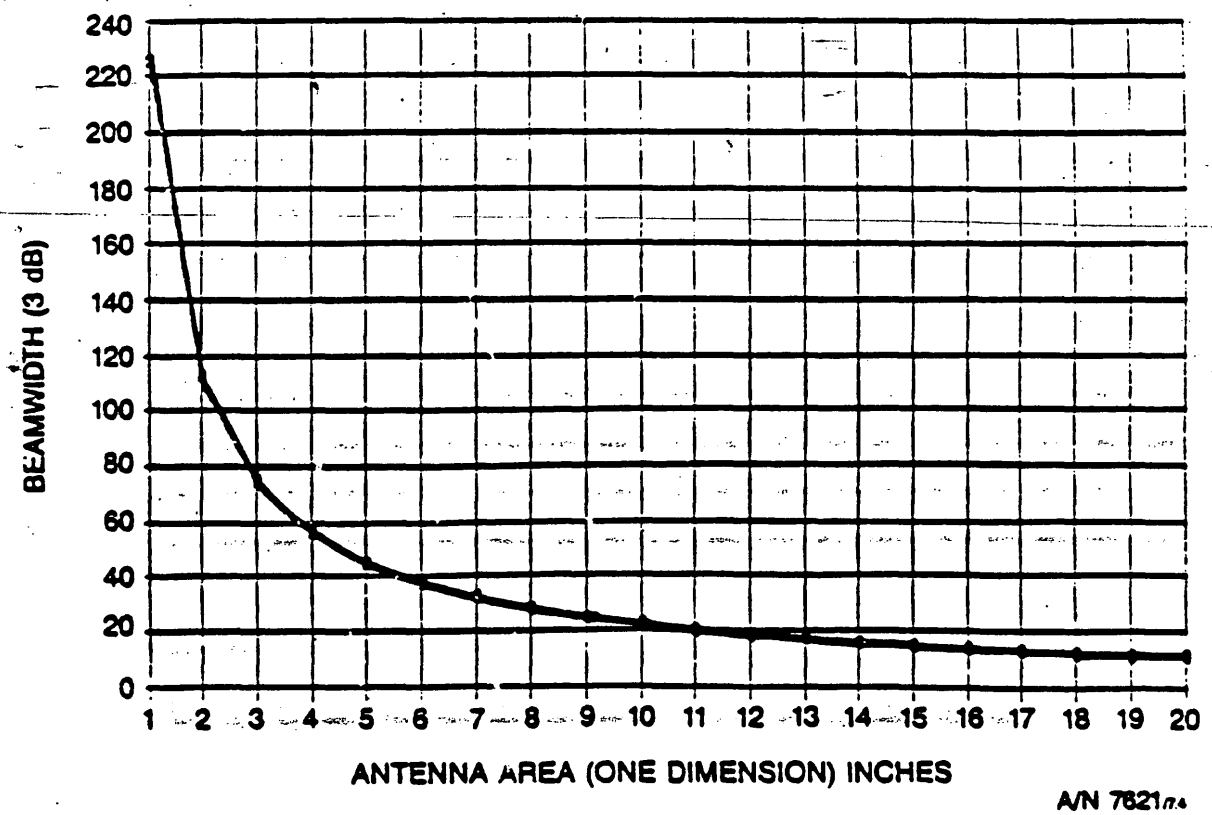



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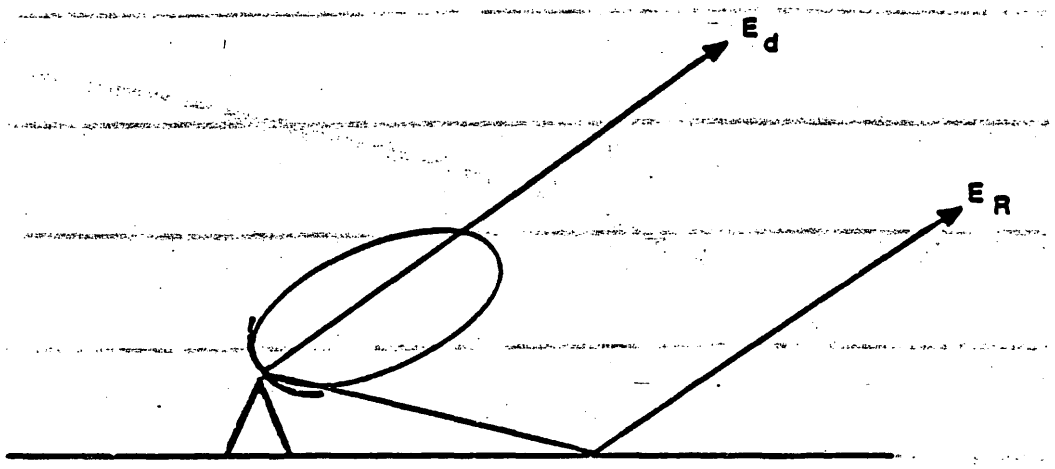
Figure 4-39 Antenna Beamwidth Versus Area




$$E_{TOTAL} = E_{DIRECT} + E_{REFLECTED}$$

Figure 4-40 Multipath

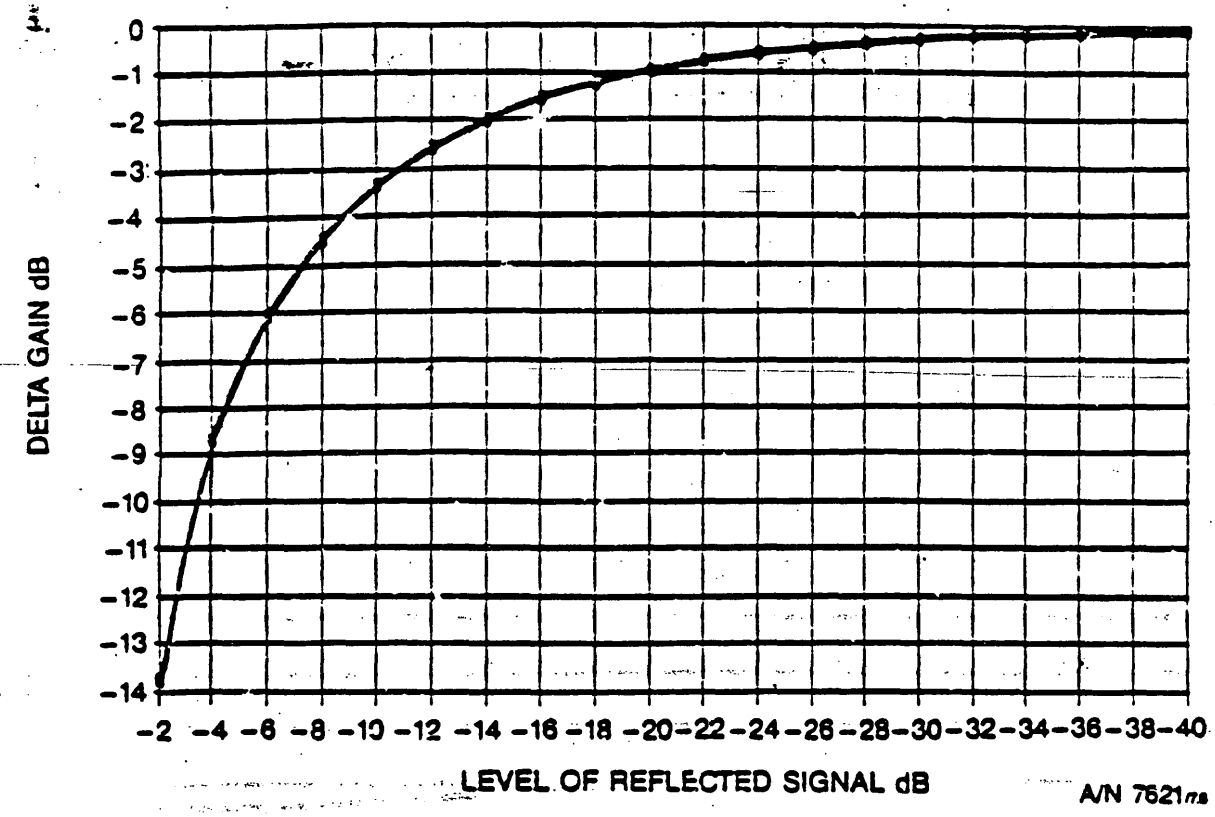



Figure 4-41 Total Signal Level Variation (Delta Gain) Versus Reflected Signal Level

reflected signal will cause the final gain to be reduced by about 3.3 dB or the antenna gain would be reduced to 13.7 dB. This is not enough to make the link. If a compromise is made to accept a 1 dB variation, the reflected signal level must be 20 dB down from the direct signal.

If the ground user has the antenna pointed at the 10 deg elevation and the reflected signal level aimed 10 deg below the horizon (assuming a flat earth) is 20 dB below the direct signal, then the 20 dB beamwidth of the antenna is 40 deg. With the assumed parabolic beam, the 3 dB beamwidth can be calculated and the required antenna gain calculated. Figure 4-42 shows the calculated required antenna gain vs. elevation angle for the 20 dB reflection. In order to obtain less than 1 dB reflection at a 10 deg elevation, the antenna gain must be at least at 20 dB. Figure 4-43 shows the relationship between required area and the elevation angle. It must be noted that these curves show the required gain and area required of the ground user for a 1 dB gain variation due to ground reflections.

Another physical factor which must be considered in the ground user equation is the path loss of the up/down link. Assuming an altitude of the spacecraft of 675 km, the slant range from the spacecraft to the user will vary from 675 km to 2100 km. The path loss to the user will vary with the elevation angle of the user. Figure 4-44 shows the variation of the path loss reduction based upon 0 dB at the 10 deg elevation angle. If 17 dB is required at 10 deg elevation angle, then when the satellite is directly above the user (elevation 90 deg), the required gain for the same system performance is about 10 dB less. Figure 4-45 shows the required gain when the path loss variation is subtracted from 17 dB.

The final physical parameter is the angle that the spacecraft must view to see the ground user. Figure 4-46 shows the scan angle that the antenna of the spacecraft must scan to view the user vs. the user elevation angle. The antenna must view a cone of about 62 deg half angle in order to view all possible 10 deg elevation users.

Summarizing for the 10 deg elevation user:

| | |
|---|---|
| required gain (20 dB) | 20 dB |
| possible reflection amplitude | 1 dB |
| required area | 14.5 in. sq |
| user beamwidth | 15 deg |
| path loss | max |
| required user link gain | 17 dB |
| spacecraft scan angle | 62 deg |
| spacecraft gain | 17 dB |

The 15 deg user beamwidth means that if the pointing of the user antenna is off by 7.5 deg, the antenna gain will be reduced by 3 dB.

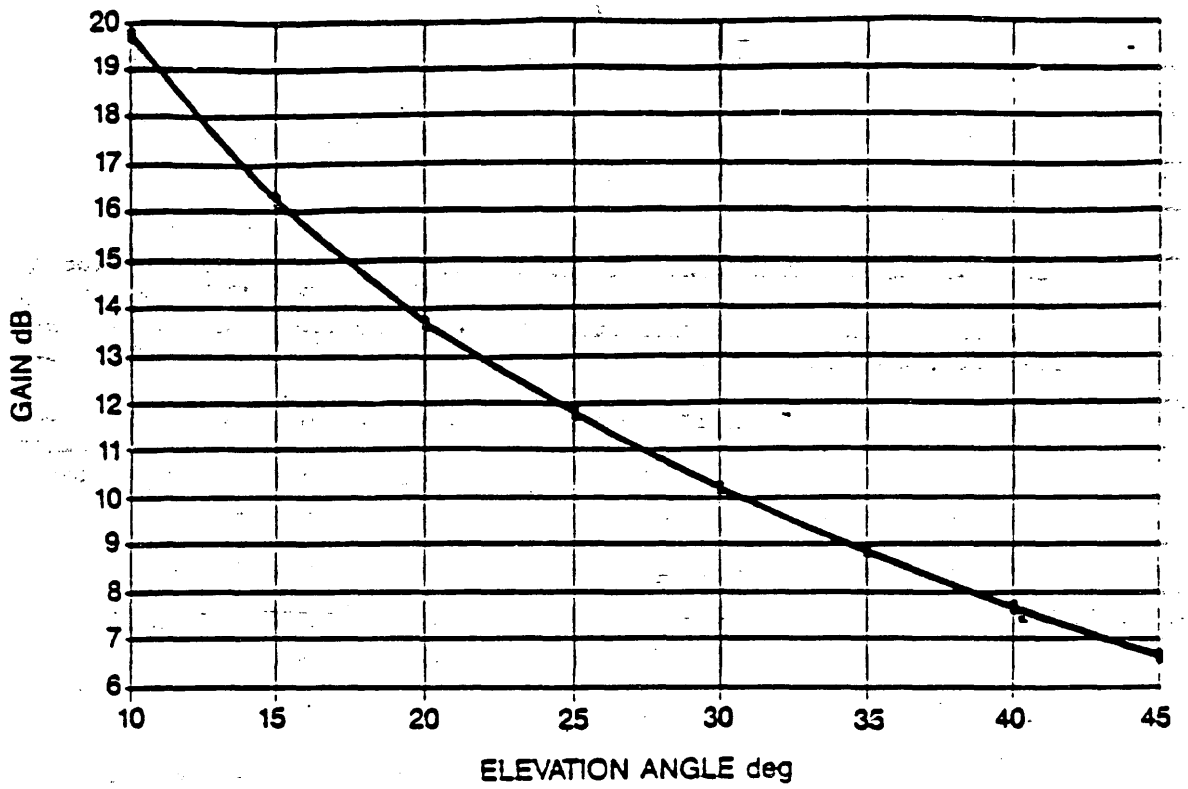



Figure 4-42 Required Gain Versus Elevation Angle for 20 dB Reflection Level

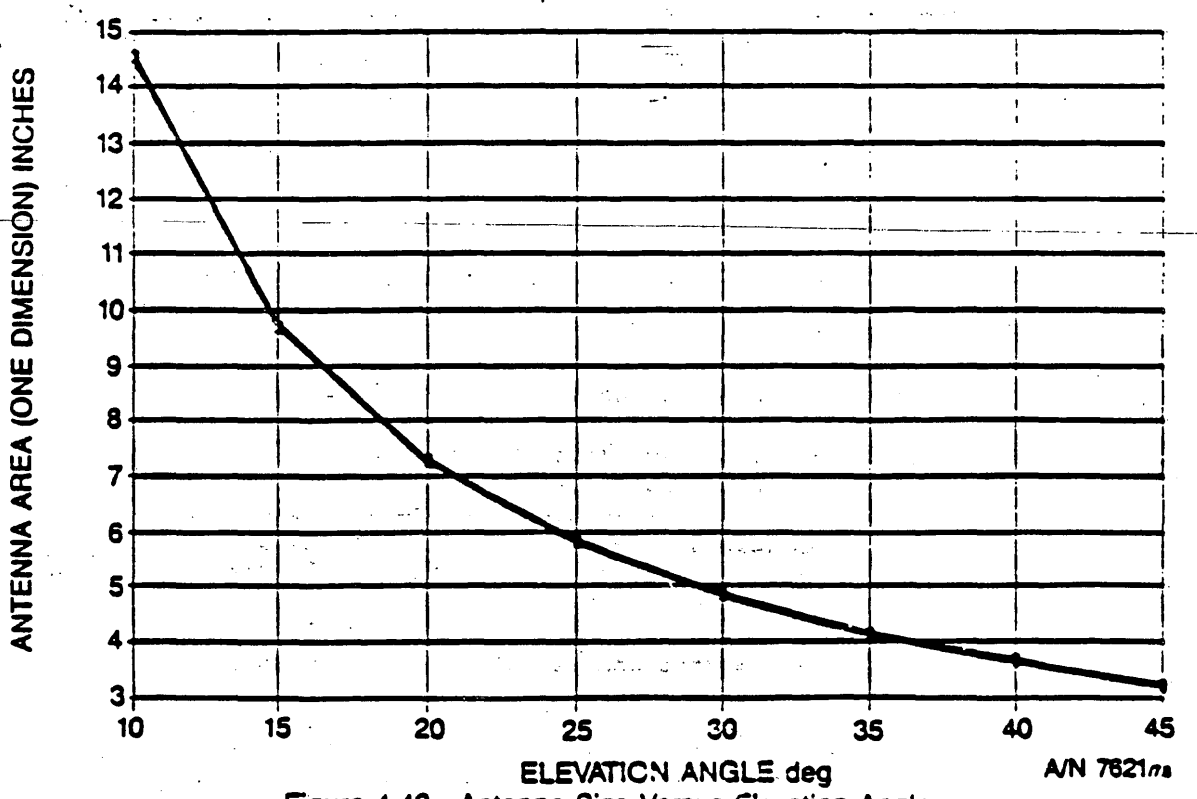



Figure 4-43 Antenna Size Versus Elevation Angle

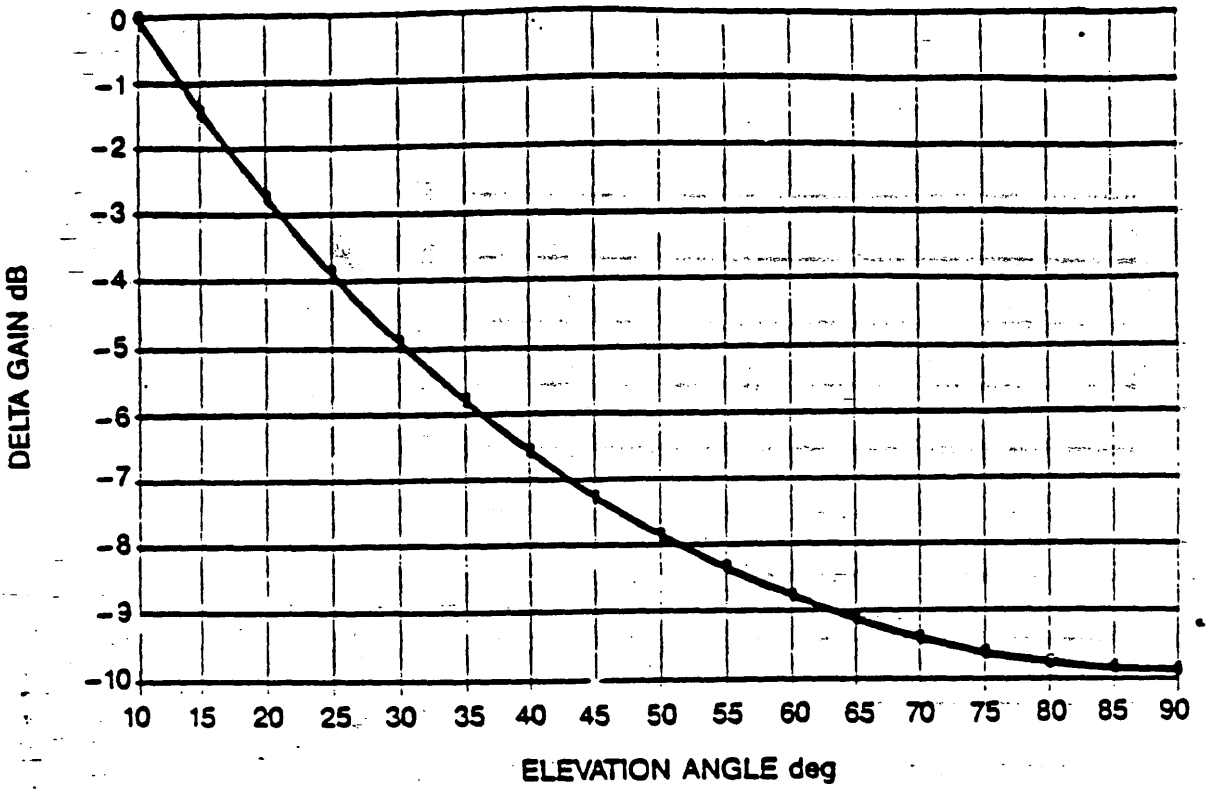



Figure 4-44 Gain Delta Versus Elevation Angle Due to Path Loss Delta wrt 10 deg Elevation

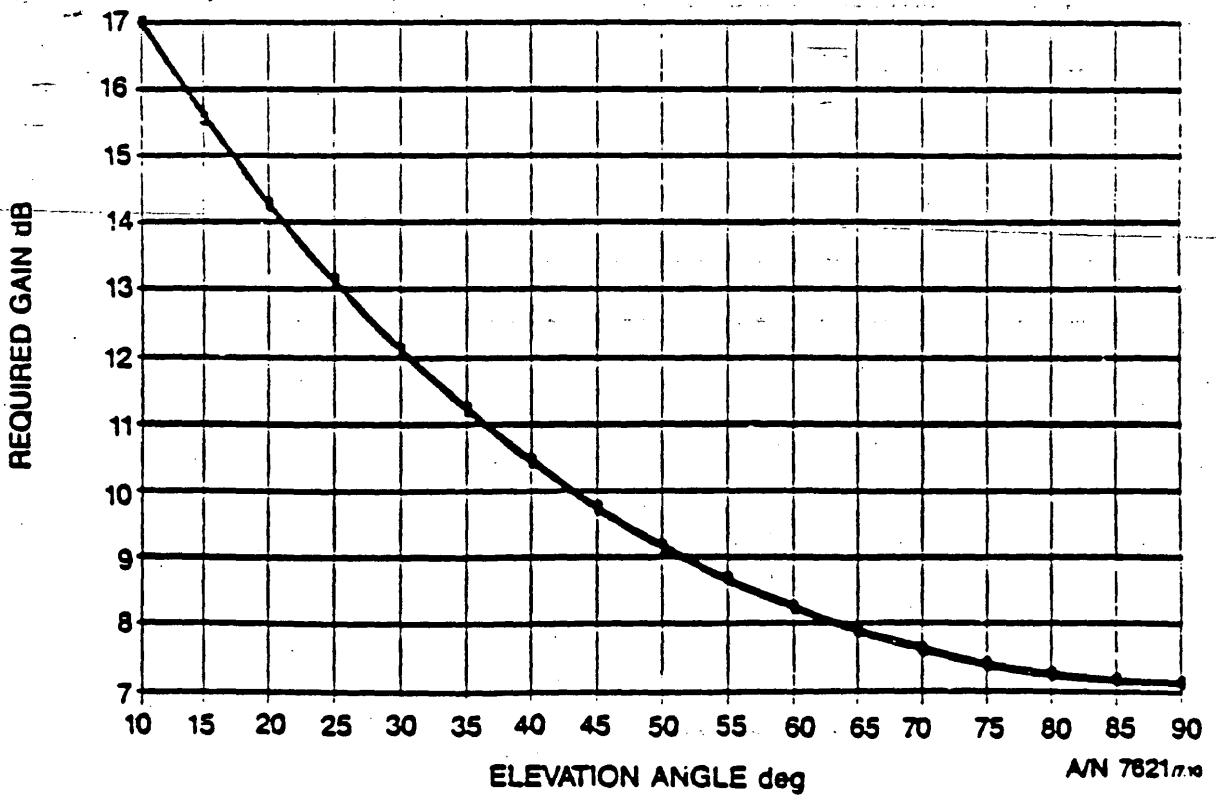



Figure 4-45 Required Antenna Gain Versus Elevation Angle Required 17 dB Gain at 10 deg Elevation

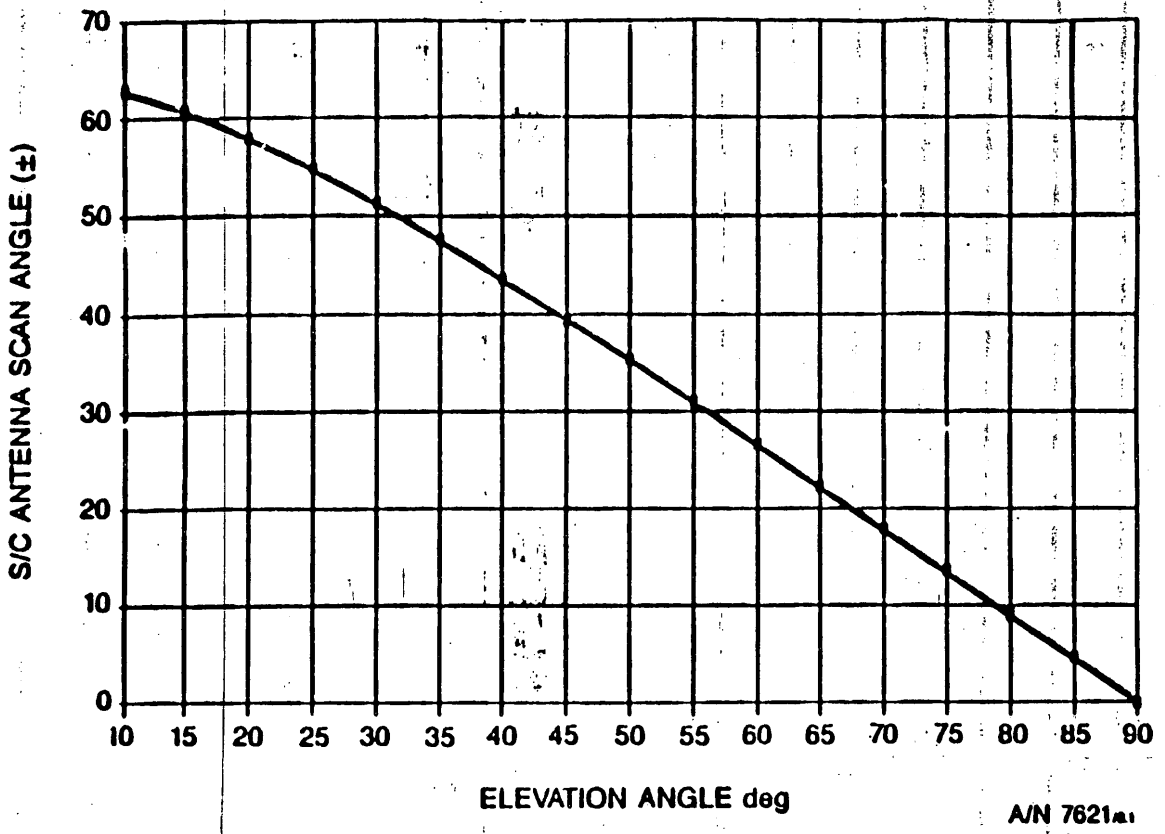



Figure 4-46 Required S/C Antenna Scan Angle Versus Ground User Elevation Angle

When the 1 dB of reflection is included, the pointing accuracy must be greater than 7.5 deg. Also, the 20dB gain antenna would compensate for atmospheric losses.

### 4.2.3 GROUND USER QUALITY FACTOR

The operational considerations of communication with low Earth orbit satellites must be factored into the trade of the performance of the ground user's antenna. The most important considerations are the number and position of the satellites which can be viewed and the scan rate required of the ground user to track the satellites.

If uniform distribution around the world is assumed, the number of satellites in view by the user will be

$$240 * 1/2 * [1 - \cos(\text{earth angle})]$$

$$\text{earth angle} = \text{Acos}[(R/R+h)*\cos(\text{elevation angle})] - \text{elevation angle}$$

or simply, the area seen by the user times the number of satellites. This plot is shown in Figure 4-47. The real case will not have perfect uniformity. However, the curve is representative of the number of satellites vs. the user elevation capability. This curve shows the cumulative number of spacecraft in view with a user view capability from 90 or overhead to a specified elevation angle. Figure 4-48 shows the number of satellites in view in 5 deg cells at any elevation angle. For the high elevation angles, there are not a lot of satellites. The probability that there will be a satellite directly overhead is very low.

Second, the user's requirement and gimbal rates for tracking the satellite must be ascertained. The satellite at 675 km altitude is moving at about 7.5 km/sec. The gimbal rate of the user will be much higher when the satellite is overhead - close to the ratio of the earth radius to the satellite altitude times the spacecraft rate. Figure 4-49 shows the user gimbal rate vs. the elevation angle. The angle of the spacecraft antenna scan angle is 90 minus the sum of the earth and user angle. Therefore, the spacecraft gimbal rate will be the user gimbal rate minus the spacecraft rate. Both scan rates will be very high at 90 deg elevation and slow at low elevation angles.

The gimbal rate shown in Figure 4-49 is for an overflight pass. When the spacecraft passes in view at angles other than an overflight, the elevation gimbal rates will be slower and the azimuth gimbal rates will be larger. Figure 4-50 shows the cumulative time for an overhead pass vs. the elevation angle. For example, if a spacecraft flew over a

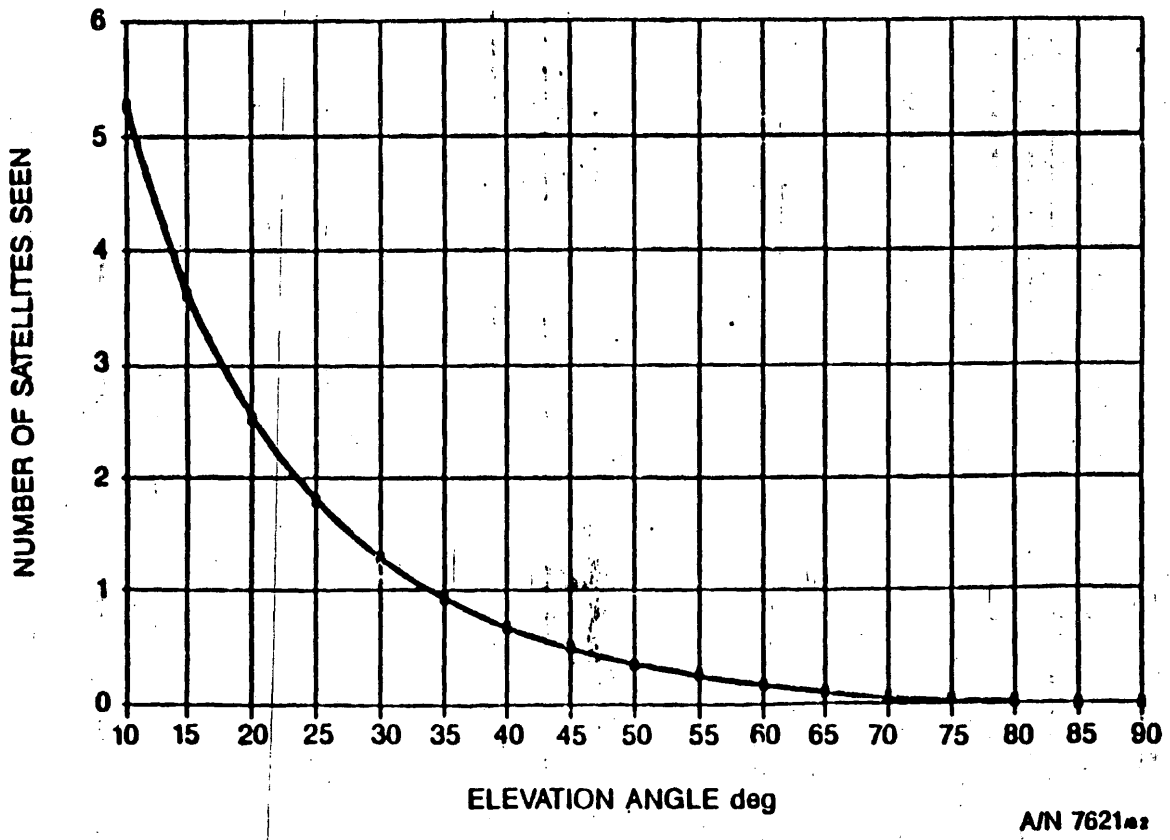



Figure 4-47 Number of Satellites Versus Elevation Angle Assumption: Uniform Satellite

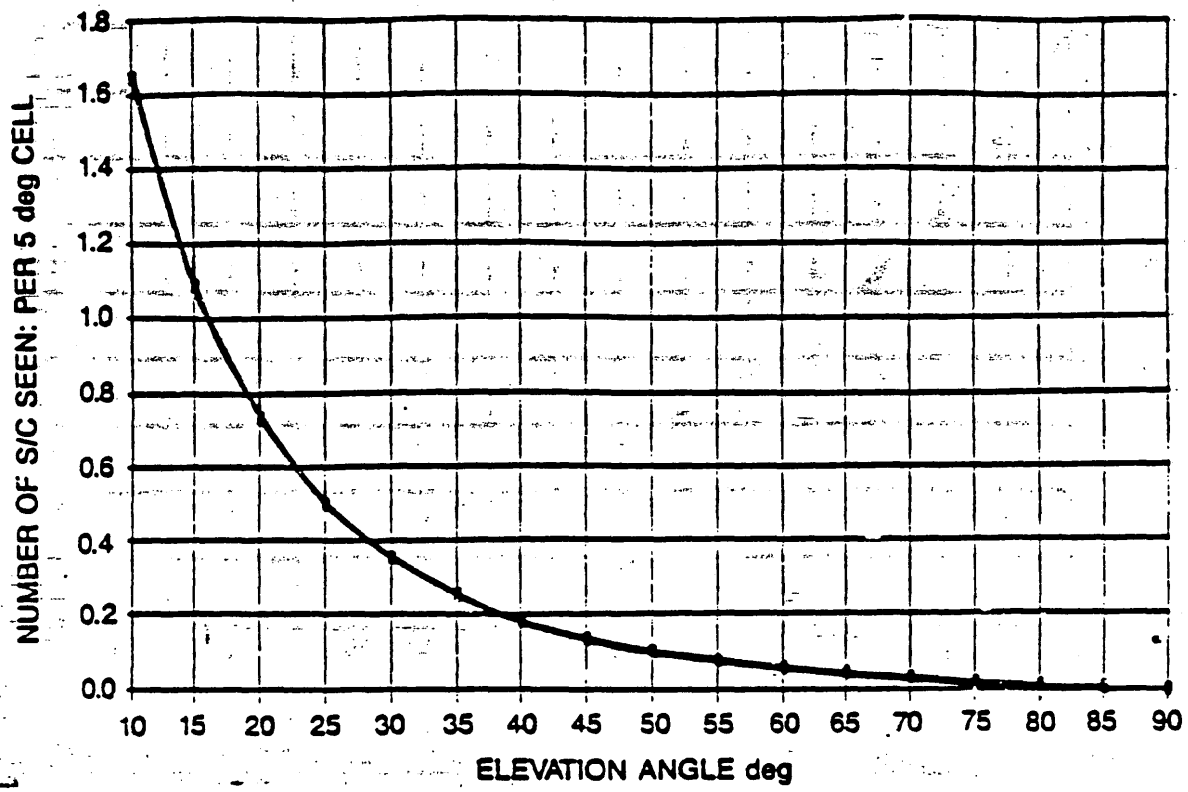



Figure 4-48 Number of Satellites Versus Elevation Angle Per Cell; Assumption: Uniform Satellite

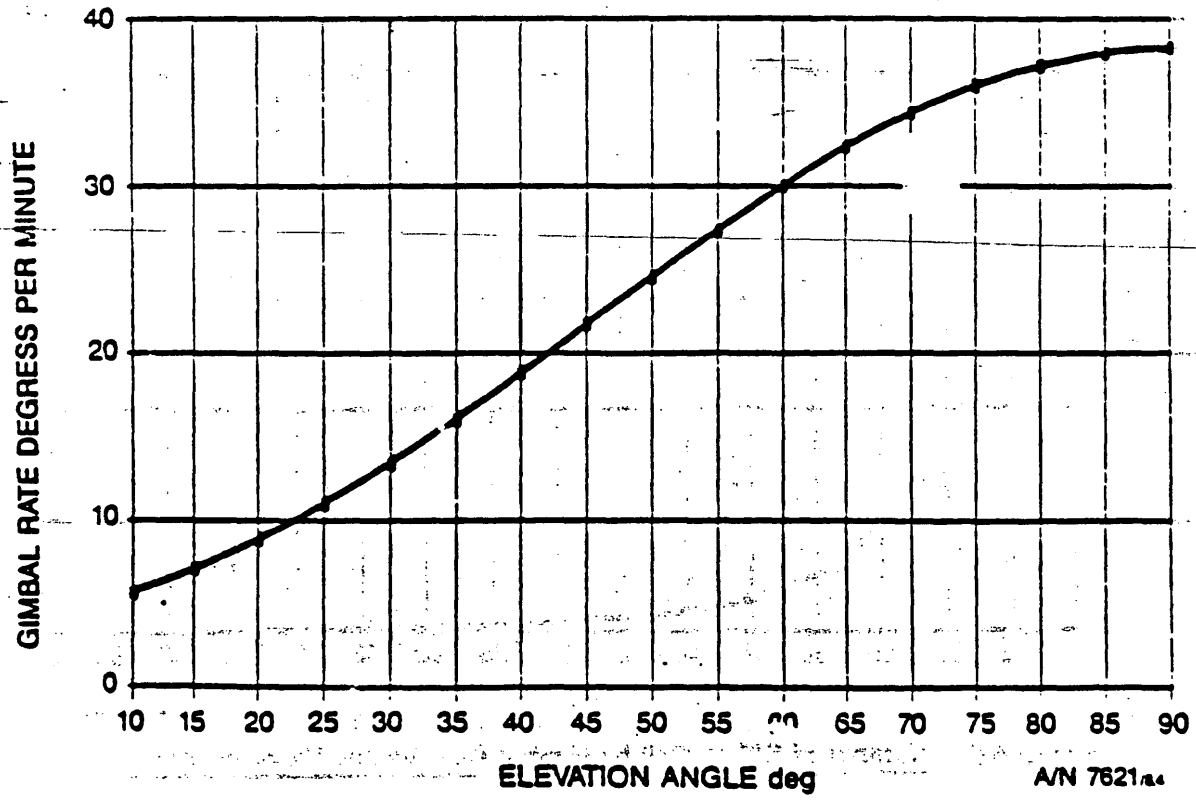



Figure 4-49 Gimbal Rate Versus Elevation Angle

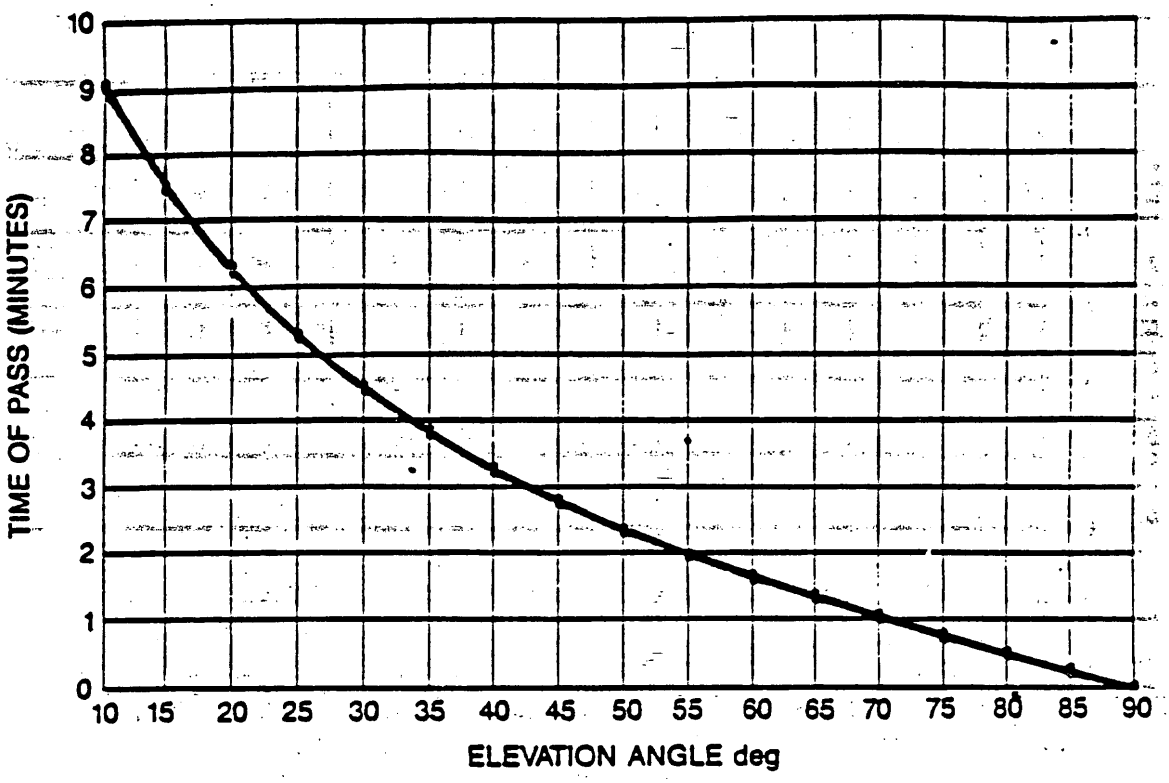



Figure 4-50 Cumulative Time Versus Elevation Angle for Spacecraft Overhead Pass

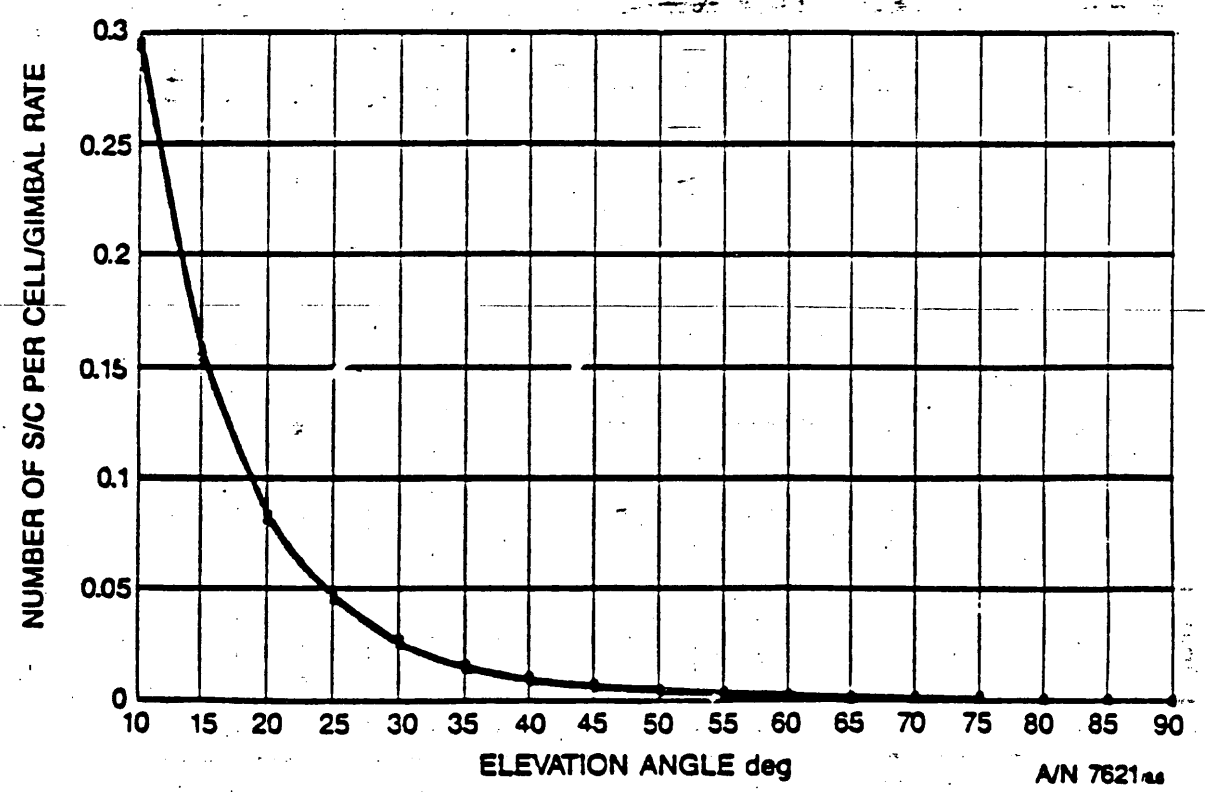



Figure 4-51 Number of S/C to Cell by Gimbal Rate Versus Elevation Angle

ground station, the time for the spacecraft to go from a 55 deg to 90 deg to 55 deg elevation is 2 minutes.

Figure 4-51 shows a ratio of the number of satellites to the gimbal. The the inexpensive ground user system performs best with a toroid-type of patterned antenna which is scanned in azimuth to detect satellites and skip the satellites which are overhead for only a short time.

A conclusion from this data is that the ground user prefers using the low elevation angle spacecraft with the higher quality factor. Therefore, several options exist for the ground user and MSSP:

a) expensive ground unit to track satellites

b) ground user use low elevation angle satellites -more slower moving satellites

c) increase spacecraft power for high elevation angle users

### 4.2.4 ANTENNA GAIN/RF POWER TRADE

The ground user link trade involves the trade between the size and scan rate capability or antenna and the RF power transmitted. The gain of the antenna will increase by almost 3 dB (double) every time the area of the antenna doubles. For small antennas, the impact of doubling the size of the antenna is insignificant. For large antennas, however, increases of 3 dB may require antenna diameter increases in terms of meters. The actual size increase for a 3 dB gain will be more than a factor of 2. The cost of RF amplifiers for increased RF power can be expensive. Also, as the magnitude of the RF power increases the efficiency of the amplifiers will decrease. For the ground user, the larger antenna and gimbal scanning system can be quite costly while the cost of an RF amplifier and power cost is not of great concern. For the satellite, the cost of RF power and larger antennas can both be costly. The optimum trade is a compromise between the RF power and antenna gain of both the satellite and ground user. The third option of increasing the satellite RF power, suggested in section 4.2.3, will require additional satellite power. An increase of the RF power by 10 dB in certain situations could produce a cost saving by increased performance capability for the total system. The link equation of the satellite/ground user is based upon 17 dB antenna gains and 10W of RF power at the 10 deg elevation angle (max path loss). If the transmit power were increased 10 dB (100W), the link equation could be written:

$$G_{sat} + G_{user} + DEL_{pl} + 10\ dB\ (\text{power delta}) = 34\ dB\ (17\ dB + 17\ dB)$$

$$\text{or} \quad G_{sat} + G_{user} + DEL_{pl} = 24\ dB$$

where DElpl = delta path loss, Gsat, Guser = gain satellite and user

Therefore, with higher power, the antenna gain and size of the satellite or ground user could be reduced. If the satellite had 17 dB antenna gain, the ground user could have a wide beam (70 deg beamwidth) 6.5 dB gain antenna to use when talking to high elevation angle (90 to 40 deg) rapidly moving satellites. This antenna would provide a dome of coverage, or a 100 deg cone about the zenith. A high gain toroidal scanned antenna would be needed to talk to the low angle satellites. The satellite and ground user would be required to use high power only when the communicating ground user is viewing the satellite at a high elevation angle.

This type of operation would be an excellent compromise. The rationale is that the ground user is required to have a high gain antenna for multipath reasons at the low angles and therefore, could use high gain rather than high power. Also, the probability of a high elevation angle satellite is low and the time period during which a satellite would be required to transmit high RF power would be short. If a satellite were passing directly overhead, the time to cross the wide beam dome (Figure 4-50) is 3.2 minutes of a 9 minute pass. Most satellite passes will be grazing off the horizon rather than overhead. Whether this user antenna cost savings would offset spacecraft power cost will depend upon the user environment and the projected user antenna cost. The main concern is the total system cost.

Another option might be to lower the nadir gain required for both the satellite and the user. Antenna gains of 14 dB with 30 deg 3 dB beamwidth, which would cover elevation angles +/- 20 deg about the 90 elevation, could be beneficial to both the satellite and ground user. Also, if the RF power outputs were 10 dB higher for all angles, and because the gain of the ground user must increase for the low elevation angles, the gain of the satellite antenna could be reduced to 14 dB. This lower gain, wider beamwidth would allow coarser pointing errors and reduced antenna size. The trade would be the larger size of the power subsystem.

### 4.2.5 CONCLUSION

This analysis was meant to state a few of the ground user concerns and possible trades. Any trade must consider the impacts on the satellite power subsystem and the satellite size. However, when compared to the costing for the ground user, some of these options seem to be of value, particularly the use of higher power for the high angle satellites. The total system cost will be greatly impacted by the large number of ground users. A final decision on the satellite/ground user interface will be made by the MSSP system engineer.

## 4.3 MSSP COMMUNICATION SYSTEM

Establishing a successful communication system for the MSS program will depend on the resolution of factors which will significantly affect the performance. These factors (such as altitude differences, number of spacecraft, pointing errors, cross link range) have been identified and are discussed in this section technically and in terms of cost. In addition, a section of the MSSP link equation has been included to provide trade graphs of the various system parameters.

### 4.3.1 SATELLITE RANGE

SPACECRAFT RANGE VS HEIGHT ABOVE EARTH

To eliminate atmospheric loss and multipath, the MSSP maximum communication range vs. S/C altitude and the minimum height above the Earth must be considered.

As the crosslink range between satellites is increased, a maximum range is obtained due to the occultation or blockage by the Earth. The communication range when communicating directly across the limb of the Earth is:

$$R_{max} = 2\,(h+r)\,\sin(\theta_{max})$$

where $\cos(\theta_{max}) = r/(r+h)$

$r$ = Earth radius (6378 km) $h$ = S/C altitude

For example $h = 675$ km $R_{max} = 6021$ km

This maximum range condition is not, however, the optimum range for a low cost communication system like MSSP. Communication directly across the Earth limb will require that the RF signal traverse the Earth's atmosphere twice. Also, the multipath reflection at the Earth's surface will cause large signal degradation. Therefore, the required signal amplitude will have to increase by several dBs as the communication ranges approach the Earth limb.

Another less costly solution would be to limit the maximum range condition to communicate across an altitude above the Earth's atmosphere or an altitude of 100 km. The maximum communication range would then be:

$$R_{max} = 2\,(h+r)\,\sin(\theta_{max})$$

where $\cos(\theta_{max}) = (r+100\text{km})/(r+h)$

For example $h = 675$ km $R_{max} = 5579$ km

The system designed for the lower range (557km) would be a more cost-effective compromise.

### 4.3.2 625/675 ALTITUDE DIFFERENTIAL

The proposed satellite orbit is 675 km. After being in orbit for several years, the satellites will drift down and new satellites will be placed in orbit at 675 km. The altitude has been established so that the satellite maximum altitude variation will be 675 km to 625 km. This altitude variation will impact the MSSP mission in that true angles between satellites will not be as designed.

To analyze the real pointing angle and range for satellites of different altitudes for comparison with results of satellites flying at the same altitude, assumptions were made that the satellites had no altitude errors, no pointing errors, and fixed beam antennas. Once the impact of the altitude differential for these satellites is known, the analysis can be extrapolated for altitude errors and the non-fixed elevation beam antennas.

#### 4.3.2.1 POINTING AND RANGE ERROR

Previous analysis has limited the angle difference from satellite A to satellite B to 4 deg with an altitude differential of 50 km. The basis for the 4 deg is shown in Figure 4-52, and was based upon an orthogonal intersection of the horizontal and the 50 km altitude difference. This analysis agrees with the 4 deg, but the relative position of the 4 deg is different from Figure 4-52.

Figure 4-53 shows the relationship between antenna pointing angle and the range between the satellites when they are at the same altitude. The pointing angle is the angle between the orthogonal to the A satellite radial and the line to the second satellite B. This is an angle down from horizontal. If the two satellites were not at the same altitude, then the range and angle would be as shown in Figure 4-54. Figure 4-54 shows the satellite A at altitude 625 km and satellite B at 675 km. Since these satellites are at different altitudes, the range and pointing angle will be different for the same "Earth angle" between them. The equations for theta prime and R prime or range prime, the real angle and range between the satellites is described in Figure 4-54. The comparison of the same altitude and 625/675 altitudes must be made for the same design angle. In other words, with the assumptions: no altitude errors or pointing errors, and fixed antenna beams, the system is designed for the same altitudes of all satellites and the impact of the altitude differences is compared at the same design angle or one-half "Earth angle".

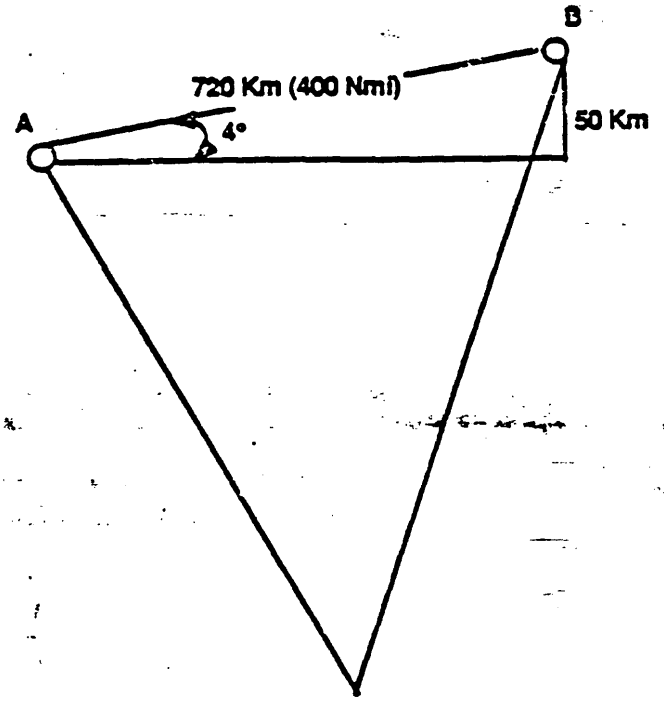



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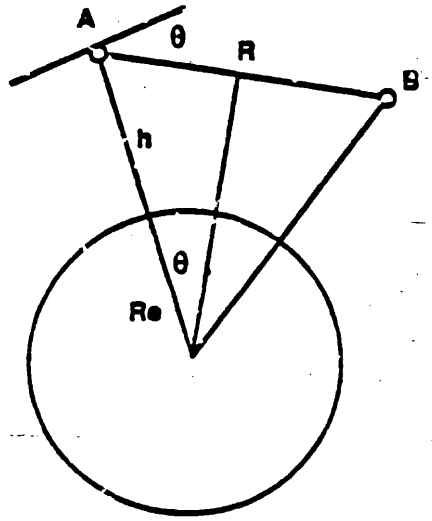
Figure 4-52 4° and 50 km Separation




$$\sin \theta = \frac{R/2}{h + Re}$$

R = RANGE
h = ALTITUDE 675Km
Re = EARTH RADIUS 6378 Km

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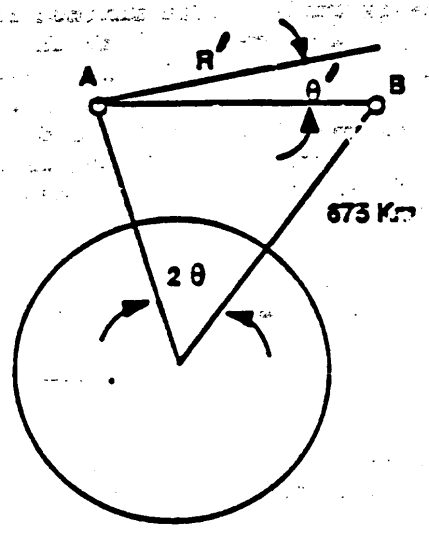
Figure 4-53 Relationship of Antenna Pointing Angle and Satellite Range




$$R'_{Km} = \text{SQRT}\left[(6378 + 675)^2 + (6378 + 625)^2 - 2(6378 + 675) \cdot (6378 + 625) \cdot \cos 2\theta\right]$$

$$\frac{\sin(90 - \theta')}{(6378 + 675)} = \frac{\sin 2\theta}{R'_{Km}}$$

$$\theta' = 90 - \sin^{-1}\left[(6378 + 675)\frac{\sin 2\theta}{R'_{Km}}\right]$$

$R'$ = REAL RANGE $\theta'$ = REAL ANGLE

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Figure 4-54 Pointing Angle and Range with Altitude Difference

Figures 4-55 and 4-56 show the ranges between the satellites for the cases where they are both 675 km and where they are at 625 and 675 km. The differences are very small except for design angles below 1 deg. Even then the difference is small. The maximum range error will be 50 km when the satellites are above one another or when the design pointing angle is 0 deg. Figures 4-57 and 4-58 show the difference between design angle and the real angle due to the satellite altitude separation. For large design angles out 22 deg, the real and design pointing angles are close. However, for angles below 3 to 4 deg, the difference becomes very large until at the design angle of 0 deg the real angle is 90 deg.

The impact of these errors will be a small increase of the link path loss, but the pointing angle difference could cause a large loss due to the antenna gain.

4.3.2.2 IMPACT OF ERRORS

The pointing error difference for the satellites at different altitudes could cause the signal amplitude to be further down on the antenna pattern. To analyze the impact of the altitude difference, a baseline link and antenna must be assumed:

Antenna: uniform illumination with sin (X)/X pattern and beamwidth of 50.8 deg/L; where L is the aperture length in wavelength.

$X = Pi * L \sin(\theta)$; $\theta$ = antenna angle

Antenna pointing at maximum range

Antenna beamwidth = 1.1 times max range 23.1 deg
It will be assumed that the maximum range is 5579 km (angle=21)

With these assumptions, the equation for the link margin with varying range (assuming that the azimuth pattern is narrow enough to provide adequate gain) is:

$$20 \log \left( \frac{5579}{R'} \right) \quad -2 * 20 \log \left( \frac{\sin X}{X} \right) \quad ; \quad X = L \sin(21-\theta)$$

Figure 4-59 shows the path loss (range) change with angle, the antenna gain change, and the total link margin change which is the path loss minus two times the antenna change for two satellites at the same altitude. These curves state that the two factors - antenna gains and path loss - compensate for each other. For this calculation the antenna beamwidth was chosen to be 21 times 1.1 or 23.1 deg. The antenna dimension was therefore, L = 2.19 wavelengths.

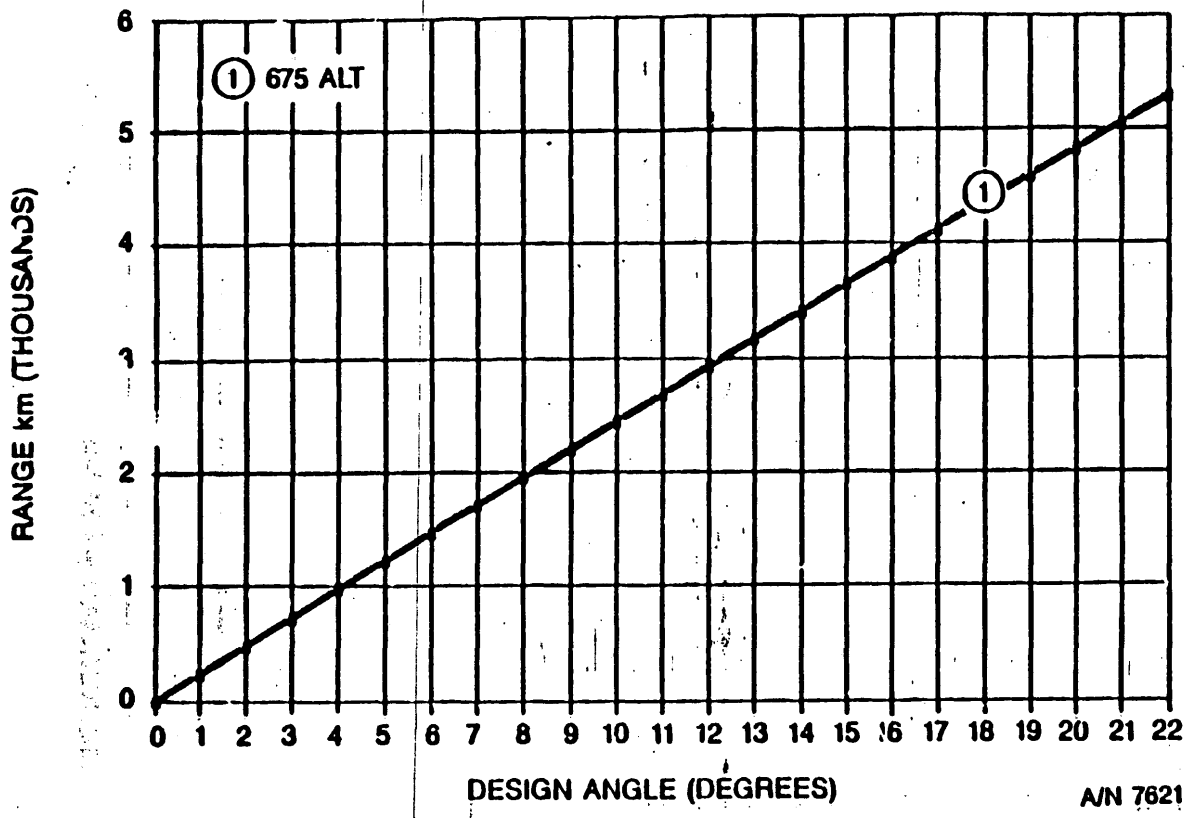


Figure 4-55 Range Versus Design Angle for 675 km Altitude

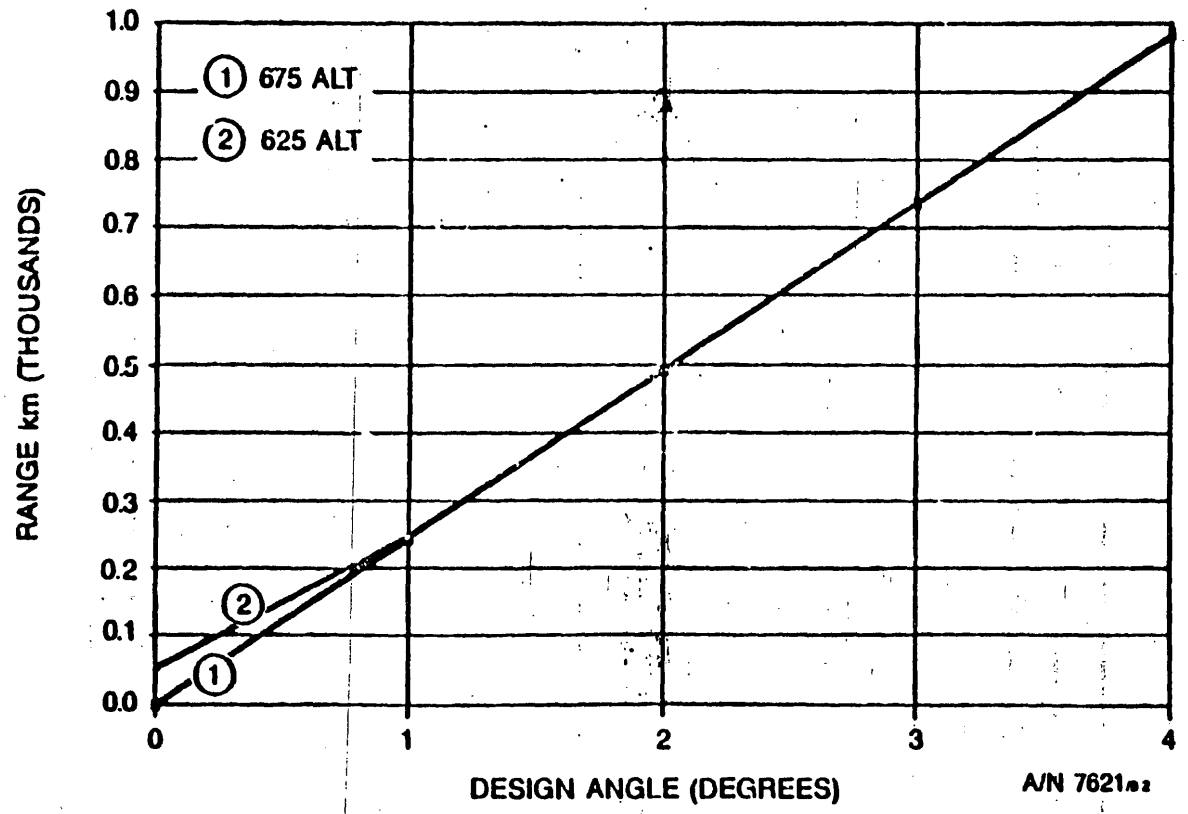


Figure 4-56 Range Versus Design Angle for 675, 625 Altitude

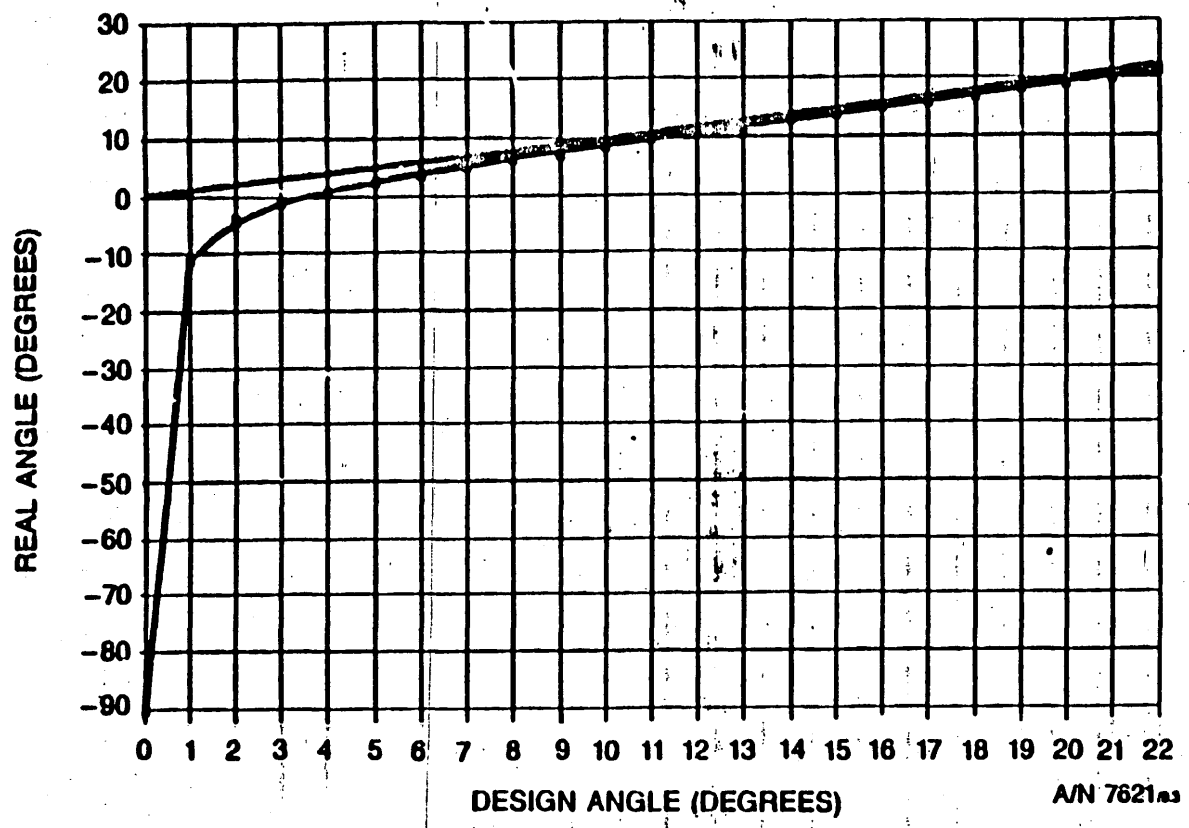


Figure 4-57 Link Real Angle Versus Design Angle Satellite 625/675 km Offset (Design Angle 0-22)

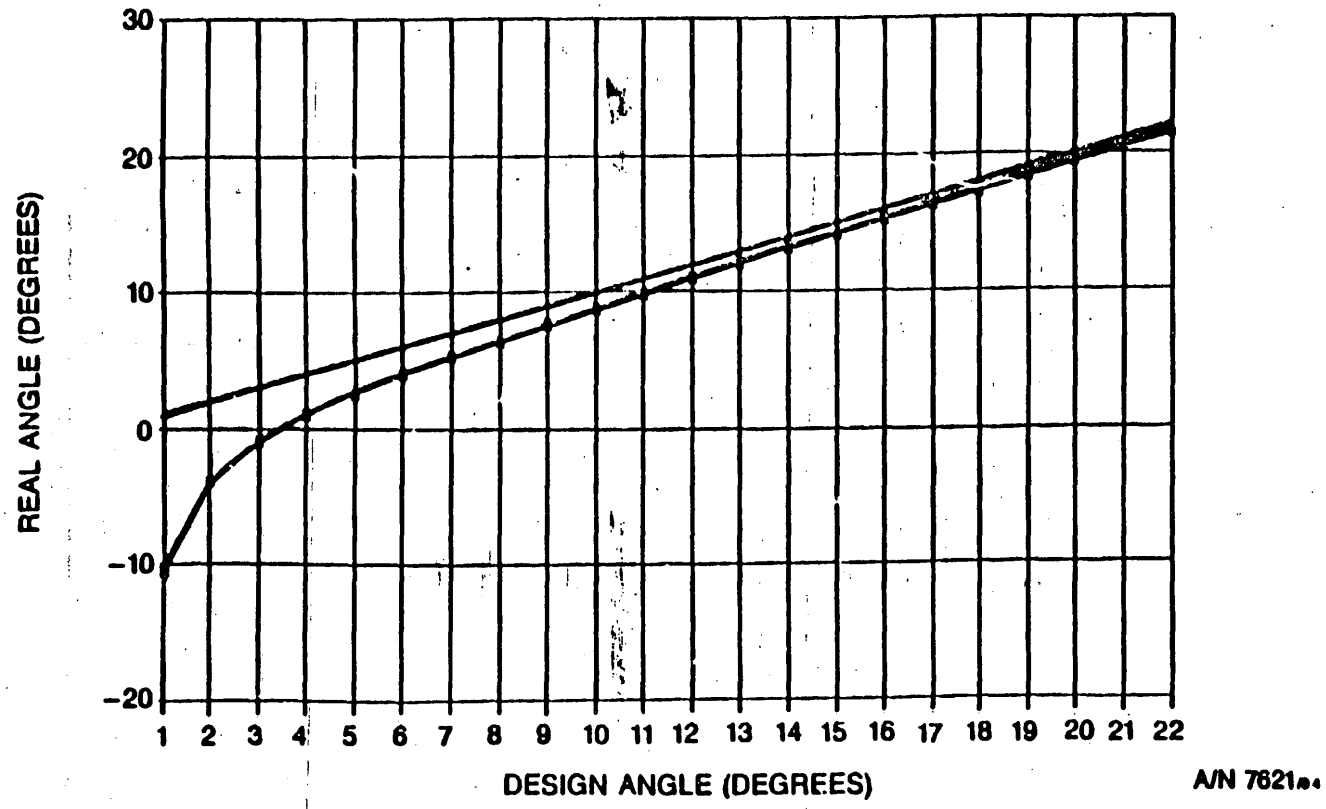


Figure 4-58 Link Real Angle Versus Design Angle Satellite 625/675 km Offset (Design Angle 1-22)

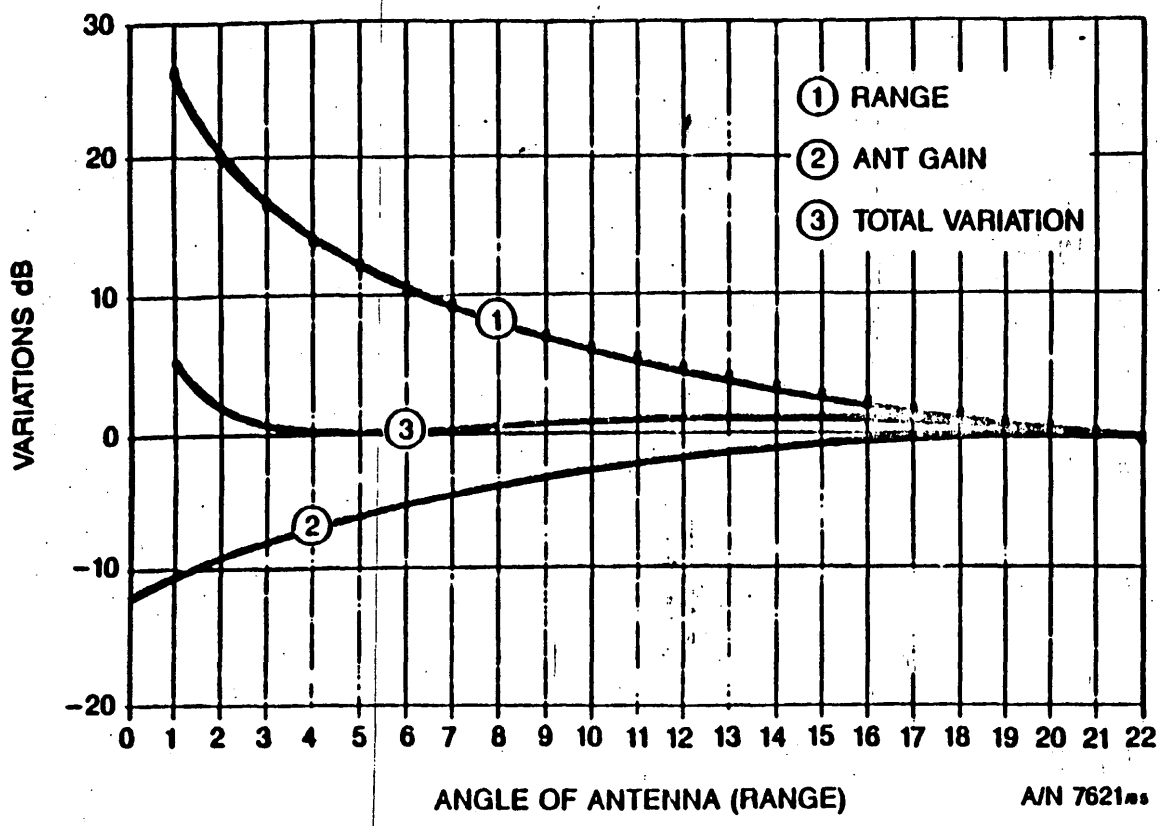


Figure 4-59 Range, Antenna Gain and Total Variation Versus Design Angle (dB of −20 to +30)

Figure 4-60 shows the same combination of path loss, antenna gain, and total variation for two antennas with the real antenna pointing angles for satellites at 625 and 675 km. Note that the antenna gain varies considerably because of the pointing angle. The antenna gain rises at about 1 deg design angle due to the antenna sidelobe. The usable link (>0dB) is limited to ranges greater than 11 deg design angle or 2700 km. This looks bad except for the fact that we have made an error.

The antenna patterns are pointed down at the maximum range of 21 deg (5579 km). Satellite B is up from the design angle for satellite A because of the 625/675 altitudes. Likewise, satellite A will be down from satellite B. Therefore, the real link margin equation should be:

$$20 \log \left(\frac{5579}{R'}\right) - 20 \log \left(\frac{\sin (XA)}{XA}\right) - 20 \log \left(\frac{\sin XB}{XB}\right)$$

Where XA is $L \sin (21-\theta)$

and XB is $L \sin (21-\theta'-(\theta'-\theta))$

$= L \sin (21-2\theta'+\theta)$

The location of satellite B will be farther "down" on the antenna pattern of satellite A because of the altitude difference, but satellite A will be farther "up" on the antenna pattern of satellite B. These two effects will almost compensate for each other. Figure 4-61 shows the link difference for the two look down case and the real or look up/down antenna cases for design angles of 2 to 21 deg. This figure was based on an antenna size of 2.19 wavelength or a beamwidth of 50.8/2.19=23.1 deg. If the beamwidth as expanded to 50.8/2.15 = 23.6 deg (L=2.15 wavelength), the calculated results will be as shown in Figure 4-62, which provides >0dB performance from almost 3 deg (720 km) to 21 deg (5055 km). The 2.19 to 2.15 antenna vertical size (8.8 in. vs. 8.6 in.) would cost the same percentage expansion of the horizontal aperture for equal antenna gain.

Satellite B is located 4 deg up from satellite A at an angle 1 deg above the satellite horizon when the design angle is 3 deg. The angle offset is 4 deg as in Figure 4-52, but 4 deg above the -3 deg.

The antenna beamwidth can be varied (with diminishing returns) to extend the close range limit but the following general conclusions can be made:

1) The altitude difference will cause a loss of gain due to the lower satellite antenna, but the higher satellite antenna will have an increase in gain almost offsetting the loss of gain.

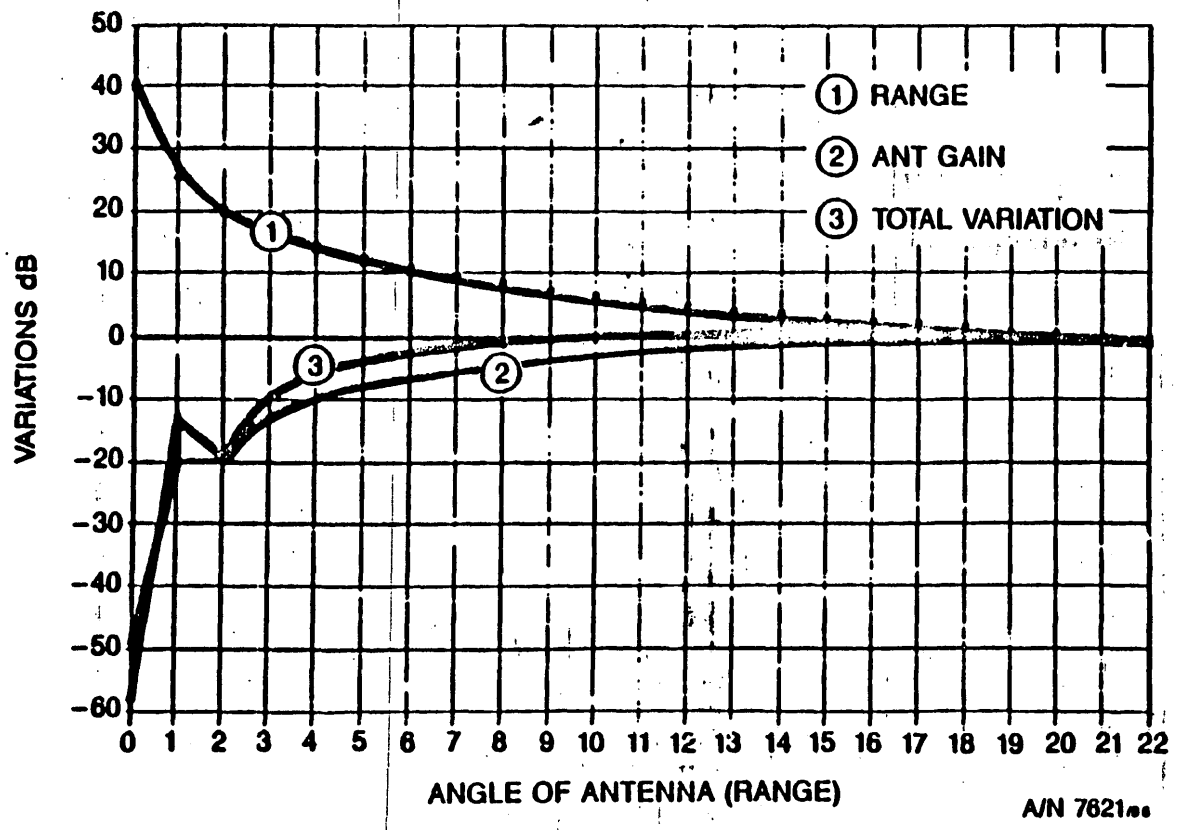


Figure 4-60 Range, Antenna Gain and Total Variation Versus Design Angle (dB of −60 to +50)

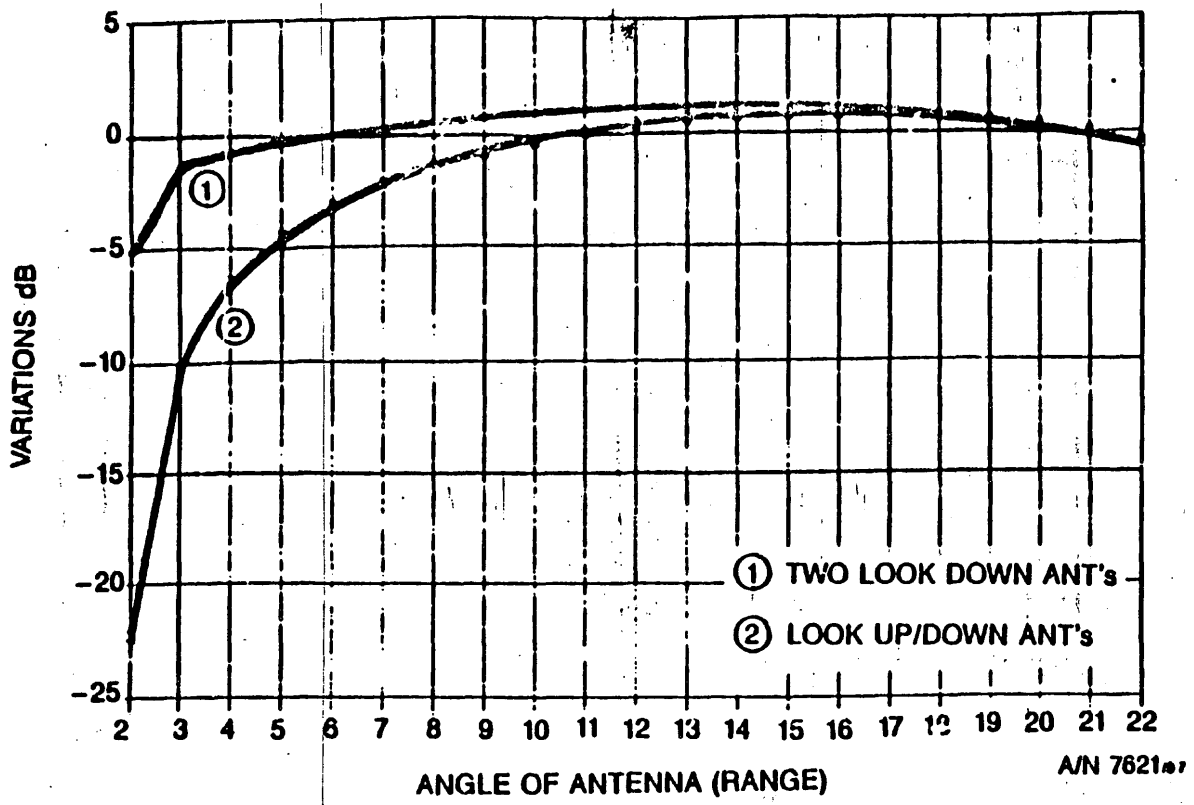


Figure 4-61 Total Link Variation Versus Design Angle for 625 and 675 km (dB of −25 to +5)

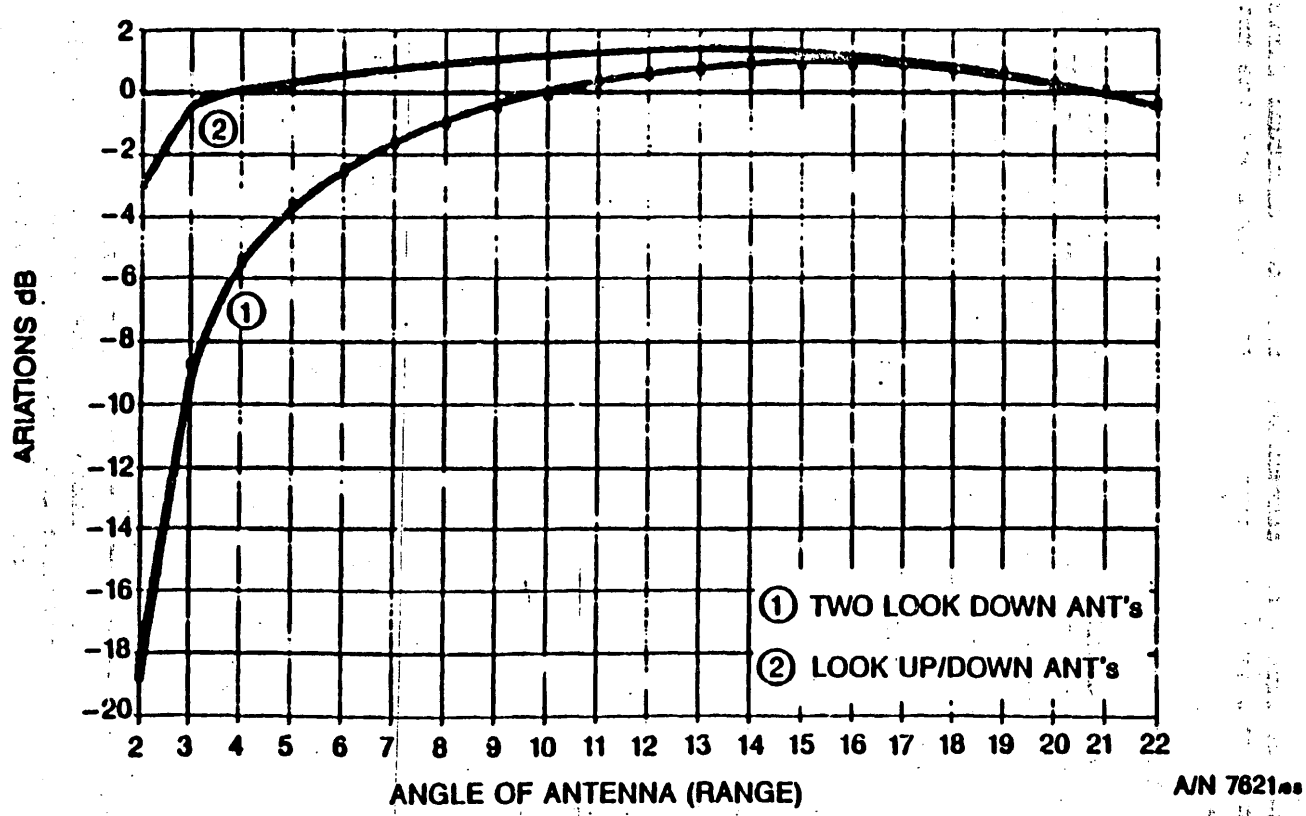


Figure 4-62 Total Link Variation Versus Design Angle for 625 and 675 km (dB of −20 to +2)

2) The altitude difference with the parameters assumed and a 2.15 wavelength size, will cause a loss of communication for angles above 3 deg down or about 750 km.

3) An antenna designed to compensate for the altitude difference would not be required to "scan" up to 4 deg above the horizon, only to 1 deg.

Attitude and Pointing Errors - Field of View (FOV)

If the effects of antenna pointing error are now included, whether caused by alignment or attitude error, the required antenna FOV can be defined. If the pointing errors are zero or small, of course, the FOV is not really defined. If the attitude error is 10 deg and if the antenna pointing error is 2 deg, then the antenna FOV required would be +/-12 deg or 24 deg about 0-21 deg (max range).

FOV = 24 deg about $\theta$max for max range

Side Note

If peak power is used by the system for long range cases, the antenna pointing and beamwidth will be modified but field of view and look angle analysis will be the same.

4.3.3 MAXIMUM CROSS LINK RANGE

During the initial analyses of the MSSP - Phase I, the communication range capability of satellite vs. the dc power requirements were analyzed. This analysis showed that the shorter ranges were more cost effective from a dc power perspective. However, when the impact of the probability of communication is considered, the maximum range capability can be seen in a different light.

Increasing the cross link range to the maximum range capability of 5579 km will result in significantly higher probabilities of useful communication experimentation in the prototype phase. During the operational phase, the total number of spacecraft required will be reduced, thereby reducing the overall system cost. This is true even if the cost per spacecraft on orbit increases due to increasing the maximum range from 2225 km to 5579 km.

The initial space demonstration of MSSP will consist of 10-20 prototype spacecraft that are placed into 675 km circular orbits by a single launch vehicle. This limited number of spacecraft cannot support an around-the-clock global communications system. However, it can provide several hours each day of communications service between North America and Northern Europe and between North America and the Western Pacific region. It can also provide up to four hours a day of regional

communication within Northern Europe. The primary purpose of the demonstration phase will be to provide on-orbit verification of most MSSP technologies. However, a complete demonstration of the network routing and ephemeris tracking technologies will not be achieved until several orbit planes have been populated.

The initial prototype demonstration can also serve as the first of three orbit planes that will form the backbone of an around-the-clock communications system. These three planes would all be at 57 deg nominal inclination with their ascending nodes separated by 120 deg of longitude. This inclination has been chosen because it provides the best North Atlantic coverage available from an ETR STS launch. Each spacecraft will be injected at a slightly different inclination and velocity so that they will continue to spread out in longitude of ascending node and true anomaly. An around-the-clock communications capability will be available in discrete latitude bands immediately after the spacecraft in the third launch have spread out 360 deg in true anomaly but before they have spread out significantly in nodal separation. For ground stations with a 5 deg elevation limit these initial latitude bands are 36.3 deg to 49.2 deg. The latitude bands are reduced to 39.9 deg to 45.6 deg for ground stations with a 10 deg elevation limit. Ground stations at other latitudes will have periodic outages at either 3 or 6 times per day until nodal spreading fills in the gaps. This will take about one year, depending upon the difference in inclination given to each spacecraft.

This around-the-clock backbone system will be sparsely populated with only 30 to 60 spacecraft. Nevertheless, this will be sufficient to provide a basic communications network after nodal spreading. This initial system will have short interruptions in availability due to the random phasing of the orbits. However, uniform coverage should be considered before examining the probabilistic factors. Figure 4-63 shows the number of spacecraft as a function of intersatellite range that are required to uniformly cover the globe. The equation for this curve is:

$$N = \frac{2/3}{\left[\sin^{-1}\left(\frac{X}{2(R_e+h)}\right)\right]^2}$$

Where X is the intersatellite range, Re = 6378 km is the radius of the Earth and h = 675 km is the orbital altitude.

The angle given by the inverse sine is expressed in radians. This equation is based upon an assumed uniform distribution of the spacecraft in a hexagonal pattern; each spacecraft has six nearest neighbors at a distance X. The results obtained with the assumed hexagonal pattern do not differ

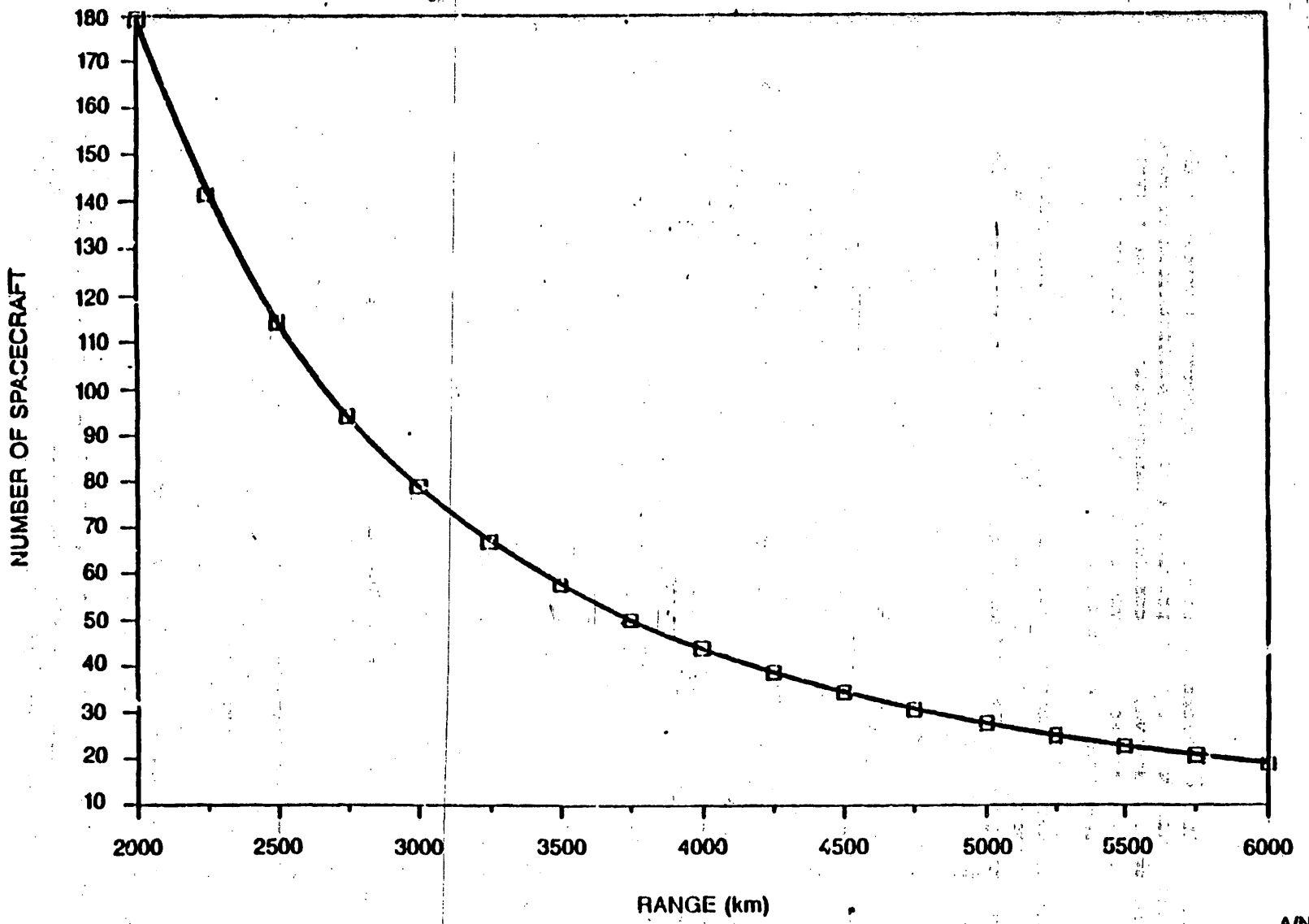


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Figure 4-63 Number of Spacecraft Versus Range for Uniform Global Distribution

appreciably from those obtained with other simple patterns such as underlapping and overlapping circles.

One of the largest multiplicative factors involved in selecting the required number of spacecraft for the constellation is the one due to random orbit phasing in true anomaly. The following approximate analysis was performed as a first cut at evaluating this effect. Consider M spacecraft in a single orbit plane that are distributed around the orbit with a uniform probability density of their true anomalies. This approximately characterizes the prototype demonstration system after some reasonable spreading in true anomaly but prior to significant nodal spreading. The probability that a particular spacecraft will be within range of a specific second spacecraft in front of it is:

$$p = \frac{1}{\pi} \sin^{-1} \left( \frac{X}{2(R_e+h)} \right)$$

where the notation is the same as in equation (1). The probability that the particular spacecraft will be within range of a specific second spacecraft that is either in front or in back of it is 2p. There are (M-1) other spacecraft to consider. Therefore, the probability of g at least one other spacecraft that is within range ahead of a particular spacecraft is

$$P_1 = 1-(1-p)^{(M-1)}$$

Similarly, the probability that there is at least one spacecraft within range ahead of and at least one spacecraft within range behind a particular spacecraft is:

$$P_2 = 1-2(1-p)^{(M-1)} + (1-2p)^{(M-1)}$$

The probability P1 approximately describes the likelihood of obtaining at least one cross link connection in the direction of a desired ground station (e.g. the probability of communicating between Northern Europe and the Persian Gulf). The probability P2 approximately describes the likelihood of having at least one cross link in each direction (e.g. the probability of Northern Europe being able to communicate simultaneously with both North America and the Persian Gulf, or alternatively of North America being able to communicate over a 3 spacecraft link with the Persian Gulf). The above interpretations are approximate because P1 and P2 refer to any spacecraft within range whereas an actual communications link depends upon the second spacecraft being at least a certain minimum range from the first.

The probabilities P1 and P2 are plotted in Figure 4-64 as a function of the number of spacecraft in the orbit plane for two maximum ranges, 2225 km and

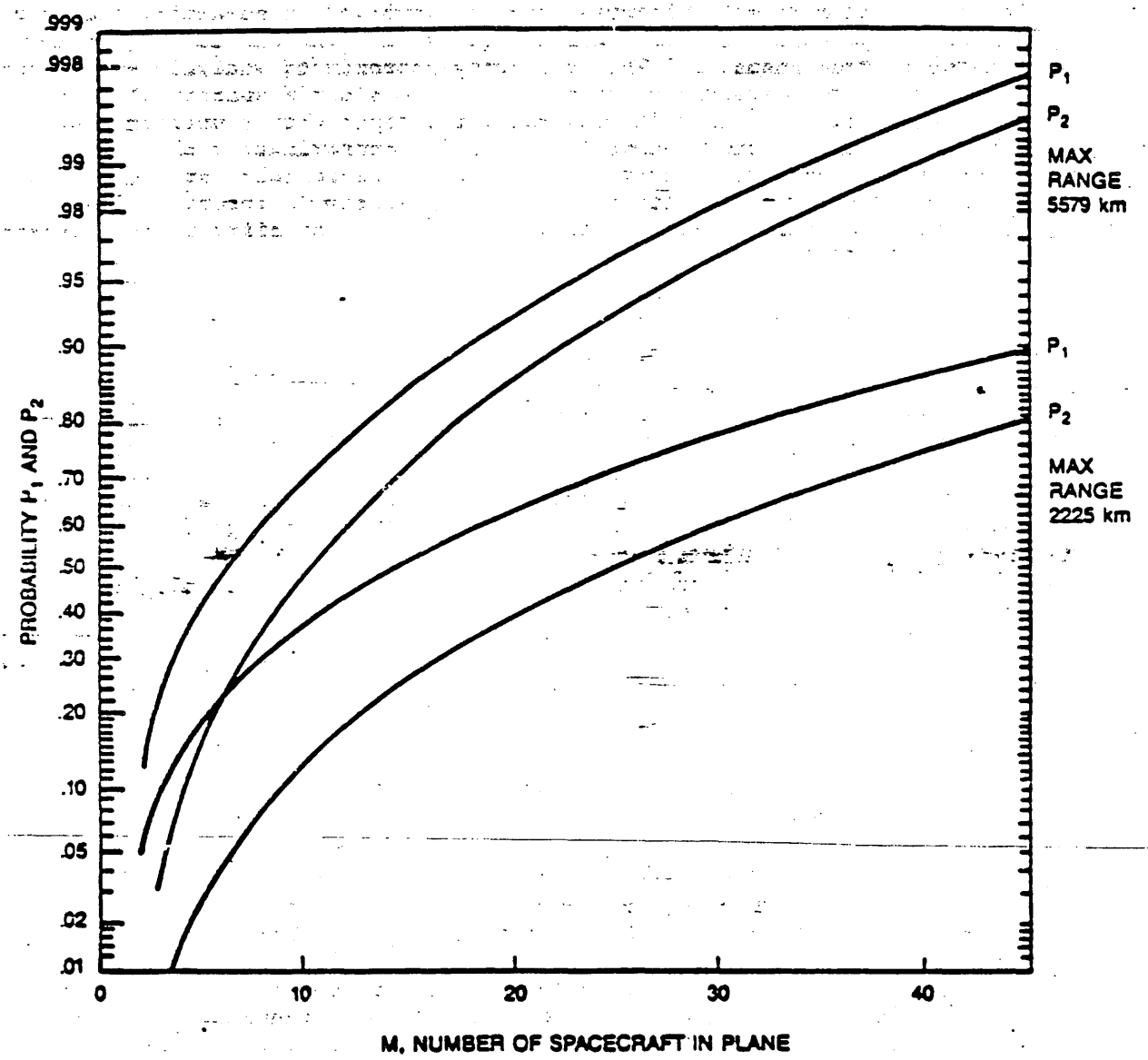


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Figure 4-64 Probability of Link to Other Spacescraft in One Direction ($P_1$) or Two Directions ($P_2$)

5579 km. Naturally the probabilities for the shorter range version are much lower and rise more slowly with an increasing number of spacecraft than do the probabilities for the longer range version. The primary purpose for deploying the prototype spacecraft is to provide on-orbit verification of the MSSP design by performing communications experiments. These experiments have not yet been defined, but it is likely that the low probabilities (P1 = 0.372, P2 - 0.129) of the shorter range version with 10 spacecraft on-orbit would be marginal. The large increase in the probabilities (P1 = 0.713, P2 = 0.493) for the 5579 km range version with 10 spacecraft on-orbit would greatly add to experiment flexibility and to the performance of the evolving communications network as more orbit planes are launched.

### 4.3.4 LINK EQUATION

This analysis was performed early in the Phase I study to identify communication with trade concerns. To accomplish the MSSP primary purpose which is establishing a successful working communication link, several design and system trades must be done to accomplish it at low cost.

The system parameters follow:

Transmit power: Drives the spacecraft size and weight cost.

Range: Affects the transmit power required, the system time delays, and the number of spacecraft necessary to complete the system.

Antenna Gain: Affects the transmit power required and the physical size of the spacecraft.

Link Margin: Adds confidence to the system capability. This margin ensures that as system components age, the system will still perform. However, this pad or excess baggage will require higher performance of the system and create higher cost.

Data Rate: Directly affects the required system transmit power required.

These parameters can be combined in a single communication link equation:

$$Pr = PtGtGr\ (\lambda/4PiR)^2$$

where:
- Pr = required received signal
- Gt,Gr = transmit and receive antenna gains
- $\lambda$ = frequency (wavelength)
- R = range or distance between satellites

The term Pr contains all of the parameters relevant to the receiver design. Many of the parameters will be dependent upon the type of communications mode used. None of the actual parameters and values are fixed yet. However, an estimate can be made which allows an analysis of the MSSP link equation and a first cut at the compromises which much be made. The equation for Pr is:

$$Pr = KT + Eb/No + NF + Loss + 10 \log (D) + Margin$$

where: KT = noise floor, 270 K = -204.3 dBW

EB/No = required signal to noise level (MA Com study) = 6.7 dB

NF = receiver noise figure (estimates) = 3.0 dB

Loss = receiver detection loss and signal to data power = 2.0 dB

D = data rate

Margin = signal level "pad" = 3.0 dB

-189.6 dBw

or, Pr = -189.6 dBW + 10 log (D)

Before proceeding any farther, additional assumptions must be made:

Gr = Gt; antenna gain transmit equal antenna gain receive

D: data rate = 1, 5, or 12.5 Mbit/sec. This assumption will provide a baseline to see the effects of data rate.

Frequency = 4 GHz

With these assumptions, the link equation can be rewritten:

$$Pr = -189.6 + 10 \log (D) = 10 \log (Pt) + 2G - 20 \log[R(Km)] - 104.5$$

$$\text{or, } 10 \log (d) = 10 \log (Pt) + 2G - 20 \log [R(Km)] + 85.1$$

With a low-cost single antenna beam system, range and antenna gain refer to the maximum range and gain. The antenna gain and the range loss will compensate at close ranges. Therefore, the term 2G - 20 log [R(Km)] is a constant. Or,

$$10 \log (D) = 10 \log (Pt) + K + 85.1$$

where K = 2G peak - 20 log [Rmax(Km)]

Figures 4-65 and 4-66 can be used to determine the relation of the link parameters. In Figure 4-65, the data rate and RF power are selected and the value K is determined. In Figure 4-67, with the determined value for K, the antenna gain is selected and the system range capability is determined. Figures 4-68 and 4-69 illustrate the system range vs. the RF power for selected data rates and antenna gains.

No hard conclusions can be ascertained from this analysis. However, several tentative conclusions can be drawn.

1. Antenna gain: Should be as high as possible and at least 15 dB.

2. Data Rate: Should be compromised to at least less than 10 Mbit.

3. Range: The system range is driven by the need to provide complete system coverage and the number of satellites. However, because of restrictions due to Earth reflections, the range might be restricted to 3000 to 5579 Km.

4. RF Power: With the above compromises, the RF power required can be lowered to less than 100W and peak power averaging used for long ranges.

4.3.5 CONCLUSION

The analyses performed in this section addressed orbital concerns of spacecraft communication. They are not all directly radio, antenna, or satellite concerns but focus on the total satellite communication system. The results of the analyses were:

a) Limit the maximum communication range to 5579 km such that the communication path is above atmosphere. Attempting to communicate across the limb of the Earth will encounter the problems of atmospheric loss and multipath.

b) The satellite altitude differentials of 625 and 675 km will require the antennas to be able to scan up to 1 deg above the horizontal for a 400 km minimum range.

c) The probability of viewing another spacecraft is increased greatly when the system is designed for maximum range communication capability; especially, during the prototype satellite phase.

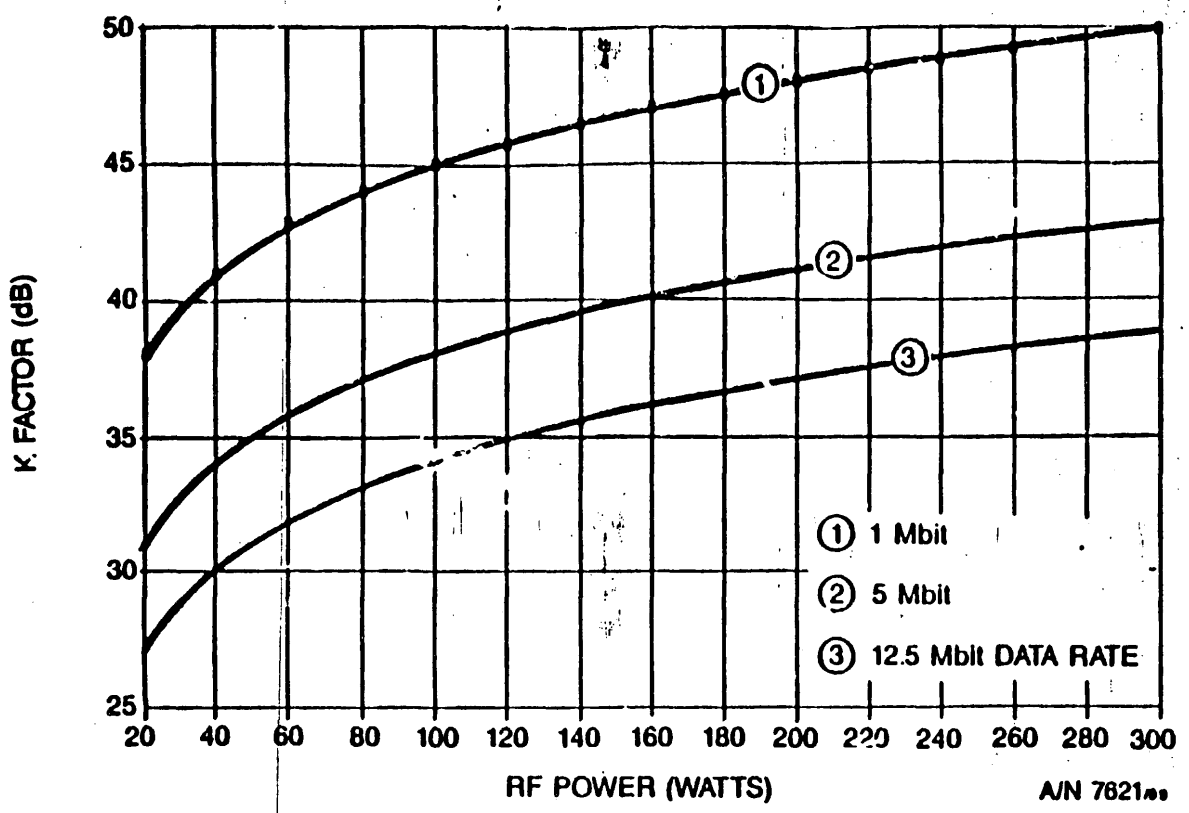


Figure 4-65 K Factor Versus RF Power for Various Data Rates

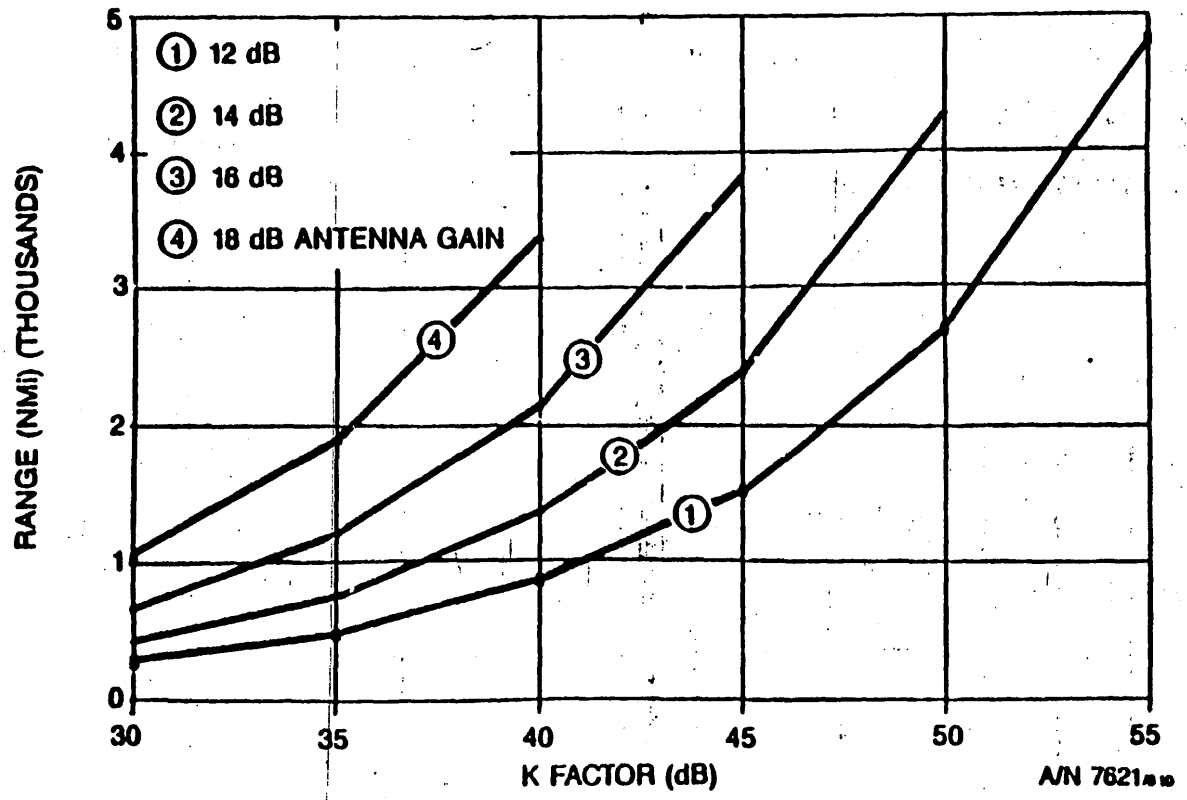


Figure 4-66 Range (NMi) Versus K Factor for Various Antenna Gains

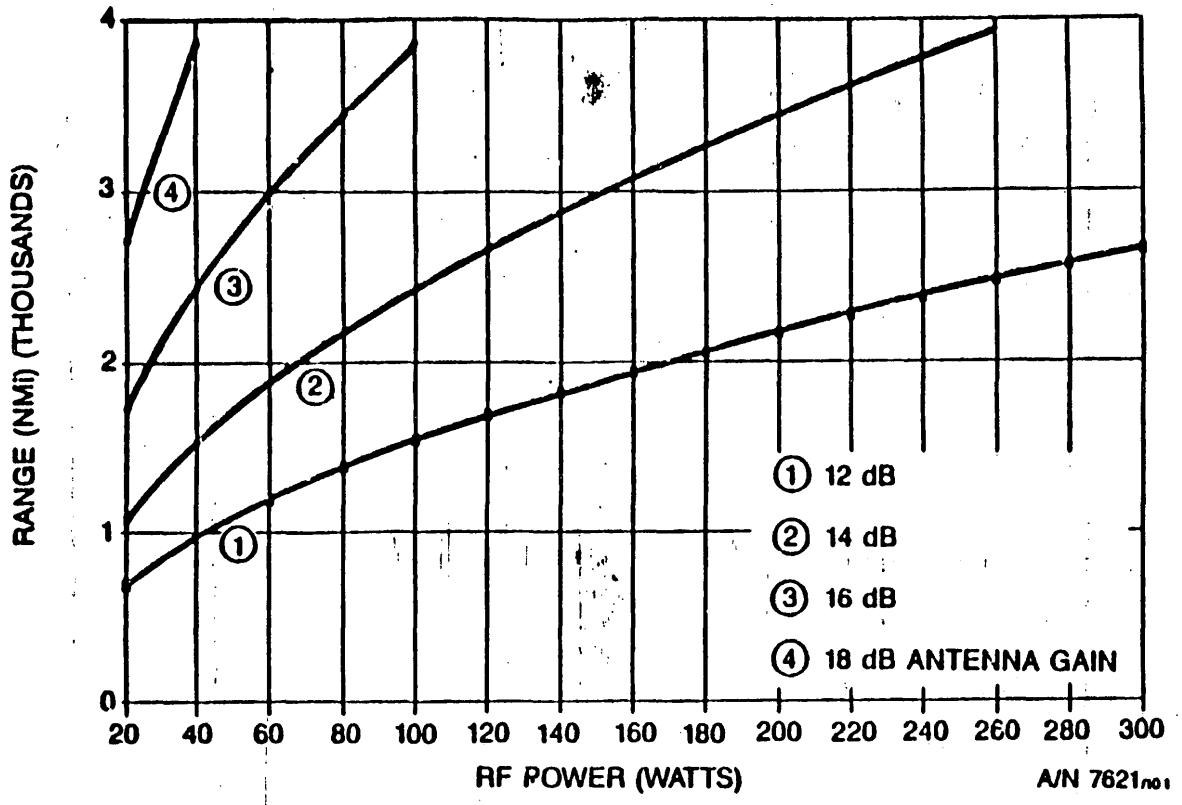


Figure 4-67 Range (NMi) Versus RF Power for 1 Mbit Data Rate

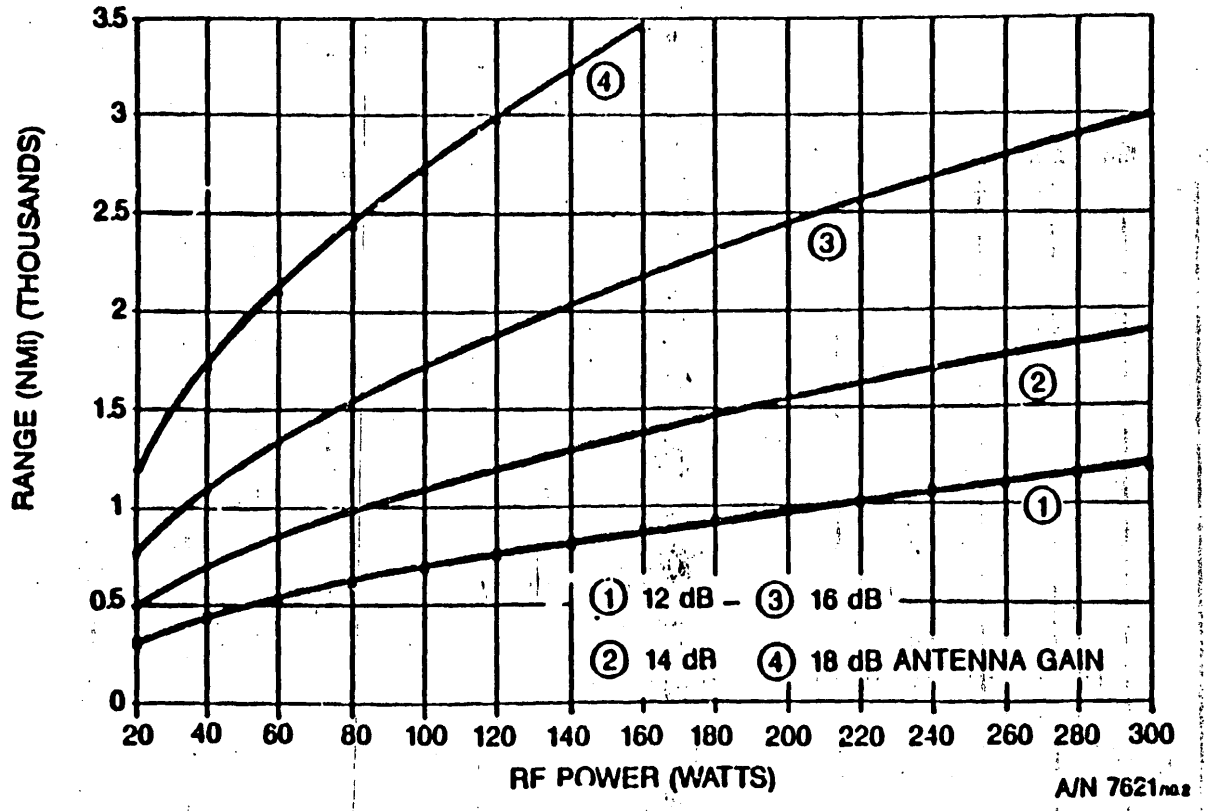


Figure 4-68 Range (NMI) Versus RF Power for 5 Mbit Data Rate

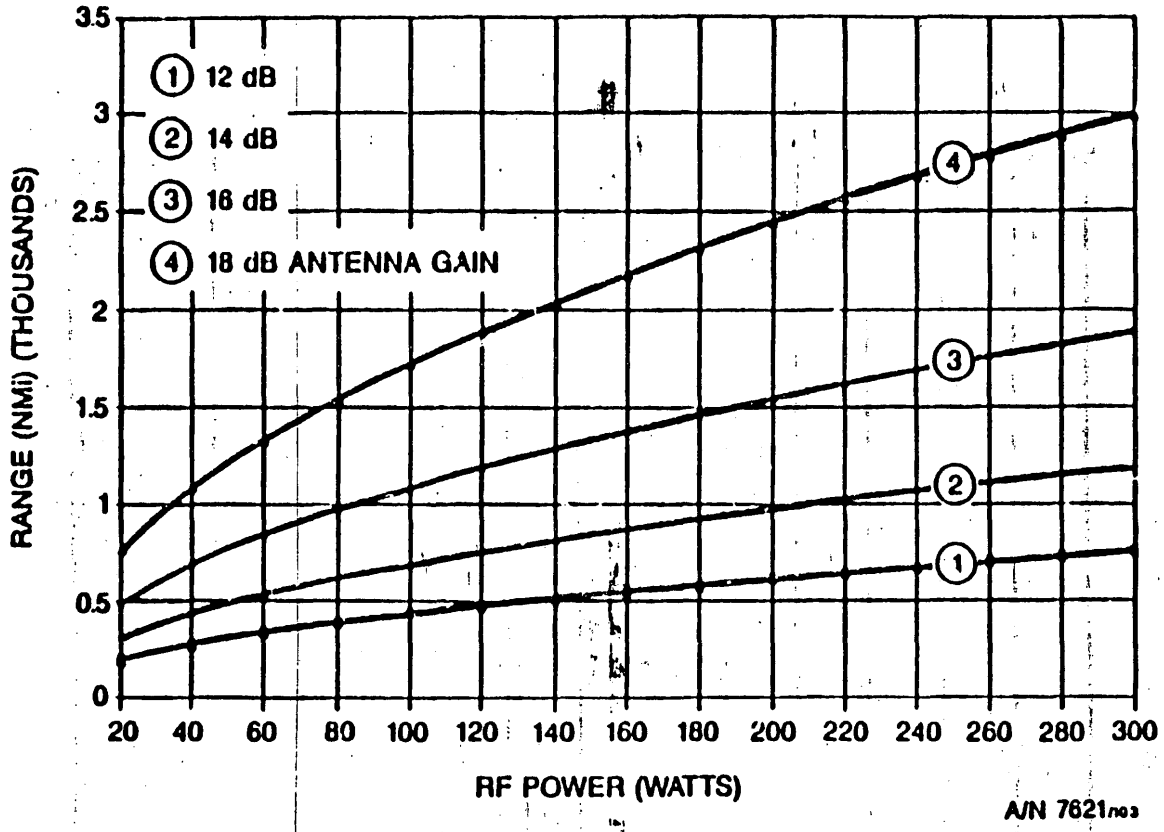


Figure 4-69 Range (NMi) Versus RF Power for 12.5 Mbit Data Rate

## 5.0 SATELLITE INTEGRATOR SUBSYSTEM TRADES

The satellite integrator subsystem trades are the areas of design study were the satellite power, attitude control, and configuration, and the satellite/antenna and satellite/radio interfaces. Communication between team members resulted in reexamination of several areas of analysis. This section will discuss the analysis based upon the final requirements from the third working group meeting (November 1986). Some parameters, however, such as satellite bus power, will be presented as trades to show the impact of the variation of the requirements. Making some of the final system trades easier to make in Phase II is the purpose of this information.

### 5.1 SATELLITE POWER

The satellite power subsystem must generate, store, control and distribute electrical energy necessary for the satellite, antenna, and radio operation. The main components of the power subsystem are:

| | |
|---|---|
| Solar Arrays | to generate energy |
| Batteries | to store energy |
| Power Controller | to control energy storage |
| DC-DC Convertor | to supply specified voltages |

To meet the MSSP criterion of low cost, the power subsystem design must provide efficient use of all of the components. The solar array mechanical configuration must maximize the energy output from a non-orientated satellite such that the amount of array surface can be minimized. Besides the cost consideration of the subsystem components, the size and weight of the components must be factored into the cost equation. The power subsystem will be the largest and heaviest element of the satellite bus. This subsystem, therefore, will increase the cost of the satellite launch and the number of satellites which can be dispersed per launch.

The MSSP Phase I power requirements listed in Table 5-1, were modified as the study progressed with communication between the team members. Initially the average power requirement was 35W, then 50W, and finally 75W. Therefore, various analyses will show calculations at differing average power levels. The analyses were performed so that final conclusions could be modified for the final requirements and be applied to any future trades in MSSP Phase II. The analyses to be presented in this section are:

* Power Cost analysis which presents an overview of the cost drivers and the impact of the power subsystem requirements

TABLE 5-1 POWER SUBSYSTEM DESIGN REQUIREMENTS

Low Cost

Adequate Orbital Average Power

Power Regulation

Adequate Lifetime

Minimal Subsystem Size

| DRIVERS | PUSHERS | OPTIMUM |
|---|---|---|
| Battery Failure | Battery Temperature<br>O/V Charge | 10 DEG C<br>Effect on Temp |
| Battery Life | Depth of Discharge | 80 - 90 percent |
| Battery Size/Weight | Battery Depth of Discharge | 80 - 90 percent |
| Solar Array Size | Power Requirement<br>Power Control | TBD<br>TBD |

* Array optimization analysis which presents the impact of the solar array configuration

* Battery sizing analysis
* Solar array trade analysis which presents a look at an unusual method of reducing the size of the solar array

Finally, a design of the power subsystem will be described for the 75W average power system.

The final conclusion for the power cost on a low cost satellite such as MSSP is $5000 per watt at the spacecraft level and $7000 per watt on orbit. The largest cost drivers are the solar arrays and the launch cost for the arrays.

### 5.1.1 POWER SUBSYSTEM COST ANALYSIS

This preliminary power system analysis will attempt to provide a rough rather than absolute analysis of size, weight, and cost for the power system. The purpose is to show the general impact of power system requirements on size, weight, and cost. Since the Satellite attitude control system was not defined, an omni solar array was assumed for this preliminary analysis.

The following assumptions, which are stated at the beginning of each analysis, were made.

* solar array: four panels spaced at 90 deg around the satellite. The panels are tilted to try to obtain uniform solar coverage (omni array)

* solar cells on both sides of panels

* projected area of solar panels assumed to be 2 panels orbital average

* the projected solar area peak area is 2.4 panels

* solar cells generate 10W per sq ft

* orbit time 96 min: 32 shade, 64 sun

* voltage at max power: 33.4V

These assumptions, while providing a somewhat broad baseline for the power subsystem, derive the subsystem cost drivers and the impact of the subsystem requirements.

### 5.1.1.1 SOLAR ARRAY SIZING

The size of the solar arrays needs to be sufficient to produce an orbital average output equal to the average system load plus a factor for the battery efficiency.

AVERAGE ARRAY OUTPUT = 1.1 * ORBITAL AVERAGE LOAD

The power output of the four panel system will be the product of the array size, output rate, and sun/shade orbit period.

average projected array area = 2 * area (ft^2) per panel

output rate = 10W/ft^2 (conservative number)

sun/shade time = 64/96

POWER OUTPUT = 2 * 10W/ft^2 * 64/96 / 1.1 = 12.11W/ft^2/panel

This number is the array orbital average output for the four panel system. In other words, if an orbital output of 12.11W is needed, the system would use four (4) one-foot square panels. Figure 5-1 shows the required size of one of the four panels vs. average required powers from 20 to 200W. With an array weight of 0.5 kg per sq ft and array cost of $6300 per sq ft (array on both sides), Figures 5-2 and 5-3 show the weight and cost impact of increasingly larger system power requirements.

CAVEAT: The pricing number used in this exercise is based upon recent proposals. Although reasonably priced, this cost does not include any estimate for high production. A closer cost examination will be done later in this analysis. The array weight number used is linear for increasing array. Actually, for larger solar arrays, the weight (and cost) would increase with more complicated structures and deployment systems.

### 5.1.1.2 BATTERY SIZING

The size and cost of the batteries required are also a function of the system's power requirements. For a low cost system, the first approach would be to size the batteries for current-limiting under the high array output condition. If the batteries are sized (amp/hr) so the maximum array output will not overheat the batteries, then a costly power control system would not be needed. The trade is the cost and weight of the additional batteries on the spacecraft.

The equation for the sizing of the batteries is:

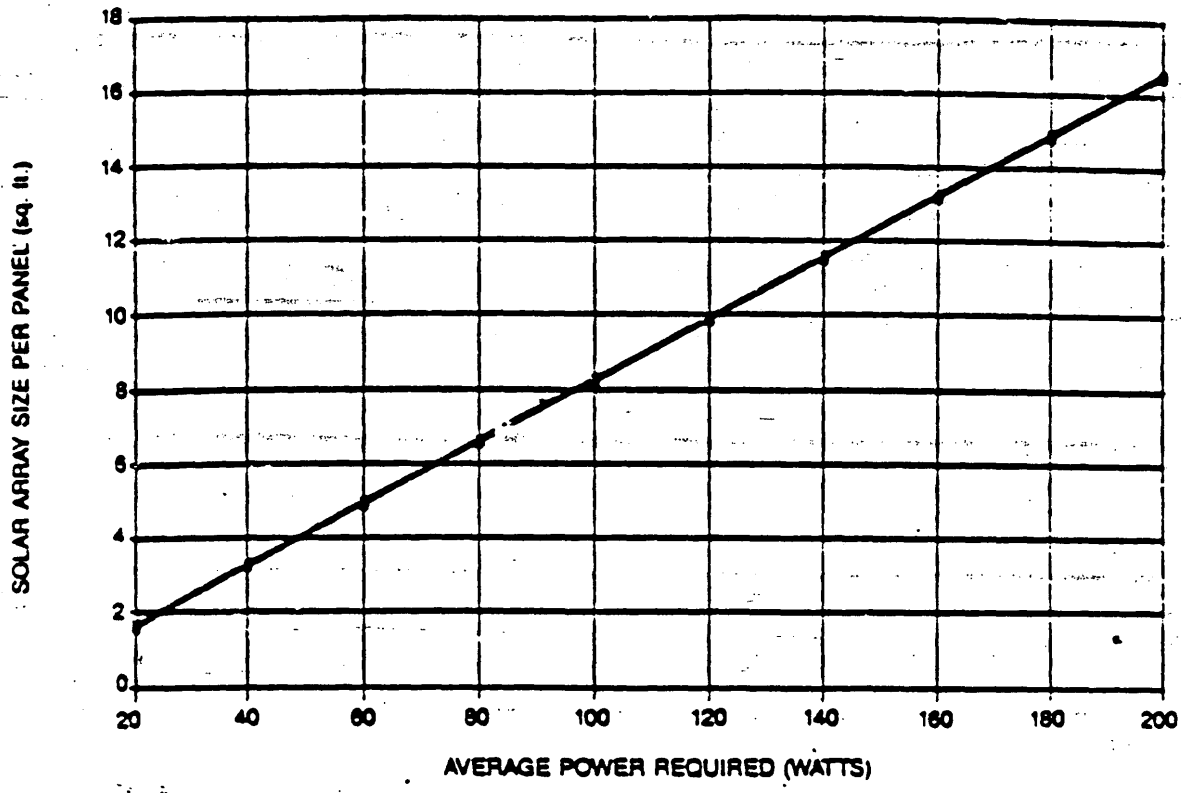


Figure 5-1 Solar Array Size (sq. ft.) Versus Average Power Required

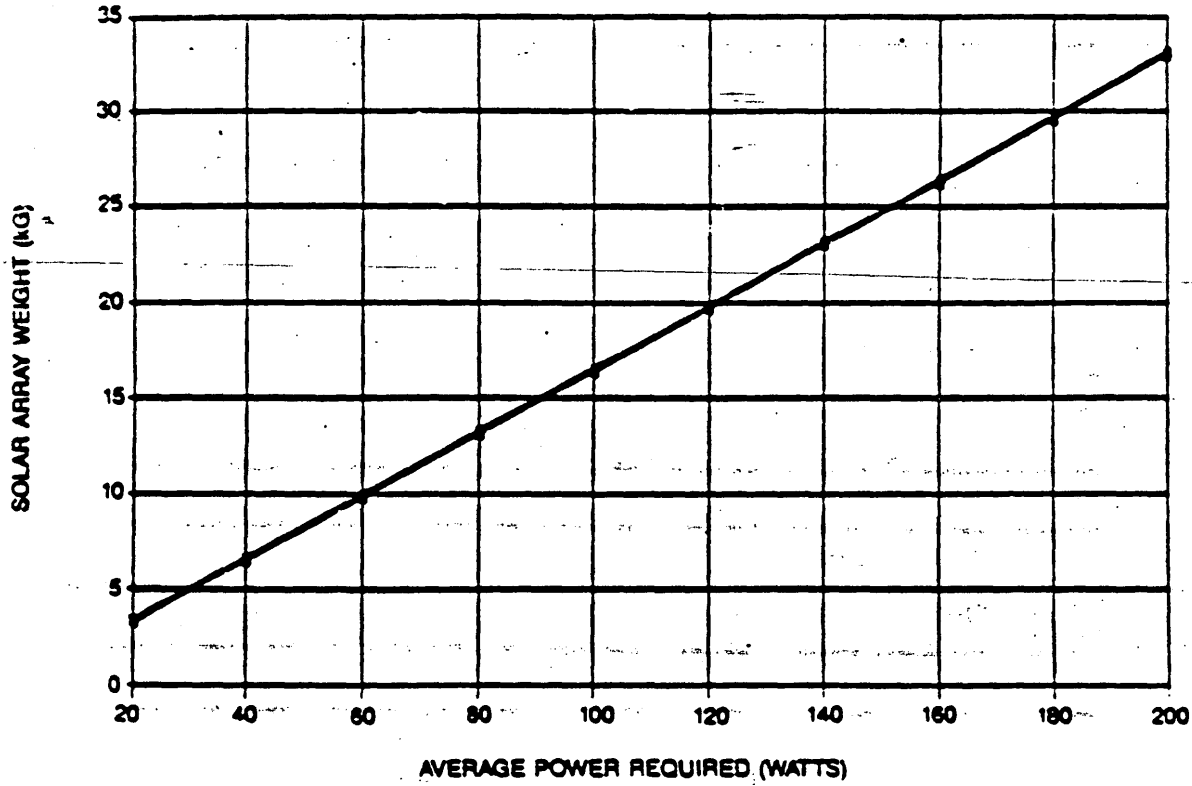


Figure 5-2 Solar Array Weight (kG) Versus Average Power Required



A/N 7621n

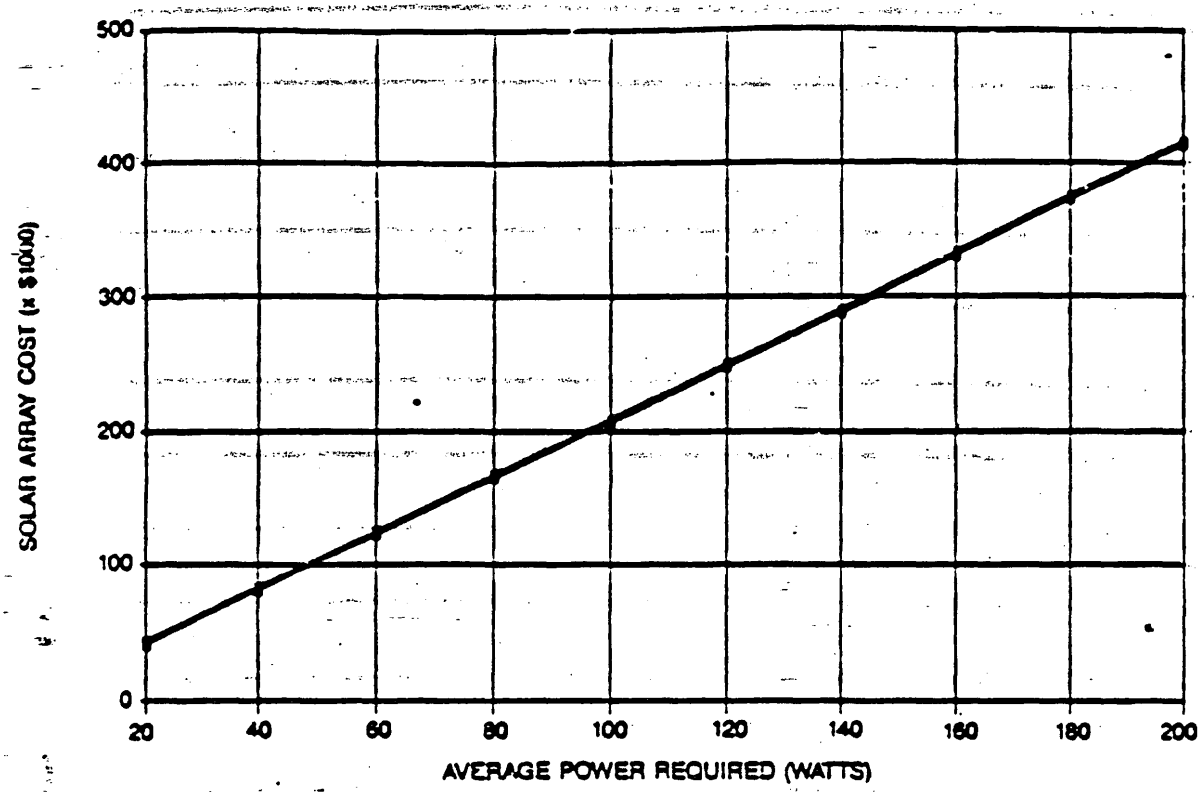


Figure 5-3 Solar Array Cost ($K) Versus Average Power Required

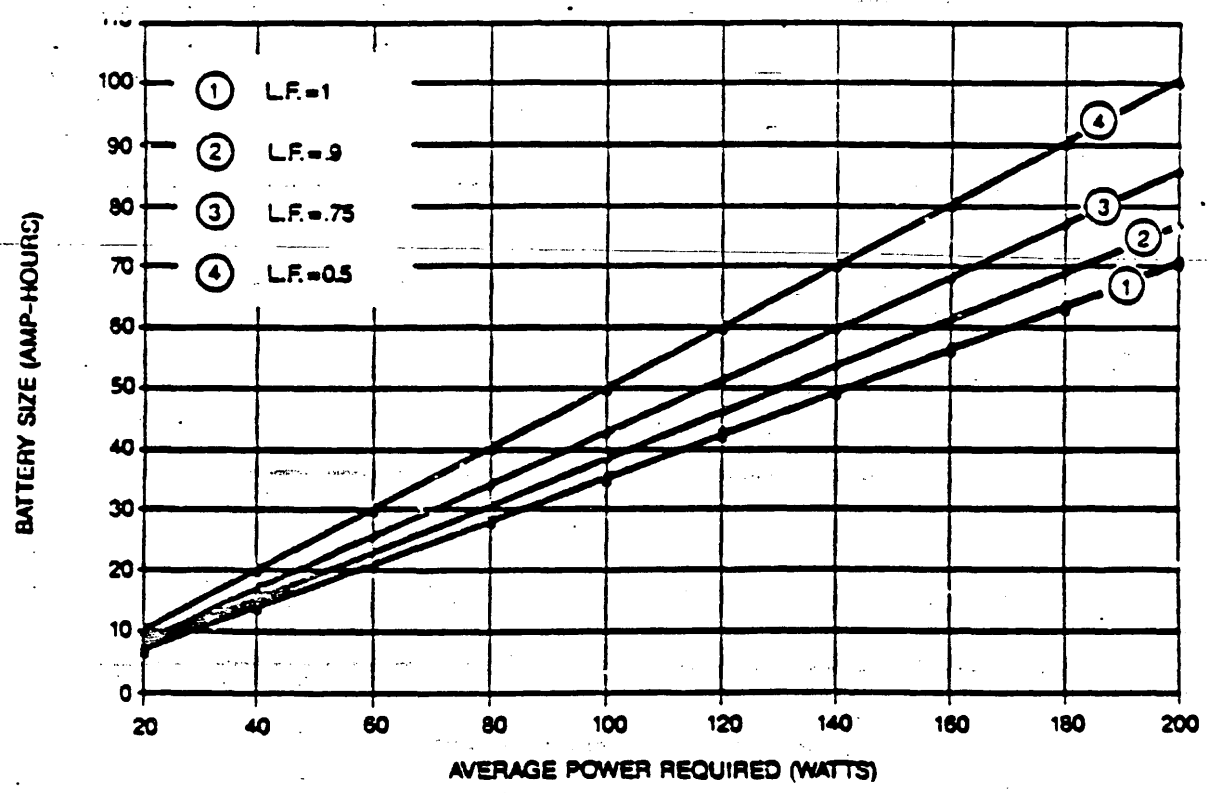


Figure 5-4 Battery Size Versus Average Power Required



A/N 7621a

C/10 = MAX ARRAY OUTPUT - MIN. LOAD REQUIRED

The C is the capacity of the batteries. The idea is to limit the maximum charge rate of the battery, which is the array maximum output minus the system minimum load requirement, to one-tenth of the battery capacity. The one-tenth value is a nominal number as a trade for battery life. The battery size is therefore:

C = 10 * (2.4 * 11 W/ft^2 * PANEL (ft^2) ) - L.F. * AVERAGE LOAD

where L.F.= load factor or the ratio of the minimum power required to average power required.

Since this equation deals with peak power, the factor 64/96 which accounts for orbital averaging is deleted. The array output is 11W per sq ft of array panel. The difference between using 11- and 10W/ft^2 is to account for the beginning vs. end of life of the array output. The peak surface area projection of the array is 2.4. The load factor (L.F. < 1.0) is used as a variable and shows the impact of the varying duty factor of the payload power requirement, e.g., transmitter on/off periods.

Figure 5-4 shows the required battery size vs. the system average power for power from 20 to 200W. The battery sizes required can be quite large. To get a concrete idea of battery size requirements these sizes (amp hr) must be converted into weight, volume, and cost. The weight was estimated at about 1.14 kg per amp hr. The expected weight per system power is shown in Figure 5-5. The battery size will be about 35 cu. in. per amp hr and is shown in Figure 5-6. The cost of NiCd batteries is between $4000 per amp hr. for high reliability and high cost units to $1000 per amp hr. for selected commercial units. Lead acid batteries cost about $200 per unit. Figure 5-7 shows the expected battery cost (L.F.=1). Figure 5-8 is a representation of three battery sizes to show the impact of larger system power requirements.

CAVEAT: The battery weights are based upon NiCd batteries. Lead acid cells will weigh somewhat more. Nickel-hydrogen batteries with pressure vessels would weigh much more. Also, for the larger batteries, some additional weight will be needed for thermal control and structure.

5.1.1.3 POWER SUBSYSTEM COST ESTIMATE

The total system cost vs. power system requirements will involve the array, battery, and system launch costs. The launch cost is dependent upon the launch vehicle used to orbit the satellite. But, since this is a preliminary analysis, the cost of launch via the space shuttle will provide a representative number. The shuttle launch costs are

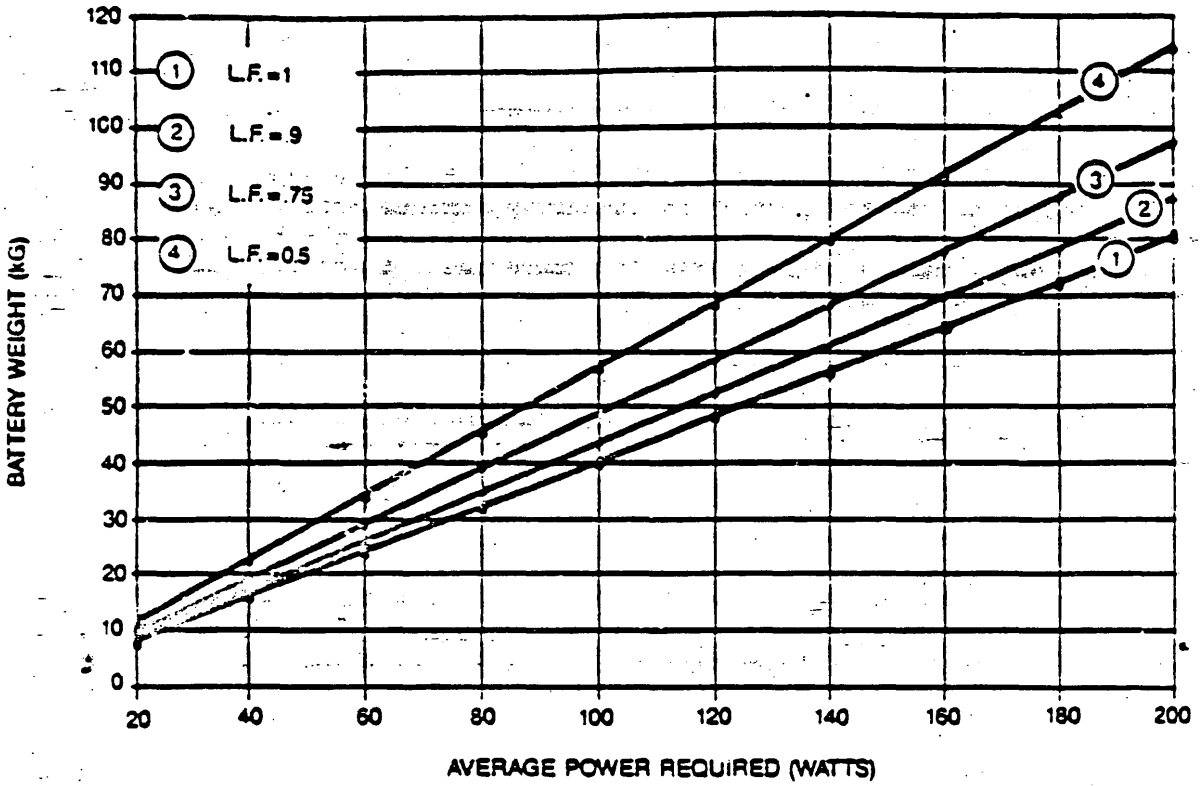


Figure 5-5 Battery Weight Versus Average Power Required

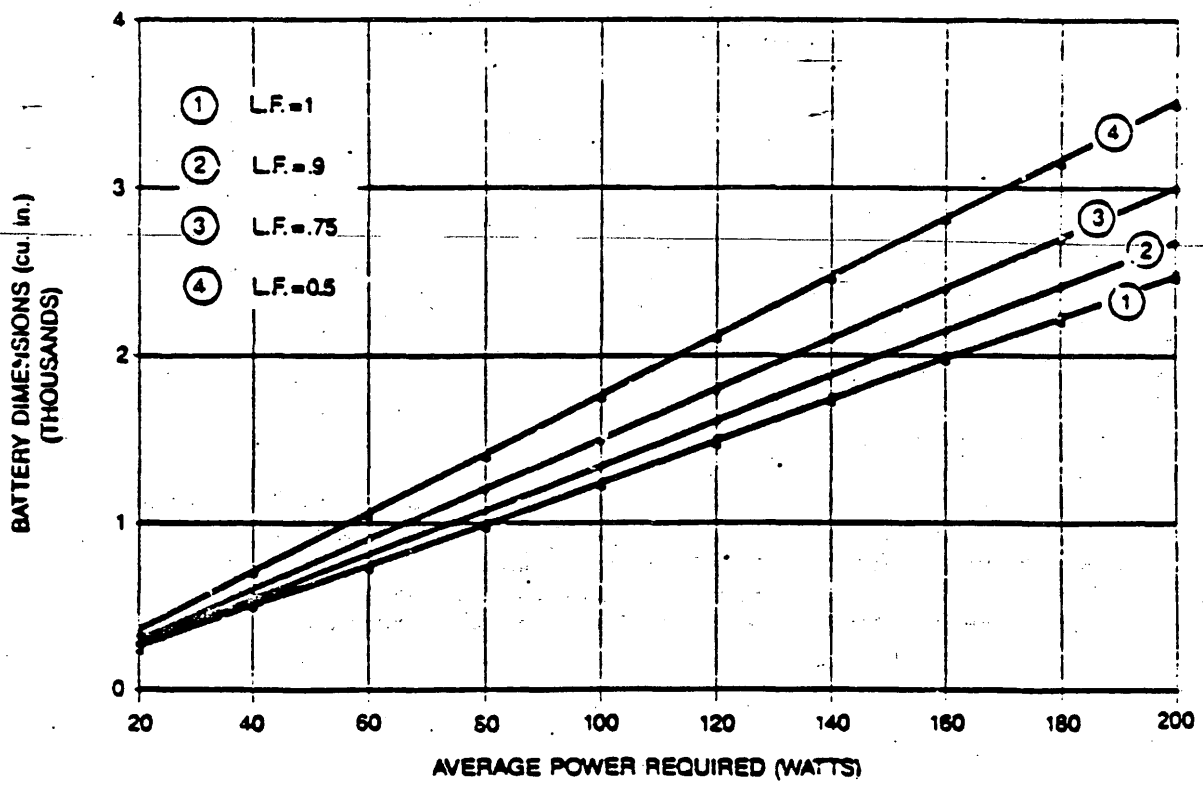


Figure 5-6 Battery Dimension (cu. in.) Versus Average Power Required



A/N 7621a

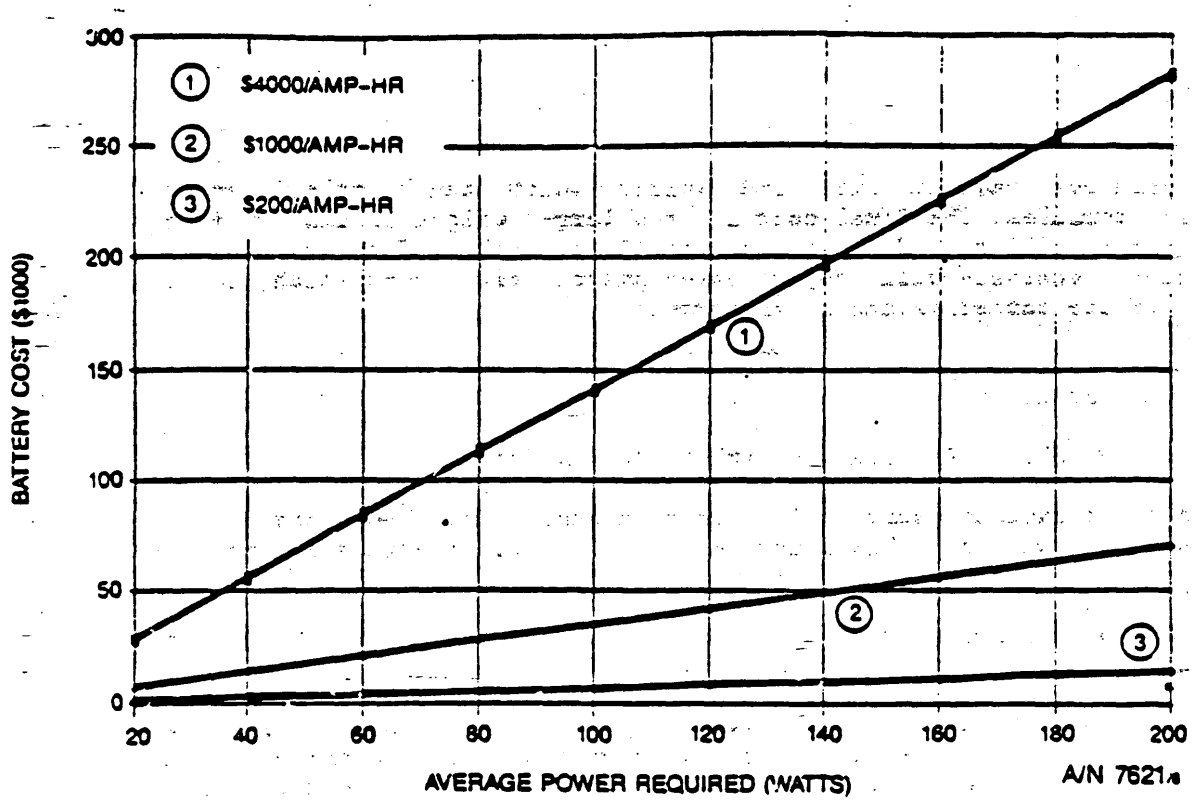


Figure 5-7 Battery Cost ($1000) Versus Average Power Required

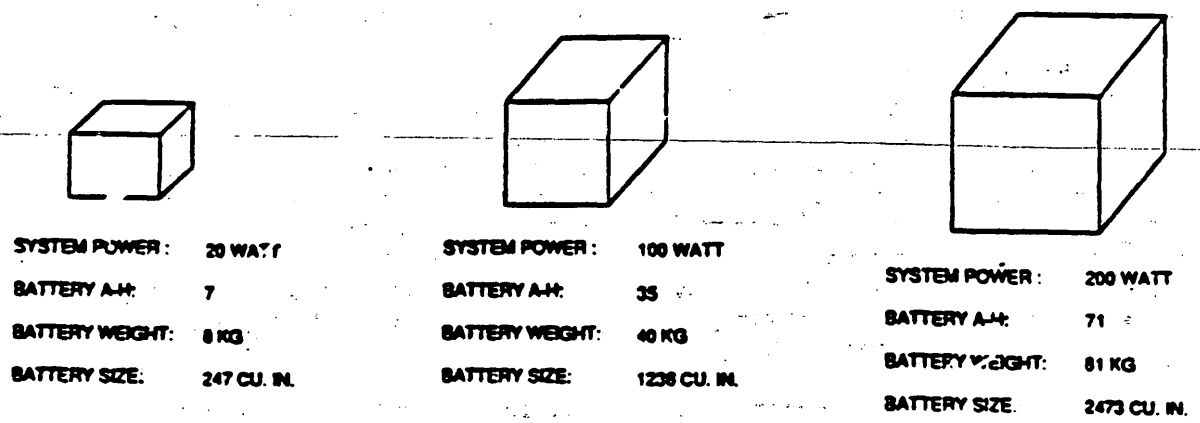


Figure 5-8 Battery Parameters Versus System Average Power

based upon both payload size and weight which can be calculated by existing formulas. The final cost is the larger weight or size. Since payload size will depend upon the satellite final configuration, this preliminary analysis will use the power system weight cost which is the weight of the batteries and solar arrays.

Based upon a shuttle payload base cost of $75M (28.5 deg launch) the cost equation is:

Cost = $7.5M. (Payload Weight)/(.75 * 65000)

or $1538 per payload pound. (The cost increases for 57 deg and 90 deg inclination launches.) The launch cost of the solar arrays and batteries vs. the power system requirements is shown in Figure 5-9.

The total system cost (array + battery + launch) based upon the use of commercial batteries is shown in Figure 5-10.

CAVEAT: As was stated above, the launch cost could be modified if system size with large solar arrays requires that the cost be based upon a size- basis. The launch costs are also based upon a $75M shuttle cost which could increase.

5.1.1.4. POWER SUBSYSTEM COST REDUCTION

Clearly, the cost, size and weight of the higher power system levels previously described are too great. Cost reduction will begin with the size and weight of the batteries. The previous analysis was performed with a low cost power system (current limiting). The battery size was the controlling function of limiting the charge ratio into the battery. By correctly limiting the current, the battery lifetime can be extended. Another, though often more expensive, method of extending battery life is to control the battery charge current based upon the temperature of the battery. This system could limit the charge ratio to the battery to C/2 rather than C/10 with the stipulation that the true charge ratio be controlled by the temperature of the batteries. If the batteries are cool and not fully charged, the charge ratio could be C/2. However, if the batteries are fully charged, they would become warm as the arrays attempt to overcharge them. At this point, the controller would sense the battery temperature and limit the battery charge to a much lower value.

Various methods of charge control are available: shunt load (dumping power into a load resistor), array switching (switching part of the arrays off), and others. The important parameter is the maximum allowable ratio and the estimated cost of the controller. If a cost-effective reliability program and adequate testing (burn-in) is performed, the cost of the controller could be limited to $20K (for large quantity builds). The charge ratio maximum could be set to C/2.

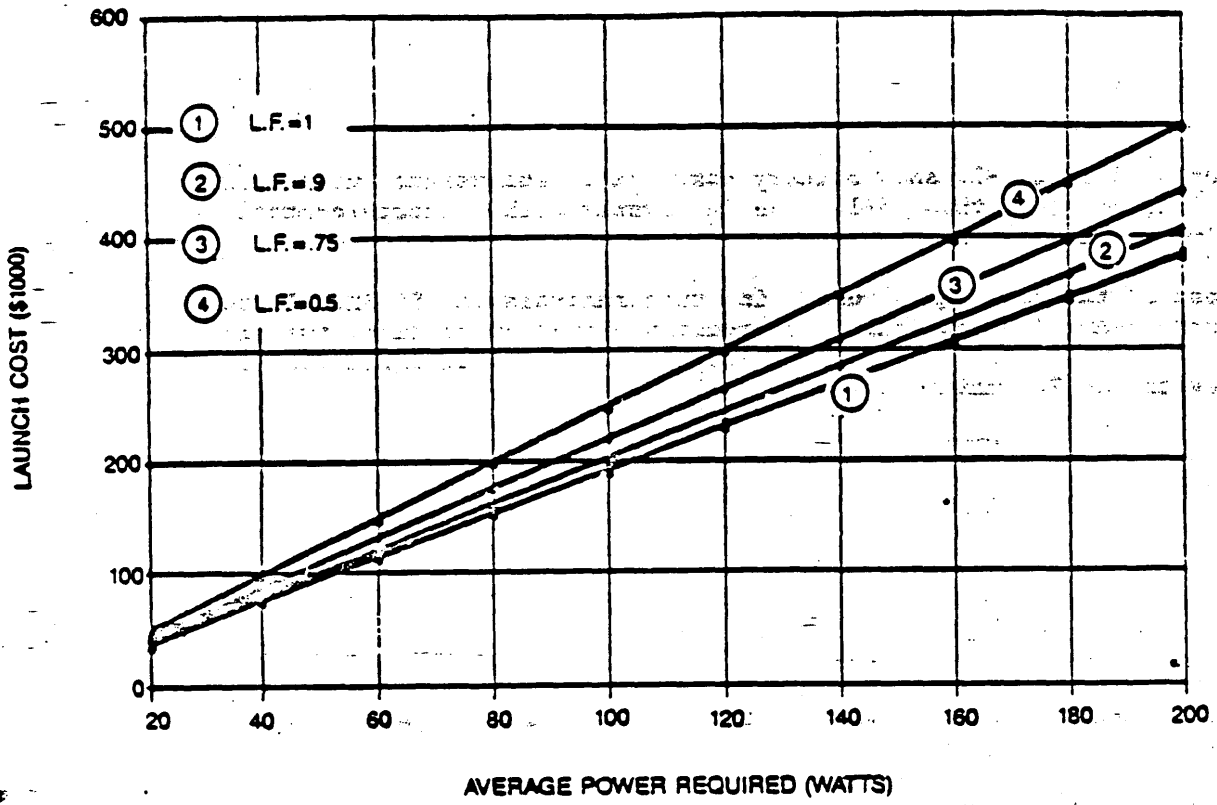


Figure 5-9 Launch Cost ($1000) Versus Average Power Required

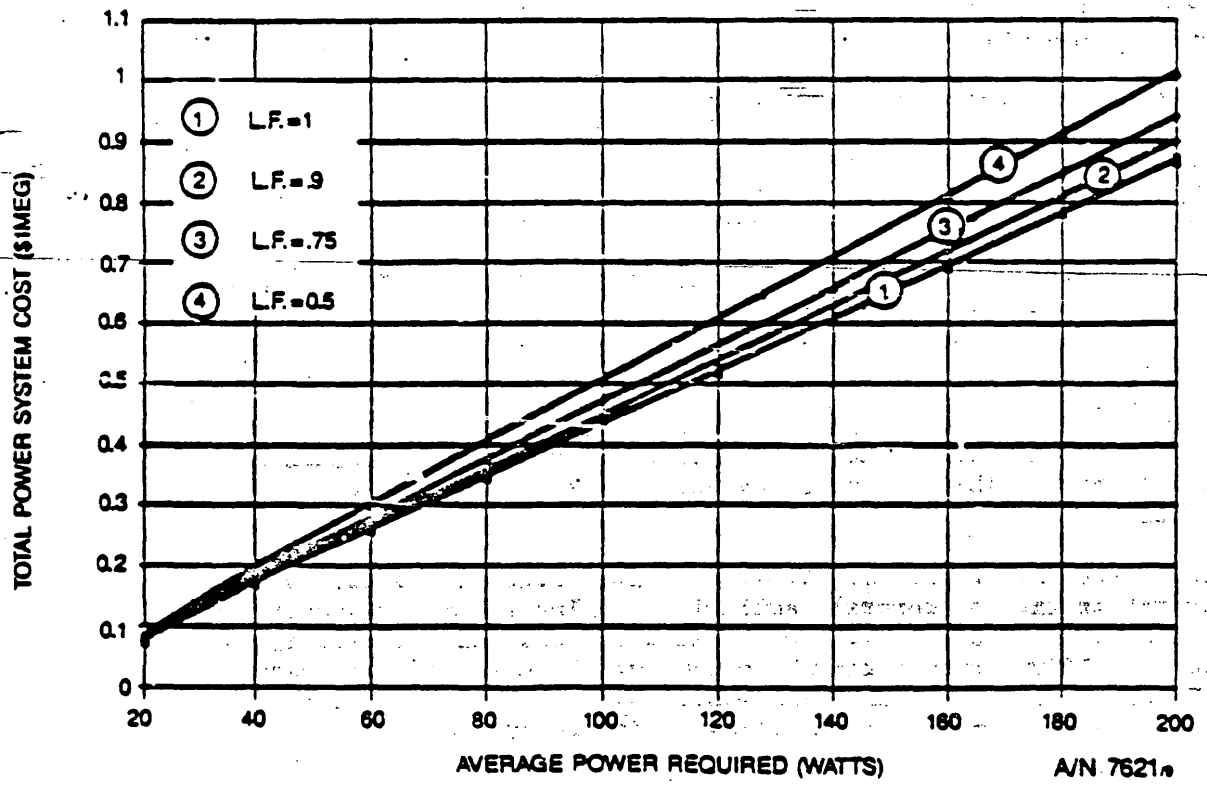


Figure 5-10 Total Power Cost ($1MEG) Versus Average Power Required

Figures 5-11 thru 5-16 show battery cost, size, and weight curves similar to Figures 5-4 thru 5-10 for a system with a charge controller limiting the maximum charge to C/2.

The cost of the solar array used in this analysis was $6.3K. If this cost were reduced by one-half by producing large quantities, the final system cost would be as shown in Figure 5-17. An optimistic cost of a 200W system in 240 units follows:

* LAUNCH: $165K
* ARRAY: $208K
* BATTERY: $ 14K
* CONTROLLER: $ 20K

Possible ways to reduce system cost follows. This analysis, performed early in the MSSP Phase 1 study, identifies areas of concern to other team members.

The largest cost drivers are the solar array and launch cost. Reducing the power required will decrease the system cost. *****

Several options exist to reduce power requirements. Two conventional and one unique approach are:

* Reduce system data rate
* Reduce range
* Reposition transmitter amplifiers

Reducing the system data rate will impact many other subsystem power requirements. The transmitter will require less RF power to complete the link, and the processing speed, (directly related to power) will be reduced.

Reduction of the system range requirement reduces the RF power required. It would also reduce the attitude control power requirements (pointing) and possibly the required antenna beamwidth or number pointing positions.

The transmitter RF amplifier could be repositioned to each antenna element rather than a central amplifier. The antenna switching and phasing components (S&P) will have losses which affect system power efficiency. If the transmitter RF output was 200W and if, for example, the S&P losses were 3 dB, the S&P would absorb 100W of the transmitter output. With a transmitter amplifier efficiency of 25 percent, the

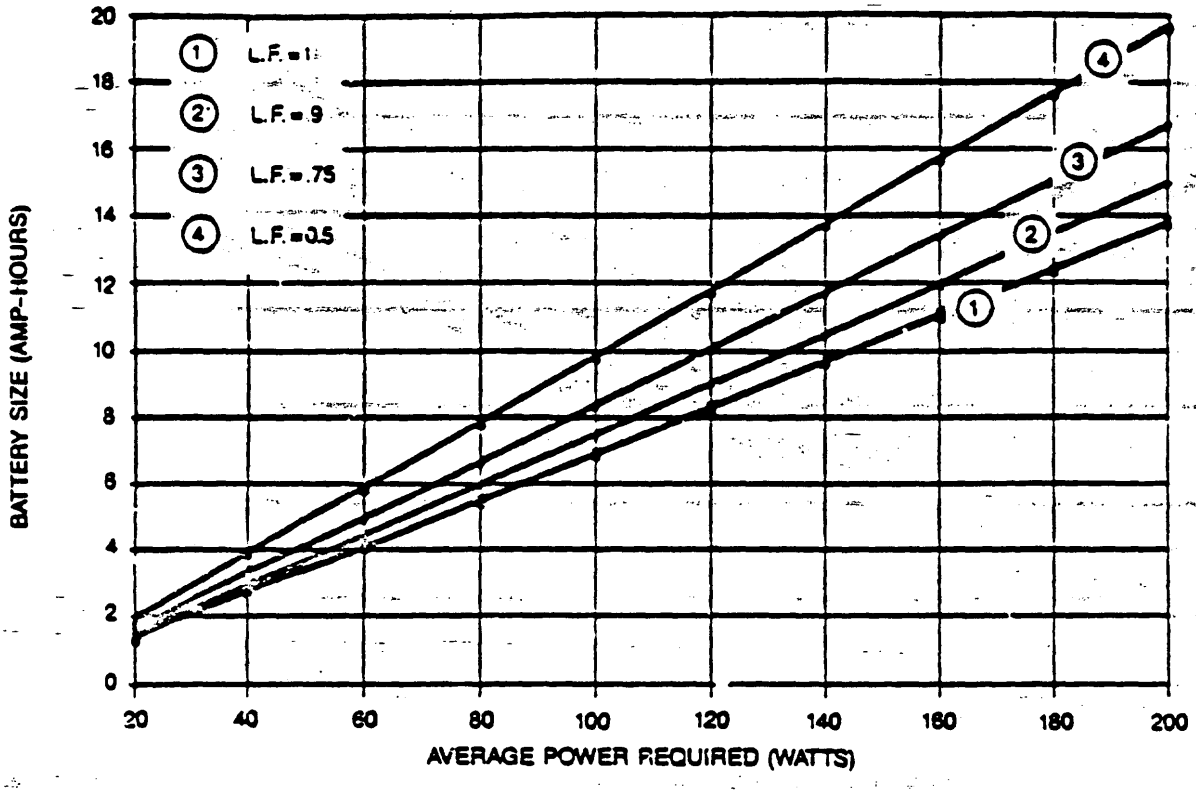


Figure 5-11 Battery Size Versus Average Power C/2 Controller

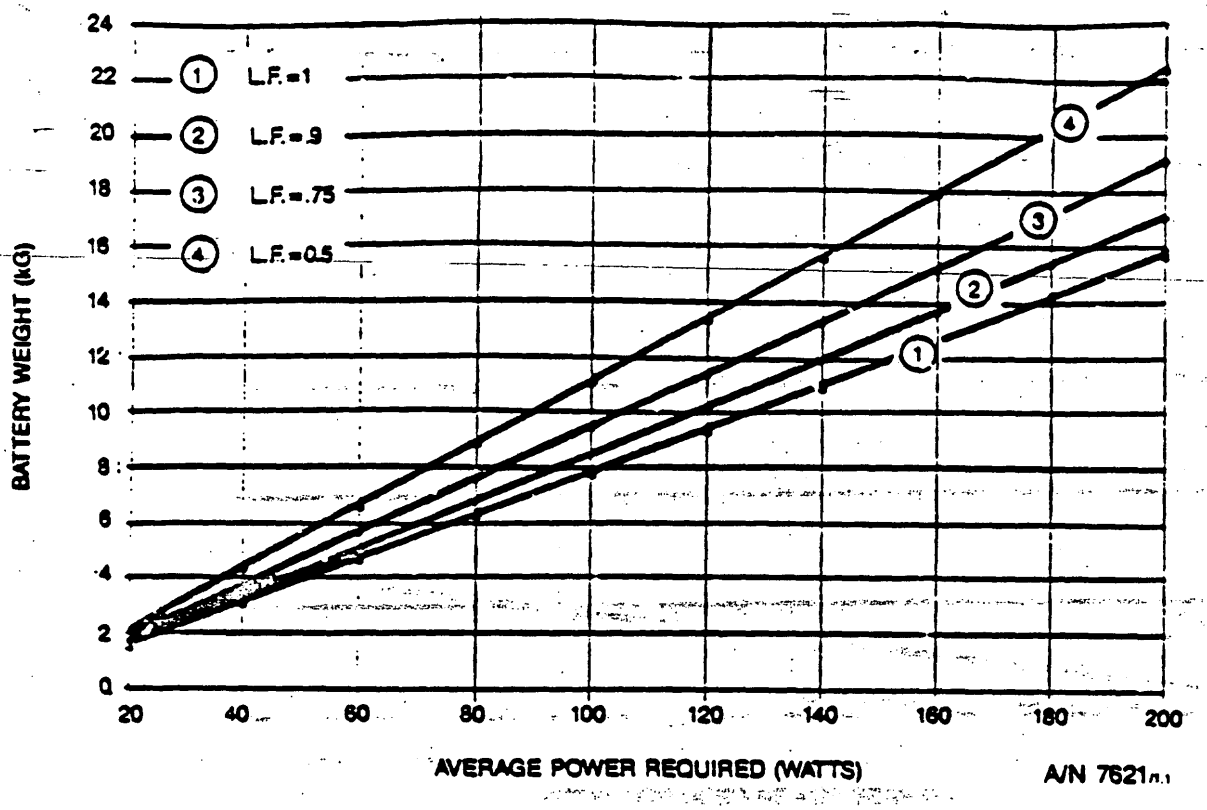


Figure 5-12 Battery Weight Versus Average Power Required (C/2 Controller)

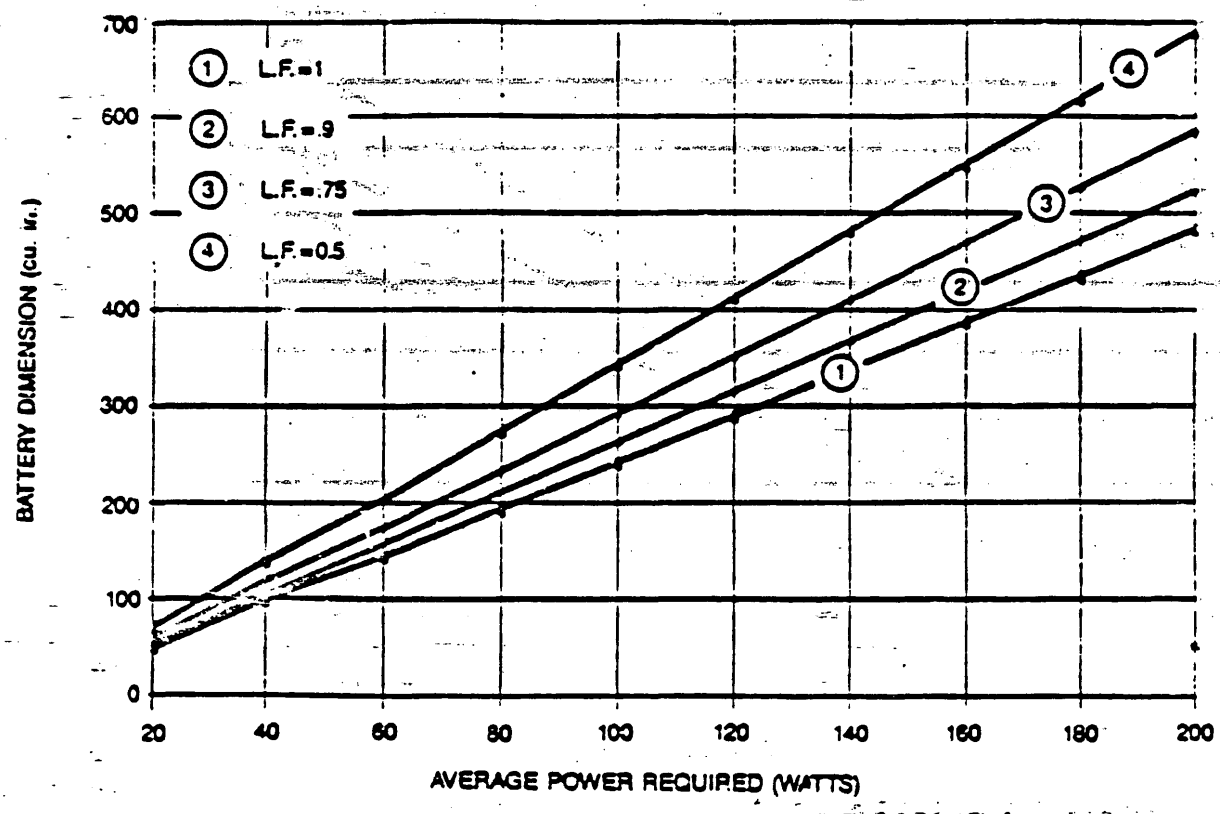


Figure 5-13 Battery Dimension (cu. in.) Versus Average Power Required (C/2 Controller)

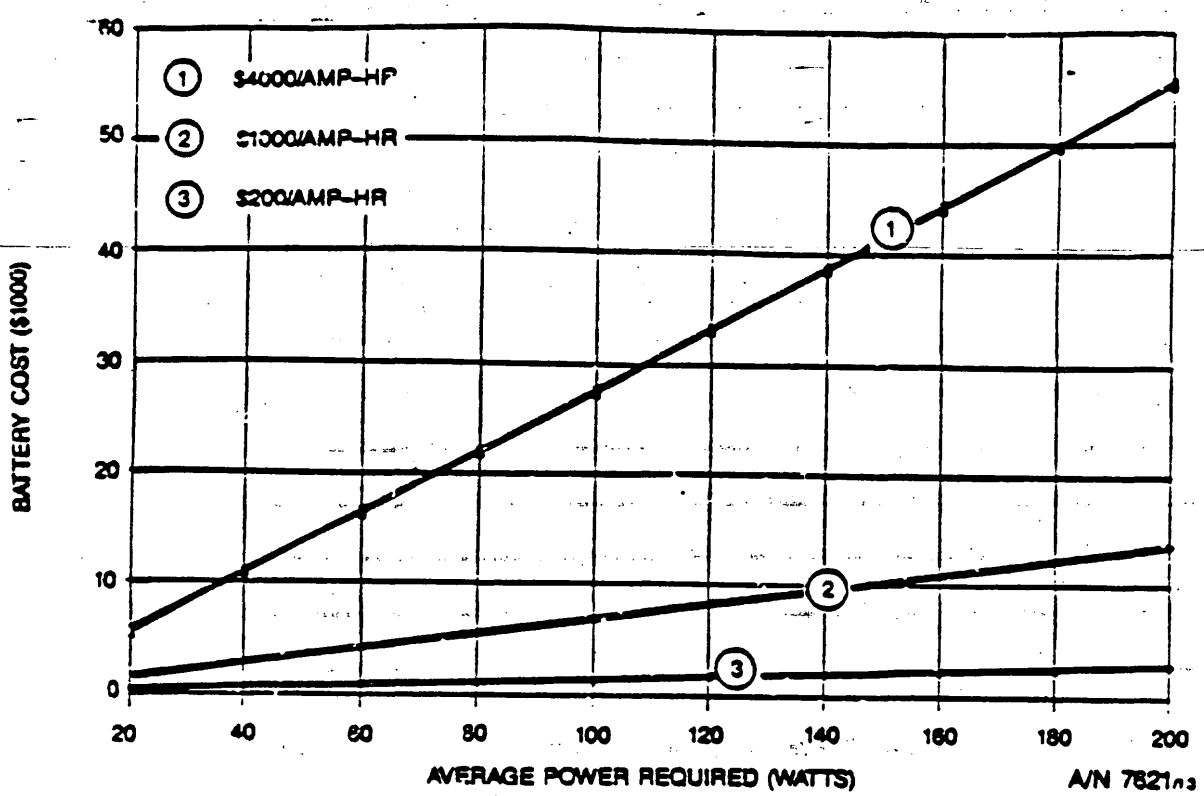


Figure 5-14 Battery Cost ($1000) Versus Average Power Required (C/2 Controller)

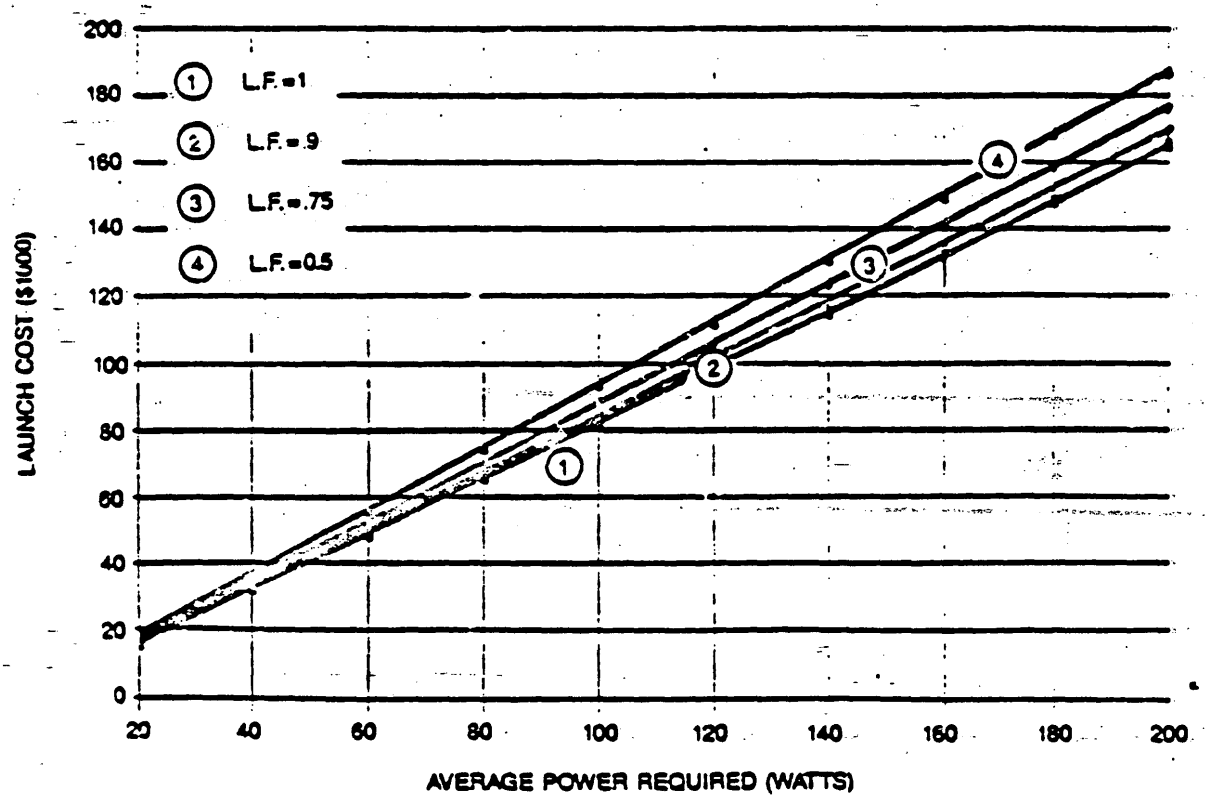


Figure 5-15 Launch Cost ($1000) Versus Average Power Required (C/2 Controller)

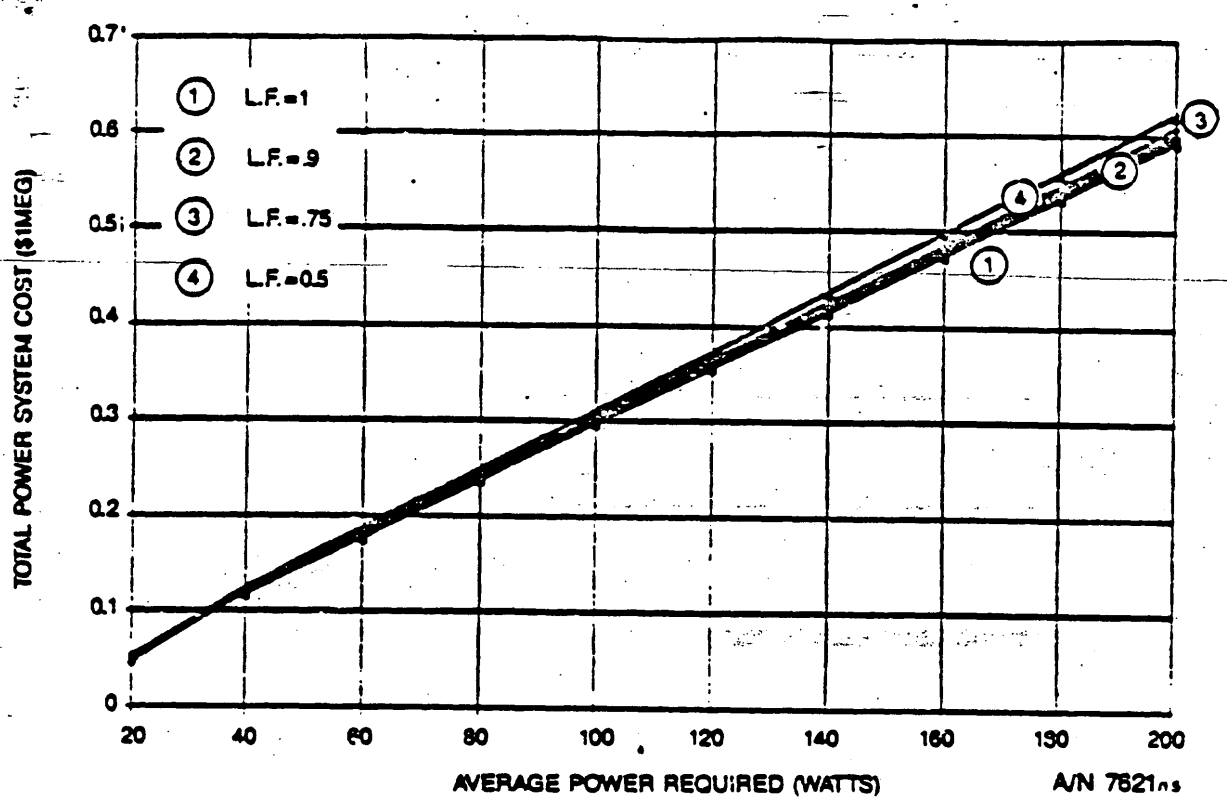


Figure 5-16 Total Power Cost ($1MEG) Versus Average Power Required (C/2 Controller)

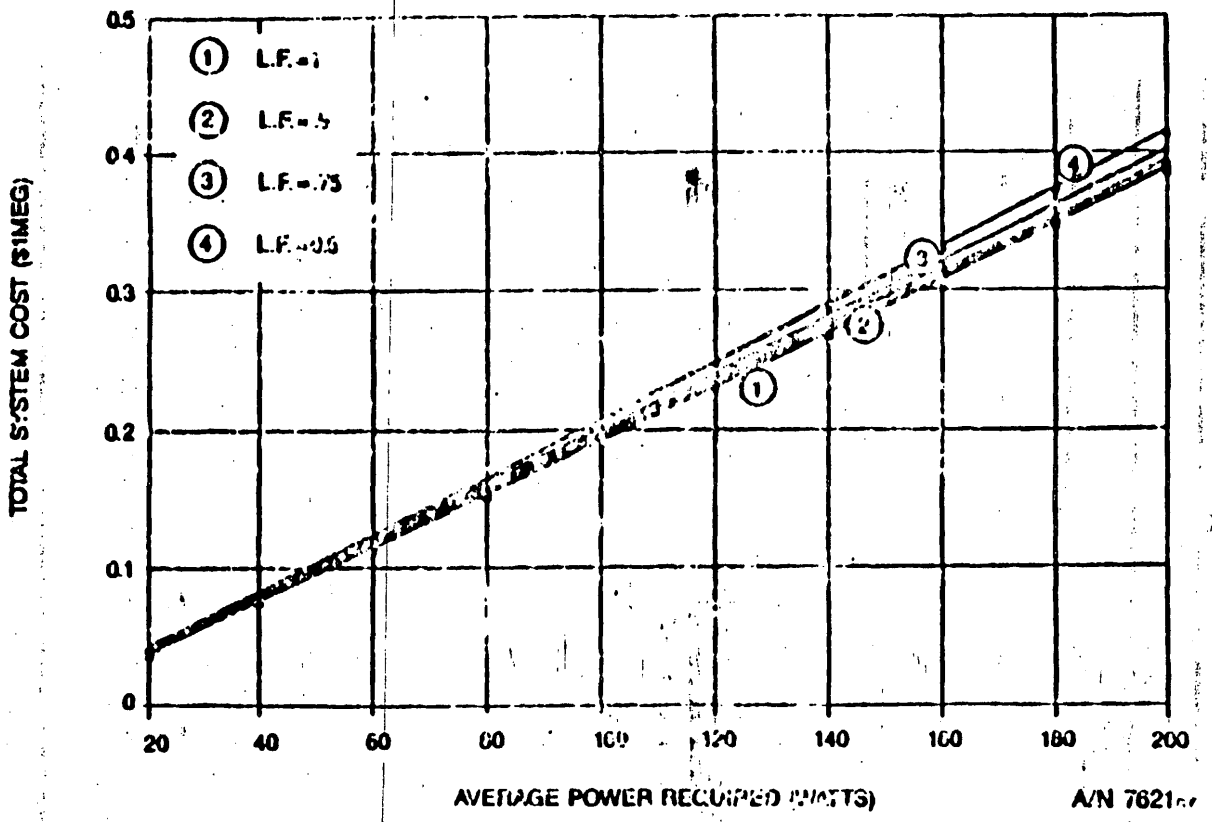


Figure 5-17 Total Cost ($1MEG) Versus Average Power Required (C/2 Controller)

effect of the S&P would be 400W. Conversely, with distributed amplifier at each antenna element, the absolute power magnitude of the S&P losses is less. Also, since the S&P is required to switch lower power levels, less power is required for the electronic switching which could possibly be faster.

### 5.1.2 SOLAR ARRAY OPTIMIZATION

The largest cost driver for the power subsystem is due to the solar arrays. A crucial parameter in designing the power subsystem will be to maximize the array energy output vs. cost. The output can be maximized by increasing the individual cell output efficiency or by increasing the orbital average projected area of the array. This section will analyze the various solar array configurations to obtain a better solar array configuration.

Now that the relative size of the battery and solar array have been analyzed for an omni-directional solar array (Section 5.1.1), an analysis to reduce the cost of the solar array can be undertaken. The omni solar array is independent of the satellite orientation. It consists of four double-sided solar panels or eight arrays. If the satellite were nadir-orientated and in a 97 deg polar orbit at 6 o'clock, the sun would always be on one side of the satellite. Therefore, two solar panels with solar cells on one side facing the sun could provide the same amount of power as the eight panels. Since the sun would be continually in view, the size of the arrays (orbital average) could be individually smaller than an omni panel. A significant cost reduction is realized. The amount of solar cells required is reduced by three-quarters. For example, if the solar arrays cost $100K for the omni, the 2 panel system would cost less than $25K. Other cost reductions would follow due to weight and size reductions. The front of the omni system would consist of two solar panels which would impact the orbital lifetime and attitude stability of the spacecraft. This frontal area (area/mass:A/M) would be close to zero (sideways to orbit direction) and cause the spacecraft to be more stabilized in yaw.

The previous example is an extreme of possible spacecraft orbits. However, it does demonstrate the approach which must be taken to reduce the cost of the MSSP spacecraft. If careful design is used, the benefits of multiple build will not be lost and the additional cost savings of optimum design realized. The omni pattern produces a very low maximum-minimum power output variation. The low variation allows for a low-cost power controller. The cost trade must be made comparing the cost of the solar array vs. the controller and considering the performance parameters of frontal area, power output, weight, and size.

Figure 5-18 shows four possible array configurations. The omni pattern array is the reference unit. The Mansard is a compromise of the omni

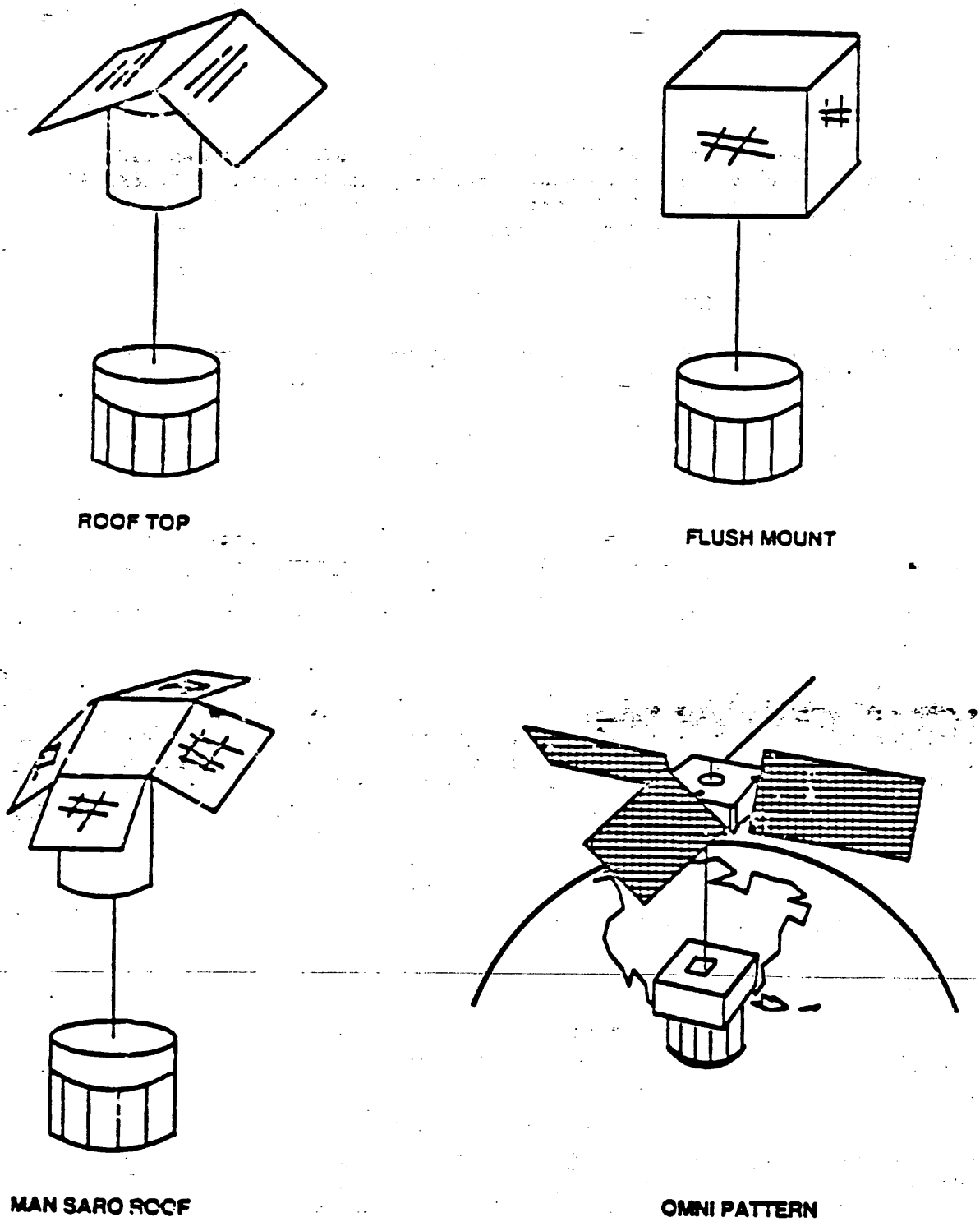


Figure 5-18 Possible Solar Array Configuration

pattern unit with the solar array on one side only. The roof top unit provides a compromise of the Mansard with a smaller frontal area. The flush unit provides the minimum frontal area by simply attaching the solar array to the top and sides of the upper spacecraft module.

A means of comparing the solar array output is necessary. The output for the omni array was 12.44W per sq ft of array panel (Section 5.1.1.1). (10W/ft^2 was a conservative number used in the preliminary analysis of Section 5.1.1.1. For later analysis a number closer to actual orbital numbers (12.44) was used.) A sq ft is actually four-sq ft panels with solar arrays on both sides. The power output per panel sq ft is:

$$\frac{12.44\ W/ft^2}{4\ \text{panels}} = 3.11W/\ ft^2\ \text{of panel}$$

In terms of solar cells, (because cells are on both sides of the panel), the array output is 1.9W per sq ft of actual solar cells. These numbers are the orbital average output power of the solar array (Cursory projection refined numbers will be presented later). Similar numbers (W/ft^2 of panel) produced by other solar array orientations will be used to compare the solar array performance.

Before describing the performance of various types of solar array configurations, it is important to describe the phenomenon of the Sun angle. Figure 5-19 shows a sketch of the Earth/Sun "solar system" and the Sun beta angle for an inclination angle of 0 deg (orbit plane is the equator). The Earth's equator is tilted about 23 deg to the Sun/Earth line; thus producing summer/winter. For a spacecraft orbiting in the equator plane with nadir orientation, the Sun's angle to the orbit plane will vary from zero deg (equinox) to +/- 23 deg. Therefore, the angle from the solar array to the Sun will vary with the time of the year. Also, the Sun/solar array angle will vary with the spacecraft inclination angle. This beta angle variation effect would not be important for an omni pattern array, except for the Earth eclipse. The effect on other solar array designs of the expected Sun/array angles will cause variations in the solar array outputs depending upon the satellite inclination angle.

Figure 5-20 shows the beta angles possible for the satellite in the equator plane. Beta angles vary from zero to 23 deg with respect to the XZ satellite orbit plane. The variation of the beta angle for the zero inclination angle spacecraft is caused by the Sun's seasonal rotation. The maximum possible beta angle for satellite inclination angles other than zero would be the sum of the inclination angle and 23 deg.

$$BETA\ (MAX) = 23 + Ai$$

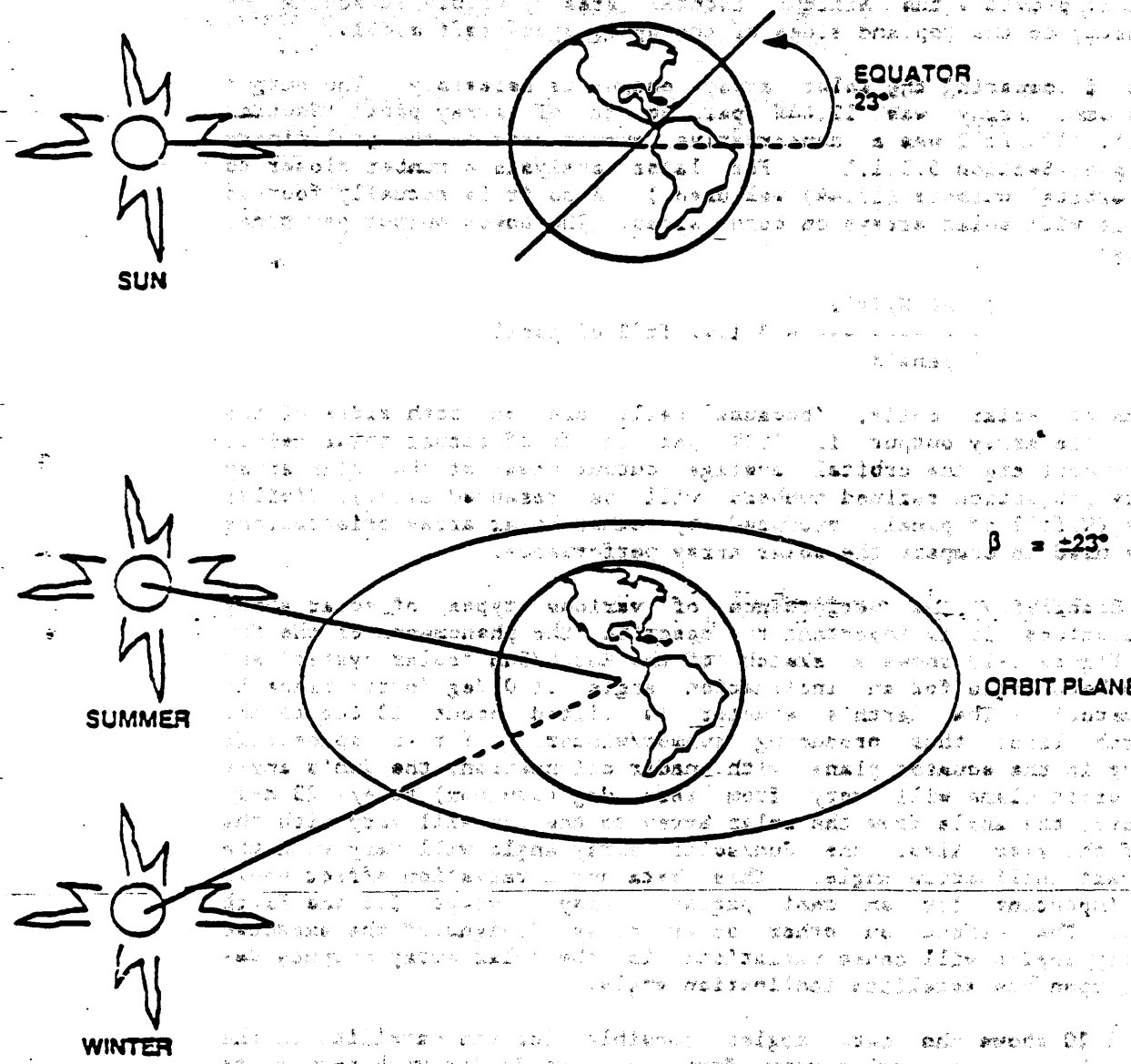


Figure 5-19 Sun $\beta$ Angle for Inclination Angle $A_I = 0$

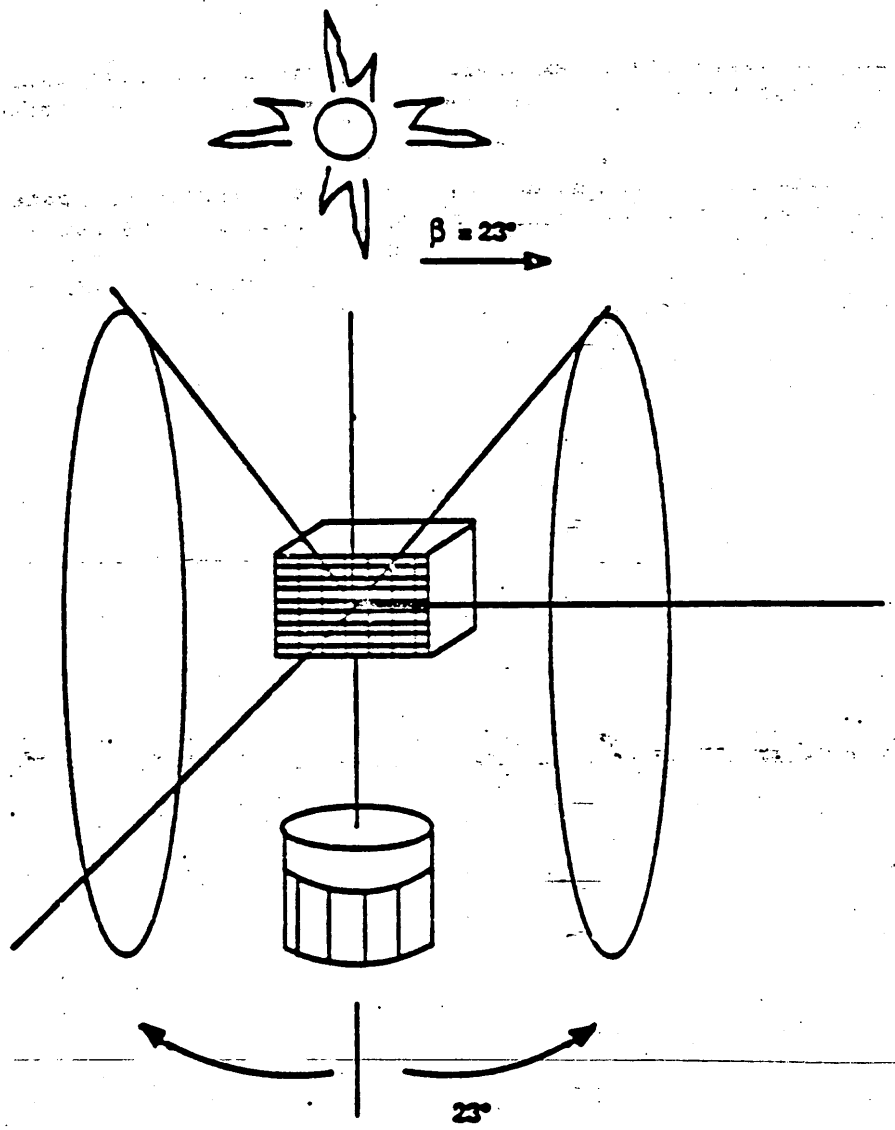


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Figure 5-20 Beta Angle Equator Satellite

For inclination angles other than zero, nodal regression will cause the orbit plane to rotate and the satellite beta angle to vary between + and- beta max.

By using the beta angle technique, analyzing the effect of spacecraft motion on the spacecraft solar array output is easier. The solar array output of any satellite will vary with inclination angle, season of the year, etc. With beta angle, an analysis can be done which reveals orbital output allowing satellite power to be properly sized.

The beta angle affects three important factors:

* solar array output

* satellite eclipse time

* satellite thermal effects

The thermal effects will not be addressed in the study. The satellite eclipse time is the time that the satellite is hidden from the Sun by the Earth. Figure 5-21 shows the satellite eclipse angle vs. the satellite beta angle. Note that for beta angles greater than 70 deg for an altitude of 650 km, the eclipse angle can be 180 deg. In other words, periods will exist for satellites with inclination angles greater than 47 deg where the satellite will view the sun continuously. Figure 5-22 shows the eclipse time as a percentage of orbit vs. the beta angle. The eclipse time in minutes vs. beta angle is exhibited in Figure 5-23. The solar array output is greatly affected by the angle of the sun to the solar array.

Rather than addressing the projected output of a particular solar array configuration, the analysis first looked at the output of single arrays tilted toward the velocity direction and toward the orbit normal (Figure 5-24). Any MSSP array configuration will be a combination of these arrays. The MSSP spacecraft is assumed to be nadir-pointing but uncontrolled in yaw.

Figure 5-25 is a brief description of the solar array orbital average power output power equations. The array output constant is assumed to be 12.44W per sq ft. This number will be dependent upon many factors, including temperature, array glint angle, and radiations effects. For this analysis, which is the array configuration, it will not be a crucial factor.

Figure 5-26 and 5-27 show, for various tilt angles, the orbital average outputs of a solar array tilted toward the spacecraft velocity direction. The top curve is the orbital average output for a zero degree tilt; the solar array is pointing anti-nadir. The zero degree

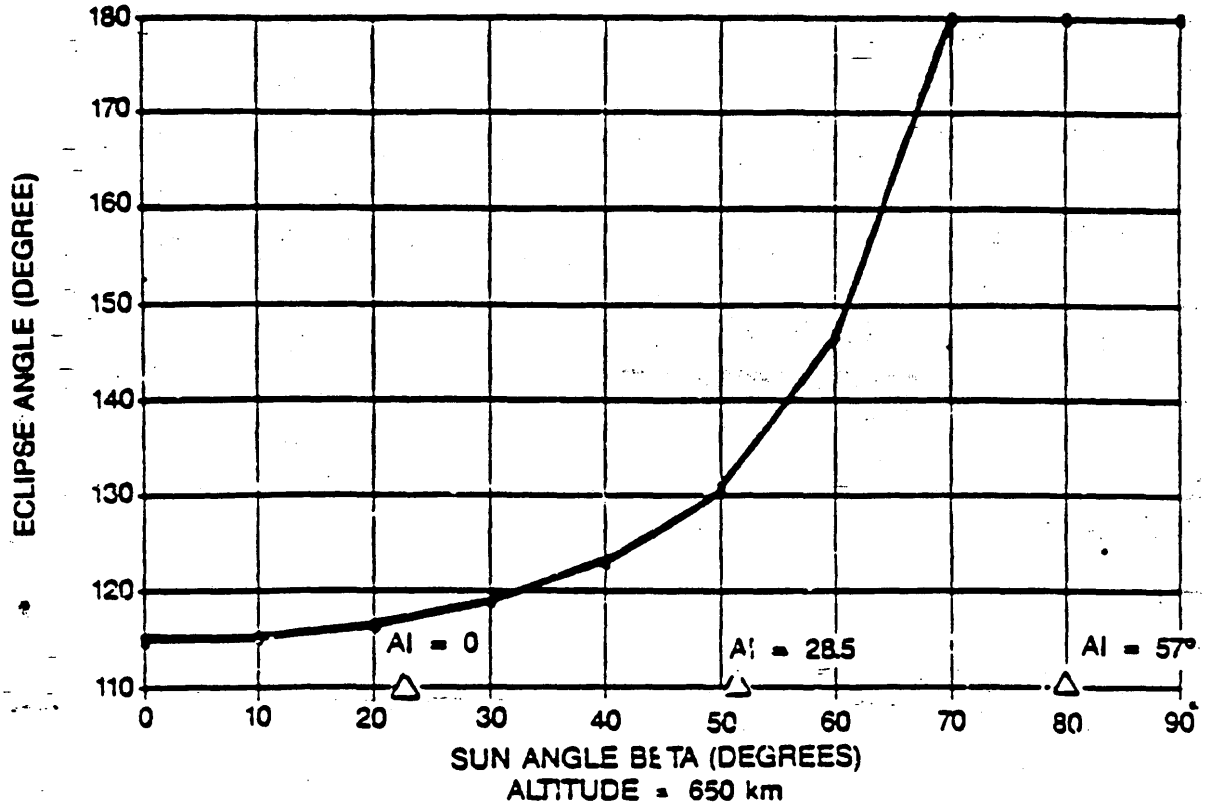



Figure 5-21 Eclipse Angle Versus Sun Angle Beta

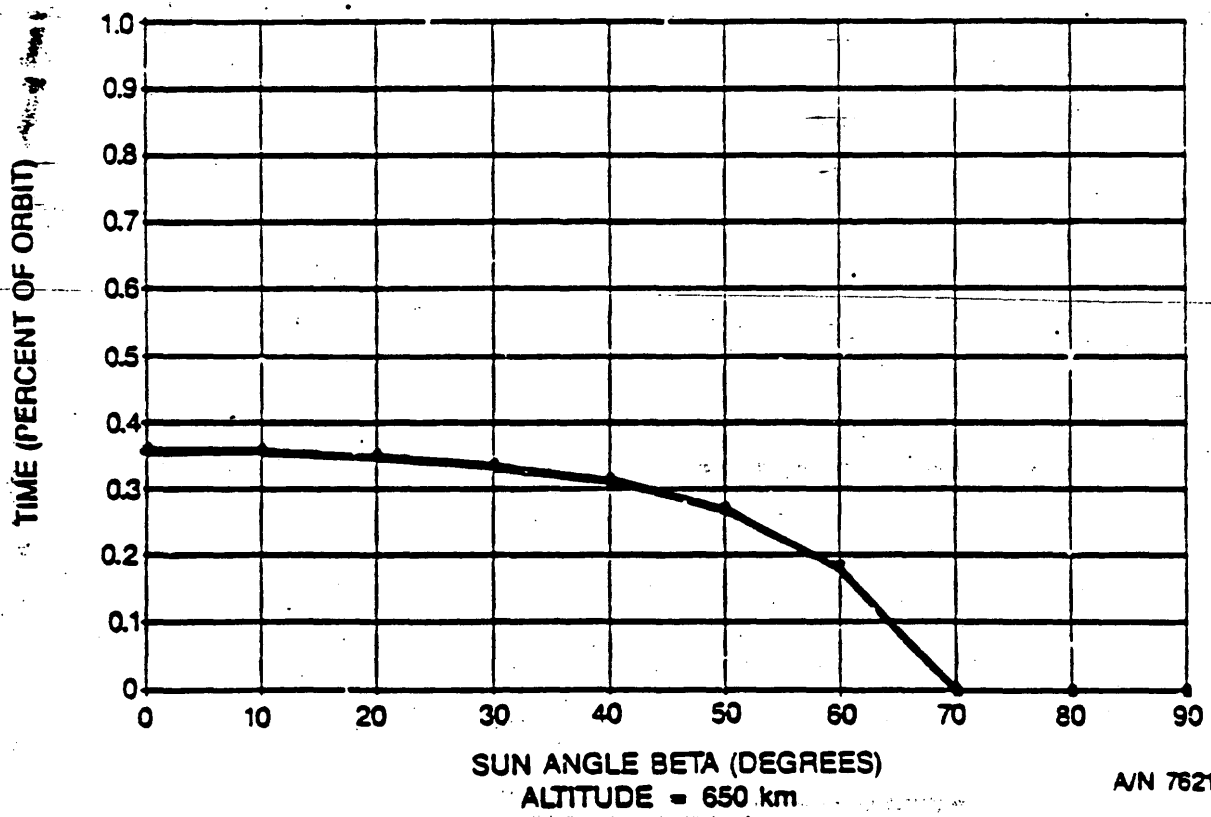



Figure 5-22 Eclipse Time (Percent of Orbit) Versus Sun Angle Beta

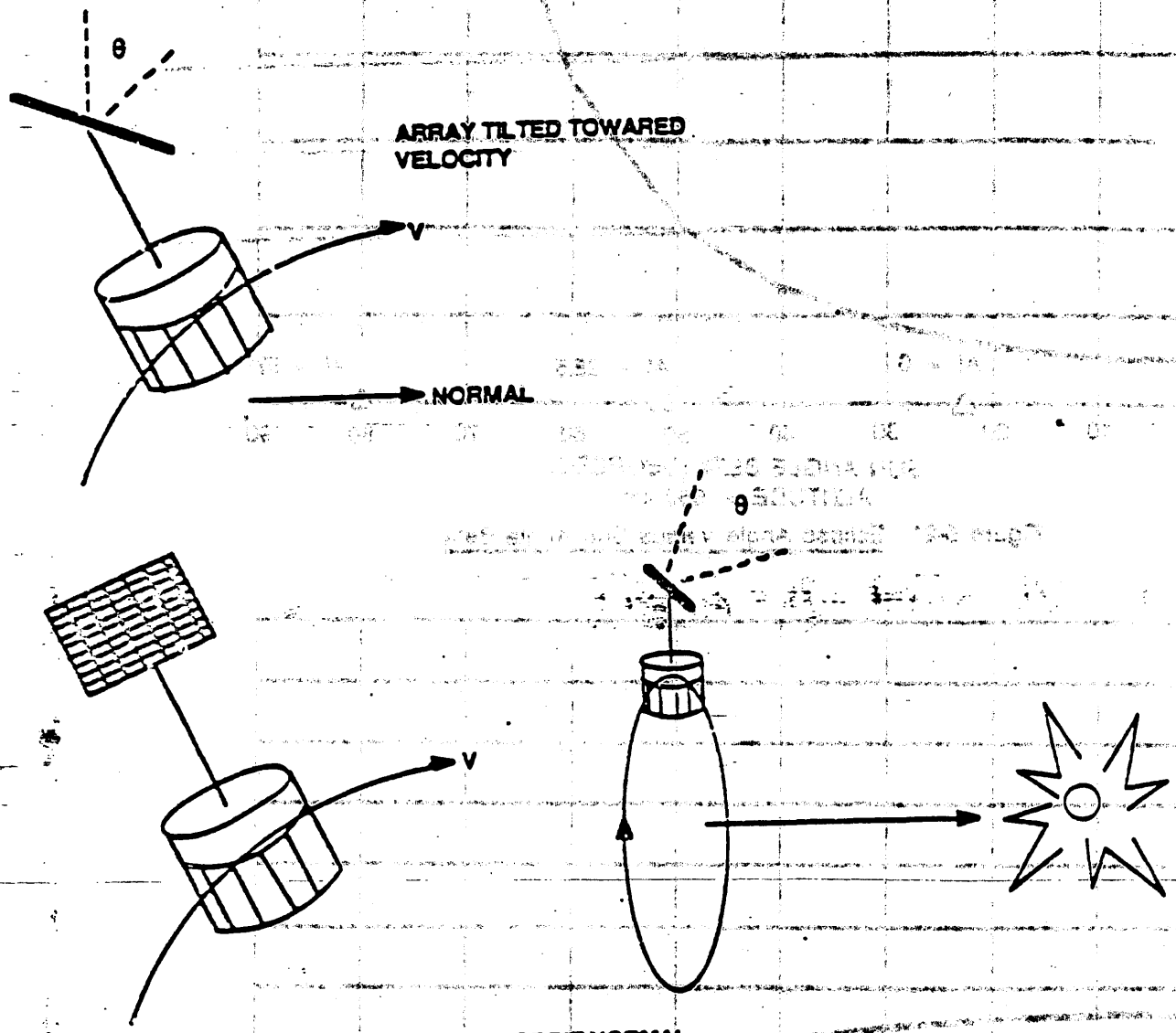
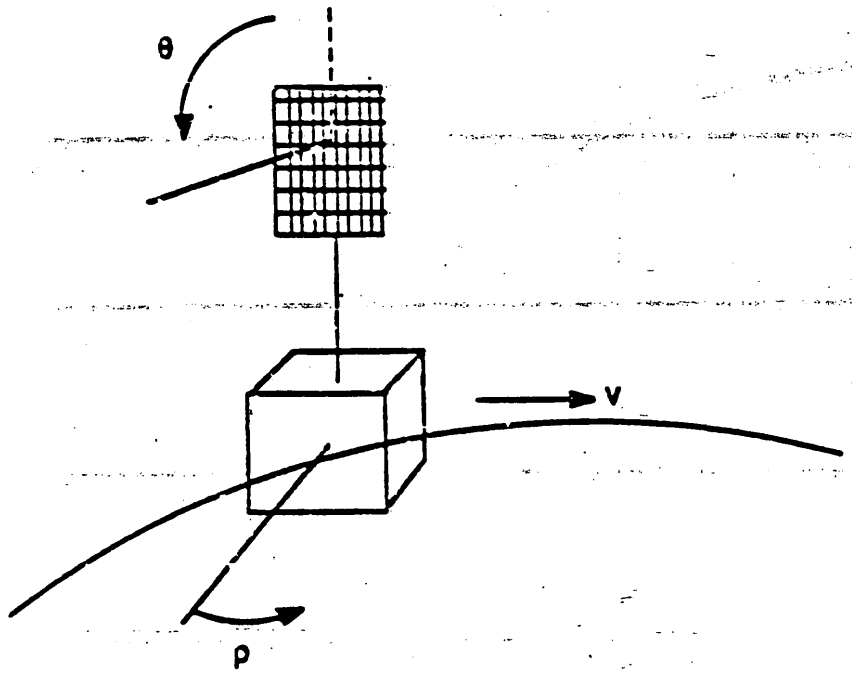



Figure 5-24 Illustration of Array Tilt




$$\text{POWER} = \text{AREA} \cdot C \cdot \frac{\int_{-\gamma}^{\gamma} \cos \varnothing \, d\gamma}{2\Pi}$$

$$\frac{\text{POWER}}{A} \left(\frac{W}{Ft^2}\right) = C \cdot \frac{\int_{-\gamma}^{\gamma} \cos \varnothing \, d\gamma}{2\Pi} \qquad ; C = 12.44 \ w/Ft^2$$

$$\cos \varnothing = \cos \beta \cos \theta \cos \gamma - \cos \beta \sin \theta \sin \rho \sin \gamma + \sin \beta \sin \theta \cos \rho$$

$\gamma$ : ORBIT POSITION (VARIABLE)

$\theta$ : TILT ANGLE

$\beta$ : SUN ANGLE

$\rho$ : ARRAY ORIENTATION

$\gamma$ LIMITED BY EARTH ECLIPSE AND ARRAY OUTPUT WITH SOLAR CELLS ON ONE SIDE.

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Figure 5-25 Power Output Equations (Orbital Average)

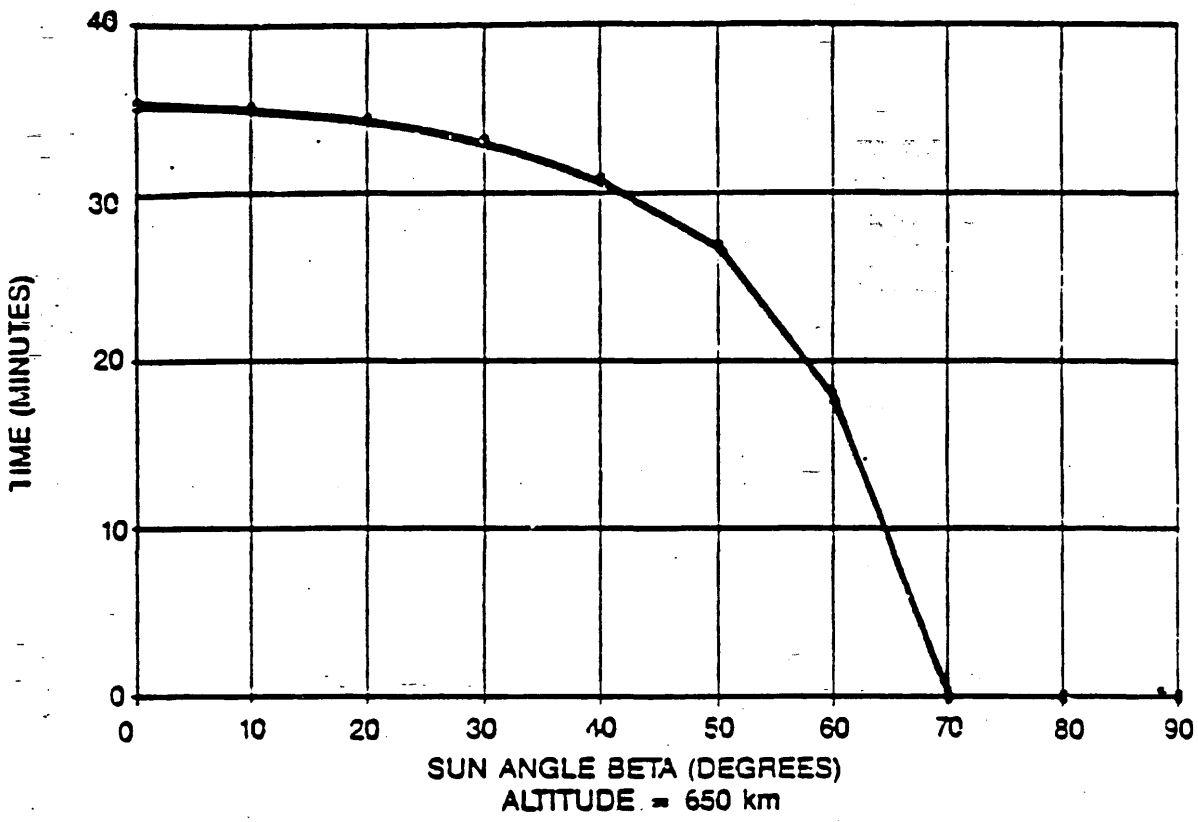


Figure 5-23 Eclipse Time Versus Sun Angle Beta

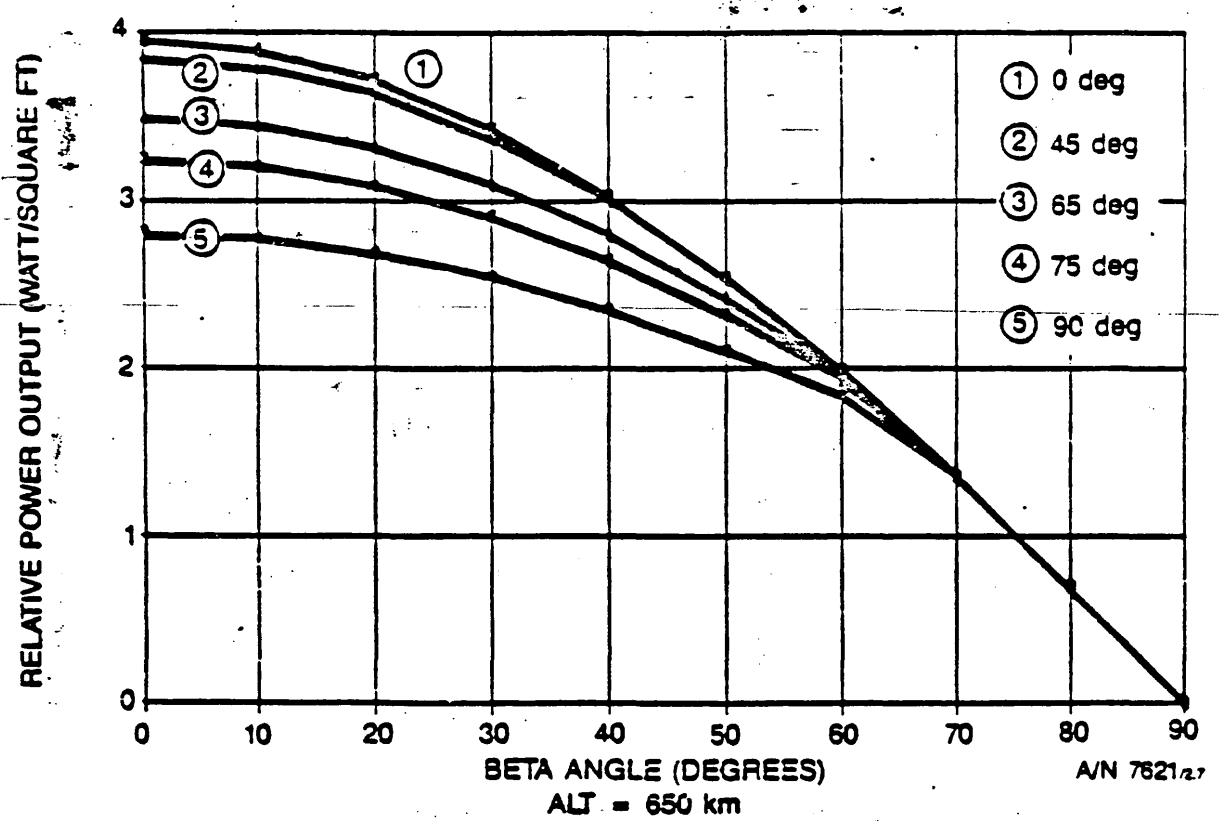


Figure 5-26 Power Output Versus Outer Panel Tilt Angle
(Panel Tilted Toward Velocity Direction 0° to 90°)

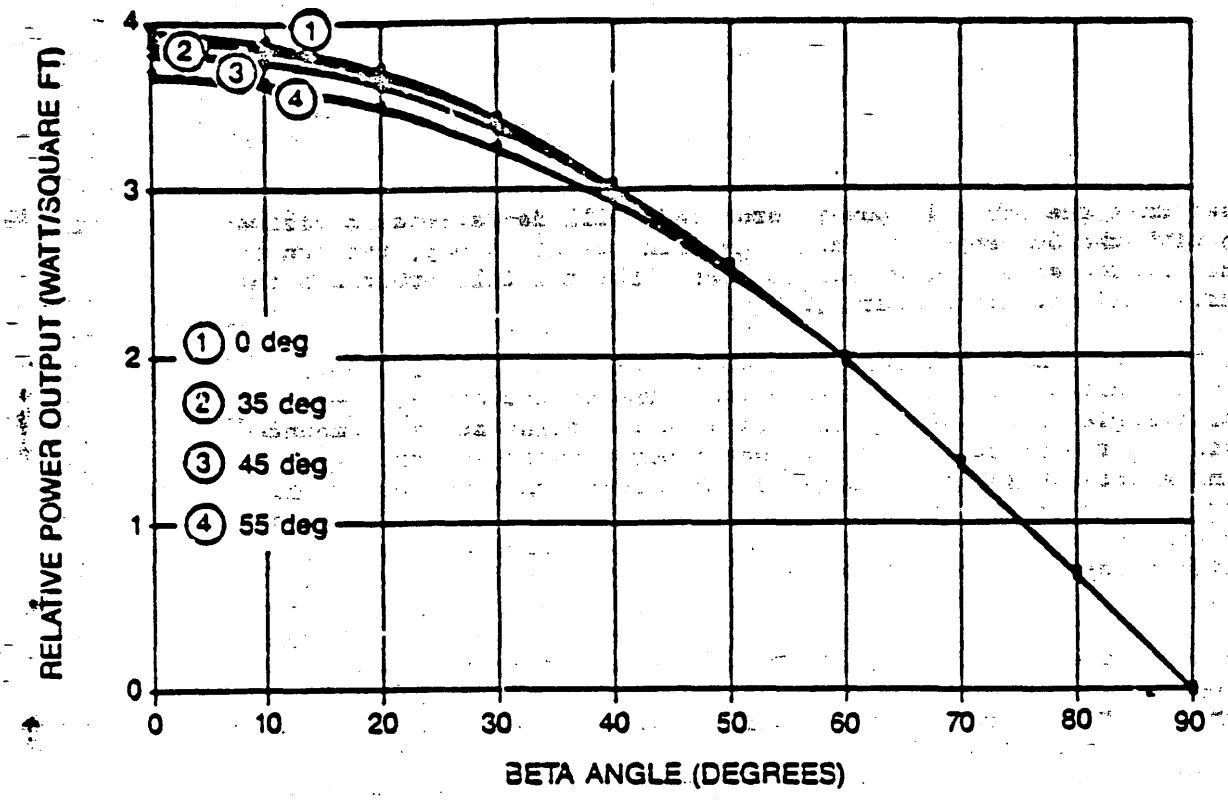


Figure 5-27 Power Output Versus Outer Panel Tilt Angle (Panel Tilted Toward Velocity Direction 0° TO 55°)

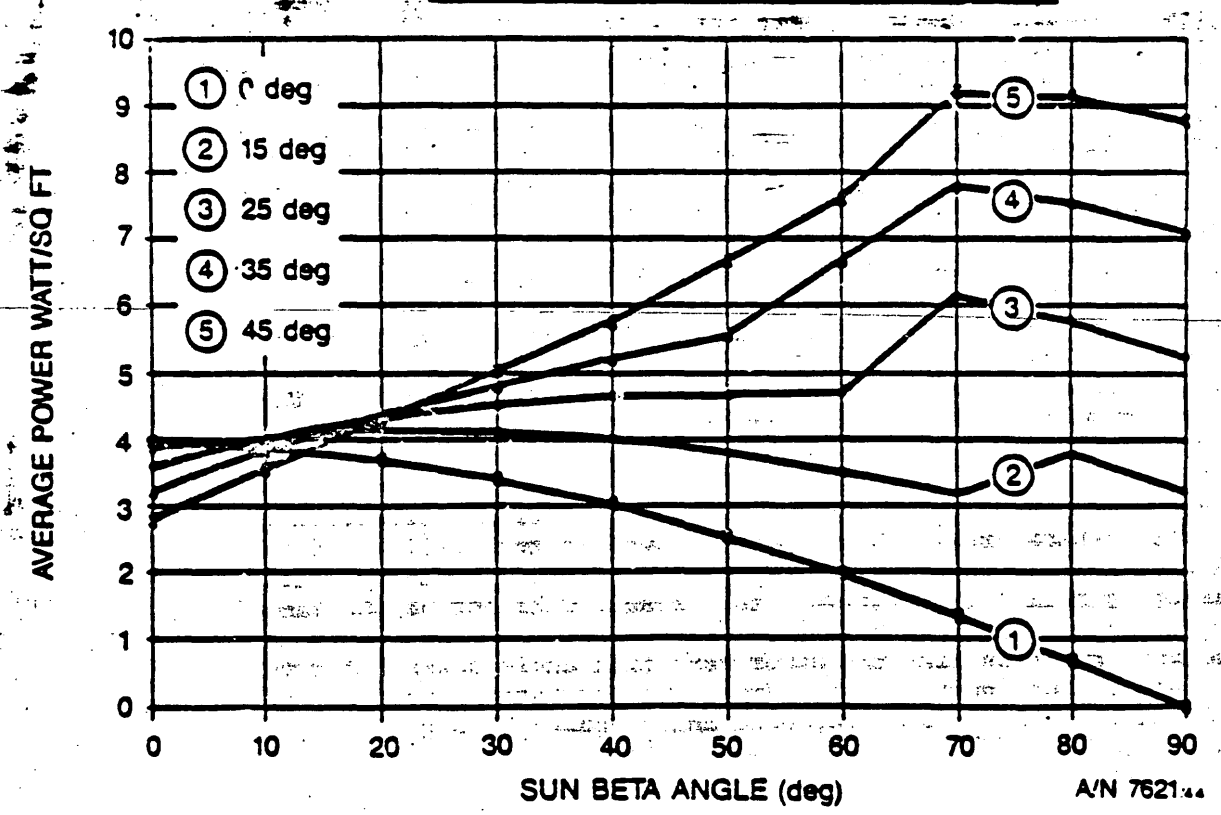


Figure 5-28 Power Per Sq. Ft. of Array Versus Beta for Various Tilt Angles (Panel Tilted Toward Orbit Normal 0° TO 45°)

tilt shows that the orbital power produced will decrease as a cosine function with the Sun angle. When the Sun is at 90 deg, the Sun is positioned at the edge of the solar array for the full orbit and the solar array will produce no output.

Figure 5-27 shows small variations of power output as the tilt angle of the array is increased. Figure 5-26 shows the variations of power for larger tilt angles. The 90 deg tilt is when the solar array is mounted to the side of the spacecraft. The power output with an array tilt of 90 deg and a beta angle of 0 deg is much lower than the 0 deg tilt. This condition is caused by the array being hidden during part of its orbit by the solar eclipse. As the beta angle approaches 90 deg, the 90 deg tilt array output approaches the output of the 0 deg tilted array. If these solar array configurations were the only choices, then for a satellite orbiting at 28.5 deg (beta max = 51.5), the optimum array choice would be a 0 deg tilt and the solar array size would be calculated using 2.5W/sq ft average array output. The sizing of the solar array must use the lowest orbital average power output. If the inclination angle were 57 deg, it would not be possible to build a suitable array with only a forward tilt because of the zero output when beta equals 90 deg.

Figures 5-28 and 5-29 show the orbital average power output from an array tilted toward the orbit normal or toward right of the satellite and the Sun as the satellite flies. The zero degree tilt has the same results as the forward tilt of zero degrees. As the angle of tilt increases, the power output for a beta angle of zero decreases and the output for the higher beta angles increases. Figure 5-29 shows that for a tilt angle of 90 deg, the power output approaches a sine wave with zero output when the beta angle is zero and 12.44 when the beta angle is 90. The 90 deg tilt condition exists when the solar array is mounted to the side of the spacecraft looking to the right side of the spacecraft. Both Figures 5-28 and 5-29 show elevated curves for beta angles of 50 deg or larger. The eclipse region is slowly being reduced while the tilt effect is being seen. For example, the 25 deg tilt power greatly increases for beta angles between 60 and 70 deg and then proceeds with the normal decrease expected for beta angles between 70 and 90 deg.

When the angle of the array tilted toward the forward direction is negative (tilted backwards), the orbital power output will be of the same form as a positive tilt angle. When the angle of the array tilted toward the array normal is negative, the result will not be the same. Figure 5- 30 shows the orbital power output for a negative angle. Again, the zero angle is like the other zero tilt angle cases. For the other tilt angle, the outputs are sine type variations down to zero output when the tilt angle plus the beta angle equals 90 deg.

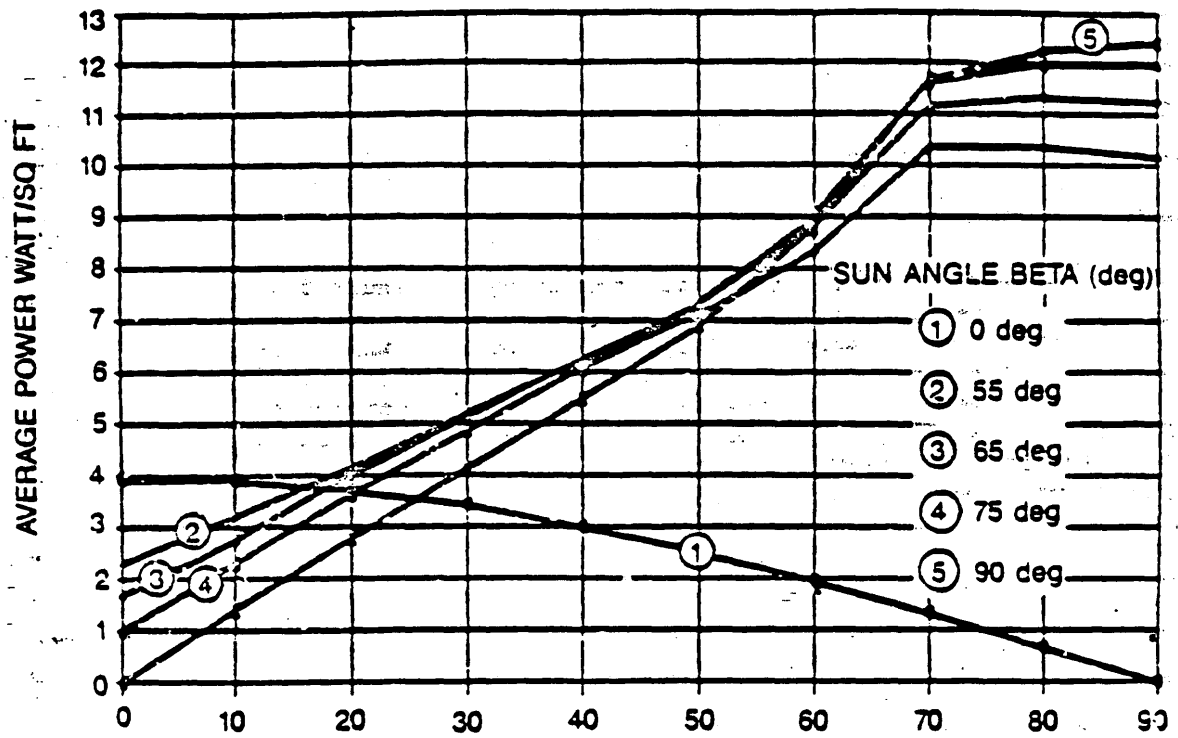


Figure 5-29 Power Per Sq. Ft. of Array Versus Beta for Various Tilt Angles (Panel Tilted Toward Orbit Normal 0° TO 90°)

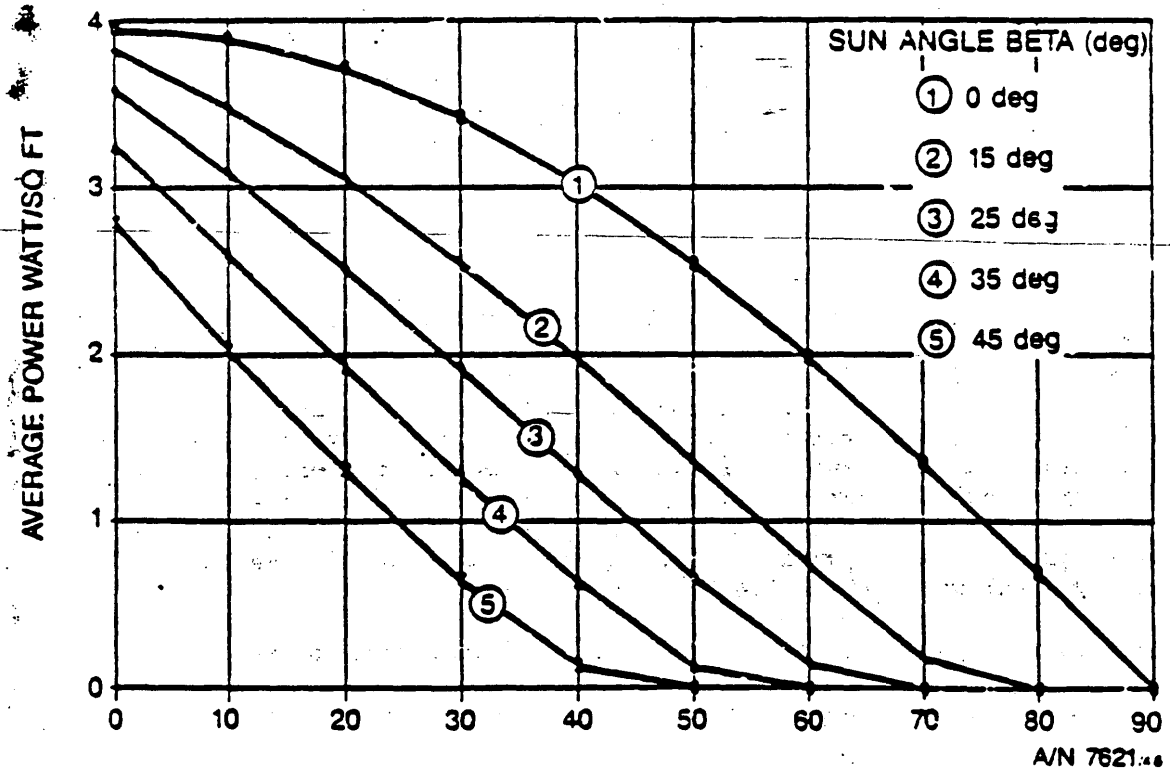


Figure 5-30 Power Per Sq. Ft. of Array Versus Beta for Various Tilt Angles (Panel Tilted Toward Orbit Normal Negative Angle 0° TO 45°)

5.1.2.1 SELECTED ARRAY CONFIGURATIONS

ROOF TOP ARRAY

The orbital average power output of the roof top array of Figure 5-18 can be calculated by using the information from Figures 5-26 through 5-30. Figure 5-31 shows this array configuration output when the spacecraft flies with the edges of the arrays in the velocity direction. The two arrays (positive and negative tilt) seem to complement each other and produce a flatter response vs. beta angles especially for the 25 deg tilt angle. However, if the spacecraft were to yaw by 90 deg, the array output would become as that in Figures 5-26 and 5-27 and would be unacceptable for the higher beta angles. Therefore, the array configuration must be symmetrical.

FLUSH MOUNT ARRAY

The flush-mounted solar array is always the most desirable from a structural and deployment standpoint. Figure 5-32 shows the orbital average power output from an array of four solar arrays around a box structure. The arrays have a 1.5W/ft^2 orbital output when the beta angle is zero and a much larger output when the beta angle is 90 deg. If solar arrays were now placed on the top of the spacecraft box, more output would be obtained for the low beta angles. Figure 5-33 shows the comparison of solar array outputs for four flush panels with their output, an equal-sized top array, and four flush panels with a top array twice the size of the side panels (weighted array). The curves are output per sq ft of array. The five panels will output more power than the four panels, but the criterion is the array efficiency. The weighted array has a higher average power and a much flatter response of output vs. solar angle.

MANSARD ROOF

The Mansard roof is an adaptation of the flush mount array. The arrays are tilted with respect to the spacecraft rather than flat as in the flush mount. Figures 5-34 and 5-35 illustrate the orbital average output of four panels tilted with respect to the spacecraft (no top panel). Figures 5-36 and 5-37 illustrate the orbital average output of a complete Mansard array configuration. The four panel arrays with a tilt angle of 55 deg seem to have the flattest power output response. The power output on an orbital average basis is about 3W per sq ft for a beta angle up to 80 deg and a minimum of 2.55. When the fifth panel or the top panel is added, the optimum tilt angle is closer to 75 deg which has a minimum output of about 2.4 W/ft^2.

If the average required spacecraft power was 66W, the four panel array would consist of 26 sq ft of solar array or four panels, each 6.5 ft sq. The five panel Mansard would require 27.5 sq ft of array or five

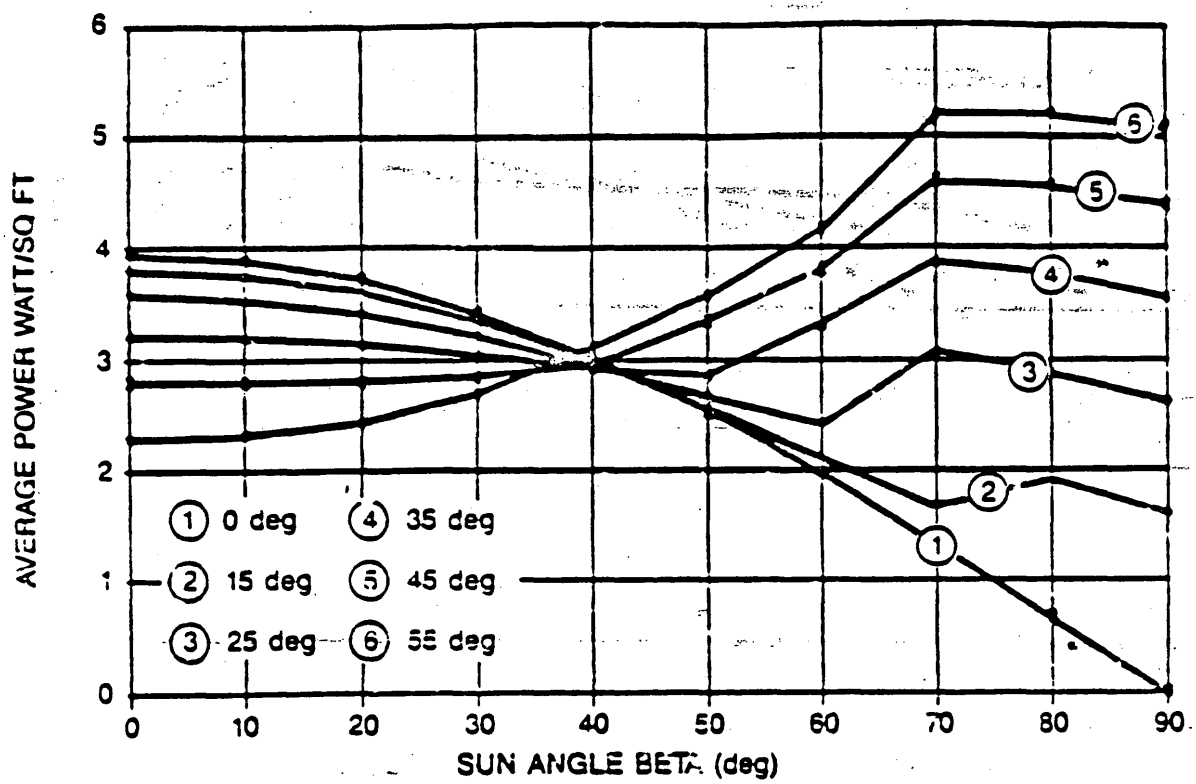


Figure 5-31 Power Per Sq. Ft. of Array Versus Beta for Various Tilt Angles (Roof Top Array 0° TO 55°)

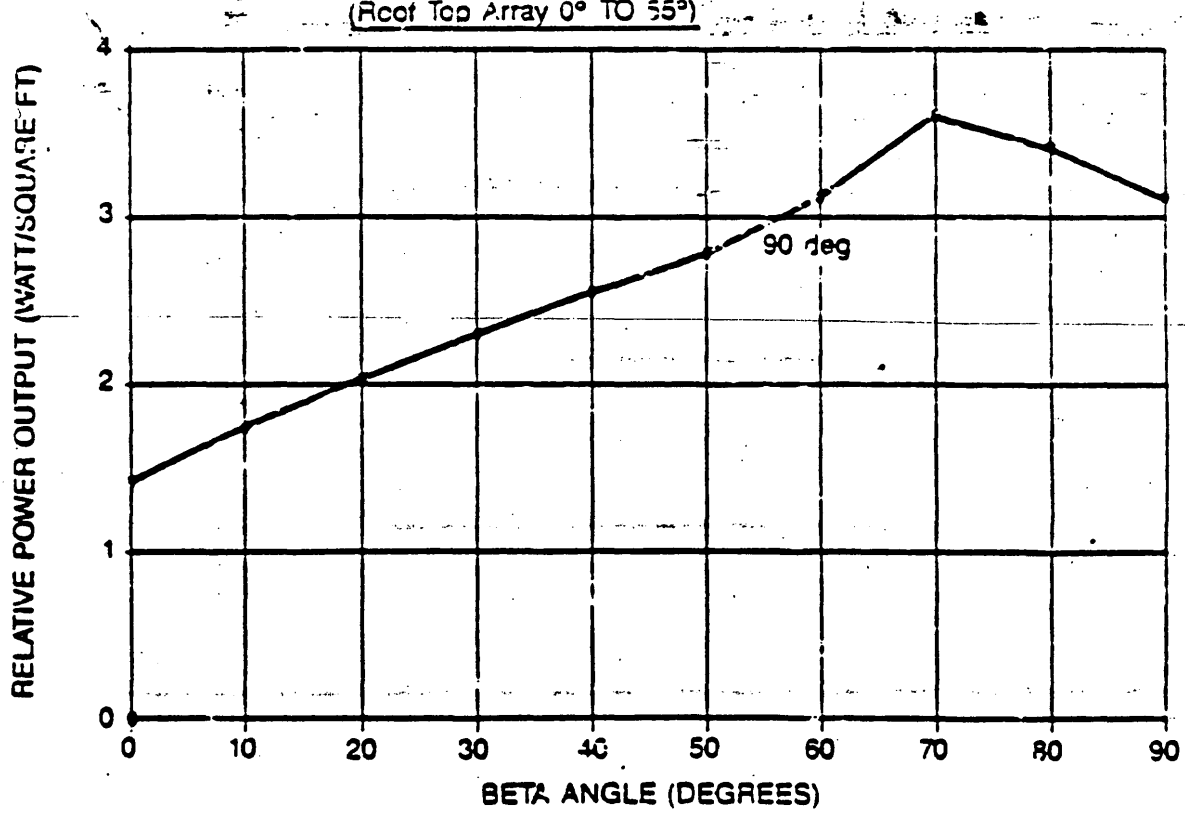


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Figure 5-32 Power Output Versus Outer Panel Tilt Angle (Power Output of Four Flush Panels)

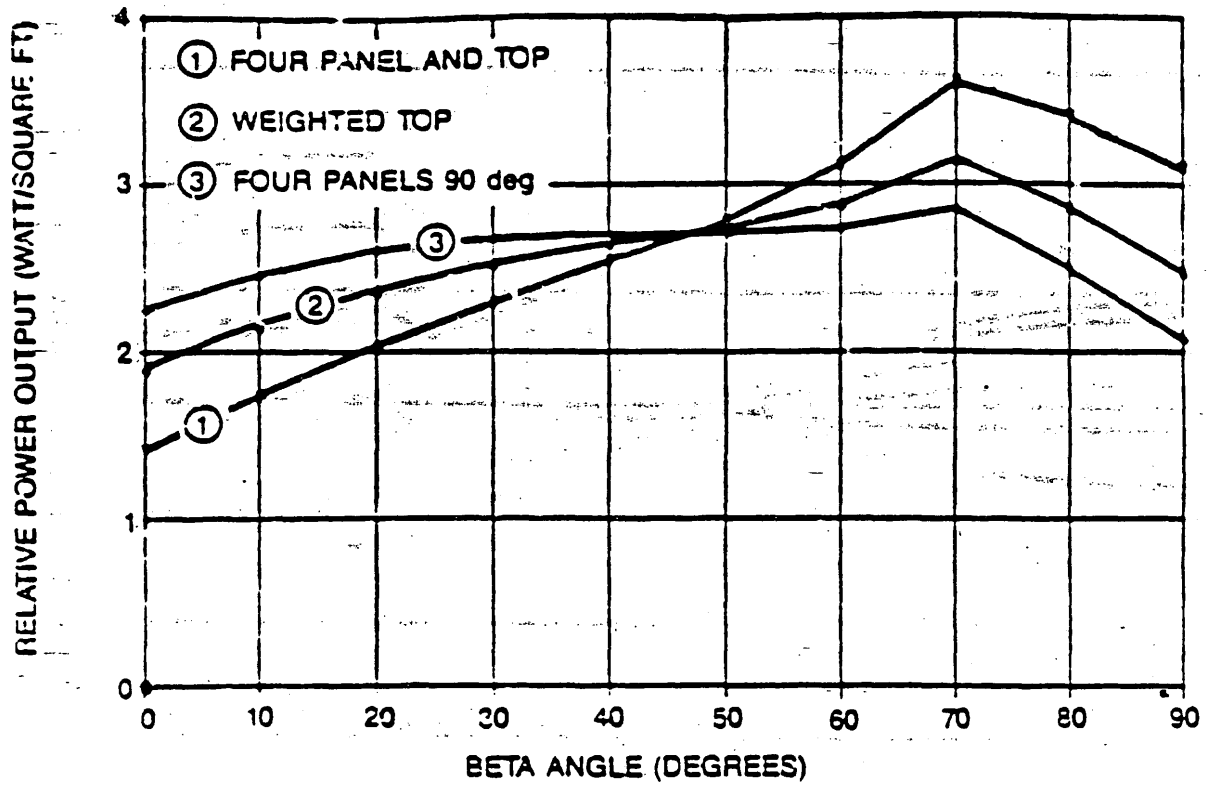


Figure 5-33 Power Output Versus Outer Panel Tilt Angle (Four Flush Arrays with Top Array)

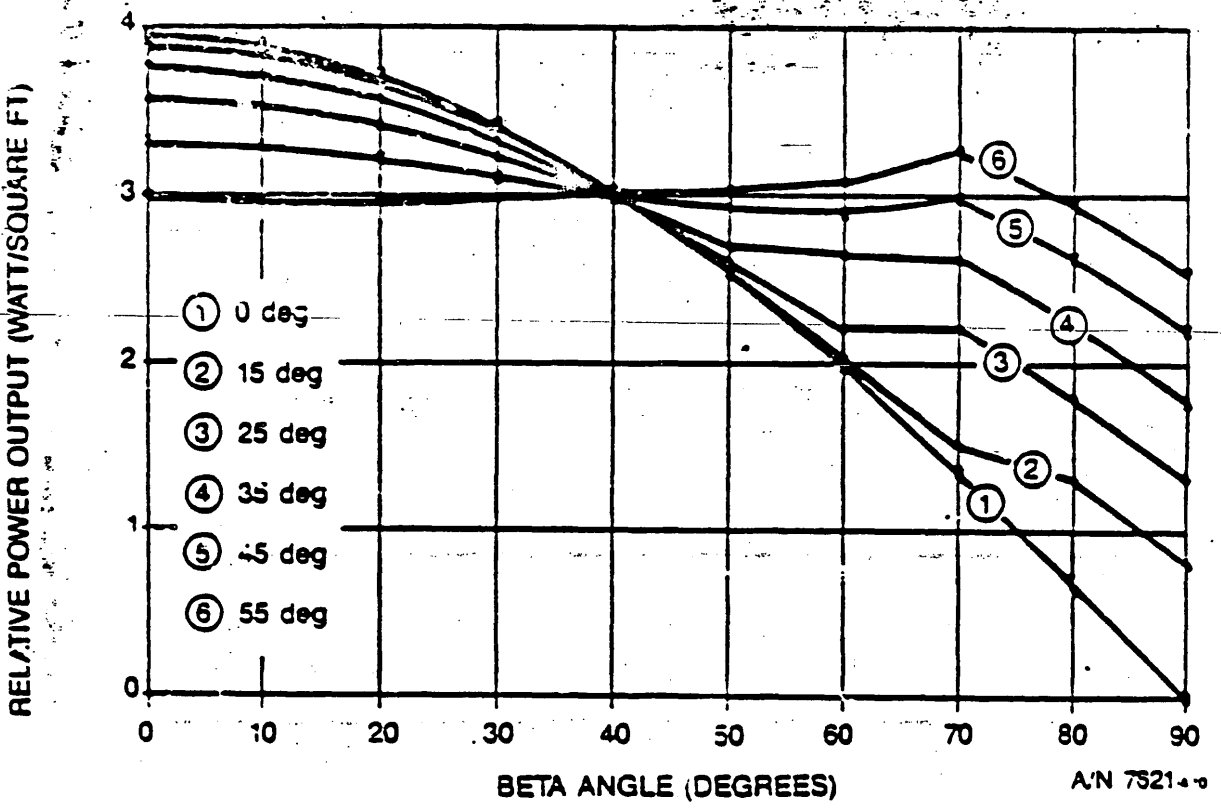


Figure 5-34 Power Output Versus Outer Panel Tilt Angle (4 Panel Tilted Array 0° to 55°)

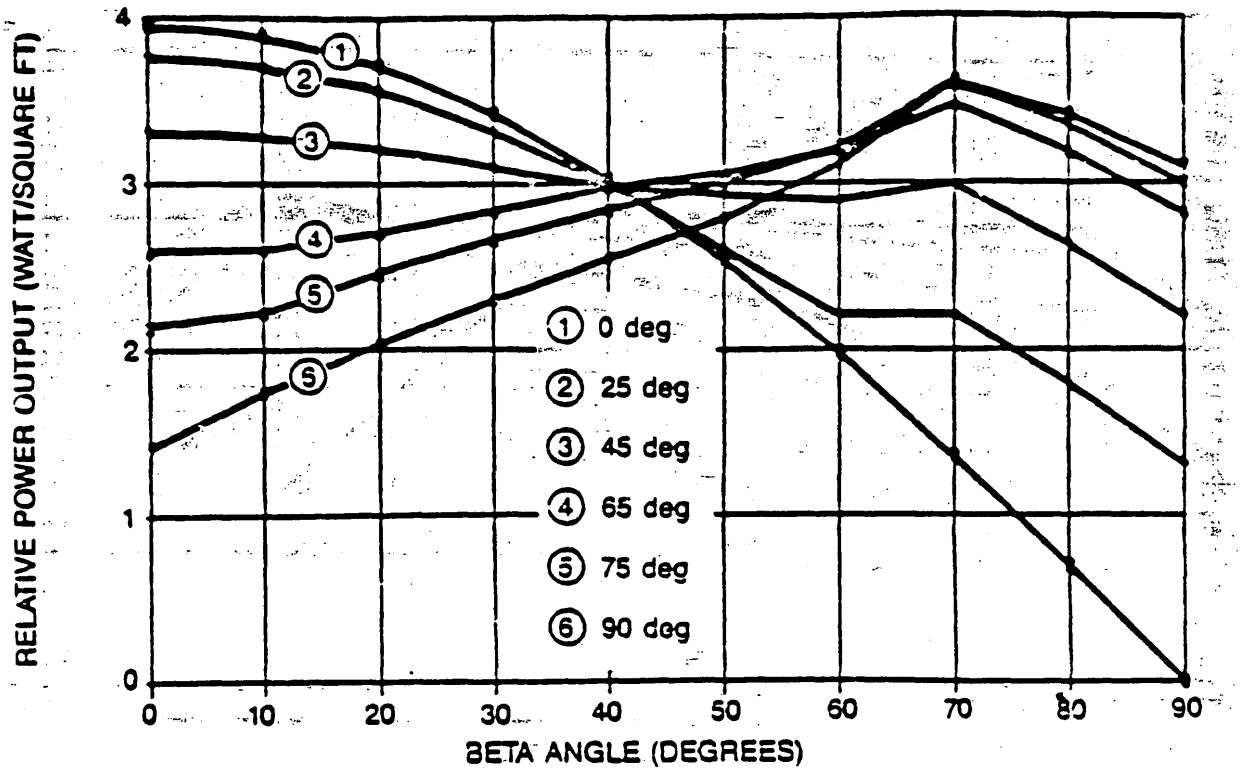


Figure 5-35 Power Output Versus Outer Panel Tilt Angle (4 Panel Tilted Array 0° to 90°)

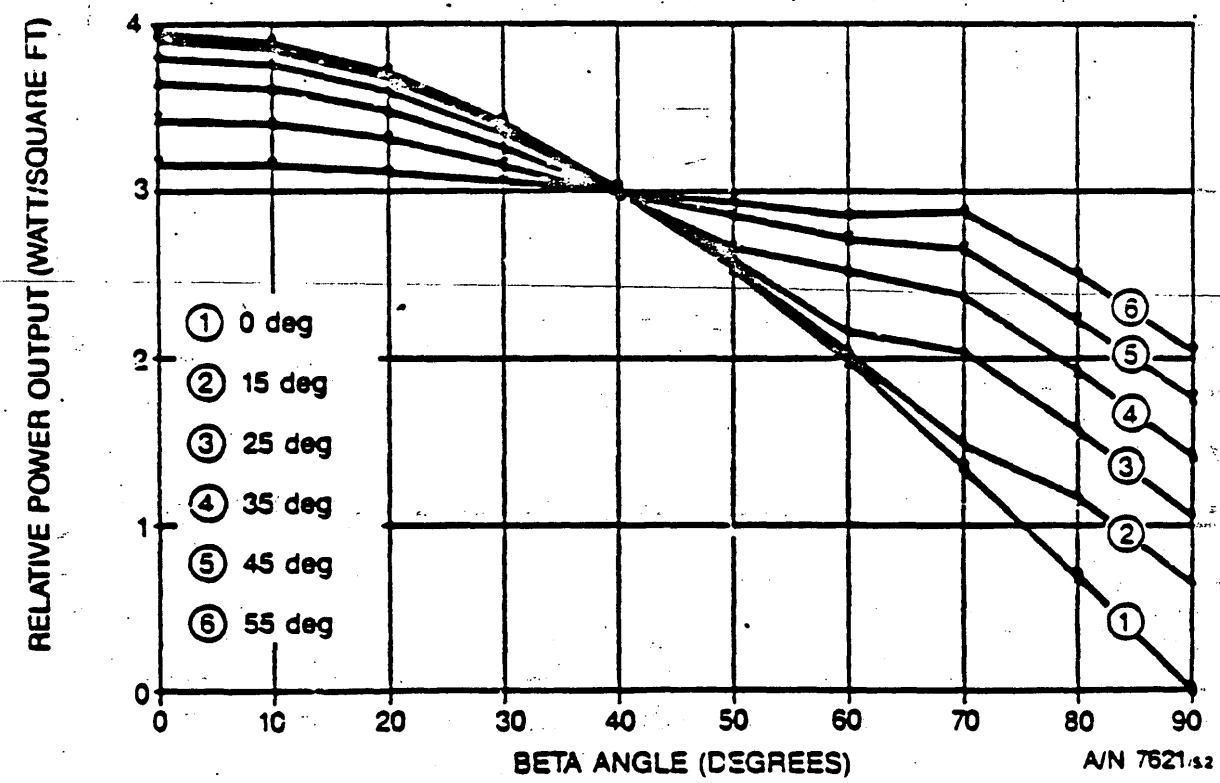


Figure 5-36 Power Output Versus Outer Panel Tilt Angle (5 Panel Mansard Array 0° to 50°)

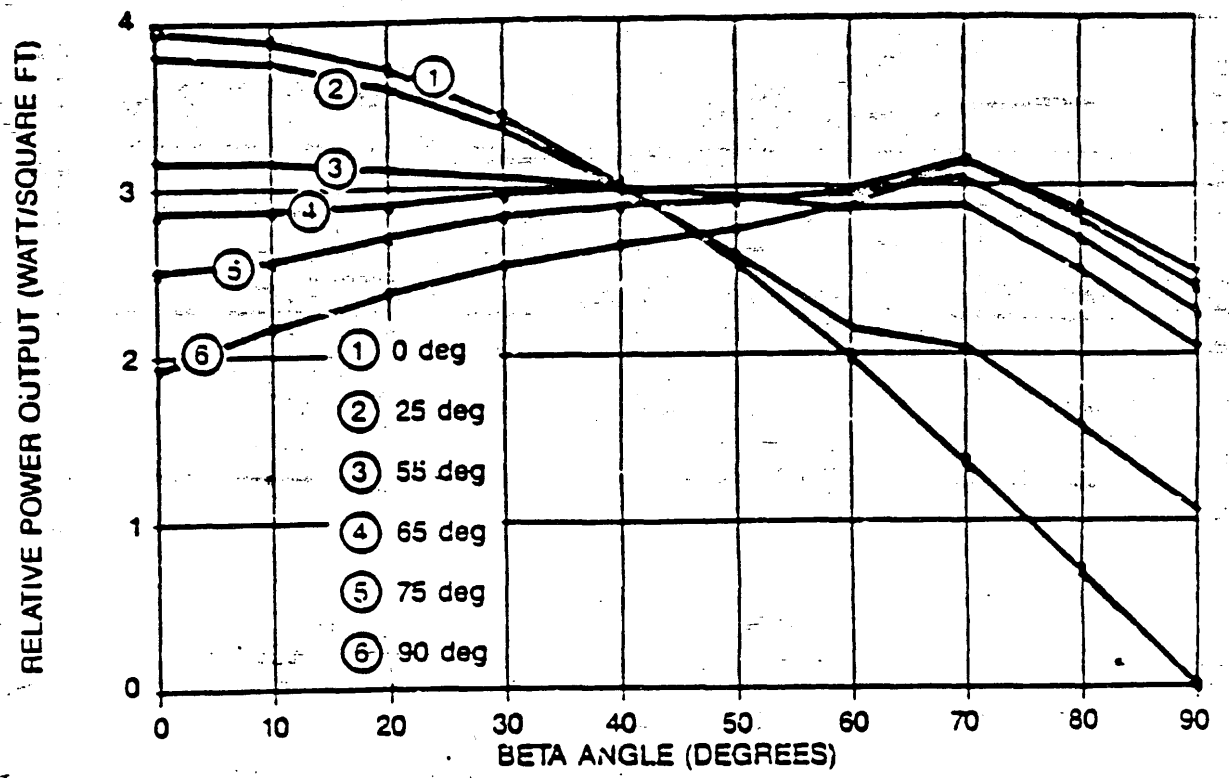


Figure 5-37 Power Output Versus Outer Panel Tilt Angle (5 Panel Mansard Array 0° to 90°)

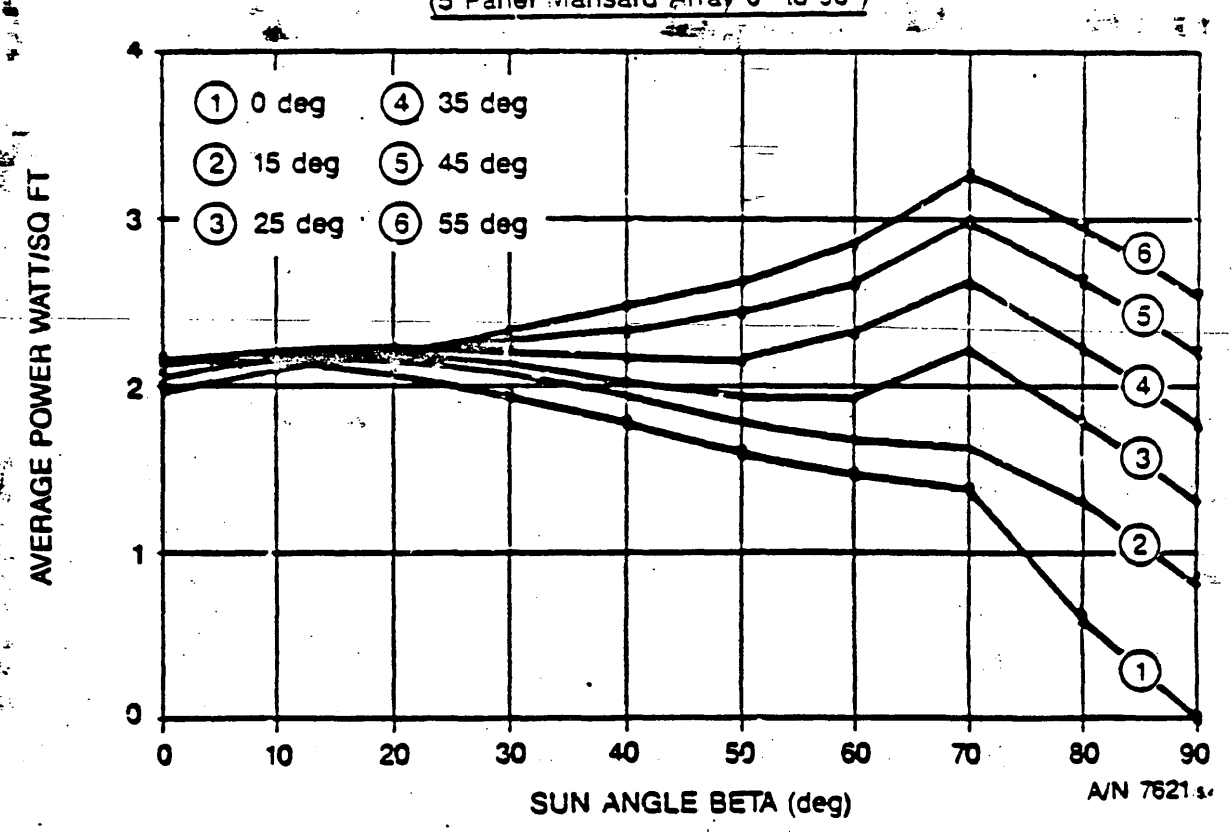


Figure 5-38 Power Per Sq. Ft. of Array Versus Beta for Various Tilt Angles ("Omni" Array Output 0° TO 55°)

panels (including the top panel) each 5.5 sq ft. The five panel Mansard is not quite as efficient as the four panel design, but the structure would be smaller producing a savings in launch cost compensating for the larger number of solar cells. The trade between a Mansard and four element array will also depend upon the spacecraft mechanical and deployment considerations.

OMNI PATTERN

The omni pattern solar array is the combination of two arrays of four panel tilted arrays (excluding the shadowing effect of the other solar arrays and the satellite). The first array is orientated upward and produces the same power outputs. The second array is orientated to nadir. The addition of the downward-looking set of solar arrays provides a more uniform power output over an extended portion of the orbit during which the upward arrays are beyond 90 deg to the Sun. The downward arrays do not produce much output for low Sun angles because the array is either blocked by eclipse or is facing the Earth rather than the Sun. When the Sun is at a high beta angle, the combination produces a uniform total power output independent of the spacecraft yaw orientation. Figure 5-38 illustrates the orbital average output of the omni array with the tilt angle from 0- to 55 deg. The optimum tilt angle is about 45 deg. The average output is 2.05W per sq ft of solar array (cells on both side of panel). This number is very close to the previous estimate of 1.9W per sq ft.

The advantage of the omni solar array is uniform total power output (excluding eclipse). The four panel array will produce a sine wave type of power output and the power controller will have to be capable of regulating the power output for varying power. The lower panels extend the power output cycle between eclipse region and the power output of the upward panels. The omni array has an additional advantage of producing about the same power output as if the spacecraft were inverted. The disadvantages of the omni array is that while the second set of arrays will increase the power output for the same panel area and smooth this power output, it is at the cost of using a set of arrays in a very low efficiency mode. The system power output per sq ft of solar array will be 2W per sq ft rather than 2.4 and this does not include blockage due to the satellite.

5.1.2.2 ARRAY OPTIMIZATION CONCLUSION

The solar array configuration must be a balance of cost and performance, especially for the multiple spacecraft. The cost of an additional battery and more expensive power controller is much less than the cost of solar arrays and deployment mechanisms. The solar array configuration must also be symmetrical and must provide the minimum frontal area to reduce spacecraft torques due to aerodynamics. The four panel Mansard array with a tilt angle of 55 deg will provide this

balance. The array size for a 55W system, assuming a 12.44 array constant and a 1.1 charge efficiency, would be:

$$55/(2.55/1.1) = 23.7 \text{ sq ft}$$

or each panel would be about 6 sq ft (30 in. x 30 in.)

The omni array area required would be:

$$(55/(2.05/1.1)) = 29.5 \text{ sq ft}$$

or each of the four panels would be about 7.4 sq ft (32.6 in. x 32.6 in.) with solar arrays on both sides.

The frontal area of the omni would be almost twice the flat array. The weight, deployment mechanism, and doubling the number of solar cells would drive up the omni cost.

The final conclusion of the size and shape of the solar arrays is a compromise between the mechanical configuration of the satellite for minimum cost through launch. The design must consider the projected temperatures which modify the array constant (12.44W/ft^2), the size of the antenna for MSSP, and the area-to-mass ratios for the gravity gradient boom stabilization. The curves presented in this analysis can be used in the final satellite configuration.

### 5.1.3 POWER SUBSYSTEM AND CONTROLLER ANALYSIS

The power requirements of the MSSP spacecraft were narrowed during the MSSP second working group team meeting to 55W. (During the third working group meeting, the power level was increased to 75W. This analysis was performed with 55W, but the conclusions are still applicable.) With this information, the power subsystem configuration could be advanced. The design cannot be finalized, but several areas impacting MSSP cost can be identified. This section will state the basic requirements of any satellite power subsystem, the MSSP power requirements, and analyze the possible subsystem design.

The three main areas of design for the power subsystem are power controller, battery size, and solar array configuration. The solar array configuration, discussed in the previous section, is a major driver in the selection of battery size and the power controller. This section will analyze the MSSP battery and power controller.

#### 5.1.3.1 BASIC TENETS OF POWER SUBSYSTEM DESIGN

The following tenets are the areas of design crucial to the performance and life of the power subsystem.

BATTERY SIZE: The battery size is controlled by the amount of power (current) outputted by the solar array at maximum output minus the minimum load current required by the satellite. The maximum current, which can be controlled by various means, must be less than the battery specified charge rate (C/x). The batteries must also be large enough to supply power for night spacecraft operation within the allowable state of battery discharge.

SOLAR ARRAY SIZE: The solar array size must be large enough for the daily orbital average power worst case to be greater than the daily power usage.

BATTERY FAILURE: Battery failure is strictly a function of the battery temperature. If allowed to become too hot the battery will fail. The battery thermal control will require the satellite surface finishes to allow the satellite to run cool. The battery charge control will have to limit battery overcharge to limit battery heating (battery size).

BATTERY LIFE: The battery life is a function of the depth of discharge that the battery is repeatedly exposed to and the total number of discharge cycles. An optimum level of depth of discharge is 80 to 90 percent. Depths greater than these will greatly shorten the expected battery life.

The design concerns to be addressed can now be listed in order of priority.

SATELLITE POWER SUBSYSTEM CONCERNS:

1- Satellite Power Required
2- Solar Array Power Output Variation With Orbit
3- Satellite Load Variation
4- Battery Weight And Size
5- Satellite Power Control System

The satellite power subsystem concerns are listed above. The first is the magnitude of the power required for the satellite, antenna, and radio. It also determines the size of the solar arrays. The second concern, the solar array power output variation, (addressed in Section 5.1.2) will require sizing the solar arrays so that the output is sufficient to supply the required power on an orbital average. The satellite power load variation will require that either the satellite batteries are large enough to prevent large overcharge currents, or more costly charge current controls will be needed. The battery size and weight, and the satellite power control system are the two parameters which must be negotiated. The battery size and weight, which impacts satellite launch cost, must be traded with the additional cost of an expensive battery charge control system.

### 5.1.3.2 MSSP SATELLITE REQUIRED POWER

The following power levels were defined during the second working meeting. Although the requirements changed later in the study, the analysis from the second working group levels will be used here. These numbers will not detract from the final conclusion of the analysis. Section 5.1.5 discusses the final design conclusions with final power requirements.

| Item | Power avg watts |
|---|---|
| Antenna | 5 |
| Radio | 30 |
| Processor/antenna | |
| Processor/baseband | |
| Satellite | |
| Power | --- |
| Thermal | --- |
| AC&DS | 10 |
| C&DH | 10 |
| Total | 55 |

### 5.1.3.3 BATTERY SIZE

The satellite battery size is dependent upon power control techniques. The simplest approach is to use current limiting, where the battery capacity divided by 10 (C/10) is equal to the peak charge capacity. This approach requires large batteries. The second approach uses a power controller to limit peak charge dependent upon the battery temperature and sizes the battery capacity at C/2. This second approach allows smaller battery capacity, but causes other problems discussed later.

The battery sizing will be dependent upon the acceptable charge rate and the solar array output capacity. From Section 5.1.2.2 with a 55W system, the solar array size is 24 sq ft. The peak array output with panels tilted 55 deg is:

$$24 \text{ sq ft} \times \frac{12.44\text{W}}{\text{ft}^2} \times \cos 55 \text{ deg} = \frac{5.12 \text{ amp}}{33.4\text{V}}$$

The batteries will first be sized with a power controller so that the peak charge capacity can be C/2. This controller would limit the charge depending upon the battery charge as indicated by the battery voltage and temperature. But, the battery is sized by the max charge rate.

C/2 = max array output - min load

C/2 = 5.12 amphr -55W/ 33.4V

C = 7.0

The simpler power control method, where battery current limiting is used to control the battery charge, would require that the battery capacity be determined by the same equation as the temperature-controlled unit, but with a maximum charge rate of C/10.

C/10 = [2.1 * 11W/ft^2 * 5.45 ft^2 -55] W / 33.4/V

or, C = 21.2 amphrs.

The battery is subjected to a trickle charge once a full charge is obtained. The simpler power control approach requires larger batteries, which increases weight and could increase satellite size.

The trade between the two controller approaches is cost and weight of the additional batteries vs. cost of the controller design, fabrication, and reliability. If the complex power controller is chosen, then the type of controller must be chosen. Another parameter affecting the trade is satellite lifetime or battery depth of discharge. The battery size must be sufficient to allow the depth of discharge of the batteries to be within the desired levels. The depth of discharge will occur during the night-time of the orbit, or for 32 min of a 96 min orbit (32 of 96 or the worst case beta angle must be chosen for this calculation). The depth of discharge will be:

55W/28V * 32 min * 1 hr/60 min = 1.05 amphr

For a 4.24 amphr battery, the depth of discharge will be 1.05/4.24 or 25 percent. Effects of the size of the battery and the depth of discharge can be seen in Table 5-2.

TABLE 2 DEPTH OF DISCHARGE VERSUS BATTERY SIZE

| DEPTH OF DISCHARGE (%) | BATTERY SIZE REQUIRED (AMPHR) |
|---|---|
| 5 | 21 |
| 10 | 10.5 |
| 15 | 7 |
| 20 | 5.2 |
| 25 | 4.2 |

If the depth of discharge is required to remain below 5 to 10 percent in order to extend the spacecraft life time, the battery size must be expanded to between 11 and 21 amphr.

The battery size needed to ensure low depth of discharge and satellite lifetime is basically the size needed for the simple power control. Both the satellite lifetime and controller requirements are met by using the larger size batteries. Because the battery size comes in units of 6 amphr, the optimum battery size would be 24 amphr. The final sizing of the batteries and power controller will depend upon the final power system requirements. The preliminary conclusion is that the simpler power controller with larger longer life batteries is the best choice for the MSSP. It will mean that no network processing capability will be required for the power system control.

### 5.1.4 POWER SUBSYSTEM TRADES

As part of the MSSP Phase I study, various techniques described in this section, were investigated as either new technology or techniques to reduce the cost of the MSSP satellites. The MSSP extra battery storage describes the possible use of larger batteries to filter the duty cycle of the transmitter on/off operation. The solar array trade describes an approach to reduce the amount of solar array by orientating the satellite and yawing the spacecraft 180 deg every 37 days or so. The larger spacecraft battery trade analyzed the use of larger batteries to lower the required solar arrays. Although the results of these analyses were not positive for the MSSP, the analyses are presented as part of the work performed during the MSSP Phase I study and as part of the open-minded approach in solving the MSSP low cost objective.

#### 5.1.4.1 MSSP EXTRA BATTERY STORAGE

The preliminary analysis of the power subsystem determined that the lowest power system was also the lowest cost. Although this was a fairly obvious conclusion, it is difficult to operate on a 40W power subsystem when the radio requires an average of 80W. A possible solution is to use extra battery storage.

Extra battery storage would filter the variations of system power required. The 80W of power will not be required continuously but depend upon how active the MSSP system is. For example, if 80W is used continuously for 20 min and then not used for 170 min, the long term 2-orbit average power is actually lower than 80W. If the non-transmitting power required is 30W, the average power would be 35W. The differential in power (30 to 80W) for the 20 min could be made up by using a larger battery which filters the power requirement fluctuations: a battery filter.

The equation for the sizing of the extra battery is:

$$\frac{(\text{Peak power} - \text{system power}) * \text{duty factor} * 3.2 \text{ hours}}{30V * \text{discharge capacity}} = \text{amphr}$$

where:

Peak power: required power during transmit

System power: nominal power system capability

Duty factor: percent of transmit time on two orbit basis

Discharge capacity: allowable depth of battery discharge

3.2 hours: ~2 orbits ;96 minutes per orbit

30: system voltage

For a 40W power system (orbital average), a 30W non-transmit power, 80W transmit power average, and a 20 percent duty factor over two orbits (3.2 hours), the additional battery capability with a 20 percent depth of discharge would be:

$$\frac{(80-40) * 0.2 * 3.2}{30V * 0.2 \text{ dod}} = 4.3 \text{ amphr}$$

Figure 5-39 shows the required battery size for transmit duty factors from 0.1 to 1 with various depths of discharge. The figure shows that for a selected battery size, if the duty factor were to increase intermittently, the depth of battery discharge would increase. The depth of discharge is regulated to increase the battery life. However, intermittent (in terms of days) depths of discharge as great as 80 percent would not greatly decrease battery life. Therefore, the battery filter is a compromise that can be stretched as needed.

The system must be able to recharge the battery filter between periods of high active duty. This capability is balanced when

(system power - nontransmit power) * 3.2 =

(transmit power required - nontransmit power) * 3.2 * duty factor

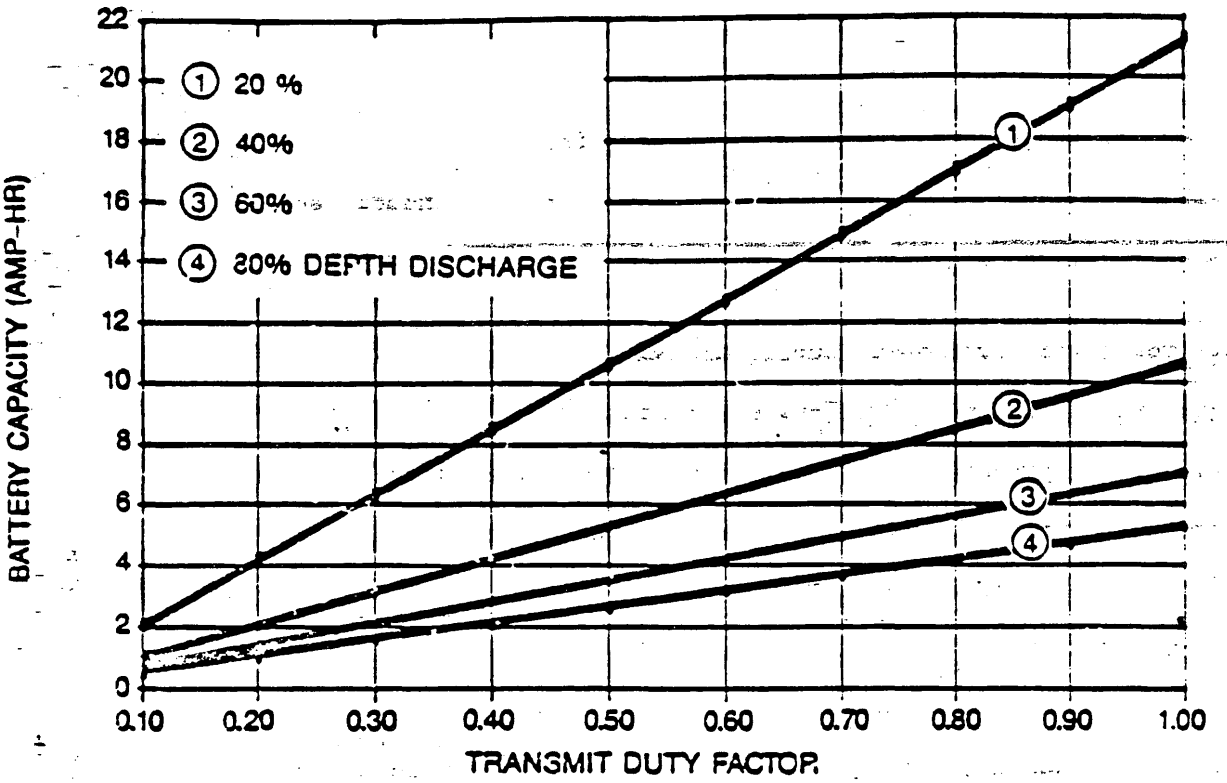


Figure 5-39 Battery Capacity Needed Versus Transmit Duty Factor (Depth of Discharge 20 to 80 Percent; Power System 40W, Transmit 80W, Non-Transmit 30W)

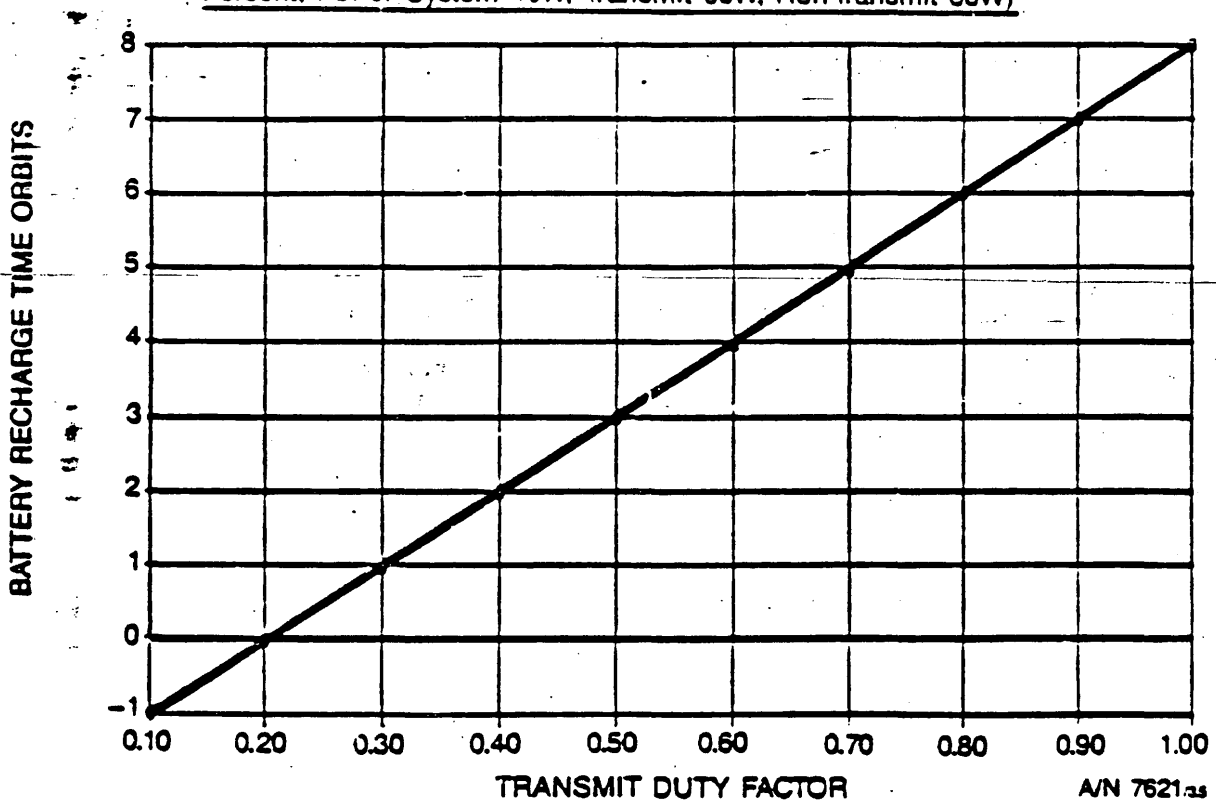


Figure 5-40 Battery Recharge Time in Orbits Versus Duty Factor (After Two Orbits At Duty Factor (40, 80, 30W System)

For the previous example:

$$(40-30) * 3.2 = (80-30) * 3.2 * 0.2$$

The system is balanced with a 20 percent duty factor. When the system operation is beyond 20 percent, several orbits will be required to recharge the batteries. Figure 5-40 shows the recharge time, in orbits, required vs. the duty factor of a two orbit time frame. The time required to recharge is obviously too long to expect the spacecraft to be "out of the system".

A possible first cut for MSSP would be a compromise of cost with inputs from other team members (radio and antennas). The compromise chosen expands the power system to 50W and uses a 40 percent duty factor. Figures 5-41 and 5-42 are plots for this system. System parameters are:

System power: 50W orbital average

Required power (non-transmit): 30W

Transmit power (total): 80W

Solar array size (orbital): 50W

Battery size (filter included): 12 amp/hr

Transmit duty: 40 percent

Depth of discharge: 20 percent

100 percent duty factor depth of discharge (two orbits): 55 percent
Recharge 100 percent duty factor: three orbits

The important parameter is that the relative duty factor will have to be defined. This factor is a cost/performance trade that must be made to lower satellite cost. A final conclusion to this trade was not obtained because the communications system parameters such as duty factor were not completely defined during Phase I of the MSSP study. The trade is presented as a possible aid during Phase II.

5.1.4.2 SOLAR ARRAY TRADE

The most costly subsystems for the spacecraft are attitude control and power. The power subsystem consists of solar arrays, batteries, and a power controller. The solar arrays comprise 70 to 85 percent of the cost of this subsystem. If a trade were made to reduce the cost of this subsystem, the emphasis must be placed on reducing the cost of solar arrays. This analysis will discuss two possible opt

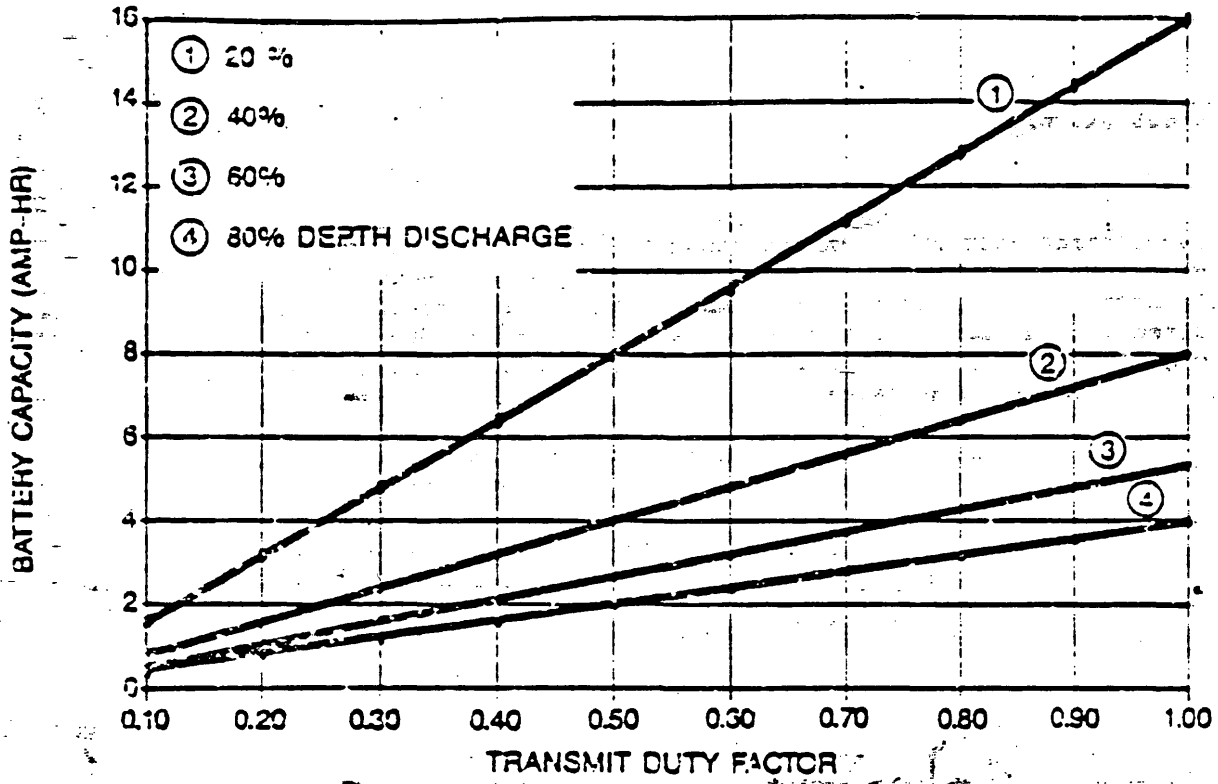


Figure 5-41 Battery Capacity Needed Versus Transmit Duty Factor (Power System 50W, Transmit 80 and Non-transmit 30W)

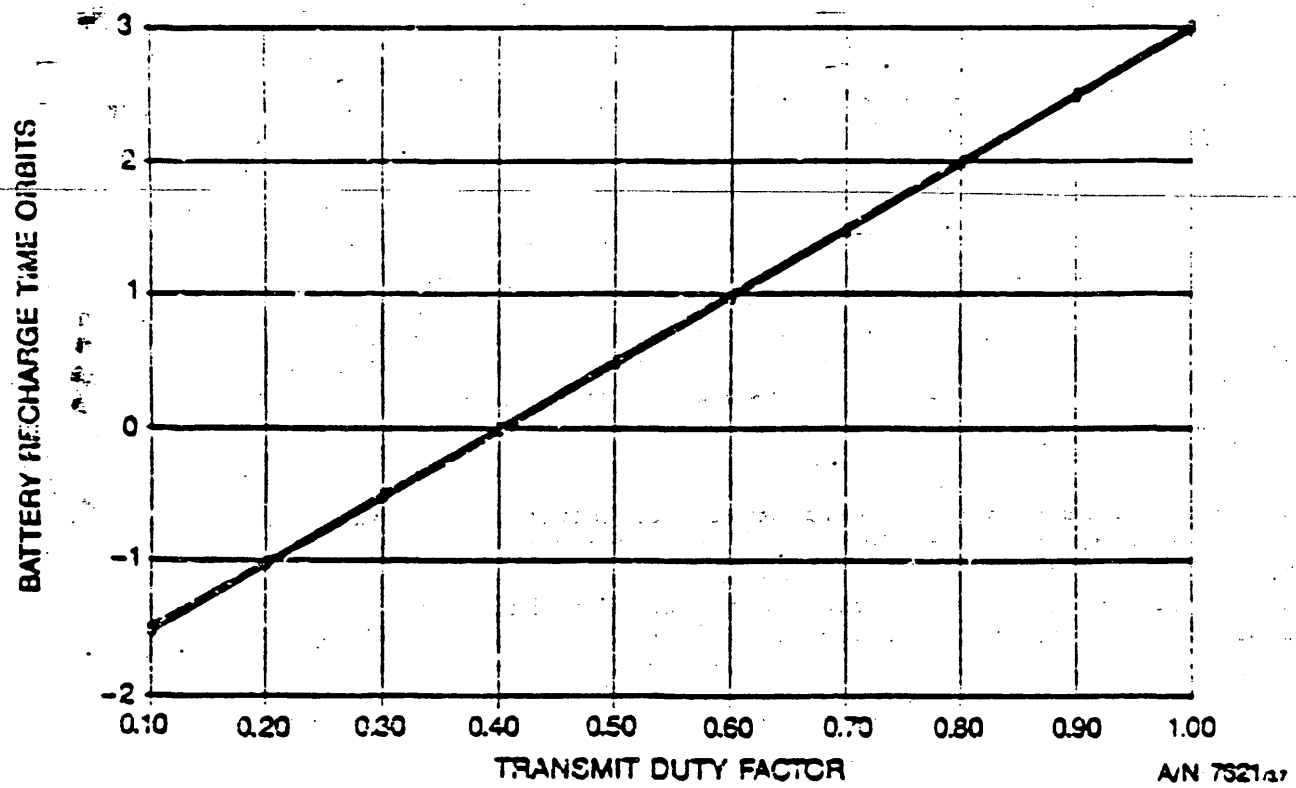


Figure 5-42 Battery Recharge Time in Orbits Versus Duty Factor (50, 80, 30W System)

orienting the arrays and spacecraft to increase the array efficiency, and using larger batteries to overcome the array efficiency during low beta angles. The spacecraft and array orientation is presented with a possible savings, but it is not entirely clear that this option should be selected for MSSP. The cost of the required attitude control does not negate the possible array savings for a 75W system. The larger battery option was analyzed and found not to be advisable for MSSP.

5.1.4.2.1 ARRAY ORIENTATION DETERMINES ARRAY EFFICIENCY

The solar array analyzed for the MSSP consisted of panels arranged around the spacecraft such that the average power was maximized at 2.5W per sq ft on an orbital average. This array configuration is the most efficient when the MSSP is designed with no yaw stabilization. A more efficient array configuration (more watts per square foot independent of beta angle) can be obtained only if the solar arrays are oriented toward the Sun. This orientation can be accomplished in two fashions: mounting the arrays on a gimbal that is pointed at the Sun, or rotating the spacecraft such that the array is pointed toward the Sun. The first option is not entirely viable because the gimbal approach has several negatives - mainly cost, reliability, and lifetime of the mechanical system.

The second option has benefits, but also cost and performance trades. This option involves rotating spacecraft in a fashion similar to the NASA/BASD Earth Radiation Budget Satellite (ERBS). The spacecraft is rotated in yaw 180 deg every 37 days (for a 57 deg inclination angle). In simple terms, if the spacecraft has four solar panels facing forward, aft, to the right, and to the left, and if the Sun were to the right, then the performance of the panel facing to the left would detract from the efficiency of the system. Likewise, if the Sun were to the left of the satellite, the right panel would detract from the efficiency. The efficiency of the arrays could be increased if the satellite and panels were rotated to orient the panels toward the Sun and "delete" the panel on the other side of the spacecraft.

Before further description of satellite rotation, the phenomenon of the Sun beta angle should be described. Figure 5-43 shows a spacecraft orbit around the Earth relative to the Sun. If the Earth were a perfect sphere, the orbit plane would be fixed relative to inertial space and the angle to the Sun would rotate once per year. However, the same phenomenon that causes the nodal regression of the satellites for dispersion will cause the orbit plane to rotate. The orbit plane rotates with respect to the Sun by the rate.

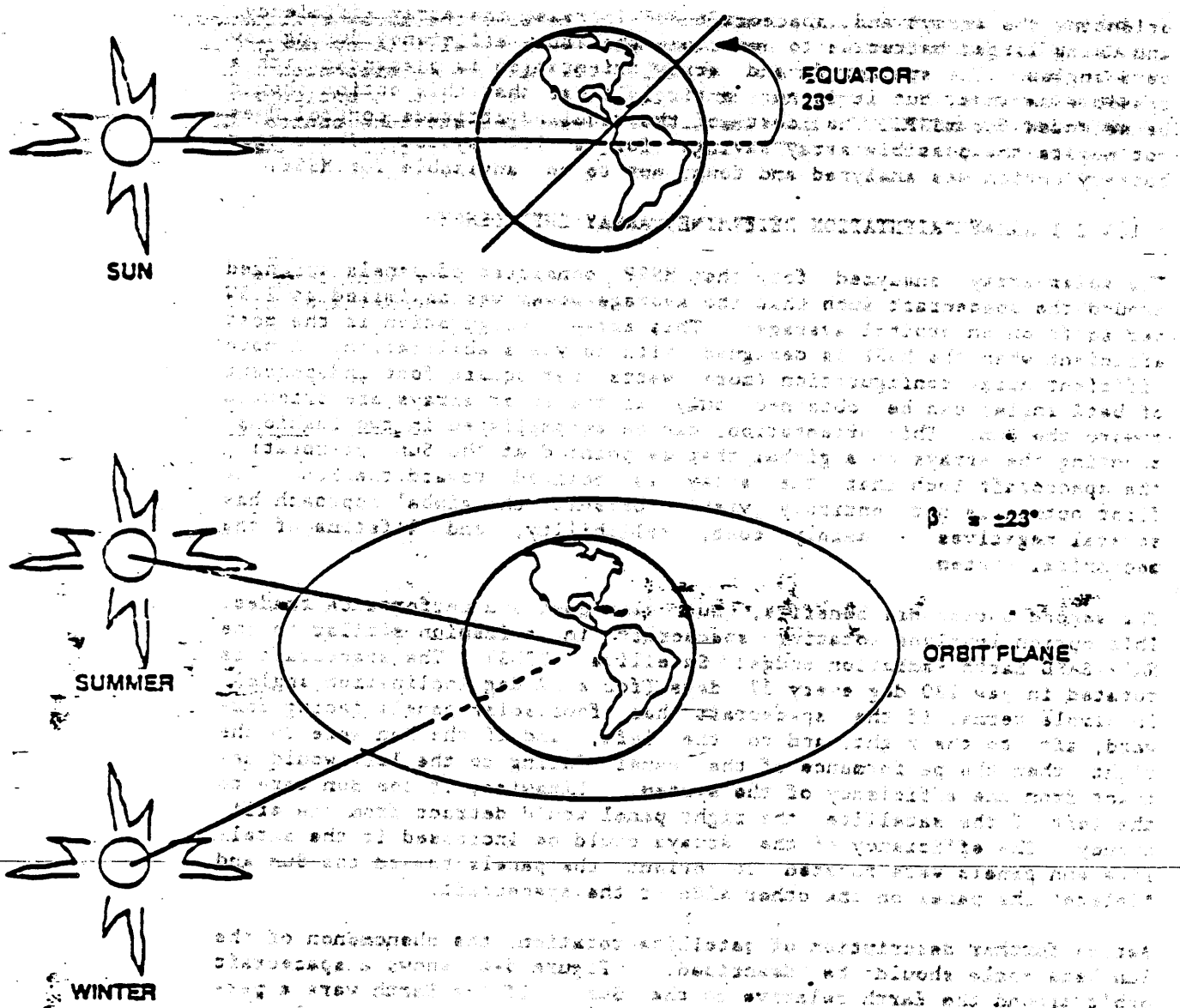


Figure 5-43 Sun β Angle for Inclination Angle

$$\omega = \frac{360}{365} + [R_e/R_e+h)]^{7/2} \cos A_i$$

$R_e$ = Earth radius; h = altitude; $A_i$: orbit inclination angle.

The angle from the orbit plane to the Sun will be of the form:

$$\beta = 23 \sin \alpha t + A_i \cos \omega t$$

Twenty-three deg is the obliquity of the ecliptic or tilt of the equatorial plane out of the ecliptic which results in a seasonal oscillation (summer/winter), and Ai refers to the inclination angle of the spacecraft orbit. Dividing 180 deg by the orbit rate gives approximately 37 days for a 57 deg orbit (approximately 81 days for 80 deg orbit). For the rest of this analysis, 37 days will be discussed. It must be recognized that different delays will occur for different orbits. If an observer were standing on the top of the MSSF, the Sun would appear to be on the left and then rise overhead and continue until it was on the right; it would then scan from the right back overhead to the left. Actually, every time the Sun passes overhead it is at opposite sides of the orbit (if a reference mark existed on the orbit), but regarding the solar arrays, it does not matter.

The angle of the Sun out of the orbit plane or beta angle will vary, as shown in Figure 5-44. The horizontal axis is time in months; the vertical axis is beta angle, both plus and minus. The curve shows a high-frequency ripple, which is the rotation of the orbit plane around the Earth and a low-frequency ripple, which is the +/- 23 deg of the ecliptic plane. The curve shown is for a satellite at a 28.5 deg inclination.

Figure 5-45 shows the power output of various solar panels. The bottom curve shows the output of a four-panel array tilted down 55 deg. This array outputs almost a constant power independent of beta angle. When the beta angle is zero, the output is reduced because of the 55 deg tilt. For high beta angles, the output is reduced because only one panel is in clear view of the Sun. The other curves show the outputs of single panels facing toward the orbit normal and tilted down 25, 45, 55, and 65 deg. These array outputs would be obtained only if the satellite could turn around and fly backwards every 37 days. The turnaround would orient the array to the right or left depending on the angle of the Sun.

The first item to note from the one-panel curves is the high output for high beta angle because the total array is on one side of the satellite facing the Sun. The second item is that for a tilt angle of 25 deg, the minimum output at beta equal to zero is 3.5W per sq ft. This output is 1.4 times the output of the four-panel array (3.5/2.5). In other words, by the simple rotation of the satellite every 37 days, the size

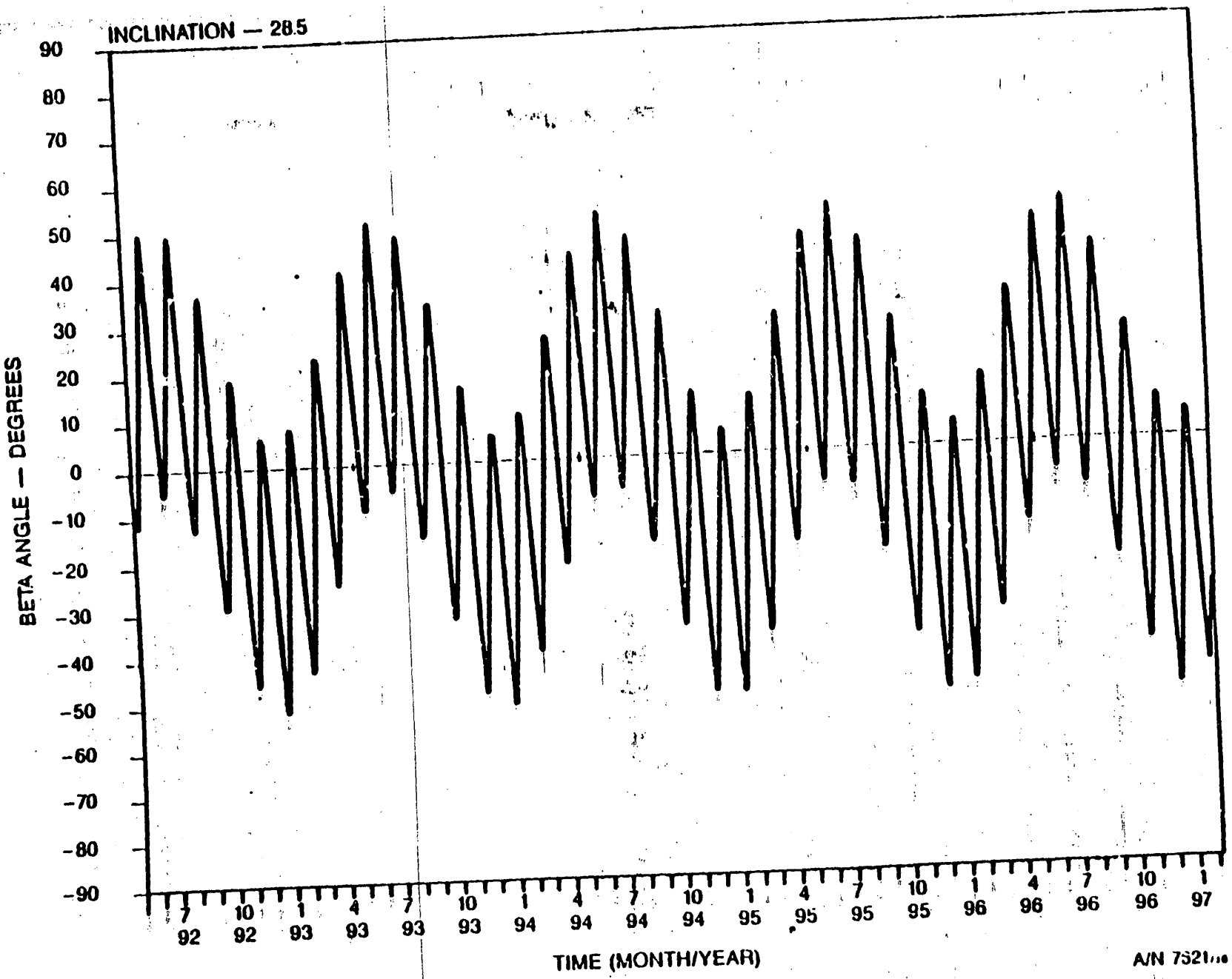


Figure 5-44 Example of Beta Angle Variation

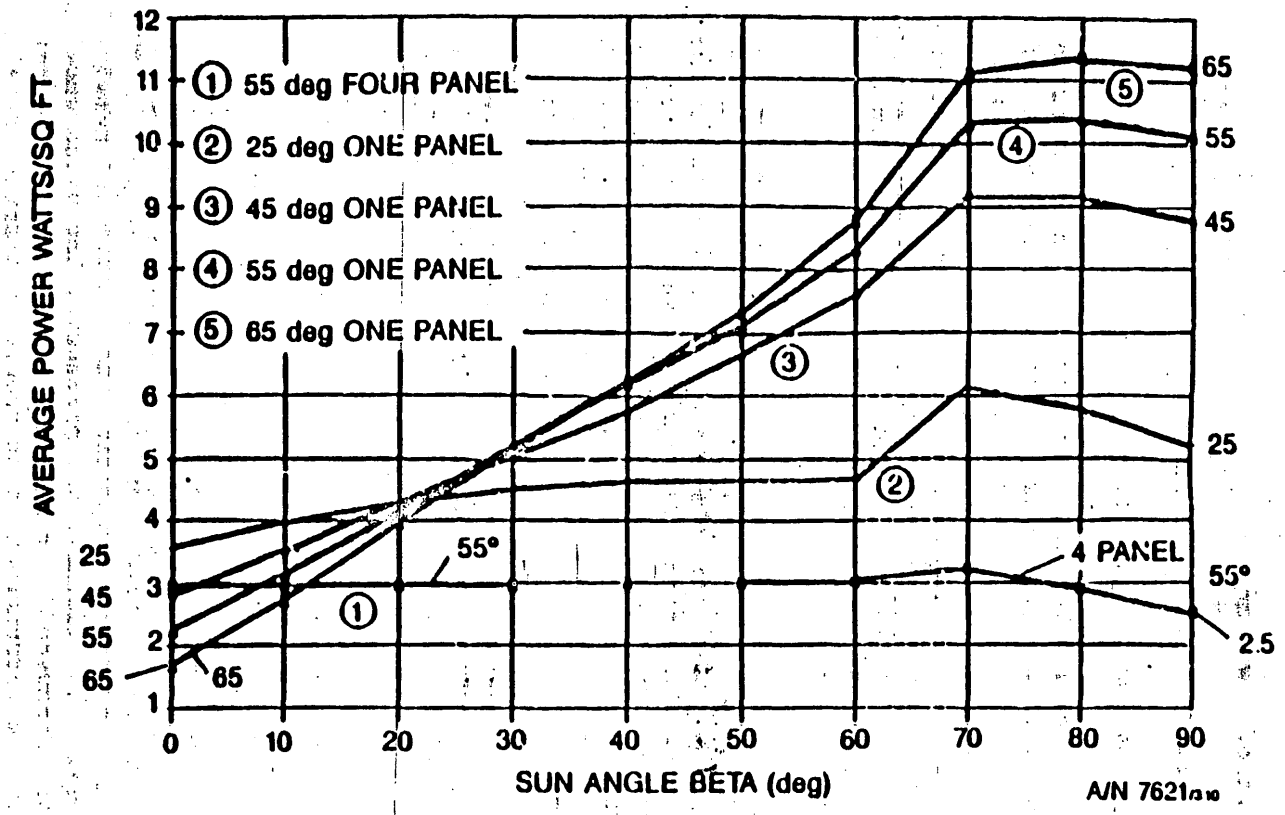


Figure 5-45 Power of Single Array Panel Versus Beta for Various Tilt Angles

of the solar arrays could be reduced by 30 percent. Since the array cost is 70- to 85- percent of the power subsystem cost, this would reduce the cost of the subsystem by 20- to 24 percent.

The problems with the yaw-around system are:

- o Required yaw control
- o Required yaw-around technique
- o Required control software

The required yaw control refers to the yaw control capability of the spacecraft. The spacecraft must be oriented toward the right or left with a +/- 20 deg orientation. The yaw control can be this unrestrained because a +/- 20 deg variation will not have much effect for the low beta angles. Also when the beta angle is large the projected solar array will be reduced, but the array output capability is large. Since the array would be deployed and oriented such that its long direction is in the satellite velocity direction, the array would tend to stabilize the spacecraft in yaw.

The required yaw-around technique refers to the means of causing the satellite to rotate 180 deg in yaw every 37 days. This could be accomplished by placing a wheel on the satellite with its spin axis oriented nadir. The wheel would spin approximately every 37 days for a certain number of revolutions (depending upon the final satellite weight and physical characteristics). The wheel would then be shut off and the satellite would rest at the new yaw position. The on/off use of the wheel would greatly extend the wheel's life, and its design would be much simpler than that for an attitude control wheel. Rotation of the satellite could extend for days because rotation is necessary when the beta angle nears zero deg impact. This wheel should also cost much less than an attitude control wheel. If the wheel cost $50K and the array savings were 22 percent of a $400K power system, the satellite could save $40K.

Two other options exist for yaw-around:

- o With a pitch wheel
- o With a magnet

Each of these options provides a viable approach for yaw-around. However, more analysis is necessary to define the amount of network processor control necessary to produce the procedure. The pitch wheel cost would be the $50K discussed for attitude control. The magnet would simply be a wound electromagnet that should cost approximately $100. The on/off spin wheel cost is therefore in the middle.

The yaw-around software refers to the added capability required of the network processor, which would be required to time the approximately 37 days between rotations and then command and count the wheel rotations. Figure 5-46 shows the array output as shown in Figure 5-45 vs. days (Ai = 57 deg and the Sun at the equator) with the beta angle in 10 deg steps. The beta angle would increase to 57 deg max or 18.5 days. The "err's" on Figure 5-46 refer to the fact that the Sun angle is limited to 57 deg. When the eccentricity of the Earth orbit is included, the number of days between rotations will vary; but as seen in Figure 5-46, the allowable error is in terms of days, not hours.

A side benefit of the yaw-around is the satellite thermal concern. Since a particular side of the spacecraft is always facing the Sun, the other side will be facing cold space. This will allow better control of the spacecraft thermal surfaces. This factor would ease the design of thermal surfaces that radiate heat from the hot components of the satellite, such as the power amplifier. If louvers are used on the spacecraft, they would be placed facing out the cold side of the spacecraft. The solar panels would have a defined cold and hot side.

Conclusion

Table 5-3 presents the possible benefits and negatives of the "yaw-around" technique. As stated previously, this technique is presented as a trade for the MSSP. It has been used on a large spacecraft with larger solar arrays, and the cost savings was considerably greater. The MSSP savings will not be as great and must be traded with the other spacecraft operation considerations. A clear-cut decision regarding the viability of this approach for MSSP cannot be made until both the amount and final cost of attitude control are determined. However, it does not appear that the savings is sufficient to warrant additional design cost.

5.1.4.2.2 LARGER SPACECRAFT BATTERIES

This trade involves the use of larger spacecraft batteries with the yaw-around approach discussed in Section 5.1.4.2.1 to reduce the size of the solar arrays. If larger batteries were used and the batteries were allowed to discharge to 60 percent depth of discharge during the periods of the beta angle close to zero deg, then the solar arrays could be designed for a higher output in watts per square foot. Since the cost of batteries (commercial type) is less than the cost of solar arrays, a total system cost savings might be gained. This trade proved not to be feasible. However, since the trade was analyzed, and since other parameters in the future might allow this trade to be possible, it is presented.

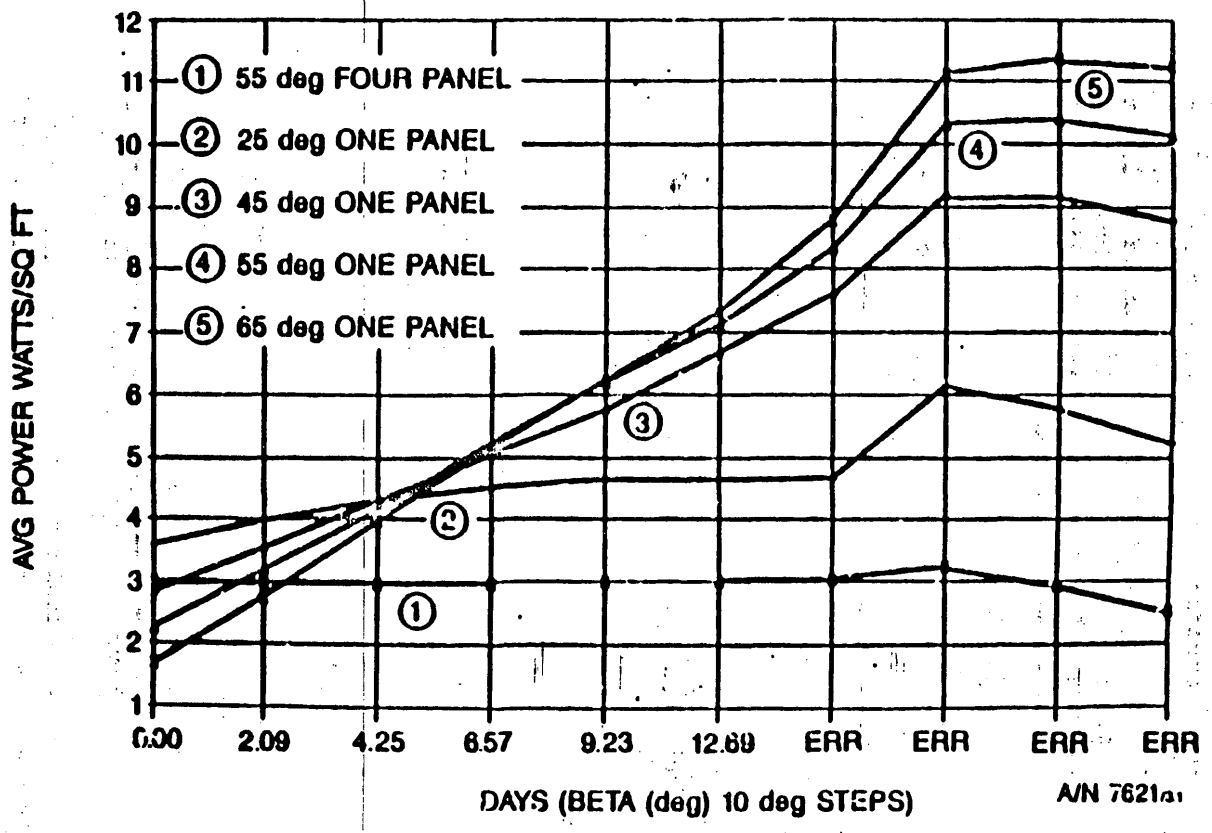


Figure 5-46 Power of Single Array Versus Days (Beta in 10° Steps)

TABLE 5-3 TRADES OF YAW-AROUND SPACECRAFT

| BENEFITS | NEGATIVES |
|---|---|
| * ARRAY EFFICIENCY<br>more watts/sq ft | * ARRAY DEPLOYMENT<br>complicated deployment |
| * DEFINED THERMAL INTERFACES<br>cold side for thermal dissipation | * ROTATION DEVICE<br>wheel needed to rotate |
| * YAW CONTROL<br>solar array oriented to help yaw | * YAW CONTROL<br>processor needed to time yaw-around +/- 20 deg |
| * SMALLER SOLAR ARRAY<br>louver launch cost | * SPACECRAFT YAW<br>impact on communication when S/C rotating |

The solar array output presented in Figure 5-45 showed that the array output increased with larger beta angles. Also, as shown in Figure 5-46, the amount of time spent at the low beta angles is relatively small because of the sine function of the beta angle variation. If larger batteries were used and the depth of discharge of the batteries was allowed to increase during low beta angle, the array could be designed with a smaller size for the larger output rate. The periodic large depth of discharge of the batteries could actually help condition the batteries and would not greatly degrade battery lifetime. The batteries would go through a large depth cycle rate once every ~37 days for a 57 deg inclination, or about 50 cycles in five years. As a baseline for a trade discussion, it is assumed that the spacecraft power needed is 75W with 80 percent convertor efficiency and a factor of 1.2 for array/battery charge efficiency. Or, the array power needed is:

$$75 * \frac{1}{0.8} * 1.2 = 112W$$

Figure 5-47 shows solar array cost vs. array output with an assumed array cost of $7K per square foot. These numbers will vary with the final satellite design and production cost variations, but the shape of the curve and the conclusions should be valid. Also shown in Figure 5-47 is the rate of cost change vs. array output. The item to note is that the impact of power output change from 2.5 to 3.5W per sq ft is located just about at the knee of the curve. Increasing the output to 4W might be an improvement, but increases beyond 4W will not produce great savings.

Using the data from Figure 5-46, a 112W array, and a battery depth of discharge of 60 percent, the battery size necessary to replace the power from the solar arrays can be calculated by integrating the data in Figure 5-46. Figure 5-48 shows the battery size needed vs. solar array output for a yaw-around system with tilt angles of 25- and 35 deg. The 25 deg tilt, for array output of less than 4.5W per sq ft, requires the least battery complement. However, the battery size for even a 4W/ft^2 output is large: 33 amp/hr. This magnitude of battery would produce both battery and weight increase costs that would exceed the savings of the solar arrays.

Conclusion

The array savings will be outweighed by the cost of the batteries and therefore is not a viable trade.

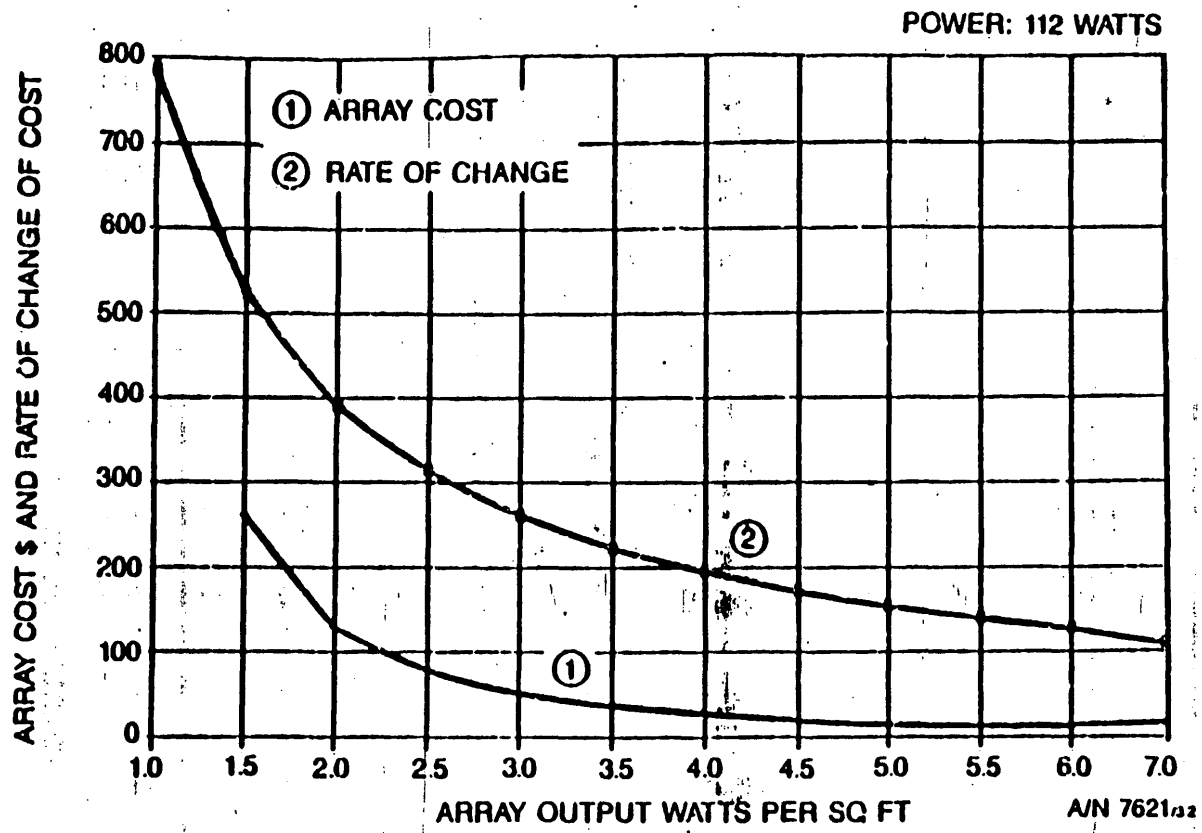


Figure 5-47 Impact of Array Output Rate (Watts/Sq.Ft.) Versus Array Cost ($7k/Sq.Ft.)

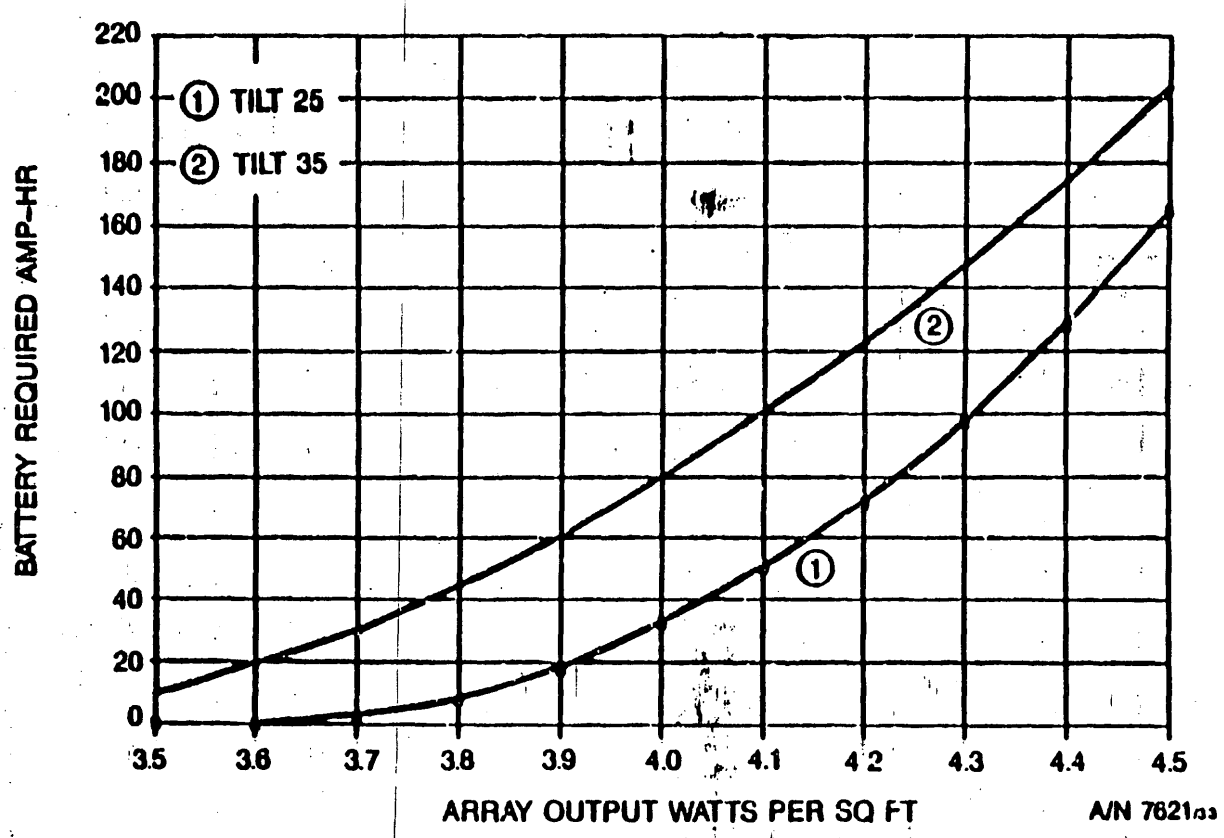


Figure 5-48 Battery Required Versus Array With Sq. Ft. Integrated Average (112 Watts 60% DOD)

### 5.1.5 POWER SUBSYSTEM CONCLUSION

The power subsystem is the main cost driver for the satellite bus for the MSSP. This point was stressed throughout the Phase I study. The main cost for the power subsystem was shown to be the solar array in terms of both base cost and additional cost for launch. The final power requirement for the MSSP was concluded to be 75W. With this power level the final parameters of the power subsystem are:

| | |
|---|---|
| Power level | 75W |
| Array type | four-panel Mansard; 55 deg tilt |
| Array size | 8.1 sq ft |
| Depth of discharge | 5 percent |
| Battery size | 36 amphr |
| Power controller | Current limiting |

Several techniques to limit the size of the solar array were analyzed to reduce the satellite system cost. The Mansard solar array design, which maximized the array configuration for power output, was very beneficial. Trades of battery size and satellite orientation for reduced solar array size were analyzed, but did not produce cost savings. However, these trades may be considered again if the requirements of the MSSP mission cause future changes in duty factor or attitude control.

The cost of power for a 75W MSSP satellite is about $3K per watt for the power subsystem components. Structure, test, and design cost will raise this value to about $5K. When launch of the power subsystem is also considered, the total cost of power would be about $7K per watt.

## 5.2 SATELLITE ATTITUDE CONTROL

The satellite attitude control subsystem must provide a stable platform to achieve communication which is the main mission requirement. Selection of the attitude control subsystem design is pivotal in a number of tradeoff studies with other subsystems (such as antenna, power, thermal, and orbit determination) in establishing the cost effectiveness of the overall mission design. Previous MSSP system studies by ESL (Technical Memorandum No. ESL-TM1632, 15 June 1983) have found gravity gradient stabilization systems to be most cost effective. There are, however, a wide variety of gravity gradient systems available. The studies presented in this section help to quantify the cost and performance of the various gravity gradient stabilization systems.

### 5.2.1 GRAVITY GRADIENT POINTING ERROR SOURCES

The basic gravity gradient system consists of a passive damping device and an extendable boom that is used to separate two end masses. The separation of the end masses typically produces transverse-to-longitudinal inertia ratios in the range of 10 to 100. This results in gravity gradient torques that restore the longitudinal axis toward the local vertical. The basic system does not provide any restraint about the longitudinal axis so that yaw is uncontrolled.

Boom design technology developed rapidly during the 1960s in an effort to minimize thermal bending effects. Thermal bending can result in instabilities that degrade pointing performance. This was particularly apparent in the early systems when long booms (20m or more) were used. The MSSP design should not be subject to thermally induced instabilities since a short (10M or less) rigid boom is planned to be used.

The damping device is required to damp out roll and pitch libration. The simplest passive damper (hysteresis rods) consists of a triad of orthogonal magnetically permeable rods. Rotation of the triad in the Earth's magnetic field produces hysteresis losses. A magnetic ball floating inside of a conducting sphere is the other commonly used damper (eddy current damper). The magnet follows the Earth's field. The rotation of the spherical shell, with respect to the magnet, produces eddy current losses in the conducting material.

The eddy current damper gives better performance than the hysteresis bars but is more expensive. Both types of dampers produce disturbance torques on the spacecraft due to the changing directions of the magnetic field in the orbital reference frame as the spacecraft goes around in orbit. The disturbances vary inversely with the damping time constant. A typical value for the eddy current damper is one degree of disturbance with a one day time constant.

A wide variety of other damping mechanisms have been proposed. Some of them were built and flown on early gravity gradient satellites. One damper that flew on four APL satellites in the early 1960s was the lossy spring between the end of the boom and the tip mass. The time varying centripetal force, resulting from libration, caused the tip mass to move in and out, thereby absorbing the libration energy in the mechanical hysteresis of the spring. The lossy spring was used in conjunction with hysteresis rods. The use of the lossy spring was discontinued after realizing that the rods used alone were effective. All of the other dampers, except hysteresis rods and eddy current dampers, have also been discontinued and are not commercially available.

Most gravity gradient systems are designed to operate in nearly circular orbits. The time varying orbital rate in an elliptic orbit tends to pump libration in the orbit plane. The orbital eccentricity (e) produces a once per orbit sinusoidal pitch oscillation with an amplitude of

$$\theta = \frac{2e}{3\sigma_y - 1} \text{ (RAD)} \quad \text{where } \sigma_y = \frac{(I_x - I_z)}{I_y}$$

For a typical gravity gradient satellite, the roll (Ix) and pitch (Iy) inertias are approximately equal and at least one order of magnitude greater than the yaw (Iz) inertia y-1. The eccentricity of MSSP will be less than 0.0015 to keep a reasonable altitude variation between satellites. This results in a pitch amplitude of less than 0.1 deg.

Two potential disturbances to the attitude of a gravity gradient-stabilized spacecraft can be minimized by careful configuration control. Both the aerodynamic drag and solar radiation pressure torques are strongly dependent upon configuration. The following design goals are given to help minimize the effect of these disturbances: 1) The required boom length is minimized by equally dividing the mass of the spacecraft between the two ends. 2) The area-to-mass ratios of the two ends should be the same to minimize aerodynamic disturbance torques. 3) The end masses should be of a convex cylindrically symmetric design eliminating variations in torque as a function of yaw attitude. 4) The surface properties of the two ends should be similar to minimize solar torque. The ideal configuration, from an attitude control viewpoint, would consist of two identical spheres as end masses.

BASD has examined a number of different spacecraft configurations with peak solar radiation and aerodynamic torques at 625 km altitude ranging from $5 \times 10^{-7}$ Nm to $5 \times 10^{-6}$ Nm. These are often the dominant disturbance torques for configurations at the high end of this range. Residual magnetism is usually the dominant disturbance for the carefully configured spacecraft at the low end.

The aerodynamic torque decreases with atmospheric density at higher altitudes, but it will still be significant at 625 km altitude. The time variations of the disturbance torques may lead to resonances. The maximum aerodynamic torque variation will be due to the diurnal bulge in the atmosphere caused by solar heating. Maximum density occurs at about 14:00 hr and minimum density at 3:00 hr local solar time. The torque will appear primarily as a biased sinusoid at the orbit rate about the orbit-normal. This torque can cause pitch oscillations at orbit rates that are several degrees in amplitude in low altitude orbits. However, this torque is not the main concern. Smaller torques may produce much larger attitude responses if they appear at the resonant frequencies $\sqrt{3}wo$ in pitch or 2wo in roll. Wo is the orbital frequency or 2Pi/period or approximately 2Pi/96 minutes.

The pitch aerodynamic torque is a function of the yaw attitude for concave configurations. The deployed solar arrays shadow each other and the main body by different amounts when viewed from different yaw angles. The aerodynamic torque in free molecular flow depends directly on the area projected into the wind. Consequently, the pitch torque will oscillate as the spacecraft rotates in yaw. The critical yaw rotation rate is 3/4 wo for a typical configuration with 4 deployed arrays. This yaw rate produces a pitch disturbance torque at the resonant frequency in pitch. The propensity of the satellite to maintain the critical yaw rate (about 0.0266 deg/sec for MSSP) is dependent upon the configuration. Some configurations result in almost no yaw torque. Others, with canted arrays, have yaw torques that vary with pitch attitude. Generalizations on this potential pitch resonance should be avoided since it is so strongly dependent upon the details of the configuration.

Roll is also subjected to a naturally occurring forcing function at resonant frequency. Solar radiation pressure has a component at the second harmonic of orbital frequency for eclipsing orbits. The amplitude of the second harmonic roll torque is a function of the location of the ascending node with respect to the sun line. Consequently, the duration of the maximum resonance conditions depends upon inclination angle which determines the nodal regression rate. J.M. Whisnant and D.K. Anand refer to a 9 deg libration amplitude being produced by this resonance in an engineering note on pages 743-744 of the June 1968 Journal of Spacecraft, Vol. 5, No. 6. Presumably, the second harmonic of an aerodynamic torque in yaw could couple into roll through the yaw rate.

It should be observed that one of the principal reasons for adding a constant speed pitch wheel to a gravity gradient-stabilized spacecraft is to break up the resonances described above. The wheel provides yaw restraint. This prevents a constant speed yaw rotation that may produce pitch disturbance torques at the pitch axis natural frequency.

The wheel also changes the natural frequency in roll so that the naturally occurring second harmonic disturbances will not be in resonance. It is a commonly held misconception that a wheel is added only to improve yaw pointing. Significant improvement in pitch and roll can be achieved by the addition of a wheel. Pitch and roll amplitudes can be limited to the order of one degree for a carefully configured design that includes a pitch wheel, such as GEOSAT-A launched in March 1985.

The following table gives the documented pointing performance of four basic gravity gradient systems. The data for OSCAR-14 and GEOS II is actual flight data as presented by D.K. Anand in the Journal of the British Interplanetary Society, Vol. 26, pages 641-661, 1973. The TRANSIT 5A flight data is from a paper by F. Mobley and R. Fischell at the Symposium on Passive Gravity Gradient Stabilization which was published in NASA SP 107, 1966. The more recent GEOSAT flight data is given in J. Hunt's paper AAS 86-052 which was presented at the 1986 AAS Guidance and Control Conference.

TABLE 5-4
PERFORMANCE OF BASIC GRAVITY GRADIENT SYSTEMS

| Spacecraft | Pointing Accuracy $3\sigma$ | Semi-Major Axis | Eccentricity | Damper | Tip Mass | Boom Length |
|---|---|---|---|---|---|---|
| CSCAR-14 | Roll 10 deg | 1.169 Re | 0.004 | Hysteresis | 1.3 kg | 30.5 |
| | Pitch 30 deg | | | Bars | | |
| TRANSIT 5A | Roll 6 deg | 1.117 Re | 0.003 | Hysteresis | 1.8 kg | 30.5 |
| | Pitch 6 deg | | | Bars and Lossy spring | | |
| GEOS-II | Vertical 7 deg | 1.208 Re | 0.032 | Eddy | 3.2 kg | 8.6 |
| | | | | Current | | |
| GEOSAT | Roll 1 deg | 1.125 Re | 0.003 | Eddy | 45 kg | 6 |
| | Pitch 1 deg | | | Current and Wheel | | |

Table 5-5 is copied directly from Anand's paper mentioned previously. It shows the percentage of time that the boom axis of GEOS II was within a specified angle of nadir. These entries are based upon 434 data points taken during two separate time intervals.

TABLE 5-5
QUANTITATIVE FLIGHT PERFORMANCE OF GEOS-II

| vert | Number of data points* | Frequency % | Cumulative frequency, % |
|---|---|---|---|
| 0 deg-1 deg | 7 | 1.6 | 1.6 |
| 1 deg-2 deg | 62 | 14.3 | 15.9 |
| 2 deg-3 deg | 130 | 30.0 | 45.9 |
| 3 deg-4 deg | 140 | 32.3 | 78.2 |
| 4 deg-5 deg | 78 | 18.0 | 96.2 |
| 5 deg-6 deg | 14 | 3.2 | 99.4 |
| 6 deg-7 deg | 3 | 0.7 | 100.0 |

*Data points recorded at one-minute intervals on days 135-138, 254 and 255, 1968.

### 5.2.2 GRAVITY GRADIENT SUBSYSTEM COST ESTIMATES

The total cost of the attitude stabilization subsystem includes hardware and launch cost, cost of providing electrical power on-orbit, plus design, integration, and test costs. The last three items depend upon the specific design configuration selected and must be priced on a case by case basis. A first approximation of cost can be made on the basis of hardware, launch, and power costs of generic components of gravity gradient stabilization systems. Table 5-6 gives the estimated size, mass, power, and cost of various components for gravity gradient stabilization of MSSP class spacecraft. The cost figure includes the estimated price from the vendor plus $3.4 K/kg for launch cost plus $5.0 K/W for on-orbit power.

The certainty of the cost numbers in this table varies greatly. In general, the costs have been adjusted by the potential vendors to reflect purchases of several hundred units in 1990 dollars. The most notable exception to this is the eddy current damper where the potential vendor could only quote a single unit price. The other significant exceptions are the hysteresis bars and torque rods. It is anticipated that the extreme simplicity of these elements will permit dramatic reductions in the vendor prices for these components. If this assumption is incorrect, then the price of a torque rod could be as much as $30K more than that shown in the table.

Note that the attitude determination equipment has been included in the table for the sake of completeness. Attitude sensor telemetry will be desired during the prototype demonstration flights to verify pointing performance. No attitude sensors are planned for the operational phase of the program.

Four different configurations of ADCS components for a gravity gradient-stabilized spacecraft are given in Table 5-7. The cost numbers for each system design are summed from the component costs given in Table 5-6. Consequently, the system cost numbers given in Table IV include launch and power cost but do not include design, integration, and test. The pointing performance entries are derived from digital simulation results presented in the next section. The four entries under performance are deviation from the local vertical, roll, pitch, and yaw, respectively. The deviation of the boom axis of the satellite from the local vertical is approximately the square root of the sum of the squares of roll and pitch. For small angles, roll is the angle about the velocity vector, pitch is the angle about the orbit-normal, and yaw is the angle about the local vertical. It should be emphasized that pointing performance depends upon the satellite mass distribution, boom length, and environmental torques. These particular results are for the specific parameter values described in the following section.

TABLE 5-6
ADCS COMPONENT CHARACTERISTICS

| COMPONENT | SIZE | MASS | POWER | COST | COMMENTS |
|---|---|---|---|---|---|
| Bi-Stem Boom | 11cm x 11cm x 15cm | 1.0 kg | - | 43K | <20 Extended |
| Hysteresis sections Bars solar | 0.6cm diameter<br>4.4 Total Length | 1.0 kg | - | 5K | 8-55cm embedded in arrays |
| Eddy Current Damper | 13cm high<br>15cm diameter<br>Mounting Flange | 3.0 kg | - | 140 K | 70,000 Dyne-cm- sec Damping Constant |
| Constant | 15cm diameter | 2.5 kg | 8.0W | 125 K | 1.0 Nms |
| Speed Wheel | 10cm length | | | | |
| Scanwheel | 17cm diameter<br>18cm length | 6.8 kg | 3.5W<br>30.0W | 144K | 5.7 Nms, 25 mNm |
| Scanwheel Electronics | 15cm x 15cm x 7.5cm | 1.6 kg | 2.5W | - | Included with Scanwheel |
| Torquerod | 2.1cm diameter<br>56cm length | 0.9 kg | 0.7W | 10K | $30Am^2$ |
| Yaw Sensor | 11.5cm diameter<br>25cm length | 3 kg | 3W | 75K | $4^o$ x $4^o$ FOV |
| Magnetometer | 7cm x 7cm x 11cm | 0.4 kg | 1.4W | 51K | 3-axis sensor |
| Magnetometer Electronics | 14cm x 14cm x 5cm | 0.8 kg | - | - | Included with Magnetometer |

TABLE 5-7
ADCS SYSTEM DESIGN COMPARISON

| CONFIGURATION | PERFORMANCE (Deg) Local Vertical, Roll, Pitch, Yaw | MASS | POWER | COST* |
|---|---|---|---|---|
| A) Boom, Hysteresis Bars | 19.4, 17.8, 7.8, 180.0 | 2 kg | 0 | $54.8K |
| B) Boom, Eddy Current Damper | 5.3, 4.5, 2.8, 180.0 | 4 kg | 0 | $196.6K |
| C) Boom, Hysteresis Bars, Constant Speed Wheel | 2.3, 1.0, 2.0, 2.5 | 4.5 kg | 8W | $228K |
| D) Boom, Eddy Current Damper, Constant Speed Wheel | 1.8, 0.2, 1.8, 0.4 | 6.5 kg | 8W | $370K |

* Cost includes hardware, launch, and on-orbit power.

### 5.2.3 SIMULATION RESULTS

BASD has developed a digital computer simulation of the attitude dynamics of an MSSP class gravity gradient-stabilized spacecraft. This simulation has been used extensively for preliminary design and performance prediction. In general, the simulated results correspond closely with the wealth of data on gravity gradient systems that has been published in conference papers, journal articles and technical reports over the last 25 years. The majority of this data is of interest only to detail design engineers and will not be recounted here. Instead, four specific simulation runs will be presented that support the pointing performance predictions that were made in the preceding section.

The orbit used in these four cases was a circular orbit at 600 km altitude with an inclination of 80 deg. The magnetic field model used in the simulation was the tenth order spherical harmonic expansion given in the 1985 IGRF Model. The atmospheric density model included a diurnal bulge $2 \times 10^{-13}$ kg/m$^2$. The peak aerodynamic torques resulting from the diurnal bulge were $7.4 \times 10^{-6}$ Nm and occurred once per orbit. The ascending node of the orbit and time of year were selected to provide an orbit with 25 percent eclipse time. This resulted in solar torque components that were nearly sinusoidal in pitch and on-off step shaped in roll. This phasing was deliberately selected to maximize the content of the second harmonic of orbit rate in the roll disturbance torque. The amplitudes of the pitch sine wave and roll step solar torques was $0.6 \times 10^{-6}$ Nm. Note that the conditions given above represent a worst case combination of environmental disturbances.

The simulated spacecraft had roll, pitch, and yaw moments of inertia of $Ix = Iy = 198.8$ kgm$^2$ and $Iz = 17.3$ kgm$^2$. For aerodynamic and solar radiation pressure torques calculations, the spacecraft was modeled as two end bodies separated by a boom of negligible area. The lower body had a cross sectional area of 1M$^2$ displaced 5 cm from the center of mass of the system. The upper body had a cross sectional area of 0.02m$^2$ and was displaced 7.3M by the boom from the system's mass center. In all four simulated cases, a residual magnetic dipole of 0.1Am$^2$ (100 Pole-cm) along the boom axis was included.

Figure 5-49 corresponds to configuration A in Table 5-7 which is the least expensive gravity gradient stabilization system. The hysteresis bar damper in this configuration consists of three orthogonal rods of AEM 4750; each rod is 0.75m long and 2.5mm in diameter. The saturation magnetization of the rods is $Is = 10^{-6}$ Tesla. The primary disturbance to the pitch and roll attitude is due to the induced magnetism in the rods. This disturbance varies considerably during the course of a day as the Earth's magnetic poles rotate in and out of the orbit plane. Yaw is unrestrained and rotates through thirteen revolutions in twenty-four hours at an average rate of slightly less than one revolution per

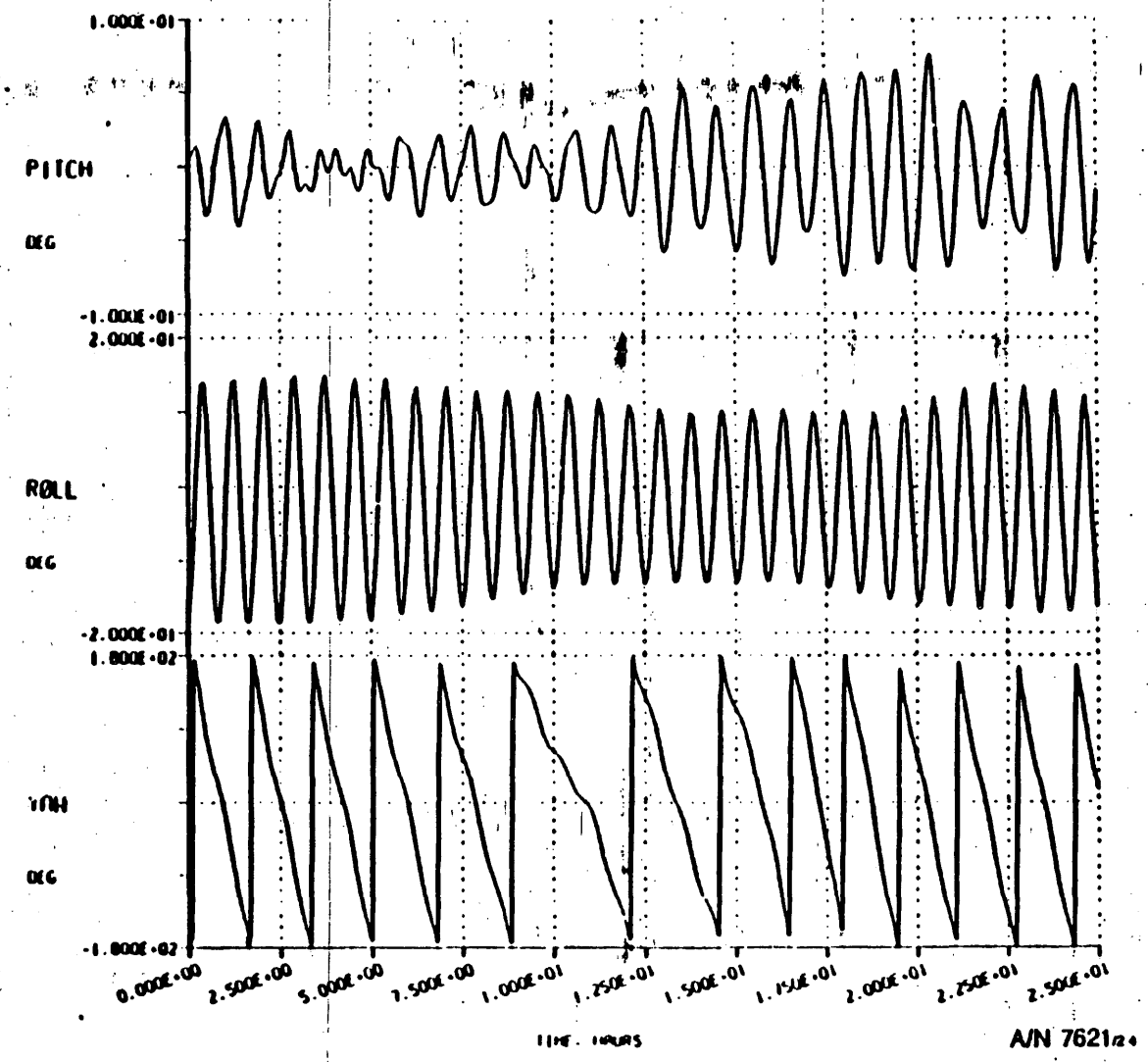


Figure 5-49 Attitude Responses (Three Rods, 0.1 AM2 Dipole, No Wheel, Solar Torques)

orbit. The maximum deviation of the boom axis from the local vertical is 19.4 deg. It should be noted that this large error is due to a worst case set of environmental torques and that design optimization may result in appreciably better pointing performance.

Figure 5-50 corresponds to configuration B in Table IV which uses an eddy current damper in place of the hysteresis bars. The eddy current damper simulated in this case has a damping constant of Kd=0.0014 Nms (14,000 Dyne-cm-sec). The pitch and roll errors are greatly reduced from the preceding case resulting in a maximum pointing error from vertical of only 5.3 deg. The yaw attitude is still unconstrained; however the average yaw rate is now reduced to approximately one revolution per day. Once again note that some pointing performance improvement is possible by optimizing the system parameters.

Figure 5-51 corresponds to configuration C in Table 5-7. This configuration uses the same hysteresis bars as configuration A and adds a small constant speed pitch wheel. The simulated angular momentum of the wheel is 0.25 Nm. The addition of the wheel has coupled the roll and yaw motion with a nutation frequency of approximately 5.2 wo. Without the wheel, the roll libration frequency was 2wo which was in resonance with the second harmonic of the disturbance torques.

Figure 5-52 corresponds to configuration D in Table 5-7. This configuration uses the same eddy current damper as configuration B and adds a 0.25 Nms pitch wheel. The pointing performance in roll and yaw is approximately five times more accurate than for configuration C. However, the pointing performance in pitch is only slightly better. Consequently, the maximum deviation from the vertical is not significantly better for the momentum bias system with the eddy current damper than it is for the momentum bias system with the hysteresis bars.

### 5.2.4 CONCLUSION

The purpose of the analysis of the satellite attitude control system was to establish the performance capability and the system cost. The performance of a gravity gradient system can be enhanced with the addition of various components. The performance is obtained in discrete steps with increases in weight and cost.

The computer simulations have shown that the pointing errors with a worse-case environment can be as large as 19 deg with uncontrolled yaw or as small as 2 deg with less than 0.5 deg yaw control. The important factor to be traded is the cost trades of attitude control vs. pointing compensation via the spacecraft antenna.

Analysis by the Phase I antenna team members has shown that the antenna and network processor can compensate for the attitude error (Section

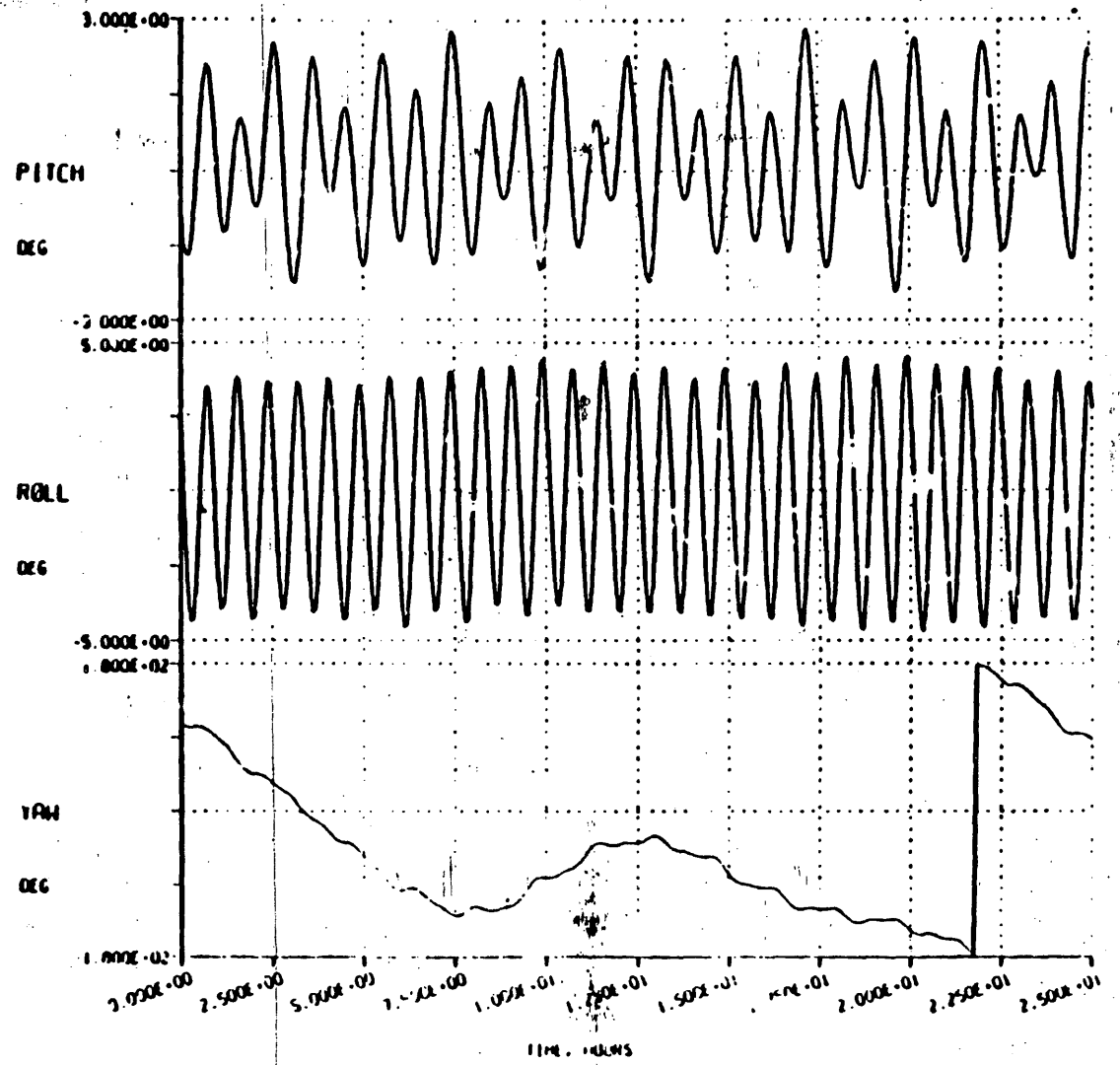


Figure 5-50 Attitude Angles — Without Wheel Eddy Current Damper

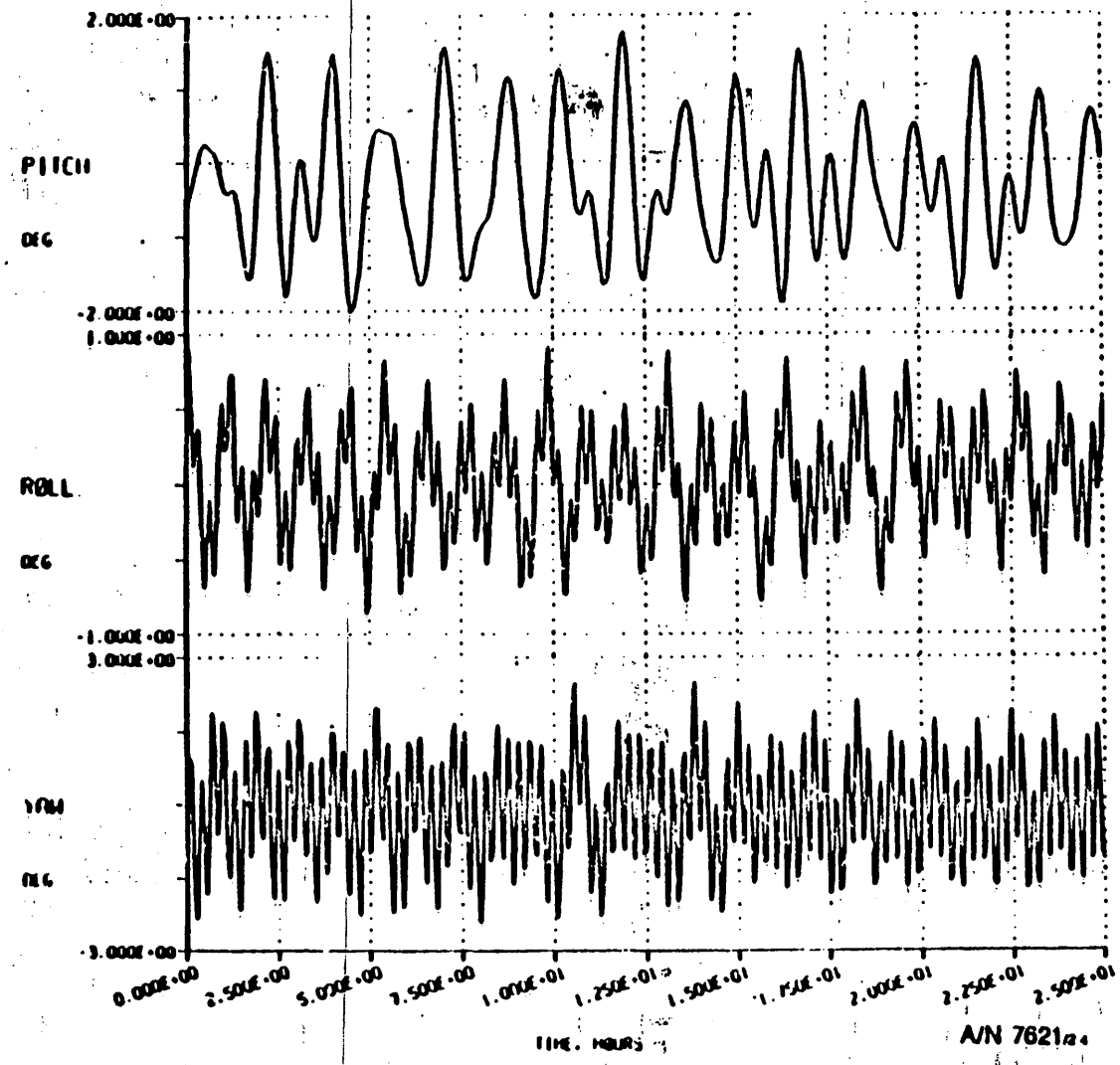


Figure 5-51 Three Rods, 0.1 AM2 Dipole, 0.25 NMS Wheel, Solar Torques

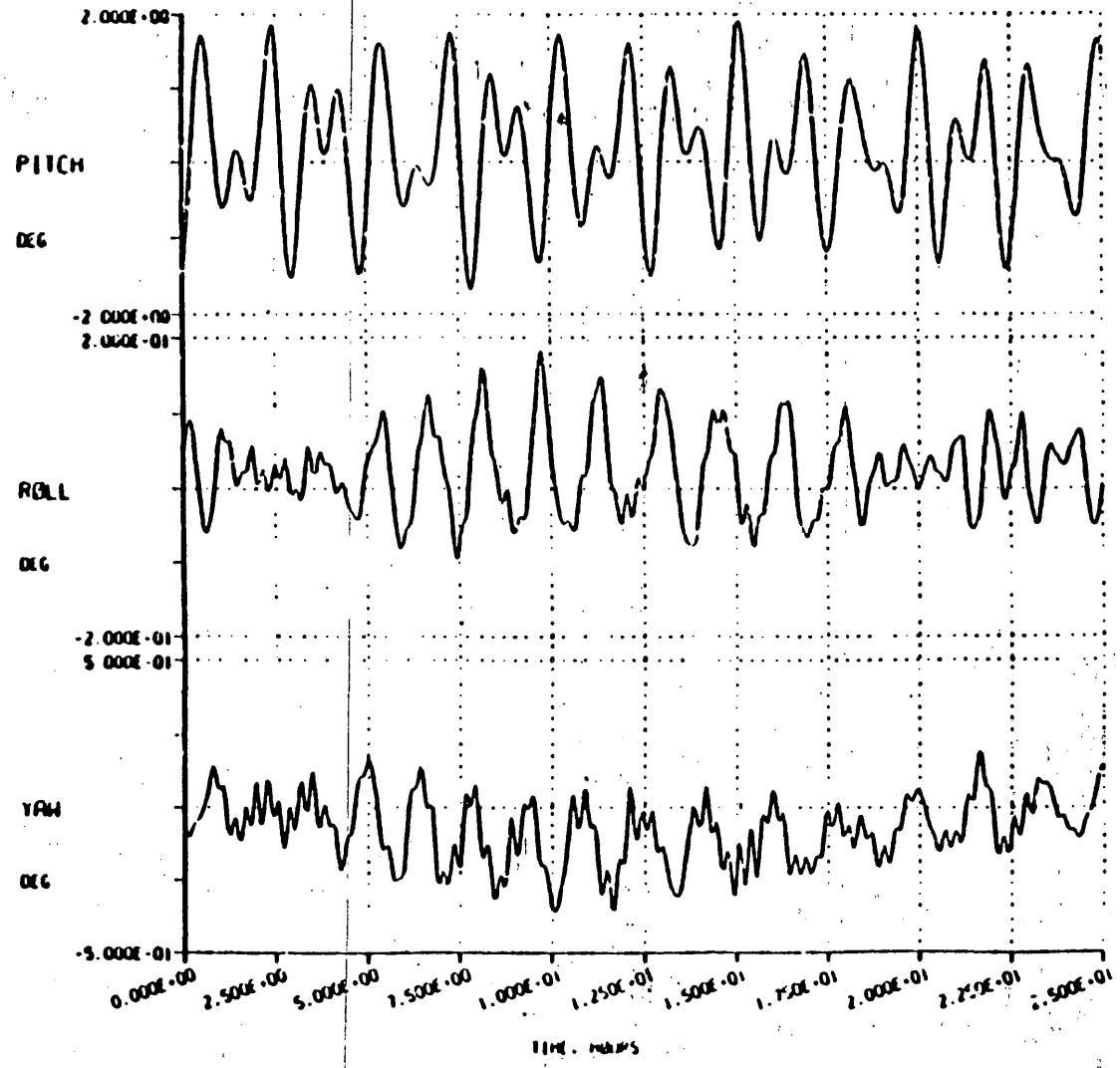


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Figure 5-52 Attitude Angles — With Wheel Eddy Current Damper

5.5) at relatively low costs. The final working group meeting estimates for the payload - radio and antenna was $100K to $200K.

The antennas can compensate for the uncontrolled yaw by their required azimuth scan capability. The antenna designs presented use the elevation scan capability of the crosslink antenna to provide complete up/down link coverage and reduce the size of the up/down antenna and therefore, the diameter of the spacecraft. These benefits - cost, performance and size - direct the conclusion that the attitude control system be a gravity gradient boom with a eddy current damper.

## 5.3 SPACECRAFT COMMAND AND TELEMETRY

A normal spacecraft command and telemetry subsystem is necessary to reconfigure the spacecraft and gather information on the status of the spacecraft. However, the MSSP spacecraft is not typical. The spacecraft payload is the antenna, radio, and network processor. The payload provides the satellite communication. Most commands or telemetry information is resonant within the network processor. Actual spacecraft reconfiguration commands were not identified during the Phase I study but, if needed, could be simply provided via an interface with the network processor. Spacecraft telemetry which is mainly health monitoring of the spacecraft and payload can be provided also via an interface with the network processor.

### 5.3.1 SPACECRAFT TELEMETRY

Spacecraft telemetry, mainly concerned with voltages and temperatures, is necessary to monitor the system and analyze spacecraft failures. The level of telemetry will be dependent upon the state of spacecraft development. During the breadboard and prototype phases, more telemetry will be required than flight model spacecraft.

Several factors, such as the magnitude of the telemetry, increasing the number of channels to be monitored, and ground station capabilities, will increase the cost of the system. The type of telemetry to be monitored will also affect the cost. Temperature data with conditioning circuits (to develop voltage for the thermistors) and analog/digital convertors are the most expensive, analog data (voltage or RF power monitors) requiring analog/digital converters are second and serial digital and bilevel or 1/0 truth data are least expensive. Temperature, voltage, and signal levels are generally more important for troubleshooting than bilevel data.

Table 5-8 is a preliminary telemetry data list and dependent on acceptance by radio and antenna personnel.

More telemetry channels may be desired but limiting the number to 40 or 320 bits (8 bit analogs) would enable data restriction to one message packet from the satellite. Another advantage is that no changes would be required between the breadboard, prototype, and flight models when testing in the thermal vacuum chambers is done.

The final amount of telemetry will depend upon what is needed, in addition to performance and health telemetry, by the radio and antenna. The ADACS subsystem has a telemetry listing for roll, pitch, and yaw attitude sensor telemetry which is only needed on the breadboard and prototype units which have sensors to verify the attitude control.

TABLE 5-8
PRELIMINARY TELEMETRY LIST

| | | | |
|---|---|---|---|
| Spacecraft | | | |
| | Power | | |
| | | Battery current | Analog |
| | | Battery cell balance | " " |
| | | Battery voltage | " " |
| | | Solar array current | " " |
| | | Regulator voltages | " " |
| | | Battery temperature | " " |
| | | Solar array temperatures | " " |
| | | Heaters | 4 bilevels |
| | Thermal | (in subsystems) | |
| | Structure | (in subsystems) | |
| | C&DH | | |
| | | Network processor temperature | Analog |
| | | Network ref bits(?) | 8 Bilevels |
| | | Network processor injection | Analog |
| | AD&CS | | |
| | | Boom position | Analog |
| | | Wheel speed | " " |
| | | Wheel Power | " " |
| | | Wheel Temperature | " " |
| | | Roll Attitude | " " |
| | | Pitch Attitude | " " |
| | | Yaw Attitude | " " |
| | | Up/down controllers | 8 bilevels |
| Radio | | | |
| | Temperature monitor | | Analog |
| | Radio AGC | | " " |
| | RF power monitor | | " " |
| | PA current monitor | | " " |
| | PA temperature | | " " |
| | Oscillator temperature | | " " |
| Antenna | | | |
| | Antenna temperature | | 3 Analogs |
| | Antenna reference bits | | 8 Bilevels |

The spacecraft telemetry is gathered by an interface box within the spacecraft stored in the network processor memory. The process for reading performance data such as the radio AGC which is only valid at certain times will have to be determined. The analog data will be converted into an 8-bit, 0 to 5 volt by a 0 to 511 count analog-to-digital converter. The telemetry can be taken by the interface at a relatively slow rate and stored into the network processor memory. When a determination has been made that the health of a spacecraft is of concern, the ground station can call the network processor which sends data stored within the processor memory simulating traffic data.

The network processor could also process the telemetry data. The health data in the spacecraft will have nominal levels for all of the data channels verified during the breadboard testing phase. Therefore, limits +/- will exist for the nominal levels for the performance and bilevel data and definitely exist for the voltage and temperature (health) data. These data ranges could be stored in the network processor memory and then verified by the network processor that all data read is within stored nominal limits. A health bit 1 or 0 stating that all checked parameters are within predetermined limits could then be inserted in the overhead of any message from the satellite.

The format for adding information to the overhead, such as health data, will have to be determined; however, incorporating such a format would greatly eliminate the amount of ground station support needed to maintain the system creating a large control cost saving and more autonomous system.

### 5.3.2 SPACECRAFT COMMANDS

Commands external to the network processor are not envisioned. Uplinks to the spacecraft such as ephemeris data, time data, or a request for spacecraft status would be handled internal to the network processor. Any external commands would require an external interface with telemetry for confirmation and would degrade system autonomy. If required, by the radio or antenna, the command function could be accomplished by the network processor and an interface circuit which would drive latching-type relays.

## 5.4 SPACECRAFT CARRIER CONFIGURATION

This analysis was performed as a means of generating an approach to deploy the satellites at the desired altitude and identifying any cost drivers which would impact the satellite design. The primary conclusion is that as the satellite size increases the number of satellites per launch decreases and the total system cost increases.

### 5.4.1 SPACECRAFT CARRIER CONFIGURATION

The carrier provides the means for launching a group of MSSP spacecraft on a single structure, thereby eliminating the need for individual propulsion systems on each spacecraft. The Orbiter Maneuvering Vehicle (OMV) places the loaded carrier in the proper orbit. The OMV then returns the empty carrier to the orbiter to be reflown at a later time. Launch costs are reduced by flying as many spacecraft as practical on a carrier to minimize the number overall of flights needed to place the system on orbit.

Various configurations were considered to try to optimize the packing density capability of the carrier. The best configuration uses a hexagonal-shaped structure rather than square. If, however, the hex structure shows by analysis to be marginal in capability or too difficult to properly manufacture to the required tolerance, then the square-based concept will have to be reconsidered.

Each of the five carriers shown in Figures 5-53 through 5-57 occupy the same amount of cargo bay length and are all fabricated using five-inch square aluminum tubing. Concepts 1 through 4 can carry 13, 16, 17 and 19 spacecraft each, respectively. Although the 17-hole carrier would be the obvious choice, loads analysis may again dictate one of the smaller versions - concept 2 or 1 - be used, as their structures employ better load paths throughout the primary structure.

The most efficient carrier, the 19-hole hex structure and spacecraft were initially selected as the spacecraft carrier configuration baseline. However, since a 28-inch antenna became a possibility, a new carrier was configured and is shown in Figure 5-57.

Launch costs per spacecraft are increased as the number of spacecraft per carrier are decreased. The impact that antenna diameter has on launch cost as a result of carrier volume being consumed by the larger antenna follows:

| Antenna Diameter | Spacecraft per Carrier |
|---|---|
| 18 in. | 24 |
| 24 in. | 19 |
| 28 in | 12 |

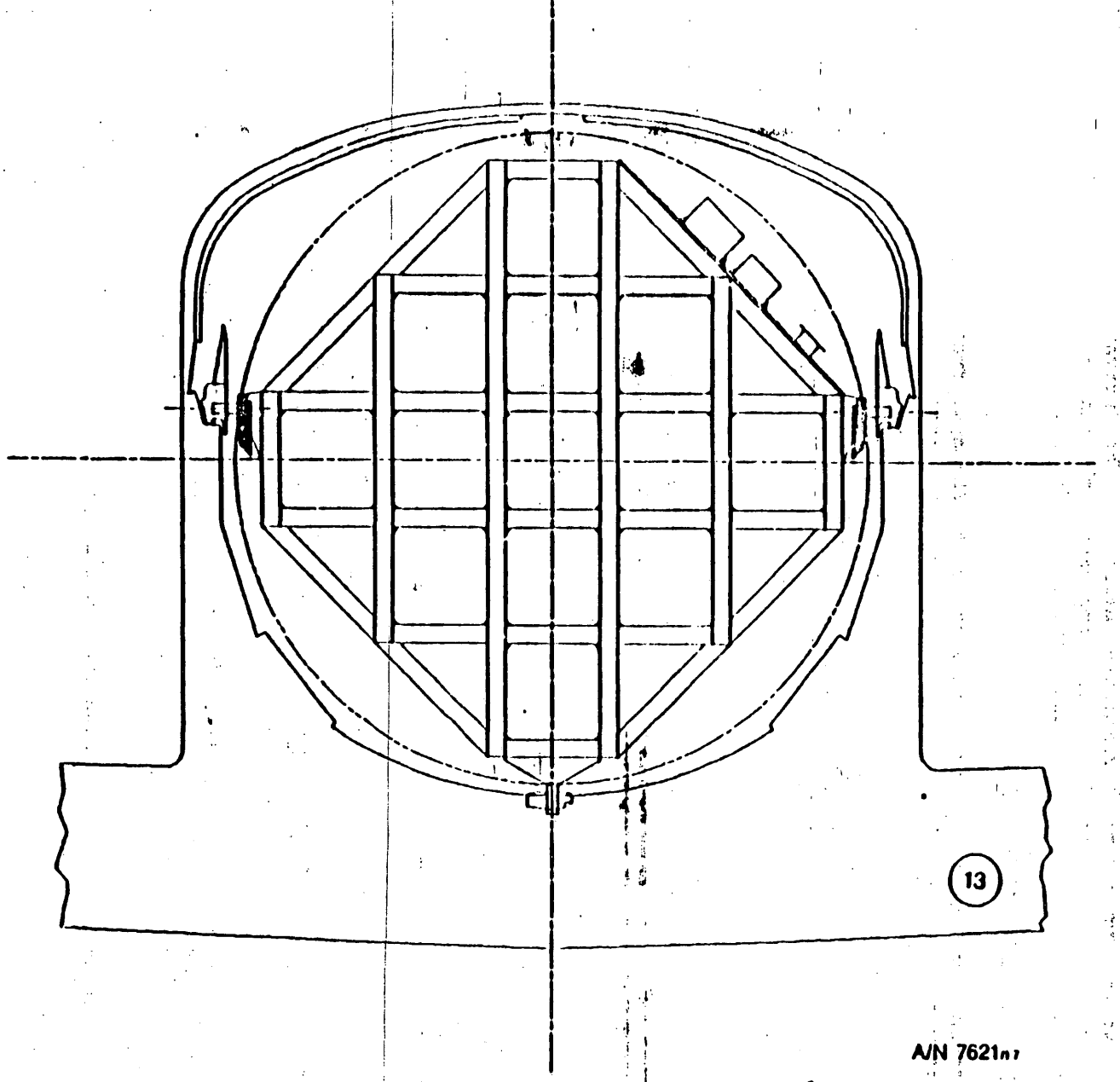


Figure 5-53 Satellite Carrier Concept 1

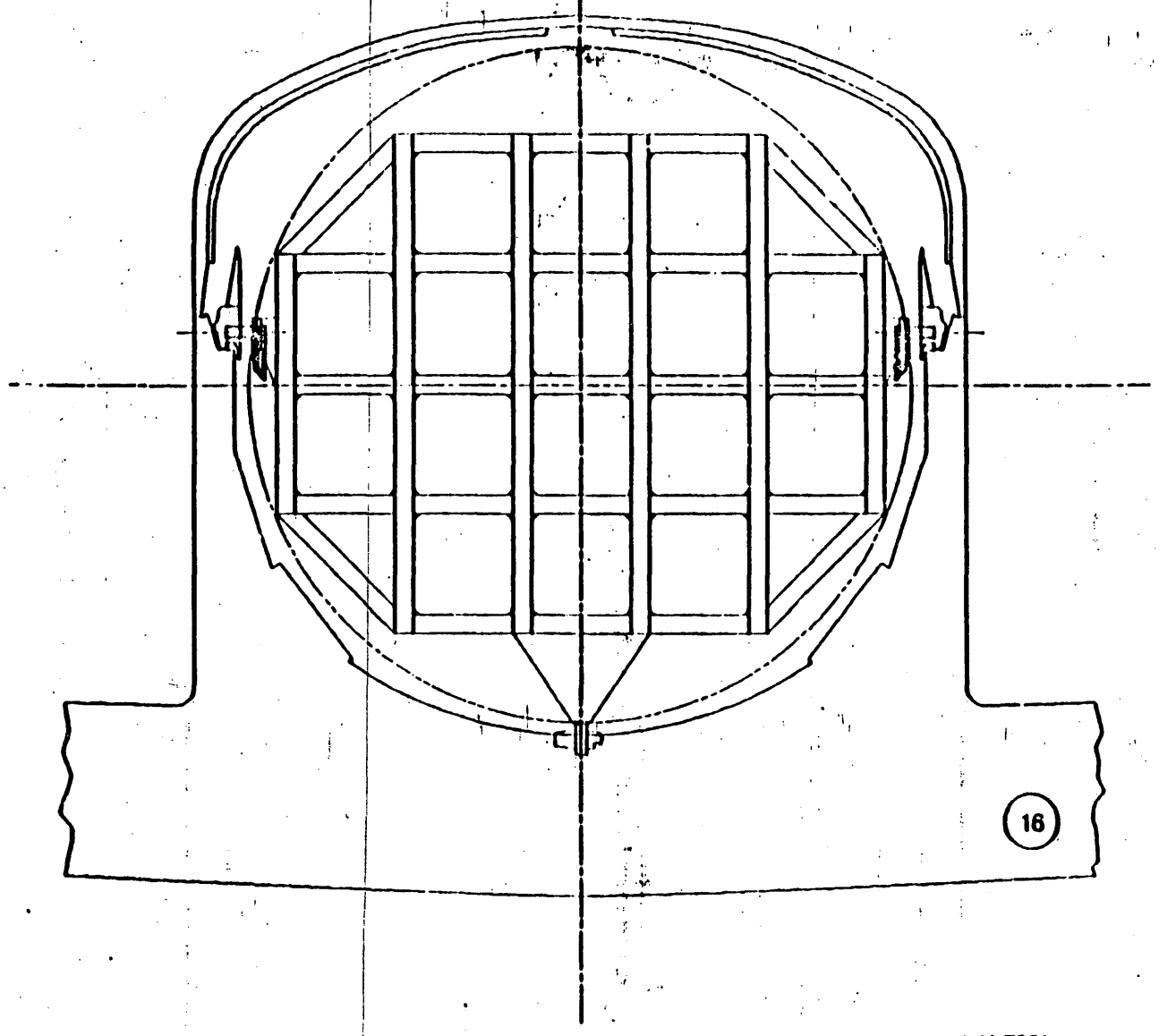


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Figure 5-54 Satellite Carrier Concept 2

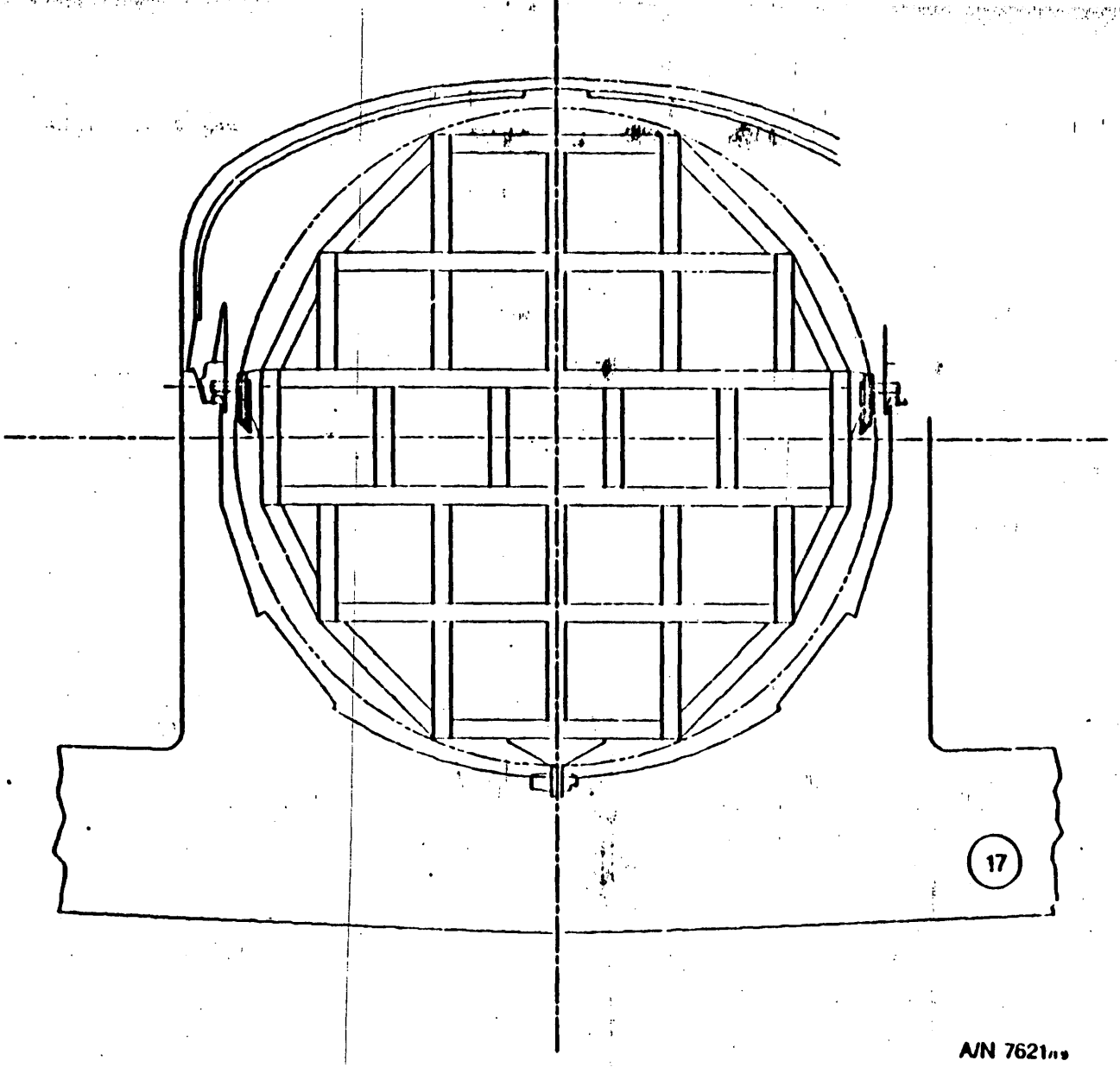


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Figure 5-55 Satellite Carrier Concept 3

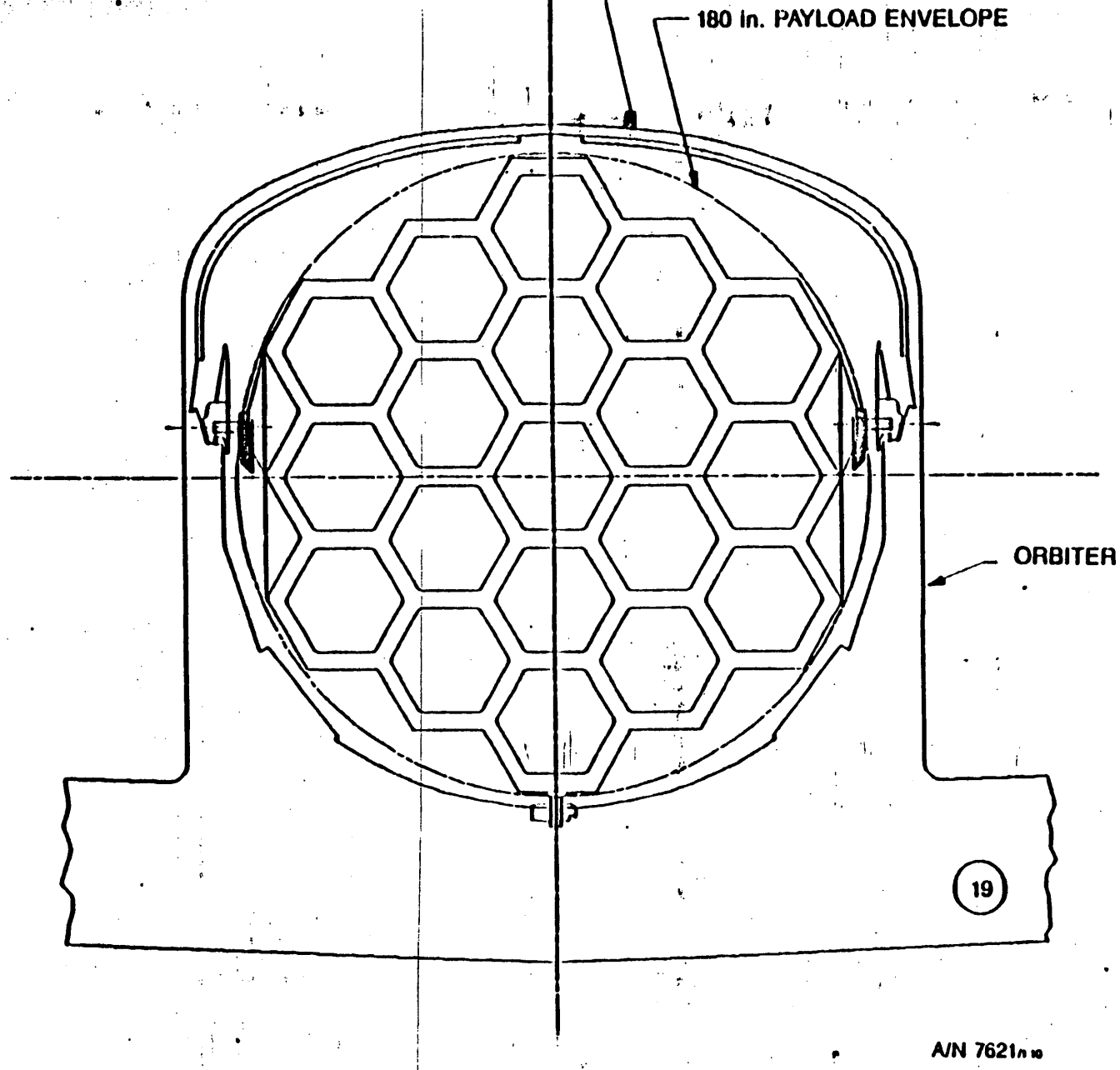
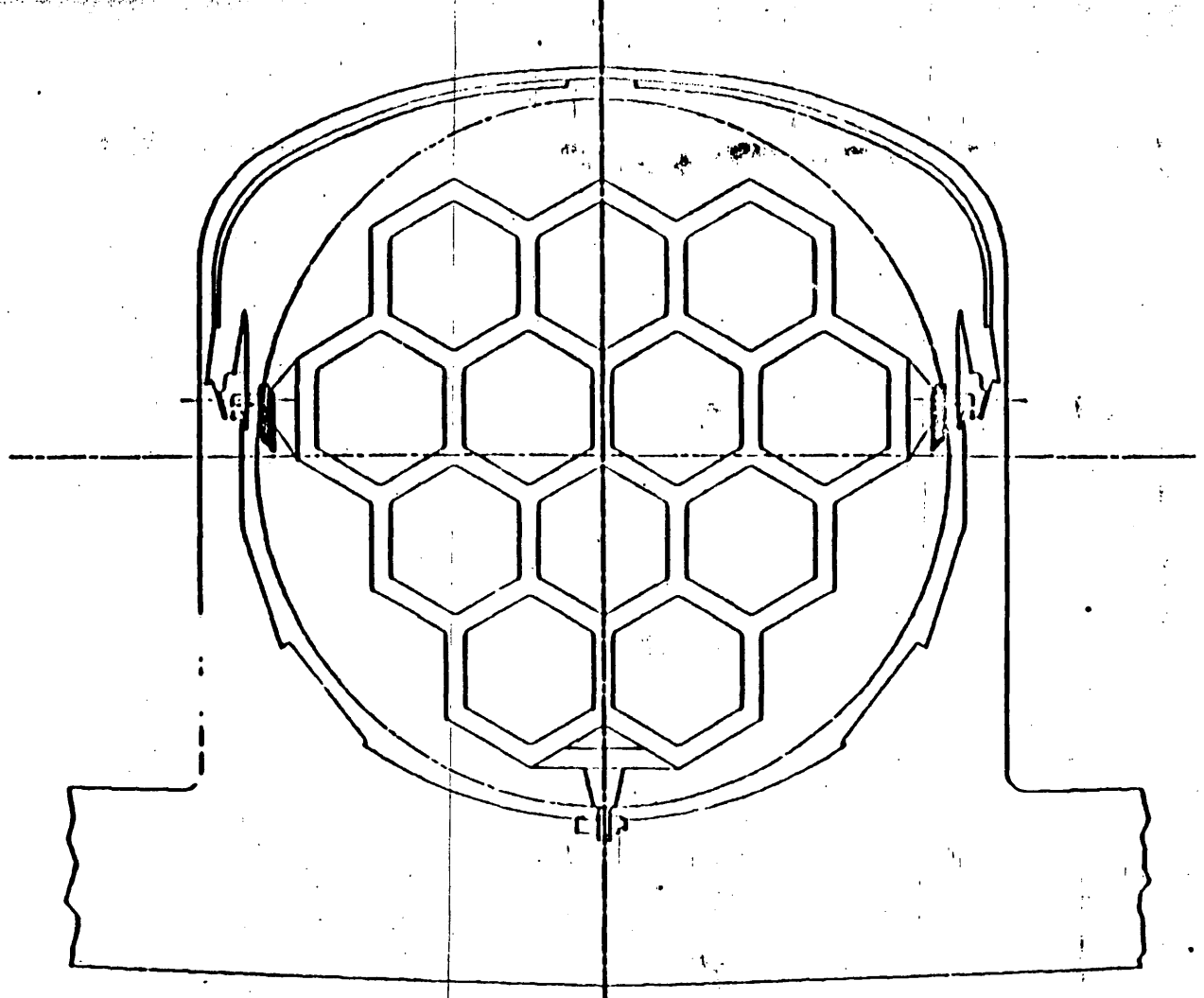



Figure 5-56 Satellite Carrier Concept 4



ESTIMATED CONFIGURATION TO ACCOMMODATE
S/C WITH 28 in. dia. ANTENNA

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Figure 5-57 Configuration to Accommodate S/C with 28 in. diameter Antenna

The impact of a size change of the MSSP is two fold. Besides the possible increase in cost due to shuttle bay length or weight increases ($1500/lb shuttle launch), a more basic factor occurs. The number of satellites per carrier on the shuttle will decrease. The phenomenon that fewer large diameter spacecraft can be launched per launch vehicle will be common for all of the possible launch methods. Figures 5-58 and 5-59 present three point curves, based upon the analysis of the three possible spacecraft diameters.

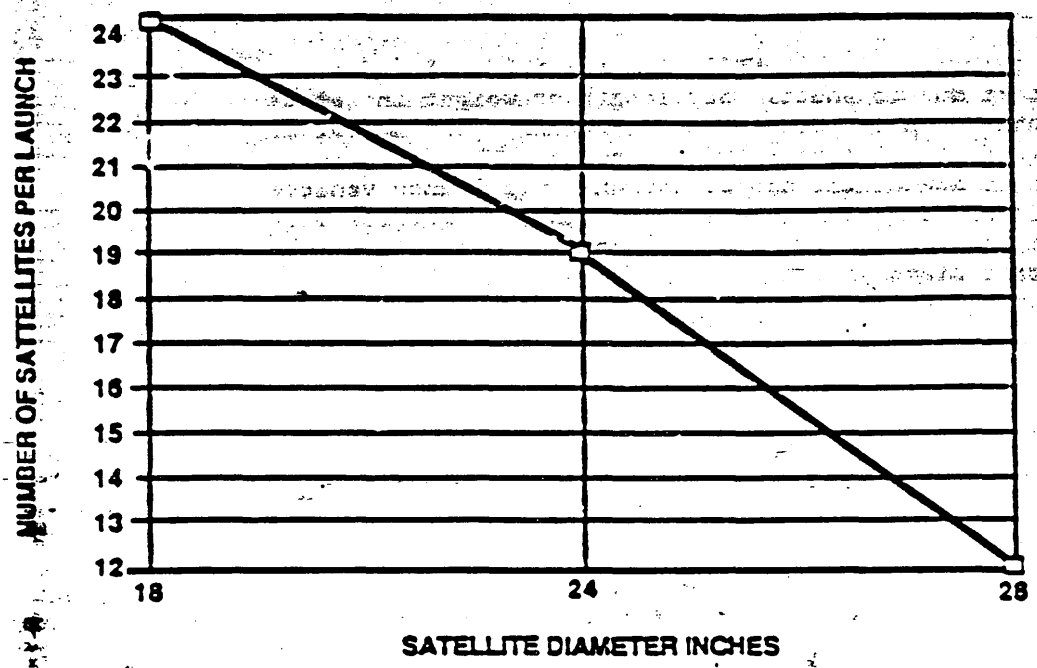


Figure 5-58 Power Output Versus Outer Panel Tilt Angle

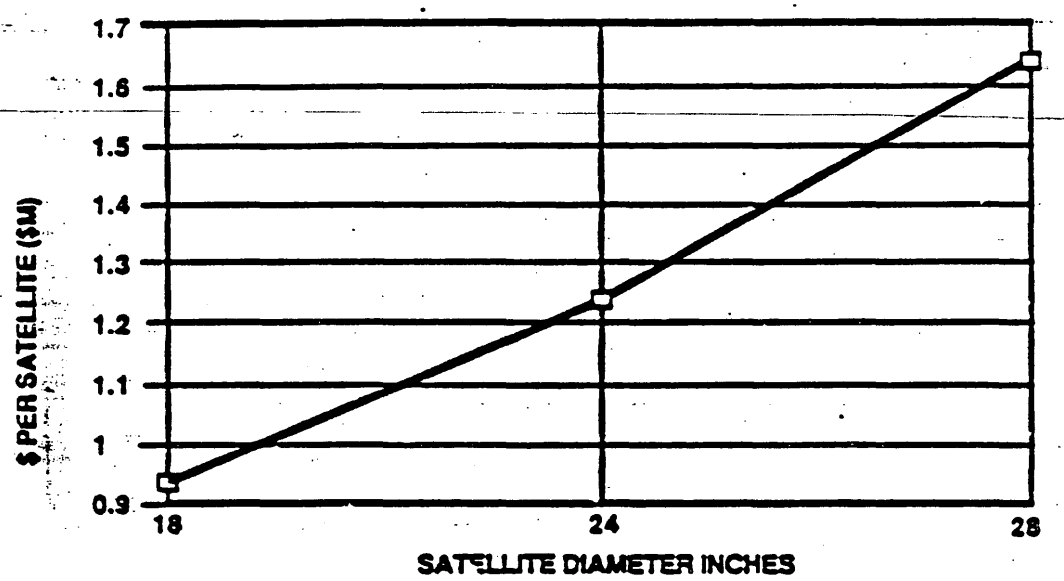


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Figure 5-59 Power Output Versus Outer Panel Tilt Angle

## 5.5 RADIO/ANTENNA INTERFACES

The analyses of the spacecraft interfaces with the radio and antenna summarized in this section were conducted to establish antenna size, antenna shape, attitude control vs. antenna pointing and the concern of operation of the antenna and radio at high power (100W) in space. These analyses were performed easier in the Phase I study effort to identify concerns. The analyses provided inputs for a satellite strawman which was presented at the early working group meetings. Many of the trade considerations were modified or the performance and interfaces of the radio and antenna were refined by those team members in the latter stages of the Phase I study.

The antenna pointing/attitude control analyses identified an interface problem which was resolved by elevation scan capability in the cross link antennas. The analyses of the possible antenna size and shape enabled a discussion with the antenna team members and an early start on the configuration analysis of the spacecraft solar arrays. The analyses of the radio/antenna high power operation demonstrated the capability of the spacecraft to operate with RF peak power averaging and identified other concerns of high RF power operation.

### 5.5.1 ANTENNA POINTING ERRORS AND ATTITUDE CONTROL

The objective of this section is to identify the antenna pointing errors and the resultant antenna pointing loss. As the error parameters are defined, a cost tradeoff of antenna pointing loss vs. attitude control can be established.

Based on antenna experience from other spacecraft, we have established two requirements for satellite antenna pointing; 1) state pointing errors and 2) show antenna gain, loss vs. error. The last requirement, loss vs. error, will determine the impact of the attitude control.

Errors associated with pointing the antenna have been determined as the following: 1. antenna mechanical error, 2. antenna pointing error 3. pointing calculation error, 4. attitude control error, spacecraft movement. The errors can be defined as follows:

1) The antenna mechanical error is the physical alignment of the antenna to the spacecraft attitude zero. The value +/- 1 deg was chosen because it can be obtained without costly alignment procedures.

2) The antenna pointing error is the real antenna pointing error or the design capability of the antenna alone.

3) The antenna pointing calculation error is the error of the digital processor to determine where to point the antenna beam.

4) The attitude control error is the satellite platform's ability to correctly orientate the antenna.

5) The spacecraft movement error is dependent upon the update time of the antenna pointing error (especially a stepped beam). It is essentially the granduiarity of the antenna beam positions.

The errors do not add to a worst case. A realistic method of calculating the total error is to calculate the root of the sum of the errors squared. The pointing error equation can be written:

$$\theta e = SQR( (\#1)^2 + (\#2)^2 + (\#3)^2 + (\#4)^2 + (\#5)^2 )$$

With careful fabrication and design the magnitudes of the mechanical, the real antenna pointing error, and the calculation error should be about 1 deg each. Therefore, the errors which are the main concern for the antenna and spacecraft are the attitude control errors and the antenna grandularity errors.

The affect of an antenna pointing error is to reduce the communication system antenna gain. This error loss will increase for higher gain antennas with narrower beamwidths. Assuming a 60 percent efficient antenna with a parabolic radiation beam shape, Figure 5-60 shows the maximum system gain vs. the pointing error. For higher antenna gains with narrower beamwidths than those shown in Figure 5-60, the final system gain with pointing losses will actually be lower. Figure 5-61 shows the system gain loss (dB) vs. pointing errors from 6 to 16 deg. Since the MSSP antennas will have gains of 18 to 20 dB the pointing error must be less than 6 deg.

The two important trades for the MSSP are the attitude control and the antenna grandularity to obtain system pointing errors less than 6 deg. The antenna grandularity is the capability of the antenna to point a beam in a particular direction. The MSSP antenna is an electrically steered or switched antenna rather than a mechanically sweeped antenna. The antenna will therefore point in stepped increments. The pointing loss will be the valleys between the beam steps as shown in Figure 5-62. The depth of the valleys will be dependent upon the number of steps which are dependent upon the physical size of the antenna. The attitude control error is dependent upon the complication of the attitude control system. For the simple gravity gradient boom, the attitude control would be uncontrolled in yaw (azimuth) and nadir-pointing (roll and pitch) within 10 deg (3 sigma). If a momentum-bias wheel is added, the attitude would be controlled in yaw and nadir-pointing (roll and pitch) within 2 deg. Thus, the capability of the attitude pointing is also grandular.

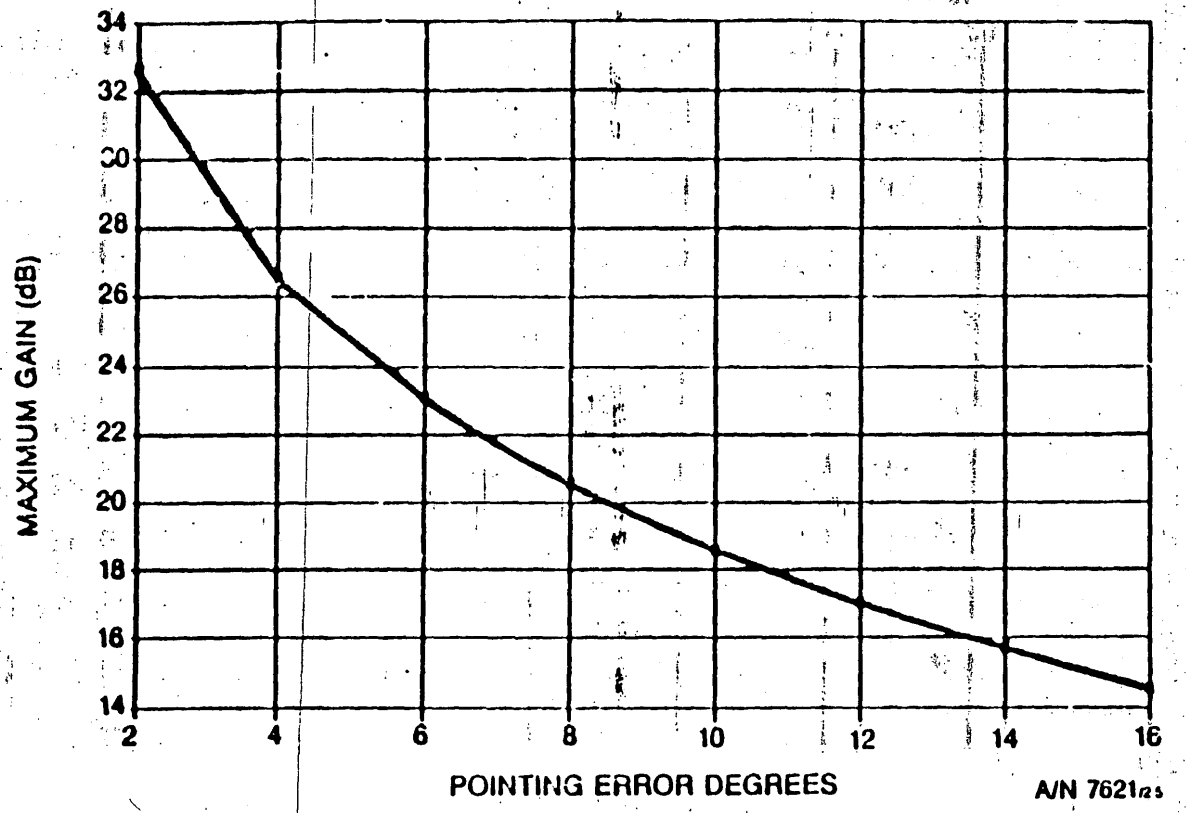



Figure 5-60 Maximum Antenna Gain Versus Pointing Error

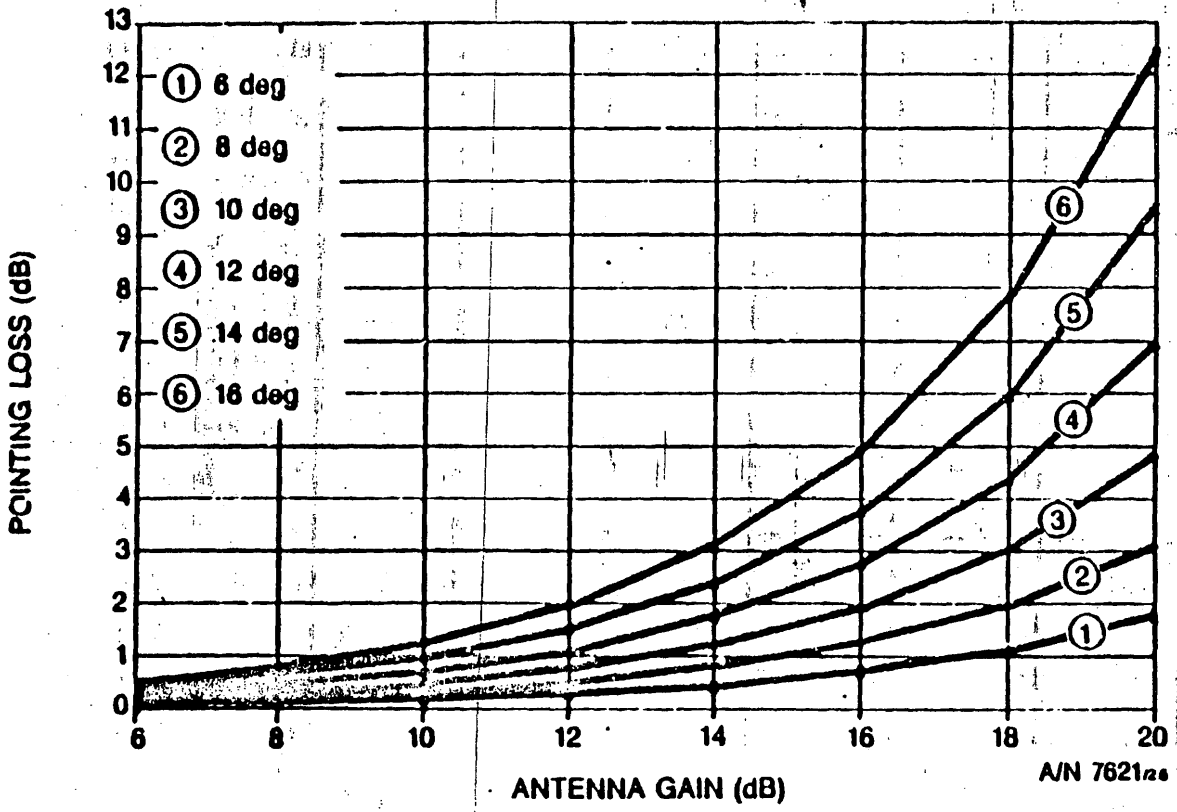



Figure 5-61 Pointing Loss vs. Gain (Pointing Errors of 6 to 16 deg)

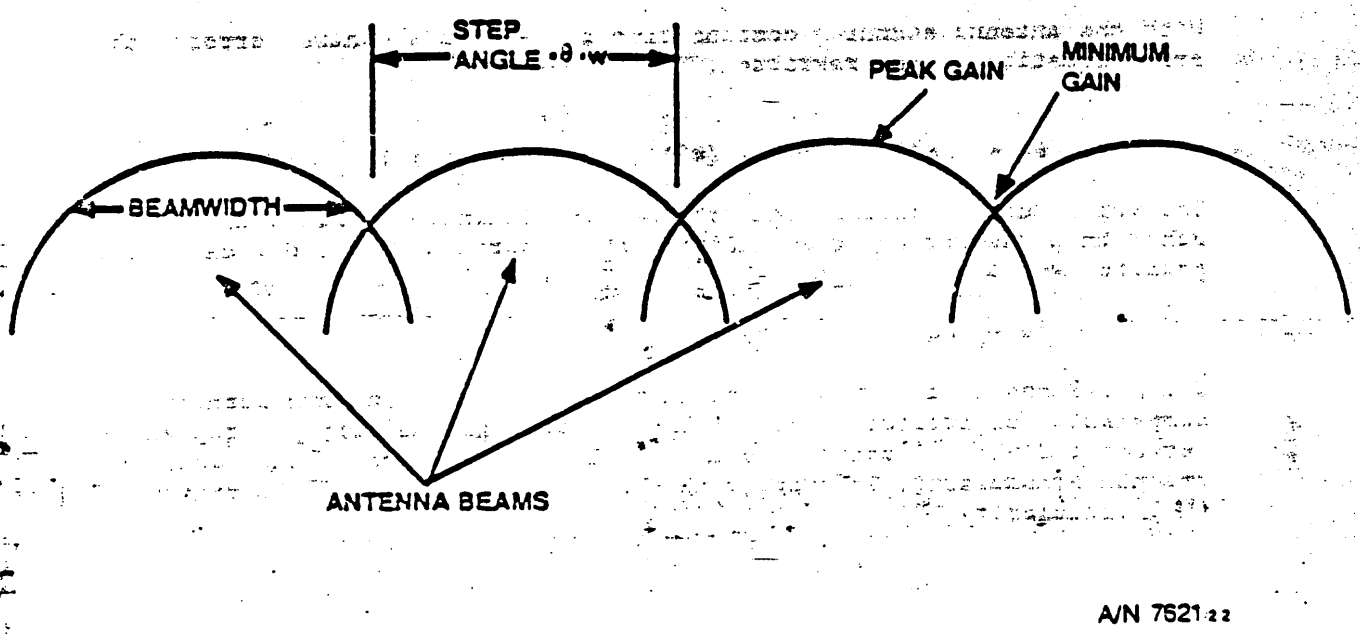



Figure 5-62 Illustration of Antenna Parameters

In addition to the trade of antenna grandularity and attitude control, is the possibility of using elevation scanning of the antenna and processing capability of the network processor. Analysis by the other team members determined that the network processor could determine the yaw, roll, and pitch orientation of the incoming RF signals. The scanning, elevation, and azimuth capability of the antenna could then be used to compensate for the attitude errors less than 10 deg (3 sigma) roll, pitch, and 360 deg yaw.

With the antenna scanning compensating for attitude control errors, the error equation can be rewritten:

$$\theta e = SQR\left( (\#1)^2 + (\#2)^2 + (\#3)^2 + (\#5)^2 \right)$$

The error due to attitude control (#4) is eliminated. Since the errors other than the antenna grandularity are approximately 1 deg each, the equation would be:

$$\theta e = SQR\left( 3 + (\#5)^2 \right)$$

Table 5-9 shows the system pointing error with antenna scanning to compensate for attitude error. The antenna grandularity is the angle between individual stepped beams. The table shows that when the antenna grandularity is greater than 2 deg, the error will be driven by the grandularity.

TABLE 5-9

System Pointing Errors vs. Antenna Grandularity

| Antenna Grandularity (deg) | Pointing Error (deg) |
|---|---|
| 1 | 2 |
| 2 | 2.6 |
| 3 | 3.5 |
| 4 | 4.4 |
| 5 | 5.3 |
| 6 | 6.2 |
| 7 | 7.2 |

The conclusion of this analysis is that communication system pointing errors must be less than 6 deg for an antenna gain of 18 to 20 dB (60 percent efficiency). The main error concerns are spacecraft attitude error and antenna grandularity. However, analysis by other team members has shown that the attitude error can be compensated by the elevation and azimuth scanning of the antenna. Therefore, the main error concern is the antenna scanning grandularity or beam step size.

The calculations of error/loss were performed assuming 60 percent antenna efficiency. If the antenna efficiency is less than 60 percent and narrower beamwidths are used to obtain the required gain, the impact of the pointing error will increase.

### 5.5.2 ANTENNA SIZE

The following analysis was performed early in the Phase II study to obtain an estimate of the size of the MSS antenna. The size impacts the required attitude control and the antenna pointing as shown in Section 5.1. Also, by estimating the size of the antenna, the analysis of the spacecraft configuration and possible solar array configuration could proceed.

To create a parametric presentation of the antenna size to determine spacecraft size, two assumptions were made:

1. The antenna is mounted on a cylinder and the antenna arrays are switched in segments around it. The grandularity of the switching around the cylinder was assumed to be $\lambda/4$.

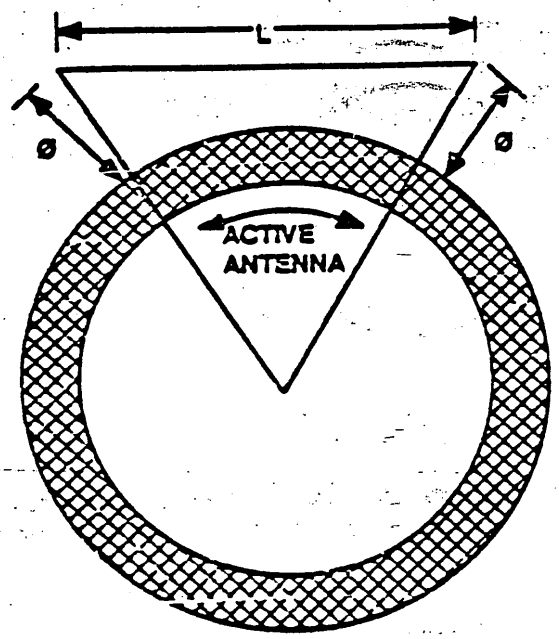
2. As an increasingly wider antenna is used, a phase error occurs (Figure 5-63) due to the displacement of the end elements off a flat plane. If the antenna were a multifaceted surface rather than cylindrical, similar considerations would have to be made. The maximum displacement was assumed to be $\lambda/4$.

Having established these assumptions, the equation for the number of antenna beam steps vs. the antenna radius is shown in Figure 5-64.

Figure 5-65 shows the antenna step angle or the number of degrees per step of the antenna beam vs. antenna radius. The projected antenna array length antenna radius is demonstrated in Figure 5-66.

Because of the allowable phase error of $\lambda/4$ at the edge of the projected antenna, the antenna 3dB beamwidth will be broader than a classical uniform array. However, with the use of the classical expression

$$Bw*L/\lambda = 52$$




L : PROJECTED ARRAY LENGTH

Ø : ASSUMED TO BE $\lambda/4$

DEFINITION OF ARRAY Ø ERROR OF $\lambda/4$

Ø : PHASE ERROR

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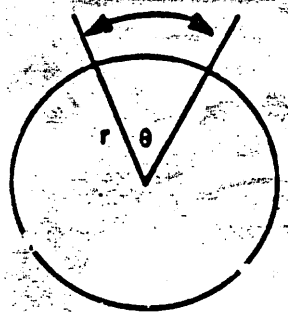
Figure 5-63 Antenna Phase Error




λ /4 ANGLE STEPPING OF ANTENNA POINTING i.e. ANTENNA POINTING CAN BE STEERED AROUND TO POINT TO DIFFERENT SPACECRAFT IN λ/4 STEPS

$$r \cdot \theta = \lambda/4 \text{ or } \theta = \frac{1}{4 \cdot r/\lambda}$$

CIRCLE IS 2 Π RADIANS

$$\text{\# OF ANTENNA POINTING STEPS} = \frac{2\Pi}{\theta} = 2\Pi \cdot 4\, r/\lambda$$

$$= 8\Pi \cdot r/\lambda$$

$$\text{DEGREES PER STEP} = \frac{360}{8\Pi \cdot r/\lambda}$$

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Figure 5-64 Antenna Beam Step: Degrees Per Step

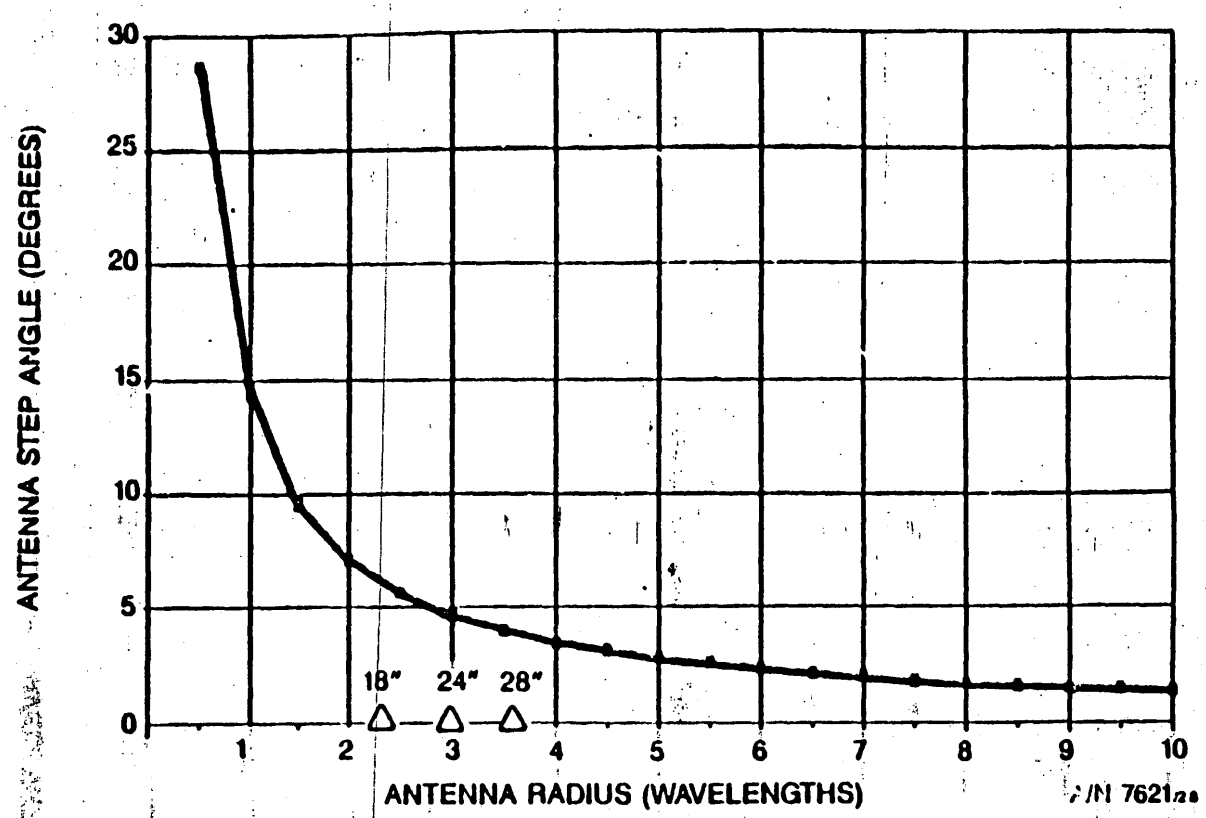



Figure 5-65 Antenna Step Angle Versus Antenna Radius (¼ Wavelength Step)

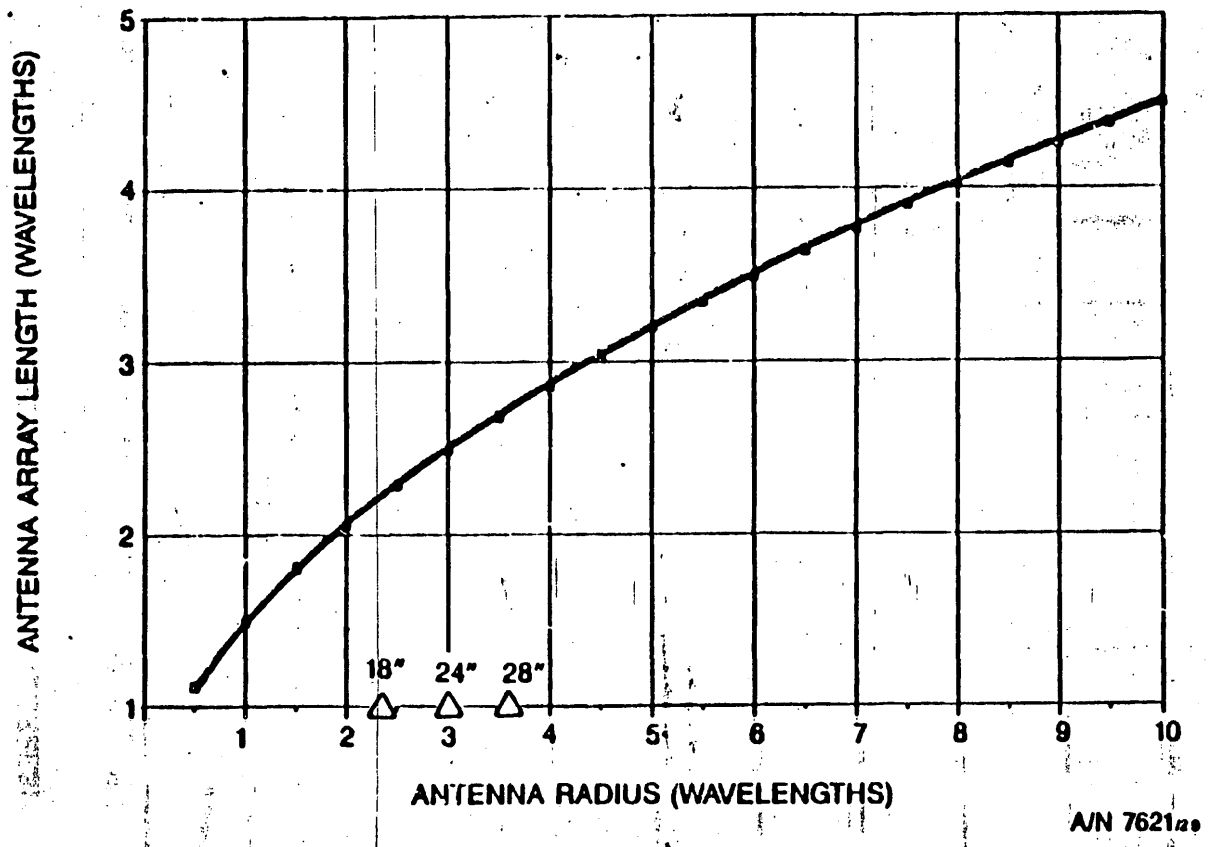



Figure 5-66 Antenna Array Length Versus Antenna Radius (¼ Wavelength Phase Error)

an approximation of the antenna beamwidth vs. the antenna radius can be obtained and is shown in Figure 5-67.

Noted on Figures 5-65 and 5-67 are the antenna dimensions of 18, 24, and 28 in. for the frequency of 3GHz. The antenna size assumed for the spacecraft analysis was 18 in. This dimension was later verified by the team antenna members. The smaller dimensions provided a lower satellite launch cost.

### 5.5.3 MSSP RF POWER CONCERNS

The major topics to be discussed include the effect of power averaging on the satellite, RF power and test considerations, receiver/PA turn-on concerns, and miscellaneous items such as the RF limiter, and PA thermal.

#### POWER AVERAGING (PA)/SPACECRAFT POWER

To vary spacecraft range, it may be necessary to vary the duty cycle and peak power of the data packets. This requires that the battery size or the rated depth of battery discharge be increased. (The battery life is dependent upon depth of discharge and number of cycles.)

The power subsystem is driven by two factors:

(a) Solar array is sized so that orbital power input is equal to orbital power required.

(b) The batteries are large enough so the drain during periods of charge below the bus requirements does not drain the batteries below the specified depth of discharge.

When the PA power is increased for short periods of time, and the orbital average bus requirement remains the same, factor (a) above is not affected, but factor (b) will require that the high power time periods be controlled. Either the high power periods can be shortened or the depth of discharge of the battery must be allowed to increase.

The power generated by the solar arrays will be dependent upon the solar array configuration, the sun angle, and orbit position. However, a simplified case of battery discharge can be derived when the Earth eclipse period is considered. For the altitude of concern, the orbit period will be about 96 min with 32 min of eclipse. The equation for the required battery capacity because of eclipse is:

$$\frac{\text{Avg power}}{28\text{V}} * \frac{32\text{ min}}{60\text{ min/hr}} = \text{Capacity} * \text{depth of discharge}$$

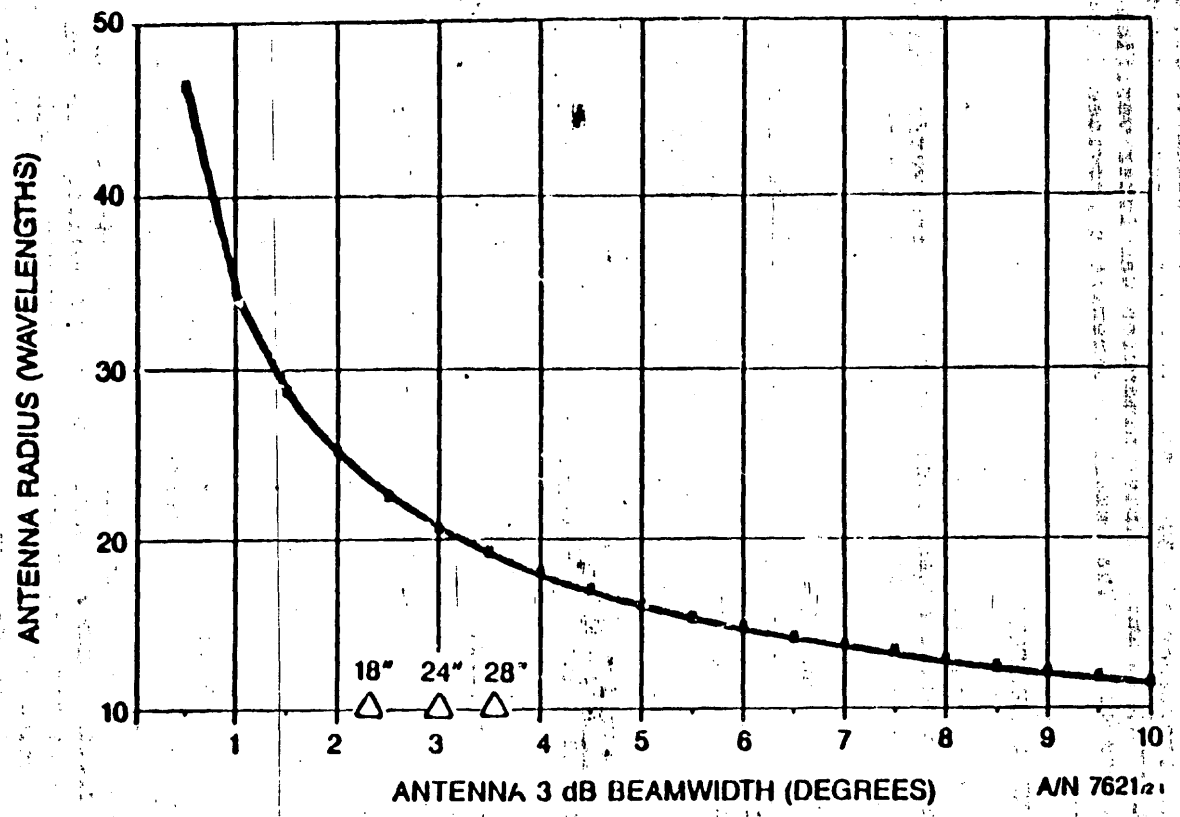



Figure 5-67 Antenna Beamwidth Versus Antenna Radius (¼ Wavelength Phase Error)

The latest power numbers for MSSP are:

PA power: 10W

Other power: 64W

74W/0.75 = 98.7W (75 percent convertor efficiency)

or a 36 amp-hr battery would work and the final depth of discharge (DoD) would be 5.2 percent.

The 10W PA is based on:

10W peak RF; 30 percent duty factor; 30 percent efficiency; 75 percent dc-dc convertor efficiency.
10 x 0.3/0.3 = 10W average or 13.33W with dc-dc

For 100W operation the PA power is:

100W peak; 10 percent duty factor; 30 percent efficiency; 75 percent dc-dc efficiency.
100 x 0.1/0.3 = 33.33W avg. or 44.4W including convertor efficiency

The bus required power during high power operation would be (64 + 33.3)/0.75 = 129.8W. The impact of high power operation would be to:

1) allow the depth of discharge (dod) to increase

$$\frac{129.8}{28} \times \frac{32}{60} \times \frac{1}{36} = 6.9 \text{ percent}$$

2) or reduce the transmit time

$$\frac{98.7}{129.8} \times 32 \text{ min} = 24.3 \text{ min}$$

Either of these options are acceptable compromises. The impact is lower than first expected because the bus power has increased to where the percentage effect of the PA power is lower. This analysis is optimistic because the orbital charge is somewhat like a sine wave rather than an on/off eclipse/charge period. The periods of solar array output less than spacecraft power requirement will be greater than the eclipse period (depending upon array orientation). If the high power operation were to start at the end of an eclipse, when the

batteries have already discharged during normal eclipse operation, and the array output has not yet increased, the operation at high power could drag the battery dod down to 8-10 percent.

Transmitting for 24.3 min with high power will affect the amount of 10W transmit time. Dead time or periods of no transmission will be required.

The length of the dead time is given by:

(33.3 - 10) * time (high power) = 10 * dead time

dead time = 57 min dead time = 2.33 * time high power

Twenty four minutes of high power operation will shorten low power time to only 15 min.

If batteries with lower allowable dod are used, the problem is worse because a larger dod is multiplied by the power difference. This could be solved by spending more money to buy larger batteries.

The impact of the high power operation will be dependent upon many orbital parameters, but the design could proceed in either of two methods:

a) state powers required (74W to bus) and high power operation time-limited to 10 min P/A power = 33W. Dead time = 24 min.

b) state required high power operation time required (TBD minutes 33W P/A anytime) and the average power 74W. The satellite designer will upgrade the power system to accomplish the desired DoD. Dead time = TBD.

Either case would define a baseline interface and designs so that trades could proceed.

Another concern of peak power averaging is the system thermal design. The low power operation of 10W average power will dissipate an average of 7W with 3W average transmitted. The high power operation will dissipate 23W and transmit 10W average. During some periods, the PA will dissipate no power. The thermal design will be required to control the satellite temperature for all of these wide dissipated power swings. The swings of 64W bus-power to 97W or about 50 percent variation, independent of orbit, may require more sophisticated forms of thermal control, such as thermal louvers, which can be expensive. The final determination is dependent upon the thermal interfaces of the radio, antenna, and satellite.

* RF POWER AND TEST CONCERNS

The normal criteria for thermal vacuum testing is that components shall be subjected to acceptance level thermal vacuum testing in lieu of thermal cycling tests if they meet any of the thermal vacuum test criteria defined below:

- o have terminals exposed to vacuum and carrying over 50V
- o parts not hermetically sealed and carrying over 500V
- o parts exposed to vacuum, dissipating over 30 percent of their in-air rating and for which a thermal tolerance analysis on the component mounting does not exist.

Above 50W RF (actually 30W with 2:1 VSWR-Voltage Standing Wave Ratio) the spacecraft, antenna, and radio must be thermal vacuum-tested. The RF transistors must be analyzed and tested to ensure that their thermal time constant is such that the 100W operation will not exceed the thermal rating in vacuum. The transistor junction temperature must be designed for a MTTF of five years minimum. The thermal properties of high power pin diode switches and phase shifters must be analyzed and tested.

The RF capability of SMA connectors is about 82W. With a safety factor of 2:1, the maximum power would be 40W. Therefore, TNC type connectors (circuitous dielectric interface) will be required. Also, to ensure rapid bleed-down at vacuum, the connectors must be vented (drilled).

* TURN ON/OFF SWITCHING TIMES

The discussions for protocols and routing has assumed time lines with transmission periods and receive periods but no switching time in-between. Switching time would be periods of transition between transmit and receive. There are concerns about the following areas of operation which would cause switch time delays.

- o P/A operation
- o T/R switching
- o Antenna switching

P/A operation: The circuit for the operating PA is shown in Figure 5-68. The turn on of the PA will require very low inductance in the battery, regulator, wiring and P/A. A solution might be to place a large capacitor as close to the P/A as possible. However, if the P/A on/off switch is before the capacitor, the turn on will be slowed due to charging of the capacitor.

Another solution would be to turn the P/A on continuously, charge all capacitors, and be ready for the transmitter on/off. The impact of this will be small because the P/A is class C and should draw low power (~1W) while in standby between transmissions.

T/R switching: Two possible RF configurations are shown in Figures 5-69 and 5-70. The system is half duplex transmitting and receiving on the same frequency. When transmitting 100W (ignoring filter losses for now), a circulator would provide 20+ dB of isolation. However, a 2:1 VSWR antenna would provide only 10 dB of reflection loss. In other words, 10W, if peak power, would be reflected back to the receiver. A T/R switch may be needed to protect the receiver front end. Also, for the switched beam or phased array antennas, end of life degradation might cause VSWRs much higher for certain beams and therefore, the T/R switch might be required to handle higher powers than 10W.

The questions to be answered include: What will the switching time of the T/R switch be? What will be the isolation of the switch? If the receiver upper dynamic range is 0 dBm, the reflected 10W will need 40 dB of switch isolation. Even then, the receiver may require time to recuperate from the 0 dBm signal.

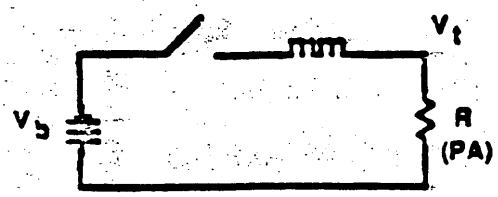
Antenna switching: The antenna is specified at <1us switching time. This specification states that a 1us dead time must be apportioned to the protocol.

The switching speed of a pin-diode switch is dependent upon 1) on: the RF power through-put and insertion loss - large current to offset the RF power, 2) off: the RF power and isolation required - high voltage to shut off diode and offset RF power, 3) the diode package design and switch package design to handle cooling of diode (due to dc and RF loss) in a vacuum, and 4) the switching speed of the switch driver, which supplies the voltages to the pin-diode. The 1us switch time for MSSP will be the sum of the pin-diode switch and switch driver time. Previous experience has shown rapid on-switching of pin diodes, but the off can be longer depending on system line capacitance.

A summary of switching concerns can be answered by the following questions:

| | |
|---|---|
| PA on or off continuous | usec |
| If off: on-time | usec |
| T/R on/off-time | usec |
| Antenna switch time | 1 usec |

Since all of these times occur in parallel, the longest time will be the system dead time.




$P = V^2/R$  $V = 28$

$P = 100/.3 = 333$ WATT

$$R = \frac{(28)^2}{333.33} = 2.35\,\Omega$$

$V_t = V_b (1 - e^{-tR/L})$

$t = 1\,\mu$ sec; $R = 2.35\,\Omega$  $V_b = .95\,V_t$  Then $L \leq 0.8\,\mu h$

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Figure 5-68 PA Turn On

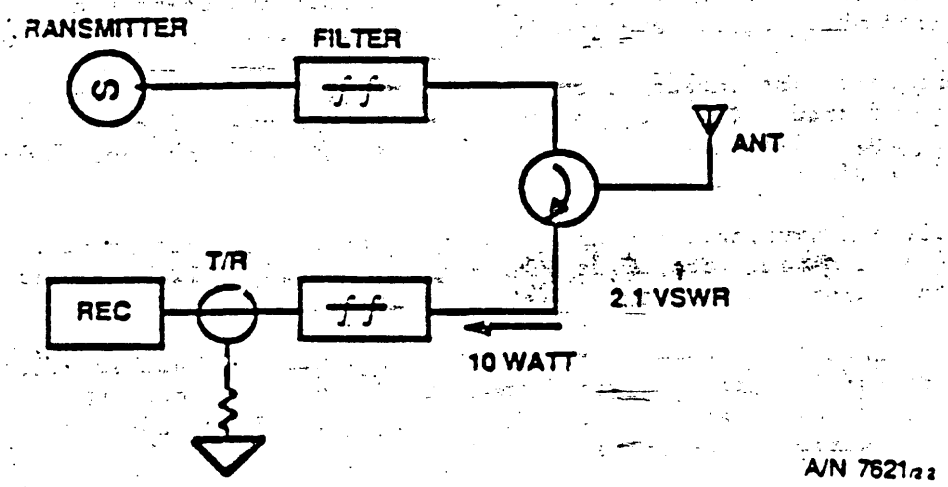


Figure 5-69 RF Configuration 1

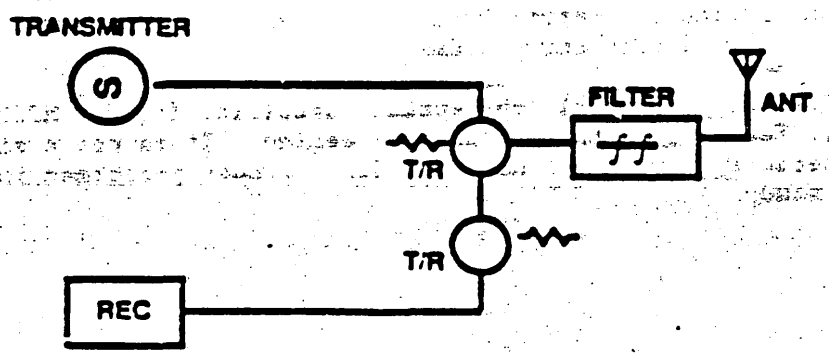


Figure 5-70 RF Configuration 2

## 6.0 Conclusions and Recommendations

The MSSP Phase I study is not a presentation of a single point satellite design but a combination of trades and analyses leading to the preliminary design of a low cost satellite communication system. The study was based upon the interaction between the Phase I team members: antenna, radio, and satellite integration such that an optimum cost-effective design could be obtained.

During this study several general conclusions were obtained:

- Orbits: The satellite orbits should be at a high inclination for the view capability by the Northern Hemisphere where most of the system users would be located. A satellite altitude of approximately 675 km would create a sufficient orbit lifetime while providing lower launch cost and lower radiation levels. A technique has also been presented for satellite dispersion using nodal regression which could save considerable system cost.

- Ground User: The impact of the system design cost due to the multitude of ground users can be great. If peak power for high elevation angle satellites is used, the system cost and complexity could be reduced.

- Satellite Communication Range: The satellite design should be capable of maximum communication range. This capability increases communication probability and system reliability.

- Power: Satellite power requirements are the largest satellite cost drivers. The analyses show the impact of cost per watt (~$5000 per watt) and how the solar array configuration was analyzed to maximize the array orbital average output.

- Attitude Control: The attitude control necessary from the satellite is dependent upon the antenna capability and cost. The primary concern addressed in the analysis is the conflict between attitude control and the antenna pointing requirements. The cost of the dual scan, elevation, and azimuth crosslink antenna estimated by the antenna team members directed the result of the analysis. The attitude control would be a simple gravity gradient boom system with a eddy current damper.

Figure 6-1 illustrates a final conceptual satellite for the MSSP and Table 6-1 lists features of the satellite design. It is not a single point design because if will depend upon final trades performed in the MSSP Phase II study.

TABLE 6-1
PROMINENT SATELLITE DESIGN ASPECTS

| | |
|---|---|
| Power: | 75W system of 6 panels deployed for maximum orbital power. 36 amphr battery for low depth of discharge and current-limiting power control. |
| Attitude Control: | Gravity gradient boom with eddy current damper for nadir-pointing 10 deg 3$\sigma$ |
| Structure: | Tubular frame, low cost construction, with thermal blanket walls. |
| Preferred Launch Mode: | Shuttle launch with OMV for final orbits, satellites deployed for nodal dispersion. |
| Command and Telemetry: | Command and telemetry provided through interface with network processor. |
| Thermal Control: | Thermal heater compensating for no-transmit periods. Passive thermal blankets and radiators. |

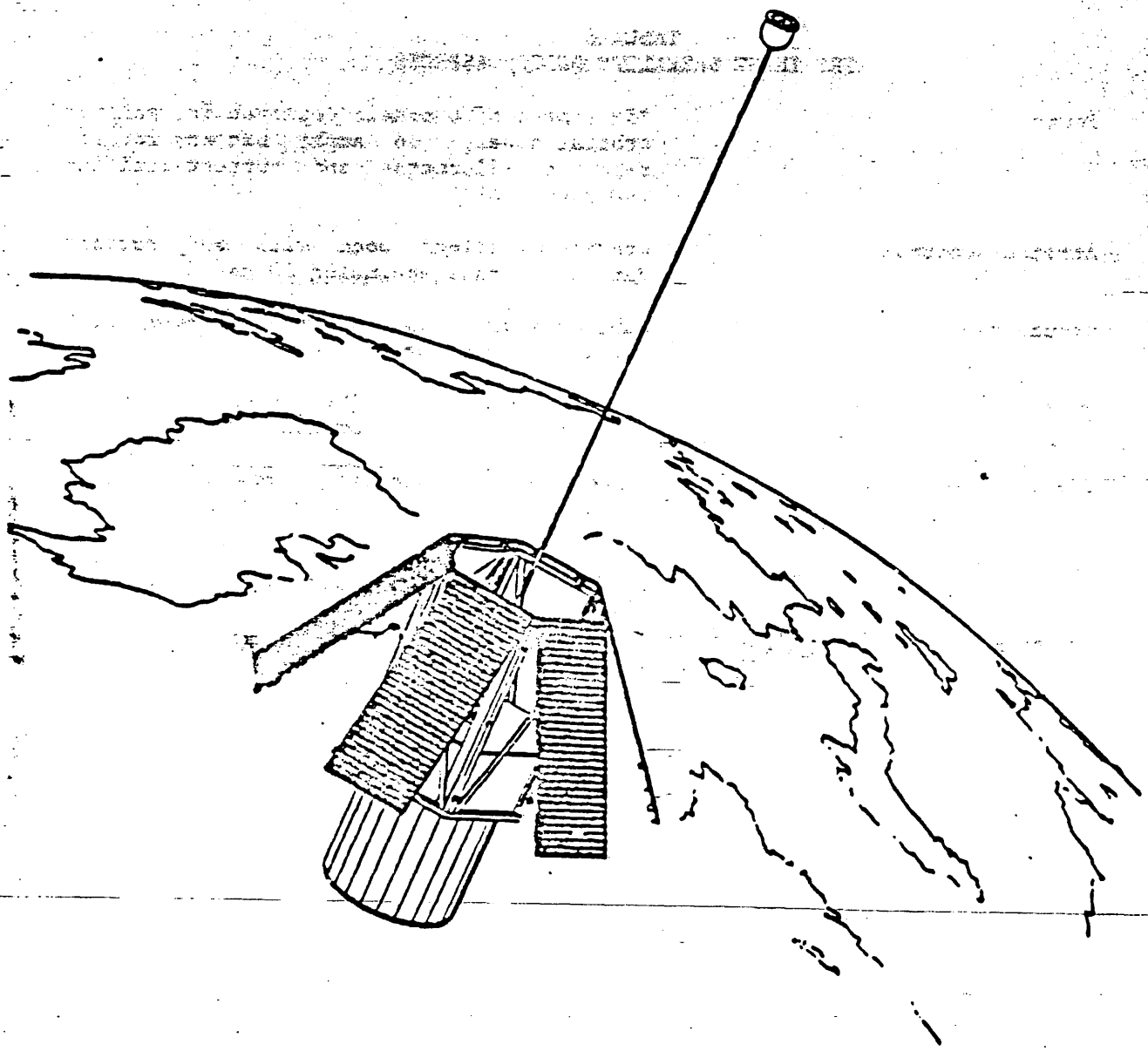

Figure 6-1 MSS Satellite Concept

1. Jacchia, L. G., "Thermospheric Temperature, Density, and Composition: New Models" Smithsonian Institution Astrophysical Observator: SAO Special Report No. 375, March 1977

2. King-Hele, Desmond "Theory of Satellite Orbits in an Atmosphere", Butterworth & Co. Ltd., London, 1964

## Acronym List

| | |
|---|---|
| AAS - | American Astronautical Society |
| ADACS | Attitude Determination and Control System |
| AGC - | Automatic Gain Control |
| Al - | Aluminum |
| AO - | Atomic Oxygen |
| Ap - | Geomagnetic Indices |
| APL - | Applied Physics Laboratory |
| A/M - | Area/Mass ratio |
| BASD - | Ball Aerospace Systems Division |
| Cd - | Coefficient of Drag |
| cm - | centimeter |
| CMOS - | Complementary Metallic Oxide Semiconductor |
| dB - | decibel |
| deg - | degree |
| ELV - | Expendable Launch Vehicle |
| ERBS - | Earth Radiation Budget Satellite |
| ETR - | Eastern Test Range |
| ft - | feet/foot |
| FOV - | Field of View |
| GEOS - | Geodetic Earth Orbiting Satellite |
| GHz - | GigaHertz |
| GSAT - | Gravity Satellite |
| hrs - | Hours |
| ICs - | Integrated Circuits |
| IGRF - | International Geomagnetic Reference Field |
| kg - | Kilogram |
| km - | Kilometer |
| LHC - | Left Hand Circular |
| min - | minutes |
| MSFC - | Marshall Space Flight Center |

## Acronym List (cont)

| | |
|---|---|
| MSS - | Multiple Satellite System |
| MSSP - | Multiple Satellite System Program |
| MTTF - | Mean Time to Failure |
| NASA - | National Aeronautics and Space Administration |
| NF - | Noise Figure |
| NiCad - | Nickel Cadmium |
| nm - | nautical mile |
| nmi - | nautical mile |
| OMV - | Orbital Maneuvering Vehicle |
| rad - | radiation units |
| RF - | Radio Frequency |
| RMS - | Remote Manipulator System |
| ROM - | Read Only Memory |
| RSS - | Root Sum Squared |
| sec - | second |
| SER - | System Engineering Report |
| SHIELDOSE - | Radiation Calculation Program |
| SMA - | RF connector Sub-Minature Type A |
| SOFIP - | Short Orbit Flux Integration Program |
| SRM - | Solid Rocket Motor |
| STS - | Space Transportation System |
| temp - | Temperature |
| TBD - | To be determined |
| TLM - | Telemetry |
| TNC - | RF Connector Type TNC |
| VSWR - | Voltage Standing Wave Ratio |