NASA CR-159,735

NASA-CR-159735
19800007836

NASA CR-159735
D180-25475-1

ELECTRIC PROPULSION

FOR

NEAR - EARTH SPACE MISSIONS

BOEING AEROSPACE COMPANY

prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA Lewis Research Center
Contract NAS 3-21346

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
CR-159735		

4. Title and Subtitle	5. Report Date
ELECTRIC PROPULSION FOR NEAR-EARTH SPACE MISSIONS	January 1980
	6. Performing Organization Code
7. Author(s) C. H. TERWILLIGER, W. W. SMITH	8. Performing Organization Report No. D180-25475-1
	10. Work Unit No.
9. Performing Organization Name and Address BOEING AEROSPACE COMPANY P.O. BOX 3999 SEATTLE, WA 98124	11. Contract or Grant No. NAS 3-21346
	13. Type of Report and Period Covered FINAL REPORT FEB. 1978 - APR. 1979
12. Sponsoring Agency Name and Address NASA/LEWIS RESEARCH CENTER 21000 BROOKPARK ROAD CLEVELAND, OH 44135	14. Sponsoring Agency Code

15. Supplementary Notes

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16. Abstract

A set of missions was postulated that was considered to be representative of those likely to be desirable/feasible over the next three decades. The characteristics of these missions, and their payloads, that most impact the choice/design of the requisite propulsion system were determined. A system-level model of the near-Earth transportation process was constructed, which incorporated these mission/system characteristics, as well as the fundamental parameters describing the technology/performance of an ion bombardment based electric propulsion system. The model was used for sensitivity studies to determine the interactions between the technology descriptors and program costs, and to establish the most cost-effective directions for technology advancement. The most important factor was seen to be the costs associated with the duration of the mission, and this in turn makes the development of advanced electric propulsion systems having moderate to high efficiencies (>50%) at intermediate ranges of specific impulse (~1000 seconds) very desirable.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
ION ROCKET PRIMARY PROPULSION EARTH ORBITAL TECHNOLOGY ADVANCEMENT COST MODELING	UNLIMITED

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
UNCLASSIFIED	UNCLASSIFIED		

* For sale by the National Technical Information Service, Springfield, Virginia 22161

N80-16096 #

FOREWORD

This document is the Final Report for the "Electric Propulsion for Near-Earth Space Missions" study. This study was performed by the Boeing Aerospace Company during the period of February 1978 thru April 1979. This study was performed for the Lewis Research Center (LeRC) of the National Aeronautics and Space Administration (NASA) under contract NAS3-21346.

Acknowledgement is made to the following personnel who contributed suggestions and information that enabled this study to be successfully completed:

NASA Headquarters:	Mr. W. Hudson
Lewis Research Center:	Mr. D. Byers
	Mr. J. Cake
	Mr. J. Pelouch
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TABLE OF CONTENTS

Page

LIST OF FIGURES

SUMMARY

The objective of the study reported herein was to identify those areas in the field of electric propulsion technology where advances in the state-of-the-art are required to allow development of propulsion systems which will meet the requirements and constraints of the probable near-Earth space mission set for approximately the next three decades, and to establish the general nature of these advances as guidelines for ensuing technology efforts. Four activities were accomplished, essentially in sequence, to meet this goal: (1) the establishment of a representative mission set; (2) the definition of mission requirements and the corresponding payload characteristics; (3) the development of a system level model for a primary electric propulsion system; and (4) the conduct of studies of the cost impacts of changes in electric propulsion technology, and in system design philosophy, over the mission set.

Reviews of available literature, in-house studies of future mission needs, forecasts of improvement trends in supporting technologies, and considerations of possible scenarios for the development of near-Earth space led to the establishment of 68 potentially desirable/feasible missions. Of these, 30 were selected as representative of a future characterized by a moderately vigorous pursuit of space activities. Programmatic and physical characteristics of each of the selected missions and their respective payloads were determined from existing documentation or mission/configuration design analyses, as necessary. The mission requirements were derived by establishing six types of trajectories and performing a number of trajectory simulations of each type to define the parameters needed for later cost modeling. A system-level model of the near-Earth transportation process was constructed, which combined simplified representations of the payloads, the mission trajectories, the electrical power source, and the Earth-launch system, with the fundamental parameters describing a generic electric propulsion system based upon ion bombardment technology.

This model was used to predict the costs and propulsive performance, across the 30 mission set, for 4 design philosophies: (1) state-of-the-art systems; (2) systems which minimize power requirements; (3) time-constrained/minimized systems; and (4) cost-optimized systems. Then, cost/mission sensitivities to the various technology parameters were established, and interactions between certain system/technology descriptors were determined.

Whereas past development efforts have emphasized reductions in the specific weights of electric propulsion components, this was seen to be less critical for future missions, in which the payloads themselves will be the greatest contributor to total system mass. The commercial nature of future missions will result in a greater importance being attached to the costs associated with the duration of the propulsive phase. To reduce mission times, the development of advanced electric propulsion systems having moderate to high efficiencies (>50%) at intermediate ranges of specific impulse (~1000 seconds) was seen to be very desirable.

1.0 INTRODUCTION

1.1 STUDY BACKGROUND AND OBJECTIVES

Historically, this nation's space program has been the cutting edge for new technology. The goals and objectives of our mission planners seem to be always sufficiently ambitious as to require continual progress in the development of scientific instruments, spacecraft subsystems, and space transportation vehicles. As a result, NASA's Office of Aeronautics and Space Technology (OAST) must continually reassess the direction of its research and development efforts to ensure that the requisite technologies will be in-place to support the goals and missions of the NASA.

It is particularly appropriate that technology needs in the field of electric propulsion be re-examined at this time for at least two important reasons. First, past and current programs have been aimed at the perfection of the 8 and 30-cm mercury ion bombardment thruster systems into useful items of mission hardware. With work on flight test hardware for the 8-cm system now in progress, and with the committment of the 30-cm system to a major flight program imminent, these goals are nearing fruition. Second, the decade of the seventies has seen the development of a powerful new means of access to near-Earth space, the Shuttle-based space transportation system (STS). With the approach of the STS era, new missions have been suggested to make use of this versatile new tool and to benefit mankind; missions which are bolder, more aggressive, and more numerous than have heretofore been attempted. In addition to the STS, many of these new missions will require advances in other supporting technologies, such as electric propulsion.

Recognizing these circumstances, NASA's Lewis Research Center in early 1978 contracted for this study. The objective of this study is to identify those areas in the field of electric propulsion technology where advances in the state-of-the-art are required to allow development of propulsion systems which will meet the requirements and constraints of the probable near-Earth space mission set for approximately the next three decades, and to establish the general nature of these advances as guidelines for ensuing technology efforts.

1.2 STUDY GUIDELINES AND CONSTRAINTS

The NASA statement of work set forth certain constraints to guide the conduct of the study. These groundrules helped ensure that the study results would be of maximum usefulness to the NASA, and would be complementary to other current investigations.

1) This study was restricted to missions in the "near-Earth region only. This constraint allowed a concentration on the missions whereby mankind will begin to utilize the space program for the betterment of conditions on Earth. Any consideration of deep space exploration missions was avoided, as their requirements were being addressed by others.

2) This study was restricted to consideration of primary propulsion applications only. The mission needs for primary propulsion functions have long been established to be sufficiently different from those of attitude control and station-keeping that the development of separate systems is generally warranted.

3) This study was originally restricted to consideration of ion bombardment electric propulsion systems only. This groundrule was considered necessary to ensure an adequate depth of investigation for the available contract resources. As the study progressed, and the effort was directed away from a "design" orientation, toward a parametric examination of system impacts and sensitivities, this guideline became of less importance. In the end, the final conclusions are believed to be valid for any type of electric propulsion system.

4) This study considers that any propulsion-dedicated power sources are photovoltaic only. This constraint forced a consideration of the effects (time and cost) of solar array degradation, and introduced additional complications (trajectory optimization and steering penalties) into the calculations of system performance, however it had little effect on the final conclusions.

5) It was originally a goal of the study to emphasize commonality in system design. As the study was directed away from a design orientation, this groundrule became of little influence.

6) This study endeavored to make maximum use of past results and of the data and experience base that exists. In particular, an extensive literature search was specifically required by the contract statement of work. In addition, a review of the current state-of-the-art (SOA) in electric propulsion technology was furnished by the LeRC at study initiation.

1.3 METHODS OF APPROACH

As originally conceived, this study was to be made up of four analytical tasks, plus two support tasks for documentation and review presentation. The inter-relationships of the original tasks are shown in figure 1-1.

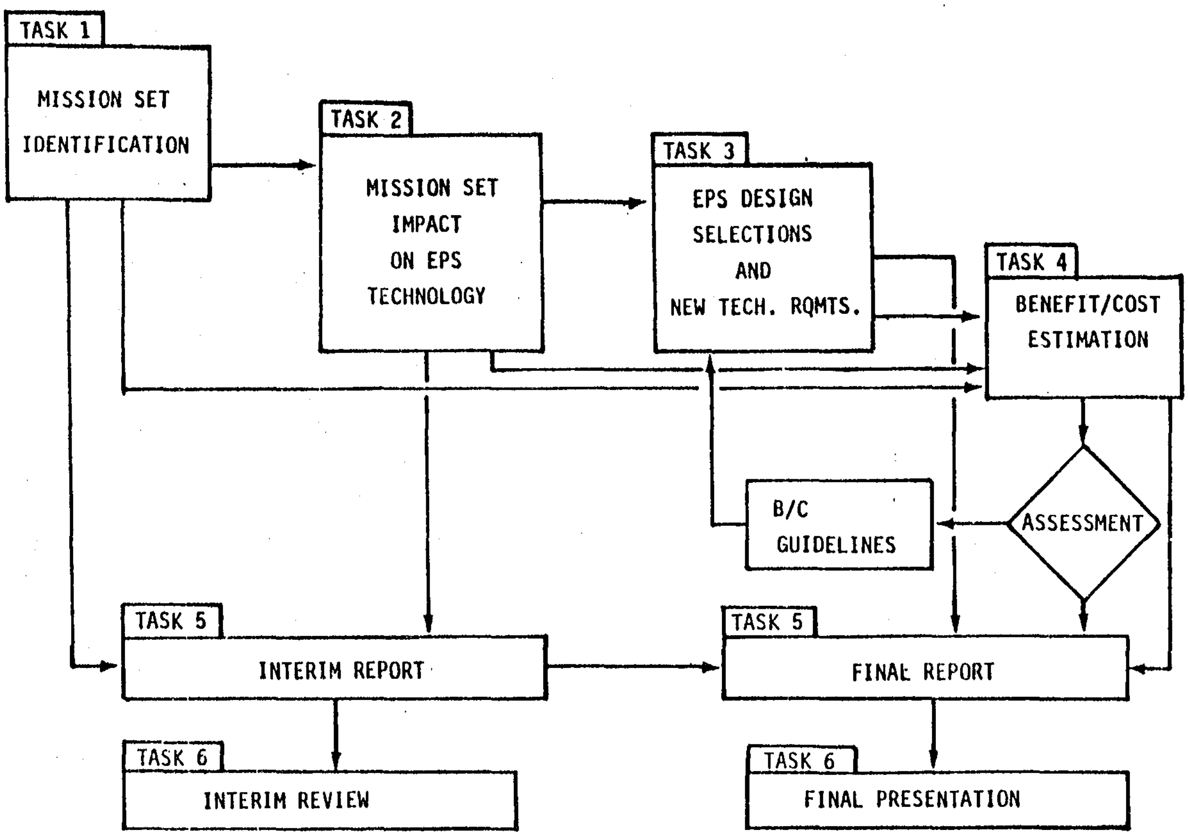

FIGURE 1-1 Original Study Task Flow

In task 1, a set of missions was identified to provide a basis for the assessment of electric-propulsion technology. Task 1 also included a review of available related literature. Section 2 of this report will discuss this effort in more detail. In task 2, comprehensive analyses of each of the selected missions was performed to define the requirements and constraints of each payload and to determine the characteristics of each of the several types of trajectories. This activity established a data base to be used for the remaining study tasks. The results of task 2 will be given in section 3.

As originally conceived, in task 3 a number of designs for advanced technology electric propulsion systems would be formulated. Thru a suitable grouping of the mission requirements, a minimum set of these designs could be selected, which would then be optimized to fit the mission set. The requirements for new technology to support the selected set of designs was envisioned as the final study output. In task 4, cost estimates for each of the potential electric propulsion designs were originally to have been developed, and the economic impacts of those systems on the overall mission set determined. These data were to be fed back into task 3 to influence the selection and optimization of the system designs, and hence their requirements for technology advancement. These activities were not implemented.

At approximately the half-way point, the approach to achieving the study objective was reassessed. It was concluded that the goal of recommending beneficial directions for technology advancement would be best served by a three-level approach as shown in figure 1-2. In the first level of analysis, the electric propulsion system would be treated as a "black box", represented only by its top-level characteristics (i.e. specific weight, cost, efficiency, etc.). In the second level, the system could be broken down into its constituent subsystems, with each represented as a "black box", and the relationships between these subsystems being of importance. Finally, the characteristics of the hardware components could be modeled for each subsystem, thus allowing study of the engineering design parameters. It was then realized that the original design-oriented approach prematurely

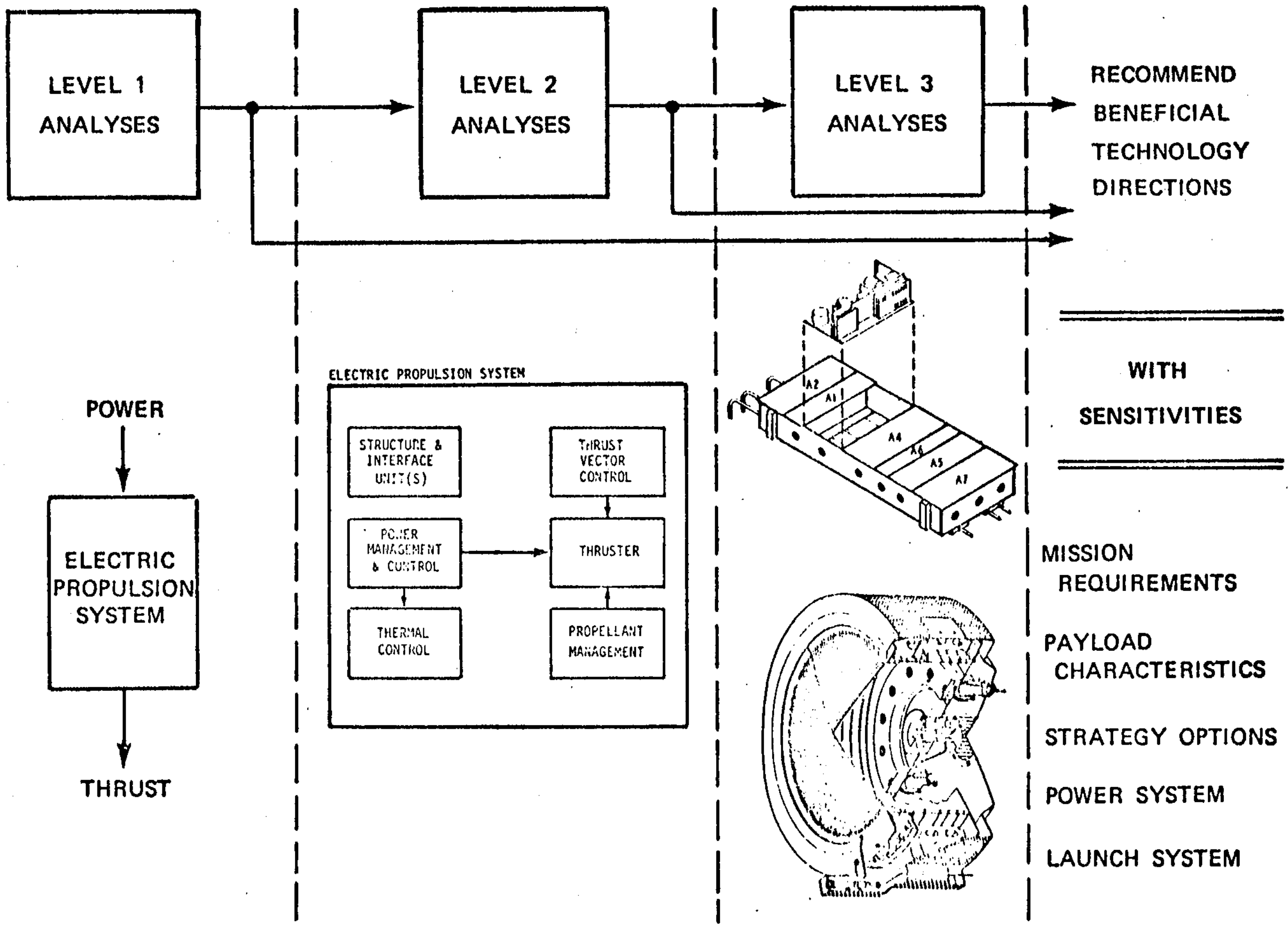

FIGURE 1-2 **Approach to Determination of Technology Needs**

focused on hardware characteristics and potential implementation options for advanced technology systems. First, an understanding of the relationships between the mission requirements and the overall system characteristics (shown as the level 1 analyses in the diagram) was needed by the NASA.

Accordingly, the remainder of the study was restructured, as shown in figure 1-3, to provide these outputs. In the revised task 3, we developed a simplified model to evaluate the cost and performance of a generic electric propulsion across the set of missions. In task 4, we then exercised that model to determine the benefits of certain changes in the elements that characterize the electric propulsion technology. Studies were also conducted to establish the sensitivity of these changes to our input assumptions, prior to an assessment of the results and a formulation of our final conclusions. A description of the analytical model, and its inputs, will be found in section 4 of this report. The results of the parametric studies will be presented in section 5.

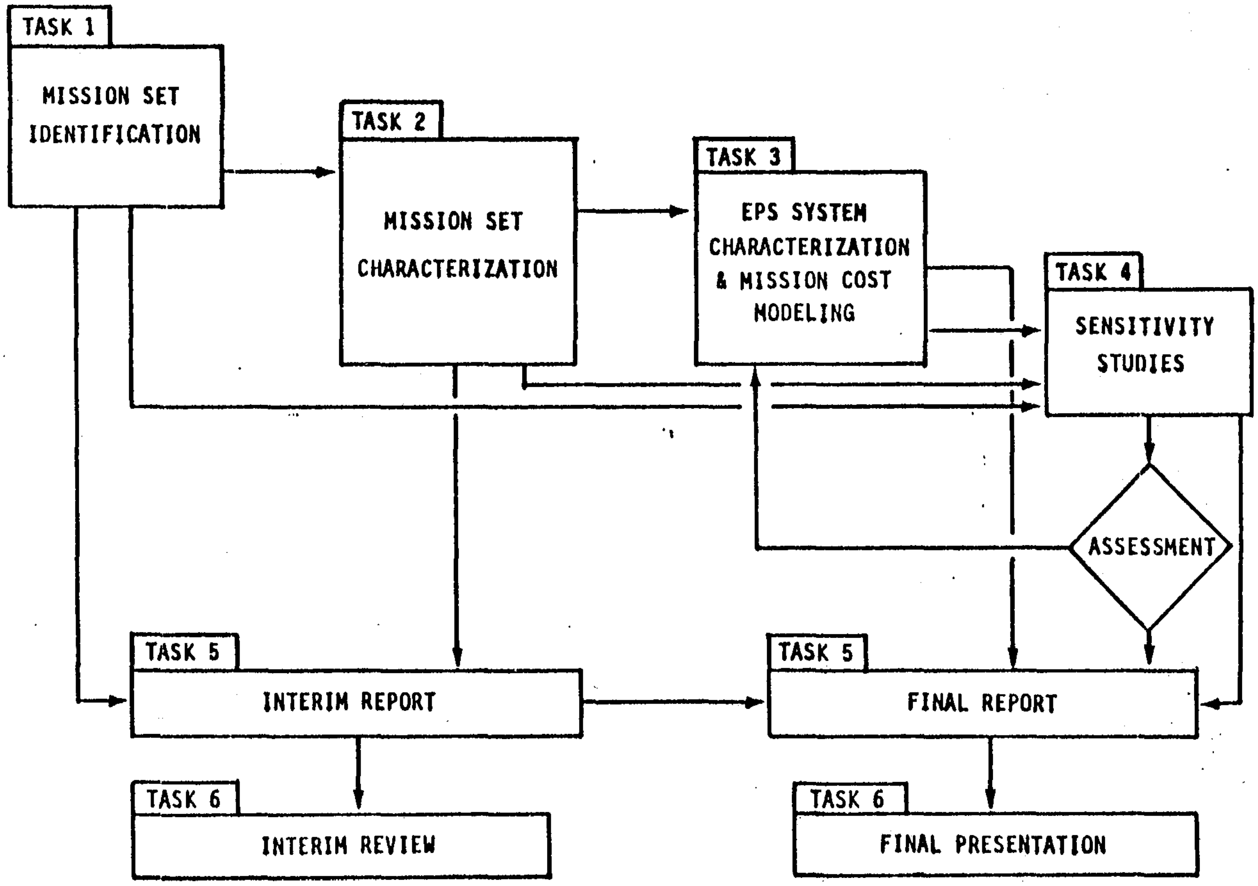

FIGURE 1-3 Reformulated Study Task Flow

2.0 MISSION SET SELECTION

To provide a basis for assessing the efficacy of potential advances in electric propulsion, it is necessary to be cognizant of the applications for this technology. Thus, the first task of the study was to establish a set of earth-orbital missions which could serve as a baseline for the remaining study efforts. The approach to this task was as illustrated in figure 2-1.

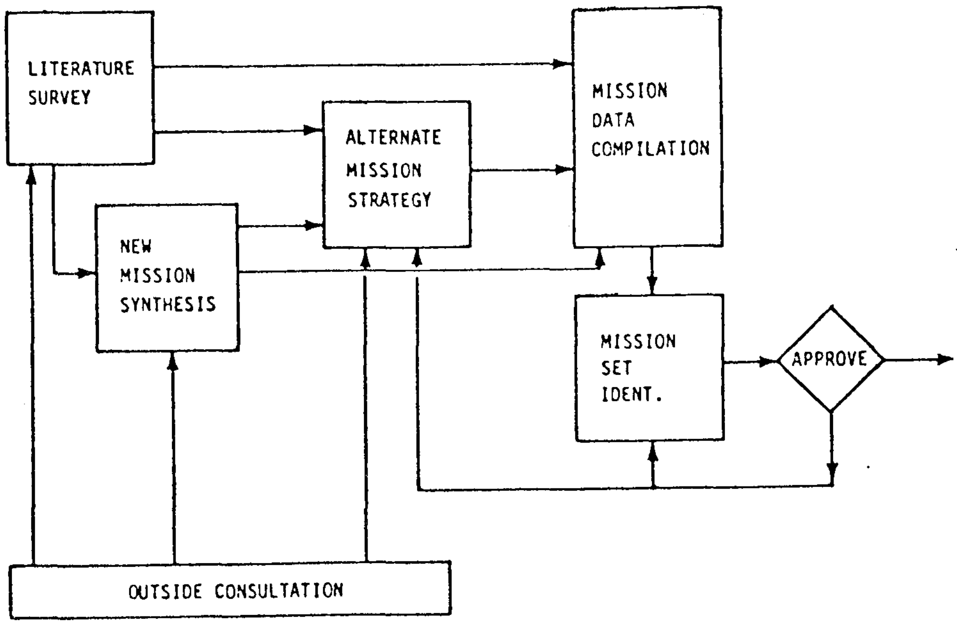

FIGURE 2-1 Study Logic for Task 1

A total of 68 literature sources were reviewed to ensure that this study benefited from existing work in the field. This review was supplemented by in-house brain-storming sessions and contacts with other researchers in the field in an effort to define new mission concepts, and new methods of accomplishing mission objectives. These activities resulted in the identification of 68 potential missions, spanning the next three decades which were felt to be feasible, desirable, and compatible with electric propulsion technology. Of these, 30 were selected to form the basis for succeeding study efforts.

2.1 LITERATURE REVIEW

The contract statement of work required a comprehensive search of available literature to provide a foundation for the study activities. This review served three purposes: (1) to gather data on potential missions previously identified; (2) to aid in estimating the feasibility of any necessary advances in supporting technology areas; and (3) to aid in estimating the potential levels for future space activities. A "minimum" list of sources was given and is reprinted as figure 2-2. In addition, our literature search suggested that the material listed in figure 2-3 had relevance to this study. These were also reviewed. Several sources which were particularly helpful are noted below:

- Reference 2, figure 2-2, provided useful insights into the scaling relationships and modeling techniques for electron bombardment ion thruster systems.

- Reference 3, figure 2-2, provided a comprehensive set of quantitative predictions of the prospects for advancements in the technologies required to implement, and to support, the space programs of the next few decades.

- Reference 1, figure 2-3, provided a description of a potential near-term electric propulsion vehicle, including costs and performance, along with the impacts of adapting earth-orbital payloads for its utilizations.

- Reference 4, figure 2-3, provided descriptions of a great many potentially feasible and desirable missions for the time period of interest.

- Reference 12, figure 2-3, provided additional data on beneficial earth-orbital applications of space and the conditions necessary to make such missions economically viable.

1. Future Space Transportation Systems Analysis Study, Final Report, December 31, 1976, Contract NAS9-14323

2. Advanced Electrostatic Ion Thruster for Space Propulsion - Contract NAS 3-20101, Midterm and Final Reports. Hughes Research Laboratories, Malibu, California

3. Outlook for Space - A Forecast of Space Technology 1980-2000 NASA July 15, 1975

4. Orbital Transportation in the 1980's and Beyond. H. P. Davis Paper # AAS 75-441

5. Requirements and Considerations in Selecting Space Tug Propulsion Systems, C. J. Cohan, AAS Paper # 75-160

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18. Grumman Aerospace Corp., Earth Observatory Satellite System Definition Study, September 1974, NASA CR 143661 through 143675, Contract NAS 5-20520

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20. Stearns, J. W., Status of Nuclear Electric Propulsion, AAS Paper # 75-164

FIGURE 2-2 Literature Sources from SOW

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2. SEP Applications and Systems Design Update, Final Report, Sept. 1977, D180-19783-5, Boeing, Contract NAS8-31444
3. Space Industrialization Study, Midterm Report, March 1977, SIS 76-03-001, Science Applications Inc., Contract NAS8-32197
4. Advanced Space System Concepts and their Orbital Support Needs (1980-2000), Dec. 1976 (7365)-1, Aerospace, Contract NASW-2727
5. Estimates of Potential Propulsion Performance thru 2000 A.D., April 1975, 2-5730-0000-139, Boeing internal
6. Solar Power Satellite, System Definition Study, Midterm Report, Aug. 1977 D180-20748-1, Boeing, Contract NAS9-15196
7. Systems Definition, Space Based Power Conversion Systems, Final Briefing, December 1976, Boeing, Contract NAS8-31628
8. Shuttle Derivative Vehicles Study, First Progress Review, July 1977, Boeing, Contract NAS8-32395
9. Space Construction Facility Study, December 1975, Boeing internal
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21. MOLTOP User's Manual, October 1973, M-240-1224, Northrop Services, Contract NAS8-21810
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24. Argon Ion Propulsion System Characteristics, May 1977, Boeing letter to Hu Davis (NASA JSC)

FIGURE 2-3 Additional Sources Reviewed

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48. Extended Performance Solar Electric Propulsion Thrust System Study, Hughes, Final Report, 5 Volumes, CR 135281, September 1977

FIGURE 2-3 Additional Sources Reviewed

In addition to the sources listed above, several classified documents were reviewed to identify the potential mission needs for primary electric propulsion system from the military arena. Our conclusions from this review were that all missions suggested to-date have requirements that are either near-duplicates of those for some civilian missions, or that are tailored for current or planned launch vehicles. Thus, while specific non-civilian applications were not studied, it is believed that the conclusions reached regarding desirable directions for EP technology advancement are valid over the full spectrum of potential earth-orbital missions.

2.2 SPACE ACTIVITY LEVEL PREDICTION

Initially, three scenarios were postulated to represent the characteristics of man's future development of space. These were chosen to encompass the extremes in levels of support/interest for space industrialization over the next few decades.

In the most pessimistic scenario, there would be only a token pursuit of space. Activities in earth-orbit are viewed primarily as a satisfaction of scientific curiosity, with little impact on the world's socio-economic conditions. Commercialization would be limited to proven fields only (primarily telecommunications), and even in these, some degree of government subsidization would be necessary. Manned activities in space would be confined to the Space Shuttle for most of the period of interest, with the establishment of our first space station being deferred until after the start of the twenty-first century. In this scenario, NASA would be the only developing institution, with no investments made by U.S. industry. Low in the nation's priorities, space missions would face a perpetual uphill battle for funding.

The most optimistic scenario would predict the era of "homo spatium". Our expansion into and utilization of near-Earth space are seen as providing the solutions to mankind's problems. Orbiting space stations would be established as soon as the Space Shuttle becomes operational, and these are followed by major colonization efforts (both orbiting and lunar) before the twentieth century ends. Early in the next century, space industrialization has become an integral part of world economy with some facet affecting the day-to-day activities of almost all individuals. In this scenario, the expansion into space has been taken over by commercial interests. This, interestly enough, leads to a retrenchment of NASA, with its role again being relegated to scientific exploration and technology advancement.

Neither of the above scenarios were judged to be suitable baselines for this study, since they represent extremes in likelihood. A third scenario was formed to cover the middle ground. This scenario was not an attempt to formulate a best guess prediction, but rather was intentionally biased toward the optimistic end of the spectrum. It was felt that this approach would produce a study output that would push technology while retaining a firm association with reality.

This scenario would predict an early recognition of the benefits of orbital activities and their active pursuit thereafter. Early Shuttle/Spacelab experiments would identify many exciting potentials for commercial benefit in space. Vigorous engineering development efforts would quickly convert many of the opportunities into profitable ventures within the next decade. The establishment of low orbit space stations in the mid-1980's would be followed by permanent geosynchronous outposts in the early 1990's. Early in the next century, we would postulate the achievement of more ambitious projects such as a Satellite Power System (SPS) and Lunar Bases (both orbiting and on the surface). In this scenario, it is anticipated that the design, development, and operation of the primary space industrialization efforts would be under commercial auspices. NASA would continue to sponsor fundamental technology advancement and would operate some of the broadly-based, common logistics and support services (launch facilities, tracking, satellite servicing, orbital debris clearance, etc.).

A reference time frame was needed against which mission, and hence technology needs, could be assessed. One measure of development timing is the date of the initial operational capability (IOC) of the major space systems. Figure 2-4 shows a set of potential milestones that was judged to be appropriate to the "middle-ground" scenario discussed above. This time frame provided a basis for the establishment of a detailed "launch schedule" for the overall set of missions to be considered in this study (see figure 2-7).

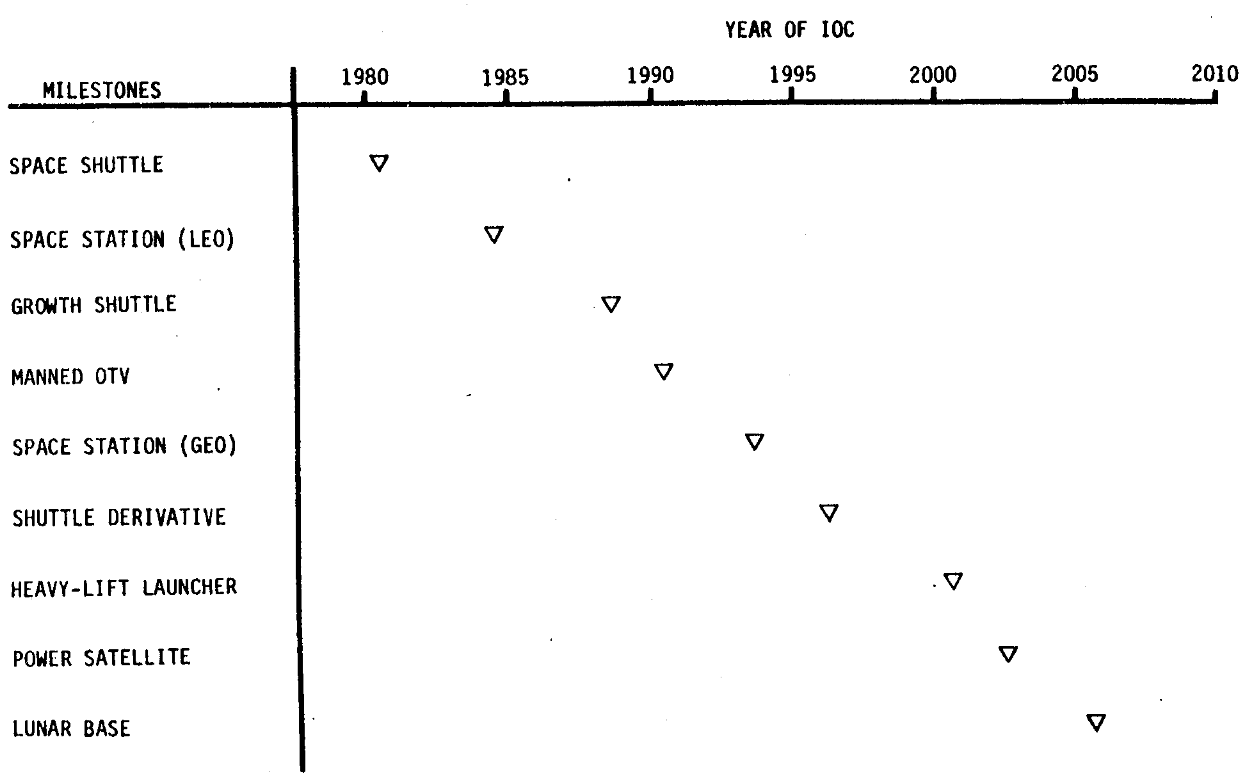

FIGURE 2-4 Schedule of Potential Milestones in the Development of Space

2.3 TECHNOLOGY FORECAST

Several studies have made extrapolations of past and present levels of various technologies to predict the likely or possible future trends. In the current study, the available reports were reviewed in an attempt to arrive at a "concensus" technology forecast. The prognostication for the key technologies required to support a beneficial Earth-orbital space program are given in figure 2-5. These predictions provided a basis for the definition of the mission and payload characteristics (section 3.0).

As is customary in all forecasting activities, certain qualifications must be stated for the clarification of the reader:

- No attempt was made to postulate break-throughs.
- A rather ambitious pursuit of each technology was assumed, without regard to prioritization of funding. This implies that the commitment to a given mission would cause the necessary funds for technology advancement to spring forth.

TECHNOLOGY	YEAR				UNITS
	1980	1990	2000	2010	
Space Telescope Aperture Size	200	340	480	620	cm.
Imaging Angular Resolution	30	10	5	3	μrad.
Space Radar Imaging Resolution	4	2	1	0.5	m.
Earth Imaging Data Return	10^{11}	10^{13}	10^{15}	10^{17}	Bits/Day
Computer (Space) Processing	3	50	400	1000	MOPS
Computer (Earth) Processing	100	10^{3}	10^{4}	10^{5}	MOPS
Data Storage	10^{11}	7×10^{12}	10^{14}	10^{15}	Bits
(S-Band) RF Output Power	800	2000	5000	7500	kw
Communications Data Rate	5×10^{8}	4×10^{9}	2×10^{10}	10^{11}	Bits/Sec
Large Structures	20	100	1000	20,000	m.
Power Levels	3	100	10^{7}	10^{9}	kw
Launch Capacity	30	50	250	500	MT
Leo Launch Costs	700	400	125	50	$/kg.
Men in Space	5	100	10^{4}	10^{6}	-

FIGURE 2-5 Projected Capabilities for Space Mission Supporting Technologies

- The urge to "adjust" the results of older studies that "missed the mark" in predicting present-day capabilities was resisted. This was in recognition of the frequent observation that forecasting activities usually tend to be over optimistic in the near-term but very conservative in the far term.

2.4 MISSION DEFINITION

From our review of the literature and in-house brain-storming activities, many potential near-Earth mission opportunities were identified. Preliminary examinations of each were performed to assure mission feasibility and to determine the alternative modes available for achieving the perceived mission objectives. This served as a pre-screening process and resulted in the tabulation of 68 missions that would support and be supported by the moderately ambitious scenario adopted. The objectives and significant features of each are synopsized as follows:

1-0. Geosynchronous Satellite Maintenance Sortie -- to perform repair, refurbishment, refueling, and equipment update on satellites in geosynchronous orbit (GEO). Sorties originate in low earth orbit (LEO) from the Shuttle, with multiple rendezvous in GEO.

1-1. Geosynchronous-based Satellite Maintenance Sortie -- similar to 1-0, except based at a space station on GEO.

1-2. Geosynchronous-based Satellite Maintenance -- similar to 1-1, except servicing performed at space station rather than at orbital station.

2-0. Geosynchronous Space Station -- to serve as a control center for geosynchronous logistics operations, to conduct scientific and technological experiments, and to monitor Earth resources and condition on a global basis. Assembled from individually transportable modules.

3-0. Orbiting Lunar Station -- similar to 2-0, except in a close (100-300 km) orbit around the moon.

4-0. Nuclear Waste Disposal -- to achieve safe and economical storage of nuclear waste material. Prepackaged material would be brought to LEO in the Shuttle and transported to a very high orbit. Other studies have looked at Earth-escape disposal options, but the high orbit option was chosen to allow EPS recovery and re-use.

5-0. Satellite Power Systems (SPS) -- to continuously and economically produce solar-derived electrical power for general commercial and industrial use on Earth. Assembly and checkout in LEO was contemplated with modular transport to GEO.

6-0. SPS Pilot Plant -- a precursor to 5-0, to demonstrate concept and technology feasibility on a reduced ($\sim 10^3$) scale.

7-0. SPS Engineering Prototype -- a tenth scale system constructed to demonstrate engineering and operational readiness, and commercial viability prior to proceeding with mission 5-0.

8-0. Forest Fire Detection -- to detect forest fires in remote regions, assist in coordination of fire-fighting efforts, and maintain surveillance of hot spots. Sensors at synchronous altitude.

9-0. Nuclear Fuel Location System -- to provide world-wide, real-time, monitoring of the location of nuclear materials/weapons, reducing the chances for nuclear blackmail. Transponders at synchronous altitude.

10-0. Border Surveillance System -- to detect overt/covert attempts at crossing a border, thus reducing levels of illegal aliens and drug trafficking. Relay antenna at GEO.

11-0. Coastal Passive Radar -- to serve as the transmitting portion of marine radar system, thus allowing pleasure craft and other surface vessels to realize the benefits of a precision radar system, with the installation of a rather inexpensive receiver. Phased array on GEO.

11-1. Marine Broadcast Radar -- similar to 11-0, except the entire radar function would be performed on-orbit. Visual images of individual radar scanned areas would be broadcast directly to conventional television receivers to decrease user costs.

12-0. Astronomical Telescope -- to extend man's knowledge of the universe by allowing examination of distant objects with very high resolution. A crossed array of mirrors, station-kept with each other and with a focal plane unit in LEO.

13-0. Atmospheric Temperature Profile Sounder -- to supply data needed for weather prediction and atmospheric modeling. Pulsed laser and detector in an intermediate altitude orbit.

14-0. Global Search & Rescue Locator -- to provide world-wide locating capability for emergency transmitters, thus improving success ratio while reducing costs of search and rescue efforts. Transponders in intermediate altitude orbits.

15-0. Urban/Police Wrist Radio -- to give real-time, secure, anti-jammable, high coverage, wide area communications to each policeman, thus resulting in increased police mobility with improved safety. Phased array transceiver on GEO.

16-0. Disaster Control Satellite -- to provide communications, command, and control to disaster area emergency personnel. Similar to 15-0 with an expanded audience.

17-0. Advanced Resource/Pollution Observatory -- to provide high quality (improvement over the current Land-Sat system), multi-spectral, earth resources and pollution data. Visible, IR, and radar sensors in sun-synchronous orbit.

18-0. Water Level and Fault Movement Indicator -- to aid in the prediction of earthquakes, floods and droughts, and improve the assessment of global water resources. Scanning laser/detector on GEO.

19-0. Ocean Resources and Dynamics System -- to maximize the yield of the world's fish protein resource by locating schools of fish and mapping the ocean's dynamic signature. IR sensors in polar orbit.

20-0. Multinational Air Traffic Control Radar -- to reduce numbers of active radar installations, while centralizing the ATC function improving coverage. Large reflectors in LEO.

21-0. UN Truce Observation Satellite -- to aid UN teams in monitoring truce agreements and weapon system dispositions, while reduce the requirements for on-site personnel. High resolution optical & IR detectors in LEO.

22-0. Synchronous Meteorological Satellite -- to collect world-wide data for global weather prediction. Multi-spectral instruments on GEO.

23-0. High Resolution Earth Mapping Radar -- to provide maps of the earth's surface with high resolution through cloud cover for the assessment of pollution and crops, water and other resources. Synthetic array radar on LEO.

24-0. Interplanetary Television Link -- to provide live reception of color images over planetary ranges in support of complex automated probes and manned settlements. Laser/Detector at GEO.

25-0. Electronic Mail Transmission -- to speed up delivery while decreasing costs of most mail services. Radio relay on GEO.

26-0. Transportation Services Satellite -- to simultaneously satisfy needs for traffic control, route surveillance, navigation, position fixing, etc. Multiple transponders at an intermediate altitude polar orbit.

27-0. Advanced Television Broadcast Satellite -- to make television (services) available to all locations (including mountainous, rural, and remote areas) with conventional, inexpensive, home receivers and antennas. Powerful transmitter in GEO.

28-0. Voting/Polling System -- to provide direct access to the entire U.S. population for voting or polling purposes. Sensitive receiver/repeater on GEO.

29-0. National Information Services -- similar to 27-0, except for a wider range (including non-video) of services.

30-0. Personal Communications Wrist Radio -- to expand two-way telephone service to individuals wherever they might be via lightweight, inexpensive, personal transceivers. Multi-channel repeater with real-time switchboard at GEO.

31-0. Diplomatic/UN Hotline -- to provide rapid, reliable, secure communications between heads of state (and/or embassies), thus reducing the potential for misunderstanding/miscalculations. Transponders on GEO.

32-0. 3-D Holographic Teleconferencing -- to reduce the need for travel to most government or private industry conferences, thus reducing costs and lost time, without a significant loss in the ability to transact business. Similar to 30-0.

33-0. Vehicle/Package Locator -- to locate vehicles or articles in transit, continuously, anywhere in the U.S., thus aiding in the prevention of theft/hijacking, and minimizing errors in shipment. Similar to 9-0.

34-0. Personal Navigation Wrist Set -- to provide accurate relative position location with very inexpensive user equipment. Narrow-beam, phased array, transmitters in GEO.

34-1. Near-Term Navigation Concept -- this is an early, less sophisticated, version of 34-0.

35-0. Aircraft Laser Beam Powering -- to provide an alternative to petroleum as a source of energy for powering commerical air transports. Clusters of steerable mirrors in LEO.

36-0. Night Illuminator -- to provide nighttime lighting without Earth-based energy pollution, unsightly street lights, cables, trenches, etc. Clusters of reflectors in GEO.

37-0. Multi-National Energy Distribution -- to distribute energy to small city users without transmission lines, and to serve many nations simultaneously. Steerable mirrors in LEO.

37-1. Power Relay Satellite -- advanced version of 37-0, more powerful, in GEO.

38-0. Energy Monitor -- to measure energy flow at a very large number of points in the distribution network, allowing near-instantaneous fine tuning of network operation. Transponders on GEO.

38-1. Utility Load Management Satellite -- a more sophisticated version of 38-0, capable of interrogating the home consumer's meter, and commanding industrial substations.

39-0. Vehicular Speed Limit Control -- to reduce traffic accidents and injuries by establishing positive speed control zones. Multi-beam transmitters in GEO.

40-0. Rail Anti-Collision System -- to prevent train collisions, with consequent reduction in losses of lives, property and productivity. Transceiver with correlation computer at synchronous altitude.

41-0. Burglar Alarm/Intrusion Detection -- to safeguard government and industrial buildings, facilities, or homes. Similar to 10-0.

42-0. Space Debris Sweeper -- to remove expended satellites and debris from the synchronous equatorial corridor, where they pose a long-term collision threat to future space activities. Reusable de-boost vehicle.

42-1. Orbital Debris Collector -- alternate means to accomplish 42-0. Mobile capture/disposal module.

43-0. Ozone Layer Replenishment/Protection -- to counteract the environmental damage being done by the release of Freon (and other pollutants) into the Earth's upper atmosphere. Large ion source dispersing binding catalyst in LEO.

44-0. Space Construction Facility -- to provide a facility for the fabrication and construction of large structures in space. Modular space station with jigs, fixtures, and logistics supports in LEO.

45-0. Unmanned Orbital Platform -- to provide a multi-purpose facility, which produces programmatic savings thru the consolidation of engineering functions. Versatile engineering support module in GEO.

46-0. Tethered Satellite -- to conduct upper atmospheric investigations, e.g., pollution surveys, thermal profiles, wind systems, ionospheric fluctuations, etc. Small autonomous satellite lowered approximately 100 km down into sensible atmosphere from LEO.

47-0. Advanced Communications Satellite -- to provide communications services with growth capacity, operational flexibility, and increased economic benefits. Multi-channel transceiver in GEO.

48-0. Gravity Gradient Explorer -- to obtain data on the higher harmonics of the Earth's gravitational field by direct observation of attitude perturbations on a large structure. Long truss (with ACS) movable to a variety of Earth orbits.

49-0. Geosynchronous Communications Platform -- to support the operation of multiple communications systems by providing common subsystems and on-board switching facilities. Structural platform for antennas with engineering services in GEO.

50-0. Earthwatch -- to provide map and assessment capability for resource management (e.g., agriculture, forestry, geology, water shed, land use, etc.) Sensor packages in 6-hr. orbit.

51-0. Orbiting Deep Space Relay Station -- to replace the existing world-wide network of Deep Space Tracking Stations. Large, precisely-pointable, antenna in GEO.

52-0. SPS Orbit Transfer System Recovery -- to reduce SPS transportation costs by returning orbit transfer hardware to LEO for refurbishment and subsequent reuse. Autonomous propulsion vehicle.

53-0. Solar Wind Sampler -- to examine the solar wind in its pristine state via an "upstream" monitoring platform. Sensor package in near-Earth helio-centric orbit.

54-0. Earth's Magnetic Tail Mapper -- to establish/monitor the characteristics of the Earth's magnetic tail. Similar in payload/orbit to 53-0.

55-0. Iceberg Dissipator -- to reduce danger for world-wide shipping by speeding the meltdown of icebergs. Mirrors in intermediate altitude orbit.

56-0. Soil Surface Texturometer -- to assist in the classification of ground materials by measurement of particle sizes, periodically, and material content. Laser scatterometer in LEO.

57-0. Tornado Tracker -- to reduce the loss of lives and property by prediction/warning of the ground tracks and touchdown points of cyclonic disturbances. Multi-spectral/RF sensors in intermediate altitude orbit.

58-0. Technology Development Platform -- to provide a versatile, long-term, test-bed facility in the geosynchronous environment. Engineering support services platform (modular, building-block approach) in GEO.

59-0. Detached Experiment Modules -- to provide an experiment platform that realizes the benefits of colocation with a manned space station, while eliminating deleterious cross-coupling interactions. Engineering/propulsion support services module near GEO.

60-0. Space Based Radar System - Near Term -- to provide a long-range, unjammable, radar surveillance capability. Large antenna, orbiting at intermediate altitude.

61-0. Space Based Radar System - Far Term -- an advanced version of 60-0, at GEO.

2.5 BASELINE SET SELECTION

A subset of the overall catalog of missions was selected for more detailed study in the later tasks. The objective of the selection process was to ensure that the baseline mission set adequately represented the range of potentialities for the next three decades. To this end, each of the candidate missions was characterized by objective, payload type, and the physical parameters of interest (i.e., orbit, and payload mass, size, power, etc.). The selection process was somewhat arbitrary in that different investigators could well arrive at a different set which would meet the study goals as well.

Examination of the total completion of missions revealed nine differentiable mission objectives, or themes (where a mission accomplished several purposes, only the primary objective was considered). These themes are listed below, along with the catalog numbers of the missions belonging to each group. The order of the list signifies whether the need is currently being satisfied by satellites (top), or if its fulfillment is merely postulated (bottom).

- SCIENTIFIC RESEARCH 12, 48, 53, 54
- INFORMATION TRANSFER 15, 24, 25, 27, 29, 30, 32, 47, 49
- ENVIRONMENTAL PREDICTION/PROTECTION . 4, 13, 17, 18, 22, 43, 46
- EARTH RESOURCES 8, 19, 23, 50, 56, 57
- LAW, ORDER & DIPLOMACY 9, 10, 21, 31, 33, 39, 60, 61
- PUBLIC SERVICE 11, 14, 16, 20, 26, 28, 34, 40, 41, 55
- TECHNOLOGY DEVELOPMENT 6, 7, 58
- SPACE LOGISTICS SERVICES 1, 2, 3, 42, 44, 45, 51, 52, 59
- ENERGY/MATERIAL PRODUCTION 5, 35, 36, 37, 38

The payloads necessary to satisfy the preceeding objectives were broadly classified into nine generic types. (There is not a one-to-one correspondence between mission objective and payload type.) The generic payload types are listed in the following page. Again, those at the top of the list represent those types which have already been realized, while those at the bottom are more far-term.

- PASSIVE REFLECTORS 20, 35, 36, 37, 55
- COMMUNICATIONS RELAYS 14, 15, 16, 24, 25, 26, 27, 29, 30, 31, 32, 33, 40, 41, 47, 49, 51
- R&D PACKAGES 6, 7, 46, 48, 53, 54, 58
- SPACE STATIONS 2, 3
- OPTICAL/IR TELESCOPES 8, 12, 17, 19, 21, 22, 50, 57
- RECEIVING ANTENNAS 9, 10
- SHAPED BEAM GENERATORS 11, 13, 18, 23, 28, 34, 38, 39, 56, 60, 61
- LOGISTICS PACKAGE 1, 42, 43, 44, 45, 52, 59
- ENERGY SOURCE 4, 5

In addition, estimates of the physical attribute of the candidate payloads were gleaned from the literature, whenever available. (These initial estimates were updated in task 2, and so will not be reported here.) The selection process then included a calculated effort to ensure that the baseline mission set would be representative of the spectrum of possibilities in terms of mass, dimensions, power, costs, and complexity.

The set finally selected was comprised of 30 missions from the original field of 68. These thirty are tabulated in figure 2-6. The selected set includes representatives of all 9 mission types, and of all 9 payload types. Over 75 percent of the selected missions are being, or have been, actively studied by various industry or government agencies. This is desirable and was considered in the selection process, because it tends to increase the amount of supporting data, advice, and counsel available and it also assures an audience that will be interested in the study results. Certain (approximately 25%) somewhat "far-out", or "just barely possible" missions were also deliberately included in the baseline set. This was done for three reasons: (1) it broadened the range of mission/system requirements; (2) it would tend to ensure the requirements for the development of advanced technology; and (3) historically, man's predictions of the future tend to be conservative.

One further datum required for tasks 3 and 4 was seen to be an estimate of the system readiness date for each mission. Therefore, launch schedules were postulated for each of the three potential levels of space activities

FIGURE 2-6 Baseline Mission Set

No.	Title
1-1	Geosynchronous-Based Satellite Maintenance Sortie
2-0	Geosynchronous Space Station
3-0	Orbiting Lunar Station
4-0	Nuclear Waste Disposal
5-0	Satellite Power Systems
6-0	SPS Pilot Plant
9-0	Nuclear Fuel Location System
11-1	Marine Broadcast Radar
12-0	Astronomical Telescope
14-0	Global Search and Rescue Locator
20-0	Multinational Air Traffic Control Radar
25-0	Electronic Mail Transmission
30-0	Personal Communications/Wrist Radio
34-0	Personal Navigation/Wrist Set
34-1	Near-Term Navigation Concept
37-1	Power Relay Satellite
38-1	Utility Load Management Satellite
44-0	Space Construction Facility
46-0	Tethered Satellite (Atmospheric Explorer)
48-0	Gravity Gradient Explorer
49-0	Geosynchronous Communications Platform
50-0	Earthwatch (Resources Mapper)
51-0	Orbiting Deep Space Relay Station
52-0	SPS Orbit Transfer System Recovery
54-0	Magnetic Tail Mapping
55-0	Iceberg Dissipator
56-0	Soil Surface Texturometer
58-0	Technology Development Platform
60-0	Space Based Radar - Near Term
61-0	Space Based Radar - Far Term

as characterized in section 2.2. The traffic projection for the nominal scenario is shown in figure 2-7. Here, the ∇ indicates when a satellite is launched or when payload becomes operational (for those cases where multiple launches are required to assemble a modular payload on-orbit). Variations on this symbol are explained below.

This schedule is probably unrealistic in that it was constructed to specifically and individually include all missions of the baseline set. It may well be that certain missions will either exclude or be combined with others (e.g. some of the communications - oriented missions may well make up a portion of a large geosynchronous communications platform). Nevertheless, this schedule represents a point-of-departure in terms of traffic levels and timing, based upon a moderately active growth in funding for space-related activities.

∇^N	-	N Payloads launched or operational in the stated year
∇^M	-	Maintenance visit
∇^U	-	System update (capability expansion) visit
$\nabla \rightarrow$	-	Start of a launch requirement that continues year after year
∇^H	-	Household-level control/monitoring capability
∇^S	-	Substation-level control/monitoring capability

FIGURE 2-7 Nominal Scenario Launch Schedule

MISSION		79 (19XX)	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	96	97	98	99	00 (20XX)	01	02	03	04	05	06	07	08	09	10
			▽STS				▽LEO MSS				▽GROWTH SHUTTLE		▽MOTV			▽GSO MSS			▽SHUTTLE DERIVATIVE				▽HLLV		▽SPS			▽LSB					
3	ORBITING LUNAR STATION																		▽														
5	SATELLITE POWER SYSTEMS																								▽		▽	▽	▽2	▽2	▽2	▽2	▽2
34-1	NEAR-TERM NAVIGATION CONCEPT									▽			▽U																				
38-1	UTILITY LOAD MGMT SATELLITE								▽S		▽S		▽H																				
46	TETHERED SATELLITE					▽		▽		▽		▽		▽		▽		▽		▽		▽		▽		▽							
49	GSO COMMUNICATIONS PLATFORM													▽	▽	▽	▽	▽	▽U →														
50	EARTHWATCH								▽2	▽2	▽2	▽2	▽2	▽2	▽2	▽2	▽2	▽2															
51	ORBITING DEEP SPACE RELAY STA.																	▽			▽												
52	SPS ORBIT TRANSFER RECOVERY																										▽ →						
54	EARTH'S MAGNETIC TAIL MAPPER								▽			▽			▽			▽			▽			▽			▽			▽			▽
55	ICEBERG DISSIPATOR																			▽		▽	▽2	▽2	▽2	▽2	▽4	▽4	▽3	▽2	▽2	▽M →	
58	TECHNOLOGY DEVELOPMENT PLATFORM										▽		▽U		▽U		▽U		▽U														
1-2	GEOSYNCHRONOUS-BASED SAT. MAINTENANCE																▽ →																
2	GEOSYNCHRONOUS SPACE STATION															▽						▽						▽					
4	NUCLEAR WASTE DISPOSAL							▽ →																									
6	SPS PILOT PLANT																			▽													
9	NUCLEAR FUEL LOCATION SYSTEM												▽2	▽2	▽4	▽4	▽2 →																
11-1	MARINE BROADCAST RADAR																	▽	▽	▽	▽												
12	ASTRONOMICAL TELESCOPE											▽				▽				▽				▽				▽A				▽A	
14	GLOBAL SEARCH & RESCUE LOCATOR													▽	▽2	▽3	▽3	▽	▽3	▽3	▽3	▽2 →											
20	MULTI-NATIONAL AIR TRAFFIC CONTROL RADAR							▽2	▽4	▽6	▽18	▽25	▽25	▽25	▽25																		
25	ELECTRONIC MAIL TRANSMISSION						▽							▽							▽							▽					
30	PERSONAL COMMUNICATIONS WRIST RADIO												▽				▽																
34-0	PERSONAL NAVIGATION WRIST SET															▽																	
37-1	POWER RELAY SATELLITE														▽	▽3	▽7	▽15	▽25	▽25	▽25	▽25	▽15	▽10									
44	SPACE CONSTRUCTION FACILITY								▽								▽					▽					▽					▽	
48	GRAVITY GRADIENT EXPLORER							▽				▽																					
56	SOIL SURFACE TEXTUROMETER										▽																						
60	SPACE BASED RADAR SYSTEM – NEAR TERM									▽	▽	▽	▽																				
61	SPACE BASED RADAR SYSTEM – FAR TERM														▽	▽	▽	▽	▽		▽		▽		▽		▽		▽		▽		▽

3.0 MISSION SET IMPACTS

For each of the near-Earth missions selected at the end of task 1, engineering analyses and further library researches were performed to: (1) identify the potentially fruitful areas for electric propulsion system (EPS) technology advancement; and (2) provide an adequate data base for the modeling and analysis activities of tasks 3 and 4. As shown in figure 3-1,

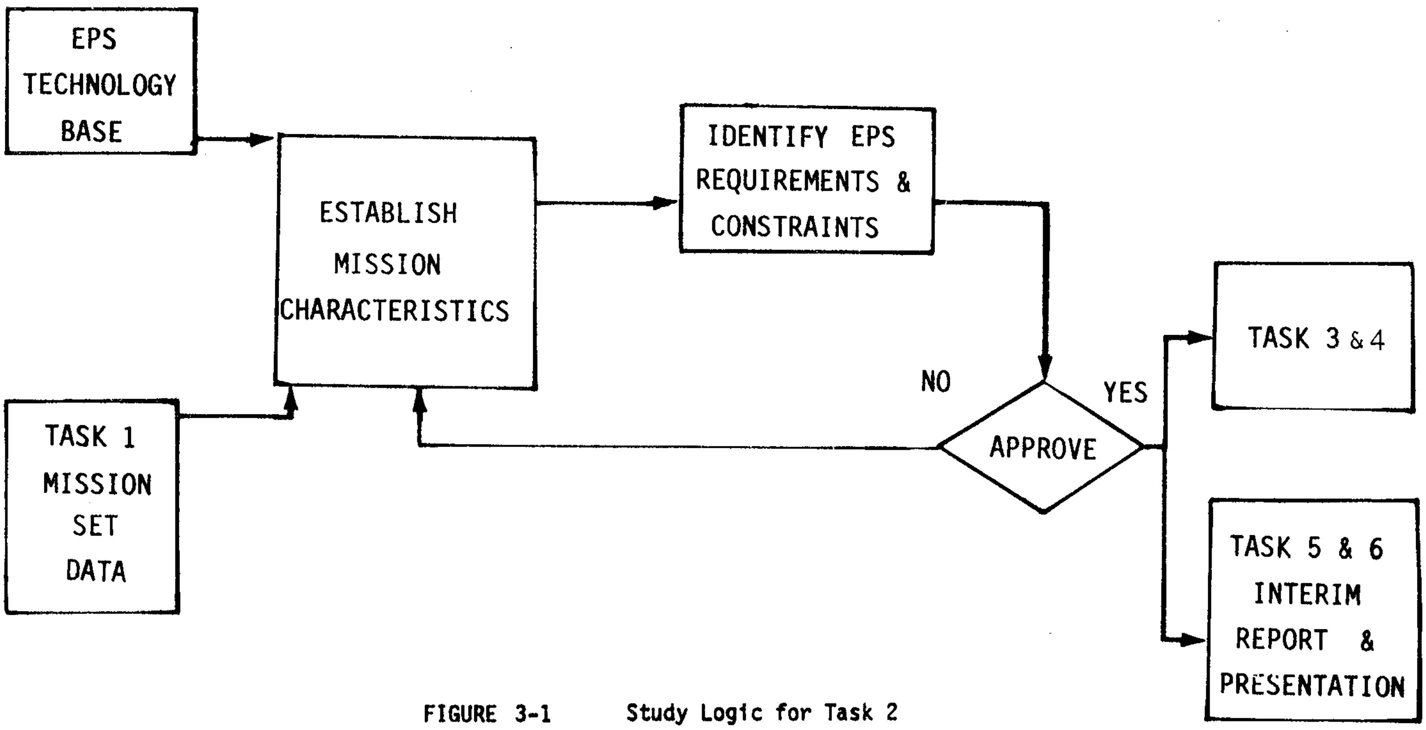

FIGURE 3-1 Study Logic for Task 2

the approach was to first determine the mission and payload characteristics that impact the choice of an EPS, and then to derive the values for these parameters for each baseline mission. In general, these efforts fell into two areas, a determination of a probable set, of physical and functional characteristics for each payload, and an evaluation of the trajectory requirements for each type of mission. The discussion below is structured accordingly.

3.1 PAYLOAD DEFINITION

Traditionally, propulsion system designers are most concerned with the mass of payload spacecraft. However, as the STS-era matures, evolving larger and more sophisticated payloads, other physical characteristics will become equally important. This is particularly true for the case where electric propulsion systems are to be used. The EPS applicability and design are profoundly influenced by the physical size and shape of the payload, its density, modularity, and the size and type of power supply onboard. In addition, its functional mode during launch and inter-orbital transport (stowed, deployed, dormant, operational, etc.) will determine the nature of the design/cost penalties that will accrue due to the EPS characteristically long transfer times.

Much of data on physical characteristics (mass, power and size) of the various payloads was available from the literature. Experience on previous studies (e.g., reference 1 of table 2-3) provided a basis for estimating the impacts of non-trivial transport times, and the mass/cost penalties associated with adapting the payload to the EPS. Where the available descriptions were either unavailable or incomplete, conceptual designs were formulated for payloads which would meet the mission objectives. An example is shown in figures 3-2 and 3-3 for the Soil Surface Texturometer mission (catalog number 56-0). Such designs were completed only to the degree necessary to estimate physical characteristics, to develop assembly and transportation concepts, and to visualize potential mission scenarios.

For each mission, the pertinent information was collected on a "Mission Data Sheet". For the selected mission set, these data sheets are reproduced as an appendix to this report. Some of the more significant mission/system parameters are summarized in figure 3-4.

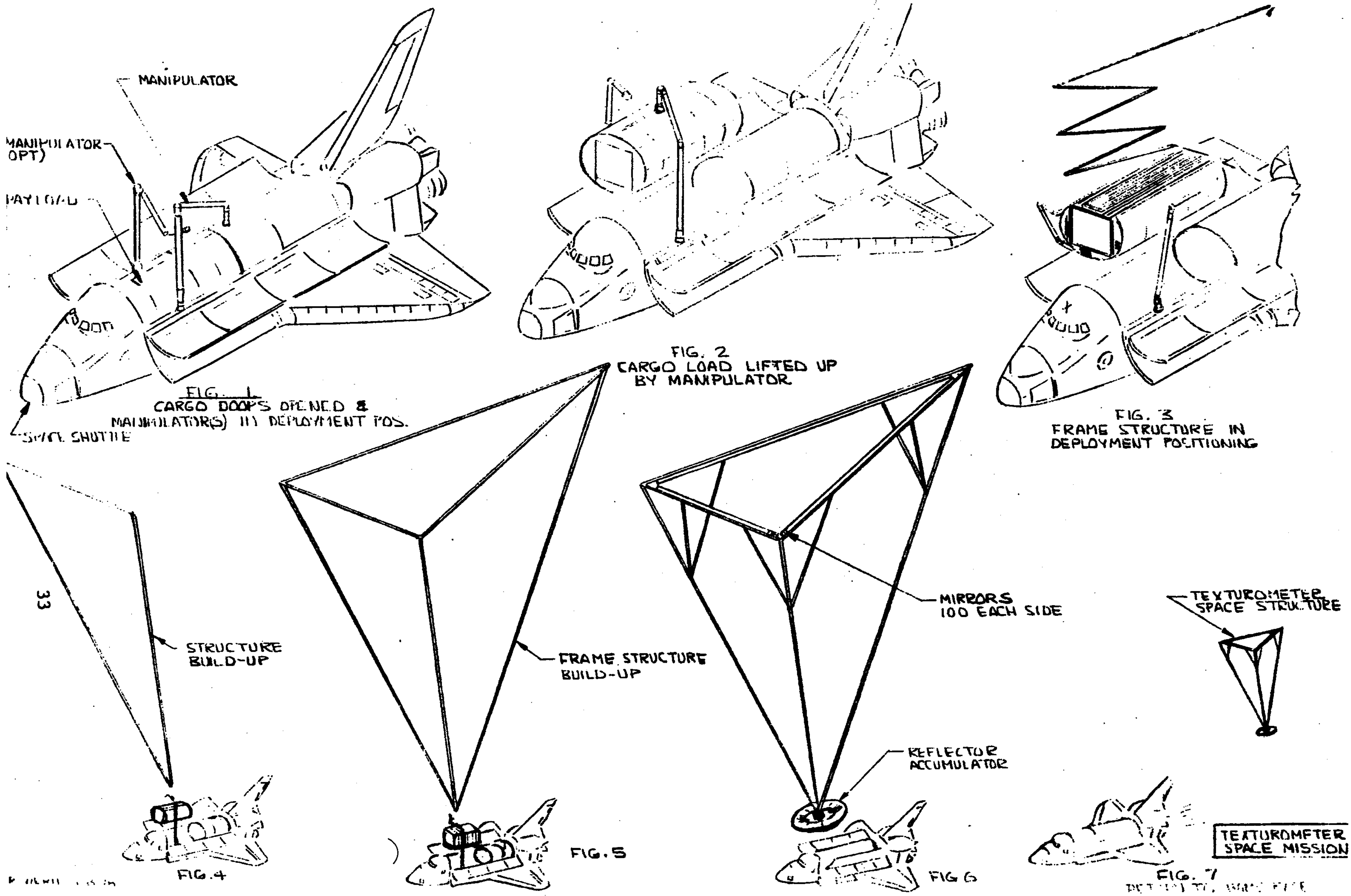

FIGURE 3-2 Assembly Sequence - Soil Surface Texturometer

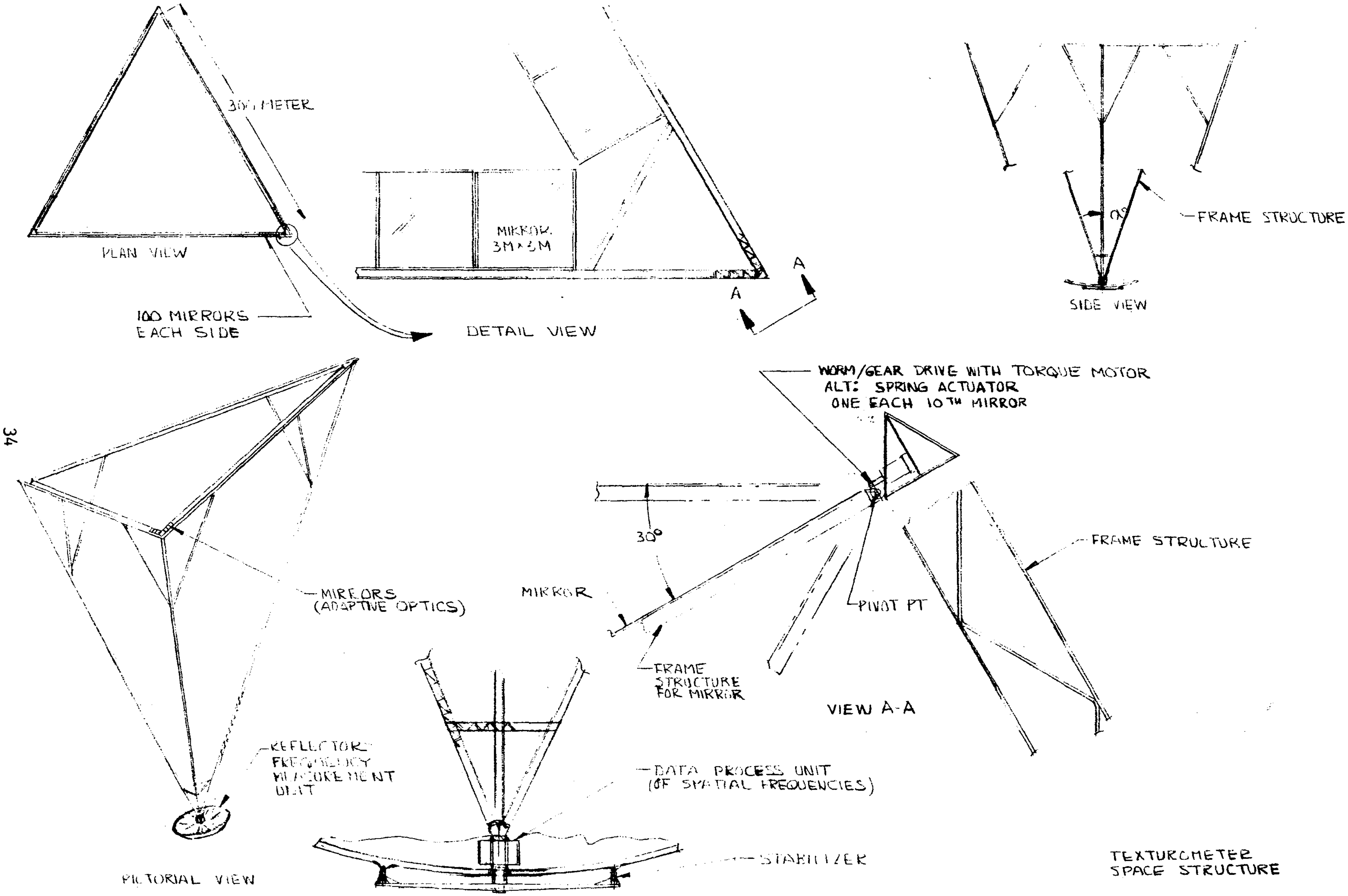

FIGURE 3-3 Construction Details - Soil Surface Texturometer

FIGURE 3-4 Mission Set Characteristics

	MISSION	ORBITAL RADIUS (10^3 KM)	ORBITAL INCLINATION (DEG)	ORBITAL ECCENTRICITY	IOC	PAYLOAD MASS (MT)	PAYLOAD POWER (kW)	MAXIMUM PAYLOAD DIMENSION (m)	PAYLOAD VOLUME (m^3)	PAYLOAD DENSITY (KG/m^3)	NUMBER OF PAYLOADS	PAYLOAD VALUE $M (Avg)	TRAFFIC
46	TETHERED SATELLITE	6.7	~ 28.5	0	1983	0.7	0	10^5	102	7	11	?	2-YR INTERVALS
25	ELECTRONIC MAIL TRANSMISSION	42.2	0	0	1984	9.1	15	61	2500	3.6	4	430	7 YR INTERVALS
4	NUCLEAR WASTE DISPOSAL	750	~ 0	~ 0	1985	3.25	50-75	3	10.5	310	100-1300	0	4/YR-1/WK.
48	GRAVITY GRADIENT EXPLORER	~ 10	~ 28.5	0	1985	5	0.5	3100	112,000	0.04	2	?	2 @ 4-YR INTERVALS
20	MULTI-NATIONAL AIR TRAFFIC CONTROL RADAR	7.0	35-50	0	1985	1.7	1	75	8500	0.2	150	2	2,4,6,18,25*4
38-1	UTILITY LOAD MGMT SATELLITE	42.2	0	0	1986	3.2	7	10	240	13	2	50	2 @ 2-YR INTERVALS
54	EARTH'S MAGNETIC TAIL MAPPER	3000	0	> 1	1986	0.375	0.2	3.5	1.7	220	9	?	3 YR. INTERVALS
50	EARTHWATCH	12.8	50	0	1986	6.5	2.5	15	550	12	20	?	2/YR. FOR 10 YRS
44	SPACE CONSTRUCTION FACILITY	6.9	35	0	1986	2500	> 100	750	3×10^6	0.8	1	?	1 ONLY
60	SPACE BASED RADAR SYSTEM – NEAR TERM	16.7	~ 90	0	1987	4	30	90	87,700	0.05	4	75	4 @ 1-YR.INTERVALS
34-1	NEAR-TERM NAVIGATION CONCEPT	42.2	0	0	1987	0.725	1	49	25	29	1	90	1 ONLY
56	SOIL SURFACE TEXTUROMETER	7.0	~ 50	0	1988	2.31	0.4	600	7.5×10^6	0.0003	1	?	1 ONLY
58	TECHNOLOGY DEVELOPMENT PLATFORM	42.2	0	0	1988	3.09	160	51	7000	0.4	1	40	1 ONLY
12	ASTRONOMICAL TELESCOPE	7.0	0	0	1989	0.8/MIRROR 1.3-2.8/FOCAL PL.	0.5/MIRROR 4.5/FOCAL PLANE	4/MIRROR 5.4/FOCAL PLANE	38/MIRROR 68/FOCAL PLANE	21/MIRROR 19-41/FOCAL PLANE	21 MIRRORS + 1 FOCAL PLANE/SYS	175	4 SYS.@ 4-YR INTERVALS
9	NUCLEAR FUEL LOCATION SYSTEM	42.2	50	0	1990	1.36	0.3	12.8	90	15	46	11	2/YR + 2 ADDITIONAL IN 1992, 1993
30	PERSONAL COMMUNICATIONS WRIST RADIO	42.2	0	0	1990	14	21	61	4600	3	2	300	2 @ 4-YR-INTERVALS
49	GSO COMMUNICATIONS PLATFORM	42.2	0	0	1991	8.2	20	430	0.6×10^6	0.1	5	~ 500 TOT.	1/YR
14	GLOBAL SEARCH & RESCUE LOCATOR	26.6	50	0	1991	0.91	1	6.1	14	66.3	20	20	2-1/4/YR.(EQUIV)
37-1	POWER RELAY SATELLITE	42.2	0	0	1992	27.5	~ 0	1100	3.6×10^6	0.008	145	36	1 + 3 (T-1992)
61	SPACE BASED RADAR SYSTEM – FAR TERM	42.2	0	0	1992	7	50	270	2.3×10^6	0.003	5	100	5 @ 1-YR INTERVALS THEN 1/2 YRS.
34-0	PERSONAL NAVIGATION WRIST SET	42.2	0	0	1993	13.6	2	1700	17,000	0.8	1	100	1 ONLY
2	GEOSYNCHRONOUS SPACE STATION	42.2	0	0	1993	16.5 EA.FOR 9	1 @ 75 8 @ 0	35.4	563 EA.	29	3 @ 9 MOD.EA.	?	6 YR. INTERVALS
1-2	GEOSYNCHRONOUS-BASED SAT. MAINTENANCE	42.2	≤ 50	~ 0	1994	1.031	0	8	36	29	< 5 SERVICERS	?	2 + 1/2 (T-1994)/YR
51	ORBITING DEEP SPACE RELAY STA.	42.2	≤ 11	0	1995	7.5	.75	100	20,000	0.34	2	?	2 @ 3-YR INTERVALS
11-1	MARINE BROADCAST RADAR	42.2	0	0	1995	6.7	25	500	6000	1.1	4	?	1/YR FOR 4 YRS.
3	ORBITING LUNAR STATION	384.4	18-28	> 1	1996	22.1	1 @ 150 9 @ 0	12.8	186 EA.	120	1 @ 10 MOD.EA.	?	1 ONLY
55	ICEBERG DISSIPATOR	9.1	60	0	1997	1750	~ 0	6000	10^8	0.01	25	?	AVG. 2/YR.
6	SPS PILOT PLANT	42.2	0	0	1997	340	15,000	373	0.7×10^6	0.5	1	-	1 ONLY
5	SATELLITE POWER SYSTEMS	42.2	0	0	2002	12,500 EA.	$< .2 \times 10^6$ EA.	2675	7×10^9	0.002	8 MOD./SPS $\times 13$	5000 EA	1 SPS/YR. (EQUIV.)
52	SPS ORBIT TRANSFER RECOVERY	42.2	0	0	2004	275	0	57	2750	100	4 OTS/MOD. 10 MOD./SPS $\times 11$	45	1 SPS/YR.

3.2 TRAJECTORY CHARACTERIZATION

The selected set can be grouped according to the type of trajectory (basically their destination orbit) each pursues. The categories used in this study are shown below. The remainder of the task 2 analyses will be discussed according to this grouping. For each trajectory type, the missions belonging to that group will be summarized, followed by the characteristics of that type of trajectory. Potential areas for technology development will be provided, as appropriate.

- *LEO to GEO*
- *Low Earth*
- *GEO to LEO*
- *LEO to Intermediate*
- *Elliptical to GEO*
- *Beyond GEO*

3.2.1 Shuttle Orbit to Geosynchronous

This class encompasses the majority of the selected missions, and indeed of the near-Earth missions foreseen by all studies. This is because synchronous orbit provides such a desirable platform from which to view the Earth. While some previous studies have investigated ascent modes involving a two stage propulsion system (chemical propulsion to transfer orbit and electric propulsion from there to GEO), this study concentrated on a direct EPS transfer. This mode will become more and more desirable as future space systems become larger and it becomes necessary to reduce the orbit transfer system acceleration levels to avoid costly mass penalties (space optimized designs). A single, direct, ascent eliminates cumbersome handover operations, and of course, reduces development costs to a minimum. Additionally, the assembly phase can be carried out, with manned assistance and a complete operational checkout, in low Earth orbit, thus enhancing the probability of mission success.

The following missions were considered to be members of this group:

- *Geosynchronous-Based Satellite Maintenance*
- *Geosynchronous Manned Space Station*
- *Power Satellite*
- *SPS Pilot Plant*
- *Nuclear Fuel Location System*
- *Marine Broadcast Radar*
- *Electronic Mail Transmission*
- *Personal Communications Wrist Radio*
- *Personal Navigation Wrist Set*
- *Near Term Navigation Concept*
- *Power Relay Satellite*
- *Utility Load Management Satellite*
- *Gravity Gradient Explorer*
- *GSO Communications Platform*
- *Orbiting Deep Space Relay Station*
- *Technology Development Platform*

The altitude and inclination time histories of a transfer from a 300 km/ 28.5° (Space Shuttle handover) orbit to a geostationary orbit are shown in figure 3-5. The effects of shadowing and solar cell radiation damage are shown explicitly. The curve labeled real also accounts for such things as Earth oblateness, seasonal variations, and steering penalties. The dif-

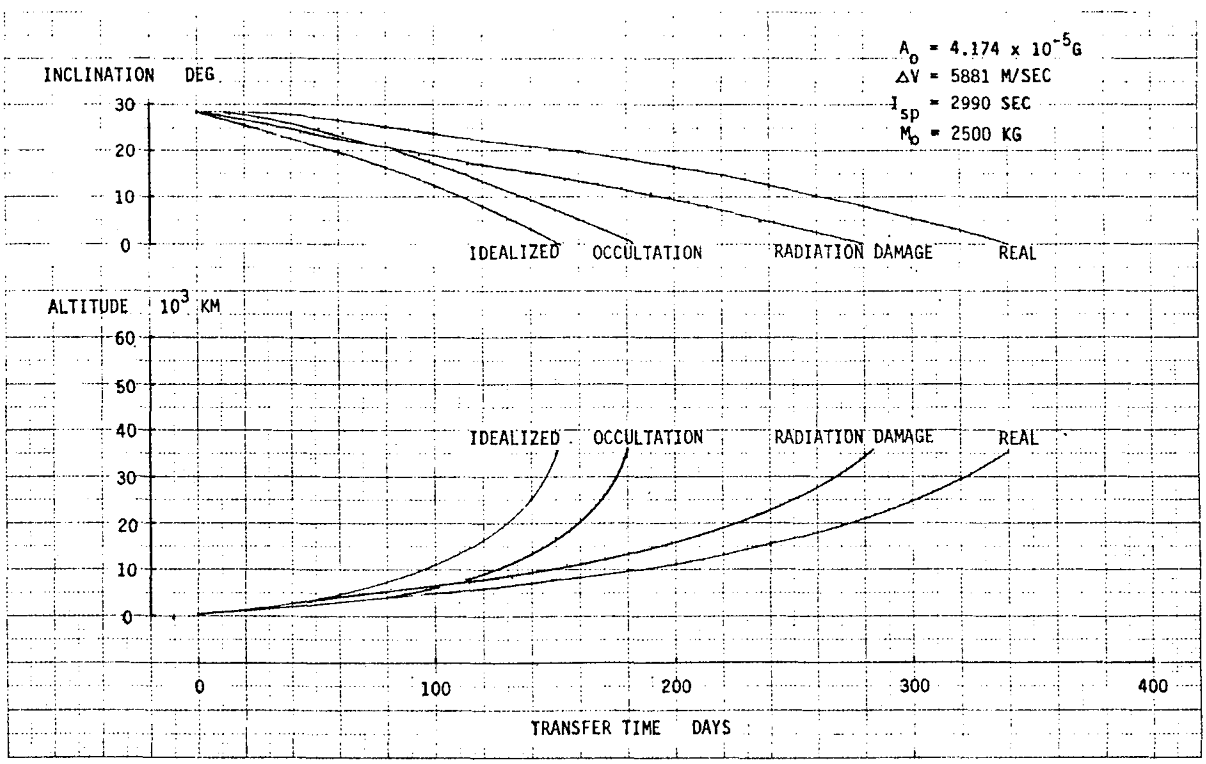

FIGURE 3-5 LEO→GEO Trajectory Time History

ferences between different pairs of curves allows a calculation of several penalty factors which were then used in the system level cost model (see section 4.2). The graph shown is for a current state-of-the-art electric propulsion system; other cases were run, but their inclusion did not affect the values of the penalty factors.

EFFECT OF LARGER PAYLOAD -- Depending on the overall mission economics, it appeared probable that some of the larger payloads would require transfer times several times longer than had previously been studied. Therefore, a small study was made to determine whether this increase in transfer time (lower acceleration levels) would significantly affect the total energy (ΔV) requirements. For a given EPS technology, initial acceleration is set by the system mass and the available electric power, as shown in figure 3-6. Figure 3-7 illustrates the relationship between this factor and the transfer energy requirements, for systems representing near-term technology (accelerations on the order of 10^{-5} g's and solar arrays that suffer over 50% degradation due to trapped particle bombardment), on a typical LEO to GEO mission. The energy requirements only increase by a few percent over

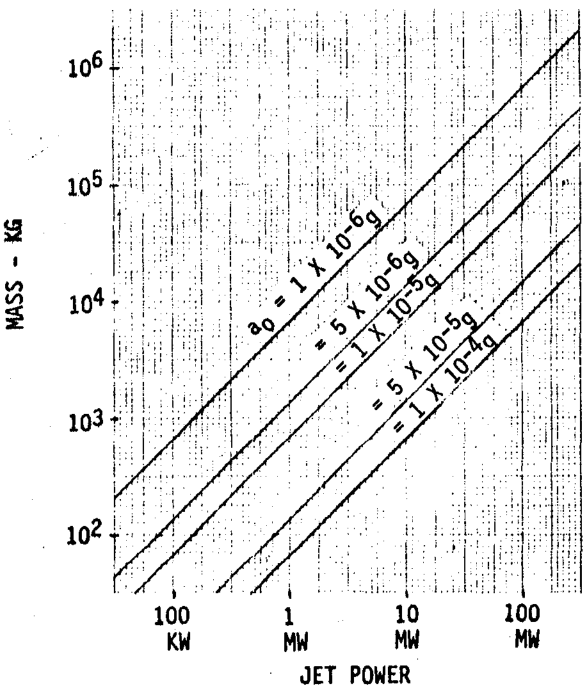

FIGURE 3-6 EPS Acceleration Characteristics

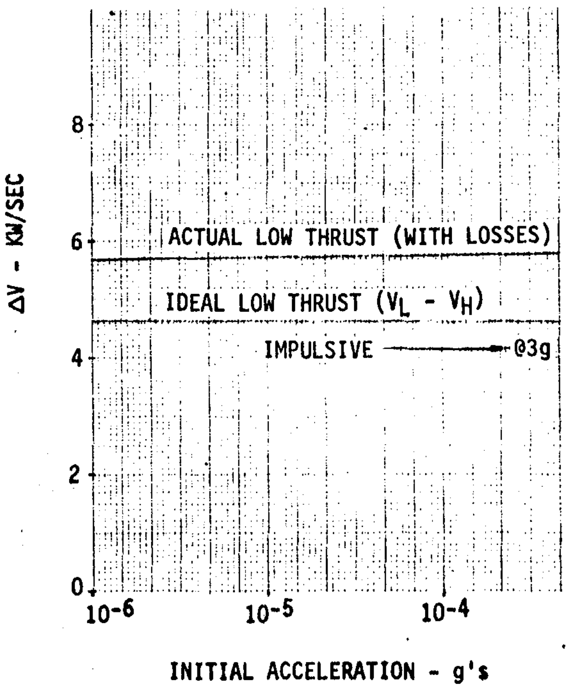

FIGURE 3-7 Transfer Energy Requirements (LEO→GEO)

the acceleration range of interest (corresponding to mission durations of a few months to a few years). This increase can be thought of as analogous to the "gravity loss" factor which must be included in analyses of high thrust transfers when finite burn times are considered.

RADIATION EFFECTS -- The lower curve of figure 3-8 shows the flux levels felt through a 3 mil cover glass during a typical one year low thrust transfer. This flux model results in an integrated fluence shown by the upper curve of the same figure. Figure 3-9 shows the effects on the power output of a state-of-the-art solar array (see section 4.3 for further description). Final power output is only about forty percent of the installed array capacity. (This is a major difference between near-Earth and planetary mission design). There would seem to be a two avenues of approach to electric propulsion system design considering the effects of the near-Earth radiation environment: design accommodations and technology improvement.

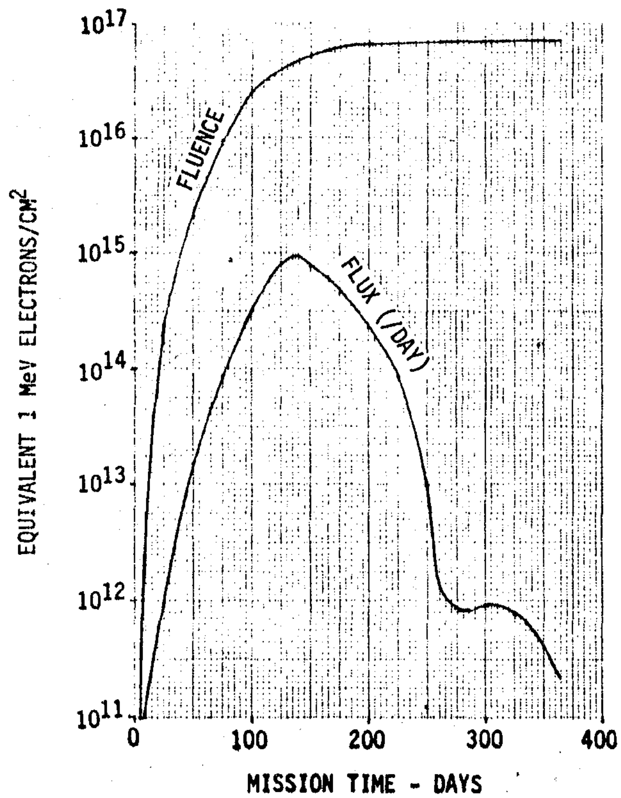

FIGURE 3-8 Radiation Environment during LEO→GEO Transfer

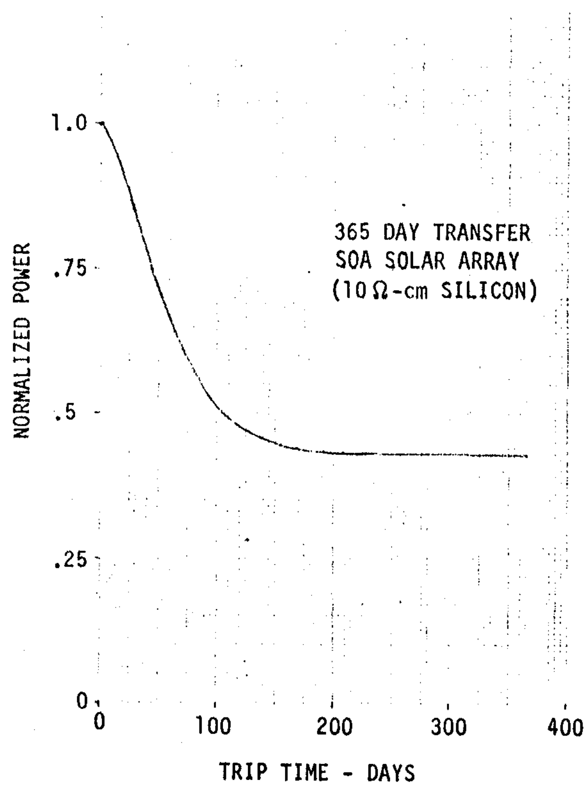

FIGURE 3-9 Degraded Solar Array Output

Design accommodations would include adding more solar array, increasing the shielding (both front and back sides) of the solar cells, and sizing the EPS for the power output expected either at the end of the mission (see section 5.5.3) or some other, intermediate, point. These solutions then would attempt to make the best of the degraded performance capabilities.

On the other hand, technology improvement would be aimed at improving the performance of the system. Possibilities include "over-powering" the EPS early in the mission, employing more radiation resistant solar cells (gallium-aluminum-arsenide (GaAlAs) or doped-silicon), and the in-flight annealing of the solar array. The recently-studied (references 1, 36, 43 and 48 of figure 2-3) concept of concentrating solar arrays combines elements of both the second and third potential improvement options. Here, reflecting surfaces are employed to produce higher than normal concen trations of sunlight on the photovoltaic surfaces. It has been reported that the use of GaAlAs as the conversion material may allow an almost continuous self-annealing process to take place at moderate operating temperatures.

Several possibilities have been suggested for annealing out the damage centers in an irradiated photovoltaic array, including bulk thermal processes and the use of beams of charged particles. Another promising technique involves the use of a laser to produce localized hot spots, and thus to anneal a degraded array incrementally. Figure 3-10 shows a concept considered in a recent study of solar power satellites (reference 6 of figure 2-3). A

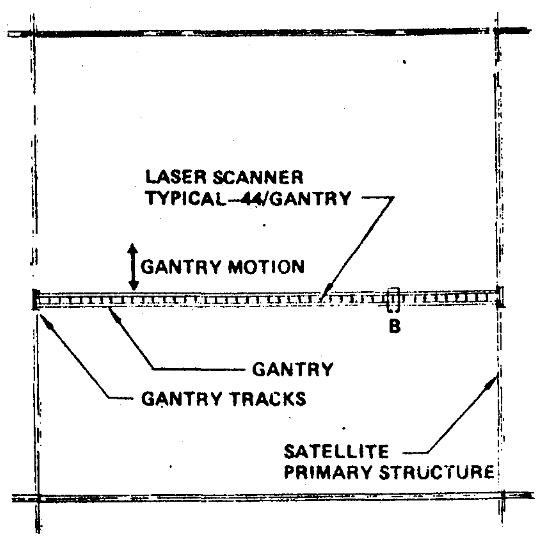

FIGURE 3-10

SPS Irradiation System

scaled down version could certainly be designed that would be more suitable for an electric propulsion vehicle. The curves of figure 3-11 show the results of recent tests by SPIRE for Boeing's SPS study. Silicon solar cells (10 Ω-cm) were irradiated with 1 MeV electrons and then annealed with five pulses from a CO_2 laser. The data shows almost a complete recovery from a degradation level of approximately thirty percent. If this technology could be developed to allow periodic in-flight annealing, we might see a power-time history similar to that of figure 3-12. This data was extrapolated from that shown in figures 3-7, -8 and -9, but does not take into account the shorter trip time and more favorable time-altitude profile that would result from the higher accelerations that would be realized through the heart of the Van Allen belts. It has been observed that the cell recovery is not total and this produces a gradual fall-off in maximum power output as indicated on the graph. Further studies are needed to determine such factors as the optimum depth of degradation to permit before annealing is initiated, and to quantify the performance gain that could be realized.

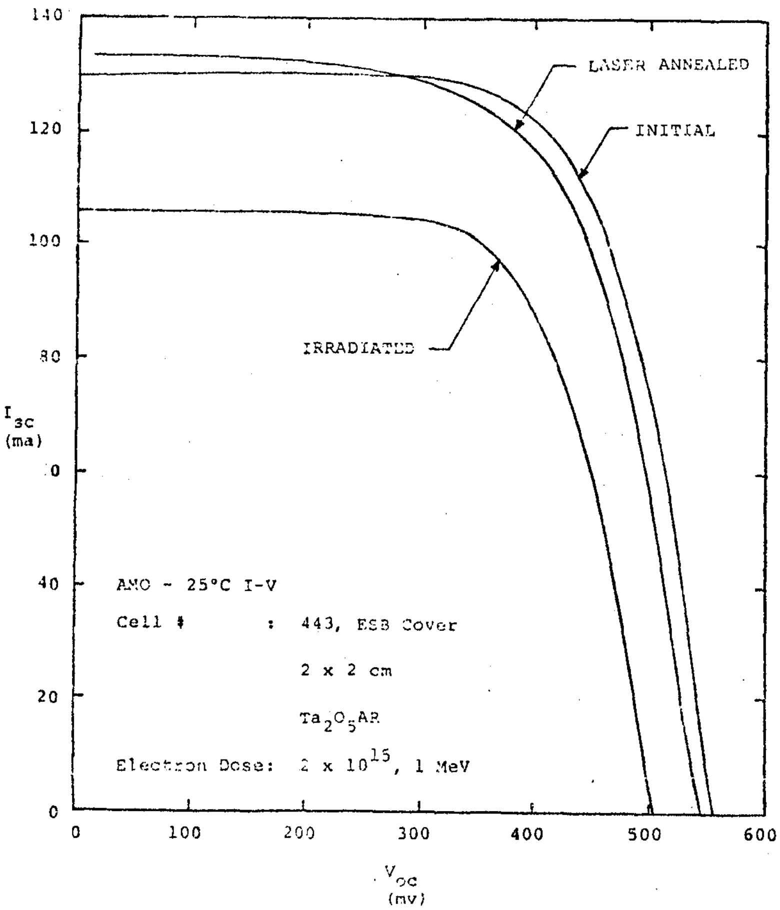

FIGURE 3-11 Cell Response

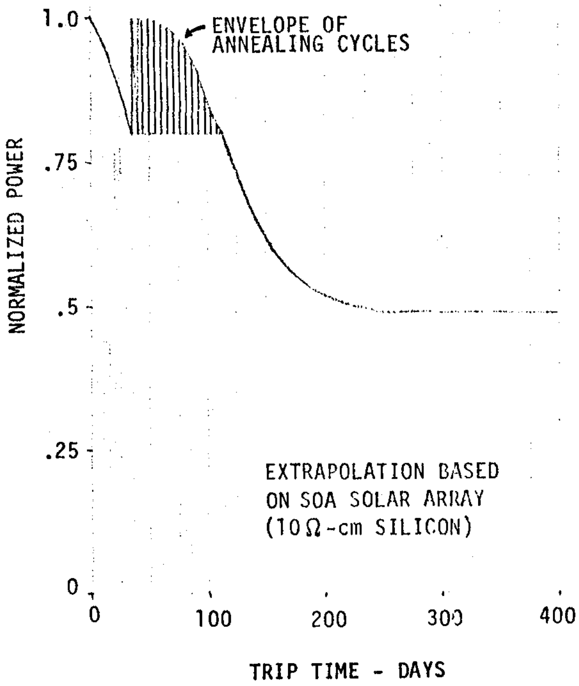

FIGURE 3-12 System Effects

In addition to degrading the EPS solar arrays, the Earth's trapped particle belts can produce damage in all other vehicle electronics. The curves of figure 3-13 show the dose that would be received by an avionics package as a function of the packaging. Typical spacecraft design practices produce an effective shielding thickness of approximately .25 cm. (100 mils) of aluminum, yielding an integrated dose of about 10^5 rad (Si) for a 180 day transfer. As can be seen from figure 3-14, this is within the damage threshold of many common electronics components. Thus, systems being designed for near-Earth utilization must consider the radiation environment in their selection of component and circuit types and may also find it necessary to include extra mass for shielding the avionics and power conditioning subsystems.

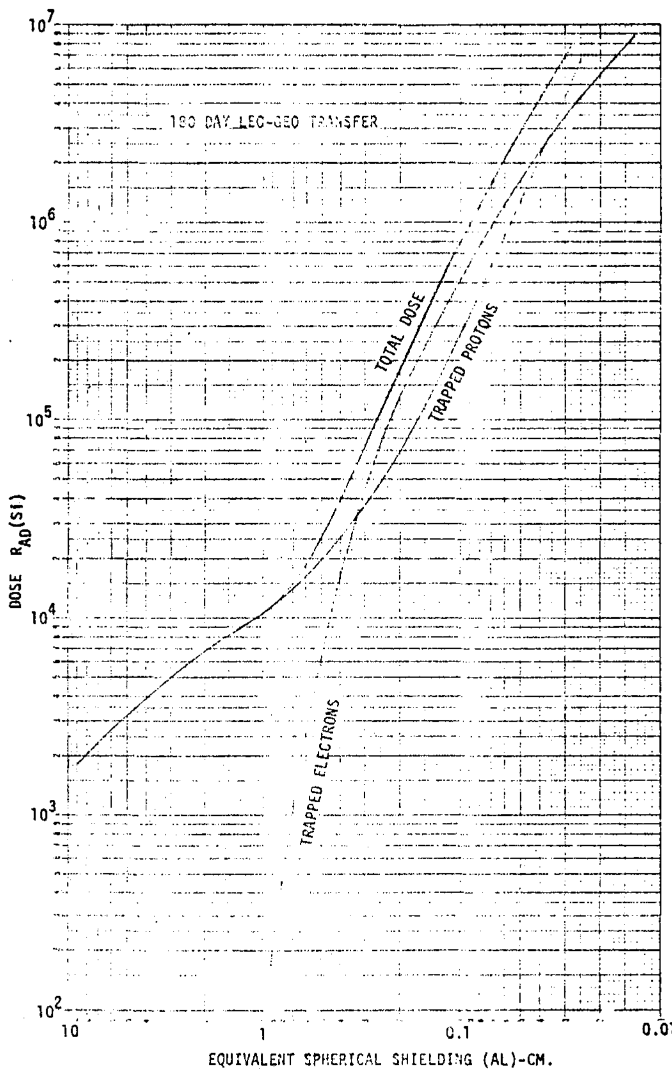

FIGURE 3-13 **Radiation Dose for LEO→GEO Transfer**

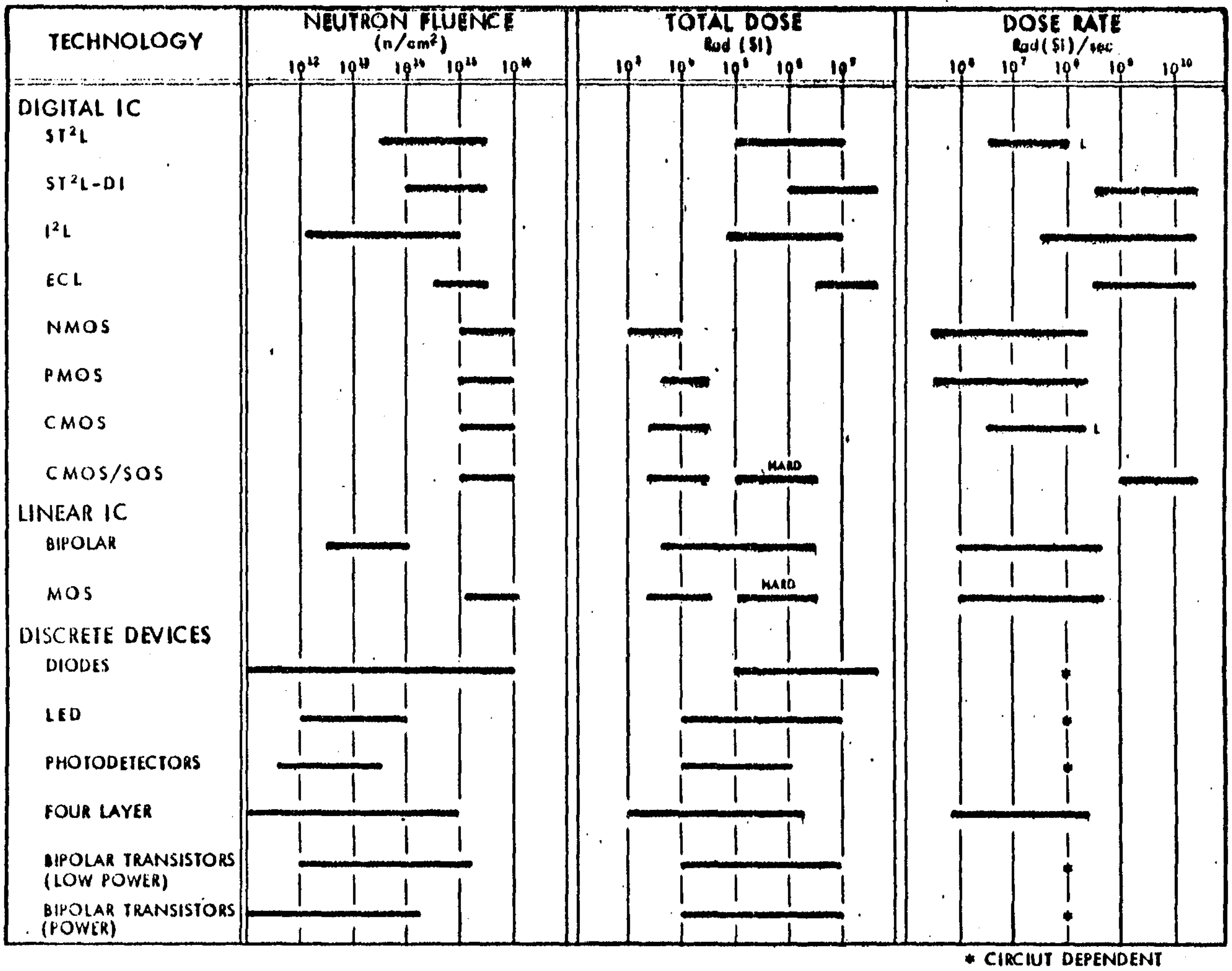

FIGURE 3-14 Ionization Test Data

OCCULTATION EFFECTS -- Earth shadowing is a significant design condition for Earth orbital missions. Since there will be no power available for the engines to function when the vehicle is in shadow, a performance loss will result. The magnitude of this potential loss can be seen in figure 3-15. Here the available thrusting (sunlight) time is plotted as a function of orbit height. The curve shown represents a maximum at the indicated altitude. The amount of occultation for any given trajectory depends on the relative alignment of the instantaneous orbit plane with the ecliptic, and may even approximate zero for the optimum choice of launch conditions. For these studies (see section 4.2), a "seasonally averaged" value was calculated by running numerous cases, and was judged to be appropriate for the highly active, space-industrialized future this study assumed.

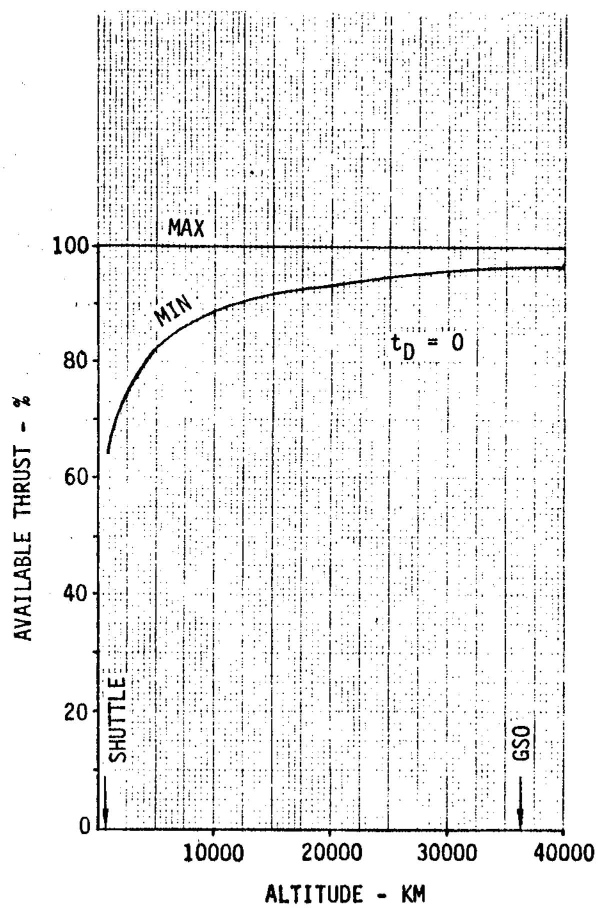

FIGURE 3-15 Mission Dependence of Occultation Penalty

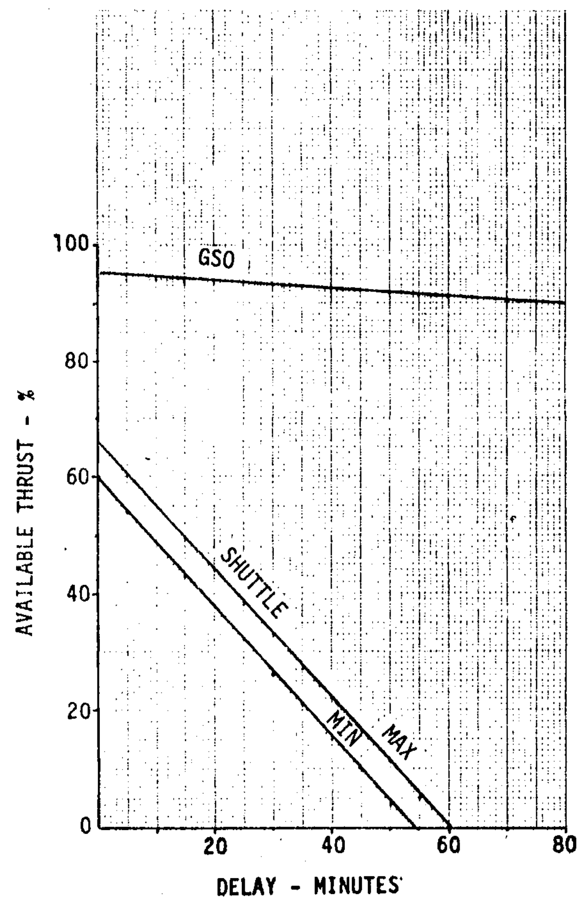

FIGURE 3-16 Technology Dependence of Occultation Penalty

In addition to thrusting time lost while in shadow, it will take a finite amount of time to start the ion engines of an EPS after array power is restored. The effect of this start-up delay is shown by figure 3-16. (The band labeled "Shuttle" represents a 28.5^{0} orbit at 300 km, and illustrates the variation that may be experienced between favorable/unfavorable launch windows.) It is obviously a significant effect, even on geostationary orbit, and may well justify the inclusion of "extra" heater circuitry in any future thruster/power processor system that is to be used for near-Earth applications. However, this modification is well within the current state-of-the-art, and imposes only a modest load on the power source. Thus only a minimal penalty was assumed in the studies reported in section 5.0.

EARTH-ORBITAL STEERING -- Typical steering profiles for a LEO to GEO transfer are shown in figures 3-17 and 3-18. (The numbers refer to the orbit number for a 340 day transfer with current technology, i.e., ~3000 seconds, and $a_0 \sim 5 \times 10^{-5}$ g's.) For initial EPS applications, the electric propulsion system will be comparable in physical size to the mission payload, and these steering requirements can be accommodated with minimal performance impact. As larger systems are developed, multi-module propulsion systems (as illustrated in figure 3-19) will be necessary to meet structural and other design considerations. In many cases non-optimal pointing for some EPS

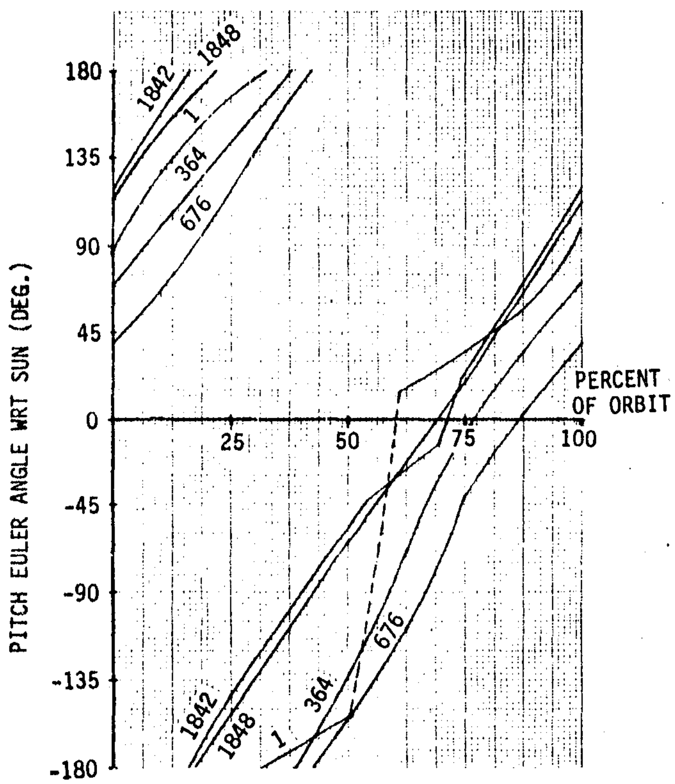

FIGURE 3-17 LEO→GEO Steering Profile (In Plane)

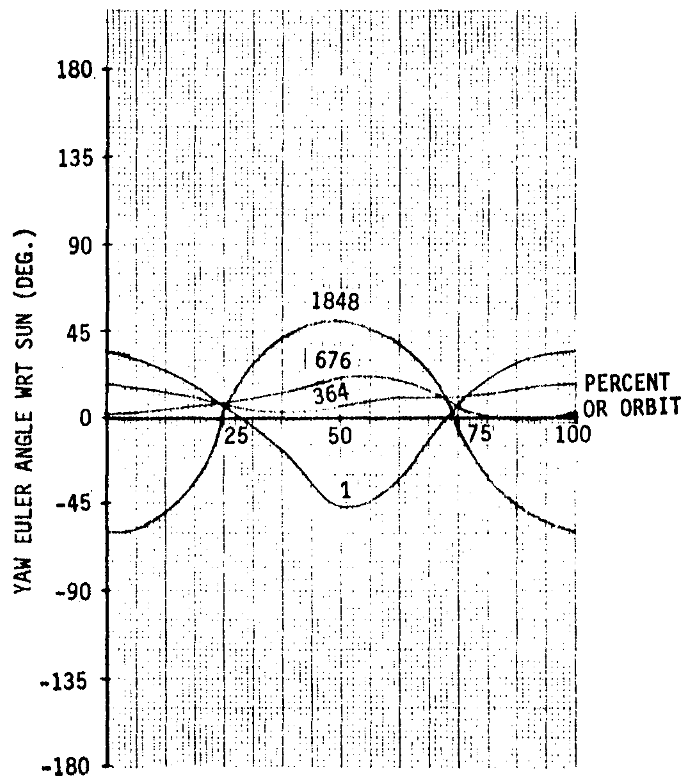

FIGURE 3-18 LEO→GEO Steering Profile (Out-of-Plane)

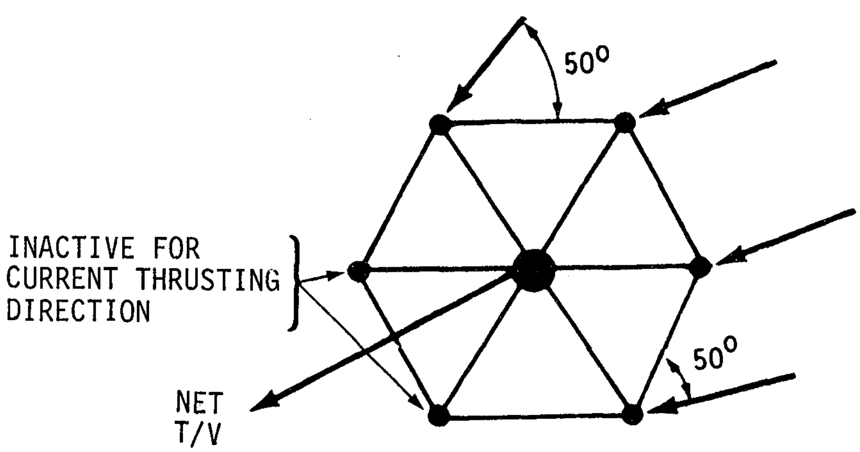

FIGURE 3-19

Thrust Vector Steering of a Distributed System

modules, and even additional thruster installations, will be necessary to allow the required freedoms in thrust vector pointing without violating plume impingement constraints. These effects were considered as performance losses in the parametric studies of task 3. Obviously, any developments which reduce the effective plume angle of ion bombardment thrusters or the harmful effects of impingement (e.g., different propellants), will decrease these losses.

3.2.2 Low Earth Orbits

After LEO to GEO transport, the next largest group of mission opportunities for an electric propulsion system lies in low Earth orbit. Missions in this group include:

- *Astronomical Telescope*
- *Multi-National Air Traffic Control Radar*
- *Space Construction Facility*
- *Tethered Satellite*
- *Soil Surface Texturometer*

Initially the primary role for an EPS in LEO was thought to be in final orbit placement, multiple-delivery economics, and in logistics support services. However, it was found that the function of orbit maintenance (drag cancellation) may be of more fundamental importance as orbiting structures increase in size.

The following series of curves were based upon the "Tethered Satellite" mission. Figure 3-20 gives the energy requirements to maintain a constant altitude for a system composed of an electric propulsion vehicle, and a small (1.4 m diameter) subsatellite suspended by a 100 km tether (approximately 1 mm in diameter). State-of-the-art characteristics (see section 4.3 and 4.5) were assumed for the EPS and its power source. The requirements for a shuttle-based system are also shown, and allow a comparison of the contribution of the EPS and the tethered satellite to the total system drag.

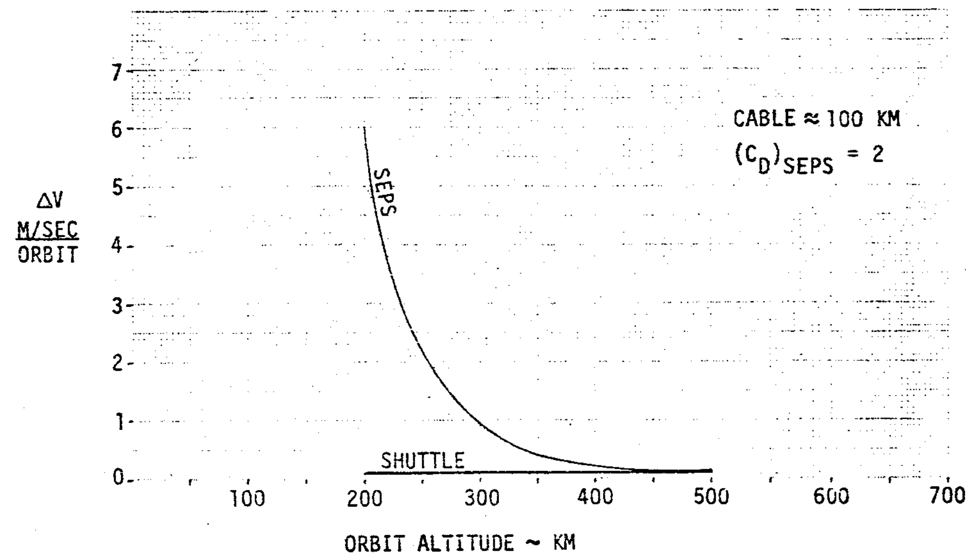

FIGURE 3-20 Altitude Maintenance Energy Requirements

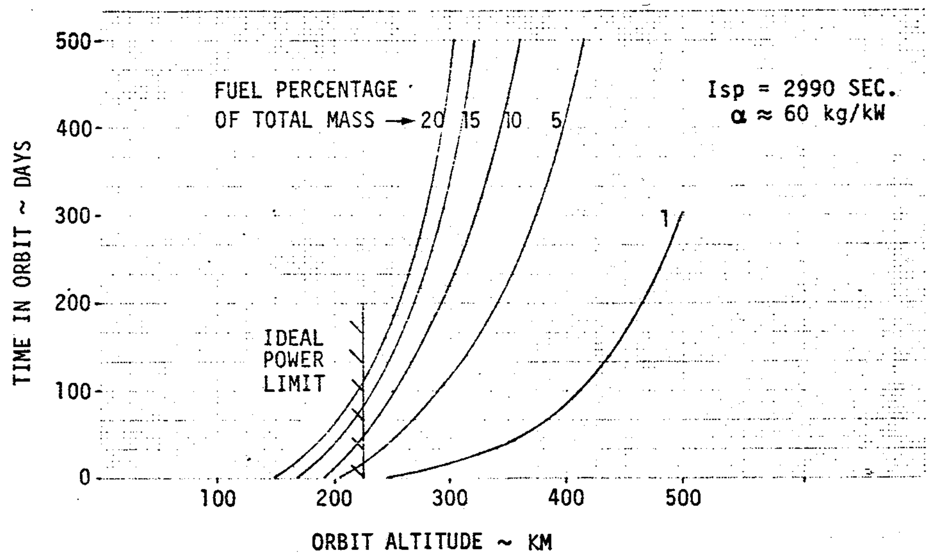

FIGURE 3-21 Orbital Stay Times

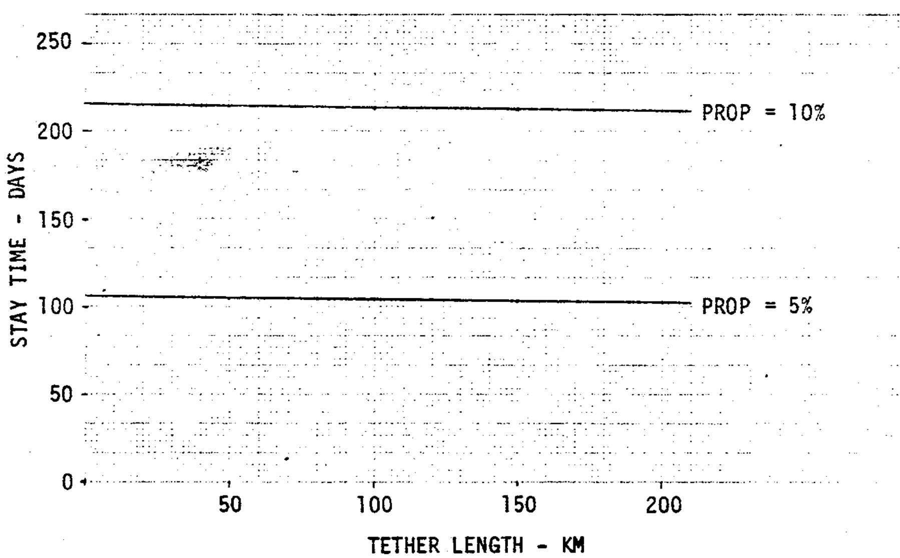

FIGURE 3-22 Sensitivity to Atmospheric Immersion

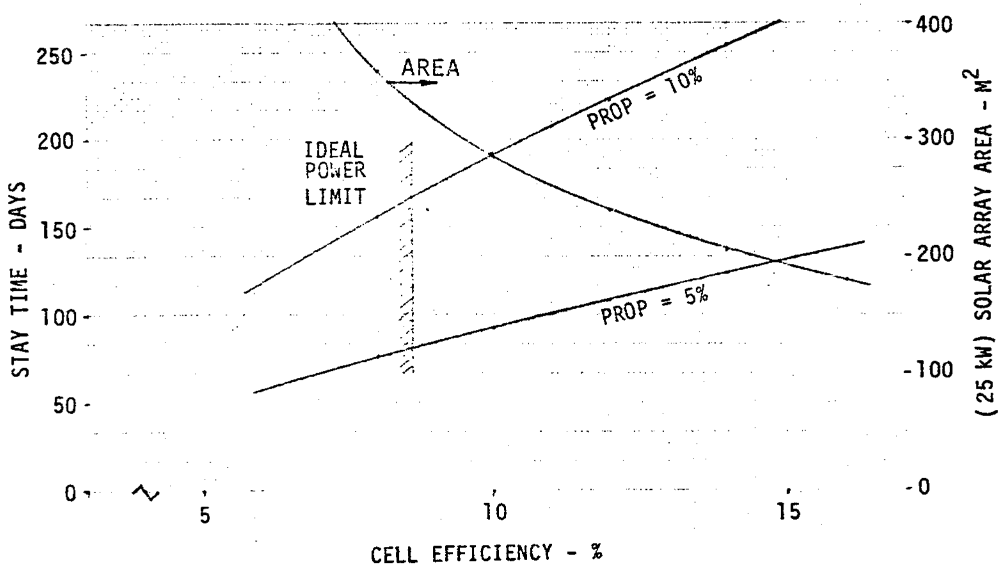

FIGURE 3-23 Sensitivity to Solar Array Size

In figure 3-21, these requirements have been interpreted in terms of the orbital stay times that are possible with various fuel loadings for the electric propulsion system. The ideal power limit represents the point at which the EPS is thrusting for 100% of the orbit (sun-synchronous or no shadowing), and thus represents the lower limit for mission feasibility. This lower limit will rise in inverse proportion to the amount of shadowing experienced in any given mission orbit.

As shown in figure 3-22, the tether length was varied while the electric propulsion vehicle was held at a constant altitude of 300 km. The change in stay time was not significant. In figure 3-23, the solar array was varied by changing the cell efficiency with a constant 100 km tether. Increased cell efficiency is then reflected in a smaller array area required to maintain a constant vehicle power level. A variable cell thickness was also postulated, thus raising the vehicle mass for higher values of cell efficiency. The effects are dramatic, suggesting that for LEO drag cancellation missions, array area rather than vehicle mass is the parameter to minimize.

3.2.3 Geosynchronous to Shuttle Orbit

The need for a "reverse LEO $\rightarrow$ GEO" transfer was represented by the mission to recover the orbit transfer hardware used to deliver a solar power satellite to geostationary orbit. This was seen to be a large and expensive hardware package, far exceeding any requirements for on-orbit attitude control and stationkeeping. Its return to LEO for refurbishment and reuse might justify the development of a recovery vehicle or could affect the optimization of the SPS delivery system.

Simulations of this mission were performed under a variety of conditions, as typified by figures 3-24, 3-25 and 3-26, to provide input data for tasks 3 and 4. (Data shown is for an Isp of 3000 seconds, an α of 60 kg/kw, and an initial acceleration of 4×10^{-4} m/sec.) No unique EPS technology drivers were noted. The viability of EPS recovery and reuse was seen to be dependent on economic assessments, as reported in section 5.

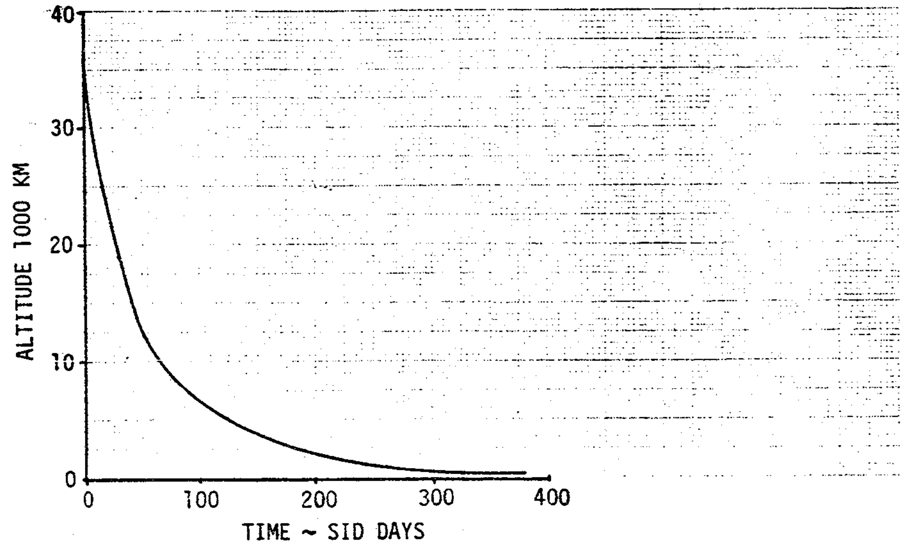

FIGURE 3-24 GEO→LEO Altitude Time History

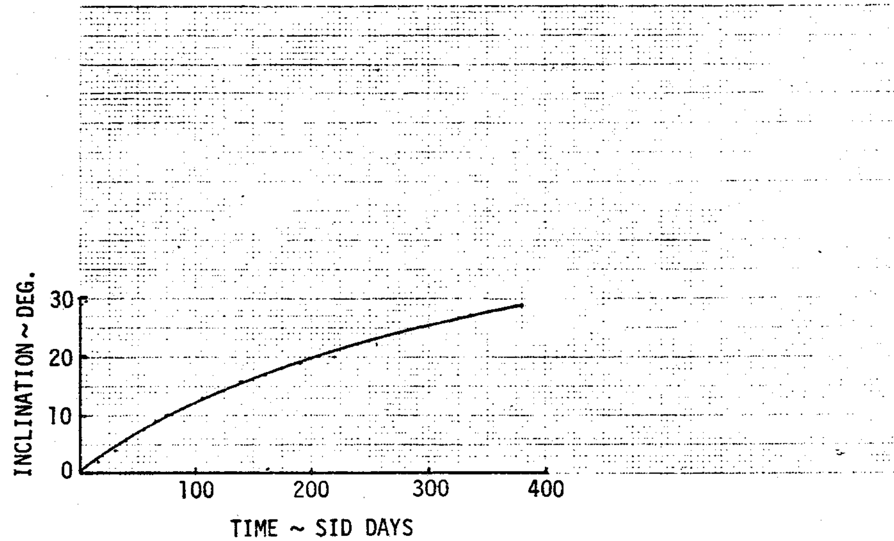

FIGURE 3-25 GEO→LEO Inclination Time History

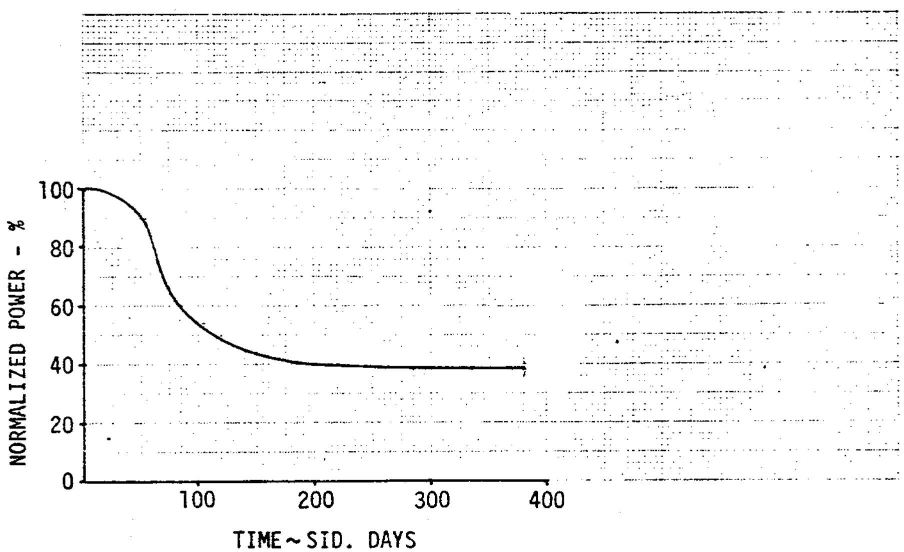

FIGURE 3-26 GEO→LEO Power Profile

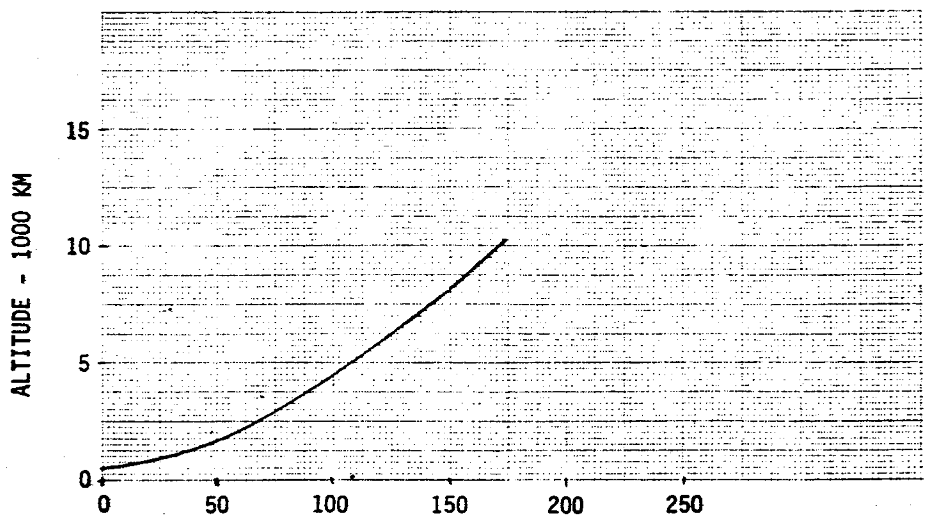

FIGURE 3-27 Altitude Time History - Earthwatch

3.2.4 Shuttle Orbit to Intermediate Orbit

Another important class of missions are those stationed in intermediate altitude orbits. Orbits below geosynchronous offer increased ground resolution and reduced beam attenuation, but generally require multiple payload emplacement (increasing propulsion opportunities) to achieve whole-Earth coverage. Missions of this group in the selected set were:

- *Global Search and Rescue Locator*
- *Earthwatch (Land-Sat Follow-on)*
- *Iceberg Dissipator*
- *Space-Based Radar (Near-Term)*

Time histories of altitude, inclination, jet power, and steering angles are presented in figures 3-27 thru 3-30, respectively, for a transfer to a 55^{o}, 11000 km orbit such as might be considered for an advanced Earth resources mission (SOA EPS). It is noted that the optimized trajectory quickly increases inclination to minimize the effects of the Van Allen belts. Propulsion system requirements are seen to be about the same as for the LEO to GEO transfers shown earlier. Interestingly enough, while the shorter mission times might suggest a greater potential for EPS reuse, it must be recognized that for this mission class, almost the entire vehicle lifetime is spent within/exposed to the radiation belts.

3.2.5 Elliptical Orbit to Geosynchronous

Some studies have suggested that use of a hybrid propulsion system might be most effective for the orbit raising of missions such as the Space Based Radar demonstration. In such an option, a medium thrust chemical propulsion system would be used to attain an intermediate altitude parking orbit after Shuttle launch and LEO assembly. Final orbital transfer and emplacement would then be performed by an electric propulsion. For this study, it was assumed that this mode would be used (whether further studies show this to be

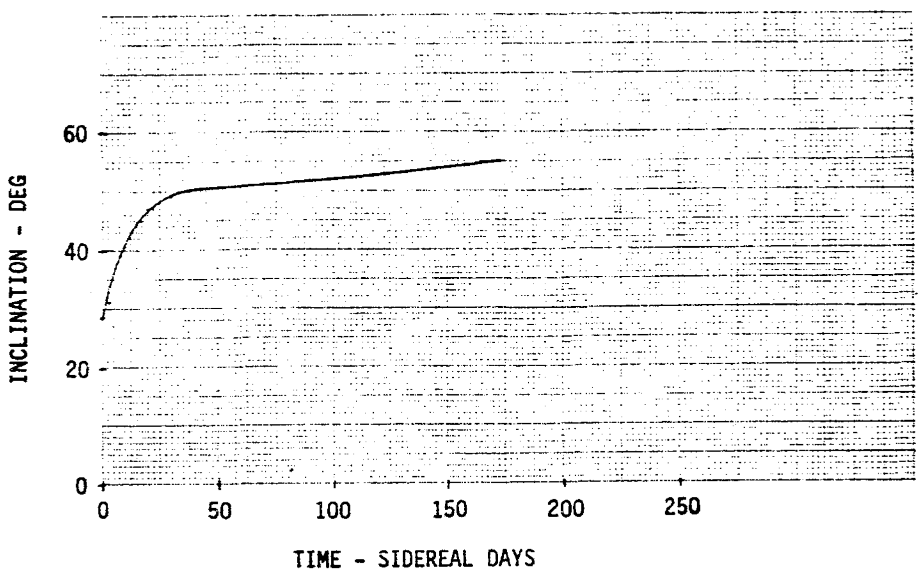

FIGURE 3-28 Inclination Time History
Earthwatch

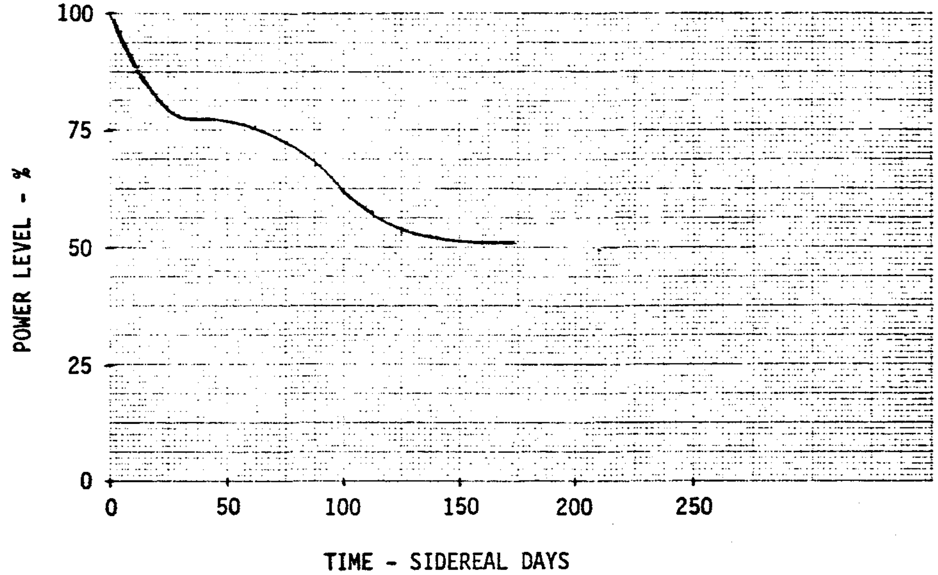

FIGURE 3-29 Power Profile - Earthwatch

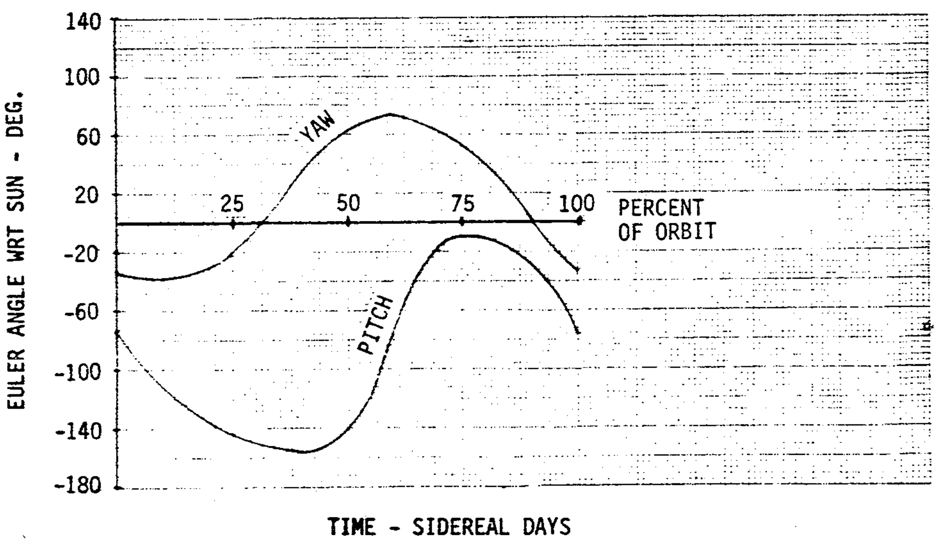

FIGURE 3-30 EPS Steering Angles
Earthwatch (near end of mission)

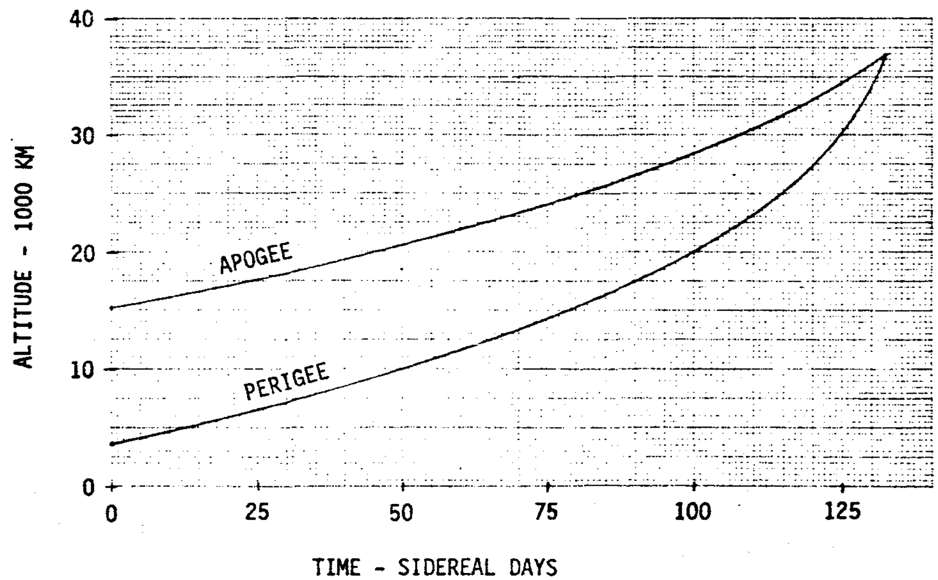

FIGURE 3-31 Altitude Time History - SBR

optimum or not) in order to ensure a consideration of any unique characteristics resulting from the high inclination/eccentricity starting condition.

Trajectory characteristics are shown in figures 3-31 thru 3-35. It is noted that the first month to month and a half are devoted primarily to raising both the apogee and perigee, in an effort to minimize the radiation damage to the solar array. Later in the mission, the thrusting pattern is modified to accomplish the necessary circularization and to reduce the eccentricity to zero. No particularly demanding requirements were noted. Penalty factors (see section 4.2) were generated for use in the final task.

3.2.6 Orbits Beyond Geosynchronous

The final class of trajectories considered those missions with destination orbits above synchronous altitude, yet with objectives still focused toward the Earth rather than on planetary explorations. From the selected set, these included:

- *Orbiting Lunar Station*
- *Nuclear Waste Disposal*
- *Magnetic Tail Mapping*

Figure 3-36 shows an altitude time-history for the initial, or "departure", phase of the above missions. This analysis assumed a "launch" from a geostationary orbit and a requirement for a coplanar transfer to an equatorial final orbit. It can be seen that approximately eight weeks are required to travel to the vicinity of the moon's orbit and an additional week to escape entirely from the Earth's sphere of influence. The data shown assume a vehicle wherein the payload mass is approximately equal to the mass of the electric propulsion system, and the system specific mass (α) is about 50 kg/kw, producing an initial acceleration of about 4 x 10^{-4} m/sec. Even more than the case of the LEO to GEO transfer, this trajectory type is sensitive to the initial acceleration (combined effect of vehicle specific power

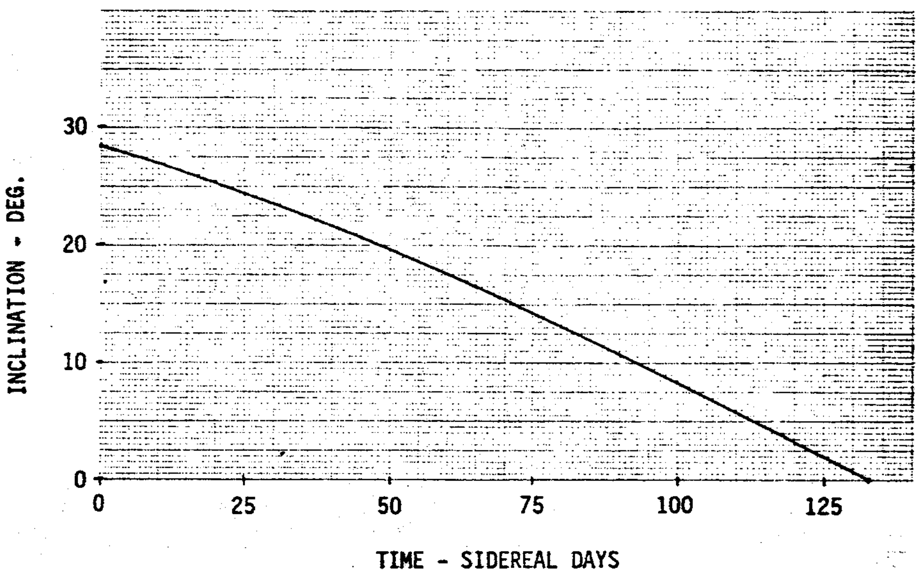

FIGURE 3-32 Inclination Time History - SBR

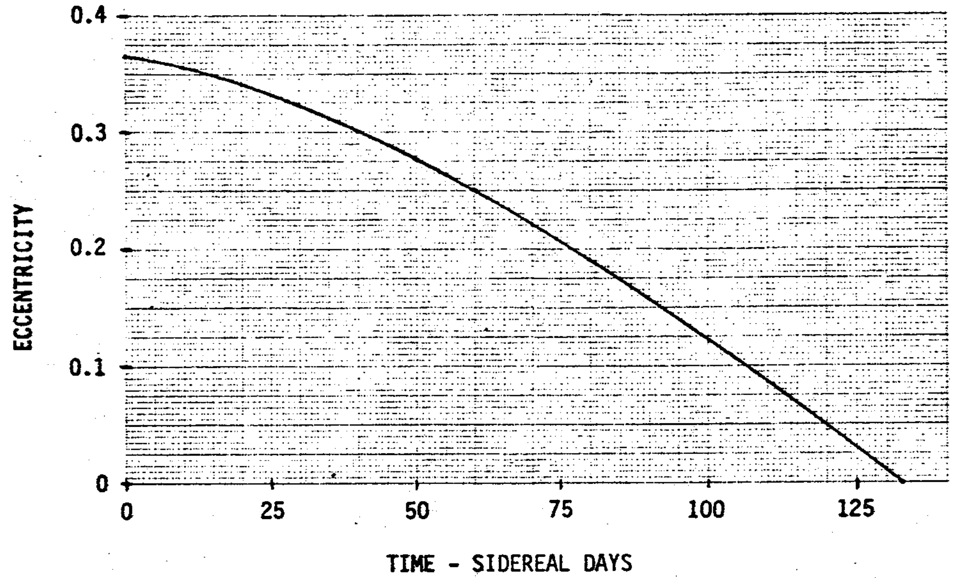

FIGURE 3-33 Eccentricity Time History - SBR

FIG. 3-34

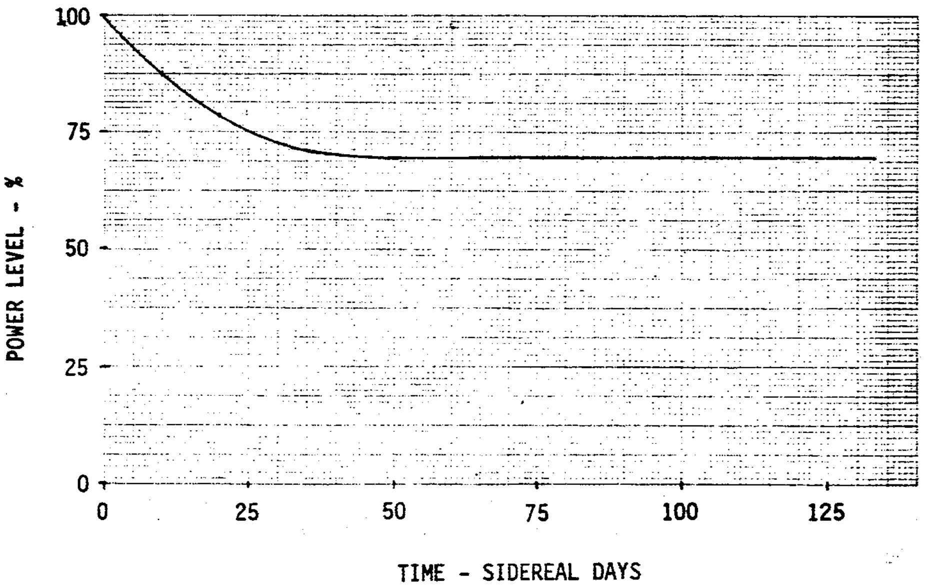

FIGURE 3-34 Power Profile - SBR

FIG. 3-35

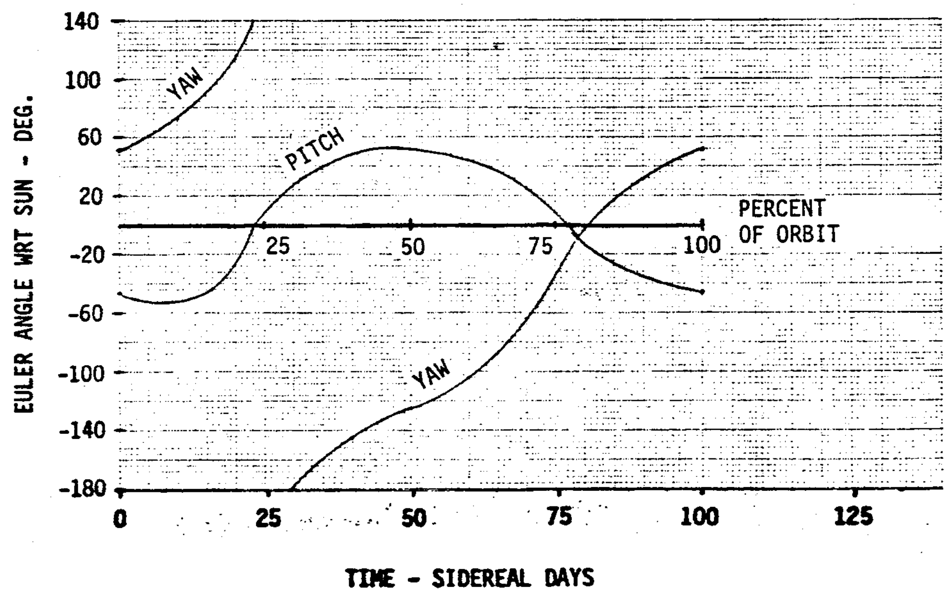

FIGURE 3-35 EPS Steering Angles - SBR (early in mission)

and payload mass) of the system. This acceleration dependence is displayed in figure 3-37 and is seen to be more important than the altitude finally attained. It was also found that these transfers are fairly cheap in terms of propellant consumption, as a direct result of the shorter transfer times. For example, to reach the moon's orbit from geosynchronous only requires about 15% as much propellant as the initial GEO to LEO transit.

For missions such as the Magnetic Tail Mapper, the station-keeping requirements to maintain a heliocentric orbit synchronized with the Earth are of interest. As evidenced in figure 3-38, the first calculations (2-body solution) ignored the effects of the Earth's gravity, but showed that such a maneuver was within the range of current electric propulsion technologies (accelerations of 10^{-5} to 10^{-4} g's). The Earth's effect was then included and is of course dependent on the relative positions of the Sun, Earth, and mission vehicle. Lunar perturbations are even more complex to illustrate but were seen to result in a maximum increase in the acceleration requirements of between 10 and 20 percent. Obviously, it is most economical to maintain a separation of about 1.5 million kilometers from the Earth, but sufficient motion for mapping purposes can be obtained with the acceleration levels possible from electrical propulsion systems. Propellant requirements for these missions can be estimated from figure 3-39, and are seen to be low enough to yield multi-year observation periods, as desired.

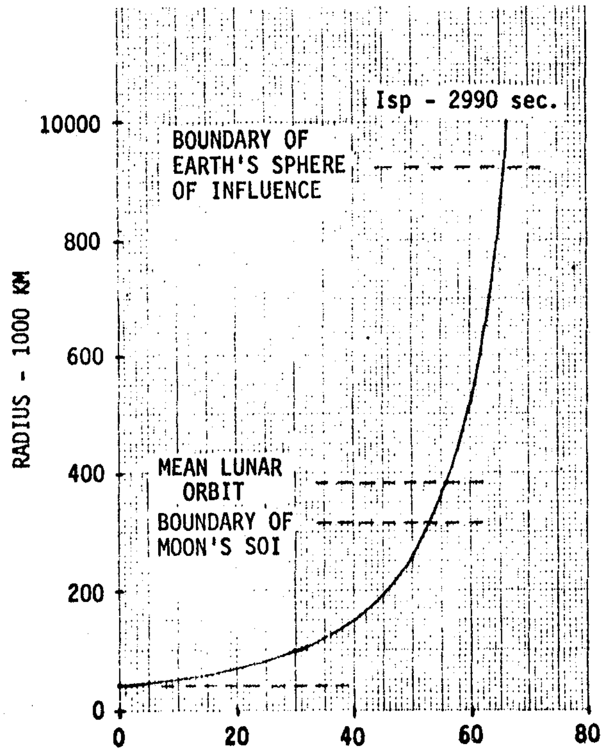

FIGURE 3-36 Transfer to Large Distances from Earth

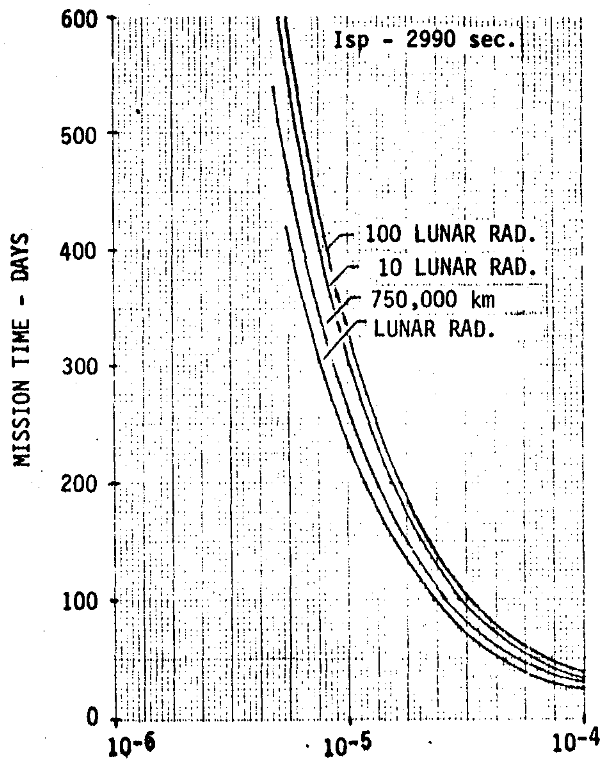

FIGURE 3-37 Sensitivities of Transfer Times for Exo-Synchronous Distances

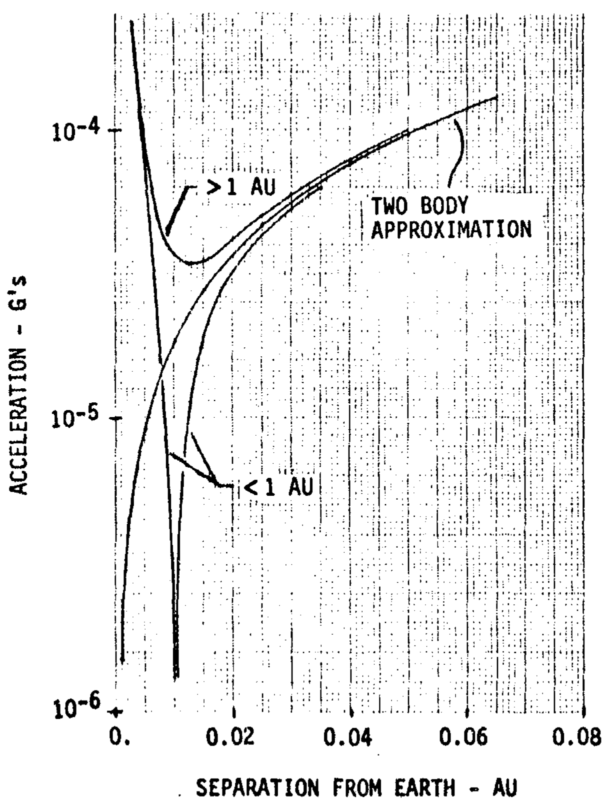

FIGURE 3-38 Radial Acceleration Requirements for Collinear Geo-Solar Stationkeeping

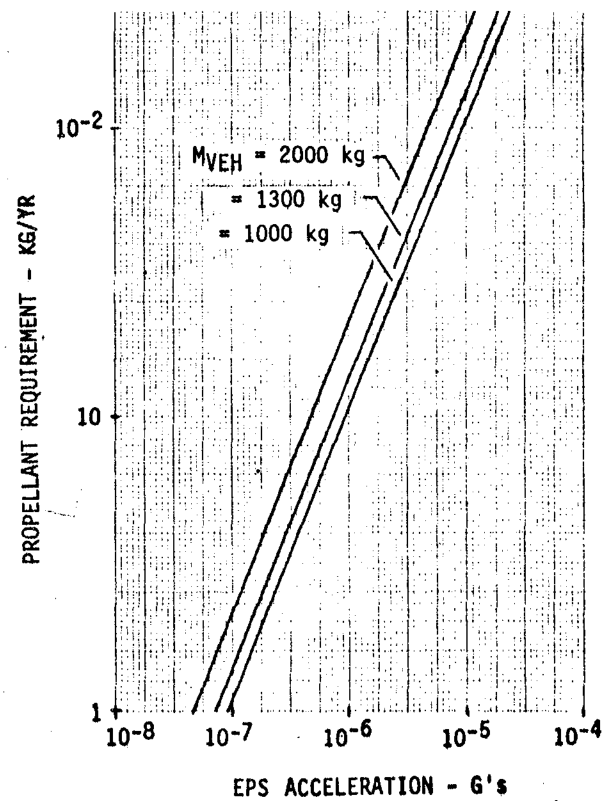

FIGURE 3-39 Propellant Requirements for Collinear Geo-Solar Stationkeeping (SOA Technology)

4.0 SYSTEM LEVEL COST MODELING

As the third task of this study, a simplified set of algorithms was developed to represent a generic electric propulsion system and to evaluate both its performance and its cost impact across the mission set. These algorithms were then implemented on an IBM 370 computer system to facilitate obtaining numerical results for the many sub-studies across the 30 mission set.

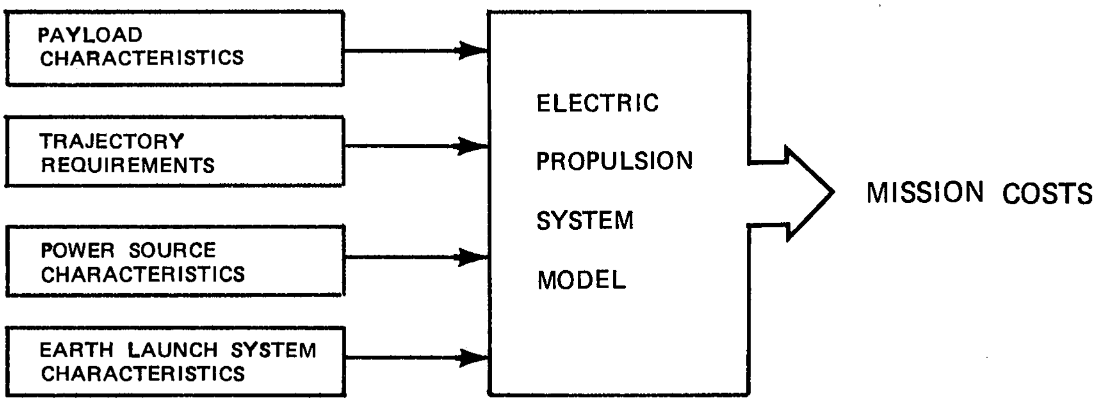

FIGURE 4-1 System Level EPS Modeling

The calculation process used is illustrated by the diagram of figure 4-1. In this process, certain parameters representing the payload, the trajectory, the power source and the Earth-to-low-orbit launch system are combined with algorithms characterizing the electric-propulsion system to produce a set of costs for each of the missions selected previously. Since this study considered only primary propulsion applications, the costs were formulated in terms of those associated with a transportation (e.g. orbit-raising) mission. In particular, the mission costs (in dollars) were expressed as:

$$C_M = C_{EPS} + C_{SA} + C_{ETO} + C_{TT} + C_P + C_{SCAR} - B , \qquad (4\text{-}1)$$

where the variables are as follows:

C_M = total mission costs, from the surface of the Earth to final destination orbit

C_{EPS} = purchase cost of the electric propulsion system (EPS)

C_{SA} = purchase cost of the power source for the EPS, generally a solar array (SA)

C_{ETO} = cost of the launch system required to place the payload and its transport system in low-Earth orbit (ETO = Earth-to-orbit).

C_{TT} = cost penalty resulting from the non-negligible transfer time

C_P = purchase cost of the propellant for the EPS

C_{SCAR} = "scar" cost associated with modifying the payload for compatibility with the EPS, and its extended transfer times.

B = cost benefit to the payload program due to utilization of some capability of EPS after arrival at the destination orbit, or to costs saving resulting from a low-thrust transfer (This factor was zero'd out in this system level study, due to difficulties in quantifying its value across the mission set, but should be included in any future studies which focus on specific payloads and specific implementation options for advanced electric propulsion systems.)

Many of the results to be shown in section 4 of this report will be expressed in terms of a characteristic value, ζ, which is the total delivery charge (usually given as \$/kg) for transportation from the Earth's surface to the final destination orbit. This is calculated as:

$\zeta = \frac{C_M}{M_{PL}}$ from the C_M above, and

from the mass of the mission payload (M_{PL})

Sections 4.1 thru 4.5 will discuss the various terms of equation 4-1, with the exception of C_{TT}. These trip time costs are given as:

$$C_{TT} = (\delta C_{PL} + \gamma_{OPS})\, T \tag{4-2}$$

where: δ= a "discount" factor which represents the cost of money to the payload program. This factor accounts for the fact that the payload sponsor's investment is "frozen" for the transfer period. The nominal value used in this study was 7% per year, although the parameter was varied from zero to 20% per year to obtain sensitivity data.

C_{PL} = purchase cost of the payload system (dollars)

γ_{OPS} = costs associated with operating the flight system (both EPS and payload) during the transfer period. The nominal value used in this study was 5 million dollars/year, and this parameter was varied over the range from 1 to 10 million dollars/year.

T = the time associated with completing the mission, as calculated by the electric propulsion system model.

Additionally, it should be noted that for three missions the discount factor (δ) was reduced to zero. These missions were the Tethered Satellite, Nuclear Waste Disposal, and the Gravity Gradient Explorer. For those cases, the trip time penalty was felt to be either non-existent or non-quantifiable.

4.1 PAYLOAD REPRESENTATION

In this system-level study, the primary characteristics of interest for each EPS payload are its mass (M_{PL}), and its cost (C_{PL}). Values of these factors were calculated for each mission during task 2 (see payload definition, section 3.1). However, for calculations of the EPS performance, a modified payload mass was used, which was defined as:

$$M_{PLD} = (1 + \alpha_{SCAR})\, M_{PL} \qquad (4\text{-}3)$$

where: α_{SCAR} = a mass penalty resulting from the modification of the payload to accommodate the EPS and EPS transfer. The nominal value used in this study was +7.5 gr/kg, and was derived from a survey of previous studies of the application of electric propulsion to specific payload programs. This parameter was varied over the range from -60 to +30 gr/kg.

As noted in equation 4-1, a "scar cost" term was included. This was calculated as:

$$C_{SCAR} = K_{SCAR}\, C_{PL} \qquad (4\text{-}4)$$

where:

K_{SCAR} = a cost penalty factor corresponding to α_{SCAR}. The nominal value was + $6.5/$1000 with a parametric variation from -$60 to +$30/$1000.

In the original study planning, it was felt necessary to divide the overall (30)

	FEATURE	GROUP 1	GROUP 2	GROUP 3	GROUP 4	GROUP 5
EPS	• DESIGN TYPE	CENTRALIZED	DISTRIBUTED	MODULAR	MODULAR	DISTRIBUTED
	• POWER SOURCE	CENTRALIZED	MODULAR	MODULAR	CENTRALIZED	DISTRIBUTED
	• POWER LEVEL	< 100 kW	> 100 kW	< 150 kW	> 100 kW	> 1 MW
	• LAUNCH VEHICLE	SHUTTLE	SHUTTLE	SHUTTLE	SHUTTLE	HLLV
	• ASSEMBLY	GROUND	SHUTTLE	SHUTTLE	GROUND	ORBITAL BASE
PAYLOAD	MASS	LIGHT	LIGHT	MODERATE	MODERATE	HIGH
	DENSITY	HIGH	MODERATE	LOW	HIGH	LOW

FIGURE 4-2 Payload Grouping Characteristics

mission set into several smaller groups which could each utilize a common electric propulsion system design. The primary payload characteristics which would affect the type of EPS were adjudged to be the total mass and the volumetric distribution (density) of that mass. The payload mass will determine the size (thrust/power level) of the EPS, while its physical extent will determine the EPS design constraints (view factors for thrust vector pointing, solar array exposure, and thermal control radiators).

Five groups were seen to be necessary to span the set of missions; their features are summarized in figure 4-2. Utilizing the familiarity with the payloads gained in task 2, the overall set was sorted into the five groups of missions that are indicated in figure 4-3. Figure 4-4 displays the range of characteristics present in each of the five groups. Specific values for each individual mission may be found in the appendix. To assist in presenting the study results, a representative mission was picked (which had "average" characteristics) for each group; these are the missions that are "boxed" in figure 4-3.

Some of the results to be presented in section 5 are shown in terms of the mission payload mass. Figure 4-5 relates the range of payload masses to the time frame in which the transportation service is first required, based on the nominally optimistic scenario used in this study. This information is helpful in developing a time scale for technology advancement.

GROUP 1

1983 - TETHERED SATELLITE
1985 - NUCLEAR WASTE DISPOSAL
1986 - UTILITY LOAD MANAGEMENT SATELLITE
1986 - EARTHWATCH
1986 - EARTH'S MAGNETIC TAIL MAPPER
1989 - ASTRONOMICAL TELESCOPE
1990 - NUCLEAR FUEL LOCATION SYSTEM
1991 - GLOBAL SEARCH & RESCUE LOCATOR
1994 - GEOSYNCHRONOUS - BASED SATELLITE MAINTENANCE

GROUP 2

1984 - ELECTRONIC MAIL TRANSMISSION
1985 - MULTI-NATIONAL AIR TRAFFIC CONTROL RADAR
1987 - SPACE BASED RADAR (NEAR TERM)
1987 - NEAR-TERM NAVIGATION CONCEPT
1988 - TECHNOLOGY DEVELOPMENT PLATFORM
1990 - PERSONAL COMMUNICATIONS WRIST RADIO
1995 - ORBITING DEEP SPACE RELAY STATION

GROUP 3

1985 - GRAVITY GRADIENT EXPLORER
1988 - SOIL SURFACE TEXTUROMETER
1991 - GSO COMMUNICATIONS PLATFORM
1992 - SPACE BASED RADAR (FAR TERM)
1993 - PERSONAL NAVIGATION WRIST SET
1995 - MARINE BROADCAST RADAR

GROUP 4

1993 - GEOSYNCHRONOUS SPACE STATION
1996 - ORBITING LUNAR STATION

GROUP 5

1986 - SPACE CONSTRUCTION FACILITY
1992 - POWER RELAY SATELLITE
1997 - ICEBERG DISSIPATOR
1997 - SPS PILOT PLANT
2002 - SATELLITE POWER SYSTEM
2004 - SPS ORBIT TRANSFER RECOVERY

FIGURE 4-3 Mission Groups and Representative Mission

CHARACTERISTIC		GROUP 1	GROUP 2	GROUP 3	GROUP 4	GROUP 5
IOC	-Year	1983-94	1984-95	1985-95	1993/96	1986-2004
ORBIT	-10^3KM	6.7-3000	7-42	7-42	42/384	6.9-42
INCLINATION	$- ^\circ$	0-50	0-90	0-50	0/18	0-60
MASS	- kg	0.4-6	0.7-14	2-14	16/22	28-12500
POWER	- kW	0-50	0.7-160	0.4-50	-0-	0-2M
MAX. DIMENSION	-m.	3-15	49-100	270-3100	13/35	57-6000
VOLUME	$- m^3$	1.7-550	25-88,000	60K-75M	186/563	28K-70B
DENSITY	$-kg/m^3$	7-310	0.05-29	0.0003-1.1	29/120	0.002-100
VALUE	- $M	1-175	2.2-430	17-488	120/145	36-7500
#MODULES/SYSTEM		1-22	-1-	-1-	9/10	1-8
TOTAL #MODULES		2-1300	1-150	1-5	10/27	1-104
MAX. #LAUNCHES/YR		1 - 50	1-25	-1-	-1-	1-25

FIGURE 4-4 Mission Characteristics by Group

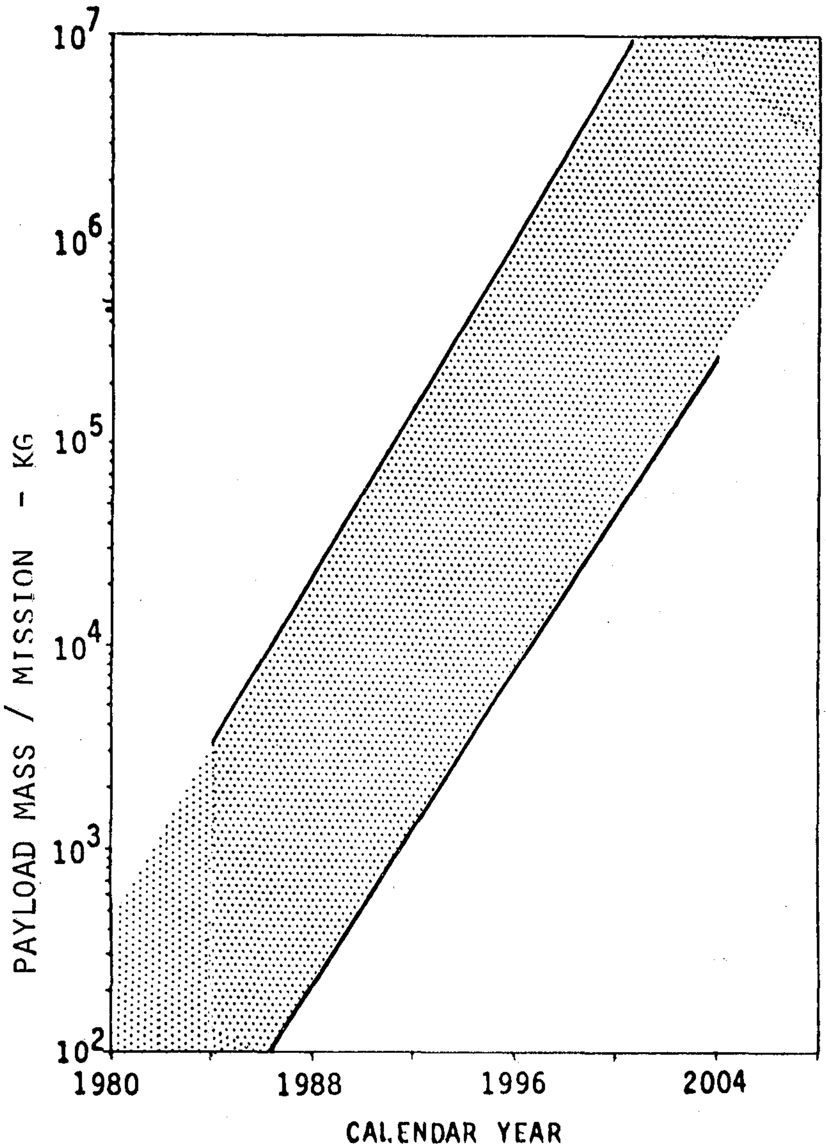

FIGURE 4-5 EPS Payload Mass Transport Scenario

4.2 TRAJECTORY CHARACTERIZATION

The traditional parameter characterizing the mission trajectory requirements is simply the velocity increment (ΔV), usually expressed in meters per second (m/s). In addition, for electric propulsion systems with photovolatic power sources, four other effects become important, and these have been modeled as penalty factors modifying certain terms in the calculations of EPS performance. They are:

R = loss factor that accounts for the decrease in solar array output due to accumulated cell-structure damage by the ionized particles trapped in the Earth's vicinity (e.g. in the Van Allen belts) - commonly referred to as radiation degradation. The factor decreases the value of the system power that is used to calculate mission time (see equation 4-13) as:

$$P_{EFF} = (1-R)P_{NOM}$$

ϕ = penalty to account for the time spent in shadow, during which no useful thrust is produced by the EPS. This factor relates the total mission time to the amount of time spent thrusting as:

$$T_{TOTAL} = (1 + \phi)T_{THRUST}$$

S = loss factor to account for the non-optimum thrust vector pointing that results from the inability to achieve the very high slew rates that are characteristic of low-Earth orbit maneuvering. This factor decreases the effective value of the thruster expellant velocity used to calculate propellant requirements (equation 4-11) as:

$$V_{EFF} = g_o I_{SP}(1-S)$$

where:

g_o = the normal value of the acceleration due to gravity. For this study, Go = 9.8 m/sec^2.

I_{SP} = the specific impulse of the EPS (in seconds).

D = penalty to account for the drag on extended surfaces (i.e. the EPS-dedicated solar array, or the payload itself for low-density cases). This factor increases the energy that must be supplied to accomplish the mission as:

$$\Delta V_{eff} = (1 + D)\, \Delta V_{req}$$

These factors represent performance penalties on EPS performance due to the various effects, and hence are not simply a direct function of a set of physical characteristics (e.g. cell type, frontal area etc.). In particular, they are a strong function of the second-order trajectory features that were not explicitly included in the system model, namely the altitudes of the initial and final orbits, the inclination/altitude profile, and the mission timing. In task 2, the characteristics of each trajectory type was calculated for conditions that fully covered the mission set, and for cases that both included and ignored each of the various physical losses, thus allowing a determination of the above loss factors. The values used for this study are given in figure 4-6. Seasonal variations have been averaged out, and radiation degradation is characteristic of the baseline solar array.

MISSION NAME	RADIATION DEGRADATION PENALTY	OCCULTATION PENALTY	STEERING PENALTY	DRAG LOSS PENALTY
Tethered Satellite	0.	2.2	0.	0.
Nuclear Waste Disposal	.40	.05	.03	.01
Utility Load Management Satellite	.47	.195	.05	.02
Earth's Magnetic Tail Mapper	.278	.077	.02	.008
Earthwatch	.36	.035	.08	.05
Astronomical Telescope	.02	1.5	.10	.07
Nuclear Fuel Location System	.47	.195	.05	.02
Global Search & Rescue Locator	.44	.15	.05	.023
Geosynchronous-Based Satellite Maint.	.02	.005	0.	0.
Electronic Mail Transmission	.47	.195	.05	.02
Multi-National Air Traffic Control Radar	.02	1.5	.10	.07
Space Based Radar (Near Term)	.39	.02	.06	.036
Near-Term Navigation Concept	.47	.195	.05	.02
Technology Development Platform	.47	.195	.05	.02
Personal Communications Wrist Radio	.47	.195	.05	.02
Orbiting Deep Space Relay Station	.47	.195	.05	.02
Gravity Gradient Explorer	.45	.195	.05	.02
Soil Surface Texturometer	.02	1.5	.10	.07
GSO Communications Platform	.47	.195	.05	.02
Space Based Radar (Far Term)	.02	0.	.005	0.
Personal Navigation Wrist Set	.47	.195	.05	.02
Marine Broadcast Radar	.47	.195	.05	.02
Geosynchronous Space Station	.47	.195	.05	.02
Orbiting Lunar Station	.28	.09	.04	.008
Space Construction Facility	.02	1.4	.10	.07
Power Relay Satellite	.47	.195	.05	.02
Iceberg Dissipator	.32	.05	.08	.06
SPS Pilot Plant	.47	.195	.05	.02
Satellite Power System	.47	.195	.05	.02
SPS Orbit Transfer Recovery	.49	.195	.05	.02

FIGURE 4-6 Nominal Values of Trajectory Characteristics

4.3 POWER SOURCE REPRESENTATION

The characteristic of paramount importance for the EPS-power source is, of course, its (electrical) size or watt-rating. Most of the analyses were performed with this value representing the power that was purchased/installed at system initialization. However, a brief examination was also made of "end-of-life" system sizing (see section 5.5.3 for a discussion). EPS power level was varied as a design parameter throughout the study.

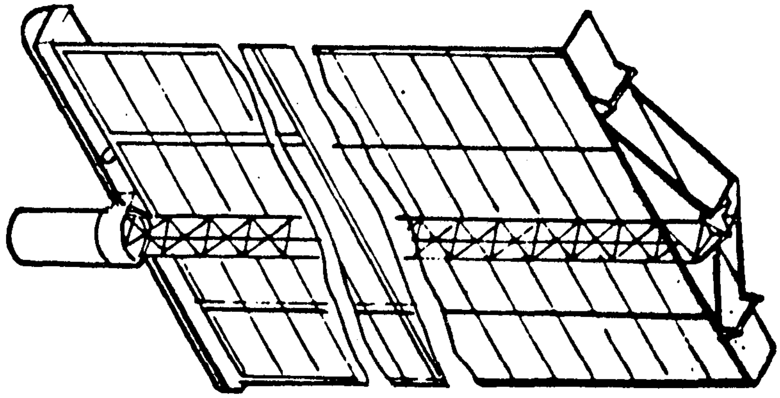

FIGURE 4-7 Baseline Solar Array

Many of the analyses reported herein are compared to a "baseline" electrical propulsion system. The power source postulated for this baseline system is shown in figure 4-7. This array is the flat-fold, deployable/retractable, flexible substrate design that has been developed by NASA's Marshall Space Flight Center over the past five years for the SEPS program. Its electrical size (P) is taken as 25kW. Its physical size is 4 x 32 meters. Conventional n-on-p silicon cells are employed, 8 mil thick with a conversion efficiency of 11.4%, and 6 milglass covers.

The second parameter of interest for the EPS power source is its mass (M_{SA}). In the simplified model, this was calculated as:

$$M_{SA} = \alpha_{SA} P \tag{4-5}$$

where:

α_{SA} = solar array specific mass. The nominal value used in this study was 15 kg/kw, corresponding to a mass of 375 kg for the baseline array. The parameter was varied from 1 to 20 kg/kw to obtain sensitivity data.

The final element characterizing the photovoltaic power source is its cost. This was calculated as:

$$C_{SA} = \gamma_{SA} P \tag{4-6}$$

where:

γ_{SA} = solar array specific cost. This parameter was initially taken as a constant \$350/watt (corresponding to a value of \$8.75M for the baseline array) and was to be varied from 50¢ to \$500/watt. However, treating this parameter as a constant produced extremely high array costs for missions with large payloads. This was seen to skew the relative magnitude of the components of equation 4-1, and hence would have distorted the study results. Therefore a variable cost function (shown in figure 4-8) was integrated into the model. It was derived from a survey of previous studies which project a "volume discount" philosophy in the solar array marketplace, with costs eventually reaching the 50¢/watt level which has been targeted for terrestial solar power.

4.4 LAUNCH SYSTEM REPRESENTATION

The Shuttle-based space transportation system (STS) was the baseline for launching each mission to a low-Earth orbit, from which the EPS operations could begin. The cost of this operation was calculated as:

$$C_{ETO} = \gamma_{STS} M_T \qquad (4\text{-}7)$$

where:

γ_{STS} = the STS specific launch cost. For this study, a Shuttle flight cost of \$20.5M was assumed, with a cargo capacity of 29,500 kg (65,000 pounds), resulting in a nominal γ_{STS} of \$700/kg. Treating this parameter as a constant also produced skewed results, since that philosophy did not recognize that launch vehicle technology would progress to support the more ambitious missions. Based upon our survey of studies involving growth versions of the STS, Shuttle-derivatives, and heavy lift launch vehicles (HLLV's), the cost function of figure 4-9 was formulated and incorporated into the model.

Also:

M_T = the total mass launched to LEO (in kg). This term was calculated as:

$$M_T = (1 + \alpha_{ADP})(M_{PLD} + K(M_{SA} + M_{EPS}) + M_P) \qquad (4\text{-}8)$$

where:

α_{ADP} = a factor to account for hardware (adapter) that is necessary to interface the STS to its cargo. The nominal value of α_{ADP} was 125 gr/kg. This parameter was varied from 0 to 250 gr/kg.

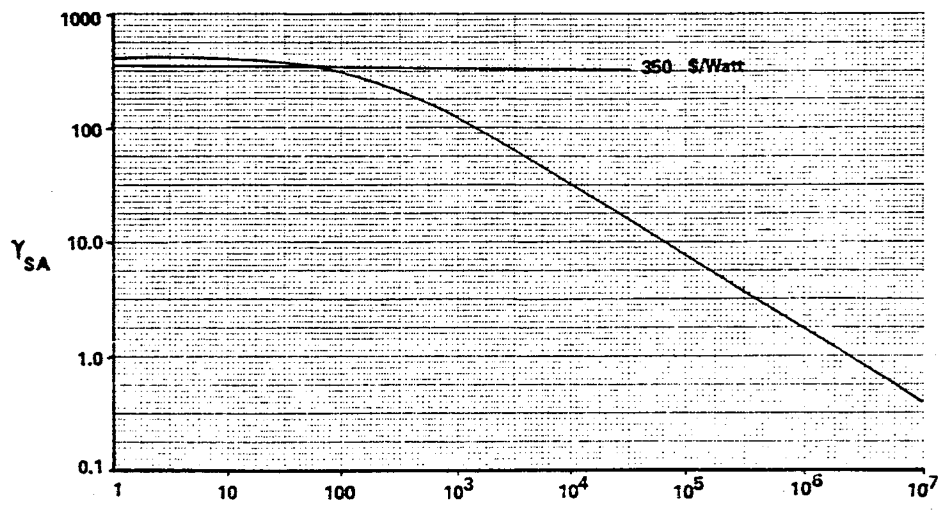

FIGURE 4-8 Solar Array Cost Function

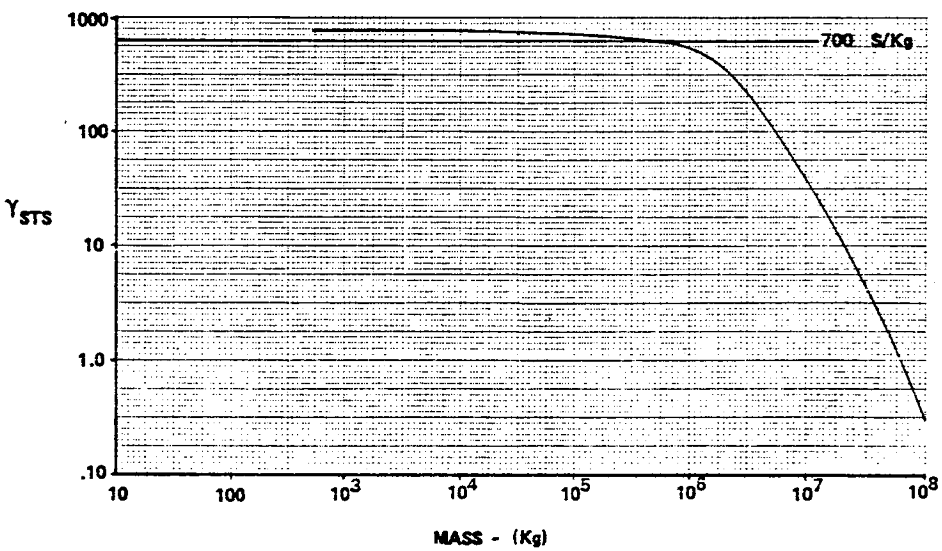

FIGURE 4-9 Earth Launch Cost Function

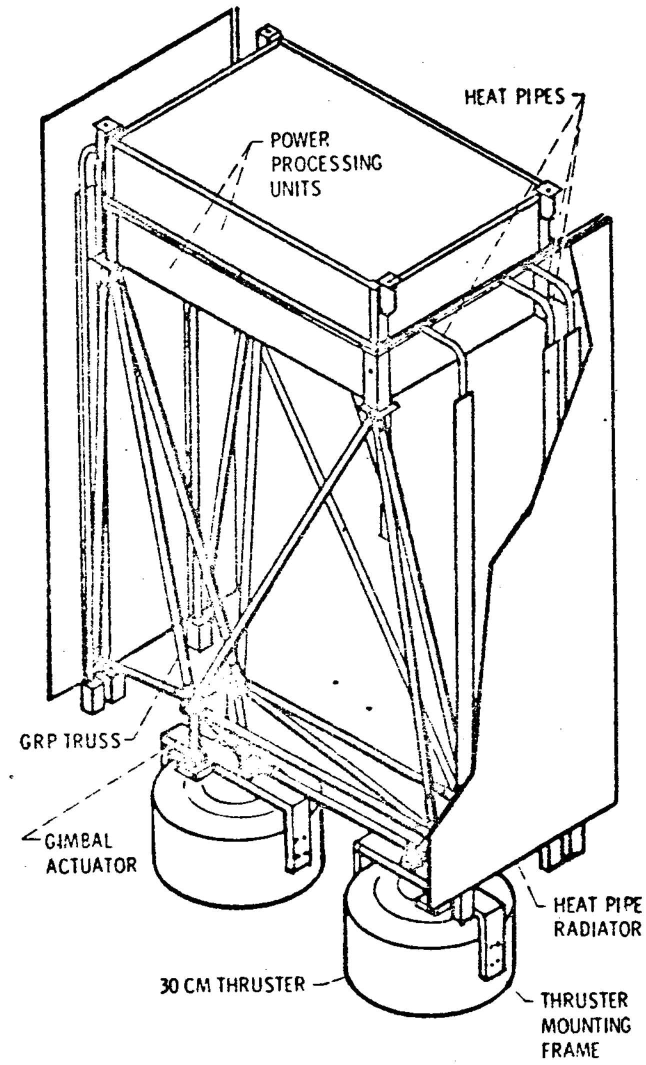

FIGURE 4-10 Bi-Mod Thrust Assembly

M_{PLD} = the modified payload mass of equation 4-3

M_{EPS} = the mass (kg) of the electric propulsion system (see equation 4-9)

M_{SA} = the solar array mass per equation 4-5.

K = a ground-based residency factor to account for reusable, space-based, EPS. No meaningful results were obtained for reusable systems during the course of this study, so K may be set to 1.

M_P = The mass of EPS propellant (see equation 4-11).

4.5 ELECTRIC PROPULSION SYSTEM REPRESENTATION

For this study, the state-of-the-art (SOA) in electric propulsion technology was considered to be that embodied in the "bi-mod" thrust assembly concept (shown in figure 4-10) being developed at the LeRC. This design was used to select nominal values for the EPS characterizing parameters. A specific impulse (I_{SP}) of 3000 seconds was used for evaluations of the baseline or SOA system, but was treated as a design parameter (range of variation = 500 to 10,000 seconds) for the majority of the analyses. The system lifetime was considered to be 15,000 hours, corresponding to the oft-quoted figure of 30,000 Ampere-hours for the SOA ion thruster.

An electric propulsion system was considered to be made up of several of these modules, the structure necessary to integrate the bi-mods to each other and to the payload, and the control avionics. The system mass (M_{EPS}) was calculated as:

$$M_{EPS} = \alpha_{EPS} P + \alpha_{STR} M_{PLD} + M_{AV} \qquad (4\text{-}9)$$

where:

α_{EPS} = that fraction of the system specific mass that accounts for the propulsion-related hardware. The nominal value used in this study was 21 kg/kw; however, the parameter was varied over the range from 2 to 30 kg/kw.

α_{STR} = that fraction of the system specific mass that accounts for the payload structural support hardware. The nominal value was 20 gr of EPS system for each kilogram of payload, and this parameter was varied from 0.1 to 100 gr/kg in the study.

M_{AV} = a factor to account for the constant mass that will be present in any EPS to accommodate system level functions. A nominal value of 200 kg was used, with variations from 0 to 500 kg.

The cost of the electric propulsion system was calculated as:

$$C_{EPS} = \gamma_{EPS} M_{EPS} \tag{4-10}$$

where:

γ_{EPS} = specific cost to produce the system (development amortization was ignored). This parameter was initially taken as a constant \$13,500/kg, and was to be varied from \$150 to \$100K/kg. However this constant treatment was seen to distort the analyses for advanced missions, since standardized, modular, systems tend to experience per-unit cost reductions when in volume production. Therefore, the variable cost function shown in figure 4-11 was formulated and integrated into our model.

The amount of propellant (M_P) required by the EPS is primarily determined by the mission requirements, and the mass of the system, and was calculated as:

$$M_P = (M_{PLD} + M_{EPS} + M_{SA}) \left(e^{\frac{\Delta V(1+D)}{I_{SP}\, go(1-S)}} - 1 \right) \tag{4-11}$$

where all variables are as previously defined in this section. This expression is derived from the familiar "rocket equation":

$$\Delta V = V_{EXH} \ln \left(\frac{M_{bo} + M_P}{M_{bo}} \right).$$

The cost of the propellant was computed as:

$$C_P = \gamma_P M_P \tag{4-12}$$

where:

γ_P = specific cost of the EPS propellant on Earth. This was originally taken as a constant \$15/kg, but the "quantity discount" function shown in figure 4-12 was later incorporated into the model (at

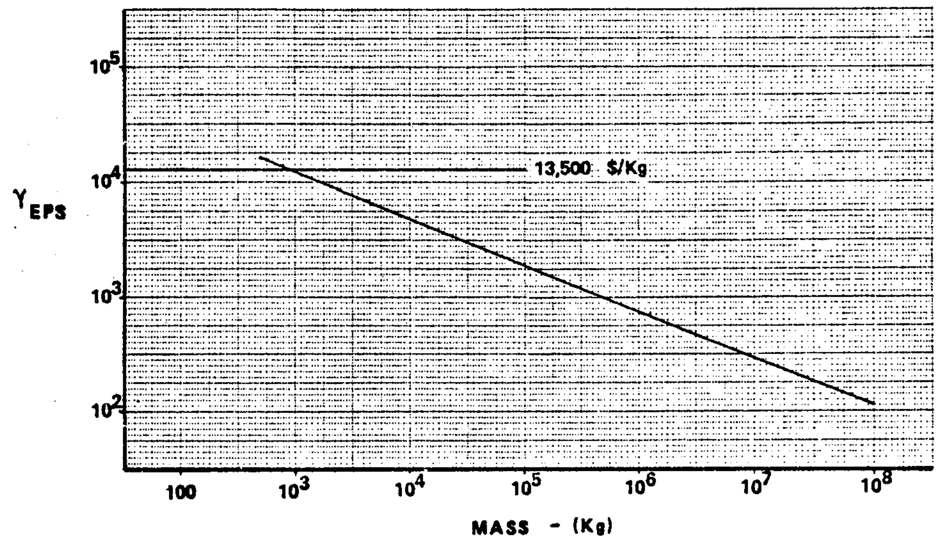

FIGURE 4-11 EPS Production Cost Function

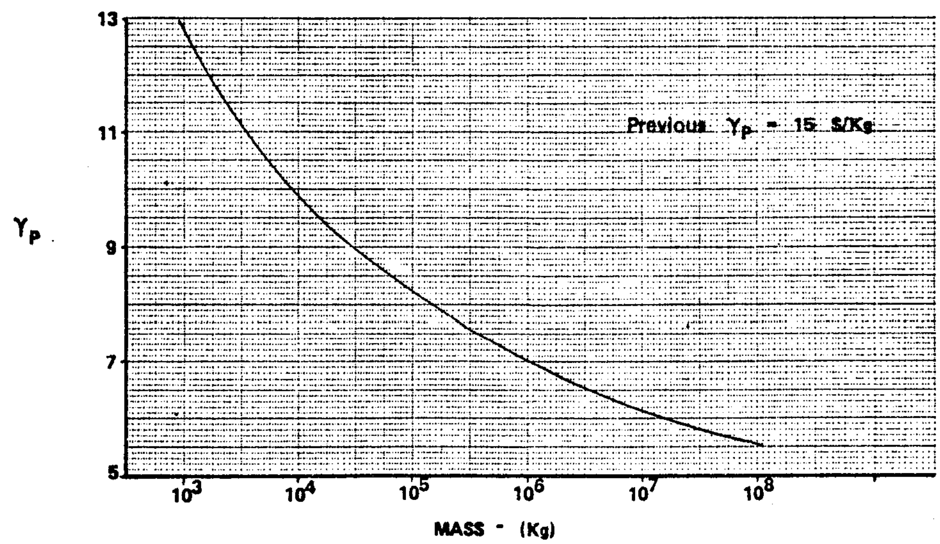

FIGURE 4-12 Propellant Cost Function

the same time as the functional relationships for γ_{STS}, γ_{SA}, and γ_{EPS} were established).

The final item of concern in modeling the EPS for Earth-orbital applications is its performance (i.e. the time which is required to achieve the mission objectives). This was calculated as:

$$T = \frac{M_P \, (g_o \, I_{SP})^2 \, (1 + \phi \, (1 + T_D))}{2 \, \eta P \, (1-R)} + T_R \tag{4-13}$$

where:

M_p, g_0, I_{SP}, ϕ, P, and R have previously been defined.

T_D = a penalty factor to account for the finite amount of time that is required to re-establish the engine systems' operating point after an eclipse period. This factor modifies the occultation penalty that was previously discussed. The nominal value was 0.23, which corresponds to a baseline start-up time of approximately 30 minutes. The range of parametric variation was from 0 to approximately 60 minutes.

T_R = the non-productive time for reusable systems from the end of one mission to the start of the next. No meaningful data was obtained for multiple-use systems in this study, therefore set T_R = 0.

η = the total system efficiency (i.e., for the thruster, the power processor, and any EPS cabling). The simplified (level 1) model recognized that efficiency was a function of the EPS operating point (Isp). For some analyses, the form of this relationship was variable, but generally:

$$\eta = \frac{1}{1 + \frac{k2}{I_{sp}\,2}} \text{, with } \eta \leq \eta_{MAX} \text{ was used.} \qquad (4\text{-}14)$$

This form has been used previously in low-thrust mission analysis programs (e.g. CHEBYTOP, etc.), and follows available empirical data fairly well. The values of the scaling constants were established by a curve-fit to the J series thruster performance predictions, given by the LeRC in February 1978. Nominal values of 1.094 (for k_1) and 6.99 X 10^6 (for k_2) were used with $\pm$20% variations studied. The limiting value of efficiency (η_{MAX}) was taken from the literature (reference 48 of figure 2-3) as 82%; this parameter was varied from 75% to 100% in the study.

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5.0 SENSITIVITY STUDIES

5.1 DESIGN POINT SELECTION

In any study aimed at identifying technology needs, such as this one, the conclusions reached can be influenced greatly by the boundary conditions that are assumed. Certainly, sensitivity studies can be performed which help one to understand the effects of these input assumptions, however, such analyses are usually limited to variations in only one or two parameters at a time, and thus sometimes do not tell the complete story. For this study, it was felt desirable to look at several conditions which represent major differences in the philosophical approach to designing the electrical propulsion system for a given mission (set). Four design points were identified. They are illustrated on figure 5-1 (wherein the mission time is presented as a function of system size) and figure 5-2 (here the total specific transportation costs - Earth to final destination - are plotted against system power level) for a relatively easy mission. These design points are:

1) the state-of-the-art system - provides an assessment of the capabilities of the current technology, and serves as a point of departure for the remaining studies.
2) the cost-optimum system - mission cost is judged to be of paramount importance and the size and operating conditions of the system are adjusted to minimize this quantity.
3) the minimum-power system - minimization of the size/cost of the power source is determined to be more critical than the mission cost here, and the system design is adjusted accordingly - specifically the thrusters are utilized to the limit of their lifetime.
4) the minimum-time system - in this case, mission time is critical, allowing a sacrifice of cost and power level. (Since true minimum time requires an infinite power source, an approximation to it is shown on the graph.) Such a case might come about through payload reliability considerations, for example.

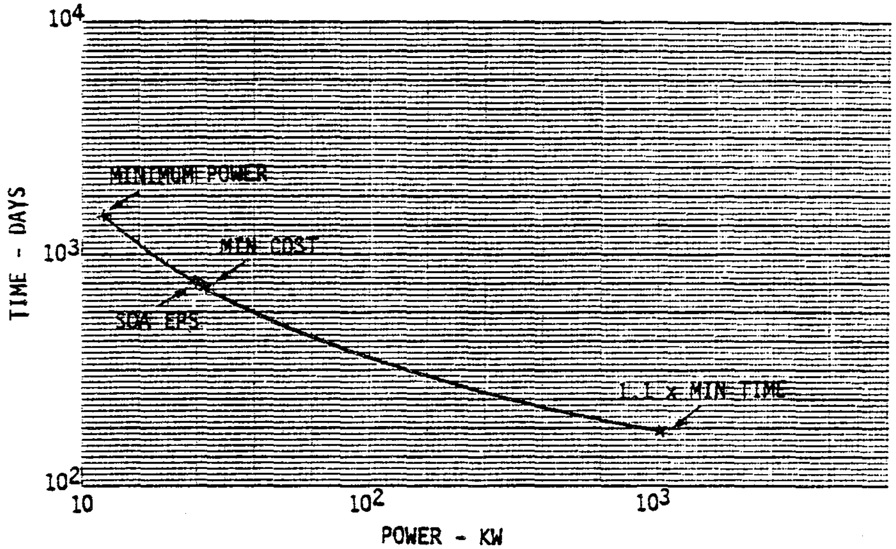


FIGURE 5-1 Time Relationships of System Design Points (Group 1)

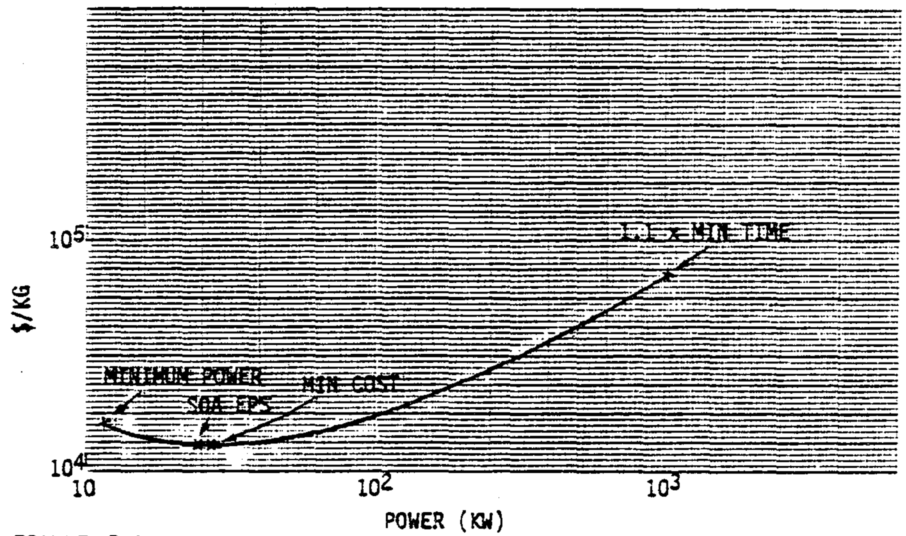


FIGURE 5-2 Cost Relationships of System Design Points (Group 1)

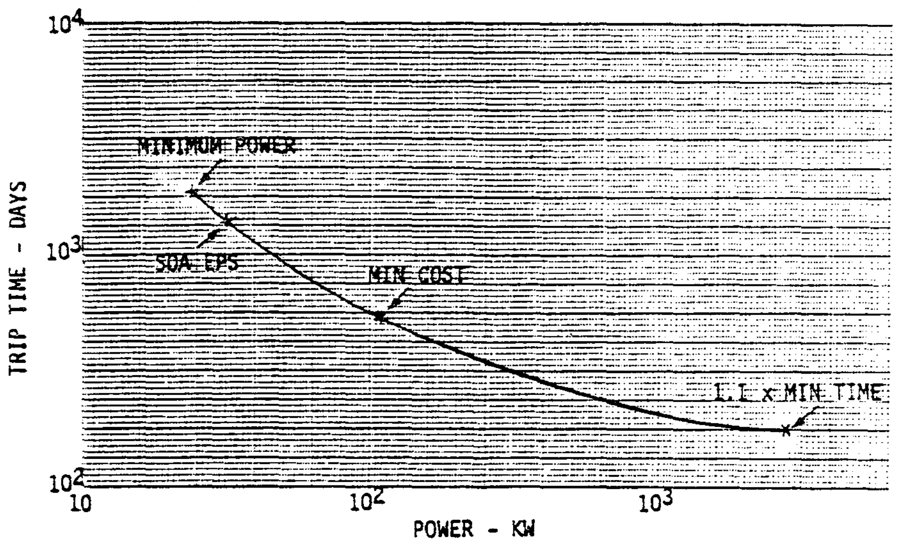


FIGURE 5-3 Time Relationships of System Design Points (Group 2)

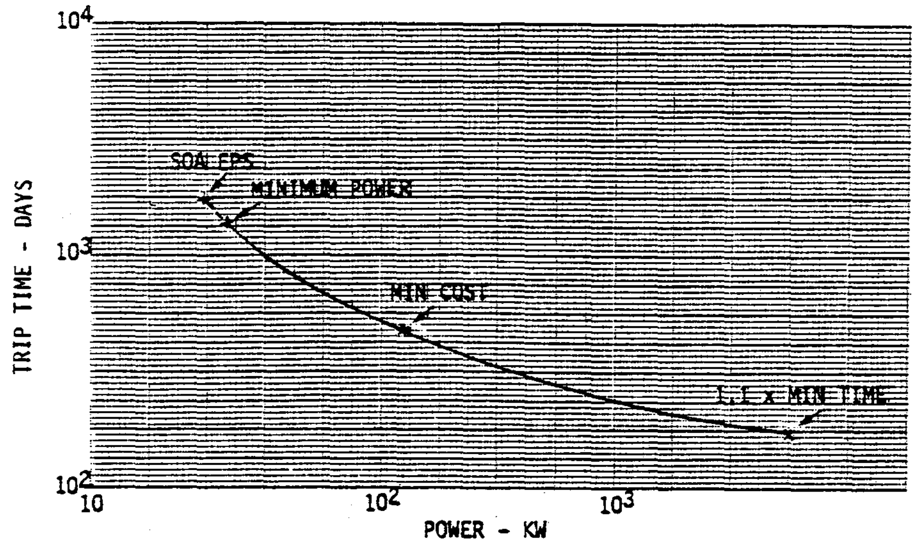


FIGURE 5-4 Time Relationships of System Design Points (Group 3)

In the remainder of this section, these design conditions will be utilized as a framework to discuss the analyses of potentially beneficial directions for EPS technology advancement. Most attention will be concentrated on the cost-optimum condition, since cost is generally perceived to be the primary design driver.

As a point of comparison, figures 5-3 thru 5-6 show the time relationships (similar to figure 5-1), and figures 5-7 thru 5-10 give the cost relationships (like figure 5-2), for missions representing the other 4 groups. It can be seen that for the near-term missions (group 1), the baseline (SOA) system is nearly cost-optimal. Further, significant reductions in mission time can be made - should that be deemed desirable - with only modest cost penalties.

As more ambitious missions are contemplated (perhaps in the "mature STS" era - groups 2 and 3), it is observed that the baseline performance approaches the minimum power design condition. Here mission feasibility is getting marginal (limited by lifetime technology), and costs could have been reduced by 50% or more. Further into the future (group 4 and 5 missions), the SOA design point has moved far to the left of the minimum power point, indicating that its use can no longer be considered - either from time feasibility or cost criteria. (Note, for the SPS Pilot Plant, use of the baseline - 25 kw/SOA EPS - system requires over 150 years.)

In retrospect, the perspective provided by these curves would have provided a more consistent set of mission groups than the criterion discussed in section 4.1. For future studies, a grouping process based upon the relationships of the 4 design conditions is suggested, since this relates to the applicability of today's technology and the motivation for further development effort.

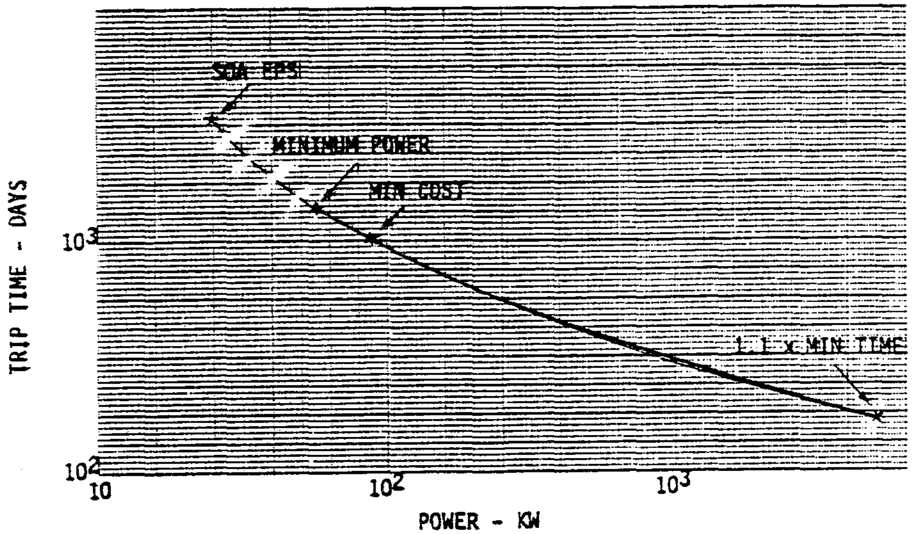


FIGURE 5-5 Time Relationships of System Design Points (Group 4)

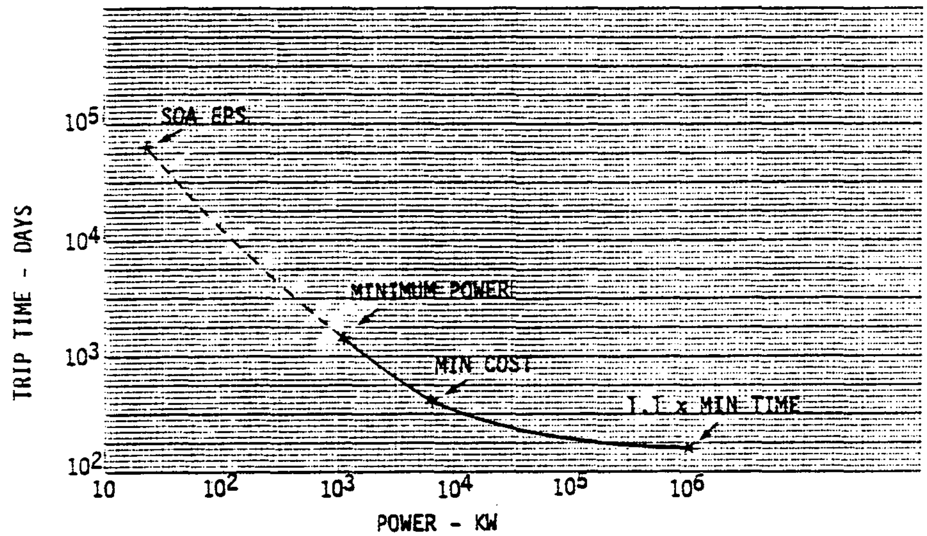


FIGURE 5-6 Time Relationships of System Design Points (Group 5)

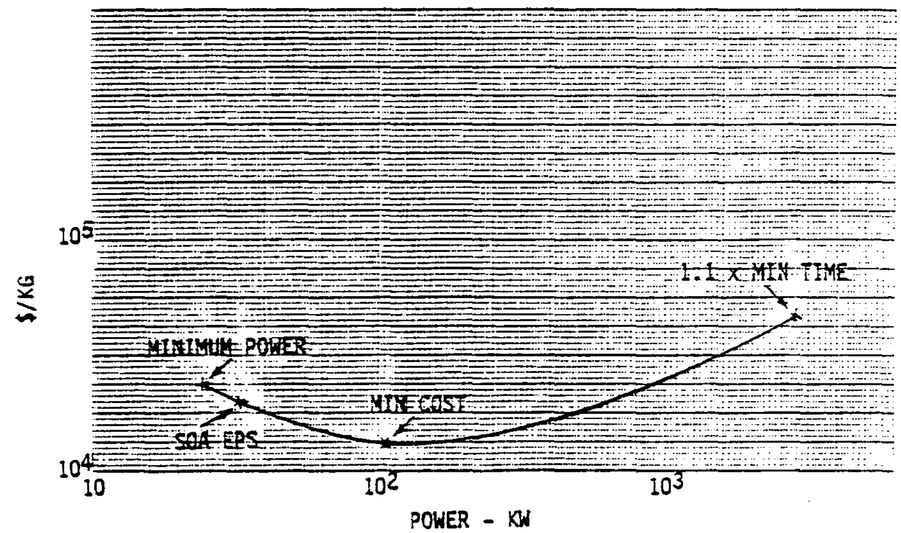


FIGURE 5-7 Cost Relationships of System Design Points (Group 2)

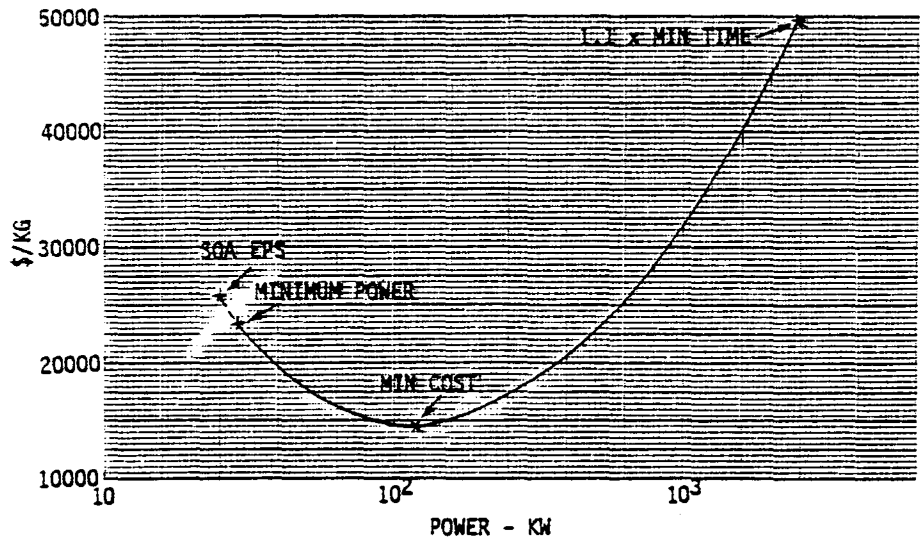


FIGURE 5-8 Cost Relationships of System Design Points (Group 3)

5.2 BASELINE SYSTEM DESIGN POINTS

A baseline electric propulsion system (also referred to as the SOA EPS) was characterized by the nominal parameter values that were given in section 4 of this report. It was representative of a system assembled from four bi-mods, a 25-kw solar array, supporting structure and a full capability avionics complement. This system was then "tried on" each member of the overall mission set.

The results have been tabulated in figure 5-11, which gives the calculated values of the components of transportation costs for each mission, and the propulsion time requirements. The column labeled "mission time" represents the total calendar time from initial orbit to final destination (all missions have been viewed as equivalent to transportation missions for these analyses). The column labeled "thruster time" represents the average "on-time" for an individual engine system (the analysis assumes that all units are cycled on and off as necessary to equalize thruster wear). The total

	MISSION NAME	MISSION TIME (DAYS)	THRUSTER TIME (HRS)	COSTS ($M)						TOTAL $/KG
				EPS	SA	LAUNCH	TRIP	PROPELLANT	SCAR	
1	Tethered Satellite	436	3270	9.893	8.750	1.722	5.967	.005	0.	37368
2	Nuclear Waste Disposal	730	10005	10.272	8.750	4.496	9.991	.015	0.	10315
3	Utility Load Management Satellite	781	8294	10.265	8.750	4.265	18.177	.013	.326	13061
4	Earth's Magnetic Tail Mapper	262	4212	9.844	8.750	1.589	3.965	.007	.049	64542
5	Earthwatch	613	9081	10.742	8.750	7.038	13.100	.014	.260	6139
6	Astronomical Telescope	531	4791	9.923	8.750	2.038	25.101	.007	1.137	52175
7	Nuclear Fuel Location System	445	4721	9.992	8.750	2.427	7.023	.008	.072	20787
8	Global Search & Rescue Locator	266	2983	9.924	8.750	1.898	4.520	.005	.130	27723
9	Geosynchronous-Based Satellite Maint.	81	1896	9.942	8.750	1.892	1.616	.004	.216	21741
10	Electronic Mail Transmission	1860	19736	11.109	8.750	10.157	178.768	.028	2.795	23253
11	Multi-National Air Traffic Control Radar	550	4962	10.043	8.750	2.701	7.767	.008	.014	17225
12	Space Based Radar (Near Term)	320	5043	10.382	8.750	4.565	8.984	.007	.488	8294
13	Near-Term Navigation Concept	330	3501	9.896	8.750	1.793	10.160	.006	.585	43021
14	Technology Development Platform	838	8891	10.248	8.750	4.235	17.893	.013	.260	13398
15	Personal Communications Wrist Radio	2757	29252	11.779	8.750	16.050	196.220	.040	1.950	16699
16	Orbiting Deep Space Relay Station	1386	14706	10.884	8.750	8.370	41.538	.021	.652	9349
17	Gravity Gradient Explorer	1191	13113	10.526	8.750	6.192	16.299	.019	0.	8368
18	Soil Surface Texturometer	648	5846	10.133	8.750	3.276	14.454	.009	.292	15980
19	GSO Communications Platform	1696	17994	10.983	8.750	9.258	181.796	.025	3.172	26096
20	Space Based Radar (Far Term)	333	7832	10.814	8.750	7.320	10.929	.013	.650	6497
21	Personal Navigation Wrist Set	2683	28467	11.728	8.750	14.650	88.159	.039	.660	9116
22	Marine Broadcast Radar	1421	15077	10.771	8.750	7.760	57.590	.021	.910	12806
23	Geosynchronous Space Station	3214	34100	12.112	8.750	17.645	117.905	.046	.780	9523
24	Orbiting Lunar Station	3843	60879	12.835	8.750	24.779	159.410	.078	.942	9357
25	Space Construction Facility	142097	1.338×10^6	119.544	8.750	640.424	86366.9	1.270	20.149	34863
26	Power Relay Satellite	5226	55448	13.507	8.750	28.521	107.591	.071	.234	6770
27	Iceberg Dissipator	271569	4.207×10^6	96.995	8.750	674.790	16728.4	3.977	1.625	10008
28	SPS Pilot Plant	62384	6.619×10^5	38.378	8.750	321.590	90524.1	.693	48.750	267477
29	Satellite Power System	2206556	2.426×10^7	310.112	8.750	155.041	222238.8	20.105	32.600	177833
30	SPS Orbit Transfer Recovery	52476	5.357×10^5	34.293	8.750	266.418	1170.9	.570	.292	6383

FIGURE 5-11 State-of-the-Art EPS Performance

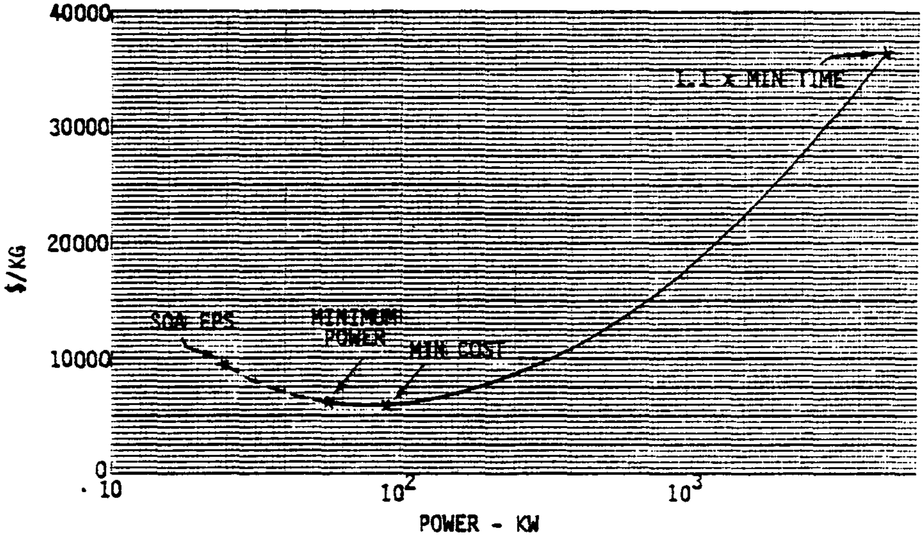


FIGURE 5-9 Cost Relationships of System Design Points (Group 4)

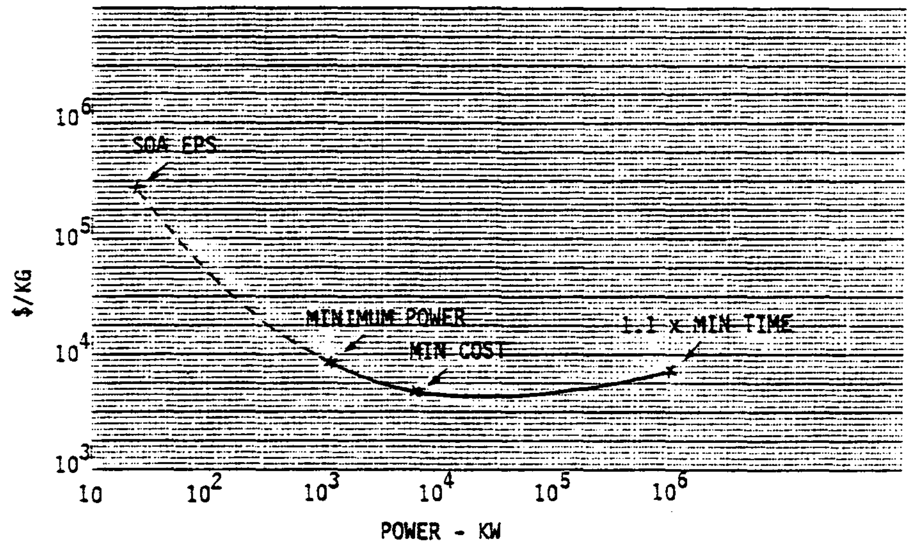


FIGURE 5-10 Cost Relationships of System Design Points (Group 5)

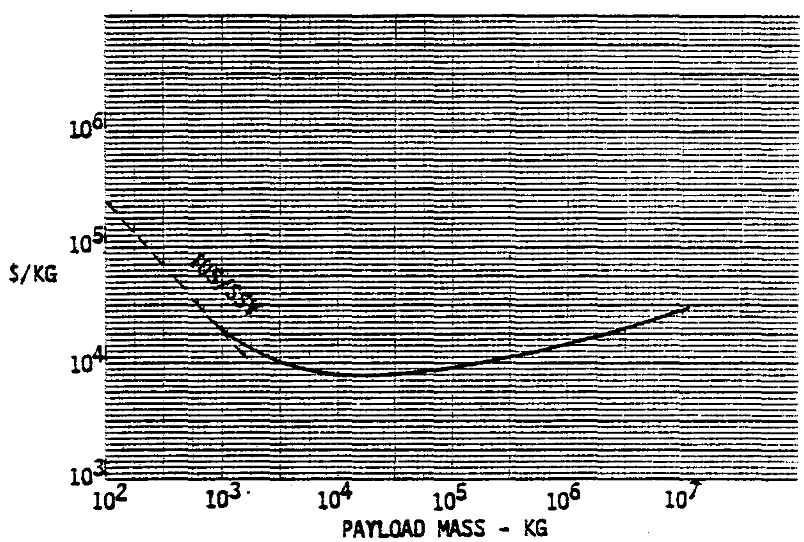


FIGURE 5-12 SOA $/kg Variation with Payload Mass

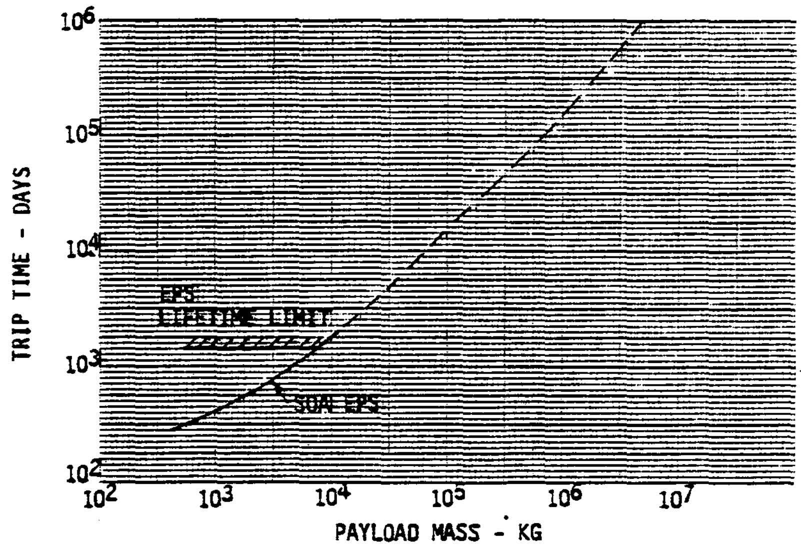


FIGURE 5-13 SOA Trip Time Variation with Payload Mass

delivery charges (last column) have been plotted against payload mass in figure 5-12. Here, the "per-kg" cost is seen to decrease with larger payloads up to about 10,000 kg (at which point the capability of the baseline system is saturated), after that the cost penalties associated with longer transfer times begin to dominate - causing specific costs to rise. As a point of comparison, it is noted that the baseline space transportation system (STS - the Space Shuttle and the Inertial Upper Stage) is expected to deliver a maximum payload of 2270 kg to geosynchronous orbit for about $11,300/kg, with increasing specific costs for decreasing levels of utilization - comparable to the EPS in that range of payload masses.

In figure 5-13, the mission time has been plotted against the mass of the payload. It is noted that missions with payloads heavier than about 7000 kg (typically requiring more than about 1400 days to complete) are not possible with the baseline 25 kw EPS system. Above this point, the SOA 15,000 hour lifetime limit is exceeded. This is the primary technical (as opposed to cost-effectiveness) limit, that will hinder application of the baseline EPS to the more ambitious missions. One way around this limitation is to increase the system size (add more solar array and engine systems), and this approach is equivalent to adopting one of the other three design philosophies (see sections 5.3, 5.4 and 5.6).

Another way around this lifetime limitation is to postulate a system wherein sufficient spare (back-up) engine systems are provisioned so that (with suitable duty cycle management) the utilization time of each individual component just equals its expected lifetime. The required redundancy factor is displayed in figure 5-14. This "sparing" philosophy was modeled by altering the factor α_{EPS}, thus increasing the mass of the electric propulsion system as shown in figure 5-15. The detail costs for each mission are tabulated in figure 5-16. Figure 5-17 illustrates the fact that the "burn-time" for the individual engine systems is restricted to be no more than the SOA lifetime. The increased EPS mass inherent in this approach increases the EPS component of mission cost and also slightly lenthens the required delivery time (increases that component of cost also). The resulting specific

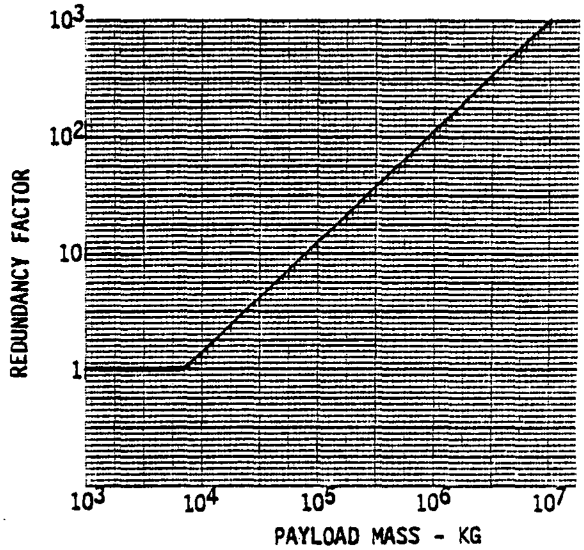


FIGURE 5-14 Redundancy Factor for SOA EPS

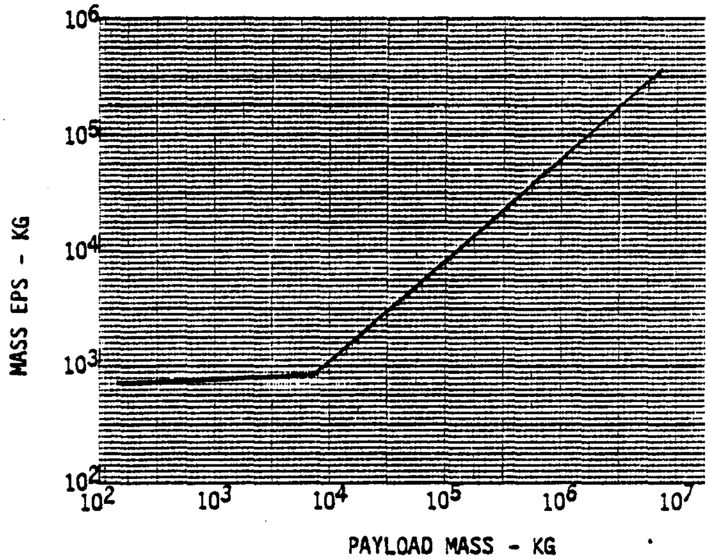


FIGURE 5-15 SOA EPS Mass Variation with Payload Mass

	MISSION NAME	MISSION TIME (DAYS)	NUMBER OF THRUSTERS	COSTS ($M) EPS	SA	LAUNCH	TRIP	PROPELLANT	SCAR	TOTAL $/KG
1	Tethered Satellite	436	8	9.893	8.760	1.722	5.967	.005	0.	37358
2	Nuclear Waste Disposal	730	8	10.272	8.750	4.496	9.991	.015	0.	10316
3	Utility Load Management Satellite	781	8	10.265	8.750	4.265	18.177	.013	.325	13061
4	Earth's Magnetic Tail Mapper	262	8	9.844	8.760	1.589	3.966	.007	.049	64542
5	Earthwatch	613	8	10.742	8.750	7.038	13.100	.014	.260	6139
6	Astronomical Telescope	531	8	9.923	8.760	2.038	25.101	.007	1.137	52176
7	Nuclear Fuel Location System	445	8	9.992	8.760	2,427	7.023	.008	.072	20787
8	Global Search & Rescue Locator	256	8	9.924	8.750	1.898	4.620	.006	.130	27723
9	Geosynchronous-Based Satellite Maint.	81	8	9.942	8.750	1.892	1.616	.004	.216	21741
10	Electronic Mail Transmission	1087	10.33	12.136	8.750	10.305	181.379	.028	2.794	23669
11	Multi-National Air Traffic Control Radar	550	8	10.043	8.760	2.701	7.767	.008	.014	17226
12	Space Based Radar (Near Term)	320	8	10.382	8.760	4.565	8.984	.007	.488	8294
13	Near-Term Navigation Concept	330	8	9.896	8.750	1.793	10.160	.006	.685	43021
14	Technology Development Platform	838	8	10.248	8.750	4.235	17.893	.013	.260	13398
15	Personal Communications Wrist Radio	2884	15.67	14.829	8.760	15.531	202.610	.041	1.949	17401
16	Orbiting Deep Space Relay Station	1386	8	10.884	8.750	8.370	41.538	.021	.552	9349
17	Gravity Gradient Explorer	1191	8	10.626	8.750	6.192	16.299	.019	0.	8358
18	Soil Surface Texturometer	648	8	10.133	8.750	3.276	14.454	.009	.292	15980
19	GSO Communications Platform	1711	9.37	11.596	8.750	9.345	183.504	.026	3.172	26389
20	Space Based Radar (Far Term)	333	8	10.814	8.750	7.320	10.929	.013	.650	5497
21	Personal Navigation Wrist Set	2767	15.14	14.621	8.760	15.105	90.898	.040	.650	9564
22	Marine Broadcast Radar	1421	8	10.771	8.750	7.760	57.590	.021	.910	12806
23	Geosynchronous Space Station	3333	18.24	16.082	8.760	18.197	122.292	.047	.780	10070
24	Orbiting Lunar Station	4120	34.18	21.710	8.750	26.566	170.916	.084	.942	10361
25	Space Construction Facility	144480	665.80	172.240	8.750	636.060	87817	1.290	20.150	35462
26	Power Relay Satellite	5402	30.01	20.967	8.750	29.992	112.881	.074	.234	6285
27	Iceberg Dissipator	295470	2422.56	265.286	8.750	657.668	18202	4.304	1.625	10937
28	SPS Pilot Plant	66544	364.07	88.647	8.750	340.408	96559	.736	48.75	285430
29	Satellite Power System	2.4426×10^6	13360	767.801	8.750	145.362	2.37×10^6	21.395	32.60	189998
30	SPS Orbit Transfer Recovery	65955	294.6	78.204	8.750	281.672	1249	.605	.293	5884

FIGURE 5-16 State-of-the-Art EPS Performance with Redundancy

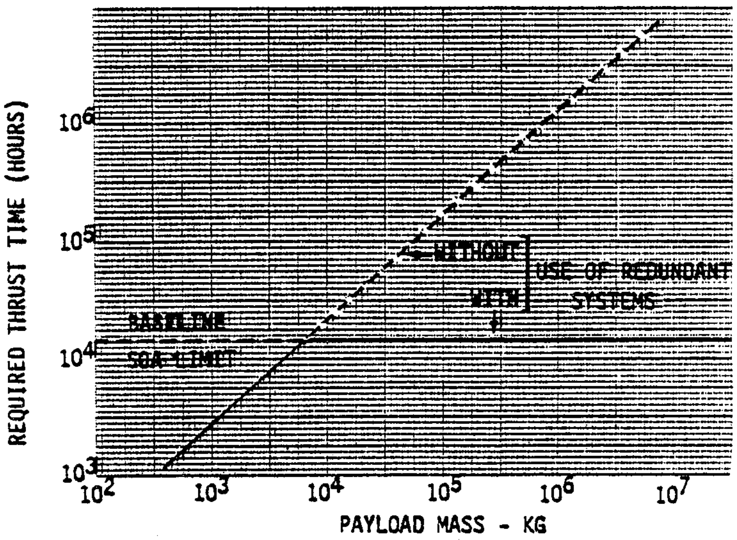


FIGURE 5-17 SOA Thruster On-Time Variation with Payload Mass

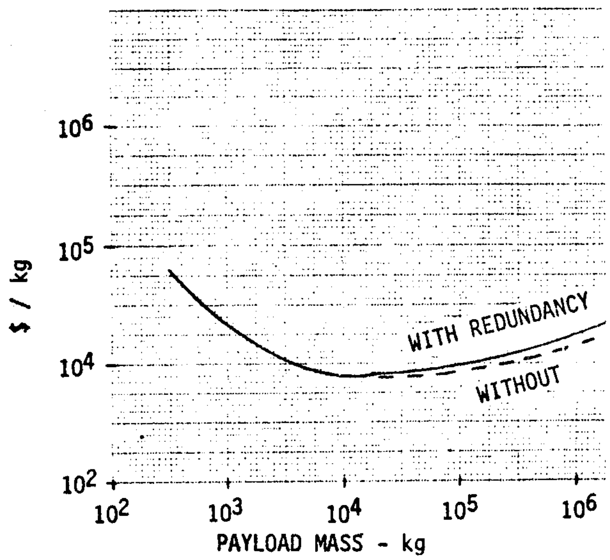


FIGURE 5-18 Delivery Costs for Baseline (SOA) EPS

cost is plotted against payload mass in figure 5-18, along with the baseline curve. It can be seen that while all missions have now been made "physically do-able," there has been no improvement in cost performance of the SOA system - in fact, costs have increased slightly. Clearly, systems larger than the 25 kw baseline will be required for the mid-to-far term missions.

Figure 5-19 illustrates the proportional relationships of the various components of mission transportation costs for representatives of each of the mission groups. (Use of redundancy to insure mission realizability is presumed.) The effect of the very long mission time is obvious.

5.3 MINIMUM POWER SYSTEM DESIGN POINTS

For the studies to be discussed in this section, it was assumed that the overriding program concern was the minimization of the size of the EPS power source. (The motivation was to provide diversity in the design conditions being examined, but conditions resulting in a limitation in the nation's solar array production capability might make such a philosophy desirable.) The minimum power condition is realized when the engine system lifetime (L) is just equal to the required (average) utilization time for that particular mission. By suitable rearrangement of the modeling equations (see section 4), the minimum power can be expressed as:

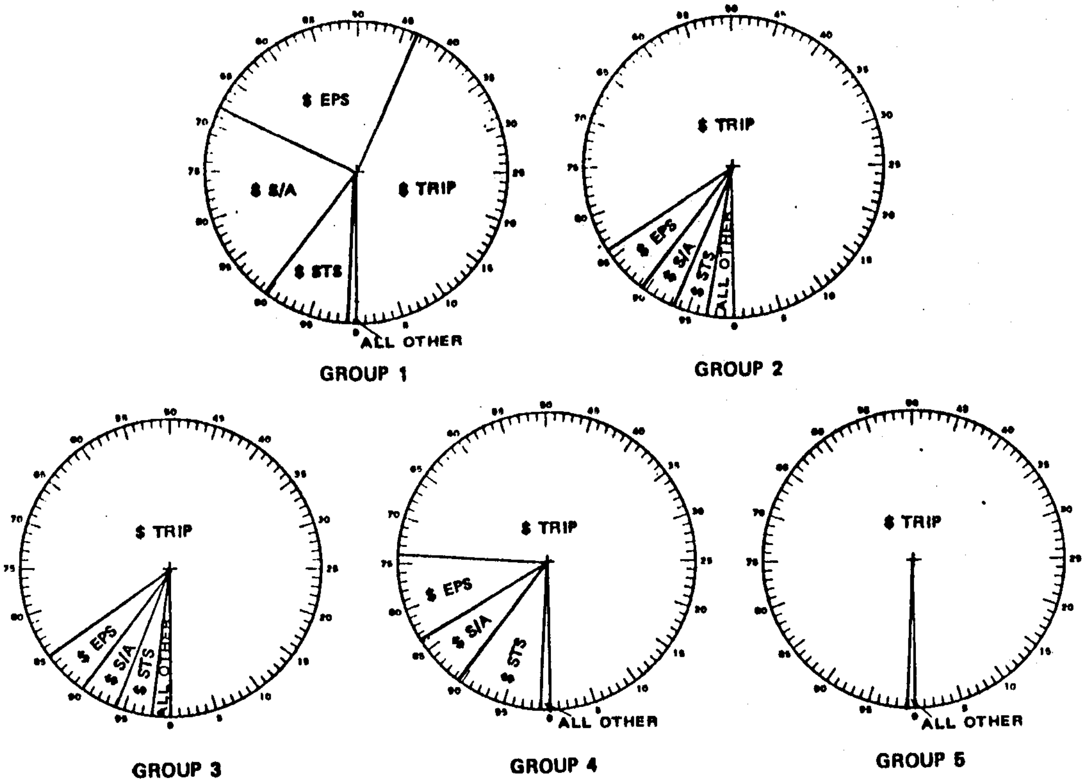


FIGURE 5-19 **Components of Transportation Costs - Baseline (SOA) System with Redundancy**

$$P_{MIN} = \frac{M_K}{\dfrac{L}{F_2\left(e^{V_K/I_{SP}} - 1\right)\left(I_{SP}^2 + K_2\right)} - \alpha_T} \qquad (5\text{-}1)$$

where:

$$M_K = (1 + \alpha_{STR}) M_{PLD} + M_{AV} \qquad (5\text{-}2)$$

$$F_2 = \frac{G_0^2}{2K_1(1-R)} \qquad (5\text{-}3)$$

$$V_K = \frac{\Delta V(1+D)}{G_0\,(1-S)} \qquad (5\text{-}4)$$

$$\alpha_T = \alpha_{EPS} + \alpha_{SA} \qquad (5\text{-}5)$$

While from this expression, it is clear that system lifetime influences the minimum power level in a straight-forward manner, it is also seen that the EPS specific impulse has an effect. As shown in figure 5-20, each mission will have an optimum specific impulse for minimum power. The

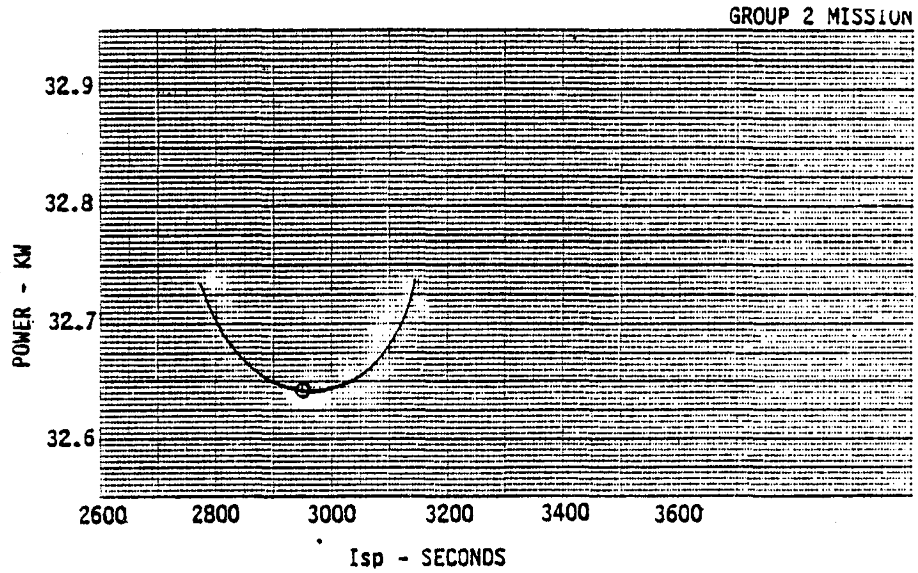


FIGURE 5-20 Minimum Power Mission - Isp Dependence

consideration of the minimum power design condition was directed toward uncovering any shifts in the optimum I_{SP} that might exist across the mission set.

The minimum power conditions were established for each member of the mission set, under the assumption of a 15,000 hour system lifetime. Figure 5-21 summarizes the results of this analysis. The mission minimum power level

	MISSION NAME	POWER LEVEL (kW)	I_{sp} (SEC)	MISSION TIME (DAYS)	THRUSTER TIME (HRS)	COSTS ($M)						TOTAL $/KG
						EPS	SA	LAUNCH	TRIP	PROPELLANT	SCAR	
1	Tethered Satellite	2.63	2900	2316	15000	5.826	9.845	.967	31.708	.003	0.	66014
2	Nuclear Waste Disposal	15.19	3000	1106	15000	8.688	6.473	4.138	16.137	.014	0.	10292
3	Utility Load Management Satellite	11.92	2950	1462	15000	8.109	4.339	3.820	34.025	.012	.326	15822
4	Earth's Magnetic Tail Mapper	3.27	3100	948	15000	5.897	1.224	.745	14.344	.004	.049	69339
5	Earthwatch	14.29	2850	1019	15000	9.062	5.164	6.738	21.762	.014	.260	6614
6	Astronomical Telescope	4.68	3000	1814	15000	6.297	1.742	1.301	85.691	.005	1.137	106860
7	Nuclear Fuel Location System	5.46	2950	1462	15000	6.547	2.028	1.749	23.097	.006	.071	24632
8	Global Search & Rescue Locator	2.98	2900	1322	15000	5.943	1.114	1.165	23.164	.004	.130	34637
9	Geosynchronous-Based Satellite Maint.	1.94	2800	642	15000	6.747	.731	1.174	12.768	.003	.211	20012
10	Electronic Mail Transmission	32.64	2950	1462	15000	12.189	11.177	10.461	140.604	.029	2.795	19467
11	Multi-National Air Traffic Control Radar	5.73	2925	1814	15000	6.668	2.128	2.053	25.603	.006	.015	21455
12	Space Based Radar (Near Term)	6.64	2800	1050	15000	7.277	2.457	4.016	29.460	.007	.488	10927
13	Near-Term Navigation Concept	3.23	2950	1462	15000	5.969	1.208	1.035	45.233	.004	.585	74517
14	Technology Development Platform	12.86	2975	1462	15000	8.257	4.668	3.809	31.223	.012	.260	15609
15	Personal Communications Wrist Radio	49.85	2950	1462	15000	14.973	16,292	15.976	104.077	.042	1.950	10951
16	Orbiting Deep Space Relay Station	23.54	2925	1462	15000	10.668	8.276	8.361	43.833	.022	.652	9561
17	Gravity Gradient Explorer	20.59	3000	1409	15000	9.848	7.301	6.035	19.287	.018	0.	8498
18	Soil Surface Texturometer	7.28	2900	1814	15000	7.089	2.690	2.691	40.486	.008	.292	23055
19	GSO Communications Platform	29.48	2975	1462	15000	11.627	10.186	9.431	156.756	.026	3.172	23317
20	Space Based Radar (Far Term)	12.29	2800	638	15000	8.808	4.470	6.974	20.953	.013	.650	5981
21	Personal Navigation Wrist Set	48.44	2950	1462	15000	14.759	15.892	15.525	48.036	.014	.650	6978
22	Marine Broadcast Radar	24.21	2950	1462	15000	10.654	8.494	7.760	59.244	.022	.910	12998
23	Geosynchronous Space Station	58.63	2950	1462	15000	16.267	18.734	18.788	53.640	.050	.780	6561
24	Orbiting Lunar Station	112.87	3050	964	15000	22.265	31.854	27.941	39.991	.086	.942	5569
25	Space Construction Facility	2097.00	2800	1736	15000	172.855	162.855	631.703	1055.126	1.393	20.150	818
26	Power Relay Satellite	97.26	2950	1462	15000	21.299	28.380	31.156	30.102	.078	.234	4045
27	Iceberg Dissipator	8079.00	3025	976	15000	273.091	274.574	642.558	60.101	4.534	1.625	718
28	SPS Pilot Plant	1195.00	2950	1462	15000	90.675	128.023	356.261	2121.574	.784	48.750	8077
29	Satellite Power System	13399.00	2950	1462	15000	775.639	507.943	137.939	1421.055	22.824	32.499	232
30	SPS Orbit Transfer Recovery	966.44	2950	1519	15000	79.980	116.234	295.296	33.904	.644	.295	1914

FIGURE 5-21 Minimum Power Case - EPS Performance

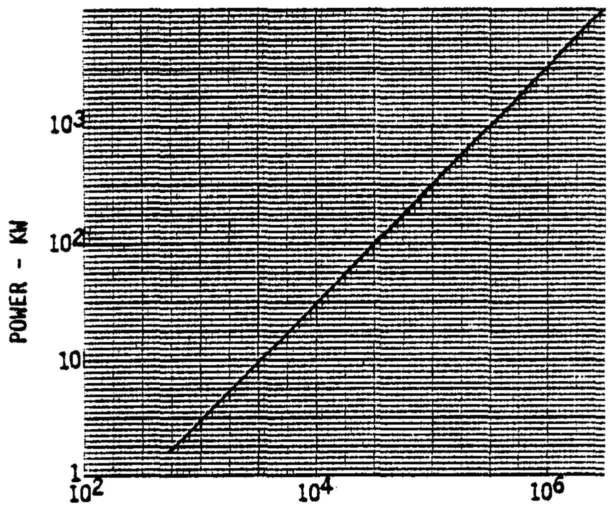


FIGURE 5-22 Mission Dependence of Minimum Power Level

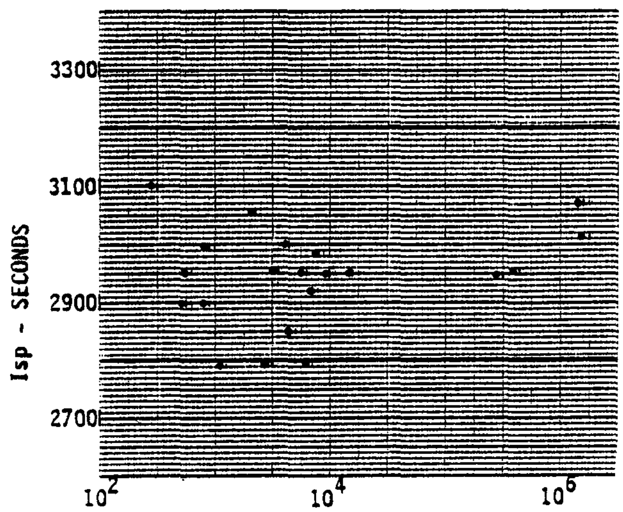


FIGURE 5-23 Mission Dependence of EPS Specific Impulse

and the corresponding specific impulse at which that minimum power occurs is plotted against payload mass in figures 5-22 and 5-23. As expected there is a direct relationship between mass and minimum power. There is no such direct correlation between the optimum specific impulse and the payload mass (it is rather more dependent on the mission energy requirement), but it is noted that all points fall in a narrow band centered about the current technology development point.

As mentioned above, the minimum power point is influenced by the system lifetime assumption. In this study, minimum power points were calculated for lifetimes from 10,000 to 50,000 hours. Within this range, the specific impulse at which minimum power occurs was not found to be affected by lifetime. This allows the conclusion that current technology development efforts are in the proper I_{SP} region, should power source minimization become of prime concern.

Figure 5-24 illustrates the relationships of the contributors to transportation costs across the mission set for the minimum power design point. The trip time charges dominate in all cases because of the concentration on reducing the size of the power source.

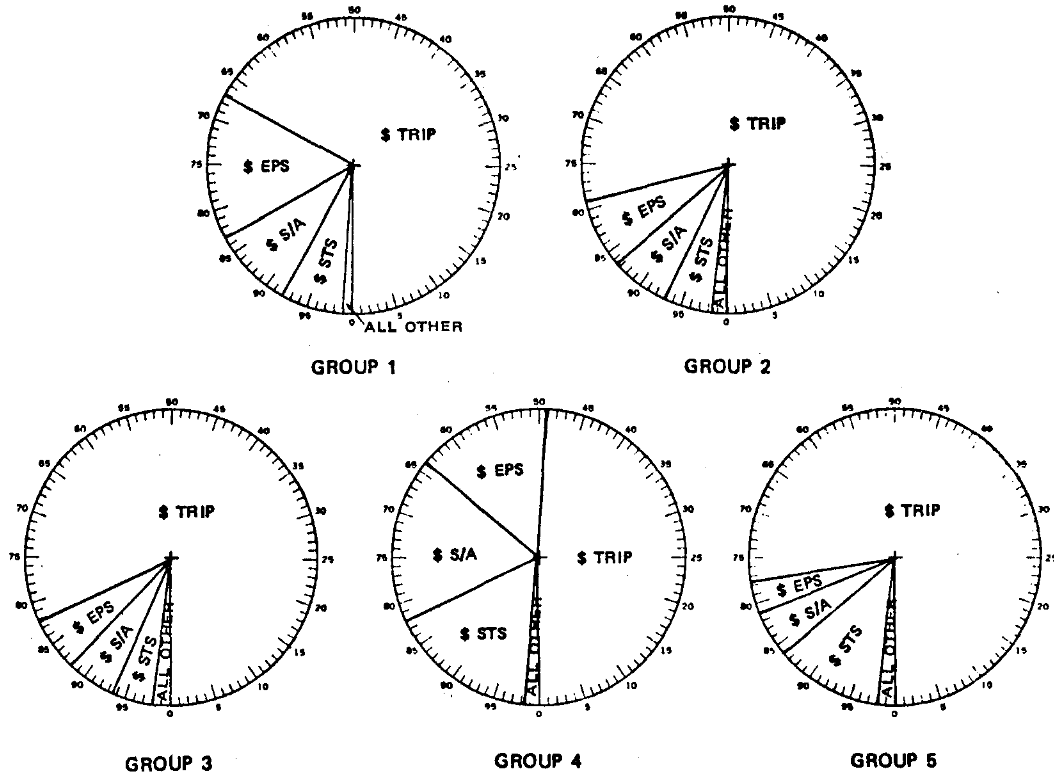


FIGURE 5-24 **Components of Transportation Costs - Minimum Power Design Point**

5.4 TIME-CONSTRAINED SYSTEM DESIGN POINTS

In this section, the technology drivers for the trip time-constrained design point will be discussed. This condition assumes that the duration of the low-thrust transfer phase is of major concern, such as might be the case if some of the payload systems had a limited lifetime (e.g. cryogenic coolers, photographic film, etc.) or if time-related cost factors were found to be even higher than those assumed in this study. Two cases will be discussed; the absolute minimization of transport time; and, the achievement of some pre-ordained, fixed, mission time.

5.4.1 Idealized Minimum Trip Times

With suitable manipulation, the equations of section 4 yield an expression for mission time that is of the form:

$$T = f_1 f_2 (\alpha_T + \frac{M_K}{P}) \qquad (5\text{-}6)$$

Obviously, the first term ($f_1 f_2 \alpha_T$) represents an absolute minimum transportation time-obtainable by the application of infinite power. (This is also equivalent to reducing the payload to zero.) The factors of this term are:

$$f_1 = \frac{(G_o I_{SP})^2 (1+\phi(1+T_D))}{2\eta\ (1-R)} \tag{5-7}$$

$$f_2 = e^{\left[\frac{\Delta V\ (1+D)}{G_o I_{SP}\ (1-S)}\right]} - 1 \tag{5-8}$$

$$\alpha_T = \alpha_{EPS} + \alpha_{SA} \tag{5-9}$$

It is noted that neither factor contains any payload-dependent parameters (M_{PL}, $\alpha_{SCAR} + \alpha_{STR}$), any system descriptors (α_{ADP}, M_{AV}, L, and of course, P_o), or any cost functions (γ's). The idealized minimum time is only a function of the trajectory requirements (ΔV, R, D, S and ϕ) and the characteristics of the electric propulsion system (I_{SP}, α_{EPS}, $\eta(=f(I_{SP}))$, T_D, and α_{SA}). This suggests that the analysis of the effects of the EPS technology parameters on mission time can be done generically (without application to a specific mission). This approach was followed, and yields insight into desirable technology directions should the prime factor in mission/transportation system design be determined to be short trip times.

Figure 5-25 shows the minimum transfer time for a LEO-to-GEO trajectory as a function of the specific impulse of the EPS. The curve labeled "REAL" represents a baseline (SOA) system, while the others show the effects of halving the specific weights of either the solar array, or the electric propulsion system, or both. The arrows point out the minima of these minimum time curves, which are seen to be rather insensitive to I_{SP}. Another way of looking at the dependence on specific weight is shown in figure 5-26, wherein specific impulse was held constant at 3000 seconds, and the ΔV = 5760 m/s curve represents the LEO-to-GEO transfer. The strong and direct relationship is obvious. As can be seen from equation 5-9, the EPS and array specific weights are equally important. A simple economic trade can thus be performed to determine whether it is more advantageous to expend development effort on reducing EPS or array weights.

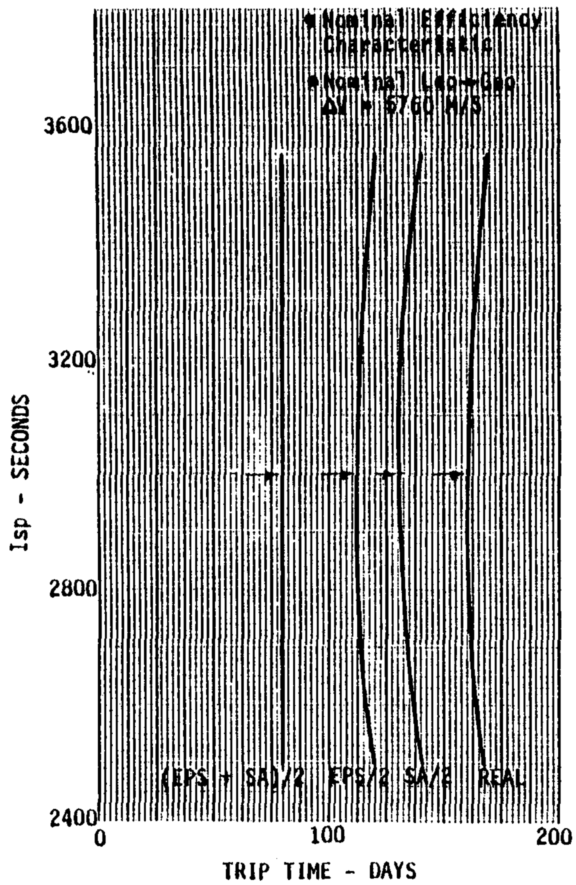


FIGURE 5-25 Idealized Minimum Trip Time LEO→GEO Transfer

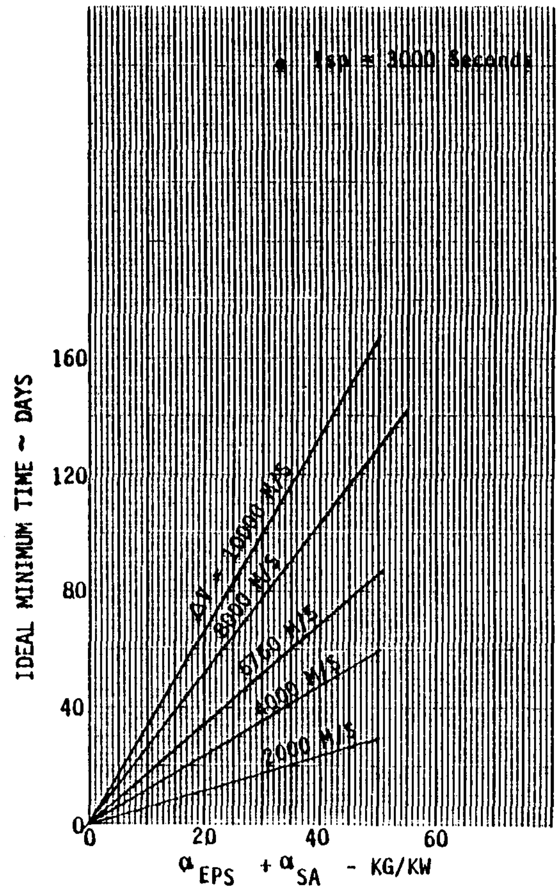


FIGURE 5-26 Minimum Trip Time Variation with Specific Masses

Figure 5-27 shows the dependence of minimum transfer time on the required velocity increment of the desired trajectory. For real missions, the total energy requirement is, of course, a function of the performance loss factors (occultation, steering, radiation degradation and drag) which depend on the exact trajectory characteristics, as well as the ΔV. For the baseline mission set for this study, the requirements all fall within the shaded band shown in the figure. If the loss factors are set to zero (e.g., an NEP system), the lower single curve results. A LEO-to-GEO trajectory was analyzed (see figure 5-28) with all the penalty factors set equal to zero (curve marked IDEAL) and for the nominal case (marked REAL). The location of the "minimum of the minimum" did not change, hence the IDEAL curves were used for the subsequent minimum time studies in order to avoid dealing with bands of data.

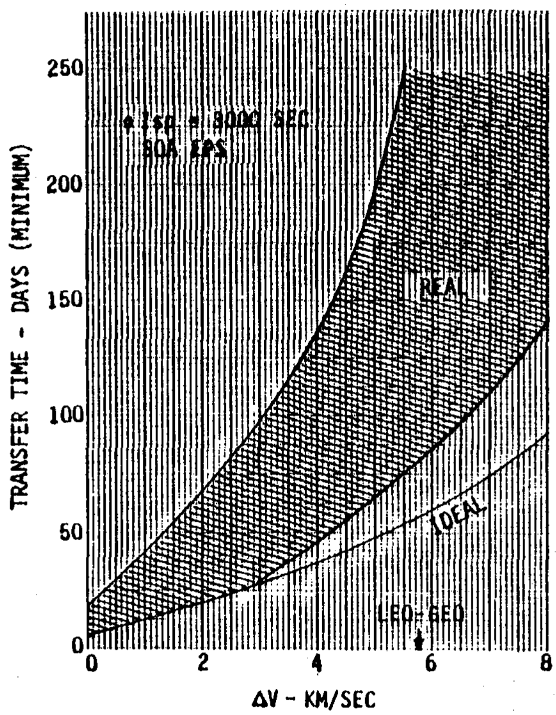


FIGURE 5-27 Minimum Trip Time Variation with Mission Energy

The major factor that determines the "best" specific impulse for minimum time transfers is the trajectory velocity increment. As shown in figure 5-29, the optimum varies from about 2750 seconds to 3150 seconds over the range of interest (the study mission set encompasses ΔV's from 1500 to 9000 meters/ second.) With this in mind, and considering the flatness of the curves, the SOA reference point of 3000 seconds thus seems a good choice for future development efforts, from a minimum time standpoint.

A similar conclusion was reached from a study of the effects of efficiency on minimized trip times. Here, both the K_1 and the K_2 factors(corresponds to the "scaling" and "translation" cases, respectively, to be discussed in section 5.6 - see figures 5-107 and 5-109), in the curve (equation 4-14) were varied by about 20%. Figure 5-30 shows that changing the slope of the efficiency curve has no discernable effect on the minimum-time specific impulse. Translating the system efficiency characteristic, on the other hand, does influence the value of the "best" I_{SP}, as can be seen in figure 5-31. However, the variations are minimal ($\pm$300 seconds) and are centered around 2950 seconds, which is very close to the state-of-the-art technology.

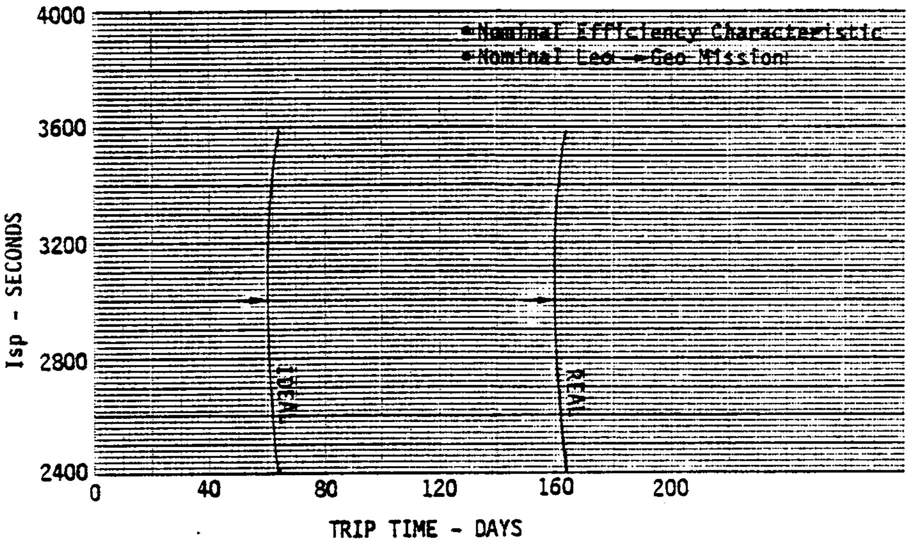


FIGURE 5-28 Effect of Performance Loss Factors

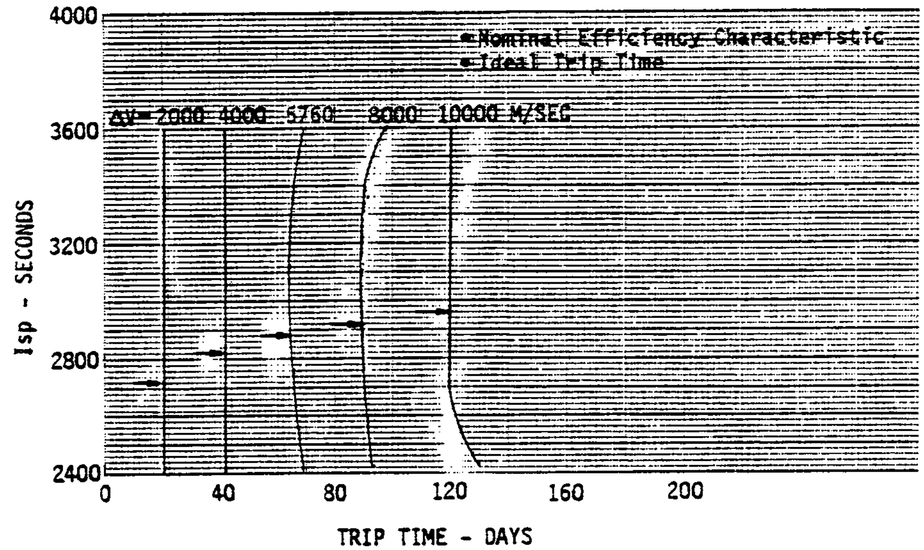


FIGURE 5-29 Minimum Time Isp Selection as a Function of Mission Requirements

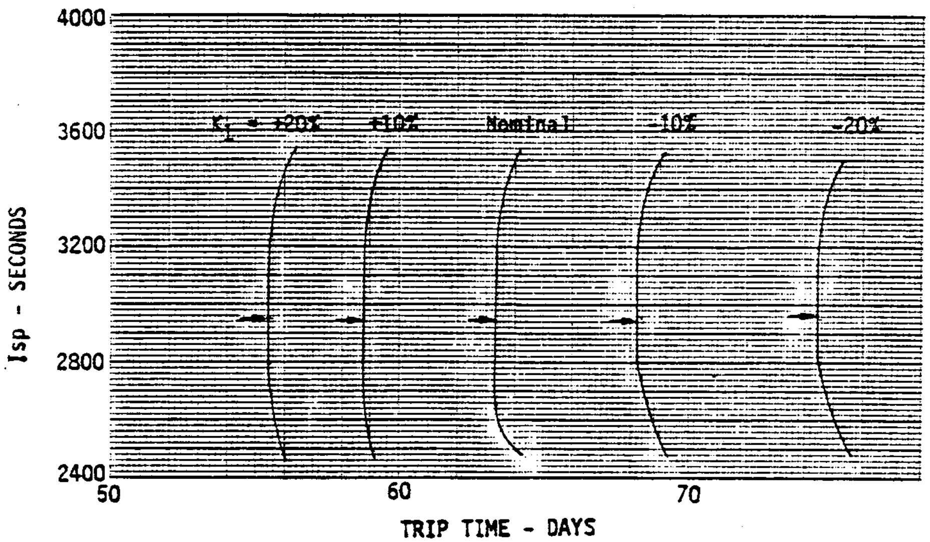


FIGURE 5-30 Impact of Scaling Efficiency Function on Isp for Minimum Time

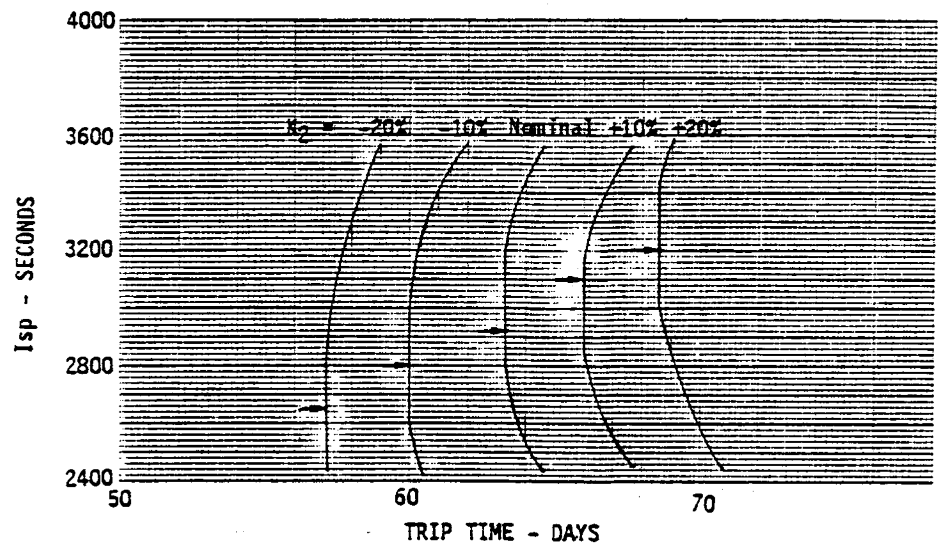


FIGURE 5-31 Impact of Efficiency Increments on Isp for Minimum Time

	MISSION NAME	POWER LEVEL (kW)	I_{sp} (SEC)	MISSION TIME (DAYS)	THRUSTER TIME (HRS)	COSTS ($M) EPS	SA	LAUNCH	TRIP	PROPELLANT	SCAR	TOTAL $/KG
1	Tethered Satellite	249	2913	215	1613	31.609	65.678	9.409	3.244	.024	0.	141793
2	Nuclear Waste Disposal	1034	3028	147	2016	72.327	119.947	41.167	2.218	.114	0.	72545
3	Utility Load Management Satellite	980	2975	160	1699	70.083	117.024	37.736	4.095	.093	.325	71674
4	Earth's Magnetic Tail Mapper	192	2700	158	2540	27.263	46.891	8.319	2.632	.034	.049	227167
5	Earthwatch	2898	2855	71	1052	103.607	166.287	68.371	1.666	.116	.260	50816
6	Astronomical Telescope	310	3017	236	2129	35.808	63.765	12.368	12.279	.038	1.137	139317
7	Nuclear Fuel Location System	448	2961	160	1698	44.323	78.897	17.305	2.780	.046	.072	105458
8	Global Search & Rescue Locator	327	2825	114	1328	36.925	65.839	12.186	2.190	.030	.130	128900
9	Geosynchronous-Based Satellite Maint.	365	2795	34	796	39.318	70.193	12.724	.736	.021	.211	119498
10	Electronic Mail Transmission	2685	2982	160	1697	127.301	180.033	102.812	16.913	.232	2.795	47262
11	Multi-National Air Traffic Control Radar	551	2925	174	1570	49.967	88.123	20.775	2.695	.049	.014	95073
12	Space Based Radar (Near Term)	1130	2815	55	867	76.239	124.895	39.684	1.712	.057	.487	60769
13	Near-Term Navigation Concept	265	2961	160	1698	32.730	57.881	10.238	5.445	.028	.585	147460
14	Technology Development Platform	464	3015	193	2048	45.293	80.373	19.874	4.527	.056	.260	48667
15	Personal Communications Wrist Radio	4102	2989	160	1698	163.784	212.581	156.854	12.528	.341	1.950	39074
16	Orbiting Deep Space Relay Station	2309	2947	141	1496	116.371	169.403	86.487	4.647	.182	.552	50352
17	Gravity Gradient Explorer	1479	3000	172	1894	89.373	140.621	58.067	2.587	.147	0.	58159
18	Soil Surface Texturometer	694	2900	168	1516	57.222	99.183	26.172	4.128	.060	.293	80977
19	GSO Communications Platform	2425	2982	160	1698	119.827	172.811	92.950	18.870	.211	3.172	49736
20	Space Based Radar (Far Term)	1859	2830	36	847	102.355	154.935	65.916	1.314	.096	.650	46466
21	Personal Navigation Wrist Set	3986	2987	160	1698	161.011	210.241	151.579	5.782	.333	.650	38941
22	Marine Broadcast Radar	1992	2980	160	1698	106.605	159.441	76.458	7.131	.176	.910	52347
23	Geosynchronous Space Station	4825	2989	160	1698	180.387	226.216	182.426	6.457	.396	.780	36161
24	Orbiting Lunar Station	6266	3005	145	2297	210.750	249.702	248.308	6.637	.657	.942	32443
25	Space Construction Facility	647000	2740	50	471	3351.714	1327.236	89.803	33.430	12.008	20.160	1943
26	Power Relay Satellite	8007	2996	160	1690	243.916	273.662	293.693	3.623	.632	.234	29664
27	Iceberg Dissipator	478280	3015	136	2107	2798.390	1191.801	100.484	9.240	36.282	1.625	2364
28	SPS Pilot Plant	98472	2898	160	1698	1089.616	678.285	452.961	255.387	6.789	48.750	7446
29	Satellite Power System	3614600	2998	160	1698	9354.924	2449.088	14.102	171.061	196.439	32.499	977
30	SPS Orbit Transfer Recovery	137390	2900	160	1633	1328.101	763.954	356.255	3.927	8.946	.292	8951

FIGURE 5-32 110% Minimum Time - EPS Dependence

Since it takes infinite power to attain the theoretical minimum trip time, the (hardware) costs would become infinite for that case also. Thus in figure 5-32, the costs are tabulated for a condition representing 10% more than the absolute minimum time (the theoretical minimum is indicated for completeness) for each member of the overall mission set. As can be seen from figure 5-33, the trip time-associated costs have been reduced to a small fraction of the total in all cases, as would be expected for a minimized transport time goal. It is also noted that the solar array now dominates the mission costs, indicating that technology development to reduce the cost of this component would be fruitful in a world in which it is desired to keep mission durations as short as possible.

5.4.2 Fixed Non-Minimum Trip Times

Because of the impracticality (both technically and from a cost-effectiveness standpoint) of implementing the absolute minimum time design condition, this study also examined the implications of constraining the trip time to be some (short) pre-ordained value. The system power level required for any fixed mission time can be calculated from equation 5-1, with the variable L replaced by T_0 (the required mission time). As previously noted,

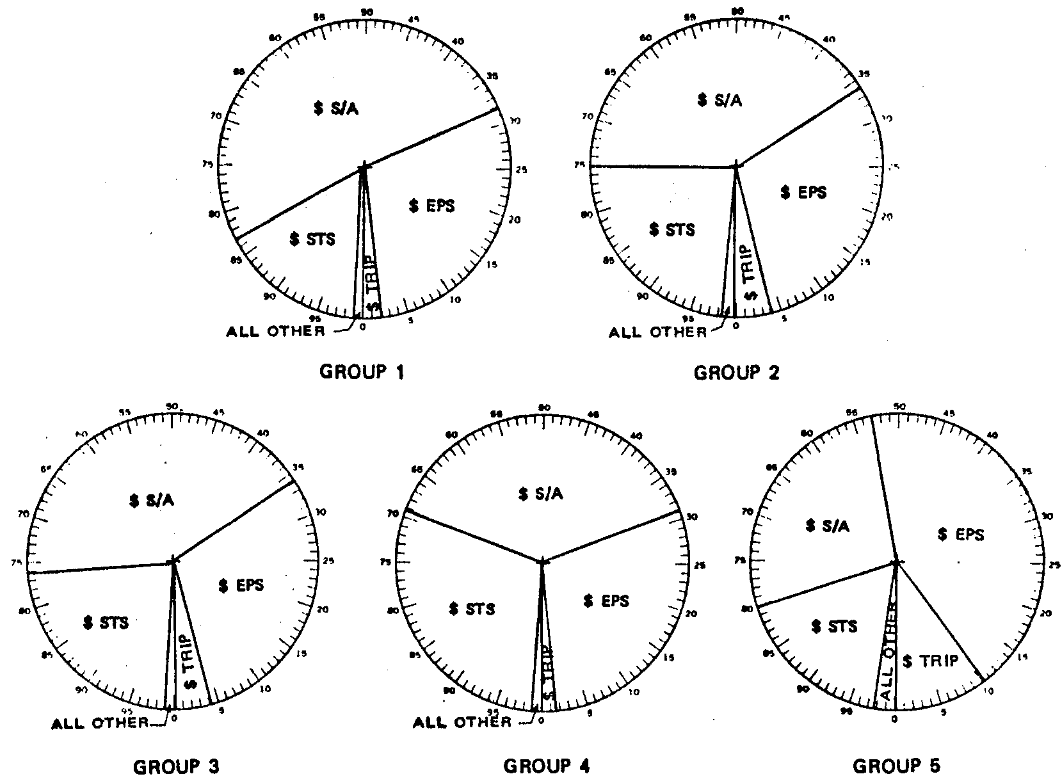


FIGURE 5-33 Components of Transportation Costs - 110% Minimum Time Design Point

there is an "optimum" value of EPS specific impulse which results in a minimum power requirement. This can be seen in figures 5-34 thru 5-38, which display the P_0-I_{SP} space for the representative mission of each of the five groups. As might be expected, the minimum power requirement is a direct function of the size of the mission payload (illustrated by figure 5-39) The "best" value of EPS specific impulse decreases slightly with larger payloads (actually this results from increased trip time charges, as will be explained in the next section), but is not impacted by the chosen duration of the mission (see figure 5-40). The total range of "best" $I_{SP's}$ for fixed-time missions is from 2900 - 3100 seconds with nominal values for all other EPS technology parameters. This coincides with the thrust of present-day developmental efforts.

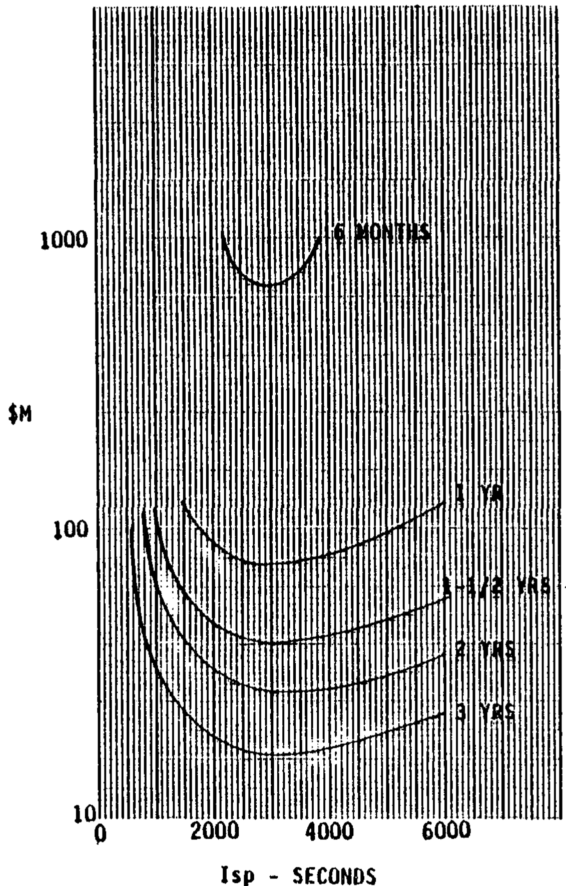


FIGURE 5-34 Isp Optimization for Fixed Mission Times (Group 1)

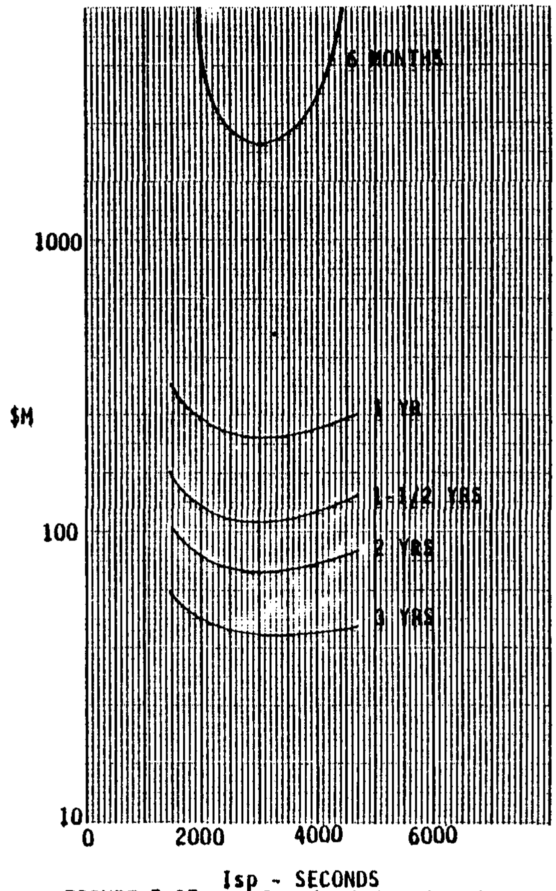


FIGURE 5-35 Isp Optimization for Fixed Mission Times (Group 2)

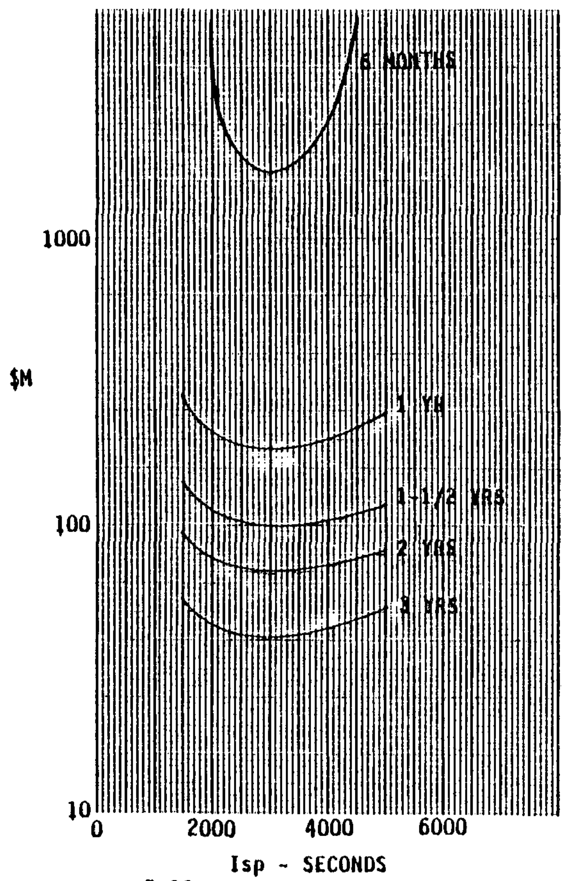


FIGURE 5-36 Isp Optimization for Fixed Mission Times (Group 3)

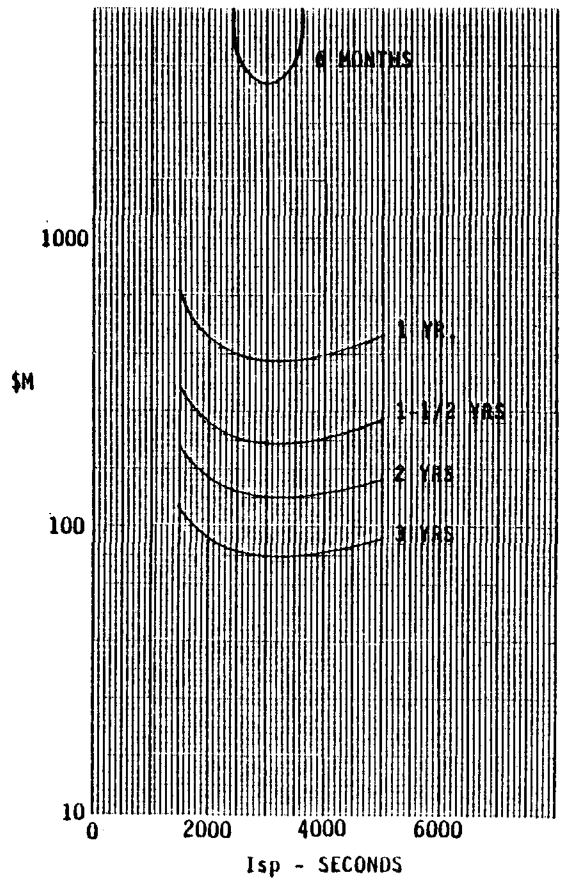


FIGURE 5-37 Isp Optimization for Fixed Mission Times (Group 4)

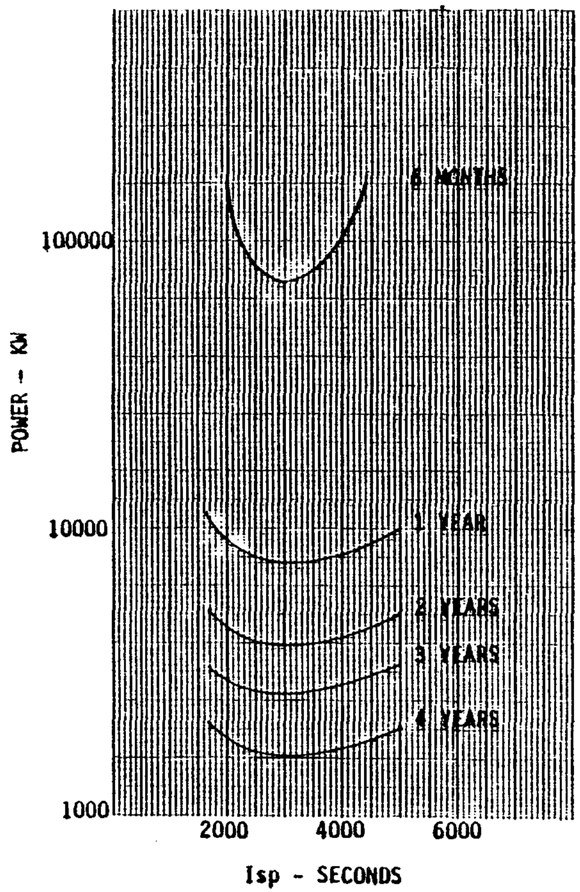


FIGURE 5-38 Isp Optimization for Fixed Mission Times (Group 5)

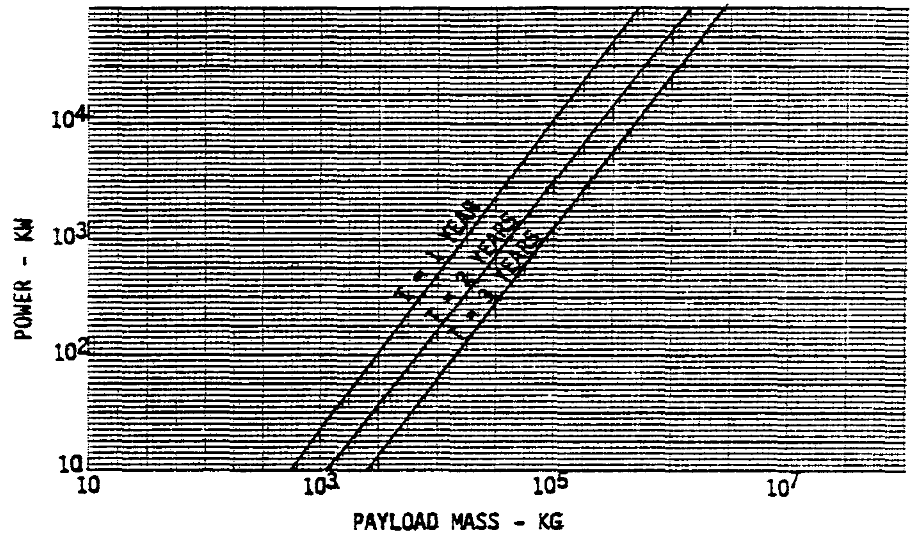


FIGURE 5-39 Power Requirements for Fixed Trip Times

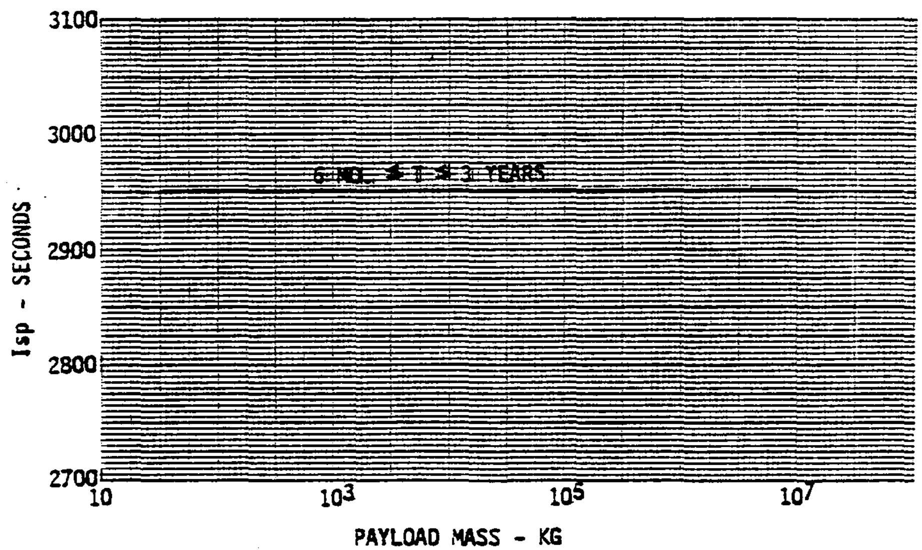


FIGURE 5-40 Optimum Specific Impulse for Fixed Trip Times

5.5 COST-OPTIMUM SYSTEM DESIGN POINTS

The final design condition to be discussed will be that of the cost-optimum solution. Here, the electric propulsion system design point is chosen so as to minimize the total transportation cost - Earth's surface to final destination orbit. This is generally perceived to be the "correct" goal for the development of new space transportation systems.

As shown in figure 5-41, for a fixed specific impulse (3100 seconds in this case), there is an optimum size for the power source. Below that optimum, the system is underpowered and the charges associated with the transportation time duration drive the mission cost up. For higher powered systems, the point of diminishing returns has been reached regarding decreasing trip time, and so increased hardware costs (for the larger solar arrays and engine systems to use that power) cause the mission cost to increase. The graph shows these effects for the delivery of the Geosynchronous Communications Platform (the group 3 representative mission), an 8200 kg payload, with all other EPS parameters fixed at their nominal (SOA) values.

For the same mission, if the size of the power source is held constant at its optimum value of 109 kw, figure 5-42 shows the impact of varying the system specific impulse. Here again, a cost-optimum design point is seen to exist. For lower values of I_{SP}, larger amounts of propellant are required, and this increases the Earth-launch costs, and also decreases the initial acceleration that can be achieved. For constant power systems, vehicle thrust level decreases with increasing specific impulse, thus the trip time duration (and costs) increases above the optimum value of I_{SP}.

By performing a two-dimensional optimization (both power and specific impulse simultaneously), the minimum cost design point was found for each member of the overall mission set. These values, as well as the corresponding components of missions costs, are tabulated in figure 5-43. The sensitivity studies to be described in this section are all "centered" about these design points.

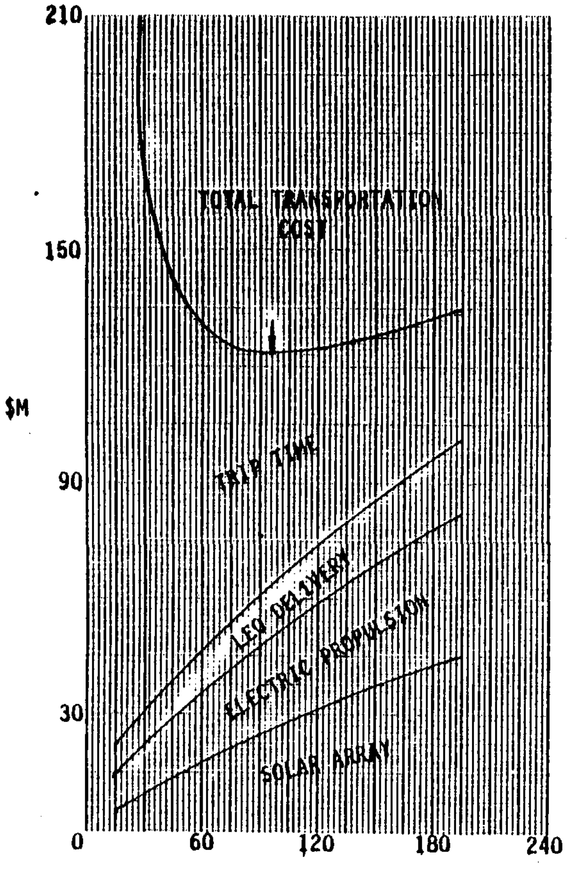


FIGURE 5-41 Cost Optimization of EPS Power

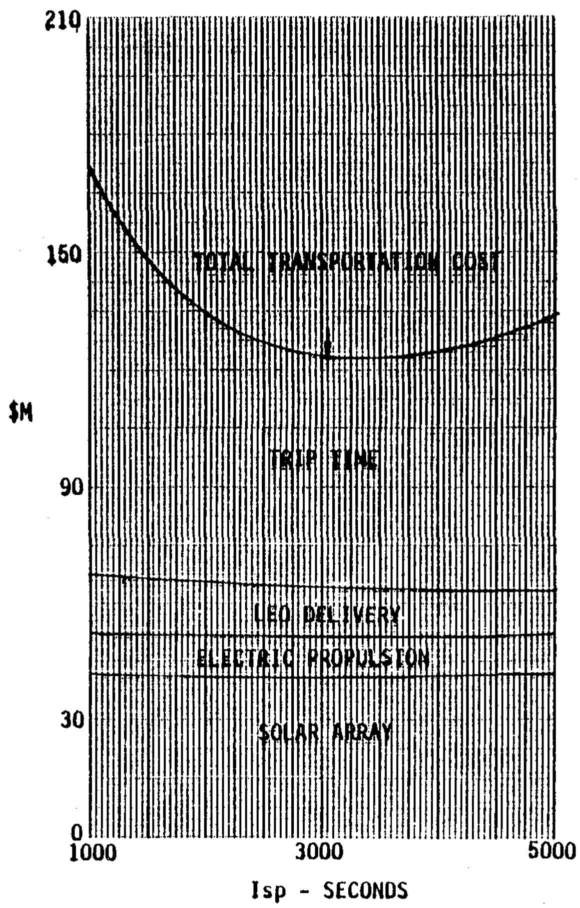


FIGURE 5-42 Cost Optimization of EPS Specific Impulse

	MISSION NAME	POWER LEVEL (kW)	I_{sp} (SEC)	MISSION TIME (DAYS)	THRUSTER TIME (HRS)	COSTS ($M)						TOTAL $/KG
						EPS	SA	LAUNCH	TRIP	PROPELLANT	SCAR	
1	Tethered Satellite	12	2980	675	6062	7.872	4.716	1.314	8.758	.004	0.	32032
2	Nuclear Waste Disposal	19	3240	916	12555	9.486	7.103	4.227	12.022	.014	0.	10107
3	Utility Load Management Satellite	27	3100	736	7815	10.718	9.716	4.334	16.654	.013	.326	13042
4	Earth's Magnetic Tail Mapper	9	3200	446	7625	7.062	3.306	.952	6.745	.004	.005	48318
5	Earthwatch	24	3040	637	9374	10.742	8.750	7.019	13.117	.014	.260	6135
6	Astronomical Telescope	26	3060	520	4300	10.232	9.396	2.097	24.076	.008	1.137	52146
7	Nuclear Fuel Location System	14	3060	668	6857	8.342	6.409	2.066	10.026	.007	.071	19043
8	Global Search & Rescue Locator	10	2980	474	6374	7.640	4.015	1.428	7.728	.004	.130	22905
9	Geosynchronous-Based Satellite Maint.	9	2880	165	3861	6.793	2.588	1.329	4.032	.003	.211	14919
10	Electronic Mail Transmission	108	3100	654	6687	20.739	31.012	12.989	52.944	.033	2.796	13243
11	Multi-National Air Traffic Control Radar	16	3040	763	6304	8.743	6.093	2.419	10.275	.007	.014	16173
12	Space Based Radar (Near Term)	19	2920	403	5760	9.601	7.103	4.417	10.828	.008	.487	8102
13	Near-Term Navigation Concept	15	3020	441	4520	8.410	6.752	1.474	13.091	.005	.585	40421
14	Technology Development Platform	26	3140	814	8347	10.652	9.396	4.256	16.878	.013	.260	13382
15	Personal Communications Wrist Radio	116	3160	721	7352	22.023	32.951	18.085	60.702	.045	1.950	8982
16	Orbiting Deep Space Relay Station	45	3100	834	8849	13.863	15.477	9.057	24.129	.002	.552	8411
17	Gravity Gradient Explorer	27	3220	1120	12298	10.825	9.396	6.167	15.322	.018	0.	8344
18	Soil Surface Texturometer	23	3020	690	5701	9.978	8.425	3.235	14.908	.009	.282	15936
19	GSO Communications Platform	109	3080	513	5294	20.657	31.012	12.114	54.994	.032	3.172	14876
20	Space Based Radar (Far Term)	21	3000	389	8776	10.616	8.097	7.252	11.782	.012	.650	5470
21	Personal Navigation Wrist Set	73	3220	1028	10673	17.639	22.618	16.076	33.804	.040	.650	6664
22	Marine Broadcast Radar	55	3120	734	7788	14.761	17.738	8.727	29.774	.023	.910	10735
23	Geosynchronous Space Station	88	3240	1033	10694	19.379	26.213	19.419	37.898	.047	.780	6287
24	Orbiting Lunar Station	106	3500	1028	16119	21.632	30.351	26.689	42.666	.074	.942	5536
25	Space Construction Facility	9700	2520	416	3573	312.870	294.902	600.764	250.062	1.6792	20.150	692
26	Power Relay Satellite	150	3620	1035	12750	23.457	33.583	30.719	26.430	.065	.234	4306
27	Iceberg Dissipator	10000	2000	825	18762	256.286	260.988	611.916	74.355	6.935	1.625	733
28	SPS Pilot Plant	6400	3280	406	4181	219.418	251.708	478.355	588.979	1.003	48.750	4671
29	Satellite Power System	76500	2960	907	9308	1023.744	621.196	127.869	877.120	24.338	32.500	217
30	SPS Orbit Transfer Recovery	700	4540	2239	35286	56.974	76.829	260.041	79.196	.392	.295	1773

FIGURE 5-43 Cost Optimum Solutions - EPS Performance

The range of the cost-optimum design-points is illustrated by figures 5-44 and 5-45. The system size (or power level) is seen to be a direct function of the mass of the payload as would be expected. The optimum specific impulse, on the other hand, does not exhibit such straight-forward behavior, since it is driven by the mission/payload cost factors and the trajectory loss factors in a rather complex manner. A mild trend toward lower specific impulse with increasing mission difficulty is shown. This is primarily a consequence of the greater payload values that tend to go along with the heavier masses. Since higher payload costs will increase the penalty associated with mission duration (see equation 4-2), and propellant launch costs were assumed to decrease with larger quantities, the optimum electric propulsion systems for large payloads tend toward lower specific impulses, to gain the benefit of the resulting higher accelerations. Figure 5-46 shows that the average thruster "burntime" also increases as the payloads become larger and for several missions approach or exceed the lifetime assumed for current (SOA) technology. Thus, the development of longer-functioning components would be beneficial to the implementation of cost-optimum electric propulsion systems for the far-term missions.

Figure 5-47 shows the trend toward decreased specific transportation costs with increasing payload size that occurs for cost-optimized electric propulsion systems. This is in sharp contrast to the cost trends for the baseline (SOA-25 kw) system (see figure 5-12). For the cost-optimum EPS, the increased hardware costs resulting from the generally larger systems is more than offset by the reduced penalties resulting from shorter mission times. The make-up of these costs can be seen in figure 5-48. The optimization process seems to drive the combined EPS and power source costs toward equality with the trip time costs. The Earth-to-low orbit launch costs are seen to increase (proportionately) with more advanced missions, suggesting the potential payoff for the development of advanced systems, such as the oft-studied Heavy-Lift Launch Vehicle.

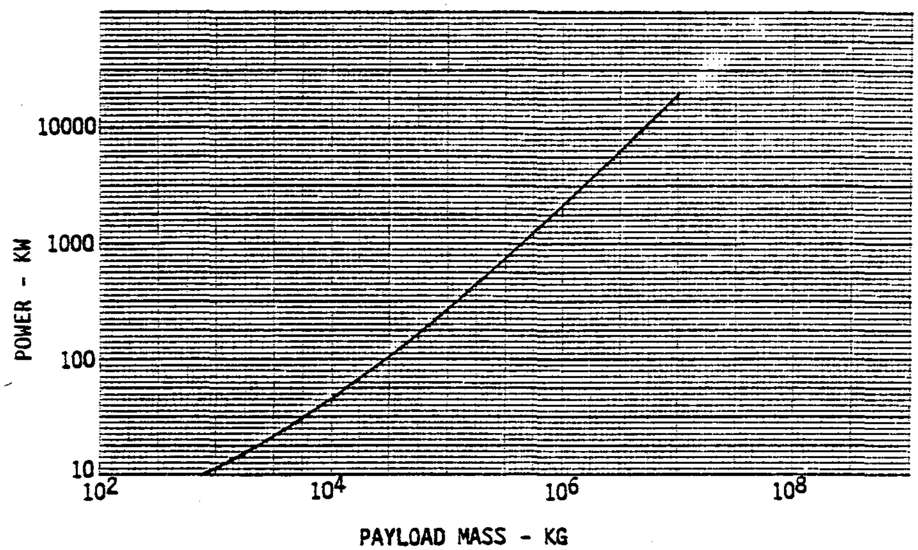


FIGURE 5-44 Mission Variation of Cost Optimum Power

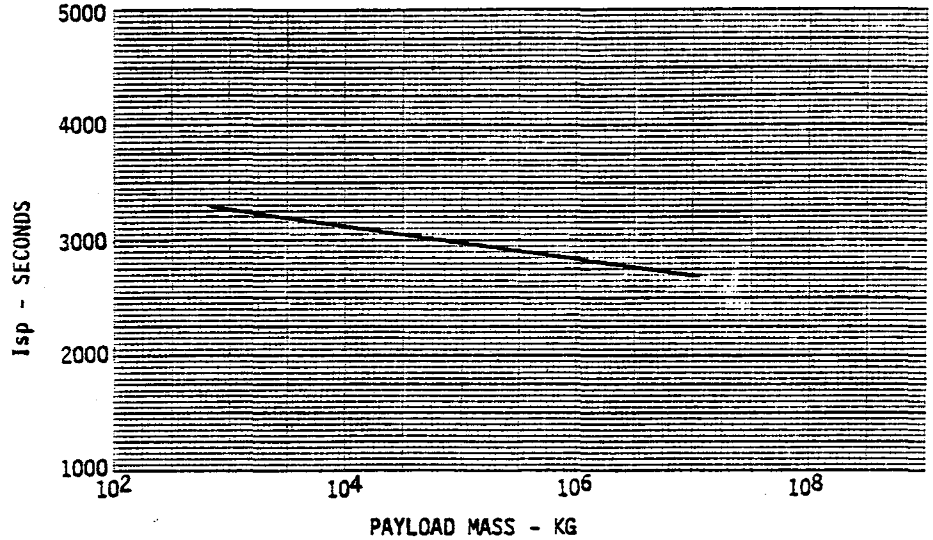


FIGURE 5-45 Mission Variation of Cost Optimum Specific Impulse

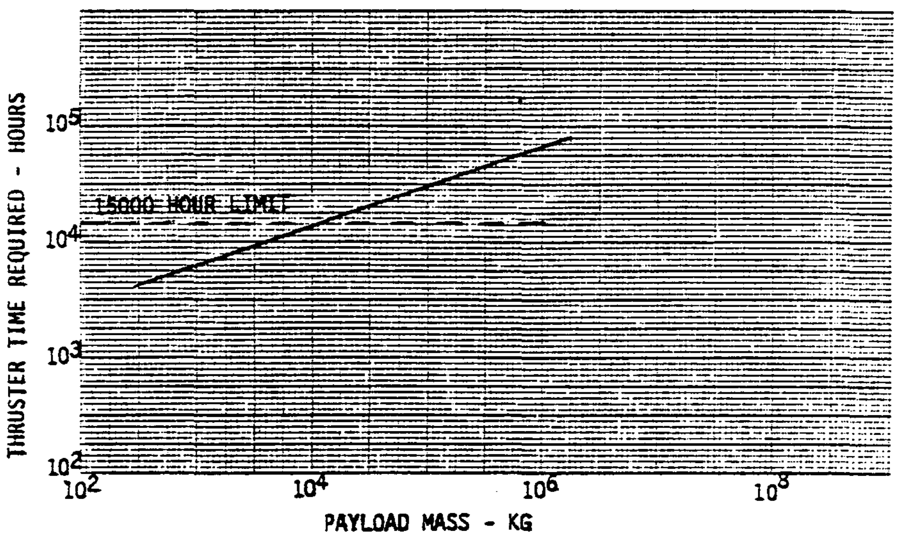


FIGURE 5-46 Mission Variation of Thruster Burn Time

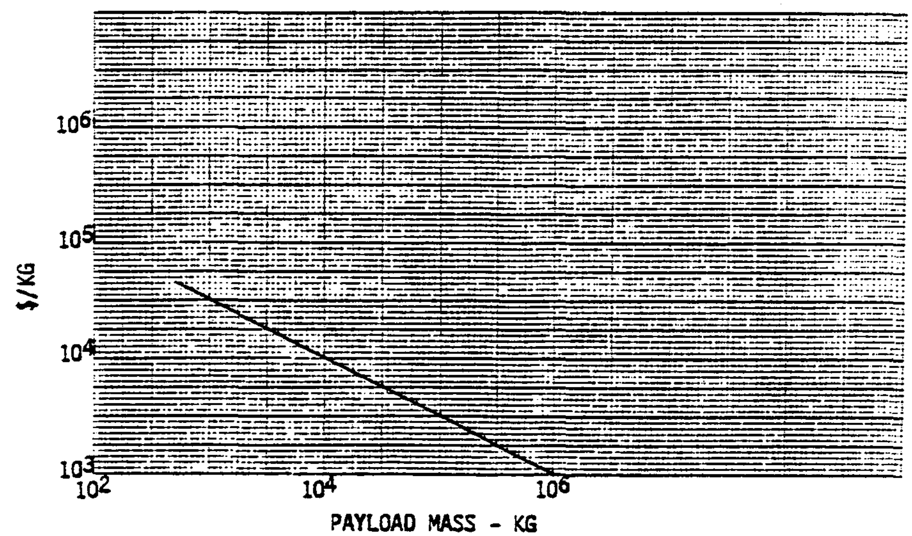


FIGURE 5-47 Mission Variation of Payload Delivery Charges

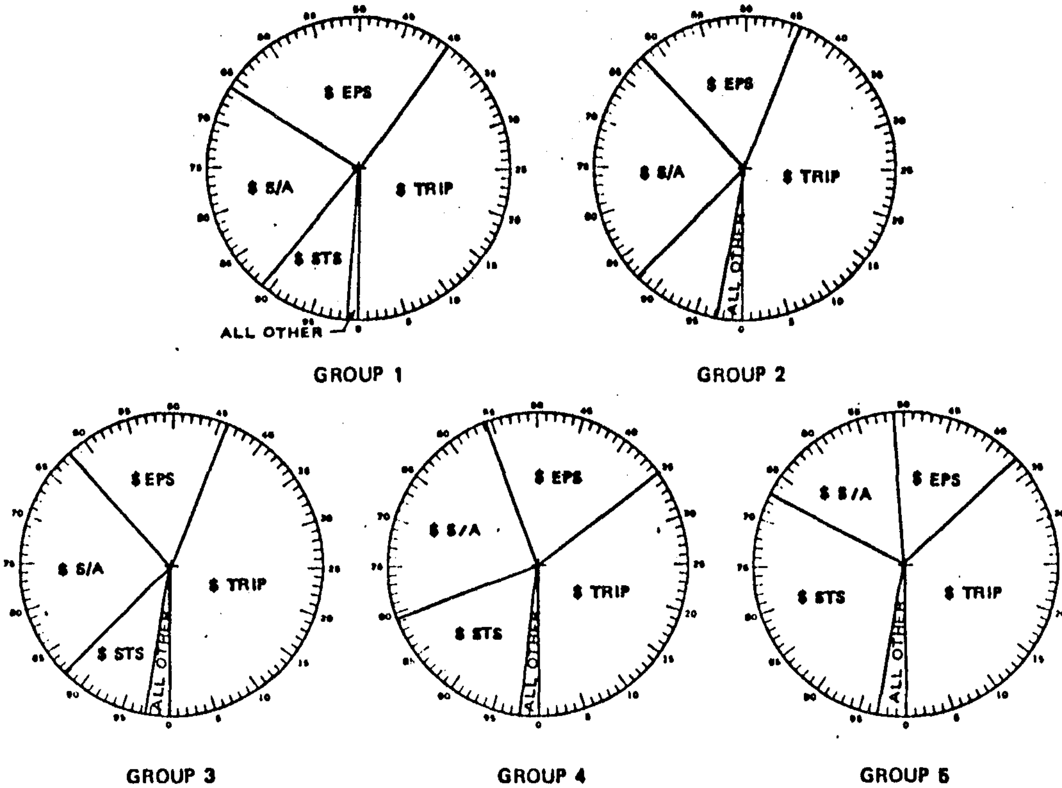


FIGURE 5-48 Components of Transportation Costs - Cost Optimum Design Points

5.5.1 Design Point Sensitivities

Having established a cost optimum design point for each mission using nominal values of the modeling parameters, it is next of interest to define the changes in those solutions that result from perturbing the input assumptions. Such a study was performed and will be summarized herein by resort to the representative mission for each of five groups.

Figure 5-49 displays the space of design points, with the circles indicating the cost-optimum design point* for each of the missions (the numbers identify the mission groups) at the nominal (baseline SOA) conditions. The directed line segments indicate the shift in the cost-optimum solution as the value of the EPS specific mass (α_{EPS}) increases from 0.1 kg/kw to 100 kg/kw (nominal = 21 kg/kw). Since lower values of α mean that system power levels can be increased without a signifi-

*NOTE: The points shown in this section were calculated using constant cost functions, and hence do not correlate with those of table 5-43. Spot checks showed the sensitivity trends to be the same as when variable cost functions are used, but a complete set of data are not available for that case.

cant increase in the EPS mass to be transported, it is seen that decreasing specific masses will drive the cost-optimum power levels up. In addition, heavier systems (greater α) tend toward lower values of specific impulse since the consequent higher thrust levels are required to produce the acceleration necessary to keep the trip times/costs down to reasonable values. However, the changes are relatively small, and thus no major shift in specific impulse goals is called for as component weights are reduced.

A similar plot of the P-I_{SP} space is given in figure 5-50 for the case where the payload-dependent component (α_{STR}) of the electric propulsion mass is varied from 0.1 to 100 gr/kg. No effect is observed due to the small relative contribution of this factor to the mass of the electric propulsion system. Similarly, no change was observed in the system design point when the constant component (M_{AV}) was perturbed (see figure 5-51). This term was varied from 0 to 500 kg and thus was only a small fraction of the EPS mass.

The design point space is again displayed in figure 5-52 to show the shifts that result from changes in the specific cost (γ_{EPS}) of the electric propulsion system. For each of the representative missions, this parameter was allowed to vary from \$150 to \$100,000 per kilogram; the circles represent the design points for a nominal \$13,500/kg value. We see that increases in the per-unit system costs cause a decrease in both the cost-optimum power level and specific impulse. Increases in the EPS per-unit costs cause the EPS component of mission costs to gain in significance relative to the trip time costs, and this increased emphasis causes the tendency toward lower powered optimized systems. The decreased specific impulses reflects the increased significance of EPS costs in relation to the Earth-to-low-orbit launch costs, and a tendency towards keeping a constant thrust level as the system power level falls. Here again, the range of variation is small, from about 3000 to 3250 seconds.

The design point sensitivity to the cost of operating the payload and the EPS during the transportation phase of the mission is shown in figure 5-53. This factor (γ_{OPS}) affects the magnitude of the penalty associ-

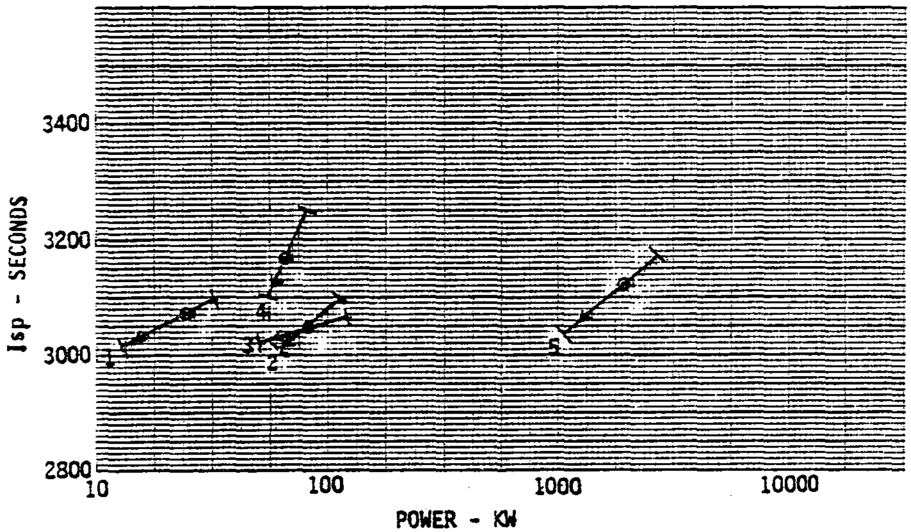


FIGURE 5-49 Sensitivity of Cost Optimum Design Points to EPS Specific Mass

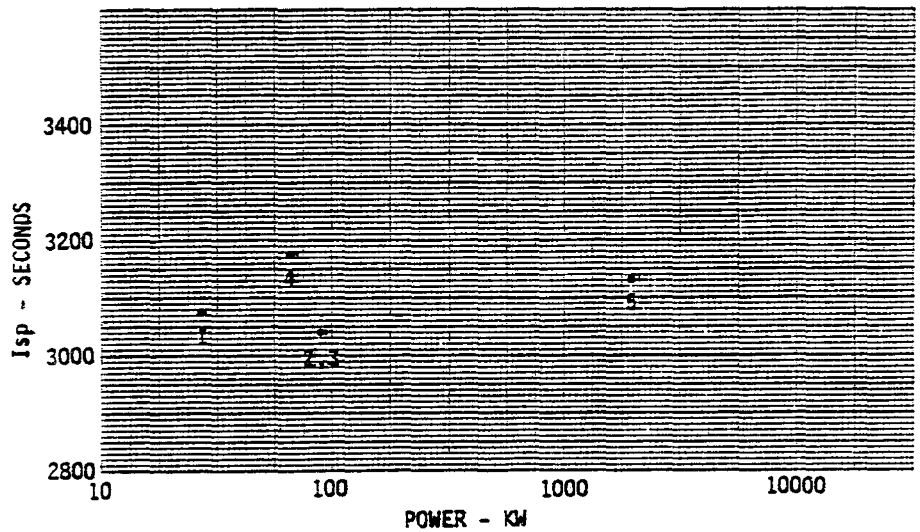


FIGURE 5-50 Sensitivity of Cost Optimum Design Points to EPS Payload - Dependent Mass

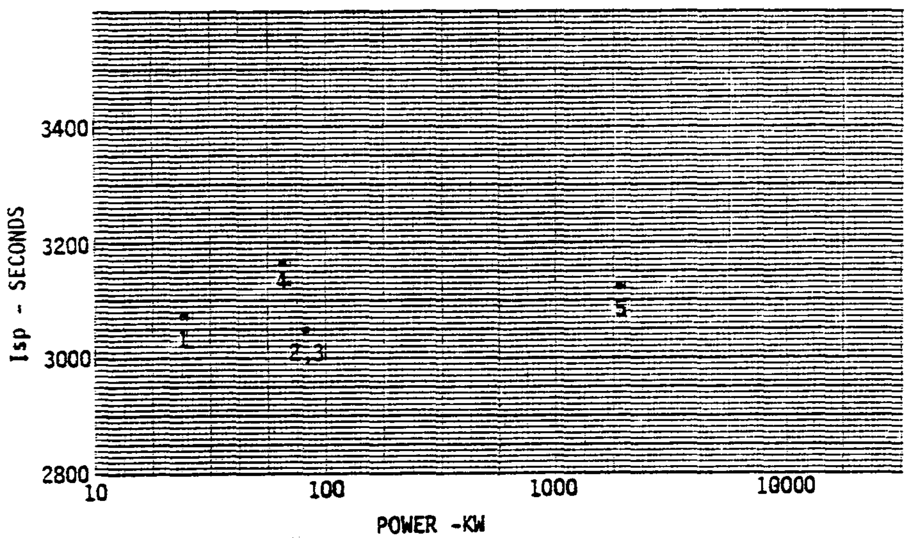


FIGURE 5-51 Sensitivity of Cost Optimum Design Points to System Constant Mass

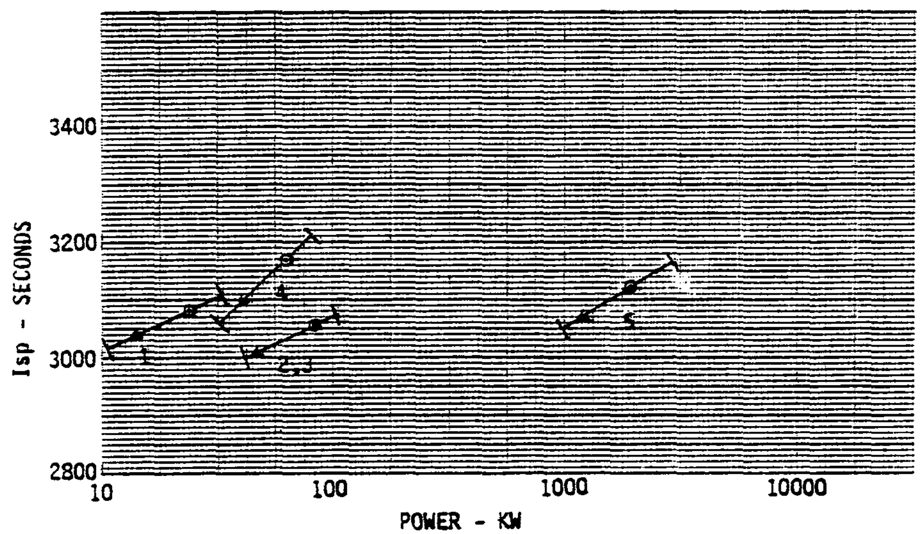


FIGURE 5-52 Sensitivity of Cost Optimum Design Points to EPS Specific Cost

ated with the duration of the electric propulsion mission. Higher costs penalties naturally tend to drive the mission times down. These shorter trip times are obtained by increasing the system power levels and decreasing the EPS specific impulses, as indicated in the plot. However, for the range of operations costs studied ($1 million to $10 million/year), the design point shifts are small.

The other parameter that impacts the trip time cost is the cost of money (δ) to the payload program. This factor enters into the optimization process in the same manner as the operations cost, and produces the same design trends (as shown in figure 5-54), that is, increases in costs will force higher power levels and lower specific impulses. However, because this "interest rate" is multiplied by the value of the payload (see equation 4-2), its leverage is greater than the operation costs, particularly for the more advanced, group 5, missions. This parameter was noted by this study to be the single most important influence on the cost-optimum design point, and with all other characteristics set at their nominal value can force a swing in specific impulse from 2900 to 3500 seconds, and a two-to-one swing in EPS power levels. Some doubt has been expressed as to whether these "interest charge" or "frozen asset" charges will really be assessed in evaluating transportation cost, but we believe that for the postulated scenario (in which commercial and economic factors motivate man to move aggressively into an expanding space program) this component of transportation costs will play a decisive role in mission-and system-level trade-offs. In the figure, the range of variation was from zero to 20% per year, with the nominal 7% values "circled"; greatest sensitivity is below 10%.

Figures 5-55 and 5-56 present the sensitivity of the cost-optimum design point to the characteristics of the power source. Within the range of 1 to 20 kg/kw, the optimum power and I_{SP} was not affected by the mass of the solar array. Not so with the solar array specific costs, which were varied from %0.50 to $500/watt ($350/watt is the nominal, circled value). Just as with the EPS specific costs, the missions will optimize to higher power levels if the costs of obtaining/utilizing that power decreases (the

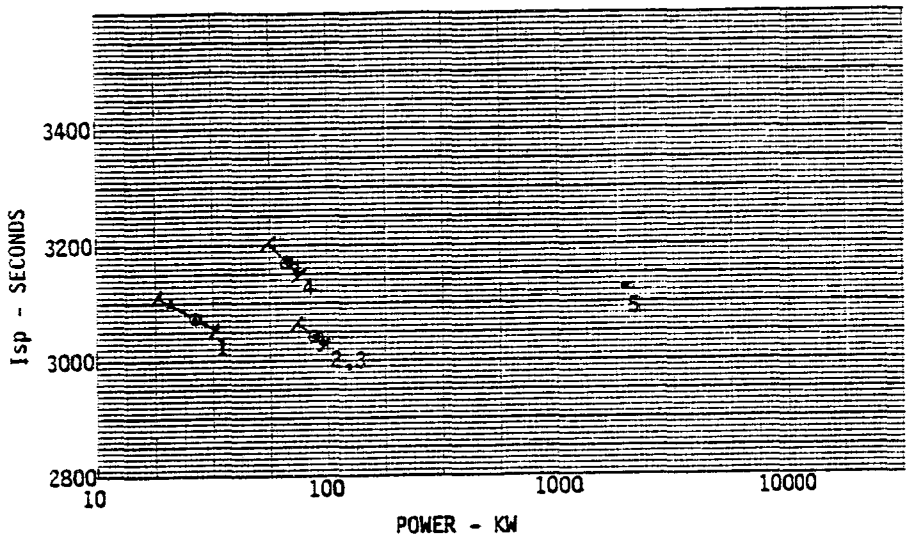


FIGURE 5-53 Sensitivity of Cost Optimum Design Points to Operations Costs

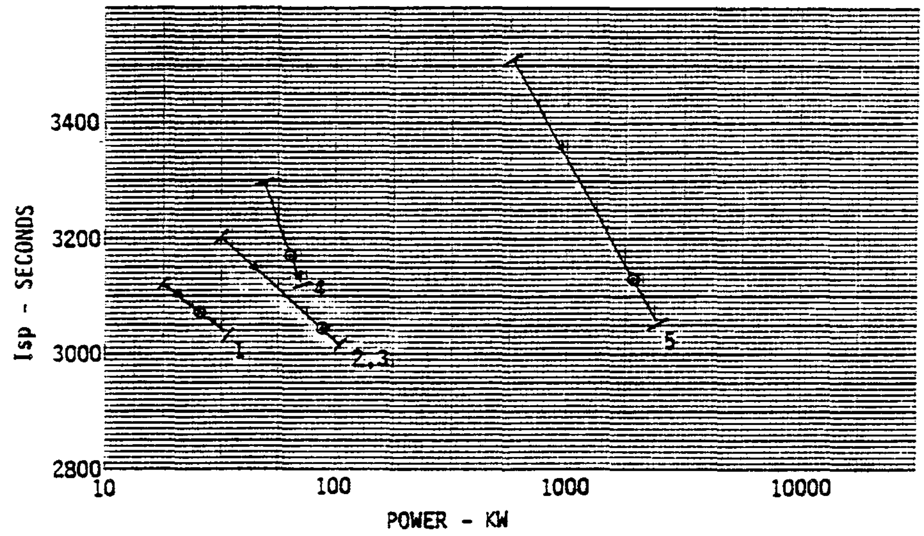


FIGURE 5-54 Sensitivity of Cost Optimum Design Points to Cost of Money

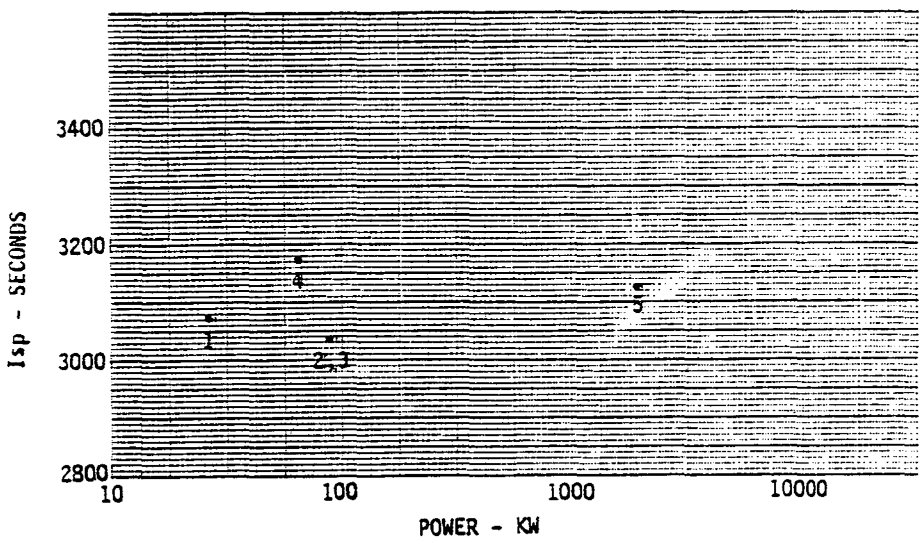


FIGURE 5-55 Sensitivity of Cost Optimum Design Points to S/A Specific Mass

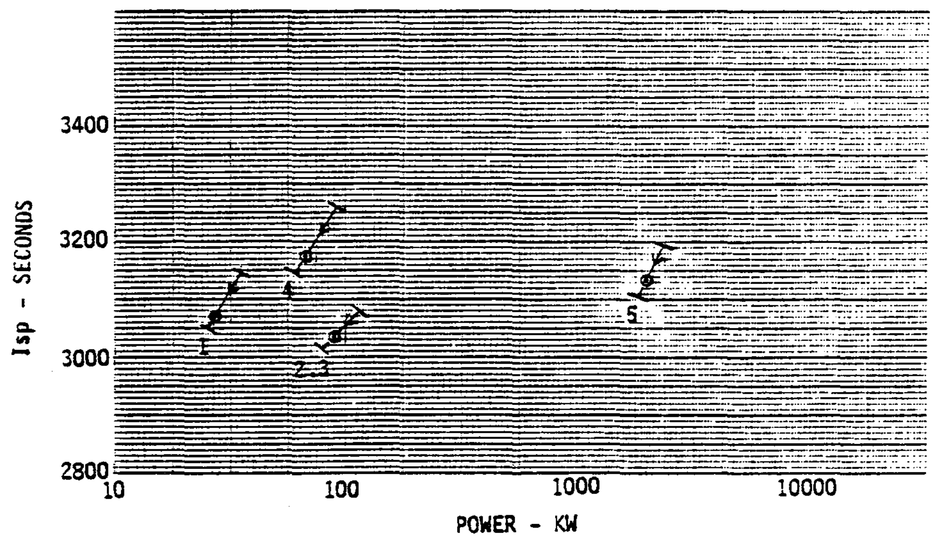


FIGURE 5-56 Sensitivity of Cost Optimum Design Points to S/A Specific Cost

"law of supply and demand" as applied to EPS mission economics). Additionally, since the higher power levels will drive the trip time/costs down, the I_{SP} may be increased, with decreasing power costs, allowing a savings in Earth-to-orbit transportation charges.

The effects of perturbations in the costs of transporting the electric propulsion system and its power source, propellant and payload to low Earth orbit is mapped in the P-I_{SP} space in figure 5-57. The range of launch costs shown are from \$25 to \$1000 per kilogram, as compared to a nominal (circled) value of \$700/kg. The arrows represent increasing costs (technology retrocession). Decreasing ETO transportation costs will emphasize the importance of the trip time penalties. To achieve shorter missions, an increase in the optimum power level is coupled with a lowering of the system specific impulse.

The final perturbation studied was that due to changes in the velocity increment (ΔV) necessary to accomplish each mission. This is also equivalent to an examination of the effects of the trajectory loss factors (for radiation degradation, occultation, start-up delay, drag and steering). In figure 5-58, the mission ΔV was increased from 3000 to 9000 m/s with the circles representing the nominal requirement (5760 meters per second) for transport to geosynchronous orbit. The higher energy missions tend to optimize at slightly large power levels to keep trip time penalties low, and at larger specific impulses, in order to keep the propellant launch charges down. It is noted that over this rather large range of mission energies, the change in the desirable I_{SP} is less than 20% of the state-of-the-art value of 3000 seconds, and well within the range of variability that has been demonstrated with current hardware.

None of the parameters examined caused any "large" changes in the set of cost-optimized design points. (The exception was the efficiency function - magnitude and shape factor - which will be discussed in section 5.6 of this report.)

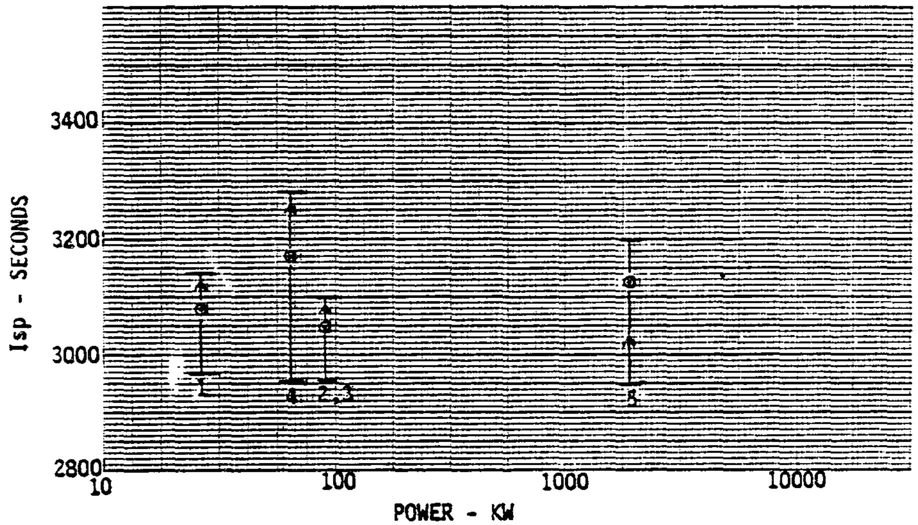


FIGURE 5-57 Sensitivity of Cost Optimum Design Points to Earth Launch Costs

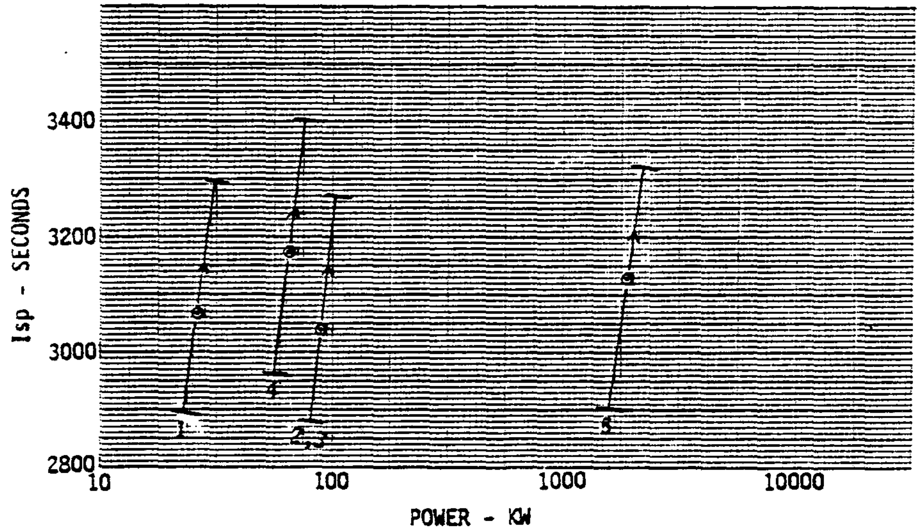


FIGURE 5-58 Sensitivity of Cost Optimum Design Points to Mission Energy Requirements

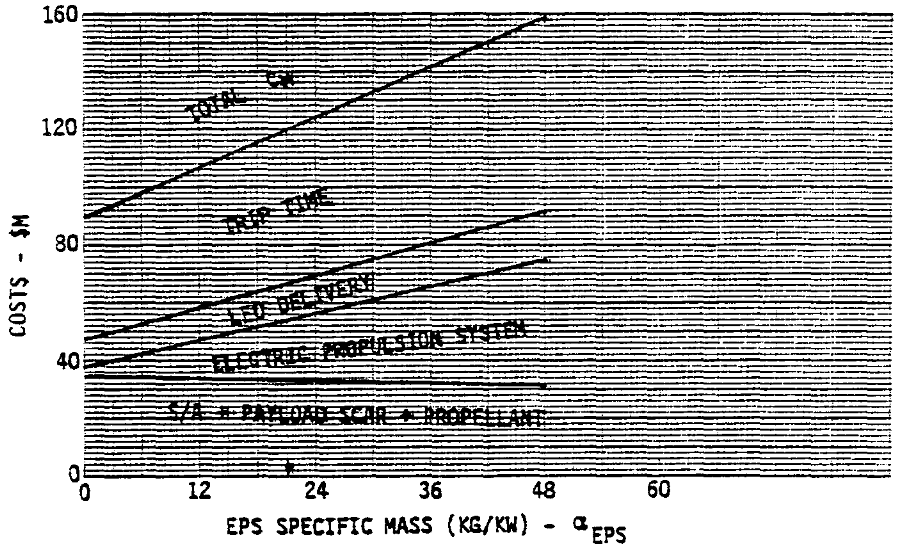


FIGURE 5-59 Transportation Cost Sensitivity to EPS Specific Mass

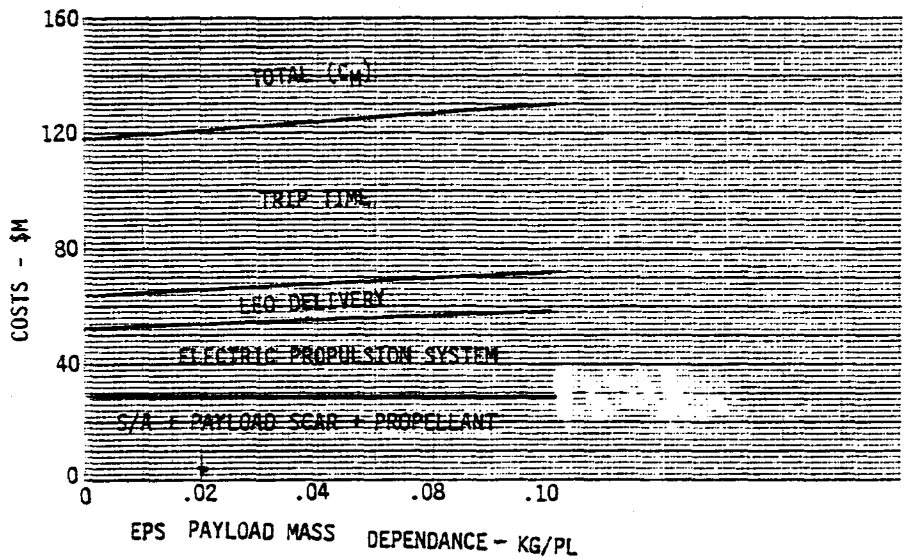


FIGURE 5-60 Transportation Cost Sensitivity to EPS Payload Dependent Mass

5.5.2 Mission Cost Sensitivities

For each member of the overall mission set, the sensitivity of the total mission cost, and each of its components, to perturbations in the modeling parameters was calculated. This data allows an assessment of the potential benefit to be gained from any contemplated technology improvement undertaking. Nominal values were used for all parameters except the one being examined. Cost optimum values were used for system power levels and specific impulses.

In figures 5-59 thru 5-61, the changes in mission costs are shown as a function of the magnitude of each of the components of the electric propulsion system mass. (Throughout this section, each of these sensitivities will be illustrated by resort to the representative mission for group 3 - the Geosynchronous Communications Platform - thus obviating the need to display 30 similar plots for each parameter.) The arrow indicates the SOA values. In each case, we note that the only cost significantly affected is that of the electric propulsion system, and this simply increases in a linear fashion.

Figures 5-62 through 5-64 show the changes in these sensitivities as a function of the payload mass. The ordinate for this set of curves is the slope of the "total cost" curves (previous 3 figures). It is given in terms of the percentage change in mission costs caused by a one percent change in the studied parameter - at the nominal value of that parameter. The effect of EPS specific mass is constant across the mission set, while the payload structural support factor tends to gain in importance for heavier payloads, as might be expected. The system constant mass becomes a smaller component of the total EPS mass as mission difficulty increases; thus, the sensitivity to it decreases.

Figure 5-65 and 5-66 show the impact of raising the per-unit cost of the electric propulsion system. The EPS cost is of course the only component of mission costs affected. The sensitivity to this parameter is essentially constant across the mission set.

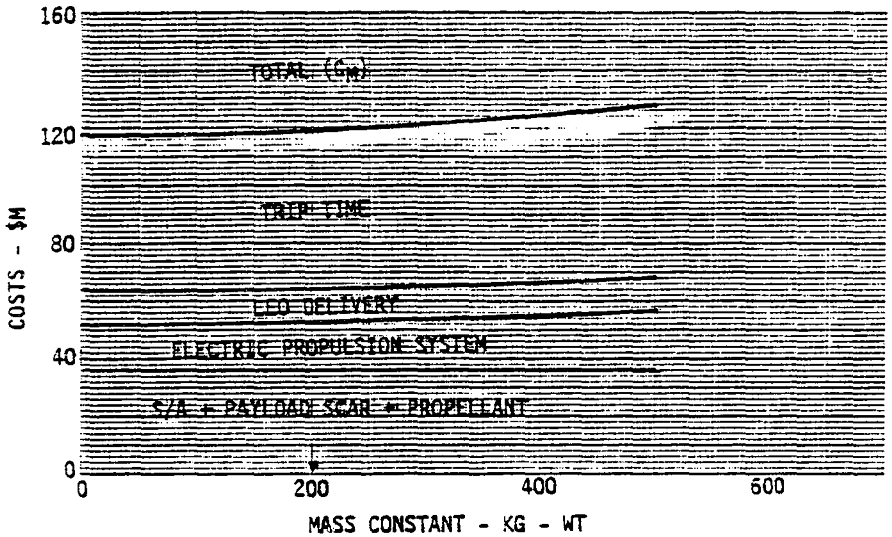


FIGURE 5-61 Transportation Cost Sensitivity to EPS Constant Mass

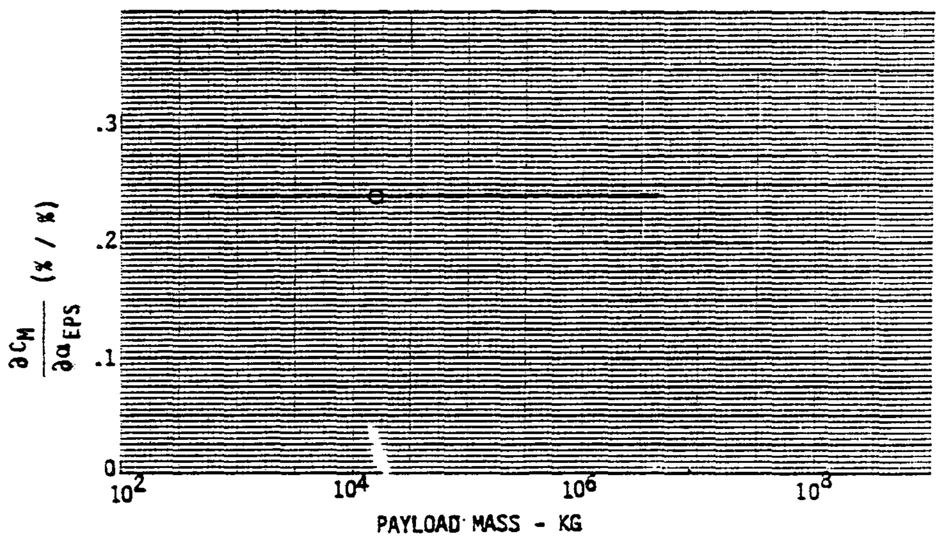


FIGURE 5-62 Mission Effects on Sensitivity to EPS Specific Mass

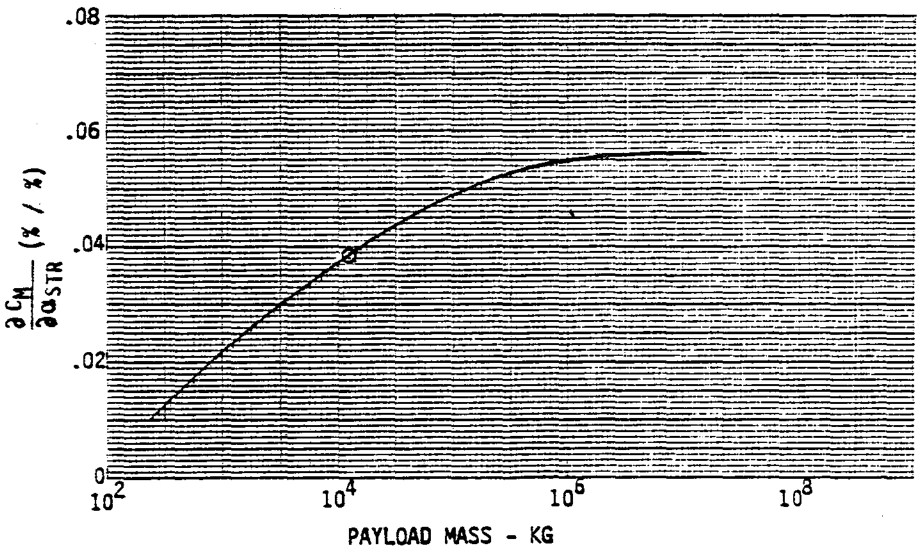


FIGURE 5-63 Mission Effects on Sensitivity to EPS Payload Mass

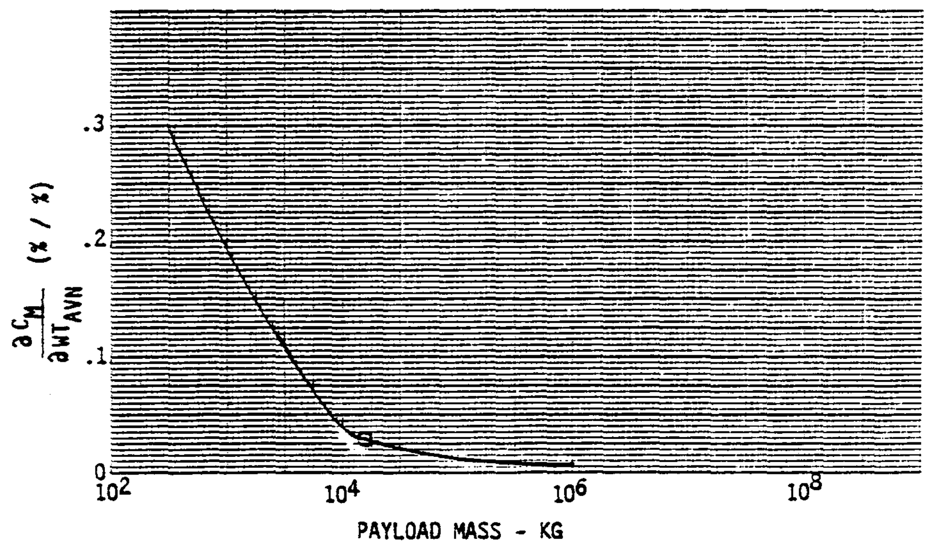


FIGURE 5-64 Mission Effects on Sensitivity to EPS Constant Mass

Since the propellant costs contribute such a small part to the total mission costs, the effects of changing these costs is essentially negligible (see figure 5-67). There is a slight increase in this sensitivity (figure 5-68) as missions become larger, but the value is still very small.

Figures 5-69 through 5-72 show the sensitivities to the power source characteristics. Increasing the solar array mass impacts the costs to launch the EPS from Earth to its initial orbit, and to a minor extent, increases the trip time penalty due to lower initial accelerations. Changing the specific cost of the solar array does not impact any other components of mission cost. Both effects remained relatively constant across the mission set.

Figure 5-73 shows the influence of the STS charges to deliver the EPS and its payload to low-Earth orbit. The impact of this factor escalates as the mission becomes more ambitious, as can be seen in figure 5-74. Obviously, Earth-launch systems with lower operational costs, or higher delivery efficiency, will be desirable for the far-term missions.

The impact of the two factors that determine the amount of penalty that is charged for long mission times is shown in figures 5-75 and 5-76. Both the "interest charges" and the system operating charges have a straight-forward relationship. The changes in these two sensitivities across the mission set are displayed in figure 5-77 and 5-78. They have been plotted against the value of the payload, since that is fundamental to the assessment of any trip time charges. The impact of the cost of money is enhanced with increased payload values, while the influence of the system operating cost decreases in relative influence.

5.5.3 Power Utilization Impacts

The baseline power utilization strategy assumed for the cost modeling in this study was that sufficient propulsive capacity would be installed to utilize all of the power coming from the energy source at the start of the vehicle lifetime. Since for the typical near-Earth mission, the solar array output will quickly be degraded by radiation damage, an excess propulsive capability will be carried (as dead weight) for a significant por-

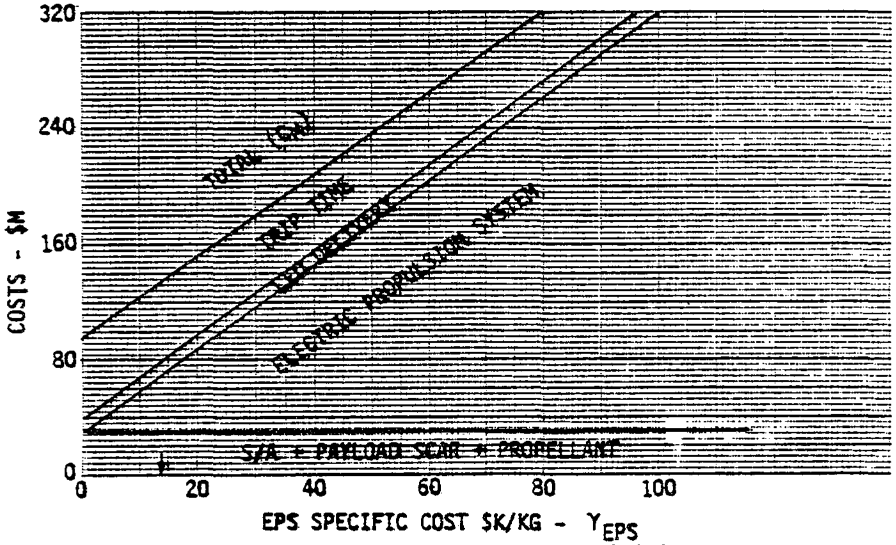


FIGURE 5-65 Transportation Cost Sensitivity to EPS Specific Cost

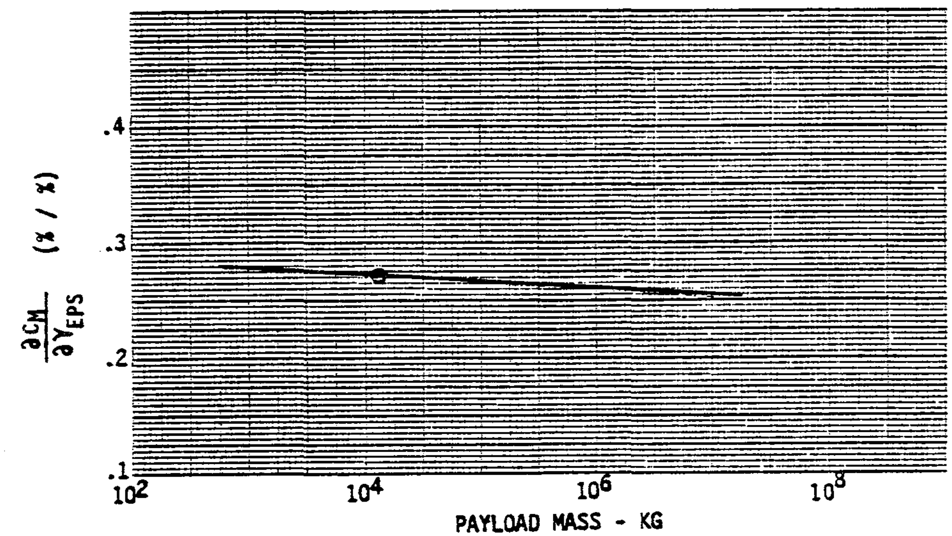


FIGURE 5-66 Mission Effects on Sensitivity to EPS Specific Cost

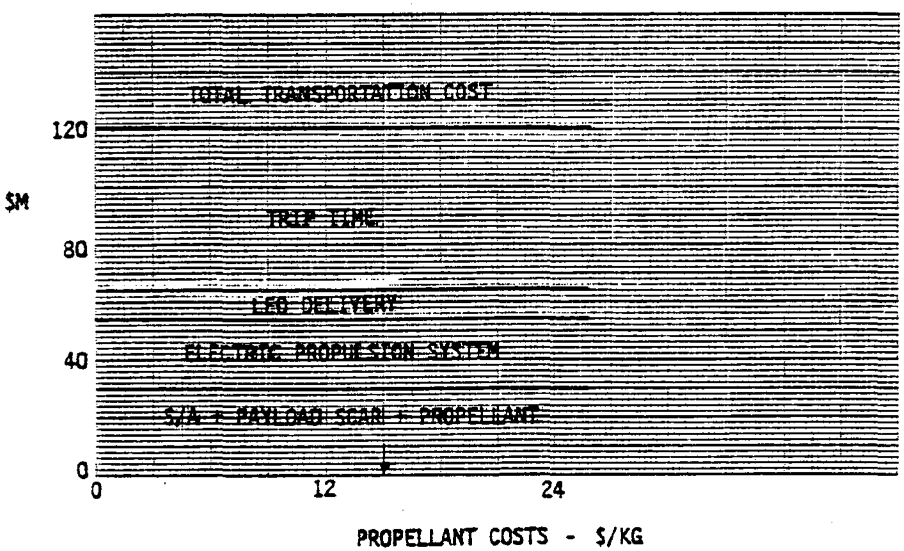


FIGURE 5-67 Transportation Cost Sensitivity to EPS Propellant Costs

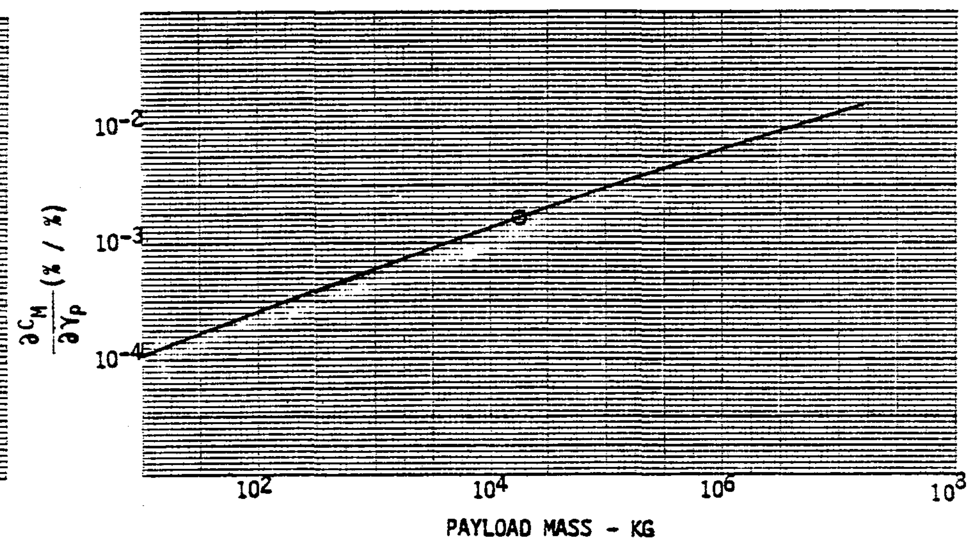


FIGURE 5-68 Mission Effects on Sensitivity to EPS Propellant Costs

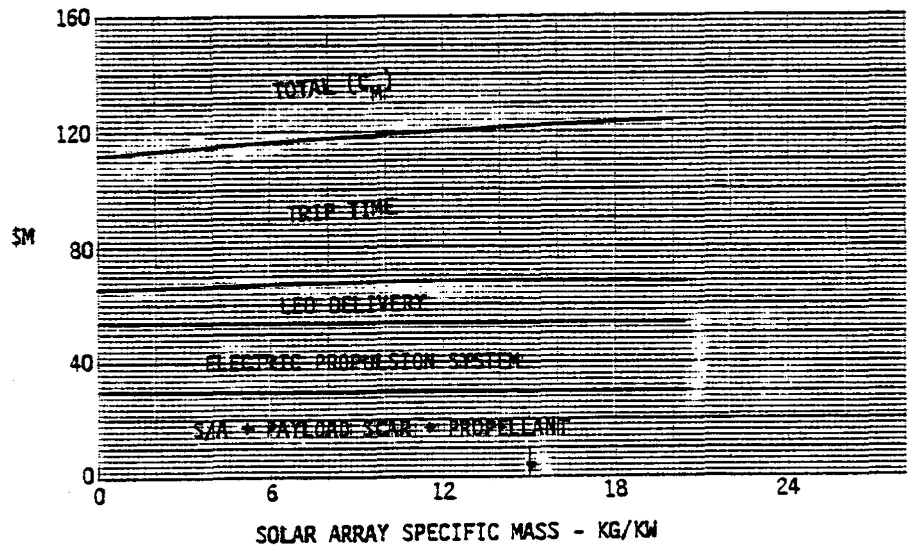


FIGURE 5-69 Transportation Cost Sensitivity to S/A Specific Mass

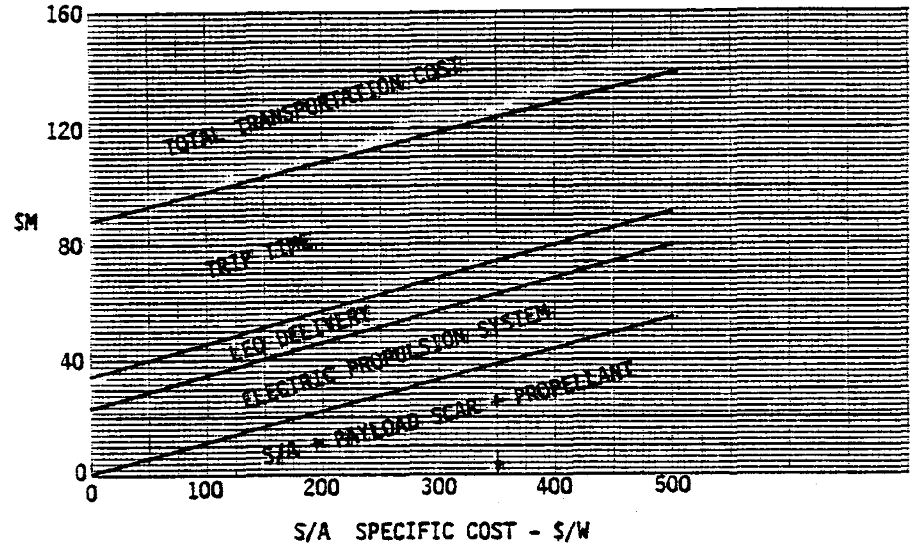


FIGURE 5-70 Transportation Cost Sensitivity to S/A Specific Cost

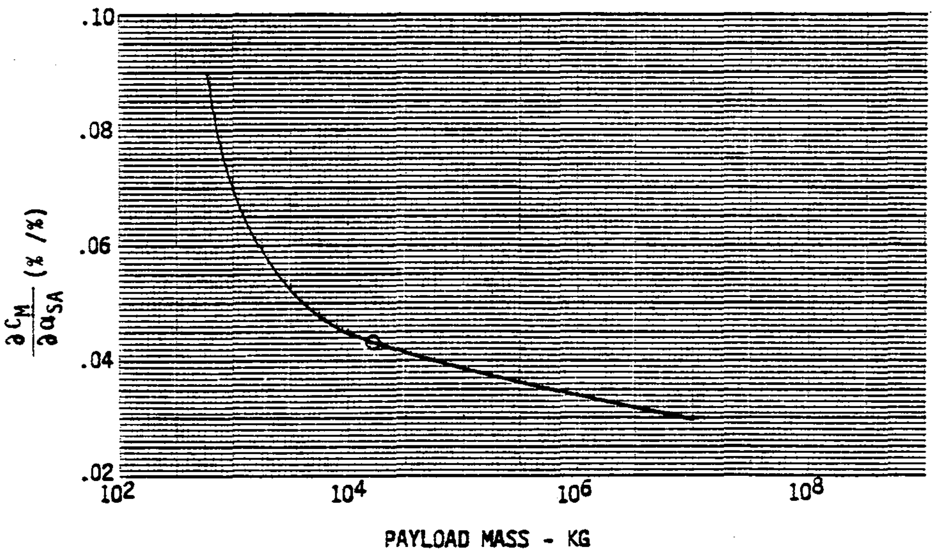


FIGURE 5-71 Mission Effects on Sensitivity to S/A Specific Mass

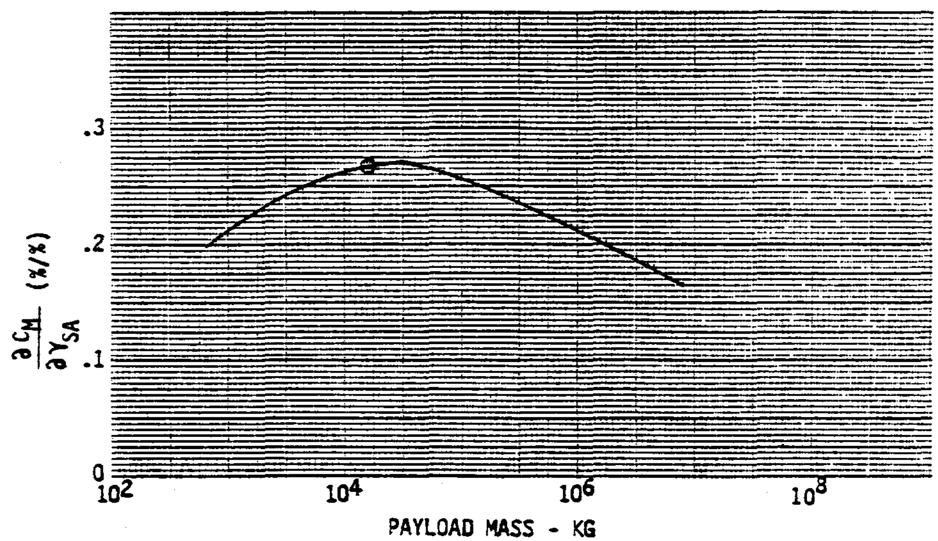


FIGURE 5-72 Mission Effects on Sensitivity to S/A Specific Cost

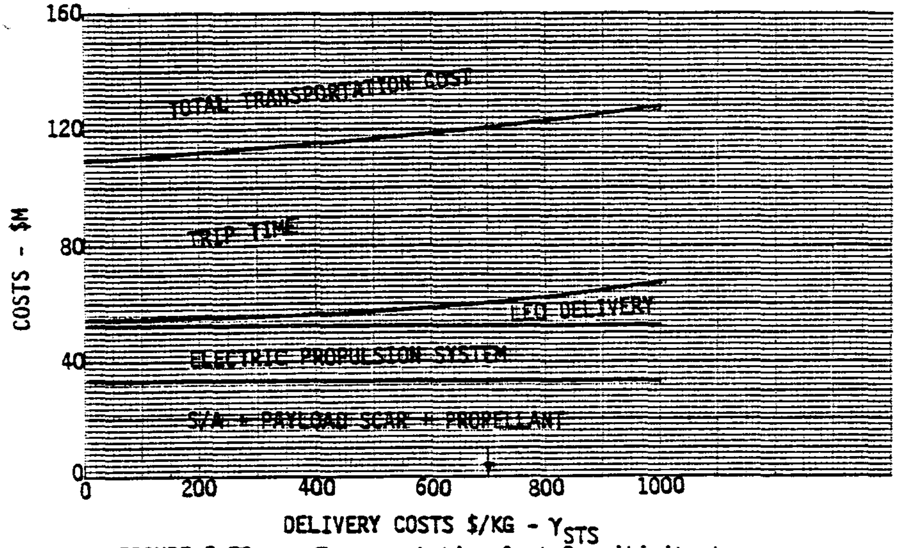


FIGURE 5-73 Transportation Cost Sensitivity to ETO Launch Costs

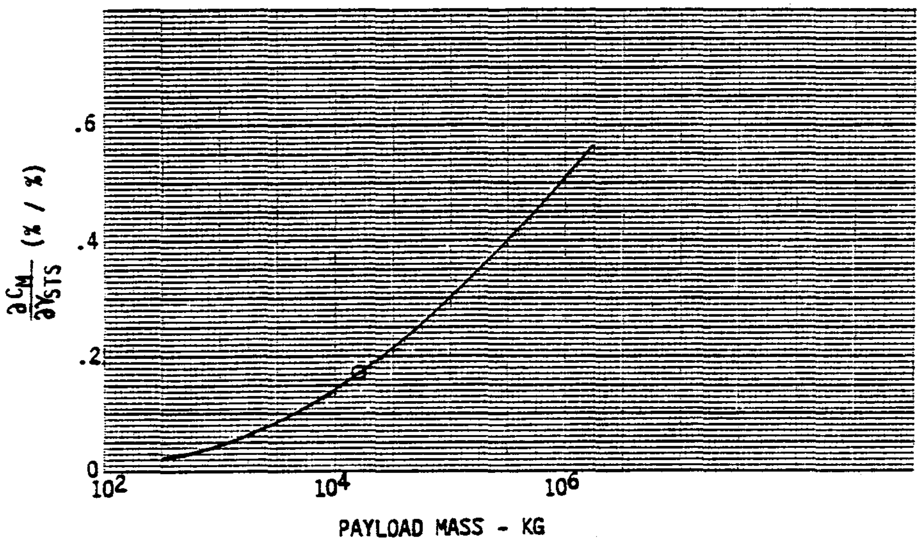


FIGURE 5-74 Mission Effects on Sensitivity to ETO Launch Costs

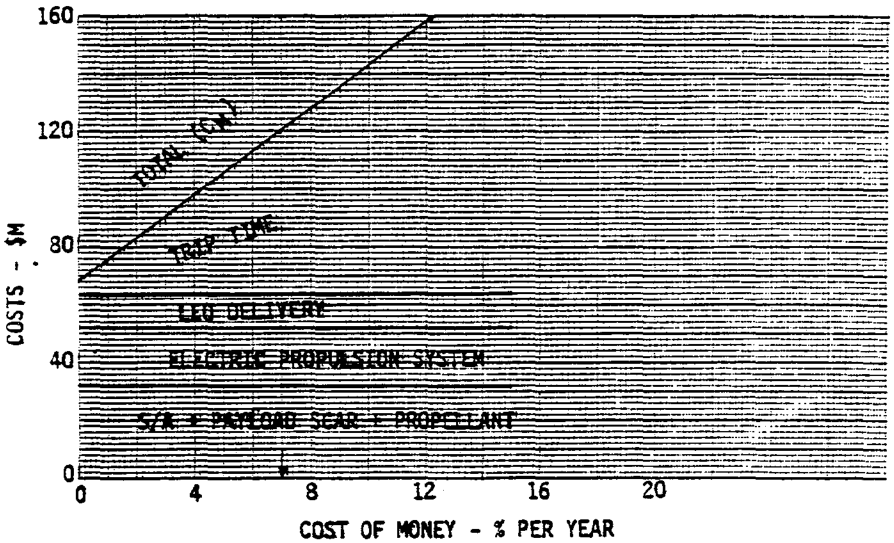


FIGURE 5-75 Transportation Cost Sensitivity to Cost of Money

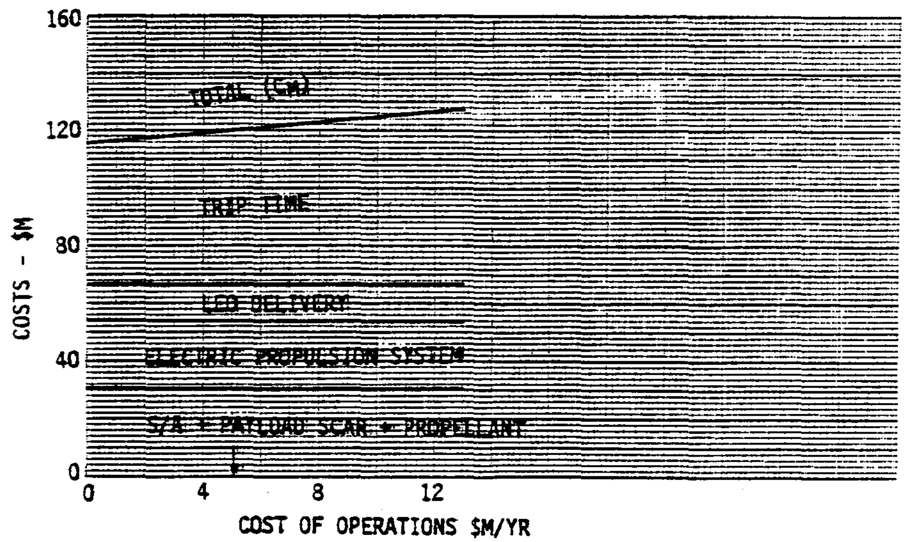


FIGURE 5-76 Transportation Cost Sensitivity to Cost of Operations

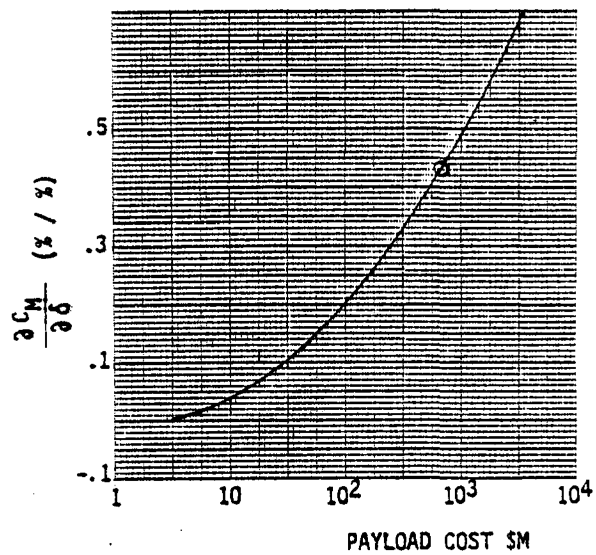


FIGURE 5-77 Mission Effects on Sensitivity to Cost of Money

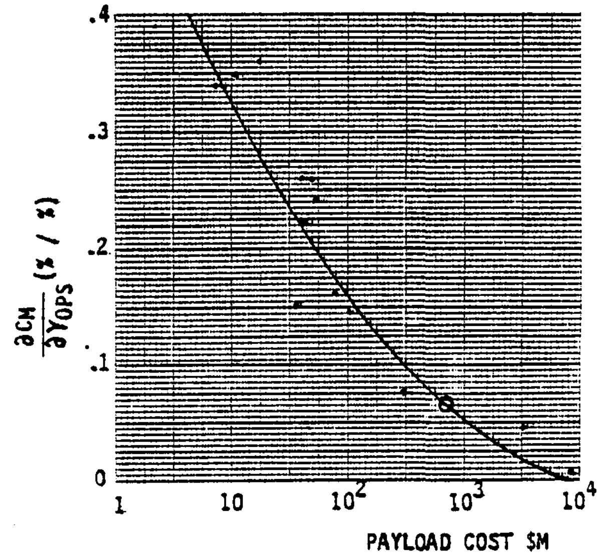


FIGURE 5-78 Mission Effects on Sensitivity to Cost of Operations

tion of the time (see figure 5-79). Recent studies have suggested that it may be more cost effective to install only enough propulsive capability to utilize the solar array output that is expected at the end of the mission. In fact, this is true, as illustrated by figure 5-80, a power study of the group 3 representative mission.

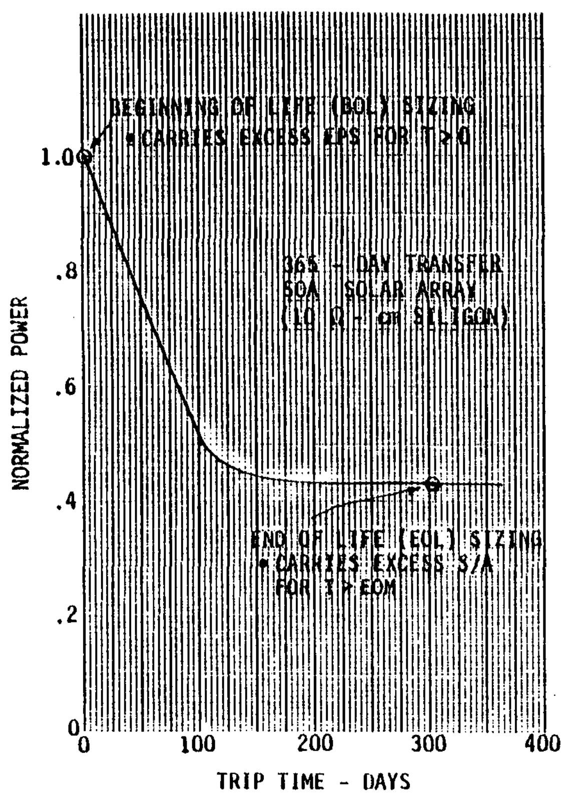


FIGURE 5-79 Alternate System Sizing Strategies

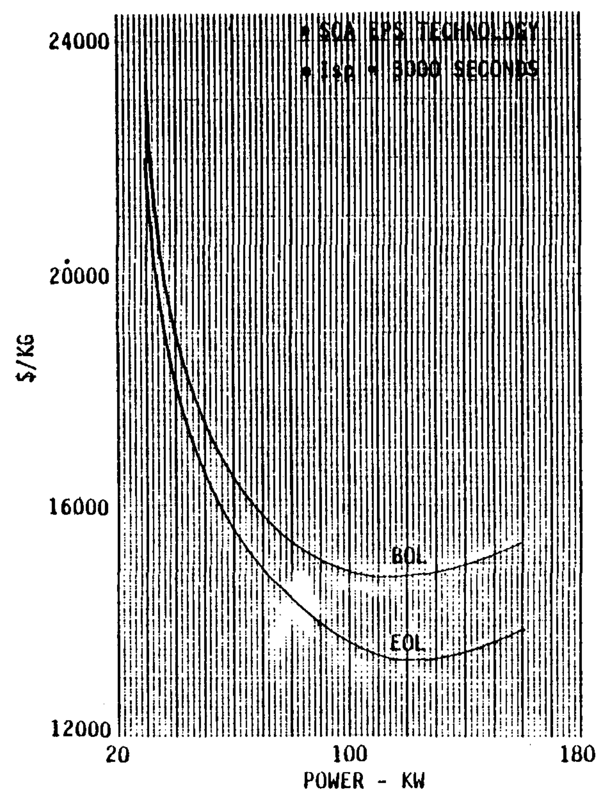


FIGURE 5-80 Mission Performance Comparison - Geosynchronous Communications Platform

	MISSION NAME	INSTALLED POWER (kW)	UTILIZED POWER (kW)	I_{sp} (SEC)	MISSION TIME (DAYS)	THRUSTER TIME (HRS)	COSTS ($M) EPS	SA	LAUNCH	TRIP	PROPELLANT	SCAR	TOTAL $/KG
1	Tethered Satellite	12	12.0	3000	676	6062	7.873	4.716	1.310	8.764	.004	0.	32032
2	Nuclear Waste Disposal	21	12.6	3250	809	11089	8.344	7.768	4.104	10.626	.013	0.	9492
3	Utility Load Management Satellite	29	15.4	3100	652	6924	8.801	10.350	4.111	14.778	.012	.326	11987
4	Earth's Magnetic Tail Mapper	9	6.5	3200	420	6752	6.712	3.662	.927	6.926	.004	.049	45810
5	Earthwatch	25	16.0	3050	599	8874	9.447	9.074	6.862	12.365	.014	.260	5848
6	Astronomical Telescope	26	25.5	3050	517	4665	10.149	9.396	2.083	23.951	.076	1.137	51899
7	Nuclear Fuel Location System	15	7.9	3050	590	6260	7.144	5.752	1.946	8.866	.006	.072	17487-
8	Global Search & Rescue Locator	11	6.2	3000	412	4800	6.714	4.367	1.364	6.743	.004	.130	21208
9	Geosynchronous-Based Satellite Maint.	6	4.9	2600	234	6476	7.336	3.662	1.436	3.020	.003	.211	14606
10	Electronic Mail Transmission	117	62.0	3100	480	6093	16.905	32.951	12.156	45.884	.007	2.795	12057
11	Multi-National Air Traffic Control Radar	16	15.7	3050	761	6866	8.685	6.093	2.407	10.254	.007	.011	16120
12	Space Based Radar (Near Term)	20	12.2	2900	373	5878	8.402	7.437	4.296	10.036	.007	.488	7661
13	Near-Term Navigation Concept	17	9.0	3000	364	3862	7.231	6.432	1.371	10.829	.005	.585	36430
14	Technology Development Platform	29	15.4	3150	699	7416	8.783	10.350	4.061	14.545	.012	.260	12293
15	Personal Communications Wrist Radio	126	66.8	3150	633	6716	17.023	35.030	17.196	44.522	.043	1.950	8269
16	Orbiting Deep Space Relay Station	49	26.0	3100	739	7841	11.181	16.618	8.703	21.419	.021	.552	7796
17	Gravity Gradient Explorer	29	16.0	3250	1009	11109	9.280	10.665	6.988	13.054	.017	0.	7784
18	Soil Surface Texturometer	23	22.5	3000	688	6207	9.903	8.426	3.223	14.867	.009	.295	15882
19	GSO Communications Platform	118	62.5	3100	442	4690	15.988	33.583	11.341	46.577	.030	3.172	13495
20	Space Based Radar (Far Term)	21	20.6	3000	389	9150	10.445	8.097	7.235	11.777	.012	.650	5457
21	Personal Navigation Wrist Set	79	41.9	3250	919	9751	14.151	24.764	16.657	29.284	.038	.650	6205
22	Marine Broadcast Radar	59	31.3	3150	652	6918	11.825	19.376	8.324	25.740	.022	.910	9875
23	Geosynchronous Space Station	94	49.0	3250	933	9899	15.440	28.320	18.770	33.356	.045	.780	5059
24	Orbiting Lunar Station	110	79.2	3500	972	15398	19.324	32.098	26.265	39.157	.072	.942	5331
25	Space Construction Facility	9800	9604	2600	410	3860	311.368	296.007	601.720	246.926	1.612	20.150	691
26	Power Relay Satellite	160	79.5	3760	990	10504	19.846	40.290	30.212	19.872	.062	.234	3992
27	Iceberg Dissipator	7794	5300	2600	994	16400	254.286	299.280	631.948	48.267	5.243	1.625	687
28	SPS Pilot Plant	7300	3869	3350	332	3523	166.893	265.750	453.463	478.786	.916	48.750	4160
29	Satellite Power System	91500	48495	2950	741	7862	815.210	662.006	131.082	716.873	23.631	32.499	190
30	SPS Orbit Transfer Recovery	700	357	4600	2203	22491	54.745	102.997	262.622	46.552	.389	.293	1690

FIGURE 5-81 End of Life Sizing - EPS Performance

New cost-optimum design points were calculated for each member of the baseline mission set, and the results are tabulated in figure 5-81. As shown in figure 5-82, there is an across-the-board reduction in total mission transportation charges of about 10%. However, from a technology development standpoint, the question of whether to employ EOL or BOL sizing is irrelevant. This is illustrated in figure 5-83, which shows the impact of the different strategies in the space of design points.

(NOTE; Except for this section, all other studies reported herein were exercised with the assumption that BOL sizing would be employed.)

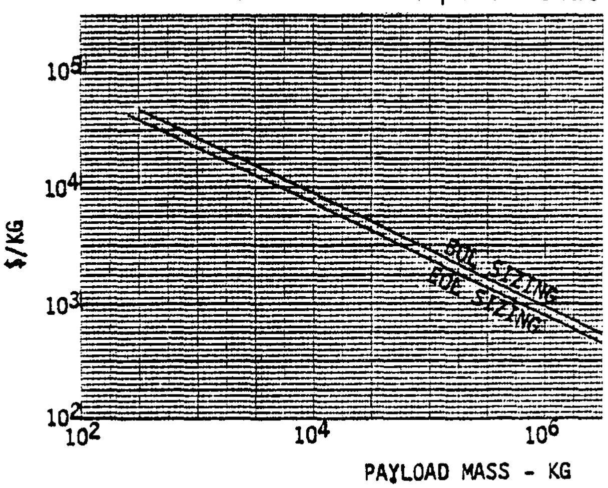


FIGURE 5-82 EOL Performance Across the Mission Set

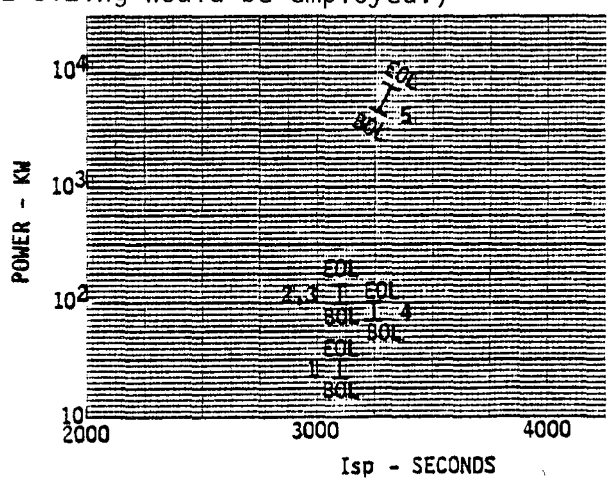


FIGURE 5-83 EOL Impact on Cost Optimum Design Points

5.5.4 Technology Parameter Interactions

In addition to the sensitivities of the mission costs to each of the characteristic parameters of the electric propulsion system, it is desirable to know if trades are possible. Such a trade might sacrifice a regression in one characteristic for an improvement in another to realize a net gain in mission performance. With this end in mind, the interactions that occur between the most significant characteristics of electric propulsion system technology (i.e., the specific mass, cost, and efficiency of the EPS, the launch costs, and the cost of power) were examined.

Figures 5-84 thru 5-88 show the nature of the interaction between the efficiency of the cost-optimum electric propulsion system and its specific mass for the missions that are being used to represent the five mission groups. The lines on the figures are isograms with respect to transportation costs. (Any point on the one marked "nominal" will yield mission costs equal to the cost optimal solution.) Thus for near-term missions, there exists the possibility of allowing a reduction in system efficiency in order to gain an improvement in EPS specific mass. The break-even point is approximately -2% for a 1 kg/kw improvement in the vicinity of the current (SOA) technology (circled). However for later missions, this is no longer true and even if the system mass could be reduced to zero, this would not pay for even a one percent loss in efficiency. This is primarily due to the much greater impact on trip time of efficiency as compared to specific mass, and the large contribution of trip time costs for the advanced missions.

The interplay between the specific mass of the electric propulsion system and its cost can be seen in figures 5-89 thru 5-93. Here again, the mission cost isograms show the potential trade-offs. (The reason that all three curves do not appear on all five plots is that it is not always possible to achieve the attempted 10% increment in mission costs by changing only the two parameters that are shown.) For early missions, it appears that an EPS cost increase on the order of 70% could be

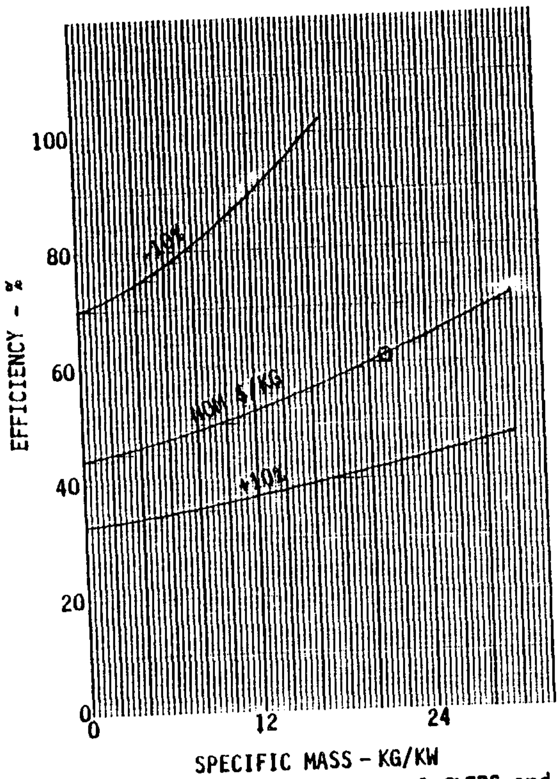


FIGURE 5-84 Interactions of αEPS and η - Group 1

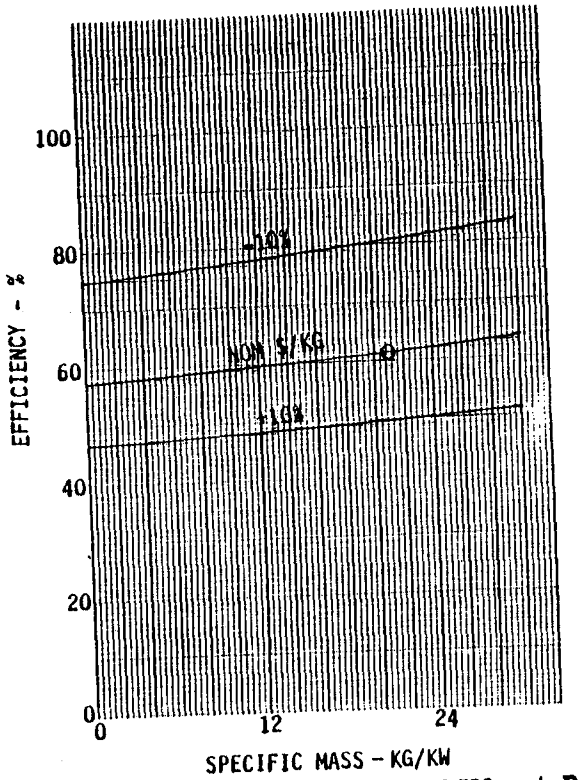


FIGURE 5-85 Interactions of αEPS and η - Group 2

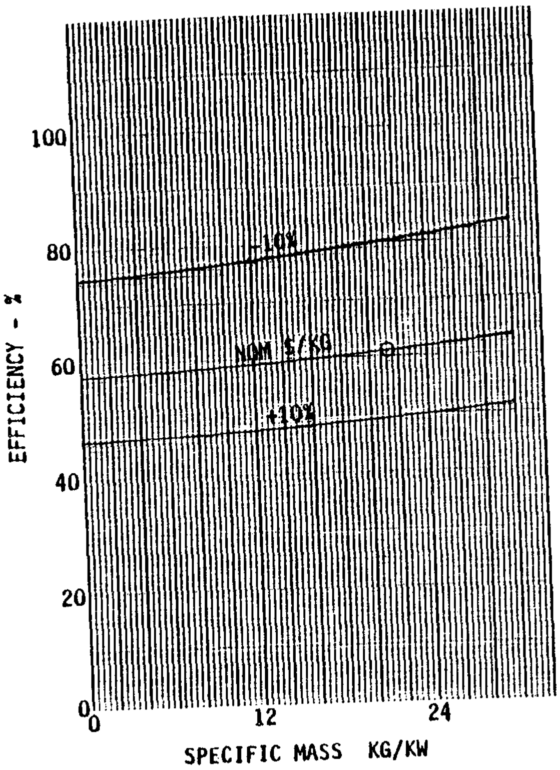


FIGURE 5-86 Interactions of αEPS and η - Group 3

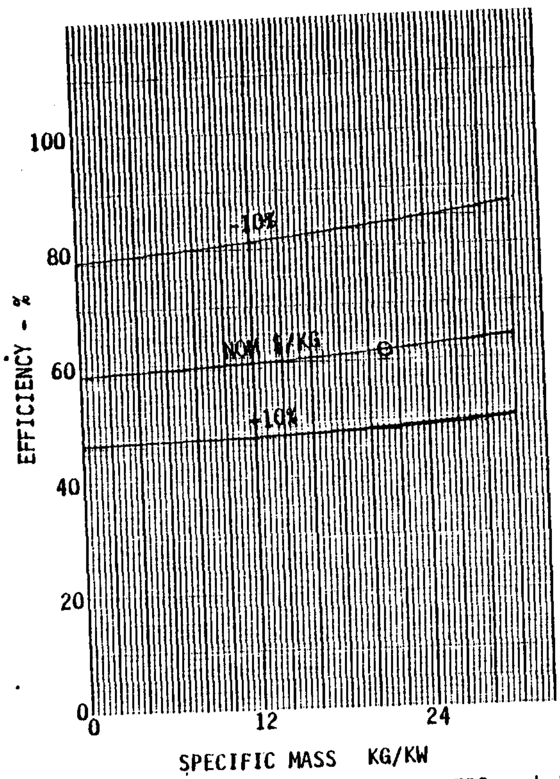


FIGURE 5-87 Interactions of αEPS and η - Group 4

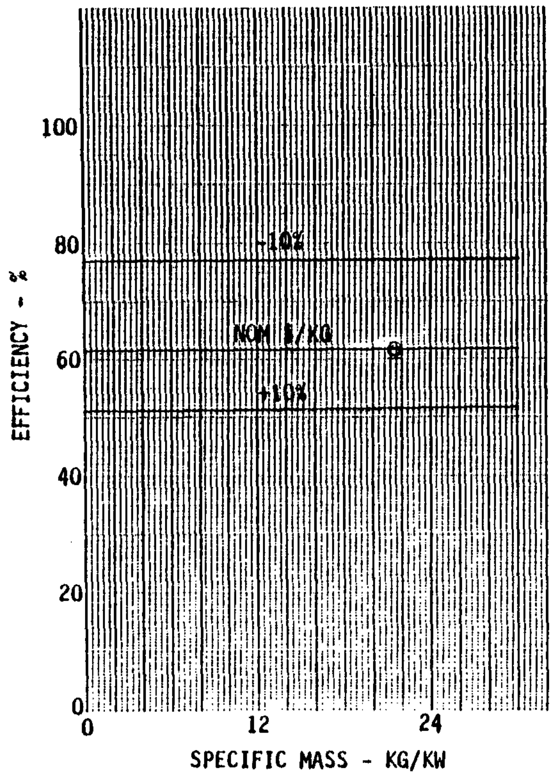


FIGURE 5-88 Interactions of αEPS and η - Group 5

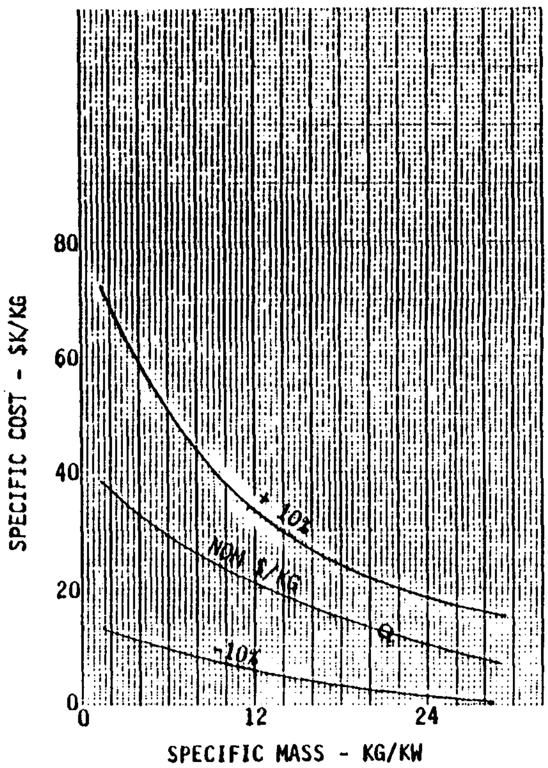


FIGURE 5-89 Interactions of αEPS and γEPS - Group 1

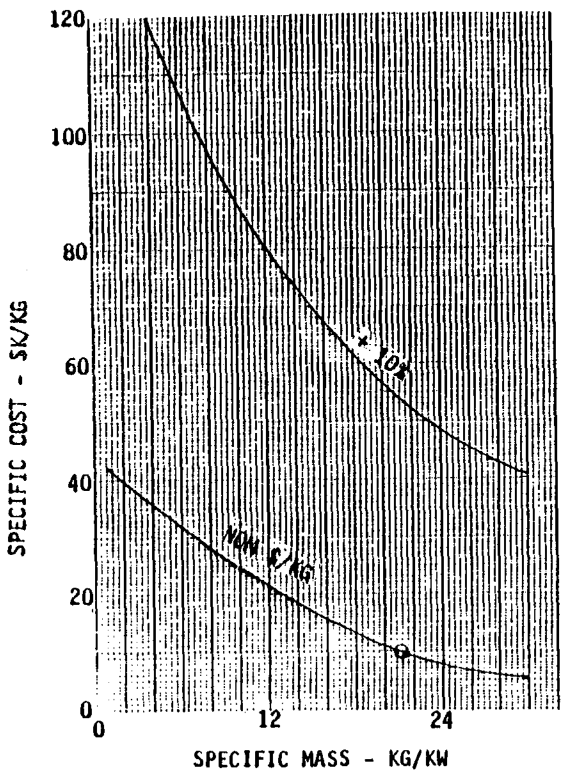


FIGURE 5-90 Interactions of αEPS and γEPS - Group 2

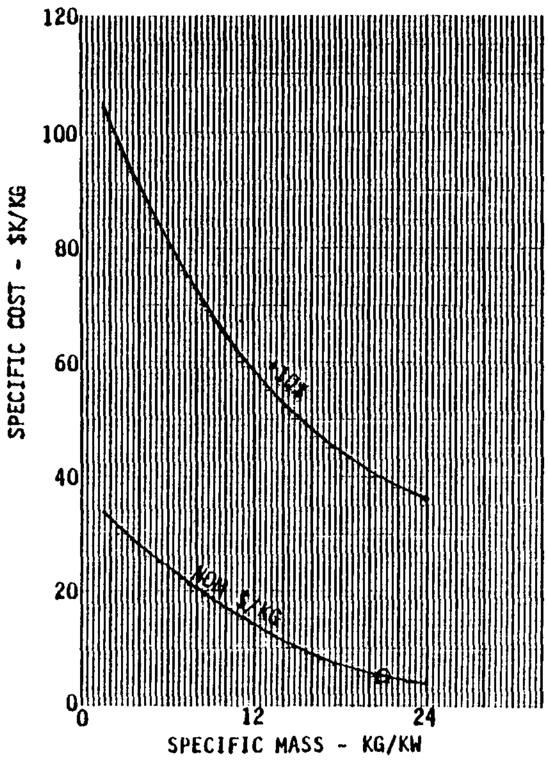


FIGURE 5-91 Interactions of αEPS and γEPS - Group 3

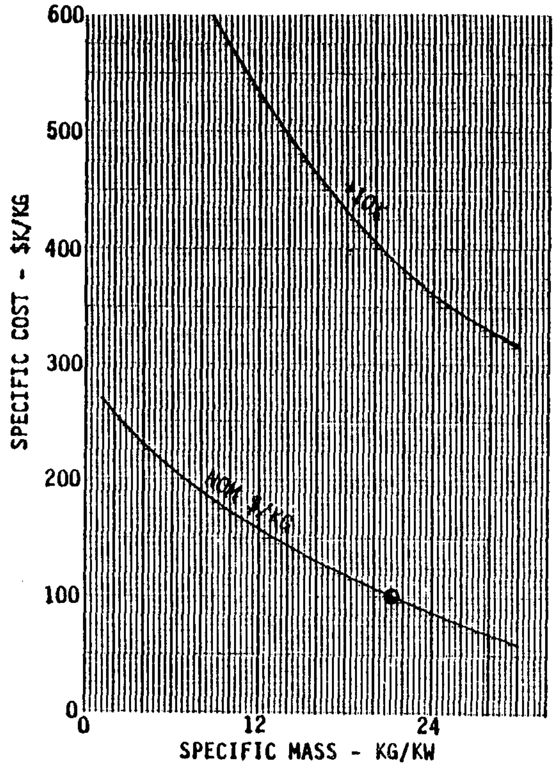


FIGURE 5-92 Interactions of αEPS and γEPS - Group 4

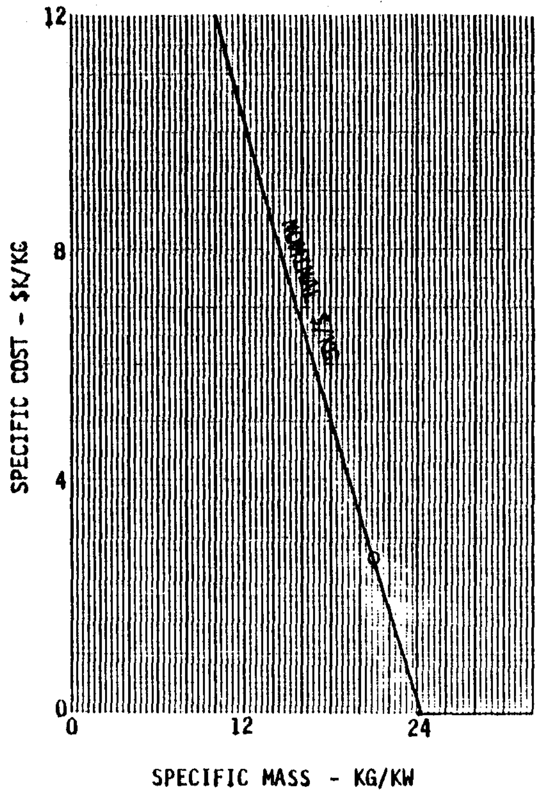


FIGURE 5-93 Interactions of αEPS and γEPS - Group 5

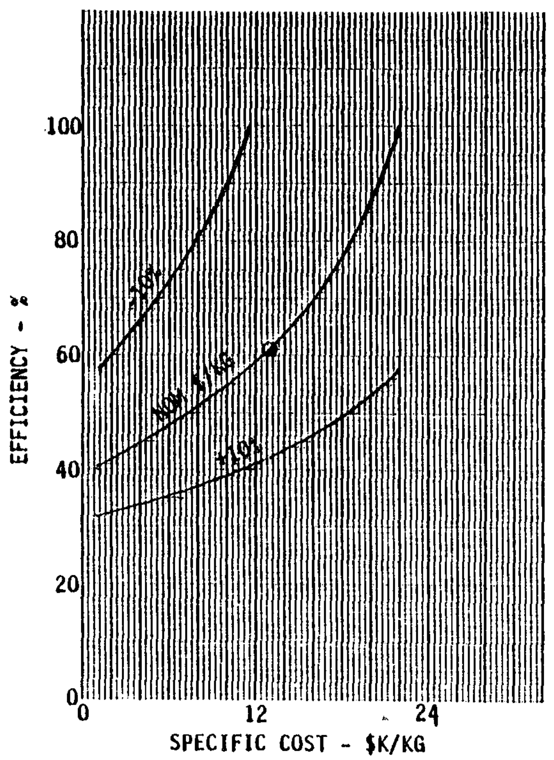


FIGURE 5-94 Interactions of γEPS and η - Group 1

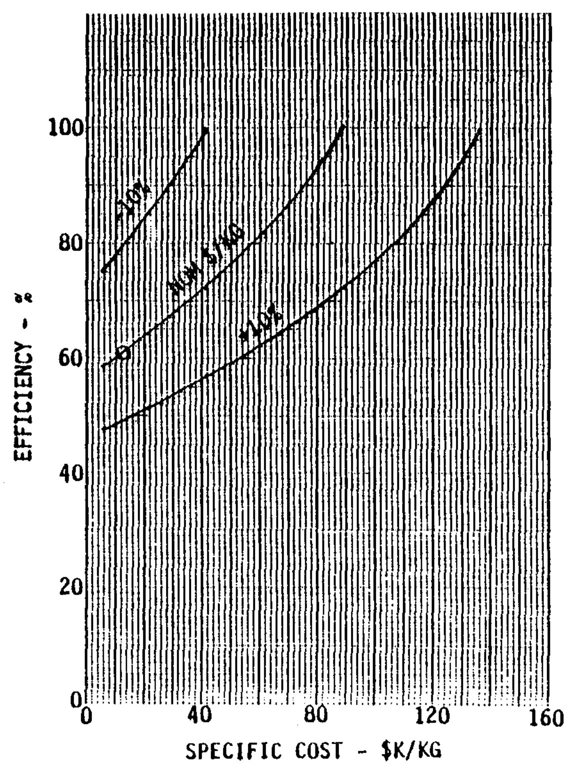


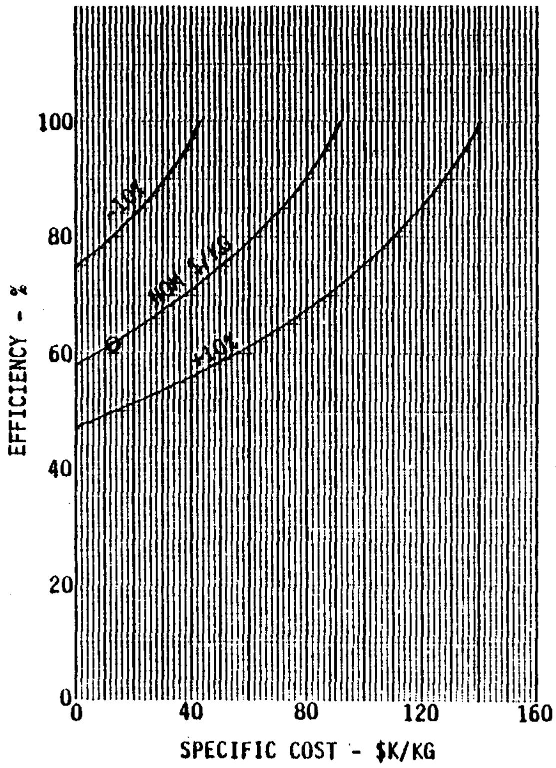
FIGURE 5-95 Interactions of γEPS and η - Group 2

afforded to realize a 50% reduction in mass. For the far-term missions, a mass reduction is even more valuable - a 400% cost growth would be an acceptable trade to halve the system weight.

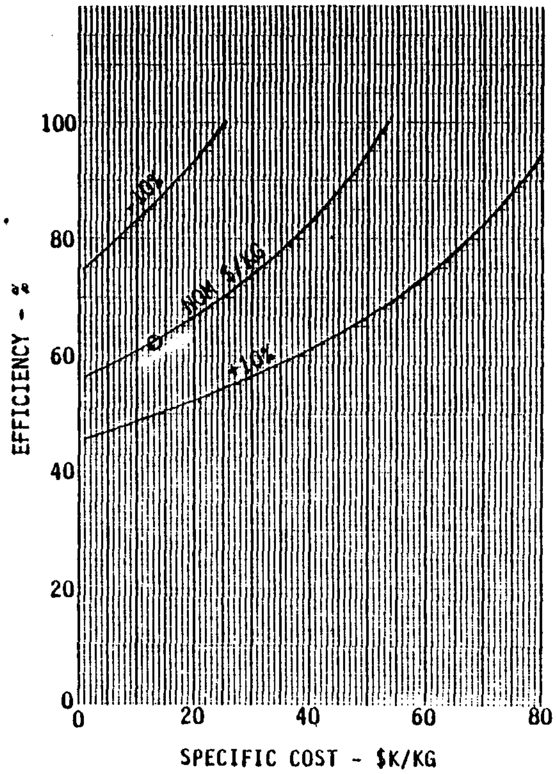
The lines of constant total mission costs are shown in figures 5-94 through 5-98 to summarize the dollar value of increased electric propulsion system efficiency. (Note the differences in the scale of the abscissa for these five graphs.) For the group 1 representative mission, the gain of one point in efficiency only warrants a 5% increase in EPS specific cost. However, the very high amounts of money involved in the time-associated costs cause a dramatic shift in emphasis for the far-term mission. For the group 5 mission (figure 5-98), attempts to maintain constant delivery costs with increasing efficiency allowed specific costs (abscissa) that were one to two orders of magnitude greater than those shown on figures 5-94 thru 5-97. The increased importance of efficiency for the far term missions is thus demonstrated.

Figures 5-99 thru 5-103 show the mission cost isograms for variation in LEO launch costs as a function of the specific mass of the electric propulsion system. It is seen that in the case of group 1 missions, some opportunity exists to trade an increase in system mass for a reduction in Earth launch costs, should this prove feasible. For the more advanced missions however, these two parameters are essentially decoupled, and no such trades are possible.

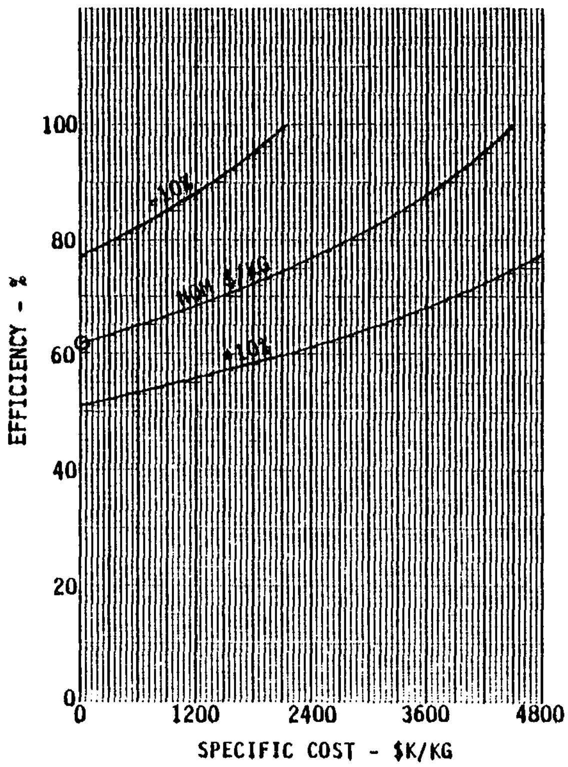
The final parameter interaction study to be reported is the synergistic coupling that was observed between the costs of the system hardware and the trip time charges. As was noted earlier, the "law of supply and demand" dictates that in a cost-optimized situation, the less expensive a quantity gets, the more of it the system will tend to utilize. Figure 5-104 illustrates this effect for the group 1 representative mission - the utility load management satellite. Note that decreasing either the EPS specific cost or the solar array specific cost will force the cost-optimum power levels to increase. This in turn results in a decrease in the time that the EPS transportation phase requires. This is shown in



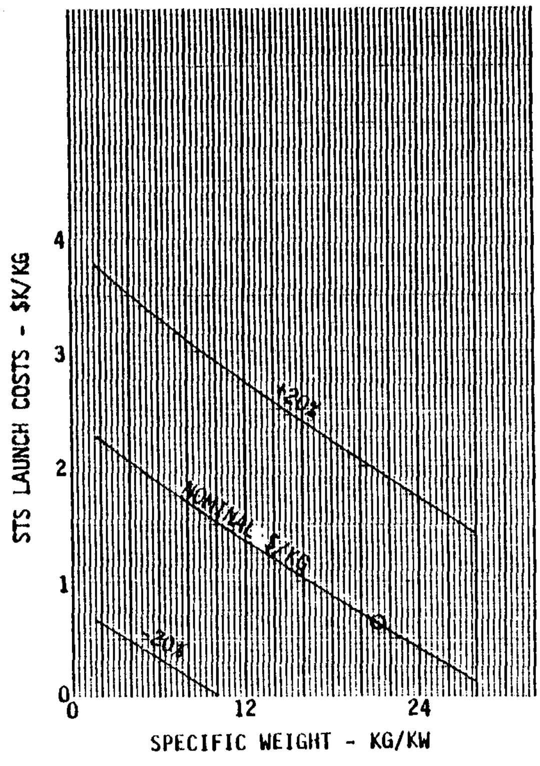
5-96 Interactions of γ_{EPS} and η - Group 3



5-97 Interactions of γ_{EPS} and η - Group 4



5-98 Interactions of γ_{EPS} and η - Group 5



5-99 Interactions of α_{EPS} and γ_{STS} - Group 1

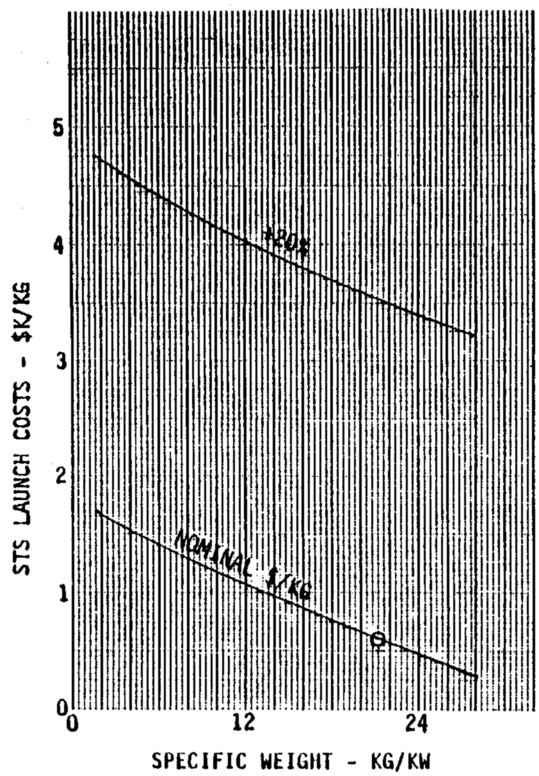


FIGURE 5-100 Interactions of αEPS and γSTS - Group 2

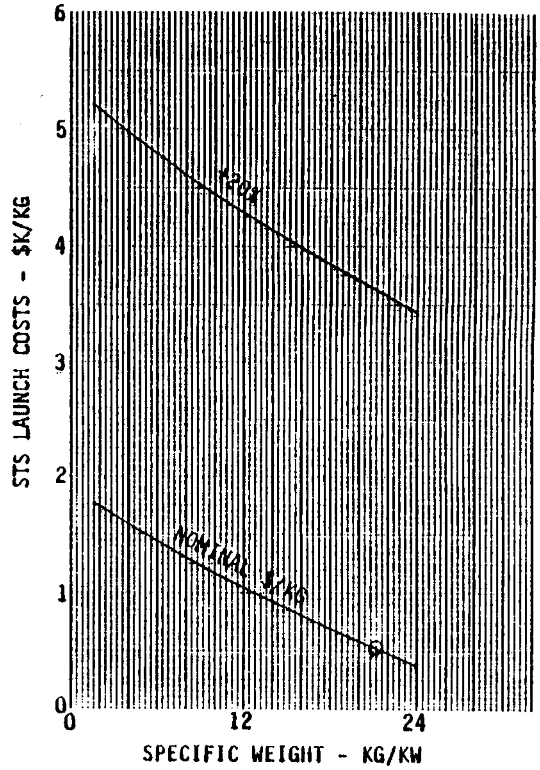


FIGURE 5-101 Interactions of αEPS and γSTS - Group 3

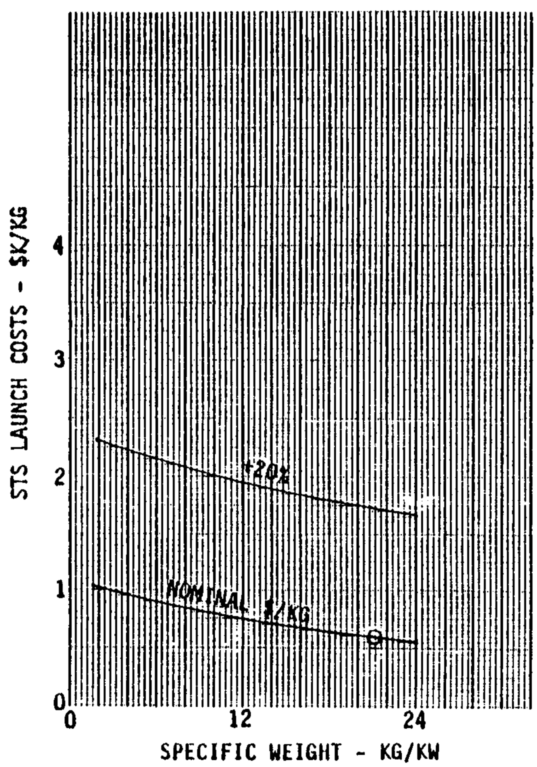


FIGURE 5-102 Interactions of αEPS and γSTS - Group 4

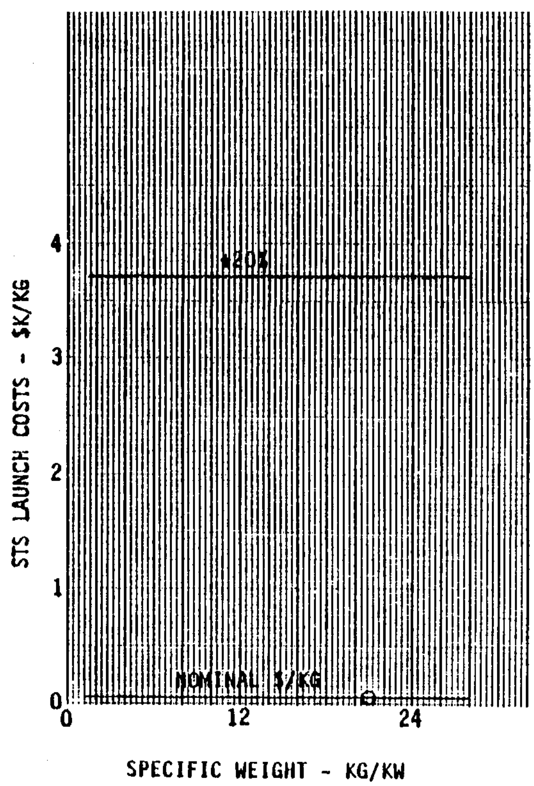


FIGURE 5-103 Interactions of αEPS and γSTS - Group 5

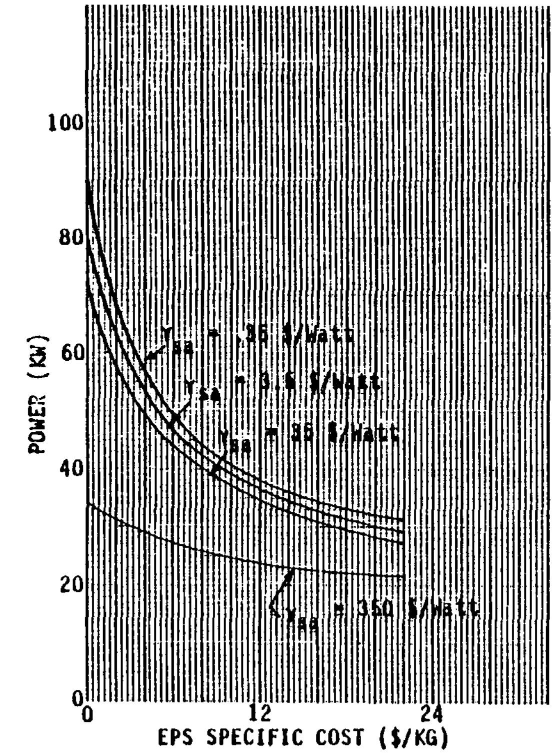


FIGURE 5-104 Cost Optimized Power Levels as a Function of Hardware Costs

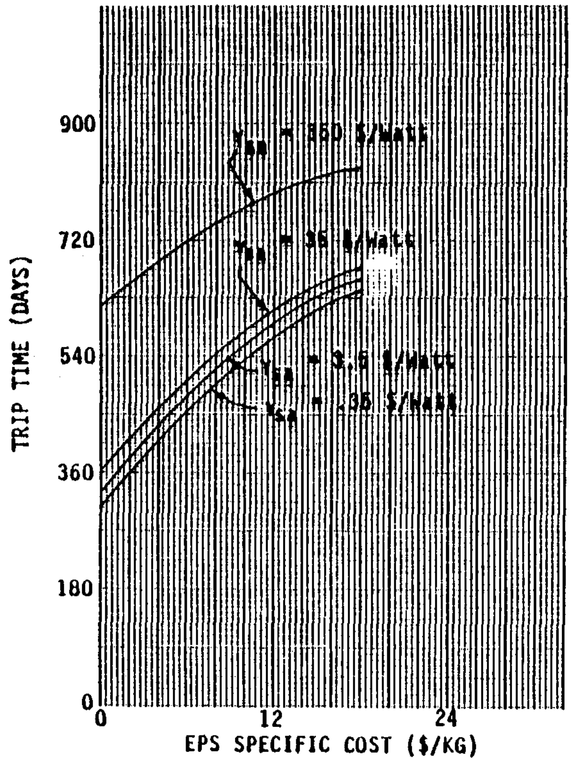


FIGURE 5-105 Cost Optimized Mission Time as a Function of Hardware Costs

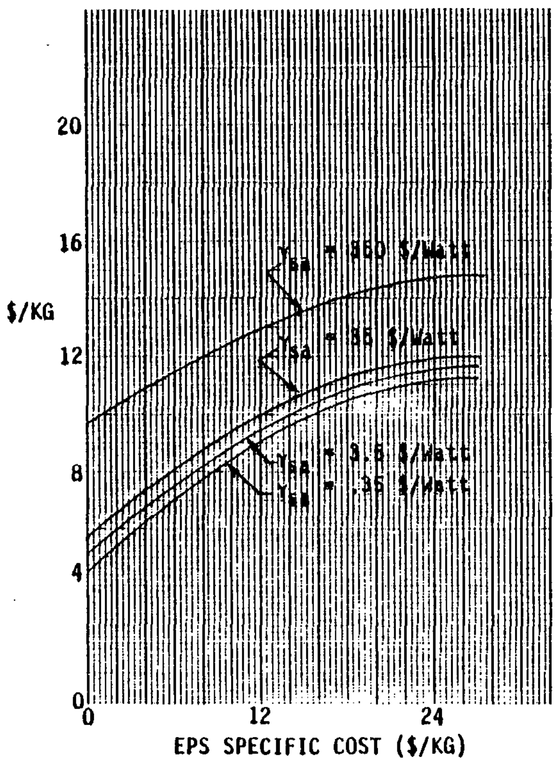


FIGURE 5-106 Cost Optimized Delivery Charges as a Function of Hardware Costs

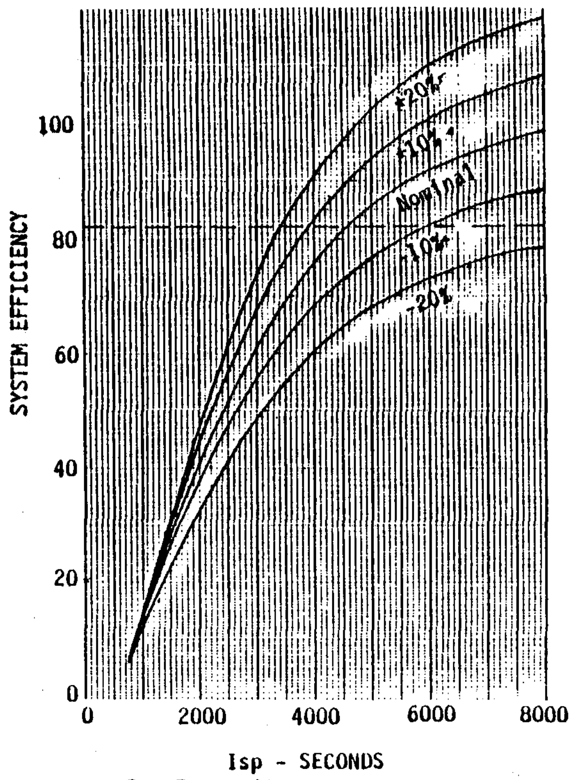


FIGURE 5-107 Effect of Scaling on Efficiency Function

figure 5-105, and of course, the reduction in mission duration also results in decreased charges for "interest" and flight operations. The total reductions in delivery charges are shown in figure 5-106, and these are a fusion of both the reduced hardware costs and the decreased trip time penalties. The total mission costs thus "benefit twice" from any decrease in the specific cost of either the electric propulsion system or the solar array.

5.6 EFFICIENCY FUNCTION IMPACTS

Throughout the study, it was noted that the optimum specific impulse for the electric propulsion systems always kept coming out in the vicinity of 3000 seconds - the nominal, state-of-the-art value. No significant changes were noted, even though all of the system parameters were varied over fairly broad ranges. The reason for this apparent "unshakability" was finally determined to be wrapped up in the characteristic shape of the efficiency curve.

Efficiency was assumed to be a function of the system specific impulse, as described by equation 4-14. This function has been well established in the literature as being a reasonably accurate representation of current mercury ion bombardment engine system technology, and this curve was fitted to the characteristics of the J-series thruster as given by the LeRC. It is thus assumed that the resulting functional relationship is an excellent starting point for this study.

Two simple modifications to this efficiency function suggest themselves. First, all points on the efficiency curve may be multiplied by a constant. This alteration is illustrated in figure 5-107, where the constant ranges from 0.8 to 1.2 and the dotted line shows the assumed upper limit (SOA = 82% of efficiency). Figure 5-108 gives the resulting shifts in the cost optimum power level and specific impulse - essentially no change.

The second simple change that may be made is to simply add a constant amount to the efficiency - across the board. This change is displayed in figure 5-109, and the corresponding shift in design points can be seen in figure 5-110, where the arrows point in the direction of increasing efficiency. A large shift in design emphasis results. Evidently, the factor that was holding the I_{SP} up around 3000 seconds is the slope of the efficiency function in that region. When a higher efficiency can be realized at lower values of specific impulse, the cost-optimization process tends to seek a lower I_{SP} in order to drive the mission duration/costs down.

To test this hypothesis, it was next assumed that the efficiency could be made independent of the EPS specific impulse as shown in figure 5-111. This resulted in a mapping into the design point space as displayed in figure 5-112. It is noted that the optimum values of the EPS specific impulse have decreased markedly. The effects on mission costs are shown in figures 5-113 and 5-114 for two different values of constant efficiency and for the group 1 representative mission. The results are similar for all members of the overall mission set. These graphs confirm the cost optimum specific impulses shown in figure 5-112 and lead to the conclusion that, if greater efficiencies can be realized at lower values of EPS I_{SP}, large savings in mission costs will accrue as a result of the decreased mission durations that become possible. This can also be seen in figure 5-115, where the shape of an efficiency characteristic that is required to attain a constant mission cost is plotted. The SOA characteristic is shown for comparison. All parameters other than efficiency are at their nominal values.

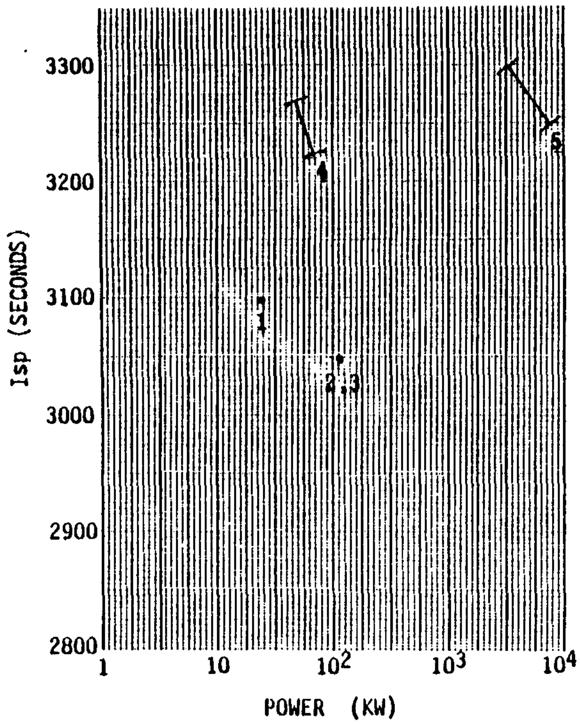


FIGURE 5-108 Effect of Scaling on Cost Optimum Design Point

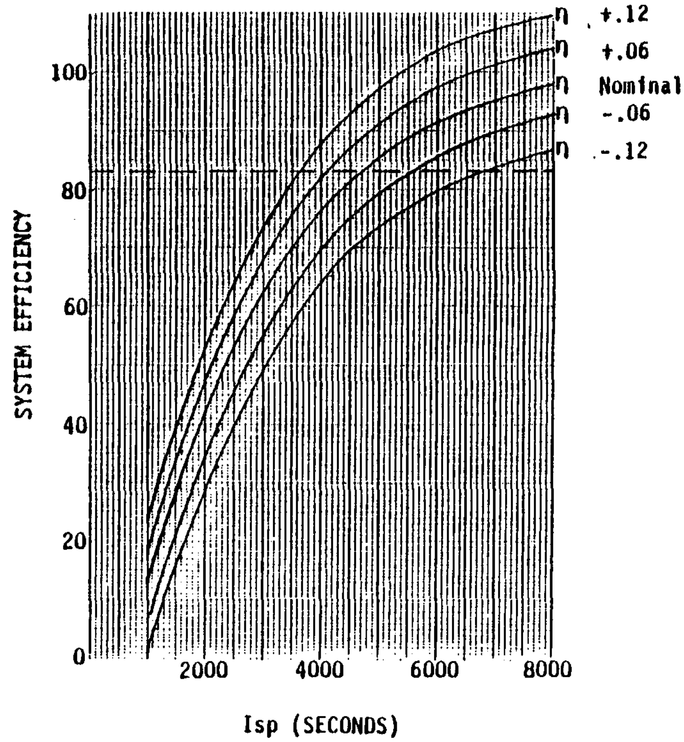


FIGURE 5-109 Effect of Translation on Efficiency Function

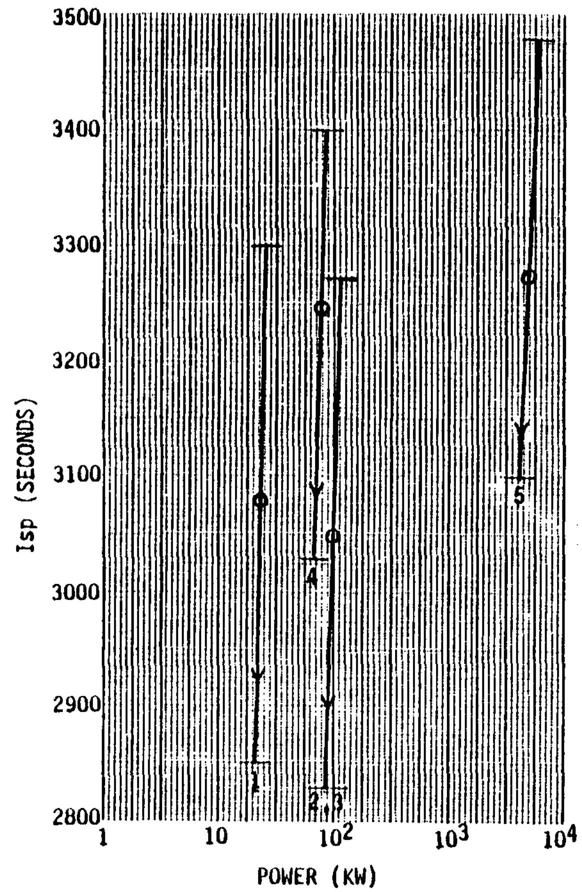


FIGURE 5-110 Effect of Translation on Cost Optimum Design Points

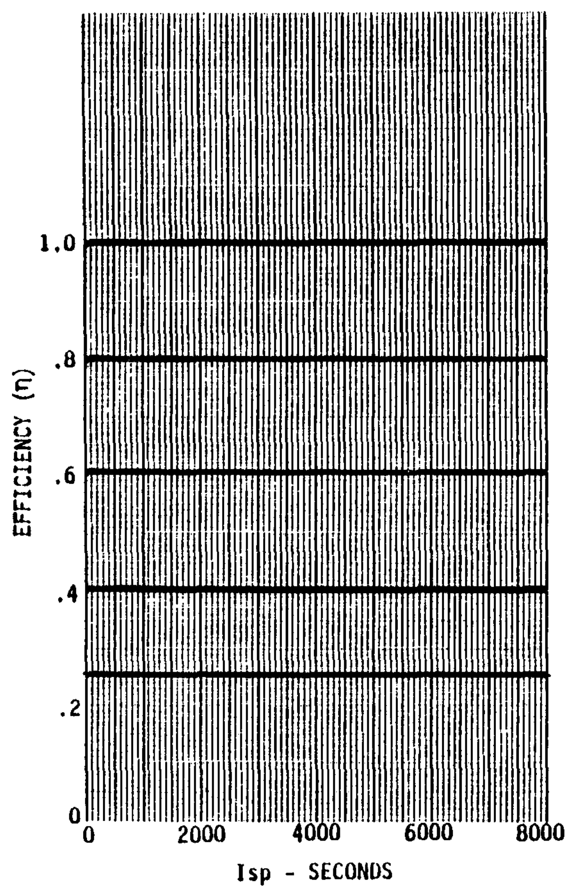


FIGURE 5-111 Constant Efficiency Functions

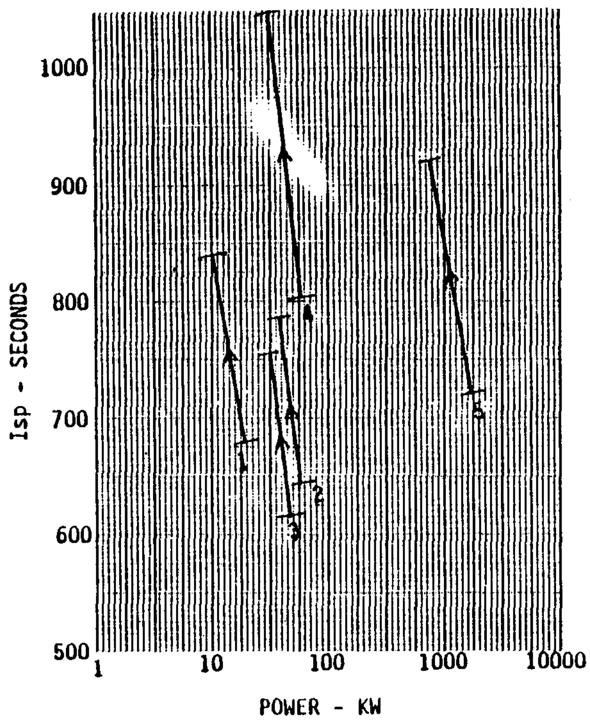


FIGURE 5-112 Effect of Constant Efficiencies on Cost Optimum Design Points

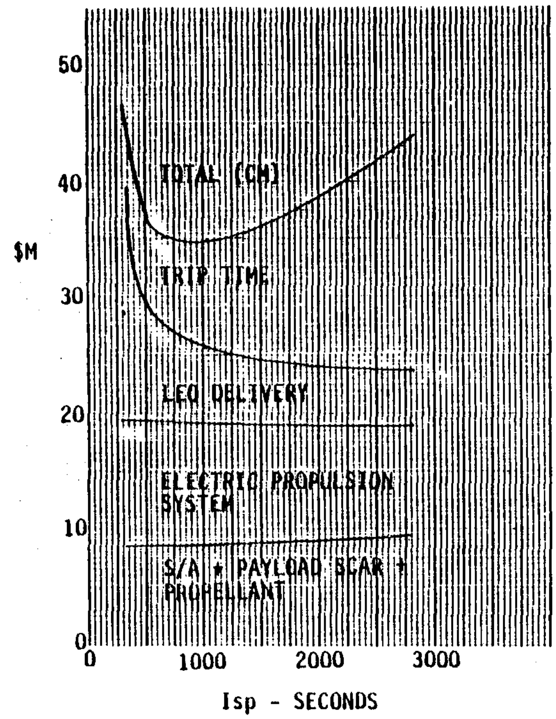


FIGURE 5-113 Transportation Costs as a Function of Specific Impulse for η = 0.5 (constant)

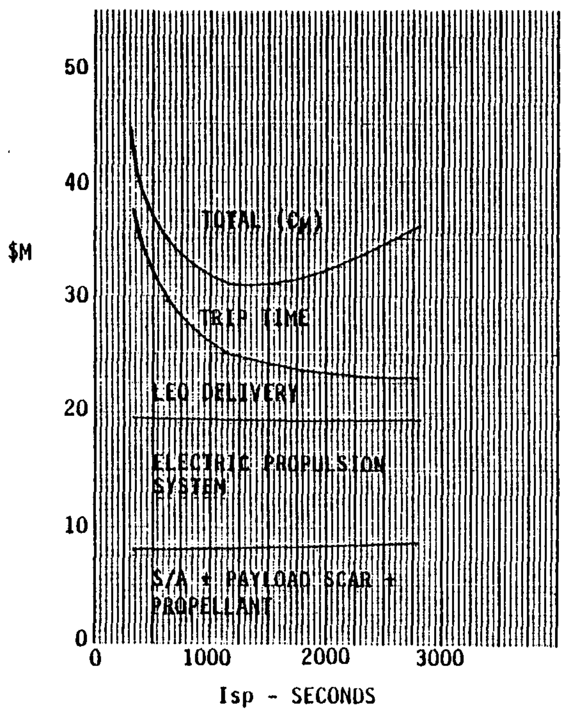


FIGURE 5-114 Transportation Costs as a Function of Specific Impulse for η = 0.8 (constant)

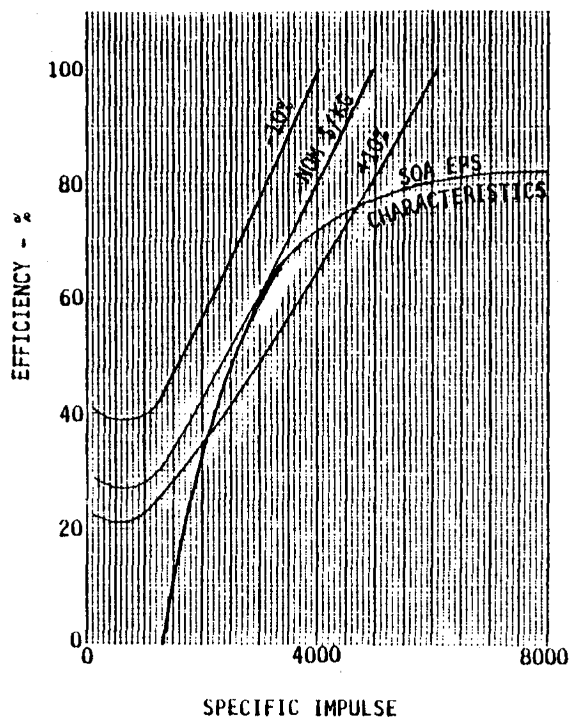


FIGURE 5-115 Efficiency Characteristics for Constant (Specific Impulse - Independent) Transportation Costs

6.0 SUMMARY OF RESULTS AND CONCLUSIONS

Missions are now being proposed wherein electric propulsion systems will be utilized for interplanetary explorations, and for auxiliary functions in Earth-orbit. Current EPS technology has been aimed toward these goals. However, as the Space Shuttle makes near-Earth space more accessible, man will attempt ever-more ambitious programs to capitalize on our present investment, and to realize the returns that are possible from space industrialization. These initiatives will require increasing quantities and qualities of propulsive support. The purpose of this study was to determine the directions for future EPS technology advancement efforts that offer the best opportunities for meeting the challenges that lie ahead. This objective was met by employing a system level cost model as a tool for evaluating the performance of a baseline electric propulsion system across a representative set of future near-Earth space missions. Sensitivities, benefits, and impacts were then established with regard to the assumptions concerning the EPS technology, the mission characteristics, and the supporting systems.

The selected mission set was comprised of 30 missions which spanned the next three decades and "orbits" that ranged from within the upper reaches of the atmosphere to beyond the Earth's sphere of influence. Payload masses ranged from a few hundred kilograms to tens of thousands of metric tons with corresponding dimensions from a little over a meter to several kilometers across. To aid in the evaluation of technology drivers, the full set was divided into 5 groups of missions. Figure 6-1 depicts the missions taken as representative of each group. Performance parameters were determined for six "types" of trajectories which encompassed the mission set. In addition to advancements to enhance EPS cost-effectiveness (to be discussed below), two other issues were seen as crucial to the applications of electric propulsion in Earth-orbit. First, the effects of solar occultations must be minimized, either via optimum launch scheduling, or by decreased ion thruster start-up time/power requirements. Second, the effects of passage thru the radiation belts must be minimized, either via the discovery of new solar cell types, by

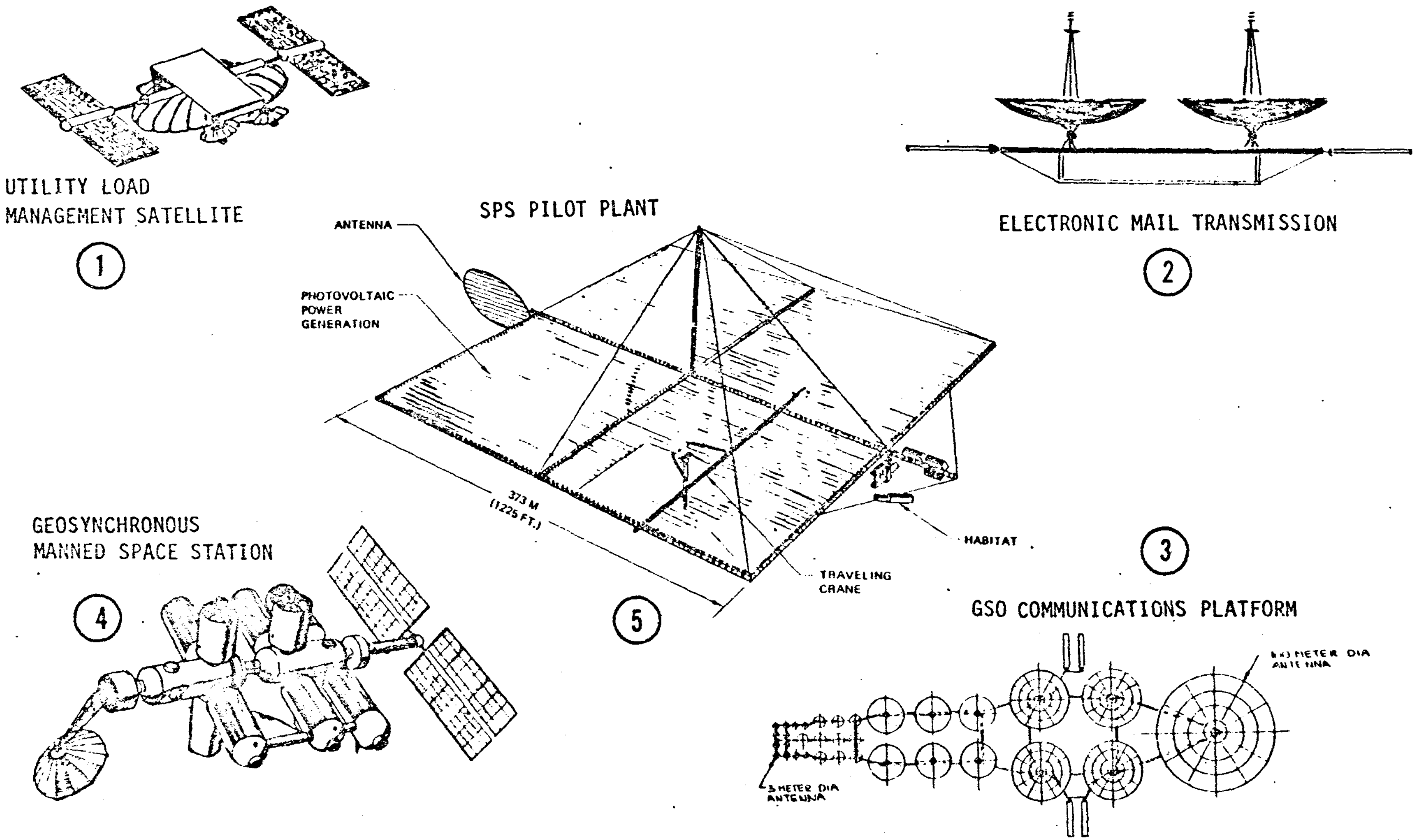


FIGURE 6-1 Five Representative Missions

including "over-powering" provisions in new EP engine systems, or by the development of techniques for in-flight annealing of the solar arrays. Drag cancellation in low-Earth orbit was seen as a good potential application for electric propulsion. It is recommended that more study be devoted to that arena in order to more fully understand this opportunity.

The cost model that was constructed treated the electric propulsion system as a "black box" which could be represented by only a handful of top-level descriptors (see figure 6-2). Appropriate characterization of the missions, their payloads, and the interfacing systems, allowed the generation of the major elements of mission costs. Initial EPS inputs corresponded to a baseline system comprised of four of the current (SOA) technology "bi-mod" engine systems powered by two 12.5 kw, flexible/fold-out, solar array wings. Results for this baseline system indicated transportation charges to GEO of

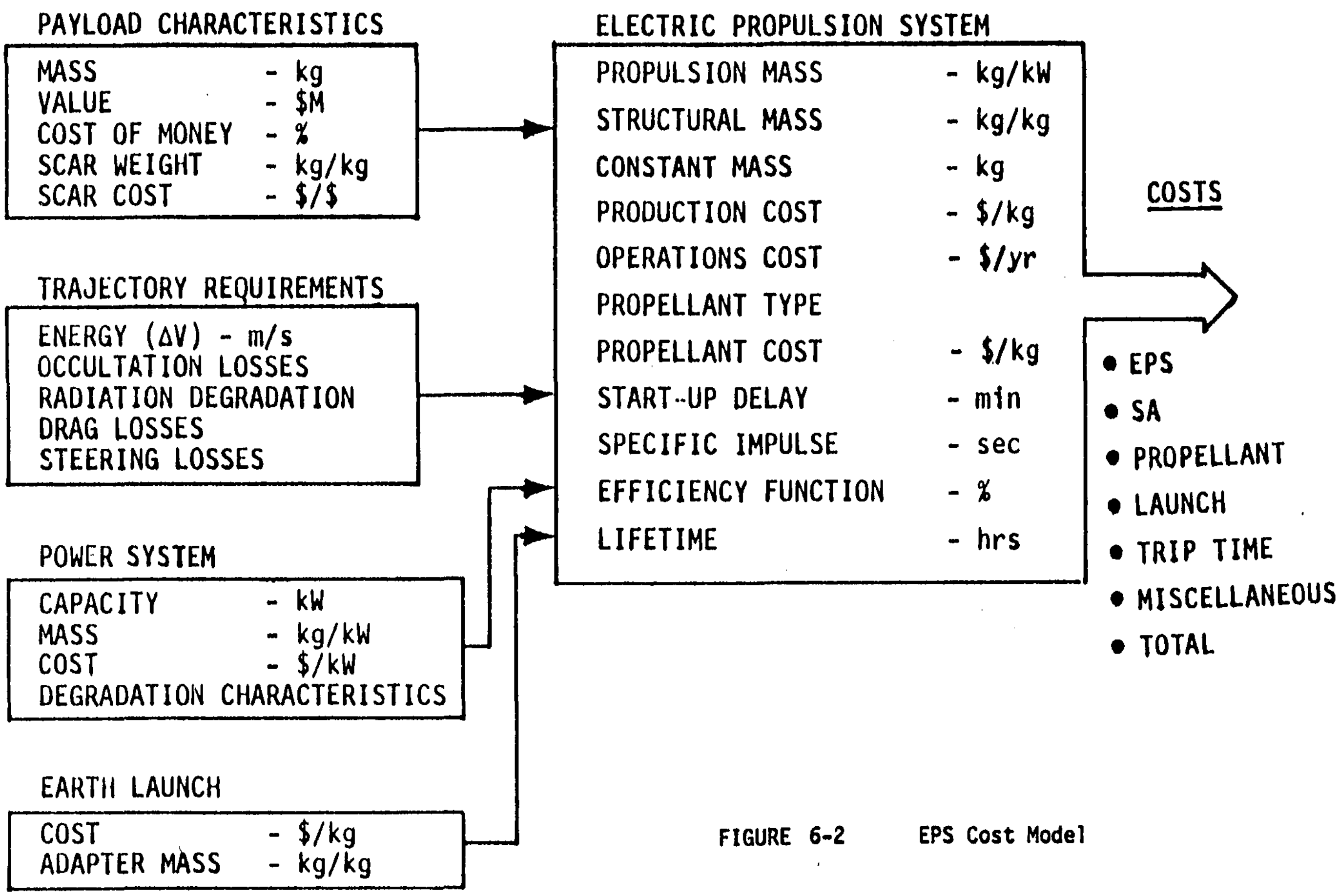


FIGURE 6-2 EPS Cost Model

the same magnitude as early STS-era projections. Payload capacities offer improvement over Shuttle/two-stage IUS capabilities by a factor of 2 to 4, primarily limited by EPS lifetime. The addition of "spare" engine systems can effectively eliminate the lifetime limit, but delivery costs become non-competitive.

Three other design philosophies were investigated for comparison to the state-of-the-art: minimum power, minimum time, and minimum cost. The first of these assumed adding sufficient amounts of solar array and EPS hardware to avoid exceeding lifetime constraints. An optimum specific impulse can be found which minimizes power source requirements. This was seen to be in the range of 2850 to 3100 seconds across the mission set, with the value of the the minimum power increasing roughly in proportion to the mass of the payload to be transported. The second philosophy assumed the availability of an infinite amount of power (and the EPS hardware to utilize it) in order to reduce the mission duration to an absolute minimum. This case seemed to be of interest since the minimum time was independent of system/payload considerations, being solely a function the trajectory parameters and the EPS technology level. An optimum specific impulse was found to exist to minimize transfer time and was seen to be in the range of 2700 to 3200 seconds for the selected mission set. A derivative of this philosophy was examined wherein mission duration was constrained to an arbitrary, but fixed value. Similar results to the time minimized case were noted regarding EPS technology. In both cases, due to the large amounts of power required, it was noted that the specific cost of the electrical energy source was a major determinant of the delivery charges, and therefore a good candidate for the expenditure of advanced development resources.

Most of the study attention was devoted to the cost-optimum design philosophy. A most favorable specific impulse and EPS power level was found to exist for each of the 30 missions under study. For this philosophy, the model predicted a monotonic decline in total transportation costs as electric propulsion systems, their power sources, and their payloads, grow ever larger. In general, optimum Isp was in the range of 2600 to 3750 seconds. For early missions, the EPS size and mass was seen to be comparable to that of the payload and the cost-optimum design point was generally quite close to the state-of-the-art. As a result, the greatest decreases in mission costs/performance were found to stem from improvements in EPS production costs and specific weights. However, for later, more difficult missions, payload sizes/costs are generally much larger than those of the EPS, and thus improvements in these factors are not nearly so beneficial. For these missions, the cost penalties associated with the long, low-thrust, mission times become most important. Investigations of the interactions (trade-off potentials) between the various electric propulsion technology parameters resulted in the conclusion that for the more advanced missions, the greatest benefit would come about from improvements in the system efficiency. It is even possible to suffer degradation in specific weights/costs to gain improved efficiency and still realize a benefit in total costs.

All substudies had shown the current (SOA) specific impulse of 3000 seconds to be nearly optimum across the mission set, for all 4 design conditions, and under all variations of other EPS technology parameters. Analysis revealed that this was the result of the shape (primarily the slope) of the efficiency function that characterizes the ion bombardment thruster. A curve was derived which produced constant mission costs, regardless of the value of Isp. This function is shown in figure 6-3, along with a plot of the state-of-the-art characteristic. The differences between the curves indicate that moderate values (>50%) of efficiencies in the lower ranges of specific impulse (around 1000 seconds) hold the potential for significant reductions in total transportation charges. Further studies are recommended to determine the development potential for propulsion components/systems in this regime.

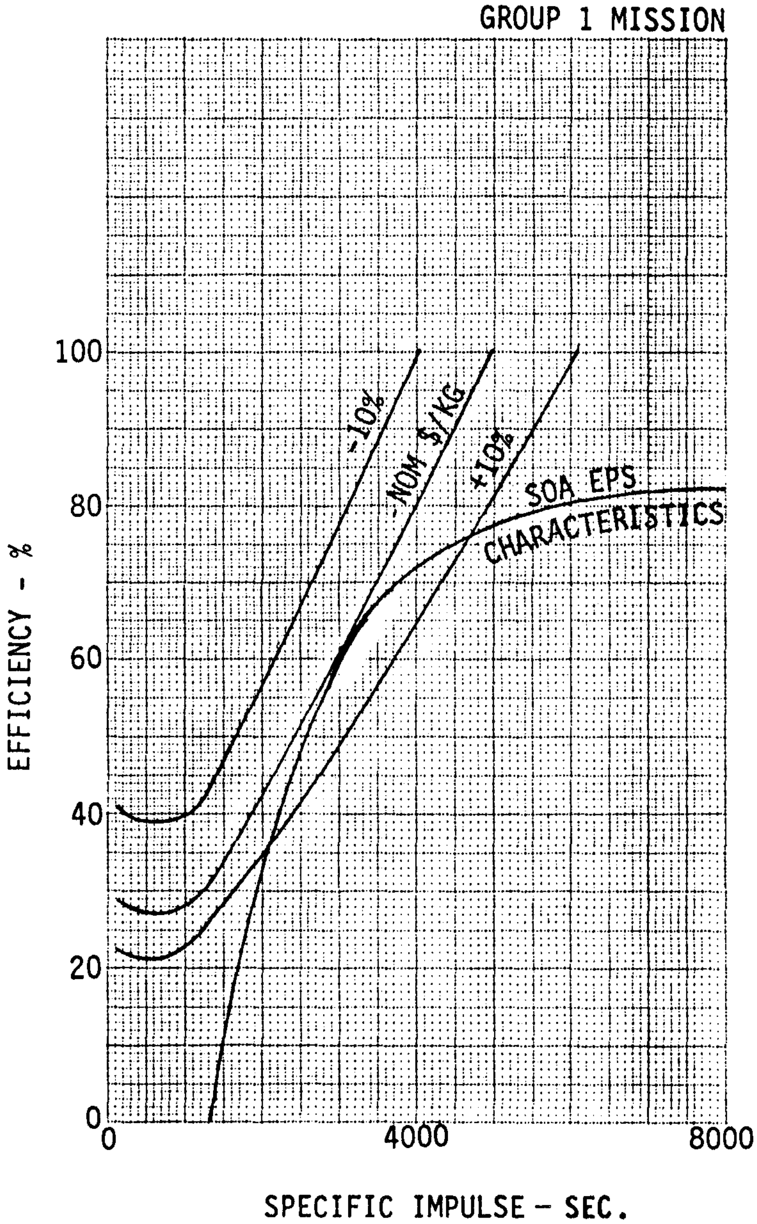


FIGURE 6-3 Efficiency Characteristics for Constant (Specific Impulse - Independent) Transportation Costs

APPENDIX A

MEMBERS OF THE BASELINE MISSIONS SET

1-1 Geosynchronous-Based Satellite Maintenance Sortie
2-0 Geosynchronous Space Station
3-0 Orbiting Lunar Station
4-0 Nuclear Waste Disposal
5-0 Satellite Power Systems
6-0 SPS Pilot Plant
9-0 Nuclear Fuel Location System
11-1 Marine Broadcast Radar
12-0 Astronomical Telescope
14-0 Global Search and Rescue Locator
20-0 Multinational Air Traffic Control Radar
25-0 Electronic Mail Transmission
30-0 Personal Communications/Wrist Radio
34-0 Personal Navigation/Wrist Set
34-1 Near-Term Navigation Concept
37-1 Power Relay Satellite
38-1 Utility Load Management Satellite
44-0 Space Construction Facility
46-0 Tethered Satellite (Atmospheric Explorer)
48-0 Gravity Gradient Explorer
49-0 Geosynchronous Communications Platform
50-0 Earthwatch (Resources Mapper)
51-0 Orbiting Deep Space Relay Station
52-0 SPS Orbit Transfer System Recovery
54-0 Magnetic Tail Mapping
55-0 Iceberg Dissipator
56-0 Soil Surface Texturometer
58-0 Technology Development Platform
60-0 Space Based Radar - Near Term
61-0 Space Based Radar - Far Term

MISSION DATA SHEET

MISSION	NO.
Geosynchronous - Based Satellite Maintenance Sortie	1-1

OBJECTIVES

- To perform repair, refurbishment, refueling, and equipment update on geosynchronous satellites.

GLOBAL IMPLICATIONS

- Assumes that a fairly large manned space station exists at geosynchronous altitude

TRANSPORTATION SCENARIO

- Maintenance vehicle and stocks of spare parts are based at GSO space station
- On each sortie, the maintenance vehicle visits one or more satellites (at or near geosynchronous altitude) and performs automated servicing in situ.
- Vehicle returns to GSO space station for resupply and storage between sorties

INITIAL ORBIT

ALTITUDE	Geostationary
INCLINATION	
ECCENTRICITY	
LONGITUDE	
OTHER	

FINAL ORBIT

ALTITUDE	35,800 km
INCLINATION	various
ECCENTRICITY	0
LONGITUDE	various
OTHER	

TRANSPORT TIME

days

REUSABLE	DISPOSABLE
X	

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

SPACE

- Geosynchronous manned space base

COST ESTIMATES

PROGRAM COST

$162M

PAYLOAD VALUE

$32.4M

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

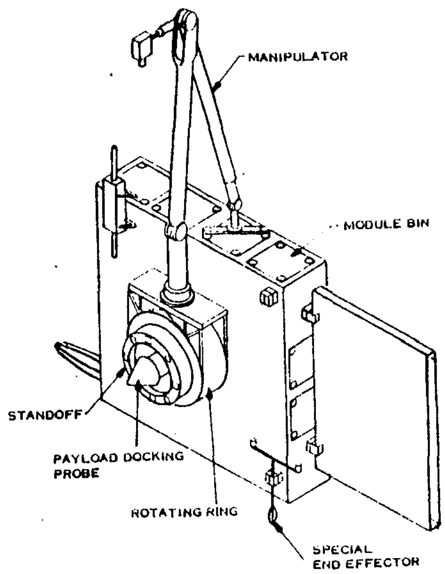
DOCUMENTATION SOURCES

Plus study report, D180-19783-2, ¶ 4.0

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION	
. Servicing unit requirements: . Power - 400 w. peak . Command - 1024 bps . Telemetry - Real-time TV . Attitude Control . Rendezvous and Docking . Man-in-the-loop control from space base	MASS: 1031 kg ▷1 SIZE: 3 x 8 x 1.5 m LIFE: MAX. Gs: 0.1 (shock) ONBOARD POWER TYPE QUANTITY None VOLTS FREQUENCY POINTING: 0.1^{o} ATTITUDE CONTROL STATION-KEEPING



▷1 Empty mass = 467 kg + 12 modules at 47 kg each

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

IOC: 1994

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION	NO.
Geosynchronous Space Station	2-0

OBJECTIVES

o Sensing of Earth resources and science measurements

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

Delivered in 9 modules, transported from LEO to GSO separately, then mated on-station.

INITIAL ORBIT

ALTITUDE 300 - 500 km

INCLINATION $28\frac{1}{2}^{0}$

ECCENTRICITY

LONGITUDE

OTHER

FINAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

GEOSTATIONARY

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

Space Shuttle

SPACE

. Station modules
. Applications/science mod.
. Crew transfer vehicle
. Resupply modules

COST ESTIMATES

PROGRAM COST

$3.2B

PAYLOAD VALUE

$635M

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

FSTSA, D180-20242-1, p. 6

& D180-20242-2, Sec. 3.2.2

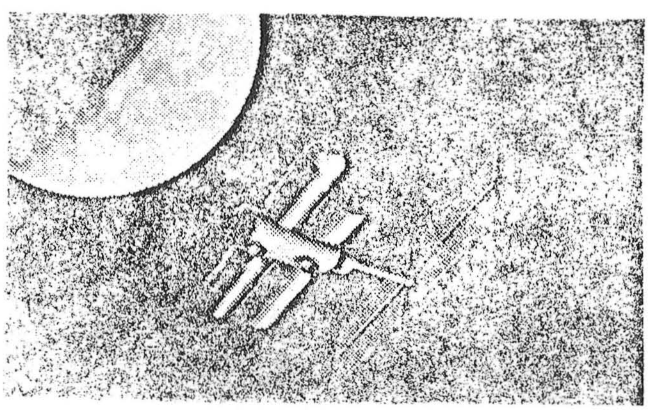
REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION Nine station modules provide quarters for the eight-man crew, supporting subsystems and consumables. The functions provided by these modules are as follows: two core modules house basic station subsystems and the docking provisions for all the other modules; two modules each provide crew quarters for four men and eight in an emergency; two modules serve as command/control centers with one also providing the radiation shelter; one module provides the electrical power system; one module is used for the galley and recreation purposes; and the final module houses cryogenics and provides storage.

A unitary station option for this mission is also described in the FSTSA technical report. The eight-man station options require crew rotation and resupply at six-month intervals. Delivery and return payloads are 25,200 kg (55,400 lb) and 14,800 kg (32,600 lb) respectively.

A brief study was made of transportation requirements for a 50-man geosynchronous station. The selected crew rotation and resupply interval was 2 months with delivery and return payloads of 40 100 kg (88,400 lb) and 23 100 kg (50,900 lb) respectively. The 50-man station delivery mass was 423 000 kg (931,000 lb).



PREVIOUS STUDY CONSTRAINTS

. Delivery in 2 pieces was dictated by size of transport system which was also to be used for the recurring function of resupply and crew rotation.

MASS 148,400 kg

SIZE 4.5 x 35.4 x 60.5 m

LIFE

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 75 kW

VOLTS

FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		
MANNED SYSTEM	X	
REPAIRABLE SYSTEM		

PERFORMANCE PARAMETERS

TRAFFIC PROJECTION

- 3 Stations in GSO eventually
- 6 Years apart (IOCs)

IOC

1993

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION	NO.
Orbiting Lunar Station	3-0

OBJECTIVES	GLOBAL IMPLICATIONS
o Perform a broad spectrum observation of the lunar surface o Support manned surface sorties o Support/control unmanned orbital and surface operations	

TRANSPORTATION SCENARIO

- o Delivered in 8 to 10 sections to LEO
- o Individual transport to lunar orbit
- o Rendezvous and docking (final assembly)
- o Transport resupply modules (one-way/two-way) periodically
- o Transport new modules when required to accommodate expansion of base operations and update obsolescent equipement

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Shuttle

FINAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Lunar

TRANSPORT TIME

REUSABLE	DISPOSABLE

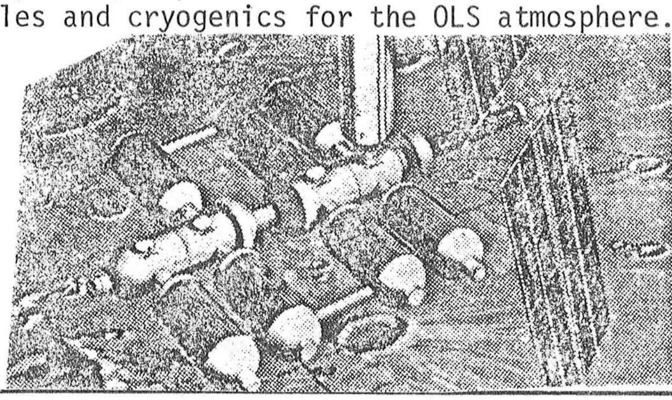
SUPPORT SYSTEM REQUIREMENTS	COST ESTIMATES	DOCUMENTATION SOURCES
GROUND	PROGRAM COST $1.45B	FSTSA, D180-20242-1, p. 9
LAUNCH	PAYLOAD VALUE $685M	
	TRANSPORTATION ALLOWANCE	
SPACE o Station modules o Lunar transport vehicle o Crew transport vehicle	REVENUE PROJECTION	REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION The flight configuration for a modular station is shown below. Ten modules are required to provide the required volume for a crew of eight, subsystems, and consumables. An eleventh module contains science equipment and sensors. A unitary OLS could also be employed and would require only one habitat module rather than nine. The unitary option is described in the FSTSA technical report.

Two LTV's each provide capability to conduct a 4-man 28-day surface exploration. Landing and ascent payloads are 14 900 kg (33,000 lbs) and 11 500 kg (25,400 lbs) respectively. Exploration payloads include a lunar vehicle (LRV) and lunar flying vehicle (LFV). The LTV's also serve as emergency vehicles to transport the OLS crew back to Earth orbit should the OLS require evaluation or to rescue a crew stranded on the lunar surface.

A combination crew rotation/resupply flight occurs at 109 day intervals. Typical delivery and return payloads are 58 400 kg (128,760 lbs) and 6 100 kg (13,400 lbs) Crew rotation is accomplished through use of a crew transfer vehicle (CTV). The CTV is sized to provide quarters for up to 8 crewmen during transits between Earth and lunar orbit. The resupply module (RM) is a pressurized container that includes bulk cargo (e.g., food, clothes, etc.) for both OLS and LTV. The module is sized for a basic resupply interval of 109 days plus 55 days for contingency. The fluid module (FM) provides propellant to completely replenish one LTV and all lunar mobility vehicles and cryogenics for the OLS atmosphere.



PREVIOUS STUDY CONSTRAINTS

MASS
221,000 kg

SIZE (30m solar arrays)
4.3 x 12.8 m

LIFE

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 150 kW

VOLTS -

FREQUENCY DC

POINTING
communications/science

ATTITUDE CONTROL
3 axis

STATION-KEEPING
lunar orbit maintenance

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		
MANNED SYSTEM	X	
REPAIRABLE SYSTEM		

PERFORMANCE PARAMETERS

TRAFFIC PROJECTION

IOC
1996

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION
Nuclear Waste Disposal

NO. 4-0

OBJECTIVES

- To achieve a safe and economical long-term storage of nuclear waste material

GLOBAL IMPLICATIONS

- Assumes that an Earth-bound storage method cannot be found which is environmentally acceptable,
- Assume nuclear wastes will continue to be (judged) valueless and thus disposible.

TRANSPORTATION SCENARIO

- Deliver to LEO via Space Shuttle
- Transport to destination orbit via electric propulsion
- Recover/reuse electric propulsion system (???)

1 Another potential destination would be a circular orbit in the middle of the Van Allen belts

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE 750,000 km 1

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Processing/repackaging center

LAUNCH

- Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

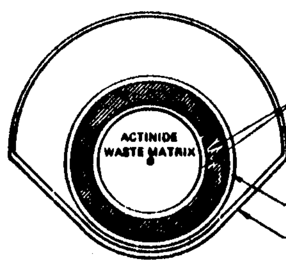
- FSTSA, D180-20242-1, p 14
- ATR-76(7365)-1, Vol. III Aerospace study, Pg 39 (CS-4)
- ATF-75(7365)-2, Pg 129

REVISION DATE: 6/2/78

PAYLOAD DATA SHEET

DESCRIPTION

- Refined and shielded actinides
- Thermionic conversion of waste heat to electricity supplies power for electrical propulsion system
- Ultra-high reliability required



PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- up to 1 mission/week

IOC

1985

REVISION DATE: 9/22/78

MASS

3250 kg

SIZE

3 m pseudo-sphere

LIFE

10^6 years

MAX. Gs

high

ONBOARD POWER

TYPE: thermionic

QUANTITY: 50-75 kW

VOLTS:

FREQUENCY: DC

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		X
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM		X

PERFORMANCE PARAMETERS

MISSION DATA SHEET

MISSION
Satellite Power Systems

NO.
5-0

OBJECTIVES

To continuously and economically produce solar-derived electric power for general commercial and industrial use on Earth.

GLOBAL IMPLICATIONS

- Significant technical advances are required.
- National commitment is required
- International agreements are required to assure safety, etc.

TRANSPORTATION SCENARIO

- Transport to low Earth orbit
- Assemble and perform initial checkout
- Transport to GSO, module by module
- Docking and final assembly/checkout in GSO

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Shuttle

FINAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Geostationary

TRANSPORT TIME

REUSABLE | **DISPOSABLE**

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Receiving antenna
- Distribution network

LAUNCH

- HLLV

SPACE

- LEO construction bases
- GSO maintenance bases
- Orbit transfer systems

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

- FSTSA,D180-20242-1, P 15
- ATR-76(7365)-1, Vol III Pg 36 (CS-1)
- ATR-75(7365)-1, Pg 127 (CS-3)

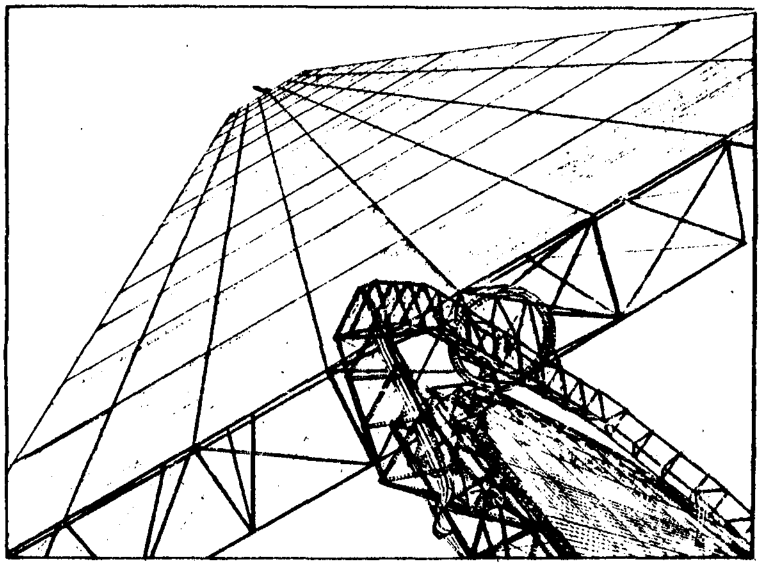
D180-20242-2, Sec. 3 . 8 and D180-24071-1 thru -7, March 1978

REVISION DATE: 9/22/78

PAYLOAD DATA SHEET

DESCRIPTION

- 224 silicon solar cells (6.55 x 7.44 cm)/panel
- 364,156 panels/bay
- 128 bays/satellite
- Thermal annealing @ 500° C by laser
- Slip rings for power transfer to MPTS
- Hexagonal antenna with Gaussian taper and integral klystron subarrays
- Transport as 8 modules (2 with antennas and 6 without)



MASS

100,000 MT

SIZE

5.35 x 21.4 x .5 km

LIFE

30 years

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 17GW

VOLTS 40kV

FREQUENCY DC

POINTING

solar & antenna control

ATTITUDE CONTROL

3 axis

STATION-KEEPING

±10 km E-W & 0.1° N-S

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

- 5 to 10 GW of electrical power output (on the ground)
- 2.4 GHz power transmission

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 1 to 4 per year after initial installation

IOC

2002

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION	NO.
SPS Pilot Plant	6-0

OBJECTIVES	GLOBAL IMPLICATIONS
To conduct an engineering demonstration of the orbital construction of a large satellite and of the generation of megawatt levels of electricity on-orbit.	• Requires partial commitment to SPS program

TRANSPORTATION SCENARIO

- Assemble in Low Earth Orbit (~ 1 year)
- Perform inital testing (~ 1 year)
- Transport to GSO
- Perform SPS test/demo program

INITIAL ORBIT

ALTITUDE / INCLINATION / ECCENTRICITY / LONGITUDE / OTHER: Shuttle

FINAL ORBIT

ALTITUDE / INCLINATION / ECCENTRICITY / LONGITUDE / OTHER: Geostationary

TRANSPORT TIME

180 days

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Rectenna 1.5 x 1.5 km

LAUNCH

- Growth Shuttle

SPACE

- On orbit Construction Crew of ~ 15 people ~ 1 year
- ~~Space station — LEO~~

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

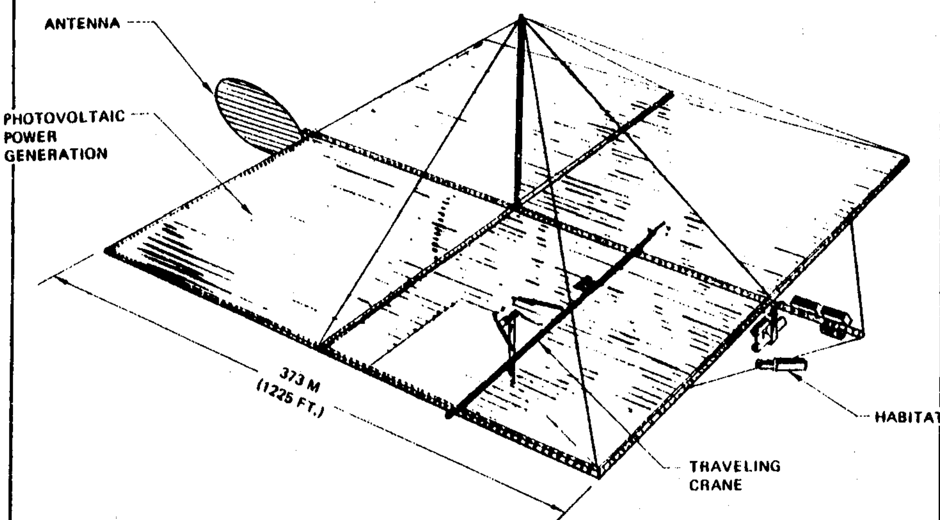
DOCUMENTATION SOURCES

FSTSA, D180-20242-1, P. 16, and D180-20242-2, Sec 3.8

REVISION DATE: 6/7/78

PAYLOAD DATA SHEET

DESCRIPTION



MASS	340 metric tons
SIZE	373 m sq x 5 m thick
LIFE	3-5 years
MAX. Gs	0.05

ONBOARD POWER

TYPE	Photovoltaic
QUANTITY	15 MW
VOLTS	
FREQUENCY	DC

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE	X	
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 1 required

IOC

1997

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION: Nuclear Fuel Location System

NO.: 9-0

OBJECTIVES

- o Real-time monitoring of location of nuclear materials to prevent proliferation of weapons and nuclear blackmail

GLOBAL IMPLICATIONS

- o Will require treaty agreements to extend coverage beyond U.S. jurisdiction

TRANSPORTATION SCENARIO

- o Launch all 4 with a single shuttle
- o Transfer satellite #1 to destination orbit (Orbit-raising, inclination & longitudinal phasing)
- o Transfer satellite #2 to destination orbit (longitudinal phasing)
- o Transfer satellite #3 to destination orbit (longitudinal phasing)
- o Transfer satellite #4 to destination orbit (longitudinal phasing)

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

SHUTTLE

FINAL ORBIT

ALTITUDE 35,800 km

INCLINATION 50^{o}

ECCENTRICITY 0

LONGITUDE <US>

OTHER

TRANSPORT TIME

Non-critical

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

- o Modified fuel rods with tamper-proof microwave transmitter (10 mW)
- o Tracking & Control Center

LAUNCH

Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

$560M

PAYLOAD VALUE

$270 M/20

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

ATR-75(7365)-2, Aerospace study, pg. 95 (CO-7)

ATR-76 (7365)-1, Vol. III, Page 16

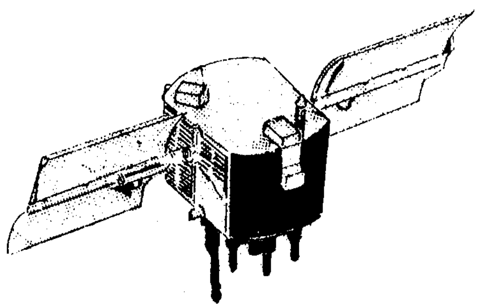
REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION

o Satellite serves simply as a microwave relay satellite. The position of fuel elements is resolved from time-difference of arrival of signals. All decoding/computation is performed at the ground station.

o 116 beams - s-band (3000 MHz)



MASS	1360 kg
SIZE	12.8 x 3m (diam)
LIFE	5 years
MAX. Gs	

ONBOARD POWER

TYPE	photovoltaic
QUANTITY	300w
VOLTS	
FREQUENCY	

POINTING	
ATTITUDE CONTROL	
STATION-KEEPING	E-W only

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		X
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM		X

PERFORMANCE PARAMETERS

o Track 10,000 fuel rods simultaneously
o Locate rods to $\pm$ 150 m every 30 seconds

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

o 4 satellites to cover U.S.
o 20 needed to obtain world-wide coverage
o Replacement

IOC

1990

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION
Marine Broadcast Radar

NO.
11-1

OBJECTIVES

To make the services of radar inexpensive and widely available to small boat operators thus increasing marine safety.

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Launch on Space Shuttle
- Assemble and checkout antenna modules in LEO via RMS and EVA
- Transport to GSO

INITIAL ORBIT
ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER
Shuttle

FINAL ORBIT
ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER
Geostationary

TRANSPORT TIME

REUSABLE | **DISPOSABLE**

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Shipboard TV broadcast receiver

LAUNCH

- Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

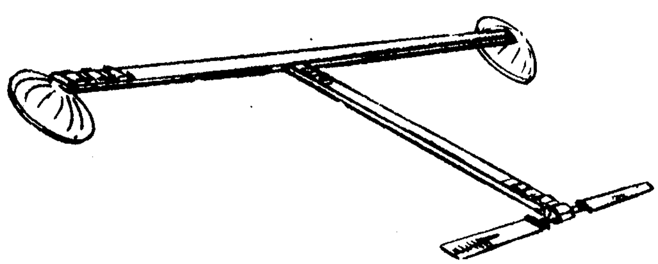
DOCUMENTATION SOURCES

REVISION DATE: 6/6/78

PAYLOAD DATA SHEET

DESCRIPTION

- Rectangular Radar Antenna
  - Slotted waveguide subarrays
  - 150 m x 500 m, longest arm oriented perpendicular to coastline of interest (3 m wide and deep)
  - Simplex transmit/receive functions
  - 150 m (dia) dish for receive
  - On-board processing
- Parabolic dish for direct broadcast
  - 150 m diameter
  - Multiple (~60) spot beams
  - Vertical polarization
  - Pre-assigned public service channel in UHF band



MASS
6700 kg

SIZE
500 m long

LIFE
10 years

MAX. Gs

ONBOARD POWER
TYPE photovoltaic
QUANTITY 25 kW
VOLTS
FREQUENCY

POINTING

ATTITUDE CONTROL
3 axis

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 4 for coverage of CONUS
- Servicing sorties every 3 years

IOC
1995

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION	NO.
Astronomical Telescope	12-0

OBJECTIVES

- To extend knowledge of universe by examination of most distant objects with even more resolution than can be provided by ST or ground based instruments.

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

1) Boost to LEO with Space Shuttle
2) Assemble mirror array in orbit
3) Modify orbit to 0^{o} inclination
4) Repeat steps 1) and 3) for focal plane unit
5) Final assembly = initialize station-keeping
6) Servicing sorties as necessary

INITIAL ORBIT

ALTITUDE	SHUTTLE
INCLINATION	
ECCENTRICITY	
LONGITUDE	
OTHER	

FINAL ORBIT

ALTITUDE	555km
INCLINATION	0
ECCENTRICITY	0
LONGITUDE	
OTHER	

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH Space Shuttle

SPACE Orbital services

COST ESTIMATES

PROGRAM COST

$690M

PAYLOAD VALUE

$430M/2

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

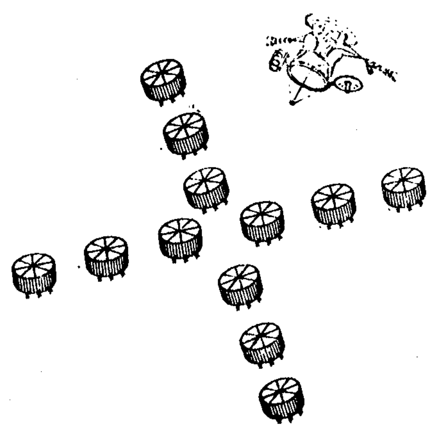
- ATR-75(7365)-2, Aerospace study, pg 101 (CO-10)
- ATR-76(7365)-1, Vol. III, page 19

REVISION DATE: 9/22/78

PAYLOAD DATA SHEET

DESCRIPTION

- A crossed array of visible light & IR (100 μm) mirrors with a station kept focal plane unit.
- Twenty-one mirrors, each 4 meter diameter
- Focal length adjusted by phase control of each individual mirror, and repositioning of focal plane unit
- 1 km separation - focal unit to mirror plane



MASS
18.1-19.6 MT

SIZE each arm
4 x 240 m

LIFE
10 years

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 15 kW

VOLTS

FREQUENCY

POINTING
$<10^{-10}$ radians

ATTITUDE CONTROL pointing & gravity-gradient control

STATION-KEEPING focal plane unit to mirror array

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE	X	
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

- Resolution $\approx 3\times10^{-9}$ radians
- Direct parallax measurements to 6500 light-years

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 2 units, 100 km separation in orbit
- Perhaps another pair, 180° around orbit (opposite side of Earth) from first pair
- Yearly servicing sorties

IOC

1989

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION	NO.
Global Search and Rescue Locator	14-0

OBJECTIVES

- To locate emergency transmitters world-wide
  - To improve success ratio of search and rescue efforts
  - To reduce search and rescue costs

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Deliver all 20 satellites with a single Space Shuttle launch
- Transport all 20 satellites to destination orbit #1
- Transport remaining 19 satellites to destination orbit #2

.
.
.
.

- Transport remaining 2 satellites to destination orbit #19
- Transport remaining satellite to destination orbit #20

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

SHUTTLE

FINAL ORBIT

ALTITUDE 20,185 km

INCLINATION 50^{o}

ECCENTRICITY 0

LONGITUDE

OTHER

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND
- small inexpensive lightweight transmitters
- ground site(s)
  - signal receivers
  - search coordination

LAUNCH

Space Shuttle

SPACE

Servicing System

COST ESTIMATES

PROGRAM COST

$700M

PAYLOAD VALUE

$350M/20

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

- ATR-75(7365)-2, Aerospace study, pg 105 (cc-1)
- ATR-76 (7365)-1, Vol. III pg 24

REVISION DATE: 10/26/79

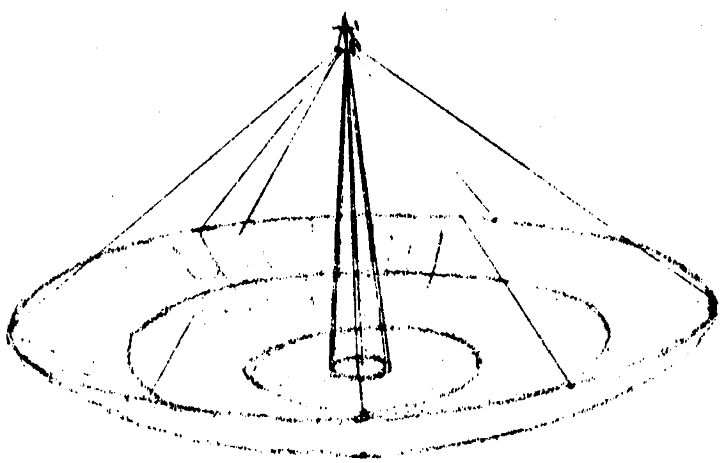
PAYLOAD DATA SHEET

DESCRIPTION

The satellites transpond the signals from the emergency transmitter and the location is computed by time-difference-of-arrival (TDOA) at the ground site.

- 10 m diameter antenna
- 1000 channel transponder

Requires 4 or more satellites to be in simultaneous view of emergency transmitter and ground station for accurate position fixing.



MASS	680-910 kg
SIZE	1.5 x 6.1 m (stowed)
LIFE	10 yrs
MAX. Gs	

ONBOARD POWER

TYPE	photovoltaic
QUANTITY	1000 w
VOLTS	
FREQUENCY	

POINTING	
ATTITUDE CONTROL	
STATION-KEEPING	

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		X
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

Characteristics of emergency transmitters	• 1 W peak (10 mW average) • 1 month life • uniquely (1 of 100) coded • self-contained • 1000 GHz

PERFORMANCE PARAMETERS

Location resolution
<±150m(X,Y, & Z)

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 20 operational simultaneously
- Servicing sorties every 3 years

IOC

1991

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION	NO.
Multinational Air Traffic Control Radar	20-0

OBJECTIVES

- To extend radar coverage beyond the line-of-sight for Air Traffic Surveillance
- To reduce numbers (i.e, costs) of active radar systems
- To centralize control of ATC functions
- To avail other countries of modernized ATC services

GLOBAL IMPLICATIONS

- New treaties will be required for multinational radar coverage
- Large structure technology required

TRANSPORTATION SCENARIO

- Pre-fab package of parts to LEO via space shuttle (15/launch)
- Assemble/deploy all arrays by RMS and astronaut EVA in proximity of Shuttle
- Transfer individual satellites to final destination orbits with low thrust system
- Use electric propulsion for long-term attitude control/orbit maintenance

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

SHUTTLE

FINAL ORBIT

ALTITUDE 555 km

INCLINATION 35-50°

ECCENTRICITY 0

LONGITUDE

OTHER

TRANSPORT TIME

non-critical

REUSABLE	DISPOSABLE
	X

SUPPORT SYSTEM REQUIREMENTS

GROUND

- 10 w beacons in all airplanes
- 3 radar/GCC sites for USA
- 0-2 sites for other countries

LAUNCH

Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

$330 M/150 arrays

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

- ATR-76(7365)-1, Vol III, Aerospace study, pg 14 (CO-5)

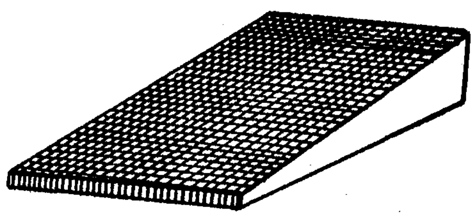
REVISION DATE: 3/20/78

PAYLOAD DATA SHEET

<table>
<tr><td colspan="2" rowspan="9">

DESCRIPTION

- Orbiting passive diffracting arrays allow large coverage from a few central radars. Orbital motion in conjunction with frequency shift accomplishes scan function.
- Array reflector face
  - Aluminized silica grid
  - 25×10^{-3} mm cloth
  - 25 x 25 mesh



</td><td colspan="3">MASS
1700 kg</td></tr>
<tr><td colspan="3">SIZE
75 m sq x 3 m thick</td></tr>
<tr><td colspan="3">LIFE</td></tr>
<tr><td colspan="3">MAX. Gs
0.1</td></tr>
<tr><td colspan="3">ONBOARD POWER
TYPE photovoltaic
QUANTITY 1 kW
VOLTS
FREQUENCY</td></tr>
<tr><td colspan="3">POINTING</td></tr>
<tr><td colspan="3">ATTITUDE CONTROL</td></tr>
<tr><td colspan="3">STATION-KEEPING</td></tr>
<tr><td>CHARACTERISTIC</td><td>YES</td><td>NO</td></tr>
<tr><td colspan="2" rowspan="4"></td><td>MODULAR CONSTRUCTION</td><td></td><td>X</td></tr>
<tr><td>CONTAMINATION SENSITIVE</td><td></td><td>X</td></tr>
<tr><td>MANNED SYSTEM</td><td></td><td>X</td></tr>
<tr><td>REPAIRABLE SYSTEM</td><td>X</td><td></td></tr>
<tr><td colspan="2">PREVIOUS STUDY CONSTRAINTS</td><td colspan="3">PERFORMANCE PARAMETERS
• Max. detection interval = 4 min.
• Scan width=1100 km
• Array ground footprint=450x1220m
• 18m diam. ground illuminator/receiver</td></tr>
<tr><td rowspan="2">TRAFFIC PROJECTION

• 150 for world-wide coverage</td><td colspan="4">IOC
1985</td></tr>
<tr><td colspan="4">REVISION DATE: 9/22/78</td></tr>
</table>

MISSION DATA SHEET

MISSION	NO.
Electronic Mail Transmission	25-0

OBJECTIVES

(1) To speed up delivery and lower costs of most mail service.

(2) To service thinly populated areas

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Space Shuttle delivery to LEO
- Assembly and checkout via astronaut EVA
- EPS transport to destination orbit

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

SHUTTLE

FINAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

GEOSTATIONARY

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND Page readers and facsimile printers at each post office

LAUNCH

Space Shuttle

SPACE

Orbital Servicing

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

$430M

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

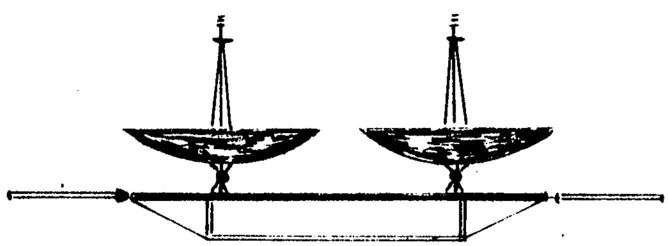
- ATR-76(7365)-1, Vol. III Aerospace study, pg 27 (CC-4)

REVISION DATE: 3-21-78

PAYLOAD DATA SHEET

DESCRIPTION

- Satellite acts as multi-channel repeater.
- Multi-beam antenna
- Multi-channel transponder, with switching for routing of data stream between receiver and transmitter sections
- LSI processor for message routing, beam steering, and traffic management
- 1000 beams
- 100 channels/beam
- 5 kW radiated power



Post-office ground station characteristics:

- 1m antenna
- Rural areas → 50 m W transmitter
- Urban areas → 5 watt transmitter

PREVIOUS STUDY CONSTRAINTS

MASS 9100 kg

SIZE 61 m diam. antenna

LIFE 10 years

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 15 kW

VOLTS

FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		
CONTAMINATION SENSITIVE		
MANNED SYSTEM		
REPAIRABLE SYSTEM		

PERFORMANCE PARAMETERS

- 10 pages (21.6x27.9 cm)/second/post office
- 100,000 post offices serviced
- Beam footprint = 74 km

TRAFFIC PROJECTION

- 1 required for CONUS coverage
- Servicing at 3 year intervals

IOC 1984

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION	NO.
Personal Communications Wrist Radio	30-0

OBJECTIVES

- To expand two-way telephone service to individuals wherever they might be. Allows users to establish voice contact either directly with other users, or with non-users via conventional telephone networks

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Launch to LEO via Space Shuttle
- Assemble and check-out
- Transfer to geosynchronous with electric propulsion system

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

SHUTTLE

FINAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

GEOSTATIONARY

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

Wrist Radios
- Max Weight = 0.1 kg
- Max. Power = 25 mW
- Battery Life $\geq$20 hours

LAUNCH

Space Shuttle

SPACE

Orbital Servicing

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

$300M

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

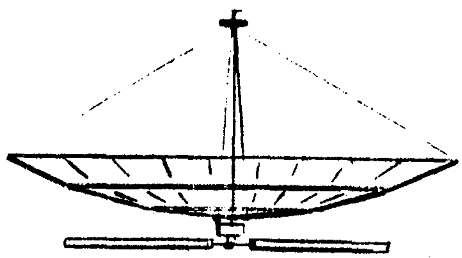
- ATR-76(7365)-1, Vol III Aerospace study, pg 32 (CC-9)
- ATR-75(7365)-2, pg 119

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION

- Multi-channel switching satellite
- 25 beams - 110 km(diam) footprint (covers 25 largest U.S. cities)
- 7 kW RF power - S band
- LSI processor for beam steering control and voice/code recognition of telephone address and message routing



MASS 14,000 kg

SIZE 61 m diameter

LIFE

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 21 kW

VOLTS

FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

- 25,000 simultaneous voice channels
- up to 100 two-way conversations/channel

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 1 required for CONUS coverage
- Servicing at 3 year intervals

IOC

1990

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION: Personal Navigation Wrist Set

NO.: 34-0

OBJECTIVES

- To provide accurate relative position location with very inexpensive user equipment.

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Launch to LEO via Space Shuttle
- Assemble and checkout using astronaut EVA
- Transport to GSO in single package via EPS
- Deployment and final assembly in GSO
- Continuous station-keeping of control unit
- Revisit as required for servicing

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Shuttle

FINAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Geostationary

TRANSPORT TIME

REUSABLE | **DISPOSABLE**

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Simple, inexpensive wrist receiver
- Fixed beacons for reference/calibration

LAUNCH

Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

$100M

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

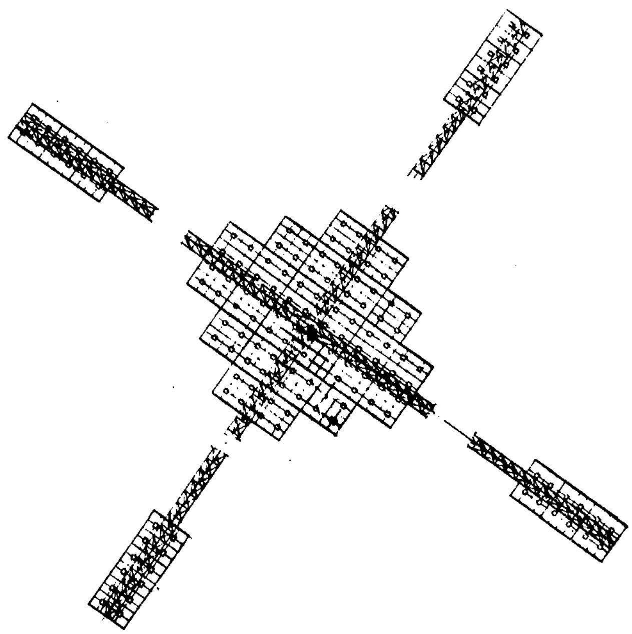
- ATR-76(7365)-1, Vol. III, Aerospace study, pg. 42 (CS-7)
- ATR-75(7365)-2, pg 141 (CS-13)

REVISION DATE: 3/22/78

PAYLOAD DATA SHEET

DESCRIPTION

- Narrow beams are swept over the U.S. by large phased arrays in space. Very simple receivers measure time elapsed between pulses received and display (N-S, E-W) distances to selected fixed points.
- Crossed arm antenna (2 arms) -"n" section phased array - ground footprint = 300 x 4500 m - different frequency for each arm.
- X-band, 4w RF output/arm
- Adaptive RF phase control for shaping and sweeping the two crossed beams.



MASS 13.6 MT

SIZE 5 x 1700 m/arm x 2 m thick

LIFE

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 2 kW

VOLTS

FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

- ~~Total satellites~~
- ~~Control unit~~

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

Wrist Set Characteristics	
	• 2 frequency receiver • Omni-antenna • clock drift $<10^{-5}$ • cost < $10.00

PERFORMANCE PARAMETERS

- Accuracy < 100 m relative to fixed site < 185 km away.
- Sweep frequency ≈ every 10 sec.

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 1 required for CONUS coverage
- Periodic revisits required for servicing

IOC 1993

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION: Near-Term Navigation Concept

NO. 34-1

OBJECTIVES

- To provide reasonably accurate relative position location services in the near term with very inexpensive ground-based equipment, thus increasing user acceptance.

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Launch to LEO via Space Shuttle
- Boost to GSO via IUS
- Automatic deployment and initiation
- Revisit for servicing as required

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Geostationary

TRANSPORT TIME

REUSABLE

DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Inexpensive user terminals
- Fixed beacons

LAUNCH

Space Shuttle

SPACE

Orbital Servicing

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

$90M

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

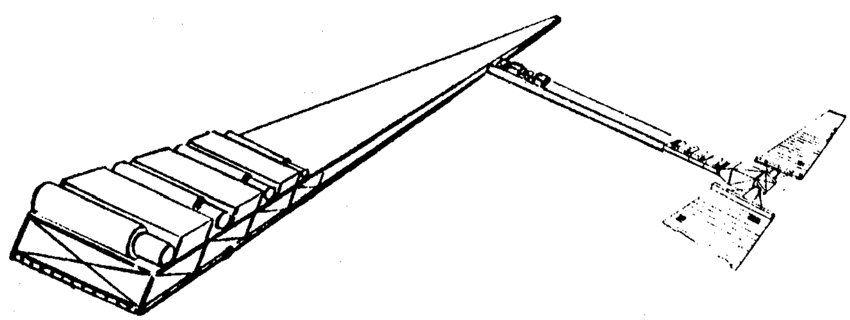
- ATR-76 (7365)-1, Vol. III, Aerospace study, pg 51 (CS-16)

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION

- Narrow beams are swept over the U.S. by phased arrays. Receivers measure time elapsed between pulses received and display distances (N-S, E-W) to fixed points.
- Pair of crossed arms, each 0.5m x 49m x .5 m
- Dual frequency X-band, one/arm.
- 100 x. RF output/arm
- Multi-section phased array/arm, ground footprint = 20 x 6000 km/arm



User Receiver Characteristics	• Dual frequency • Omni-antenna • Clock accuracy $\approx 10^{-5}$ • Cost <$10.00 (mass production)

MASS: 725 kg

SIZE: 49 x 49m

LIFE:

MAX. Gs:

ONBOARD POWER

TYPE: photovoltaic

QUANTITY: 1 kW

VOLTS:

FREQUENCY:

POINTING:

ATTITUDE CONTROL:

STATION-KEEPING:

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		
CONTAMINATION SENSITIVE		
MANNED SYSTEM		
REPAIRABLE SYSTEM		

PERFORMANCE PARAMETERS

- Position location to $\pm$ 1 km every 10 sec

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 1 required for CONUS coverage
- Periodic servicing sorties

IOC: 1987

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION Power Relay Satellite

NO. 37-1

OBJECTIVES

- To provide for transmission of electrical power from one area on Earth to another without unsightly and inefficient transmission lines.
  - Allows power generation to be confined to remote regions, minimizing environmental impact
  - Allows ground solar power plants on the day side of Earth to supply loads on the night side.

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Launch to LEO via HLLV
- Assemble and checkout in LEO
- Transfer to GSO via EPS

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Geostationary

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Transmission sites with power sources.
- Receiving substations

LAUNCH

Heavy lift launch vehicle

SPACE

Orbital servicing

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

$36M

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

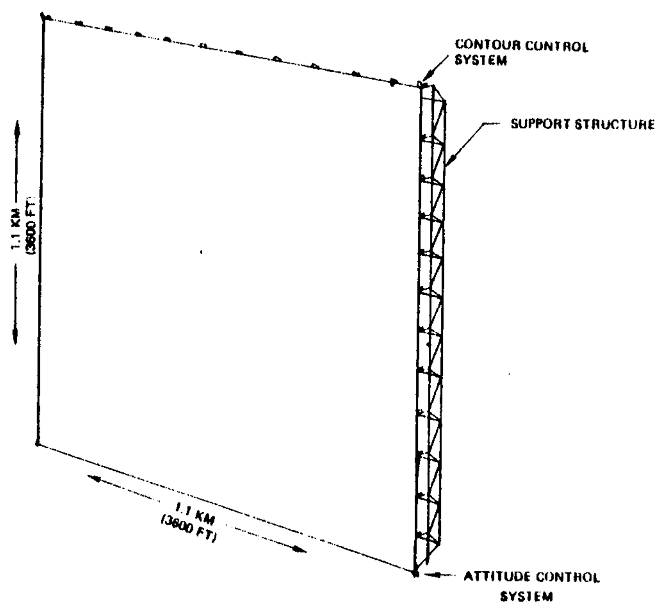
- ATR-76(7365)-1, Vol. III Aerospace study, pg 50 (CS-15)

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION

- Source power is converted to a microwave beam, bounced off an orbiting reflector, and reconverted to electricity at a receiving antenna on the ground.



MASS

27.5 MT

SIZE

1.1 km sq x 3 m thick

LIFE

MAX. Gs

low

ONBOARD POWER

TYPE tapped from beam

QUANTITY ?

VOLTS

FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

- 10 km square antennas (transmit & receive)
- 53% efficiency

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 100 satellites estimated to correspond to power transfer equivalent to 10% of U.S. consumption
- Periodic servicing sorties

IOC

1992

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION Utility Load Manangement Satellite

NO. 38-1

OBJECTIVES

- Improve the capital and energy efficiency of the electric utility system
- Reduce reserve requirements for generation and transmission capacity
- Improve reliability of service to essential loads
- Allow remote motor reading
- Allow institution of time-of-day (demand cycle) rate structures
- Allow centrally-controlled load shedding

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Space Shuttle launch
- Assemble/check-out in LEO
- Electric propulsion transfer to GSO
- Initial deployment monitors/commands to substation level
- Later update extends capability to individual househould level

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Shuttle

FINAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Geostationary

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Satellite control station
- Utility monitoring and control station
- Monitoring tranceivers

LAUNCH

- Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

$650 M

PAYLOAD VALUE

$50 M each

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

$10M/year

DOCUMENTATION SOURCES

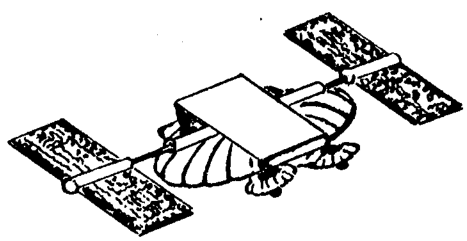
- D180-20791-1, Boeing. October 1977, study brief

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION

- 4 beam antenna - 4000 load-control groups/beam - 1000 load-control blocks/group - 1000 meters/block-beam footprint = 1250 km (E-W)
- Interrogation band with = 50 kbps. Meter response bandwidth = 500 bps
- 4 interrogation/response frequency pairs/beam
- Antenna diameter = 10 meters



PREVIOUS STUDY CONSTRAINTS

MASS
3200 kg

SIZE
10 m (dia) x 3 m

LIFE

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 7 kW

VOLTS DC

FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		
CONTAMINATION SENSITIVE		
MANNED SYSTEM		
REPAIRABLE SYSTEM		

PERFORMANCE PARAMETERS

- 17 day cycle time for 6 x 10^7 meters/region
- 1 minute response time to turn off up to 6 x 10^7 load blocks

TRAFFIC PROJECTION

- 2 required (1 spare) for CONUS coverage

IOC

1986

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION Space Constuction Facility	NO. 44-0

OBJECTIVES

- To provide a facility for the fabrication and construction of large structures in space.

GLOBAL IMPLICATIONS

- Requires commitment to large scale program such as satellite power system.

TRANSPORTATION SCENARIO

- 4 Shuttle launches initially - limited equipment and 8 man space station - for test bed/proof-of-concept/development of techniques
- Later launches to upgrade facility (size, personnel complement and construction capability and thru-put) as required to support emerging program requirements
- Major propulsion requirement will be for drag cancellation and to overcome gravity-gradient torques

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE 500 km

INCLINATION 28-35^{o}

ECCENTRICITY 0

LONGITUDE

OTHER

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

- Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

$3.1B

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

- D180-19399-1, Boeing IR & D study, 12/75

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION

- Incorporates a space station (probably modular) to provide living quarters for up to 100 people, and to serve as engineering and operations control centers
- Includes (limited) space manufacturing facilities to complete the fabrication of those items that cannot be boosted intact due to launch vehicle payload density limitations, and to repair/recondition tools and other equipment.
- Requires manipulators, positioning devices and holding fixtures for final assembly of major structural elements.
- A variety of logistics support vechicles will be necessary to:
  - manage floating storage yards
  - transport materials and supplies
  - ferry personnel - individually, and as construction crews (e.g. shift change)
- Some parts of the facility may have to be isolated from other parts, and have separate power supplies and environmental controls.

MASS: 2500 MT

SIZE: 18.3 x 230 x 750 m

LIFE:

MAX. Gs:

ONBOARD POWER

- TYPE: photovoltaic
- QUANTITY: >100 kW
- VOLTS:
- FREQUENCY:

POINTING:

ATTITUDE CONTROL: large requirements

STATION-KEEPING:

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM	X	
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 1 or a few depending upon development of "driver programs"

IOC: 1986

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION	Tethered Satellite	NO. 46-0

OBJECTIVES

- To conduct upper atmospheric investigations; i.e.,
  - pollution
  - thermal profile
  - wind systems
  - ionospheric fluctuations

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Launch via Space Shuttle
- Unroll tether (deploy satellite)
- Deploy SEPS/satellite from Shuttle
- Fly SEPS in drag cancelling mode
- Revisit periodically with Shuttle to resupply/refurbish/replace

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

< Shuttle

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

- Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

- H. Liemohn, Private Communication, 4/12/78
- Journal of the Astronautical Sciences; Vol. 26, No. 1; January 1978; page 1.

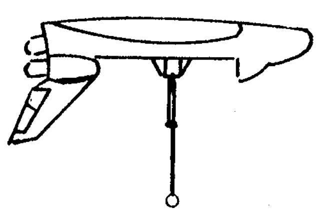
REVISION DATE: 9/22/78

PAYLOAD DATA SHEET

DESCRIPTION

- A small satellite is suspended in the upper atmosphere via a cable (1 mm dia x 100 km)

1 Satellite = 175 kg
Tether = 200 kg
Mounting Hardware = 330 kg



PREVIOUS STUDY CONSTRAINTS

- Shuttle based (limits stay-time)

TRAFFIC PROJECTION

- 1 (experimental)

MASS 705 kg 1

SIZE 144 cm

LIFE

MAX. Gs

ONBOARD POWER

TYPE battery

QUANTITY 121 w (average)

VOLTS

FREQUENCY DC

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		X
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM		X

PERFORMANCE PARAMETERS

IOC 1983

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION: Gravity Gradient Explorer

NO. 48-0

OBJECTIVES

- To obtain data on the higher harmonics of the Earth's gravitational field by direct observation of attitude perturbations experienced by a large structure in orbit.
- Research - follow-on to the Grav Sat currently planned for mid '80's)
- On-orbit assembly - precurson to SPS (technology demonstration)

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Transport to LEO via Space Shuttle
- Assemble on-orbit via RMS and EVA
- Transport to higher orbits (e.g. geosynchronous) for mapping operations of spherical harmonics of Earth's gravity field
- Supply attitude control forces (in a precisely measurable fashion) to overcome gravity gradient torques

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Shuttle

FINAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Variable

TRANSPORT TIME

REUSABLE	DISPOSABLE
	X

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

- Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

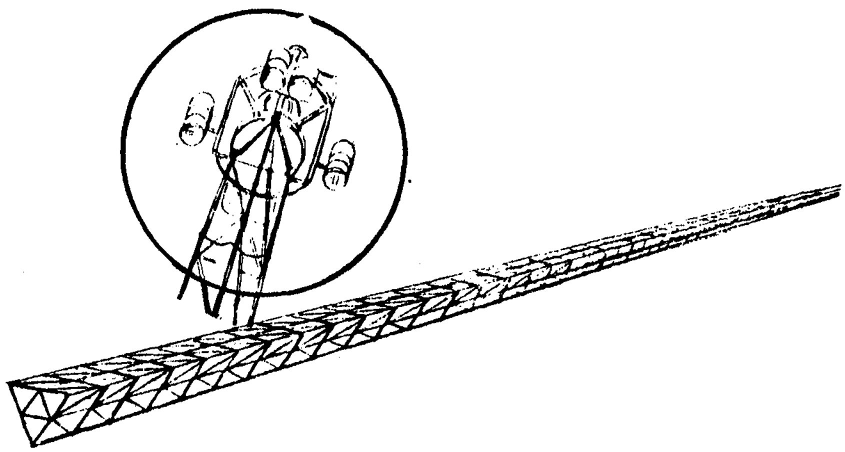
REVENUE PROJECTION

DOCUMENTATION SOURCES

REVISION DATE: 4/18/78

PAYLOAD DATA SHEET

DESCRIPTION	
• Long, skinny, truss structure (4,500 kg) • Minimized potential for thermal distortions • Use electric propulsion to provide restoring torques • Celestial attitude sensors with accuracies of $\sim 10^{-6}$ radians • Requires capability to periodically revise orbital parameters (e.g. reposition in longitude - at geosynchronous altitude)	MASS: 5000 kg SIZE: 6 x 6 x 3,100 m LIFE: 7 years MAX. Gs:



ONBOARD POWER

TYPE: photo Voltaic

QUANTITY: 500 w.

VOLTS:

FREQUENCY: DC

POINTING:

ATTITUDE CONTROL: 3-axis precision

STATION-KEEPING:

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

2 For complete mapping

IOC

1985

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION: Geosynchronous Communications Platform	NO. 49-0

OBJECTIVES

- To support the operation of multiple communications satellite systems while providing subsystems support and on-board switching facilities.
- To achieve lower costs/circuit-year (both development and operational)
- To conserve orbital space and reduce the building-up of geosynchronous debris

GLOBAL IMPLICATIONS

- Requires resolution of institutional responsibilities (may require establishment of a national or international agency to sell space, allocate channels, define and maintain interfaces, etc.)

TRANSPORTATION SCENARIO

- Boost to LEO via 3 Space Shuttles
- Assemble and test on-orbit
- Transfer to GSO via EPS

INITIAL ORBIT

ALTITUDE: 500 km

INCLINATION: 28.5^{0}

ECCENTRICITY: 0

LONGITUDE:

OTHER: 95 min period

FINAL ORBIT

ALTITUDE:

INCLINATION:

ECCENTRICITY:

LONGITUDE:

OTHER:

Geostationary

TRANSPORT TIME

months

REUSABLE	DISPOSABLE
X	X

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

$488 M

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

- "A Switchboard-in-the-Sky Concept for Domestic Satellite Communications" L. Jaffe, S. Fordyce, & E. C. Hamilton; 3/3/78

REVISION DATE: 4/28/78

PAYLOAD DATA SHEET

DESCRIPTION

- Antenna diameters to 100 meters
- C-band example - 33 spot beams (108 km Dia footprint)
- Onboard processing (message routing/switching)
- Very high redundancy/reliability levels
  - Structure/Mechanisms
  - Power (solar array/batteries)
  - Attitude Control/Stationkeeping
  - Thermal conditioning
  - Command/Telemetry
  - Programmable computer
- Total Peak RF power = 3200 watts
- Gimballed antennas to control pointing to $\pm 0.1^{\circ}$
- Unload momentum wheels once daily (36 n.m/s per axis)

Services to be Carried

- C-band, Ku-band, K-band, S-band — Fixed Communications (Point-to-point)
- L-band, VHF, UHF — Mobile Communications
- S-band, Ku-band — Direct Broadcast
- Space-to-space (TDRS)

Antenna Options

- Phased Array
- Microwave lenses
- Parabolic reflectors with offset feeds
- Cross polarization on multiple spot beams

[1] Not including antennas

MASS: 8200 kg

SIZE [1]: 430 x 175 x 15 m

LIFE: indefinite

MAX. Gs:

ONBOARD POWER

TYPE: photovoltaic

QUANTITY: 20 kW

VOLTS: 28/200 VDC±5%

FREQUENCY: DC

POINTING: $\pm 0.5^{\circ}$ Earth-pointing

ATTITUDE CONTROL: 3 axis

STATION-KEEPING: $\pm 0.5^{\circ}$ N/S & E/W

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 5 to support free-world traffic
- Servicing visits as required (~ 5 years)

IOC: 1991

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION Earthwatch (Resources Mapper)	NO. 50-0

OBJECTIVES

- Agriculture - Crop Production Forecasting
- Range Management - Grazing Potential
- Forestry - Timber Stand Volume Estimates
- Geology - Resources Location
- Land Use - Pseudo-census - taking
- Water shed - Resources Monitor
- Enviroment - Air/water pollution
- Disaster - Abrupt Event Assessment

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

. Multiple launch to LEO via Space Shuttle

. Transfer satellite #1 to destination orbit

.

.

.

. Transfer sattelite #η to destination orbit

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE 6390 km

INCLINATION $> 50^0$

ECCENTRICITY 0

~~LONGITUDE~~ repeating ground tracks

OTHER 6 hour period

TRANSPORT TIME

REUSABLE	DISPOSABLE
X	

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

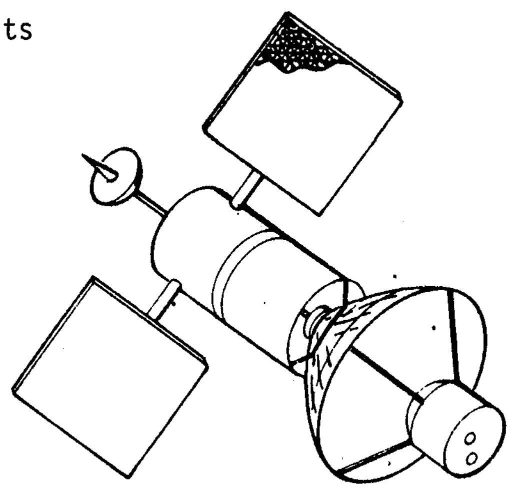
- Post-Landsat Advanced Concept Evaluation (PLACE) Midterm Briefing, 12/77

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION

- 2 pointable optical sensors
  - Hi-resolution for quick-look
  - Med-resolution for mapping
- Antenna is frequency-shared by synthetic aperture antenna and radiometer
- Visibile/IR imaging system
  - 3 to 6 m resolution
  - 30 m resolution
- Synthetic aperture radar
  - 10 to 25 m resolution
  - X/S/L-bands
- Passive radiometer
  - X-band - 12 km resolution
  - S-band - 60 km resolution
  - L-band - 120 km resolution
- Requires hardened solar arrays due to placement inside Van Allen belts



MASS
6500 kg

SIZE 15 m (dia) x 10 m antenna

LIFE

MAX. Gs

ONBOARD POWER
TYPE photovoltaic
QUANTITY 2.5 kW
VOLTS
FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		X
CONTAMINATION SENSITIVE	X	
MANNED SYSTEM		X
REPAIRABLE SYSTEM		X

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 20 satellites for continuous global coverage

IOC
1986

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION: Orbiting Deep Space Relay Station	NO. 51-0

OBJECTIVES

- To supplement/replace the existing world-wide network of Deep Space tracking Stations to:
  - Update obsolete, non-automated high maintenance, equipment
  - Increase performance
  - Decrease dependance on international politics (foreign sites)

GLOBAL IMPLICATIONS

- Presumes a continuing and expanding program of planetary exploration

TRANSPORTATION SCENARIO

- Boost to LEO via Space Shuttle
- Assemble and check-out via RMS and EVA
- Transfer to GSO via low-thrust
- Revisit as required for servicing

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE 35,800

INCLINATION $\leq 11^{0}$

ECCENTRICITY

LONGITUDE

OTHER

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Central command and data reception center

LAUNCH

- Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

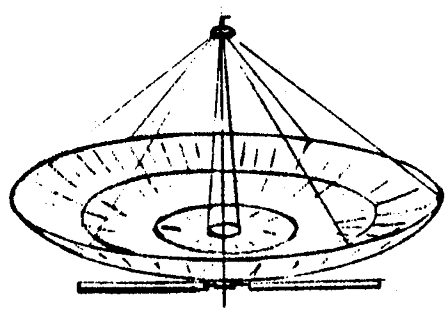
- ODSRS Study Plan, JPL, February 1978 (PRELIM)

REVISION DATE: 9/22/78

PAYLOAD DATA SHEET

DESCRIPTION

- TBD (Study in progress at JPL)



PREVIOUS STUDY CONSTRAINTS

MASS

7500 kg

SIZE

100 m (dia) x 30 m

LIFE

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 750 W

VOLTS

FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

- Received bit rates
- Frequency availability
- Navigation accurancy

TRAFFIC PROJECTION

- 2 required for ΔVLBI measurements
- Servicing sorties for maintenance and equipment update

IOC

1995

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION SPS Orbit Transfer System Recovery	NO. 52-0

OBJECTIVES

- To return the SPS orbit transfer system hardware to LEO for refurbishment and subsequent reuse; thus reducing transportation costs

GLOBAL IMPLICATIONS

- Assumes commitment to production of SPS system

TRANSPORTATION SCENARIO

- After transportation of an SPS or SPS module to GSO, the propulsion hardware is detached, and is returned to LEO via an autonomous propulsion vehicle.

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Geostationary

FINAL ORBIT

ALTITUDE 500 km
INCLINATION $28\frac{1}{2}^{0}$
ECCENTRICITY 0
LONGITUDE
OTHER

TRANSPORT TIME

REUSABLE	DISPOSABLE
X	

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

$45 M

TRANSPORTATION ALLOWANCE

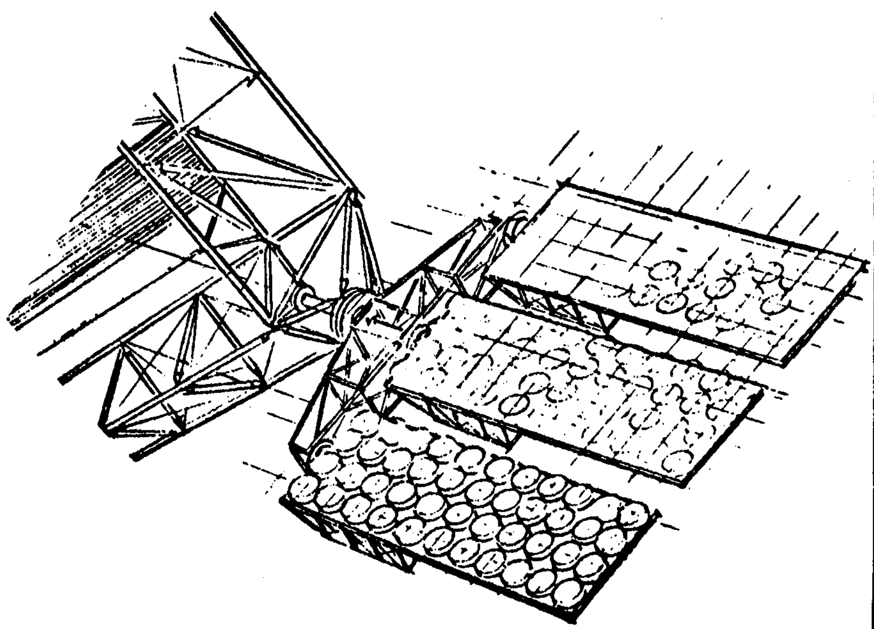
REVENUE PROJECTION

DOCUMENTATION SOURCES

- AIAA 78-695, D. Grim, April 1978

REVISION DATE: 4/26/78

PAYLOAD DATA SHEET

DESCRIPTION	
• SPS orbit transfer hardware is assumed to be electric propulsion • Modular construction is assumed to allow retention of some fraction of the propulsion hardware to fulfill the on-orbit attitude control requirements  * With antenna, 275 MT without antenna module ** With antenna, 24 x 38 x 3 m without antenna module	MASS: 725 MT * SIZE: 48 x 57 x 3 m ** LIFE: MAX. Gs: ONBOARD POWER TYPE: QUANTITY: none VOLTS: FREQUENCY: POINTING: ATTITUDE CONTROL: STATION-KEEPING:

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		X
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

• Sufficient to support production of 1-4 SPS per year

IOC: 2004

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION
EARTH'S MAGNETIC TAIL MAPPER

NO. 54-0

OBJECTIVES

To establish the characteristics and extent of the Earth's magnetic tail and to monitor its fluctuations in response to solar-terrestial phenomena.

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

. Launch to LEO via Space Shuttle
. Spiral out to Earth escape
. Cruise to desired observation postiion
. Transition to station-keeping mode to maintain position relative to Earth
. Change position as desired to map magnetic streamlines and follow desired monitoring schedule

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Shuttle

FINAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

1.01 - 1.03 au
Heliocentric

TRANSPORT TIME

REUSABLE	DISPOSABLE
	X

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH
Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

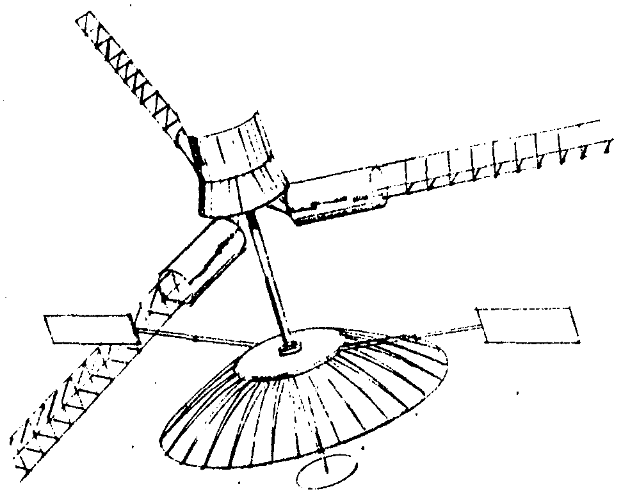
DOCUMENTATION SOURCES

REVISION DATE: 4/11/78

PAYLOAD DATA SHEET

DESCRIPTION

Neutral Mass Spectrometer	-	10 kg
Ion Mass Spectrometer	-	10 kg
Electron Spectrometer	-	3 kg
Magnetometers (2)	-	3 kg each
Solar Wind Analyzer	-	9 kg
Plasma Wave Detector	-	5 kg
Thermal Plasma Detector	-	3 kg
IR Spectrometer	-	8 kg
UV Spectrometer	-	4 kg
X-Ray Spectrometer	-	8 kg
γ-Ray Spectrometer	-	10 kg
Science Booms (2-6 m.ea.)	-	5 kg each
Data Processor	-	14 kg
Tape Recorder	-	8 kg
+ Engineering Support		



MASS

375 kg

SIZE

.7 x .7 x 3.5 m

LIFE

3 years

MAX. Gs

REQ'D ONBOARD POWER

TYPE

QUANTITY 125w

VOLTS 28

FREQUENCY DC

POINTING

Spin axis to Sun

ATTITUDE CONTROL

Spin

STATION-KEEPING

with Earth

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION		X
CONTAMINATION SENSITIVE	X	
MANNED SYSTEM		X
REPAIRABLE SYSTEM		X

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

One in service at a time

IOC

1986

REVISION DATE: 10/26/79

MISSION DATA SHEET

MISSION

ICEBERG DISSIPATER

NO.

55-0

OBJECTIVES

- To speed the meltdown of icebergs that have a (potential) danger to world wide shipping.

GLOBAL IMPLICATIONS

. An advanced Earth observatory satellite with resolution sufficient to spot "calving" icebergs and to map ocean currents is a desirable adjunct to this program

TRANSPORTATION SCENARIO

- Launch via Space Shuttle
- Assemble/check-out in LEO
- Transfer to destination orbit via electric propulsion

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE 2710 km

INCLINATION 60^{o}

ECCENTRICITY

LONGITUDE

OTHER

TRANSPORT TIME

REUSABLE

DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

- Coast Guard Ice Watch Command (existing)

LAUNCH

- Space Shuttle

SPACE

- Earth observatory satellite

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

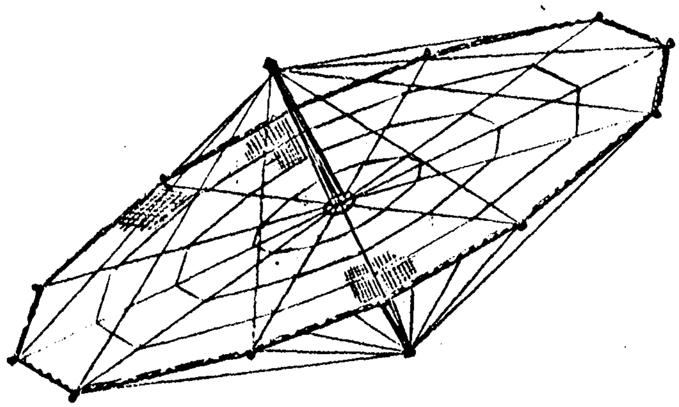
DOCUMENTATION SOURCES

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET

DESCRIPTION

A disc of reflecting material with a commandable pointing system.



MASS	1,750 MT
SIZE	4.5 m x 6 km (dia)
LIFE	10 years
MAX. Gs	

ONBOARD POWER

TYPE	
QUANTITY	
VOLTS	
FREQUENCY	

POINTING	$\pm 0.1^{\circ}$
ATTITUDE CONTROL	3 axis
STATION-KEEPING	N/R

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE	X	
MANNED SYSTEM		X
REPAIRABLE SYSTEM		

PERFORMANCE PARAMETERS

1/3 sun illumination over 18 km diameter area on Earth

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

~25 for global coverage

IOC

1997

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION	NO.
Soil Surface Texturometer	56-0

OBJECTIVES

To measure the texture of the Earth's surface to assist in the classification of ground materials.

- Identification of vegetation
- Measurement of particle size
- Ground periodicity

GLOBAL IMPLICATIONS

TRANSPORTATION SCENARIO

- Transport to LEO via Space Shuttle
- Assemble and checkout via astronaut EVA
- Transfer to destination orbit via electric propulsion

INITIAL ORBIT

ALTITUDE

INCLINATION

ECCENTRICITY

LONGITUDE

OTHER

Shuttle

FINAL ORBIT

ALTITUDE 600 km

INCLINATION 50^{o}

ECCENTRICITY 0

LONGITUDE

OTHER

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

- Space Shuttle

SPACE

- RF scatterometer

COST ESTIMATES

PROGRAM COST

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

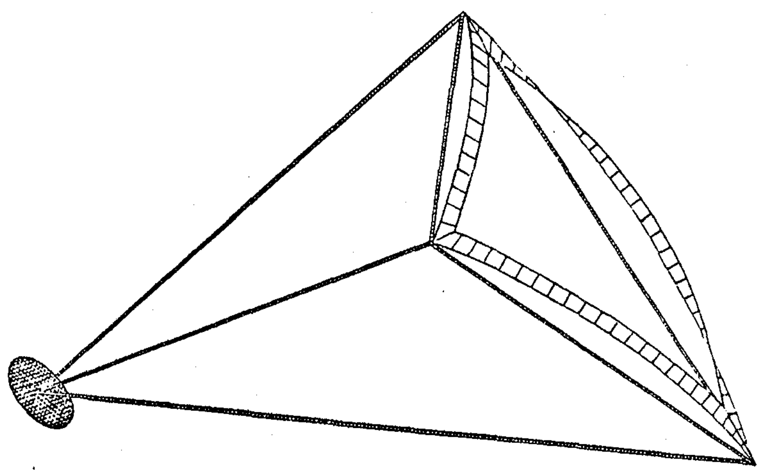
- PLACE midterm briefing

REVISION DATE: 4/10/78

PAYLOAD DATA SHEET

DESCRIPTION

- Visible/IR lasers used as scatterometer
- On-board statistical analyzer to examine ground returns and reduce data link (satellite to ground) requirements
- Adaptive optics - 3 lines of mirrors (60^0 apart) - 100 in each line
- Individual mirror is 3 m square (focal length ~ 600 m)
- Image motion compensation
- Picosecond pulses - visible through IR



1 Tetrahedral, 300 m sides for base and 600 m to apex

MASS

2310 kg

SIZE

1 600 m

LIFE

5 years

MAX. Gs

ONBOARD POWER

TYPE photovoltaic

QUANTITY 400 w

VOLTS

FREQUENCY DC

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE	X	
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

- Resolutions ranging from 10^{-3} to 1.0 m, as commeasurate with atmospheric scattering

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 1 required

IOC

1988

REVISION DATE: 9/22/78

MISSION DATA SHEET

MISSION	NO.
Technology Development Platform	58-0

OBJECTIVES

- To provide a long-term test-bed facility in the geosynchronous environment.

GLOBAL IMPLICATIONS

- Supports commitment to large scale space program

TRANSPORTATION SCENARIO

- Transport to LEO via Space Shuttle
- Assemble and checkout via RMS
- Transport to orbital destination via electrical propulsion
- EPS provide engineering support services as required
- Revisit as necessary to reconfigure/update experiment equipment

INITIAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Shuttle

FINAL ORBIT

ALTITUDE
INCLINATION
ECCENTRICITY
LONGITUDE
OTHER

Geostationary

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

LAUNCH

- Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

$40 - 50 M

PAYLOAD VALUE

TRANSPORTATION ALLOWANCE

REVENUE PROJECTION

DOCUMENTATION SOURCES

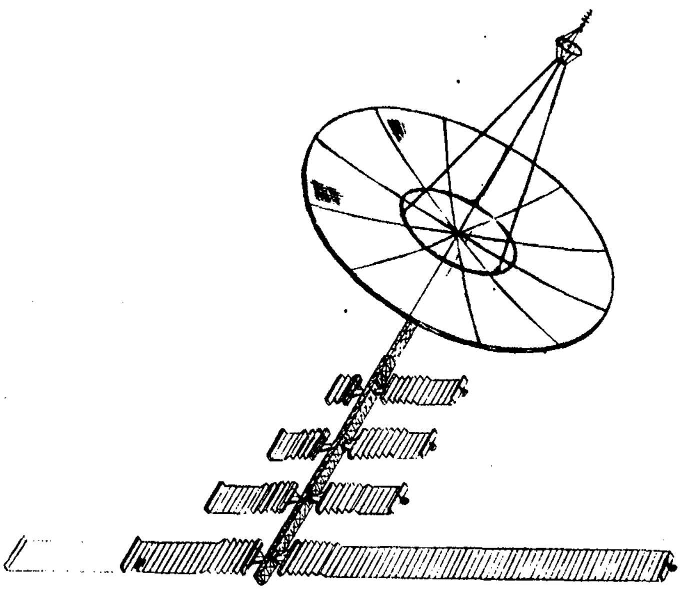
- D180-19783-3, PLUS final report, July 1976

REVISION DATE: 6/7/78

PAYLOAD DATA SHEET

DESCRIPTION

- Square frame structure
- Multiple SEPS solar arrays
- 30 m furlable antenna
- Docking subsystem for EPS attachment
- Manipulator reconfiguration aids
- Monopulse fine pointing system (pilot beam)
- Accommodate a pair of 70 kW (RF output) klystrons and their associated electronics



MASS
3090 kg

SIZE 1 x 1 x 51 m
Ant = 30 m (dia)

LIFE
10 years

MAX. Gs

ONBOARD POWER

TYPE: photovoltaic
QUANTITY: 160 kW
VOLTS: 40 kV
FREQUENCY: DC

POINTING
Antenna - 2π FOV

ATTITUDE CONTROL
3 axis/gravity gradient

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		
MANNED SYSTEM		X
REPAIRABLE SYSTEM	X	

PERFORMANCE PARAMETERS

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

- 1 in service at any one time
- Revisits as necessary

IOC
1988

REVISION DATE: 9/22/78

MISSION DATA SHEET (U)

MISSION	NO.
SPACE BASED RADAR SYSTEM-NEAR TERM	60-0

OBJECTIVES

To provide USAF with the capability for long-range, unjamm·able, radar surveillance of aircraft, spacecraft, and missiles.

GLOBAL IMPLICATIONS

Presumes Shuttle flight test of antenna deployment test model (in LEO)

TRANSPORTATION SCENARIO

- Assumes a Polar Launch via STS
- Assemble/check-out in LEO
- EPS to final orbit

INITIAL ORBIT

ALTITUDE

INCLINATION Shuttle

ECCENTRICITY

LONGITUDE

OTHER

FINAL ORBIT

ALTITUDE 10,355 km

INCLINATION ~90^{0}

ECCENTRICITY 0

LONGITUDE

OTHER

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

Ground station (1) in CONUS

LAUNCH

Space Shuttle

SPACE

COST ESTIMATES

PROGRAM COST

$550M

PAYLOAD VALUE

$75M each

TRANSPORTATION ALLOWANCE

$76M Shuttle/IUS

REVENUE PROJECTION

DOCUMENTATION SOURCES

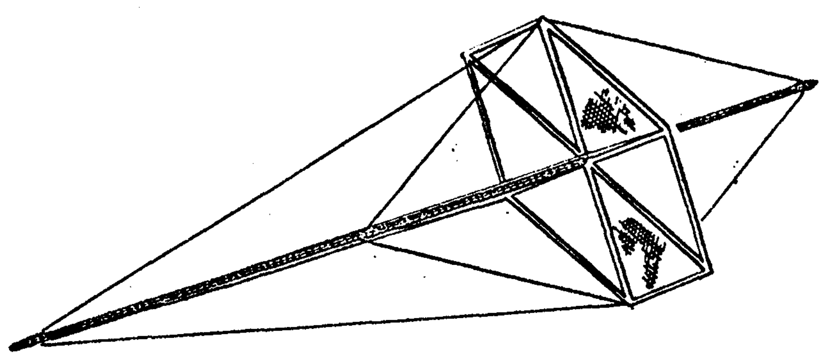
SAMSO TR-77-78, May 1977, Space-Based Radar Surveillance System study final report (U)

REVISION DATE: 10/26/79

PAYLOAD DATA SHEET (U)

DESCRIPTION

. Space-fed phased array len s antenna
. Electronically scanned pencil beam
. Single L-band beam
. 75,000 individual modules/antenna
. Solid state transmitter/sidelobe canceller/ on-board signal processor
. Satellite-to-satellite relay
. Circular solar arrays (2) of 14 m dia. mounted on upper systems package (USP)



PREVIOUS STUDY CONSTRAINTS

MASS
4000 kg

SIZE
90 m mast
61 m dia

LIFE
5 years

MAX. Gs

ONBOARD POWER
TYPE Photovoltaic
QUANTITY 30 kW
VOLTS 120 v.
FREQUENCY DC

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		
MANNED SYSTEM		
REPAIRABLE SYSTEM		

PERFORMANCE PARAMETERS
Classified

TRAFFIC PROJECTION

. 4 satellites in service simultaneously
. First satellite goes up by itself for a 1 year demonstration phase prior to further deployment

IOC
1987

REVISION DATE: 9/22/78

MISSION DATA SHEET(U)

MISSION	NO.
SPACE BASED RADAR SYSTEM - FAR TERM	61-0

OBJECTIVES

To provide USAF with the capability for long-range, unjammable, radar surveillance of aircraft, spacecraft, and missiles.

GLOBAL IMPLICATIONS

Probable implementation = prior acquisition of SBR system-near term

TRANSPORTATION SCENARIO

- Delivery to LEO via multiple Shuttle launches
- Assemble and check-out in LEO
- Transfer to intermediate elliptical orbit via low-thrust chemical propulsion
- Separation of CPS/deployment of electric propulsion system
- Low-thrust transfer to geosynchronous orbit via electric propulsion

INITIAL ORBIT

ALTITUDE	10,000-21,500
INCLINATION	28.5°
ECCENTRICITY	.36
LONGITUDE	
OTHER	

FINAL ORBIT

ALTITUDE	Geostationary
INCLINATION	
ECCENTRICITY	
LONGITUDE	
OTHER	

TRANSPORT TIME

REUSABLE	DISPOSABLE

SUPPORT SYSTEM REQUIREMENTS

GROUND

Ground station (1) in CONUS

LAUNCH

Space Shuttle (2/satellite)

SPACE

COST ESTIMATES

PROGRAM COST

$700M

PAYLOAD VALUE

$100M

TRANSPORTATION ALLOWANCE

$190M Shuttle/IUS

REVENUE PROJECTION

DOCUMENTATION SOURCES

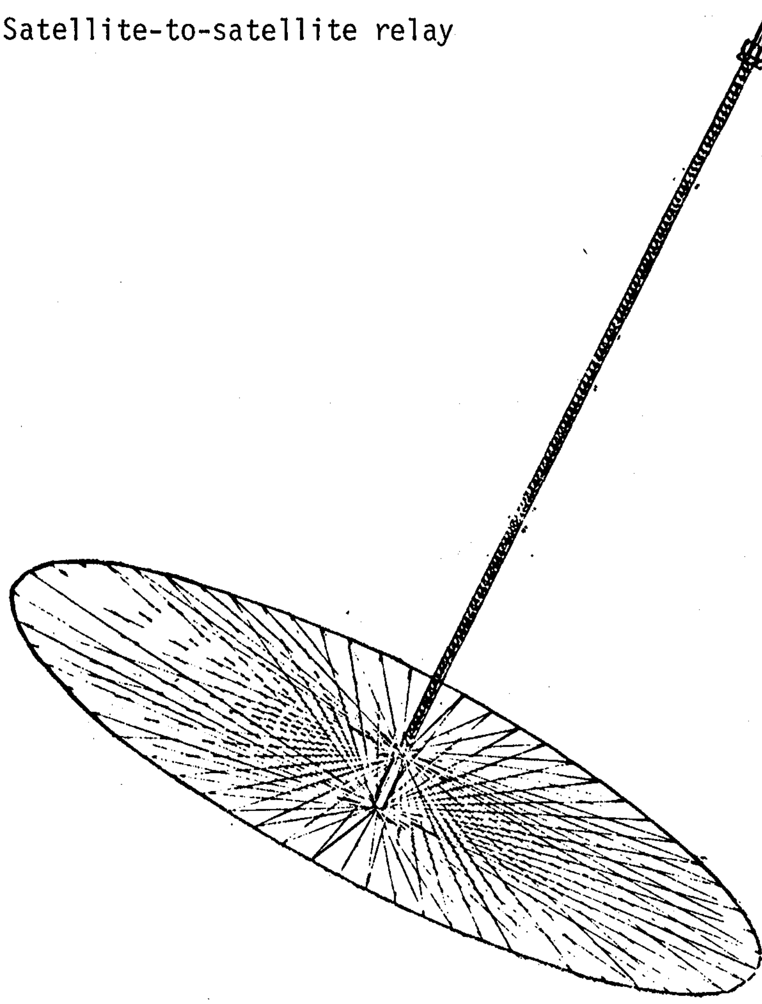
SAMSO TR-77-78, May 1977, Space-Based Radar Surveillance System study final report (U)

REVISION DATE: 9/22/78

PAYLOAD DATA SHEET

DESCRIPTION

. Space-fed phased array lens antenna
. Electronically scanned pencil beam
. 8 L-band beams
. 51,000 individual modules/antenna
. Satellite-to-satellite relay



MASS
7,000 kg

SIZE 270m mast
180 m diam

LIFE 7 years

MAX. Gs

ONBOARD POWER
TYPE Nuclear
QUANTITY 50 kW
VOLTS
FREQUENCY

POINTING

ATTITUDE CONTROL

STATION-KEEPING

CHARACTERISTIC	YES	NO
MODULAR CONSTRUCTION	X	
CONTAMINATION SENSITIVE		
MANNED SYSTEM		
REPAIRABLE SYSTEM		

PERFORMANCE PARAMETERS

Classified

PREVIOUS STUDY CONSTRAINTS

TRAFFIC PROJECTION

5 Satellites in service simultaneously

IOC

1992

REVISION DATE: 9/22/78

APPENDIX B - SYMBOLS AND ABBREVIATIONS

a_o - Initial system acceleration

ACS - Attitude control system

C_{EPS} - EPS purchase costs

C_{ETO} - Launch costs

C_M - Total mission costs

C_P - EPS propellant costs

C_{PL} - Payload value

C_{SA} - Solar array purchase costs

C_{SCAR} - Costs derived from payload modification for EPS

C_{TT} - Mission duration associated costs

D - Mission performance penalty for atmospheric drag

EP - Electric propulsion

EPS - Electric propulsion system

ETO - Earth to orbit

GaAlAs - Gallium-aluminum-arsenide

GEO - Geosynchronous orbit (also GSO)

g_o - Gravitational constant

IOC - Initial operational capability

IR - Infrared

I_{sp} - Specific impulse

IUS - Inertial Upper Stage

K - Ground-based residency factor

kg - kilogram

K_{SCAR} - Payload cost penalty for EP compatibility

kW - kilowatt

k_1 - Curve fit parameter for η relationship

k_2 - Curve fit parameter for η relationship

LEO - Low Earth Orbit

LeRC - Lewis Research Center

M_{AV} - EPS supporting subsystem mass

M_{bo} - System mass at end of mission (burn out)

M_{EPS} - Total mass of electric propulsion system

M_P - EPS propellant mass

M_{PL} - Payload mass

M_{PLD} - Modified (for EPS) payload mass

M_{SA} - Solar array mass

MT - Metric ton

M_T - Total mass launched to LEO

NASA - National Aeronautics and Space Administration

NOM - Nominal value of ...

OAST - Office of Aeronautics and Space Technology

P - Solar array output power (also P_o)

P_{EFF} - Effective value of SA power

P_{NOM} - Nominal value of SA power

R - Mission performance penalty for SA degradation

RF - Radio frequency

S	-	Mission performance penalty for thrust vector steering
SA	-	Solar array
SBR	-	Space-based radar
SEPS	-	Solar electric propulsion system
SOA	-	State-of-the-art
SOW	-	Statement of Work
SPS	-	Satellite power system
SSV	-	Space Shuttle vehicle
STS	-	Space Transportation System
T	-	Mission time
T_D	-	Mission performance penalty for EP start-up
TT	-	Trip/transfer time
V_{EFF}	-	Effective propellant discharge velocity (also V_{EXH})
YR	-	Year
\$K	-	Thousands of dollars
\$M	-	Millions of dollars
α	-	EPS specific mass (total)
α_{ADP}	-	Specific mass of support equipment for STS launch
α_{EPS}	-	EPS propulsion specific mass
α_{SA}	-	Specific mass of SA
α_{SCAR}	-	Payload mass penalty for EP compatibility
α_{STR}	-	EPS structural support specific mass

γ_{EPS} - EPS specific (production) costs

γ_{OPS} - Mission operating costs

γ_{P} - Propellant specific costs

γ_{SA} - Specific costs of solar array

γ_{STS} - Specific costs of launch to LEO

δ - Cost of money (discount rate)

ΔV - Mission velocity increment

ζ - Delivery charges ($/kg)

η - System efficiency

η_{MAX} - Maximum value of system efficiency

ϕ - Mission performance penalty for occultations

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