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MISSION ROLES FOR THE SOLAR ELECTRIC PROPULSION STAGE (SEPS) WITH THE SPACE TRANSPORTATION SYSTEM

VOLUME II – SYSTEM ANALYSIS AND EVOLUTION OF DESIGN AND OPERATIONAL CONCEPTS

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PREPARED FOR:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Program Development Directorate

Under Contract NAS8–30742

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REVIEWED AND APPROVED BY:

David M Hammock

David M. Hammock
Program Manager

FOREWORD

This volume, Volume II, presents the Northrop Services, Inc., SEPS System Analysis and Evolution of Design and Operational Concepts.

The complete final study report is composed of four volumes:

Volume I	Executive Summary
Volume II	System Analysis and Evolution of Design and Operational Concepts
Volume III	Design Reference Mission Description and Program Support Requirements
Volume IV	Traffic Model and Flight Schedule Analysis Techniques and Computer Programs.

The study, Mission Roles for the Solar Electric Propulsion Stage, with the Space Transportation System, was conducted under Contract NAS8-30742. Mr. Robert E. Austin of the Marshall Space Flight Center was the Contracting Officer's Representative for NASA. Mr. David M. Hammock was Northrop Services, Inc.'s, Study Program Manager.

The study was accomplished under Contract NAS8-30742 during the period from 20 May 1974 to February 1975, and was funded at $130,000.

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Section I

SUMMARY

1.1 GENERAL

The Solar Electric Propulsion Stage (SEPS) is a space propulsion stage that achieves high specific impulse (Isp) by converting solar energy to electrical energy which is used in an electrostatic particle accelerator to produce a high velocity ion beam. A parallel beam of electrons is produced so that diffusion of electrons into the ion beam produces a neutral plasma jet obviating any ion return flow problems. A specific impulse of more than 30,000 seconds is feasible with this general type of space propulsion system. The desirable Isp range for missions contemplated for the period 1979 to 1991 is in the range of 2,500 seconds to 5,000 seconds. Technology programs from 1967 to the present have demonstrated long life, continuous operation (in this Isp range) of flight suitable thrusters in laboratory tests and in research vehicle flight tests.

Previous SEPS mission and system definition studies have concentrated primarily on planetary exploration. As the Space Transportation System (STS) configuration and its mission employment was defined in greater detail, it became obvious that a SEPS type vehicle with its high Isp, relatively unlimited stay time in space, small propellant weight requirement, and operational flexibility would greatly augment the Shuttle, Interim Upper Stage (IUS), and Tug capabilities in the areas of transport to high energy orbits, orbital taxi functions, and servicing functions.

In 1974, the National Aeronautics and Space Administration (NASA), entered that phase of SEPS concept definition where significant funding would be committed to design definition and Supporting Research and Technology (SRT) projects oriented to specific SEPS configuration concepts. NASA considered it an appropriate time to:

- Critically review design defining trade studies and "optimization" results of past studies
- Ensure that system requirements and "baseline" system configuration characteristics derived from past studies were valid

- Ensure credibility of the cost effectiveness of SEPS as an added element of the STS.

Therefore, NASA, through its George C. Marshall Space Flight Center, implemented the "Mission Roles for SEPS with the Space Transportation System" study to quantify SEPS potential capabilities and transportation cost savings.

1.2 STUDY OBJECTIVES

The primary objectives of the SEPS study were to:

- Define mission roles that are major contributors to transportation cost reduction when SEPS is operated as an element of the Space Transportation System
- Generate concepts for and perform operations analyses on:
  - Payload exchanges with Shuttle, IUS, and Tug
  - Multiple payload deployment and retrieval
  - Payload maintenance and servicing in space
- Develop conceptual designs of payload handling and servicing equipment
- Identify SEPS interfaces with Shuttle, IUS, Tug, ground flight control centers, and launch support systems
- Define requirements not identified in prior studies and assess resultant design impacts on subsystems proposed in earlier studies.

Contributing secondary objectives of the SEPS study were to:

- Quantify transport cost effectiveness of SEPS with STS relative to a NASA supplied mission model
- Define a system operational profile with individual payloads assigned to specific flights to occur on specific dates
- Identify operational requirements and define SEPS program support
- Establish SEPS transport performance and show potential for improvement
- Identify benefits to IUS, Tug, and payload operations resulting from SEPS use
- Estimate operational costs of SEPS
- Identify problem areas for future investigation.

1.3 RELATION TO OTHER NASA EFFORTS

The reference mission model for quantifying the transportation cost savings and the definition of the "baseline" STS without SEPS were generated by the Marshall Space Flight Center. The "baseline" SEPS configuration ground rule for this study was the culmination of 3 years of NASA sponsored studies by Rockwell International Space Division, as generally defined in the final reports of their two latest studies*.

The performance of the power conversion units and thruster elements were based upon values from the Lewis Research Center's thruster subsystem control documents provided by MSFC in June 1974. Mr. Charles H. Guttman, MSFC, was the Contracting Officer's Representative for the Rockwell International Space Division studies.

Concurrent NASA in-house technology programs and other NASA sponsored studies contributing to the data base for this study were:

- Lewis Research Center's ongoing technology programs in solar electric propulsion power processors and thrusters
- Jet Propulsion Laboratory's thruster subsystem integration programs
- MSFC's ongoing programs in solar arrays and navigation and guidance analysis
- MSFC's Baseline Space Tug System Definition
- Hughes Research Labs' and TRW's engineering model development and improvement programs for thrusters and power processors under Lewis Research Center's sponsorship
- McDonnell Douglas' "Payload Utilization of Tug" and Follow-On (NAS9-29743 MSFC) and "IUS/Tug Payload Requirements Compatibility Study" (NAS8-31013 MSFC)
- International Business Machine's IUS and Tug Orbital Operations and Mission Support Study
- NASA supplied STS (other than SEPS) operational cost data.

1.4 STUDY APPROACH

The study effort was divided into five principal tasks. The systematic output of the tasks at a given level of detail allowed selection of competing

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*(1) *Feasibility Study of a Solar Electric Propulsion Stage for Geosynchronous Equatorial Missions, DRL No. MA04 DPD304, Contract NAS8-27360, dated February 1973.*

(2) *Extended Definition Feasibility Study for a SEPS Concept Definition, DR No. MA04 DPD369, dated December 21, 1973.*

concepts with a minimum of defining details of concepts later to be rejected. Successive iterations of the study were used to improve the concept of the selected system approach and to improve the accuracy of quantitative values used to support certain decisions.

The five study tasks were:

1. Mission Roles Identification and Analysis of STS Baseline Configuration Selection
2. Mission Operations and Systems Requirements Analysis
3. System and Subsystem Design Impacts Analysis
4. Interface Analysis
5. Cost Analysis.

The first step in establishing the transportation cost effectiveness of SEPS was to establish the maximum credible performance (minimum number of Shuttle flights) of STS without EO SEPS as the reference base for cost comparisons. To do this NSI evaluated transportation capabilities of the NASA defined baseline STS in operational modes that would maximize its transportation efficiency. NSI assumed modified forms of operational modes and equipment concepts evolved for STS with EO SEPS that if applied to baseline STS would justify removal of arbitrary restrictions on the number of payloads that could be carried on any flight.

The sensitivity of cost savings to various operational constraints such as multiple payload packaging restraints and arbitrary restrictions on numbers of payloads on a given flight that had been used in other studies were determined. Transportation cost savings resulting from more compact Tug designs, higher Isp in SEPS, and higher SEPS power were investigated.

A concerted attempt to compare maximum capability STS operation to maximum capability STS with SEPS was made so that the transportation cost savings attributed to SEPS would be extremely conservative and as realistic as the mission model.

In Task II, design reference mission descriptions were generated to establish design requirements for flight articles and to define ground support requirements for the flight operations. Operational modes, organizations, and facility concepts that would minimize the cost for the total SEPS Program Support were generated and defined.

In Tasks III and IV, new approaches and new applications of older ones were conceived for SEPS payload transport and for handling and servicing functions. New approaches were conceived for General Purpose Mission Equipment (GPME) for Tug and IUS that simplified IUS and Tug operations. Conceptual design of the equipment required by the approaches were developed.

Interfaces between SEPS and other STS elements and payloads were identified and defined to the extent warranted by the present level of design definition of the elements (or to the extent necessary to identify the desirable characteristics of the interface).

Assessments were conducted of technology areas that would have significant influence on the recommended SEPS and GPME configuration or on their operational modes with the STS.

Task V study requirements were to update NASA supplied "baseline" SEPS program costs by generating cost deltas resulting from the study's recommended changes to SEPS baseline subsystem. Recommendations from this study and NASA in-house activities indicated that a better approach to costing was to generate new independent cost estimates. Estimated program costs were significantly reduced by new configurations and new operational modes evolved during this study.

1.5 STUDY LIMITATIONS

Certain areas of the study were limited by the following guidelines or constraints:

- Cost effectiveness of SEPS was limited solely to STS transport cost savings for accomplishment of "The October 1973 Space Shuttle Traffic Model," NASA TMX-64751, Revision 2, dated January 1974. No cost advantage of other SEPS mission capabilities such as onorbit servicing, maneuvering payloads in orbit, or the great increase in allowable payload weights for high energy earth orbital missions and planetary missions was considered. The mission model covers the years from 1981 to 1991.

- The "baseline" STS was defined as the Shuttle, an expendable tran-stage (IUS) through 1983, and the MSFC (June 1974) baseline Tug from 1984 to 1991.
- Planetary mission roles were not investigated except to ensure that configurations and characteristics defined for the SEPS earth orbital functions would provide equal or enhanced planetary mission capabilities relative to the NASA supplied baseline SEPS configuration.

1.6 SIGNIFICANT SYSTEM CHARACTERISTICS AND STUDY CONCLUSIONS

Solar electric propulsion stages have radically different physical and performance characteristics than the familiar chemical propulsion stages. These characteristics influence every facet of the associated developmental and operational phases. Although the difference in physical characteristics is rather obvious, the tremendous potential from exploiting these differences (and some limitations) are often overlooked even by experienced space system planners and concept evaluators.

Depending upon the evaluator's recognition of the influence of certain physical and performance differences of SEPS and conventional stages, the conclusions and other results of this study may be accepted as so obvious as to hardly warrant their statement, or they may be summarily rejected.

Because of these factors, the following rather unorthodox order of presentation will be used:

- Primary characteristics and resulting first order influences of system differences
- Study conclusions
- System concepts and data generated
- Technology assessments.

1.6.1 Primary Characteristics and Influences

Isp AND THRUST

The feasible range of specific impulse (Isp) for mercury ion systems is 2,000 to 30,000 seconds. Demonstrated designs have SEPS operating in the

2,000- to 5,000-second range. For negligible weight and cost penalty, selectable high thrust and low Isp, or high Isp and low thrust operating modes can be designed into the system. Selection of the combination best suited to each mission phase can be made in flight.

The potential of SEPS high Isp can be inferred from the following comparisons. A characteristic high performance (450-second Isp) Space Tug configuration with 22,676 kg of O_2/H_2 propellant and a 2,585 kg inert weight can provide a 1,814 kg payload with a 8,016 m/sec change in velocity. A 3,000 second Isp SEPS with 959 kg of mercury propellant and a burnout weight of 1,297 kg can provide the same ΔV to a similar 1,814 kg payload. The SEPS loaded weight (2,206 kg) is only 9 percent of the chemical stage weight (25,260 kg).

The ΔV just described is approximately the ΔV for a round trip from Shuttle orbit to geosynchronous and return. If that were the mission and SEPS executed it, SEPS low thrust would result in "gravity losses" such that its idealized ΔV requirement would be 1.5 times an impulse stage's ΔV or 12,024 m/sec. For the SEPS to accomplish that ΔV, its initial weight would be 2,793 kg (11 percent of the chemical stage mass) and it would have to tank 1,546 kg of mercury. If SEPS were designed to operate through the Van Allen belts with radiation resistant, self-annealing solar cells, the solar cell "blankets" might increase 30 to 40 percent in cost and increase in weight by approximately 70 kg.

SEPS specific impulse is proportional to the square root of screen voltage; therefore, Isp could be increased by operating at higher thruster screen voltages (Vs). Assume an operation at 2 times the screen voltage. SEPS Isp is now $\sqrt{2} \times 3000$ sec. = 4243 sec. Initial stage weight is only 2,273 kg and only 1,025 kg of mercury would have to be tanked. Initial stage weight for the 4,243 Isp stage is just 81 percent of a 3000 Isp stage.

SEPS receives its energy from the sun, so increasing the energy per unit mass of propellant (increasing Isp) in order to reduce the total required propellant for a mission will reduce the initial total weight, but will increase the mission time. To shorten trip times, SEPS energy collection and conversion rate to electrical power must be increased. Within ranges of interest for SEPS, power is limited only by the cost of solar arrays required to produce the higher power levels. Masses increase but they are within launch capability of a single STS flight.

As a result of the physical phenomena by which SEPS functions, it has the unique capability to trade increased mission accomplishment time against reduced gross weight as was just illustrated. Its mercury propellant is so dense (specific gravity over 13) and tank pressures so low (21 n/cm^2) that excess capacity tanks can be designed into the system at minor weight penalty. If this is done, planned increases in payload masses or more demanding total impulse missions, not originally planned for the vehicle, can be accomplished simply by allowing longer times for accomplishing the missions and tanking more propellant at initiation of the mission.

In the power ranges desirable for the 1984 to 1991 time frame (25 kw up to 100 kw), the power level chosen for development has relatively small influence on the development cost of the system. Solar arrays may represent about 25 percent of the production cost of the complete stage. If oversized arrays for planetary missions are produced when the power and extra payload mass ability is not required, a cost penalty of about 10 percent of the base production cost of those of planetary vehicles could be incurred.

BASIC PROPULSION POWER CONVERSION CONSIDERATION

The SEPS thruster is a simple electrostatic charged particle accelerator as shown schematically on Figure 1-1. The operating Isp (proportional to $\sqrt{Vs}$) is set by the voltage level of the screens (Vs). The thrust level and current flow of the thruster are dominantly responsive to the density of the plasma (ion population per unit volume). Therefore, primary thrust control is by control of the temperature of the main and cathode mercury propellant vaporizers which determine the plasma pressure inside the thrusters. Of the total electrical power to the thruster, (depending on screen voltage) 80 to 90 percent goes into ion beam energy. This "screen power" or "beam power" only needs to be direct current, relatively free of ripple currents and at approximately the voltage corresponding to the Isp desired for the particular mission or mission phase. The solar arrays are nearly ideal sources for direct supply of this power. Their use avoids loss of power due to processor inefficiencies and reduces weight and cost associated with screen power processing.

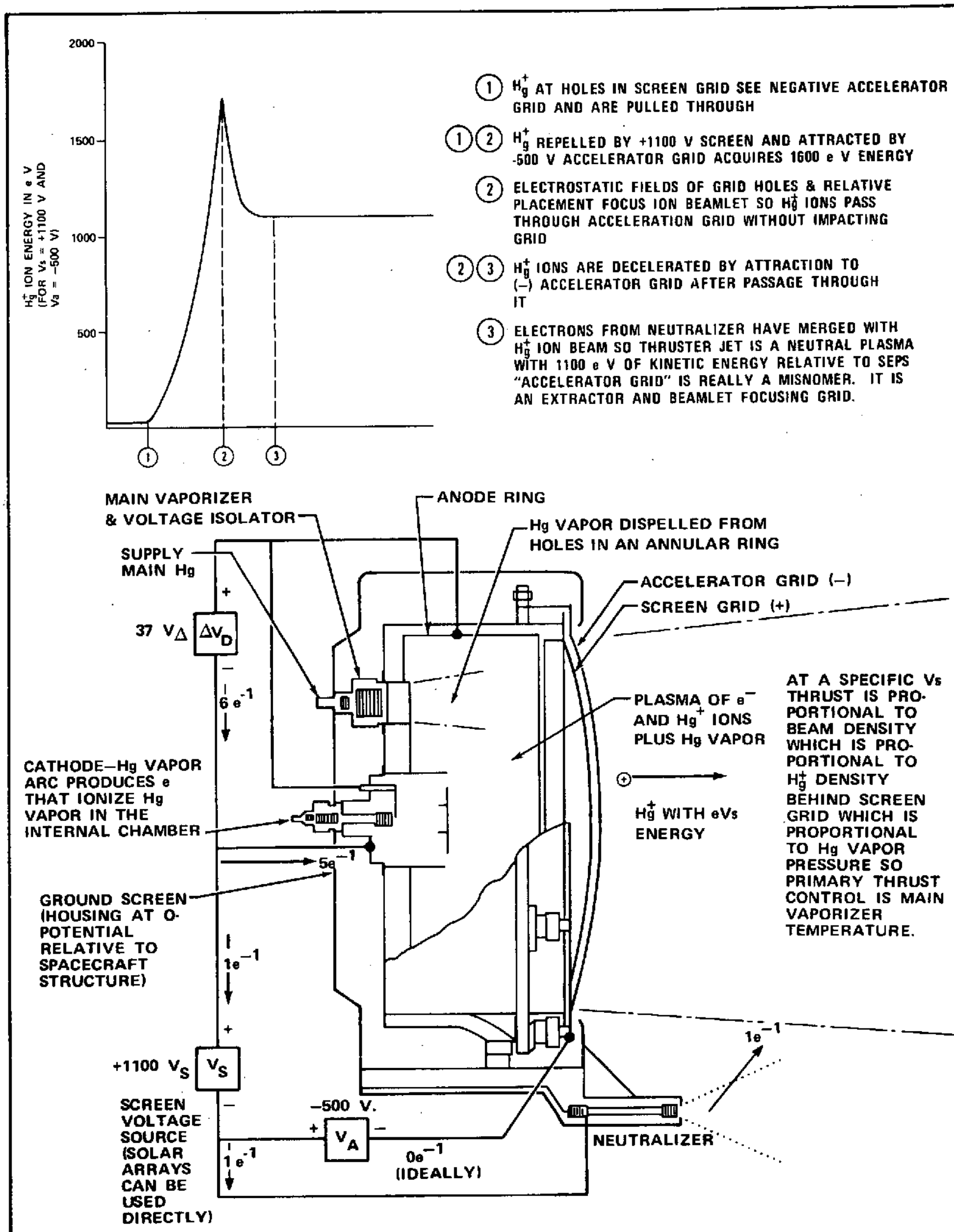


Figure 1-1. SEPS THRUSTER SCHEMATIC

PHYSICAL SIZE AND TEST CHARACTERISTICS INFLUENCING SUPPORT REQUIREMENTS

The SEPS dimensions when packaged for transportation or in the launch configuration are 3 meters by 3 meters by 5 meters. A variety of surface or air transport options exist for transport from manufacturing site to operations support center and to launch site without requirement of special vehicles or handling gear.

The SEPS is essentially an electrical device with relatively simple mechanical subsystems. No expensive test devices, other than vacuum chambers now in existence and used only in initial thruster subsystem acceptance tests and for Design, Development, Test and Evaluation (DDT&E), are required. The operational and sustaining engineering force and facilities required for SEPS total program support is therefore small.

1.6.2 The Space Transportation System with SEPS As A Transport Element

The system elements are shown on Figure 1-2. No physical changes or additions to the Shuttle are required for SEPS operation in the system. A standard family of "kick stages" should not be defined until more information exists on the character of payloads and specific mission requirements. For this study, a representative kick stage that could be fitted with different numbers of solid rocket motors was assumed. For earth orbital missions, SEPS eliminates the need for any kick stages or payload velocity addition ability in the payloads themselves for achieving initial mission position, or for retrieval of payloads after mission accomplishment. For other missions, planetary and earth escape, SEPS reduces auxiliary propulsion performance requirements without placing any demands or constraints on the kick stages. SEPS offers the potential for recovery of Tug instead of expending it for many missions. The scope of this study did not allow investigation of that potential.

The study ground rules supplied by NASA defined an Interim Upper Stage (IUS), which is a "stretched tank" transtage for use through 1983 and a baseline Space Tug defined by MSFC for use from 1984 onward. SEPS requires no characteristics of these vehicles that are not required for their missions when operated independently of SEPS. Because SEPS can always accomplish the remaining portions of any combined SEPS plus IUS or Tug missions by extensions of the required SEPS trip time, SEPS removes the development schedule and cost risks that are associated with meeting burnout weight and propulsion performance goals from the IUS and Tug programs. The use of SEPS reduces the number of IUS and Tug flights required to accomplish the reference mission model.

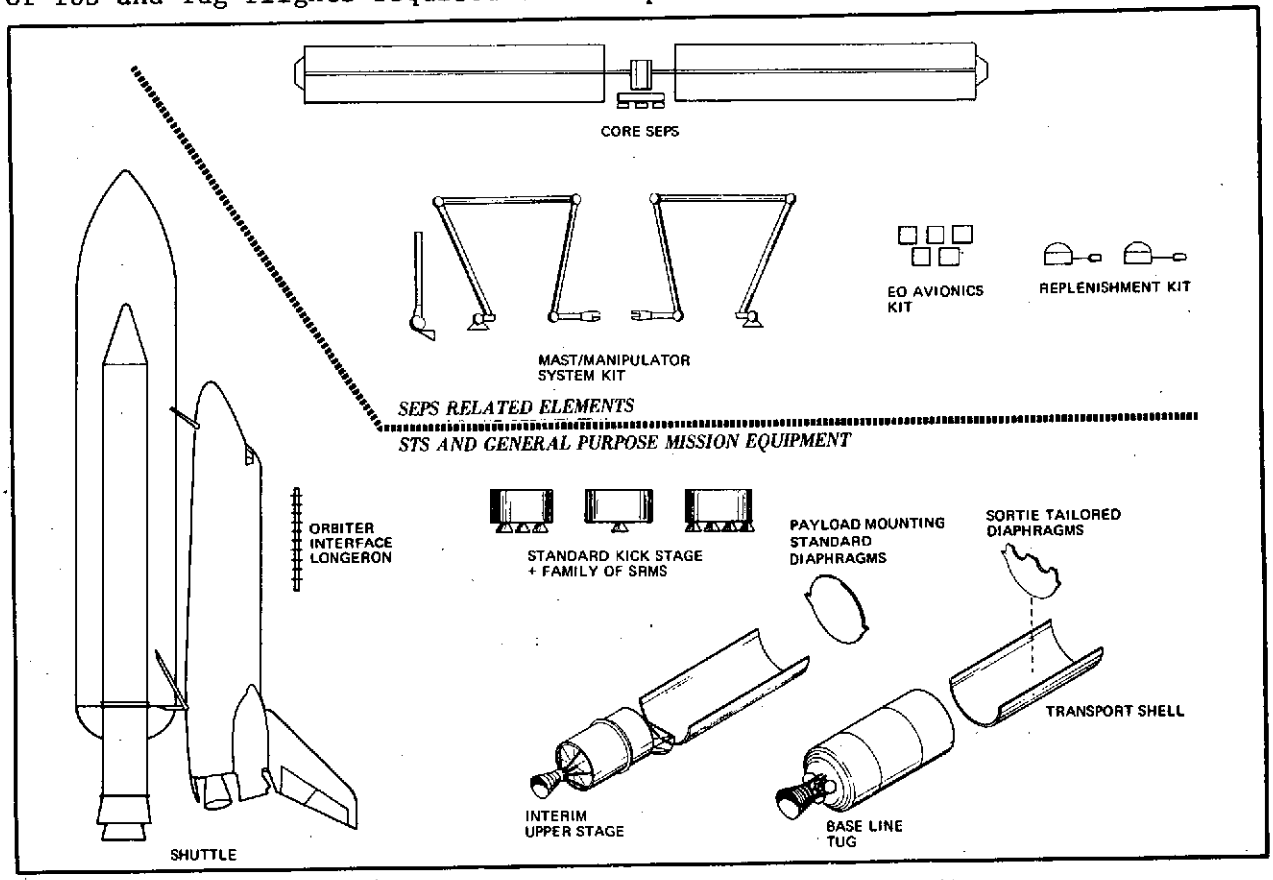


Figure 1-2. STS WITH SEPS SYSTEM ELEMENTS

The system characteristics and programmatic cost factors identified in this study indicate that a single core SEPS vehicle should be developed. NASA has directed that this study concentrate on the operational characteristics of a 25 kw power level SEPS. NSI, for reasons to be described later under principal trade studies, believes that greater power levels are desirable. Except for trade study discussions, all SEPS configuration, performance, and operational characteristics discussed in this volume are those of a 25 kw power level configuration.

The core vehicle is produced in a single continuous production run to minimize production cost of the 11 flight articles and one test article which is refurbished to provide the second spare vehicle for the program. There are eight SEPS committed to four (dual launch) planetary missions and three to earth orbital (with one spare considered as an earth orbital vehicle). The planetary missions are 1981 Europe Rendezvous, 1981 Jupiter Orbiter, Metis Rendezvous and Mercury Orbiter. The communication, navigation and guidance, and data management subsystems of the core vehicle are standard although they are operated in different modes for the planetary mission and the earth orbital missions. Major blocks of the software are naturally different.

For the earth orbital kit the avionics system contains four TV cameras, two located on the manipulator arms and two located on the scanning platforms with other core vehicle navigation and guidance sensors. The earth orbital function utilizes a scanning LADAR for rendezvous with payloads and other elements of the STS. The scan platform mounted TV cameras can serve as backup for the LADAR. The core SEPS is capable of autonomous navigation and guidance on planetary missions. With the addition of a horizon sensor or an Interferometric Landmark Tracker (ILT) and radar altimeters, the SEPS has autonomous navigation and guidance capability for earth orbit missions.

The extendable payload mast and manipulator system kit, to be described later, provides near universal adaptability for in-space handling, servicing, retrieval, and maintenance of payloads without forcing severe configuration or geometric arrangement constraints on payload developers. The software required to prevent human operators from commanding manipulator functions that could cause equipment damage, and the software which allows simplified manipulator hand steering to desired locations, requires less than 32,000 word of computer memory (a SUMC memory block 3.7 cm x 25 cm x 25 cm). The combined mechanisms required for the full range of payload and multiple payload transport functions is simpler with manipulators than with any other system providing even the basic capabilities.

The economy of the STS operation to accomplish the total NASA supplied reference mission model in the years 1981 to 1991 demands multiple payload deployments on each Tug-Shuttle flight. For example, 83 percent of the payloads can be arranged in flight manifests for a Shuttle comprising five or more individual payloads. Figure 1-3 shows the frequency of Shuttle flight manifests versus the number of individual payloads on the manifest. On some flights, some of these individual payloads go to intermediate orbits and are not transported by SEPS.

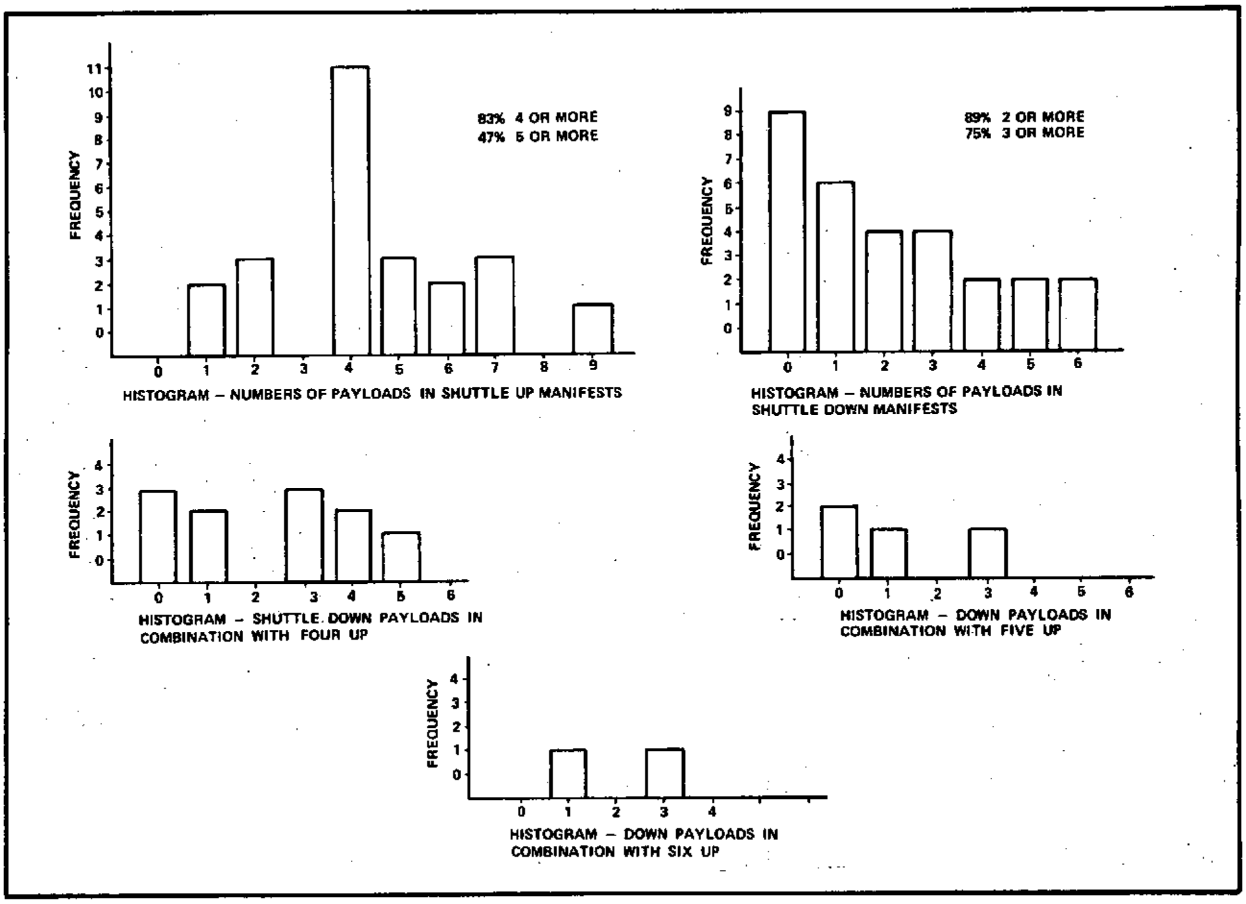


Figure 1-3. FREQUENCY OF OCCURRENCE VERSUS NUMBER OF INDIVIDUAL PAYLOADS IN CARGO MANIFESTS

In order to isolate Shuttle and Tug operations from the potential delays of launch preparation associated with the integration of four or more payloads into a single launch package and to provide payload users with simple, easy access to their payloads, NSI generated a standard transport shell and payload mounting diaphragm concept. This concept allows all payloads for a specific flight to be integrated into a single package prior to mating the package to the Tug. The Tug plus "package" is then mated to the Orbiter as a single payload.

Since each payload is mounted directly to a diaphragm, interactions between the individual payloads are minimized, and access to individual payloads is simplified.

The payload transport shell is a lightweight half cylinder, honeycomb core, monocoque structure. The standard diaphragms for payload mounting have multiple payload mount structural attach points and are reusable General Purpose Mission Equipment (GPME). Specially tailored payload mount diaphragms are fabricated for those infrequent conditions where unusual payload attach requirements exist.

Satellite systems are presently being designed for 10-year operational lifetimes. Several presently operating satellites have been in orbit for 6 to 9 years. SEPS operational life for each mission cycle was assumed for cost analysis purposes to be 5 years. The expected operational life is much longer. If propellant for the total lifetime in space is carried on early SEPS sorties, trip times are unnecessarily long. To shorten average trip times, methods for replenishing expendables must be implemented. SEPS has only two expendables, the main propellant (mercury), and the attitude control system propellant (N_2H_4). Both propellant supply subsystems are N_2 accumulator pressurized so that replenishment may be accomplished by simply forcing propellant from the replenishing tank into the depleted storage tanks which recompresses the expulsion gases during the replenishment. The SEPS manipulators provide the inherent ability for self-servicing on any payload delivery mission where Tug brings up an expendables replenishment kit with the payload group to be transferred to SEPS. The probable limiting factor on SEPS operational life in space is thruster lifetime. Technology programs directed toward extending thruster life are highly desirable.

1.6.3 SEPS Configuration and Functional Characteristics

The foregoing discussions described the elements composing an STS plus SEPS transport system. At the beginning of any discussion on SEPS configuration, several basic factors should be emphasized. The active elements of SEPS are very compact. Once operational in space, the greatest acceleration that SEPS is ever exposed to results from its attitude control system thrusters. Their absolute thrust level requirement for control and docking is extremely low. The level is therefore chosen based on accelerations that make for operator convenience and reduce the time that mission control centers must be involved in SEPS operations. Peak accelerations are in the range of 0.002g to 0.01g. Any desired deployed geometry in space can therefore be implemented at a very small penalty in structural mass increase. The active elements of SEPS have no preferential orientation except to meet the condition that solar arrays must be orientable normal to the sunline, and radiation cooling panels must have at least one face orientable to dark space. Many equally attractive arrangements of SEPS power production and thrust producing components are possible.

The decision controlling factors regarding SEPS overall characteristics, therefore, are primarily related to the functional interfaces with the payloads, and STS General Purpose Mission Equipment (GPME). In summary, the decision controlling factors are:

- STS transportation efficiency depends on multiple payload deliveries and multiple retrievals
- Cost effectiveness requires that GPME be usable on successive flights without modification and with few special payload adapter items
- The GPME must simplify Shuttle-Tug operations
- Multiple payload transport must place minimum constraints on payload designers
- Design should provide for easy replenishment of expendables
- GPME mass increase to simplify other STS operations does not reduce SEPS plus Tug net payload capability; modest trip time increases allow SEPS to make up for Tug's lower payload transfer orbit ability
- SEPS capabilities are almost directly proportional to design power level in the range of 25 to 100 kw.

With the characteristics controlling factors identified, selection of criteria for choosing a SEPS configuration must be established. These criteria derive from national and NASA policy decisions rather than technical facts. No configuration choice is defensible without final reference to some of these criteria. The selection choices are to configure for:

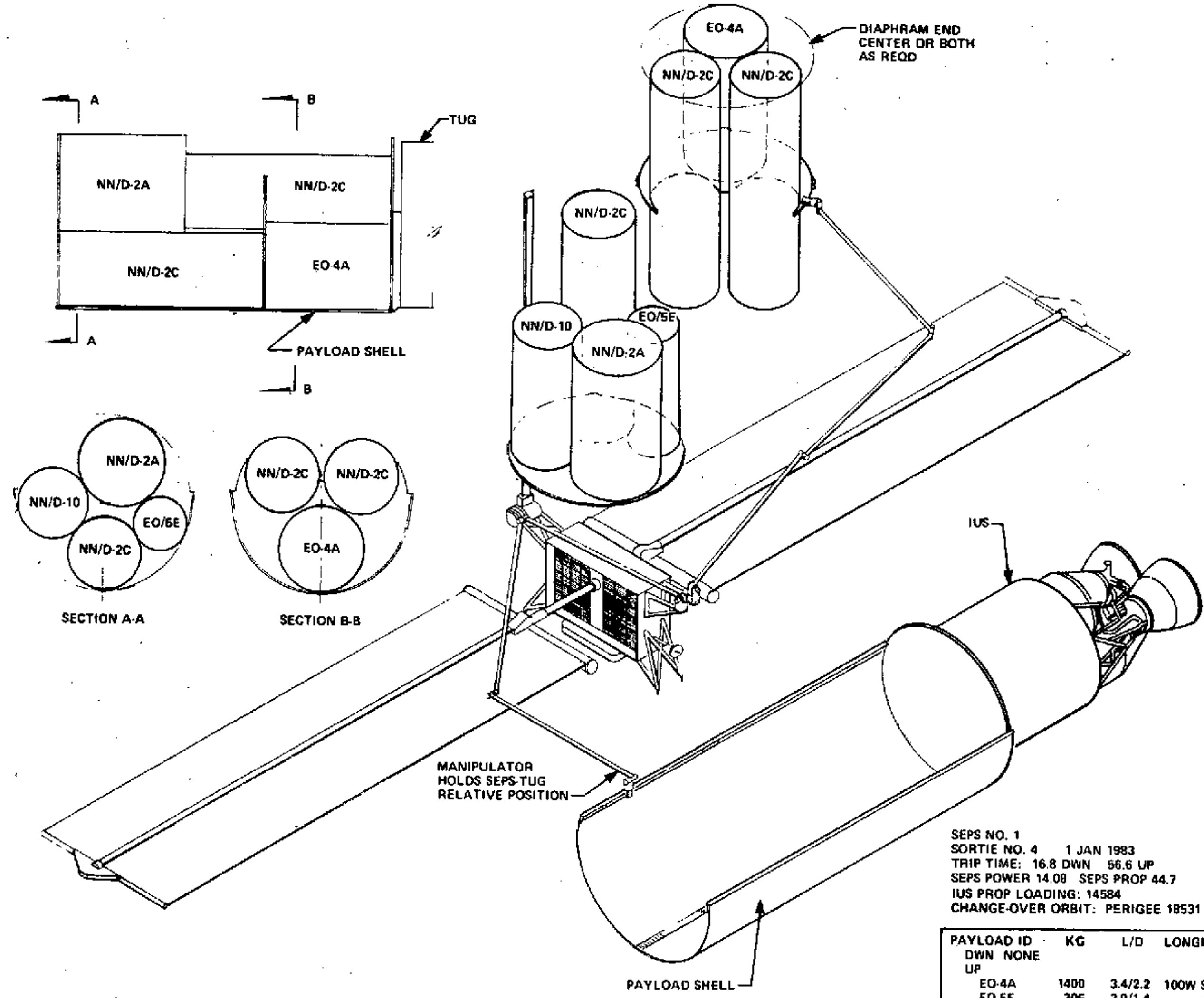
- The minimum to meet absolute mission needs for some reference mission model existing on a certain date, or
- Cost effectiveness against a reference mission model considering only transport vehicle operational cost savings, or
- Total cost effectiveness plus those low cost characteristics that greatly enhance functional capability and mission versatility, since mission models and payload concepts are at present inadequately defined and are constantly changing as the value of new missions and concepts are recognized.

Based on the analyses of this study, the foregoing decision factors, and NSI's belief that the last criteria above is the logical choice, the conclusions regarding SEPS configuration and Space Transportation System GPME associated with SEPS sorties are:

- A standard payload transport shell to facilitate Tug handling of independently mounted multiple payloads should be developed.
- A manipulator/extendable payload support mast system for SEPS will result in low operational cost and impose the minimum design constraints on payload developers.*
- Screen power direct from the solar arrays with inherent Isp option to match specific mission requirements will reduce the size of required solar arrays for a given thrust, improve reliability and reduce radiator panel size.
- SEPS transportation capability within a specified trip time is almost directly proportional to power. SEPS development costs are only slightly increased by power level and operational costs are reduced. SEPS should be developed with power level greater than 25 kw.

The basic configuration recommended for SEPS and GPME is shown on Figure 1-4. To illustrate the recommended system's capability, one of the sorties from the baseline 25 kw SEPS System Operational Profile will be briefly described. The sortie is a 1983 flight from one of the master schedules generated to accomplish the reference mission model where the Interim Upper Stage (IUS) brings 7 payloads up to payload transfer orbit to meet SEPS. The seven net payload masses SEPS will deploy at its final mission destination total about 3860 Kg.

**A detachable mission kit of these items for Tug would provide desirable capability for quick response services.*



SEPS NO. 1
SORTIE NO. 4 1 JAN 1983
TRIP TIME: 16.8 DWN 56.6 UP
SEPS POWER 14.08 SEPS PROP 44.7
IUS PROP LOADING: 14584
CHANGE-OVER ORBIT: PERIGEE 18531 APOGEE 47905 INC 4.2

PAYLOAD ID	KG	L/D	LONGITUDE	DESCRIPTION
DWN NONE				
UP				
EO-4A	1400	3.4/2.2	100W	SYNC EARTH OBS
EO-5E	306	2.9/1.4	–	APPLICATION EXP.
NN/D-2C	441	5.5/1.9	N/A	US DOM SAT C
NN/D-2C	441	5.5/1.9	N/A	US DOM SAT C
NN/D-2C	441	6.5/1.9	N/A	US DOM SAT C
NN/D-2A	493	3.4/2.3	N/A	US DOM SAT A
NN/D-10	366	3.1/1.8	140E	GEOSYNC OPER MET SAT

Figure 1-4. PAYLOAD TRANSFERS SEPS NUMBER 1, SORTIE NUMBER 4

The expendable IUS without SEPS could deliver only about one-half this net payload weight to geosynchronous orbit and would have to deploy all payloads at one point. Each payload would therefore have to be designed to independently maneuver to its final mission destination. Without SEPS two IUS plus Shuttle flights would be required to deploy these seven payloads.

On Figure 1-4 each cylinder represents the envelope dimensions of a payload from the reference mission model. The reference mission model and payload dimensions were supplied by NASA as guidelines for the study. The code letters on each cylinder correspond to a payload whose mass, dimensions, and descriptive title are given in the legend.

This particular example is sortie number 4 for the first SEPS which was launched in 1981. After completion of sortie number 3 SEPS had been dormant in geosynchronous orbit awaiting commands to initiate actions for implementation of sortie number 4. In response to preplanned schedules, the SEPS cruise down to the elliptical rendezvous orbit (18,520 km perigee by 47,967 km apogee) was initiated some 17 days previously. In accordance with the mission plan, Shuttle with IUS and payloads was launched and through the standard mission procedures IUS was targeted on the known conditions of SEPS. IUS achieves the target conditions within its navigation and guidance system accuracy.

Ground tract may order an IUS correction or SEPS may initiate final rendezvous action immediately.

To shorten rendezvous times SEPS will use a combination of its chemical Attitude Control System (ACS) and ion propulsion system thrusters. SEPS will be the active partner in the rendezvous and payload transfer operation with IUS. For this operation with Tug, Tug will be the active partner until station alongside SEPS at 100 to 300 meters is achieved. After this time SEPS is the active partner until completion of the payload transfers.

SEPS closes on the IUS which is passive but in an attitude hold mode. Closing is based on range, range rate, and line of sight data from the LADAR and/or the scan platform mounted TV system.

At the option of the SEPS Operations Center (SEPSOC) flight control final approach maneuvers are controlled by onboard systems in an autonomous manner or by a payload transfer controller on SEPSOC. Final closing is accomplished in a parallel or other nonintersecting velocity vector mode so that human or other errors do not result in catastropic conditions. When on station alongside Tug or IUS, the ground command pilot steers a manipulator end effector (hand) out to position to grasp the payload shell. Views from TV cameras, body mounted on SEPS and on each manipulator arm, are employed as visual aids in accomplishing this action. After the manipulator "hand" grasping the payload shell has been clamped, the attitude control system of both vehicles are deactivated to conserve propellants. If a preferred space orientation is desired for any reason, such as a special lighting effect, one of the vehicles' ACS would hold attitude. The manipulator arm holds the vehicles in their original relative geometric positions.

The other manipulator hand is steered to one side of the transport shell to release the latch holding the diaphragm to which the first group of payloads are mounted. The manipulator then deploys a payload mast clamp on the diaphragm and releases the payload umbilical through which the IUS/Tug supplied the payload electrical and data system connections, and then releases the diametrically opposite latch and grasps the diaphragm for transfer on the first payload set to the payload transport mast.

The payload transport mast comprises a pair of preformed biconvex sections edge welded so that, when wound on a drum, the edge welded sections collapse into parallel metal ribbons held on the drum by the combination of winding tension and forces resulting from the geometry of the housing. When the drum is driven in the (unwind) extend mast direction, the ribbons spring to their preformed shape. The biconvex sections are suprisingly strong in bending and have high torsional rigidity because of the edge welding of the ribbons.

This payload transport mast is commanded out to any position required for mounting of both payload sets. The diaphragms have spring loaded clamps that lock onto the mast when pushed against it.

The manipulator grasps the diaphragm containing the first payload set at a location where the TV camera on the arm can be slewed so that its field of view contains the diaphragm edge where the mast clamp is located. The payload transfer controller (teleoperator) commands the manipulator to lift the payload set out and place it on the payload mast. For direct control, the visual aids provided are the scan platform mounted TV on the mast side, the scan platform mounted TV on the manipulator side, the TV on the back of the manipulator hand holding the payload shell which can be slewed to see along or into the IUS-payload shell, and the previously mentioned TV on the back of the manipulator holding the diaphragm.

The manipulator's detailed joint motion and arm segment positions required to achieve "hand" motion along a desired path are controlled by the computer. The ground controller flies the "hand" in the sense that he commands translational rates of the hand and rotational rates about its three rotational axes. The computer also provides damage avoidance by forbidding any geometry of the arms that will result in collisions of any type. The computer also prevents acceleration of masses being translated by the arms to velocities greater than those the manipulator system can brake before the mass contacts any element of the combined spacecraft and payloads system.

The system has flexibility in the degree of automation which can be selected for any operation. For example, if after the first hand is steered to grasp the payload shell at the beginning of the transfer function, the grasp position is given to the computer along with the shell geometry, payload geometry, initial diaphragm positions in the payload shell, and desired attach locations on the SEPS transport mast, then the computer could execute the desired payload transfers without active participation by ground controllers. The memory block size (32,000 words) required for the full automation option is equal to the memory block size required for the autonomous navigation and guidance system plus all other SEPS functions and therefore may be considered as a fully redundant memory block for the SEPS central computer.

Trade studies which led to choice of the manipulator mast system as the simplest for the combined functions of transport, deployment, retrieval, transfer, and servicing of payloads are summarized in Section IV.

Again referring to Figure 1-4, after SEPS has completed the payload transfer operation, the manipulator still holding the payload shell and attached IUS is used to push the space vehicles apart so that neither vehicle's ACS thrusters are used.

After the vehicles have separated adequately, if the mission were conducted with Tug, Tug begins preparation for initiating the phasing orbit and transfer orbit maneuvers to return it to the Orbiter.

SEPS initiates cruise mode. For the sortie payload group used in the Figure 1-4 example it requires 57 days to achieve geosynchronous orbit. With SEPS autonomous navigation and guidance accuracies, the only demands on STDN during this 57-day period are weekly status checks on SEPS STDN determined status versus its own autonomously determined status. Payload data requirements may dictate more frequent STDN data link usage. Many payload developers will have facilities such that for appreciable parts of the trip time direct communications with SEPS will be possible.

Because of SEPS low acceleration it does not use phasing orbits, but is started on trajectory profiles so that continuous thrusting for the minimum length of time will bring it to the desired rendezvous or payload deployment point. The terminal phase of SEPS to a target point for deployment of a payload, or to a rendezvous, is just an extension of the cruise phase as indicated on Figure 1-5a. For sunlit targets, the SEPS, with information from the ground as to target payload position, can acquire the target at distances up to 7,223 km and begin line of sight tracking. Figure 1-5a shows the relative motion of SEPS approaching a target geosynchronous payload when only the ion thrusters are used in order to conserve ACS propellants. Times are times before station alongside the payload at relative velocity 0. The arrows indicate the direction of thrust. Figure 1-5b shows added details of the last few hours.

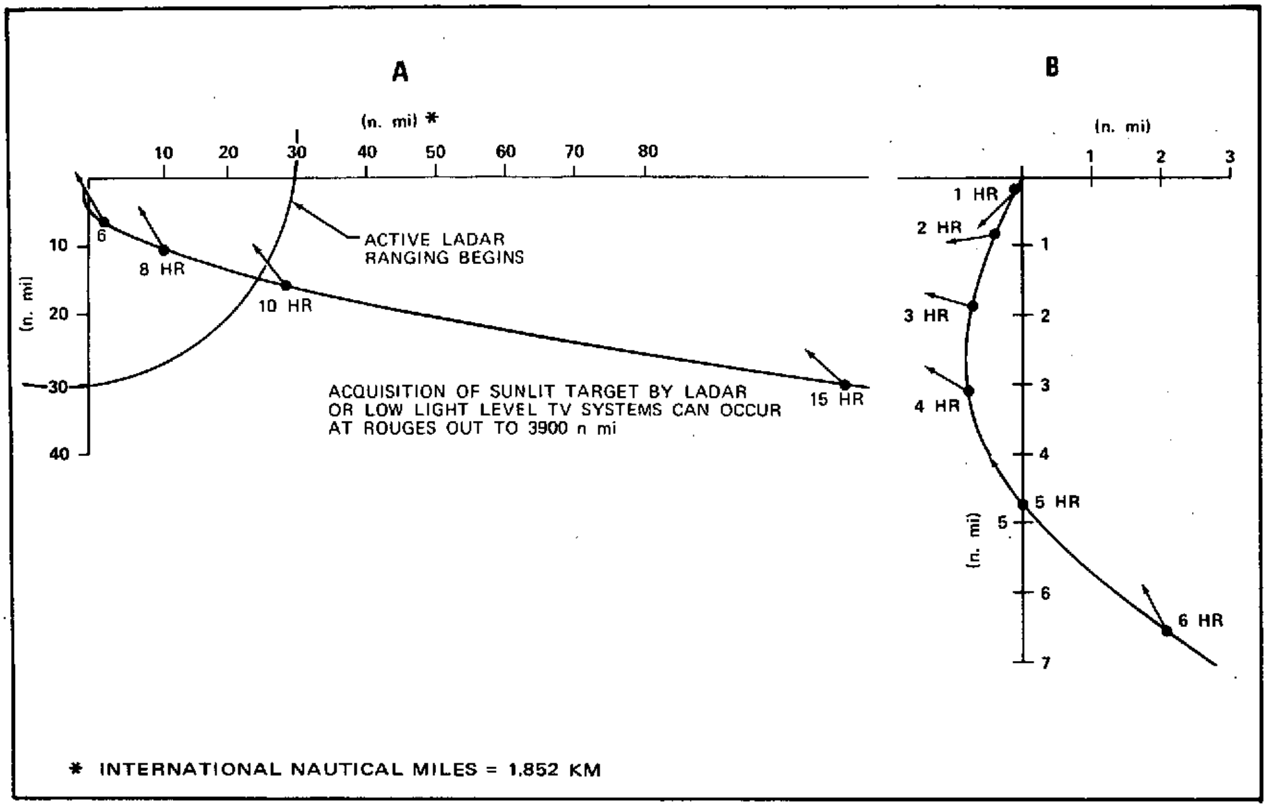


Figure 1-5. SEPS RELATIVE MOTION APPROACHING TARGET

The SEPSOC flight control center would not need to be fully manned prior to about 2 hours before payload deployment or retrieval was to begin. Conversely, if it is desired to compress the last 6 hours of the operation, ACS thrusters can be utilized. These thrusters, combined for additive thrust in the same direction as the ion system, provide about 100 times the acceleration of the ion system.

During a typical mission cycle, usually 10 or more sorties, SEPS may be refueled 3 times.

Replenishment of ACS and mercury propellant will not be described in any detail since, from the payload transfer discussion and the sketches on Figures 1-4 and 1-6, SEPS inherent capability for self-replenishment is obvious. The relatively small amounts of ACS propellant (N_2H_4) and the high density of the mercury propellants result in such small volumes for the replenishment kits

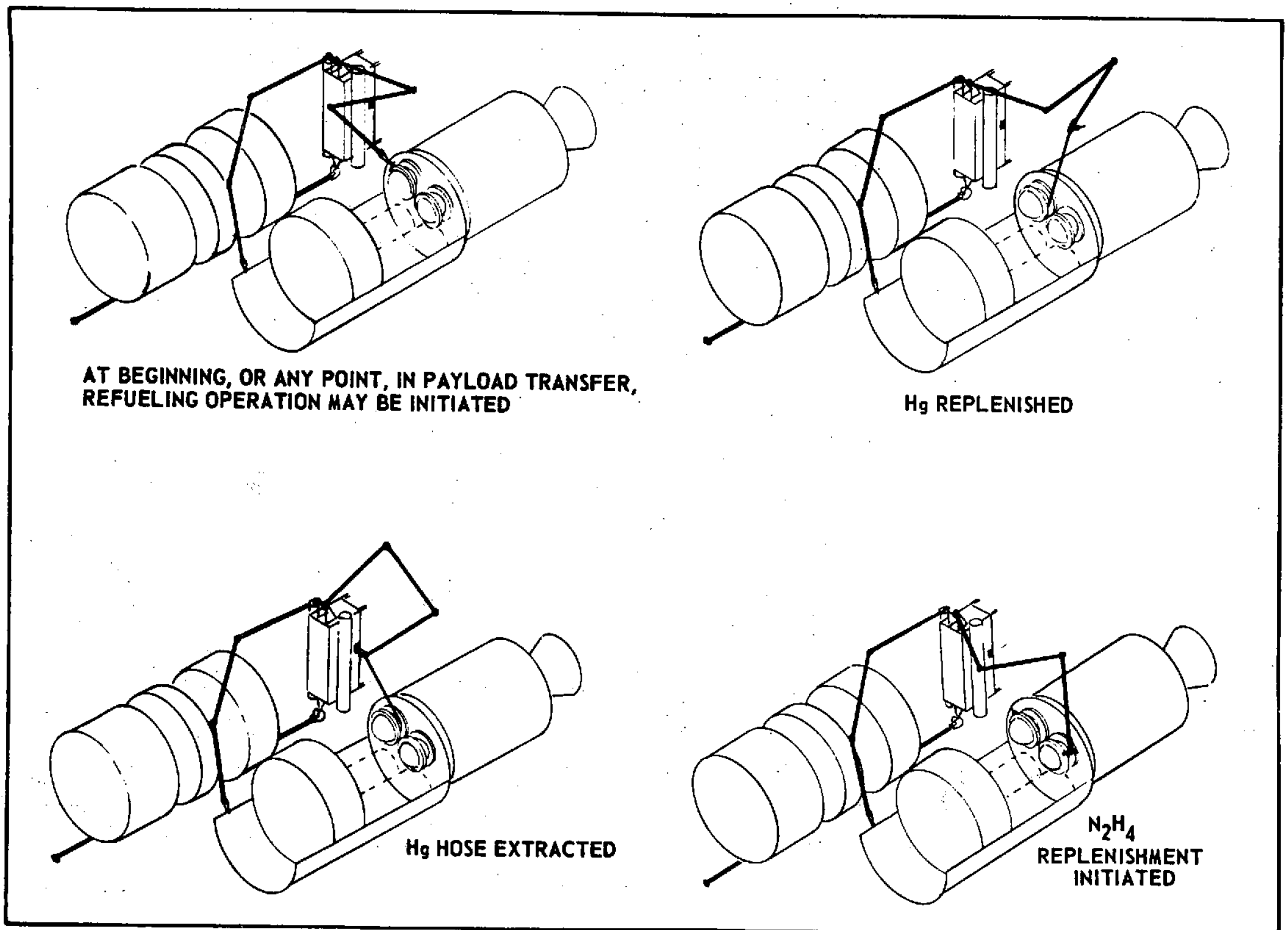


Figure 1-6. REFUELING SEQUENCE

that they have frequent opportunities to be carried on IUS-Tug sorties where the payloads are not utilizing all the available cargo space. Thus, flights dedicated solely to SEPS replenishment were never required throughout the entire 1981 to 1991 time frame encompassed by the reference mission model. Design concepts for the refueling equipment are described later in this report.

SEPS has a significant potential for self-repair as well as for servicing and maintenance of other satellites. The manipulators with a set of in-space changeable hands or end effectors are extremely versatile payload servicers, payload element deployment assistors, and malfunction repair tools. The broad range of applications of manipulators in automated production and assembly operations and their uses in nuclear reactor core and fuel element recycling attest to the well developed state-of-the-art.

NSI does not believe that the high reliability and long service life expectancy of properly designed SEPS subsystems warrants design for in-space maintenance in a spacecraft that can be retrieved and returned to earth for repair. If further analysis indicates in-space maintenance to be desirable, SEPS physical and functional characteristics are such that it has the inherent potential to be an "erector set" type spacecraft. Various subsystems can be attached to a core structure. Figure 1-7, a modification of some NASA technology program designs, illustrates this. Specific design for in-space maintenance, if it were an initial program requirement, should not be expected to increase total DDT&E program cost and could actually reduce total program cost if program management exploited the resultant characteristics of the system in a diligent cost reduction effort. Without further discussion, Figure 1-7 is presented so that the program concept assessor, with a little imaginative consideration of design detail offered by present technology, can envision the flexibility of the manipulators for many types of functions:

- space experiment interchange on laboratory type spacecraft
- servicing and repair of other spacecraft
- replacement of SEPS components if such design approach should later prove warranted.

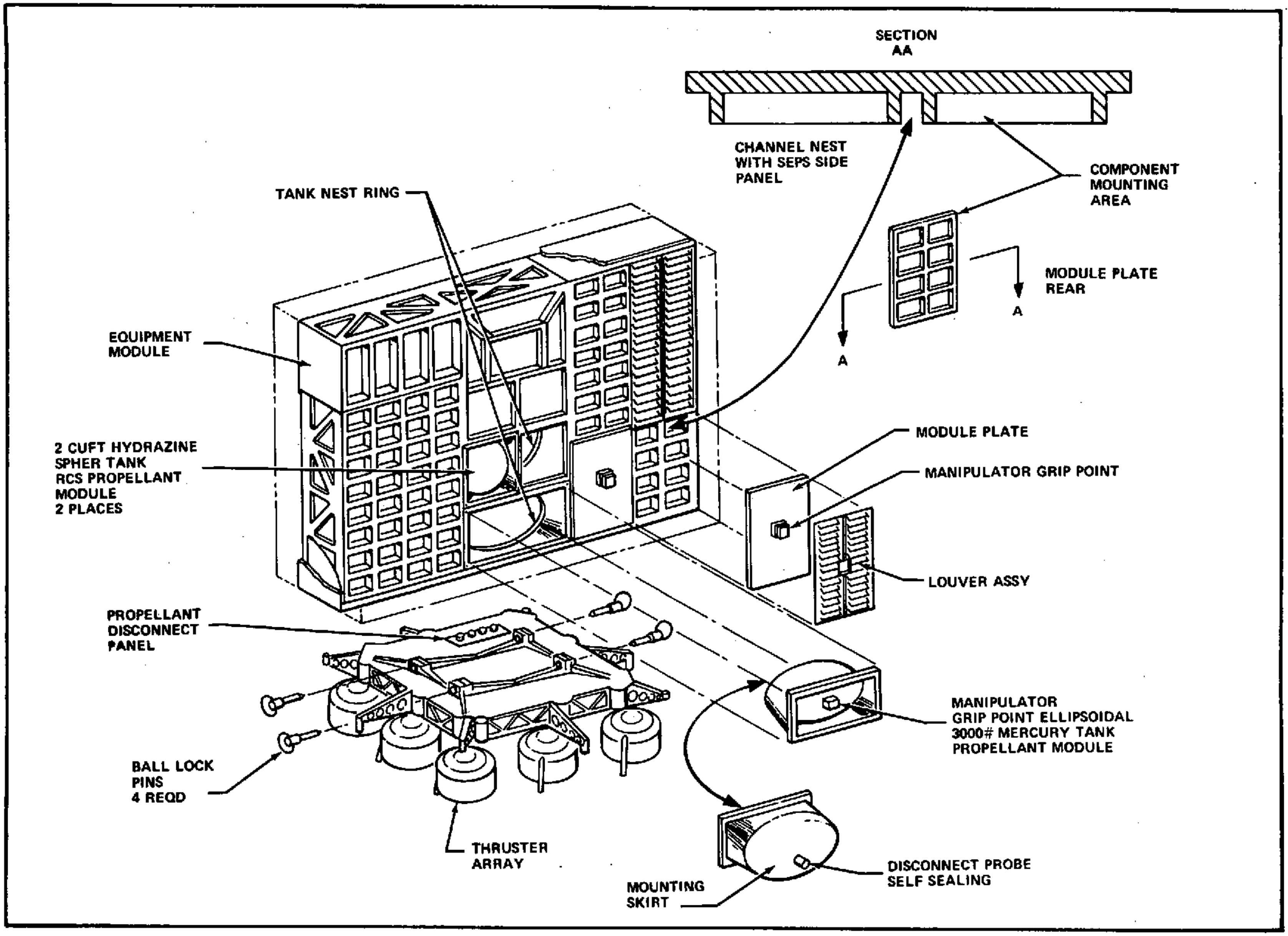


Figure 1-7. SEPS POTENTIAL FOR IN-SPACE MAINTENANCE

1.6.4 Mission Roles for SEPS in Accomplishing the NASA Reference Mission Model

The reference mission model was derived from "The October 1973 Space Shuttle Traffic Model" (NASA TMX-64751 Revision 2 dated January 1974) by considering all flights from year 1981 through year 1991. SEPS functions in accomplishing the mission model are summarized as follows:

- SEPS-Tug combined missions to geosynchronous orbit with intermediate orbit payload deliveries comprised 124 payload deployments or retrievals which represented 93 percent of all geosynchronous payload missions and 47 percent of all intermediate orbit payload missions
- SEPS accomplishes four of the 16 planetary missions. Because backup planetary spacecraft are flown, the four missions require eight SEPS launches
- Tug alone accomplishes only 7 percent of the geosynchronous missions but 53 percent of the intermediate orbit missions.
- Low earth orbit missions are feasible for SEPS but we found no significant cost savings for this transport role.

A summary of the total mission model and SEPS utilization in accomplishing it is shown in Table 1-1.

Table 1-1. ACCOMPLISHMENT OF PAYLOAD MISSIONS REQUIRING UPPER STAGES

Total Payload Missions **879**
- **Shuttle Only** **644**
- **Requiring Upper Stage** **235**

MISSION CATEGORY	MISSION IN EACH CATEGORY	DIFFERENT PAYLOAD TYPES	TUG ALONE		TUG WITH SEPS RENDEZVOUS	
			No.	%	No.	%
GEOSYNCHRONOUS	133	17	9	7	124	93
ESCAPE	45	22	39	87	6	13
POLAR EO	33	5	33	91	0	0
HIGH ENERGY EO	9	3	9	100	0	0
INTERMEDIATE EO	15	2	8	53	7	47
TOTAL	235	49	95	40	137	58

Mission roles for SEPS with the Space Transportation System are seen to be predominantly in the geosynchronous orbit delivery, retrieval, and payload servicing area. In the study NSI was directed to establish cost effectiveness of an earth orbital SEPS strictly on the basis of direct transportation cost savings. Many other obvious benefits occur from SEPS capability.

Direct transportation cost savings derive from the fact that with SEPS the required number of earth orbital Shuttle-Tug flights is 15 less than required to accomplsih the mission model without SEPS. Other minor factors such as fewer expended IUS and kick stages result in a net transport cost saving of $126 million after all earth orbital SEPS development, production, start up, and operations costs are amortized. The $126 million saved represents a 217 percent return on the delta $58 million investment in SEPS for earth orbital operations. The total STS with SEPS Operational Profile to accomplish the mission model is shown on Figure 1-8. The comparison of cost for earth orbital STS transport functions that require upper stages with and without SEPS are summarized in Table 1-2.

Table 1-2. STS COMPARED TO STS WITH SEPS FOR TRANSPORTATION COST EFFECTIVENESS -- EARTH ORBITAL FLIGHTS REQUIRING UPPER STAGES

COST ELEMENT (DOLLARS IN MILLIONS)	BLSTS		BLSTS BLSEPS (20 KHR-REFUELED)	
	$10^6$$	NUMBER	$10^6$$	NUMBER
SHUTTLE FLIGHTS @ $11.09	1508.	136	1342.	121
IUS EXPENDED @ $5.17	103.	20	98.	19
IUS WITH KICK STAGE @ $6.37	13.	2	13.	2
TUG RECOVERED FLTS @ $.96	87.	91	74.	77
TUG RECOVERED EXPENDED KS @ $2.16	15.	7	15.	7
TUG EXPENDED @ $14.16	0.	0	0.	0
TUG AND KS EXPENDED @ $15.36	92.	6	92.	6
TOTAL TRANSPORTATION COST	1818.		1634.	
$ SAVED IN TRANSPORT COST	--		184.	
VEHICLE INVENTORY COST SEPS @ (VARIES WITH PRODUCTION)	110.	9*	146.	11**
SEPS DEVELOPMENT & OPERATIONS	122.		144.	
TOTAL SYSTEM COST	2050.		1924.	
NET $ SAVED	--		126.	

*8 PLANETARY VEHICLES PLUS ONE SPARE

**8 PLANETARY VEHICLES PLUS ONE SPARE PLUS TWO EARTH ORBITAL VEHICLES

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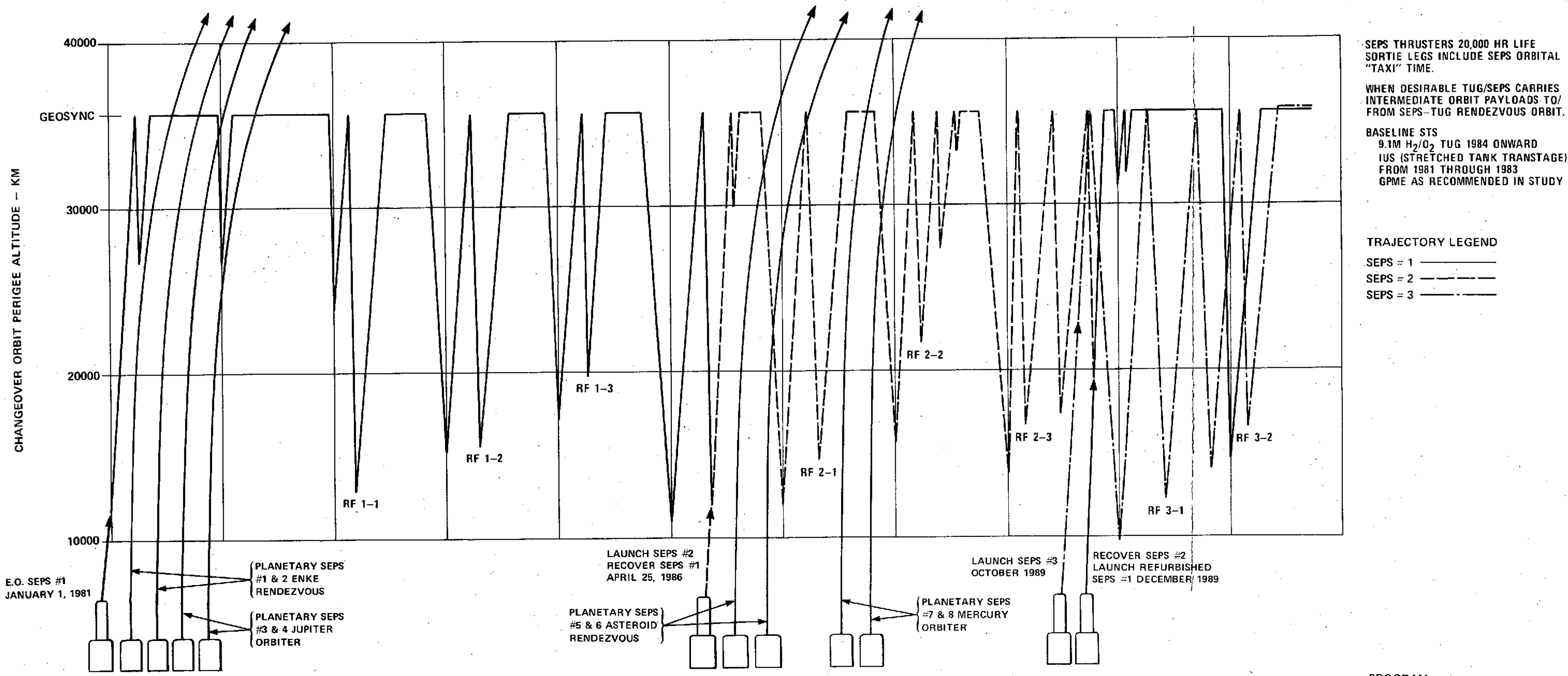

												PROGRAM TOTALS
SHUTTLE	19	22	28	27	33	32	30	32	30	30	32	315
SHUTTLE/TUG	6	2	8	11	13	12	11	4	6	9	10	92
SHUTTLE/TUG/ EO SEPS	2	1	2	2	2	3	2	4	5	5	2	30
YEAR	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	437

Figure 1-8. SYSTEM OPERATIONAL PROFILE (9.1-METER BASELINE TUG + 25 KW SEPS WITH 20,000 HOUR THRUSTER LIFE - REFUELABLE)

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FOLDOUT FRAME 1

FOLDOUT FRAME 2

In the above comparisons the STS operating without SEPS was given every advantage to assure that its full potential was utilized. No constraints were placed on Tug operating alone in regard to the number of payloads Tug could return in a single trip even though Tug would have to have equipment not presently planned for it that is capable of multiple payload retrieval. This equipment might be similar to a SEPS manipulator set plus a payload transport shell. Any of the practical alternates we investigated had nearly equivalent weight and complexity but a great deal less mission flexibility. Transport assumptions favorable to STS operating without SEPS in a transport role were:

- Tug payload transport and retrieval gear weight total was only 136 kg (more realistic weight penalties are 272 kg).
- All multiple payload retrieval flights had payloads collected at one point by some arbitrary means so Tug did not have to taxi around geosynchronous orbit to collect them.
- All multiple payloads to geosynchronous orbit were deployed at one location in geosynchronous orbit and the payloads provided their own propulsive power to move to their final mission locations.

In other studies conducted on STS without SEPS various analysis groups have made arbitrary assumptions as to the payload packaging geometry that would be allowed for multiple payload flights and also as to the total number of up and down payloads to be allowed on one flight in order to reflect Tug's limited ability when not equipped with payload handling gear such as SEPS's. The effect of some of these assumptions on Shuttle flights required to accomplish the mission with and without SEPS as a transport element are shown in Table 1-3.

Table 1-3. COMPARISON OF STS FLIGHTS REQUIRED VERSUS ALLOWED PACKAGING SYSTEM TO ACCOMPLISH ALL MISSIONS REQUIRING UPPER STAGES

STS VARIANT/PACKAGING SYSTEM	TANDEM	SIDE BY SIDE	THREE DIMENSIONAL	THREE DIMENSIONAL
BASELINE STS	156	150	150	136
STS WITH SEPS	146	129	125	121
STS FLIGHTS SAVED	10	21	25	15

NOTES:
1. *Number of payloads for Tug operating alone limited to three up and one down on each sortie for all cases except those in the last column*
2. *General purpose mission equipment designs evolved in this study make any number of payloads per sortie feasible up to STS volume or mass limits*
3. *SEPS high performance essentially removes payload weight per sortie limits*
4. *Available payload volume in Orbiter cargo bay becomes the significant limiting factor.*

NSI therefore believes that the cost saving equivalent to a reduction in Shuttle-Tug flight requirements by 15 flights is an extremely conservative estimate of transportation savings occurring from operation of the SEPS as an STS transport element. NSI believes that considerably more than the previously presented 217 percent return on EO SEPS development and operational start up cost investment would be achieved for actual operations conducted under the general management and operational concepts described in this study final report. Shuttle flights and STS cost savings are not the only benefits SEPS provides. Its real potential is in the major capabilities not taxed by this mission model and in its versatility for missions not yet identified.

1.6.5 SEPS Benefits to IUS, Tug, and Payloads

In addition to the transportation cost saving defined earlier, SEPS provides other programmatic cost savings and operational simplifications.

BENEFITS RELATIVE TO IUS

- The IUS is not required to have a navigation and guidance system capable of active participation in rendezvous operations even if it is a recoverable system.
- Costly research and development programs to improve propulsion capability or reduce inert stage weights are not required since SEPS can make up any IUS performance deficit.
- IUS flight preparations are greatly simplified. Payloads can be individually mounted into the transport shell. The multiple payloads in the transport shell package can be checked for flight readiness combined with IUS in a single mating operation. IUS plus multiple payloads are presented to Shuttle as a single payload.
- It is feasible to recover IUS on many missions if it is equipped with the proper avionics equipment.

BENEFITS RELATIVE TO TUG

- Schedule and cost risk associated with high performance requirements of the Tug program are removed.
- Tug operations are simplified. Multiple payloads are presented to Tug as a single package ready for flight.
- Tug docking and payload interface, other than electronic, may be developed for a single payload interface rather than for multiple docking and retrieval operations.
- Fifteen to 27 fewer Tug flights are required to accomplish the mission model.

- Tug does not have to be designed for the long stay times in space necessary to perform orbital taxi missions for multiple payload deployment or retrieval.

BENEFITS RELATIVE TO PAYLOADS

- Reduction in transportation cost prorated to each payload. Average number of payloads per flight in SEPS case is approximately four and for Tug alone is less than two.
- Essentially removes weight restrictions for payloads. Development cost increases to solve missed initial program weight goals will not be incurred.
- Higher initial payload weight allowances can be used to reduce development cost, improve reliability, or to provide for functional capabilities not feasible for payloads delivered by Tug alone.
- SEPS can deploy various payload elements (or undeploy them for retrieval) to either backup payload on-board systems or relieve the payload entirely from self-deployment requirements. This should considerably reduce the development cost of some payloads.
- Most payload failures prior to end of design life are of the infant mortality type. SEPS can maintain station alongside a recently deployed payload with its TV cameras transmitting visual records of the payloads deployment and initial functional test responses to the payload developer's ground control commands. SEPS can assist in correction of the malfunctions. Upon ground command SEPS can return the payload on the next rendezvous with Tug, if onorbit correction of the malfunction was not possbile.
- SEPS can service payloads by providing for substitution of new sensor packs, or different experiments that may extend the usefulness of large optical or other instrument platforms without requiring their recovery or replacement in space.
- SEPS can provide replenishment services for payload expendables.
- For planetary missions SEPS allows significantly greater payload mass and may provide power, communication, attitude, and thermal conditioning support to the payload. For some planetary orbiting payloads, SEPS can modify orbital parameters to conduct complete surface mapping operations plus mapping of fields and particle physical phenomena in space around the planet.
- Combination of science packages with SEPS can provide nearly ideal spacecraft for comprehensive surveys and continuous monitoring of earth's magnetosphere and near earth solar system space. "Out-of-the-ecliptic" missions are examples of the latter. New spacecraft do not need to be developed for these missions. SEPS itself may be considered a "standard" spacecraft.

- Where the payload scientific objectives require mission orbits so greatly separated in energy level that it is not practical to provide spacecraft propulsion to accomplish the change, SEPS can taxi the spacecraft to its new orbit, thus saving a new Shuttle launch of a new spacecraft.

1.6.6 New Mission Applications for SEPS

This study, by work statement requirements, was directed primarily toward earth orbital mission roles, development of payload handling concepts, and analysis of operation support requirements. Roles in accomplishing the mission model with STS were described in some detail. Other potential applications of SEPS are:

- Spacecraft host supplying power to a direct broadcast satellite for educational TV and general communications to family units and villages in remote areas of the US or of the world. A valuable function of the system is its use in the event of hurricanes, ice storms, or any natural emergency that isolates communities by interruption of their normal communications channels. The system could serve ships at sea, small fishing craft, and oil or other geodetic exploration units. The system would provide one-way TV and two-way voice communication.
- Support and provide space mobility for a high resolution earth observing satellite providing high data rate real time information on weather or other local phenomena. High resolution optics and other sensors could switch systematically from locality to locality providing detailed scan information for each area for the time the local area was under observation.
- Collection of space debris and removal from frequently used areas of near earth space by return to ground via Shuttle and Tug or transfer by SEPS to higher infrequently used space areas.
- Transportation of very large space structures from their initial assembly positions in low earth orbit to final functional positions.
- Mobile teleoperated assembly device for construction of large space structures.

1.6.7 Trade Studies and Technology Assessments

As in all systems, trade studies can be conducted at every level of the system's functional design detail. A principal objective of this study was to establish the first level trade of any system; namely, is its existence and operation justified on the basis of cost effectiveness, other identifiable benefits, and predictable future benefits?

The priority and scientific work of the planetary, cometary, and solar space exploration missions justifies initiation of the basic SEPS program. Investigations conducted during this study indicate that a reasonable case for initiation of the program can be made solely on the basis of its value for earth orbital missions and its cost effectiveness as an element of the Space Transportation System. NSI believes the combination of values for solar system exploration and earth orbital applications justifies high priority for early implementation of a SEPS development program.

Given a baseline SEPS, high cost effectiveness from its operation as an element of STS was established. Within the scope of this study it appeared that several major configuration trade studies and reassessments of baseline subsystem definitions were warranted.

The major trade study was evolution of the General Purpose Mission Equipment (GPME) concepts that simplify Tug operations with multiple payloads, simplify Shuttle Orbiter interfaces, and also provide SEPS with a highly flexible payload support and servicing subsystem. The results of that study evolved the concept presented earlier. The key element of the concept was SEPS manipulator system. Considerations leading to the selection are summarized in Table 1-4.

CHOICE OF SEPS POWER LEVEL

The next most significant configuration definition choice is associated with SEPS power level. The decision becomes largely a matter of judgement since no clear mission requirement sets a definite minimum power level in the range of practical choices and no technology factor or cost factor produces a sharp step in development difficulty or cost as power increases.

The transport capability and operational flexibility of SEPS with the STS is almost directly proportional to power level. To demonstrate this, NSI developed complete System Operational Profiles for accomplishing the reference mission model. The 25 kw NASA baseline profile was shown on Figure 1-8. Figure 1-9 shows the sortie trip times required by a 25 kw SEPS to accomplish

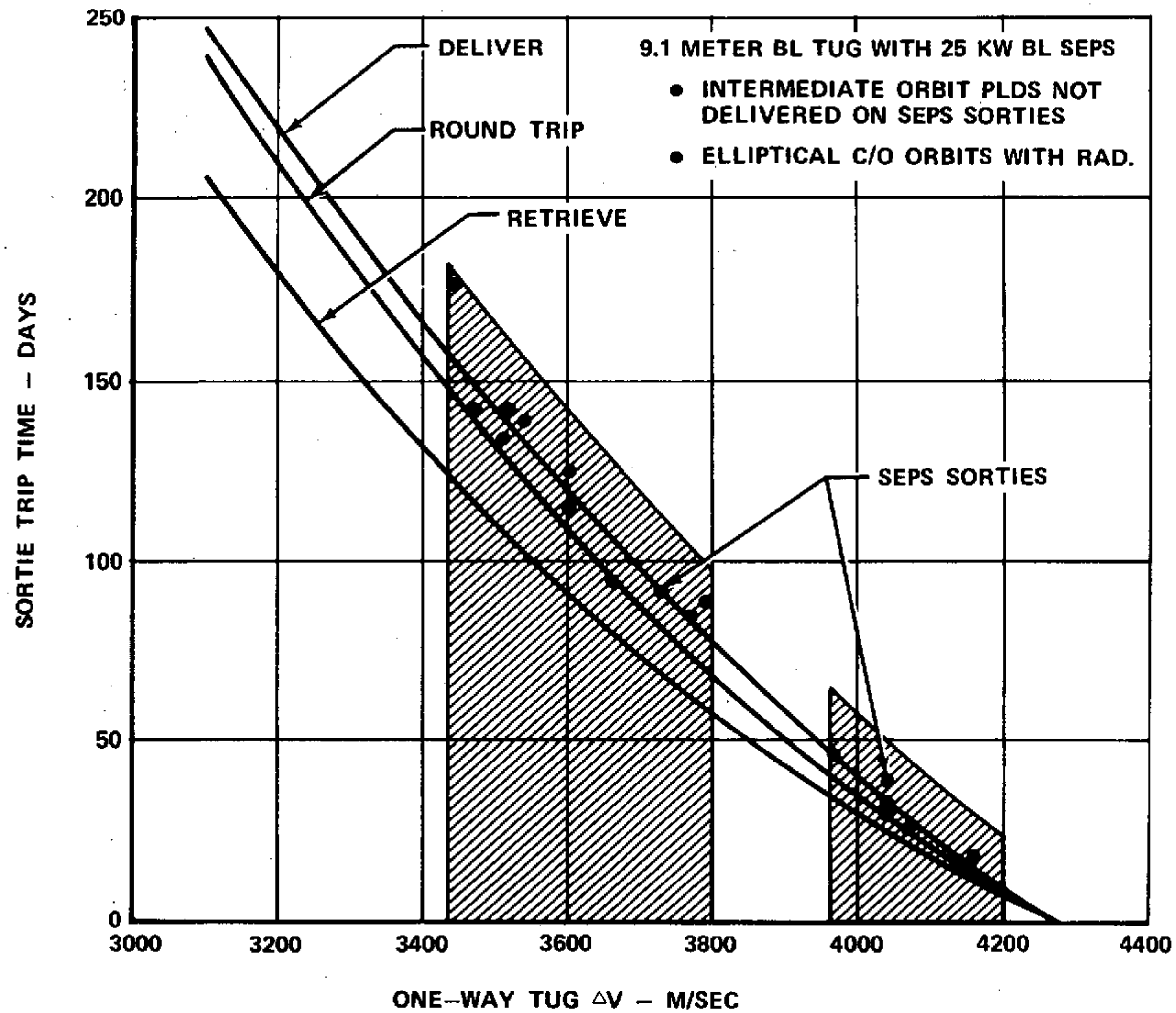

Figure 1-9. 25 KW 3000 Isp BASELINE SEPS SORTIE TRIP TIMES

Table 1-4. PAYLOAD SUPPORT, HANDLING AND SERVICING CONCEPT COMPARISON

ARTICULATED DOCKING FRAME AND ARTICULATED MULTIPLE PAYLOAD SUPPORT STRUCTURES	TRANSPORT SHELL, EXPENDABLE BOOM AND SIMPLIFIED MANIPULATOR	TRANSPORT SHELL, PAYLOAD MAST AND MANIPULATOR SYSTEM
ADVANTAGES	**ADVANTAGES**	**ADVANTAGES**
• SIMPLEST ONBOARD SOFTWARE	• MODERATE ONBOARD SOFTWARE REQUIREMENT • SIMPLEST PAYLOAD TRANSFER FUNCTION	• GREATEST INHERENT CAPABILITY FOR PAYLOAD SERVICES AND MAINTENANCE • MINIMIZES DESIGN CONSTRAINTS ON PAYLOADS • SIMPLEST AND MOST FLEXIBLE INFLIGHT OPERATIONS • SIMPLEST GPME & TUG PAYLOAD INTEGRATION FUNCTION • HIGHEST MISSION SUCCESS PROBABILITY
DISADVANTAGES	**DISADVANTAGES**	**DISADVANTAGES**
• MOST COMPLEX FLIGHT OPERATION • MOST COMPLEX FLIGHT HARDWARE • LIMITED GPME - REQUIRES TAILORING OF TUG MISSION EQUIPMENT & ORBITER TO PL ADAPTERS FOR EACH SORTIE • EITHER SERIOUS PL DESIGN CONSTRAINT OR VERY LIMITED SERVICING ABILITY • NOT ADAPTABLE TO UNFORESEEN OR UNPLANNED MISSION EVENTS • TOTAL COMPONENTS REQUIRING POSITIONING & FEEDBACK INFO EXCEED OTHER SYSTEMS	• LIMITED SERVICING AND ONORBIT MAINTENANCE ABILITY • INTERMEDIATE ADAPTABILITY TO UNPLANNED MISSION EVENTS	• ONBOARD SOFTWARE REQUIRES 32K WORD MEMORY STORAGE

delivery and retrieval missions in conjunction with a 9.1 M H_2/O_2 high performance Tug. The solid curves are the theoretical times required for SEPS to complete a mission with the maximum payloads that Tug could bring to the SEPS/Tug rendezvous orbit for the Tug one-way velocity increments shown by the abcissa.

The cross-hatched areas indicate the range of Tug velocity increments actually required to accomplish the mission model. The black dots are individual sortie trip times calculated with radiation degredation effects, and so forth. Figure 1-10 shows the sortie trip time savings of a 50 kw SEPS relative to the 25 kw SEPS. The system operational profile, as illustrated in Figure 1-8, does not utilize the full capability of a 25 kw SEPS until 1989 and does

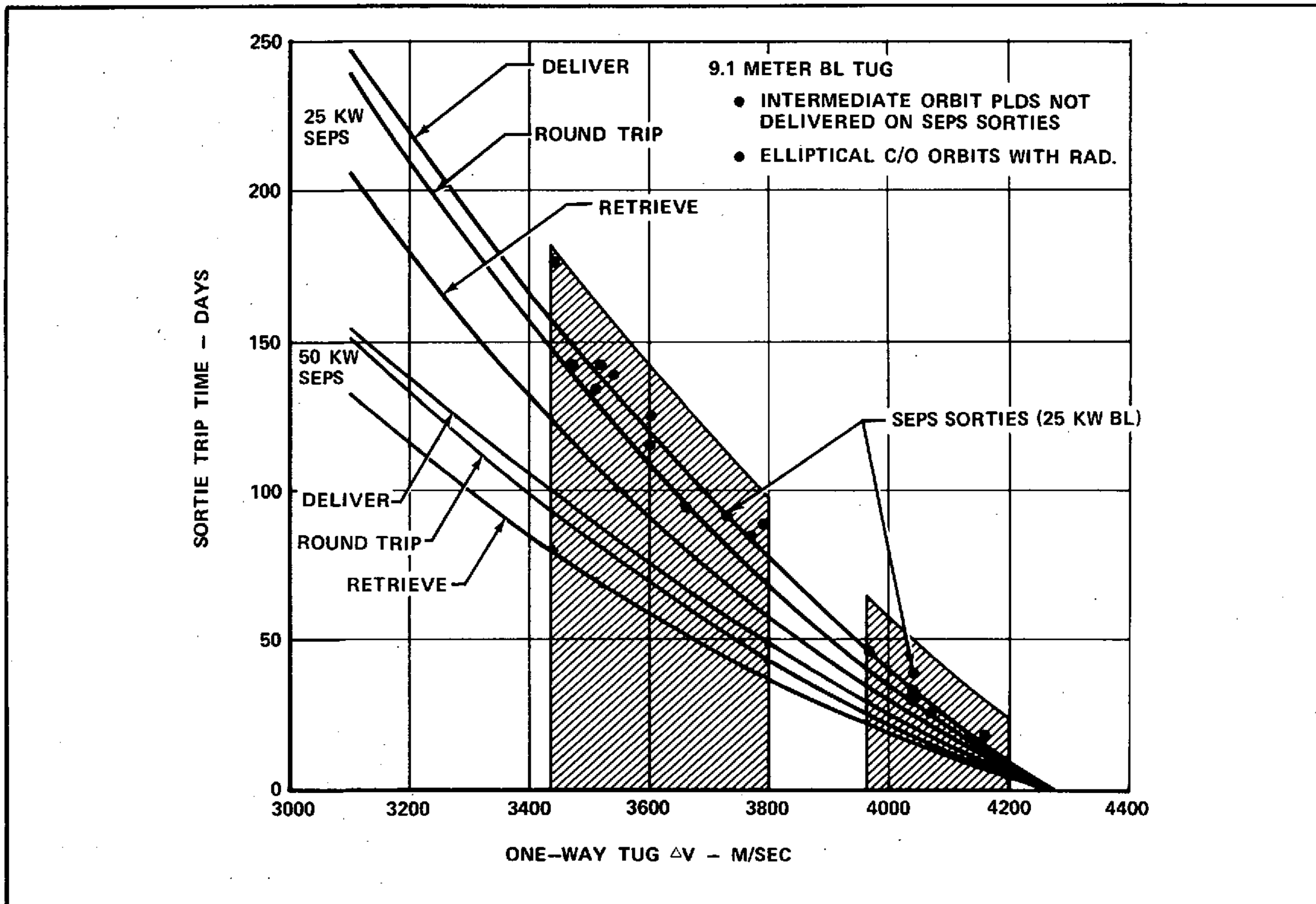

Figure 1-10. 50 KW 4243 Isp SEPS TRIP TIME SAVINGS RELATIVE TO THE 25 KW 3000 Isp BASELINE SEPS

not require two SEPS on orbit until 1990. Therefore, use of a 50 kw SEPS saves only 2 more shuttle flights than a 25 kw SEPS. The advantage of increased power for earth orbital operations with the reference mission model is therefore due only to:

- Reduction of the time required for execution of individual sorties
- The speed with which SEPS could respond to unplanned revisions of flight schedules
- Quick response to special demands for maintenance and/or retrieval of malfunctioning satellite.

Conversely, the DDT&E cost to develop a 50 kw SEPS was estimated by NSI to be only 7.5 percent greater than for a 25 kw SEPS so that a very small additional investment produced a transport vehicle of nearly twice the inherent capability. Figure 1-11 shows a size comparison between 50 kw and 25 kw power level SEPS. Table 1-5 shows a comparison of 25 kw and 50 kw basic costs.

Table 1-5. COMPARISON OF 25 kw TO 50 kw BASIC COSTS

COST ELEMENT	DEVELOPMENT		FIRST UNIT COST	
	25 kw	Δ FOR 50 kw	25 kw	Δ FOR 50 kw
STRUCTURES & THERMAL CONTROL	$ 4.8		$ 1.2	0.1
PROPULSION	9.1		2.0	0.8
POWER DISTRIBUTION	1.0		0.4	
SOLAR ARRAY	7.8		5.8	6.1
DATA MANAGEMENT	3.4		1.0	
COMMUNICATION	2.2		1.2	
ATTITUDE CONTROL/N&G	9.2		2.0	0.2
INTEGRATION & TEST CHECKOUT	6.7	1.0	1.1	1.0
TEST HARDWARE	21.3	6.5		
GSE	5.0			
SOFTWARE	4.5			
LOGISTICS	0.5			
SE&I	6.8		1.4	
PROGRAM MANAGEMENT	6.9		1.4	
BASIC SEPS	$89.2	Δ7.5	$17.5	Δ8.2
Δ FOR EARTH ORBITAL FUNCTIONS	8.3		1.0	
	97.5		18.5	
Δ FOR TUG PAYLOAD SHELL AND DIAPHRAGMS	2.5		0.8	
	$100.0	Δ% 7.5	$ 19.3	Δ% 42

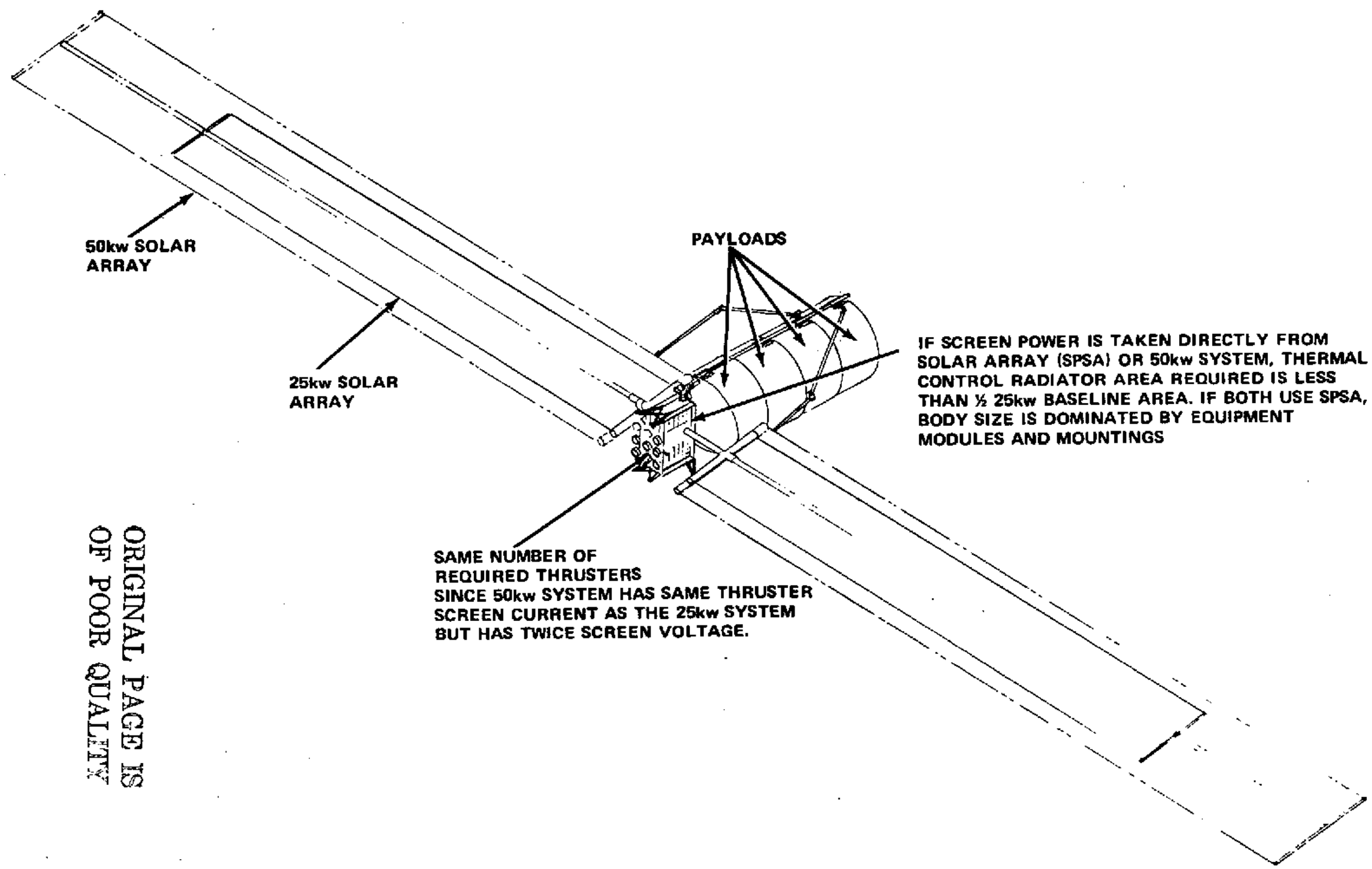

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Figure 1-11. SEPS SIZE COMPARISON 25 KW 3000 Isp BL TO 50 KW 4243 Isp SPSA

For the planetary missions the rate of gain in usable net scientific payload as power level increases varies considerably with the mission. In addition, the gains are sensitive to the mass-to-power ratio so that design approaches for SEPS thruster subsystem that result in high mass-to-beam power ratio or unjustifiably conservative mass estimates will cause apparent "optimum" power levels to be considerably lower than the true optimums. Even on the most conservative basis for mass-to-power ratio, such as used in Rockwell International 1972 and 1973 studies, trends for continuing growth in available net payload are indicated as power levels extend beyond 25 kw.

The planetary science packages conceived for most of these missions do not indicate the need for the higher payloads associated with the higher powers desirable for a SEPS operating in earth orbit. It is the opinion of this author at least, that the planned sciences packages are rather minimal and that a great deal more useful information would be obtained if the available payload mass allowed by the higher powered SEPS were used to fly on the planetary missions, some modification of the higher resolution, versatile sensors and instruments contained in proposed satellites such as the Synchronous Earth Observing Satellite (SEOS) and other environment determination and monitoring satellites. Figure 1-12 presents a review of typical planetary missions from earlier SEPS work by Rockwell International. The curves that show parametrically the influence of trip time and power level, the ordinates labeled "Approach Net Mass" are all masses (SEPS nonpropulsive and gross payload) in addition to the mass of the solar arrays and the thruster subsystem. If a standard core SEPS were used as the spacecraft bus, the gross payload would be approximately net mass minus 500 kilograms. For the Jupiter Orbiter the payload must include the chemical retro rockets for capture maneuver into a highly elliptical Jovian orbit.

The four sets of mission charts demonstrate two salient features. In all cases, increased power increases payload. For the mission beyond 1 AU power, SEPS can provide only limited payload support power if developed at the 25 kw of solar power level.

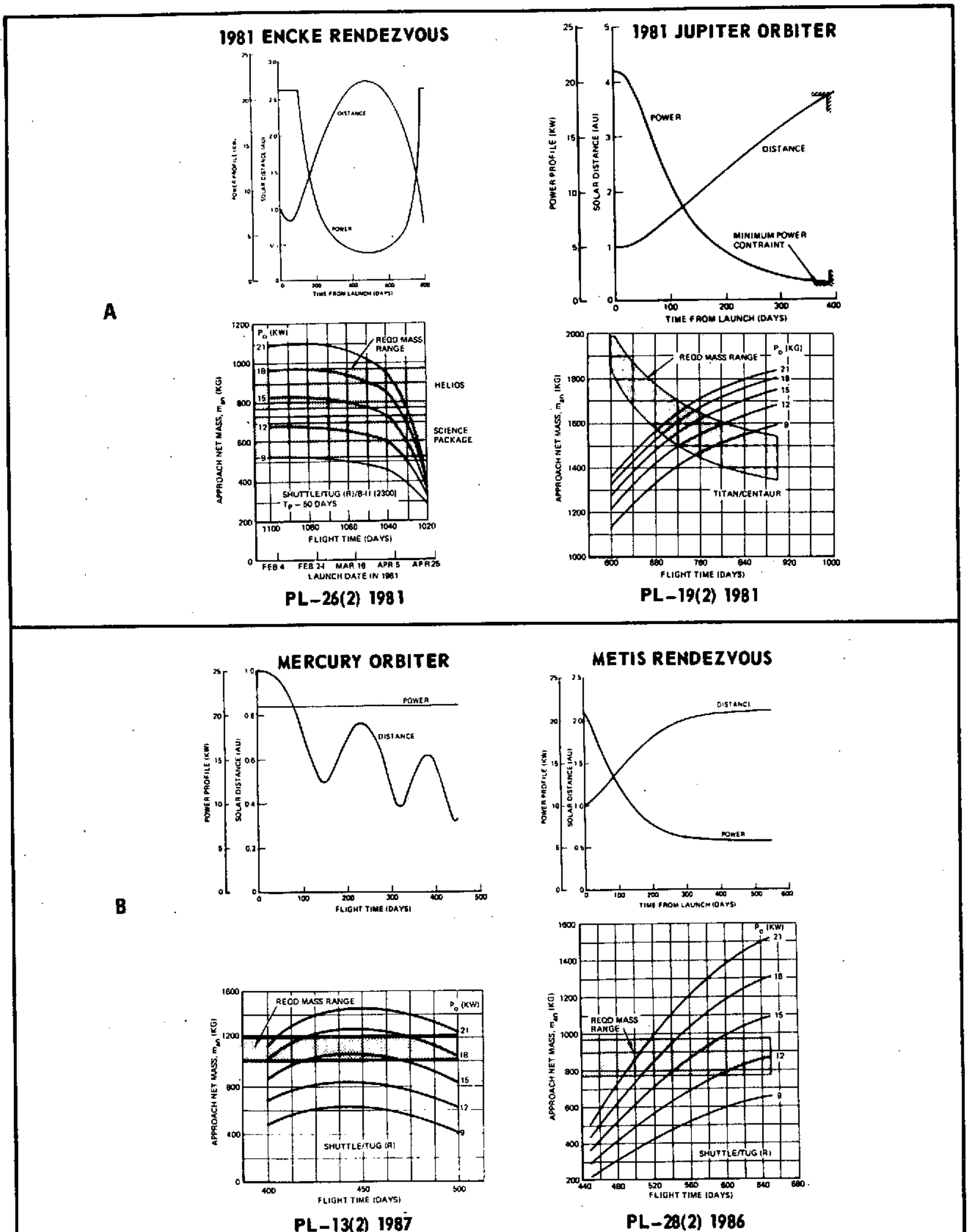

Figure 1-12. TYPICAL PLANETARY MISSIONS

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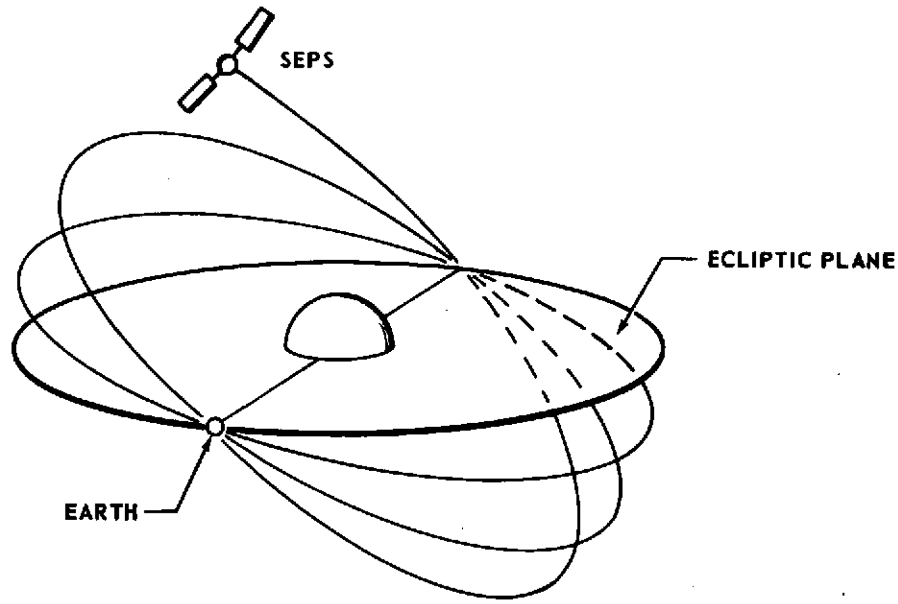
In the case of the Jupiter Orbiter mission, increased power beyond 25 kw would allow SEPS thrusters to operate during the approach to Jupiter, aiding in the capture maneuver, and also allow SEPS to modify the Jovian orbit for close inspection of each Jovian moon. When not thrusting, more power is available for communications so that high resolution imaging can be conducted in shorter periods of time. All of the RI work presented on Figure 1-12 was conducted with very conservative mass-to-power ratios based on processing screen power with associated losses and weight penalties. The Jupiter missions, which chemically retro SEPS into the capture orbit, will benefit greatly from improved (lower) mass-to-power ratios.

Figure 1-13 shows NSI's analyses of SEPS potential for an exciting new set of "out-of-the-ecliptic" missions that allow examination of the solar magnetosphere and solar surface with high resolution instruments over the entire solar sphere. In the particular example shown, the SEPS is launched by a Titan Centaur vehicle. The curves demonstrate the effect of three parameters. The curve showing the higher heliographic inclination versus mission time illustrates the advantages of increased power, better power-to-mass ratio by taking thruster screen power directly from the solar arrays, and the value of the option of operating at a factor of 2 greater (2200 Vs/1100 Vs) thruster screen voltage to achieve an Isp of 4243 seconds rather than a baseline 3000 seconds. The higher achievable inclination for the upper curve is due solely to the higher Isp and lower mass-to-power ratio from direct use of solar array power for screen power.

A design approach similar to that used on the 50 kw system but at 25 kw level would finally achieve the 80-degree inclination but in a much longer trip time.

This discussion has not covered all the implications of Figures 1-12 and 1-13. Thoughtful perusal of these figures will indicate that desirable characteristics for a standard core SEPS to achieve enhanced planetary mission suitability are:

- Improved average thrust-to-mass ratios



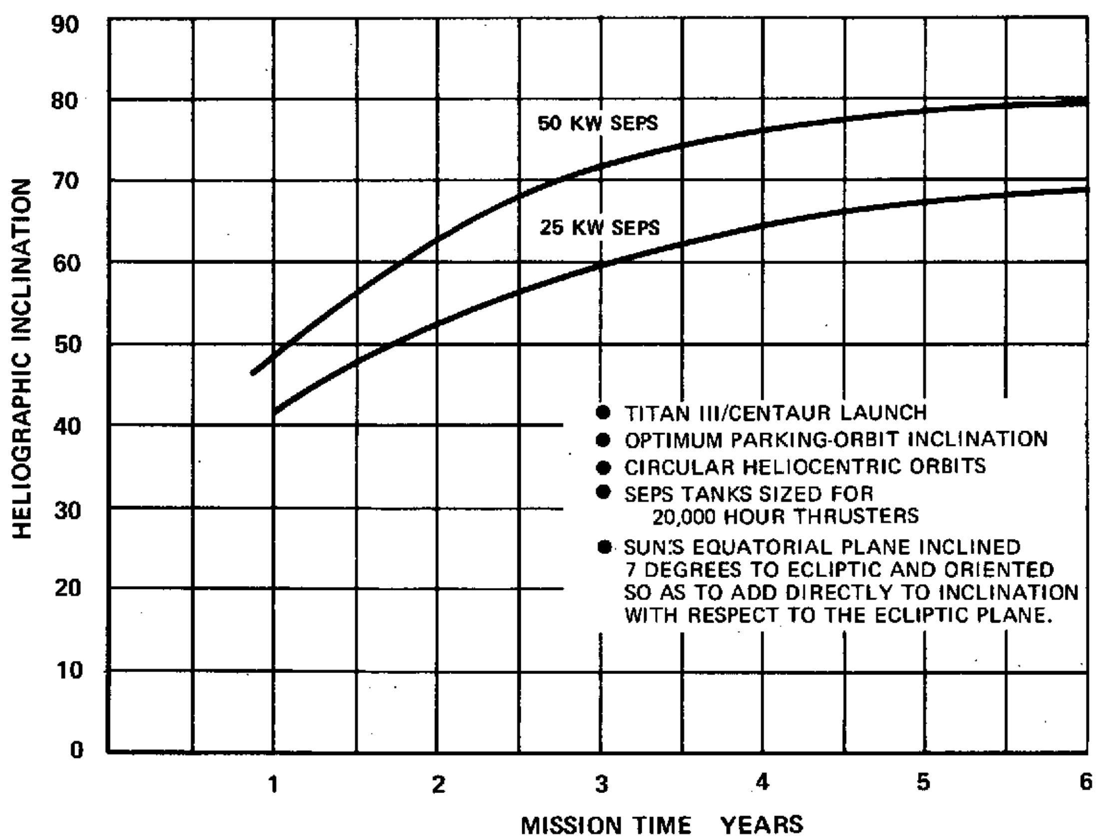


Figure 1-13. "OUT-OF-THE-ECLIPTIC" MISSIONS FOR SEPS

- Option to operate at high or low Isp to match requirements of a specific mission
- Reserve power to support larger payloads and higher communications rates at extended distances from the sun.
- Maneuver power to extend scientific mission capabilities after arrival at the target planet.

Improved average thrust-to-mass ratio can be achieved by:

- Increased solar array area and higher kw/kg values for the arrays by fuller exploitation of present technology
- Taking thruster screen power directly from the solar arrays and improving power processor efficiency for the remaining ≈20 percent of the power
- Fuller utilization of the ion thruster's inherent capabilities indicated by the last several years of NASA's technology program.

RELATED TECHNOLOGY ASSESSMENTS

NSI has reviewed the available technology base derived from NASA's thruster technology and research programs, has reviewed industrial developments of devices suitable for solid state power processing, and has reviewed the literature on solar cell technology. The conclusions of this assessment are:

- Thrusters have the inherent ability to operate over screen voltage ranges of about 800 v to more than 2800 v and at beam currents corresponding to .05 amp to 4 amps in a 30 centimeter thruster
- Solar arrays are both feasible and desirable direct sources of thruster beam power
- Higher voltage solar arrays (400 v up to 1100 v) are both feasible and desirable
- The potential exists for lower cost and higher reliability solar arrays than those assumed in prior studies
- Higher voltage power processors than those baselined for prior studies (200 v to 400 v) are feasible
- Exploitation of the technology base will provide a SEPS of significantly greater mission flexibility than the baseline derived from previous studies.

1.7 IMPACT OF SEPS OPERATION WITH STS ON ORBITER, IUS, TUG PHYSICAL INTERFACE REQUIREMENTS

1.7.1 General Considerations

The delivery to or retrieval of SEPS from typical IUS/Tug payload transfer orbits imposes no additional physical interface requirements since SEPS as an individual payload to be delivered has very modest support requirements well within the design capabilities proposed for IUS and Tug or those baselined for the Orbiter.

Figure 1-8, the System Operational Profile, showed that only four scheduled SEPS launches and one retrieval were required to accomplish the reference mission model from 1981 through 1991.

SEPS augmentation of IUS-Tug transportation capabilities allows the use of the GPME concepts described earlier, which greatly simplifies the Orbiter, IUS, and Tug ground operations involvement in multiple payload delivery operations. The transport shell always presents a single structural payload interface to the IUS, Tug, and Shuttle Orbiter. Because all payload inertial loads are distributed into the shell which distributes the total load to the Orbiter's cargo bay longerons in an acceptable way, loads on IUS and Tug are lower than design limit loads derived from certain individual payloads carried by IUS and Tug.

The additional interface requirements for STS elements therefore derive from the fact that with SEPS in the system multiple payload cargo manifests may contain up to seven or eight payloads instead of three to four. The primary impact, as might be expected, is in the avionics support areas of telemetry, command, and power supply.

Other potential added demands are in the areas of propellant dumping, venting, and RTG cooling, or other payload environmental factors. None of these represent extra requirements since the character of the multiple payloads with SEPS does not present a greater requirement than some of the more complex single and dual payloads transported without SEPS. Manifolding of multiple payload requirements on the transport stage results in interfaces equivalent to a single payload.

Safety and interface discussions will be considered in the following sequence:

- SEPS as one of a multiple payload group for delivery in terms of Orbiter safety requirements and interfaces
- Multiple payload avionics potential requirements
- Gases and liquids venting and dumping requirement

1.7.2 SEPS Safety and Interface Considerations in Relation to Orbiter

Figure 1-14 shows SEPS with other schematically represented payloads in a transport shell with Tug in the Orbiter cargo bay. IUS would mount similarly. The transport shells for IUS and Tug are essentially identical and could be developed for interchangeability. SEPS is mounted on a standard GPME diaphragm and has no direct structural interface with the Orbiter or IUS-Tug.

SEPS, if nominally fueled for the initial deployment mission, has a mass of about 2725 kilograms (6,000 pounds). SEPS contains only four fluids: pressurizing N_2, battery fluids, mercury, and hydrazine.

The pressurizing N_2 for the mercury expulsion system has a peak charged pressure of 58 N/m^2 (40 psia). The N_2 is contained inside the mercury propellant tank; the tank design limit load is controlled by the 9g Shuttle crash load factor. Design for containment to peak cargo bay temperatures is a negligible mass penalty. Pressure relief venting to the cargo bay interior is acceptable. No caution and warning (C&W) signals or control from the orbiter is required.

The N_2 for ACS has a peak charge pressure of 290 N/cm^2 (200 psia) and is also within the pressure shell of the N_2H_4 tanks. The tanks contain 109 kg (240 pounds) of N_2H_4. The tanks will be designed for containment of N_2 and N_2H_4 at peak cargo bay temperatures. Backup N_2 pressure relief vent to the cargo bay will be used for added safety. No propellant dump for this quantity of N_2H_4 is required. No C&W or command lines to or from the Orbiter are required.

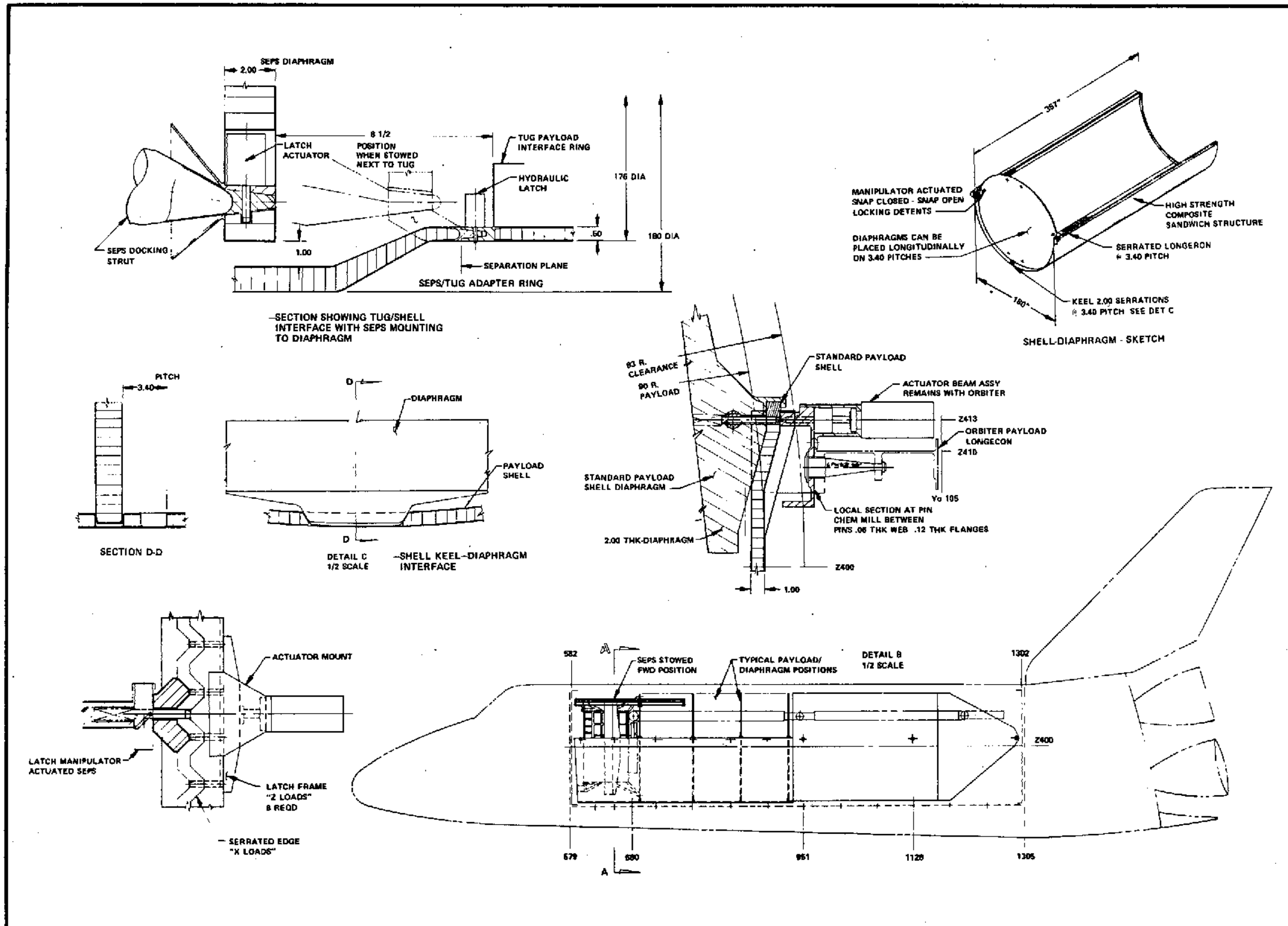


Figure 1-14. SEPS AND INITIAL PAYLOAD GROUP IN THE ORBITER CARGO BAY

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Because of the space thermal requirement both propellant tanks are insulated. No condition that has not destroyed the Orbiter will cause monopropellant decomposition of the N_2H_4 in SEPS.

SEPS, like most long-life spacecraft, uses Nickel-Cadmium batteries which are sealed. The batteries will be designed for containment. No C&W or command lines to or from the Orbiter are required.

SEPS is designed to have no separation or deployment ordnance. All separation functions are controlled by reversable motors or with the aid of the manipulators. Orbiter may derive status information and command control for latchings.

1.7.3 IUS—Tug Avionics Support to SEPS

NSI believes the most desirable approach to avionics support for all payloads mounted on Tug is from Tug, since the support must be continued after separation from the Orbiter. During ascent, Orbiter must support Tug by provision of primary power and data links into the Tug.

The following requirements for avionics support of SEPS from Tug exist:

- During prelaunch after transport shell has been mated to Tug and after installation in Orbiter:
  - 150 watts power and 1,000 kbits/sec digital data during brief flight readiness status check periods. Thermal control power of about 200 watts could be required depending on temperature of Orbiter's N_2 purge gases
- During Orbiter ascent:
  - Nominally no support; 200 watts periodically if required for thermal control
- During Tug deployment parking orbits and ascent to SEPS initial parking orbit:
  - 200 watts primary power for thermal control
- SEPS initial startup and transfer of initial payload to SEPS payload mast:
  - 600 watts, 10,000 bits/sec digital TV data and telemetry. Uplink data rate 10 kbits/sec. This support requirement would last approximately 1 hour. 1000 watt power required. Total energy required 3 kw/hr.

This deployment and initial payload transfer sequence is shown schematically on Figure 1-15. All of the above requirements are within Tug proposed capability. As indicated on Figure 1-15, one of the SEPS phased array antennas is exposed and SEPS' own systems can supply the capability.

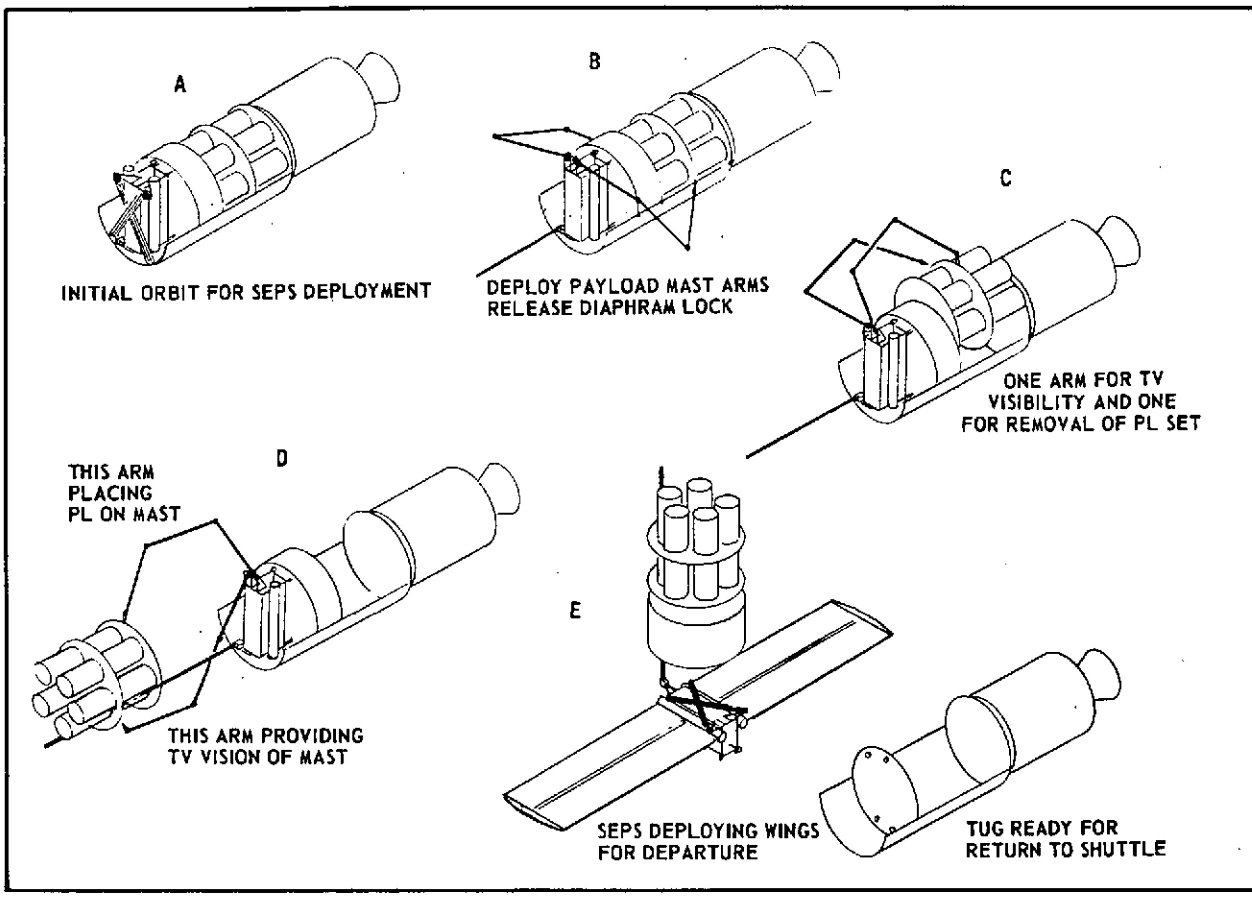

Figure 1-15. PAYLOAD TRANSFER INITIAL SEPS SORTIE

1.7.4 Tug—IUS Support to Payloads in Transport Shell

McDonnell Douglas and General Electric, teamed for the MSFC directed "IUS/Tug Payload Requirements Compatibility Study," reported in their midterm review the results of a payload design engineering committee analysis to determine nominal, maximum, and minimum values of Tug payload support requirements.

Consider that peak power and peak data rates are part of the final deployment functional checks and would be conducted on SEPS after SEPS had achieved the payload mission deployment conditions. SEPS, in this case, relieves Tug of ever having to meet the peak power and data rate requirements indicated by the committee analysis.

In further analysis the committee changed their approach to checkout test while still onboard a transport vehicle. Only payload status checks will be conducted until the payload spacecraft are deployed. All spacecraft payload demands indicated are therefore reduced to data rate levels of ≈1 kbit/sec and power levels to 200 or less watts. SEPS data rate capabilities are in the megabit range so this poses no problems for SEPS.

1.8 PROGRAM SUPPORT AND COST ESTIMATES

1.8.1 Program Support

SEPS is relatively simple. It is nearly all electrical. It has compact dimensions for transport and storage. Very modest buildings and checkout equipment will support its few launch preparation and refurbishment activities. The largest cost in SEPS operations is for mission planning and flight control personnel. These personnel must know SEPS configuration, functions, subsystems, and components in detail. The personnel that support the launch preparation functions, the one or two refurbishments, and the sustaining engineers must know the system intimately.

Reference to Figure 1-8, the System Operational Profile shows that in 11 years there are only eight planetary and three earth orbital launches to accomplish the reference mission model. There is only one SEPS refurbishment for relaunch. There are only 30 earth orbital sorties by SEPS over the 11-year period. Recall the SEPS autonomous cruise and autonomous terminal approach phase of the rendezvous (when desired) capability so that a sortie, typically 90 days or less total time, has only four periods of peak activity where the mission planning and flight control crews are fully utilized. These periods of peak activity are associated with the following functions:

- Detail planning of the next sortie in conjunction with the payload sponsors and developers and Shuttle flight planners.
- Systematic retrieval of the payloads to be returned to earth by Tug and orbiter, and initiation of the cruise phase down to the Tug rendezvous orbit

- Rendezvous with Tug, delivery of down payloads, acceptance of up payloads, and initiation of the ascent cruise phase to deploy up payloads at their mission conditions
- Deployment of payloads at their mission station and performance of servicing functions for any other payloads requiring that function.

Readers interested and experienced in mission planning and flight control recognize those four functions in the past space experience as time consuming and demanding of a large investment in man-hours. For this SEPS group, however the longest involvement of any intense activity is with the payload sponsors in the detail mission planning. Other functions require two to three days' full utilization of a 16-man team around some key flight operation. A small investment in time and people (in spite of past experience) can accomplish in the SEPS program the four functions described on the preceding page, because:

- 13.2 million dollars is allocated for initial software (onboard \$4.5 M) and flight control center (\$8.7 M) software to automate the mission planning and flight control
- The group does only the SEPS specific detail planning. Two other principal groups providing controlling event sequences and system function timelines to which SEPS must perform. The advance planning input comes from the Shuttle/STS Utilization and Master Scheduling Center. The detailed specific mission timeline event sequence for activities influencing Shuttle is established by the Shuttle Operations Center.

In view of the above factors, NSI believes that a small 45-man team, organized as shown on Figure 1-16, can accomplish the complete program support. Volume IV of this series, Design Reference Mission and Program Requirements, discusses the subject in some detail. Reference to Volume III will provide a fuller understanding of the complete sortie and mission cycle for SEPS.

SEPS transportation due to its small packaged size (3 m x 3 m x 5 m) and light unfueled packaged mass (1814 kg) is convenient and inexpensive. The total supporting equipment and facilities investment is \$8.8 million, \$5.3 million of which are allocated to computers and peripheral equipment. Computers are under-utilized except for the previously defined periods of peak activity and could be utilized by the SEPS operations center (SEPSOC) host institution for its other functions.

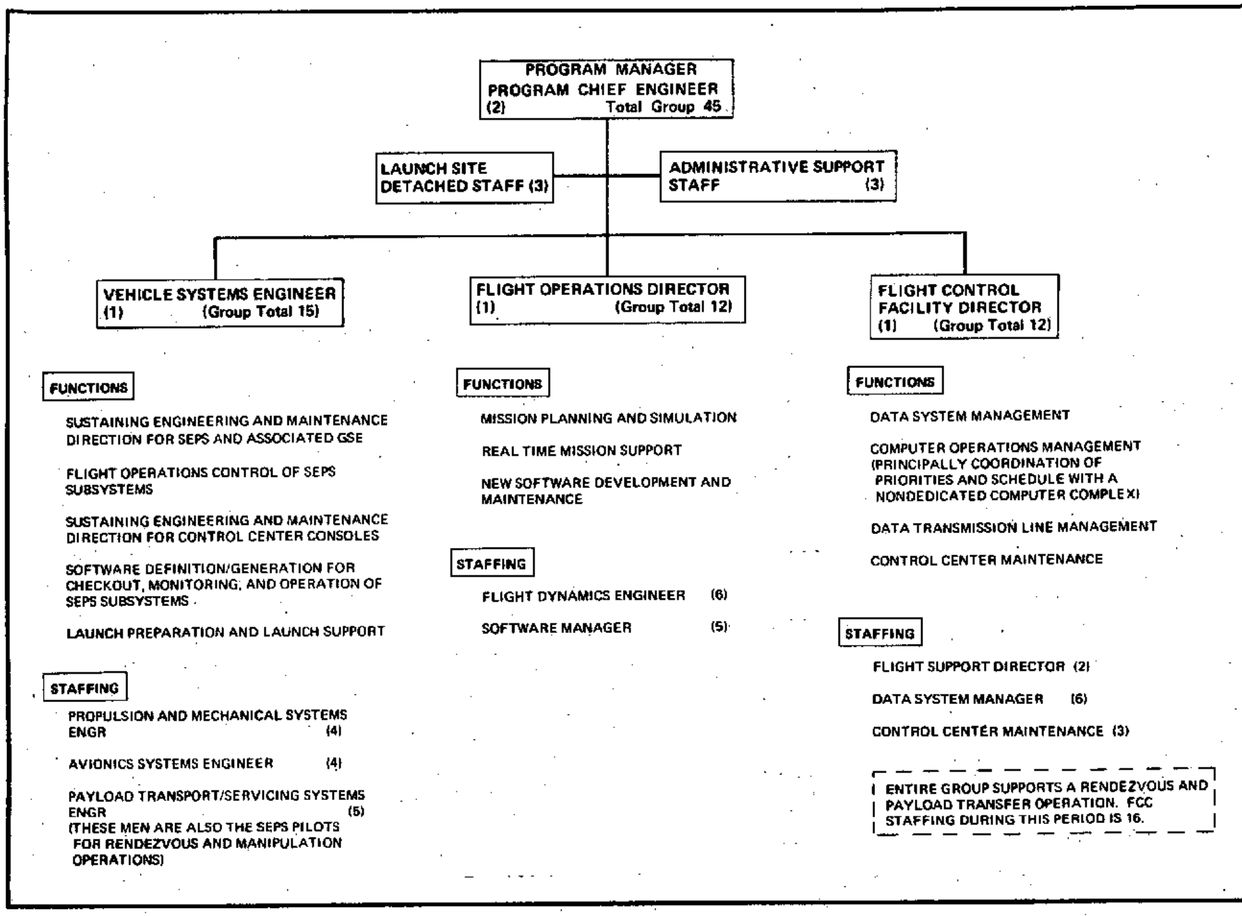

Figure 1-16. SEPS PROGRAM SUPPORT ORGANIZATION

Because of the above factors, NSI believes that SEPSOC facility and equipment cost factors should not control the location of SEPSOC. To accomplish the program cost savings indicated by the 45-man total program support team, the SEPSOC must be located at the center that is given the total program responsibility for SEPS.

1.8.2 Program Cost Summary

The cost estimation assumptions used in the analysis are as follows:

There will be a single SEPS DDT&E and production program managed by one organization. The basic core vehicle will be capable of accomplishing either the earth orbital functions or the deep space mission when certain components and sensors are added. This will, on occasion, result in SEPS implementing missions which do not require its full capability in solar array power or thrusters. NSI strongly believes it is false economy to have tailored, reduced capability vehicles just to save a few hardware production dollars on a specific production vehicle. Therefore, the single DDT&E program will phase into production at the most economical rate for the total inventory. Each SEPS, after production, will undergo a rigorous flight readiness check as a part of the final acceptance testing. Then it will be stored in a hermetically sealed, inert gas filled container with its status check and power supply hard lines used in ascent flight carried through the container walls to a test umbilical. As each SEPS is completed, accepted and installed in its storage container it goes to the launch site for immediate launch or to the SEPSOC for inventory storage.

When production of inventory and refurbishment spares are complete, the DDT&E/production contract is terminated. There is no sustaining engineering support team at any contractor or subsystem supplier's plant included in these cost estimates after production is complete. This does not preclude NASA from electing to have SEPSOC operated by a contractor and the DDT&E contractor may be the successful bidder for the SEPSOC support.

It is management wise and technically feasible that the 45-man program support team at the SEPSOC make any modifications or system changes found later in the program to be desirable.

Other assumptions are:

- Production is continuous for 11 vehicles. The first vehicle is delivered 30 months after authority to proceed (ATP).
- All $ are 1974 $.
- There are four planetary missions, each flown with a backup spacecraft requiring a total of eight planetary SEPS. Only two EO SEPS are required. One production spare is planned and the integrated system test article is refurbished at the end of production to provide a second spare.
- Two refurbishments are included in the cost estimates which would extend the SEPS capability beyond the 1991 operational time ground rules for this cost effectiveness study.
- No costs are included for mission special planetary spacecraft sensors.
- The center given responsibility for the science package and mission operation will assume flight control of SEPS and the science package

at some time after cruise mode is established for the initial planetary trajectory. Only periodic advice or consultation from SEPS vehicle systems specialists will be provided on request of the planetary control groups after cruise mode is established.

Table 1-6 presents the SEPS total program costs including planetary vehicle core development costs and the launch support operation for eight planetary vehicles.

Table 1-6. SEPS SUMMARY COSTS

STAGE DDT&E		97.5
EO Functions (Transport Mast & Manipulators)	(8.3)	
Basic Stage	(89.2)	
STS GPME DDT&E		2.5
PL Shell & Diaphragms		
FLIGHT ARTICLE PRODUCTION		145.9
8 Planetary Vehicles	(97.6)	
3 EO Stages	(39.6)	
STS GPME	(1.5)	
Stage Refurbishment and Maintenance	(7.2)	
SEPS OPERATIONS CENTER INITIAL COSTS		17.9
Facility and Equipment	(8.8)	
Initial Software Package	(8.7)	
Initial SEPSOC Spares	(0.4)	
SEPS SYSTEMS OPERATIONS		26.2
Personnel (45 men 11 years)	(23.7)	
Computer Support	(2.1)	
Flight Article Consumables	(0.4)	
TOTAL PROGRAM COSTS		290.0

Table 1-7 is the DDT&E cost broken down by major subsystem and functional area of the program.

Table 1-7. SEPS DEVELOPMENT COSTS

	TOTAL DDT&E	CORE VEHICLE	PLANETARY PECULIAR	EO PECULIAR
STRUCTURES & THERMAL CONTROL	$ 4.8	$ 4.8		
PROPULSION	9.1	9.1		
POWER DISTRIBUTION	1.0	1.0		
SOLAR ARRAY	7.8	7.8		
DATA MANAGEMENT	3.4	3.4		
COMMUNICATION	2.2	1.4	$ 0.5	$ 0.3
NAVIGATION & GUIDANCE/ATTITUDE CONTROL	9.2	6.0	2.2	1.0
INTEGRATION & TEST CHECKOUT	6.7	6.7		
TEST HARDWARE	21.3	19.8	1.1	0.4
STAGE GSE	5.0	4.0	0.2	0.8
SOFTWARE	4.5	4.5		
LOGISTICS	0.5	0.1		0.4
S.E.&I.	6.8	6.8		
PROGRAM MANAGEMENT	6.9	6.9		
BASIC SEPS	89.2	82.3	4.0	2.9
Δ FOR EARTH ORBITAL FUNCTIONS OR (PAYLOAD MAST & MANIPULATOR)	8.3			8.3
TOTAL	97.5			

Figure 1-17 shows the prime contractor's total manloading versus time for DDT&E and production for the first 36 months of the contract. Beginning at 30 months into the contract, SEPS are delivered at the rate of three per year until delivery of the 12th SEPS (the refurbished test article). Total DDT&E plus production duration is approximately 6 years.

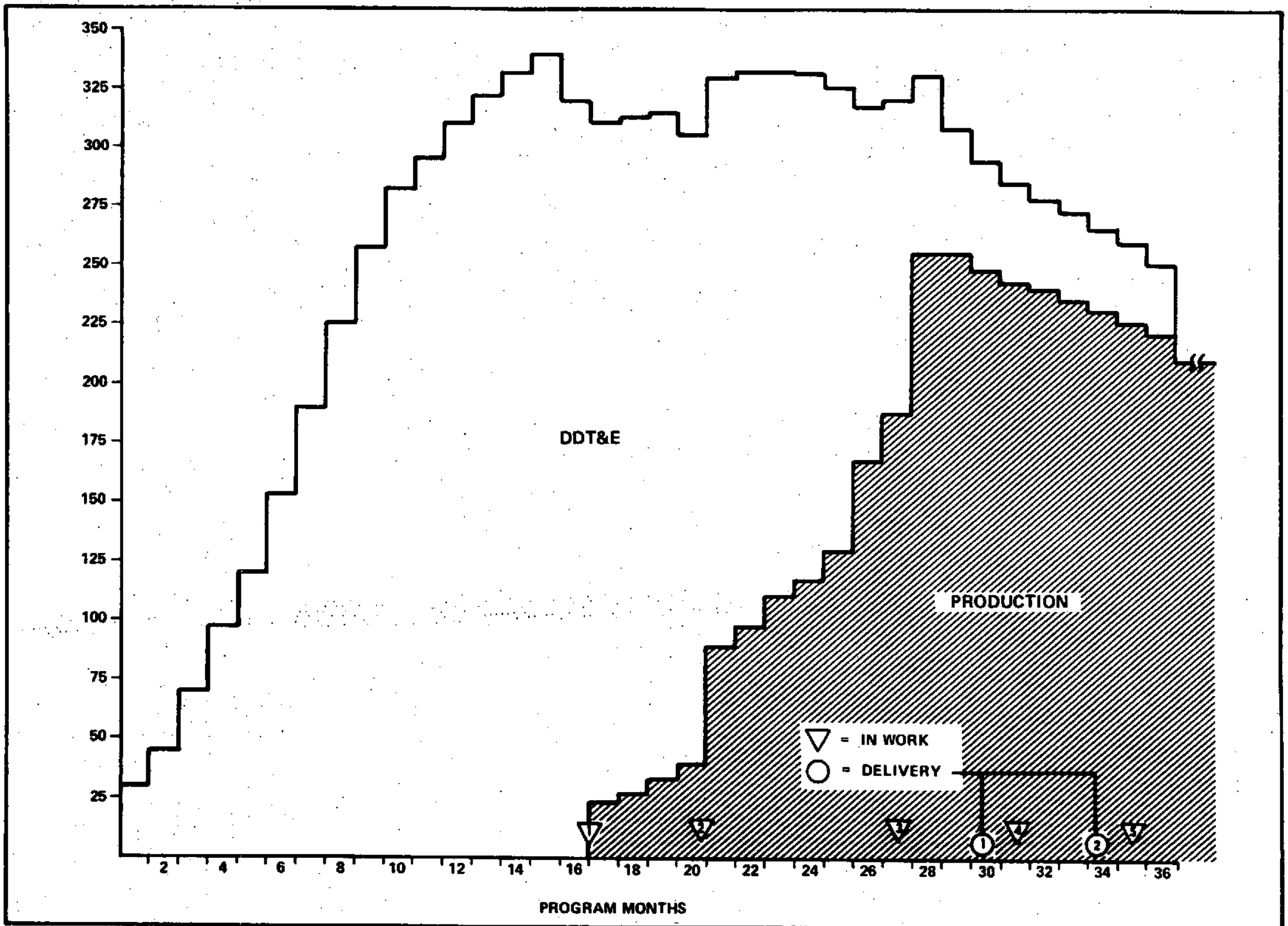

Figure 1-17. TOTAL SEPS PROGRAM MANLOAD, MONTHS 1 THROUGH 36 ONLY

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II TRAJECTORY & TRAFFIC MODEL ANALYSIS

Section II

TRAJECTORY AND TRAFFIC MODEL ANALYSIS

The prinicpal objectives of the traffic model analysis were:

- o To provide a data base for SEPS transport cost effectiveness by establishing the minimum number of Shuttle flights that would accomplish the mission model for an STS without Earth Orbital SEPS and for an STS with Earth Orbital SEPS.
- o To determine the sensitivity of the Shuttle flights required by various operational ground rules such as method of payload packaging or specification of an arbitrary limit on the number of payloads on Tug in a single flight.
- o To establish the sensitivity of the Shuttle flights to Tug performance and length.
- o To support trade studies on SEPS power level and specific impulse.
- o To identify the number of individual payloads and the mix of different types to be delivered, retrieved, and serviced on each Tug-SEPS sortie so that GPME and E O SEPS equipment functional requirements could be identified.

In order to meet these objectives, NSI formulated an analysis technique that identified the ordered series of cargo manifests (list of individual payloads assigned to a specific flight) that would result in the minimum number of Shuttle-Tug kick stages and SEPS sorties to accomplish the reference mission model. NSI refers to this ordered series of flights as a traffic model or System Operational Profile.

This analysis effort required assignment of payloads to each flight within the restrictions of the Shuttle or Tug payload capability and the Shuttle cargo bay size limits. The number of SEPS vehicles required as well as the flight schedules to support the mission model are dependent on the SEPS sortie trip times as soon as flight frequencies require full utilization of SEPS.

The determination of sortie trip times evolves generation of SEPS low-thrust trajectories and changeover orbit characteristics. Two computer programs were used for this work. Payload flight assignments, SEPS trip time calculations, and flight scheduling were done by the WHATIF program. This program was jointly developed by MSFC and NSI. It is a basic program used by MSFC for the generation of STS traffic models, cost effectiveness analysis of STS, and trade studies to define Tug characteristics. SEPS trajectories and changeover orbits were generated by the MOLTOP program. Major modification of the WHATIF program was necessary to provide SEPS performance and scheduling capability.[1]

Four mission roles were initially envisioned for SEPS where SEPS could effectively augment the performance of Shuttle and Tug. For reasons discussed in the following sections, the practical SEPS-Tug sorties become composites that include the two major earth orbital roles. Only the planetary mission role remains distinctly different.

Traffic model analyses with and without SEPS were done for a number of Tug and SEPS configurations, principally Tugs shorter than the 30-foot baseline and SEPS with higher power and specific impulse than the baseline 25 kw SEPS. Results of these analyses show the value of SEPS and the effect of configurations other than the baseline on the Space Transportation System cost (expressed as number of flights required by the mission model).

A similar analysis assessed the impact on STS cost of the following SEPS operational modes and constraints:

- In-space refueling of SEPS
- Elliptical versus circular changeover orbits
- Delivery of payloads at intermediate orbital altitudes by Tug on the way to changeover orbit
- SEPS maximum trip time limits
- Payload packaging constraints (end-to-end, side-by-side, three-dimensional)
- Limits on number of payloads per flight.

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[1] *A description of this program as modified for this study is contained in Volume IV of this report.*

The traffic models also provided data for construction of system operational profiles showing yearly activity of the onorbit SEPS, Shuttle, and Tugs required by the mission model. SEPS launches, retrievals, and refuelings are included in the operational profile along with sortie durations and Shuttle launch dates to support SEPS sorties.

Based on traffic model analysis, a representative SEPS sortie was synthesized for identification of operations support requirements. A reference trajectory profile was then developed for this sortie showing event times (timeline) on the Shuttle, Tug, and SEPS trajectories. This design reference trajectory is discussed in Section III of this volume and in Volume III.

2.1 REFERENCE MISSION MODEL

The reference mission model (supplied by NASA) used throughout this study to measure the transport effectiveness of SEPS as part of the STS was the NASA October 1973 "Best Mix" mission model.[2] This model was developed by NASA by selecting from alternate payload concepts those payload configurations which produced the least total cost for payload development and procurement plus transportation cost when the STS consisted of Shuttle and Tug without SEPS. This payload cost versus transportation cost trade resulted in a "best mix" of current reusable, current expendable low-cost expendable, and intermediate payload designs which was optimized for Shuttle-Tug capability and as such is biased against showing the true SEPS potential. By the ground rules in this study, SEPS cost effectiveness considers only STS operational costs. No credit is taken for potentially lower payload cost. One example of the way the "best mix" analysis affected definition of payloads from geosynchronous orbit is that it is difficult and expensive for Tug to complete round trip missions with low cost reusable payloads, usually requiring separate delivery and separate retrieval flights thus requiring two Shuttle launchs. SEPS can deliver/retrieve these payloads with just one Shuttle-Tug launch and thereby save a Shuttle flight. Since these payloads have high transportation costs, they were all but eliminated from the reference mission model in NASA's "best

[2] *MSFC TMX-64751, Rev. 2, "The October 1973 Space Shuttle Traffic Model," January 1974.*

mix" optimization. The use of this reference model and the limitation of cost effectiveness quantitative numbers to STS operational cost savings only, does not present a true picture of SEPS cost effectiveness nor of its real value to NASA's overall program plan for the 1981-1991 years.

The mission model[2] and Space Shuttle Payload Description data books[3] specify launch environment, communication, power requirements, and deployment pointing accuracies for the payloads in addition to launch schedule, size and weight, and orbital parameters. Information is also supplied about the compatibility of a payload with other payloads for packaging on the same flight. Sequences such as retrieving a payload, refurbishing a payload, and launching the same payload for a second mission cycle are identified.

Data pertinent to the traffic model analysis are shown in Tables 2-1 and 2-2. Table 2-1 lists the NASA payload designation, payload dimensions, up and down weights, and orbital parameters (delta velocity above Shuttle parking orbit in the case of escape payloads). Payload compatibility restrictions and special delivery requirements are noted where they apply. Retrieval payloads are identified by an R following the payload designation. Payload ID numbers were serially assigned by the WHATIF program for convenient identification of the payloads. Table 2-2 is the launch schedule for the payloads in Table 2-1 during the 11 years analyzed in this study, 1981 through 1991.

There are 864 missions in the 11 years of the mission model. Total number of missions in each year are shown at the end of Table 2-2 (note that several payloads included in Table 2-1 are not actually scheduled in the 1981-1991 period). Payloads planned for launch on expendable launch vehicles in 1981 and 1982 are not included in this mission model except for two plantary missions. Department of Defense payloads are excluded by the study guidelines and therefore are not considered in this study.

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[3] *MSFC, "Summarized Payload Descriptions - Automated Payloads," and "Payload Descriptions, Vol. 1 - Automated Payloads," July 1974.*

Table 2-1. PAYLOAD CHARACTERISTICS

ID	NASA NO	DIAM-FT	LGTH-FT	UP WT-LBS	DN WT-LBS	PL APO-NM	PL PER-NM (Δv-FPS)	PL INCL-DEG	CANNOT BE LAUNCHED WITH	
1	AST-1A	2.6	12.2	650.	0.	297.	297.	28.5		CR EXPLORER - LOW EARTH ORBIT
2	AST-1A R	2.6	12.2	0.	641.	297.	297.	28.5		CR EXPLORER - LOW EARTH ORBIT
3	AST-1B	2.6	12.2	650.	0.	19323.	19323.	28.5		CR EXPLORER - SYNC.
4	AST-1B R	2.6	12.2	0.	641.	19323.	19323.	28.5		CR EXPLORER - SYNC.
5	AST-3	11.6	13.1	4282.	0.	270.	270.	28.5		LCR SOLAR MAX SATELLITE
6	AST-3 R	11.6	13.1	0.	4146.	270.	270.	28.5		LCR SOLAR MAX SATELLITE
7	AST-4	9.0	18.1	6064.	0.	250.	250.	28.5		CR HEAO C
8	AST-4 R	9.0	18.1	0.	6064.	250.	250.	28.5		CR HEAO C
9	AST-5	14.0	17.5	17434.	0.	200.	200.	28.5		CR HEAO D AND E
10	AST-5 R	14.0	17.5	0.	17214.	200.	200.	28.5		CR HEAO D AND E
11	AST-5V	14.0	5.0	3500.	3500.	200.	200.	28.5	ANOTHER NO. 11 PLD SHUTTLE LAUNCH ONLY	CR HEAO D AND E REVISIT
12	AST-6	12.0	36.3	20161.	0.	340.	340.	28.5		CR LARGE SPACE TELESCOPE
13	AST-6 R	12.0	36.3	0.	20087.	340.	340.	28.5		CR LARGE SPACE TELESCOPE
14	AST-6V	14.0	5.0	3500.	3500.	340.	340.	28.5	ANOTHER NO.13 PLD SHUTTLE LAUNCH ONLY	CR LARGE SPACE TELESCOPE REVISIT
15	AST-7	15.0	58.5	27034.	0.	190.	190.	28.5		CR LARGE SOLAR OBSERVATORY
16	AST-7 R	15.0	58.5	0.	26912.	190.	190.	28.5		CR LARGE SOLAR OBSERVATORY
17	AST-7V	14.0	5.0	3500.	3500.	190.	190.	28.5	ANOTHER NO. 17 PLD SHUTTLE LAUNCH ONLY	
18	AST-8	10.0	25.0	2786.	0.	38646.	38646.	28.5		CR LARGE RADIO OBSERVATORY
19	AST-8 R	10.0	25.0	0.	2640.	38646.	38646.	28.5		CR LARGE RADIO OBSERVATORY
20	AST-8V	14.0	5.0	3000.	3000.	38646.	38646.	28.5	ANOTHER NO. 20 PLD SHUTTLE LAUNCH ONLY	CR LARGE RADIO OBSERVATORY REVISIT
21	AST-9A	14.0	17.5	17434.	0.	270.	270.	28.5		CR FOCUSING X RAY TELESCOPE (MISSI
22	AST-9A R	14.0	17.5	0.	17214.	270.	270.	28.5		CR FOCUSING X RAY TELESCOPE (MISSI
23	AST-9AV	14.0	5.0	3500.	3500.	270.	270.	28.5	ANOTHER NO. 23 PLD SHUTTLE LAUNCH ONLY	CR FOCUSING X RAY-REVISIT
24	AST-9B	14.0	53.0	24136.	0.	270.	270.	28.5		CR FOCUSING X-RAY TELESCOPE (MISSIO

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Table 2-1. PAYLOAD CHARACTERISTICS (Continued)

ID	NASA NO	DIAM-FT	LGTH-FT	UP WT-LBS	DN WT-LBS	PL APO-NM	PL PER-NM (Δv-FPS)	PL INCL-DEG	CANNOT BE LAUNCHED WITH	
25	AST-9B R	14.0	53.0	0.	23872.	270.	270.	28.5		CR FOCUSING X-RAY TELESCOPE (MISSIO
26	AST-9BV	14.0	5.0	3500.	3500.	270.	270.	28.5	ANOTHER NO. 26 PLD SHUTTLE LAUNCH ONLY	CR FOCUSING X-RAY -REVISIT
27	PHY-1A	4.0	13.3	1588.	0.	1900.	140.	90.0		CR EXPLORER - UPPER ATMOSPHERE
28	PHY-1A R	4.0	13.3	0.	1046.	1900.	140.	90.0		CR EXPLORER - UPPER ATMOSPHERE
29	PHY-1B	5.0	12.8	853.	0.	20000.	1000.	28.5		CR EXPLORER - MEDIUM ALTITUDE
30	PHY-1B R	5.0	12.8	0.	848.	20000.	1000.	28.5		CR EXPLORER - MEDIUM ALTITUDE
31	PHY-1C	6.1	10.4	1226.	0.		(14373.)	.0		LCE EXPLORER - HIGH ALTITUDE
32	PHY-2A	12.5	13.6	2514.	0.	500.	500.	90.0		LCE GRAVITY/RELATIVITY SAT. - MISSIO
33	PHY-2B	9.3	12.0	1373.	0.		(22218.)	.0		LCE GRAVITY/RELATIVITY SAT. - MISSIO
34	PHY-3A	7.0	15.8	3846.	0.	6900.	6900.	55.0		CR ENVIRONMENTAL PERTURBATION SAT 1
35	PHY-3A R	7.0	15.8	0.	3588.	6900.	6900.	55.0		CR ENVIRONMENTAL PERTURBATION SAT 1
36	PHY-3B	10.0	17.3	9845.	0.	6900.	6900.	55.0		CR ENVIRONMENT PERTUBATION SAT. - M
37	PHY-3B R	10.0	17.3	0.	9290.	6900.	6900.	55.0		CR ENVIRONMENT PERTUBATION SAT. - M
38	PHY-4	10.0	10.5	635.	0.		(28946.)	.0		CE HELIOCENTRIC AND INTERSTELLAR S7
39	PHY-5	14.0	43.5	46758.	0.	200.	200.	28.5		CR COSMIC RAY LAB
40	PHY-5 R	14.0	43.5	0.	31999.	200.	200.	28.5		CR COSMIC RAY LAB
41	PHY-5V	14.0	5.0	3500.	3500.	200.	200.	28.5	ANOTHER NO. 41 PLD SHUTTLE LAUNCH ONLY	CR COSMIC RAY LAB REVISIT
42	PL-7	14.7	23.5	10490.	0.		(12157.)	.0		LCE MARS SURFACE SAMPLE RETURN
43	PL-8	14.7	51.5	16419.	0.		(12560.)	.0		LCE MARS SATELLITE SAMPLE RETURN (PH
44	PL-10	8.4	11.5	2772.	0.		(13869.)	.0		LCE INNERPLANETARY FOLLOW-ON
45	PL-11	14.7	19.4	13485.	0.		(12258.)	.0		LCE VENUS RADAR MAPPER
46	PL-12	14.7	17.3	20617.	0.		(12661.)	.0		LCE VENUS BUOYANT STATION
47	PL-13	14.7	34.9	8498.	0.		(12000.)	.0		LCE MERCURY ORBITER
48	PL-14	14.7	25.0	6129.	0.		(12560.)	.0		LCE VENUS LARGE LANDER

Table 2-1. PAYLOAD CHARACTERISTICS (Continued)

ID	NASA NO	DIAM-FT	LGTH-FT	UP WT-LBS	DN WT-LBS	PL APO-NM	PL PER-NM	PL INCL-DEG	CANNOT BE LAUNCHED WITH	
49	PL-17	10.0	10.5	1146.	0.		(27741.)	.0		CE PIONEER SATURN PROBE
50	PL-18	10.0	10.5	1146.	0.		(27641.)	.0		CE PIONEER SATURN / URANUS FLYBY (U
51	PL-19	14.7	25.0	6888.	0.		(15279.)	.0		LCE MARINER JUPITER ORBITER
52	PL-20	10.0	10.5	1169.	0.		(25940.)	.0		CE PIONEER JUPITER PROBE
53	PL-21	14.7	39.0	4988.	0.		(24528.)	.0		LCE MARINER SATURN ORBITER
54	PL-22	15.0	25.0	2137.	0.		(30847.)	.0		CE MARINER URANUS PROBE/NEPTUNE FLY
55	PL-23	14.7	48.3	35795.	0.		(11827.)	.0		CE JUPITER SATELLITE ORBITER/LANDER
56	PL-26	14.7	19.9	4978.	0.		(16448.)	.0		LCE COMET ENCKE RENDEZVOUS
57	PL-27	12.2	13.5	2074.	0.		(12969.)	.0		LCE COMET HALLEY FLYBY
58	PL-28	14.7	20.8	4583.	0.		(13416.)	.0		LCE ASTEROID RENDEZVOUS (VESTA)
59	LUN-2	7.8	11.2	2475.	0.		(11033.)	.0		LCE AUTOMATED LUNAR ORBITER
60	LUN-3	10.0	24.0	8700.	0.		(11033.)	.0		CE AUTOMATED LUNAR ROVER
61	LUN-4	14.7	19.1	4633.	0.		(11033.)	.0		LCE HALO SAT.
62	LUN-5	10.0	24.0	11500.	0.		(11033.)	.0		CE LUNAR SAMPLE RETURN
63	LS-1	2.2	13.0	683.	0.	300.	300.	28.5	PLD NO. 64	LCR LIFE SCIENCES MODULE
64	LS-1 R	2.2	13.0	0.	656.	300.	300.	28.5	PLD NO. 63	LCR LIFE SCIENCES MODULE
65	EO-3A	10.2	36.0	8630.	0.	300.	300.	99.0		LCR EARTH OBSERVATION SATELLITE - MI
66	EO-3A R	10.2	36.0	0.	6213.	300.	300.	99.0		LCR EARTH OBSERVATION SATELLITE - MI
67	EO-3AV	14.0	5.0	3500.	3500.	300.	300.	99.0	ANOTHER NO. 67 PLD SHUTTLE LAUNCH ONLY	LCR EARTH OBSERVATION SATELLITE - MI
68	EO-3B	10.2	36.0	8630.	0.	300.	300.	99.0		LCR EARTH OBSERVATION SATELLITE - MI
69	EO-3B R	10.2	36.0	0.	6213.	300.	300.	99.0		LCR EARTH OBSERVATION SATELLITE - MI
70	EO-3BV	14.0	5.0	3500.	3500.	300.	300.	99.0	ANOTHER NO. 70 PLD. SHUTTLE LAUNCH ONLY	LCR EARTH OBSERVATION SATELLITE - MI
71	EO-3C	10.2	36.0	8630.	0.	300.	300.	99.0		LCR EARTH OBSERVATION SATELLITE - MI
72	EO-3C R	10.2	36.0	0.	6213.	300.	300.	99.0		LCR EARTH OBSERVATION SATELLITE - MI

Table 2-1. PAYLOAD CHARACTERISTICS (Continued)

ID	NASA NO		DIAM-FT	LGTH-FT	UP WT-LBS	DN WT-LBS	PL APO-NM	PL PER-NM	PL INCL-DEG	CANNOT BE LAUNCHED WITH	
73	EO-3CV		14.0	5.0	3500.	3500.	300.	300.	99.0	ANOTHER NO. 73 PLD SHUTTLE LAUNCH ONLY	LCR EARTH OBSERVATION S ATELLITE - MI
74	EO-3D		10.2	36.0	8630.	0.	300.	300.	28.5		LCR EARTH OBSERVATION S ATELLITE - MI
75	EO-3D	R	10.2	36.0	0.	6213.	300.	300.	28.5		LCR EARTH OBSERVATION S ATELLITE - MI
76	EO-4A		7.4	11.0	3085.	0.	19323.	19323.	.0		CR SEOS - R AND D
77	EO-4A	R	7.4	11.0	0.	2996.	19323.	19323.	.0		CR SEOS - R AND D
78	EO-4B		7.4	11.0	3085.	0.	19323.	19323.	.0		CR SEOS - OPERATIONAL
79	EO-4B	R	7.4	11.0	0.	2996.	19323.	19323.	.0		CR SEOS - OPERATIONAL
80	EO-5A		4.7	9.7	676.	0.	19323.	19323.	.0		LCE SPECIAL PURPOSE SAT ELLITE - SYNC
81	EO-5A	R	4.8	9.7	0.	670.	19323.	19323.	.0		LCR SPECIAL PURPOSE SAT ELLITE - SYNC
82	EO-5B		4.7	9.7	676.	0.	3000.	300.	90.0		LCE SPECIAL PURPOSE SAT ELLITE - SYNC
83	EO-5B	R	4.8	9.7	0.	670.	3000.	300.	90.0		LCR SPECIAL PURPOSE SAT ELLITE - SYNC
84	EO-5C		4.7	9.7	676.	0.	280.	280.	90.0		LCE SPECIAL PURPOSE SAT ELLITE - SYNC
85	EO-5C	R	4.8	9.7	0.	670.	280.	280.	90.0		LCR SPECIAL PURPOSE SAT ELLITE - SYNC
86	EO-5D		4.7	9.7	676.	0.	400.	400.	90.0		LCE SPECIAL PURPOSE SAT ELLITE - SYNC
87	EO-5D	R	4.8	9.7	0.	670.	400.	400.	90.0		LCR SPECIAL PURPOSE SAT ELLITE - SYNC
88	EO-5E		4.7	9.7	676.	0.	19323.	19323.	.0		LCE SPECIAL PURPOSE SAT ELLITE - POLA
89	EO-5E	R	4.8	9.7	0.	670.	19323.	19323.	.0		LCR SPECIAL PURPOSE SAT ELLITE - POLA
90	EO-6		8.0	15.3	1717.	0.	790.	790.	102.0		CR TIROS N-P
91	EO-6	R	8.0	15.3	0.	1615.	790.	790.	102.0		CR TIROS N-P
92	EO-7		7.2	10.9	1077.	0.	19323.	19323.	.0		LCE SYNCHRONOUS METEORO LOGICAL SAT.
93	EOP-3		14.7	18.3	3030.	0.	325.	325.	90.0		LCE SEASAT-B
94	EOP-4		10.9	12.8	3792.	0.	16200.	16200.	90.0		LCE GEOPAUSE
95	EOP-5		14.7	30.2	10236.	0.	108.	108.	90.0		LCE GRAVITY GRADIOMETER
96	EOP-6A		1.6	1.6	225.	0.	350.	350.	28.5		CE MINI - LAGEOS - 28. 5

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Table 2-1. PAYLOAD CHARACTERISTICS (Continued)

ID	NASA NO	DIAM-FT	LGTH-FT	UP WT-LBS	DN WT-LBS	PL APO-NM	PL PER-NM	PL INCL-DEG	CANNOT BE LAUNCHED WITH	
97	EOP-6B	1.6	1.6	225.	0.	350.	350.	55.0		CE MINI - LAGEOS - 55
98	EOP-6C	1.6	1.6	225.	0.	350.	350.	90.0		CE MINI - LAGEOS - 90
99	EOP-8	6.2	10.4	1209.	0.	216.	216.	90.0		LCR VECTOR MAGNETOMETER SATELLITE
100	EOP-8 R	6.2	10.4	0.	1080.	216.	216.	90.0		LCR VECTOR MAGNETOMETER SATELLITE
101	EOP-9	5.8	10.2	915.	0.	1080.	540.	28.0		LCR MAGNETIC MONITOR SA TELLITE
102	EOP-9 R	5.8	10.2	0.	815.	1080.	540.	28.0		LCR MAGNETIC MONITOR SA TELLITE
103	NN/D-2C	6.3	17.9	974.	0.	19323.	19323.	.0		CR TRACKING AND DATA R ELAY SATELLIT
104	NN/D-2CR	6.3	17.9	0.	892.	19323.	19323.	.0		CR TRACKING AND DATA R ELAY SATELLIT
105	ST-1	14.0	35.5	10200.	0.	270.	270.	28.5		CR LONG DURATION EXPOS URE FACILITY
106	ST-1 R	14.0	35.5	0.	10200.	270.	270.	28.5		CR LONG DURATION EXPOS URE FACILITY
107	NN/D-1	8.3	12.2	4498.	0.	19323.	19323.	.0		CR INTELSAT
108	NN/D-1 R	8.3	12.2	0.	4347.	19323.	19323.	.0		CR INTELSAT
109	NN/D-2A	7.6	11.1	1057.	0.	19323.	19323.	.0		LCE U.S. DOMCOMSAT (MI SSION A)
110	NN/D-2B	8.3	12.2	4498.	0.	19323.	19323.	.0		CR U.S. DOMCOMSAT (MI SSION B)
111	NN/D-2BR	8.3	12.2	0.	4347.	19323.	19323.	.0		CR U.S. DOMCOMSAT (MI SSION B)
112	NN/D-3	8.2	11.4	2054.	0.	19323.	19323.	.0		LCR DISASTER WARNING SA TELLITE
113	NN/D-3 R	8.2	11.4	0.	2017.	19323.	19323.	.0		LCR DISASTER WARNING SA TELLITE
114	NN/D-4	10.3	12.5	1422.	0.	19323.	19323.	.0		LCE TRAFFIC MANAGEMENT
115	NN/D-4 R	10.5	12.6	0.	1423.	19323.	19323.	.0		LCR TRAFFIC MANAGEMENT
116	NN/D-5	5.8	12.2	982.	0.	19323.	19323.	.0		CR FOREIGN COMSAT
117	NN/D-5 R	5.8	12.2	0.	831.	19323.	19323.	.0		CR FOREIGN COMSAT
118	NN/D-6	11.6	13.1	3871.	0.	19323.	19323.	.0		LCE COMMUNCATIONS R AND D SATELLITE
119	NN/D-8	10.2	12.4	2025.	0.	920.	920.	103.0		LCR ENVIRONMENTAL MONIT ORING SATELLI
120	NN/D-8 R	10.2	12.4	0.	1900.	920.	920.	103.0		LCR ENVIRONMENTAL MONIT ORING SATELLI

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Table 2-1. PAYLOAD CHARACTERISTICS (Continued)

ID	NASA NO	DIAM-FT	LGTH-FT	UP WT-LBS	DN WT-LBS	PL APO-NM	PL PER-NM	PL INCL-DEG	CANNOT BE LAUNCHED WITH
121	NN/D-9	6.0	10.3	807.	0.	19323.	19323.	.0	CR FOREIGN SYNCHRONOUS METEOROGICAL
122	NN/D-9 R	6.0	10.3	0.	765.	19323.	19323.	.0	CR FOREIGN SYNCHRONOUS METEOROGICAL
123	NN/D-10	6.0	10.3	807.	0.	19323.	19323.	.0	CR GEOSYNCHROUS OPERATIONAL METEORO
124	NN/D-10R	6.0	10.3	0.	765.	19323.	19323.	.0	CR GEOSYNCHROUS OPERATIONAL METEORO
125	NN/D-11	10.2	36.0	8630.	0.	300.	300.	97.0	LCR EARTH RESOURCES SAT. - LEO
126	NN/D-11R	10.2	36.0	0.	6213.	300.	300.	97.0	LCR EARTH RESOURCES SAT. - LEO
127	NN/D-12	7.4	11.0	3085.	0.	19323.	19323.	.0	CR EARTH RESOURCES-SYNC
128	NN/D-12R	7.4	11.0	0.	2996.	19323.	19323.	.0	CR EARTH RESOURCES-SYNC
129	NN/D-13	7.4	11.0	3085.	0.	19323.	19323.	.0	CR FOREIGN SEOS
130	NN/D-13R	7.4	11.0	0.	2996.	19323.	19323.	.0	CR FOREIGN SEOS
131	NN/D-14	12.7	13.7	5062.	0.	200.	200.	98.0	LCR GLOBAL EARTH AND OCEAN MONITOR S
132	NN/D-14R	12.7	13.7	0.	4745.	200.	200.	98.0	LCR GLOBAL EARTH AND OCEAN MONITOR S
133	AST10A	14.0	50.0	31857.	30225.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P
134	AST10B	14.0	45.0	28526.	26894.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P
135	AST10C	14.0	30.0	30811.	29179.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P
136	AS10D73	14.0	47.0	27287.	25655.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P
137	AS10D79	14.0	47.0	27287.	25655.	120.	120.	90.0	SORTIE STELLAR ASTR., 7 DAY, P
138	AS10D33	14.0	54.0	40200.	30570.	162.	162.	28.5	SORTIE STELLAR ASTR., 30 DAY, P
139	AS10D39	14.0	54.0	40200.	30570.	120.	120.	90.0	SORTIE STELLAR ASTR., 30 DAY, P
140	AST10E	14.0	40.0	25460.	23828.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P
141	AST10F	14.0	40.0	55019.	31387.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P
142	AST10G	14.0	10.0	13005.	11373.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P
143	AST10H	14.0	52.0	41512.	32000.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P
144	AST10I	14.0	54.0	29168.	19538.	162.	162.	28.5	SORTIE STELLAR ASTR., 7 DAY, P

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Table 2-1. PAYLOAD CHARACTERISTICS (Continued)

ID	NASA NO	DIAM-FT	LGTH-FT	UP WT-LBS	DN WT-LBS	PL APO-NM	PL PER-NM	PL INCL-DEG	CANNOT BE LAUNCHED WITH	
145	AST10J	14.0	45.0	23519.	21887.	162.	162.	28.5		SORTIE STELLAR ASTR., 7 DAY, P
146	AST10K7	14.0	48.0	29637.	28005.	162.	162.	28.5		SORTIE STELLAR ASTR., 7 DAY, P
147	AST10K3	14.0	55.0	42702.	31190.	162.	162.	28.5		SORTIE STELLAR ASTR., 30 DAY, P
148	AST10L	14.0	57.0	41402.	31890.	162.	162.	28.5		SORTIE STELLAR ASTR., 7 DAY, P
149	AST10M	14.0	37.0	40146.	30634.	162.	162.	55.0		SORTIE STELLAR ASTR., 7 DAY, P
150	AST11B	14.0	50.0	24771.	23039.	210.	210.	28.5		SORTIE SOLAR PHYSICS, 7 DAY, P
151	AST11C7	14.0	40.0	30298.	28566.	210.	210.	28.5		SORTIE SOLAR PHYSICS, 7 DAY, P
152	AST11A	14.0	25.0	21055.	19323.	210.	210.	28.5		SORTIE SOLAR PHYSICS, 7 DAY, P
153	AST11C3	14.0	47.0	41363.	31751.	210.	210.	28.5		SORTIE SOLAR PHYSICS, 30 DAY, P
154	AST11D7	14.0	25.0	23871.	22139.	210.	210.	28.5		SORTIE SOLAR PHYSICS, 7 DAY, P
155	AST11D3	14.0	32.0	36784.	27054.	210.	210.	28.5		SORTIE SOLAR PHYSICS, 30 DAY, P
156	AST11E7	14.0	45.0	31004.	29272.	210.	210.	28.5		SORTIE SOLAR PHYSICS, 7 DAY, P
157	AST11E3	14.0	52.0	41612.	32000.	210.	210.	28.5		SO-T*E SOLA- PHYS*CS, 30 DAY, P
158	PHY6A/B	14.0	55.0	31227.	28242.	120.	120.	28.5		SORTIE HIGH ENERGY, 7 DAY, P
159	PHY6C	14.0	30.0	22506.	20434.	120.	120.	55.0		SORTIE HIGH ENERGY, 7 DAY, P
160	PHY6D	14.0	27.0	20720.	18138.	120.	120.	28.5		SORTIE HIGH ENERGY, 7 DAY, P
161	PHY6E3	14.0	45.0	39218.	30598.	120.	120.	28.5		SORTIE HIGH ENERGY, 30 DAY, P
162	PHY7A	14.0	60.0	29002.	28238.	200.	200.	28.5		SORTIE ATMO. SPACE PHYSICS, 7 DAY, L+P
163	PHY7B	14.0	60.0	29002.	28238.	200.	200.	55.0		SORTIE ATMO. SPACE PHYSICS, 7 DAY, L+P
164	PHY7C	14.0	60.0	29002.	28238.	180.	180.	90.0		SORTIE ATMO. SPACE PHYSICS, 7 DAY, L+P
165	LS2A7	14.0	58.5	37532.	30185.	150.	150.	28.5		SORTIE LIFE SCIENCE, 7 DAY, L
166	LS2A3	14.0	58.5	37532.	30185.	150.	150.	28.5		SORTIE LIFE SCIENCE, 30 DAY, L
167	ST2A	14.0	60.0	25296.	24532.	200.	200.	55.0		SORTIE SORTIE SPACE TECH., 7 DAY, L+P
168	ST2B	14.0	60.0	25296.	24532.	200.	200.	55.0		SORTIE SPACE TECH., 7 DAY, L+P

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Table 2-1. PAYLOAD CHARACTERISTICS (Concluded)

ID	NASA NO	DIAM-FT	LGTH-FT	UP WT-LBS	DN WT-LBS	PL APO-NM	PL PER-NM	PL INCL-DEG	CANNOT BE LAUNCHED WITH	
169	ST2C	14.0	60.0	25296.	24532.	200.	200.	55.0		SORTIE SPACE TECH., 7 DAY, L+P
170	ST2D	14.0	60.0	25296.	24532.	200.	200.	55.0		SORTIE SPACE TECH., 7 DAY, L+P
171	OA1A75	14.0	60.0	27002.	26138.	180.	180.	55.0		SORTIE OFFICE OF APPLIC., 7 DAY, L+P
172	OA1A79	14.0	60.0	27002.	26138.	160.	160.	90.0		SORTIE OFFICE OF APPLIC., 7 DAY, L+P
173	OA1B75	14.0	60.0	25402.	24538.	180.	180.	55.0		SORTIE OFFICE OF APPLIC., 7 DAY, L+P
174	OA1B79	14.0	60.0	25402.	24538.	160.	160.	90.0		SORTIE OFFICE OF APPLIC., 7 DAY, L+P
175	SP1A	14.0	60.0	26084.	25320.	180.	180.	28.5		SORTIE OFFICE OF APPLIC., 7 DAY, L+P
176	ND17A73	14.0	60.0	26502.	25638.	180.	180.	28.5		SORTIE EARTH OBS., 7 DAY, L+P
177	ND17A79	14.0	60.0	26502.	25638.	180.	180.	90.0		SORTIE EARTH OBS., 7 DAY, L+P
178	NND17B	14.0	45.0	26798.	25166.	162.	162.	28.5		SORTIE ASTM, 7 DAY, P
179	NND17C	14.0	60.0	26482.	25718.	200.	200.	28.5		SORTIE GPL 1, 7 DAY, L+P
180	NND17D	14.0	60.0	26261.	25497.	200.	200.	28.5		SORTIE GPL 2, 7 DAY, L+P
181	SP1B	14.0	5.0	6171.	5239.	160.	160.	28.5		SORTIE OFFICE OF APPLIC., 7 DAY, P
182	SP1C	14.0	5.0	5121.	4189.	160.	160.	28.5		SORTIE OFFICE OF APPLIC., 7 DAY, P
183	NND16A	14.0	5.0	6171.	5239.	160.	160.	28.5		SORTIE SPACE MFG., 7 DAY, P
184	NND16B	14.0	5.0	5121.	4184.	160.	160.	28.5		SORTIE SPACE MFG., 7 DAY, P

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Table 2-2. PAYLOAD SCHEDULE

ID	NASA NO		YEAR									
		81	82	83	84	85	86	87	88	89	90	91
1	AST-1A	1	1	1	1	1	1	1	1	1	1	1
2	AST-1A R	2	1	0	0	0	0	1	1	1	1	1
3	AST-1B	0	0	1	0	1	0	1	0	0	0	0
4	AST-1B R	0	0	0	1	0	1	0	0	0	0	0
5	AST-3	0	1	0	1	0	1	0	1	0	1	0
6	AST-3 R	0	1	0	1	0	1	0	1	0	1	0
7	AST-4	0	0	0	0	0	0	0	0	0	0	0
8	AST-4 R	0	0	0	0	0	0	0	0	0	0	0
9	AST-5	0	1	0	0	0	1	1	0	0	0	1
10	AST-5 R	0	0	0	0	0	1	0	0	0	1	1
11	AST-5V	0	0	1	0	1	0	1	0	2	0	0
12	AST-6	0	0	1	0	0	0	0	1	0	0	0
13	AST-6 R	0	0	1	0	0	0	0	1	0	0	0
14	AST-6V	1	1	0	1	1	1	1	0	1	1	1
15	AST-7	0	0	0	0	1	0	0	0	0	0	0
16	AST-7 R	0	0	0	0	0	0	0	0	0	0	0
17	AST-7V	0	0	0	0	0	1	1	1	1	1	1
18	AST-8	0	0	0	0	1	0	0	0	0	0	0
19	AST-8 R	0	0	0	0	0	0	0	0	0	0	0
20	AST-8V	0	0	0	0	0	0	1	0	1	0	1
21	AST-9A	0	0	1	0	0	0	0	0	0	0	1
22	AST-9A R	0	0	0	0	0	0	1	0	0	0	0
23	AST-9AV	0	0	0	1	1	0	0	0	0	0	0
24	AST-9B	0	0	0	0	0	1	0	0	0	0	0
25	AST-9B R	0	0	0	0	0	0	0	0	0	0	1
26	AST-9BV	0	0	0	0	0	0	0	1	0	1	0
27	PHY-1A	0	0	0	1	0	0	0	1	1	1	1
28	PHY-1A R	0	0	0	1	0	0	0	0	1	1	1
29	PHY-1B	1	0	0	1	0	0	0	1	1	1	1
30	PHY-1B R	0	0	0	1	0	0	0	0	1	1	1
31	PHY-1C	0	1	1	0	1	1	1	0	0	0	0
32	PHY-2A	0	0	1	0	0	0	0	0	0	0	0
33	PHY-2B	0	0	0	0	0	1	0	0	0	0	1
34	PHY-3A	1	0	0	1	0	0	0	0	0	0	0
35	PHY-3A R	0	0	0	1	0	0	0	0	0	0	0
36	PHY-3B	0	0	0	0	0	0	1	0	0	1	0
37	PHY-3B R	0	0	0	0	0	0	0	0	0	1	0
38	PHY-4	0	0	0	0	0	0	0	1	0	0	0
39	PHY-5	0	0	0	0	0	0	1	0	0	0	0
40	PHY-5 R	0	0	0	0	0	0	0	0	0	0	0

Table 2-2. PAYLOAD SCHEDULE (Continued)

ID	NASA NO		YEAR 81	82	83	84	85	86	87	88	89	90	91
41	PHY-5V		0	0	0	0	0	0	0	1	1	1	1
42	PL-7		0	0	0	2	0	0	0	0	0	0	0
43	PL-8		0	0	0	0	0	0	0	0	0	1	1
44	PL-10		2	0	1	0	0	1	0	0	0	0	0
45	PL-11		0	0	2	0	0	0	0	0	0	0	0
46	PL-12		0	0	0	0	2	0	0	0	0	0	0
47	PL-13		0	0	0	0	0	0	2	0	0	0	0
48	PL-14		0	0	0	0	0	0	0	0	2	0	0
49	PL-17		0	0	0	0	0	0	0	0	0	0	0
50	PL-18		1	0	0	0	0	0	0	0	0	0	0
51	PL-19		2	0	0	0	0	0	0	0	0	0	0
52	PL-20		0	0	0	2	0	0	0	0	0	0	0
53	PL-21		0	0	0	0	2	0	0	0	0	0	0
54	PL-22		0	0	0	0	0	2	0	0	0	0	0
55	PL-23		0	0	0	0	0	0	0	0	0	1	1
56	PL-26		0	0	0	0	0	0	0	0	0	0	0
57	PL-27		0	0	0	0	1	0	0	0	0	0	0
58	PL-28		0	0	0	0	0	2	0	0	0	0	0
59	LUN-2		0	0	0	1	0	1	0	0	0	0	0
60	LUN-3		0	0	0	0	0	0	1	1	0	0	0
61	LUN-4		0	0	0	0	0	0	0	0	1	0	0
62	LUN-5		0	0	0	0	0	0	0	0	0	1	1
63	LS-1		2	2	2	2	2	2	2	2	2	2	2
64	LS-1	R	2	2	2	2	2	2	2	2	2	2	2
65	EO-3A		0	0	1	0	0	0	1	0	0	0	0
66	EO-3A	R	0	1	0	0	0	1	0	0	0	0	1
67	EO-3AV		0	0	0	0	1	0	0	0	0	1	0
68	EO-3B		0	0	0	0	1	0	0	0	1	0	0
69	EO-3B	R	0	0	1	0	0	0	1	0	0	0	0
70	EO-3BV		0	1	0	0	0	0	0	0	0	0	0
71	EO-3C		0	0	0	0	0	1	0	0	0	0	1
72	EO-3C	R	0	0	0	0	1	0	0	0	1	0	0
73	EO-3CV		0	0	0	1	0	0	0	1	0	0	0
74	EO-3D		0	0	0	0	0	0	0	0	0	0	0
75	EO-3D	R	0	0	0	0	0	0	0	0	0	0	0
76	EO-4A		1	0	1	0	1	0	0	0	0	0	0
77	EO-4A	R	0	0	0	0	0	1	0	0	0	0	0
78	EO-4B		0	0	0	0	0	0	2	0	2	0	2
79	EO-4B	R	0	0	0	0	0	0	0	0	0	0	0
80	EO-5A		0	0	0	0	0	0	0	0	0	0	0

Table 2-2. PAYLOAD SCHEDULE (Continued)

ID	NASA NO		YEAR 81	82	83	84	85	86	87	88	89	90	91
81	EO-5A	R	0	0	0	0	0	0	0	0	0	0	0
82	EO-5B		0	0	0	0	0	0	0	0	0	0	0
83	EO-5B	R	0	0	0	0	0	0	0	0	0	0	0
84	EO-5C		0	0	0	1	0	0	1	0	0	1	0
85	EO-5C	R	0	0	0	0	0	0	0	0	0	0	0
86	EO-5D		0	0	0	0	1	0	0	1	0	0	1
87	EO-5D	R	0	0	0	0	0	0	0	0	0	0	0
88	EO-5E		0	0	1	0	0	1	0	0	1	0	0
89	EO-5E	R	0	0	0	0	0	0	0	0	0	0	0
90	EO-6		0	0	0	0	0	0	1	0	0	0	0
91	EO-6	R	0	0	0	1	0	0	0	0	0	0	0
92	EO-7		0	0	0	0	0	0	1	0	0	0	0
93	EOP-3		0	0	0	0	0	0	0	0	0	0	0
94	EOP-4		0	0	0	0	0	0	0	0	0	0	0
95	EOP-5		0	0	0	0	0	0	0	0	0	0	0
96	EOP-6A		0	0	0	0	2	0	0	0	0	0	0
97	EOP-6B		0	0	0	0	2	0	0	0	0	0	0
98	EOP-6C		0	0	0	0	2	0	0	0	0	0	0
99	EOP-8		0	0	0	0	0	3	0	0	0	3	0
100	EOP-8	R	0	0	3	0	0	0	3	0	0	0	0
101	EOP-9		1	0	0	0	0	1	0	0	0	1	0
102	EOP-9	R	0	0	0	1	0	0	1	0	0	0	0
103	NN/D-2C		0	0	3	0	0	0	0	3	0	0	0
104	NN/D-2C	R	0	0	0	0	0	0	0	0	0	0	0
105	ST-1		0	1	0	1	0	1	0	1	0	1	0
106	ST-1	R	1	1	0	1	0	1	0	1	0	1	0
107	NN/D-1		0	0	2	3	2	2	0	0	2	3	2
108	NN/D-1	R	0	0	0	0	0	0	1	1	2	3	0
109	NN/D-2A		2	2	1	0	0	0	0	0	0	0	0
110	NN/D-2B		0	0	0	1	1	2	2	3	2	2	1
111	NN/D-2B	R	0	0	0	0	0	0	0	0	0	0	0
112	NN/D-3		1	1	0	0	1	0	0	0	0	1	0
113	NN/D-3	R	0	0	0	0	0	1	1	0	0	0	0
114	NN/D-4		2	1	1	1	0	1	0	1	0	1	0
115	NN/D-4	R	0	0	0	0	0	0	0	0	0	0	0
116	NN/D-5		1	1	1	1	1	1	1	1	1	1	1
117	NN/D-5	R	0	0	0	3	1	0	0	1	1	1	1
118	NN/D-6		0	0	0	0	1	0	0	1	0	1	0
119	NN/D-8		0	0	0	0	1	1	1	1	0	1	1
120	NN/D-8	R	0	0	0	3	0	0	1	1	1	1	0

Table 2-2. PAYLOAD SCHEDULE (Continued)

ID	NASA NO	YEAR 81	82	83	84	85	86	87	88	89	90	91
121	NN/D-9	1	1	0	1	0	1	0	1	0	1	0
122	NN/D-9 R	0	0	0	1	1	1	0	1	0	1	0
123	NN/D-10	1	1	1	0	1	0	1	1	1	0	1
124	NN/D-10R	0	0	0	3	1	1	1	0	1	0	1
125	NN/D-11	0	0	1	1	1	1	1	1	1	1	1
126	NN/D-11R	0	0	2	1	1	1	1	1	1	1	1
127	NN/D-12	0	0	0	0	0	0	0	2	0	2	0
128	NN/D-12R	0	0	0	0	0	0	0	0	0	0	0
129	NN/D-13	0	0	0	0	0	0	0	1	2	0	1
130	NN/D-13R	0	0	0	0	0	0	0	0	0	0	0
131	NN/D-14	0	0	0	0	0	3	0	3	0	3	0
132	NN/D-14R	0	0	0	0	0	0	0	0	3	0	3
133	AST10A	1	1	1	0	0	0	0	0	0	0	0
134	AST10B	0	1	1	0	0	0	0	0	0	0	0
135	AST10C	0	0	0	1	1	0	0	0	0	0	0
136	AS10D73	0	0	0	1	0	0	0	0	0	0	0
137	AS10D79	0	0	0	0	1	0	0	0	0	0	0
138	AS10D33	0	0	0	0	0	0	1	1	0	0	1
139	AS10D39	0	0	0	0	0	1	0	0	1	1	0
140	AST10E	0	0	0	1	1	0	0	0	0	0	0
141	AST10F	0	0	0	0	1	1	0	0	0	0	0
142	AST10G	0	0	0	0	1	1	0	0	0	0	0
143	AST10H	0	0	0	0	0	1	0	0	0	0	0
144	AST10I	0	0	0	0	0	1	0	0	0	0	0
145	AST10J	0	0	0	0	0	0	1	0	0	1	0
146	AST10K7	0	0	0	0	0	0	1	0	0	0	0
147	AST10K3	0	0	0	0	0	0	0	1	1	1	1
148	AST10L	0	0	0	0	0	0	0	1	0	0	1
149	AST10M	0	0	0	0	0	0	0	1	1	0	0
150	AST11B	0	1	2	2	1	1	1	0	0	0	0
151	AST11C7	0	0	0	0	2	0	0	0	0	0	0
152	AST11A	1	0	0	0	0	0	0	0	0	0	0
153	AST11C3	0	0	0	0	0	1	0	0	0	0	0
154	AST11D7	0	0	0	0	0	0	1	0	0	0	0
155	AST11D3	0	0	0	0	0	0	1	0	0	0	0
156	AST11E7	0	0	0	0	0	0	0	1	2	1	2
157	AST11E3	0	0	0	0	0	0	0	1	1	1	1
158	PHY6A/B	0	1	1	1	1	0	0	0	0	0	0
159	PHY6C	1	1	1	1	1	1	1	1	1	1	1
160	PHY6D	1	1	1	1	1	1	1	1	1	1	1

Table 2-2. PAYLOAD SCHEDULE (Concluded)

ID	NASA NO	YEAR 81	82	83	84	85	86	87	88	89	90	91
161	PHY6E3	0	0	0	0	0	1	1	1	1	1	1
162	PHY7A	1	1	0	0	1	1	0	1	1	1	1
163	PHY7B	0	0	0	1	0	1	1	1	0	1	0
164	PHY7C	0	0	1	2	2	2	2	2	2	2	2
165	LS2A7	2	2	0	0	0	0	0	0	0	0	0
166	LS2A3	0	0	2	2	2	2	2	3	3	3	3
167	ST2A	1	1	1	1	1	1	1	1	1	1	1
168	ST2B	1	1	1	1	1	1	1	1	1	1	1
169	ST2C	1	1	1	1	1	1	1	1	1	1	1
170	ST2D	1	1	1	1	1	1	1	1	1	1	1
171	OA1A75	1	1	1	0	1	0	1	0	0	1	0
172	OA1A79	0	0	0	1	0	1	0	1	1	0	1
173	OA1B75	1	1	1	1	0	1	0	1	0	0	1
174	OA1B79	0	0	0	0	1	0	1	0	1	1	0
175	SP1A	1	1	1	1	1	1	1	1	1	1	1
176	ND17A73	1	1	0	0	0	0	0	0	0	0	0
177	ND17A79	0	0	1	1	1	1	1	1	1	1	1
178	NND17B	1	1	1	1	1	1	1	1	1	1	1
179	NND17C	1	1	1	1	1	1	1	1	1	1	1
180	NND17D	0	0	1	0	1	0	1	0	1	0	1
181	SP1B	2	6	6	6	6	6	6	6	6	6	6
182	SP1C	2	6	6	6	6	6	6	6	6	6	6
183	NND16A	0	0	0	0	2	4	2	4	2	4	2
184	NND16B	0	0	0	0	2	4	2	4	2	4	2
	TOTAL	47	53	67	81	84	93	82	91	84	99	83

The mission model contains two kinds of payloads: automated payloads that operate independently of the Shuttle (ID numbers 1 through 132 in Table 2-1) and sortie lab payloads which are dependent on Shuttle and remain in the cargo bay (ID numbers 133 through 184). The last four of these (181-184) do not have a preferred orbit and can be launched to any orbit within Shuttle capability. Automated payloads fall into six mission classes: earth escape missions which include lunar, planetary, and interplanetary missions, and five earth orbital mission classes.

The five earth orbital classes are: geosynchronous equatorial missions, polar and sun synchronous missions at inclinations from 90 to 103 degrees, 55-degree inclination missions, high energy 28.5 degree missions at or above geosynchronous altitude, and 28.5 degree missions at low and intermediate orbital altitudes. The number of each class of mission is shown in Table 2-3.

Table 2-3. MISSION MODEL SUMMARY NUMBER OF MISSIONS 1981-1991

SORTIE MISSIONS			425
AUTOMATED PLD MISSIONS			
ESCAPE		45	
EARTH ORBIT			
GEOSYNCHRONOUS	133		
POLAR AND SUN SYNCHRONOUS	97		
55° INCLINATION	8		
28.5° HIGH ENERGY	9		
28.5° LOW AND INTERMEDIATE ORBIT	147		
TOTAL EARTH ORBIT MISSIONS	394	394	
TOTAL AUTOMATED MISSIONS		439	439
TOTAL MISSIONS IN MODEL			864

In the earth orbit mission classes, 24 polar and sun synchronous missions are beyond Shuttle-alone capability and require an upper stage or a propulsion capability integrated into the payload. Six of the 55-degree inclination missions and 15 of the 28.5-degree low and intermediate orbital missions require upper stages. The earth orbital mission classes become somewhat indistinct in traffic models generated by the WHATIF program, particularly on missions with upper stages. Where assigning payloads to flights, the WHATIF program, in order to make maximum use of available cargo bay volume, can choose payloads in any mission class subject only to the constraints and restrictions already mentioned. The 28.5-degree low and intermediate orbit and high energy missions are frequently combined on flights with geosynchronous missions which also require 28.5-degree Shuttle launches. On combined SEPS-Tug sorties where SEPS augments Tug performance (because of the Tug plane change capability at higher altitudes) 55-degree missions are occasionally combined with 28.5-degree high energy and intermediate orbital missions. Polar and sun synchronous missions are never assigned to flights with any other class of missions because of the large plane changes involved. Escape missions are dedicated flights, each one requiring its own Shuttle and Tug (in some cases multiple Shuttles and Tugs). Their large energy requirements prohibit combining them with other escape or earth orbital missions except for the any-orbit sortie missions which stay with the Shuttle.

2.2 SEPS MISSION ROLES

When used as a transportation stage in conjunction with Shuttle and Tug, SEPS transport only effectiveness can be indicated by a reduction in Shuttle flights required to deliver the payloads in the mission mode. Given enough time, SEPS can deliver any payload or combination of payloads that can be loaded in the Shuttle cargo bay. Thus, SEPS is able to reduce Shuttle flights by allowing more payloads per flight than would otherwise be possible and by eliminating the requirement for tandem Tugs and dual Shuttle launches on high energy missions. Previous studies[4] using an earlier mission model identified four potential mission roles or classes of missions where SEPS capabilities resulted in significant Shuttle flight savings. These mission roles were: planetary missions, polar and sun synchronous missions just beyond Shuttle capability, orbital taxi missions in geosynchronous orbit, and geosynchronous delivery and retrieval missions. For the study reference mission model, effective use of SEPS-Tug sorties combined intermediate orbital delivery,

[4] *Rockwell International Corporation Report SD 72-SA-0132-2-3, "Extended Definition Feasibility Study for a Solar Electric Propulsion Stage Concept Definition," 21 December 1973.*

retreival, transport to and from geosynchronous orbit, and orbital taxi roles. The following discussion illustrates this point.

ESCAPE MISSIONS

Analysis of planetary missions using SEPS was done only to the extent necessary to ensure that recommended SEPS configuration characteristics for earth orbital missions did not compromise planetary ability. The six planetary missions that are currently planned with SEPS are in the mission model, and the Shuttle launches required for them are included in the traffic model analysis results. Two additional planetary SEPS missions in 1981 are planned for expendable vehicle launch and are not included, but they do not affect Shuttle flight requirements. Table 2-4 is a summary of launch vehicles required by the 45 escape missions in the mission model. This table was constructed from traffic model results using an IUS (expendable transtage) in 1981-1983 and the 30-foot baseline Tug in 1984-1991. Sixty Shuttles are required to launch these missions. Of the 45 escape missions, 7 are lunar missions, 8 interplanetary (heliocentric and so forth) and 30 are planetary.

Table 2-4. ESCAPE MISSIONS, NUMBER OF MISSIONS 1981-1991

	1 SHUTTLE/MISSION		ORBITAL ASSY. REQ'D-2 SHUTTLES/MISSION					
YEAR	IUS	IUS-BII	IUS	IUS-BII	IUS-IUS	IUS-IUS-BII	MISSIONS	SHUTTLE LAUNCHES
81	2	2				1	5	6
82	1						1	1
83	2				2		4	6
	Tug	Tug-BII	Tug	Tug-BII	Tug-Tug	XTug-BII		
84	3	2					5	5
85	2				2	2*	6	10
86	5	1				2	8	10
87	2		2*				4	6
88	1	1					2	2
89	3						3	3
90	1			1*		1*	3	5
91	1	1		1*		1*	4	6
						TOTAL	45	60

* *Payload too long to fit in cargo bay with Tug*

- *Expended IUS 1981-1983*
- *30' Baseline TUG 1984-1991*

Fifteen of the planetary missions require dual shuttle launches with assembly of the upper stages and payloads in Shuttle parking orbit. If SEPS could be used on these missions to reduce mission ΔV (V_∞) to within the capability of a single upper stage, 15 Shuttle flights could be saved. An examination of payload dimensions in Table 2-1 shows that the payloads on eight of these missions are too long to fit the cargo bay with the Tug. An earlier study by Rockwell International Corporation[5], has shown that of the remaining seven missions, four are feasible with SEPS (two PL-11 in 1983 and two PL-12 in 1985). Thus, of the 15 potential Shuttle flight savings, 4 are actually possible with the present payload size definitions. It would be necessary to expend the SEPS on these four flights. Cost analysis indicates that the cost of SEPS is about the same as the cost of a Shuttle launch; therefore, there is no motivation to use SEPS unless the payload and missions are redefined to exploit the greater allowable payload mass and maneuver capability provided by SEPS.

POLAR AND SUN SYNCHRONOUS MISSIONS

There are 97 polar and sun synchronous missions in the mission model. A total of 24 of these are at altitudes above the 500 nautical mile Shuttle limit in polar and near-polar inclinations. The need for a Tug on these missions could be eliminated by using SEPS to make the necessary altitude and plane changes after the Shuttle had delivered the payloads to a suitable parking orbit within its capability. Fewer Shuttle launches would be required since the extra room in the cargo bay could be used for additional payloads on each flight. The highest altitude at which SEPS can operate in low-earth orbit missions is limited by radiation trapped in the Van Allen belt. This radiation becomes intense above 1000 nautical miles. To avoid crippling degradation of the solar arrays, SEPS must operate below this altitude or use higher cost self-annealing solar cells operated at temperatures that significantly reduce efficiency. Nine of the 24 missions are above a 1000-nautical mile altitude, leaving 15 missions within the range of SEPS operation. Traffic

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[5] *Rockwell International Corporation Letter 73MA4936, "Application of SEP Stage to Planetary Missions," 13 September 1973.*

model results show that 10 of these 15 missions are included on Tug flights required for delivery of the nine missions above 1000 nautical miles. The five remaining missions require Tug flight in each of the years 1985, 1986, and 1987. Since these three flights are in three different years they cannot be combined to save a Shuttle launch, and the most SEPS can do in this mission role is save three Tug sorties. Dedicating a SEPS for 3 years to deliver five payloads and save three Tug sorties at $0.96 million each did not appear to be cost effective, and this SEPS mission role was dropped from the traffic model analysis.

The polar and sun synchronous mission role was briefly reevaluated near the end of this study. There are 73 polar and sun synchronous missions within the Shuttle's capability. These missions are at orbital inclinations of 90, 97, 98, 99, 102, and 103 degrees. Since the Shuttle essentially has no onorbit plane-change capability, payloads at different inclinations can not be mixed on the same flight. If SEPS were used to make the plane changes, these payloads could be more efficiently assigned to flights and fewer Shuttles would be required. To assess this potential it was assumed that the lowest altitude for SEPS operation would be 200 nautical miles (this limit is where atmospheric drag on SEPS is equal to its thrust, and it is somewhere between 200 and 300 nautical miles - the uncertainty is due to large variations in atmospheric density at these altitudes). It was also assumed that SEPS would be able to do what was demanded of it by the payload combinations on each Shuttle flight within reasonable trip times.

Accordingly, the destination orbits for these 73 payloads were redefined to a common Shuttle parking orbit of 200 nautical miles at a 98-degree inclination. A traffic model was then generated with the WHATIF program. The result is shown in Table 2-5 for the years 1982 through 1991. There are no polar missions in 1981. Without SEPS, 39 Shuttle launches are required; 12 of these include Tugs for delivery of the 24 payloads above the Shuttle's capability. Twenty-nine Shuttle flights are required with SEPS, nine of which include Tugs. The three Tugs saved are those previously mentioned. The total of ten Shuttle flights saved is an optimistic estimate of SEPS potential in this

Table 2-5. POLAR AND SUN SYNCHRONOUS FLIGHTS

YEAR	WITHOUT SEPS		WITH SEPS		
	SHUTTLES	TUGS	SHUTTLES	TUGS	SORTIES
1982	1	--	1	--	0
1983	5	--	3	--	1
1984	4	2	3	2	1
1985	4	1	2	0	2
1986	4	1	3	0	1
1987	4	1	2	0	2
1988	4	1	3	1	1
1989	4	2	4	2	0
1990	5	2	4	2	1
1991	4	2	4	2	0
TOTAL	39	12	29	9	9

mission role since the limits on SEPS trip time and the necessity of launching and retrieving SEPS are not considered.

A problem largely ignored in this and past traffic modeling exercises is that of the relative orientation of the line-of-nodes of these orbits. Though not presently included in the mission model, nodal orientations for these missions will almost certainly be specified, particularly for the sun synchronous missions which will have some preferred orientation with respect to the earth sun line. Even for those missions without specified nodal directions, precession during the time they are in orbit (which in general will not be the same for any two payloads) will result in widely separated nodes at retrieval time. Nodal shifts at low altitudes can easily tax Tug performance capability. Nodal shifts are possible with SEPS, but trip times become unacceptably long. While one may occasionally arrange to deliver several payloads on one flight, retrieval of more than one is unlikely. Thus, delivery and retrieval of multiple payloads in this class of missions will not be the rule; and assumed flight savings made in this way are likely to be more imaginary than real.

In all probability, the direction of the line-of-nodes for missions other than sun synchronous will not be important, and in general, not specified. The expectation of being able to deliver on one flight as many of these payloads as can be loaded into the cargo bay without exceeding the launch vehicle performance capability is a reasonable one. However, orbital precession makes it doubtful that more than one payload per flight can be retrieved with either Shuttle or Shuttle-Tug in any class of missions except, of course, geosynchronous. This has been ignored in the solar and sun synchronous simplified traffic model studies above. It is assumed throughout this study that multiple payload retrievals are possible on both Shuttle and Shuttle-Tug flights.

GEOSYNCHRONOUS ORBITAL TAXI MISSION POTENTIAL

Geosynchronous payloads must be stationed over specified longitudes. When groups of these payloads are delivered or retrieved by Tug, it must make a series of longitude shifts (or it must be assumed the payloads themselves have this maneuver capability) to position the up payloads and gather together the down payloads. Tug propellant consumed by these onorbit maneuvers is inversely proportional to the time allowed for them. In the limit, any longitudinal shift can be made in infinite time with zero propellant. However, the Tug has an onorbit lifetime of roughly 7 days, and the propellant required for longitudinal shifts with this time constraint can markedly reduce the Tug's already limited payload retrieval capability. In recognition of this requirement it is usually conceded in traffic model analysis that the maximum number of geosynchronous payloads that can be handled on one Tug flight is three up and one down even when available orbiter cargo bay volume allows more payloads. An orbital taxi SEPS placed in geosynchronous orbit to position and gather up payloads could relieve Tug of this requirement and allow it to deliver and retrieve as many payloads as it could without exceeding its performance limit. To get an indication of the worth of SEPS as a geosynchronous orbital taxi, traffic models with and without the three up and one down constraint are compared in Table 2-6. The number of upper stage flights required by the mission model is shown for geosynchronous, polar and sun synchronous, and other mission classes (these other are the 28.5-degree high energy and intermediate missions and the 55-degree missions). This comparison shows that the orbital

Table 2-6. ORBITAL TAXI MISSION ROLE

	Number of Upper Stage Flights - IUS and 9.1 M BL Tug					
	NO SEPS (3 UP, 1DOWN CONSTRAINT)			ORBITAL TAXI SEPS		
YEAR	GEOSYNC	POLAR	OTHER	GEOSYNC	POLAR	OTHER
1981	4	--	1	3	--	1
1982	3	--	--	2	--	--
1983	5	--	--	5	--	--
1984	9	5	3	5	2	3
1985	5	1	1	4	1	1
1986	6	1	1	6	1	1
1987	5	1	2	5	1	2
1988	7	1	--	6	1	--
1989	8	2	1	6	2	1
1990	9	2	2	8	2	2
1991	4	2	1	4	2	1
TOTAL	65	15	12	54	12	12

taxi SEPS can save 11 flights in delivery and retrieval of geosynchronous payloads assuming none of them had self-taxiing ability.

Most of the geosynchronous orbit payloads, due to their requirement for mission stationkeeping and attitude control, have the inherent capability for self-taxiing. Their ACS usually provide for both attitude control and stationkeeping propulsion functions. If the payloads' ACS propellant supplies are increased from about 2 to 8 percent (depending on the specific payload) more than the nominal requirement, the payloads are self-taxiing.

In order to be ultraconservative and realistic, NSI's System Operational Profile for STS without an earth orbital SEPS does not arbitrarily limit Tug alone sorties to three payloads up and one payload down on any individual flight. Tug's multiple payload package delivery and retrieval capability is limited only by Orbiter's characteristics and Tug's performance. Since STS without EO SEPS' unconstrained performance is the reference for cost effectiveness analyses, the 11 flights mentioned above are not included in the SEPS transportation cost savings development.

COMBINATION TUG-SEPS SORTIES FOR MAXIMUM STS TRANSPORT EFFECTIVENESS

In this study, the orbital taxi mission role was considered to be an essential and integral part of the SEPS geosynchronous mission role. Thus, the time and propellant required by SEPS to do the longitudinal shifts dictated by the payloads being carried on a sortie are included in performance calculations when SEPS is used as a transport stage for the delivery and retreival of geosynchronous payloads.

Some longitudinal position data is specified in the level B Space Shuttle Payload Data (SSPD) and in the reference mission model, but not in sufficient detail for traffic model analysis. Using the SSPD as a guide, and based on information supplied by Marshall Space Flight Center, Table 2-7 was developed. Table 2-7 specifies by year the west longitude for the geosynchronous deliveries and retrievals in the mission model. Delivery longitudes are shown above the diagonal, retreival longitudes below. When several deliveries or retrievals of one type of payload are specified in a year, their respective longitudes are shown by more than one entry above or below the diagonal.

There are 133 geosynchronous missions in the mission model, 102 of these are deliveries, 31 are retreivals. This number is more than five times the number in any other mission class that requires upper stages. This provides the opportunity for SEPS to demonstrate its effectiveness when used to augment the Tug's performance as a transportation stage. This study shows that the most effective mode of operation for SEPS is a space-based mode with refueling. SEPS is launched and remains in space until the end of its useful life at which time it is retrieved for refurbishment and reuse. Once launched, SEPS repeatedly shuttles back and forth between geosynchronous orbit and changeover orbit where it meets and exchanges multiple payload packages with Tug. SEPS performs all taxi functions between specific geosynchronous longitudes. Traffic models were developed with SEPS in the geosynchronous mission role for

Table 2-7. WEST LONGITUDE OF GEOSYNCHRONOUS PAYLOADS

ID	NASA NO.	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991
76 / 77	EO–4A	100		100		100	100					
78 / 79	EO–4B							100, 105		100, 105		100, 105
88	EO–5E			73			87			100		
92	EO–7							95				
103	NN/D–2C			40, 105, 170					40, 105, 170			
107 / 108	NN/D–1			40, 180	45, 185, 35	50, 175	30, 190	/ 40	/ 180	55, 25 / 45, 185	60, 20, 170 / 35, 50, 175	65, 195
109	NN/D–2A	98, 130	104, 120	108								
110	NN/D–2B				85	88	91, 94	97, 103	106, 109, 112	118, 121	124, 127	130
112 / 113	NN/D–3	94	124			94	/ 94	/ 124			124	
114	NN/D–4	52, 180	162	175	29		162		52		175	
116 / 117	NN/D–5	0	94	5	99 / 0, 94, 5	0 / 99	96	10	86 / 0	350 / 96	106 / 10	0 / 86
118	NN/D–6					96			100		96	
121 / 122	NN/D–9	225	220		215 / 225	/ 220	225 / 215		220 / 225		215 / 220	
123 / 124	NND–10	80	90	120	/ 80, 90, 120	80 / 80	/ 80	90 / 90	120	80 / 120		100 / 80
127	NN/D–12								80, 115		80, 115	
129	NN/D–13								0	0, 60		60

several Tug and SEPS configurations. These were compared to baseline traffic models generated without SEPS to evaluate their effect on transportation cost. The effect of various SEPS and Tug operational constraints were similarly investigated. In particular, the delivery of payloads from the low and intermediate orbit class by Tug on its way to changeover orbit with geosynchronous payloads was found to be desirable. In each case, traffic models were made for the complete mission model. This allowed payloads in one class of missions to be loaded with those from another whenever it resulted in saving Shuttle flights. More efficient use is made of available cargo space and vehicle performance that would be the case if the mission model had been segregated into classes. A discussion of the traffic model results will be presented after a discussion of trajectory analysis to maximize SEPS-Tug performance and the traffic modeling methods.

2.3 TRAFFIC MODELING METHOD

Traffic modeling is the determination of the number of Shuttle Tug flights (with their cargo manifests specified) in each year required to deliver and retrieve the payloads specified by the mission model and the sequencing of SEPS sorties by date. Payloads are added to a Shuttle flight until no more will fit in the Shuttle cargo bay or the Shuttle or Tug performance limit is exceeded. The WHATIF computer program[6] developed by Northrop has been used for previous traffic model studies by both NSI and MSFC. With the addition of SEPS to the STS as a transportation stage in the geosynchronous mission role, traffic modeling takes on several new aspects. Without SEPS, Shuttle flights use discrete and independent events and their scheduling is relatively straightforward. In fact, WHATIF does not actually assign launch dates to Shuttle flights but simply provides a list of the necessary flights and payload assignments in each mission year and these are subsequently scheduled.

SEPS sorties can take as much as half a year and they are not independent. SEPS performance on a sortie depends on the propellant and power (as affected

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[6] *Ivory, L. R., "Shuttle and Tug traffic Scheduling Program," Northrop Services, Inc., Huntsville, Alabama, Informal Memorandum 9240-73-158, April 1973.*

by radiation) remaining at the end of the previous sortie. Thus, scheduling of SEPS sorties and the launch, retrieval, and refueling of onorbit SEPS requires keeping track of the status and availability of each SEPS. Changeover orbital data is also required for the determination of sortie trip times. To provide this capability, a number of additions were made to the WHATIF program. These additions resulted in what is, for practical purposes, a two part program with each part largely independent of the other. In order to minimize the number of Shuttle flights required, the maximum possible utilization of the orbiter's cargo bay volume must be accomplished.

The first part of the two part program packages payloads in the available cargo volume forward of Tug for the ascent part of the sortie. If payload retrievals or service round trip missions occur in that year, a descent package is determined. If Tug alone cannot accomplish that sortie it is assigned to a Tug-SEPS sortie. The first part then determines Tug-SEPS changeover orbits and trip times; schedules Shuttle-Tug launches, by day number to support the SEPS sorties; and determines the necessity of launching, retrieving, or refueling SEPS. After all full-volume or Tug performance limited missions have been assigned to SEPS-Tug sorties, the second part (the original WHATIF program) then assigns to Shuttle or Shuttle-Tug flights the remaining payloads.

With one exception, for the second part functions, the operation and capability of the original program for scheduling of Shuttle flights without SEPS was not changed. The MOLTOP computer program[7] was used to generate SEPS trajectory and changeover orbit data. This data was then included (in the form of tables) in the SEPS part of the WHATIF program. Data input and output routines, payload packaging routines, and the method of assigning payloads to flights are common to the two parts of the program. The following paragraphs describe:

- The methods used in the WHATIF part of the program
- Tug-SEPS performance calculations for delivery of geosynchronous payloads

[7] *Williams, D. F., "MOLTOP Users Manual," Northrop Services, Inc., Huntsville, Alabama, Memorandum M-240-1224, October 1973.*

- Generation of changeover orbit data using the MOLTOP program
- The operation of the SEPS part of the program.

WHATIF METHOD

The mission model is analyzed by years. It is assumed that all the payloads to be scheduled in a year are ready and available on the first day of the year. No carryover payloads from one year to the next are allowed either at the beginning or end of a year. This seemingly unimportant assumption in the computerized analysis probably results in STS with SEPS traffic models containing more Shuttle flights than necessary. Many year-end flights were only partially loaded. If payload missions from the next year could have been brought forward, the full capabilities of year-end flights could have been utilized. The assessment of SEPS savings is again conservative by the potential of three to five flights saved..

The payloads are first classified according to the upper stage required for their delivery or retrieval one at a time. The order in which upper stages are considered in this classification can be anything, but it is usually specified in order of increasing performance from no upper stage (Shuttle-alone) through one upper stage, one upper stage plus kick stage, and finally tandem upper stages requiring two Shuttle launches. In this way, each payload is classified by the lowest performance stage that can deliver it. The payloads are then ordered in a list with those requiring the highest performance upper stage at the head of the list. These are followed by the rest of the payloads in order of decreasing upper stage performance ending with those that can be delivered by the Shuttle alone.

The first payload in the list is then loaded into the cargo bay along with the necessary upper stage. An attempt is then made to load each of the following payloads in succession. At each attempt a number of tests are made:

- Is the payload compatible with those in the bay?
- Will it fit in the remaining available volume?
- Can the vehicle deliver it along with others already loaded without exceeding its performance limit?
- Is the Shuttle up or down weight within limits?

If the answer to all of these questions is yes, the payload is added to the flight; and the next payload in the list is considered in a like manner. If any test is failed, the payload is rejected and the next payload is considered. This procedure is continued to the end of the list, each payload being considered in turn.

The flight and the payloads assigned to it are then scheduled. The procedure then returns to the first payload in the list that has not been assigned. It makes up the next flight the same way. When all payloads have been assigned to flights the procedure is repeated on the list of payloads to be scheduled in the next year and so on through the mission model.

Shuttle performance capability used in these calculations is shown on Figures 2-1 and 2-2. These are the Shuttle payload curves from the Shuttle Payload Accommodations document[8] and were in the ground rules for this study. Upper stage performance calculations use impulsive ΔV approximations and idealized rocket equations. Orbit transfer ΔV calculations assume that the line-of-nodes of the orbits are aligned, and if the orbits are elliptical that the line of apsides are along the line-of-nodes. These are the conditions necessary for minimum energy transfer between inclined orbits. The optimism of these assumptions has been mentioned previously. Provisions are made for the calculation of performance for reusable or expendable combinations of reusable and expendable stages for either earth orbit or escape missions. Upper stage performance calculations are limited to no more than two stages (for example, tandem Tugs plus a kick stage cannot be handled).

Payload packaging in the cargo bay can be done in any one of three ways, (1) end-to-end, (2) side-by-side on Shuttle vertical centerline, and (3) three-dimensional. Because of the way study computer programs were derived from complex existing programs, some limitations of the earlier programs remained. We are aware of no places where these limitations made significant differences in STS compared to STS with SEPS.

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[8] *Johnson Space Center, JSC 07700, Volume XIV, Appendix B, Rev. A., "Space Shuttle Payload Accomodations," 16 July 1973.*

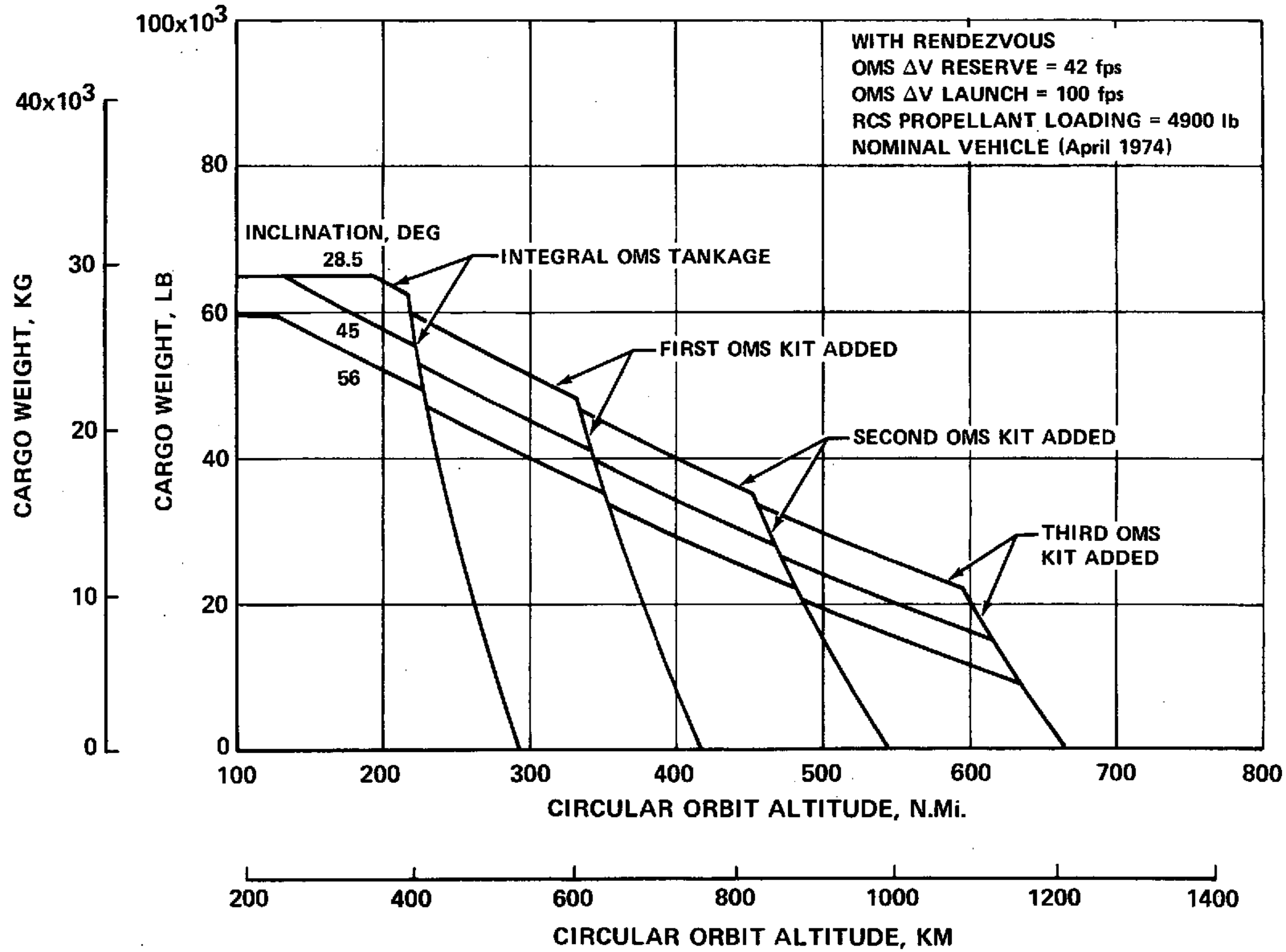

Figure 2-1. SHUTTLE PERFORMANCE - KSC LAUNCH, DELIVERY, AND RENDEZVOUS

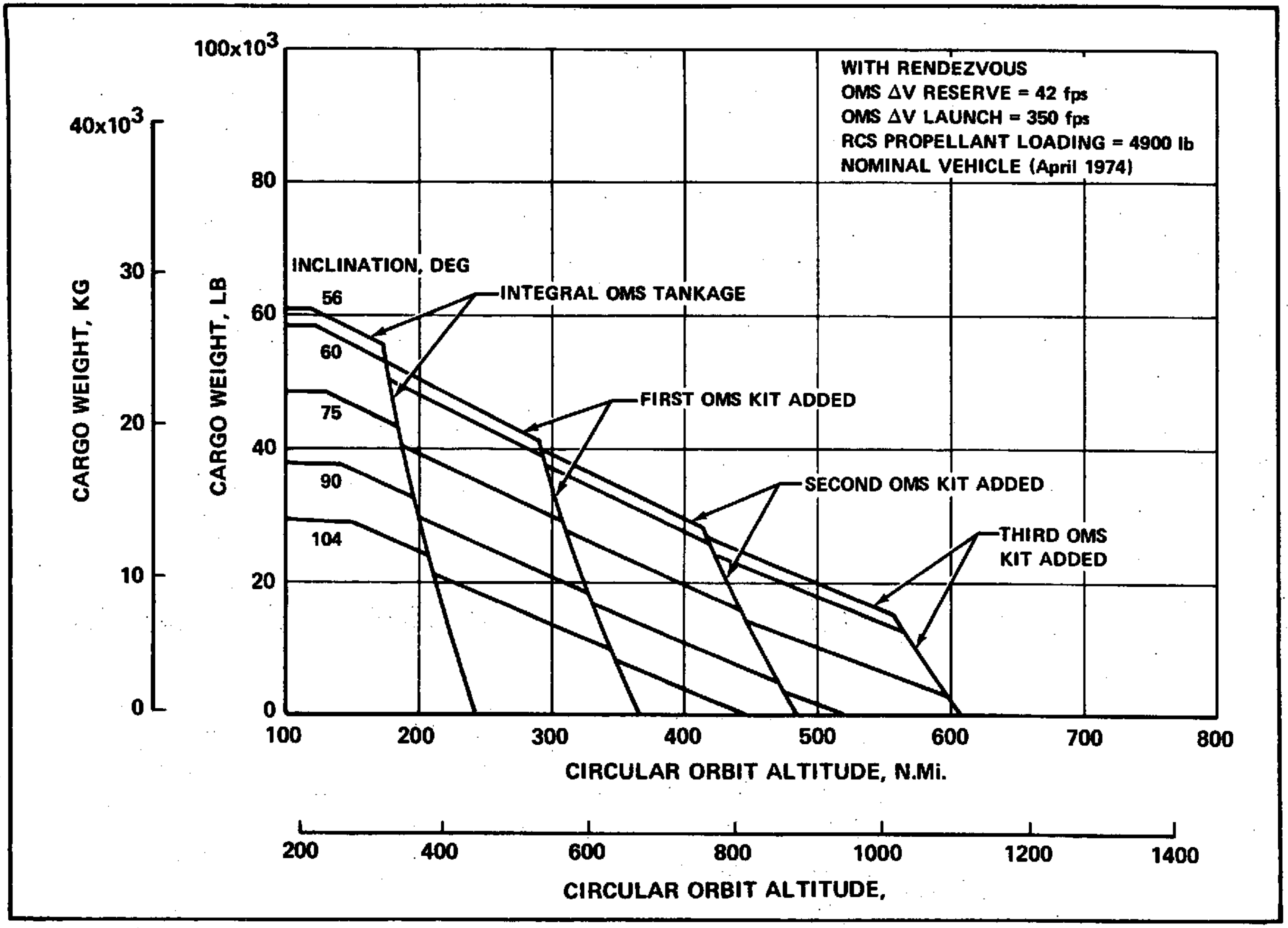

Figure 2-2. SHUTTLE PERFORMANCE - KSC LAUNCH, DELIVERY, AND RENDEZVOUS

When the program makes up cargo manifests for Shuttle flights and Shuttle-Tug flights not involving SEPS, in all cases the maximum number of payloads allowed on one flight (total of up plus down) is six. For example, if there are five up payloads, there will be only one down payload. This constraint applies only to Shuttle and Shuttle-Tug flights assigned by the WHATIF program. SEPS flights are assigned by the SEPS part of the program and the number of payloads per SEPS flight is subject only to the restrictions of cargo-bay volume and the limitations of the various packaging routines. The end-to-end routine can handle six up plus six down, the side-by-side four up plus four down, and the three-dimensional can pack ten up plus ten down. The three-dimensional packaging was added in this study primarily for SEPS where it was felt that the original routines were too restrictive to take advantage of the SEPS capability which is not performance limited. Though added for the SEPS part of the program, the three-dimensional packaging can be used by the WHATIF part, and it is the one area where the original capability was extended. Shuttle cargo center-of-gravity position restrictions are not checked in any of these packing routines. Some control of cg location is possible because of the freedom to rearrange individual packages.

The WHATIF method does not guarantee the minimum number of required Shuttle flights; it is a heuristic attempt to achieve something like a minimum. Experience has shown that changes in the upper stage preference order or a change in the order in which payloads are considered can result in plus or minus one or two flights required over the 11 years of the mission model. When used for trade studies of various STS concepts, differences of one or two flights either way are probably not significant in most instances.

2.4 TUG–SEPS PERFORMANCE AND TRAJECTORY ANALYSIS FOR GEOSYNCHRONOUS PLUS INTERMEDIATE ORBIT TRAFFIC

The Tug-SEPS trajectory profile is shown on Figure 2-3. If an intermediate payload is being delivered, the Tug first transfers from the 160-nautical mile and 28.5-degree parking orbit to the intermediate orbit also at 28.5-degree inclination (by definition of intermediate payloads). The Tug then burns for transfer to the changeover orbit. This burn must be made at the line-of-nodes of the parking and intermediate orbits since the changeover

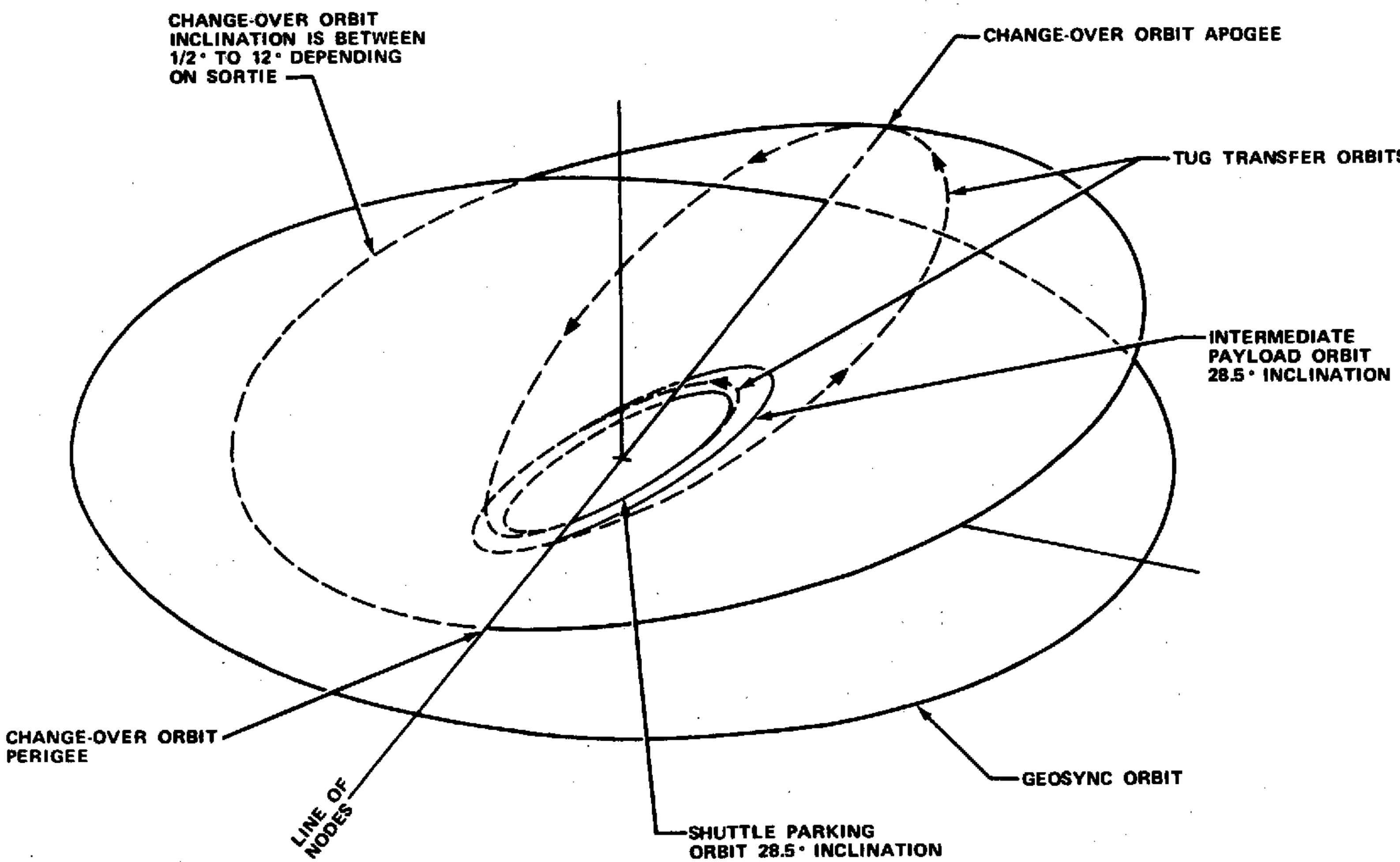

Figure 2-3. TUG-SEPS TRAJECTORY PROFILE

orbit inclination is less than 28.5 degrees, and a plane change is required. To minimize the ΔV for this transfer, some of the plane change is done during this burn, typically 1.5 to 2.0 degrees. If the changeover orbit is elliptical, as shown on Figure 2-3, the minimum energy transfer further requires that the Tug transfer to apogee of the changeover orbit where the remainder of the plane change is done. This means that the line of apsides of the changeover orbit is along the line-of-nodes. In the changeover orbit, Tug and SEPS rendezvous, exchange payloads, and the Tug deboosts to Shuttle parking orbit or to an intermediate orbit if an intermediate payload is being retrieved. SEPS then slowly changes the size and shape of its orbit from changeover orbit to geosynchronous equatorial orbit. If an expendable stage (IUS) is being used it is expended in changeover orbit. This is the trajectory profile that was evolved in this study. The intermediate orbits complicate the following discussion of Tug and SEPS performance calculations, and a discussion of this effect will be deferred until later.

It is known that the ΔV required for orbital transfer with low-thrust vehicles such as SEPS is essentially independent of thrust-to-weight and specific impulse (this is analogous to the impulsive ΔV approximation used for high-thrust vehicles). Thus, the propellant consumption for any SEPS orbital transfer can be calculated as

$$M_p = M_o\left(1 - e^{\frac{-\Delta V_s}{V_{ex}}}\right) .$$

Where M_o is the initial mass, including payload; ΔV is the SEPS ΔV required for the transfer; V_{ex} is the SEPS exhaust velocity ($g_c I_s$). For electric propulsion the propellant flowrate is given by

$$\dot{M} = \frac{2P}{V_{ex}^2}$$

where P is the exhaust beam power in watts and V_{ex} is in m/sec. The burn time required for the transfer is

$$T_b = \frac{M_p}{\dot{M}}$$

in the absence of earth shadow or power degradation due to radiation. It has been found by others and also in this study, that passage of SEPS through the earth's shadow increases trip time by an average of four percent (Figure 3-22, Section III of this volume). In the presence of radiation, power is not constant and calculation of transfer time, assuming constant $\dot{M}$ as above, is not applicable. This will be discussed later.

Since SEPS trip times can be long, it is desirable to use as much ΔV as Tug can provide. When the Tug payload is specified, along with its initial mass, its one-way ΔV capability is

$$\Delta V_{Tug} = - V_{ex_{Tug}} \ \ell n \frac{M_{fu}}{M_o}$$

where

$$M_{fu} = 1/2 \quad \Delta P + \quad \Delta P^2 + 4 \, M_o \, M_f$$

$$\Delta P = Pld_{up} - Pld_{down}$$

$$M_f = Pld_{down} + M_o$$

$$M_D = \text{Tug burnout mass.}$$

This Tug ΔV defines a three-parameter family of changeover orbits to which the Tug can transfer from parking orbit. The optimum changeover orbit is the one characterized by the r_a, r_p, and i that minimizes SEPS ΔV. This SEPS ΔV minimization can be carried out for the range of Tug ΔV's of interest. These results for elliptical and circular changeover orbits will be presented in a later paragraph.

INTERMEDIATE ORBITS

When intermediate payloads are delivered or retrieved by Tug on SEPS sorties, the Tug ΔV cannot be calculated as just described. In addition, the optimum changeover orbits will depend on the particular intermediate orbits involved. That the set of optimum changeover orbits determined for transfers from Shuttle orbit will be nearly optimum for transfer from the intermediate orbits. An iteration is done to find the "highest" (least SEPS ΔV) changeover for each orbit in this optimum set that the Tug can reach after transferring to the intermediate orbits.

SEPS PERFORMANCE CALCULATION WITH RADIATION

Because radiation damage to the solar arrays reduces available thruster power, minimizing SEPS ΔV is not equivalent to minimizing trip time. However, as will be seen in the next paragraph, when SEPS trip time is minimized in the presence of radiation, the optimum changeover orbit parameters and SEPS ΔV are nearly unaffected. Because of the power variation along the trajectory and the resulting variable mass flowrate, the SEPS burn time for a given Tug ΔV is dependent on the thrust-to-mass ratio and the ratio of the initial power at the beginning of the transfer to the degraded power. The change in the accumulated fluence which determines the power degradation during a sortie is also a function of these two parameters. This also depends on whether the transfer is an ascent or descent. Data from MOLTOP trajectories are shown on Figures 2-4 and 2-5 for descent and ascent at a Tug ΔV of 3000 m/sec. Similar data was generated at other Tug ΔVs and parameterized for inclusion in the SEPS part of the WHATIF program. This data allow trip times and power degradation to be calculated for SEPS sorties in the traffic models.

CHANGEOVER ORBIT DATA

Optimum changeover orbits were generated by the MOLTOP computer program. Three kinds of optimum changeover orbits were investigated, (1) circular with a minimum radius of 20,000 kilometers, (2) an elliptical with a minimum perigee radius of 20,000 kilometers, and (3) an elliptical without constraints. The first two were optimized without radiation simulated, the last included simulation of solar cell power degradation due to radiation. The radiation model was supplied by MSFC's Space Sciences Laboratory. Radiation flux in equivalent 1 MEV electrons is shown on Figures 2-6 and 2-7 for two orbit inclinations of interest. This is the flux contribution from one side of the array and was doubled for total flux. This data is treated as instantaneous values of flux along the trajectory and is integrated to obtain the accumulated fluence. Power degradation with accumulated fluence for the 8-mil N/P silicon solar

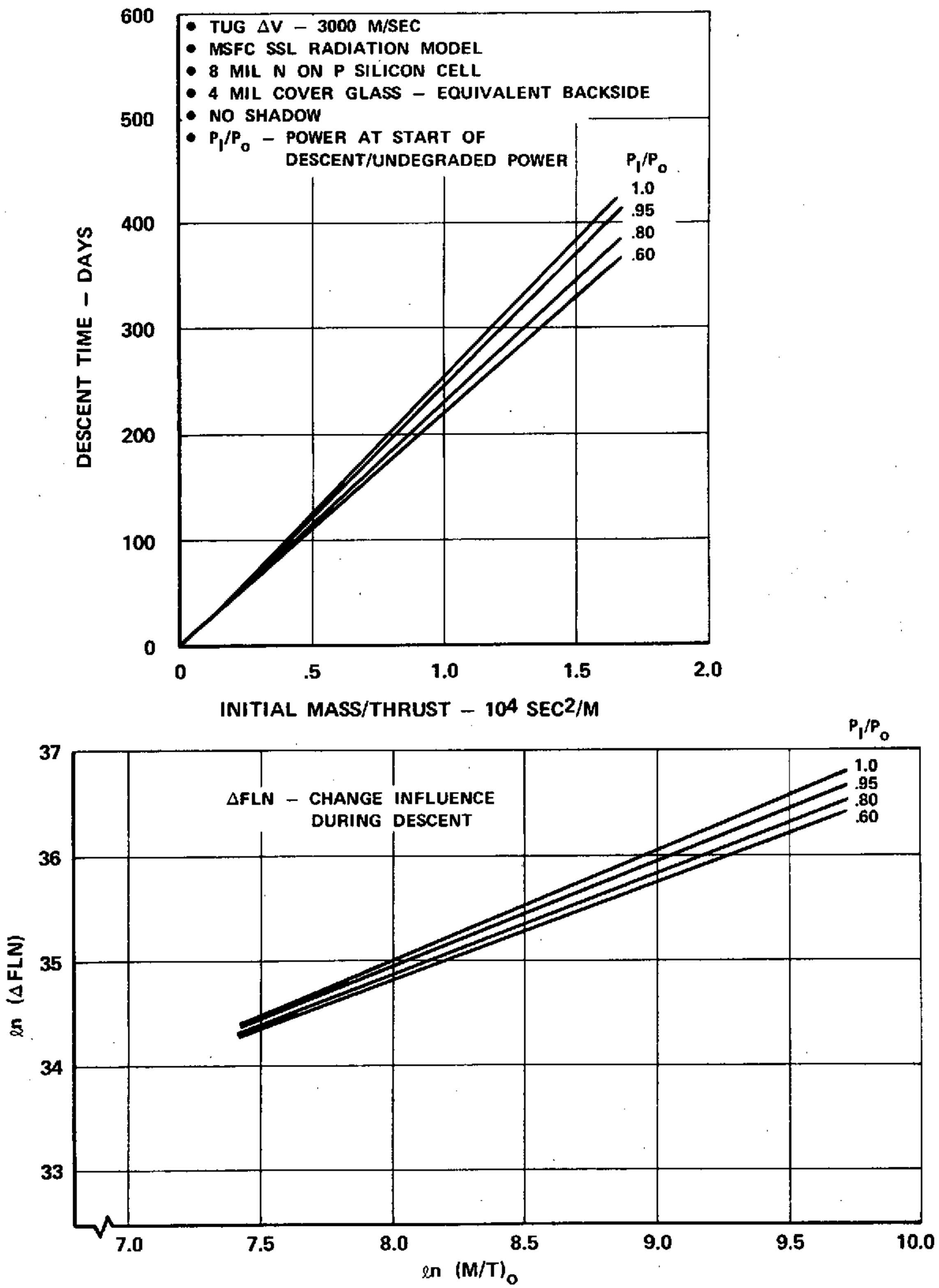

Figure 2-4. SEPS DESCENT TO OPTIMUM CHANGEOVER ORBIT WITH RADIATION

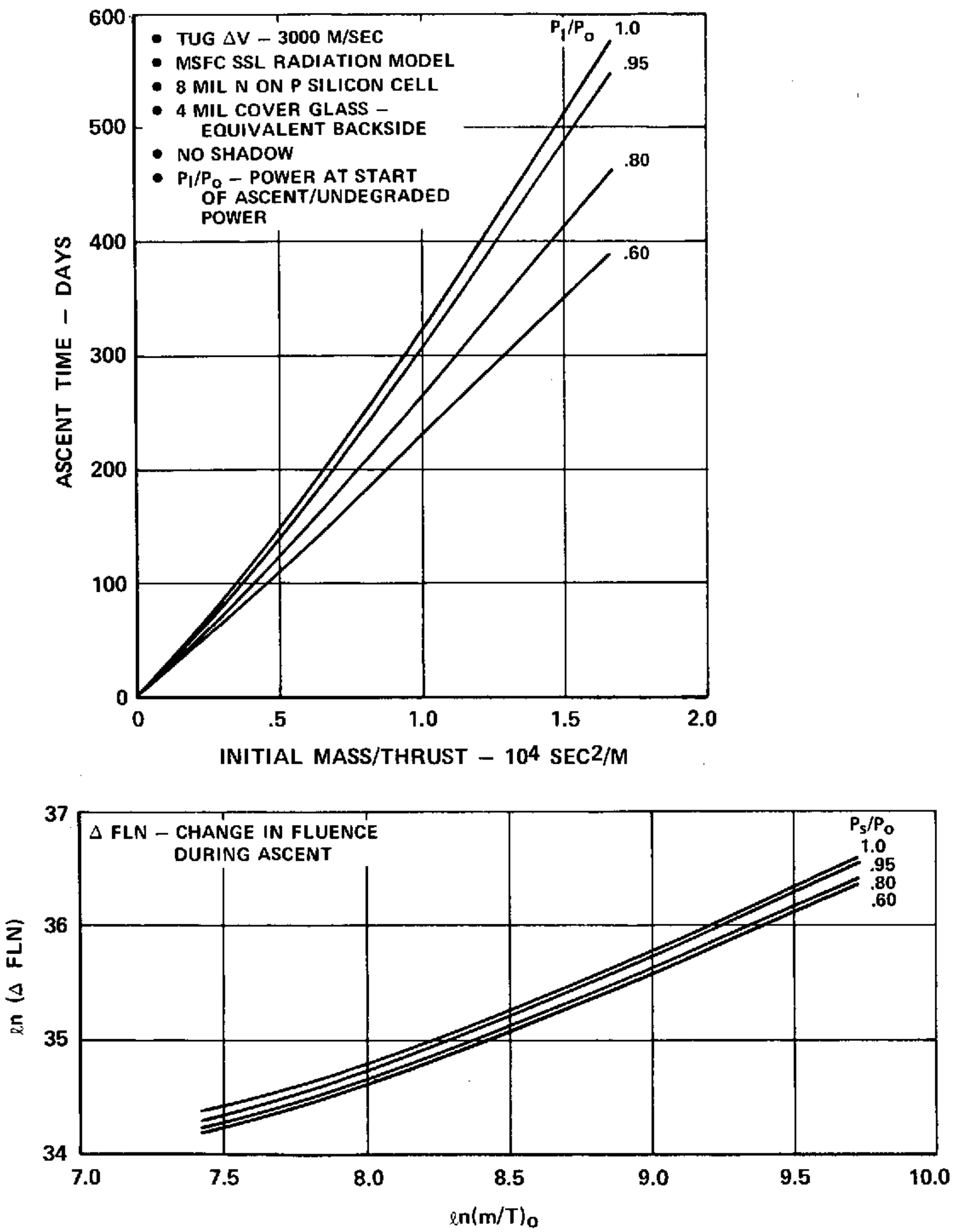

Figure 2-5. SEPS ASCENT FROM OPTIMUM CHANGEOVER ORBIT WITH RADIATION

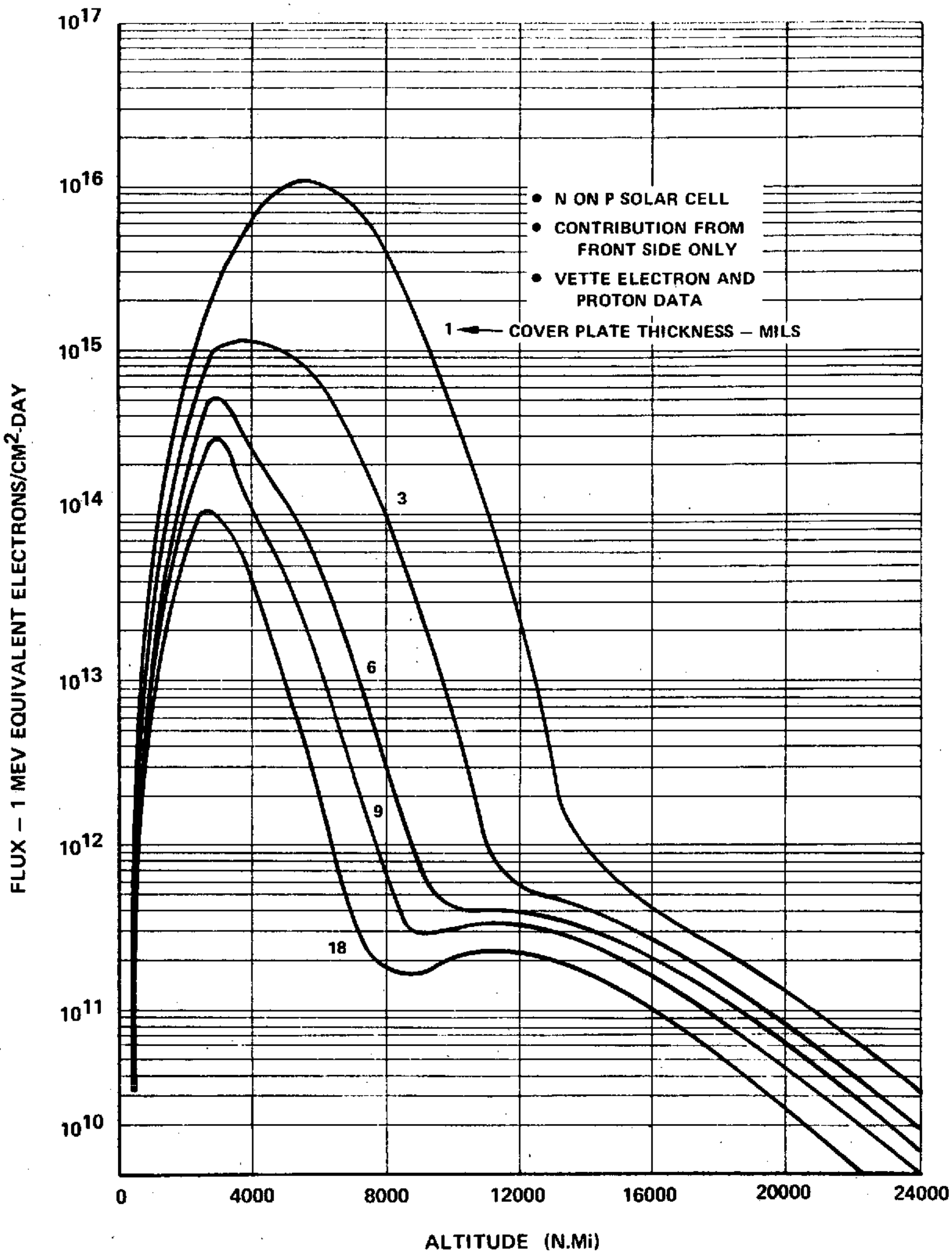

Figure 2-6. TRAPPED RADIATION 0° INCLINATION ORBIT

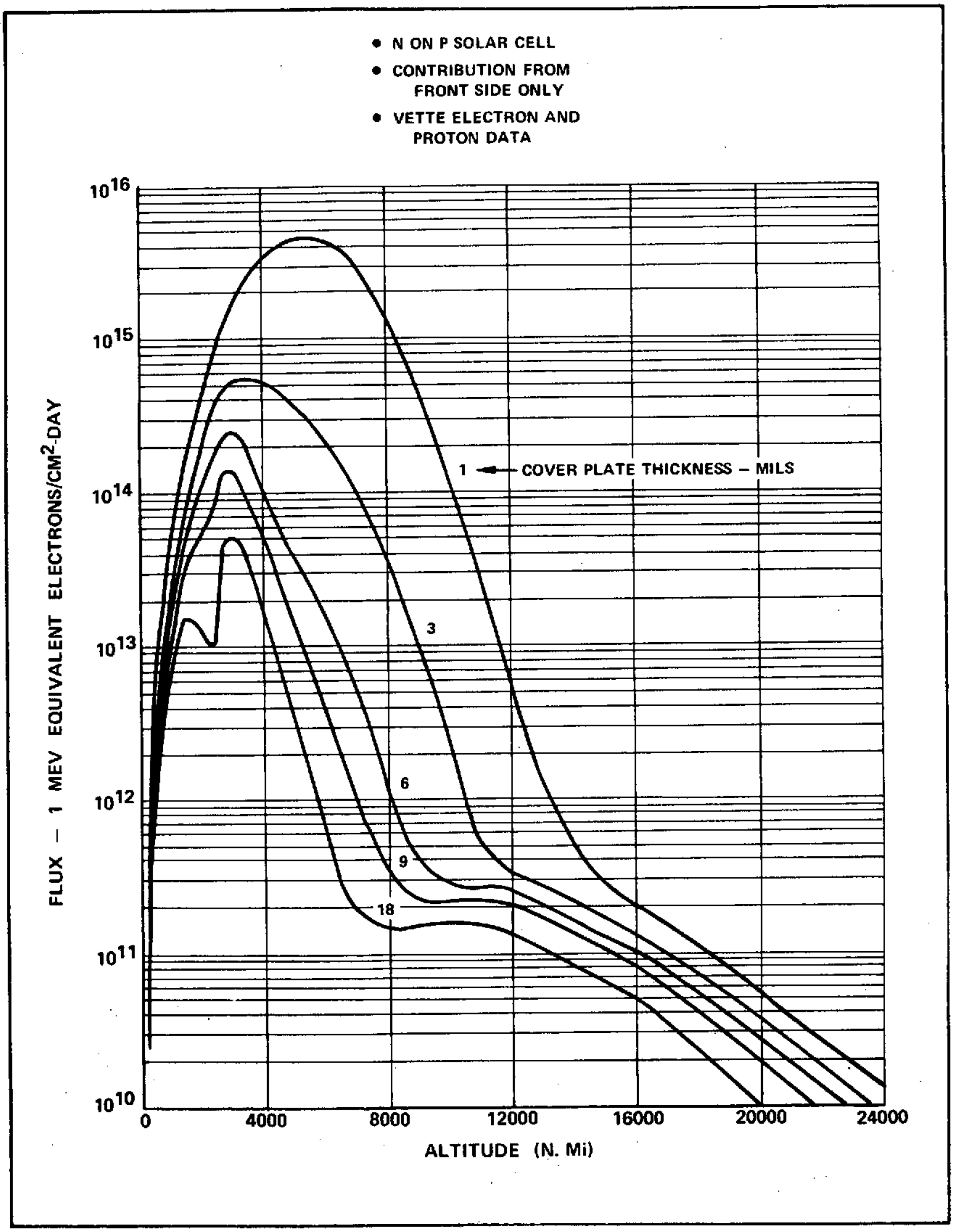

Figure 2-7. TRAPPED RADIATION ENVIRONMENT 30° ORBIT INCLINATION

cell used on the baseline SEPS is shown on Figure 2-8. A 4-mil cover glass plus equivalent backside protection was used.

SEPS ΔV as a function of Tug ΔV is shown on Figure 2-9 for the three kinds of changeover orbits. The sudden upturn in the constrained curves is where the optimization hits the minimum altitude boundary. Changeover orbit parameters are shown on Figures 2-10, 2-11, and 2-12. Notice that the SEPS ΔV savings with elliptical orbits compared to circular orbits are due to the ability of the Tug to make more of the required plane change at the high apogees of the elliptical orbits. Since it is easier for SEPS to raise the perigee altitude than make a plane change, the ellipitical orbits require less SEPS ΔV for a given Tug ΔV.

SEPS PROGRAM METHOD

The SEPS part of the program assigns payloads to SEPS sorties using the method of the WHATIF program, except that the payloads are restricted to geosynchronous payloads. After all possible geosynchronous payloads have been assigned, the program tries to add intermediate payloads. These payloads are delivered in order of increasing altitude and are retrieved in order of decreasing altitude. Tug ΔV is calculated as previously described. Four SEPS modes are provided: (1) new SEPS launch; (2) normal down-up sortie: (3) deliver a new SEPS, retrieve an onorbit one (the exchange mode); and (4) refuel. A new SEPS is launched anytime there are none available onorbit (they are all busy) and the traffic requires it. SEPS are exchanged when an onorbit SEPS has been refueled the maximum number of times or it has exceeded its five year onorbit lifetime or the maximum thruster life has been exceeded. Refueling is done a specified number of times that the need to refuel is determined (by comparing propellant remaining at the end of a sortie with the average propellant consumption per sortie for the particular SEPS since its last refueling). When the propellant remaining is less than the average, the SEPS is scheduled to be refueled on its next trip to changeover orbit. Shuttle-Tug launch dates are assigned for each SEPS sortie. In this program, it is assumed that all payloads are launched within the year if the Shuttle was launched within the year. The SEPS and payload may not reach geosynchronous orbit until sometime the next year.

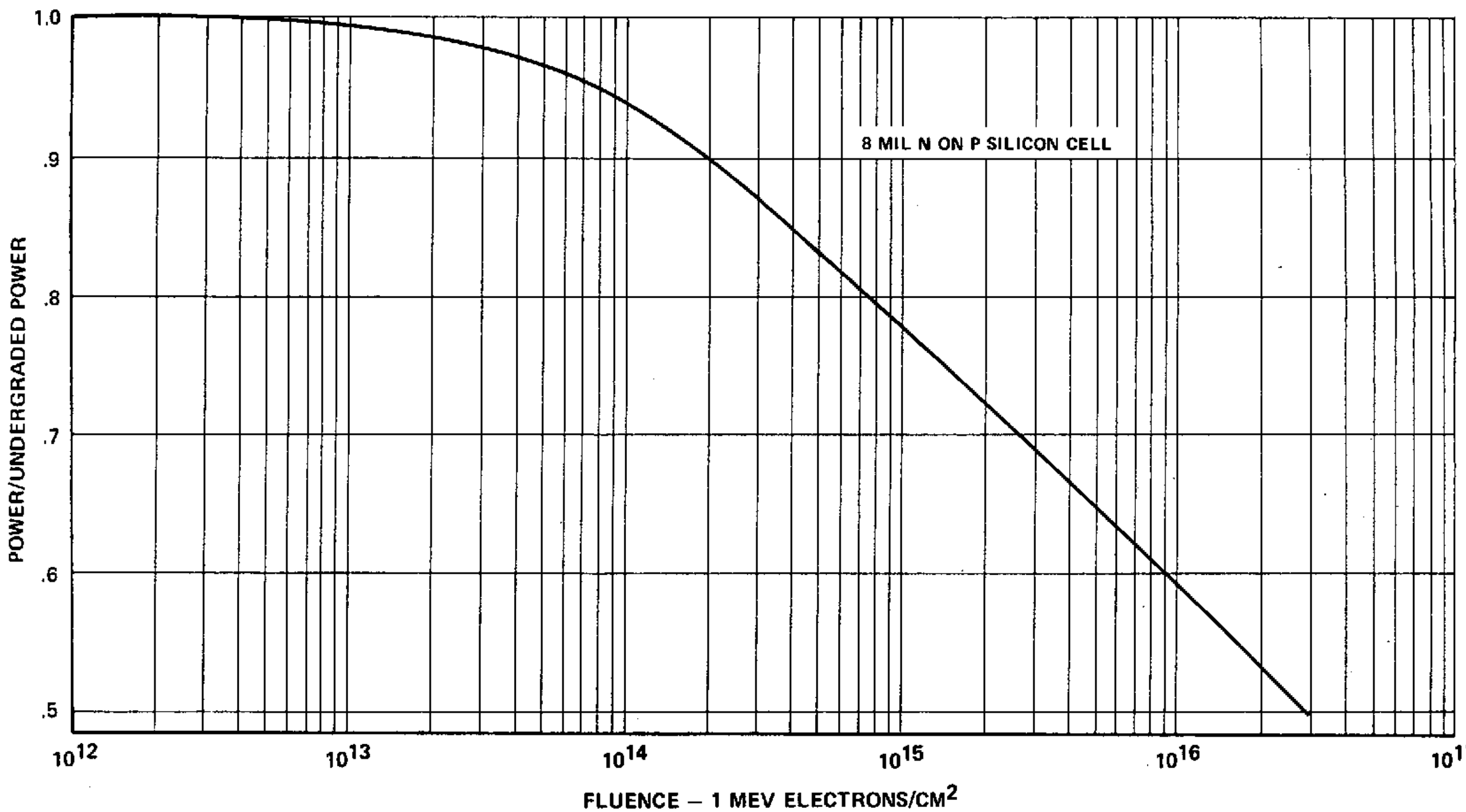

Figure 2-8. SOLAR CELL POWER DEGRADATION

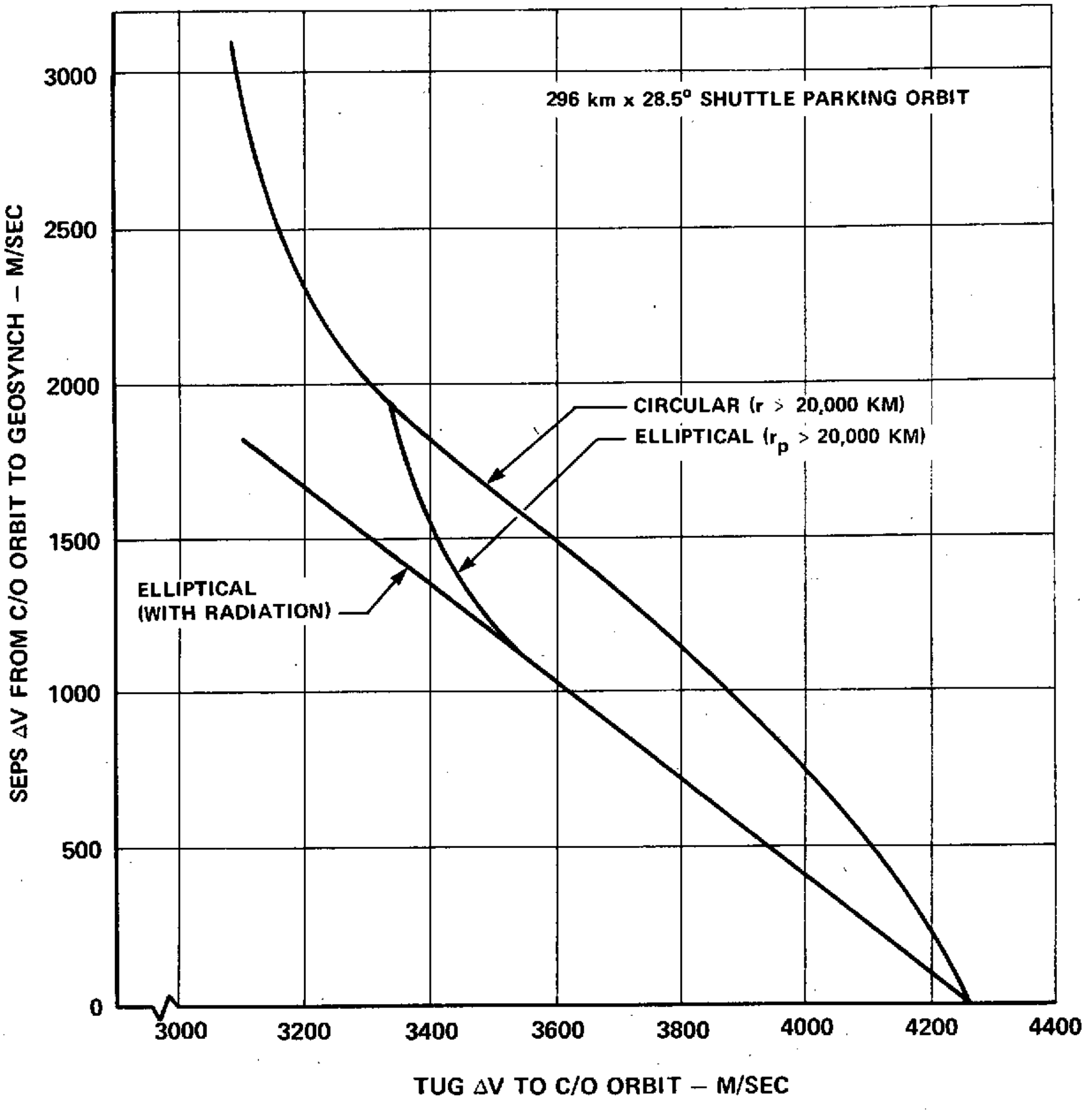

Figure 2-9. SEPS ΔV FOR OPTIMUM CHANGEOVER ORBITS

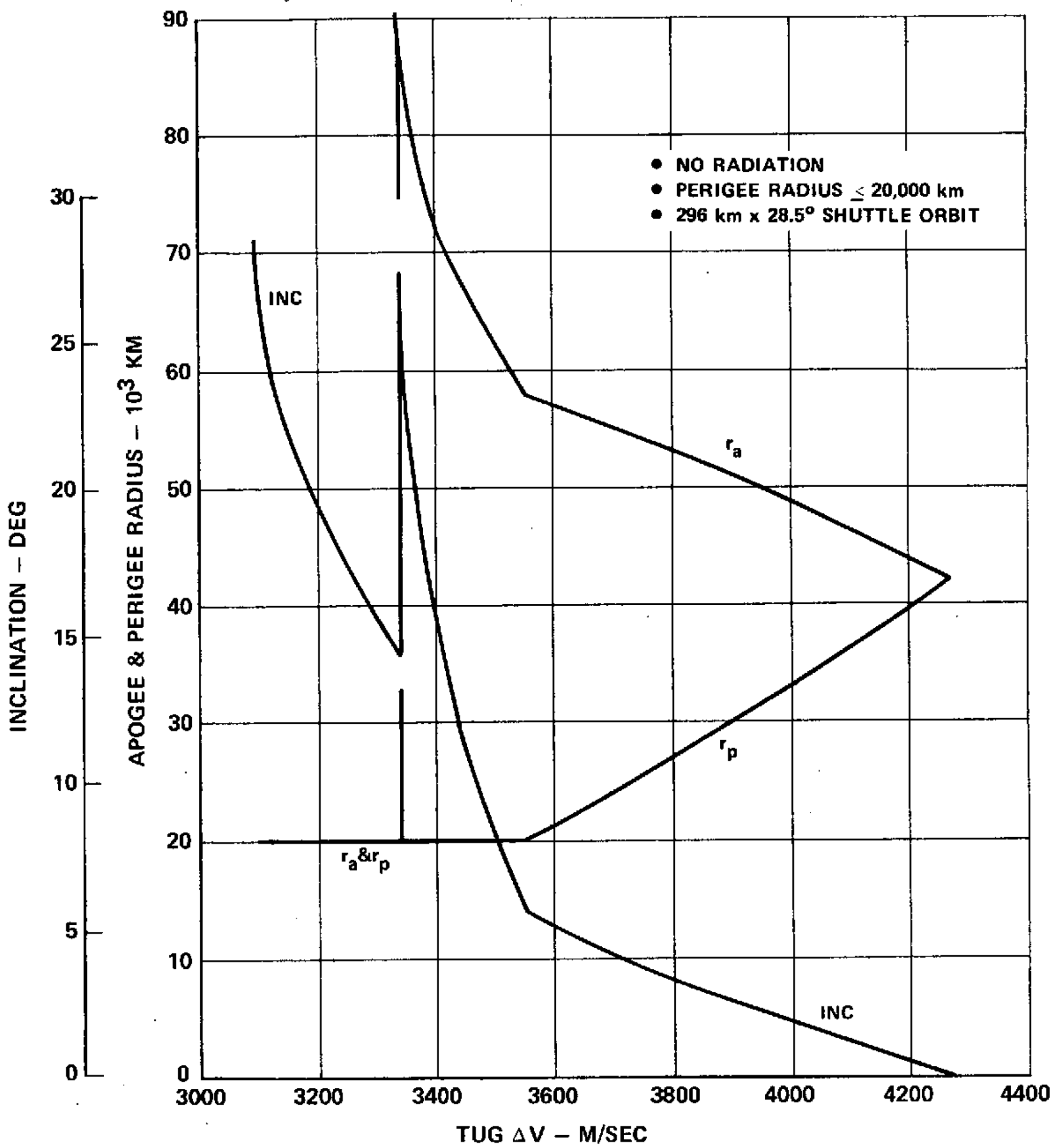

Figure 2-10. OPTIMUM ELLIPTICAL CHANGEOVER ORBITS WITHOUT RADIATION

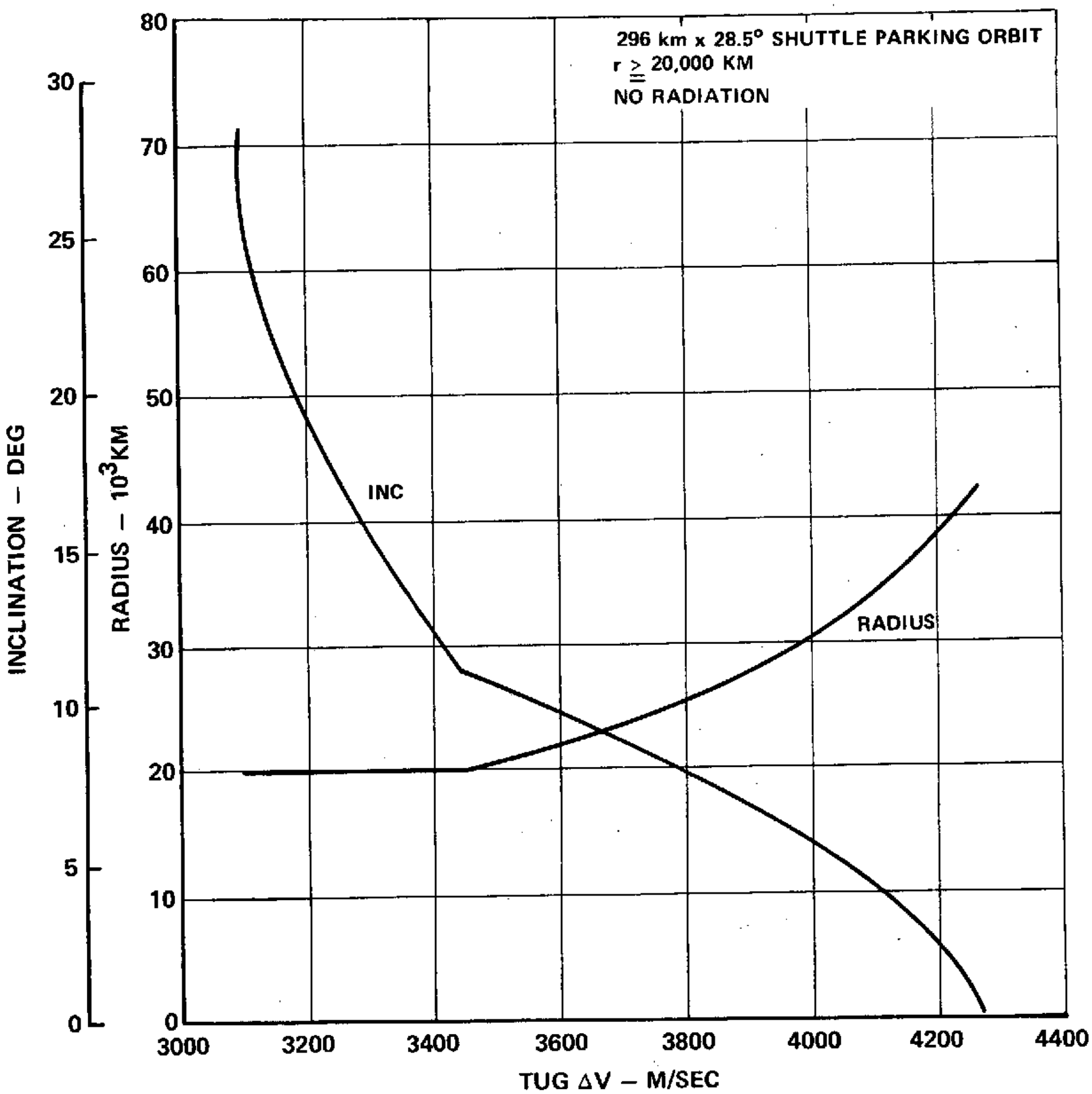

Figure 2-11. OPTIMUM CIRCULAR CHANGEOVER ORBITS WITHOUT RADIATION

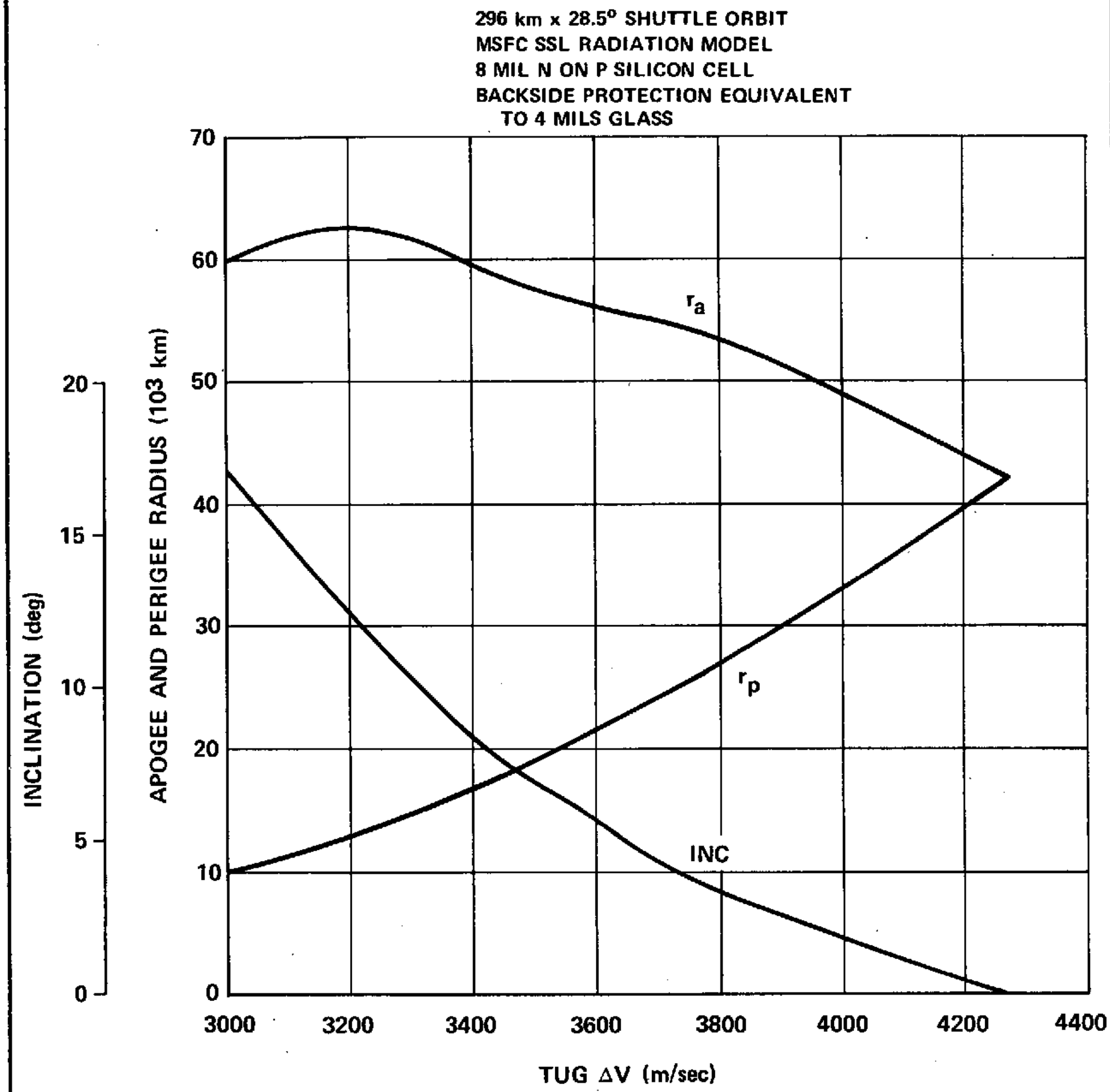

Figure 2-12. OPTIMUM ELLIPTICAL CHANGEOVER ORBITS WITH RADIATION

Payloads that cannot be combined to require a SEPS (that is the changeover orbit is equal to or greater than geosynchronous orbit) are returned to the WHATIF program to be scheduled on Shuttle and Tug flights. This occurs when there are not enough payloads left in the year to make up a SEPS sortie, or they cannot be packaged densely enough to exceed Tug payload delivery capability.

ORBITAL TAXI LONGITUDE SHIFT PERFORMANCE

The time and propellant required for SEPS to make longitude shifts in geosynchronous orbit are computed as shown on Figure 2-13. This data is based on data contained in Rockwell International[9] and NSI[10] studies. Figure 2-14 shows longitude shift times for the upper and lower extremes of SEPS thrust-to-weight ratios. Sortie trip times are based on payload longitudes shown on Table 2-7.

2.5 TRAFFIC MODEL RESULTS

Traffic models with SEPS in geosychronous mission role were generated for several Tug and SEPS configurations. Traffic models were also generated without SEPS to provide a reference for comparisons which would show the effectiveness of SEPS in the transportation system. Study ground rules specified that an expendable Interim Upper Stage (Transtage) would be used from 1981 through 1983, and the high-performance reusable Tug from 1984 through 1991. Weight and performance data for the IUS and 9.1 meter Tug baselined for the study are listed in Table 2-8. Also shown in this table are data for three other Tug configurations that were investigated. The 9.1 meter ΔRL-10 Tug is the baseline Tug with a lower performance and lower development cost engine. The 7.6 meter and 6.4 meter toroidal tank Tugs are compact high-performance Tugs based on design studies by General Dynamics.

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[9] *Rockwell International Corporation Report SD 72-SA-0199-2-1, "Feasibility Study of a Solar Electric Propulsion Stage for Geosynchronous Equatorial Missions," 23 February 1973.*

[10] *Greenleaf, W. G., "Solar Electric Propulsion Stage Geosynchronous Terminal Rendezvous Geometry, Propulsion, and Guidance Compatibility Analysis," Northrop Services, Inc., Huntsville, Alabama, Memorandum M-240-1215, May 1973.*

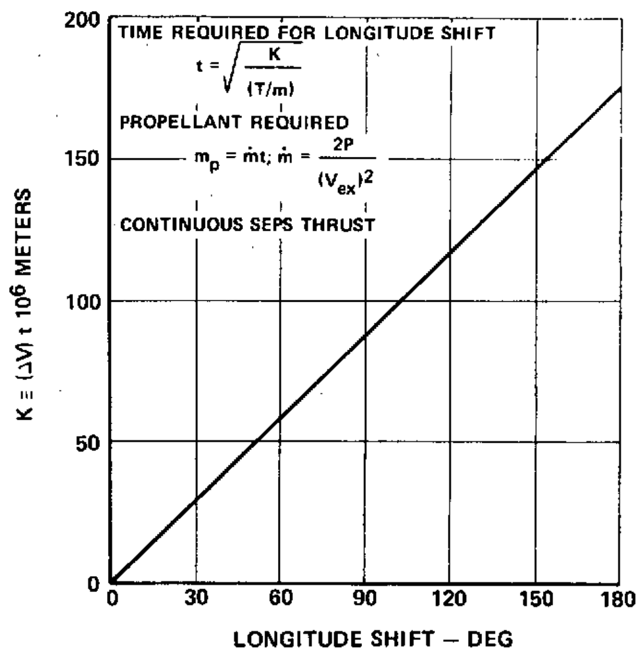


Figure 2-13. TIME AND PROPELLANT REQUIRED FOR STS GEOSYNCHRONOUS LONGITUDE SHIFT

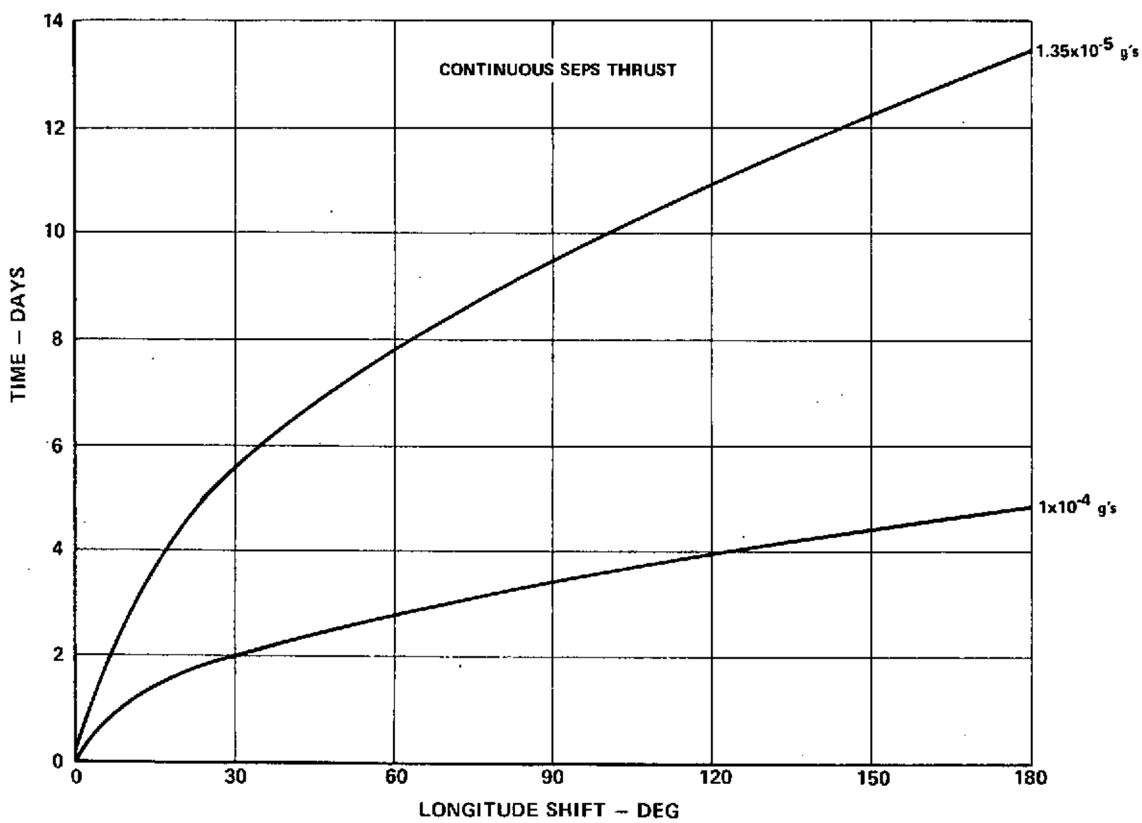


Figure 2-14. SEPS GEOSYNCHRONOUS ORBIT LONGITUDE SHIFT

Table 2-8. TUG AND IUS PERFORMANCE DATA

PARAMETER \ CONFIGURATION	IUS (Transtage)	9.1 METERS BASELINE TUG	9.1 METERS TUG ΔRL-10	7.6 METERS TOROIDAL TANK	6.4 METERS SHORTENED TOROIDAL TANK
Drop Wt-kg	2116.	2747.	2747.	2883.	2784.
Usable Prop. WT-kg	14586.	23008.	23008.	24329.	18641.
Specific Impulse-sec (effective)	308.2	449.0	430.8	449.4	447.2
Thrust-kg	7258.	6804.	6804.	6804.	6804.
Length-m	5.85	9.14	9.14	7.62	6.40
Shuttle Interface Wt-kg	1361.	862.	862.	862.	862.

An earth-orbital SEPS configuration had been evolved in earlier studies by Rockwell International. By NASA direction, this configuration was taken as the baseline SEPS for this study. This SEPS had a 25 kw solar array and nine thrusters; it used eight at a time with a 10,000-hour life each, giving it a maximum total thrust time of 11,250 hours. Shortly after the beginning of this study, the baseline thruster lifetime was increased to 20,000 hours in view of the results from the thruster technology program tests.

In-space refueling of SEPS was selected because of its advantages in trip time savings and the potential savings in the Shuttle flights. Reduction in trip times occur because of the smaller average propellant load.

Performance and summary weight data for the original 25 kw configuration and the baseline 25 kw SEPS are shown in Table 2-9a. Data for three higher power SEPS investigated in this study are also shown in Table 2-9a. Table 2-9b provides a weight breakdown of these SEPS variants. Screen power for the thrusters is taken directly from the solar array in these three configurations. The higher efficiency of these SEPS is due to the elimination of power processing losses for screen power. The 50 kw configuration with higher specific impulse gets an additional boost in efficiency because of increased thruster efficiency at the higher screen voltages used to power the higher specific impulse. The 100 kw configuration is equipped with radiation resistant cells that degrade to about 85 percent of their new output and then remain at this level due to their self-annealing property. This configuration was called upon to operate through the radiation belts, and it was assumed that its average solar array power for performance calculations was 85 kw.

Table 2-9a. SEPS PERFORMANCE DATA

PARAMETER \ CONFIGURATION	25 kw 10000 hr Thrusters Not Refueled	25 kw Baseline SEPS	50 kw BL Isp SEPS	50 kw 4158 sec BL Isp SEPS	100 kw SEPS
Power to Thruster – kw Subsystem (Undegraded)	24.	24.	49.	49.	85.*
Overall Efficiency	0.649	0.649	0.691	0.766	0.691
Beam Power – kw (Undegraded)	15.58	15.58	33.86	37.53	58.74*
Specific Impulse – sec	2,940	2,940	2,940	4,160	2,940
Empty Weight – kg	1,256	1,243	1,743	1,552	3,043
Propellant Capacity – kg	1,520 (Not refueled)	771 (Refueled 3X)	1,542 (Refueled 3X)	817 (Refueled 3X)	1,631 (Refueled 3X)
Length – m	2.59	2.59	3.66	3.66	8.63
Max Thrust Time – hrs	11,250**	22,500**	20,000	20,000	20,000

**Minimum power – Fully degraded radiation resistant cells*
***1 spare thruster*

Table 2-9b. SEPS WEIGHT BREAKDOWN

CHARACTERISTIC \ VEHICLE	25 kw SEPS BL 2940 Isp	50 kw SEPS SPSA 2940 Isp	50 kw SEPS SPSA 4158 Isp
Thrusters and Related Elements	154	274	137
Thruster Power Processing	165	74	131
RCS	24	35	35
Solar Arrays, Solar Power Distribution & Related Elements	428	855	855
Energy Storage & Distribution	82	82	82
Thermal Control (other than for Power Processors)	10	10	10
Guidance and Navigation	44	44	44
Command Computer	11	11	11
Communications	61	61	61
Data Storage	15	15	15
Hg Propellant System	39	39	39
Mechanisms and Structures	41	41	41
Structures Associated with Launch Interface Loads	39	47	44
Docking and Manipulation	87	87	87
Miscellaneous	7	7	7
Dry SEPS Weight	1209	1694	1519
Mercury Propellant	907[1]	907[1]	907[1]
N_2H_4	66	66	66
Wet SEPS Weight	2181	2665	2492
Ref Isp	2940	2940	4158

[1] *For refueled space operation basic tank capacity was scaled from this value*

The traffic model for the baseline Tug was generated using three-dimensional packaging without limiting the number of payloads on Tug flights to three up and one down. The number of flights in this model, which was to serve as the reference throughout the study, are shown in Table 2-10. A total of 452 Shuttle flights are required by the mission model from 1981 to 1991. Upper stages are necessary on 136 flights. The column headed "OTHER" in this table are flights required by the 28.5 degree intermediate and high-energy missions and the 55 degree missions. Since SEPS in the geosynchronous mission role does not affect the number of Shuttle-only flights, traffic moldel results for the various STS configurations are compared using the number of required upper stage flights. The PL-18 planetary payload in 1981 could not be scheduled by the WHATIF program since it requires more than two upper stages for delivery. The two flights for this payload are not included in the traffic model results for any of the STS configurations. In order to reduce computer run times and establish gross effects, the initial evaluation of the STS configurations was done using the following simplified ground rules:

1. Payloads packaged three-dimensionally with no prespecified limit on the number per Tug flight.
2. Elliptical changeover orbits on SEPS sorties with no constraint on minimum perigee altitude and radiation effects included.
3. SEPS trip time limited to less than 90 days per leg, 180 days maximum sortie trip time.
4. SEPS configurations with 20,000 hour thrusters refueled three times.
5. Intermediate payloads not delivered by Tug on SEPS flights.

The effect on the model of each of these ground rules will be discussed later.

Upper stage flights from the traffic models for each configuration investigated are shown in Table 2-11. With the assumed ground rules, the 25 kw baseline SEPS used with the 9.1 meter baseline Tug could save ten flights. The maximum number of flights saved was with the 7.6 meter Tug and 50 kw SEPS. The ΔRL-10 Tug could not deliver one PL-23 payload in 1990 and 1991 under the simplified rules. This is in addition to the PL-18 which cannot be delivered by the IUS in 1981. Because of the length of the PL-23 payload, three Shuttle flights would be required for its delivery with the ΔRL-10 Tug. In the 6.4 meter and 7.6 meter Tug combination configuration, planetary payloads were

Table 2-10. TRAFFIC MODEL FOR 9.1 METER BL TUG WITHOUT SEPS
NUMBER OF FLIGHTS,1981-1991

YEAR	UPPER STAGE FLIGHTS				SHUTTLE-ONLY FLIGHTS		TOTAL
	ESCAPE	GEOSYNC	POLAR	OTHER	AUTOMATED	SORTIE	
81	4*	3	-	1	2	17	27
82	1	2	-	-	3	19	25
83	6	5	-	-	7	21	39
84	5	5	2	3	4	23	42
85	10	4	1	1	6	27	49
86	10	6	1	1	6	26	50
87	6	5	1	2	5	25	44
88	2	6	1	-	4	29	42
89	3	6	2	1	3	27	42
90	5	8	2	2	4	26	47
91	6	4	2	1	5	27	45
TOTAL	58	54	12	12	49	267	452
	Total upper stage flts = 136						

* *NOTE: Payload PL-18 in 1981 cannot be delivered by tandem expendable IUS in 2 shuttle flights. This payload requires tandem IUS + kickstage in 2 shuttle flights*

Table 2-11. STS CONFIGURATION TRADES

SEPS \ TUG	NUMBER OF UPPER STAGE FLIGHTS, 1981–1991				
	9.1M BL TUG	9.1M TUG (ΔRL–10)	7.6M TUG	6.4M & 7.6M TUGS	REUSABLE TRANSTAGE
NO SEPS	136	150*	139	–	–
25 KW SEPS 10 KHR THRUSTERS	127	127*	–	133	–
BL SEPS 20 KHR THRUSTERS	126	–	123	133	–
50 KW SEPS 20 KHR THRUSTERS	124	–	122	125	–
50 KW SEPS Isp = 4,160 20 KHR THRUSTERS	124	–	122	126	–
100 KW SEPS 20 KHR THRUSTERS	–	–	–	–	138**

- 90-DAY TRIP TIME LIMIT FOR SEPS
- ELLIPTICAL CHANGEOVER ORBITS, PERIGEE ALTITUDE NOT CONSTRAINED
- RADIATION EFFECTS INCLUDED
- SEPS CONFIGS. WITH 20 KHR THRUSTERS REFUELED 3 TIMES
- INTERMEDIATE ORBIT PAYLOADS NOT DELIVERED ON SEPS FLIGHTS

**PLD PL–23 Jupiter Satellite Orbiter Lander could not be delivered.*
***Requires tandem transtage + kickstage for some planetary PLDs, PL–8 and PL–23 could not be delivered in 1990 and 1991.*

delivered with the 7.6 meter Tug, and all other payloads were delivered with the 6.4 meter Tug. This configuration would have made a considerably better showing if the 6.7 meter Tug were used only for SEPS sorties.

The reusable transtage is the IUS used in a recovered mode. Because of the limited performance of this stage and the large weight of the 100 kw SEPS used with it, changeover orbits were at relatively low altitudes with perigees practically at Shuttle parking orbit altitude. This system was configured with radiation resistant solar cells to maintain its power level when operating through the Van Allen belt. Tandem expendable transtages plus a kickstage are required for delivery of the majority of the planetary payloads, and one PL-8 and PL-23 payload in each of the years 1990 and 1991 could not be delivered by this system with two Shuttle flights. In order to achieve the 138 flights with this system, SEPS is required to deliver the 55 degree payloads and the 28.5 degree intermediate and high-energy payloads in addition to performing its geosynchronous mission role.

The impact of operational modes and constraints as reflected by the ground rules on traffic model flight requirements was investigated. The effect of increasing the trip time limit to 180 days per leg is shown in Table 2-12. Since the trip time constraint limits the number of payloads that can be carried on a sortie, it is expected that increasing the allowed trip time will result in a reduction of flights. Table 2-12 shows that the baseline configuration is not significantly constrained by the 90-day limit. However, the 6.4-meter Tug configuration which provides more room in the cargo bay for payloads would benefit with longer allowed trip times or by a SEPS with even more power than the 50 kw SEPS.

A parametric investigation of the trip time benefits of higher power SEPS was conducted. The payload delivery and retreival capability of the 9.1 meter baseline Tug is shown on Figure 2-15. Payload weights carried on SEPS sorties taken from the traffic model of the 25 kw baseline SEPS with this Tug are spotted on the plot. These sorties lie primarily in the region between Tug ΔV's of 3,400 to 4,200 meters per second. Tug ΔV capability determines the changeover

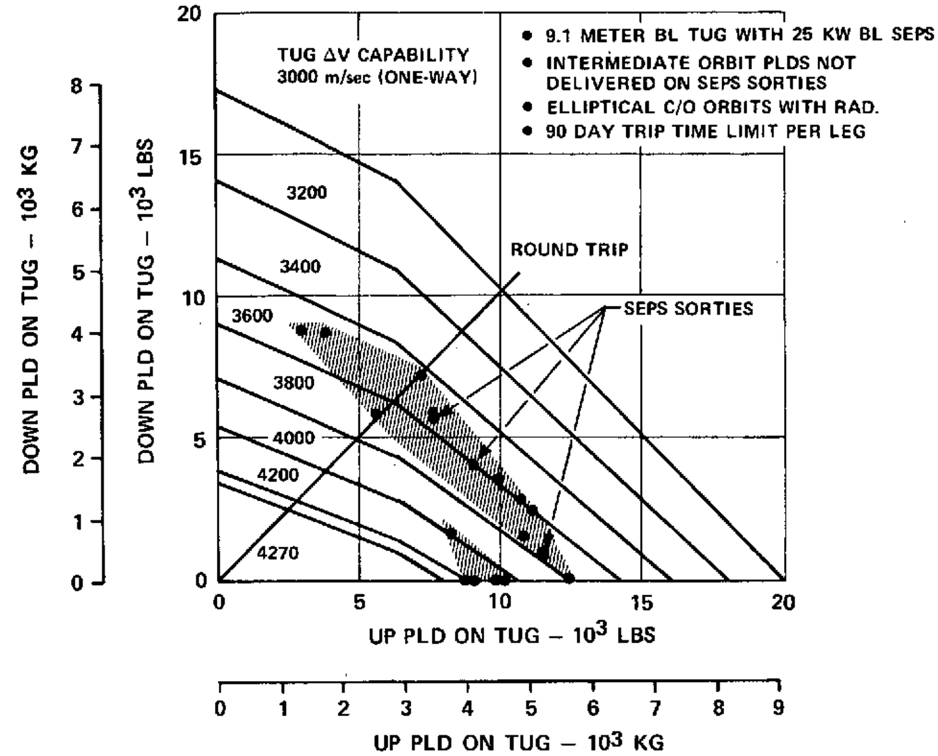


Figure 2-15. TUG C/O ORBIT PERFORMANCE

Table 2-12. TRIP TIME LIMIT COMPARISON, NUMBER OF UPPER STAGE FLIGHTS, 1981-1991

CONFIGURATION		UPPER STAGE FLIGHTS REQUIRED MAX TRIP TIME/LEG	
TUG	SEPS	90 DAYS	180 DAYS
9.1 m	25 kw BL	126	125
7.6 m	50 kw 4158 sec Isp	122	121
6.4 m 7.6 m	50 kw 4158 sec Isp	126	121

NOTES: 1. *Elliptical changeover orbits with radiation.*
2. *SEPS refueled 3 times.*
3. *Intermediate orbit payloads not delivered on SEPS flights.*
4. *3-D packaging without 3 up-1 down limit on Tug.*

orbit and, hence, the SEPS ΔV required from changeover orbit to geosynchronous orbit. SEPS ΔV and the payload weights then determine the SEPS sortie trip time. Sortie times are plotted on Figure 2-16 for the 25 kw baseline SEPS and 50 kw 4158 BL Isp SEPS when used with the baseline Tug. SEPS sorties for the 25 kw baseline SEPS fall within the shaded areas. The curves are plotted neglecting solar array power degradation due to radiation. The fact that some of the actual SEPS sorties lie above the curves indicates trip time increase caused by power degradation.

Sortie trip time savings with the 50 kw SEPS are shown on Figure 2-17. Trip time reductions of 25 to 55 days are possible in the region of most frequent SEPS operation with the higher power SEPS configuration.

The type of changeover orbit determines the SEPS ΔV required for transfer from geosynchronous orbit to changeover orbit and back. These ΔV's are shown on Figure 2-9 for the three kinds of changeover orbits considered in this study. That figure shows that elliptical changeover orbits require significantly less SEPS ΔV than circular, particularly in the region of most frequent SEPS operation. Since trip time is determined by the required SEPS ΔV, it would be expected that the use of elliptical changeover orbits would allow more payloads to be delivered per SEPS sortie within the trip time limit. A reduction in the number of flights should be the result. Upper stage flights required by traffic models generated with the three different kinds of changeover orbits are shown in Table 2-13. Table 2-13 shows that the type changeover orbit has little effect on the number of Shuttle flights. This is because these configurations are not constrained by the 90-day trip time limit as was shown in Table 2-12. In a mission model more demanding of SEPS capability, shorter trip time limits would be desired to reduce the number of onorbit SEPS required to handle the traffic in high volume years. In this case, trip time limits much less than 90 days would constrain these configurations and result in an increase in flights.

Sortie trip time reductions with elliptical changeover orbits are shown on Figure 2-18 for the baseline configuration. Recall that the majority of

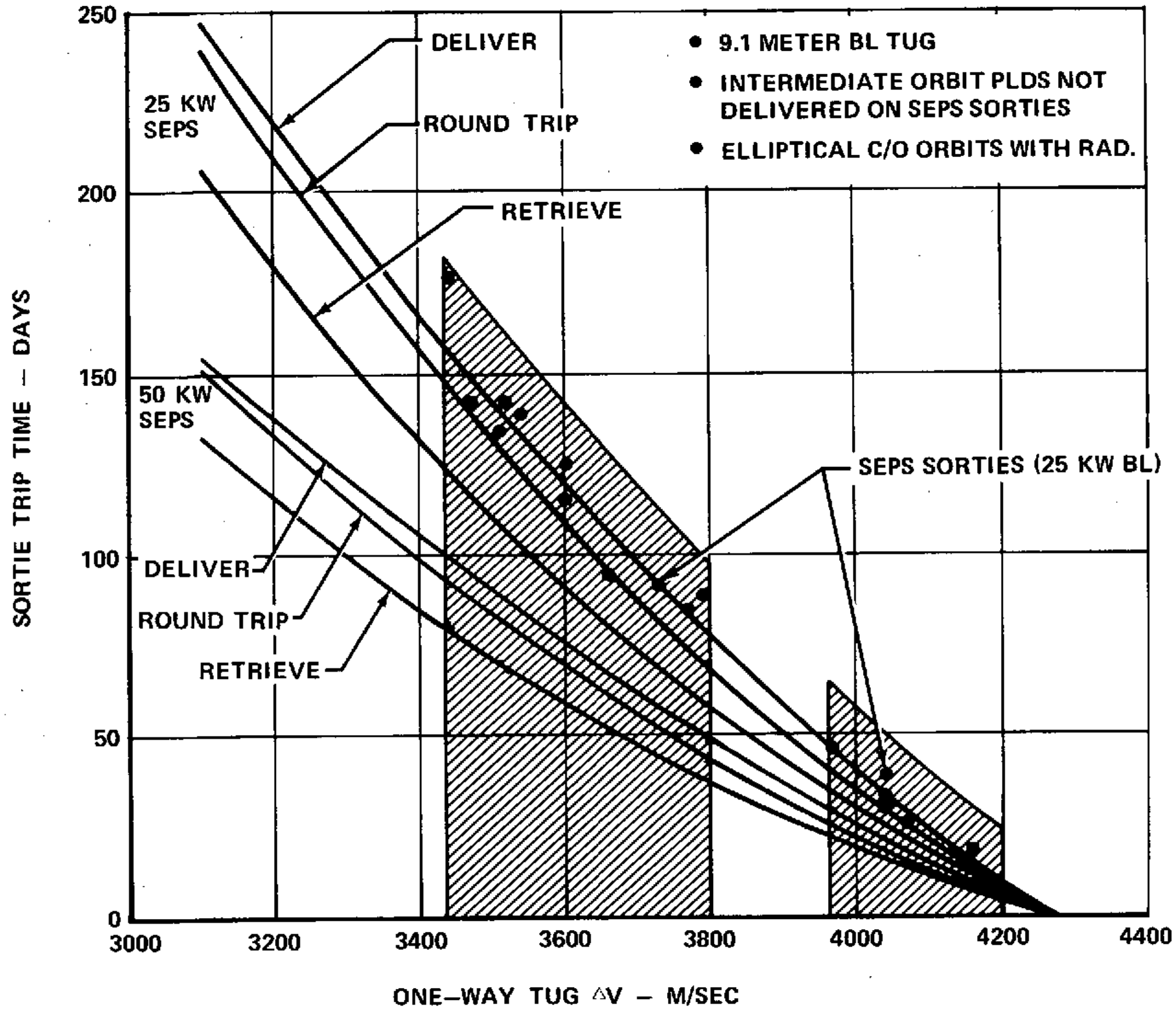


Figure 2-16. SORTIE TRIP TIMES REQUIRED BY 25 KW SEPS AND 50 KW SEPS IN CONJUNCTION WITH A 9.1-METER BL TUG

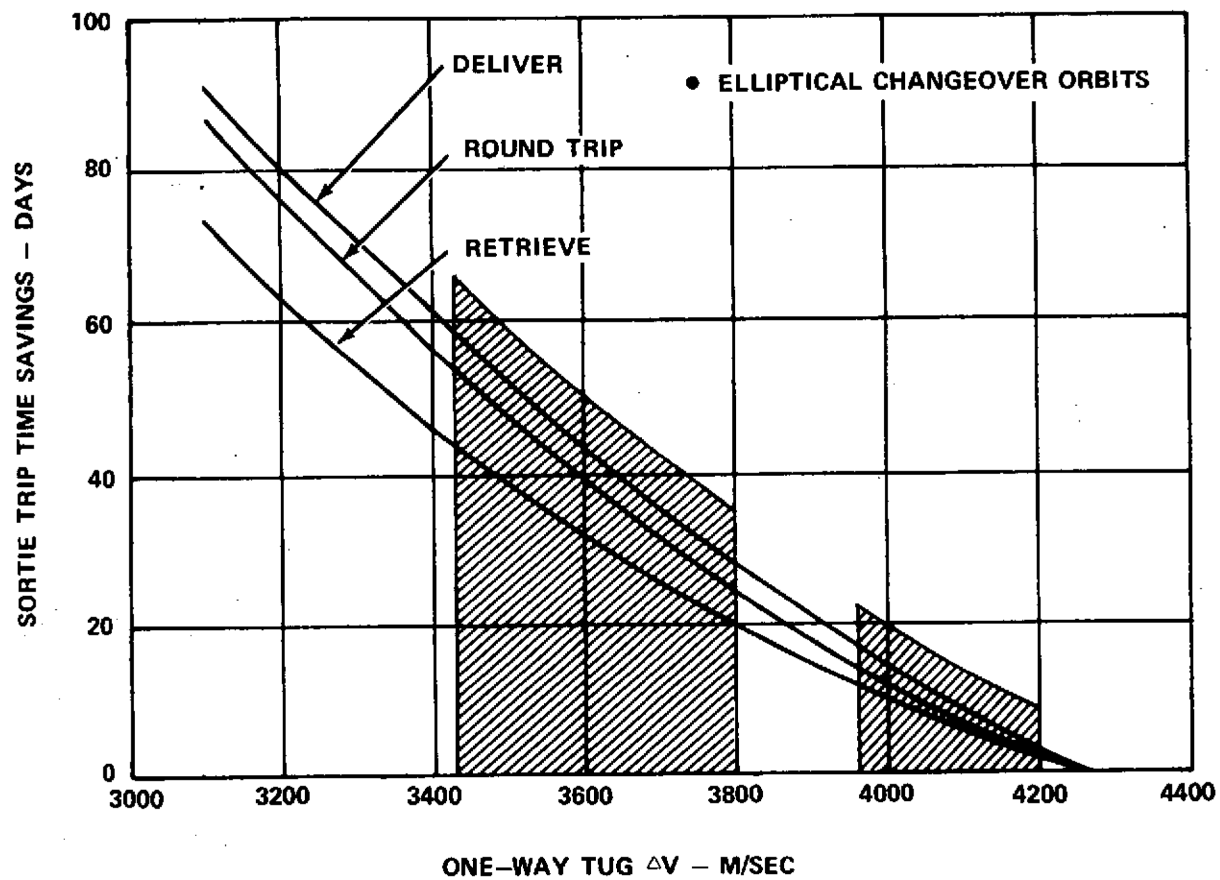


Figure 2-17. SORTIE TRIP TIME SAVINGS OF 50 KW SEPS RELATIVE TO 25 KW SEPS

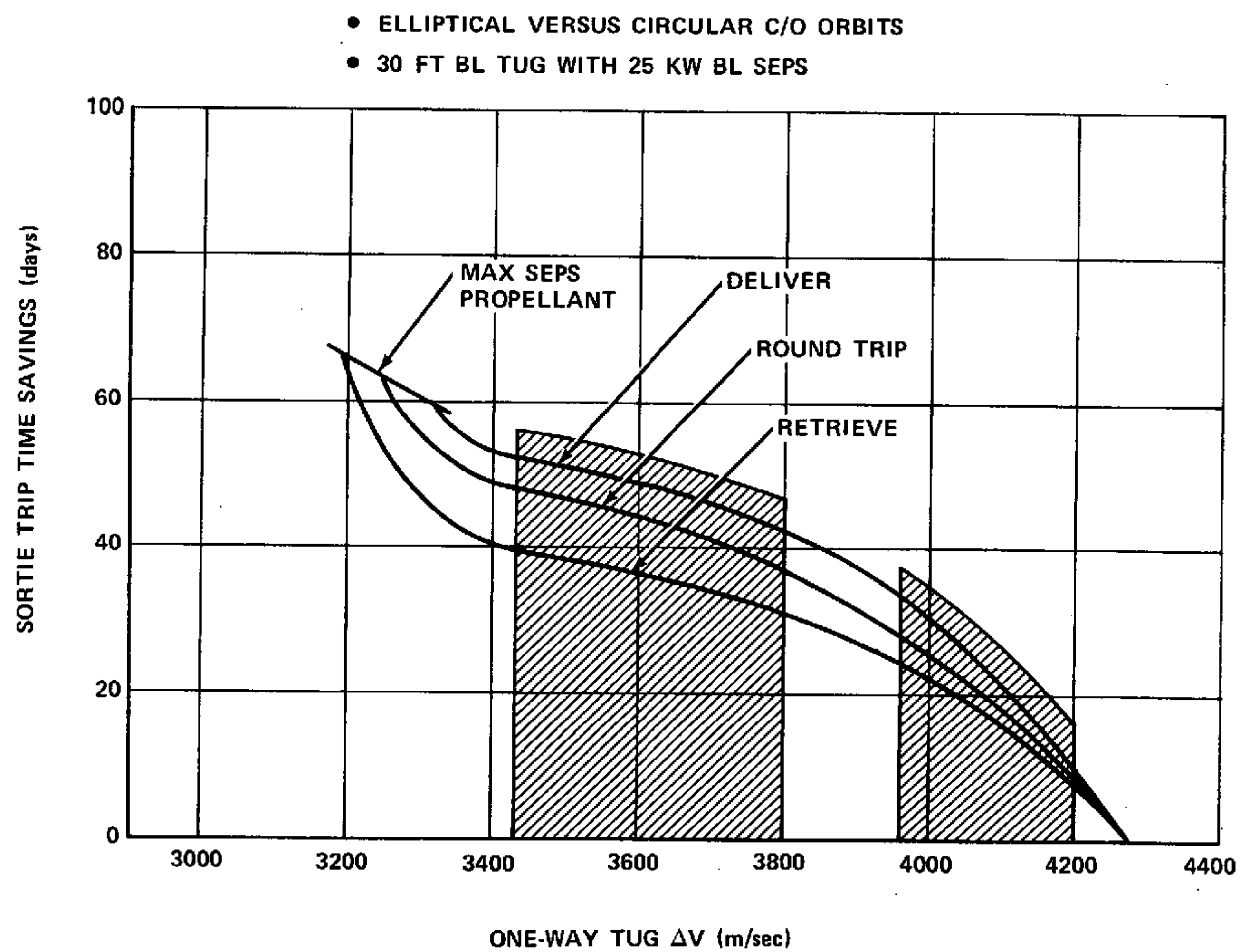


Figure 2-18. TRIP TIME SAVINGS WITH ELLIPTICAL C/O ORBITS

Table 2-13. CHANGEOVER ORBIT COMPARISON, NUMBER OF UPPER STAGE FLIGHTS, 1981-1991

CONFIGURATION		NUMBER OF UPPER STAGE FLIGHTS		
TUG	SEPS	C/O ELLIPTICAL W/RADIATION UNCONSTRAINED PERIGEE	C/O ELLIPTICAL $r_p \geq 20{,}000$ KM	C/O CIRCULAR $r \geq 20{,}000$ KM
9.1 M BL	25 KW BL	126	126	127
9.1 M BL	50 KW, 4158 Isp	124	124	124

NOTES: 1. *90 day trip time unit per leg.*
2. *SEPS refueled 3 times.*
3. *Intermediate orbit payloads not delivered on SEPS flights.*
4. *3-D packaging without 3 up-1 down limit on Tug.*

SEPS sorties fall in the shaded areas and between the deliver and round-trip curves. Ten to 50 days can be saved using elliptical instead of circular changeover orbits.

Payloads in the intermediate orbit class of missions can be delivered on Tug flights along with geosynchronous payloads since both of these mission classes require 28.5 degree Shuttle launches. In the baseline traffic model without SEPS, Table 2-10, 12 flights were dedicated to delivering these payloads and the 55 degree payloads. When SEPS is used to deliver the geosynchronous payloads and the Tug is not allowed to deliver intermediate orbital payloads on the way to changeover orbit, the result for the 9.1 meter BL Tug with 25 kw BL SEPS upper stage flights between 1981 and 1991 is as follows:

Planetary	58
Geosynchronous	37
Polar	12
Other	19
Total	126

Comparing these numbers with the totals in Table 2-10, it is seen that SEPS has reduced the number of geosynchronous flights by 17, but that the number of flights required to deliver intermediate and 55 degree payloads has increased by seven for a net savings of only 10 flights. If Tug is allowed to deliver and retrieve intermediate payloads enroute to and from SEPS rendezvous, some of the seven flights lost in the "other" category can be regained. This is shown in Table 2-14. Delivery of intermediate payloads by the Tug on SEPS flights saves an additional five flights.

Table 2-14. INTERMEDIATE ORBIT COMPARISON, NUMBER OF UPPER STAGE FLIGHTS 1981-1991

CONFIGURATION		UPPER STAGE FLIGHTS REQUIRED	
TUG	SEPS	INTERMEDIATE PLDS NOT DELIVERED	INTERMEDIATE PLDS DELIVERED
9.1 M BL	25 KW BL	126	121
9.1 M BL	50 KW 4158 SEC Isp	124	120

NOTES: 1. *90 day trip time limit per leg.*
2. *SEPS refueled 3 times.*
3. *Elliptical changeover orbits with radiation.*
4. *3-D packaging without 3 up-1 down limit on Tug.*

The shortest trip times are achieved if SEPS is refueled on every sortie to take full advantage of higher average thrust-to-weight ratio resulting from light propellant loads. However, in the WHATIF program which uses the history of average propellant consumption per sortie to indicate the impending need to refuel or retrieve SEPS, any attempt to refuel each sortie or even alternate sorties will result in SEPS being stranded in geosynchronous orbit without enough propellant to get down to changeover orbit for refueling or retrieval. As it turns out, the number of refuelings allowed for each SEPS in its onorbit lifetime do not significantly affect the number of Shuttle flights. In this investigation, the refueling propellant loads were sized so that the allowed number of refuelings (three), along with the original propellant load, would provide roughly 20,000 hours of thruster operation. That this could not be achieved exactly, was due to the average refueling criteria used in the WHATIF program. SEPS were usually refueled when they still had several hundred pounds of propellant left. Table 2-15 shows the refueling results.

With the 9.1 meter baseline Tug there is 9.1 meters of cargo space available for payloads in the Shuttle cargo bay. The number of payloads that can be loaded in this volume depend on the kind of packaging allowed and the limits that are imposed on the number of payloads that can be handled on one

Table 2-15. NUMBER OF REFUELINGS COMPARISON, NUMBER OF UPPER STAGE FLIGHTS, 1981-1991

CONFIGURATION		UPPER STAGE FLIGHTS		
		NUMBER OF REFUELINGS		
TUG	SEPS	1	2	3
9.1 M BL	25 KW BL	122	123	121

NOTES: 1. *90 day trip time limit per leg.*
2. *Elliptical changeover orbits with radiation.*
3. *Intermediate orbital payloads are delivered on SEPS flights.*
4. *3-D packaging without 3 up-1 down limit on Tug.*

Tug flight. The reasons for restricting the number of payloads per flight and their relationship to this study were discussed earlier. The effect on traffic model results of three methods of payload packaging with the three up-one down limit was investigated. These results are compared to three-dimensional packaging without the payload limit in Table 2-16.

Table 2-16. PAYLOAD PACKAGING COMPARISON, NUMBER OF UPPER STAGE FLIGHTS, 1981-1991

CONFIGURATION		PACKAGING METHOD			
TUG	SEPS	END-TO-END 3 UP-1 DOWN	SIDE-BY-SIDE 3 UP-1 DOWN	3-D 3 UP-1 DOWN	3-D NO LIMIT
9.1 M BL	NO SEPS	156	150	150	136
9.1 M BL	25 KW BL	146	129	125	121
FLIGHTS SAVED		10	21	25	15

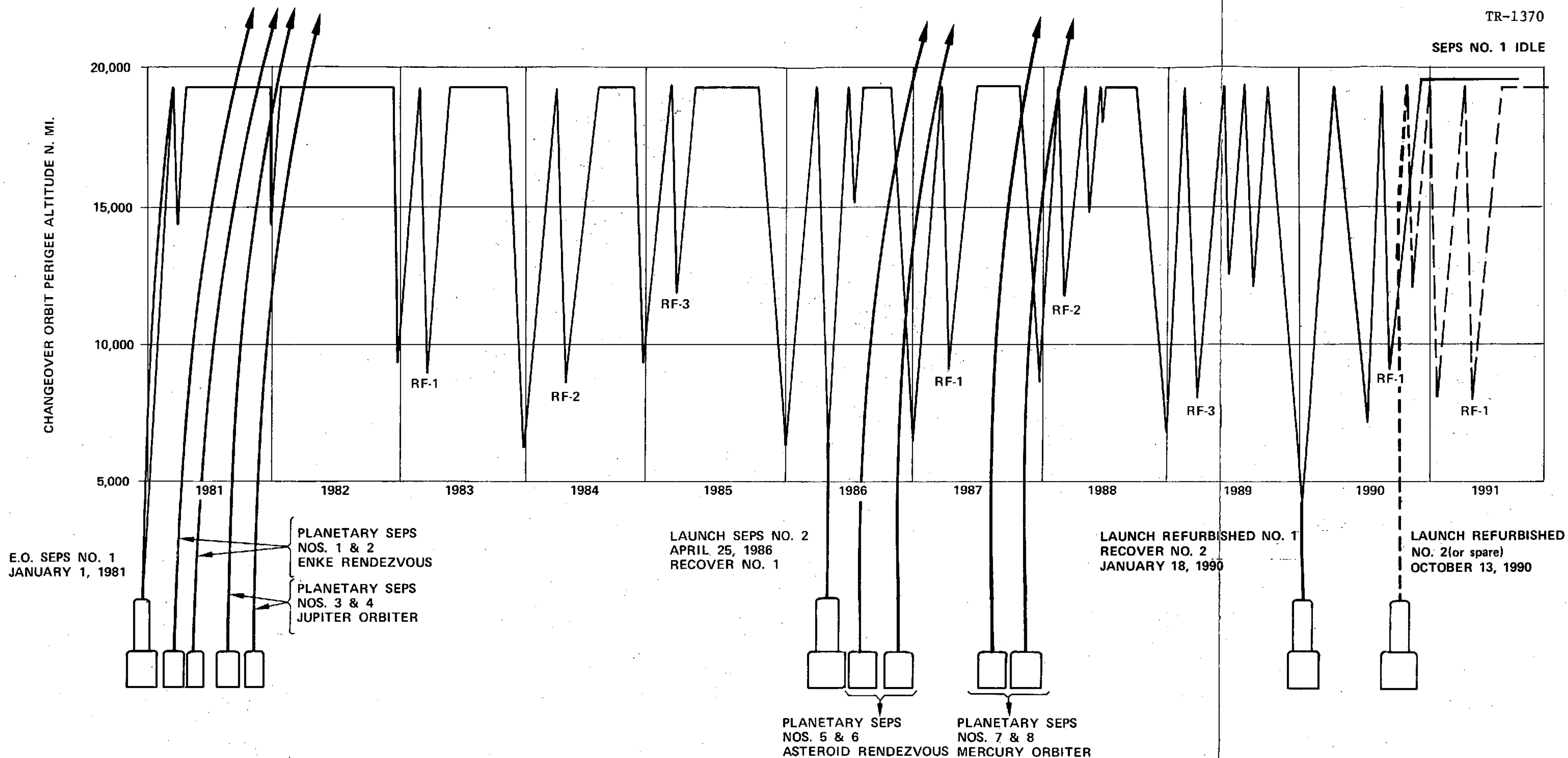
NOTES: 1. *90 day trip time limit per leg.*
2. *Elliptical changeover orbits with radiation.*
3. *SEPS refueled 3 times.*
4. *Intermediate payloads are delivered on SEPS flights.*

The last column in Table 2-16 is the assumption used throughout this study. This gives every advantage to the baseline STS without SEPS, and the resulting fifteen Shuttle flights saved with SEPS in the geosynchronous mission role is conservative. These operational trade studies have demonstrated the advantages of removing the three up-one down restriction, three-dimensional payload packaging, and delivery of intermediate payloads for both STS baseline and STS with EO SEPS. These studies also showed that the 90-day trip time limit was not so short as to cause a significant increase in the number of flights, at least for the baseline Tug. The type of changeover orbits and the number of SEPS refuelings were seen to have a nearly inconsequential effect on required Shuttle flights. These last two factors would reduce Shuttle flights in a mission model that demanded fuller utilization of SEPS capability.

At this point in the study, investigations were narrowed to two STS configurations: the 25 kw baseline SEPS and the 50 kw 4158 sec BL Isp SEPS, both used with the 9.1 meter baseline Tug. In the remainder of this discussion, it is assumed that:

1. Intermediate payloads are delivered on SEPS flights.
2. Elliptical changeover orbits are used.
3. SEPS are refueled three times.
4. Sortie trip times are limited to no longer than 90 days.

A system operational profile for the 25 kw SEPS was shown on Figure 2-19. The operational profile graphically shows SEPS sorties by years. Each sortie is represented by a V, the bottom of the V being the perigee altitude of the changeover orbit and the width of the top being the sortie trip time. SEPS launches and refuelings are indicated in the table at the bottom of Figure 2-19, along with other STS activity as represented by the number of Shuttle flights, Shuttle-Tug flights, and SEPS sorties. The horizontal lines at geosynchronous altitude represent time between sorties when SEPS is idle in geosynchronous orbit. That SEPS is under-utilized is apparent; not until 1989 is the traffic volume great enough to keep it busy the full year. Figure 2-20 is a system operational profile for the 50 kw SEPS. The shorter trip times achieved with this configuration, coupled with the light traffic, result in even more SEPS idle time than is the case with the 25 kw SEPS. The total weight of the geosynchronous payloads carried on the down and up legs of each sortie are shown on Figure 2-21. For the 25 kw SEPS and on Figure 2-22 for the 50 kw SEPS. Sortie trip time and thruster beam power at the end of a sortie are also shown on Figures 2-21 and 2-22 as they were affected by radiation damage to the solar arrays. Since the beginning points of the beam power curves are at the end of the first sortie, the radiation damage incurred on that sortie causes the initial points to be less than 15.6 kw for the 25 kw system or less than 37.5 kw for the 50 kw system.



SHUTTLE	19	22	28	27	33	32	30	32	30	31	32
SHUTTLE/TUG	6	2	8	11	13	12	10	4	6	10	10
SHUTTLE/TUG/SEPS	2	1	2	2	2	3	2	4	4	5	2
	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991

NO. OF SHUTTLE FLIGHTS	316
NO. OF SHUTTLE/TUG FLIGHTS	92
NO. OF SHUTTLE/TUG/SEPS FLIGHTS	29
TOTAL STS FLIGHTS	437

Figure 2-19. SYSTEM OPERATIONAL PROFILE (9.1M BASELINE TUG + 25 KW SEPS)

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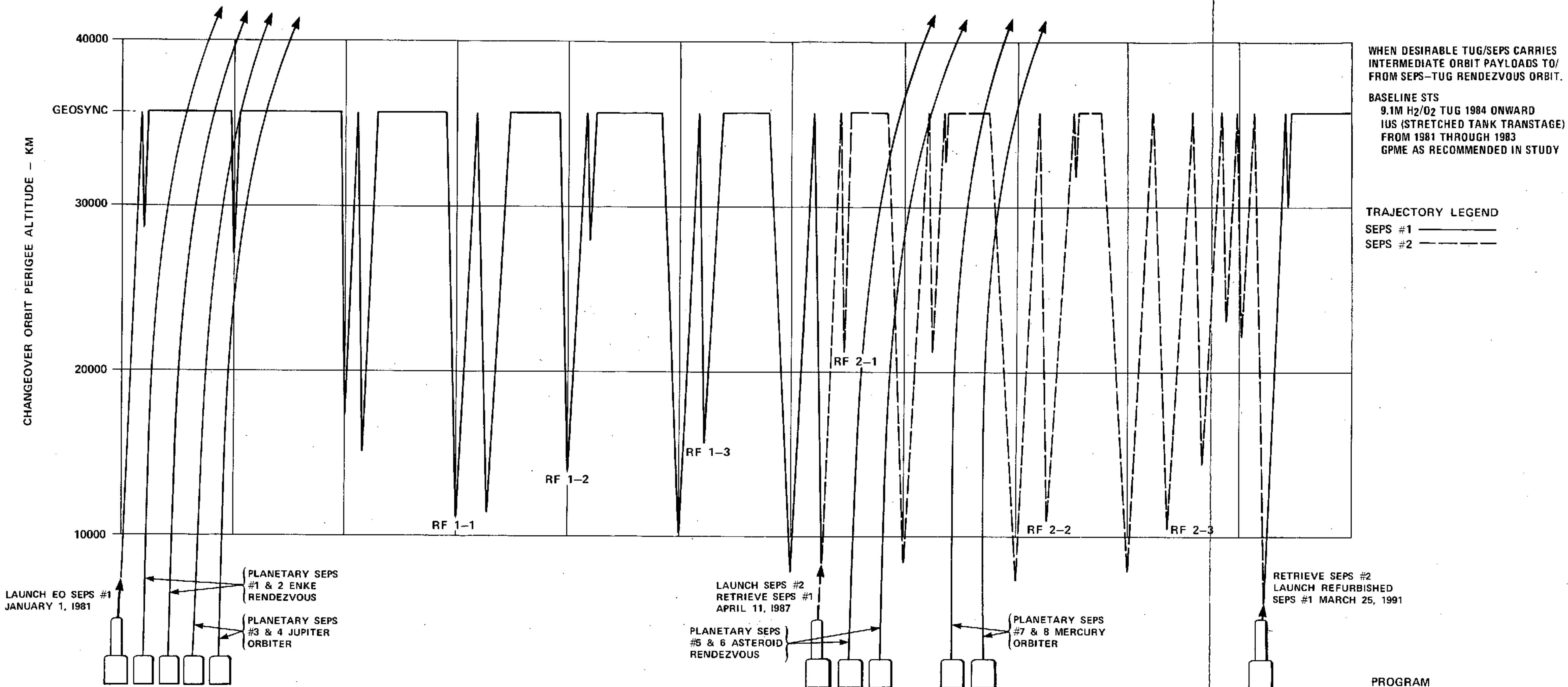

												PROGRAM TOTALS
SHUTTLE	19	22	28	27	33	32	30	32	30	30	32	315
SHUTTLE/TUG	5	2	8	11	13	13	10	5	7	10	9	93
SHUTTLE/TUG / EO SEPS	2	1	2	2	2	2	3	3	3	4	3	27
YEAR	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	435

Figure 2-20. SYSTEM OPERATIONAL PROFILE (9.1M BASELINE TUG WITH 50 KW, 4158 SEC Isp SEPS)

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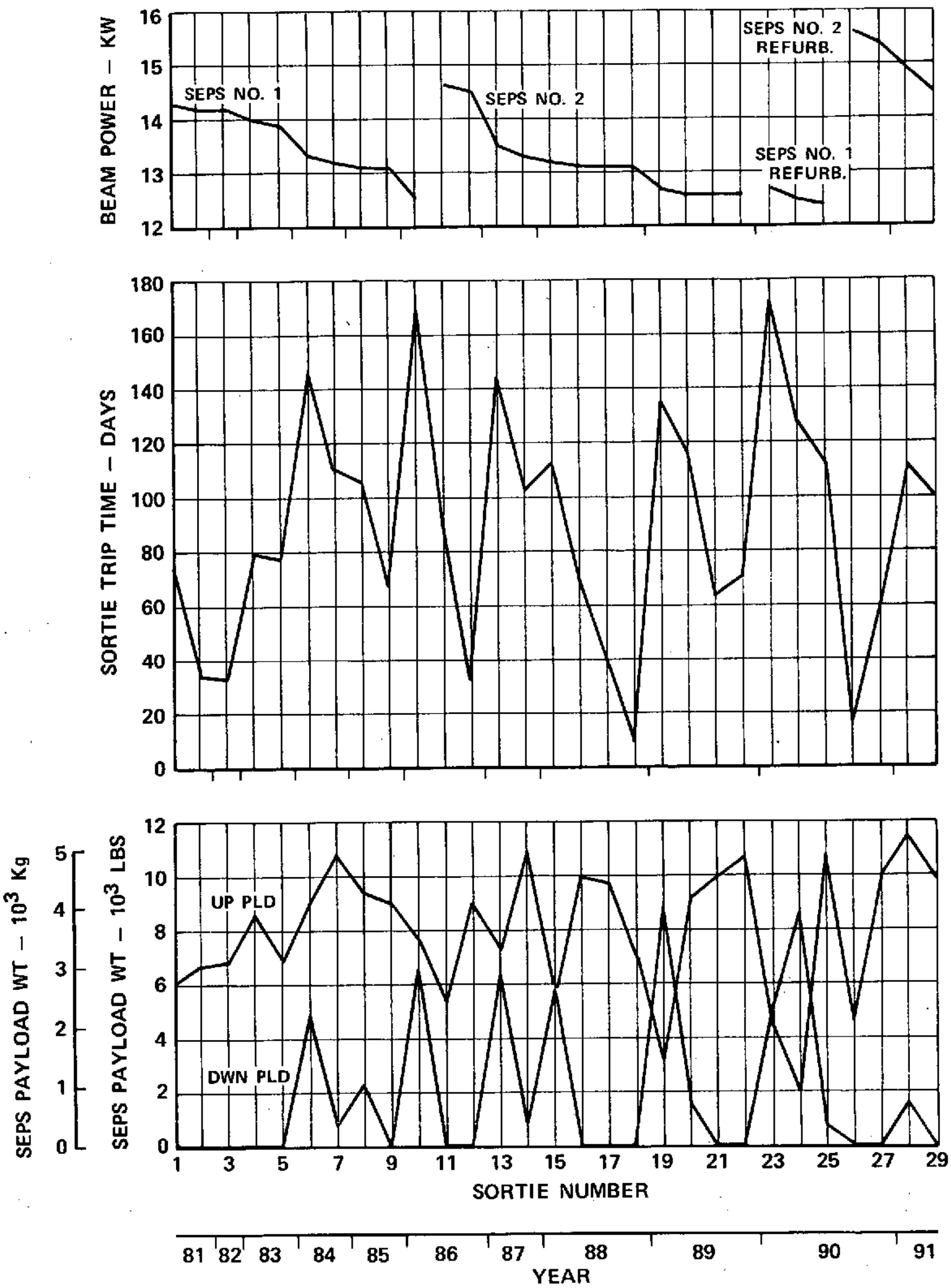


Figure 2-21. 25 KW SEPS PAYLOAD, TRIP TIME, AND POWER

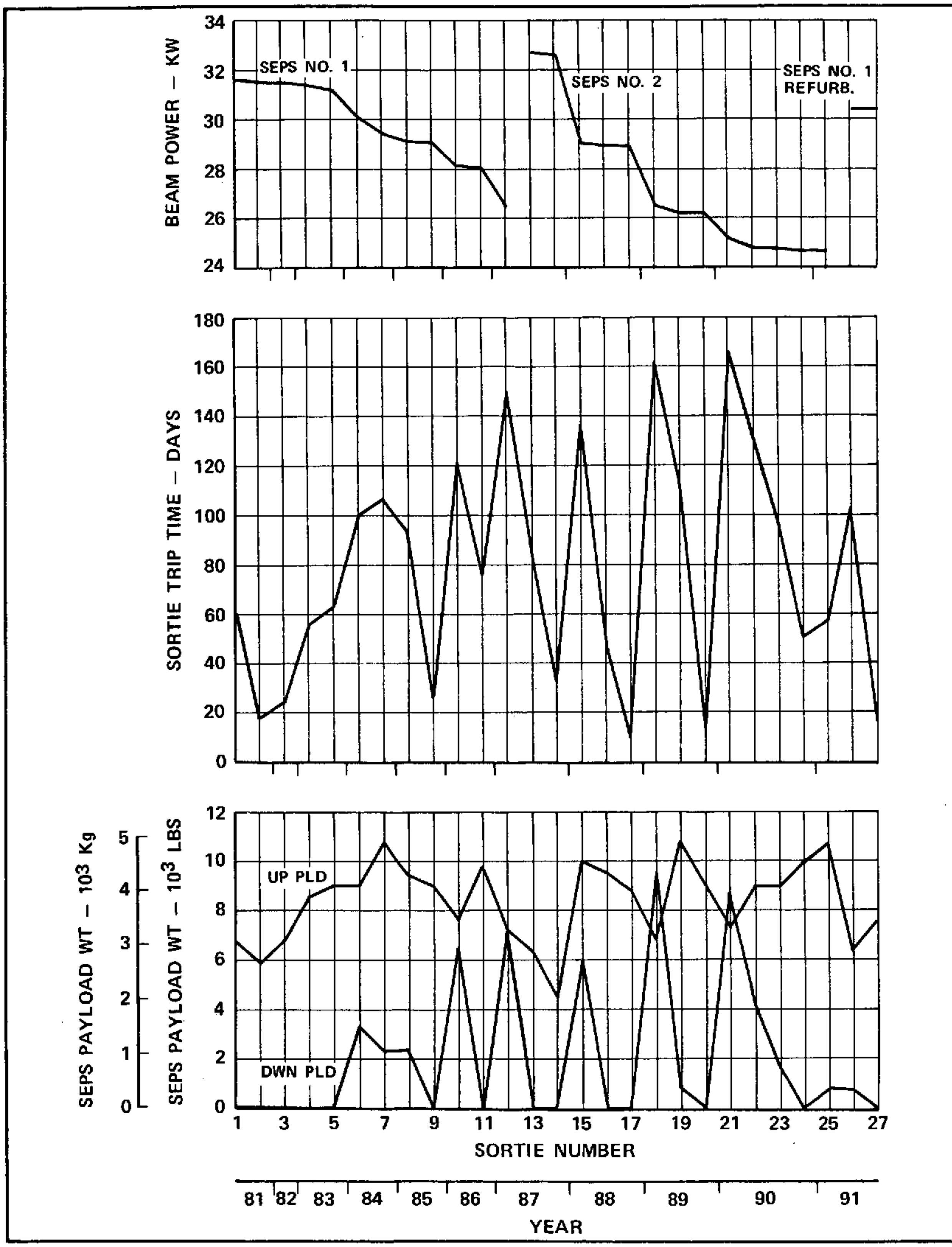


Figure 2-22. 50 KW SEPS PAYLOAD, TRIP TIME, AND POWER

Typical changeover orbits for the 25 kw SEPS are drawn to scale on Figure 2-23. The two orbits shown represent the extremes of low-energy and high-energy changeover orbits encountered in the traffic model. The high energy orbit is the one for the third sortie in 1986, and the low energy is for the recovery of SEPS No. 2 and the launch of refurbished SEPS No. 1 at the atart of 1990. The Tug ΔV's associated with these changeover orbits (as noted on the figure) are less than those in the areas of most frequent SEPS operation shown on Figure 2-16. Delivery of intermediate payloads were not allowed on the SEPS sorties spotted on Figure 2-16. In general, the Tug ΔV required for intermediate payloads results in lower changeover orbits. The question arises, if SEPS can operate from these lower changeover orbits within allowed trip times, why not take off the intermediate payloads and use the extra ΔV to deliver more geosynchronous payloads? The answer is that the number and sizes of payloads in a year do not afford the opportunity to pack enough geosynchronous payloads on Tug to take full advantage of SEPS capability even with three-dimensional packing. The WHATIF program's logic is inadequate here. The heuristic approach of loading payloads on a flight as they are encountered in a preordered list does not in all cases yield the best payload combinations. It is felt that an alternative method in which all possible combinations of the payloads to be delivered in a year are considered would result in a greater average number of payloads per sortie and thus a smaller total number of flights.

In order to cost the STS configurations and determine SEPS cost effectiveness it was necessary to provide data on the number of each kind of flight vehicle required in the traffic model. The number of IUSs, Tugs, Shuttle launches, and SEPS sorties are contained in the traffic model summaries. The traffic model summary for the 25 kw SEPS is shown in Table 2-17. Supplemental information on the number of SEPS launches, retrievals, and refuelings in each year and the number of geosynchronous payloads on SEPS sorties is also included. Tables 2-18 and 2-19 are traffic model summaries for the 50 kw SEPS and the baseline Tug without SEPS. Comparisons of the 25 kw traffic model (Table 2-17) and the 50 kw traffic model (Table 2-18) with the traffic model without SEPS (Table 2-19) for the years 1981 and 1982 show that SEPS did not save any

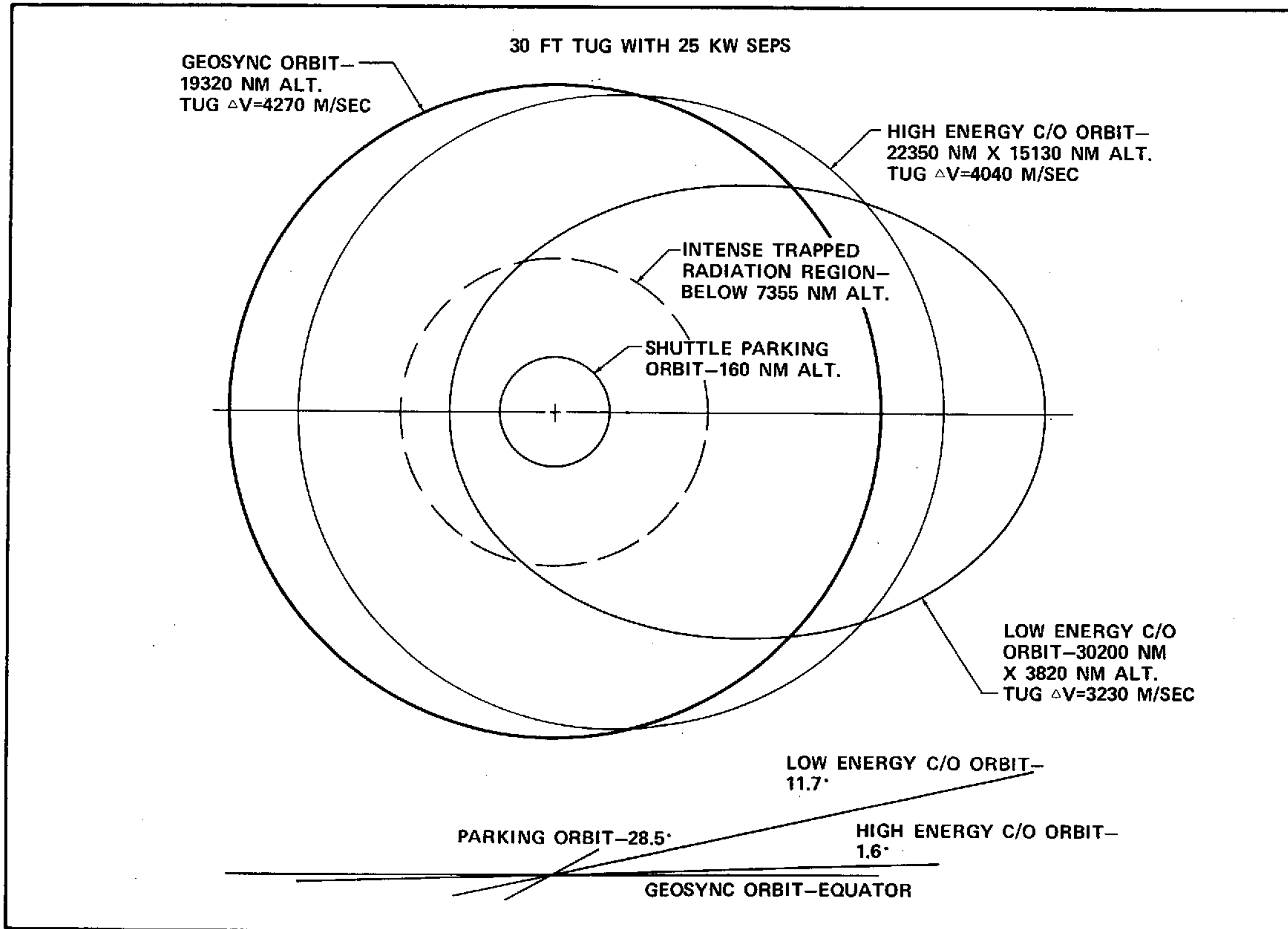


Figure 2-23. RANGE OF ELLIPTICAL CHANGEOVER ORBITS

Table 2-17. TRAFFIC MODEL SUMMARY, 9.1M BL TUG WITH 25 KW SEPS

YEAR	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
TOTAL STS FLIGHTS	27	25	38	40	48	47	42	40	40	46	44	437
IUS (EXPENDED) NOT INCLUDING IUS FOR SEPS SORTIES	4	2	8									14
IUS – BII (EXPENDED)	2											2
TUG NOT INCLUDING TUG FOR SEPS SORTIES				9	9	7	8	3	6	6	5	53
TUG – BII				2		1		1		1	2	7
SHUTTLE FLIGHTS WITH PAYLOADS REQUIRING ORBITAL ASSY WITH TUG					2	2	2			2	2	10
XTUG – BII (EXPENDED)					2	2				1	1	6
SEPS SORTIES	2	1	2	2	2	3	2	4	4	5	2	29
TOTAL UPPER STAGE FLIGHTS	8	3	10	13	15	15	12	8	10	15	12	121
SEPS LAUNCHES	1					1				2		4
SEPS RETRIEVALS						1				1		2
SEPS REFUELINGS			1	1	1		1	1	1	1	1	8
GEOSYNCHRONOUS PAYLOADS ON SEPS	9	6	10	13	9	12	10	16	15	15	9	124

Table 2-18. TRAFFIC MODEL SUMMARY, 9.1M TUG WITH 50 KW, 4158 Isp SEPS

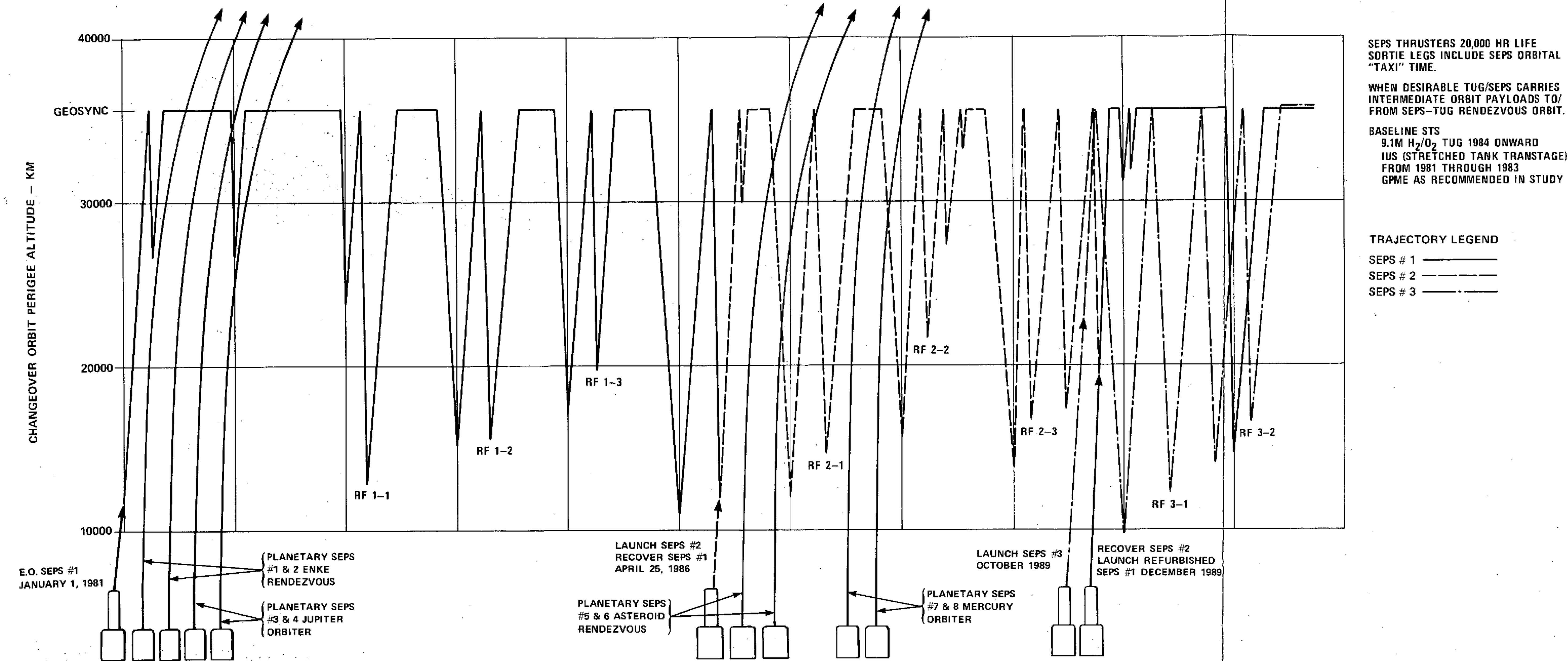
YEAR	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
TOTAL STS FLIGHTS	26	25	38	40	48	47	43	40	40	44	44	435
IUS (EXPENDED) NOT INCLUDING IUS FOR SEPS SORTIES	3	2	8									13
IUS – BII (EXPENDED)	2											2
TUG NOT INCLUDING TUG FOR SEPS SORTIES				9	9	8	8	4	7	6	4	55
TUG – BII				2		1		1		1	2	7
SHUTTLE FLIGHTS WITH PAYLOADS REQUIRING ORBITAL ASSY WITH TUG					2	2	2			2	2	10
XTUG – BII (EXPENDED)					2	2				1	1	6
SEPS SORTIES	2	1	2	2	2	2	3	3	3	4	3	27
TOTAL UPPER STAGE FLIGHTS	7	3	10	13	15	15	13	8	10	14	12	120
SEPS LAUNCHES	1						1				1	3
SEPS RETRIEVALS							1				1	2
SEPS REFUELINGS				1	1	1		1	1	1		6
GOESYNCHRONOUS PAYLOADS ON SEPS	9	6	9	13	9	11	10	15	13	15	10	120

Table 2-19. TRAFFIC MODEL SUMMARY, STS WITHOUT EO SEPS

YEAR	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	TOTAL
TOTAL STS FLIGHTS	27	25	39	42	49	50	44	42	42	47	45	452
IUS (EXPENDED) NOT INCLUDING IUS FOR SEPS SORTIES	6	3	11									20
IUS – BII (EXPENDED)	2											2
TUG NOT INCLUDING TUG FOR SEPS SORTIES				13	12	13	12	8	12	13	8	91
TUG – BII				2		1		1		1	2	7
SHUTTLE FLIGHTS WITH PAYLOADS REQUIRING ORBITAL ASSY WITH TUG					2	2	2			2	2	10
XTUG – BII (EXPENDED)					2	2				1	1	6
SEPS SORTIES												
TOTAL UPPER STAGE FLIGHTS	8	3	11	15	16	18	14	9	12	17	13	136
SEPS LAUNCHES												
SEPS RETREIVALS												
SEPS REFUELINGS												
GEOSYNCHRONOUS PAYLOADS ON SEPS												

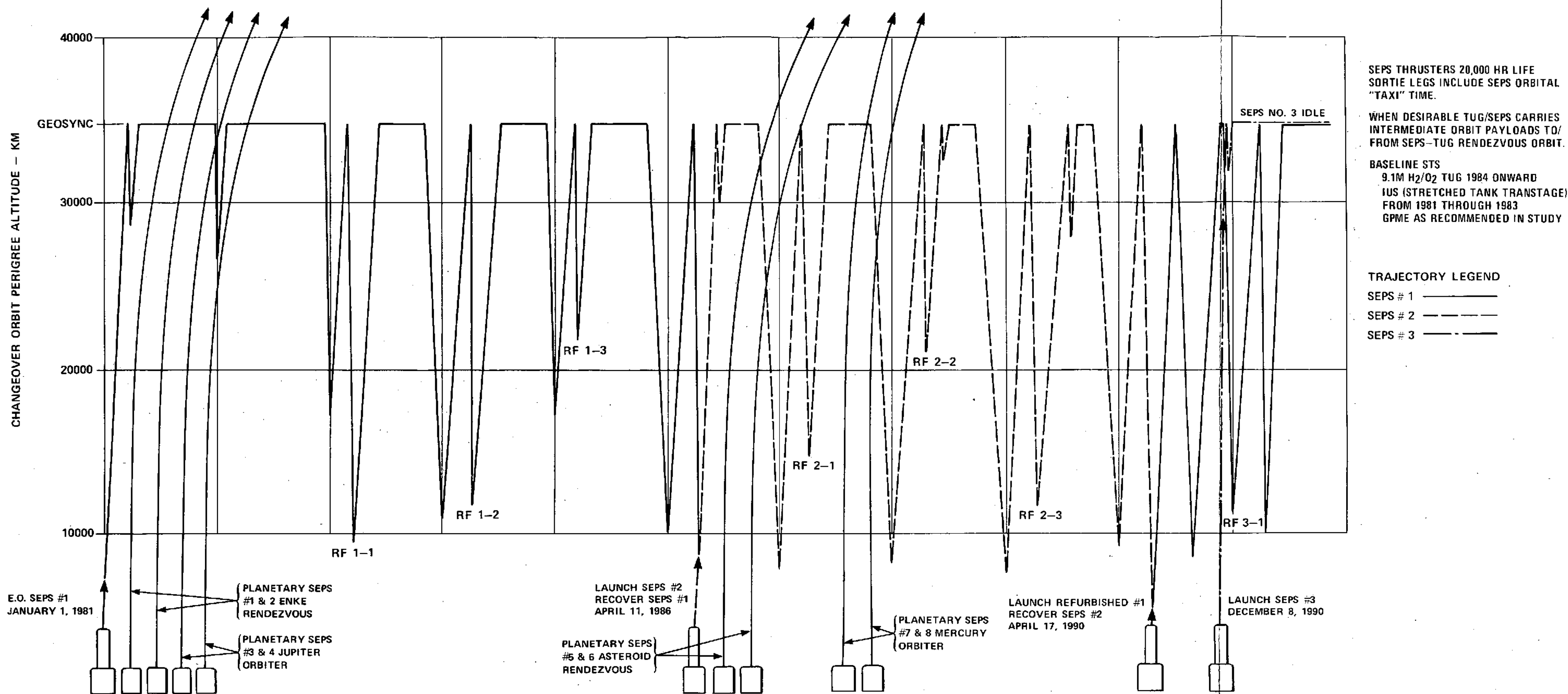
flights in these years even though it flew several sorties. Launch of the first SEPS could be deferred until the beginning of 1983 with no effect on the number of flights in the 25 kw traffic model.

It was expected that when the orbital taxi mission role was combined with the SEPS geosynchronous transport role that the only impact on the traffic model would be earlier launches and retrievals of SEPS because of the time and propellant used by the geosynchronous orbit maneuvers. The system operational profiles for this case are shown on Figures 2-24 and 2-25. The width of the top of the V's includes the time for SEPS to gather up payloads in geosynchronous orbit for the downleg and the time to place payloads at their intended longitudes on the upleg. A comparison of these figures with Figures 2-19 and 2-20 shows that the anticipated earlier launches do occur. Unfortunately, for the 25 kw SEPS this places the launches that were in 1990 in the high traffic year of 1989. This costs an extra SEPS sortie. This could have been avoided by anticipating the need for an extra SEPS in 1989 and launching it in 1988. Because of the light traffic in 1988, this SEPS launch could have been accommodated without an additional flight. In most years, SEPS is idle enough of the time so that it can do the orbital taxi maneuvers without impacting the traffic model. WHATIF program printouts of the 25 kw and 50 kw traffic models that include the orbital taxi mission role are in Appendix A of Volume IV of this report. These printouts show the payloads assigned to SEPS sorties, the changeover orbits, SEPS propellant and power remaining at the end of each sortie, and the up and down trip times. It will be noticed that some of these trip times are greater than 90 days. The 90 day limit was applied only to the transfer time to and from changeover orbit and does not include the additional time required for onorbit maneuvers. Shuttle and Shuttle-Tug flights required by the other missions in the mission model are listed after the SEPS sorties. Payloads assigned to these flights are shown along with propellant loadings and ΔV requirements.



												PROGRAM TOTALS
SHUTTLE	19	22	28	27	33	32	30	32	30	30	32	315
SHUTTLE/TUG	6	2	8	11	13	12	11	4	6	9	10	92
SHUTTLE/TUG/EO SEPS	2	1	2	2	2	3	2	4	5	5	2	30
YEAR	1981	1982		1984	1985	1986	1987	1988	1989	1990	1991	437

Figure 2-24. SYSTEM OPERATIONAL PROFILE 9.1M BL TUG WITH 25 KW BL SEPS INCLUDING ORBITAL TAXI



												PROGRAM TOTALS
SHUTTLES	19	22	28	27	33	32	30	32	30	30	32	315
SHUTTLE/TUG	5	2	8	11	13	12	10	5	7	10	10	93
SHUTTLE/TUG/ EO SEPS	2	1	2	2	2	3	2	3	3	5	2	27
YEAR	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	435

Figure 2-25. SYSTEM OPERATIONAL PROFILE 9.1M BL TUG WITH 50 KW 4158 SEC SEPS INCLUDING ORBITAL TAXI

FOLDOUT FRAME 1

FOLDOUT FRAME 2

III MISSION OPERATIONS & SYSTEM REQMT ANALYSIS

Section III

MISSION OPERATIONS AND SYSTEM REQUIREMENTS ANALYSIS

3.1 INTRODUCTION

This analysis, Task 2 of the contract work statement, had the following objectives:

1. Determine events of critical flight and ground operations for the SEPS (for earth orbital missions only)
2. Investigate flight and ground operations for the SEPS in payload exchange, multiple payload delivery and retrieval, and payload servicing
3. Identify operational modes and potential hardware concepts to implement objectives 1 and 2 and provide conceptual designs
4. Develop mission operations and ground services requirements
5. Define the characteristics of an earth orbital test flight for SEPS.

The basic concepts for operations and generation of the primary system requirements were evolved from identification of the system characteristics and functions required to:

- Execute SEPS multiple mission roles in a cost effective manner
- Provide a system for multiple payload transportation, deployment, and retrieval that would simplify overall STS operations
- Provide for the servicing and maintenance of payloads in a way that will not constrain the payload developers' options in fulfilling payload functional requirements
- Provide for the retrieval of malfunctioning or totally incapacitated satellites
- Provide for deployment of payloads from their high density passenger configuration for transport in the Orbiter and on Tug to their in-space operational configuration
- Provide for repackaging certain space configurations for retrieval
- Provide a SEPS system that has almost universal adaptability to the assembly of large spacecraft and satellites that are transported to earth orbit in modular form by separate flights of the STS.

Objectives 1, 2, and 4 comprise the principal elements of a design reference mission description. Therefore, NSI has elected to document the results of the analysis in a separate volume: Volume III - "Design Reference Mission and System Requirements."

This section will summarize a representative sortie. At appropriate points, capabilities beyond those required for the specific operation will be discussed. Short subsections are devoted to related topics such as the STDN coverage, sun illumination, circular versus elliptical changeover orbits, and times required for taxi trips around geosynchronous orbit. Subsection 3.4 describes a recommended approach to an earth orbital test flight.

3.2 MISSION ROLES AND CHARACTERISTIC PROFILES FOR SEPS WITH THE STS

Section II described in considerable depth the SEPS roles in accomplishing the reference mission model supplied by NASA for establishing the transportation cost effectiveness of SEPS. The predominant transportation roles as indicated by Table 1-1 of the Summary and Section II are:

- Transportation of multiple payload packages to geosynchronous orbit
- Collection of payloads to be retrieved from geosynchronous orbit into multiple payload packages that are transported down to a SEPS/Tug changeover orbit for Tug/Orbiter return to earth
- Combined SEPS-Tug sorties to accomplish intermediate orbital payloads in conjunction with delivery and retrieval of geosynchronous payloads.

For maximum efficiency of STS operations, all available space in the Orbiter's cargo bay must be utilized. Full utilization must be reasonably consistent with the desired launch schedule for each individual payload. When all available cargo space is utilized, Tug usually does not have the capability to deliver (or retrieve) the multiple payload package to geosynchronous orbit. Tug therefore delivers them to a lower energy orbit where the payloads are transferred to SEPS. SEPS then supplies any deficiency in Tug transport capability, delivering the individual payloads to their final mission destination.

Because SEPS always makes up any deficiency, Tug can transport payloads to any intermediate orbits of less energy than the changeover orbit with SEPS while enroute to the Tug/SEPS rendezvous. Payloads to any intermediate orbit requiring greater energy than the Tug/SEPS rendezvous orbit will be delivered by SEPS.

Table 1-1 shows that for maximum STS transport efficiency, 93 percent of all geosynchronous payload missions are accomplished by combined SEPS/Tug sorties; and 60 percent of all intermediate orbital missions are accomplished in this manner.

Figure 1-4 depicts the number of payloads in Shuttle up and down cargo manifests. A total of 83 percent of all individual up payloads requiring upper stages for delivery were transported in multiple payload packages that contained 4 or more payloads. A total of 75 percent of individual payloads were returned to earth in multiple payload packages comprising 3 or more individual payloads.

The study work statement had envisioned 4 distinct mission roles (MR) for SEPS:

MR-1	IUS/Tug performance augmentation for payload delivery/retrieval to geosynchronous orbit
MR-2	Onorbit multiple payload delivery/retrieval/servicing at geosynchronous orbit (orbit taxi)
MR-3	Low earth orbit missions just beyond the capability of Shuttle, primarily in polar and sun synchronous orbits
MR-4	Planetary missions.

Earth orbital mission descriptions and profiles were to be defined for further operations analysis, evolution of SEPS configuration concepts, and development of ancillary mission equipment (General Purpose Mission Equipment (GPME)) concepts. MR-4 planetary missions were investigated only to the extent necessary to ensure that desirable features and capabilities that are added for earth orbital functions would not degrade planetary mission capabilities.

As indicated in Sections I and II of this volume and in the foregoing discussion, MR-1 and MR-2 type functions were typically required to merge into sorties that combined both roles if STS effectiveness was to be maximized. For this reason, other operational discussions in this volume and in Volume III are generally related to representative SEPS/Tug sorties rather than to mission roles. Some specific missions and phases of missions are discussed in greater detail to illustrate desirable characteristics of the recommended SEPS configuration or of the recommended STS GPME.

Low earth orbital missions were investigated only to establish SEPS basic capabilities. SEPS can accomplish these missions; however, there appears to be little transportation cost effectiveness gain compared to accomplishing them by use of Shuttle plus the addition of a standard chemical propulsion package to the payloads.

3.2.1 System Operational Profile with the Complete Mission Model

A total STS with SEPS System Operational Profile to accomplish the reference mission model was shown in Figure 2-24 and discussed in some detail. A SEPS mission cycle is defined as the cycle of operations beginning with the SEPS removal from inventory storage and continuing through its onorbit operations until it is retrieved for refurbishment and returned to inventory. In the cost effectiveness analysis, it was assumed that refurbishment would occur at about 20,000 hours of thruster operation. On that basis, 2 1/2 SEPS mission cycles were required to complete the mission model. Present technology indicates that the expected life of SEPS thrusters that will be in operation in the 1980's will probably be 50,000 or more hours.

Figure 2-24 shows that 2 operational SEPS and 1 spare are adequate to accomplish the mission model from 1981 through 1991. SEPS No. 1 is launched in 1981 and remains in orbit accomplishing 10 sorties before it is retrieved with about 20,000 hours on the thrusters in 1986. SEPS No. 1 has its mercury and ACS N_2H_4 replenished three times during this mission cycle.

Figure 2-24 is somewhat misleading in that the sloped ascent and descent lines indicating elapsed time for the ascent or descent leg of a sortie also include the time for taxiing around geosynchronous orbit to collect retrieved payloads from, or to deploy individual payloads to, their specific mission longitudes. Times to travel to a satellite and service it when that is a designated function of a specific sortie are also parts of the ascent line. The horizontal lines at the geosynchronous altitude represent the time SEPS is idle on geosynchronous orbit. SEPS No. 1 is idle for about 50 percent of the time onorbit. Only in the last few years, 1988-1991, is one onorbit SEPS fully utilized.

On the Tug sortie that retrieves SEPS No. 1, SEPS No. 2 is deployed with its initial payload set. SEPS No. 1 is refurbished and returned to inventory. SEPS No. 2 stays in orbit from 1986 to 1989, accomplishing 10 sorties. Because of a groundrule that required every individual payload to be launched in its specified year, the spare SEPS No. 3 was launched in late 1989 to accomplish a sortie that SEPS No. 2 could not complete in that year.

On the Shuttle flight that retrieved SEPS No. 2, refurbished No. 1 was carried to changeover orbit to begin its mission cycle. SEPS No. 2 is refurbished to become the spare inventory item. Except for three sorties in 1990/1991, SEPS No. 1 is idle in geosynchronous orbit. Mission model requirements do not demand its services. SEPS No. 3 accomplishes all remaining sorties to complete the reference mission model.

Volume IV, "Traffic Model and Flight Schedule Analysis Techniques and Computer Programs," contains a computer printout giving the sequence of flights depicted in the Systems Operational Profile (Figure 2-24) just described. The cargo manifests for each flight are given with a description of individual payloads and their destinations. Manifests are also provided for the flights that did not involve SEPS to indicate the level of other STS activity. This other Shuttle and Tug activity proceeding concurrently with Tug/SEPS sorties was a principal reason for NSI's emphasis on evolving GPME that would simplify Tug-Shuttle operations for multiple payload operations even when SEPS was not involved in a sortie. The GPME concepts evolved (described in Sections IV and V) are designed to the extent practicable to allow launch preparation activities of Shuttle, Tug, and the multiple payload package to be carried out independently.

3.2.2 Reference Sortie Profile

An arbitrary reference sortie profile was established that contained one example of each function that SEPS would be required to execute in any earth orbital role. At significant phases of this reference sortie, the envelope of capabilities or range of required functions for other similar phases will be discussed.

The general functional flow of an earth orbital SEPS mission cycle is shown on Figure 3-1. This flow is discussed in some detail in Volume III. Recall that SEPS remains in orbit and executes 10 or more sorties by rendezvous with Tug in a changeover orbit before it is returned to earth.

Section II of this volume describes the advantages of elliptical changeover orbits in terms of trip time savings. Figure 3-2 shows the range of elliptical orbits used in accomplishing the complete mission model. There were very few of the low energy changeover orbits required in accomplishment of the mission model during the years 1981-1991, so that radiation damage to SEPS solar arrays, while significant, was not severe.

In order to develop a reference sortie profile, the following payload manifest was used. This manifest does not actually occur in the traffic model. It is a synthesized composite to illustrate the general Tug-SEPS sortie.

SORTIE PAYLOAD MANIFEST - SHUTTLE LAUNCH: MARCH 1986

Payload ID	Weight Kg	Length/ Dia.(M)	Longitude	Apogee Alt - Km	Perigee Alt - Km	Inc Deg
Intermediate Up Payloads						
EOP-9	414	3.1/1.77	--	2,000	1,000	28.0
Geosynch Up Payloads						
NN/D-1	2,039	3.7/2.5	30°W	35,785	35,785	0
NN/D-4	645	3.7/3.1	162°W	35,785	35,785	0
NN/D-9	366	3.1/1.8	135°E	35,785	35,785	0
Geosynch Down Payloads						
EO-4A	1,359	3.3/2.6	100°W	35,785	35,785	0
NN/D-10	347	3.1/1.8	80°W	35,785	35,785	0
Intermediate Down Payloads						
AST-1A	291	3.7/.8	--	550	550	28.5

The changeover orbit is generally chosen for compatibility with intermediate orbital payload requirements and to minimize SEPS transfer time. The changeover orbit used as the timeline base has the following characteristics:

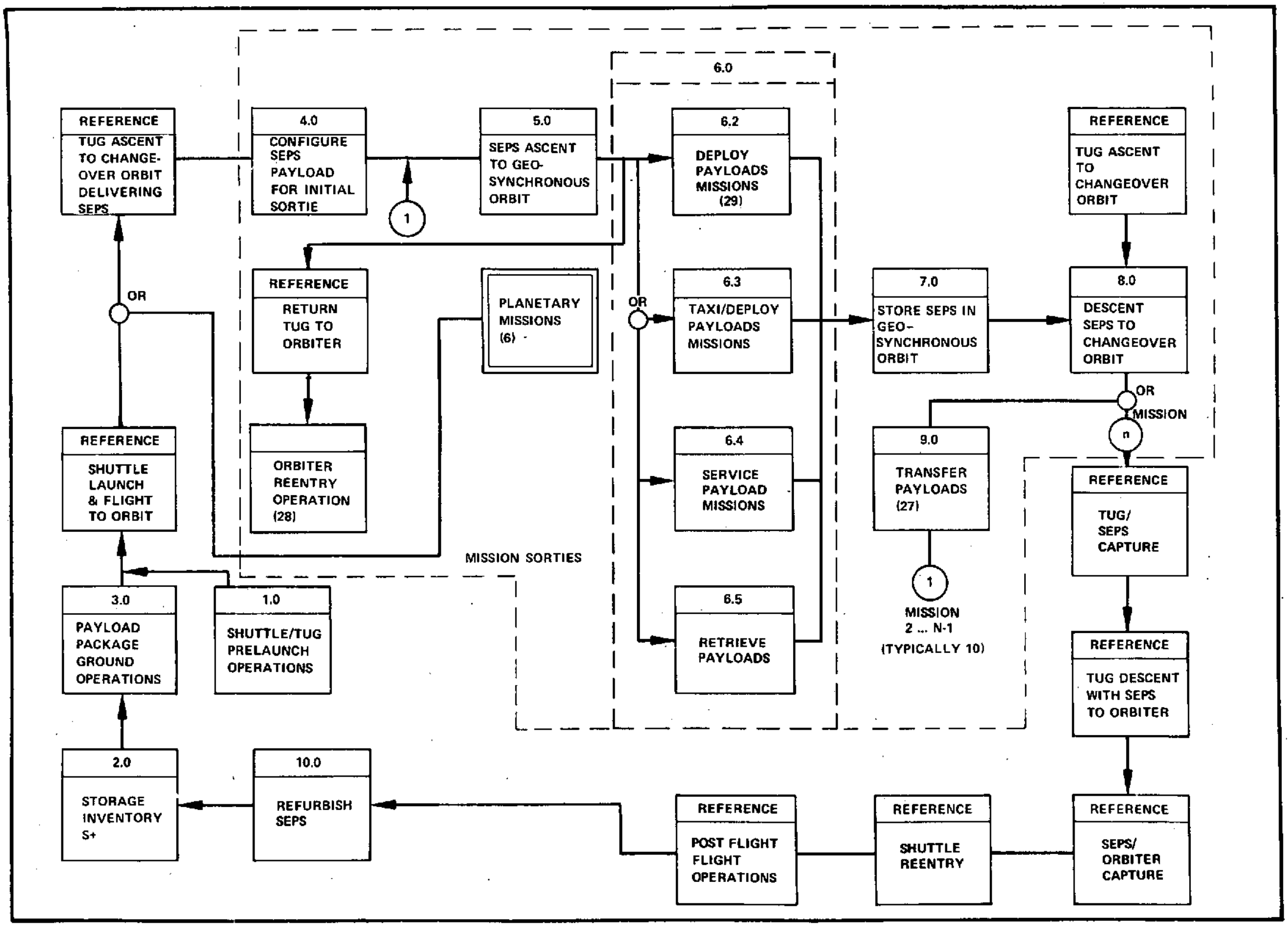


Figure 3-1. MISSION CYCLE FUNCTIONAL FLOW

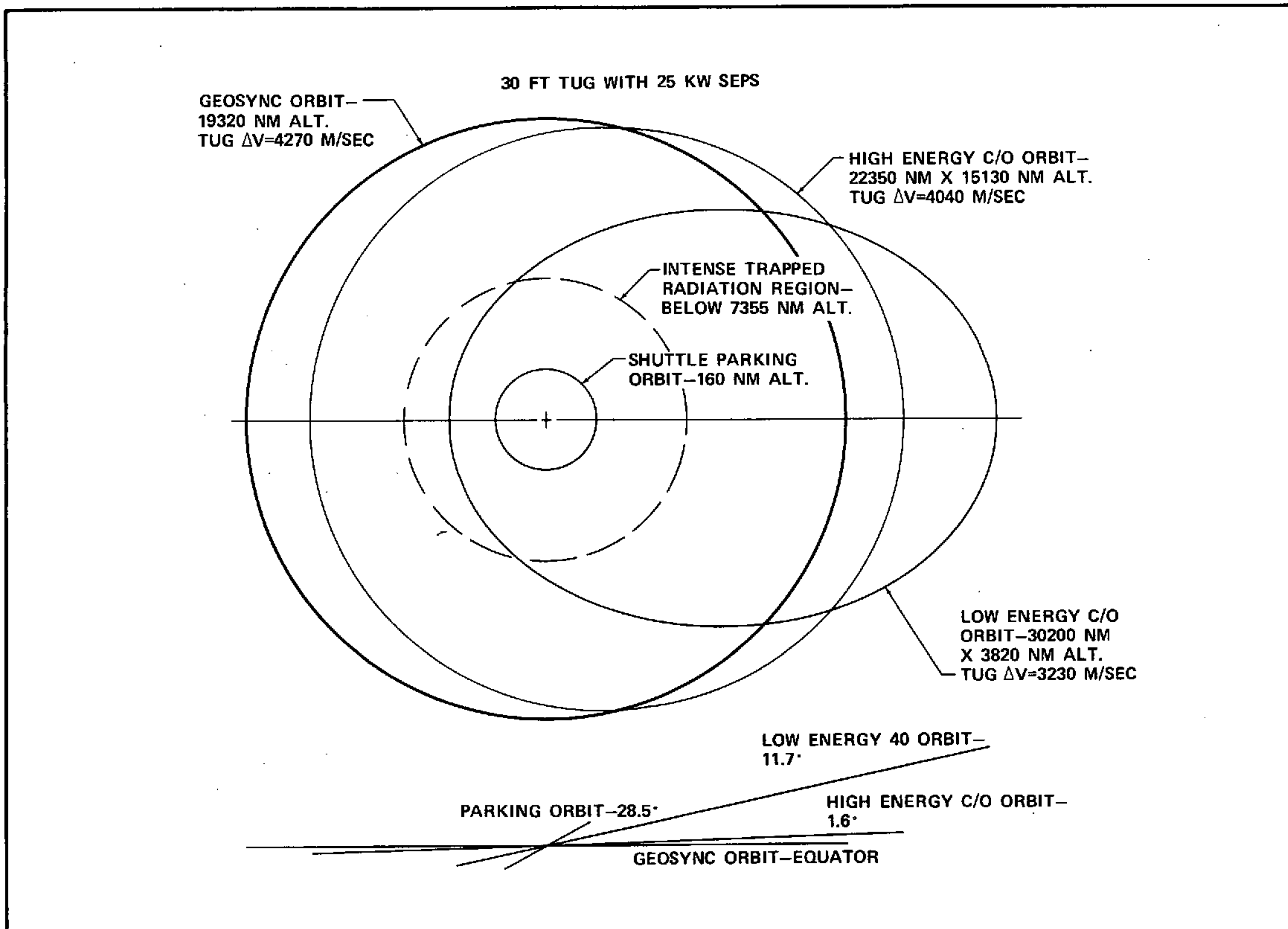


Figure 3-2. RANGE OF ELLIPTICAL CHANGEOVER ORBITS

apogee altitude - 48,475 km

perigee altitude - 17,203 km

inclination - 4.7 deg

The Shuttle, Tug, and SEPS characteristics are:

Shuttle

Payload at 296 km x 28.5 degrees	- 28,656 Kg (63,100 pounds)
Maximum Down Payload	- 14,532 Kg (32,000 pounds)

Tug

Empty Weight	- 2,750 Kg (6,055 pounds) including flight GPME
Usable Propellant	- 23,035 Kg (50,724 pounds)
Specific Impulse	- 456.5 sec
Thrust	- 66,735 N (15,000 pounds)

SEPS

Beam Power (undegraded)	- 15.67 kw
Specific Impulse	- 2,940 sec
Empty Weight	- 1,243 Kg (2,740 pounds)
Propellant Capacity	- 771 Kg (1,700 pounds)

Table 3-1 is a listing of event times for the sortie. It includes contingency times allowing several opportunities for each chemical stage burn.

The sortie events may be summarized as follows.

The master scheduling function has established the deployment dates of the up payload set and the retrieval dates for those payloads being retrieved on a scheduled basis a year or more in advance of the sortie. The specific, detailed mission plan for the sortie can respond to retrieval requirements caused by the malfunction of a payload within a few days of the time the last planned retrieval payload is collected in geosynchronous orbit just before SEPS begins its descent trip to rendezvous with Tug. Because of SEPS' high ΔV capability, the mission profile can be replanned for SEPS to return to geosynchronous orbit, even after the descent maneuver is in progress, to retrieve an additional high priority satellite that may have failed after descent began.

Table 3-1. EVENT TIMES ON REFERENCE TRAJECTORY PROFILE

MISSION TIME (Days)	TIME FROM SHUTTLE LAUNCH	EVENT	MASS (Kg)	PROPELLANT MASS (Kg)	BURN TIME	
SEPS DESCENT						POWER (kw)
0.00	-38.9 days	SEPS docked with payload at 80°W Longitude Longitude shift (20°W)	2,194	604	2.3 days	15.3
2.30	-36.6 days	SEPS docked with payload at 100°W Longitude Descent to changeover orbit	3,546	597	36.6 days	15.3
38.90	0.0 days	SEPS and payloads at changeover orbit	3,430	481		15.2
SHUTTLE ASCENT						
38.90	0.0 hours	Shuttle Launch				
39.02	2.9 hours	Orbiter injection on park orbit over 154° West Longitude				
TUG ASCENT						ΔV (m/sec)
39.02	2.9 hours	Start coast to descend node (1.32 revs)	28,622	22,409		
39.10	4.9 hours	Initiate transfer to 540 n mi (ΔV_1)	28,622	22,409	80.1 sec	191.
39.14	5.7 hours	Inject on 540 x 1080 x 28.0° orbit (ΔV_2) Drop intermediate payload and coast to ascend node (1 rev)	24,907 24,492	18,695 18,695	165.0 sec	421.
39.22	7.6 hours	Inject on phasing orbit (ΔV_3) Coast to ascend node (1 rev)	18,568	12,770	391.0 sec	1219.
39.39	11.8 hours	Initiate transfer to changeover apogee (ΔV_4)	18,568	12,770	240.0 sec	960.
39.70	19.3 hours	Inject on changeover orbit (ΔV_5)	11,828	6,030	204.8 sec	1026.
40.15	30.0 hours	Rendezvous with SEPS (1/2 rev)	11,828	6,030		
TUG DESCENT						
40.36	35.0 hours	Interchange Tug and SEPS payloads and coast to descend node (1/2 rev)	10,482	6,030		
40.60	40.8 hours	Initiate transfer to 297 n mi x 28.5° (ΔV_6)	10,482	6,030	147.5 sec	1056.
40.91	48.3 hours	Inject on phasing orbit (ΔV_7) Coast to ascend node (1 rev)	6,233 6,233	1,780 1,780	132.9 sec	1232.
41.05	51.5 hours	Inject on 297 n mi x 28.5° orbit (ΔV_8)	4,652	200	104.4 sec	1288.
41.12	53.1 hours	Rendezvous with intermediate payload (1 rev) Retrieve intermediate payload Coast to phase with orbiter (10-1/2 revs)	4,652 4,942	200 200		
41.82	69.9 hours	Initiate transfer to Shuttle orbit (ΔV_9)	4,942	200	5.2 sec	71.
41.85	70.7 hours	Inject on Shuttle orbit (ΔV_{10})	4,785	42	5.2 sec	72.
SHUTTLE DESCENT						
41.91	72.2 hours	Rendezvous with Tug (1 rev) Deorbit				
SEPS ASCENT						POWER (kw)
40.36	35.0 hours	Begin ascent from changeover orbit Ascent to geosynchronous orbit	4,776	481	50.4 days	15.2
90.76	51.9 days	SEPS and payloads in geosynchronous orbit at 30° West Longitude Deploy payloads at 30° West Longitude Longitude shift (132° West)	4,628 2,588	334 334	6.5 days	15.0
97.26	58.4 days	SEPS and payloads at 162° West Longitude Deploy payload at 162° West Longitude Longitude shift (63° West)	2,567 1,922	313 313	3.9 days	15.0
101.16	62.3 days	SEPS and payload at 135° East Longitude Deploy payload at 135° East Longitude	1,910 1,544	301 301		15.0

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In the reference sortie illustrated in Table 3-1, at the planned start time 39 days before Shuttle will be launched with Tug, SEPS proceeds to retrieve the first down payload.

Because of SEPS' low acceleration it does not use phasing orbits, but is started on trajectory profiles so that continuous thrusting for the minimum length of time will bring it to the desired rendezvous or payload deployment point. The terminal phase of SEPS' approach to a target point for deployment of a payload, or to a rendezvous, is just an extension of the cruise phase as indicated on Figure 3-3. For sunlit targets, the SEPS, with information from the ground as to target payload position, can acquire the target at distances up to 7,223 kilometers and begin path adjustments. Figure 3-3a shows the relative motion of SEPS approaching a target geosynchronous payload when only the ion thrusters are used in order to conserve ACS propellants. Times shown are times before station alongside the payload at relative velocity 0. The arrows indicate the direction of thrust. Figure 3-3b shows added details of the last few hours.

The SEPS flight control center would not need to be fully manned prior to about 2 hours before payload deployment or retrieval was to begin. Conversely, if it is desired to compress the last 6 hours of the operation, ACS thrusters can be utilized. These thrusters, combined for additive thrust in the same direction as the ion system, provide about 100 times the acceleration of the ion system. ACS-produced acceleration is 0.06 to 0.3 m/sec^2 depending on payload mass.

The manner in which the manipulator system grasps the payload and places it on a diaphragm on the SEPS transport mast is described in Sections IV and V of this volume.

After collecting the second payload, SEPS cruises to the changeover orbit to meet Tug. This consumes about 36 days for the reference profile. After the cruise phase has been initiated, the SEPSOC flight control is manned only one day each week for a status check on SEPS trajectory progress and on the functional status of subsystems. SEPS has an autonomous navigation and guidance system. The navigation system operates on the basis of establishing a

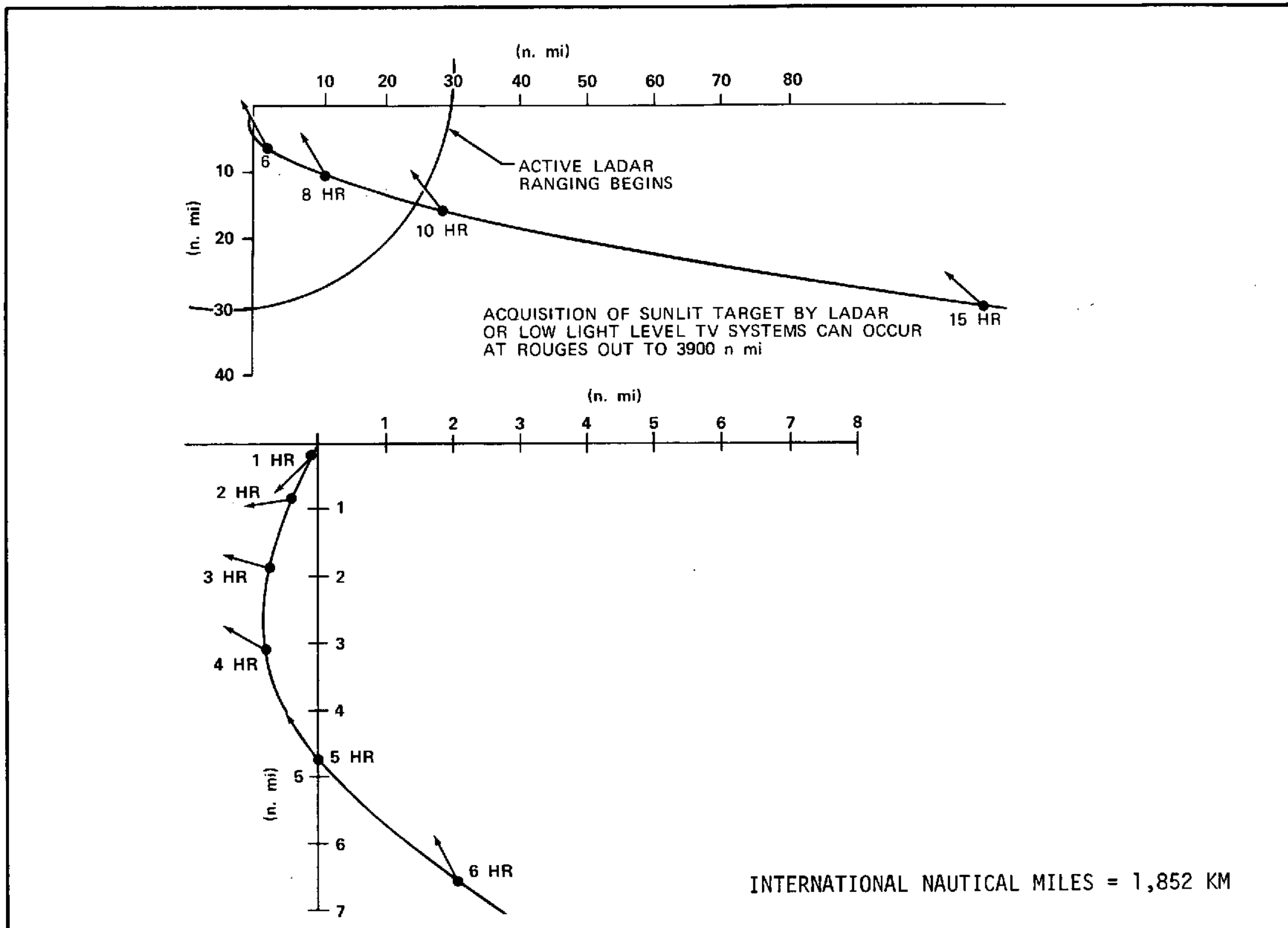

Figure 3-3. SEPS RELATIVE MOTION APPROACHING TARGET

continuing series of SEPS positions from data collected by onboard sensors. Errors are, therefore, not cumulative. The expected system accuracy is position within 1 km and velocity within 0.1 m/sec. The guidance computer with onboard software determines the thruster pointing directions to maintain the position track along the preplanned profile.

Since STDN tracking and ground computation of SEPS position are for status check only, and are not required as a part of the nominal path-keeping navigation and guidance function, this weekly status check can be shifted to accommodate other higher priority activities of STDN or the SEPS program support group when and if necessary.

Since SEPS has propulsion capability and can be planned (commanded) to be at a specific point in the changeover orbit at a specific time, the Shuttle and Tug ascent maneuvers can be planned for nominal execution with a minimum of phasing orbit time delays. This can minimize the time Shuttle and Tug must be in orbit for a sortie. Figures 3-4 and 3-5 are general illustrations of the trajectory profiles that may be used to allow Tug to deliver and retrieve an intermediate orbit payload enroute to and from the payload changeover orbit with SEPS. Some phasing orbits not normally required are shown in the figures. The representative times are given in Table 3-1. The intermediate orbital payload delivery and retrievals have been shown in 28.5° inclination orbits. There is nothing that restricts these orbits to a 28.5° inclination, and different payloads may be deployed and retrieved at different orbits enroute. As plane change requirements demanded of Tug for multiple intermediate orbit retrieval increase, less demanding changeover orbits of lower altitude must be planned. In order to avoid radiation damage, operational choices will be limited when changeover orbits approach circular orbits near the intense radiation zone of the Van Allen belt.

At the ascending node of the last intermediate orbit, the Tug burns to initiate transfer to apogee of changeover orbit and accomplish the required plane change. If the mission is properly planned, a phasing orbit will not be necessary for Tug rendezvous with SEPS.

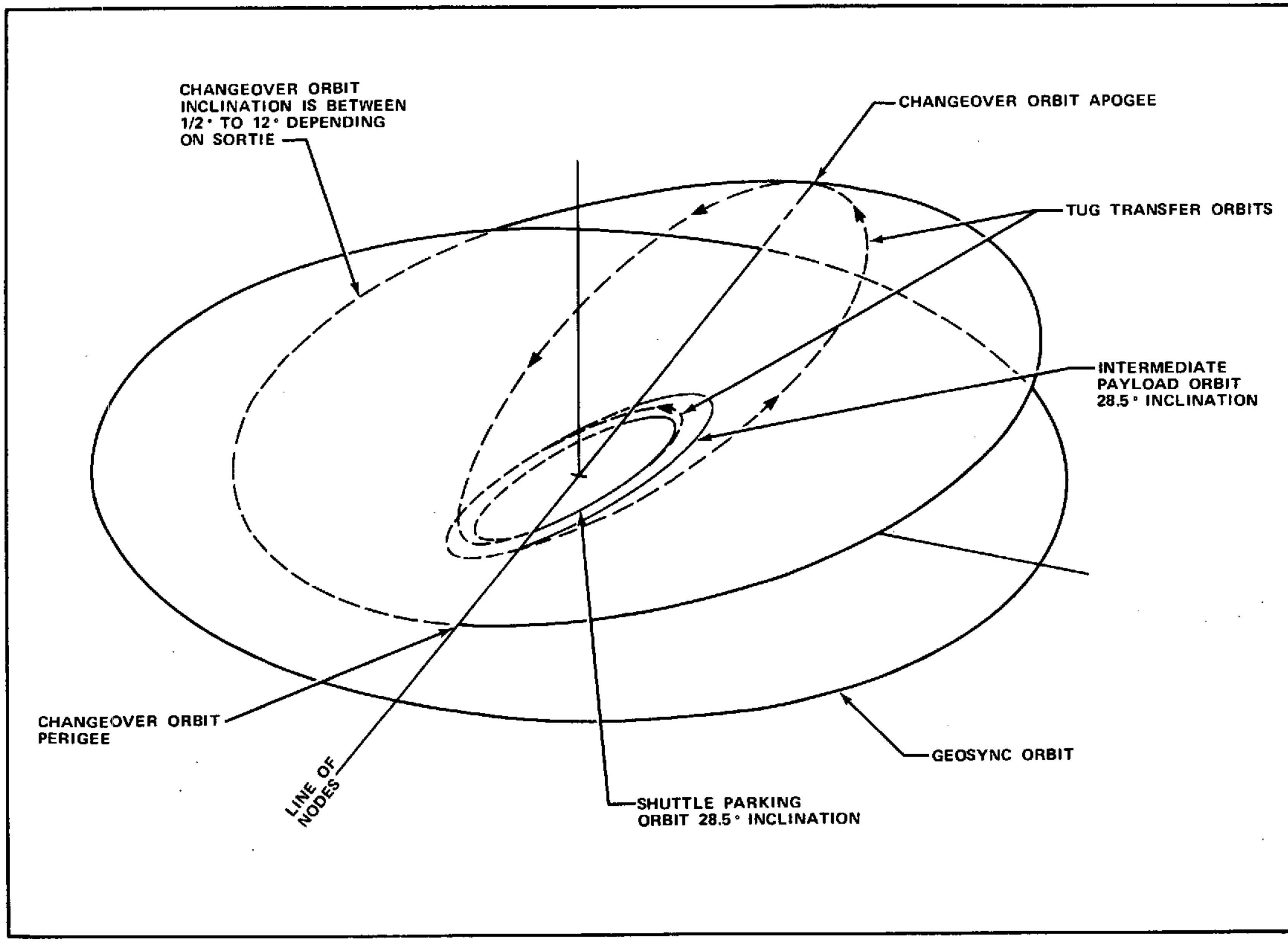

Figure 3-4. REFERENCE TRAJECTORY PROFILE

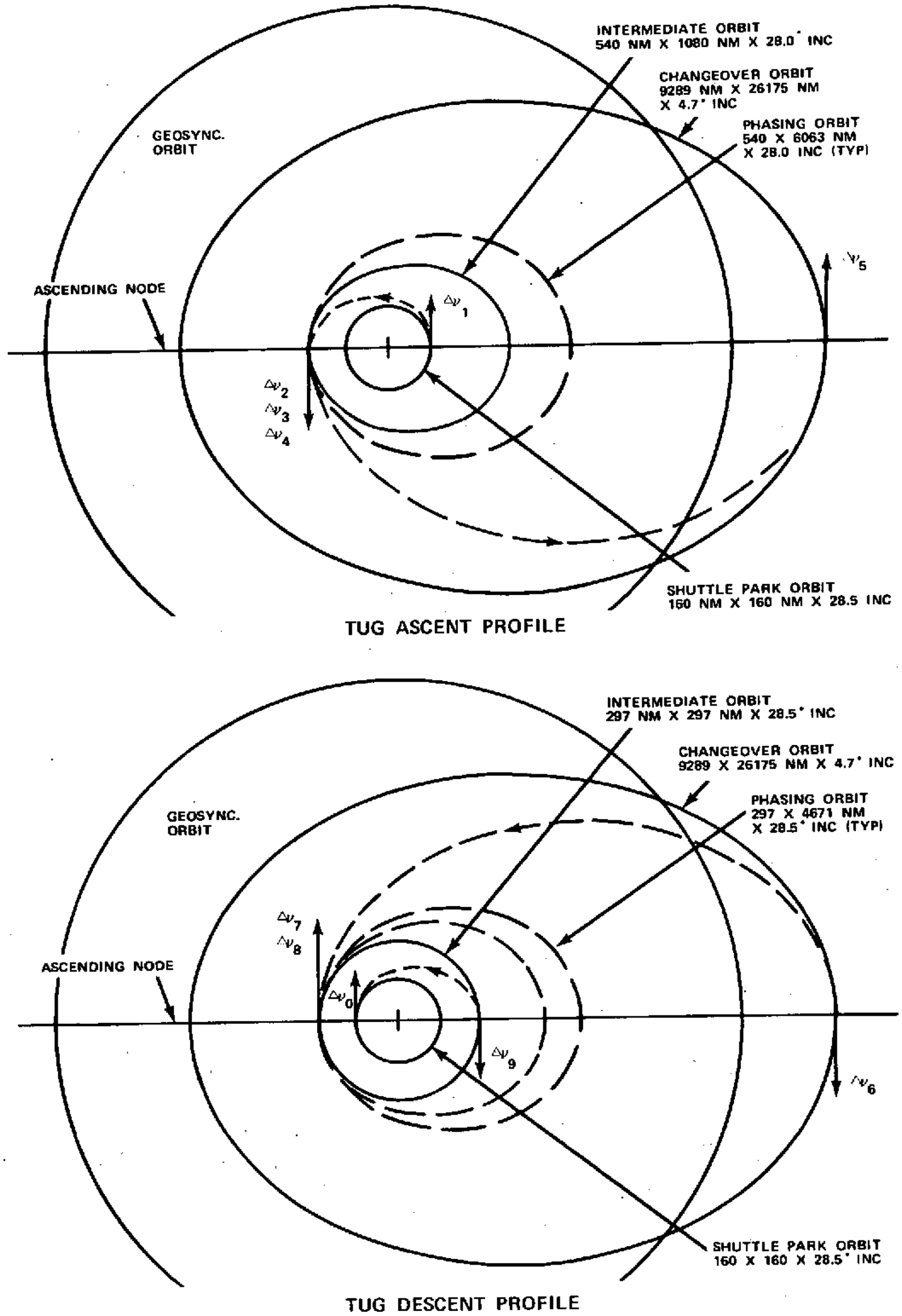

Figure 3-5. REFERENCE TRAJECTORY PROFILE

Tug burns at apogee of transfer orbit to complete plane change and inject on changeover orbit. If the Tug navigation and guidance system is operating normally, Tug and SEPS will be within active LADAR range. Either vehicle can be the active rendezvous partner. After final closure and docking of Tug with SEPS, up payloads on Tug are exchanged with down payloads on SEPS.

If an intermediate orbit is to be retrieved during Tug's return to Orbiter, the Tug burns at apogee of changeover orbit to a phasing orbit for retrieval of intermediate payloads and then burns to rendezvous with the retrieval payload. This requires that the line of nodes of the intermediate orbit be aligned with the nodal line of the changeover orbit. This can be arranged for one intermediate orbit. In general it cannot be expected that the line of nodes of several intermediate orbits will be coincident. In the case of an elliptical intermediate orbit, it is also necessary that the major axis lie in the line of nodes; any other orientation of either the nodes or major axis requires excessive Tug ΔV. Multiple intermediate orbit retrievals by Tug will occur infrequently.

After retrieval of the intermediate payload, Tug burns to transfer to the Shuttle parking orbit. A phasing orbit maneuver by either Shuttle or Tug may be required. Shuttle returns to ground with Tug and retrieved payloads.

Following exchange of payloads with Tug, SEPS begins transfer from changeover orbit to geosynchronous orbit. After 50 days, SEPS deploys the first up payload in geosynchronous orbit. In geosynchronous orbit, SEPS assumes an orbital taxi role and spaces the individual payloads around the orbit at their intended longitudes. SEPS takes 4 to 7 days between deployment of payloads in geosynchronous orbit if ion propulsion is the only thrusting used. SEPS is then free to begin the next sortie.

Detailed discussions of the mechanics of payload transfers and other related subjects are contained in other sections of this volume; therefore, they were omitted in the above discussions.

Descriptions of SEPS self-servicing and its potential for self-maintenance capability along with payload handling descriptions indicate the near universal adaptability of the SEPS manipulator systems to onorbit servicing.

3.3 RELATED MISSION PLANNING AND ENVIRONMENTAL CONSIDERATIONS

3.3.1 Comparison of Tug Ascent and Descent Profiles for Three Major Classes of Sortie Profiles

Basically all of the Tug profiles fit into three cases:

1. Tug has sufficient performance capability to carry the multiple payload package to geosynchronous orbit. SEPS taxies individual payloads to their specific mission locations.

2. The changeover orbit is an inclined circular orbit.

3. The changeover orbit is an inclined elliptical orbit.

In all cases, the Tug must ascend from a low-earth parking orbit to a target orbit (either geosynchronous or changeover), rendezvous with the target, perform specified operations while coasting in the target orbit, return to the parking orbit and rendezvous with the waiting Shuttle orbiter. For the geosynchronous SEPS mission, the target orbit will always have an inclination less than that of the parking orbit. Independent of the type of target orbit, the flight profile (beginning with the ignition of the Tug rocket engine in the parking orbit) will probably consist of six major burns, with additional terminal maneuvers performed during each rendezvous and short correction burns added to adjust the apogees or perigees of the phasing orbits and transfer conics. Only the major burns are considered in this discussion.

3.3.1.1 Basic Flight Profiles

Three burns are used in the ascent portion of the flight; Figure 3-6 illustrates the ascent sequence.

For maximum efficiency, each burn is performed at the line of intersection between the parking and target orbital planes. The length of the first burn is used to adjust the size (and thus, the period) of the upphasing orbit so that the Tug will arrive at the apogee of its up-transfer conic at the same time as the target. The length of the second burn must provide the precise ΔV needed to produce an up-transfer conic with an apogee altitude equal to the altitude of the target orbit. For gross adjustments in the rendezvous time, the Tug may need to coast for one or more additional revolutions in its parking orbit before initiating the first burn.

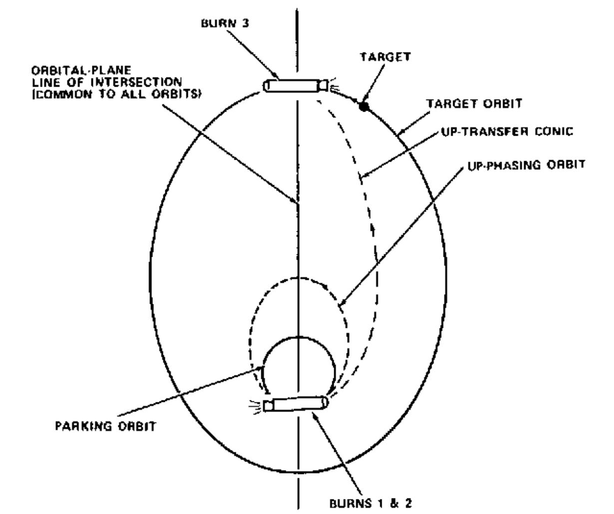

Figure 3-6. ASCENT PROFILE

For maximum efficiency, some amount of the required plane change is made on each burn. In the practical cases of interest here, the target-orbit altitudes are high enough so that the entire plane change can be made at the apogee of the transfer conic (third burn on ascent) with a negligible increase in total ΔV. It should be noted that the inclusion of an up-phasing orbit in the flight profile will reduce the gravity losses by splitting into two parts the burn required to obtain target-orbit altitude. The optimum split may not produce a phasing orbit with the desired period; however, the increase in losses produced by a nonoptimum split are negligible in practical cases. It is important that the nominal period of the up-phasing orbit be at least twice the period of the parking orbit. Then, in the event the first Tug burn cannot be made at the nominal time, the Tug can simply coast for one revolution in the parking orbit and reduce the up-phasing orbit.

When the target orbit is elliptical, its line of apsides must be aligned* with the orbital plane's line of intersection, and the rendezvous must occur at the apogee of the target orbit. In this case, a rendezvous opportunity occurs only once per target-orbit period, at the time the target reaches its apogee. The ΔV penalty for rendezvousing at the perigee of the target orbit is excessive for target orbits of substantial eccentricity, particularly when a plane change is required. When the target orbit is circular, there exists a rendezvous opportunity every half period, when the target crosses the orbital-plane's line of intersection. This is an advantage for the circular target orbit as regards the operational flexibility of the Tug's flight profile. In general, however, circular target orbits are less efficient and therefore require more total sortie time than elliptical orbits.

As in the case for the ascent, three major burns are used in the descent; Figure 3-7 illustrates the descent sequence.

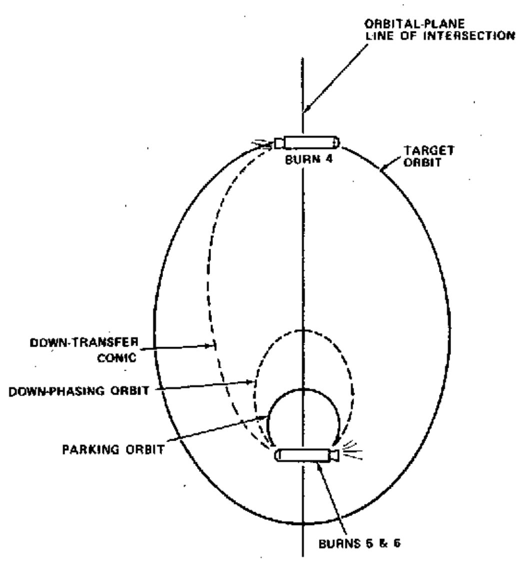

Figure 3-7. DESCENT PROFILE

Throughout the discussion the conditions set forth are those necessary for a minimum Tug ΔV. Deviations from these ideal conditions produce penalties which will be discussed later.

For the descent, the total plane change can be made on burn No. 4. The length of the fifth burn is used to produce a down-phasing orbit so that the Tug will return to the perigee of this phasing orbit at the same time as the waiting orbiter. The sixth burn accomplishes the final rendezvous.

When the target orbit is substantially eccentric, the deboost burn (burn No. 4) should only be made at the apogee of the target orbit. Therefore, unless the Tug can return immediately after rendezvousing with the target, it must coast for an entire period in the target orbit until it returns again to the apogee. When the target orbit is circular, a deboost opportunity occurs every half-period.

It should be noted that, due to the earth's oblateness, the orbits experience periodic and secular perturbations which alter their shapes and relative orientations. The magnitudes of these perturbations must be considered in the definitions of operational trajectories, but they are small enough to have no significant effect on the comparisons being made in this discussion.

3.3.1.2 Launch Opportunities and Windows

Because of the unique characteristics of the geosynchronous target orbit, there is a continuum of Shuttle launch opportunities for this orbit. Since the angular rate of the target in a geosynchronous orbit is equal to the earth's rotational rate, and since the inclination of the geosynchronous orbit is zero, the relative orientations of the Tug's parking orbit and the phasing relationship of the Tug and target will be identical regardless of the launch time.* The Tug must, however, wait in the orbiter parking orbit or a phasing orbit for periods up to 14 hours depending upon the geosynchronous delivery longitude as illustrated in Figure 3-8.

**It is assumed here that the shuttle ascent trajectory is always nominal and independent of launch time.*

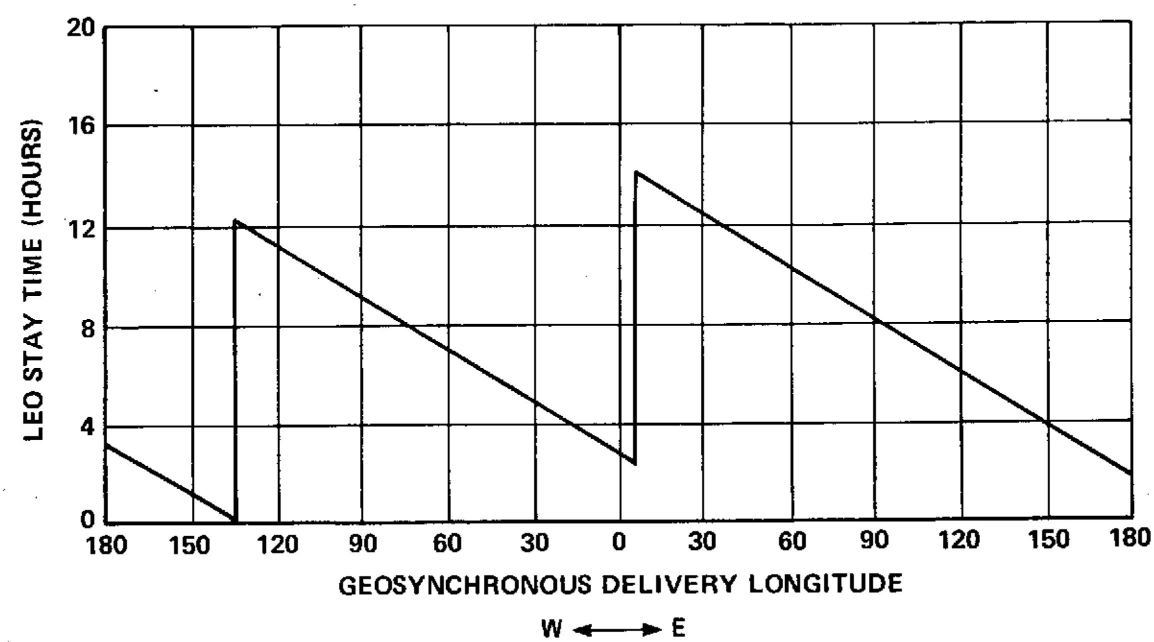

Figure 3-8. LEO STAY TIME VERSUS DELIVERY LONGITUDE IN GEOSYNCHRONOUS ORBIT

In order to make quantitative comparisons, two inclined changeover target orbits (one elliptical and one circular) have been selected from a geosynchronous SEPS System Operational Profile, defined by NSI for the 1981-1991 time period. Both changeover orbits were selected as orbits which could be reached by the Tug with a one-way ΔV of 3390 meters/second. The elliptical orbit was selected from a family of <u>unconstrained</u> changeover orbits as the one requiring the minimum SEPS ΔV, and the circular orbit was selected from a family of <u>constrained</u> <u>circular</u> changeover orbits as the one requiring the minimum SEPS ΔV. Basic data concerning these two changeover orbits (as well as the geosynchronous target orbit and the parking orbit) are contained in Table 3-2.

Since both changeover orbits have nonzero inclinations, there is a ΔV penalty for launching at a nonoptimum time. In each case, there is one launch opportunity in each 24-hour period when the minimum ΔV can be attained. The basic reason for the ΔV penalties at other launch times is the increase which

Table 3-2. ORBITAL CHARACTERISTICS

	GEOSYNCHRONOUS	CIRCULAR CHANGEOVER	ELLIPTICAL CHANGEOVER	TUG PARKING ORBIT
Apogee Radius	42,164 km	20,000 km	59,332 km	6674 km
Perigee Radius	42,164 km	20,000 km	16,723 km	6674 km
Inclination	0.00 deg	13.00 deg	8.22 deg	28.5 deg
2-Way Tug ΔV (parking orbit to/from target)	8468 m/s	6780 m/s	6780 m/s	--
2-Way SEPS ΔV (target to/from geosynchronous)	0	3640 m/s	2720 m/s	--
Orbital Period	23.93 hours	7.82 hours	20.50 hours	1.51 hours

results in the angle between the planes of the parking orbit and target orbit. For elliptical changeover orbits, there is an additional effect which adds to the penalty; the line of intersection of the parking and changeover orbital planes rotates, forcing the rendezvous with the target to occur at a point other than the apogee of the target orbit. Figure 3-9 illustrates the relevant parameters of the launch geometry.

The parking orbit nodal shift, $\Delta\Omega$, is related to a launch time delay, Δt_L, (in hours) as follows:

$$\Delta\Omega = 15.04 \ \Delta t_L \text{ degrees}$$

The total angle between the parking orbit and changeover orbit planes, σ, is related to $\Delta\Omega$ as follows:

$$\sigma = \cos^{-1}[\cos i_c \cos i_p) + (\sin i_c \sin i_p) \cos\Delta\Omega]$$

where i_c and i_p are the inclinations of the changeover and parking orbits, respectively.

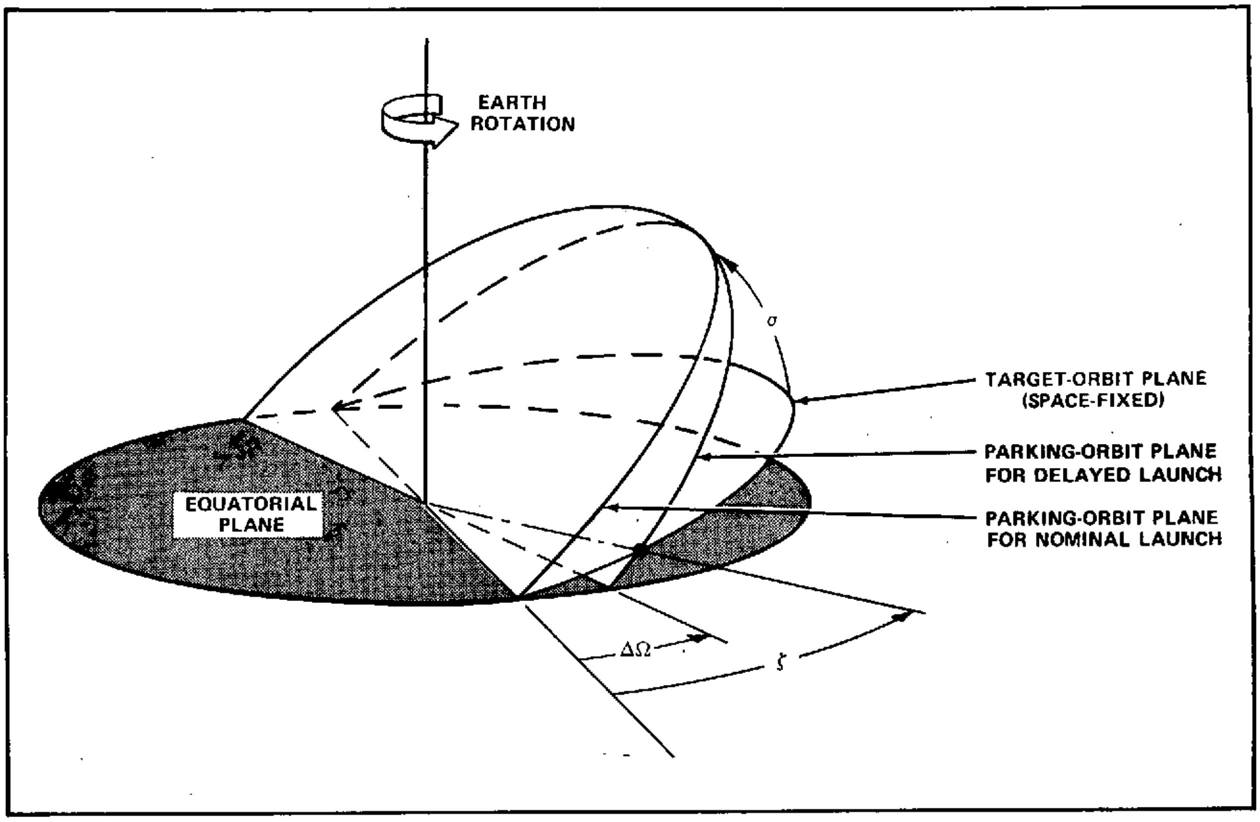

Figure 3-9. EFFECT OF DELAYED LAUNCH ON ORBITAL-PLANE GEOMETRIES

The rotation of the line of intersection between the parking orbit and changeover orbit planes, ζ, (measured in the plane of the changeover orbit) is related to $\Delta\Omega$ as follows:

$$\zeta = \cos^{-1} \frac{1}{\sin\sigma} [\cos i_c \sin i_p) \cos\Delta\Omega - (\sin i_c \cos i_p)]$$

Table 3-3 gives $\Delta\Omega$, σ, and ζ for several launch time delays for each changeover orbit.

Table 3-3. EFFECTS OF LAUNCH DELAY

		CIRCULAR CHANGEOVER ORBIT (i = 13 deg)		ELLIPTICAL CHANGEOVER ORBIT (i = 8.22 deg)	
Δt_L (hr)	$\Delta\Omega$ (deg)	σ (deg)	ζ (deg)	σ (deg)	ζ (deg)
0	0	15.50	0	20.28	0
0.5	7.52	15.70	12.78	20.38	10.39
1.0	15.04	16.28	25.99	20.66	20.52
1.5	22.56	17.18	38.14	21.12	30.45
2.0	30.08	18.35	49.30	21.75	40.21

The forward rotation of the orbital plane intersection line for late launches provides some phasing compensation; for the elliptical changeover orbit there is actually an overcompensation, and when the launch is delayed the period of the up-phasing orbit must be increased. For launch time delays of up to 2 hours, there are no phasing problems for either of the changeover orbits which cannot be corrected by the adjustment of the up-phasing orbit's period.

Figure 3-10 gives a comparison of the ΔV penalties incurred for off-nominal launch times for the two example changeover orbits.

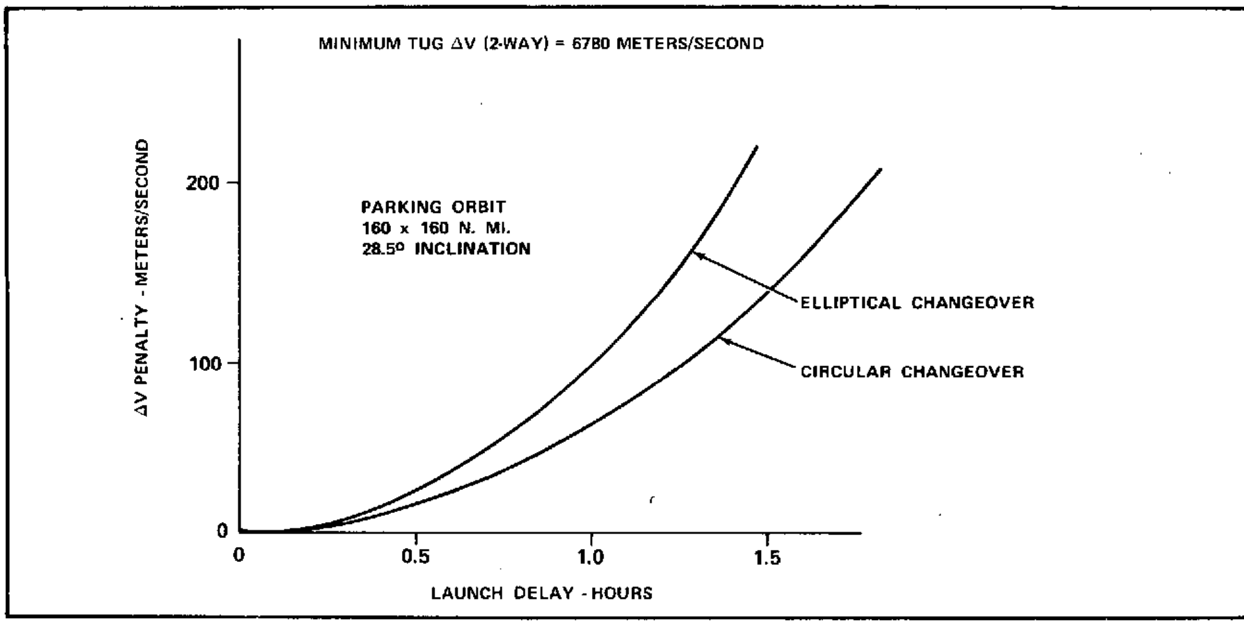

Figure 3-10. LAUNCH DELAY PENALTIES FOR INCLINED CHANGEOVER ORBITS

Although some small portion of ΔV penalties shown on Figure 3-10 is due to a change in the phasing relationships caused by a late launch, the phasing adjustments (which have been discussed) reduce this portion to an insignificant amount.

3.3.1.3 Time Away from the Shuttle Orbiter

A small advantage of the example circular changeover orbit is that the Tug is away from the Shuttle orbiter for a shorter period of time. If the rendezvous maneuvers and orbital operations which the Tug must perform in the circular changeover orbit require no more than 3.9 hours, it can make the deboost burn one-half period after it has injected into the changeover orbit.

Assuming that the up-phasing and down-phasing orbital periods are about 3 hours each, the Tug would be away from the Shuttle orbiter a total of only about 14 hours. For the example elliptical changeover orbit, however, the Tug must spend about 20.5 hours in the changeover orbit and will be away from the Shuttle orbiter for a total of about 43 hours.

This difference in time away from the orbiter of 29 hours is a distinct advantage of the circular changeover orbit. To reduce this difference, the Tug would have to initiate its deboost burn as soon as possible from the elliptical changeover orbit and take the ΔV penalty associated with a burn that is not made on the line of intersection of the changeover and parking orbital planes. For example, a wait in the elliptical changeover orbit of 2 hours would result in a ΔV penalty of about 500 meters/second. To keep the penalty this small, an additional burn would have to be inserted into the profile. Figure 3-11 illustrates the geometry produced by the off-nominal deboost burn.

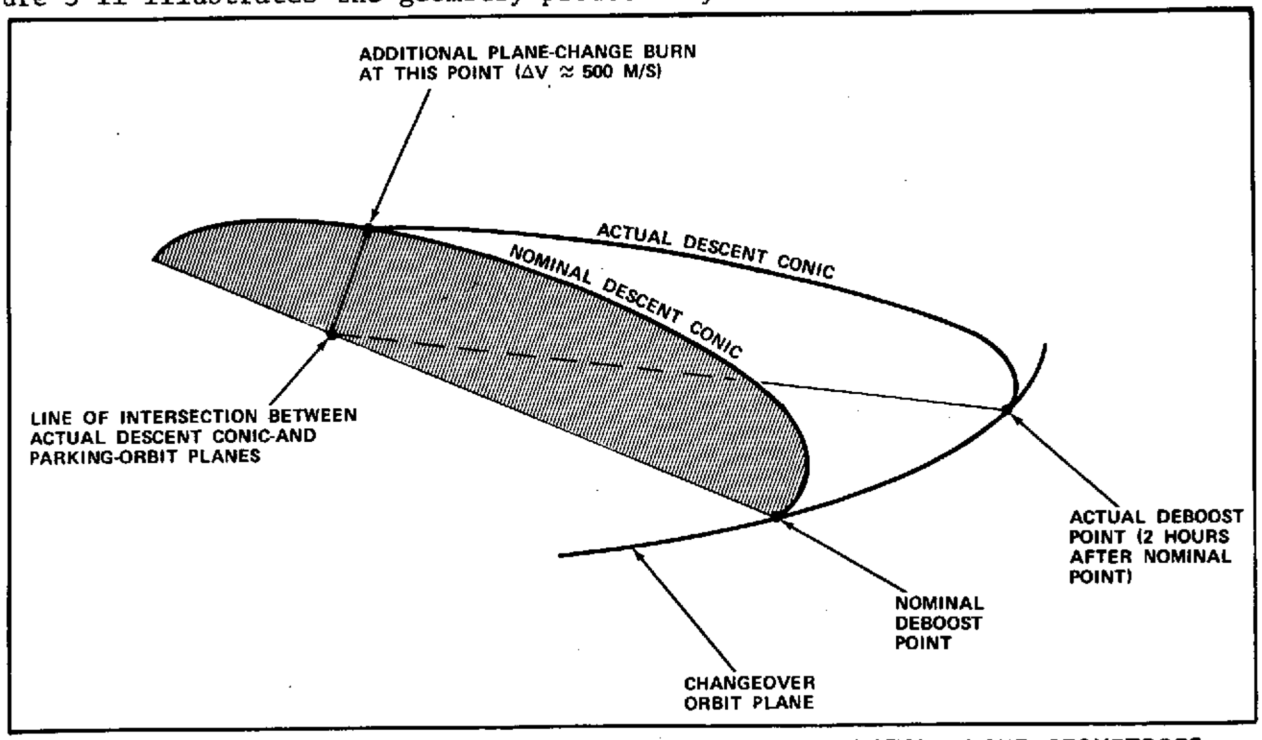

Figure 3-11. EFFECT OF OFF-NOMINAL DEBOOST ON ORBITAL-PLANE GEOMETRIES

When the target orbit is the geosynchronous orbit (SEPS in geosynchronous taxi mode only), the Tug can deboost after remaining in the target orbit only one-half period (about 12 hours). In this case, the total time away from the Shuttle orbiter is about 28.5 hours (about half-way between the times for the elliptical and circular changeover orbits) plus up to 14 additional hours for some target longitudes.

It should be pointed out that the particular examples chosen for the circular and elliptical changeover orbits result in the maximum difference in the Tug-time away from the orbiter. For those missions requiring higher Tug ΔV's, the periods of both the optimum elliptical changeover orbits and the constrained circular changeover orbits become longer and move closer together. In that period (that is, when SEPS is a taxi only), the elliptical changeover orbit becomes circular and equal to the circular changeover orbit, both being geosynchronous. In a particular case, the selection of the optimum (elliptical) changeover orbit or the constrained circular changeover orbit would be made by trading the increased Tug sortie time against the reduction in the required SEPS ΔV. Figure 3-12 shows these parameters as a function of the Tug two-way ΔV requirement. For any given Tug ΔV, Figure 3-12 shows the cost in mission time, and the reduction in SEPS ΔV and SEPS thrust time to be obtained by opting for an unconstrained elliptical changeover orbit instead of a circular changeover orbit. In a theoretical sense, as shown on Figure 3-12, when both orbits become geosynchronous, there is a 12-hour difference in Tug mission time because the Tug is coasting for an entire revolution in the elliptical changeover orbits and only a half revolution in the circular changeover orbits. In the practical sense, because both orbits are identical, Tug could retrogress at the half revolution point.

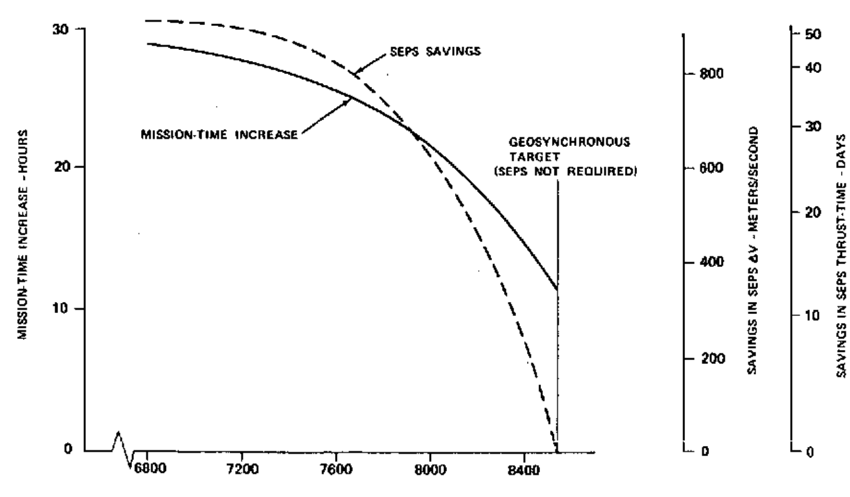

Figure 3-12. MISSION-TIME INCREASE AND SEPS SAVINGS (WITH ELLIPTICAL CHANGE-OVER ORBITS)

3.3.2 SEPS Potential for Operation Into Intense Radiation Zones of the Van Allen Belt

As an example of SEPS capabilities in this area, NSI investigated the accomplishment of the mission model with a recoverable Interim Upper Stage (IUS), no Tug, and a SEPS with radiation resistant, self-annealing solar cells. This results in a requirement for elliptical changeover orbits that have perigees deep in the high intensity zone of the Van Allen belt. The results of the analysis indicated an STS comprised of Shuttle, a recoverable IUS, and a 100 kw SEPS could accomplish the mission model with only 10 more Shuttle flights than an STS comprised of Shuttle, expendable IUS, and Tug.

As an alternate to radiation resistant cells, the effect on trip time of rolling up the array for protection in the high intensity radiation zones was investigated by NSI in a related study. For this analysis, power available to the SEPS thruster subsystems at the beginning of the sorties was 21 kw. Radiation damage effects are included.

When SEPS operates between low-energy elliptic changeover orbits and geosynchronous orbit (GSO) the SEPS thrust can be terminated at low altitudes where it is relatively ineffective in changing the orbit's size and inclination. The total SEPS ΔV requirement for a transfer between ESO and a specified changeover orbit will thereby be reduced. When the SEPS thrusters are turned off, the solar panels can be rolled in to prevent the substantial radiation damage which would occur at the low altitudes. With reduced radiation damage, the SEPS thrust remains high; and the total mission time is actually reduced from that obtained when there are no thrust terminations or solar panel roll-ins.

The Simplex version of the MOLTOP computer program (with the SSL radiation model) has been used to determine the optimum changeover orbits and the associated SEPS descent trajectories (for a typical SEPS T/M) for a range of chemical stage ΔV capabilities. At radii below 20,000 km, SEPS thrust termination and solar panel roll-in were simulated. The starting orbit for the chemical stage was a 220 nautical mile, 28.5 degree-inclined Shuttle orbit. The chemical-stage ΔV capability (one-way) was varied between 2400 and 3000 meters/second. The SEPS started in GSO with 13.44 kw of beam power, a specific impulse of 3000 seconds, an undamaged solar array, and a thrust/mass of 2.0435×10^{-4} m/s^2.

Figures 3-13 through 3-16 show several mission parameters versus chemical stage ΔV. Figure 3-13 shows the optimum changeover orbit parameter values. Figure 3-14 shows the SEPS ΔV's and times required for the descents from GSO to the optimum changeover orbits. Figure 3-15 shows the number of SEPS thrust terminations required for the descents to the optimum changeover orbits. Figure 3-16 shows the percentage reductions in SEPS exhaust power caused by radiation damage during the descents.

The SEPS ΔV and mission time values in Figure 3-14 can be used to estimate the SEPS trajectory parameters where the SEPS has a different T/M than the one used in the MOLTOP simulations, and where an ascent trajectory is desired instead of a descent trajectory. A particular transfer of interest is the delivery of a 3857-kg payload to GSO from a 220 nautical mile, 28.5-degree inclined Shuttle orbit. The chemical stage is a transtage having an inert mass of 2117 kg, a maximum propellant usage of 14586 kg, and a specific impulse of 308.2 seconds. After taking the payload to the optimum changeover orbit, the transtage must return to the Shuttle orbit. The transtage ΔV requirement for this mission is computed to be 2565 meters/second. For this ΔV, Figure 3-13 shows the optimum changeover orbit to have an apogee radius of 44,000 km, a perigee radius of 7,300 km, and an inclination of 22.75 degrees. The SEPS which meets the transtage in the changeover orbit has to have enough propellant to deliver the payload to GSO and to return to some changeover orbit for refueling. This SEPS will also have some radiation damage at the time it takes the payload from the transtage. Typical estimates for the SEPS propellant loading and percentage reduction in undamaged exhaust power are 528 kg and 10 percent, respectively. The resulting T/M of the SEPS/payload combination in the optimum changeover orbit is 1.461×10^{-4} m/s^2. Figure 3-14 shows that a SEPS with an initial T/M of 2.0435×10^{-4} m/s^2 requires about 143 days to descend to the optimum changeover orbit associated with a chemical stage ΔV of 2565 m/s. The descent time required for the SEPS with a lower T/M is approximated as:

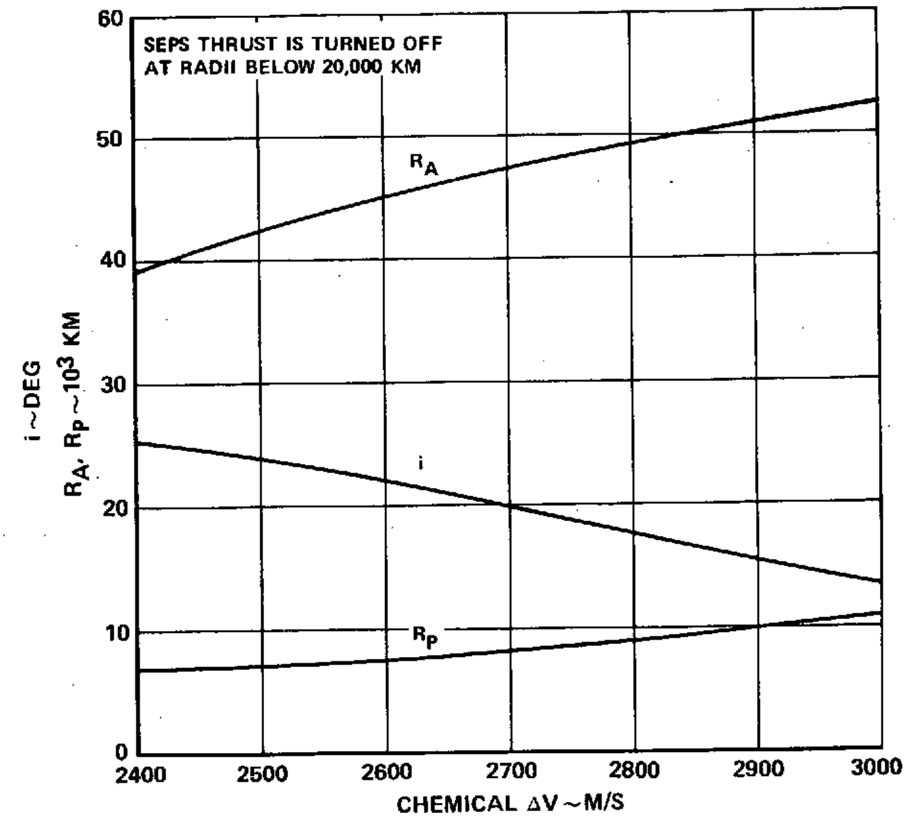

Figure 3-13. OPTIMUM CHANGEOVER ORBITS (SHUTTLE ORBIT ALTITUDE = 220 N. MI.)

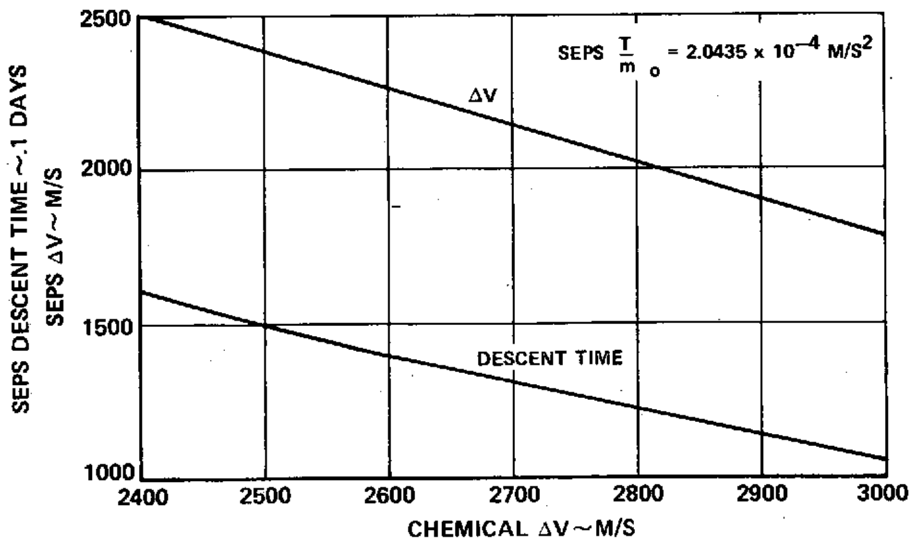

Figure 3-14. SEPS PERFORMANCE

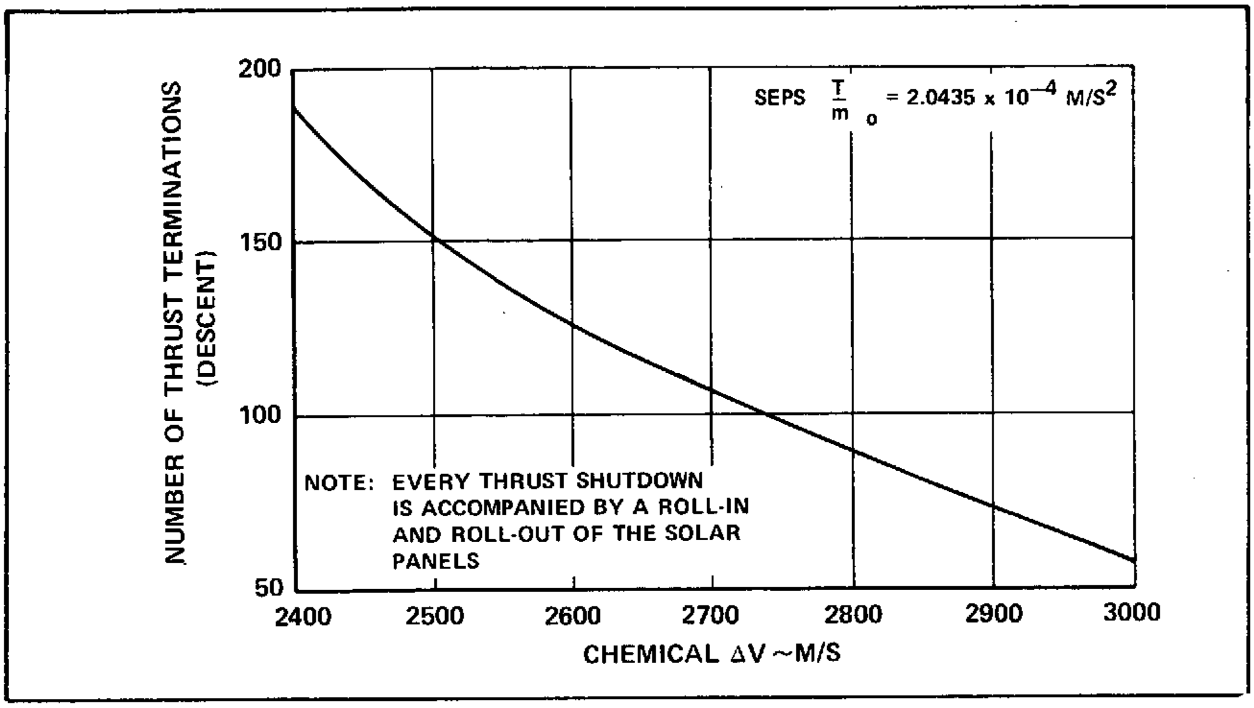

Figure 3-15. NUMBER OF THRUST SHUTDOWNS REQUIRED DURING SEPS DESCENT

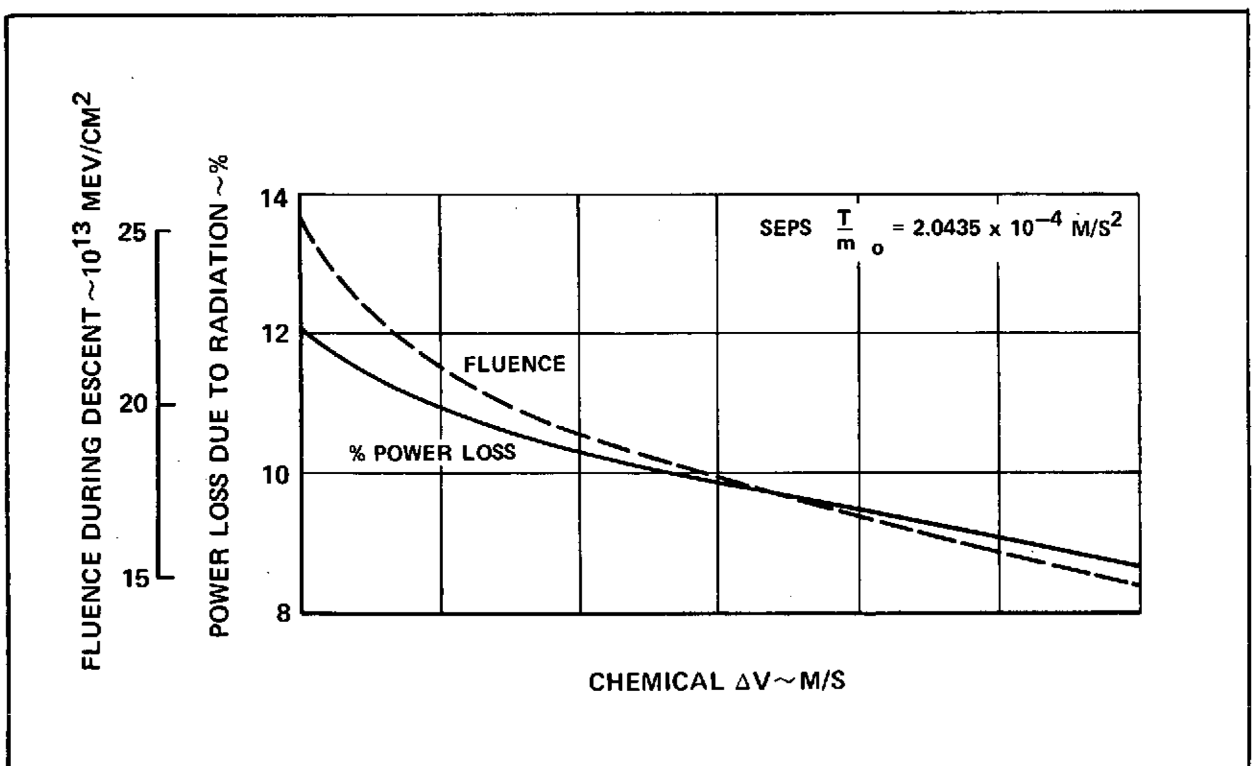

Figure 3-16. RADIATION DAMAGE EFFECTS

$$t_{DESCENT} \approx \frac{2.0435 \times 10^{-4}}{1.461 \times 10^{-4}} \times 143 \approx 200 \text{ days}$$

Previous analyses have shown that in the presence of radiation the SEPS ascent time between a given changeover orbit and GSO is greater than the descent time. Table 3-4 contains the estimates for the delivery of the 3857 kg payload to GSO.

Table 3-4. MISSION PARAMETERS FOR TRANSTAGE/SEPS DELIVERY OF 3857 KG PAYLOAD TO GSO FROM 220 N MI SHUTTLE ORBIT

Payload	3857 kg
Transtage ΔV	2565 m/s
Changeover Orbit	
Apogee Radius	44,000 km
Perigee Radius	7,300 km
Inclination	22.75 deg
SEPS	
Initial T/M	1.461×10^{-4} m/s^2
Ascent ΔV	2315 m/s
Ascent Propellant	426 kg
Ascent Time	≈ 212 days
Number of Thrust Terminations	≈ 200

3.3.3 Parametric Analysis of Times for Orbital Taxiing in Geosynchronous Orbit

In order to provide estimates of taxiing time around the GSO, the following data from a simplified parametric study are presented. The actual sortie terminal approaches that NSI investigated used optimum steering laws.

The data were generated by using a spiralling technique for shaping the trajectory profiles. These spiral trajectories were simulated by directing the SEPS thrust vector along or opposite the velocity vector depending on whether altitude is to be increased or decreased. The results indicate that SEPS can maneuver a 3000-pound payload from any geosynchronous longitude to any desired longitude in a maximum time of 11 days at a cost of less than 50 pounds of SEPS propellant.

The assumptions used in this study are as follows:

- The initial gross mass of SEPS is 1542 kg
- Thrust and Isp are 0.9136 newtons and 3000 seconds
- Continuous thrust is applied until the desired longitude shift has been achieved.

The data presented in Figure 3-17 were generated by starting the SEPS transfer maneuver 180 degrees away from the desired longitude and directing the SEPS thrust along the negative velocity vector (retrograde) until the phase angle (longitude shift) was equal to 90 degrees. At this point, the SEPS has spiralled into an orbit lower than geosynchronous, and the thrust is reoriented to a point along the velocity vector (posigrade). Thrusting is continued in this direction until the phase angle between the SEPS and the desired longitude goes to zero. At this time, the SEPS is back near geosynchronous altitude. Three SEPS spiralling trajectories were generated for achieving 180-degree longitude shifts for 1000-, 2000-, and 3000-pound payloads. The data obtained from these trajectories were used to construct the graph presented in Figure 3-17.

An additional 2 days (an overly conservative estimate) are added to the flight time to account for the short stay time in the earth's shadow (no thrusting) and the time required to perform navigation updates prior to executing the terminal rendezvous sequence of maneuvers. The data were generated assuming the desired longitude was always ahead of the initial SEPS longitude, but these data are completely symmetrical for the case in which the desired longitude is

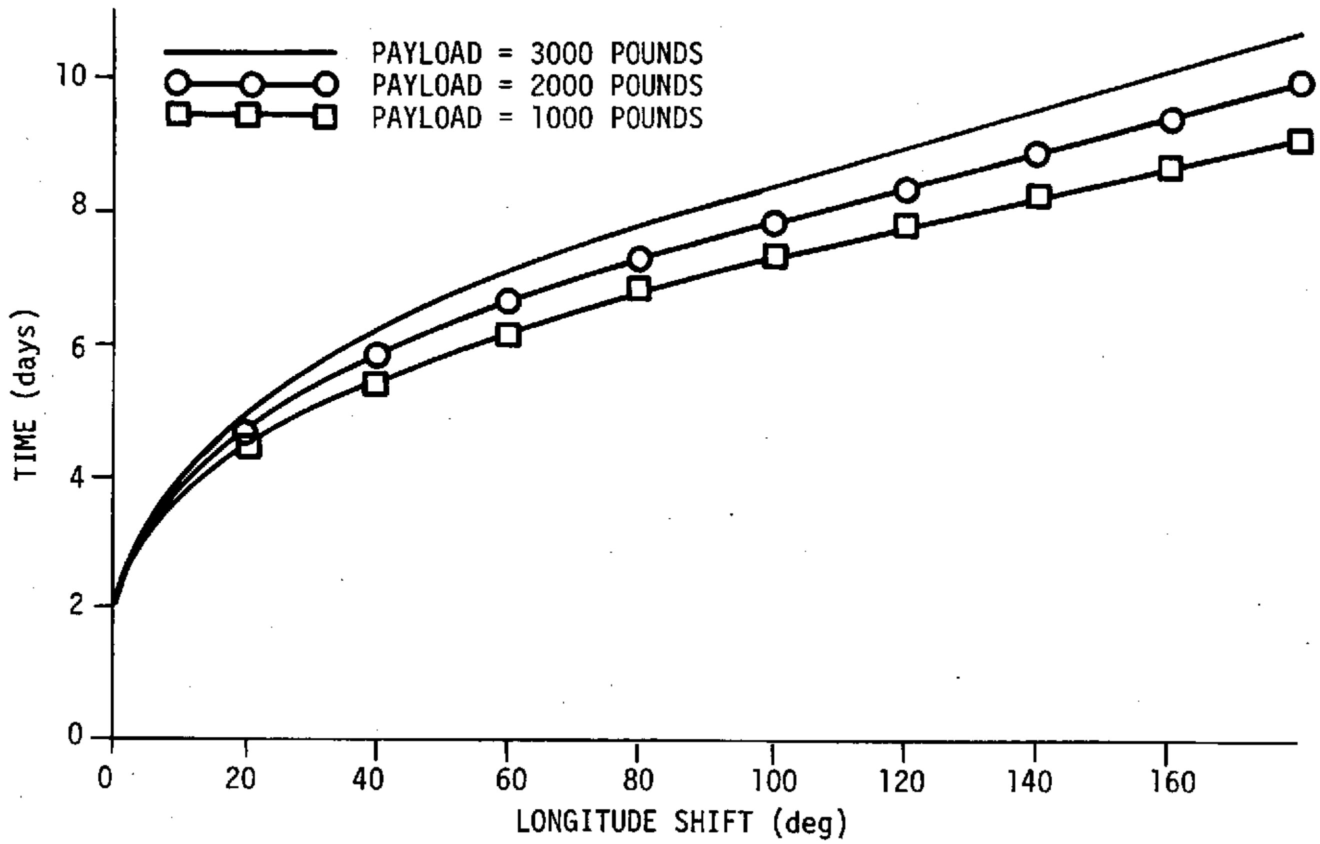

Figure 3-17. SEPS GEOSYNCHRONOUS PAYLOAD PLACEMENT TIME ESTIMATE

behind the initial SEPS longitude. The SEPS would simply spiral upward by thrusting along the velocity vector, and near the half-way point it would reverse the thrust direction.

An example of how to interpret the data presented in Figure 3-17 will be given through an illustration. The time required of the SEPS to shift a 1000-pound payload through 120 degrees of longitude is approximately 8 days (Figure 3-17). The geometry selected to accomplish this longitude shift is illustrated by Figure 3-18. The SEPS begins retrograde thrusting at position 1 in geosynchronous circular orbit 120 degrees away from the desired longitude. After three days of retrograde thrusting, the SEPS arrives at position 2 (59 degrees closer to the desired longitude) and begins posigrade thrusting. After 3 more days of thrusting, the SEPS arrives at position 3. Position 3 represents a condition in which the SEPS is below the desired stationary longitude and 2 degrees behind. After about 1-1/2 days in a coasting (catch-up) mode, the SEPS would start the terminal rendezvous maneuvers.

3.3.4 Spaceflight Tracking Data Network (STDN) Coverage of Changeover Orbits

The unshaded area of Figure 3-19 shows STDN coverage of objects that are at least a 5,586-nautical mile altitude. In order to avoid unnecessary duplication of figures, the ground tracks of three elliptical changeover orbits are plotted on the earth's equator to illustrate the continuous coverage available. The positions of the ground track's starting longitude on the equator has no significance. The locations were simply chosen to avoid overlay of the ground tracks on the illustration. Since these changeover orbits will normally be planned to enhance direct communication into the flight control centers, the figure illustrates that there is no tracking or communications problem.

SEPS would require no addition to STDN. Figure 3-20 shows communications coverage at low orbit altitudes for Tug phasing orbits. In this figure the shaded areas represent areas of STDN coverage. Tug can be contacted for adequate periods on each orbital pass. For Tug ascent to changeover orbit trajectory corrections, STDN coverage is essentially as shown on Figure 3-19.

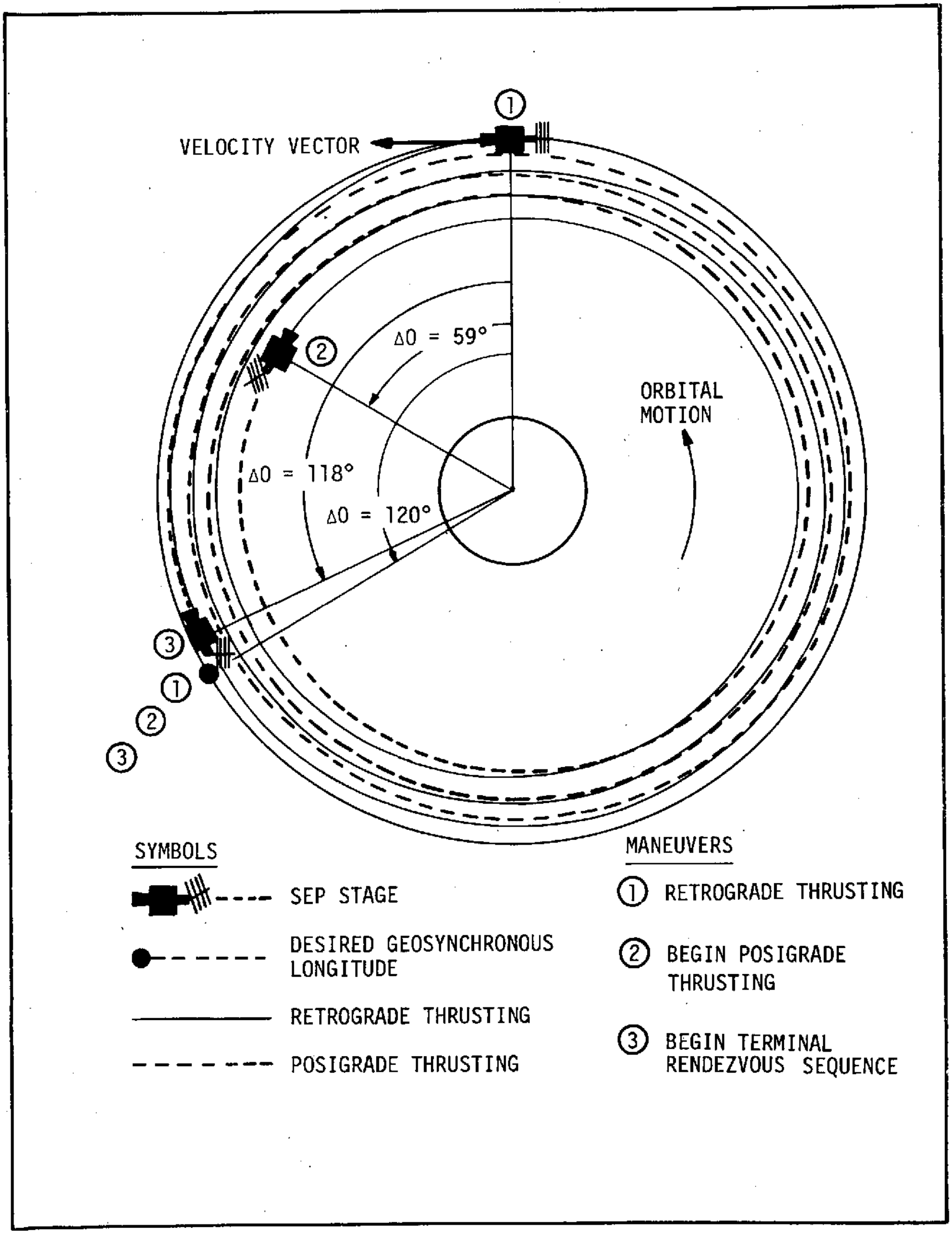

Figure 3-18. SEPS PAYLOAD PLACEMENT GEOMETRY

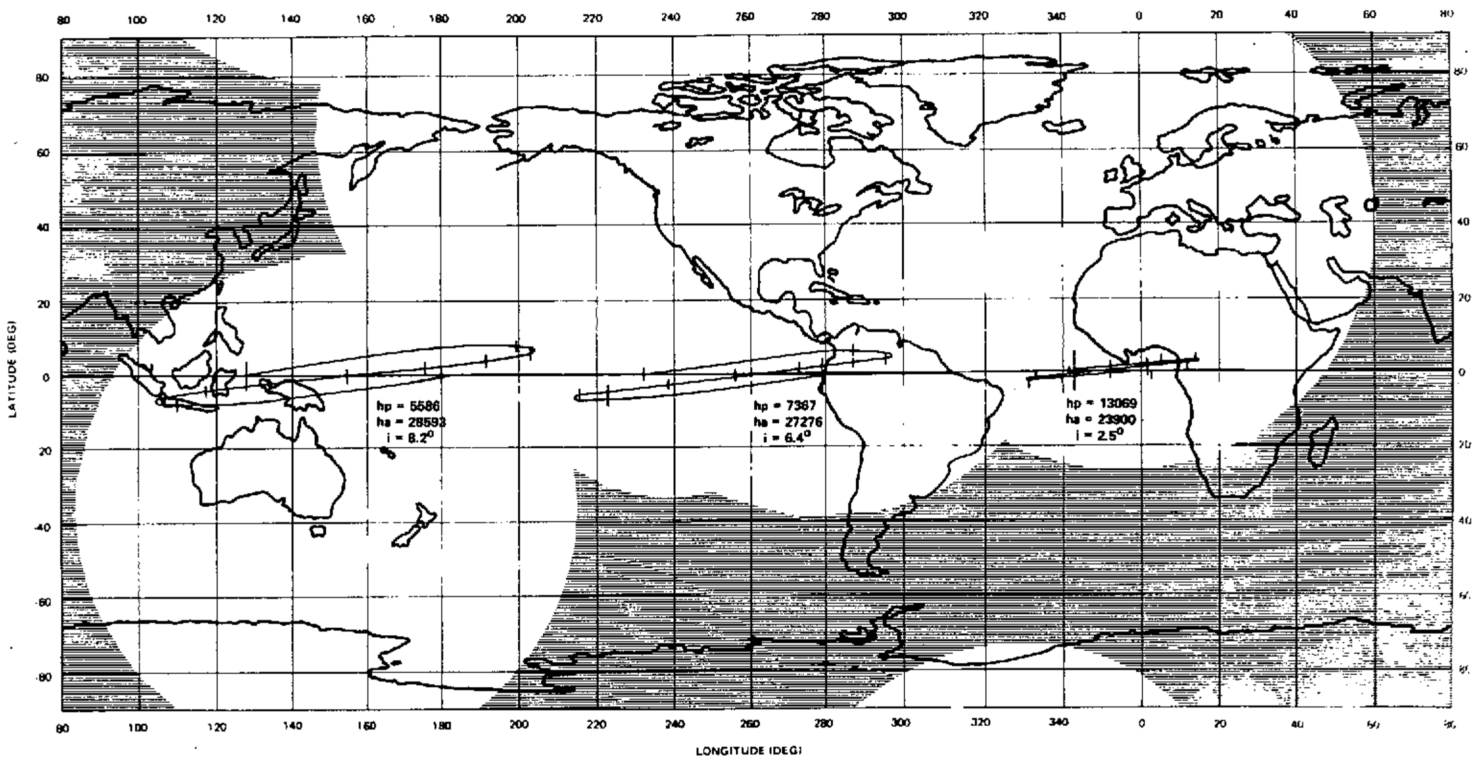

Figure 3-19. STDN COVERAGE FOR 5586 NAUTICAL MILE ORBITAL ALTITUDES

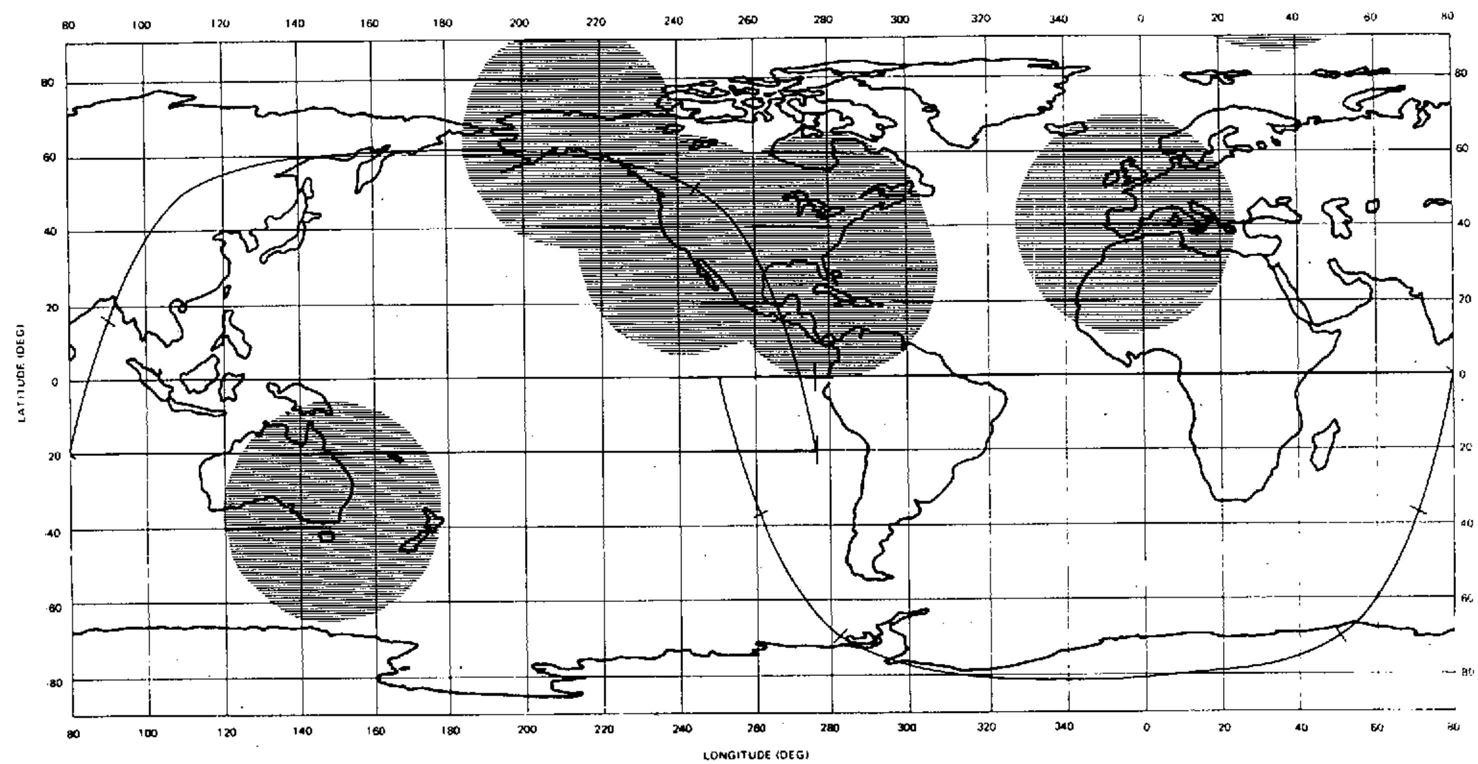

Figure 3-20. STDN COVERAGE FOR 494 NAUTICAL MILE ORBITAL ALTITUDES

3.3.5 Sunlight and Shadow Times for the Elliptical Changeover Orbits

Figure 3-21 shows the maximum percent of orbital periods that three different orbits (representative of those required to accomplish the reference mission model) will be shadowed. Even in the worst cases, ascending node locations can be chosen so that less than 7 percent of the orbital period is shadowed. Figure 3-22 shows the average yearly percent of orbital time periods that are shadowed. The figures illustrate that neither the payload transfer activity nor the SEPS propulsion time available is significantly influenced by shadow periods.

In NSI's analysis of shadow periods, seven orbits were considered. Three of the orbits were circular, with low altitudes and high inclinations. They are:

i = 99°	h = 494 n. mi.
i = 102°	h = 790 n. mi.
i = 103°	h = 920 n. mi.

Four of the orbits were elliptical, with low inclinations and large semimajor axes. They are*:

i = 0°	ha = 19,366 n. mi.	hp = 19,257 n. mi.	year 1986
i = 2.5°	ha = 23,900 n. mi.	hp = 13,069 n. mi.	year 1988
i = 6.4°	ha = 27,276 n. mi.	hp = 7,367 n. mi.	year 1989
i = 8.2°	ha = 28,593 n. mi.	hp = 5,586 n. mi.	year 1988

For each orbit, the time per revolution in the earth's shadow was computed, because the inertial positions of the orbit and the sun were varied. For the three circular orbits, the maximum shadow time per revolution is not a function of the orientation of the orbit. The maximum shadow time per revolution for these orbits occurs when the solar vector lies in the orbital plane. The values are:

	Max. Shadow Time/Rev.	Percent of Nodal Period
i = 99° , h = 494 nm	35.00 min.	0.338
i = 102°, h = 790 nm	34.92 min.	0.3021
i = 103°, h = 920 nm	34.88 min.	0.289

The argument of perigee is assumed to be zero in each case.

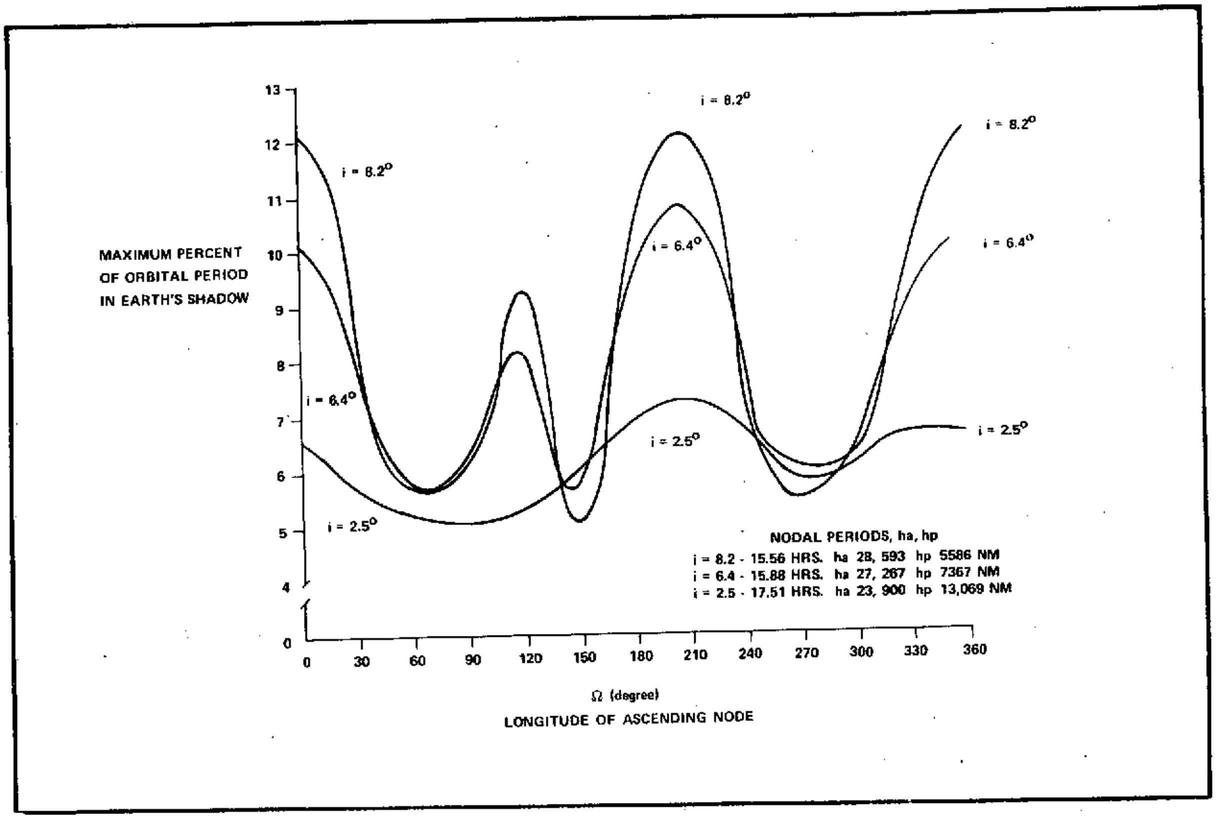

Figure 3-21. LONGITUDE OF ASCENDING NODE

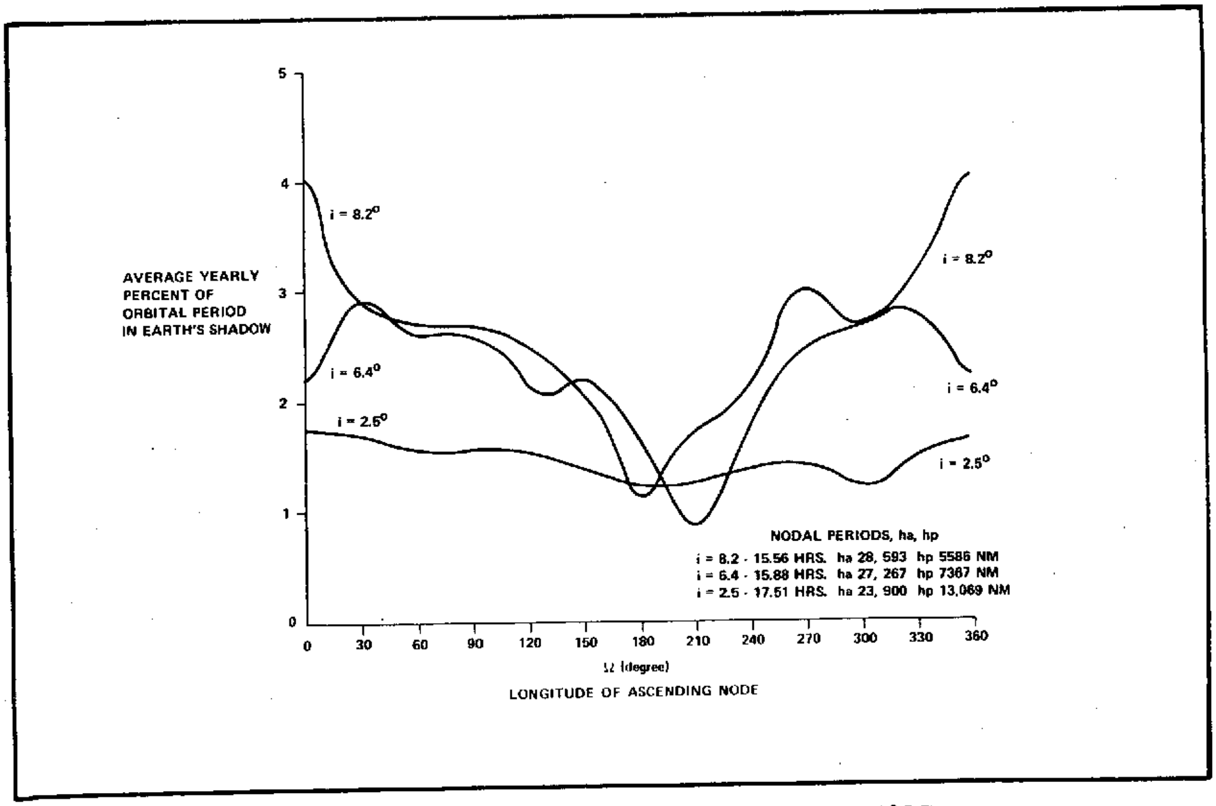

Figure 3-22. LONGITUDE OF ASCENDING NODE

For the elliptical orbits, (i>0), the maximum percent of an orbital revolution spent in the earth's shadow is shown on Figure 3-21 as a function of the right ascension of the ascending node, r. For the i = 0 orbit, the maximum time in the earth's shadow is 1.038 hours, or 5.56 percent of a nodal period.

It is possible to specify an orbit-sun orientation for each orbit, except the i = 99° case, which produces zero shadow time during a revolution regardless of the value of r. For the i = 99-degree orbit, the minimum percent of a nodal period spent in the earth's shadow is plotted versus r in Figure 3-23.

The shadow time per revolution depends upon the angle between the solar vector and its projection on the orbital plane. In the case of elliptical orbits, another important consideration is the orientation of the apogee of the orbit to the shadow zone. When the apogee of the orbit is in the shadow, the time spent in the shadow is a maximum. This situation causes the peaks in the curves of Figure 3-21 near r = 0° and r = 180°. Since the argument of perigee is assumed to be zero, when the ascending node coincides with an equinox, a date may be selected during which the apogee of the orbit lies in the midst of the shadow zone.

Another consideration in the selection of an orbit to minimize shadow time, is the fact that the maximum possible angle between the orbital plane and the ecliptic increases as the inclination increases. The result is an increase in both the range of ascending nodes and the times during the year which allow an orbit with zero shadow time to be achieved.

3.3.6 Operations Analysis to Define Program Support

This subject is discussed in some detail in Volume III.

3.4 EARTH ORBITAL TEST (EOT) SORTIE

The objectives of the EOT sortie are to demonstrate SEPS ability, using the GPME concepts evolved in this study, for:

- Multiple payload transfer
- Multiple payload retrieval

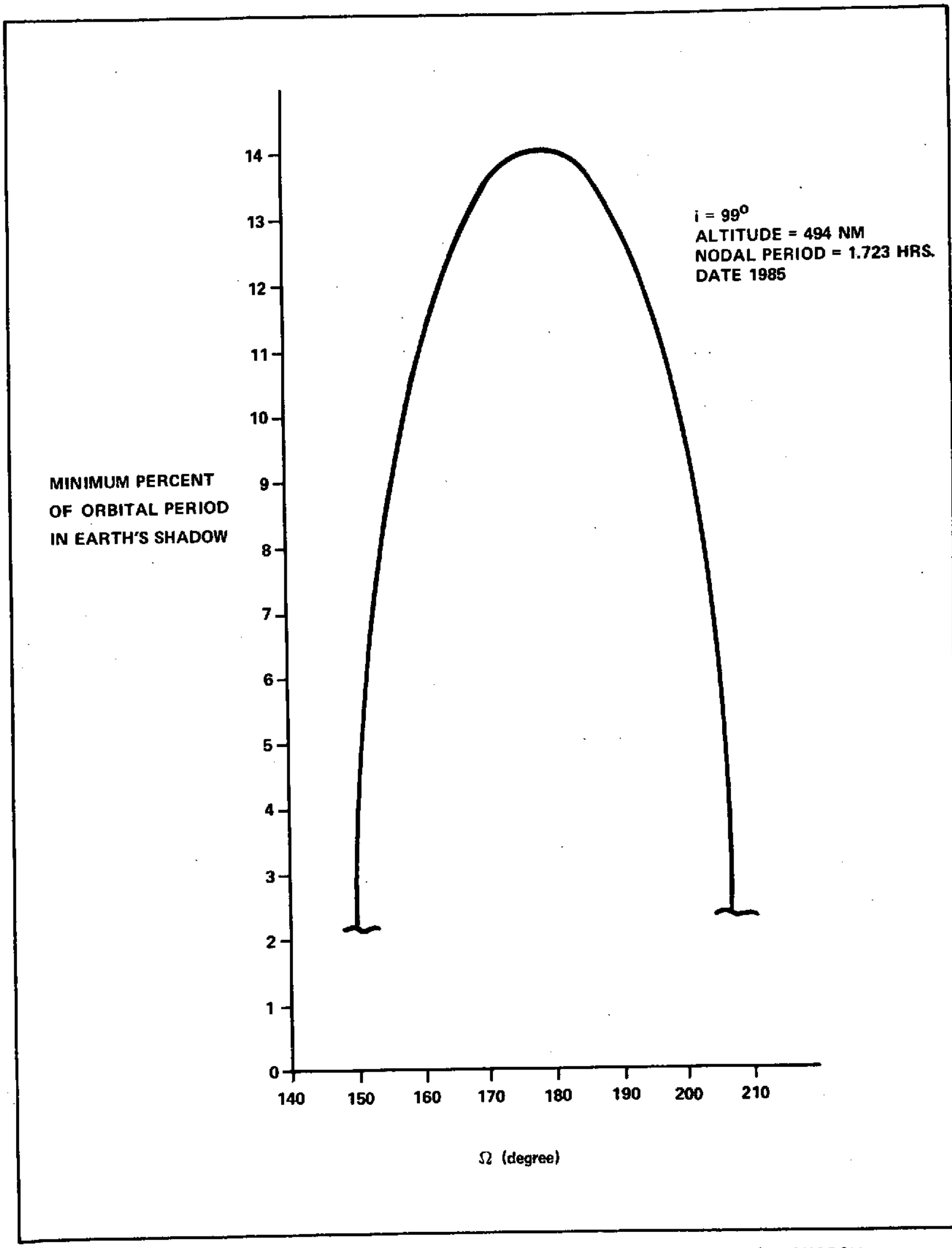

Figure 3-23. MINIMUM PERCENT OF ORBITAL PERIOD IN EARTH'S SHADOW

- Self-replenishment of expendables
- Near universal adaptability to payload servicing and maintenance functions
- Recovery of an unstabilized, noncooperative simulated satellite
- Validation of SEPSOC operational procedures
- Ability of solar arrays to function at partially deployed conditions; at each design screen voltage (Isp) level; and at each design power level contemplated
- Operation at simulated environmental extremes.

The SEPS thruster subsystem is relatively simple compared to chemical stage systems. Its attitude control, communications, navigation and guidance components and subsystems are, in general, proven elements or proven design concepts from spacecraft that will be operational before this SEPS test flight. NSI's assessment is that there is very small risk that the basic core SEPS vehicle with its manipulators will not perform in an acceptable manner even though it may not perform exactly as expected.

It is considered probable that the tests will show that many details such as: TV camera location on the manipulator arms; end effector to payload test device interface; payload to transport diaphragm attach details, and so forth, need design changes to improve operational flexibility or convenience, or both. Most of these changes can be expected in those items of GPME that are returned to earth at completion of each sortie.

In summary, NSI's assessment is that all technology areas are mature enough that SEPS No. 1 can be expected to be an acceptable operational vehicle even though certain retrofit modifications are performed on it during refurbishment at the end of its first mission cycle. The earth orbital test vehicle (SEPS No. 1) is, therefore, planned to become the first operational SEPS. The first sortie of SEPS No. 1 is planned such that intermediate orbital payloads that can be deployed independent of SEPS are the only operational payloads that are carried on this flight. The general test sortie sequence follows.

3.4.1 EOT Configuration and Payload

The configuration is comprised of an operational Shuttle, IUS, and integrated multiple payload package. The payload package consists of SEPS No. 1, operational intermediate orbital payloads, test payloads, and the full GPME set.

SEPS No. 1 is the full operational configuration described in subsection 6.2 of this volume and depicted on Figures 6-3 through 6-5.

The GPME is the full set recommended as a result of this study. It consists of:

STS GPME

- The standard payload transport shell and payload mounting diaphragms
- Transport shell to Orbiter adapter longeron that remains with Orbiter
- IUS-to-Orbiter adapter cradle (provided as baseline input to this study).

SEPS Unique GPME

- Propellant replenishment kits
- A set of optional end effectors for the manipulators.

The test payloads are composite devices designed to allow SEPS to demonstrate all of the payload support, servicing, deployment and refolding, maintenance, transfer, and retrieval functions envisioned for the full operational time of the first generation SEPS (1981-1991).

3.4.2 Sortie Sequence

1. Shuttle ascends to a 300-km earth orbit and deploys IUS. Viability of payloads is checked before IUS deployment.

2. IUS ascends to intermediate orbits and deploys operational payloads.

3. IUS ascends to SEPS deployment orbit.

4. The initial testing sequence begins with full activation of SEPS. SEPS is mounted to the most forward diaphragm of the transport shell. Transport shells can be designed with full splices so that shortened shells may be

used when desired. The transport shell does not extend beyong this diaphragm, so SEPS' solar cell array, payload mast, navigation and guidance sensors, and so forth, can be fully deployed as desired. The activation sequence begins with SEPS' switch to internal power. From this point forward, SEPS (though still attached to IUS) is functioning as an independent spacecraft.

SEPS command data system and computer functions are validated.

SEPS solar arrays are deployed to about one-quarter span, and the power supply and distribution system function is validated.

SEPS navigation and guidance sensor platforms are deployed, sensor function checks are made, and ACS function checks are made. Payload mast and manipulators are deployed.

This completes the initial test sequence validating SEPS ability to function as an independent vehicle. The probability of failure to achieve independent functional ability is almost zero due to the high level of redundancy in critical subsystems. The only requirements are:

- An up-down data link
- At least 1 kw of solar array power
- Central computer and one memory bank
- ACS system in minimal mode
- Housekeeping power supply and distribution critical circuits only.

5. With SEPS ability to function as an independent stage validated, IUS releases the payload transport shell with SEPS attached. With SEPS supplying power to IUS the functions demanded of IUS have not been limited by the IUS' small capacity storage system.

6. SEPS full navigation and guidance subsystem functions are now checked out in detail and the gyros initialized.

7. Full checkout of the payload mast and manipulator system is accomplished in parallel with other stagekeeping subsystems.

8. The manipulators are used to demonstrate their capability to accomplish the following:

a. Remove a module from the test payload and substitute another for it.

b. Remove a test payload from one diaphragm and secure it to another diaphragm in the payload shell.

c. Deploy and refold simulated or actual elements of test payload such as solar panels, antenna, scientific instrument booms, and so forth.

d. Using refueling kits, simulate the replenishment of payload expendables by filling some tanks in a test payload.

e. With manipulators, demonstrate the ability to remove and replace various items of real or simulated test equipment (and perform functions) such as:

- Experiment packs and instruments substitutions
- Power supply module replacement
- Solar cell panel replacement
- Mechanical device and scan platform replacement
- Cut and splice a structural element
- Operate various types of spring loaded clamps, latches, and so forth.
- Repeat several cycles of plugging and unplugging various types of developmental and experimental electrical umbilicals.

f. Conduct test evaluations on several complete competitive concepts for payload support umbilical systems.

9. All onboard software, computer functions, data system and communications link functions are checked out.

10. The autonomous navigation and guidance system functions are checked by comparison of the onboard SEPS position with the STDN SEPSOC determined position functions.

11. All ACS functions are demonstrated.

12. Solar arrays are fully extended.

13. SEPS grasps the payload shell with one manipulator. With the other, it disengages its launch support structure from its mounting diaphragm and disconnects the test payload support umbilicals.

14. SEPS, without ever having released the transport shell, transfers the payload shell and test payloads to its payload transport mast and reconnects the test payload support umbilical.

15. At least one of the GPME diaphragms will be designed for rotating a test payload to satisfy thermal environment conditions. Functioning of rotary transformers, slip rings, and other devices for transmitting alternating and direct current power, and RF power through rotating joints will be evaluated.

16. SEPS is prepared for cruise to geosynchronous orbit and cruise is initiated.

17. In geosynchronous orbit, SEPS is run through a set of maximum design capability maneuvers with the ion engines. These maneuvers include combinations of operating condition and sunlight at the design limit angles for both thruster and main body thermal control. These maneuvers will be planned to verify (or develop the basis for new analyses) the design analyses that predicted the operational environments and operational capabilities of all SEPS components and subsystems.

18. The transport shell with test payloads will be released, and a limited test series will be run with SEPS as a bare stage to test thermal and other effects when SEPS cruises with no payloads.

19. SEPS will rendezvous with the transportation shell, take one of the test payloads from the transport shell, and, using the manipulator, push it in a posigrade direction. Another payload will be pushed in the retrograde direction.

20. SEPS will release the payload transport shell and retrieve first one and then the other payload, thus demonstrating the ability to retrieve unstabilized, totally inactive payloads.

21. SEPS will again rendezvous with the payload shell and install the test payloads on diaphragms in the transport shell.

22. The transport shell remains in geosynchronous orbit for the training of new flight controllers who may come into the program during the subsequent years and as an evaluation device for payload related testing, new GPME testing, or for new operational concepts.

These earth orbital tests will have accomplished several significant objectives.

1. SEPS design goal capabilities will have been validated in all respects except wearout life and radiation damage sensitivity. Desirable design modifications will have been identified for the remaining SEPS production inventory.
2. STS and SEPS unique GPME functional capability and operational suitability will be validated, and data for improved designs will be obtained.
3. Operational procedures for the total system will have evolved, and software packages will be validated.
4. A general purpose training, GPME technology demonstration device with an emergency store of SEPS expendables will be in geosynchronous orbit for future use.

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IV FLIGHT SYSTEM CONSIDERATIONS

Section IV

SEPS FLIGHT SYSTEM CONFIGURATION PAYLOAD SUPPORT, GPME, AND INTERFACE CONSIDERATIONS

4.1 OBJECTIVES

The SEPS configuration, as discussed in the Summary, is dictated primarily by design considerations associated with maximizing its capabilities for:

- Multiple payload delivery and deployment assistance to each individual payload as it is deployed
- Multiple payload retrieval
- In-space servicing of payload and maintenance of payloads.

Using the concepts found most desirable in this study, SEPS has no direct interface with any STS element except Tug. Even that interface is restricted to the avionics system.

The decision controlling factors regarding SEPS overall configuration, therefore, are primarily related to the functional interfaces with payloads and STS General Purpose Mission Equipment (GPME). In summary form, the decision controlling factors are:

- STS transportation efficiency depends on multiple payload deliveries and multiple retrievals
- Cost effectiveness requires that GPME be usable on successive flights without modification and with few special payload adapter items
- The GPME must simplify Shuttle-Tug operations
- Multiple payload transport must place minimum constraints on payload designers
- SEPS staytime in space is limited only by wear out. Design should provide for easy replenishment of expendables
- GPME mass increase to simplify other STS operations does not reduce SEPS plus Tug net payload capability; modest trip time increases allow SEPS to make up for Tug's lower payload transfer orbit ability
- Earth orbital SEPS has no ΔV limit within mission model requirements
- SEPS capabilities are almost directly proportional to design power level in the range from 25 to 100 kw. Development at higher power levels causes less than 10 percent increase in development cost.

The nature of these decision controlling factors so interrelates the SEPS configuration and GPME that some of the objectives of Task II of the original study statement of work were transferred to Tasks III and IV. This section describes the analyses, rationale, compromises, and evolution of concepts best fulfilling the following objectives of the original Tasks II, III, and IV:

- Identify and develop design requirements and modifications to the NASA-provided baseline SEPS that enhance mission performance
- Establish performance capabilities and limitations for different mission modes such as delivery, delivery/retrieval, and multiple payload placement/retrieval/servicing/maintenance
- Develop conceptual designs or recommended systems of payload handling, servicing, and ancillary hardware
- Develop conceptual designs of recommended docking interfaces
- Evaluate SEPS compatibility with Shuttle-IUS-Tug safety requirements
- Identify necessary or desirable changes in specific subsystems
- Evaluate techniques leading to a preferred operational concept for man-in-the-loop or autonomous N&G subsystems for terminal approach to the rendezvous/docking functions
- Define rendezvous and docking implementation requirements, STS interfaces, and ground system interfaces
- Investigate onorbit versus ground-based servicing/refurbishment of SEPS
- Identify subsystems design impacts.

4.2 IDENTIFICATION OF THE MOST DESIRABLE PAYLOAD SUPPORT ANCILLARY GEAR AND GPME

Past study approaches to arriving at the "best" configurations on SEPS and on Tug for fulfilling the objectives described in subsection 4.1 appear to have considered each function: docking, payload transfer, retrieval, servicing, and maintenance as separate entities as if the simplest implementation for each function would lead to the "best" accumulation of equipment and the simplest inflight system operation. In NSI's first consideration of this problem, it appeared obvious that some multifunction system would be simpler than a hodgepodge of "best" single-function systems. Furthermore, it appeared that a system capable of accommodating payload configurations not known at the time the SEPS design was frozen and capable of accommodating operations not initially envisioned must necessarily be highly desirable for implementation on SEPS. Inherent adaptability to new payloads without placing undue design constraints on the payload designer appeared necessary.

4.2.1 Articulated Docking Frame and Articulated Tug Transport Frame

Study work by previous contractors for SEPS and Tug concentrated on docking devices and various mission peculiar structural frames that required articulation. When single or dual payloads are the only requirements and the servicing function is ignored, such approaches can result in desirable systems. The STS with SEPS problem is, however, quite different. As presented in Section I histograms, the most cost effective transport system utilization results in multiple payload Shuttle flights such that 83 percent of the individual payloads are delivered in groups of four or more, and 47 percent in groups of five or more. Ninety percent of the down payloads are retrievable in groups of two or more, 75 percent in groups of three or more. When such large numbers of payloads must be handled, docking frames and articulated support frames are not promising. NSI took the docking/payload frame system, at MSFC direction, as a point of reference for trade studies and tried to generate the best concept of that type which met all the requirements.

Except for the first sortie when SEPS is launched with the payloads, all other sorties begin with SEPS in the orbit where it last performed a mission function. Generally, this is a geosynchronous orbit. When a sortie requires retrieval of down payloads for return to earth by Shuttle through rendezvous with Tug, SEPS first function is collection of the payloads and transporting them to Tug in its lower energy orbit.

The simplest hardware and operations system we could envision for this operational sequence is shown on Figure 4-1. SEPS has an articulated square docking frame, similar to those evolved by McDonnell Douglas in MSFC-directed studies, and one extendable payload mast such as the one NSI selected for the SEPS manipulator/mast system. Figure 4-1 does not show all steps of the sequence. The omitted steps will be identified.

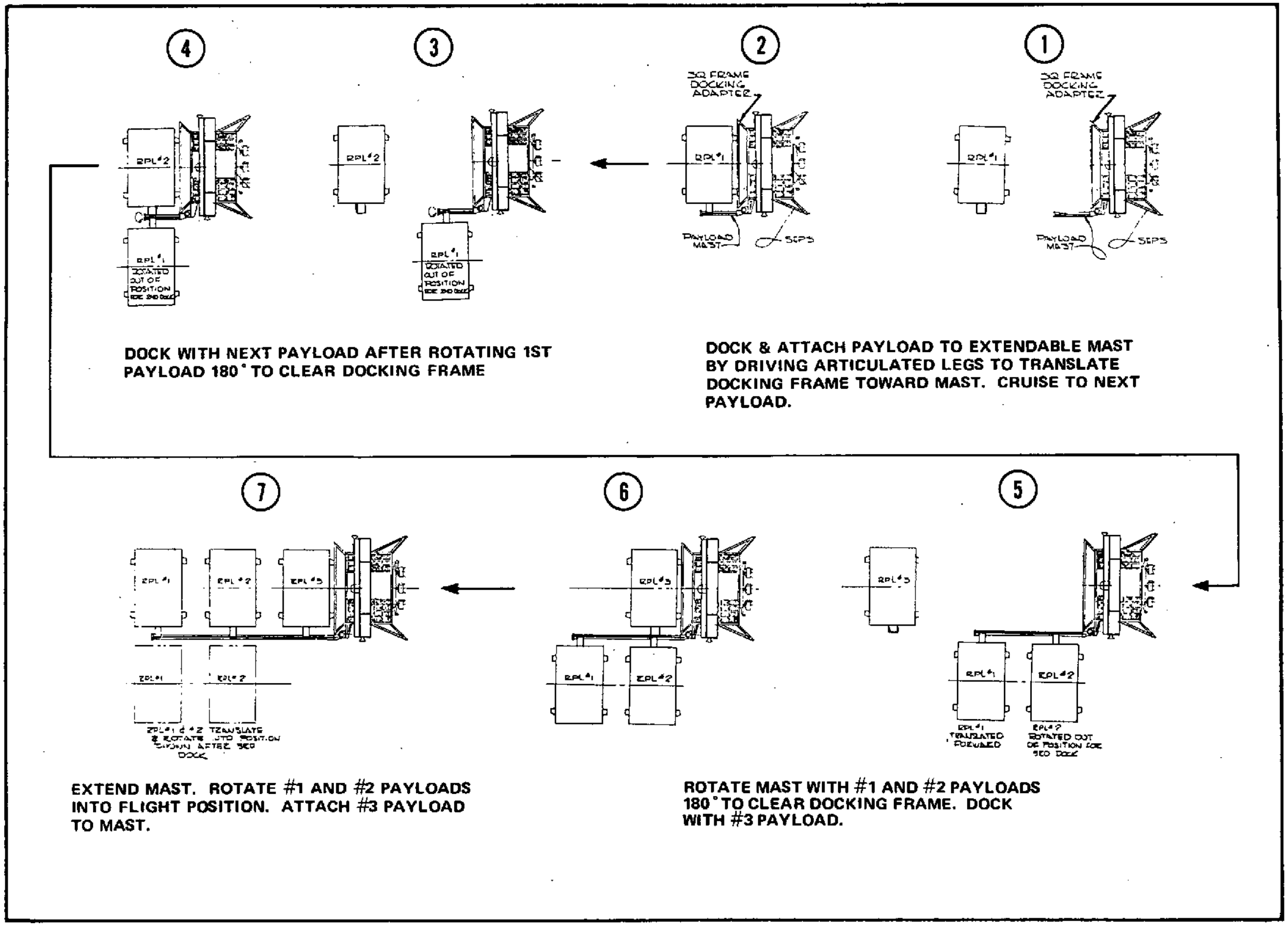

Figure 4-1. INTERRELATIONSHIP OF COORDINATE SYSTEMS

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The payloads must be designed with docking rings (or have other provisions for engaging docking latches) on two ends. The payloads also are equipped with spring-loaded clamps so that when the properly oriented clamp is pressed against the SEPS payload mast it will spread and snap over the cusps of the biconvex section mast. These clamps may be similar to ones described later for the recommended system.

The payloads must be stabilized. If docking rings are used on the payloads and they have no protuberances beyond a 2.3 meter (7.5 foot) radius, a stable spinning satellite may be retrieved. An unstable tumbling satellite cannot be retrieved.

As a first step of the sequence, SEPS cruises to the rendezvous point, commands the variable length sections of the docking frame to the proper geometry, maneuvers into position for docking, and then moves in and docks with the payload. Some parts of this sequence may be autonomous. All are monitored by ground controllers who can override the autonomous operation if necessary. The extended square frame docking device is partially retracted; the supporting struts of the docking frame have motor driven, screw activated, telescoping sections in addition to their shock absorber sections. If each of these struts is driven to the appropriate length, the square frame docking mechanism can be tilted, translated axially, and translated laterally. This articulated docking frame requires 12 struts with position-controllable linear actuators. Eight of these struts also contain shock absorbers.

After the first payload is docked to the frame in 1 of Figure 4-1, the capture latches can be commanded to a "loose clamp" position and a friction drive wheel can be engaged with the payload docking ring. The payload is rotated until its mast clamp is properly oriented with the mast on SEPS.

The articulated docking frame struts are driven to positions that translate the frame laterally about 0.25 meter until the payload mast clamp snaps over the mast. The SEPS payload configuration is as indicated in 2 of Figure 4-1. The SEPS cruises to rendezvous with payload (PL) #2.

As SEPS approaches PL #2, the docking frame latches are released, the payload mast is extended a short distance forward, and the mast assembly with attached payload #1 is rotated 180 degrees, thus leaving a clear path to the docking frame. The terminal approach configuration to PL #2 is shown in (3). The capture sequence, (4) , for PL #2 is similar to that for PL #1. The payload transport mast is extended until PL #1 on the mast will clear PL #2 on the frame when the mast is rotated 180 degrees. PL #2 is rotated until its mast clamp is in position. The docking frame is translated laterally until PL #2 mast clamp snaps onto the mast. SEPS then cruises to rendezvous with PL #3.

SEPS approach configuration to PL #3 is shown in (5) of Figure 4-1. The steps in achieving final configuration for cruise to rendezvous with Tug, (7) , are obvious after the foregoing discussions.

Somewhat simpler mechanical implementations were conceived, but the multiple payload retrieval function then involved more complex flight maneuvers. These maneuvers used more ACS propellants, they required payloads to have at least attitude holding ability throughout the full multiple payload collection phase, or they involved constraints on the payload designers. For the previous sequence, each payload was passivated after initial clamping to the SEPS mast.

Tug's problem of bringing the multiple payload group up to the rendezvous orbit with SEPS is illustrated on Figure 4-2. Sequence (1) is the configuration as deployed from Orbiter. Up PL #3 is attached to Tug's docking frame which is designed to support it through the abort and crash load safety criteria of the manned Orbiter.

Up payloads #1 and #2 are supported on articulated L-frames shown in simplified schematic form. The minimum articulation requirements of these frames are that they can be extended in and out along the long leg direction of the L and that they can be clamshell style, opened and closed. For lateral rigidity the short legs of the L-frames must have structural load-carrying latches where they meet at the extended axial center line of Tug.

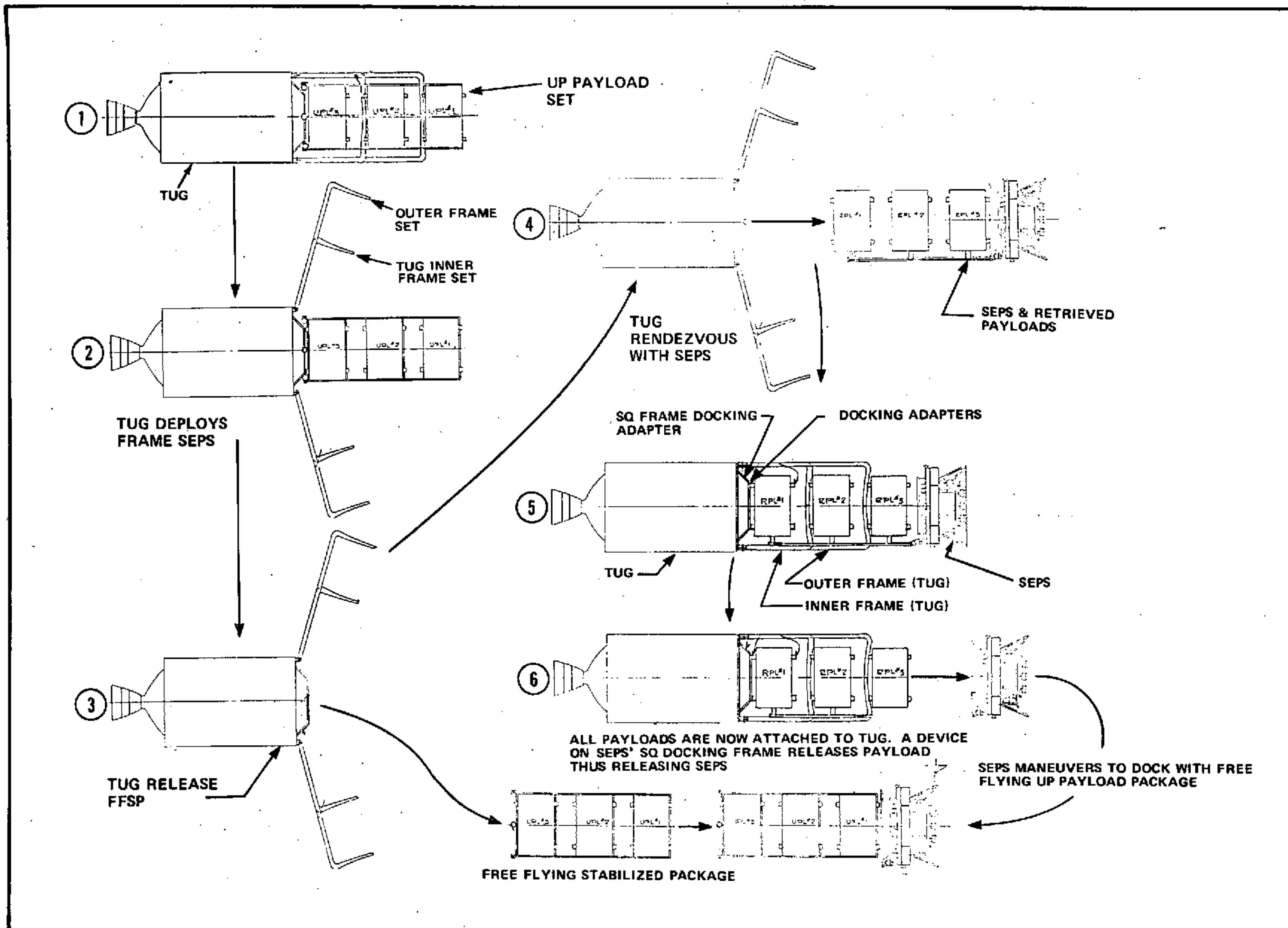

Figure 4-2. DOCKING FRAME PAYLOAD TRANSFER SEQUENCE

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The L-frame long legs are actually part of a cylindrical surface, and the short legs are pie-shaped segments of a disk to provide area for mounting payloads in a stable manner. With honeycomb cores and high strength fiber/epoxy surface sheets, these L-frames can be relatively low in mass if PL #1 and PL #2 are supported against Orbiter crash load requirements by brackets to the Orbiter structure so that the L-frames only have to resist Tug's freeflight loads. The pairs of L-frames are all shown rotated into the plane of the schematic. They would in fact be at 90 degrees to each other.

When Tug comes to the rendezvous position with SEPS the L-frames are opened, (2) , and the up payloads released, (3).

The up payloads are attached to a light, tubular, flexible frame which supported the Tug umbilical lines to the payloads. The tubular frame has a simple attitude hold and RCS to stabilize the up payload package for later retrieval by SEPS. This tubular frame and ACS is expendable. The tubular frame and other attach elements must be tailored for each payload package.

After release of the up package Tug moves over to dock with the retrieved payload set and SEPS, (4). The SEPS payload mast concept, derived from the recommended system concept to be described later, has adequate rigidity and strength to sustain docking loads.

Three shock-absorbing factors reduce non-nominal docking loads. These factors are:

- Tug docking frame shock-absorbing struts
- The payload-to-SEPS mast clamps are friction-hold clamps. If axial force exceeds design slip, the clamps slide down the mast.
- The SEPS mast can be designed for normal overdrive windup into its housing at loads that approach critical buckling for the mast column.

After Tug docks with retrieved payload (RPL) #1, the short pair of L-frames are closed and their tips are latched to each other where they meet at the Tug's extended center line. The axial legs of the L-frames are extended until the payload capture latches on the L-frame short legs capture the payload docking rings of RPL #2. A similar sequence is performed for RPL #3. The configuration status is now as shown in 5.

SEPS retracts its payload mast. As the retraction force exceeds the payload clamp friction force, the mast slides through the clamps until it clears RPL #3's clamp and is fully housed. SEPS releases its docking frame latches to RPL #3 and backs away as shown in 6. SEPS proceeds to complete the docking exercise with the up payload group 7. SEPS payload support umbilical is driven to engagement with that of the payload package. SEPS now initiates cruise to deliver the up payloads to their mission stations.

This system has simple individual devices, but there are many of them. Most of them require position command, command implementation means, and position status reporting. Many of them must work in coordinated relative geometric patterns to accomplish their functions. The system requires TV visual aids for docking, monitoring, and verification of clamp attachments, and laser radar for terminal approach to docking.

The system requires that each payload have a mast clamp and have docking rings at each end. The system does, however, provide for independent mounting of payloads so that no payload needs to be designed for structural rigidity and strength necessary to support other payloads on its docking rings.

The Tug operating alone, if fitted as described and also equipped with a payload mast like SEPS, would be capable of multiple payload retrievals for those instances where payload weights were low enough for Tug performance to allow it.

This scheme fulfills the transport requirement except that it has no capability for retrieval of payloads whose attitude control systems are malfunctioning or depleted, and it is very difficult to accommodate more than three payloads.

The scheme has no in-space servicing or maintenance capability. To meet that requirement a servicer kit such as the one MDAC proposed in its Payload Utilization of Tug studies might be adopted. The kit concept is shown on Figure 4-3. The kit has a rotating spare module table with module jack-out, jack-in ability. The spare module table is first mated to the payload with the module to be replaced over a vacant module position in the table. The defective module is jacked out. A replacement module is rotated into place and jacked into the payload. The module table scheme appears simple at first, but as implementation details are examined it becomes more complex. Further, either the payloads must all be constrained to meet the interface of a standard servicer-maintainer or the servicer-maintainer must be tailored to every payload. The system has no flexibility for unplanned situations and has very limited capability.

NSI considered this approach to be unacceptable because of the constraint to payload designers and developers, the limited servicing capability, and the fact that the culmination of its many apparently simple devices and operations makes it the most complex overall system.

The scheme does not appreciably simplify prelaunch ground functions involved in mating multiple payload packages with Shuttle and Tug nor does it decouple the multiple payload package integration and flight readiness check from Shuttle/Tug launch preparation activities.

The scheme does not appreciably reduce the amount of mission special interface devices required.

4.2.2 Boom–Manipulator – Payload Transport Shell Scheme

One highly desirable objective in any scheme for handling multiple payloads is to provide a system where the multiple payload package can be integrated into a single structural package, with single avionics and fluids (if required) interfaces. The multiple payload group is then presented to Tug as a single package. Tug plus package is presented to Shuttle as a single payload with only Tug's standard interfaces.

Ideally, Shuttle would see every Tug flight-to-rendezvous with SEPS as a standard physical and procedural interface. Only the level of raw power

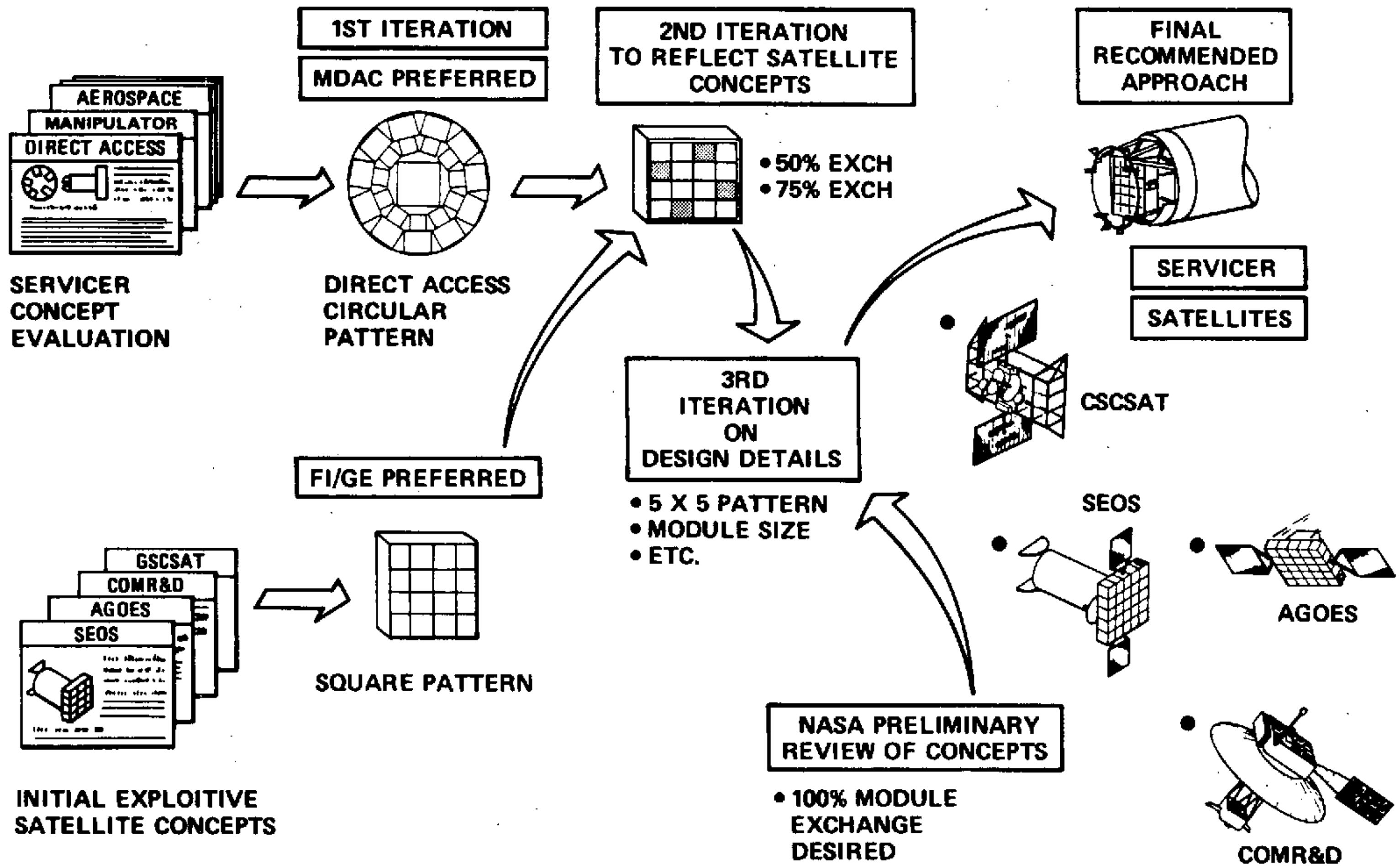


Figure 4-3. SERVICER CONCEPT FOR DOCKING FRAME SCHEME

support and the information which data management systems transferred across interfaces would be different as seen from Shuttle.

The system concept depicted on Figure 4-4 shows the potential for meeting the above objectives to the extent practicable. The system will not be described in any detail because most of its elements have nearly one-to-one correspondence with some equivalent element in the recommended system to be described later. Briefly the systems operation is as follows.

SEPS always carries a payload shell except for sorties that will not require multiple payload delivery or retrieval. The payload shells are equipped with diaphragms to which individual payloads are mounted. Payload shells may occasionally be left in "storage" in geosynchronous orbit. Each payload retains the structure and mounting/docking ring that attached it to the launch support diaphragms for its ascent flight.

SEPS has an extendable boom similar in structural characteristics to the payload mast of the recommended scheme. The shoulder mount of this boom is on a base plate that can be rotated. The angle of the boom to the base plate can be commanded, and the entire boom mechanism is rotatable upon command. At the outboard end of the boom, a joint with two degrees of rotational freedom supports an extendable forearm section to which a manipulator "wrist" and "hand" are attached. This device is, in essence, a manipulator with extendable arm segments.

A sortie sequence begins with SEPS in geosynchronous orbit with the payload shell that was used to deliver the payloads of the previous sortie. The diaphragms that up payloads were mounted on have been retained.

SEPS cruises to a station alongside a payload to be recovered. It then relocates the diaphragm equipped with latches that match that payload's docking ring to an appropriate position in the shell. The diaphragm was equipped before launch of the previous payload set with a set of contact-actuated, spring-loaded latches such that when SEPS presses the planned retrieval payload's docking ring onto the latches, they will capture the ring.

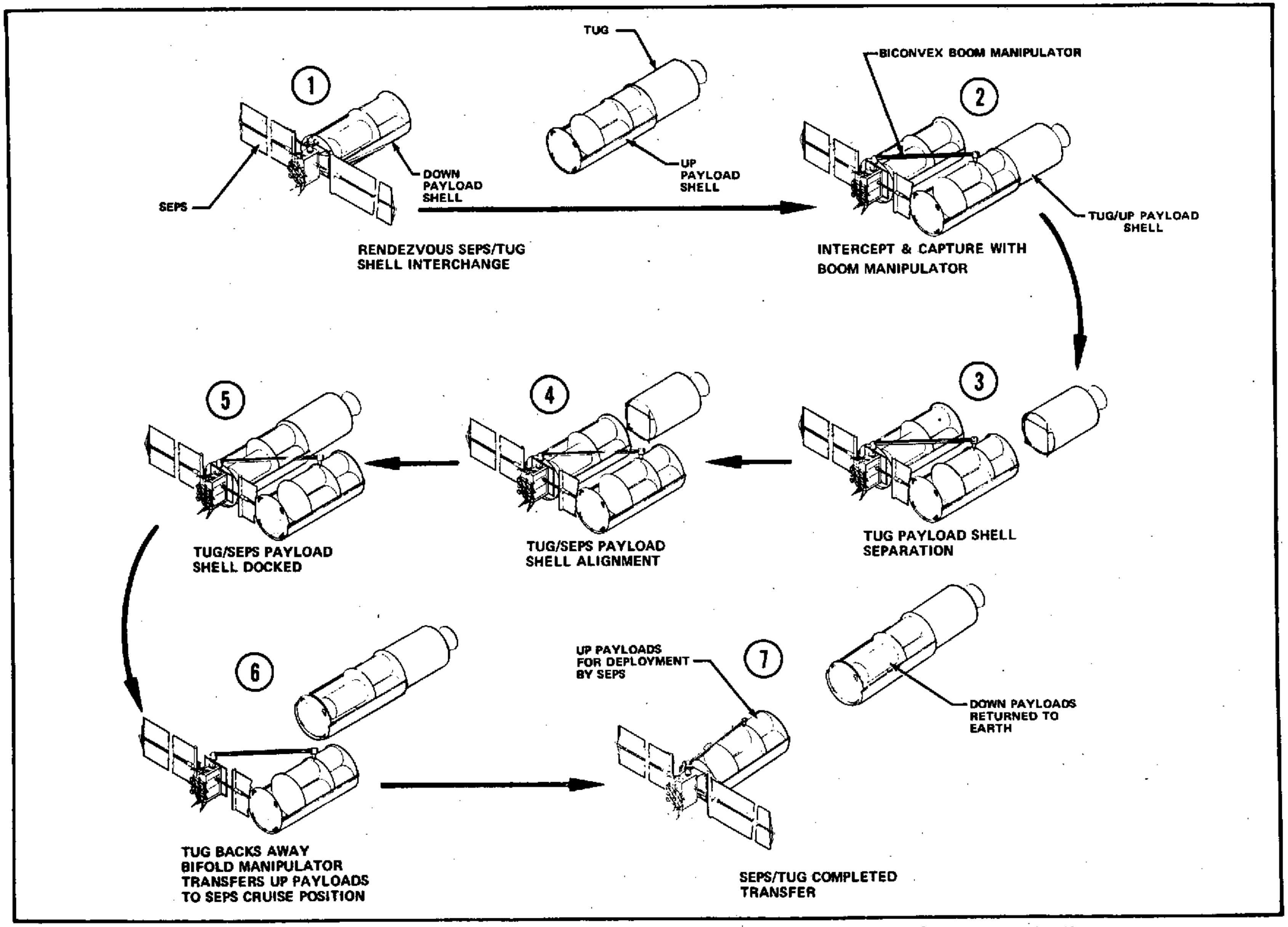


Figure 4-4. CANDIDATE SEPS/GPME SYSTEM MAKING PAYLOAD TRANSFER WITH TUG

SEPS, using the manipulator/boom, grasps the payload at any one of several built-in grasp points (or any point of adequate structural rigidity) and places it such that its docking ring trips the capture latches on the diaphragm.

SEPS successively captures each payload and cruises to meet Tug at the rendezvous point. Tug to SEPS relative positions at rendezvous are shown on Figure 4-4 (1).

Either Tug or SEPS maneuvers until they have the relative position (2) of Figure 4-4. SEPS grasps and holds a diaphragm of the payload shell brought up by Tug with its manipulator/boom.

Tug releases from the up payload shell, (3) of Figure 4-4, backs away, and moves into position to dock with the down payload shell attached to SEPS, (4) .

Tug docks with the shell, (5). SEPS releases the down payload shell to Tug, and Tug (or SEPS) backs away, (6). Tug proceeds to rendezvous with Orbiter.

(7) SEPS places the up payload shell on its docking frame and proceeds on the ascent maneuver to deploy the up payloads in their respective positions.

This scheme is compatible with the baseline Tug as defined by MSFC. Figure 4-5 presents local detail of the forward structural skirt of the MSFC baseline Tug, details of a McDonnell Douglas concept of a baseline Tug docking capture ring, and the transition parts of the NSI-proposed transport shell. Every active element portrayed on Figure 4-5 is an element of the MSFC baseline Tug.

To execute servicing or maintenance functions with this scheme, the SEPS would capture a payload and place it on a diaphragm in the transport shell with the area of the payload needing maintenance in the position providing the best teleoperator visibility and manipulator access.

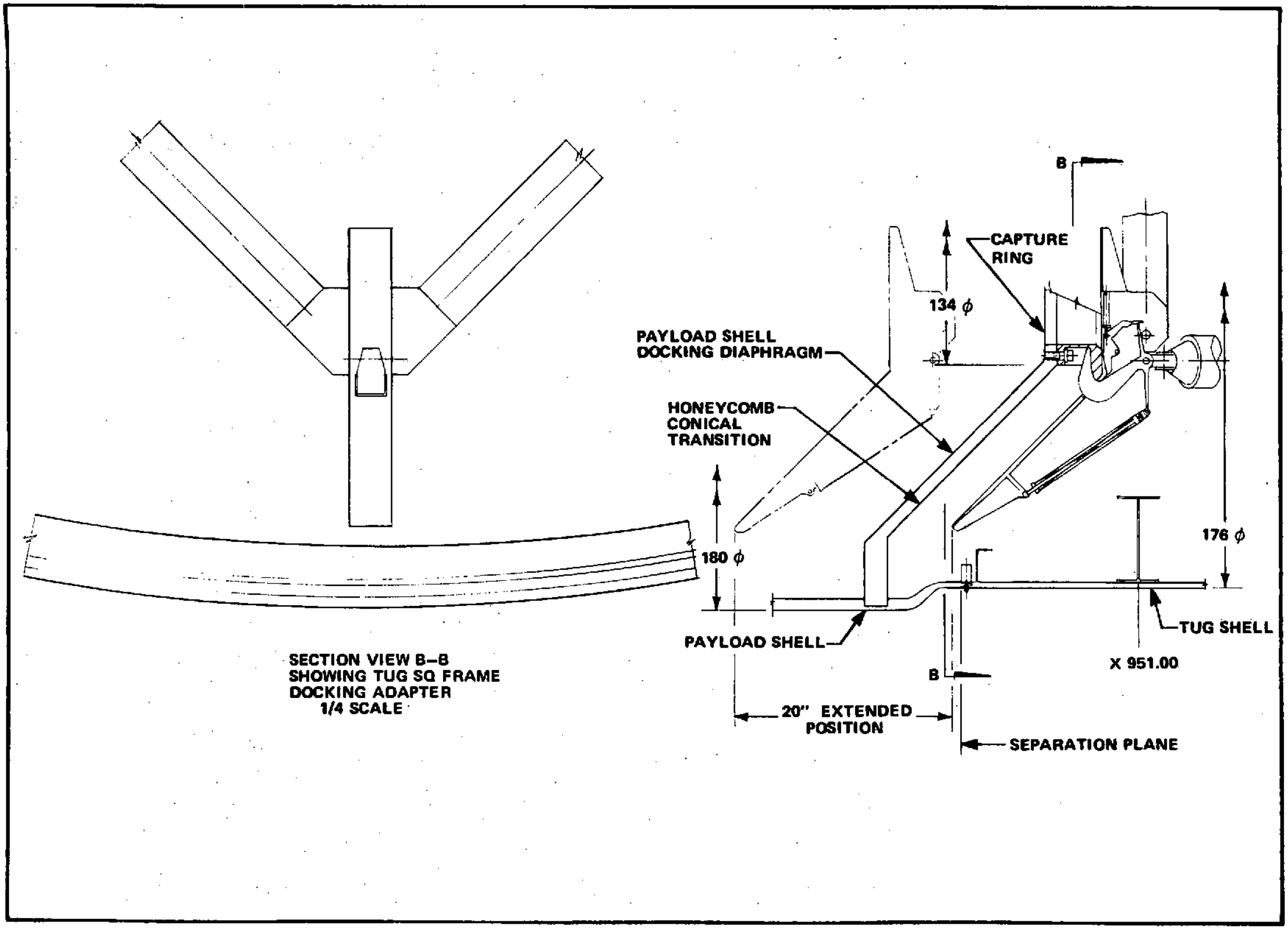


Figure 4-5. FORWARD STRUCTURAL SKIRT OF BASELINE TUG

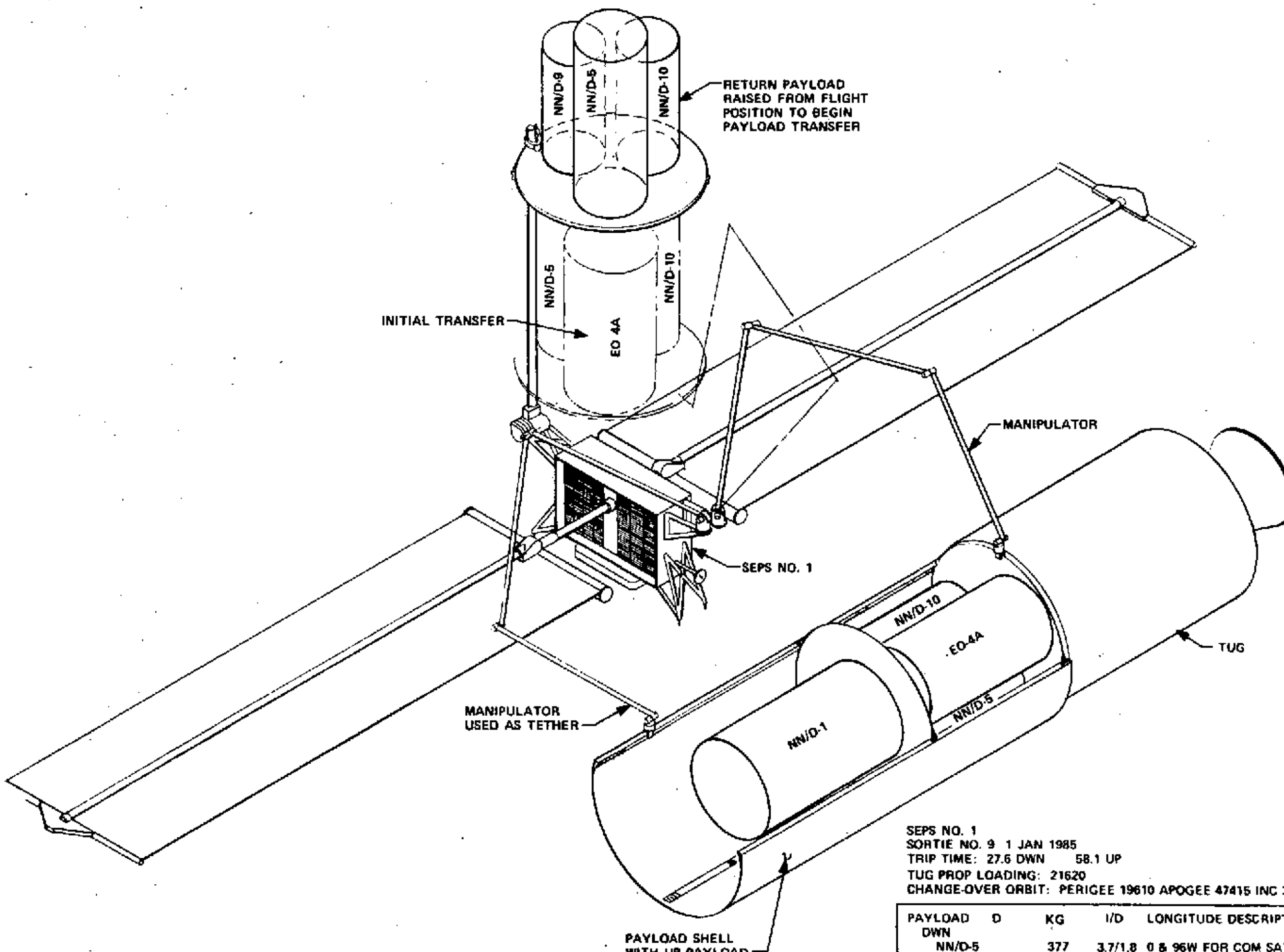
This NSI-developed concept is the simplest one that meets all of the requirements that appear most desirable. Its principal shortcomings are lack of some adaptability for unplanned, quick response maintenance or retrieval functions, and lack of a general capability to assist in deployment (or refolding for retrieval) of certain elements of payloads. It lacks the general servicing and maintenance capability of a two-arm, full manipulator system.

4.2.3 Recommended SEPS Configuration With Recommended STS GPME

NSI's operations analyses and cost effectiveness assessments indicated that the SEPS system operating with STS should meet the following criteria:

- Minimize constraints on payload designers and developers
- Simplify Tug interfaces and functions for payload transport and recovery. Provide for any arbitrary size and number of payloads that can be accommodated by Orbiter's cargo bay
- Minimize STS specialized transport gear. Use only standardized equipment plus individual payload structural attach mountings
- Standardize interface of payload packages to Shuttle
- Decouple prelaunch activity schedules of Shuttle, Tug, and the multiple payload packages to the extent practicable. Avoid large numbers of even minor mission special adapting devices on Tug or Orbiter so that substitution of the package to other STS flight articles could be made to meet priority rescheduling
- Provide ability in orbital taxi role to deploy, retrieve, and service payloads in any arbitrary sequence as SEPS moves around geosynchronous orbit
- Provide ability to deploy (or refold) elements of payloads as backup to onboard systems or to allow elimination of deploy/refold driver devices in order to reduce DDT&E costs to payload developers
- Provide ability to transport, retrieve, and service payloads not yet defined at time of SEPS first launch without significant design constraints on the payloads
- Provide capability to retrieve failed unstabilized satellites.

A system which essentially meets all of the criteria is shown on Figure 4-6. The cylinders represent the envelope dimensions of the designated payloads from the NASA-supplied mission model. The particular payload grouping is a specific Shuttle flight cargo manifest (SEPS - Tug sortie #9)



SEPS NO. 1
SORTIE NO. 9 1 JAN 1985
TRIP TIME: 27.6 DWN 58.1 UP
TUG PROP LOADING: 21620
CHANGE-OVER ORBIT: PERIGEE 19610 APOGEE 47415 INC 3.7

PAYLOAD D	KG	I/D	LONGITUDE DESCRIPTION
DWN			
NN/D-5	377	3.7/1.8	0 & 96W FOR COM SAT A
NN/D-9	347	3.1/1.8	140E FOR SYNC MET. SAT.
NN/D-10	347	3.1/1.8	140E GEOSYNC OPER MET. SAT.
UP			
EO-4A	1400	3.4/2.2	100W SYNC. EARTH OBS.
NN/D-1	2040	3.7/2.5	40W & 180W INT. COM SAT.
NN/D-5	446	3.7/1.8	0 & 96W FOR COM SAT B
NN/D-10	366	3.1/1.8	140E GEOSYNC OPER MET SAT

Figure 4-6. PAYLOAD EXCHANGE SORTIE NO. 9

taken from an STS System Operational Profile for accomplishment of the total mission model.

The sequence of this particular sortie, No. 9, for SEPS flight article No. 1 which has been operating in space for 4 years is as follows.

SEPS had retained the diaphragms to which the payloads for the previous sortie (No. 8) were mounted. The GPME diaphragms have a multifunction pattern of payload attach holes through which payload struts are secured to the diaphragms. The attachment will withstand the Shuttle's 9 g crash load criteria. Concepts for these GPME items will be described later.

SEPS cruises up to a payload to be retrieved and takes station alongside it. A manipulator, under ground control, grasps the payload and mounts it on a diaphragm. SEPS collects each successive payload to be retrieved in similar fashion. Then it begins the descent to rendezvous with Tug. In this specific sortie all of the retrieved payloads are mounted on a single diaphragm. More than one diaphragm can be used. As another option, if multiple payload package arrangements of several successive sorties make it desirable, the sorties might have been accomplished by transfer of complete payload shells.

The diaphragm clamped to SEPS payload transport mast would be located near the tip of the mast. Diaphragms from the previous sortie unused in the retrieval procedure are stored on the mast just below the one to which the payloads are mounted. The mast, throughout the payload collection operations and the return cruise to meet Tug, has been partially retracted so that the composite PL center of gravity (c.g.) is nearer SEPS (c.g.), reducing their combined moment of inertia to facilitate maneuvers.

When SEPS is in (or nearly so) the rendezvous orbit, the Shuttle is launched and Tug proceeds to the rendezvous point with SEPS. Either vehicle can execute the final station attainment maneuver.

When on-station SEPS grasps the Tug's payload transport shell and maintains the relative geometric positions of SEPS and Tug, the attitude control systems of both crafts are deactivated at this point.

The payload mast is extended until space is available underneath the down payload set for placing the first group of new payloads on the mast. The other manipulator unlatches the diaphragm clamp and Tug payload support umbilical to the diaphragm. It then grasps the diaphragm to begin transferring it to the payload mast. This is the system state depicted on Figure 4-6. Phantom lines show position of the first group of payloads after they are attached to the mast.

The mast is extended until space is available to mount the last up payload and diaphragm on the mast. SEPS repeats the previous sequence, and all payloads are now on SEPS. The manipulator now plugs a SEPS payload support umbilical into each diaphragm so that SEPS now provides the payload support previously supplied by Tug.

The down payload set on the diaphragm is then installed in Tug's payload shell by the manipulator. Diaphragms can be located at any position in the shell that is desired, providing a means for c.g. location control for Orbiter's descent flight. Spare diaphragms from the previous sortie are mounted in the shell just forward of return payloads. These spare diaphragms provide added protection against a retrieval payload becoming detached and smashing into the Orbiter crew compartment during a crash landing.

The first manipulator (which has maintained the relative geometric positions of Tug and SEPS throughout the above procedure) or both manipulators gently shove the Tug away.

When adequate clearance between the two spacecraft exists, Tug proceeds to rendezvous with Shuttle and SEPS proceeds to mission orbits desired for each payload.

Three candidate concepts have been described. The simplest mechanically and operationally is the two manipulator arm system. That system also has the most basic capability and versatility. The one area where it appears more complicated is in the requirement for computer memory and onboard software. A summary comparison of the systems is given in Table 4-1.

Table 4-1. PAYLOAD SUPPORT, HANDLING, AND SERVICING CONCEPT COMPARISON

ARTICULATED DOCKING FRAME AND ARTICULATED MULTIPLE PAYLOAD SUPPORT STRUCTURES	TRANSPORT SHELL, EXPENDABLE BOOM AND SIMPLIFIED MANIPULATOR	TRANSPORT SHELL, PAYLOAD MAST AND MANIPULATOR SYSTEM
ADVANTAGES	**ADVANTAGES**	**ADVANTAGES**
• SIMPLEST ONBOARD SOFTWARE	• MODERATE ONBOARD SOFTWARE REQUIREMENT • SIMPLEST PAYLOAD TRANSFER FUNCTION	• GREATEST INHERENT CAPABILITY FOR PAYLOAD SERVICES AND MAINTENANCE • MINIMIZES DESIGN CONSTRAINTS ON PAYLOADS • SIMPLEST AND MOST FLEXIBLE INFLIGHT OPERATIONS • SIMPLEST GPME & TUG PAYLOAD INTEGRATION FUNCTION • HIGHEST MISSION SUCCESS PROBABILITY
DISADVANTAGES	**DISADVANTAGES**	**DISADVANTAGES**
• MOST COMPLEX FLIGHT OPERATION • MOST COMPLEX FLIGHT HARDWARE • LIMITED GPME - REQUIRES TAILORING OF TUG MISSION EQUIPMENT & ORBITER TO PL ADAPTERS FOR EACH SORTIE • EITHER SERIOUS PL DESIGN CONSTRAINT OR VERY LIMITED SERVICING ABILITY • NOT ADAPTABLE TO UNFORESEEN OR UNPLANNED MISSION EVENTS • TOTAL COMPONENTS REQUIRING POSITIONING & FEEDBACK INFO EXCEED OTHER SYSTEMS	• LIMITED SERVICING AND ONORBIT MAINTENANCE ABILITY • INTERMEDIATE ADAPTABILITY TO UNPLANNED MISSION EVENTS	• ONBOARD SOFTWARE REQUIRES 32K WORD MEMORY STORAGE

4.3 GENERAL PURPOSE MISSION EQUIPMENT ASSOCIATED WITH SEPS OPERATIONS

The principal items of equipment that are kit attachments to SEPS are the manipulator and mast subsystems. The other elements of the payload transport and support equipment set are STS GPME. They will also serve to simplify STS operations that do not involve SEPS.

Throughout this study, NSI has continuously received suggestions to show design detail to accomplish various major and many minor functions. Within the 6,500 man-hour scope of the contract it is not possible to create thoroughly analyzed design concepts. Further, NSI believes many alternate detail design concepts for components are workable and reliable. The "best" (optimum) design of components is that one which makes the total system most effective in accomplishment of its desired objectives. Design detail is therefore best optimized along with detail design of the total system.

The design concepts presented here (except for minor detail) are definitely workable and are believed to be valid candidates, at least, for implementation in the STS/SEPS system.

4.3.1 Manipulator Subsystem

The attachment to SEPS and the reach of the manipulators is shown on Figure 4-7. They can reach any location around the complete circumscribing cylinder of the 9.1m long, 4.6m diameter volume available for cargo after Tug has been installed in the Orbiter's cargo bay. The manipulators are such that they can reach any area around or underneath SEPS for self-maintenance, servicing, or self-inspection with the TV cameras that would be mounted on the wrist.

Figure 4-8 shows characteristics of the manipulator. The structural strength of the manipulator is dictated by rigidity requirements. Providing motors and harmonic drives to supply 500 foot-pounds of torque at the joints allows unloaded 1 g ground testing. Only 50 foot-pounds of torque are required in space. Each manipulator can change the end effector of the other. For special functions on specific payloads specialized end effectors in addition to the standard set may be sent up to SEPS with the special payloads service items.

Figure 4-9 shows some joint concepts considered. Parallel stowage is desirable for SEPS. The offset joint offers many advantages in drive mechanism implementation. It is not inherently limited to ± 180 degrees rotation. The centerline offset of the joint is not a significant disadvantage in SEPS applications.

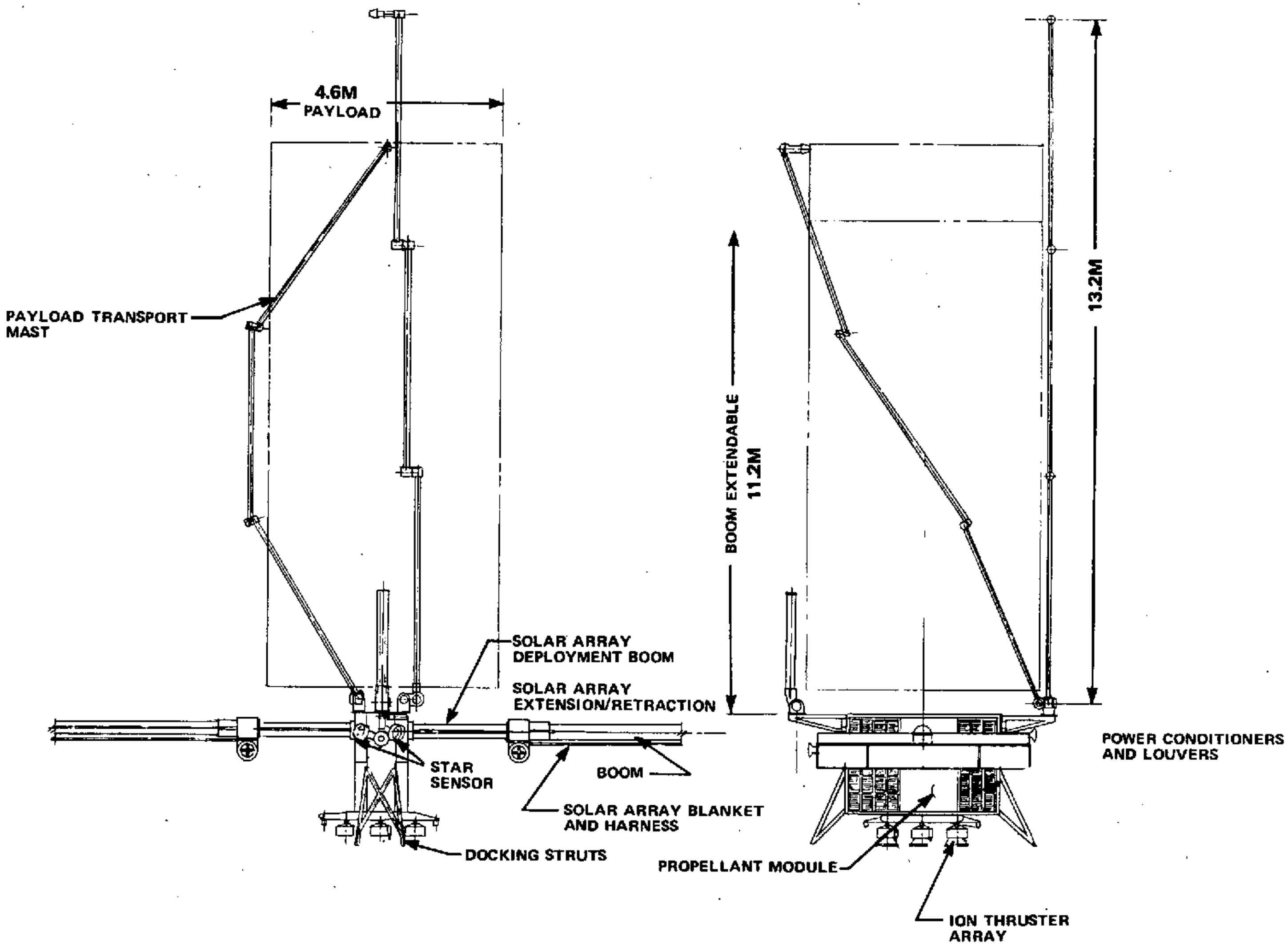
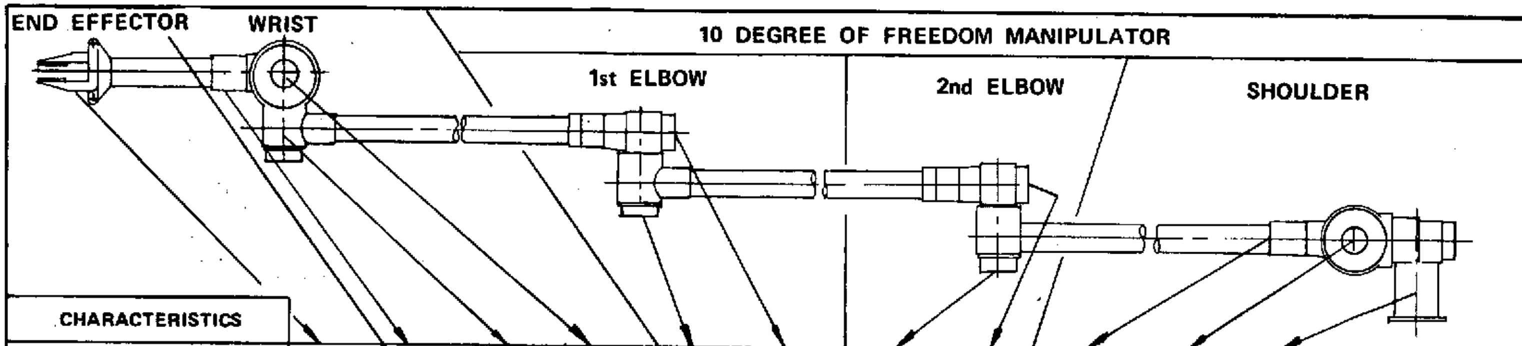


Figure 4-7. MANIPULATOR REACH AND MOUNTING POSITION ON SEPS



CHARACTERISTICS / JOINTS & MOTIONS	GRIPPER	ROLL	YAW	PITCH	YAW	ROLL	YAW	ROLL	ROLL	PITCH	YAW	
JOINT OUTPUTS												
TORQUE (FT LBS)		500									500	
TRAVEL DEGREE	CONT	± 180									± 180	
SPEED DEGREE SEC UNLOADED		10°/SEC									10°/SEC	
JOINT INPUTS												
BACKLASH ARC-SEC	2	6									6	
BACK DRIVEABLE	NO	YES									YES	
FRICTION IN GEAR TRAIN	HIGH	MIN									MIN	

Figure 4-8. RECOMMENDED MANIPULATOR CHARACTERISTICS

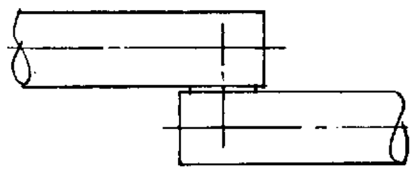
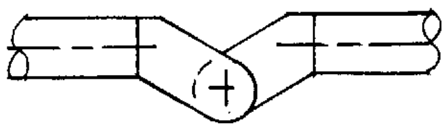
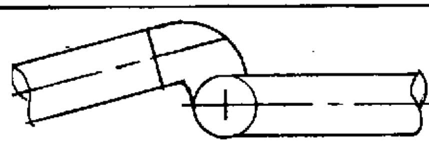
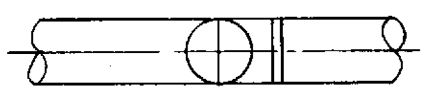
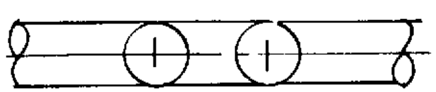
NO.	CONCEPT SCHEME	DESCRIPTION	ADVANTAGES	DISADVANTAGES
1		OFF SET ELBOW JOINT TOP VIEW	GREATER THAN 180° TRAVEL WITH PARALLEL STOWAGE	CENTER LINE OFFSET
2		DUAL TRANSITION SECTION	PERMITS 180° TRAVEL WITH PARALLEL STOWAGE	MAXIMUM REACH DISTANCE WHEN 3 PITCH ANGLES CO-LINEAR
3		SINGLE TRANSITION SECTION DESIGNED INTO ONE ARM SEGMENT	LESS THAN 180° TRAVEL WITH PARALLEL STOWAGE	ARM SEGMENTS NON-SYMMETRICAL
4		INLINE JOINT WITH YOKE STRUCTURE	SEGMENT SYMMETRY	ANGULAR TRAVEL LIMITED TO LESS THAN 180° REQUIRES LARGER YOKE DOES NOT PERMIT PAR. STOWAGE
5		DUAL PIVOT POINTS WITH EACH CAPABLE OF 90° ROTATION	SEGMENT SYMMETRY WITH AND PARALLEL STOWAGE	MORE COMPLEX HIGHER WGT. ADDITIONAL CONTROL FUNCTIONS

Figure 4-9. ELBOW JOINT CONCEPTS

Figure 4-10 is an inboard profile of the manipulator indicating the application of the harmonic drives and torque motors.

Figure 4-11 shows an isometric cutaway of a joint. Figure 4-12 is a block diagram of the manipulator's electronics and the interfaces with ground control and SEPS computer.

4.3.2 Payload Transport Mast

Figure 4-6 shows the installation of the payload transport mast on SEPS.

Figure 4-13 shows the recommended general design approach. A study of Figure 4-13 indicates the potential of this type mechanism for very high reliability. Its drive system is extremely simple and easily provided with several levels of redundancy, as indicated in the figure.

The mast section is collapsible onto the storage drum as rotation of the drum produces the forces that flatten its free form cross-section shape. Driving the drum in the extend direction will extend the mast. Each unit length will assume its free-form cross-section as it passes through the restraining sections of the housing.

This mast concept has very compact stowage for long mast lengths. It is simple, has high torsional rigidity for a collapsible system, has high bending strength, and good column characteristics. SEPS high Isp performance is not very sensitive to inert mass. The small, if any, mass penalties associated with use of these biconvex, edge welded, collapsible masts is more than offset by many other desirable features including high reliability and predictability of dynamic structural behavior. NSI also recommends this approach for the solar array spars as indicated on later drawings.

4.3.3 Payload Transport Shell and Diaphragm

The transport shell and one diaphragm are shown on Figure 4-14. The shell is a simple monocoque, honeycomb core sandwich, half cylinder. Its only

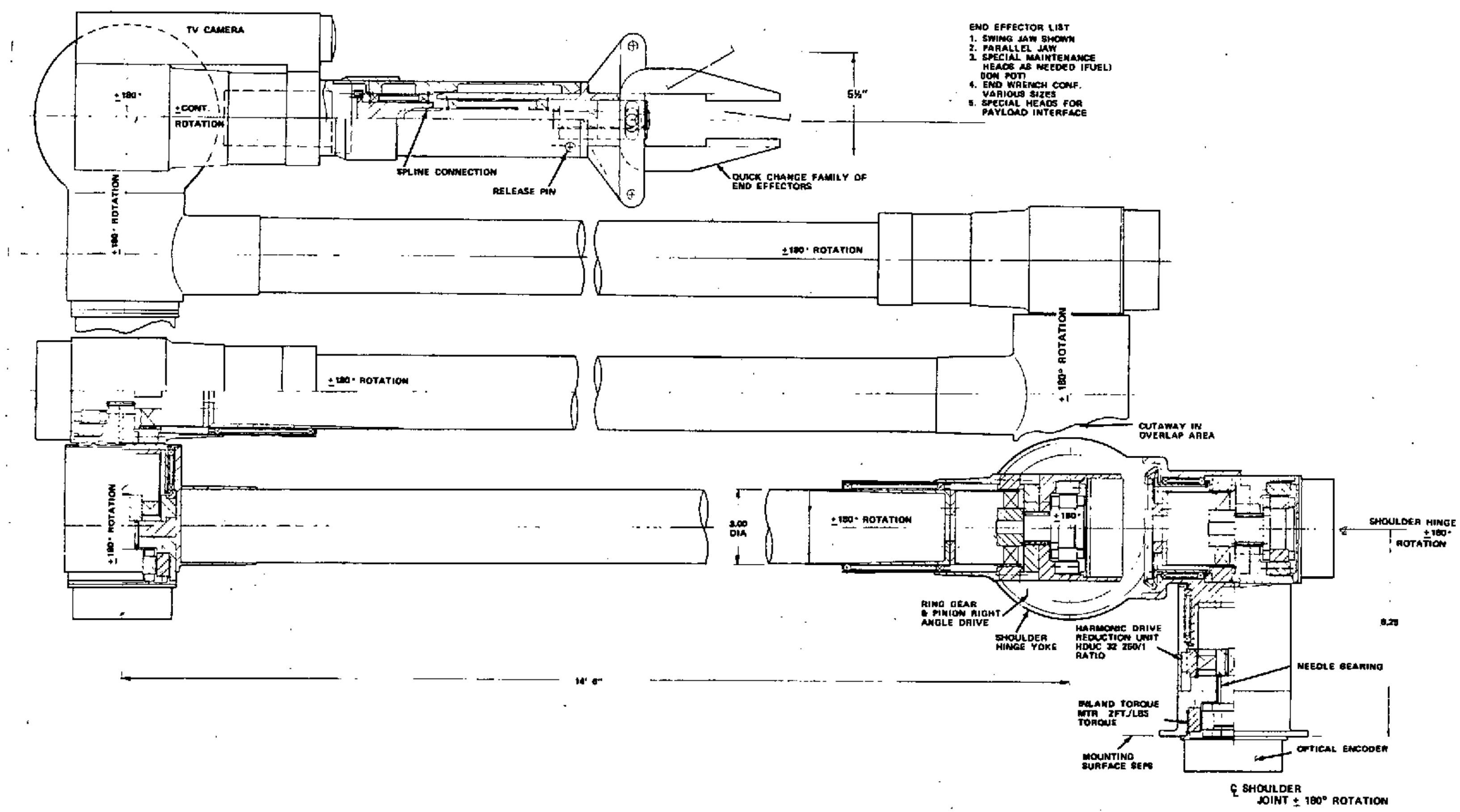


Figure 4-10. SEPS MANIPULATOR ARM

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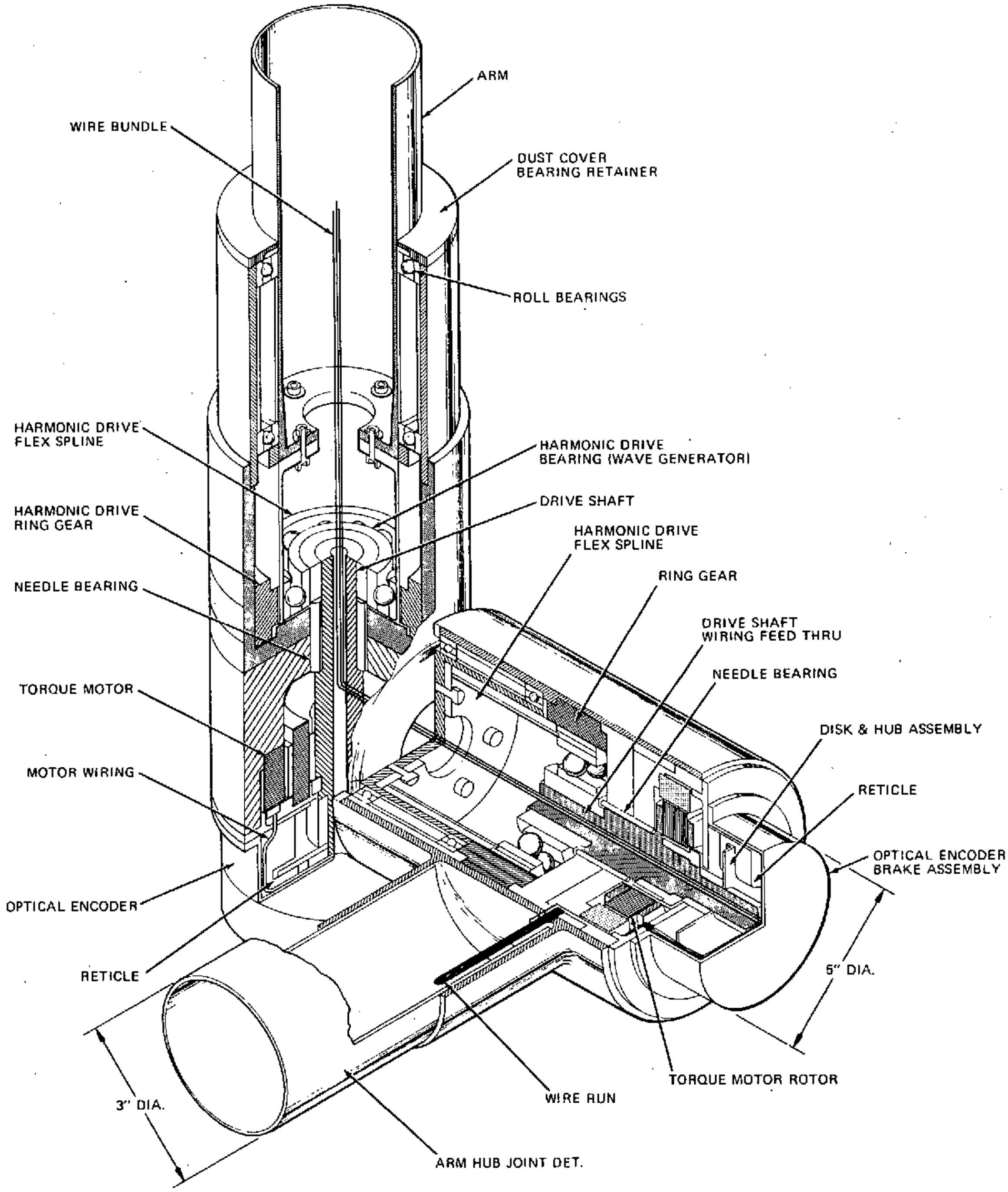

Figure 4-11. ISOMETRIC CUTAWAY OF MANIPULATOR JOINT

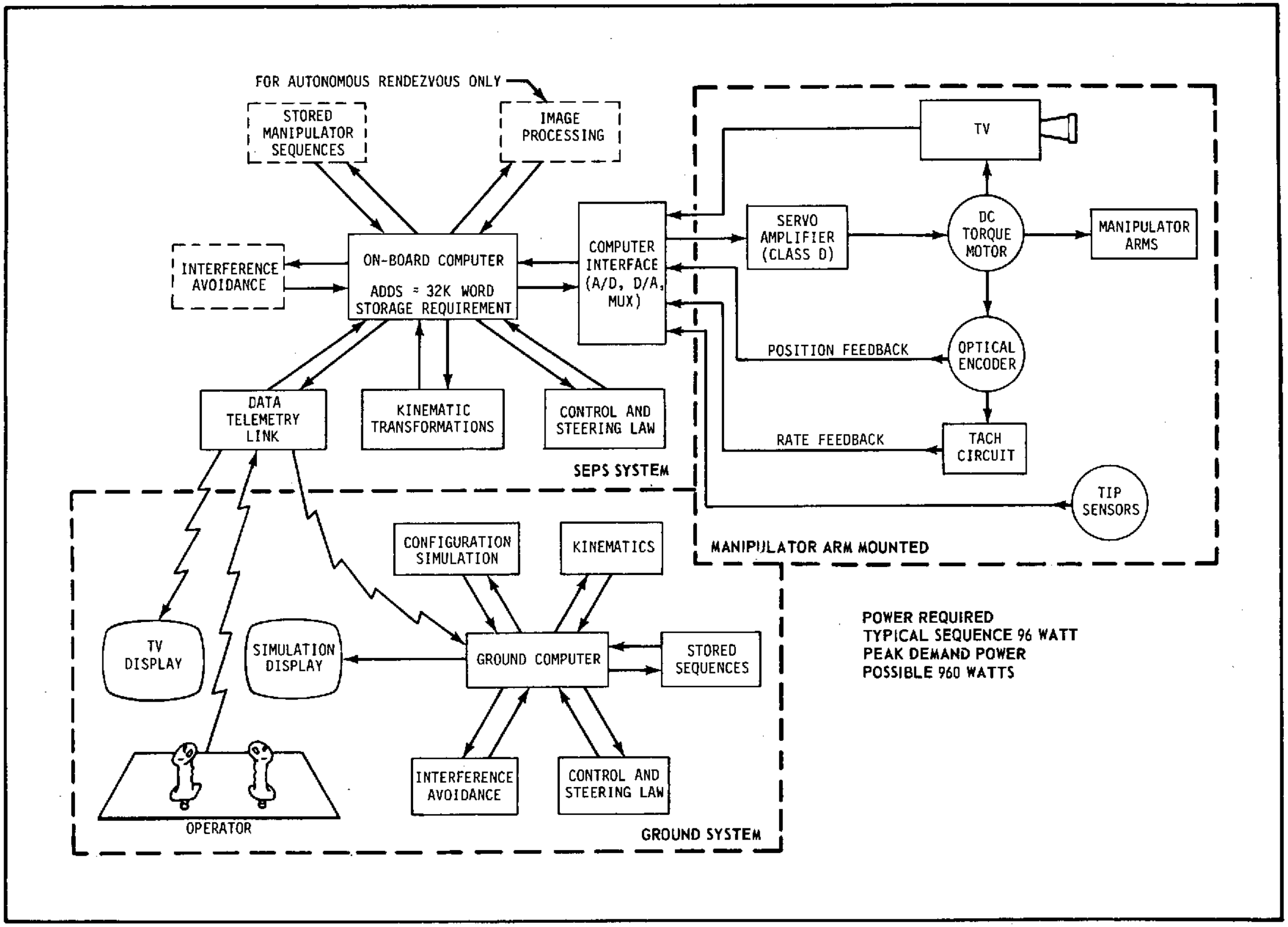

Figure 4-12. MANIPULATOR SYSTEM SCHEMATIC

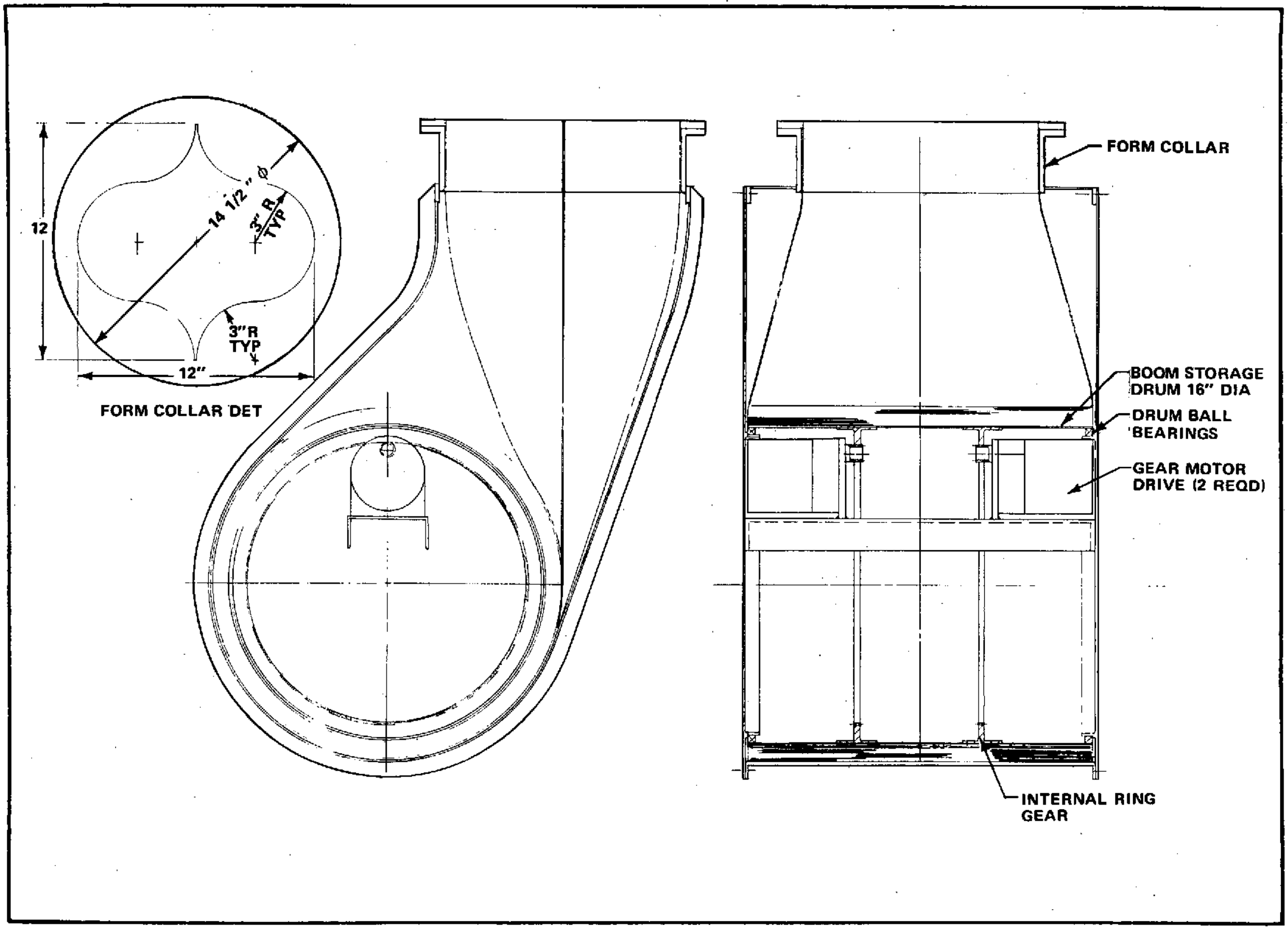

Figure 4-13. BICONVEX PAYLOAD TRANSPORT MAST

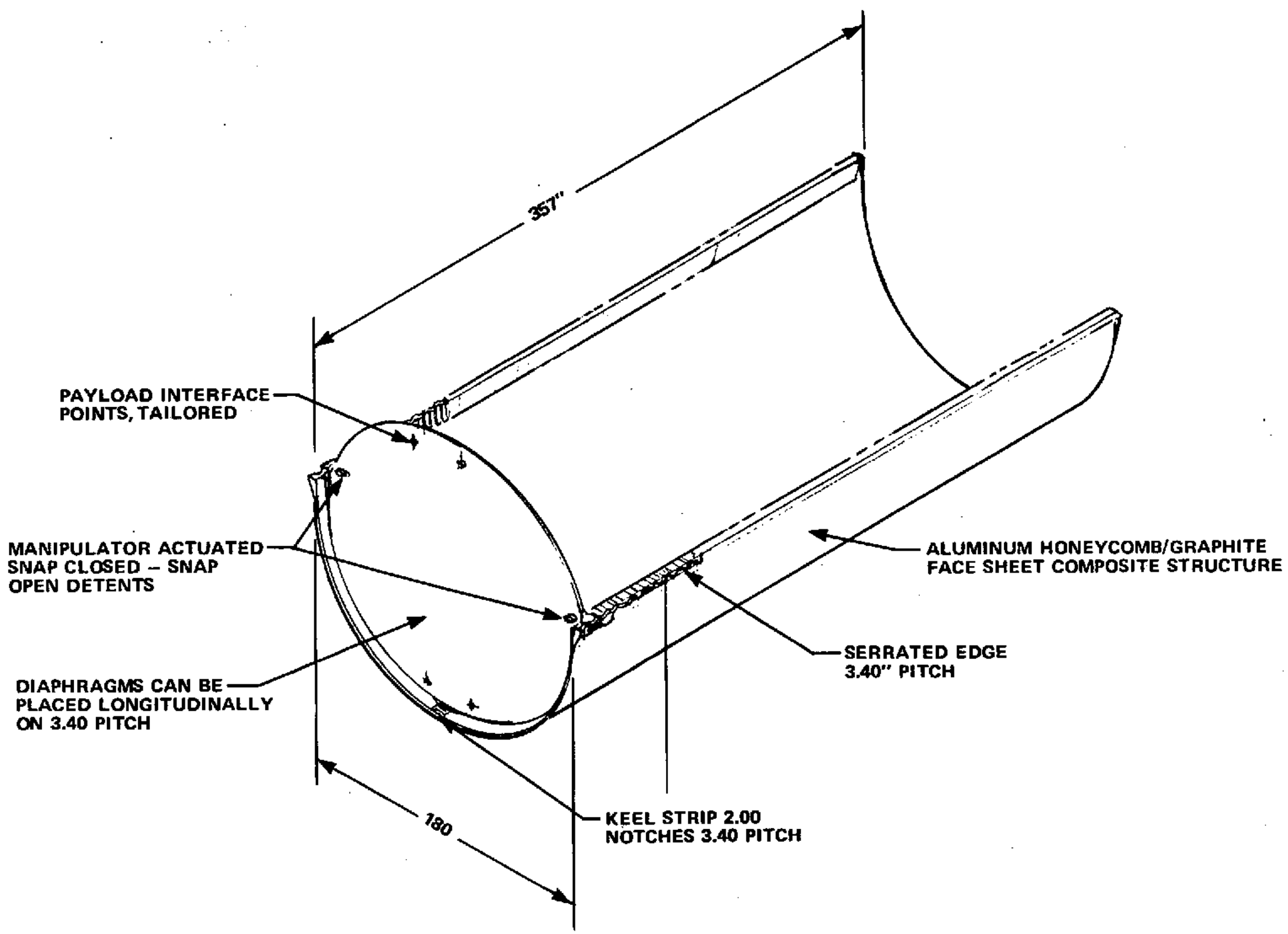

Figure 4-14. SHELL DIAPHRAGM SKETCH

unique features are a centerline keel strip and corrugations at gunwale edges. Notches in the keel plus the corrugations allow the payload mounting diaphragms to be located at any desired location within ± 8.6 cm pitch position. Payloads are mounted to the diaphragms.

One approach to standard diaphragm design is shown on Figure 4-15. Typical payload mounting hole locations are indicated by plus (+) marks at the corners of a 1-foot square grid pattern. A cross-section through a hole is shown. Honeycomb cells in the area of the hole flanges are filled with high crushing load polyurethane foam or other compressive load-bearing material.

The terminal end of a payload strut that goes through this hole is indicated on Figure 4-16. When the worm wheel nut is driven in the unscrew direction, it lifts the locking surface off the inside face sheet of the diaphragm. This leaves the strut free to go further through the hole. As the nut is further unscrewed by the impact wrench inside the manipulator hands, it lifts the "T" bar, collapsing the spring-loaded fingers which can then be withdrawn through the hole.

To attach a payload with mounting struts terminating in this device, the "worm nut" is in an intermediate position so the fingers are sprung open. If the nose of the strut is placed in a diaphragm hole and pushed toward the hole, the slope of the hole walls collapse the spring-loaded fingers, and the strut end with the folded fingers slides through the hole until the fingers clear the back side of the hole. The fingers then spring open and the payload strut is loosely attached to the diaphragm. The payloads are firmly fastened to the diaphragms by driving the worm wheel nut until the backing surface is firmly seated to the inner face sheets of the diaphragm.

Figure 4-17 shows a payload transport mast clamp housed in a 2-inch thick section of the diaphragm in the keel tang area. When the diaphragm is lifted from the payload transport shell, springs force it out to a position ready for attachment to the payload mast. When pressed against the payload mast section, the clamp arms spread further until the rollers snap over the mast cusps. The mast is now trapped by the clamp as shown.

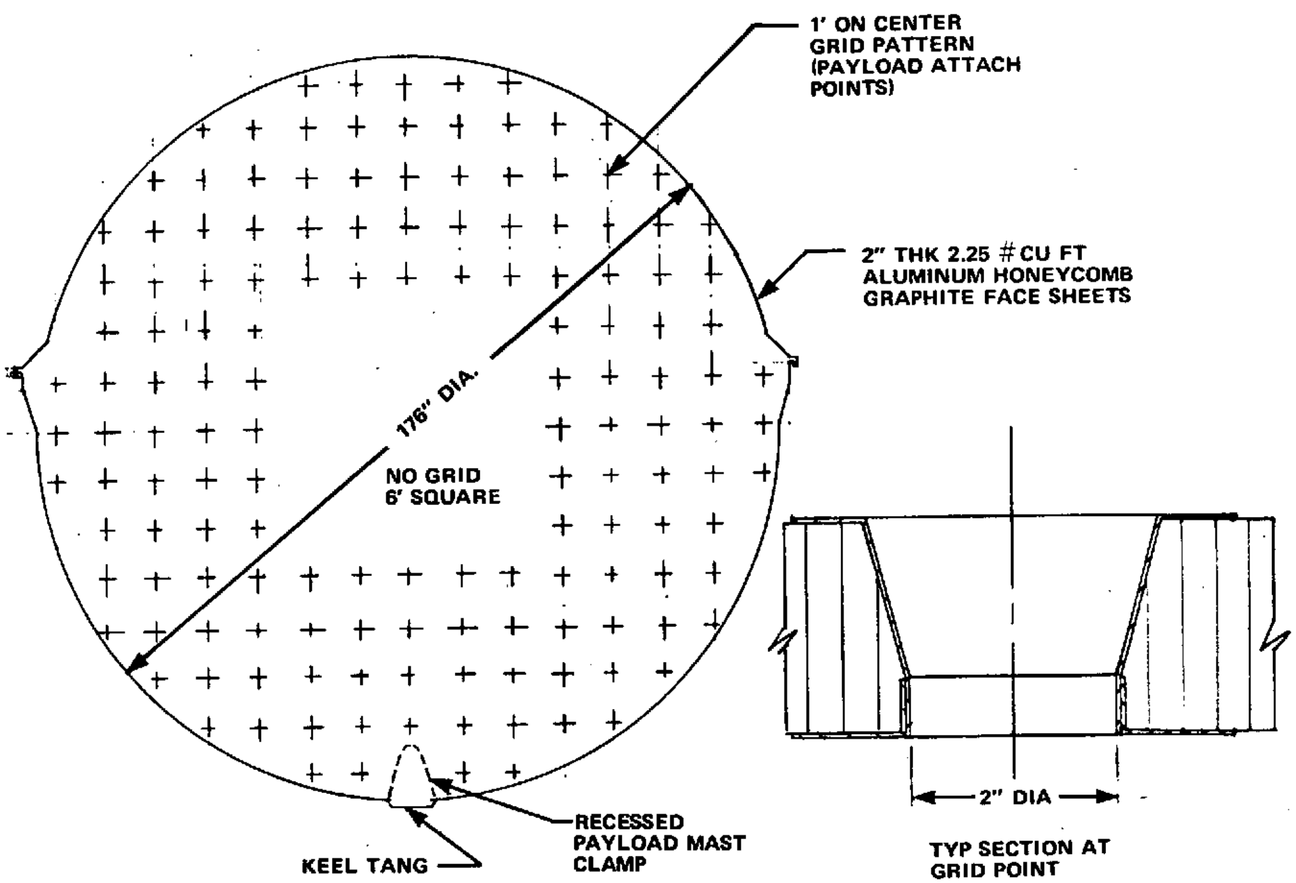

Figure 4-15. INTERFACE DETAIL GPME PAYLOAD MOUNTING DIAPHRAGM

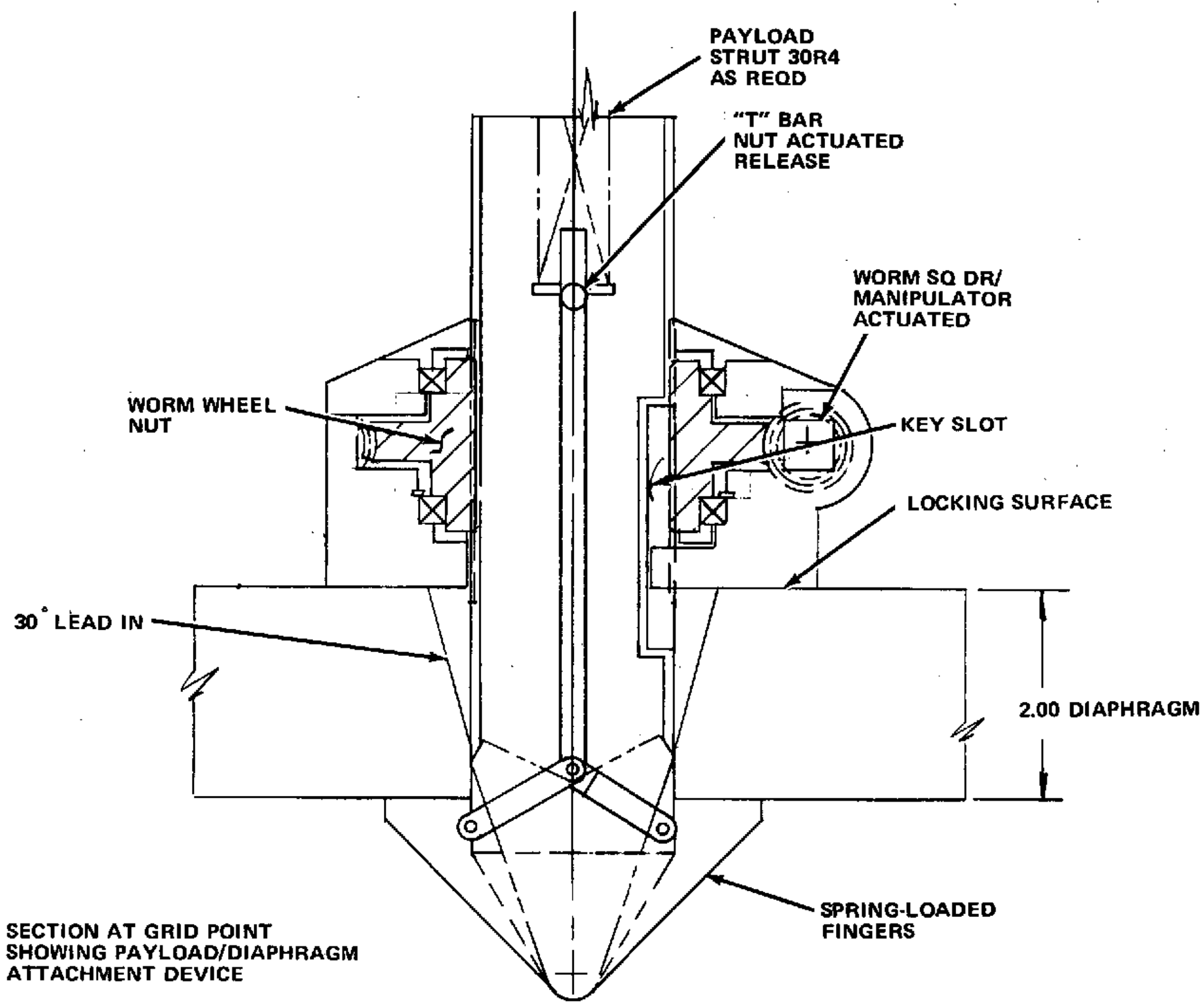

Figure 4-16. INTERFACE DETAIL PAYLOAD MOUNTING DIAPHRAGM MAST CLAMP

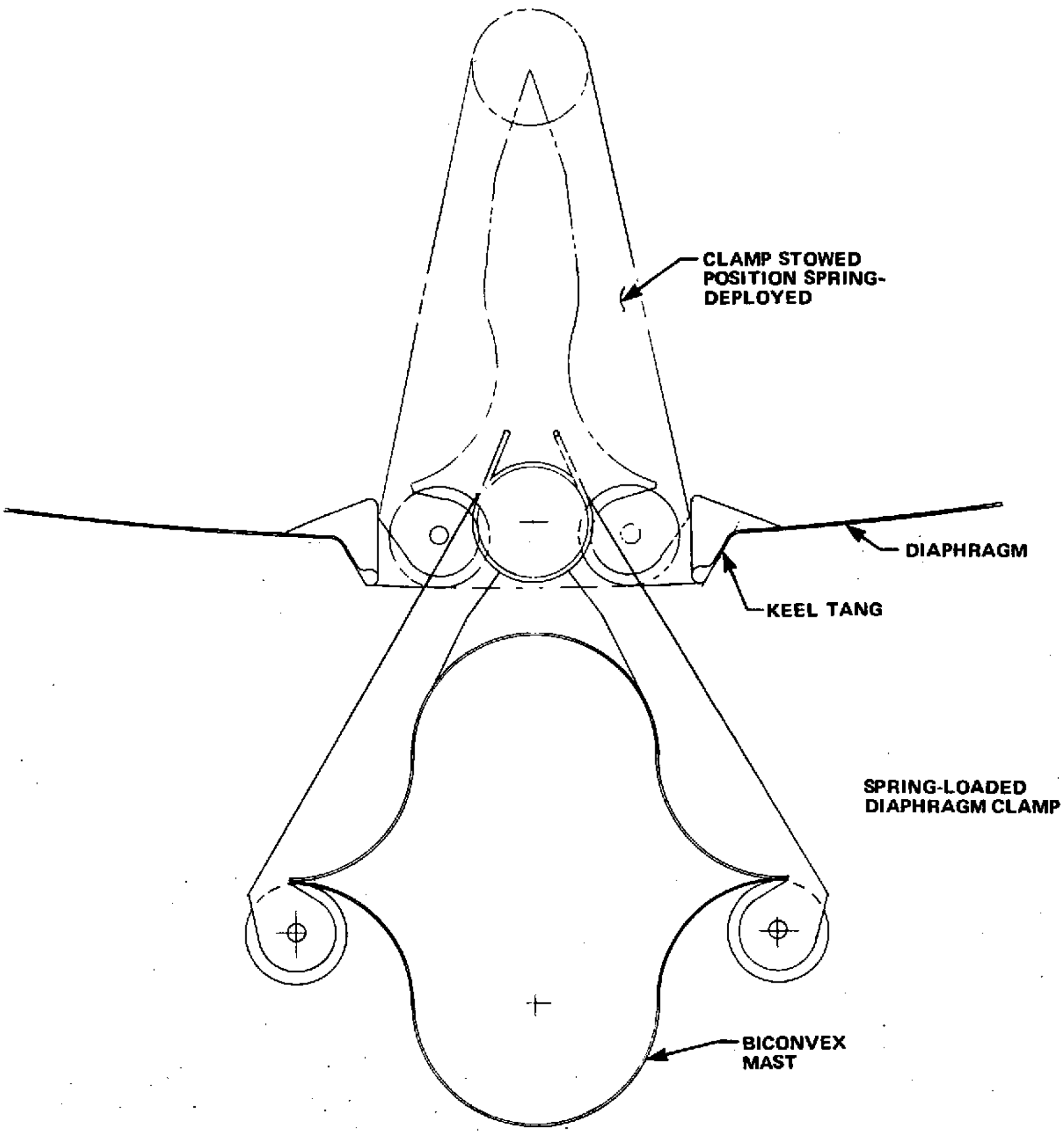

Figure 4-17. INTERFACE DETAIL PAYLOAD MOUNTING DIAPHRAGM MAST CLAMP

Figure 4-18 shows the interface longeron that is mated to the Orbiter's cargo mounting longeron. They extend 9.1 m down either side of the Orbiter's cargo bay. The shell-Orbiter interface longeron is retained by the Orbiter as long as it is using the transport shell to support cargo missions. The interface longeron's attachment requires no modification to the Shuttle, being attached or removed by use of the Orbiter's standard payload attachment pins. The corrugated edges of the transport shell gunwale fit into the corrugations of the adapter longerons. Each corrugation is designed to carry part of the 9g Orbiter crash load. In this way no concentrated loads are transmitted to the transport shell. It can therefore be a very light weight structure. At selected areas near the Orbiter's attachment pin locations, the interface longerons have Z-load locking bars which are pushed through holes in the adapter longeron into matching holes in the corrugated gunwale section of the transport shell.

The complete GPME set described in the preceding section is completely compatible with IUS, Tug, and Orbiter. The GPME set allows, to the extent practicable, the decoupling of Tug, Shuttle, and multiple payload package prelaunch operations.

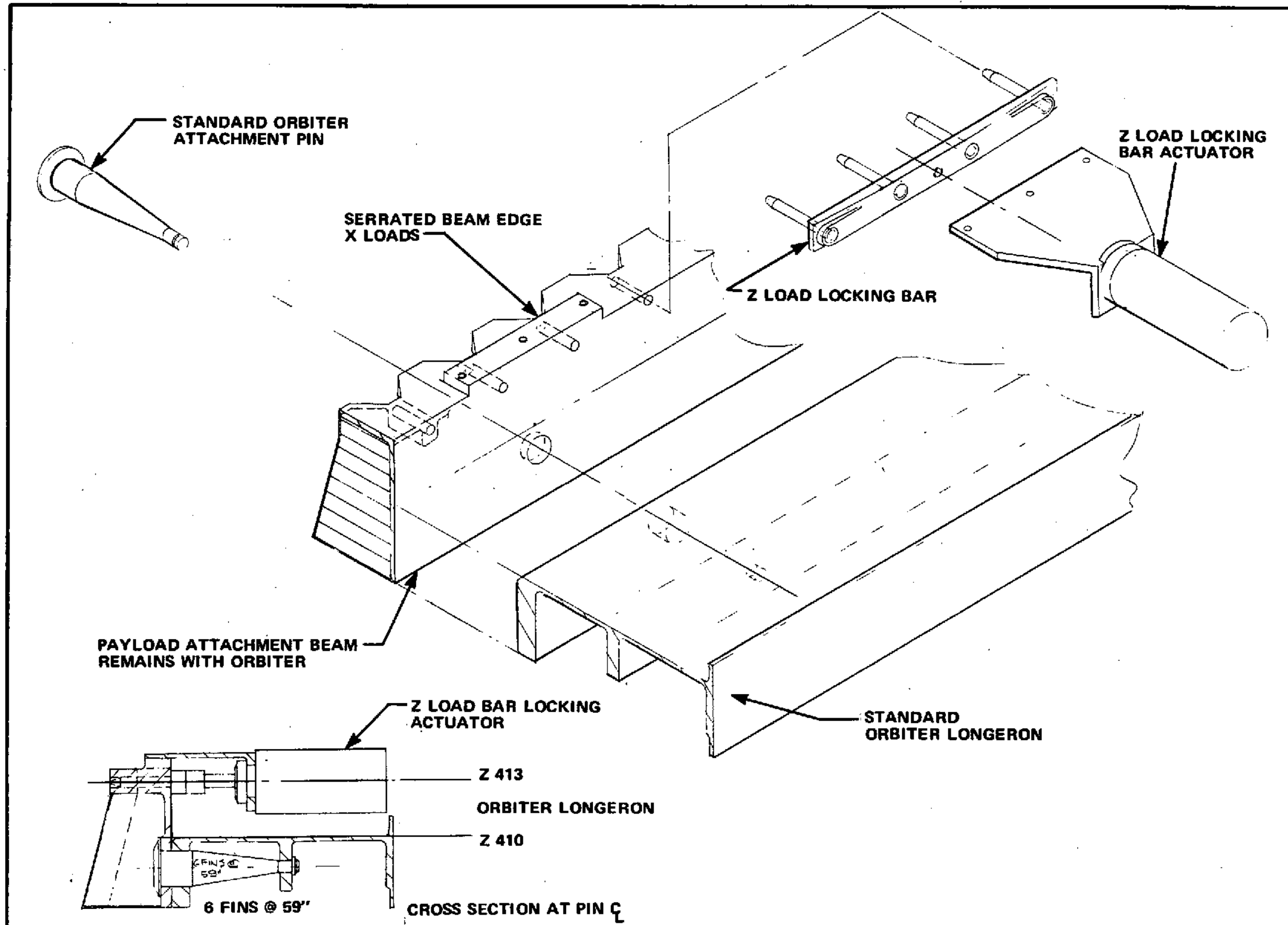

Figure 4-18. INTERFACE DETAIL PAYLOAD SHELL -- ORBITER

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V IMPACT OF SEPS
INTERFACE REQUIREMENTS

Section V

IMPACT OF SEPS OPERATION WITH STS ON ORBITER, IUS, AND TUG PHYSICAL INTERFACE REQUIREMENTS

5.1 GENERAL CONSIDERATIONS

The delivery to or retrieval of SEPS from typical IUS/Tug payload transfer orbits imposes no additional physical interface requirements. SEPS as an individual payload to be delivered has very modest support requirements well within the design capabilities proposed for IUS and Tug or those baselined for the Orbiter.

Figure 1-9, the System Operational Profile, showed that only three scheduled SEPS launches and one retrieval were required to accomplish the reference mission model from 1981 through 1991.

SEPS augmentation of IUS-Tug transportation capabilities allows the use of the GPME concepts described earlier, which greatly simplifies the Orbiter, IUS, and Tug ground operations involvement in multiple payload delivery operations. The transport shell always presents a single structural payload interface to the IUS, Tug, and Shuttle Orbiter. Because all payload inertial loads are distributed into the shell which distributes the total load to the Orbiter's cargo bay longerons in an acceptable way, loads on IUS and Tug are lower than design limit loads derived from certain individual payloads carried by IUS and Tug.

The additional interface requirements for STS elements, therefore, derive from the fact that with SEPS in the system multiple payload cargo manifests may contain up to seven or eight payloads instead of three or four. The potential primary impact, as might be expected, is in the avionics support areas of telemetry, command, and power supply.

Other potential added demands are in the areas of propellant dumping, venting, RTG cooling, and payload contamination protection. None of these represent extra requirements since the character of the multiple payloads to be delivered with Tug-SEPS sorties does not present a greater requirement than some of the more complex single and dual payloads transported without SEPS. Combining of multiple payloads on the transport stage results in an interface equivalent to a single payload. Avionics factors will be discussed in more detail later.

5.2 IUS–SEPS INSTALLATION IN ORBITER

Figure 5-1 shows the IUS with a payload shell holding a SEPS for its initial launch into space, and as added payloads, a SEOS payload and a communications satellite. In Section IV we described the payload to diaphragm and transport shell to Orbiter interfaces.

The Transtage is mounted to the Orbiter in accord with the baseline STS system design. Since that interface is not affected by SEPS it is not depicted. The IUS is not structurally attached to the transport shell during Orbiter ascent. A small gap exists between the shell and adapter structure during Orbiter ascent; therefore, no loads due to Orbiter flexing from flight loads or airframe heating are transferred from shell to IUS. For deployment from Orbiter and for IUS freeflight to its maximum energy orbit with this payload, IUS is attached to one adapter diaphragm whose outer edge is fabricated to a large L section ring frame. Eight electric motor driven screw jacks operate clamping latches to clamp the transport shell to the L-frame just prior to deploying from the Orbiter. The latches and a crosssection through the structure just described is shown in detail A of Figure 5-1. They are actuated by IUS power on command received through IUS.

The adapter diaphragm, of different diameter in its upper and lower halves, is permanently attached to IUS through its standard interface for attachment of individual payloads. Therefore, no modifications are required for IUS to operate with SEPS or for compatibility with the recommended GPME.

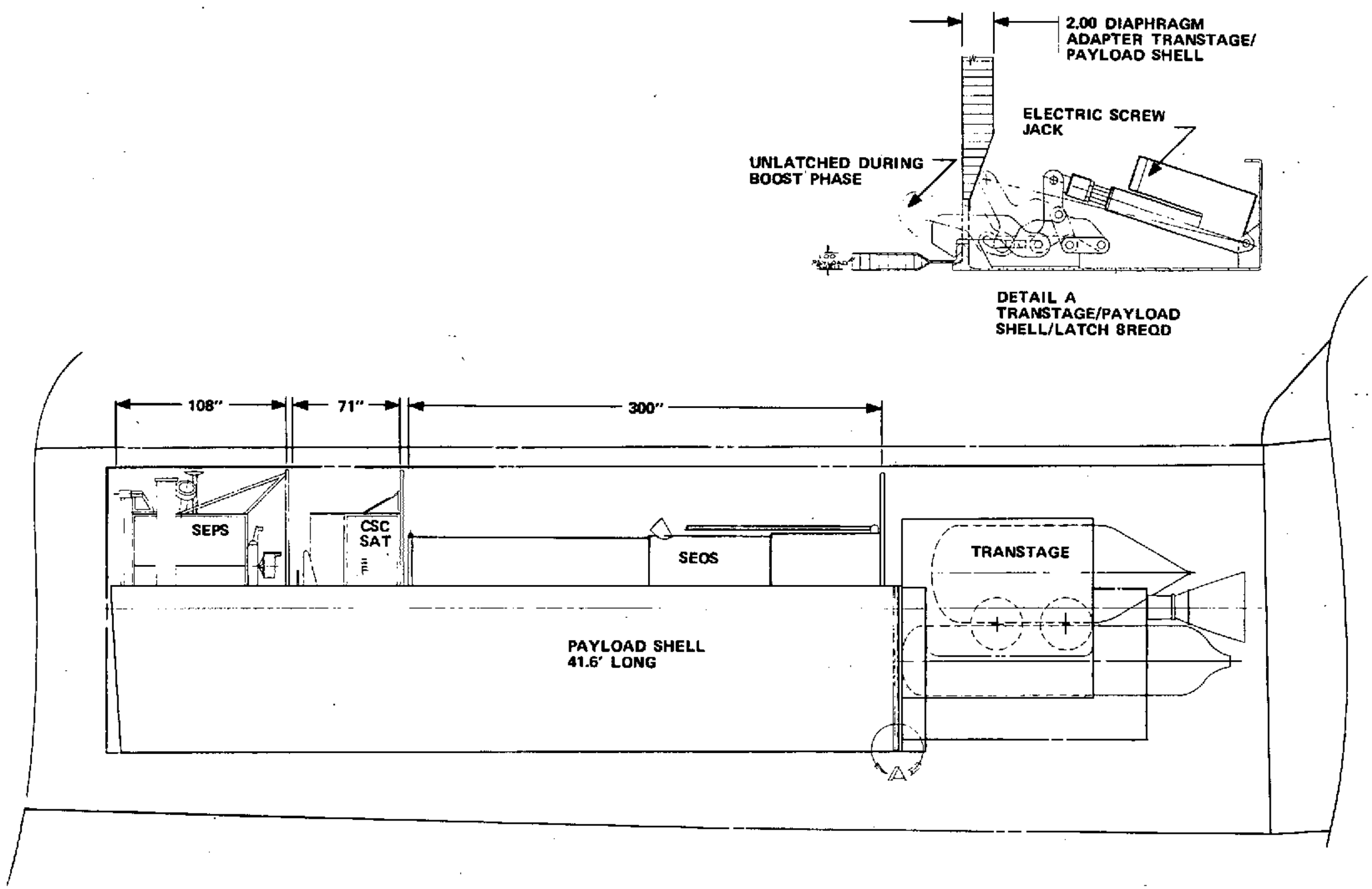

Figure 5-1. TRANSTAGE/PAYLOAD SHELL ARRANGEMENT

The system has several advantages.

1. It is the shortest method for load transistion thus providing more net available payload installation space in Orbiter's cargo bay. Mass, from a cursory examination, appears to be nearly as low as for an optimal system under flight loads from IUS main engine thrust.

2. SEPS has the option of carrying payloads to geosynchronous orbit in the payload shell rather than transferring them individually.

3. The transport shell does not need to run the full length of the cargo bay. If IUS were made recoverable, then when it returns to Orbiter the payload shell can be mounted further forward in the Orbiter than it was for the ascent phase. The empty IUS is cantilevered from the shell-diaphragm assembly for the return to earth. This allows the Orbiter some degree of control over descent payload c. g. location. On many of the flights it is feasible to recover IUS. As a matter of passing interest a 100 kw SEPS operating with an IUS alone can accomplish the total mission model with only 26 more flights than is required for SEPS + IUS + Tug.

4. By use of field splicing on the adapter ring, the shell can be retrofitted for use with Tug.

5.3 TUG–SEPS INSTALLATION IN ORBITER

Figure 5-2 is a similar layout for SEPS + Tug with an arbitrary depiction of payloads. The interface of the shell and baseline Tug are tailored so no modification to the baseline is needed to match the baseline Tug. The detail equivalent to detail A of Figure 5-1 was shown in Figure 4-5.

The soft latching for Orbiter ascent is also achieved with Tug. Similar options to those described for IUS are available.

5.4 SOME PAYLOAD–TUG–SHELL SPECIAL INTERFACES

One of the primary advantages of the payload shell concept is that multiple payloads are presented to Tug and Orbiter as single packages. The shell diaphragm mount arrangement also has the advantage that access to individual payloads is made easier. Payloads requiring contamination shrouds or other individual treatment can be accommodated readily since each payload is base mounted.

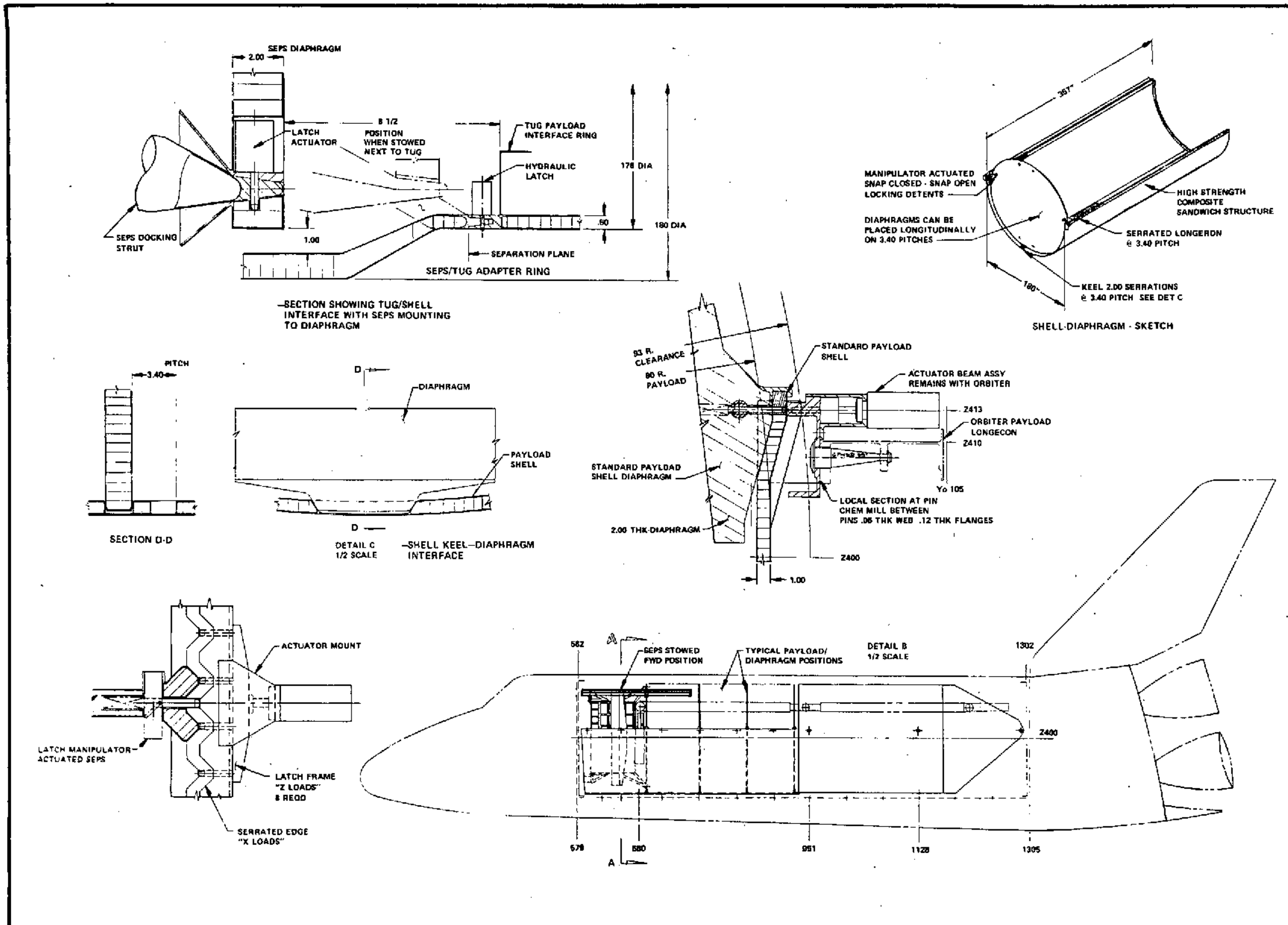

Figure 5-2. TRANSPORT HALF SHELL DESIGN CONCEPT

One of the special treatments required by some payloads is provision of contamination shrouds and filtered clean air to maintain the high cleanliness level required by some sensors and instruments.

Figure 5-3 shows schematically a cutaway of payloads mounted in the half shell. The double wall plastic bags when inflated form enclosures over only those PLs requiring protection. Shrouds can be installed before or after diaphragms are installed in the shell. Figure 5-3 also shows a shroud where the diaphragms at each end form the end closures.

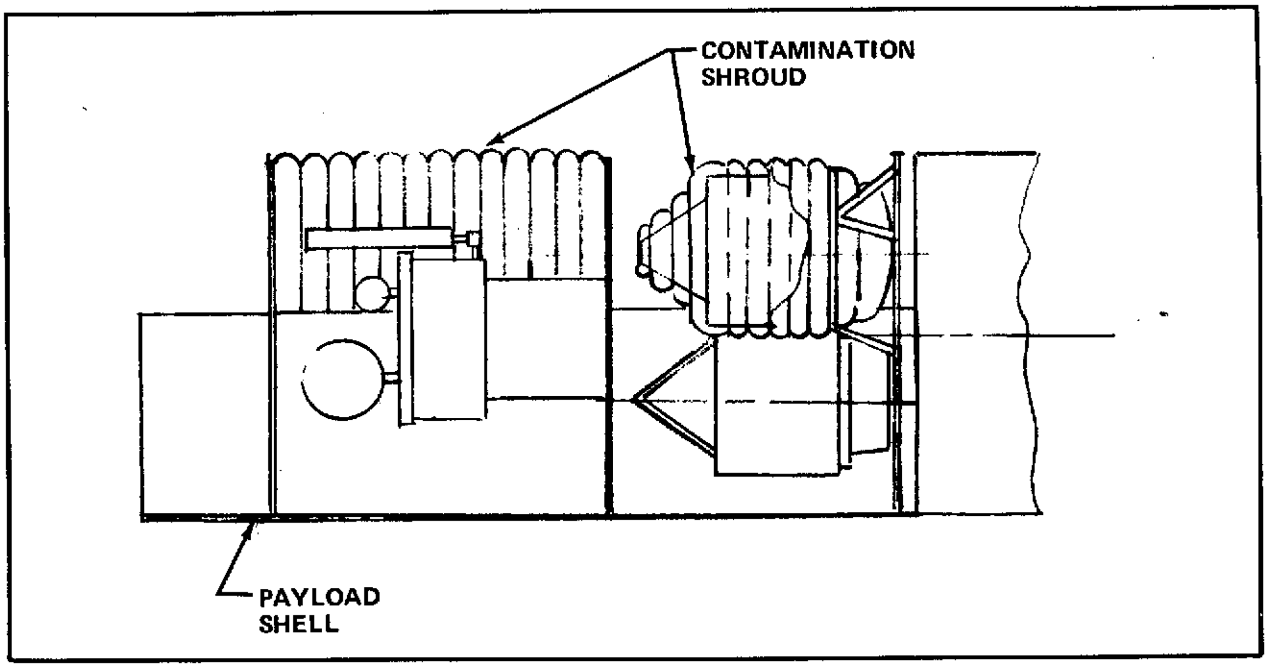

Figure 5-3. SCHEMES FOR PAYLOAD CONTAMINATION PROTECTION

Payloads can be located on the diaphragms to maximize accessability to those most likely to require adjustments or servicing after their installation.

Figure 5-4 shows an attractive alternate that may be used when found desirable. The containment shroud is formed by taping down a plastic sheet at the points where it contacts the diaphragms and along the gunwale section of the transport shell. This converts an entire longitudinal section to a contamination protected volume in a simple manner that provides easy access if required.

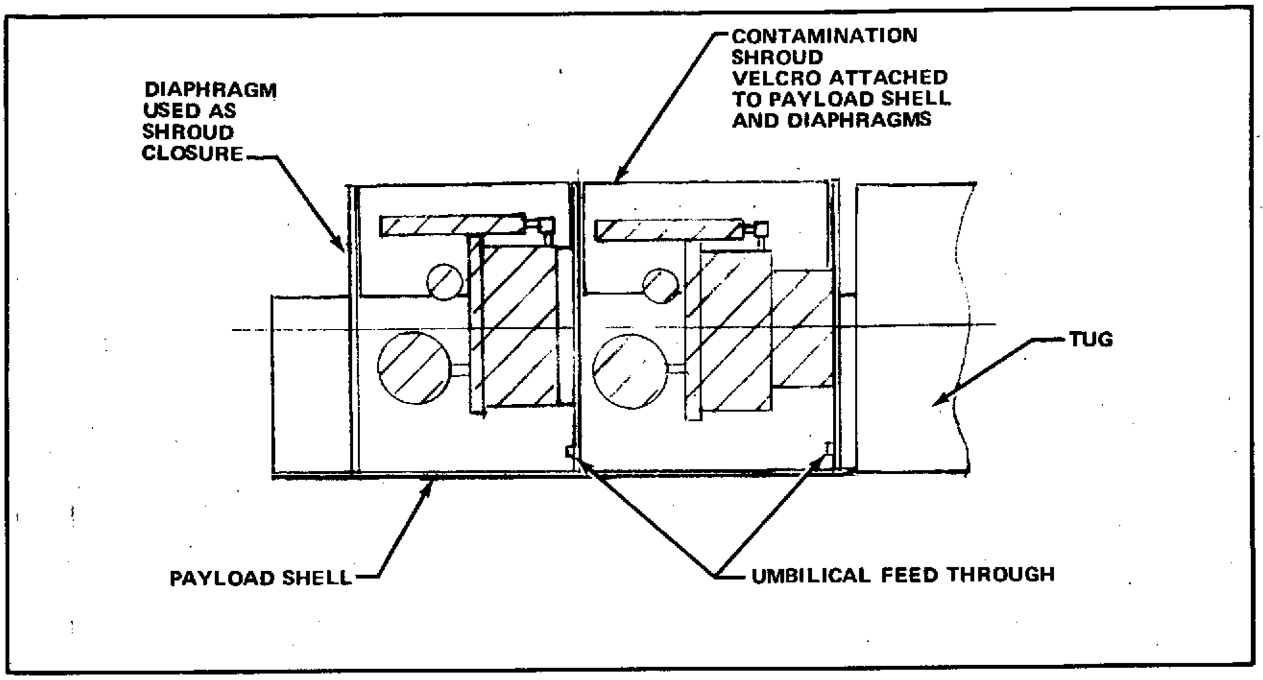

Figure 5-4. CONTAMINATION SHROUD ARRANGEMENT

A few of the planned payloads carry propellants in large enough quantities to require venting. Figure 5-5 shows three alternate means for venting these propellants through the Tug or the Orbiter. In keeping with the objective of decoupling multiple payload integration, Tug prelaunch activity, and Orbiter prelaunch activity, NSI recommends that all payload package support should be through Tug or Orbiter and then overboard.

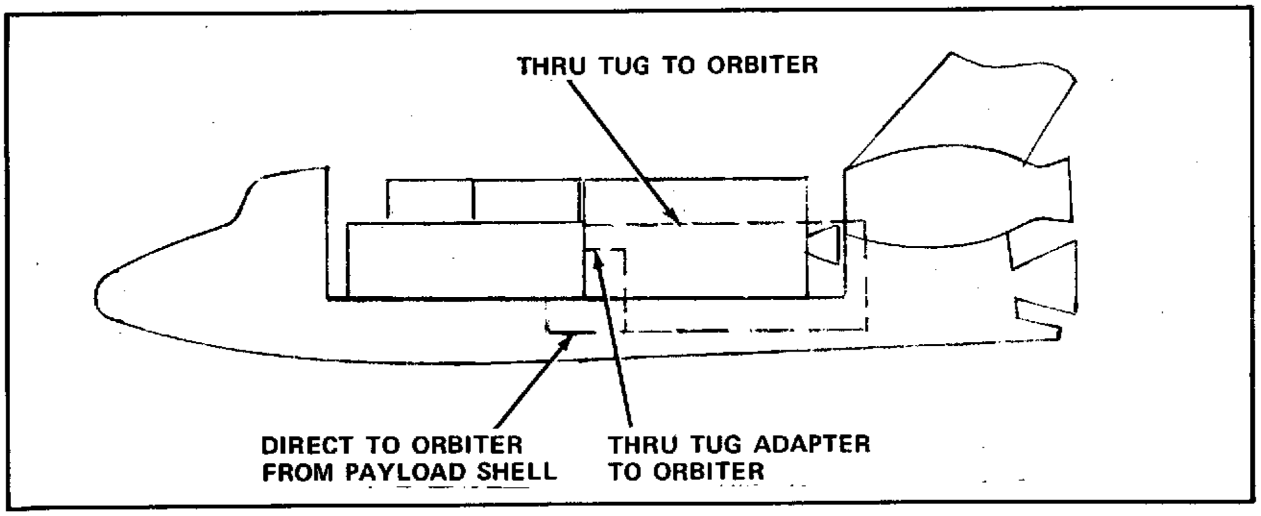

Figure 5-5. SPACECRAFT PROPELLANT VENTING OPTIONS

If more than one payload in the transport shell requires venting, then collection lines will be fabricated to channel all vents to the Tug umbilical point for vented propellants.

5.5 SEPS SAFETY AND INTERFACE CONSIDERATIONS IN RELATION TO ORBITER

Safety and interface discussions will be considered in the following sequence:

- SEPS as one of a multiple payload group for delivery in terms of Orbiter safety requirements and interfaces
- Multiple payload avionics potential requirements
- Gases and liquids venting and dumping requirement.

Figure 5-1 shows SEPS with other schematically represented payloads in a transport shell with IUS in the Orbiter cargo bay. Tug would mount SEPS similarly. The transport shells for IUS and Tug are essentially identical and could be developed for interchangeability. SEPS is mounted on a standard GPME diaphragm and has no direct structural interface with the Orbiter or IUS-Tug.

SEPS, if nominally fueled for the initial deployment mission, has a mass of about 2725 kilograms (6,000 pounds). SEPS contains only four fluids: pressurizing N_2, battery fluids, mercury, and hydrazine.

The pressurizing N_2 for the mercury expulsion system has a peak charged pressure of 28 N/cm^2 (40 psia). The N_2 is contained inside the mercury propellant tank; tank design limit load is controlled by the 9 g Shuttle crash load factor. Design for containment to peak cargo bay temperatures is a negligible mass penalty. Pressure relief venting to the cargo bay interior is acceptable. No caution and warning signals or control from the orbiter is required.

The N_2 for ACS has a peak charge pressure of 138 N/cm^2 (200 psia) and is also within the pressure shell of the N_2H_4 tanks. The tanks contain 109 kg (240 pounds) of N_2H_4. The tanks will be designed for containment of N_2 and N_2H_4 at peak cargo bay temperatures. Backup N_2 pressure relief vent to the cargo bay will be used for added safety. No propellant dump for this quantity of N_2H_4 is required. The only way in which the N_2H_4 can cause overpressure is by thermal heating to boiling temperatures, catalytic decomposition, or spontaneous decomposition at high temperatures. Catalytic decomposition would occur when the catalyst is first inadvertently introduced so it is not an orbiter inflight problem. Heat required for the remaining two catastrophic situations (with insulated tanks) requires a fire in the cargo bay.

Because of the space thermal requirement, both propellant tanks are insulated. No condition that has not destroyed the Orbiter will cause monopropellant decomposition of the N_2H_4 in SEPS. No C&W or command lines to/from the Orbiter are required.

SEPS, like most long-life spacecraft, uses Nickel-Cadmium batteries which are sealed. The batteries will be designed for containment. No C&W or command lines to/from Orbiter are required.

SEPS is designed to have no separation or deployment ordnance. All separation functions are controlled by reversable motors or with the aid of the manipulators. The Orbiter may require status information and command control for latching.

5.6 IUS–TUG AVIONICS SUPPORT TO SEPS

NSI believes the most desirable approach to avionics support for all payloads mounted on Tug is from Tug, since the support must be continued after separation from the Orbiter. During ascent, Orbiter must support Tug by provision of primary power and data links into Tug.

The following requirements for avionics support of SEPS from Tug exist:

- During preluanch after the transport shell has been mated to Tug and after installation in Orbiter:
    - 150 watts power and 1,000 kbits/sec digital data during brief flight readiness status check periods. Thermal control power of about 200 watts could be required depending on temperature of Orbiter's N_2 purge gases. Presumably such low temperature N_2 will not be used.

- During Orbiter ascent and onorbit prior to Tug deployment:
  * Nominally no support; 200 watts periodically if required for thermal control
- During Tug deployment, parking orbits and ascent to SEPS initial parking orbit:
  * 200 watts primary power for thermal control
- SEPS initial startup and transfer of initial payload to SEPS payload mast:
  * 600 watts, uplink data rate 1 kbit/sec. This support requirement would last approximately 1 hour, 1,000 watt peak power required, total energy required 3 kw-hours. SEPS own communications system provides the required TV and other down data rates.

This deployment and initial payload transfer sequence is shown schematically in Figure 5-6. All of the above requirements are within Tug proposed capability. As indicated in Figure 5-6, one of the SEPS phased array antennas is exposed and SEPS' own systems can supply the capability.

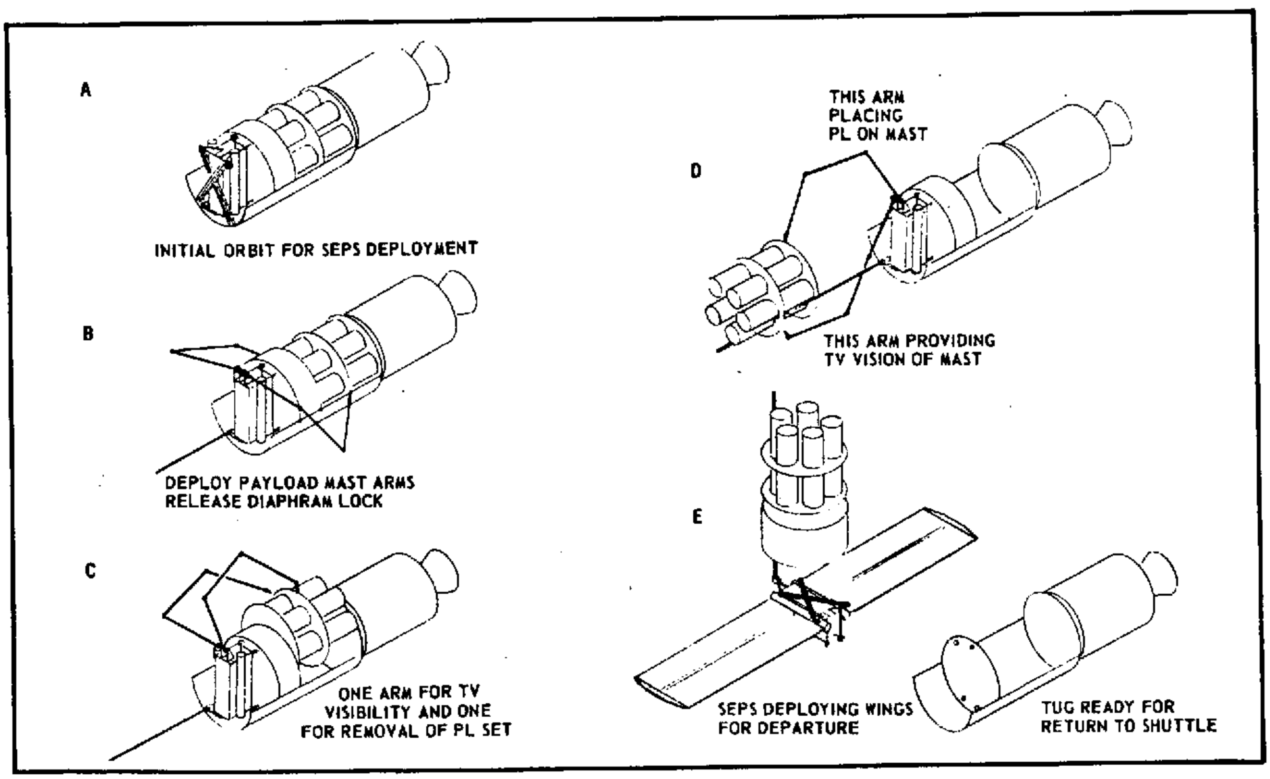

Figure 5-6. PAYLOAD TRANSFER INITIAL SEPS SORTIE

5.7 TUG—IUS SUPPORT TO PAYLOADS IN TRANSPORT SHELL

McDonnell Douglas and General Electric, teamed for the MSFC directed "IUS/Tug Payload Requirements Compatibility Study," reported in their midterm review the results of a payload design engineering committee analysis to determine nominal, maximum, and minimum values of Tug payload support requirements. The committee was composed of a group of experienced payload design engineers selected from the GE staff to provide specific support for that study group. Recent results of this study indicate that only payload status and subsystem viability checks will be conducted until the payload spacecraft are deployed. All spacecraft payload demands, on that basis, are reduced to data rate levels of less than 1 kbit/sec and power levels to 200 or less watts.

The Tug and IUS proposed baseline capability is therefore adequate for operation with the larger number of payloads that will be on Tug for its payload transfer mission to SEPS.

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VI EO CONFIGURATION & SUBSYSTEM DESIGN ANALYSIS

Section VI

EARTH ORBITAL SEPS CONFIGURATION AND SUBSYSTEM DESIGN IMPACT ANALYSIS

6.1 BACKGROUND AND GENERAL CONSIDERATIONS

The original study objective for this task was to:

- Adapt the payload handling, servicing, transporting, and maintenance concepts to be developed in the study to the initial baseline SEPS derived from previous studies
- Assess the "design impact" that the adaptation above, the interface influences with STS, and the support of payloads during delivery, would have on "baseline" subsystems.

Several situations existed and more developed which resulted in a departure from the original concept. First, the previous study documents purporting to define the baseline SEPS did not establish a clear "baseline" at the subsystem level or did not provide enough design definition to allow a meaningful "impact" assessment to be made.

Second, this study's assessments of technology and evolution of new concepts, plus NASA's in-house evolving concepts of the subsystems, so departed from the rather nebulous initial baseline that it was no longer a meaningful reference standard.

Due to these factors, this section will discuss the rationale for selection of certain configuration characteristics and/or the technology assessments leading to NSI's suggested approach to a subsystem design. Reference to a "baseline" SEPS will simply mean reference to a 25 kw power level SEPS with the thruster subsystem performance specification provided by NASA, and to mass characteristics derived from Rockwell International's prior "Exhibit E" studies.

NSI, for reasons described in several sections of this document, recommends that SEPS design minimum power level at 1 astronomical unit (AU) should be at least 50 kw. NASA, however, directed that emphasis be placed on 25 kw power level configurations. Discussions in this document and configurations shown are at the 25 kw level except for discussions of trade studies.

In our technology assessments and our search for germain design detail we studied large volumes of material some of which contained "trade studies" that were largely statements of engineering judgement or preference by the individuals authoring the reference document. There were several cases where we did not challenge the data base or what the principal contending design approaches were, but we did disagree with the conclusions and resultant recommended design concepts. Simply put, our assessment of the source data and the state of technology plus our engineering judgment led us to different conclusions than those presented by the authors of the source documents.

6.2 EARTH ORBITAL (EO) SEPS CONFIGURATION DESCRIPTION

The 25 kw configuration evolved in this study is shown on Figures 6-1(a) through 6-1(c). The configuration is dictated by considerations of flexibility in mission application as a payload servicing and transport element of STS, a spacecraft bus for scientific missions, and for earth orbital multimission technology applications. Little real conflict in desired characteristics occurred between these missions with the exception of the requirement to place certain sensor packages on deployable structures.

The deployable structures are necessary so that sensors can see around the payload packages. Essentially all of SEPS structural mass except the ACS tanks, certain sections of the power processor support structure, the extendable mast, and the extendable section of the solar array support spars is dictated by the Orbiter safety requirement that structures remain intact under the Orbiter's 9g crash load criteria. The deployable structures are, therefore, very rigid for any loads they may see during space operation. Structures which support sensors must be insulated to avoid thermal distortion when varying areas are exposed to direct sunlight or to dark space.

Figure 6-1(a) is an end view of SEPS looking in on the payload transport mast side. This component was described in Section 4. It is mounted on a structure that allows it to be hinged inboard for SEPS initial launch and retrieval. Once in space and deployed, the mast housing and support structure remain in place throughout a complete mission cycle.

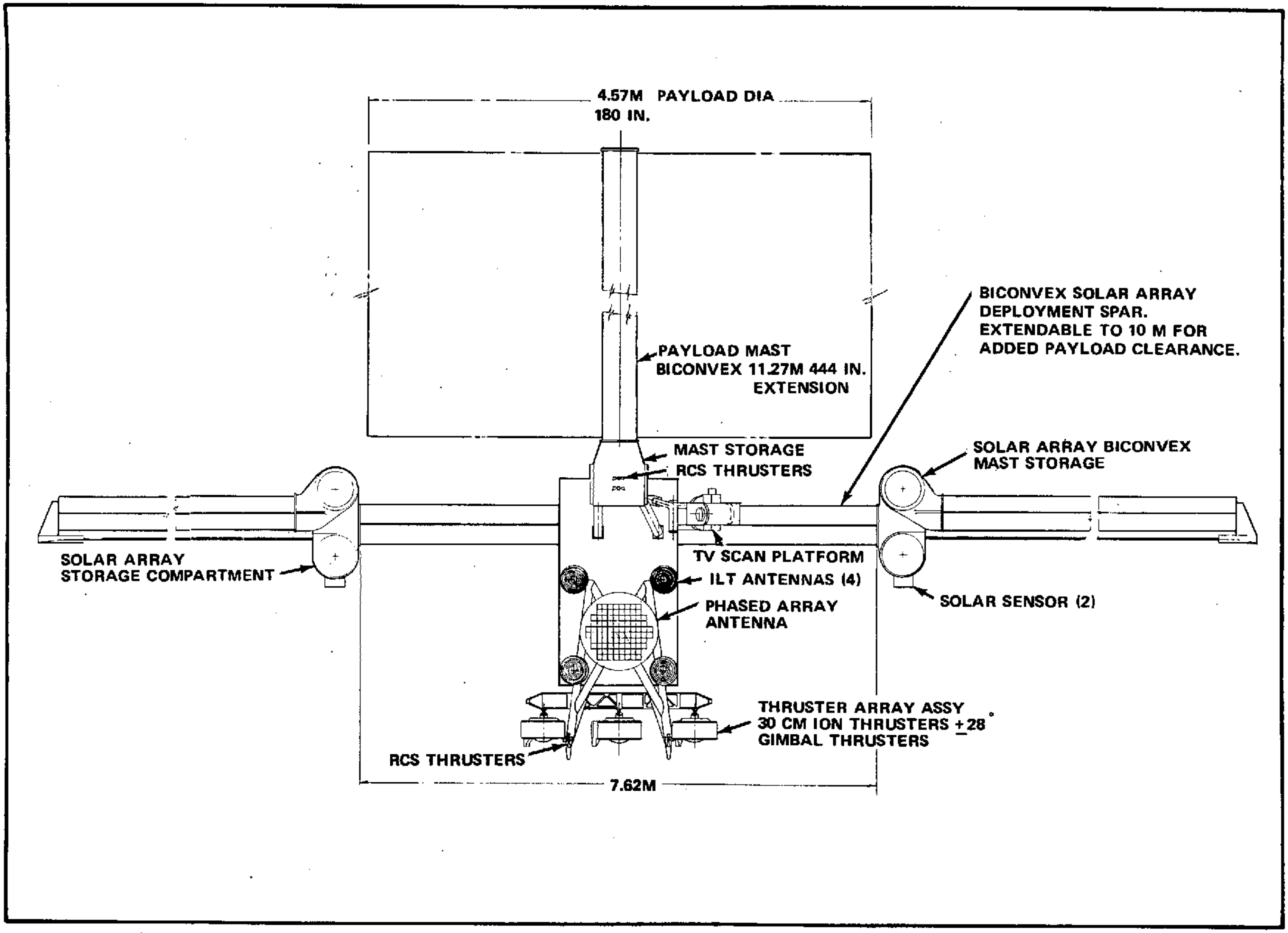

Figure 6-1 (a). RECOMMENDED SEPS CONFIGURATION

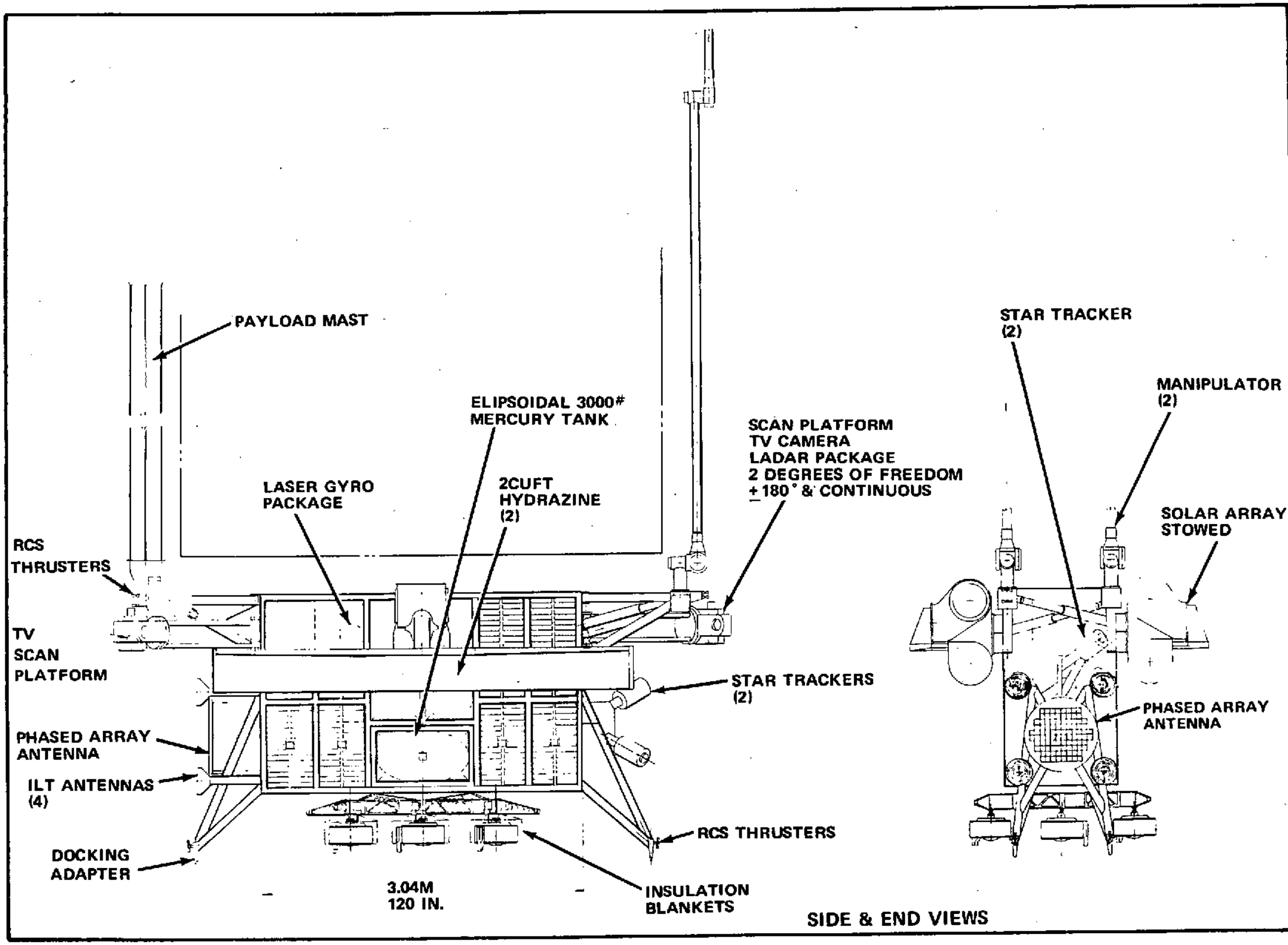

Figure 6-1 (b). RECOMMENDED SEPS CONFIGURATION

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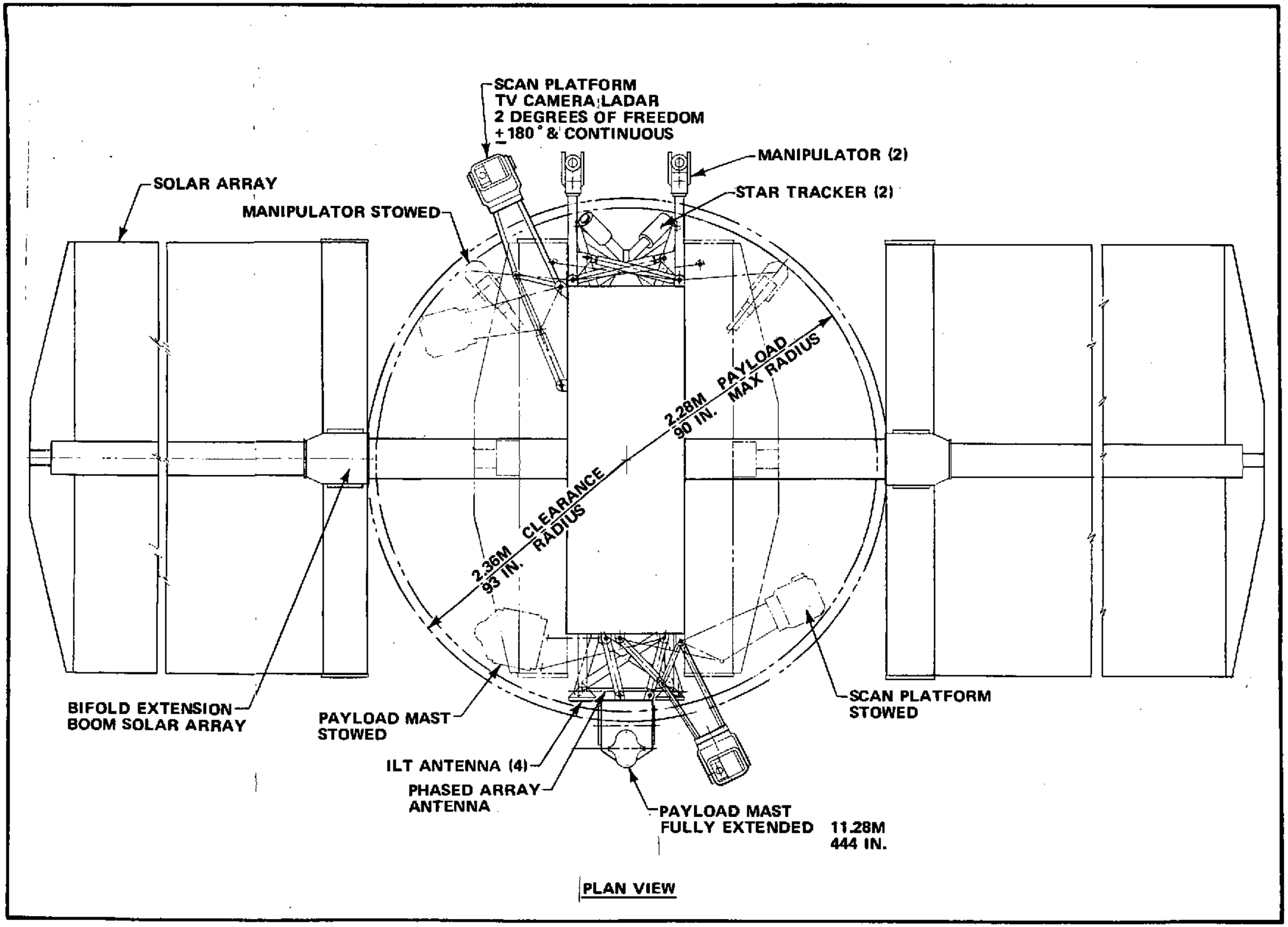

Figure 6-1 (c). RECOMMENDED SEPS CONFIGURATION

Each solar array wing is deployed on two spars. The spars are identical in concept to the transport mast. On Figure 6-1(a), the spar which deploys the solar array wing from the launch position to an inflight position is shown deployed to allow the wings to clear a 4.6 meter (15-foot) diameter payload. The spars can be extended further to clear elements of a payload that require deployment outside the 4.6-meter launch envelope during the final checkout of the payload before SEPS releases it. The housing and extension-retraction drive of the spar is located inside SEPS body and is not visible on Figure 6-1.

The solar array wing assembly, mounted at the outboard end of the deployment spar, is an independent assembly comprised of the rotation mechanism that allows it to be oriented normal to the sunline, the solar blanket storage cylinder, the wiring harnesses and switch assembly, and the biconvex spar solar blanket deployment and retraction mechanism.

Biconvex spars were selected for these assemblies because of their simplicity and their high rigidity in torsion, bending, and compression relative to other storable mast concepts. We assessed them as having the highest potential reliability of any of the mast concepts described in past studies or in published articles that we surveyed. Considering the fact that the blanket spars do not require an EI in the direction parallel to the blanket as high as in the normal direction, these biconvex, edge welded spars were as low in mass as other concepts. SEPS effectiveness is not particularly sensitive to inert mass; it is very sensitive to reliability.

The high gain antenna is a phased-array, and the beam is electronically steered. The phased-array and the Interferometric Landmark Tracker (ILT) are located as far outboard as feasible without requiring mounting on a deployable structure. The inherent redundancy in phased arrays and their lack of moving parts resulted in extremely high reliability.

There are two scan platforms, each mounted on a deployable structure, and located on opposite ends of SEPS. They would normally be used in conjunction but missions can be completed with only one functional platform. This combination of dual scan platforms and dual antenna arrays provides a fail operational and fail acceptable combination for fulfilling earth orbital sorties.

The equipment module mounted above the thruster subsystem's power processors and control electronics is an independent module. The equipment module contains all of the systems' intelligence, housekeeping, and payload support subsystems. The equipment module structure is attached to the thruster subsystem structure such that the two structures after final assembly form an integrated airframe. Figure 6-1(b) shows a side view and a view looking in on the manipulator mounting end of SEPS. The manipulators described earlier are mounted on deployable structures to locate their bases outside the 4.6-meter diameter payload accomodation area. In this end view, the solar arrays are shown in the fully stowed position as they would be for launch.

The star trackers are located as far forward as clearance with the manipulator mount deployment structure permits. The second phased-array antenna is mounted just below the star trackers. Missions can be completed with only one active antenna, but some otherwise unnecessary attitude maneuvers may be required. Figure 6-1(c) is a top view of the EO SEPS configuration.

The submodules of the thruster subsystem power conditioning and control system have no preferred orientations as long as the orientation does not interfere with maintaining their proper thermal environment, test, and maintenance accessibility. The same is true of the thrusters themselves except that their installation pattern must be such that flight control torques are efficiently applied. Many suitable arrangements are possible with little, other than personal preferences, to dictate a choice between them. The best arrangement will be a function of the detail design characteristics of the submodules.

The square 3 by 3 thruster array shown, with each thruster fully gimballed, is as attractive a general purpose array, all things considered, as any other. Insulation around the thrusters and other elements of the structure to which ACS components requiring thermal conditioning are attached, is not shown on the figures. The 3 by 3 thruster array was a Rockwell International concept and a characteristic of the initial study baseline designated by MSFC. NSI invites system planners interested in detail assessments of configuration evolution to review Figure 6-1(a,b,c) thoughtfully for its other merits and faults. It is requested that you contact the study manager and discuss with him suggestions for improvement.

6.3 EXPENDABLES REPLENISHMENT

The value of replenishing SEPS' mercury propellants is obvious from comparison of the propellant mass required to utilize the specification 20,000-hour thruster life (2,900 kg) to the total dry weight of SEPS (1,260 kg). Since many multiple payload packages are in the range around 3,000 kg, carrying a nearly full propellant tank in the first few sorties increases trip time by more than 50 percent. NSI's assessment of the technology is that most SEPS thrusters will have actual lifetimes of 50,000 or more hours if a moderately well-funded thruster technology program were oriented toward guaranteeing it.

Developers of payloads planned for the operational period from 1981 onward expect their satellites to have functional lifetimes of 10 years or more. Several satellites now in orbit have been functional from 6 to 9 years. No item of SEPS is required to function through a large number of cycles. Only 130 payloads are deployed in a total of 29 sorties to accomplish the 10-year long mission model. SEPS performs other servicing and possibly independent space bus missions in addition to the transport sorties, but the total number of cycles for any mechanical device is low in terms of cycle life for modern mechanical devices. Although the program inventory is not planned on the basis of 10-year life expectancy for SEPS, NSI considers it probable that a 10-year operational life could be achieved or exceeded. SEPS #1 may have some early failures as a result of design oversights or due to incorrect information on the design environment of some components, but retrofitted SEPS #2 and successors should achieve life goals.

In view of the simplicity that can be achieved in the propellant storage systems and in methods for their replenishment, it appears highly desirable that the reduced trip time potential and capability for longer stay time on orbit should be exploited by providing for replenishment.

From previous descriptions of the manipulator system and SEPS configuration, SEPS inherent capability for self-replenishment is obvious. The sequence is shown on Figure 6-2. The relatively small amounts of ACS propellant (N_2H_4) and the high density of the mercury propellants result in such small volumes for the replenishment kits that they have frequent opportunities to be carried on IUS-Tug sorties where the payloads are not using all the available cargo space. Thus, flights dedicated solely to SEPS replenishment were never required throughout the entire 1981 to 1991 timeframe encompassed by the reference mission model.

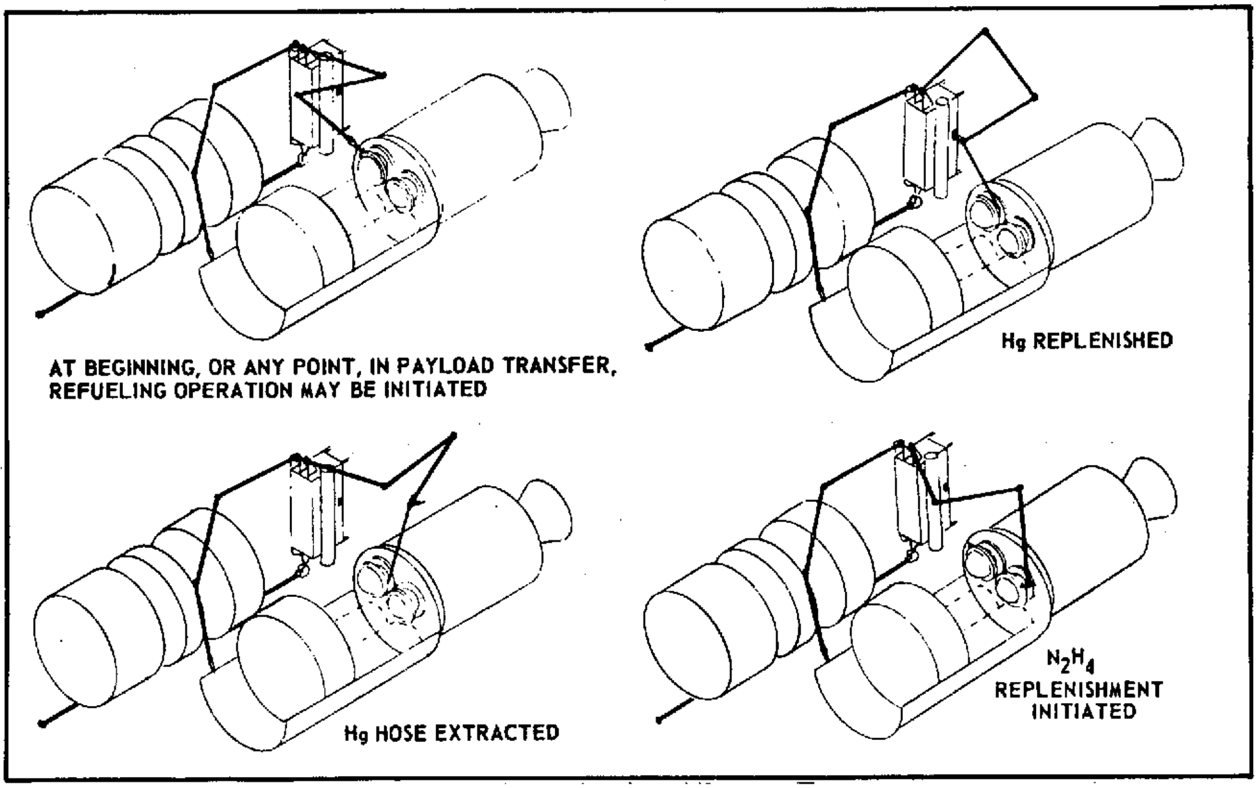

Figure 6-2. REFUELING SEQUENCE

The simplicity of the refueling functions can be envisioned when the reader considers the characteristics of the gas (N_2) pressurized, blow down propellant supply systems. Forcing the replenishment propellants into the tanks automatically compresses the N_2 to its original pressure. The N_2 is not expendable. The tanks have an internal flexible barrier separating propellants and gases. When fully fueled, the barriers are expanded against internal perforated tank bulkheads which prevent the flexible barriers from being overpressurized by the refueling systems. The mercury system operates anywhere in the range from 0.42 kg/cm^2 to 2.1 kg/cm^2 and the ACS system in the range from 3.5 kg/cm^2 to 7 kg/cm^2.

The refueling kits are simple blow down N_2 pressure tanks like the SEPS systems; they refuel with blow down pressures equal to the SEPS fully charged system pressures. The refueling tanks are mounted in bearing rings with the hose storage drums fabricated onto the tanks. Hose tensioning clock springs hold them in the wound tight condition. Figure 6-3 shows the mercury replenishment kit. The N_2H_4 replenishment kit is similar.

For refueling, a manipulator simply grasps the refueling probe at the end of the hose and pulls out the required length of hose to insert the probe into the proper refueling receptable on the SEPS side panel. Flow limiters prevent too rapid refueling of the systems in the initial phase when the pressure differences between supply and SEPS tanks are moderately high. Refueling is complete in about 2 minutes. The probe is retracted from the SEPS panel and released, the refuel kit tensioning spring rewinds the hose on the drum, and the operation is complete. Since the tanks and hose drum rotate together, there are no sliding or rotating liquid or gas seals. The only potential leak point is when the probe slides into the SEPS receptacles. Proper design can make the risk of payload contamination from spillage negligible.

An alternate approach to replenishment is the interchange of a full propellant supply kit for an empty tank in SEPS. This approach is equally effective with the hose refueling technique but was rejected because the manipulator operations required for tank interchanging are more complex than for the hose replenishment system. A single potential leak source (tank's probe into supply line) also exists for this approach. Figure 6-4 shows the component configuration for an interchangeable tank.

6.4 GROUND MAINTENANCE VERSUS SPACE MAINTENANCE

The manipulators with a set of in-space changeable hands or end effectors are extremely versatile payload servicers, payload element deployment assistors, and malfunction repair tools. The broad range of applications of manipulators in automated production and assembly operations and their uses in nuclear reactor core and fuel element recycling attest to the well developed state-of-the-art. SEPS difference is that an RF data link is inserted between hand

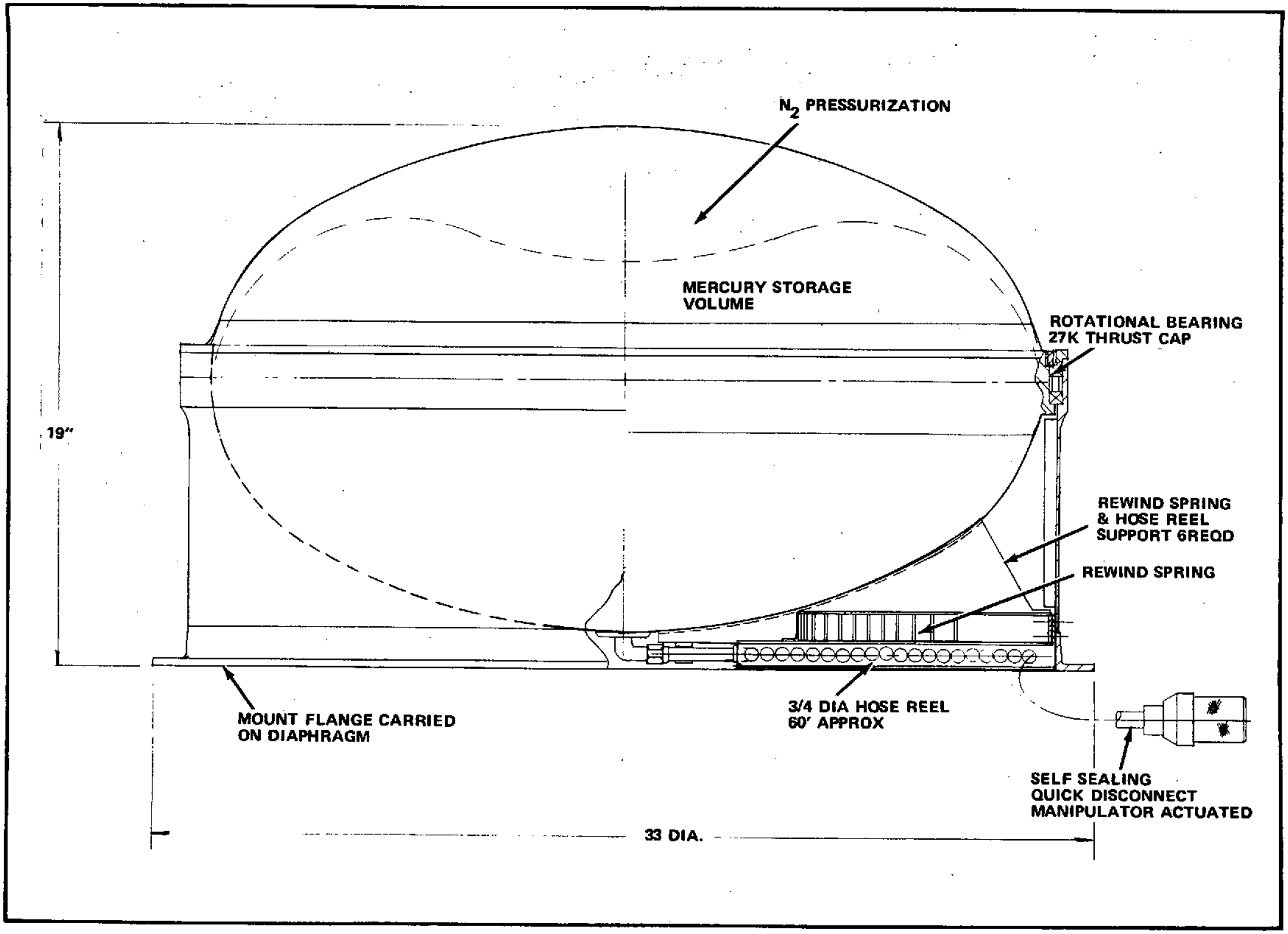

Figure 6-3. MERCURY REPLENISHMENT KIT

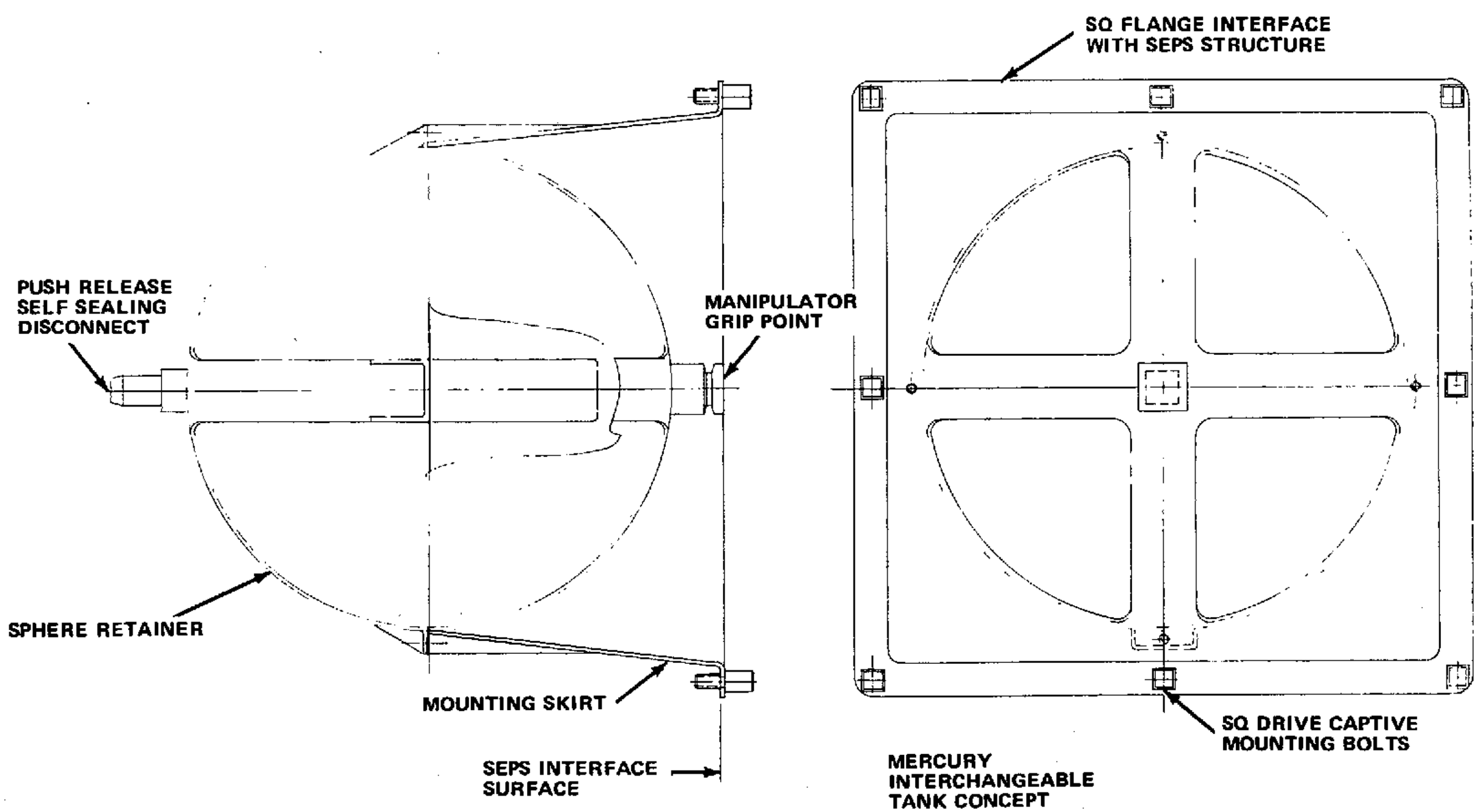

Figure 6-4. ALTERNATE APPROACH TO SEPS PROPELLANT REPLENISHMENT

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controllers, computers, and TV cameras that are hard wired to the operators' console in the industrial operations mentioned above. SEPS self-maintenance is certainly feasible.

NSI does not believe that the high reliability and long service life expectancy of properly designed SEPS subsystems warrant design for in-space maintenance in a spacecraft that can be retrieved and returned to earth for repair. If further analysis indicates in-space maintenance to be desirable, SEPS physical and functional characteristics are such that it has the inherent potential to be an "Erector Set" type spacecraft. Various subsystems can be attached to a core structure. Figure 6-5, a modification of some NASA technology program designs, illustrates this. Specific design for in-space maintenance, if it were an initial program requirement, should not be expected to increase total program cost and could actually reduce DDT&E program cost if program management exploited the resultant characteristics of the system in a diligent effort to reduce the cost of development, integrated systems life tests, and flight readiness tests. Design for in-space replacement of selected modules or equipment assemblies may be found desirable as detailed flight systems development programs are initiated.

Without further discussion, Figures 6-6, 6-7, and 6-8 are presented so that the program concept assessor, with a little imaginative consideration of design detail offered by present technology, can envision the flexibility of the manipulators for many types of functions: space experiment interchange on laboratory type spacecraft, spacecraft servicing, repair of other spacecraft, and replacement of SEPS components if such design approach should later prove warranted.

6.5 CHOICE OF POWER LEVEL FOR SEPS

The next most significant configuration definition choice is associated with SEPS power level. The decision becomes largely a matter of judgment since no clear mission requirement sets a definite minimum power level in the range of practical choices, and no technology factor or cost factor produces a sharp step in development difficulty or cost as power increases.

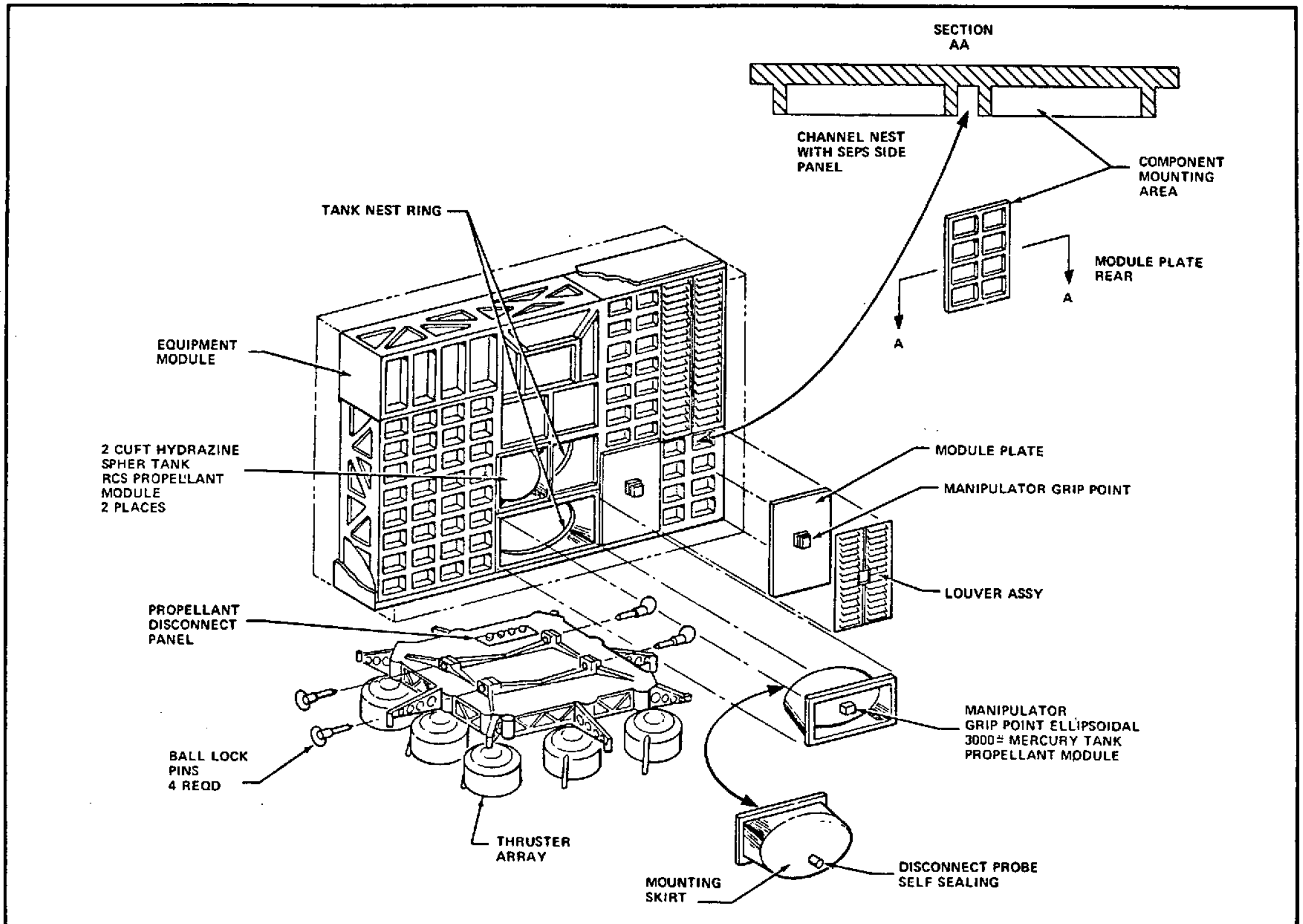

Figure 6-5. SEPS POTENTIAL FOR IN-SPACE MAINTENANCE

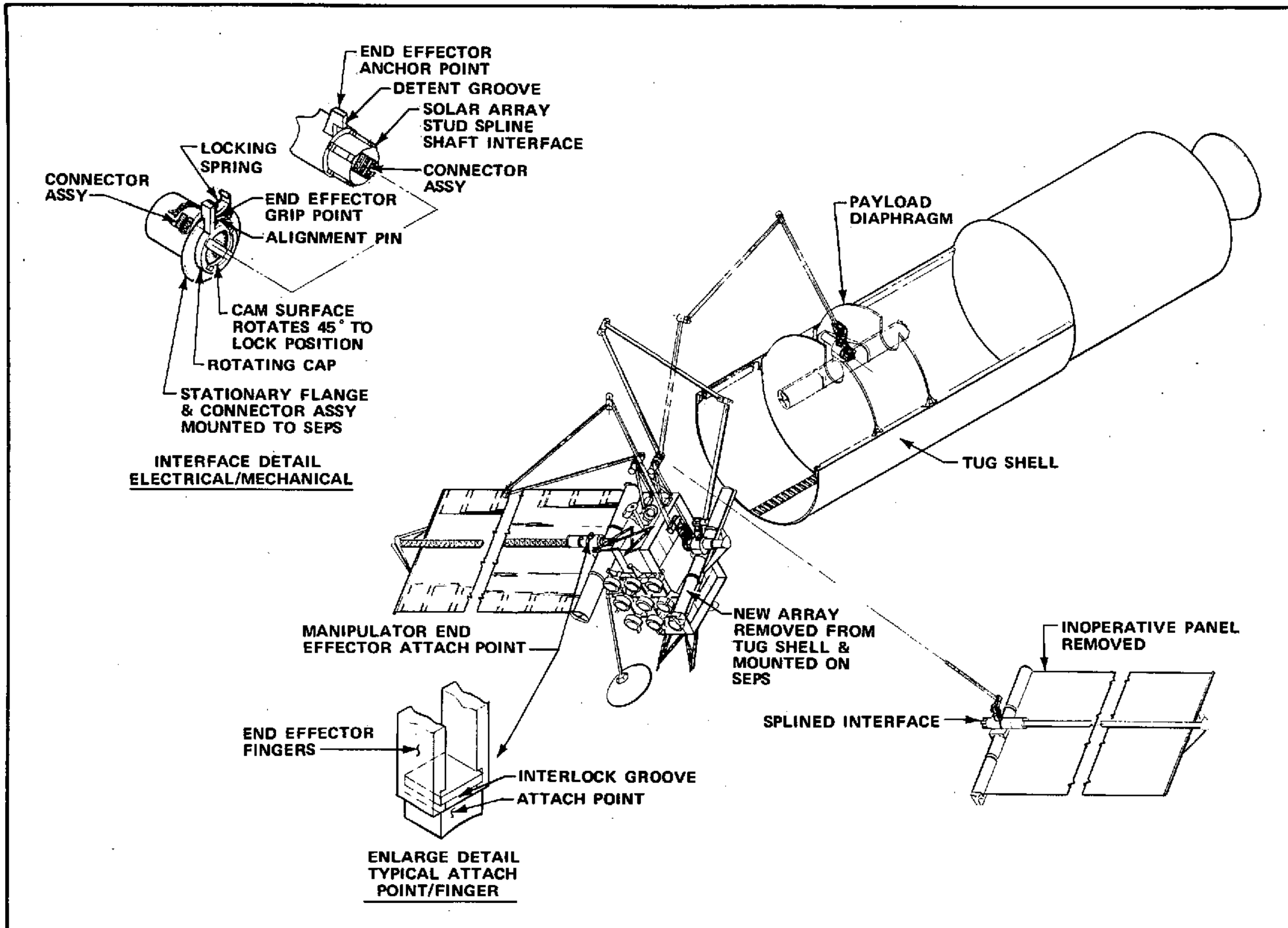

Figure 6-6. POTENTIAL IN-SPACE REPLACEMENT OF SOLAR ARRAY

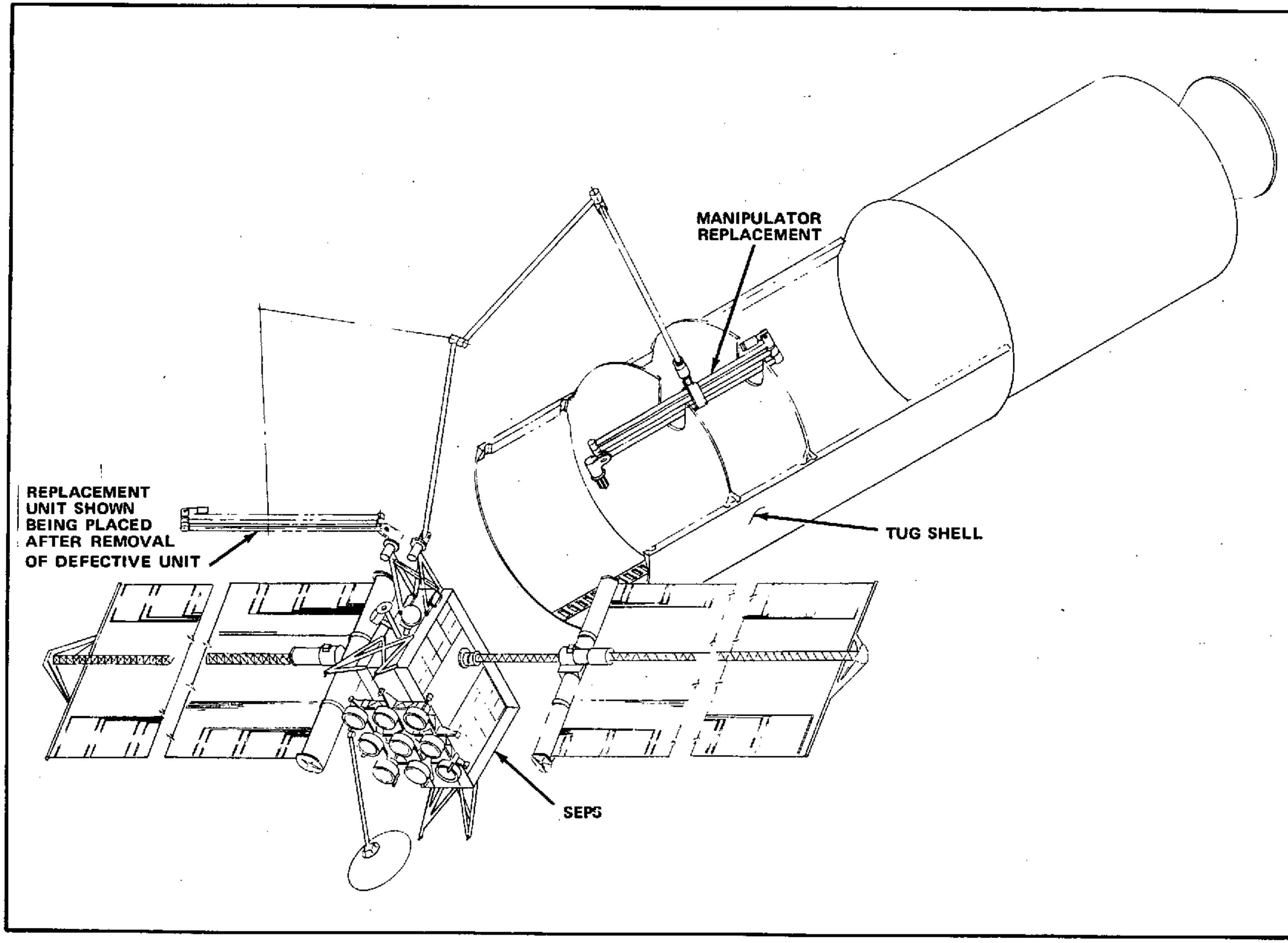

Figure 6-7. MANIPULATOR POTENTIAL FOR EXCHANGE FROM TUG SHELL TO SEPS

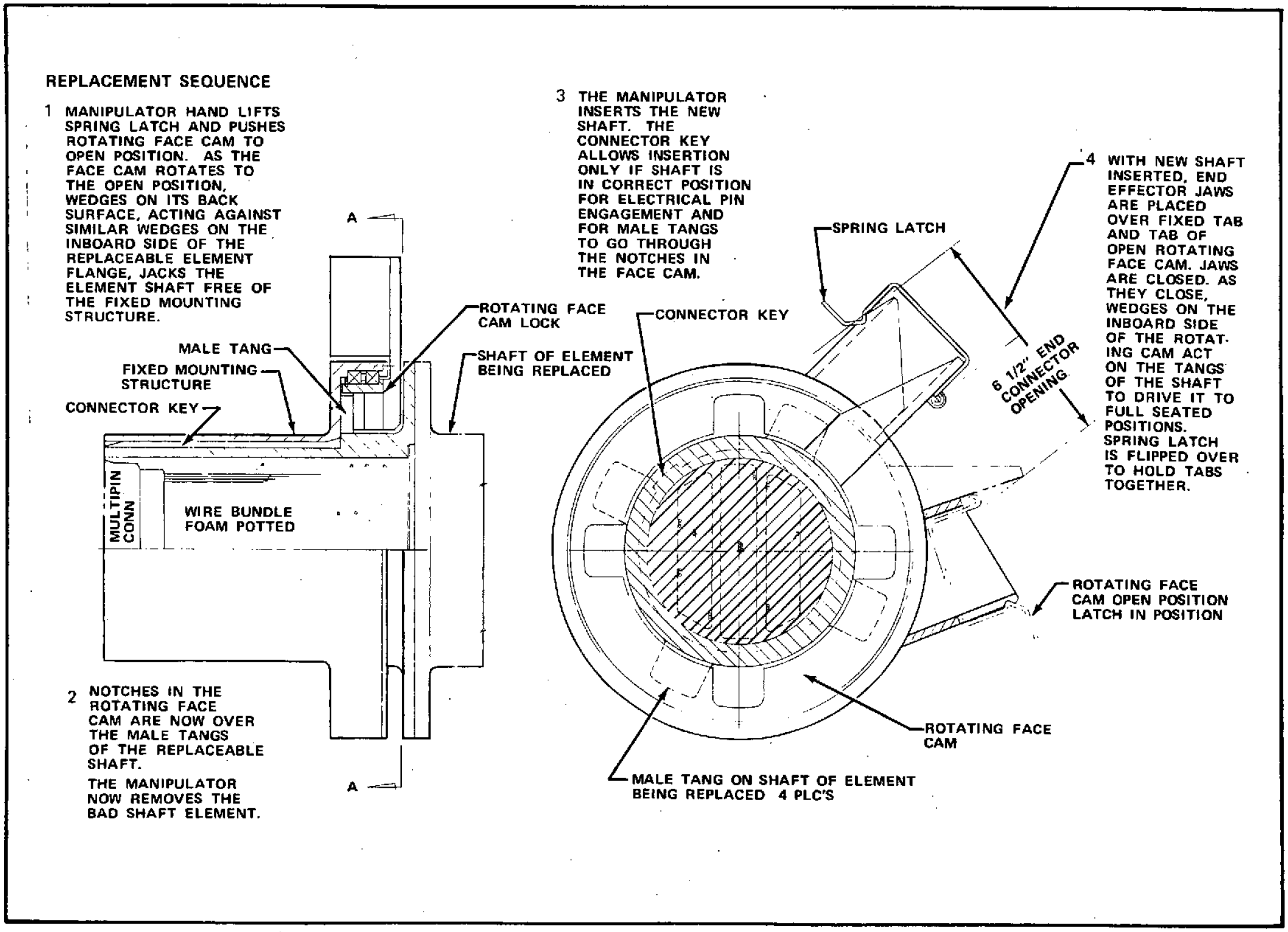

Figure 6-8. TYPICAL SOLAR ARRAY/MANIPULATOR REPLACEABLE ELEMENT

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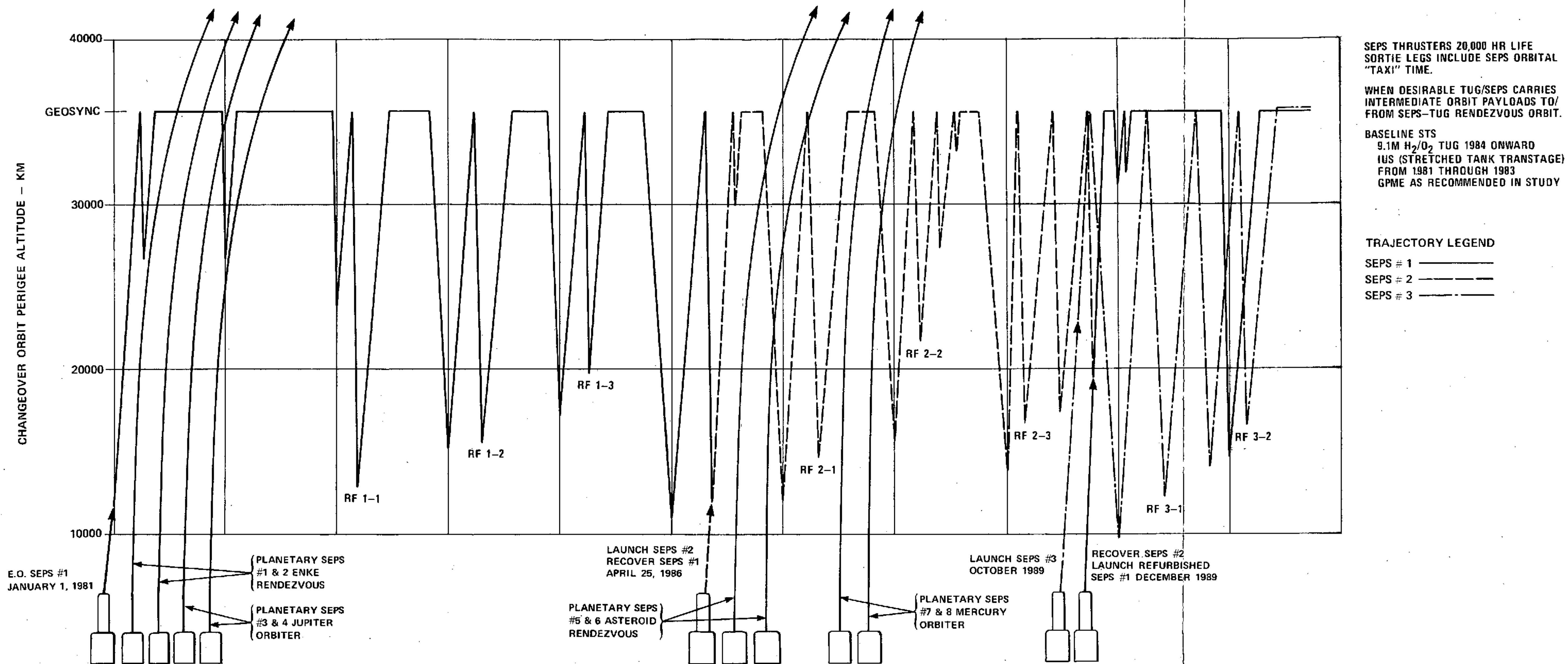
The transport capability and operational flexibility of SEPS with the STS is almost directly proportional to the power level. To demonstrate this, NSI developed complete Systems Operational Profiles for accomplishing the reference mission model. The 25 kw NASA baseline profile is shown on Figure 6-9. A profile for a 50 kw SEPS is shown on Figure 6-10. Figure 6-11 shows the sortie trip times required by a 25 kw SEPS to accomplish delivery and retrieval missions in conjunction with a 9.1-meter H_2O_2 high performance Tug. The solid curves are the theoretical times required for SEPS to complete a mission with the maximum payloads Tug could bring to the SEPS/Tug rendezvous orbit for the Tug one-way velocity increments shown by the abcissa.

The cross-hatched areas indicate the range of Tug velocity increments actually required to accomplish the mission model. The black dots are individual sortie trip times calculated with radiation degradation effects. Figure 6-12 shows the sortie trip time savings of a 50 kw SEPS relative to the 25 kw SEPS. The system operational profile, as illustrated on Figure 6-9, does not utilize the full capability of a 25 kw SEPS until 1989 and does not require two SEPS in orbit until 1990. Therefore, use of a 50 kw SEPS saves only two more shuttle flights than a 25 kw SEPS. The advantage of increased power for earth orbital operations with the reference mission model is therefore due only to:

- Reduction of the time required for execution of individual sorties
- The speed with which SEPS could respond to unplanned revisions of flight schedules
- Quick response to special demands for maintenance or retreival of a malfunctioning satellite.

Conversely, the DDT&E cost to develop a 50 kw SEPS was estimated by NSI to be only 7.5 percent greater than for a 25 kw SEPS so that a very small additional investment produced a transport vehicle of nearly twice the inherent capability. Figure 6-13 shows a size comparison between a 50 kw and a 25 kw power level SEPS. Table 6-1 shows a summary of DDT&E cost breakdown with the incremental cost for development of the 50 kw system. Note that the cost increase is essentially all in propulsion areas. The majority of that cost is due to the present high cost of solar cells which can be drastically reduced with a technology program aimed at production cost reduction for both the solar cells and their assembly into arrays.



												PROGRAM TOTALS
SHUTTLE	19	22	28	27	33	32	30	32	30	30	32	315
SHUTTLE/TUG	6	2	8	11	13	12	11	4	6	9	10	92
SHUTTLE/TUG/ EO SEPS	2	1	2	2	2	3	2	4	5	5	2	30
YEAR	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	437

Figure 6-9. SYSTEM OPERATIONAL PROFILE (9.1-METER BASELINE TUG + 25 KW SEPS WITH 20,000 HR THRUSTER LIFE - REFUELABLE)

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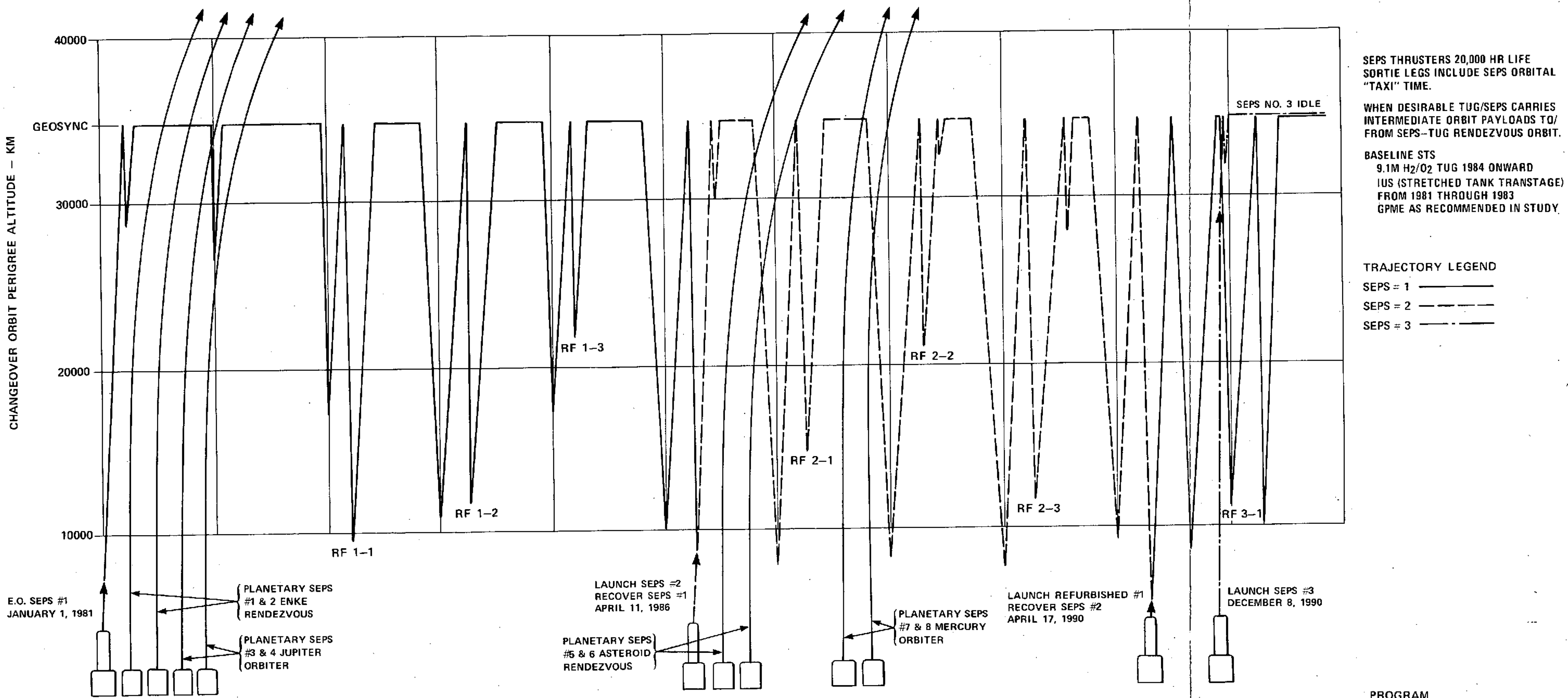

												PROGRAM TOTALS
SHUTTLES	19	22	28	27	33	32	30	32	30	30	32	315
SHUTTLE/TUG	5	2	8	11	13	12	10	5	7	10	10	93
SHUTTLE/TUG/EO SEPS	2	1	2	2	2	3	2	3	3	5	2	27
YEAR	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	435

Figure 6-10. SYSTEM OPERATIONAL PROFILE (9.1-METER BASELINE TUG + 50 KW SEPS WITH 20,000 HR THRUSTER LIFE - REFUELABLE)

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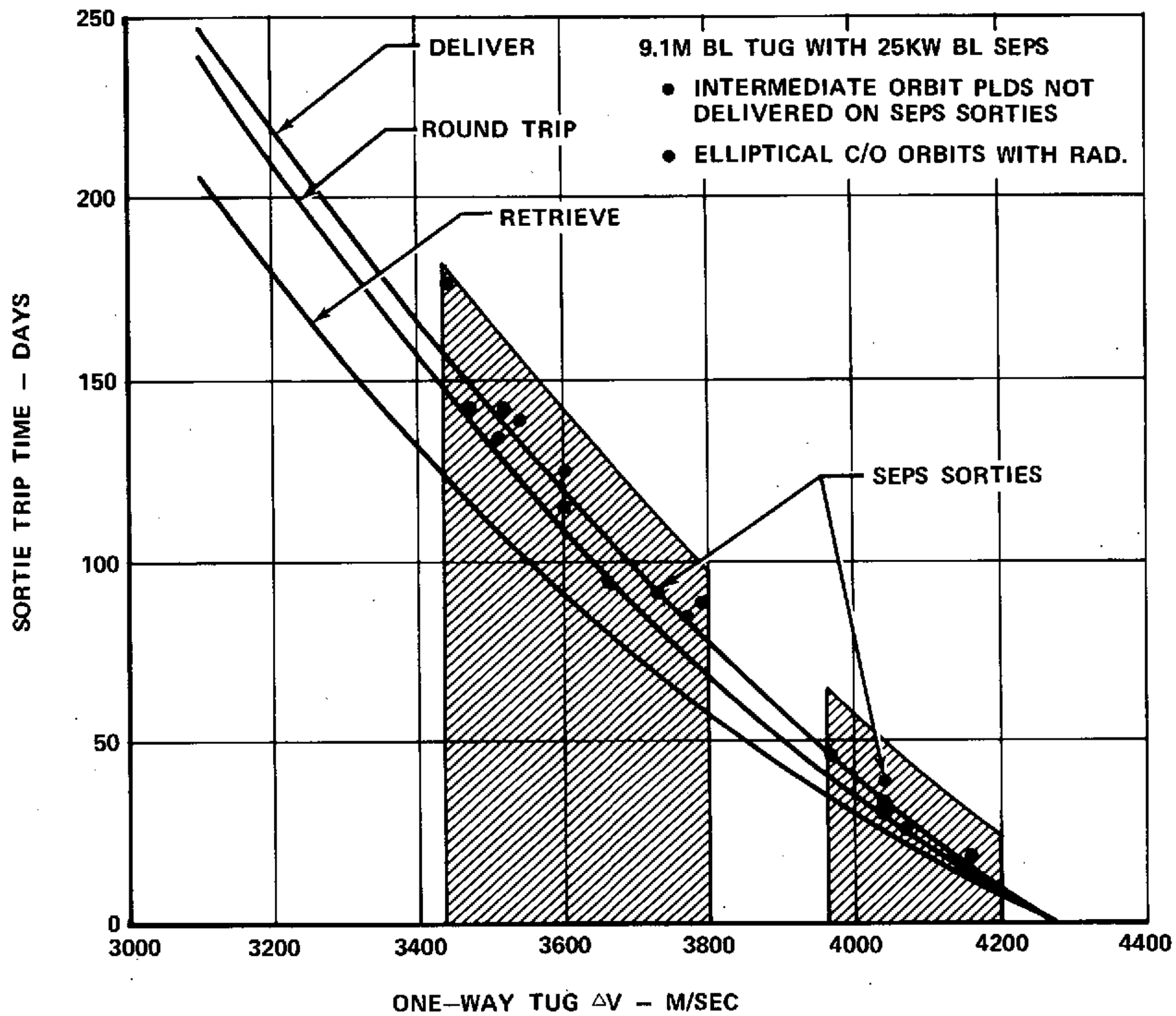

Figure 6-11. SORTIE TRIP TIMES REQUIRED BY 25 KW SEPS TO ACCOMPLISH DELIVERY AND RETRIEVAL MISSIONS IN CONJUNCTION WITH A 9.1-METER H_2O_2 HIGH PERFORMANCE TUG

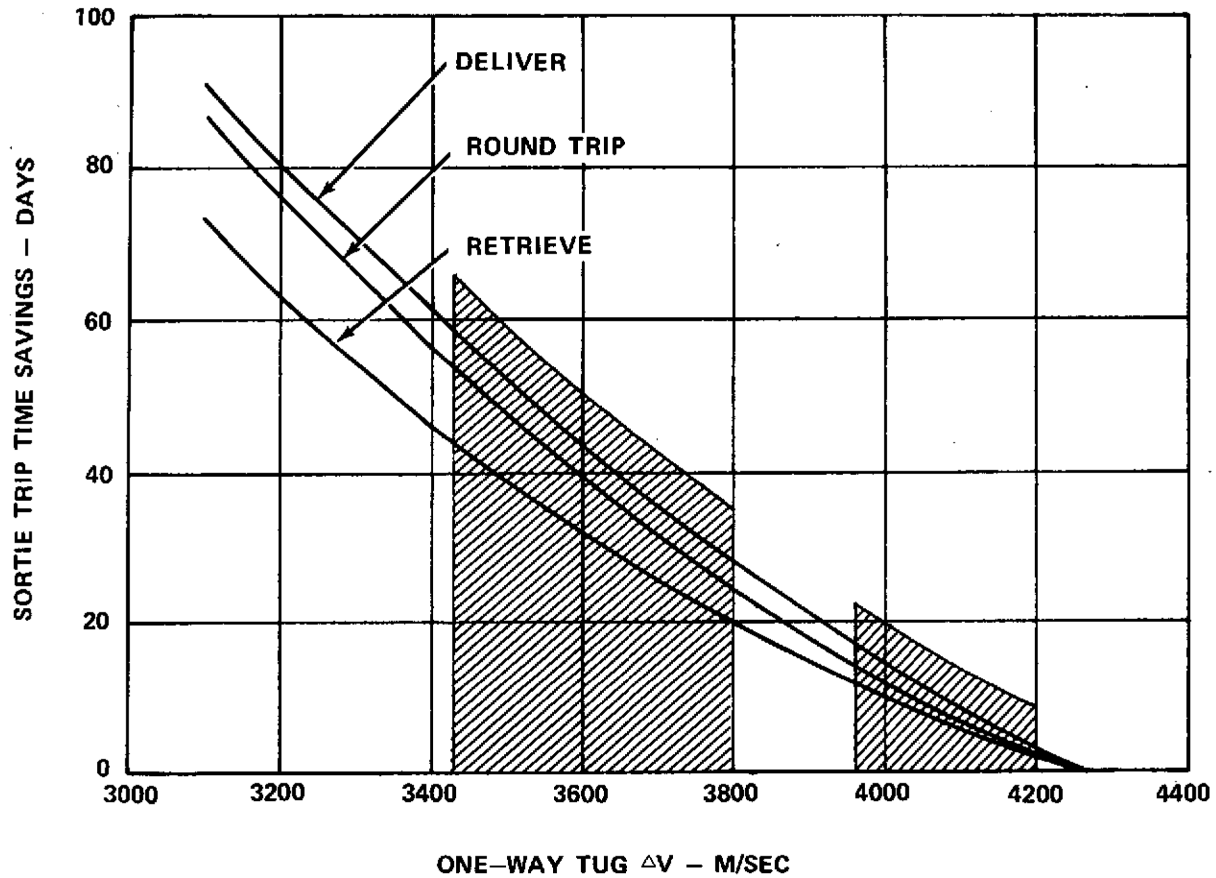

Figure 6-12. SORTIE TRIP TIME SAVINGS OF 50 KW SEPS RELATIVE TO 25 KW SEPS

Table 6-1. COMPARISON OF 25kw TO 50kw BASIC COSTS (SEPS DEVELOPMENT AND 1ST UNIT COSTS)

(Dollars in Millions)

COST ELEMENT	DEVELOPMENT		FIRST UNIT COST	
	25 kw	Δ FOR 50 kw	25 kw	Δ FOR 50 kw
STRUCTURES & THERMAL CONTROL	$ 4.8		$ 1.2	0.1
PROPULSION	9.1		2.0	0.8
POWER DISTRIBUTION	1.0		0.4	
SOLAR ARRAY	7.8		5.8	6.1
DATA MANAGEMENT	3.4		1.0	
COMMUNICATION	2.2		1.2	
ATTITUDE CONTROL/N&G	9.2		2.0	0.2
INTEGRATION & TEST CHECKOUT	6.7	1.0	1.1	1.0
TEST HARDWARE	21.3	6.5		
GSE	5.0			
SOFTWARE	4.5			
LOGISTICS	0.5			
SE&I	6.8		1.4	
PROGRAM MANAGEMENT	6.9		1.4	
BASIC SEPS	$89.2	Δ7.5	$17.5	Δ8.2
Δ FOR EARTH ORBITAL FUNCTIONS	8.3		1.0	
	97.5		18.5	
Δ FOR TUG PAYLOAD SHELL AND DIAPHRAGMS	2.5		0.8	
	$100.0	Δ% 7.5	$ 19.3	Δ% 42

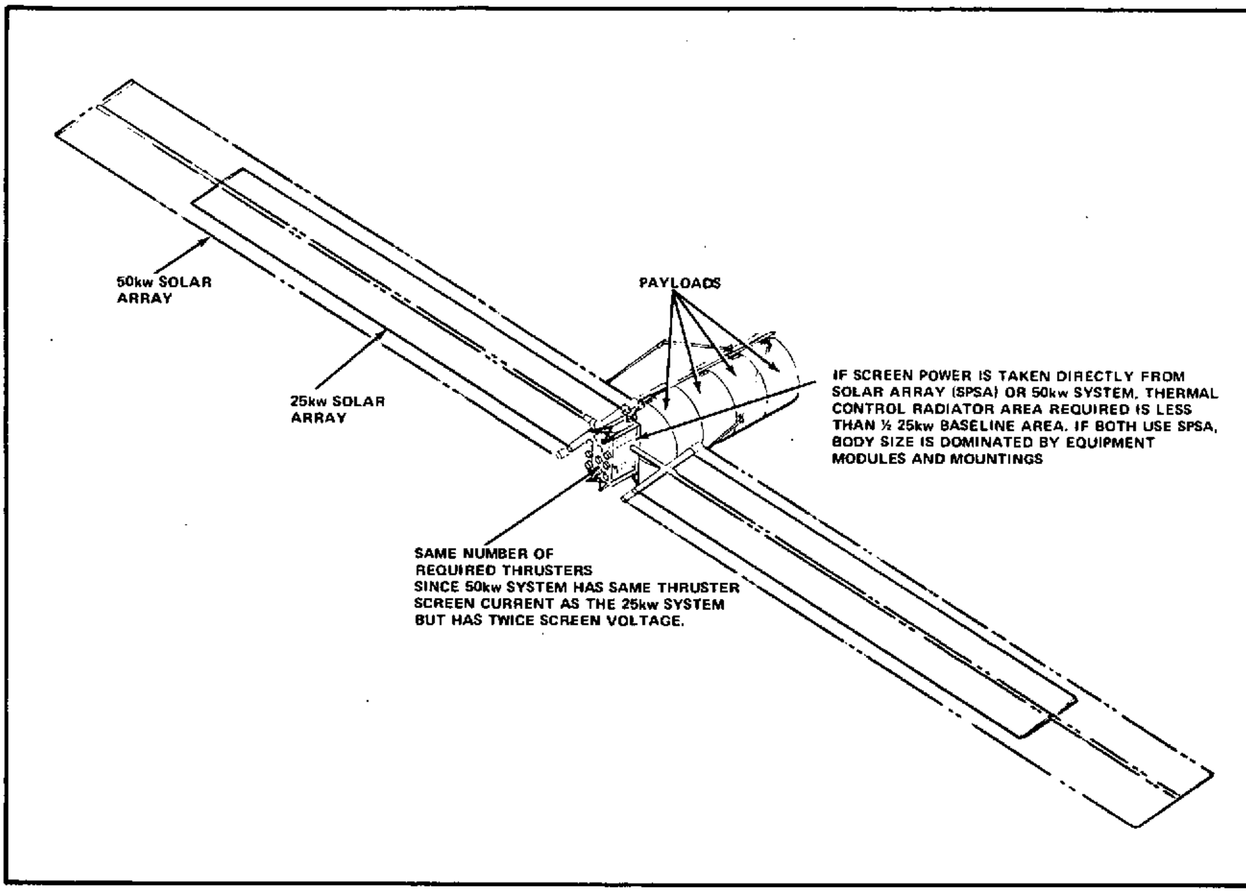

Figure 6-13. SIZE COMPARISON BETWEEN 50 KW AND 25 KW POWER LEVEL SEPS

For the planetary missions the rate of gain in usable net scientific payload as power level increases varies considerably with the mission. In addition, the gains are sensitive to the mass-to-power ratio so that design approaches for SEPS thruster subsystem that result in high mass-to-beam power ratios, or unjustifiably conservative mass estimates, will cause apparent "optimum" power levels to be considerably lower than the true optimums. Even on the most conservative basis for mass-to-power ratio, such as used in the Rockwell International 1972 and 1973 studies, trends for continuing growth in available net payload are indicated as power levels extend beyond 25 kw.

The planetary science packages conceived for most of these missions do not indicate the need for the higher payloads associated with the higher powers

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desirable for a SEPS operating in earth orbit. It is the opinion of this author, at least, that the planned science packages are rather minimal and that a great deal more useful information would be obtained if the available payload mass allowed by the higher powered SEPS were used to fly some modification of the higher resolution, versatile sensors and instruments contained in proposed satellites such as the Synchronous Earth Observing Satellite (SEOS) and other environment determination and monitoring satellites.

Figure 6-14 presents a review of typical planetary missions from earlier SEPS work by Rockwell International. The curves show parametrically the influence of trip time and power level. The ordinates labeled "Approach Net Mass" are all masses (SEPS nonpropulsive plus gross payload) in addition to the mass of the solar arrays and the thruster subsystem. If a standard core SEPS were used as the spacecraft bus, the gross payload would be approximately net mass minus 500 kilograms. For the Jupiter Orbiter the payload must include the chemical retrorockets for a capture maneuver into a highly elliptical Jovian orbit.

The four sets of mission charts demonstrate two salient features. In all cases, increased power increases payload. For the missions beyond 4 AU, SEPS can provide only limited payload support power if developed at the 25 kw of solar power level.

In the case of the Jupiter Orbiter mission, increased power beyond 25 kw would allow SEPS thrusters to operate during the approach to Jupiter, aiding in the capture maneuver, and also allow SEPS to modify the Jovian orbit for close inspection of each Jovian moon. When not thrusting, more power is available for communications so that high resolution imaging can be conducted in shorter periods of time. All of Rockwell International's work presented on Figure 6-14 was conducted with very conservative mass-to-power ratios based on processing screen power with the associated losses and weight penalties. The Jupiter missions, which chemically retro SEPS into the capture orbit, will benefit greatly from improved (lower) mass-to-power ratios.

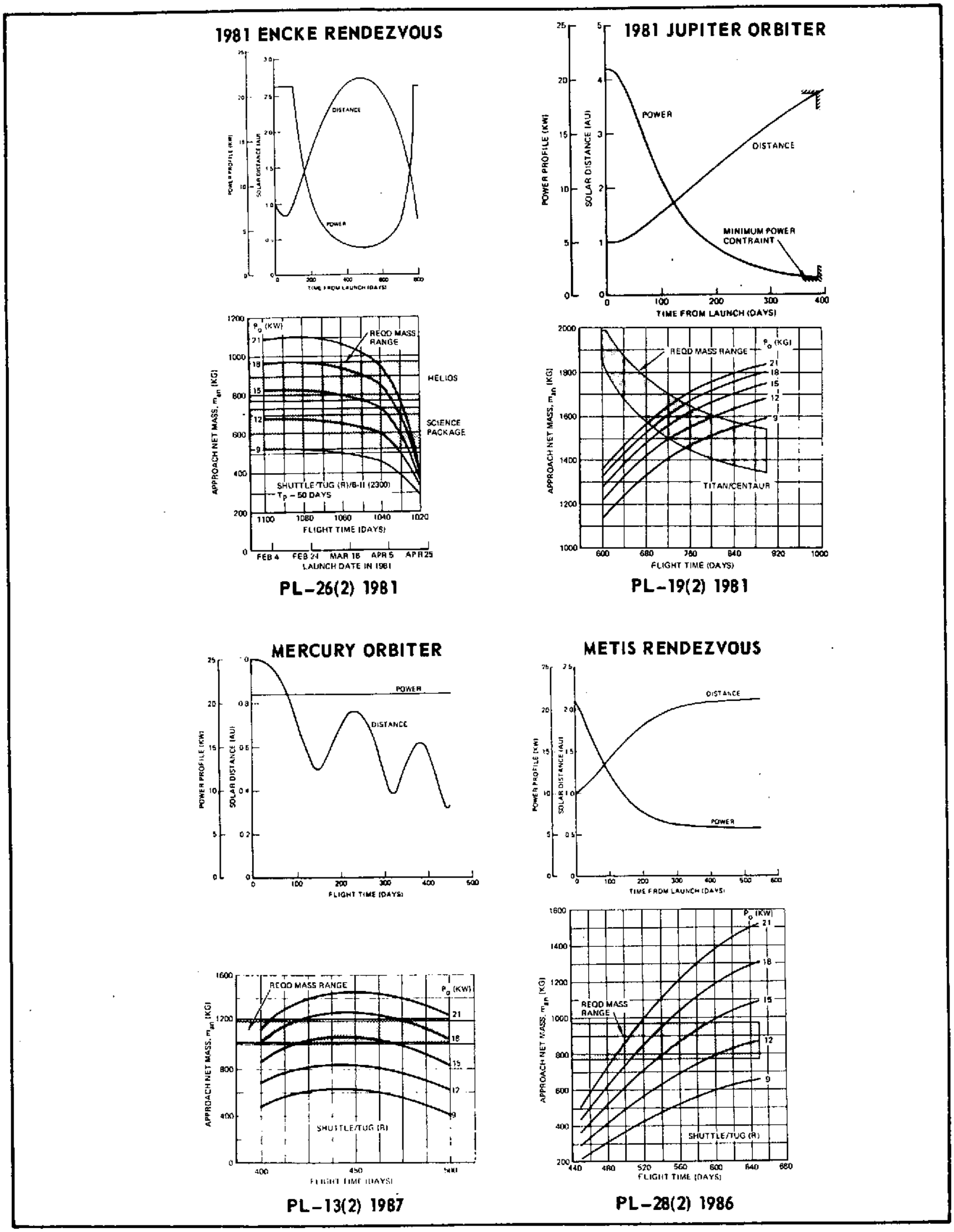

Figure 6-14. REVIEW OF TYPICAL PLANETARY MISSIONS FOR SEPS

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Figure 6-15 shows NSI's analyses of SEPS potential for an exciting new set of "out-of-the-ecliptic" missions that allow examination of the solar magnetosphere and solar surface with high resolution instruments over the entire solar sphere. In the particular example shown, the SEPS is launched by a Titan Centaur vehicle. The curves demonstrate the effect of three parameters. The curve showing the higher heliographic inclination versus mission time illustrates the advantages of increased power, better power-to-mass ratio by taking thruster screen power directly from the solar arrays, and the value of the option of operating at a factor of 2 greater (2200 Vs/1100 Vs) thruster screen voltage to achieve an Isp of 4243 seconds rather than a baseline 3,000 seconds. The higher achievable inclination for the upper curve is due solely to the higher Isp and lower mass-to-power ratio from direct use of solar array power for screen power.

ADVANTAGES OF HIGHER POWER & Isp

	25 KW BASELINE	50 KW $\sqrt{2}$ BL Isp
SEPS		
INERT MASS	1373 KG	1713 KG
PROPELLANT	2289 KG	2240 KG
BURN TIME	18955 HOURS	15534 HOURS
COAST TIME	7324 HOURS	10746 HOURS
CENTAUR		
INERT MASS	1859 KG	1859 KG
PROPELLANT	7722 KG	7437 KG
BURN TIME	258.90 SEC	249.15 SEC

NOTE: SUN'S EQUATORIAL PLANE INCLINED 7 DEGREES TO ECLIPTIC AND ORIENTED SO AS TO ADD DIRECTLY TO INCLINATION WITH RESPECT TO THE ECLIPTIC PLANE.

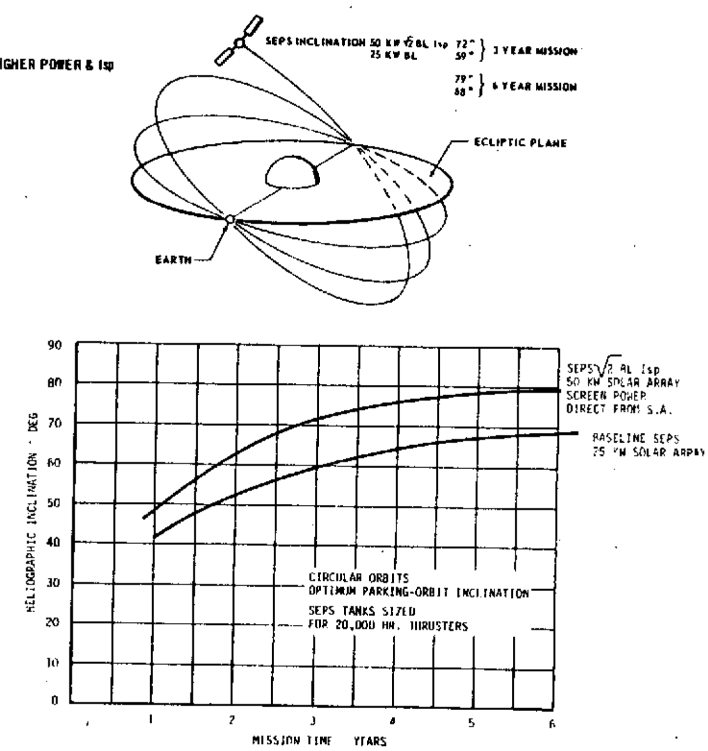


Figure 6-15. "OUT-OF-THE-ECLIPTIC" MISSIONS FOR SEPS

A design approach similar to that used on the 50 kw system but at a 25 kw level would finally achieve the 80-degree inclination but in a much longer trip time.

This discussion has not covered all the implications of Figures 6-14 and 6-15. Thoughtful perusal of these figures will indicate that desirable characteristics for a standard core SEPS to achieve enhanced planetary mission suitability are:

- Improved average thrust-to-mass ratios
- Option to operate at high or low Isp to match requirements of a specific mission phase
- Reserve power to support larger payloads and higher communications rates at extended distances from the sun
- Maneuver power to extend scientific mission capabilities after arrival at the target planet.

Improved average thrust-to-mass ratio can be achieved by:

- Increased solar array area and higher kw/kg values for the arrays by fuller exploitation of present technology
- Taking thruster screen power directly from the solar arrays and improving power processor efficiency for the remaining ≈20 percent of the power
- Fuller utilization of the ion thruster's inherent capabilities indicated by the last several years of NASA's technology program.

6.6 RELATED TECHNOLOGY ASSESSMENTS

NSI has reviewed the available technology base derived from NASA's thruster technology and research programs, has reviewed industrial developments of devices suitable for solid state power processing and has reviewed the literature on solar cell technology. The conclusions of this assessment are:

- Thrusters have the inherent ability to operate over screen voltage ranges of about 800 v to more than 2800 v and at beam currents corresponding to 0.5 amp to 4 amps in a 30-centimeter thruster
- Solar arrays are both feasible and desirable direct sources of thruster beam power
- Higher voltage solar arrays (1200 v to 2400 v) are both feasible and desirable
- The potential exists for much lower cost, higher reliability, and higher efficiency solar arrays than those assumed in prior studies
- Higher input voltage power processors than those baselined for prior studies (200 v to 400 v) are feasible
- Exploitation of the technology base will provide a SEPS of significantly greater mission flexibility than the baseline derived from previous studies.

In support of the thruster conclusions, Figure 6-16 shows operating characteristics of 30-centimeter thrusters in NASA technology program tests compared to the baseline specification for thruster performance.

6.7 THRUSTER SCREEN POWER DIRECTLY FROM SOLAR ARRAYS WITH SELECTABLE Isp

This subsection presents NSI's rationale for recommending the use of thruster screen power taken directly from the solar arrays. Detail designs of the alternate approaches are beyond the scope of this study due to the funding level of $130,000 and the broad coverage of the system and its operation required by the work statement. NSI reviewed the basic physics and characteristic phenomena associated with the functioning of both the thruster and the solar array. The factors involved in the engineering design and operation of the stage with thruster screen power taken directly from the solar arrays were assessed. The assessment showed that several strong factors motivated the direct screen power approach and only relatively weak considerations were against it.

6.7.1 Thruster Functional Characteristics

A proper assessment of the pros and cons of screen power supply alternates depends upon an understanding of the thruster's operation and control. An exhaustive definition of thruster functioning is not necessary. The reviewer with command of a little basic physics can establish the details to the extent he desires by analysis and extrapolation of the characteristics of the thruster depicted on Figure 6-16. Voltages indicated are for operation at baseline nominal condition (3,000 sec Isp).

The significant physical factors are:

1. The screen power is approximately 75 to 85 percent of total power supplied to the thruster depending upon the screen voltage (Vs) level selected. Efficiency increases significantly as screen voltage increases; this is illustrated on Figure 6-17. The screen power is used to pump electrons out of the thruster's internal enclosure (the perforated screen grid is the aft closure of this volume) to the neutralizer so that the internal mixture of Hg vapor, electrons and Hg^+ ions are maintained at the positive voltage level, Vs, above

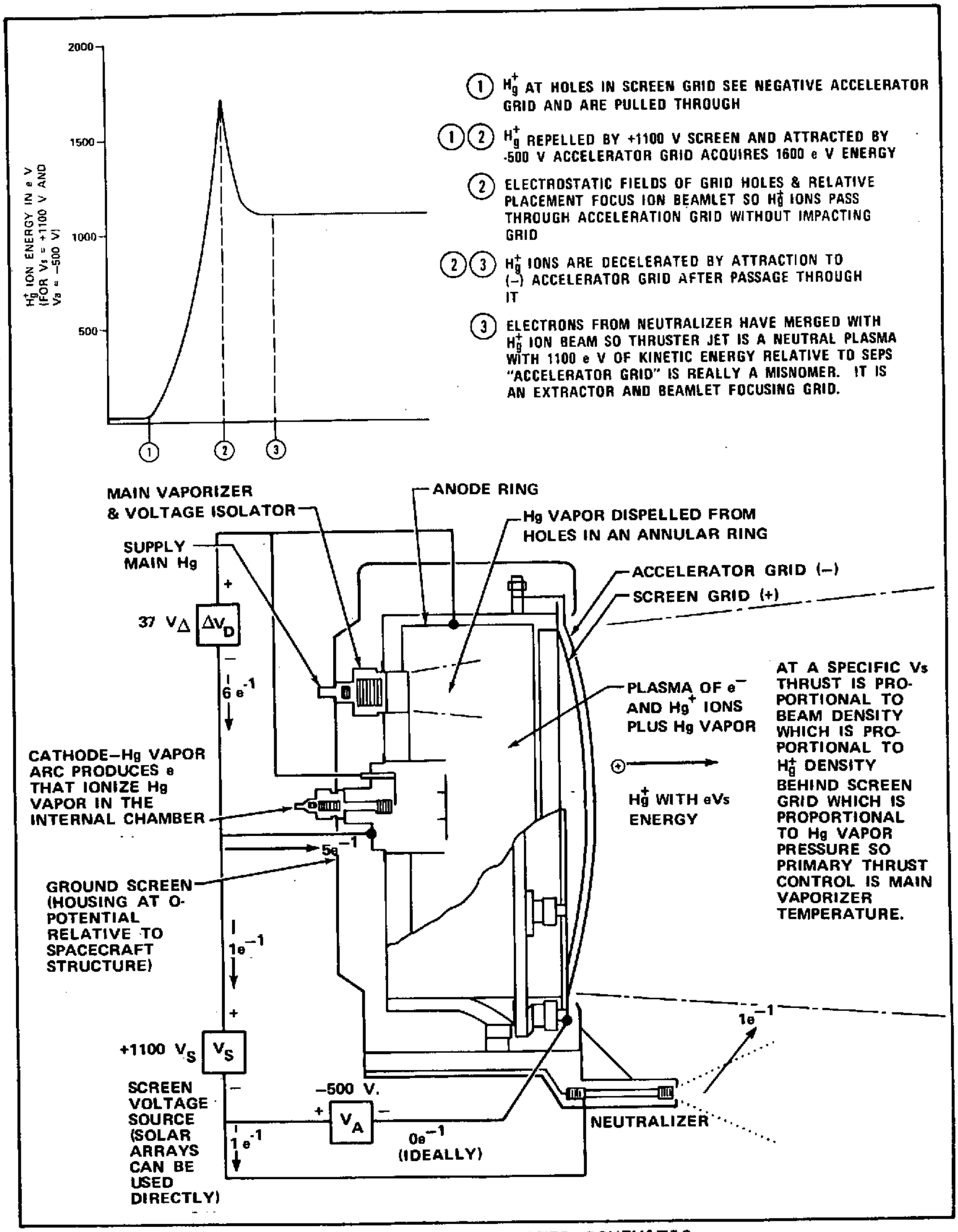
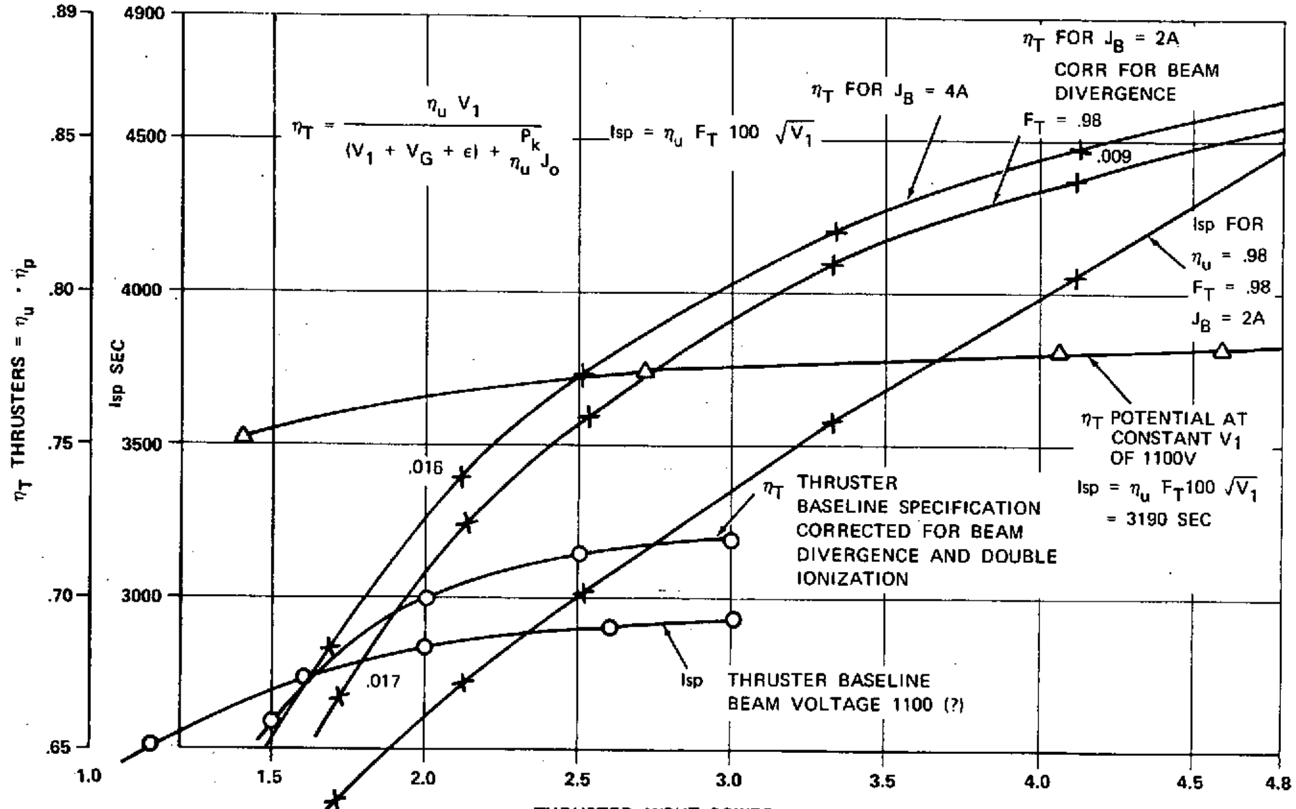


Figure 6-16. SEPS THRUSTER SCHEMATIC



SOME SOURCE REFERENCES:

1 – PERFORMANCE OF 30 cm ION THRUSTERS WITH DISHED ACCELERATOR GRIDS V. K. RAWLIN NASA LEWIS AIAA PAPER 73-1053
2 – MEASUREMENT OF BEAM DIVERGENCE OF 30 cm DISHED GRIDS, R. L. DONILOWICZ, V.K. RAWLIN, B. A. BANKS, E.G. WINTUCKY NASA LEWIS AIAA PAPER 73-1051
3 – STUDIES OF DISHED ACCELERATOR GRIDS FOR 30 cm ION THRUSTERS, V.K. RAWLIN NASA LEWIS AIAA PAPER 73-1086
4 – A 30 cm DIAMETER BOMBARDMENT THRUSTER WITH A VARIABLE MAGNETIC BAFFLE, R. T. BECHTEL NASA LEWIS AIAA PAPER 72-489

DEFINITION OF TERMS:

V_1 = NET ACCELERATING VOLTS
$\eta_u J_o = J_B$ = BEAM CURRENT
P_k = THRUSTER FIXED POWER LOSSES
η_u = PROPELLANT UTILIZATION EFF.
ϵ = DISCHARGE LOSS ev/ION
V_G = NEUTRALIZER TO BEAM VOLTAGE

Figure 6-17. ION PROPULSION DEMONSTRATED POTENTIAL COMPARED TO SEPS BASELINE FOR STUDY

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the thruster outer housing potential which is also the stage potential. The current in this circuit results from the rate at which Hg^+ ions are extracted through the screen grid from the thruster internal enclosure. The screen voltage, Vs, is essentially the net accelerating voltage. In some descriptions of the functions of electron bombardment, Hg ion thrusters, Vs is referred to as the net accelerating voltage because the net energy of the ions in the thruster discharge beam is due to their repulsion from the positive screen grid after the ions have been extracted through it by the negative electrostatic field of the accelerator grid.

2. The aftermost thruster grid, usually referred to as the "accelerator grid," is misnamed. Its real function is to extract the Hg^+ ions from the internal cavity of the thruster; focus their paths so that the ions do not impinge on the solid parts of either the screen grid or the accelerator grid; and focus the small individual beamlets so that the composite, neutralized total thruster beam is, as nearly as practicable, a cylindrical beam.

In the ideal case, no power is required to maintain the accelerator grid potential because the positive work done in accelerating ions toward the grid is equal to the negative work done in decelerating the ions after they have passed through the accelerator grid. This is illustrated by the plot of ion energy versus position relative to the grids shown on Figure 6-18. In the practical case, the ion beamlet focusing is not altogether perfect so some ions do impinge on the accelerator grid. Furthermore, there is some finite vapor pressure of the un-ionized Hg atom that causes them to leak through the holes of both grids. When neutral atoms with this thermal energy are impacted by an accelerated high energy ion a "charge exchange" may take place so that the high energy ion becomes a neutral atom and the low energy atom becomes a single or multiple charged ion. This new charge exchange ion will be accelerated toward the negative accelerator grid in an unfocused manner and will impact it causing spluttering damage to the grid. Except during start transients, current flow due to the unfocused ions results in only a few milliamps of current in the accelerator grid circuit of a properly functioning thruster.

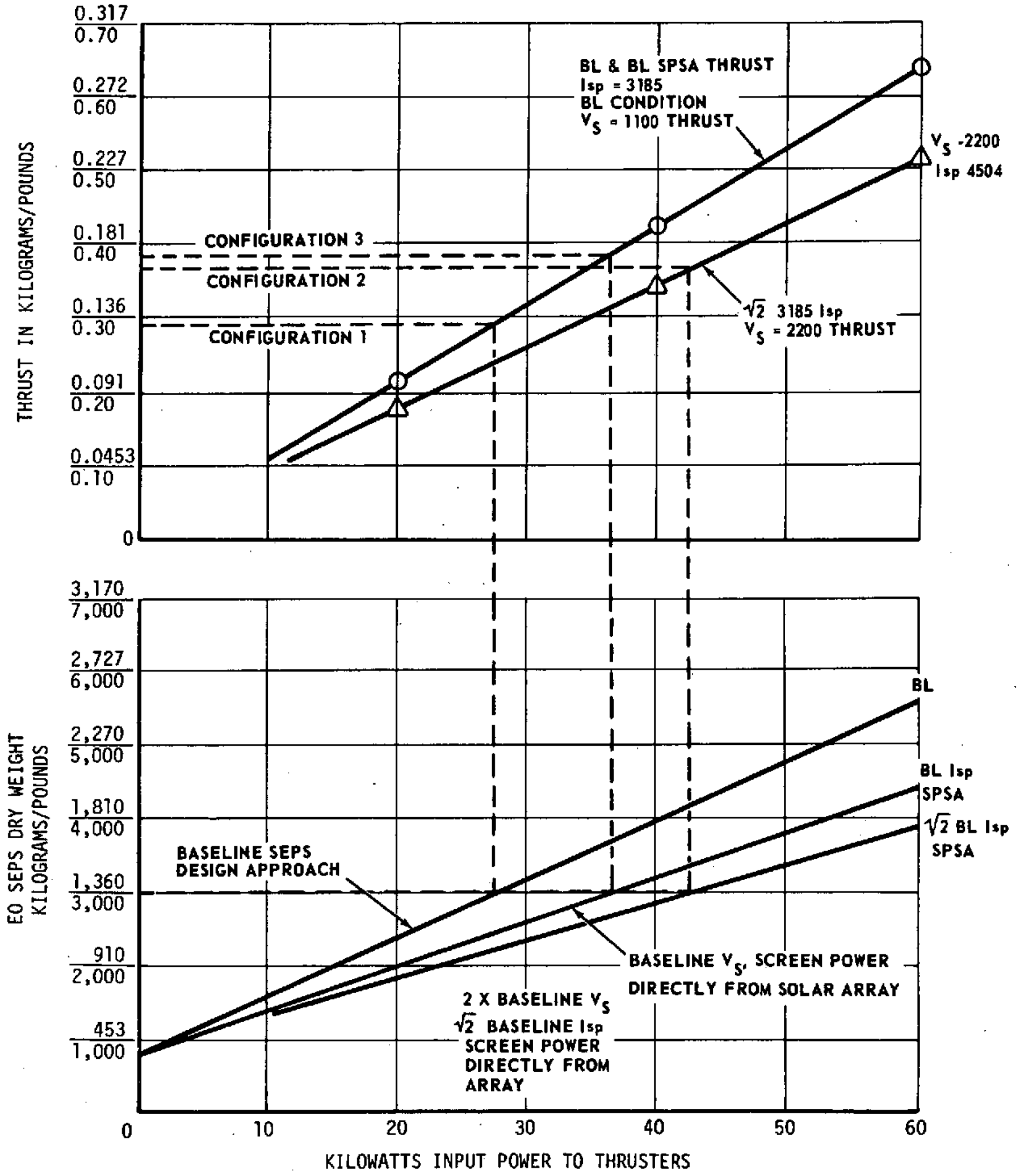

Figure 6-18. EARTH ORBITAL SEPS PARAMETRIC WEIGHT & THRUST DATA VS THRUSTER INPUT POWER FOR BL & ALTERNATE APPROACHES

3. Because of complex plasma charge and electrostatic field effects, the negative accelerator grid potential has only a second order influence on the rate at which ions are extracted through the screen grid holes from the internal enclosure of the thruster. The first order influence on the number of ions extracted is the Hg^+ ion density just behind the screen grid holes.

4. Thrust is proportional to the $\dot{m}$ of ions extracted and the square root of the ion energy, $\sqrt{eVs}$, screen current is proportional to the $\dot{m}$ of extracted ions.

Because of the above factors, the following situation exists. At a specific Vs, thrust is proportional primarily to beam density which is proportional to Hg^+ density internal to the thruster which is proportional to the Hg atom vapor pressure, assuming a minimum required number of bombardment electrons is produced by the cathode discharge arc. Therefore, both the thrust and resultant screen current are controlled by main vaporizer temperature control.

Consider the characteristics of the device just described. Its operation is stable. Large surge currents can not be produced in either its screen grid circuit or its accelerator grid circuit by voltage peaks.

Screen current is controlled by a rate of ion production primarily controlled by a rate of vaporization of main feed Hg propellant so that no large instantaneous current surge can be demanded of its power source. Screen voltage need only be DC, desirably ripple free. Screen voltage does not need to be controlled closely since it is not a primary control of the thruster.

Thruster specific impulse is directly proportional to the $\sqrt{Vs}$; therefore, the specific impulse at which the thruster operates can be selected simply by switching to a selected Vs.

Although not obvious from the schematic on Figure 6-16, it is a fact that the beamlet focusing for thruster operation at minimum design Vs and Va establishes the screen and accelerator grid geometry tolerances. In general, thruster beam optics, efficiency, and lifetime are improved by operation at higher voltages; and a given thruster may be operated at voltages up to 3 to 4 times the design minimum with improving efficiency and lifetime effects.

Thrusters are subject to a transient phenomena referred to as "arcing." This arcing, caused by a buildup of conductive contamination particles and possibly splutter-generated particles, occurs between the closely spaced screen and accelerator grids. Since the accelerator power supply circuit is designed for currents of about 0.2 amps and normally operates at a few milliamps, the arc must be extinguished to prevent overload of this circuit and the vaporization of material from the screens.

6.7.2 Motivation Factors For Use Of Screen Power Directly From the Solar Arrays

Briefly, the motivation factors for use of direct screen power are:

- Screen power processors are only 92 percent efficient.
- Screen power is 75 percent to 85 percent of total thruster power. Screen power processors, if used, are about 70 percent of the total power processor weight; and they require about 70 percent of the thermal control devices.
- Solar arrays are the most expensive single subsystem. Array cost and weight will increase by about 9 percent due to inefficiency of the power processors.
- Power processors will be more reliable, lower in cost, and lower in weight if they are not required to process screen power.
- Stage Δ mass saving from all sources (reduced solar array weight, less thermal control and PC weight, less stage structure, and so forth) as a result of using direct screen power is about 20 percent, or, conversely, the Δ power gain for the same mass is about 26 percent.
- Desired Isp ranges may be selected to match those desirable for each mission phase of a specific mission without the penalty associated with power processors that must operate over combined ranges of both high output voltage and high currents.

Figure 6-18 shows parametrically the relationship between SEPS configurations with three different approaches to the thruster subsystem. The basis for the weight scaling laws were SEPS weights from Rockwell International's Exhibit E studies in 1972 and 1973. The three approaches are:

1. All thruster power is processed with input voltage from the solar arrays to the PP in the range of 200V to 400V. Screen voltage is 1100 Vs, so nominal Isp is 3,000 sec. (Baseline system.)

2. Thruster screen power is taken directly from the solar arrays, but array panels are switched to keep screen voltage in the vicinity of 1100 Vs so nominal Isp is 3000 sec. Weight growth is less than 1 above because 75 percent of array input power is not processed and solar arrays are about 8 percent smaller.

3. Thruster screen power is taken directly from the solar arrays, but panels are switched to keep screen voltage in the vicinity of 2200 Vs, so nominal Isp is ($\sqrt{\frac{2200}{1100}}$ x 3000 sec) 4243 sec. Weight growth with thruster input power is less than 1 or 2 because 85 percent of array power is not processed (thrusters have higher electrical efficiency at higher voltages) and only 50 percent as many thrusters and associated elements are required as for 1 or 2. The solar array area is about 13 percent less than for 1.

6.7.3 Some Aspects of Thruster Power Directly From the Solar Arrays Considered Negative in Past Studies

NSI has conducted a diligent search to discover any significant negative factors that offset the advantages described in the preceding paragraphs. None of the negative factors were assessed as significant by NSI. The reviewer is invited to investigate and make his own assessments.

The first negative factor presented was that "space plasma" will cause more "leakage" over the face of solar arrays operating in the 1100V to 2200V range than one operating in the 200V to 400V range. Space plasmas are insignificant leakage sources above 300 km. SEPS will never operate below 300 km. Furthermore, 0.025 mm of clear FEP sprayed or bonded over the solar array provides added mechanical strength and protection plus an insulation capability to about 6000V.

The second negative factor presented was that switching the array panels led to reduced reliability. If all power sources for thruster operation are taken directly from the solar arrays (no power processing at all) switching controls on the arrays can become quite complex. NSI suggests that only screen power be taken directly from the solar arrays. Since the other miscellaneous power requirements are small, the control convenience of power processors for control circuits justifies the small losses associated with them. The "baseline" system of past studies involved eight power processors, any one of which could be switched to any one of nine thrusters. This involves Vs switching at 1100 Vs. The two solar wings each had two main panels that could be switched from parallel to series.

NSI suggests that each solar wing have three main panels and switching arrangements that allow the wings to be series connected and allow selected desired series-parallel arrangement of the panels to be switched. The thrusters each have access to a common solar-array supplied bus. The required switching is less than for the "baseline," and reliability is improved.

Some studies infer that power processors are required so that a deep space mission needing to produce the maximum screen current (maximum thrust) for the limited available power at large solar distances can be accommodated. Because of the thruster grids beam focusing characteristics previously described, there is a minimum suitable Vs for a given thruster design. This limits the lower Vs range, thus limiting the maximum current that can be used when available power is low.

If the three major panels per wing previously suggested were designed for 600V per panel at 1 AU, the equivalent 1 AU operating Vs conditions would be 600 Vs (not desirable), 1200 Vs, 1800 Vs, 2400 Vs, or 3600 Vs. Thrust level and Isp could be selected anywhere in this range to match the best choice for any specific phase of a deep space mission (or earth orbital mission). As the SEPS cruises out from the sun the available power decreases (refer to the previous discussion in this section with charts of planetary mission characteristics), but the solar cells are getting colder and their efficiency and output voltage is increasing. The output of the 600V panels is progressively rising. When their output reaches 800V to 900V all six panels could be paralleled to provide maximum current and therefore maximum thrust for that low power level. Power processors with their losses offer no apparent advantages and some very apparent disadvantages in even greater weight and significantly lower efficiencies if the range of Vs available from the arrays were to be provided by power processors.

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VII NAVIGATION, GUIDANCE &
RENDEZVOUS CONTROL SYS.

Section VII

NAVIGATION, GUIDANCE, AND RENDEZVOUS CONTROL SYSTEM

7.1 INTRODUCTION

In this section, the system selected by NSI for control of attitude and guidance during cruise and rendezvous is presented. In making this selection, use was made of previous studies by NSI and other organizations, so the final system selection represents the result of an evolutionary process. The requirements and baseline systems for Space Tug were also examined. Since Tug and SEPS will coexist in about the same time frame, NSI suggests that the two systems have as much commonality as is feasible in view of the differing mission requirements.

The systems selected by NSI are described in the following subsections, along with the rationale for the selections. The Guidance, Navigation, and Control (GN&C) avionics are described in subsection 7.2, and the Reaction Control System (RCS) in subsection 7.3. Factors which affect the requirements for these and related systems are described in subsection 7.4. In subsection 7.5, discussions of the related considerations of low earth orbit operations and level of autonomy trade-offs are presented.

7.2 GUIDANCE, NAVIGATION, AND CONTROL (GN&C) HARDWARE

In the selection of hardware for the GN&C system, consideration was given to the GN&C system planned for the Space Tug, which will be operational over essentially the same time frame. It is desirable that as much commonality as possible be maintained between the two systems to permit the sharing of development costs. To this end, designs for the Space Tug as defined in the Baseline Space Tug Configuration Definition MSFC 68-M00039-2 and the General Dynamics First Formal Performance Review Meeting, 11 December 1974 were examined. In the former study, avionics hardware includes:

- IMU with accelerometers (6)
  (6 laser gyros in a "pair and spare" configuration)
- Laser rate gyros (6)
- Star scanners (2)

- Sun sensors (2)
- Scanning ladar
- Slow-scan, low light-level TV with strobe lamps
- SUMC modular computer
- Steerable high-gain antenna.

The laser gyro unit is a Sperry development and is currently being tested at MSFC. Bendix image dissector star trackers and Adcole sun sensors are used. The Adcole sun sensor was also recommended by NSI.

The scanning ladar has already been baselined for SEPS. It can passively acquire a target (in sunlight) at a 2,222 km range, and actively track and range at 54 km.

The SUMC modular computer is an MSFC development, and is characterized by a building block structure that can be configured for the specific needs of the mission.

The General Dynamics design is similar, but uses:

- Dodecahedron laser gyro configuration
- Interferometric Landmark Tracker (ILT)
- Electronically steerable, phased-array antenna

The dodecahedron configuration was previously recommended by NSI using conventional gyros. It has the advantage that it is operational with any three gyros failed. With up to two failures, faulty gyros can be detected and isolated.

In order to perform autonomous navigation, it is necessary to determine the line of sight to the earth, as well as to inertial references. Horizon scanners can perform this task, but with limited accuracy. Also, horizon scanners require rotating components which give weight and reliability problems. General Dynamics uses the ILT for this purpose. The ILT uses four antennas in a square pattern, tracking with a high degree of accuracy, and can function with one antenna failure. It has been demonstrated, using a dedicated beacon, at synchronous altitude on ATS-F. The device has also been proposed by IBM for use by SEPS.

To obtain the gain required for the slow-scan TV without using high-power amplifiers or steerable antennas, the General Dynamics Tug design uses an electronically steered, phased-array antenna, consisting of 25 elements, each driven by a 1-watt transmitting module. This antenna has a gain margin of 3 db when transmitting at a 50 kbit/sec rate. Since each element is separately driven, redundancy is very high.

The TV units used by General Dynamics are 500x500 CCD devices as recommended by NSI for SEPS. The scan rate used is 15 seconds per frame. This is acceptable for SEPS during rendezvous, since SEPS itself has very long time constants. However, the scan rate would have to be more like one frame per second during payload handling, unless this is automated.

The sensor field of view requirements of SEPS are stringent because it is not spin stabilized (which would tend to ensure periodic viewing of reference bodies) and yet must function in arbitrary attitudes as demanded by the thrust vector and solar pointing requirements.

This implies that all sensors should have a 4π solid angle viewing capability. However, attempts to achieve this with sensor-out capability results in large numbers of sensors, and difficulties in selecting mounting locations. The interference of payloads further complicates the problem, and requires remote mounting of the sensors.

This problem can be alleviated if the requirement for continuous viewing is dropped in favor of guaranteed periodic viewing, for example, once per orbit. In addition, the need for high redundancy can be satisfied by permitting multifunction operation of sensors as backup for other units. For example, if suitable optics are provided, the spacecraft can be operated with somewhat reduced performance by using one of the TV units as backup for a failed sun sensor, star tracker, ladar, or ILT (using the TV as a horizon sensor).

The NSI design for SEPS uses essentially the same sensor hardware as described here for Tug, with such changes as are necessary to reflect the differences in missions. Primarily, the SEPS has less stringent accuracy requirements than Tug, but more stringent reliability requirements.

The NSI GN&C sensor configuration is shown on Figure 6-1. Six laser rate gyros are used in a dodecahedron configuration. (The second set used in the General Dynamics Tug design and in the Baseline Space Tug Configuration Definition is not needed.) No accelerometers are used. Instead, the thrust level used in the navigation Kalman filter is estimated from ion engine voltage and current. The 500x500 volt charge-coupled TV units are used, but with scan rate increased to one hertz during payload handling. To accommodate the higher bit rate, the phased array antenna is enlarged to 100 elements.

Two of the four TV cameras are mounted on gimballed computer-controlled scan platforms. This outboard mounting provides greatly increased flexibility of the cameras, and also relieves the problem of payload obscuration. The ladar is mounted on the upper scan platform, along with the TV camera, to which it is boresighted. This platform mounting of the ladar greatly improves the flexibility of the system during operations near rendezvous. The attitude of the platform is obtained by an optical angle encoder mounted on the gimbals. A spacing of 4096 steps per revolution (12 bits) gives a resolution of 0.09 degree. Alignment bias errors are removed by the data filtering.

The remote mounting of the TV units and ladar introduces certain problems of thermal control, data interfaces, reliability, and sensor alignment accuracy. However, the improved field of view represents a significant advantage. Note that failure of the platform drive mechanism would not completely disable the sensors.

The use of the ILT presents certain problems. It places more stringent conditions upon the attitude determination system. Also, the device may require additional support hardware. An IBM study indicates that horizon scanners and a radar altimeter may be needed as well. In spite of this, NSI has tentatively baselined the ILT because of the advantages it offers, under the assumption that the additional sensors are not required. Further study is necessary, and if it is found that horizon scanners are required, NSI would propose to use these without the ILT.

The number of sun sensors has been reduced from previous NSI designs to two -- one on each solar panel. These units serve essentially to direct the solar panels to the sun (not, however, directly -- they interface with the guidance computer). As a consequence, high accuracy and a wide field of view are not required.

The two star trackers provide the high accuracy attitude reference, and are mounted with a 90 degree included angle to optimize the accuracy provided.

Although it is still experimental at this time, magnetic bubble memory is suggested for bulk storage in lieu of tape recorder or similar mechanical devices, which do not have a good history of reliability. The bubble memory technology is almost certain to be sufficiently advanced to warrant its being baselined for SEPS. In fact, it is rumored that bubble memory will be the bulk storage system for the new generation of a major manufacturer, soon to be announced.

A block diagram of the NSI NG&C system is shown on Figure 4-11. All sensors feed the Kalman filter, which is a six-degree-of-freedom filter, simultaneously estimating attitude and orbital state. Processing of the TV outputs is provided to permit their use as backup sensors.

The General Dynamics configuration for the computer uses two 32-bit CPU's and a 48 k word semiconductor memory. The hardware used for SEPS may be different, because for this application reliability is a more critical factor than speed. NSI suggests the use of triple CPU's and a larger memory size. The 32-bit format is useful for SEPS.

7.3 REACTION CONTROL SYSTEM

A reaction control system (RCS) using 26 thrusters was proposed by Rockwell International in their Exhibit E document. The configuration is shown symbolically on Figure 7-1. Note that the system has four thrusters directed along the $\pm$ x-axis, six along $\pm$ y and two along $\pm$ z. In terms of torques, it can deliver couples from three pairs of thrusters about the $\pm$ x- and $\pm$ y-axes, and two about $\pm$ z. The number of thrusters used may appear excessive, but represents the minimum number which permits normal operation with any single thruster failed. The Rockwell RCS configuration has been retained by NSI, with minor adjustments in mounting.

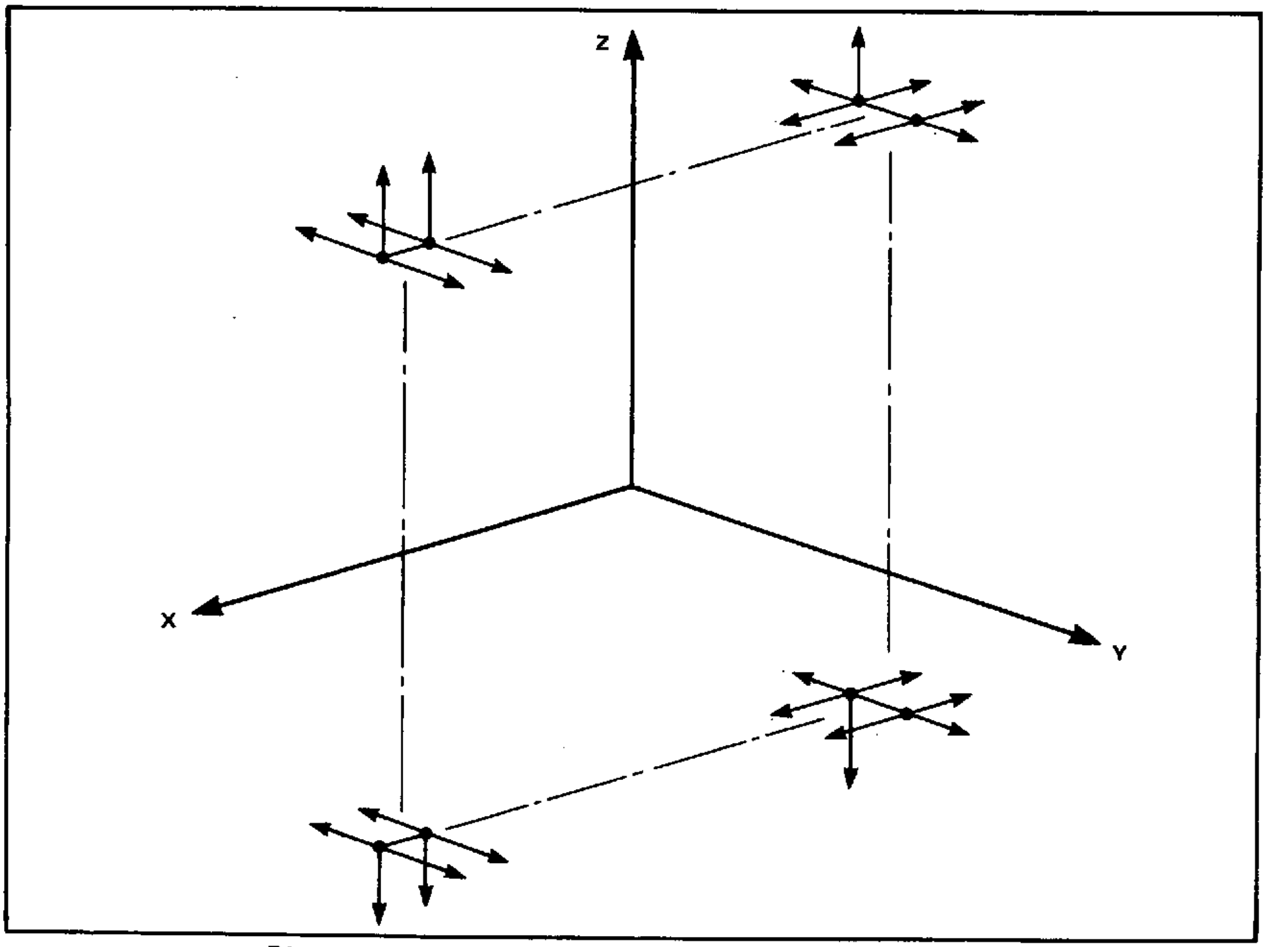

Figure 7-1. RCS THRUSTER CONFIGURATION

It should be noted that not all of the y- and z-axis translation thrusters can be used effectively. Since the payload is of necessity cantilevered beyond the physical boundaries of the RCS thrusters, it is impossible to perform a y or z translation without inducing a couple, which must be bucked by a pair of thrusters in pitch or yaw. In general, to obtain a translational force equivalent to one thruster will require the firing of three others, effectively lowering the specific impulse of the RCS fuel by the same factor. This situation is unavoidable, and the only solution is to avoid y-axis or z-axis translations. There are no specific requirements upon the RCS thruster size.

The SEPS is a low-thrust vehicle with long mission durations, low acceleration and large, flexible solar arrays. For these reasons, rapid maneuvers in either rotation or translation are neither required nor feasible. The driving consideration for the RCS system capability is related to the man-in-the-loop maneuvers performed during rendezvous, docking, and payload transfer operations. The times associated with these maneuvers must be in a range in which man can provide effective control.

On this basis, NSI has established the admittedly arbitrary condition that maneuvers must be performed within a time period of 5 to 10 minutes. NSI experience with man-in-the-loop simulations indicates that this time frame is within the range of effective control.

For the SEPS spacecraft, almost all maneuvers are likely to be limited by the available force or torque. Thus, maneuvers will tend to be time optimal. For such a maneuver, the time and energy required can be evaluated as follows.

Consider a body moving with constant acceleration. The governing equations are

$$v = a\,t$$

$$x = \frac{1}{2}\,a\,t^2 \quad .$$

The time required to accelerate from rest to a point

$$x = \frac{\ell}{2}$$

is given by

$$t = \sqrt{\frac{2x}{a}} = \sqrt{\frac{\ell}{a}} \quad .$$

Since the same time is required to decelerate to rest at $x = \ell$, the total maneuver time is given by

$$\tau = 2\sqrt{\frac{\ell}{a}}$$

$$\tau = 2\sqrt{\frac{\ell m}{F}} \quad .$$

A similar relationship holds for rotation:

$$\tau = 2\sqrt{\frac{I\theta}{L}} \quad .$$

Although few SEPS maneuvers will be of such a simple form with constant acceleration in one parameter, the time constants obtained using the above equations provide a good measure of control effectiveness.

The Rockwell RCS design used thrusters with 0.136 kg thrust (decaying to 0.068 kg after blowdown). This gives thrusts and torques shown in Table 7-1.

Table 7-1. ROCKWELL'S RCS SYSTEM CAPABILITY

F_x = 0.272 kg
F_y = 0.136 kg (effective)
F_z = 0.068 kg (effective)
L_x = 0.622 kg m
L_y = 0.622 kg m
L_z = 0.415 kg m

Using a translational distance, ℓ, of 100 feet and a rotation angle, θ, of 180 degrees (typical values), the corresponding time constants for a loaded SEPS with wings retracted are given in Table 7-2.

Table 7-2. TIME CONSTANTS USING ROCKWELL'S RCS SYSTEM

MANEUVER	τ, MINUTES
x translation	8.6
y translation	12.1
z translation	17.1
x rotation	2.8
y rotation	6.3
z rotation	7.8

The capabilities indicated by this table are marginal. Since the thrusters are quite small, little weight penalty is incurred by enlarging them, and the biconvex mast solar array structure has enough rigidity to tolerate larger thrusters. Therefore, in the NSI design the RCS thrusters have been increased to 2.3 kg units. This reduces the longest time constant to less than 3 minutes.

The RCS propellant requirements have been estimated against the baseline mission model. The results are shown in Table 7-3. Note that the largest entry in this table (except for contingency fuel) is that for rendezvous translational motion. This is also the least accurately known quantity, so a large contingency has been included.

Table 7-3. RCS PROPELLANT BUDGET

PURPOSE	FUEL REQUIRED (kg)
Cruise Attitude Control	13.6
Rendezvous (for 6 rendezvous)	
Velocity matching	27.2
Translational maneuvering	4.1
Rotational maneuvering	0.68
100% Contingency	45.4
TOTAL	90.9

7.4 DESIGN DRIVING OPERATIONS

The SEPS operations involving the use of GN&C can be separated into two parts -- cruise operation and rendezvous. The factors in each of these parts which affect the GN&C system are discussed in the next two subsections.

7.4.1 Cruise Operations

During SEPS cruise periods, only two factors generate requirements upon the SEPS attitude control system. These are the perturbation due to gravity-gradient torque, and the requirement for attitude changes imposed by the thrust vector and solar panel steering constraints.

The gravity-gradient torque acting on SEPS was evaluated and found not to be a significant factor. The analysis is given in subsection 7.4.1.1

The requirements due to steering constraints are significant. They involve the phenomenon of so-called "gimbal lock," which is a consequence of the single rotational degree of freedom between the solar panels and the spacecraft. A detailed analysis of this problem is given in Appendix A.

The effect of the gimbal lock phenomenon depends more upon the operational philosophy than upon hardware considerations. Basically, if the system is required to point the solar panels directly at the sun, it can easily be shown that attitude control can be lost regardless of the torque capability of the system. If, on the other hand, a suboptimal steering program, which permits angular errors in solar panel pointing, is adopted, control can always be maintained with a certain amount of degradation in SEPS performance. The extent of this degradation can be estimated rather easily, and does not appear to be serious. However, as discussed in the subsection on low earth orbits (subsection 7.5), a definitive determination calls for the development of new analysis software, and is outside the scope of this effort.

For attitude maneuvering during cruise, it is desirable to use the control torque available by gimballing the main engines, rather than using RCS propellant. The control authority of these engines is computed in subsection 7.4.1.2.

It appears from the results that RCS propellant will not be required except at rendezvous, and during shadow periods.

7.4.1.1 Gravity-Gradient Torque. The gravity-gradient torque on a rigid body is given by

$$\underline{L} = 3\, n^2\, \underline{\rho} \times \underline{\underline{I}}\, \underline{\rho}$$

where $\underline{\rho}$ is a unit vector directed to the earth, $\underline{\underline{I}}$ is the inertia tensor, and n the mean orbital motion. If $\underline{\underline{I}}$ is diagonal, that is

$$\underline{\underline{I}} = \begin{pmatrix} A & 0 & 0 \\ 0 & B & 0 \\ 0 & 0 & C \end{pmatrix}$$

then

$$L_x = 3\, n^2 (C - B)\rho_y\, \rho_z$$

$$L_y = 3\, n^2 (A - C)\rho_z\, \rho_x$$

$$L_z = 3\, n^2 (B - A)\rho_x\, \rho_y \quad .$$

The maximum values of these torques occur at angles of 45 degrees, for which, for example,

$$\rho_y\, \rho_z = (.707)(.707) = .5 \quad .$$

Thus

$$L_{x_{max}} = k|C - B|$$

$$L_{y_{max}} = k|A - C|$$

$$L_{z_{max}} = k|B - A|$$

where $k = \frac{3}{2} n^2$.

For near earth orbits (period = 90 minutes)

$$n = 1.164 \times 10^{-3} \text{ rad/sec},$$

so $k = 2.03 \times 10^{-6} \text{ sec}^{-2}$.

The mass properties of SEPS were estimated using the NSI digital computer program CIPP (Composite Inertia Properties Program). This program permits the computation of the inertia properties of a complex body by describing it as a collection of simple geometric shapes. The SEPS was approximated by the simplified form shown on Figure 7-2. The parameters for each portion of this shape are shown in Table 7-4.

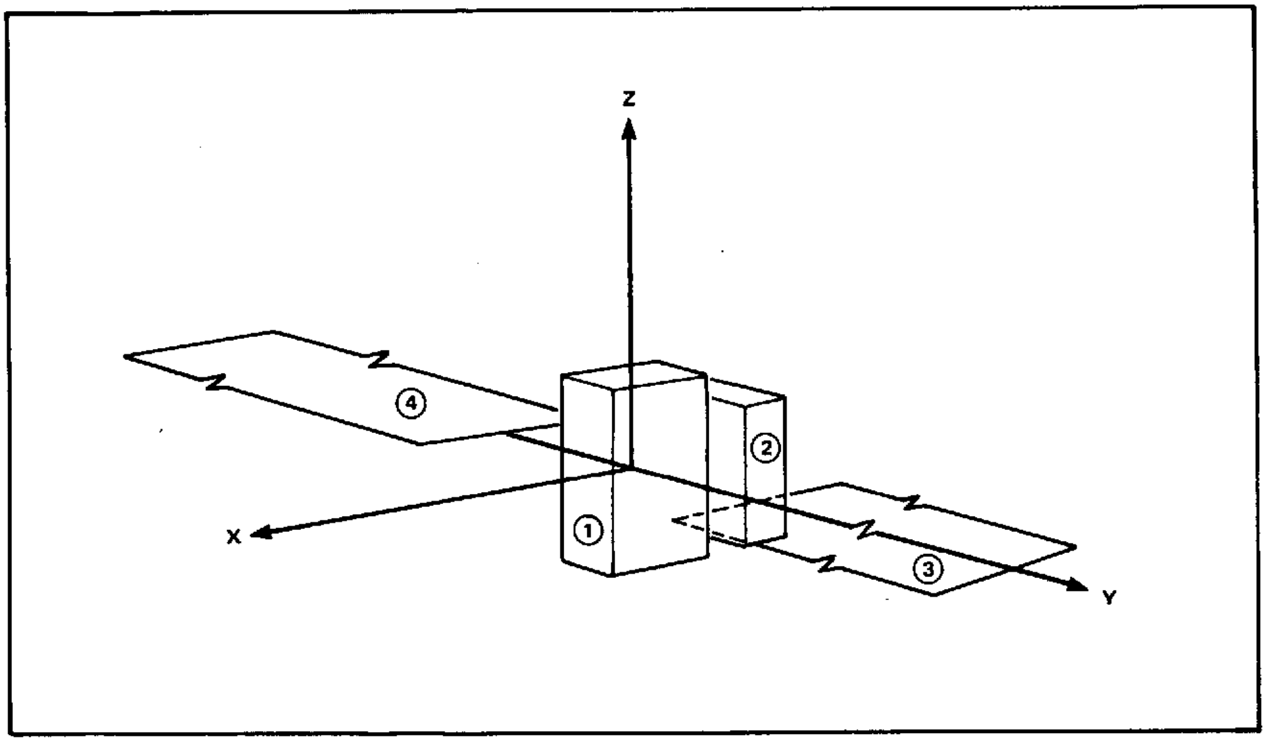

Figure 7-2. SHAPE USED FOR INERTIA ANALYSIS

Table 7-4. COMPONENTS OF SIMPLIFIED SHAPE

BODY	WEIGHT, kg	X LENGTH, cm	Y WIDTH, cm	Z HEIGHT, cm	COMPONENT
1	666	188	61	305	MAIN BODY
2	181	53	146	146	ENGINES
3	189	427	2662	0	LEFT WING
4	189	427	2662	0	RIGHT WING
5	4535	914	457	457	PAYLOAD (Cylindrical)

The inertia properties were also computed for the case in which a payload was attached. A heavy cylindrical payload was assumed as shown on Figure 7-3. The mass of the mercury propellant was ignored in computing the inertia properties, since this mass will be located near the composite center of gravity and contributes little to the moments of inertia. The resulting inertia properties are shown in Table 7-5.

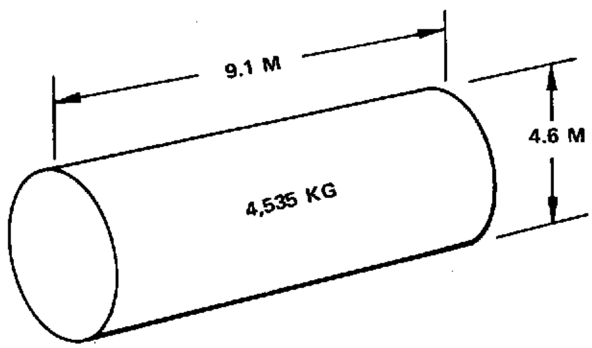

Figure 7-3. REFERENCE PAYLOAD

For the empty and loaded SEPS, the resulting torques are found to be

$$\left.\begin{aligned} L_{x_{max}} &= 0.0229 \text{ kg m} \\ L_{y_{max}} &= 9.67 \times 10^{-5} \text{ kg m} \\ L_{z_{max}} &= 0.0228 \text{ kg m} \end{aligned}\right\} \text{empty}$$

$$\left.\begin{aligned} L_{x_{max}} &= 0.0229 \text{ kg m} \\ L_{y_{max}} &= 0.0091 \text{ kg m} \\ L_{z_{max}} &= 0.011 \text{ kg m} \end{aligned}\right\} \text{loaded}$$

The most troublesome torque is that about the x-axis, which is also the axis for which control authority is smallest. However, all these torques, including the roll torque, are within the capability of the gimballed main engines.

Table 7-5. INERTIA PROPERTIES

NO PAYLOAD, WINGS RETRACTED

$$\underline{\underline{I}} = \begin{pmatrix} 220.5 & 0 & 0 \\ 0 & 158.0 & 0 \\ 0 & 0 & 266.7 \end{pmatrix} \qquad \underline{R}_{CG} = \begin{pmatrix} -.177 \\ 0 \\ 0 \end{pmatrix}$$

w = 1,224 kg

NO PAYLOAD, WINGS FULLY EXTENDED

$$\underline{\underline{I}} = \begin{pmatrix} 11{,}421 & 0 & 0 \\ 0 & 158.0 & 0 \\ 0 & 0 & 11{,}467 \end{pmatrix} \qquad \underline{R}_{CG} = \begin{pmatrix} -.177 \\ 0 \\ 0 \end{pmatrix}$$

w = 1,224 kg

10,000 LB PAYLOAD, WINGS RETRACTED

$$\underline{\underline{I}} = \begin{pmatrix} 1{,}432 & 0 & 0 \\ 0 & 7{,}187 & 0 \\ 0 & 0 & 7{,}296.5 \end{pmatrix} \qquad \underline{R}_{CG} = \begin{pmatrix} 4.3 \\ 0 \\ 0 \end{pmatrix}$$

w = 5,760 kg

10,000 LB PAYLOAD, WINGS FULLY EXTENDED

$$\underline{\underline{I}} = \begin{pmatrix} 12{,}632 & 0 & 0 \\ 0 & 7{,}188 & 0 \\ 0 & 0 & 18{,}497 \end{pmatrix} \qquad \underline{R}_{CG} = \begin{pmatrix} 4.3 \\ 0 \\ 0 \end{pmatrix}$$

w = 5,760 kg

INERTIA IN KG-M-SEC^2
DISTANCE IN M
MASS IN KG

If the SEPS is to be restricted to operations above the Van Allen belt (12,964 km), then k, and consequently the torques, are reduced by a factor of 25, and are no longer a significant factor.

7.4.1.2 Gimballed Main Engines. SEPS has nine ion engines with a thrust per engine of 0.0139 kg. With all nine engines operating at full power, this gives a total thrust of 0.125 kg. The ion thrusters are nominally mounted on a 3 by 3 matrix array, on 69 cm centers. They are gimballed in two axes with a maximum deflection of 28 degrees. Roll torque requires a couple to be generated between pairs of thrusters. The two thruster locations are defined to be Type A (corner) and Type B (side) locations, as shown on Figure 7-4.

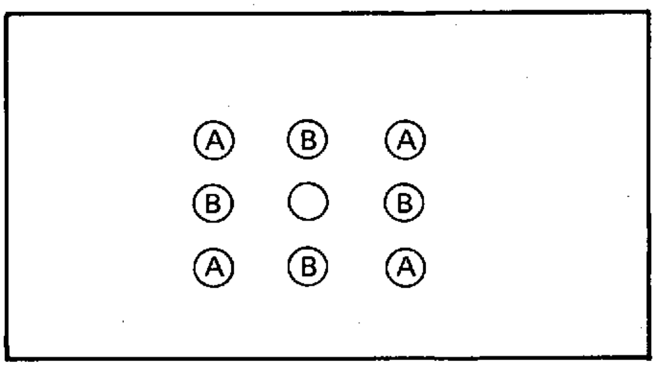

Figure 7-4. THRUSTER GEOMETRY

For the locations and gimbal angles stated above, the torque available from a pair of Type B thrusters is

$$L_B = 0.897 \text{ kg m}.$$

The torque from a pair of Type A thrusters is obtained when the gimbal angles generate a vector 45 degrees from the x-y plane. If one writes

$$T_y = T \sin \theta_1 \cos \theta_2$$

$$T_z = T \sin \theta_2$$

and requires these to be equal, one obtains

$$\sin \theta_1 = \tan \theta_2 \ .$$

For θ_1 = 28 degrees, this gives θ_2 = 25.1 degrees, and

$$T_y = T_z = .425 \ T \ .$$

The torque is then

$$L_A = 1.146 \times 10^{-2} \text{ kg m.}$$

The roll torque available from all eight thrusters is

$$L_x = 0.0409 \text{ kg m.}$$

The pitch and yaw torques available depend upon the moment arm distance between the engine gimbal plane and the composite center of gravity. This distance has been found to be

$$\ell = \begin{cases} 1.29 \text{ m} & \text{(empty)} \\ 5.77 \text{ m} & \text{(loaded)} \end{cases} \ .$$

The resulting pitch and yaw torques are, then

$$L_y = L_z = \begin{cases} 0.076 \text{ kg m} & \text{(empty)} \\ 0.340 \text{ kg m} & \text{(loaded)} \end{cases} \ .$$

The angular accelerations available using the gimballed main engines are shown in Table 7-6.

Table 7-6. ANGULAR ACCELERATIONS USING MAIN ENGINES

	EMPTY	LOADED
roll (x)	3.593×10^{-6} sec^{-2}	3.248×10^{-6} sec^{-2}
pitch (y)	4.834×10^{-4}	4.744×10^{-5}
yaw (z)	6.661×10^{-6}	1.844×10^{-5}

7.4.2 Rendezvous Operations

To evaluate the needs of the NG&C system during rendezvous, an analysis of the rendezvous maneuver was performed. This analysis is described in subsection 7.4.3. It does not appear that rendezvous imposes severe requirements upon the attitude control system, and it can be performed almost to contact using the main engines. However, operational constraints such as antenna steering may require the RCS system to be used near the target, and an allotment of RCS propellant is provided for this purpose.

7.4.3 Rendezvous Maneuvering

At the termination of a rendezvous trajectory, a terminal maneuver must be executed to match velocity with the target satellite. To avoid the unnecessary use of RCS propellant, it is desirable that the SEPS main engines be used for as large a portion as possible of this terminal maneuver. Factors which may limit the use of the main engines are:

- Requirement for rapid thrust vector direction changes near rendezvous
- Effect of ion engine plume impingement upon payload.

To investigate these considerations, it is necessary to consider the low-thrust rendezvous maneuver. The study of the maneuver is more difficult than in the case of chemical propulsion.

For a vehicle with chemical propulsion, the terminal maneuver is essentially impulsive, and simplifying approximations can be made. For a low-thrust vehicle, the terminal maneuver can take place over a period of many orbits, and the orbital dynamics and attitude maneuvering must be taken into account. Theoretically, the optimum terminal maneuver is given automatically in a natural way by use of a low-thrust optimization program such as MOLTOP with appropriate end conditions and constraints. In practice, however, a complete, three-dimensional optimization of the total trajectory is an inefficient way to study the terminal maneuver. Aside from the expense of a number of time-consuming runs, a terminal maneuver generally has little effect on the total fuel used in the mission, and thus will be only loosely optimized. Also, the three-dimensional optimization tends to call for rather extreme out-of-plane thrusts at termination, which tend to obscure the effects sought. To avoid these problems, a method of studying the terminal maneuver alone is needed.

Several studies have been conducted of suboptimal terminal maneuvers for low-thrust vehicles. Typically, these studies involve the assumption of an a priori pitch program, with constants determined by iteration in order to satisfy the boundary conditions. An NSI study* has treated the terminal rendezvous maneuver. In this study, it was assumed that truly optimum steering would not be used, but instead some empirical steering law. A linear pitch profile was used for the study. Also, an initially circular orbit was assumed. Because of these assumptions, the results are not as generally applicable as might be desired; however, some useful results were obtained. Figure 7-5 shows one trajectory from the study, a burn-coast-burn rendezvous. Note that since the coast period is 9.6 hours, this is essentially the low-thrust analog to a Hohmann transfer. The need for a coast period is open to question. It proved to be more nearly optimum in the study cited. However, this may be a consequence of the linear pitch profile assumed.

During the SEPS effort, an alternate approach, suitable for the study of continuous thrusting, was developed by NSI, and is outlined in Appendix B. In this technique, the radial position time history is specified a priori. This is used to find the pitch program for a continuous thrusting which yields the commanded radial motion.

Example approach trajectories obtained through this method corresponding to the exponential function described in Appendix B, are shown on Figure 7-6. Of this family, the most attractive trajectory appears to be that obtained when λ has its maximum value of n/2. This particular case, which is detailed on

Greenleaf, W. G., "Solar Electric Propulsion Stage (SEPS) Geosynchronous Rendezvous Geometry, Propulsion, and Guidance Compatibility Analysis," NSI Memo M-240-1215, May 1973.

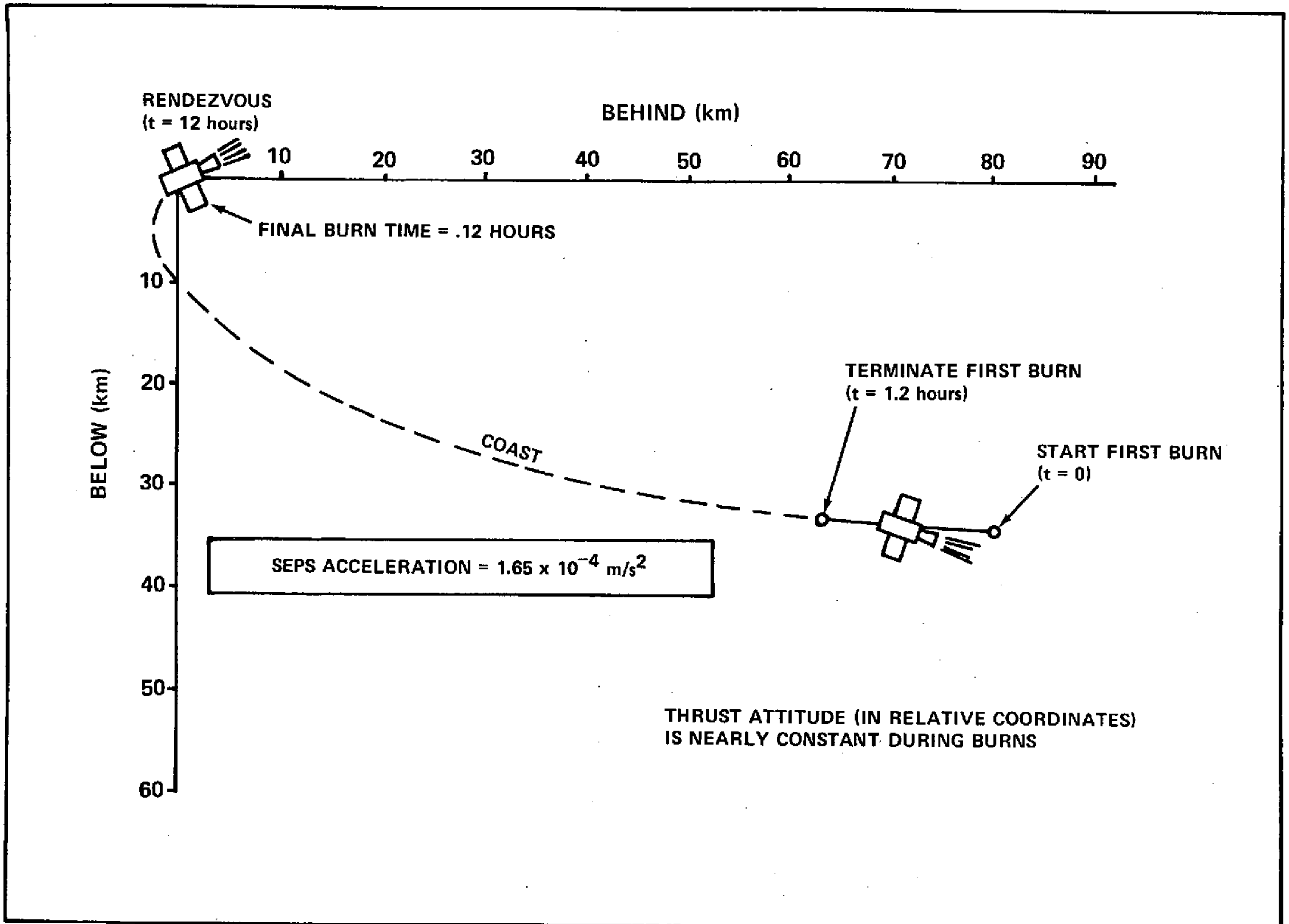

Figure 7-5. BURN-COAST-BURN RENDEZVOUS USING SEPS

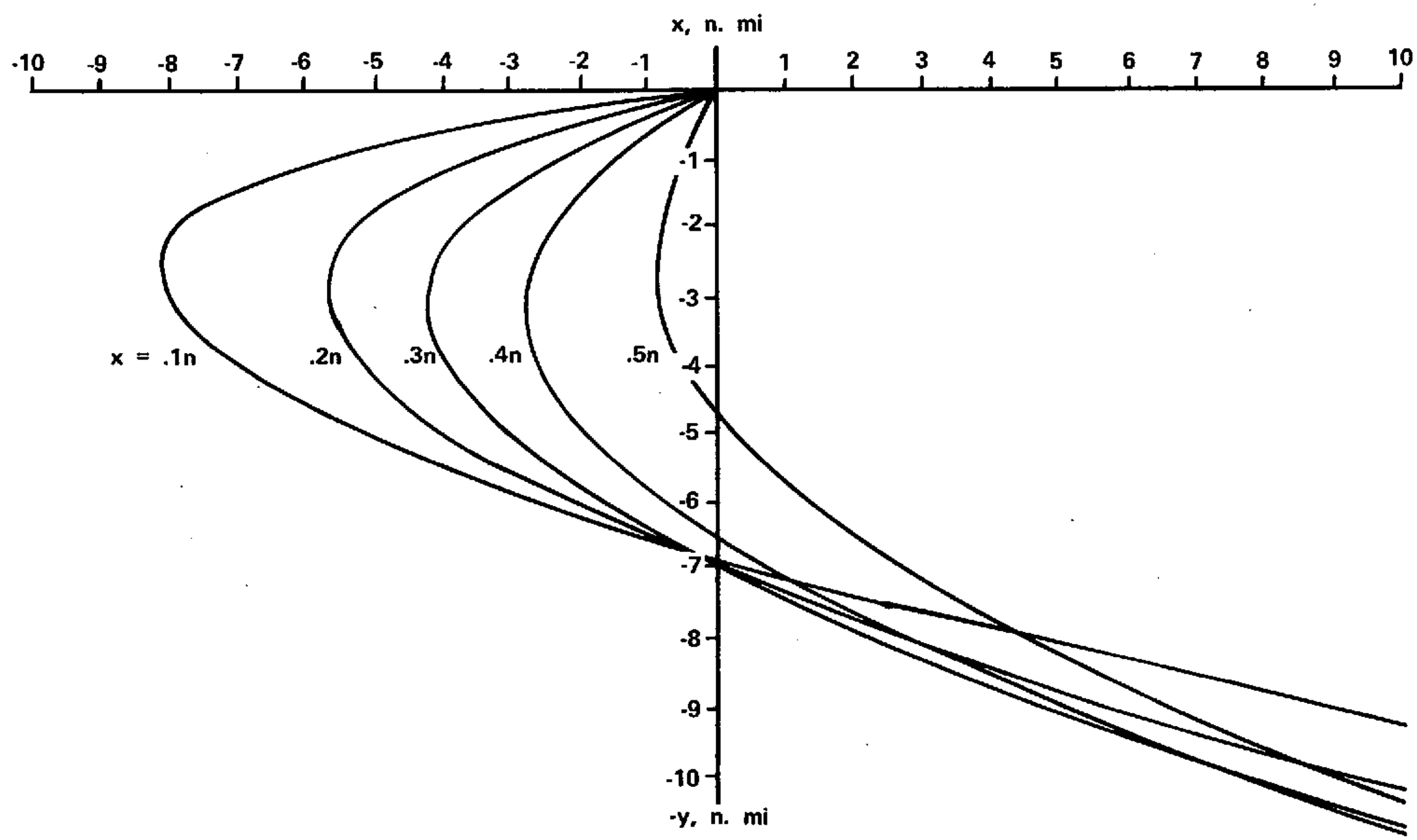

Figure 7-6. RENDEZVOUS TRAJECTORIES

Figure 7-7, is characterized by a final thrust vector which is directly vertically down. At a large distance from the target, the vehicle angle of attack (pitch angle from the horizontal) for the $\lambda = \pi/2$ case is zero, and the SEPS is operating in an orbit-raising mode. Beginning about ten orbits from rendezvous, the angle of attack begins to increase, reaching a maximum of about 62 degrees at 6 hours before rendezvous. This behavior represents something of a surprise. It may be a consequence of the choice of the exponential function used. However, it is felt at this time that this behavior is more universal than that, and is required in continuous-thrusting cases to avoid inducing eccentricity into the final orbit. However, note that it gives favorable geometry since the angle between the thrust vector and the line of sight to the target remains relatively constant over a large portion of the approach. In the last few hours of the maneuver, the SEPS begins to pitch down again, and has an angle of attack of -90 degrees at rendezvous. This pitch motion presents no difficulty to the attitude control system, but does complicate the laser radar tracking, ground communications, and so forth. In practice, it is probable that the ion engines will be shut down at some point, and final approach will be accomplished using the RCS thrusters. If these thrusters are used exclusively within 5 nautical miles (5 hours) of the target, they must supply about 2.286 m/sec ΔV capability, which in turn requires about 9.1 kg of RCS fuel for the loaded SEPS.

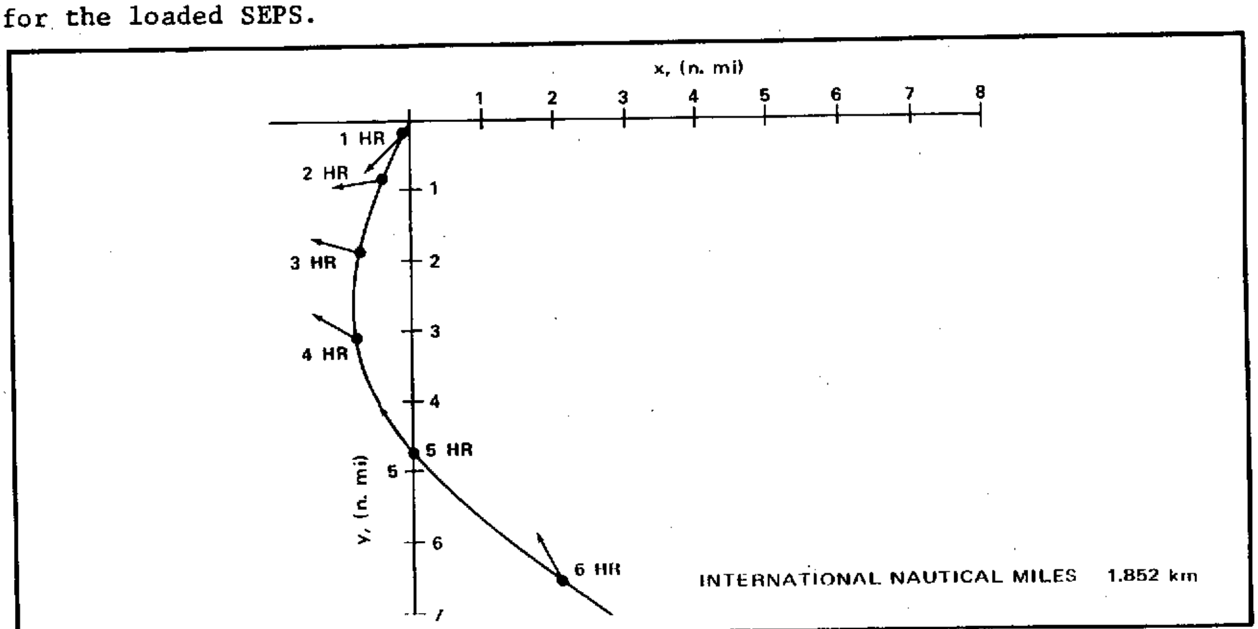

Figure 7-7. SEPS RELATIVE MOTION APPROACH SHOWING FINAL FEW HOURS

At the other extreme, the SEPS main engines could be used as long as possible, say to within one hour (457.2 meters) of rendezvous. To accomplish this, the ladar would have to be gimballed. This is done in the NSI design by mounting the ladar on one of the TV scan platforms. In this case, RCS requirements can be reduced to <0.3 m/sec and <0.9 kg fuel. In practice, the approach used will fall between these two extremes. NSI is allotting 4.5 kg of RCS fuel for each rendezvous.

The technique developed by NSI for this study appears to be quite useful since it permits the rapid generation of large families of candidate approach trajectories by defining functions of a single variable. Further study is warranted to extend the range of useful functions.

7.5 LOW EARTH ORBIT OPERATIONS

NSI has performed preliminary investigations as to the feasibility of SEPS operation in low earth orbit (LEO). Certain problems occur when operating in this mode. One of these is the rather large angular accelerations called for to meet thrust vector and solar panel pointing constraints. An analysis was performed of this problem, and it is detailed in Appendix A. It was found that an acceleration factor can be defined which is a function of the thrust vector slew rate $\dot{\lambda}$ and the minimum thrust vector/solar vector angle θ. The parameter $\dot{\lambda}$ is determined by the steering control system, and can be very large (in fact, infinite). However, in general it will be proportional to the orbital mean motion. For the case in which $\dot{\lambda} = n$ (an important special case), the values of $\ddot{\phi}_{max}$ are shown as functions of orbit altitude on Figure 7-8. Values of the acceleration factor (which is related to the misalignment angle) from 1.0 through 5.0 are shown. The dashed line represents the roll acceleration available to SEPS using gimballed main engines. One method to limit the angular acceleration to an acceptable value is to deliberately steer for a misalignment angle. If this is done, the required angular error can be directly related to altitude by a cross-plot of Figure 7-8. The result is shown on Figure 4-11. As can be seen, low earth orbit (300 to 1000 nautical miles) operation is feasible for the case $\dot{\lambda} = n$ with a misalignment of 26 degrees, corresponding to a 10 percent power

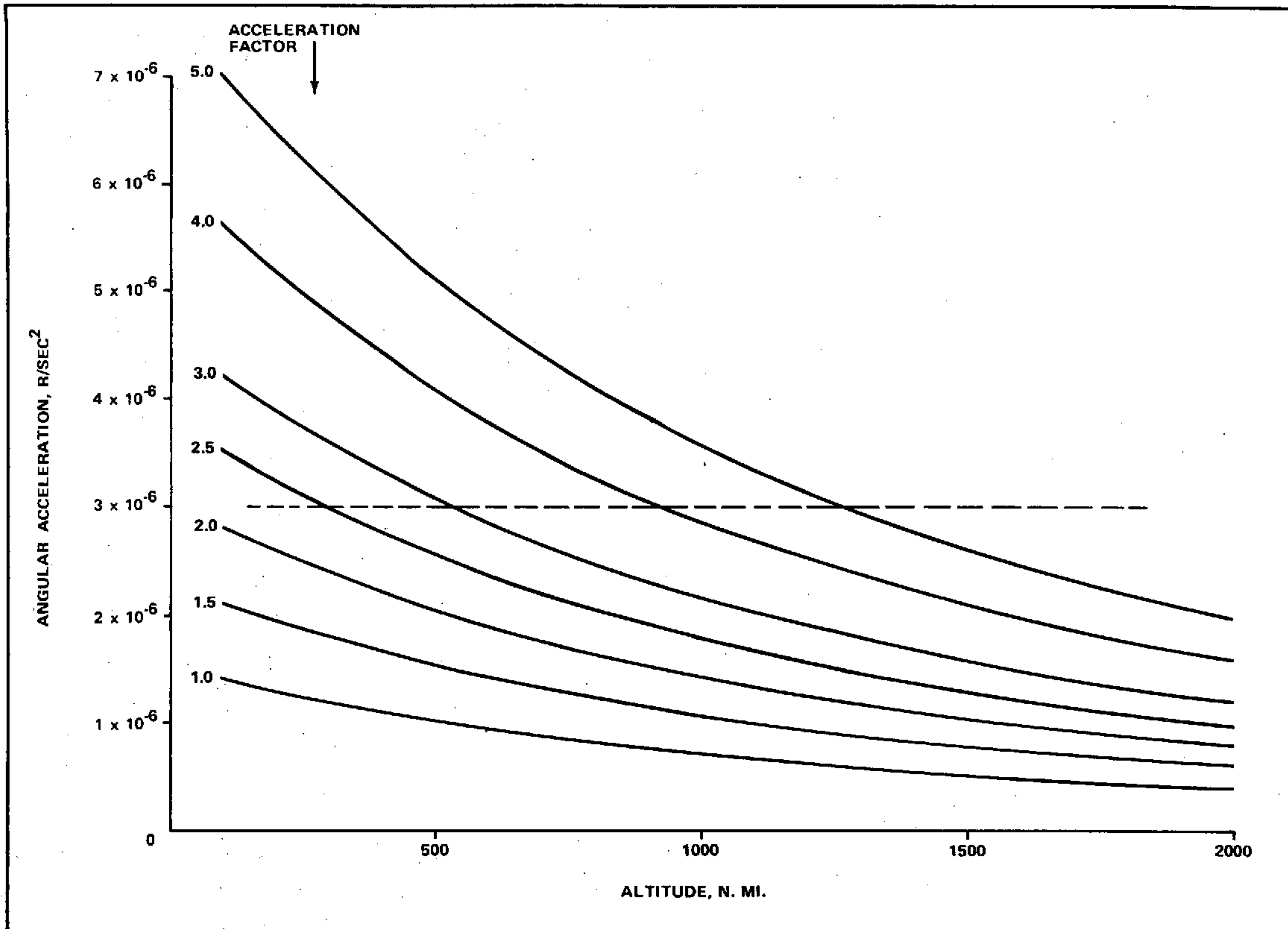

Figure 7-8. MAXIMUM ANGULAR ACCELERATION

loss. Rockwell has also studied this problem and arrived at a similar conclusion. Their "softened" steering and "alternate" steering methods, when applied together, will reduce the misalignment angles to roughly half those shown on Figure 7-9.

It should be noted that maneuvers faster than those at the orbital rate are required. For plane change maneuvers, for example, attitude changes that are essentially instantaneous are called for by the trajectory optimizer. In practice, however, the maneuver time need only be short compared to an orbital period. More will be said about this later.

Another problem occurring in LEO is that of shadowing. Rockwell has correctly pointed out that if shadowing is taken into account, the pitch angle maneuvers called for are much more violent than those of a continuous orbit-raising process (in which the assumption $\dot{\lambda} = n$ holds). This is especially true if the start-up time after shadow emergence is long. The Rockwell results indicate a serious degradation in fuel expenditure (by a factor of three) and mission time (by a factor of five). Some of the conclusions, however, may be artifacts of the method used.

For example, consider the shadowed trajectory shown on Figure 7-10. It is well known that the optimum thrust profile for an orbit-raising operation is to thrust normal to the radius vector as shown by the four arrows. If this same steering is done in an orbit that is shadowed, the thrust loss in the shadowed segment (segment D on Figure 7-10) causes an unbalanced condition. The orbit eccentricity increases, with the apogee being located on the shadowed side. The optimum place for application of thrust to raise the perigee is, of course, at apogee, but this is not possible.

There are two factors which reduce the severity of the problem. First, if the orbit is inclined to the equator (as most of the LEO orbits are), precession will cause the apogee to move out of the shadow, thus alleviating the problem. More directly, the orbit-raising process can be rebalanced by shutting down the engines in segment B. The result is a series of thrusts

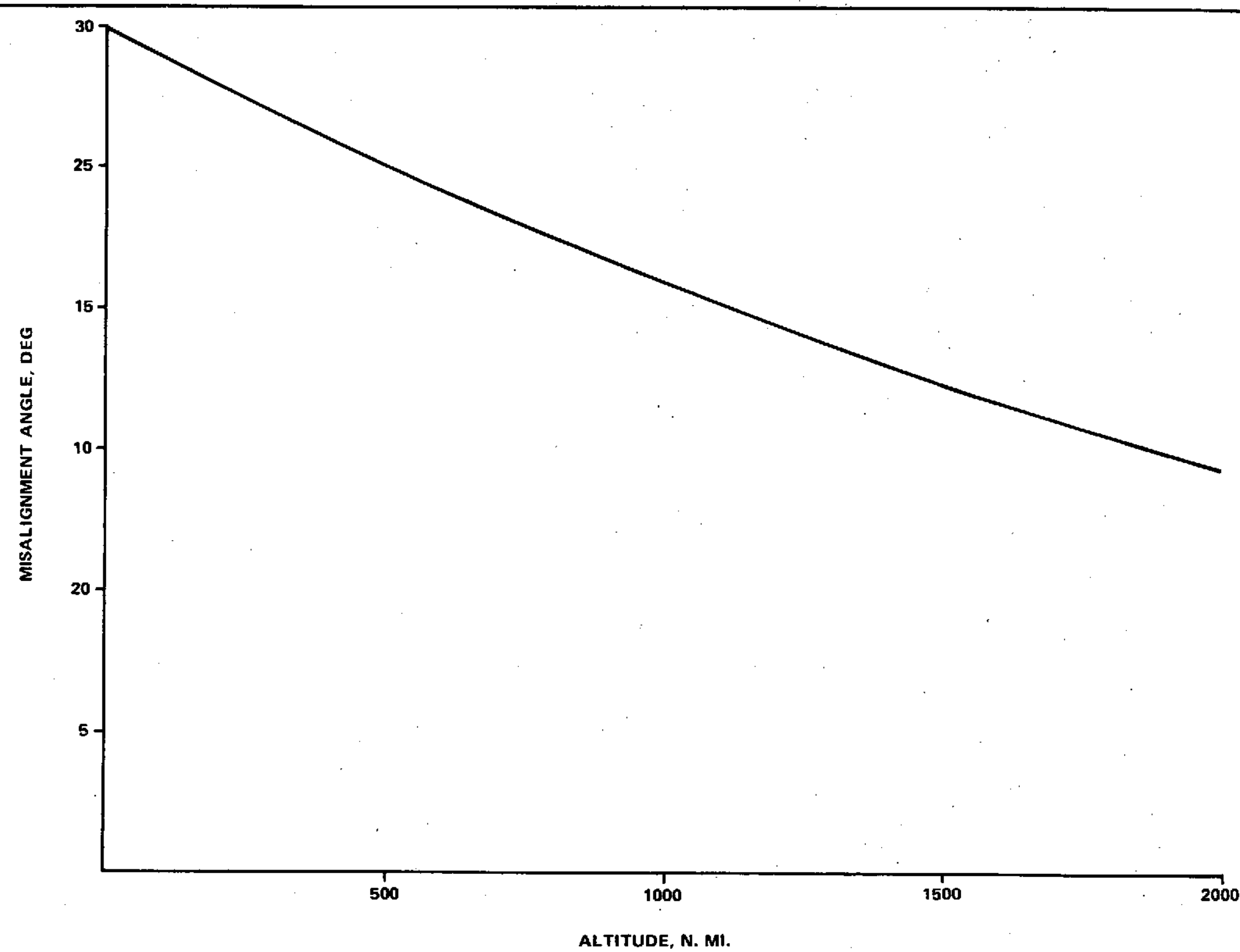

Figure 7-9. MISALIGNMENT ANGLE FOR SOFTENED STEERING

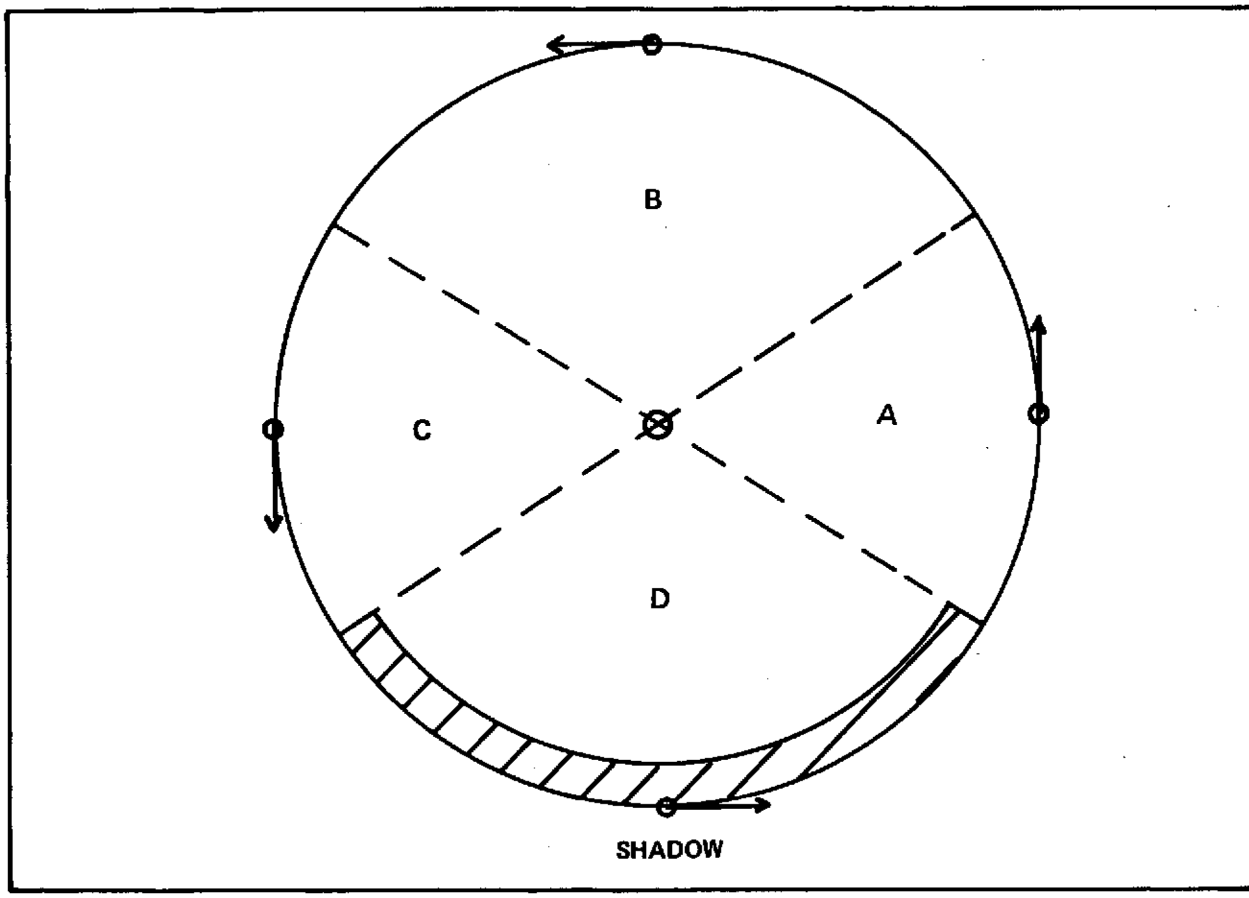

Figure 7-10. THE EFFECTS OF SHADOWING DURING PITCH ANGLE MANEUVERS

applied in segments A and C, and yielding a set of modified Hohmann transfers. This result did not appear in the Rockwell International study because of the low-thrust optimization program used. MOLTOP cannot generate the coasting subarcs. Examination of the results, however, will show that it did the next best thing: it called for 180-degree maneuvers of the thrust vector in segment B, thus effectively averaging the thrust in that segment to zero.

Of course, the use of coast segments in the mission cannot and will not improve the mission time over the Rockwell International results (although it will not increase it appreciably). However, it should greatly reduce the fuel expended to nearly the level of the unshadowed cases.

The high angular accelerations which continue to appear in studies of SEPS in LEO are similar artifacts. In a typical study, a 3-D trajectory optimization program such as MOLTOP is used to generate optimum trajectories. The thrust vector time history is then used to generate an attitude history. The high accelerations which arise are really a consequence of the fact that the thrust vector was constrained to lie *exactly* along the optimum direction, and the solar panels to point *exactly* at the sun. Rockwell International has already correctly pointed out that the required accelerations can be appreciably reduced by using "softened" steering. More correctly, the limited degree of attitude control authority available should be treated as a constraint in the optimization. The fact that the 3-D solution calls for unattainable accelerations is merely a statement that the constrained optimum is different from the unconstrained one.

The attitude control and optimum thrust factors also interact in that solar panel pointing errors affect the available power, and hence thrust level. This should not, however, be treated as a hard constraint on solar panel misalignment. (There may be other constraints on error, such as solar heating of the power processors. However, these can be modified if necessary by design changes, such as the use of heat pipes.) Rather, one should include the effects of misalignment upon engine thrust. Optimization would then automatically tend to keep the misalignment small.

The conclusion of NSI is, then, that while there are no compelling economic reasons that can be identified for the use of SEPS in LEO, it cannot be concluded from the studies to date that such an operation is infeasible. In order to establish the feasibility with confidence, a new trajectory optimization program is required. This program should be a 6-D attitude/translation optimization in which the engine gimballing for attitude control, effect of solar panel misalignment upon thrust, and oblateness effects are accounted for. The development and use of such a program is recommended only if a clear-cut advantage to SEPS over Tug in LEO can be identified.

7.6 LEVEL OF AUTONOMY

One factor of interest in several areas of spacecraft operations is the level of autonomy to be used, that is, the trade-offs between manual, ground-based control such as was used with early unmanned spacecraft, and automated navigation and control of the various functions. NSI has investigated these trade-offs for SEPS operations with respect to their impact upon SEPS hardware requirements. After considering some of these trade-offs, it has become the opinion of NSI that the difference between the approaches are so minimal that such trade-offs should not be attempted at this time, and in some cases, optimums may not even exist. As an example, consider the case of operation of the manipulator arms. Although completely ground-based (man-in-the-loop control may be baselined), there will be some operations requiring onboard control. For example, the operator will likely command composite operations such as end-effect commands rather than individual joint motions. It will be necessary, then, that joint feedback be provided to the SEPS onboard computer, and used to transform the operator commands into torque motor commands. To protect the spacecraft in the event of operator errors or telemetry malfunctions, it also would be desirable for SEPS to have a capability for avoiding interference between the arms and other parts of the SEPS or payload.

If, on the other hand, SEPS controls the arms autonomously, we would still insist on the ability to monitor the operation from the ground and override if necessary. For either extreme of operation, then, essentially the same hardware and software would be needed, namely:

- TV link with ground
- Arm control from ground
- Joint feedback to onboard computer
- Autonomous interference avoidance
- Onboard geometric transformations.

Similar considerations apply for other trade-offs between autonomous and man-in-the-loop procedures. It is rare that such trade-offs affect hardware requirements except with respect to onboard computer capacity. With regard to the onboard computer, advances in computer technology are proceeding at such a rapid rate that the task of estimating future capability is precarious.

In recent years, the technology of digital computer hardware production has made great advances, and digital computers are now commercially available with price, size, reliability, and performance figures which were not dreamed of a few years ago. In the past year alone, the following advances have been made:

- Several manufacturers (for example, Intel, Texas Instruments, Motorola, General Automation) have marketed 8-bit central processor units (CPU's) on a single integrated circuit chip
- 16K-bit read only memory (ROM) chips are now commercially available
- 4K-bit random access memory (RAM) chips are now available, with 16K expected within the year
- Fairchild has announced charged-coupled image devices (100x100 array) for TV service. Higher resolutions are expected shortly
- Charged-coupled "bucket brigade" shift registers for analog delay are commercially available. Modified versions for digital use (an easier task) are under development
- Experimental magnetic bubble memories are now operating with very high storage densities, high reliability, and low power.
- Several companies are now competing to be the first to announce nonvolatile, high-speed semiconductor memories.

Because of these recent developments and expected advances in the near future, it is feasible for the first time to consider an onboard control computer of true large-scale capacity. Estimates of weight, cost, and power requirements are difficult because of the rapid progress being made. However, even the most pessimistic estimates result in values that are essentially negligible compared to other SEPS subsystems.

Certain studies have tended to indicate that the reliability of the onboard computer may be marginal if a large memory is used. NSI cannot agree with these results. An increase in the amount of hardware permits an increase in the redundancy, error checking, and self-test and repair (STAR) capabilities which can be added, and thus increase, rather than decrease, reliability. With respect to cost, it should be pointed out that the overwhelming factor in computer-related costs is that of software development, which for a small computer is more difficult, and hence more costly, than for a large one.

The conclusion reached by NSI is that the level of autonomy used impacts only marginally the hardware requirements for SEPS, except in the area of on-board computer capacity. Since the choice of this capacity itself has only a marginal impact upon the SEPS cost, weight, and reliability, it is NSI's recommendation that sufficient hardware, including computer capability, be baselined to permit a high level of autonomy. The final trade-off between autonomy levels can then be deferred to a point at which more definitive data are available.

VIII COST ANALYSIS

Section VIII

COST ANALYSIS

8.1 BACKGROUND

The credibility of these cost estimates depends strongly upon the program reviewer's understanding of the system. As the reviewer compares these costs against his experience with past chemical stage programs and past satellite programs he should continually consider those physical and operational characteristics that allow SEPS to be delivered, produced, and operated with fewer people and a smaller range of disciplines than was possible for many reference programs. SEPS high Isp, 3,000 to 5,000 second range, results in the fact that its performance is relatively insensitive to increased mass. Reliable flight proven avionics from other space programs can be used without the necessity of additional development cost to reduce component weight or power consumption. New component development can be provided generous mass budgets that will allow reductions of cost in achieving program reliability, life, and performance goals.

SEPS is relatively simple. It is nearly all electrical. It has compact dimensions for transport and storage. Small buildings and small checkout equipment will support its few launch preparation and refurbishment activities. The largest cost in SEPS operations is for maintaining the range of disciplines for mission planning and flight control personnel. These personnel must know SEPS configuration, functions, subsystems, and components in detail. The personnel that support the launch preparation functions, the one or two refurbishments, and the sustaining engineering must know the system intimately. For the first 3 years of the program only six earth orbital sorties and two planetary missions (four SEPS with back-to-back launches) are flown. By the time flight frequency picks up to four sorties a year, the team will have had time to wring out all the bugs in their mission planning and operations programs and to establish streamlined manpower conserving computer aided procedures. The system operational profile (Figure 1-8) shows that in 11 years there are only eight planetary and four earth orbital launches to accomplish the reference mission model. Only one SEPS refurbishment for relaunch is required. Two are costed on the basis that retention of two program spares is desirable. There are only 30 earth orbital sorties by SEPS over the 11-year period. Recall the SEPS autonomous cruise and autonomous terminal approach phase of the rendezvous (when desired) capability so that a sortie, typically 90 days or less total time, has only four periods of peak activity where the mission planning and flight control crews are fully utilized. These periods of peak activity are associated with the following functions:

1. Detail planning of the next sortie in conjunction with the payload users and Shuttle flight planners
2. Systematic retrieval of the payloads to be returned to earth by Tug and Orbiter, and initiation of the cruise phase down to the Tug rendezvous orbit
3. Rendezvous with Tug, delivery of down payloads, acceptance of up payloads, and initiation of the ascent cruise phase to deploy up payloads at their mission conditions
4. Deployment of payloads at their mission station and performance of servicing functions for any other payloads requiring that function.

Readers interested and experienced in mission planning and flight control recognize those four functions in past space experience as time consuming and demanding of a large investment in man-hours. For this SEPS group, however, the longest involvement of any intense activity is with the payload sponsors in the detail mission planning. Other functions require 2 to 3 days' full utilization of a 16-man team around some key flight operation. A small investment in time and people (in spite of past experience) can accomplish in the SEPS program the four functions described previously, because:

- 13.2 million dollars is allocated for initial software. This breaks down to checkout and onboard ($4.5 million) and flight control center ($8.7 million) to automate the mission planning and flight control
- The group does only the SEPS specific detail planning. Two other principal groups provide controlling event sequences and transportation system function timelines to which SEPS must perform. The advance planning input comes from the Shuttle/STS Utilization and Master Scheduling Center. The detailed specific mission timeline event sequence for activities influencing Shuttle is established by the Shuttle Operations Center.

The total operations plus program support concept was selected to minimize personnel cost. Flight control peak activity with its rare but sometimes necessary requirement for dual shifts and backup of certain critical personnel sets the minimum number of personnel in the team. Flight control is required only about 5 percent of the time over an 11-year period. The operating concept uses a single facility for all program functions. The personnel will be cross trained to be competent in several program functions. This approach allows flight operations personnel to assist with engineering or have primary responsibility for accomplishing launch preparation, mission planning, refurbishment and other sustaining functions during SEPS idle periods onorbit and during autonomous flight periods. The analysis indicates that 45 people organized as shown in Figure 8-1 can accomplish all SEPS functions during the operational phase. If the SEPS flight unit is not autonomous during cruise periods, more people will be required. If the work is decentralized and responsibilities divided, more people would be required. In either case, the recurring costs would be higher.

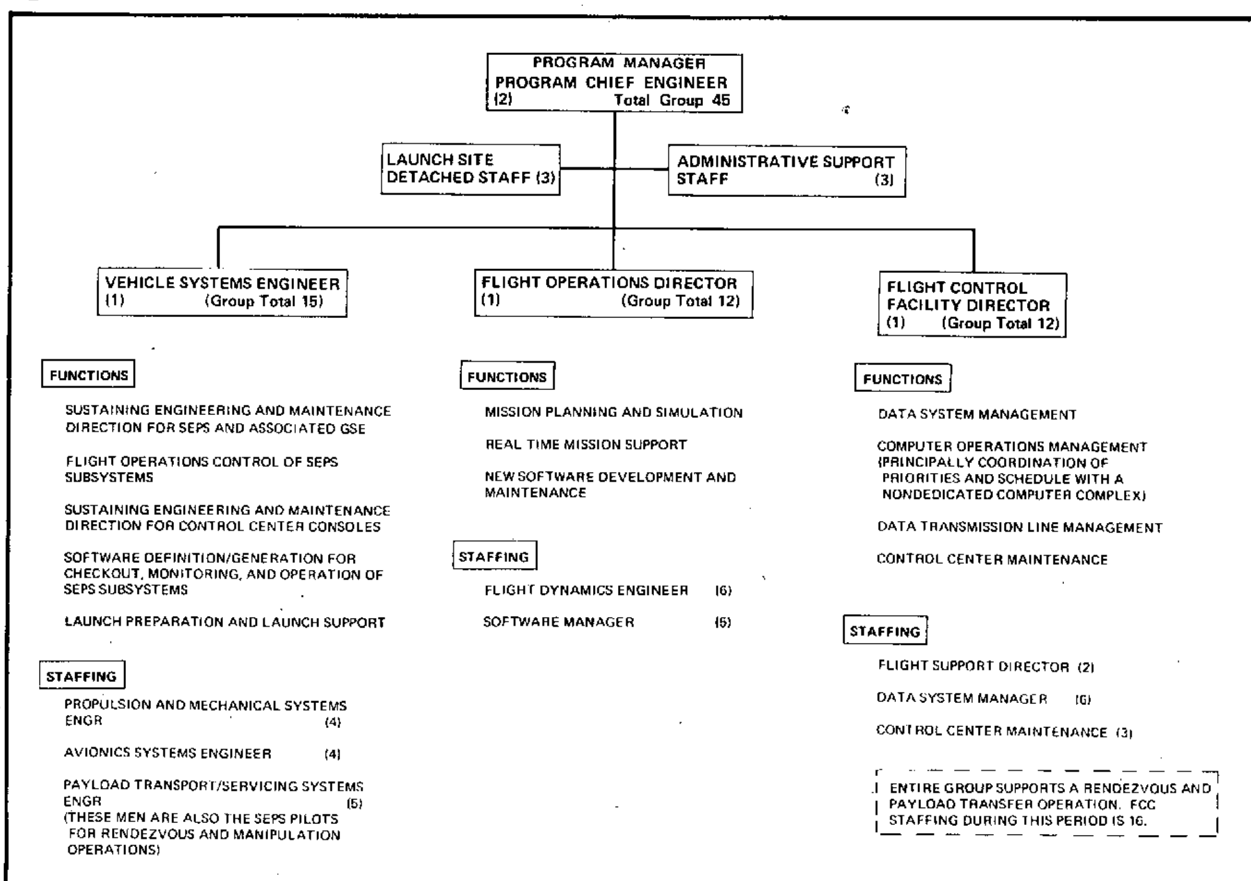

Figure 8-1. SEPS PROGRAM SUPPORT ORGANIZATION

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Reference to other sections of this volume and to Volume III, Design Reference Mission Description and Program Support Requirements, will provide a fuller understanding of a complete sortie and mission cycle for SEPS.

SEPS transportation due to its small packaged size (3m x 3m x 5m) and light unfueled packaged mass (1814 kg/2 tons) is convenient and inexpensive. The total supporting equipment and facilities investment is about $9 million, $5.3 million of which are allocated to computers and peripheral equipment. Computers are underutilized except for the previously defined periods of peak activity and should be utilized by the SEPS Operation Center (SEPSOC) host institution for its other functions. Computer systems are therefore costed to the SEPS program start-up; but computer operations personnel, assumed to be the host centers', are charged only for the estimated times they are required to support SEPS.

Because of the above factors, NSI believes that SEPSOC facility and equipment cost factors should not control the location of SEPSOC. To accomplish the program cost savings indicated by the 45-man total program support team, the SEPSOC must be located at the center that is given the total program responsibility for SEPS.

8.2 PROGRAM COST SUMMARY

The cost estimation assumptions used in the analysis are as follows:

There will be a single SEPS DDT&E and production program managed by one organization. The basic core vehicle will be capable of accomplishing either the earth orbital functions or the deep space mission when certain components and sensors are added. This may, on occasion, result in SEPS implementing missions with minimum objectives which do not require its full capability in solar array power or thrust. Extra capability in SEPS is bound to have some significant benefits to the science package either by allowing expanded objectives or by cost/reliability savings accruing due to relief of mass constraints.

NSI believes it is pennywise and dollar foolish economy to have tailored, reduced capability vehicles just to save a few hardware production dollars on a specific production vehicle since many of the deep space mission science packages will be ill defined when SEPS is produced to fly that mission and the extra capability of a standard SEPS core vehicle could be put to desirable uses. Further, the science packages to be flown will depend upon data from missions that are not available until after production is complete. It is very expensive to retain production and sustaining engineering on standby to produce mission special planetary SEPS. Therefore, the single DDT&E program will phase into production at the most economical rate for the total inventory. Each SEPS, after production, will undergo a rigorous flight readiness check as a part of the final acceptance testing. Then it will be stored in a hermetically sealed, inert gas filled container with its status check and power supply hard lines used in ascent flight carried through the container walls to a test umbilical. As each SEPS is completed, accepted, and installed in its storage container it goes to the launch site for immediate launch or to the SEPSOC for inventory storage.

When production of inventory and refurbishment spares are complete, the DDT&E/production contract is terminated. There is no sustaining engineering support team at any contractor or subsystem supplier's plant included in these cost estimates after production is complete. This does not preclude NASA from electing to have SEPSOC operated by a contractor, and the DDT&E contractor may be the successful bidder for the SEPSOC support.

It is technically feasible that the 45-man program support team at the SEPSOC make any modifications or system changes found later in the program to be deisrable.

Other assumptions are:

- Production is continuous for 11 vehicles. The first vehicle is delivered 30 months after authority to proceed (ATP).
- All \$ are 1974 \$.
- There are four planetary missions, each flown with a backup spacecraft requiring a total of eight planetary SEPS. Only two EO SEPS are required. One production spare is planned, and the integrated system test article is refurbished at the end of production to provide a second spare.

- Two refurbishments are included in the cost estimates. This extends the SEPS operating inventory adequacy beyond the 1991 operational time ground rules for this cost effectiveness study to 1997 if we assume there were no flight failures that caused a planetary mission repeat.
- The center given responsibility for the science package and mission operation will assume flight control of SEPS and the science package at some time after the cruise mode is established for the initial planetary trajectory. Only periodic advice or consultation from SEPS vehicle systems specialists will be provided on request of the planetary control groups after the cruise mode is established.

NSI's SEPS concept is one basic system, referred to in Section VIII as the "Core" SEPS plus equipment peculiar to planetary and earth orbital (EO) missions. In addition to the EO equipment, additional costs for the payload handling and servicing system (manipulator arm system and biconvex mast) are shown separately as "EO functions".

Example Using First Unit Cost Data:

"Core" Vehicle $16.75M	+	Planetary Peculiar $0.75M	=	Planetary SEPS $17.5M		
"Core" Vehicle $16.75M	+	EO Peculiar $0.75M	+	EO Functions $1.0M	=	EO SEPS $18.5M

8.3 COST SUMMARY

Table 8-1 presents the SEPS total program costs including planetary vehicle core development costs and the launch support operation for eight planetary vehicles.

8.4 DEVELOPMENT, DESIGN, TEST AND EVALUATION COSTS

The DDT&E cost shown in Table 8-2 was based on a single development program for the planetary and earth orbital SEPS. A core SEPS with all common systems would be developed. This basic stage would cost $89.2 million. The planetary and earth orbital deltas to common systems is included in the base price.

Table 8-1. SEPS TOTAL PROGRAM COST SUMMARY

STAGE DDT&E		97.5
EO Functions (Transport Mast & Manipulators)	(8.3)	
Basic Stage	(89.2)	
STS GPME DDT&E		2.5
PL Shell & Diaphragms	(2.5)	
FLIGHT ARTICLE PRODUCTION		145.9
8 Planetary Vehicles	(97.6)	
3 EO Stages	(39.6)	
STS GPME	(1.5)	
Stage Refurbishment and Maintenance	(7.2)	
SEPS OPERATIONS CENTER INITIAL COSTS		17.9
Facility and Equipment	(8.8)	
Initial Software Package	(8.7)	
Initial SEPSOC Spares	(0.4)	
SEPS SYSTEMS OPERATIONS		26.2
Personnel (45 men 11 years)	(23.7)	
Computer Support	(2.1)	
Flight Article Consumables	(0.4)	
TOTAL PROGRAM COSTS		290.0

Table 8-2. STAGE DDT&E COSTS
(All Figures are Dollars in Millions)

	TOTAL DDT&E	CORE VEHICLE	PLANETARY PECULIAR	EO PECULIAR
STRUCTURES & THERMAL CONTROL	$ 4.8	$ 4.8		
PROPULSION	9.1	9.1		
POWER DISTRIBUTION	1.0	1.0		
SOLAR ARRAY	7.8	7.8		
DATA MANAGEMENT	3.4	3.4		
COMMUNICATION	2.2	1.4	$ 0.5	$ 0.3
NAVIGATION & GUIDANCE/ATTITUDE CONTROL	9.2	6.0	2.2	1.0
INTEGRATION & TEST CHECKOUT	6.7	6.7		
TEST HARDWARE	21.3	19.8	1.1	0.4
STAGE GSE	5.0	4.0	0.2	0.8
SOFTWARE	4.5	4.5		
LOGISTICS	0.5	0.1		0.4
S.E.&I.	6.8	6.8		
PROGRAM MANAGEMENT	6.9	6.9		
BASIC SEPS	89.2	82.3	4.0	2.9
Δ FOR EARTH ORBITAL FUNCTIONS OR (PAYLOAD MAST & MANIPULATOR)	8.3			8.3
TOTAL	97.5			

The earth orbital SEPS will have an additional system for payload transport and handling. This system is composed of a payload transport mast and a manipulator system. Its cost of $8.3 million is presented in Table 8-3.

Table 8-3. SEPS PAYLOAD AND TRANSFER SERVICING SUBSYSTEM COST ESTIMATE

	DDT&E
PAYLOAD TRANSPORT AND SERVICING SYSTEM*	$ 1.9 M
POWER DISTRIBUTION	0.1
DATA MANAGEMENT	0.6
COMMUNICATION	0.2
INTEGRATION AND TEST/CHECKOUT	0.9
TEST HARDWARE	1.3
GROUND SUPPORT EQUIPMENT (GSE)	0.8
SOFTWARE	0.8
LOGISTICS	0.9
SE&I	0.7
PROGRAM MANAGEMENT	0.1
TOTAL	$ 8.3 M

*(*new Category) Manipulators/Payload Mast*

The SEPS system recommended by NSI contains general purpose mission equipment which supports payloads during STS flight operations. The equipment includes a payload half shell and support diaphragms. The development cost of $2.5 million is presented in Table 8-4.

Table 8-4. TUG PAYLOAD TRANSPORT SHELL AND DIAPHRAGMS COST ESTIMATE
(Dollars In Millions)

	DDT&E
TRANSPORT SHELL AND DIAPHRAGMS*	$ 0.2
TEST HARDWARE	1.2
INTEGRATION AND TEST CHECKOUT	0.3
S.E.&I.	0.8
	2.5

(NEW CATEGORY - COST SHARED WITH TUG)

The following manpower items (not involved directly with component and subsystem detail design and development) for the various engineering and nontechnical disciplines are provided for visibility of total DDT&E manpower requirements.

These manpower costs form the basis for the labor estimates for the DDT&E program.

(Costs in Millions of Dollars)

	DDT&E $	LABOR $
INTEGRATION AND TEST CHECKOUT	$ 6.7	$ 3.4
GSE STAGE	5.0	1.7
SOFTWARE (STAGE/TEST)	4.5	4.5
LOGISTICS	0.5	0.5
S.E.&I.	6.8	6.8
PROGRAM MANAGEMENT	6.9	6.9
TOTAL LABOR		$23.8

Initial production support is also shown to give visibility of the transition of personnel from DDT&E to production.

It should be noted that as the production pipeline becomes full, the balance of production support average manpower carrying through in the following categories is:

INTEGRATION AND TEST CHECKOUT	50
S.E.&I.	80
PROGRAM MANAGEMENT	80
TOTAL	210

Figures 8-2 through 8-6 present a breakout of the manpower by program month.

8.5 WORK BREAKDOWN STRUCTURE FOR STAGE DDT&E

The cost of the DDT&E phase for the stage is presented by the work breakdown structure shown on Table 8-5.

8.6 FLIGHT ARTICLE PRODUCTION

The assumptions and conditions are:

- Costs are in 1975 dollars.
- A single configuration core vehicle is produced.
- Production is continuous and includes 11 units. The DDT&E test and production sustaining engineering vehicle is refurbished at the end of production to provide a total of 12 vehicles.
- No material handling has been added to the subsystem costs. Particularly in the aerospace industry, there is a wide divergence in the treatment of expenses as overhead items or direct contract charges. Items sometimes considered separately as "Material Handling" and many items often considered "General and Administrative" expense have been included in the Program Management category.

8.6.1 Production Cost Summary

The 10 flight articles and 2 spares will be produced in a single production run at the most economical rate. Standard planetary kit items will be incorporated in eight vehicles and standard EO equipment will be incorporated

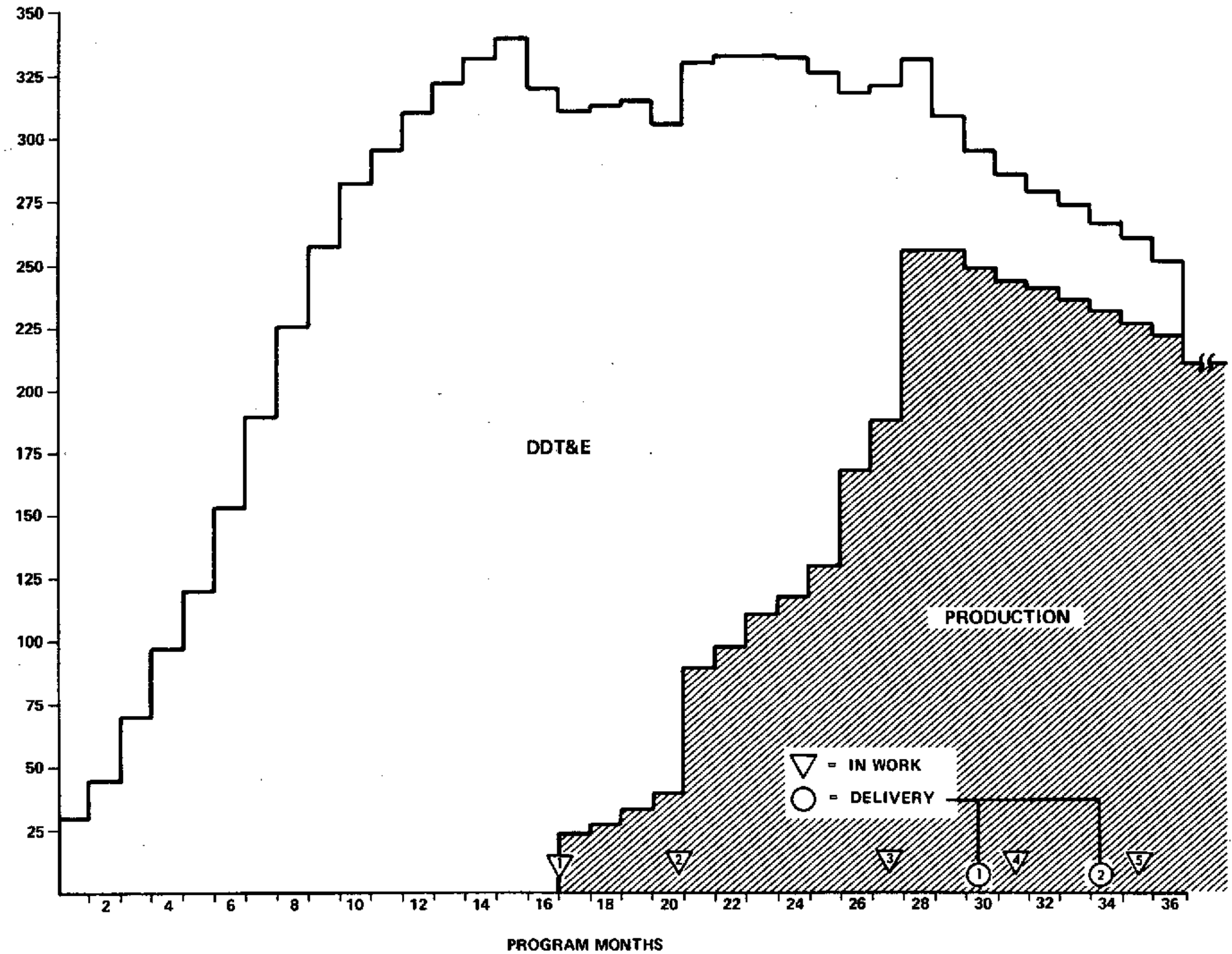


Figure 8-2. TOTAL SEPS PROGRAM MANLOAD, MONTHS 1 THROUGH 36 ONLY

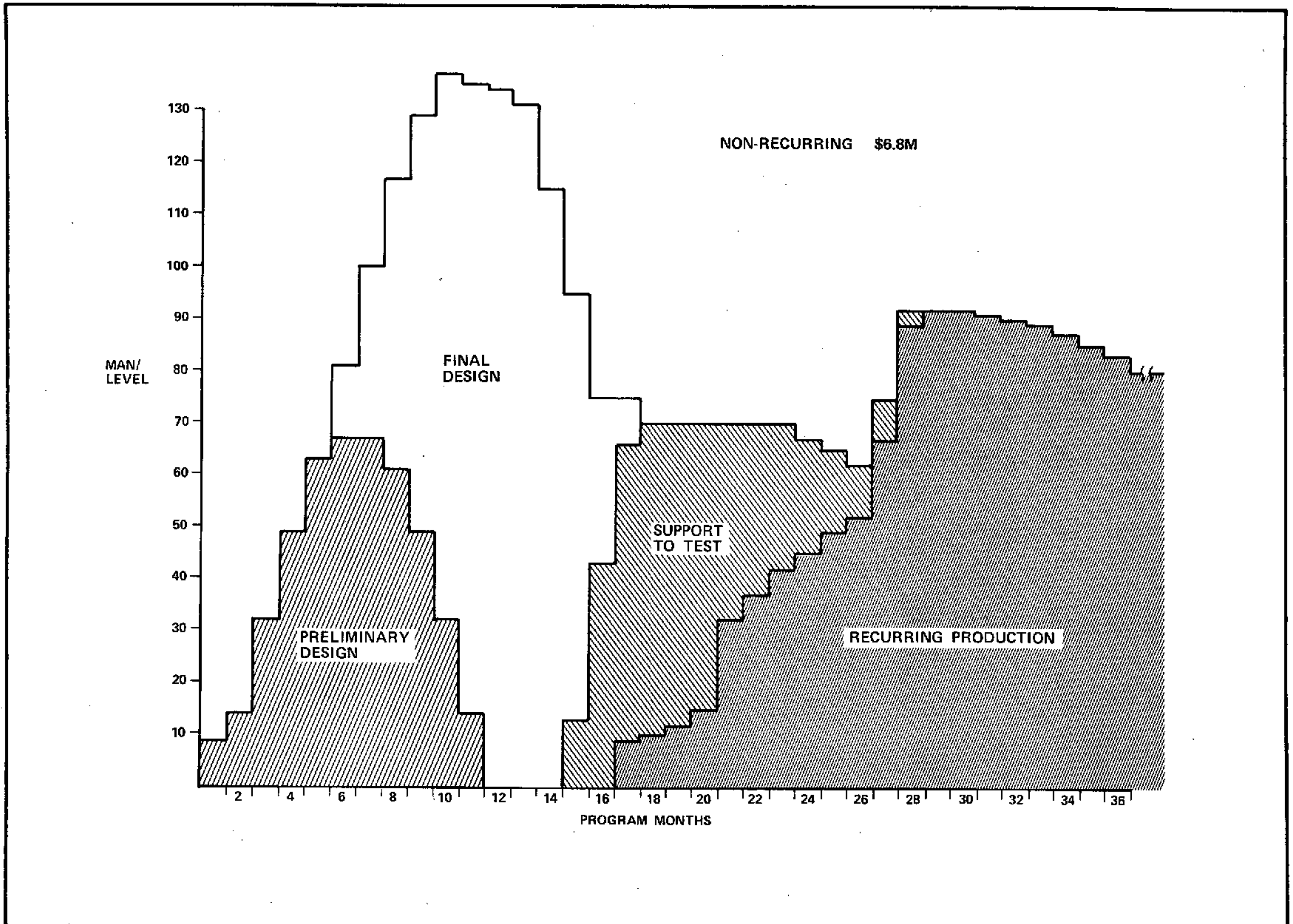


Figure 8-3. SE&I - TOTAL MANLOAD BY MONTH

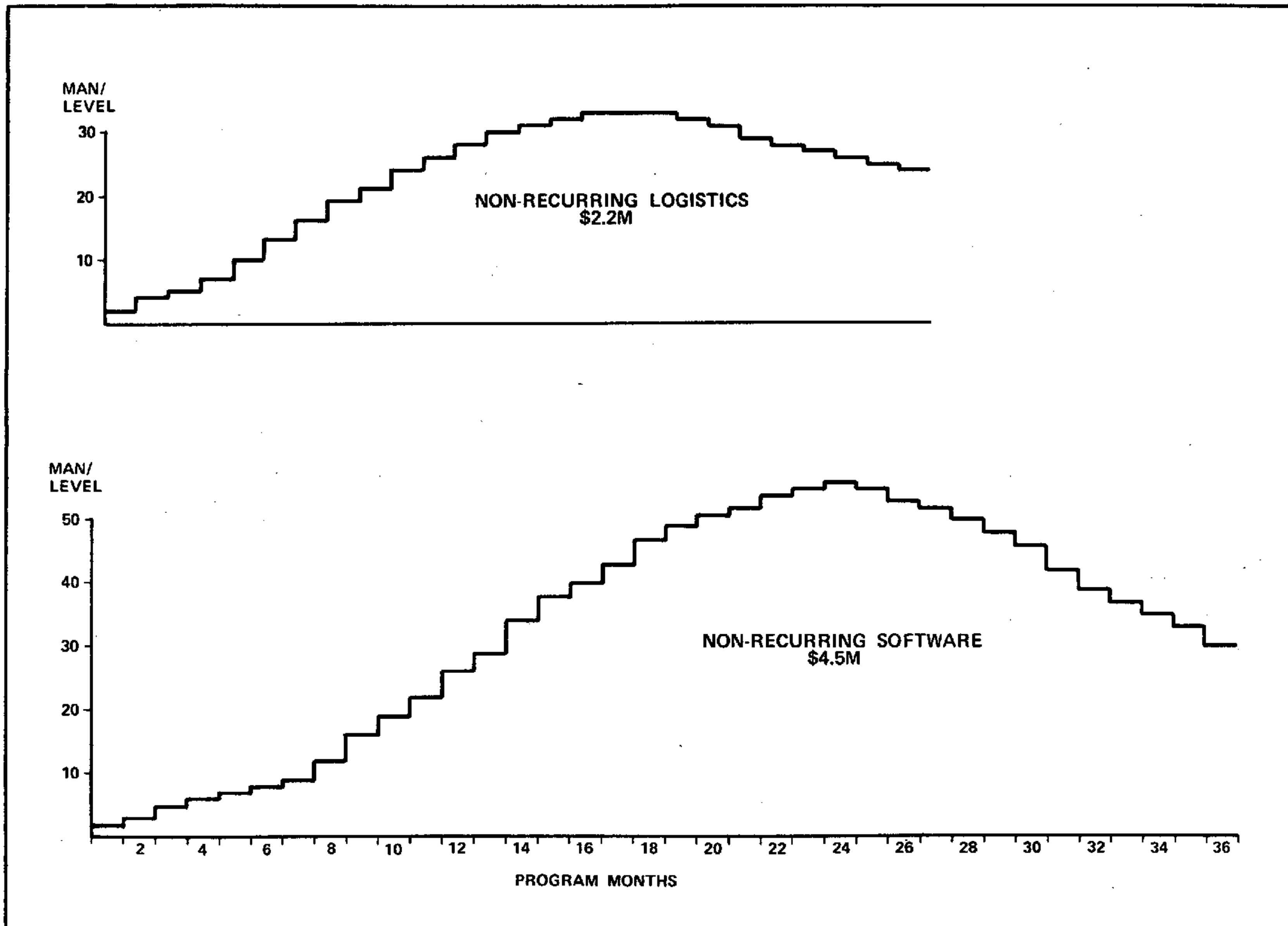


Figure 8-4. NON-RECURRING LOGISTICS, GSE, AND SOFTWARE COSTS

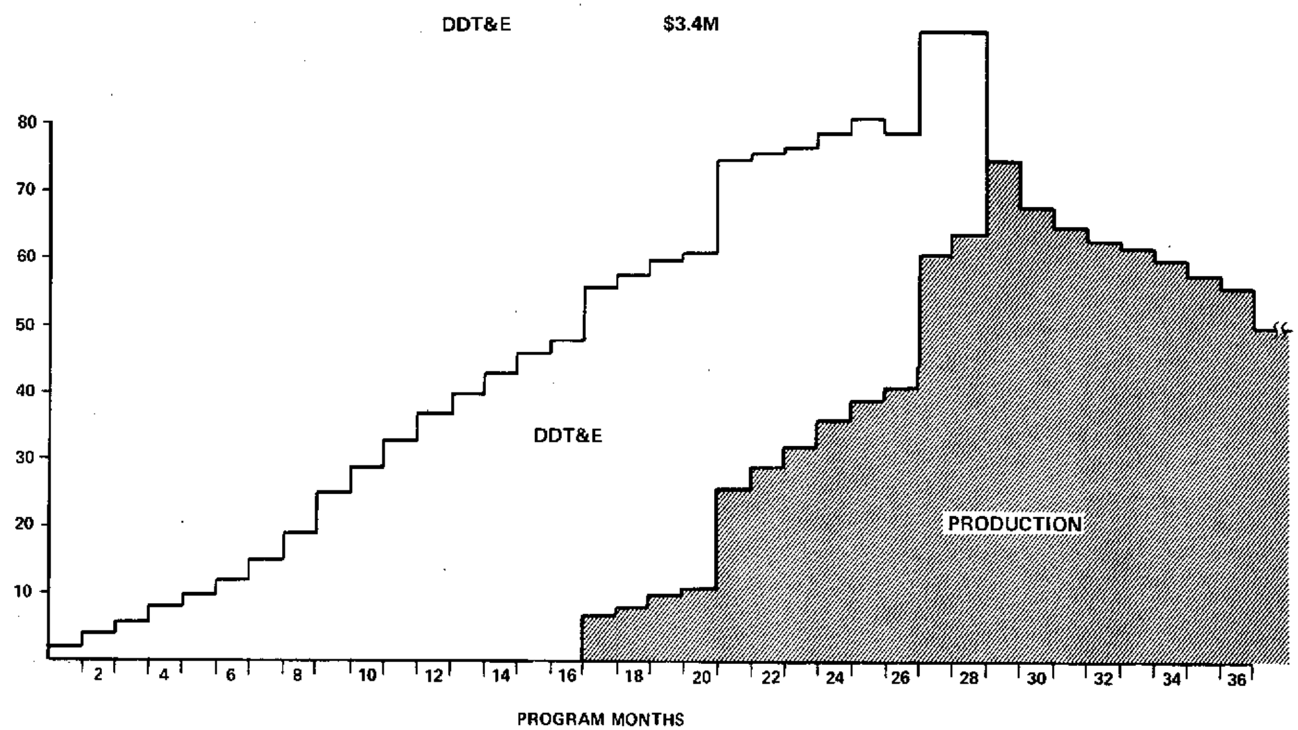


Figure 8-5. INSTALLATION AND TEST CHECKOUT, TOTAL MANLOAD BY MONTH

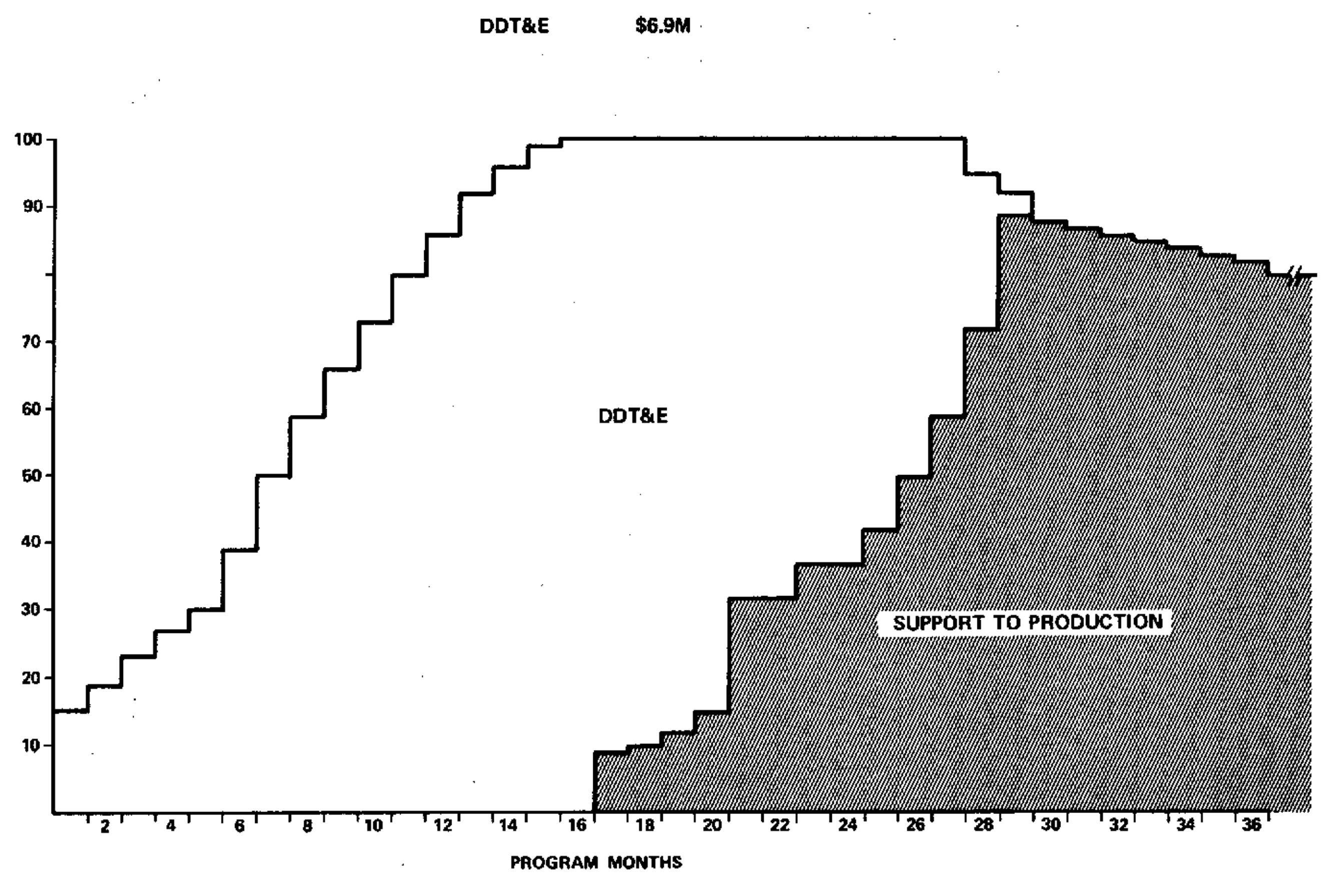


Figure 8-6. PROGRAM MANAGEMENT

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY

STUDY TITLE ____________
CONTRACT NO. ____________

COST DATA FORM – A(1)

NON-RECURRING (DDT&E)

DATE ____________
PAGE ____ OF ____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03	Stage	4	89.2				
This element includes item related to the design and development of the SEPS stage.							
Cost related to this element are non-reoccurring.							
Items in this element include all elements listed in Table 8-2 except the Δ for earth orbital							
functions.							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________
CONTRACT NO. ____________

COST DATA FORM – A(1)
NON-RECURRING (DDT&E)

DATE ____________
PAGE ____ OF ____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-02	Ion Propulsion	5	9.1				
Thruster support structure							
9 30 CM Hg thrusters							
Insulation							
Gimbal assemblies							
Actuators							
Hg storage tank and accessories							
Refueling receptacles							
Power processors							
Switch matrix							
Instrumentation							
Cabling							
Fittings and lines							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________
CONTRACT NO. ____________

COST DATA FORM — A(1)

NON-RECURRING (DDT&E)

DATE ____________
PAGE _____ OF _____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-01	Structural and Thermal	5	4.8				
	Control						
Primary structure							
Phased array antenna supports							
Tankage support							
Solar array inboard wing spars							
Solar array deploy retract							
Substructures							
Scanning platform moments							
Manipulator moments							
SEPS mounting structure							
Insulation blankets							
Radiator louvers							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________
CONTRACT NO. ____________

COST DATA FORM – A(1)

NON-RECURRING (DDT&E)

DATE ____________
PAGE ____ OF ____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-03	Energy Storage and Power	5	1.0				
	Distribution						
Solar array distribution panel							
Stage power distribution panel							
2 1000 W DC-DC converters							
4 batteries NiCAD							
2 regulators							
2 chargers							
Mounting and integration							
Wiring businesses for ES&P							
System only - Subsystem wiring businesses are parts of each subsystem							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________
CONTRACT NO. ____________

COST DATA FORM – A(1)
NON-RECURRING (DDT&E)

DATE ____________
PAGE ____ OF ____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-04	Solar Array	5	7.8				
Solar array wing							
Power take off connector							
Solar cells							
Deployment interface							
Wing deployment and retraction mechanisms							
2 sun sensors							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________

CONTRACT NO. ____________

COST DATA FORM – A(1)

NON-RECURRING (DDT&E)

DATE ____________

PAGE _____ OF _____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-05	Data Management	5	3.4				
Remote multiplexer, A/D converters, Signal conditioners							
Remote command unit							
Central computer (SOM-C)							
Data storage							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________

CONTRACT NO. ____________

COST DATA FORM – A(1)

NON-RECURRING (DDT&E)

DATE ____________

PAGE _____ OF _____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-06	Communication	5	2.2				
Antenna subsystems							
R. F. subsystems							
Command decoders and TM							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________

CONTRACT NO. ____________

COST DATA FORM – A(1)

NON-RECURRING (DDT&E)

DATE ____________

PAGE ____ OF ____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-07	Attitude Control	5	9.2				
	Navigation and Guidance						
ACS thrusters and accessories							
N_2H_4 supply system							
Thermal control							
N&G							
2 TV cameras (S.P.)							
2 Ladars							
2 Star trackers							
24 Sun sensors							
1 IMU							
2 TV cameras (on manipulators)							
Electronics							
2 Horizon sensors							
2 ILT							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ______________
CONTRACT NO. ______________

COST DATA FORM – A(1)

NON-RECURRING (DDT&E)

DATE ____________
PAGE _____ OF _____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-08	Integration and Test	5	6.7				
	Checkout						
This element contains the development test, engineering, and integration testing							
necessary to verify and flight qualify the flight test unit.							
This WBS element also includes the sustaining engineering testing associated with							
correction of any faults discovered in the earth orbital test of SEPS #1 and in-							
corporating the changes into the production item.							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________

CONTRACT NO. ____________

COST DATA FORM – A(1)
NON-RECURRING (DDT&E)

DATE ____________

PAGE _____ OF _____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-09	Test Hardware	5	21.3				
This element contains the cost of material, fabrication, reliability and quality							
assurance to produce the flight test unit. The cost of modifications during the							
qualification test period are included.							
Test hardware associated with modifications resulting from the EOT and support to							
production test is included.							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ______________
CONTRACT NO. ______________

COST DATA FORM – A(1)
NON-RECURRING (DDT&E)

DATE ______________
PAGE ______ OF ______

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-010	Stage GSE	5	5.0				
This element contains the cost of the engineering and production of 2 sets of							
manufacturing acceptance test equipment, one set will be used at the SEPSOC							
during launch preparation. Major items include:							
1. Test control console							
2. Computer terminal							
3. Air table to support solar arrays during 1g deployment test.							
The cost of handling equipment for the launch site is also included.							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ____________
CONTRACT NO. ____________

COST DATA FORM – A(1)

NON-RECURRING (DDT&E)

DATE ____________
PAGE ____ OF ____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-11	Software	5	4.5				

This element includes the development of the computer execution and operating system software. The cost of applications software to support the qualification test program is also included. The onboard N&G initial program sets are included.

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE ______________

CONTRACT NO. ______________

COST DATA FORM – A(1)

NON-RECURRING (DDT&E)

DATE ______________

PAGE _____ OF _____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-12	Logistics	5	0.5				
This element contains the analytical cost of identifying the Line Replaceable							
Units necessary to support the operational phase. It includes the hardware							
cost of repair parts for the qualification test program.							

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Continued)

STUDY TITLE __________
CONTRACT NO. __________

COST DATA FORM – A(1)
NON-RECURRING (DDT&E)

DATE __________
PAGE ____ OF ____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-12	SE&I	5	6.8				
This element includes all analytical tasks to define the SEPS system. The effort required to integrate related technical functions and interfaces to optimize the system design is included. The element includes the following detail tasks.							
	a. System/Subsystem definition and integration						
	b. System documentation						
	c. Safety analysis						
	d. Tug/Space shuttle interface definition						
	e. Payload interface						
	f. Maintainability Analysis						
	g. Reliability Analysis						
	h. Payload interface definition						

Table 8-5. DESIGN DEVELOPMENT COST SUMMARY (Concluded)

STUDY TITLE ______________
CONTRACT NO. ______________

COST DATA FORM – A(1)
NON-RECURRING (DDT&E)

DATE ______________
PAGE ____ OF ____

IDENTIFICATION NUMBER	WBS IDENTIFICATION	WBS LEVEL	EXPECT. COST	CONFID. RATING	T_d	T_s	SPREAD FUNCT.
A-03-13	Program Management	5	6.9				
This element covers program management for the DDT&E phase. It includes the following cost categories.							
	a. Engineering Administration						
	b. Business Management						
	c. Qualification Test Management						
	d. Configuration Management						
	e. Quality Assurance Management						

in three vehicles. The second spare will not be equipped with either kit in the production program. At the end of the run, the production contracts will be terminated. Cost of the hardware is shown in Table 8-6.

Table 8-6. FLIGHT ARTICLE PRODUCTION COST SUMMARY
(Costs shown are dollars in millions)

	FIRST UNIT COST	REQ'D QTY.	IMPROVEMENT CURVE FACTOR (%)	AVERAGE UNIT COST	TOTAL PRODUCTION COST
Core Vehicle	16.75	11	69.4	11.6	127.6
Planetary Peculiar	0.75	8	75.3	0.6	4.8
EO Peculiar	0.75	3	91.4	0.7	2.1
EO Functions	1.00	3	91.4	0.9	2.7
Tug P/L Shell & Diaphragms	0.80	2	94.9	0.75	1.5
					138.7
Stage Refurbishment and Spares for Operational Refurbishment					7.2
GPME (Tug P/L Shell and Diaphragms)					1.5

Planetary SEPS Average Cost

8 Core Vehicles @ 11.6 = 92.8
8 Planetary Peculiar @ 0.6 = 4.8
97.6 ÷ 8 = 12.2 Average Cost

Earth Orbital SEPS Average Cost

3 Core Vehicles @ 11.6 = 34.8
3 EO Peculiar @ 0.7 = 2.1
3 EO Functions @ 0.9 = 2.7
2 Tug P/L & Diaphragms @ 0.75= 1.5
41.1 ÷ 3 = 13.7 Average Cost

A breakout of these costs is presented in Table 8-7. These vehicle production costs are based on the estimated first unit costs and curves of percentage reduction in unit cost versus number of units produced. The curves are based on Northrop's experience with a wide range of electromechanical, electronic, and aircraft production programs.

Table 8-7. SEPS FIRST UNIT COST

	"CORE" SEPS	PLANETARY PECULIAR	EARTH ORBITAL PECULIAR
Structure & Thermal	1.20		
Propulsion	2.00		
Power Distribution	0.40		
Solar Array	5.80		
Data Management	1.00		
Communications	0.90	0.30	0.30
Reaction Control System	0.90	0.45	
Guidance & Navigation	0.65		0.45
Integration & Test Checkout	1.10		
System Engineering	1.40		
Program Management	1.40		
TOTALS	16.75	0.75	0.75

CORE SEPS	16.75
Planetary Peculiar	0.75
TOTAL Planetary SEPS	17.50

CORE SEPS	16.75
EO Peculiar	0.75
Subtotal	17.50
Add to EO Functions	1.00
TOTAL EO SEPS	18.50

8.6.2 Cost Improvement Curve

The recommended improvement curve represents a composite curve based on NSI cost experience in the areas of labor, material, installation, and test.

The SEPS first unit subsystem costs (NSI recommended first unit costs) were analyzed for material and labor content. These items along with integration and test checkout, were projected down the appropriate curve to obtain the weighted composite improvement curve.

The historical data used to prepare the individual labor, material, installation, and test checkout curves were gathered from the following Northrop programs:

- Polaris/Poseidon Missile Test and Readiness Eq. (Electronics)
- C-5 Navigation Systems (Electronics)
- TISEO (Target Identification Selection Evaluation Optics) (Electro-optical)
- Hawk Missile Loaders/Launchers/Missile Wings/Actuators (Mechanical)
- F5/T38 Aircraft (Airframe)

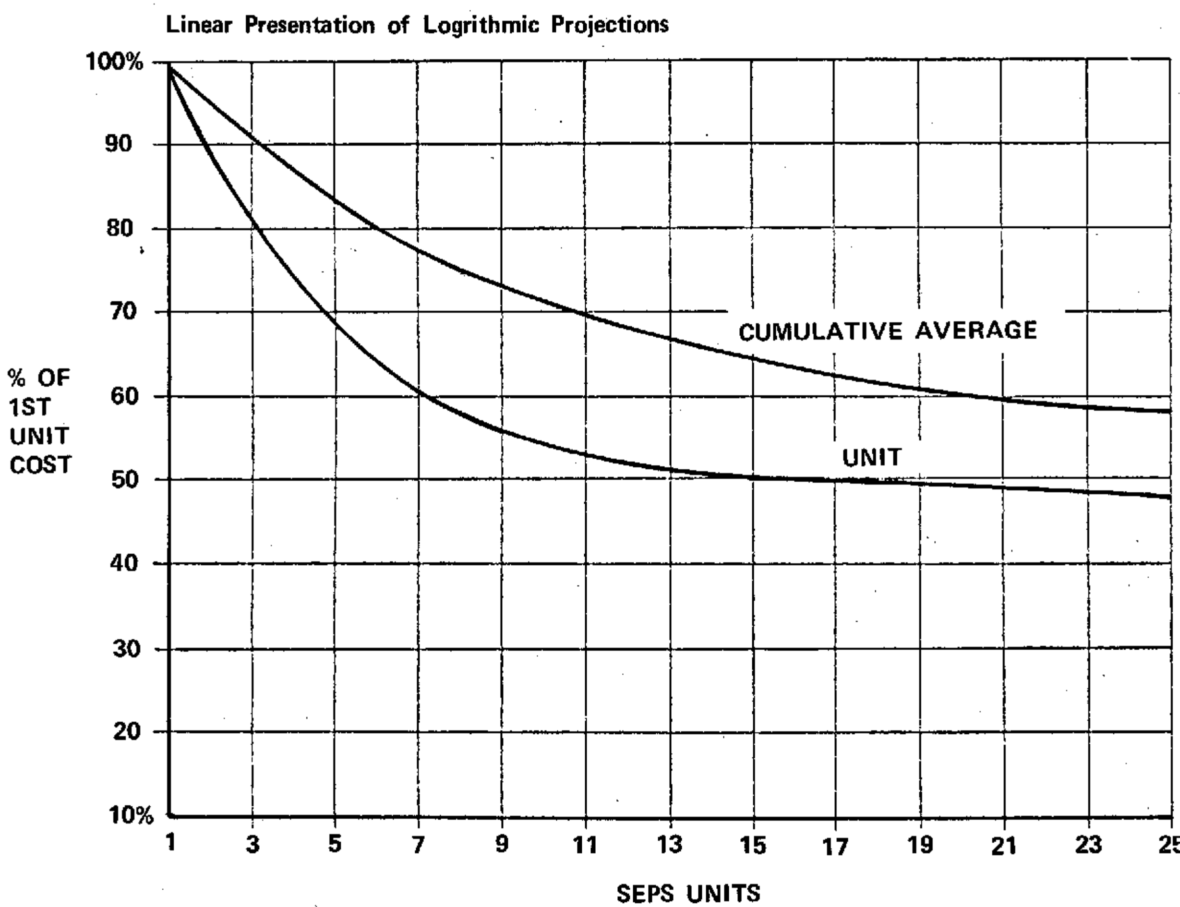
The above programs all demonstrated similar characteristics as the NSI recommended improvement curve with variations dependent upon labor and material mix.

The cost improvement curves are presented on Figures 8-7 through 8-10.

8.7 SEPS OPERATIONS CENTER INITIAL COSTS

Almost all of the SEPS operational phase functions will be accomplished at a single operations center. This includes launch preparation, flight control, refurbishment, and mission planning. The single exception is the integration of SEPS into a payload transport shell. This will occur at the launch site.

A SEPSOC is required. The basic building will provide space for each function at a cost of $0.7 million. The flight control equipment includes a



PROJECTIONS AS A PERCENT OF 1ST UNIT COST

	UNIT	CUM AVG
UNIT 1	100.0%	100.0%
2	89.8	94.9
3	84.5	91.4
4	75.3	87.4
5	69.0	83.7
10	53.7	71.1
15	50.7	64.6
20	49.3	60.9
25	48.2	58.4

Figure 8-7. SEPS RECURRING HARDWARE PROJECTIONS BASED ON CONTINUOUS RATE PRODUCTION

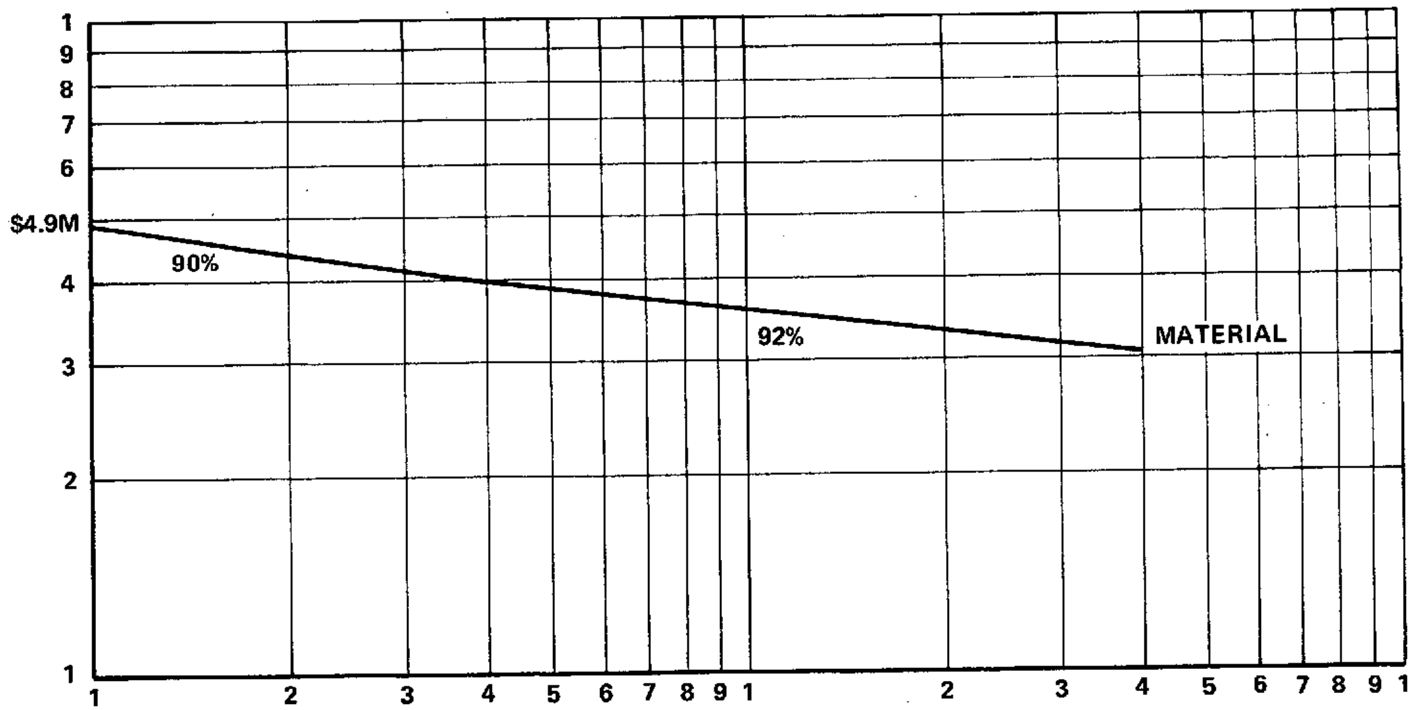


Figure 8-8. PROJECTED MATERIAL IMPROVEMENT CURVE

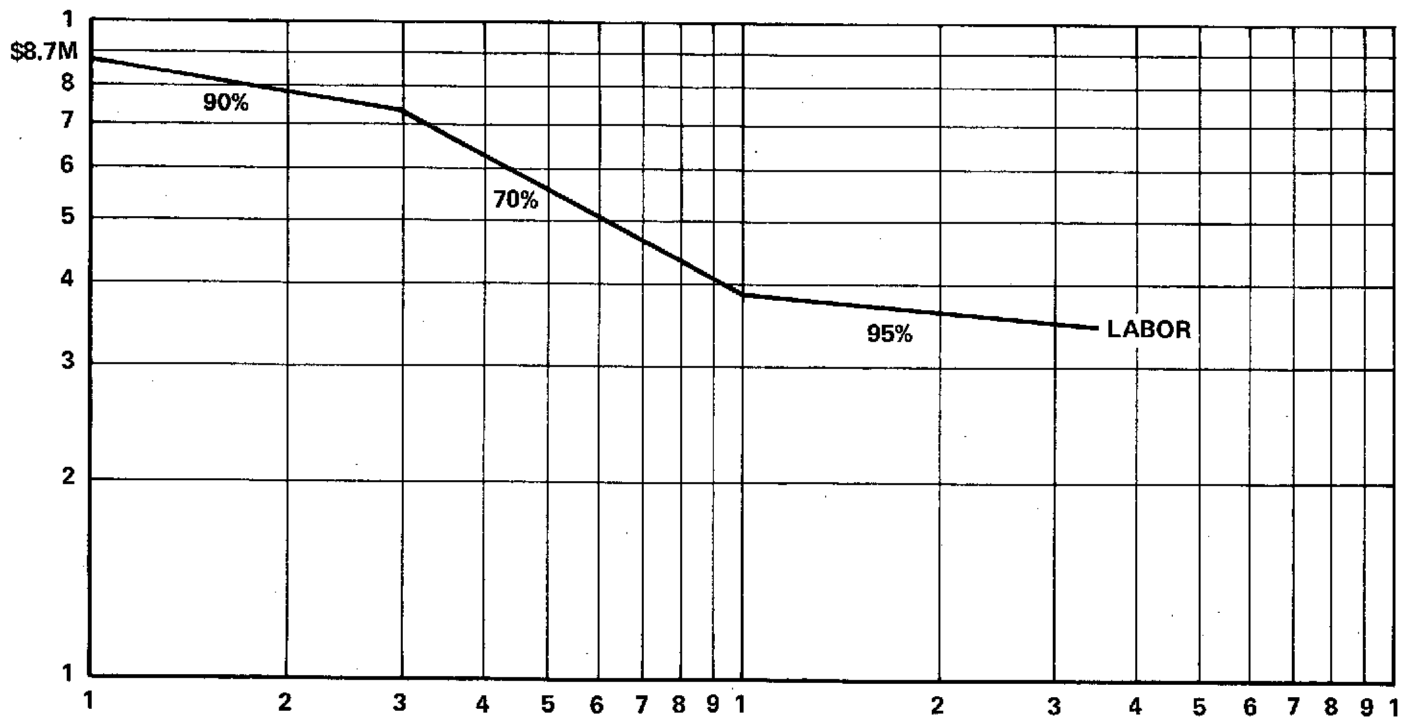


Figure 8-9. PROJECTED LABOR IMPROVEMENT CURVE

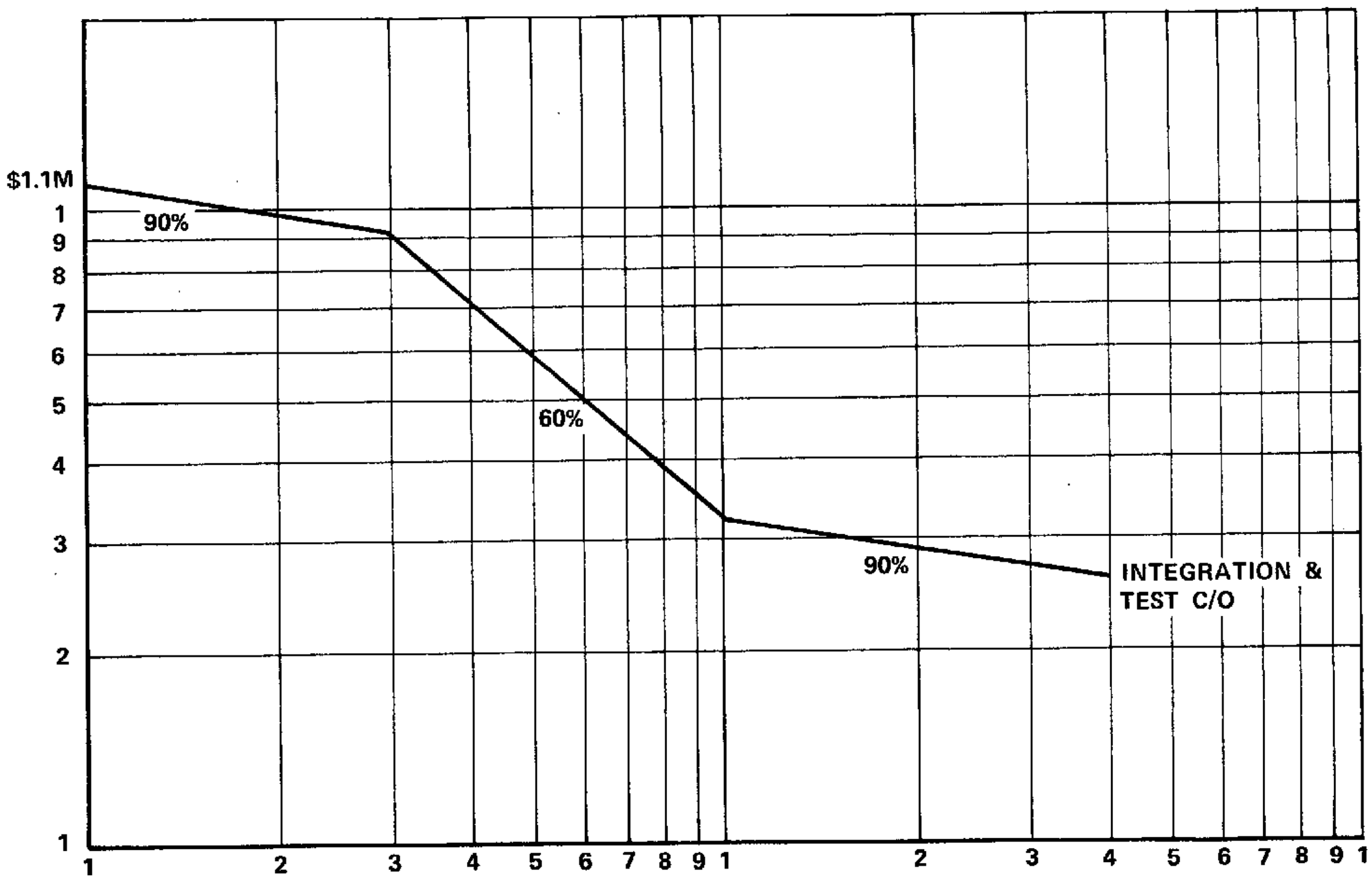


Figure 8-10. PROJECTED INTEGRATION AND TEST C/O IMPROVEMENT CURVE

computer at $5.3 million, and control consoles and displays at $1.1 million. Spare parts for the control consoles and displays will cost $0.4 million.

Application software, which will enable a small (45 man) group to accomplish all program functions, will cost $8.7 million.

The total SEPSOC costs are $17.9 million. A breakdown is presented in Table 8-8.

Table 8-8. SEPS OPERATIONS CENTER COSTS

GSE SYSTEMS ENGINEERING	$1.7M
EQUIPMENT (COMPUTER 5.3M + CONSOLES AND DISPLAYS 1.1M)	6.4M
FACILITY	0.7M
SOFTWARE (OPERATIONS)	8.7M
SEPSOC SPARE PARTS	0.4M
	$17.9M

The mission model requires the use of earth orbital SEPS in 30 sorties over an 11-year period. In this same period, 12 launches are required - 8 planetary and 4 earth orbital. In addition, the flight test article and one earth orbital SEPS must be refurbished.

Under the NSI operations concept, a 45-man organization can accomplish all functions except computer operations. This organization will cost $23.7 million over 11 years. Table 8-9 shows the portions allocated to each function.

Table 8-9. SEPS PERSONNEL ALLOCATION

FUNCTION	% OF TOTAL	PLANETARY	EARTH ORBITAL
LAUNCH PREPARATION	12.0	8.0 %	4.0 %
PROGRAM MANAGEMENT	10.0	2.5 %	7.5 %
REFURBISHMENT	12.0	-	12.0 %
FLIGHT OPERATIONS	8.0	-	8.0 %
PLANNING	58.0	14.5 %	43.5 %
	100.0 %	25.0 %	75.0 %
PERSONNEL COST	$ 23,700,000	$ 5.9M	$ 17.8M

The SEPS program will purchase the computer for use by the host NASA center. SEPS operations will then purchase $2.1 million in computer operations support from the host center.

The flight units will consume $0.4 million for mercury and hydrazine in the accomplishment of flight missions.

A breakdown of the $26.2 million operations cost is presented in Table 8-10.

Table 8-10. SYSTEM OPERATIONS COST

PERSONNEL	
45 MAN OPERATIONS ORGANIZATION FOR 11 YEARS AT $48K per man year	23.7 Million
COMPUTER OPERATIONS (11 YEARS)	2.1 Million
FLIGHT ARTICLE CONSUMABLES (MERCURY 5400 POUNDS + 750 POUNDS N_2H_4)	0.4 Million
	$26.2 Million

8.8 COST EFFECTIVENESS OF EARTH ORBITAL SEPS

A planetary only SEPS program is estimated to cost $232 million. The recommended planetary plus earth orbital SEPS program will cost an additional $58 million. Its use will result in a gross transportation cost savings of $184 million. This is the result of reducing the number of Shuttle flights by 15 and saving $18 million in STS hardware costs.

The addition of the earth orbital SEPS is, therefore, cost effective (Table 8-11), with a net savings of $126 million. This $126 million represents a 217 percent return on the investment in an earth orbital program.

Table 8-11. COST EFFECTIVENESS SUMMARY (DOLLARS IN MILLIONS)

NET COST OF EO SEPS	58
COST PER SORTIE (29 SORTIES)	2
NET SAVINGS OF STS WITH EO SEPS VERSUS STS WITH PLANETARY SEPS ONLY	126
RETURN ON NET COST	217%

Table 8-12 compares the total STS costs with and without the earth orbital SEPS. Table 8-13 depicts the allocation of SEPS program costs between the planetary and earth orbital SEPS. The earth orbital deltas are for additional hardware, software, and personnel to accomplish payload handling functions.

Table 8-12. STS COMPARED TO STS WITH SEPS FOR TRANSPORTATION COST EFFECTIVENESS - EARTH ORBITAL FLIGHTS REQUIRING UPPER STAGES

COST ELEMENT (DOLLARS IN MILLIONS)	BLSTS		BLSTS BLSEPS (20 KHR-REFUELED)	
	$10^6$$	NUMBER	$10^6$$	NUMBER
SHUTTLE FLIGHTS @ $11.09	1508.	136	1342.	121
IUS EXPENDED @ $5.17	103.	20	98.	19
IUS WITH KICK STAGE @ $6.37	13.	2	13.	2
TUG RECOVERED FLTS @ $.96	87.	91	74.	77
TUG RECOVERED EXPENDED KS @ $2.16	15.	7	15.	7
TUG EXPENDED @ $14.16	0.	0	0.	0
TUG AND KS EXPENDED @ $15.36	92.	6	92.	6
TOTAL TRANSPORTATION COST	1818.		1634.	
$ SAVED IN TRANSPORT COST	--		184.	
VEHICLE INVENTORY COST SEPS @ (VARIES WITH PRODUCTION)	110.	9*	146.	11**
SEPS DEVELOPMENT & OPERATIONS	122.		144.	
TOTAL SYSTEM COST	2050.		1924.	
NET $ SAVED	--		126.	

*8 PLANETARY VEHICLES PLUS ONE SPARE

**8 PLANETARY VEHICLES PLUS ONE SPARE PLUS TWO EARTH ORBITAL VEHICLES

Table 8-13. ALLOCATION OF PROGRAM COSTS

	PLANETARY	ΔEO	TOTAL
DDT&E			100
PLANETARY STAGE	89		
EO PAYLOAD SYSTEMS		11	
PRODUCTION			146
9 PLANETARY UNITS	110		
2 EO UNITS PLUS SPARES AND GPME		36	
OPERATIONS			44
START UP	15		
SOFTWARE Δ		3	
PERSONNEL	18	8	
	232	58	290

Cost effectiveness is based upon comparison of the cost required to accomplish the reference mission model (which contains a planetary SEPS program) with the baseline Space Transportation System without an Earth Orbital SEPS to the cost required if the program described in this document were implemented.

8.9 DDT&E AND PRODUCTION PROGRAM COMPARISON OF A 25 kw SEPS TO A 50 kw SEPS

A cost of a DDT&E and Production Program for a 25 kw SEPS is compared to one for a 50 kw SEPS in Table 8-14. This comparison covers the system from DDT&E through first unit production costs.

It is estimated that the costs of production will follow the "cost improvement curves" in subsection 8.6. The operations costs will not change significantly.

Table 8-14. COMPARISON OF 25kw TO 50kw BASIC COSTS (SEPS DEVELOPMENT AND 1ST UNIT COSTS)

(Dollars in Millions)

COST ELEMENT	DEVELOPMENT		FIRST UNIT COST	
	25 kw	Δ FOR 50 kw	25 kw	Δ FOR 50 kw
STRUCTURES & THERMAL CONTROL	$ 4.8		$ 1.2	0.1
PROPULSION	9.1		2.0	0.8
POWER DISTRIBUTION	1.0		0.4	
SOLAR ARRAY	7.8		5.8	6.1
DATA MANAGEMENT	3.4		1.0	
COMMUNICATION	2.2		1.2	
ATTITUDE CONTROL/N&G	9.2		2.0	0.2
INTEGRATION & TEST CHECKOUT	6.7	1.0	1.1	1.0
TEST HARDWARE	21.3	6.5		
GSE	5.0			
SOFTWARE	4.5			
LOGISTICS	0.5			
SE&I	6.8		1.4	
PROGRAM MANAGEMENT	6.9		1.4	
BASIC SEPS	$89.2	Δ7.5	$17.5	Δ8.2
Δ FOR EARTH ORBITAL FUNCTIONS	8.3		1.0	
	97.5		18.5	
Δ FOR TUG PAYLOAD SHELL AND DIAPHRAGMS	2.5		0.8	
	$100.0	Δ% 7.5	$ 19.3	Δ% 42

Review of Table 8-14 shows that no DDT&E costs of the 50 kw system are different from those of the 25 kw system except in the areas of integration and test checkout and in the costs of test hardware. The rationale for the assessment is simple. Except for the areas of the deployed solar wing, SEPS at 25 kw and SEPS at 50 kw are so similar in size that they can use identical facilities, similar handling transport, and so forth.

The same number of engineers design, test, manage, and so forth, the various aspects of each program. The biggest individual component of either vehicle can be held by one man. The numbers of components required by a 25 kw or a 50 kw SEPS are the same (except for solar cells); therefore, only material costs would be expected to be different for DDT&E.

Production costs are greater primarily in the solar array subsystem because individual cells are expensive, and twice as many are required for a 50 kw system as for a 25 kw system.

Appendix A

SOLAR PANEL GIMBAL LOCK

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In the SEPS vehicle, the solar panels must be directed toward the sun, while the engines must be directed along some direction determined by the navigation control requirements. These two conditions, which must be satisfied simultaneously, determine the attitude of the vehicle. When the commanded thrust vector passes close to the solar vector, high angular accelerations are called for, which can lead to excessive torque commanded to the attitude control system. The geometry is shown on Figure A-1. The coordinate system X Y Z is an inertially fixed system with the Z-axis directed toward the sun (motion of the sun is ignored).

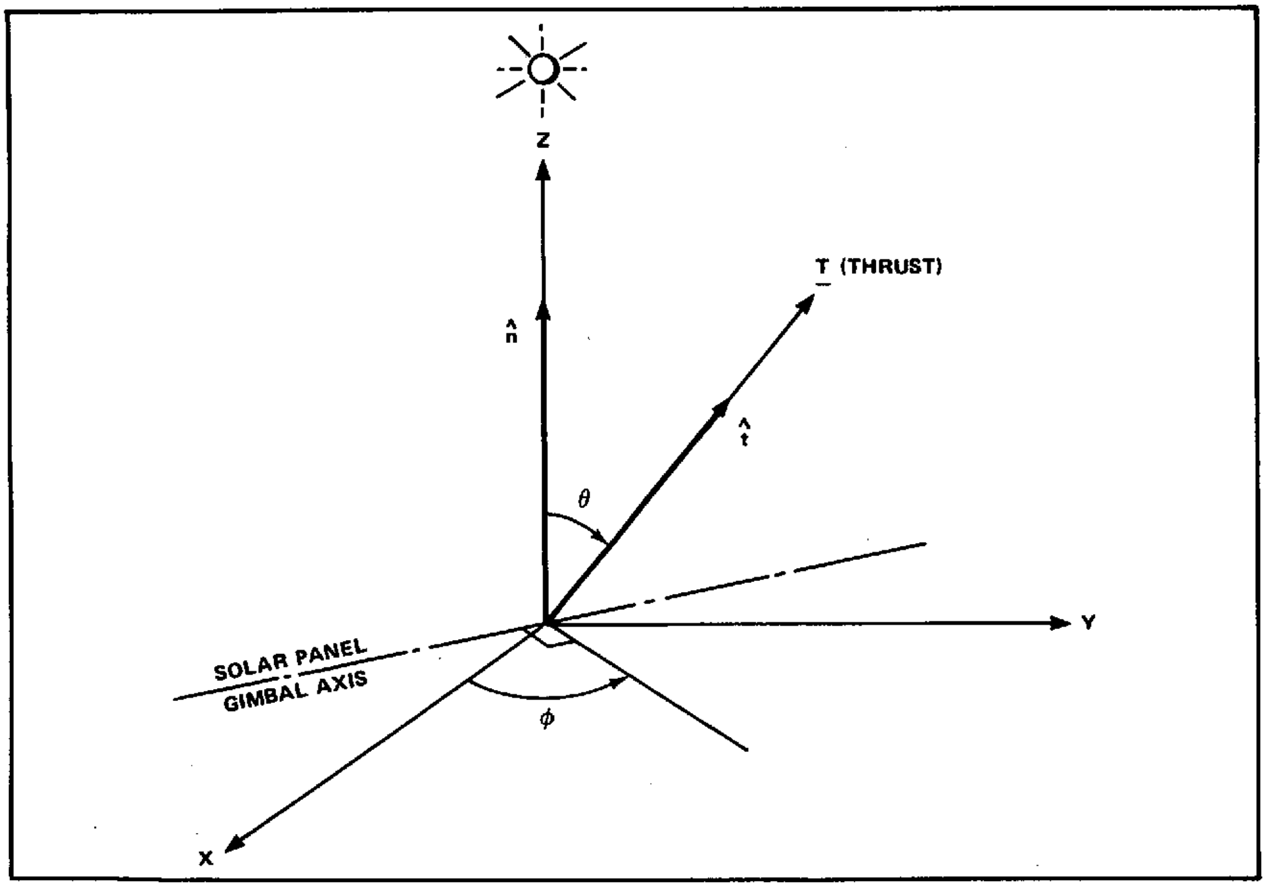

Figure A-1. SEPS GEOMETRY

On Figure A-1, $\hat{n}$ is a unit vector normal to the solar panels; and $\hat{t}$ is a vector along the thrust axis. The angles are given by:

$$\tan \phi = \frac{t_Y}{t_X} \tag{A-1}$$

$$\cos \theta = t_Z \tag{A-2}$$

or

$$\sin \theta = \sqrt{t_X{}^2 + t_Y{}^2} \quad . \tag{A-3}$$

Now, consider the manner in which $\hat{t}$ changes during a slewing maneuver. This geometry is shown on Figure A-2.

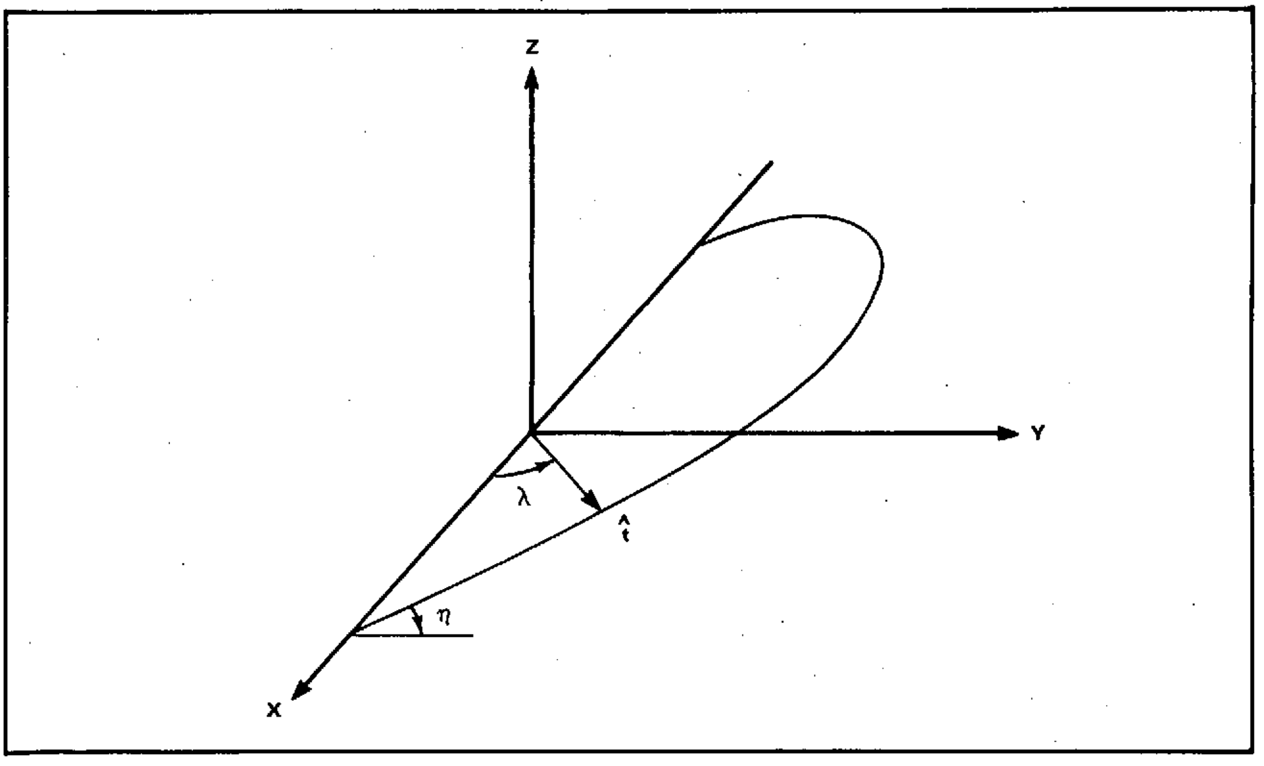

Figure A-2. THRUST VECTOR GEOMETRY

It is assumed that $\hat{t}$ moves along a great circle, in the plane of inclination η, shown on Figure A-2. Since the direction of the X-axis has not yet been defined, there is no loss of generality in assuming it to be along the line of nodes between the X-Y plane and the plane of $\hat{t}$.

The components of $\hat{t}$ are given by

$$t_X = \cos\lambda$$

$$t_Y = \sin\lambda\cos\eta \tag{A-4}$$

$$t_Z = \sin\lambda\sin\eta\quad .$$

Substituting these relations into Eqs. (A-1), (A-2), and (A-3), obtain

$$\tan\phi = \tan\lambda\cos\eta \tag{A-5}$$

$$\cos\theta = \sin\lambda\sin\eta \tag{A-6}$$

or $$\sin\theta = \sin\lambda\sqrt{\cos^2\eta + \cos^2\lambda}\quad . \tag{A-7}$$

Note that the maximum value of θ is given by

$$\sin\theta_m = \cos\eta.$$

Note also that when $\eta = 90°$, $\theta_m = 0$ and ϕ becomes indeterminate. This is the "gimbal lock" phenomenon.

Now, suppose $\hat{t}$ is moving in its plane at a constant rate $\dot{\lambda}$, and η is constant. Consider the derivatives of θ and ϕ. From Eq. (A-6), obtain

$$-\sin\theta\,\dot{\theta} = \cos\lambda\sin\eta\,\dot{\lambda}$$

$$\dot{\theta} = -\frac{\cos\lambda\sin\eta}{\sin\theta}\dot{\lambda}\quad . \tag{A-9}$$

Eq. (A-5) gives

$$\sec^2\phi\,\dot{\phi} = \cos\eta\sec^2\lambda\,\dot{\lambda}\quad .$$

But $$\sec^2\phi = \frac{1}{\cos^2\phi} = 1 + \tan^2\phi\quad . \tag{A-10}$$

Hence, one has

$$(1 + \tan^2\phi)\dot{\phi} = \cos\eta\sec^2\lambda\,\dot{\lambda}$$

$$(1 + \tan^2\lambda\cos^2\eta)\dot{\phi} = \cos\eta\sec^2\lambda\,\dot{\lambda}$$

$$\dot{\phi} = \frac{\cos\eta\,\dot{\lambda}}{\cos^2\lambda\,(1 + \tan^2\lambda\,\cos^2\eta)}$$

$$= \frac{\cos\eta\,\dot{\lambda}}{\cos^2\lambda + \sin^2\lambda\,\cos^2\eta}$$

$$= \frac{\cos\eta\,\dot{\lambda}}{\cos^2\lambda + (1 - \cos^2\lambda)\,\cos^2\eta}$$

$$= \frac{\cos\eta\,\dot{\lambda}}{\cos^2\lambda(1 - \cos^2\eta) + \cos^2\eta}$$

$$\dot{\phi} = \frac{\cos\eta\,\dot{\lambda}}{\cos^2\lambda\,\sin^2\eta + \cos^2\eta} \quad .$$

Using Eq. (A-8), one may write

$$\dot{\phi} = \frac{\sin\theta_m\,\dot{\lambda}}{\sin^2\theta_n + \cos^2\theta_m\,\cos^2\lambda} \quad . \tag{A-11}$$

Consider the maximum value of this rate. It is maximum when the denominator is minimum, that is, when

$$\frac{\partial}{\partial\lambda}\left[\sin^2\theta_m + \cos^2\theta_m\,\cos^2\lambda\right] = 0 \; .$$

Differentiating gives

$$-\,2\cos^2\theta_m\,\sin\lambda\,\cos\lambda = 0$$

$$\sin\lambda\,\cos\lambda = 0$$

or

$$\lambda = \frac{n\pi}{2} \quad . \tag{A-12}$$

The roots for n even correspond to minimums in $\dot{\phi}$. Those for n odd give the desired maximums. Setting $\lambda = 90°$ in Eq. (A-11), obtain

$$\dot{\phi}_{max} = \frac{\sin\theta_m\,\dot{\lambda}}{\sin^2\theta_m}$$

or

$$\dot{\phi}_{max} = \frac{\dot{\lambda}}{\sin\theta_m} \quad . \tag{A-13}$$

These results indicate that the maximum rotation rate about the solar vector is related to $\dot{\lambda}$ by a multiplication factor csc θ_m. When $\theta_m = 0$ (that is, the system passes through "gimbal lock"), $\dot{\phi}_{max}$ is infinite. In Table A-1, the value for this factor is given as a function of θ_m.

Table A-1. RATE MULTIPLICATION FACTOR

θ_m	csc θ_m
90°	1.
60°	1.155
45°	1.414
30°	2.000
20°	2.924
10°	5.759
5°	11.47
1°	57.30

A factor perhaps more important than the rate $\dot{\phi}$ is the corresponding acceleration $\ddot{\phi}$, since this is directly related to control torques. Returning to Eq. (A-11), it may be seen that Eq. (A-14) may be written

$$\dot{\phi} = \frac{k}{f(\lambda)} \tag{A-14}$$

where $k = \sin \theta_m \dot{\lambda}$

$$f(\lambda) = \sin^2 \theta_m + \cos^2 \theta_m \cos^2 \lambda \ . \tag{A-15}$$

Differentiating Eq. (A-14) gives

$$\ddot{\phi} = - \frac{k\, f' \dot{\lambda}}{f^2} \tag{A-16}$$

where $f' = \dfrac{\partial f(\lambda)}{\partial \lambda}$. (A-17)

The maximum acceleration occurs when

$$\frac{\partial}{\partial\lambda}(\ddot{\phi}) = - k \frac{\partial}{\partial\lambda} (f'/f^2) = 0$$

which gives

$$\frac{f''}{f^2} - 2 \frac{(f')^2}{f^3} = 0$$

or

$$f\ f'' - 2\ (f')^2 = 0\ . \tag{A-18}$$

The derivatives are:

$$f' = -2 \sin\lambda \cos\lambda \cos^2\theta_m$$

$$f'' = -2(\cos^2\lambda - \sin^2\lambda)\cos^2\theta_m\ . \tag{A-19}$$

Eq. (A-18) becomes

$$-2(\sin^2\theta_m + \cos^2\theta_m \cos^2\lambda_m)(\cos^2\lambda_m - \sin^2\lambda_m) \cos^2\theta_m$$

$$- 2[-2 \sin\lambda_m \cos\lambda_m \cos^2\theta_m]^2 = 0$$

$$(\sin^2\theta_m + \cos^2\theta_m \cos^2\lambda_m)(\cos^2\lambda_m - \sin^2\lambda_m)$$

$$+ 4 \sin^2\lambda_m \cos^2\lambda_m \cos^2\theta_m = 0$$

$$(\sin^2\theta_m + \cos^2\theta_m \cos^2\lambda_m)(\cos^2\lambda_m - \sin^2\lambda_m)$$

$$+ 4 \sin^2\lambda_m \cos^2\lambda_m \cos^2\theta_m = 0\ . \tag{A-20}$$

This may be rewritten

$$(\sin^2 \theta_m + \cos^2 \theta_m \cos^2 \lambda_m) \; [2 \cos^2 \lambda_m - 1]$$

$$+ 4(1 - \cos^2 \lambda_m) \cos^2 \lambda_m \cos^2 \theta_m = 0$$

$$(2 \cos^2 \theta_m - 4 \cos^2 \theta_m)\cos^4 \lambda_m + (2 \sin^2 \theta_m - \cos^2 \theta_m$$

$$+ 4 \cos^2 \theta_m)\cos^2 \lambda_m - \sin^2 \theta_m = 0$$

$$- 2 \cos^2 \theta_m \cos^4 \lambda_m + (2 \sin^2 \theta_m + 3 \cos^2 \theta_m)\cos^2 \lambda_m - \sin^2 \theta_m = 0$$

$$2 \cos^2 \theta_m \cos^4 \lambda_m - (2 + \cos^2 \theta_m)\cos^2 \lambda_m + \sin^2 \theta_m = 0 \; . \tag{A-21}$$

This is a quadratic in $\cos^2 \lambda_m$, with

$$\begin{aligned} a &= 2 \cos^2 \theta_m \\ b &= -(2 + \cos^2 \theta_m) \\ c &= \sin^2 \theta_m \; . \end{aligned} \tag{A-22}$$

Using these, one obtains

$$\begin{aligned} b^2 - 4\,a\,c &= (2 + \cos^2 \theta_m)^2 - 8 \cos^2 \theta_m \sin^2 \theta_m \\ &= (4 + 4 \cos^2 \theta_m + \cos^4 \theta_m) - 8 \cos^2 \theta_m (1 - \cos^2 \theta_m) \\ &= 4 + 4 \cos^2 \theta_m + \cos^4 \theta_m - 8 \cos^2 \theta_m + 8 \cos^4 \theta_m \\ &= 9 \cos^4 \theta_m - 4 \cos^2 \theta_m + 4 \end{aligned}$$

$$b^2 - 4\,a\,c = 9 \cos^4 \theta_m + 4 \sin^2 0_m \; . \tag{A-23}$$

Then one obtains

$$\cos^2 \lambda_m = \frac{2 + \cos^2 \theta_m \pm \sqrt{9 \cos^4 \theta_m + 4 \sin^2 \theta_m}}{4 \cos^2 \theta_m} .$$

Factoring the term under the radical gives

$$9 \cos^4 \theta_m + 4 \sin^2 \theta_m = 9 \cos^4 \theta_m \left[1 + \frac{4}{9} \frac{\sin^2 \theta_m}{\cos^4 \theta_m} \right]$$

$$= (3 \cos^2 \theta_m)^2 \left[1 + \left(\frac{2}{3} \tan \theta_m \sec \theta_m \right)^2 \right] .$$

then

$$\cos^2 \lambda_m = \frac{2 + \cos^2 \theta_m \pm 3 \cos^2 \theta_m \sqrt{1 + \left(\frac{2}{3} \tan \theta_m \sec \theta_m \right)^2}}{4 \cos^2 \theta_m}$$

or

$$\cos^2 \lambda_m = \frac{\frac{1}{4} \left[2 \sec^2 \theta_m + 1 \pm 3 \sqrt{1 + \left(\frac{2}{3} \tan \theta_m \sec \theta_m \right)^2} \right]}{4 \cos^2 \theta_m} . \qquad \text{(A-24)}$$

The negative sign must be used for the root, since the positive one leads to

$$\cos^2 \lambda_m \geq \frac{1}{4}[4 + 2 \sec^2 \theta_m] > 1 . \qquad \text{(A-25)}$$

Having obtained λ_m from Eq. (A-24), one can determine $f(\lambda_m)$ and $f'(\lambda_m)$ from Eqs. (A-15) and (A-19). This then gives $\ddot{\phi}_{max}$ using Eq. (A-16). Because of the complexity of Eq. (A-24), it is not practical to seek an explicit formula for $\ddot{\phi}_{max}$. A tabular result must suffice.

In Table A-2, the value of $\ddot{\phi}_{max}$ is given as a function of θ_m. Note that for $\theta_m < 40°$, $\ddot{\phi}_m/(\dot{\lambda})^2$ is greater than unity, and becomes large very rapidly as θ_m decreases. For $\theta_m = 5°$, the multiplication factor is 85. A graph of this function is given on Figure A-3.

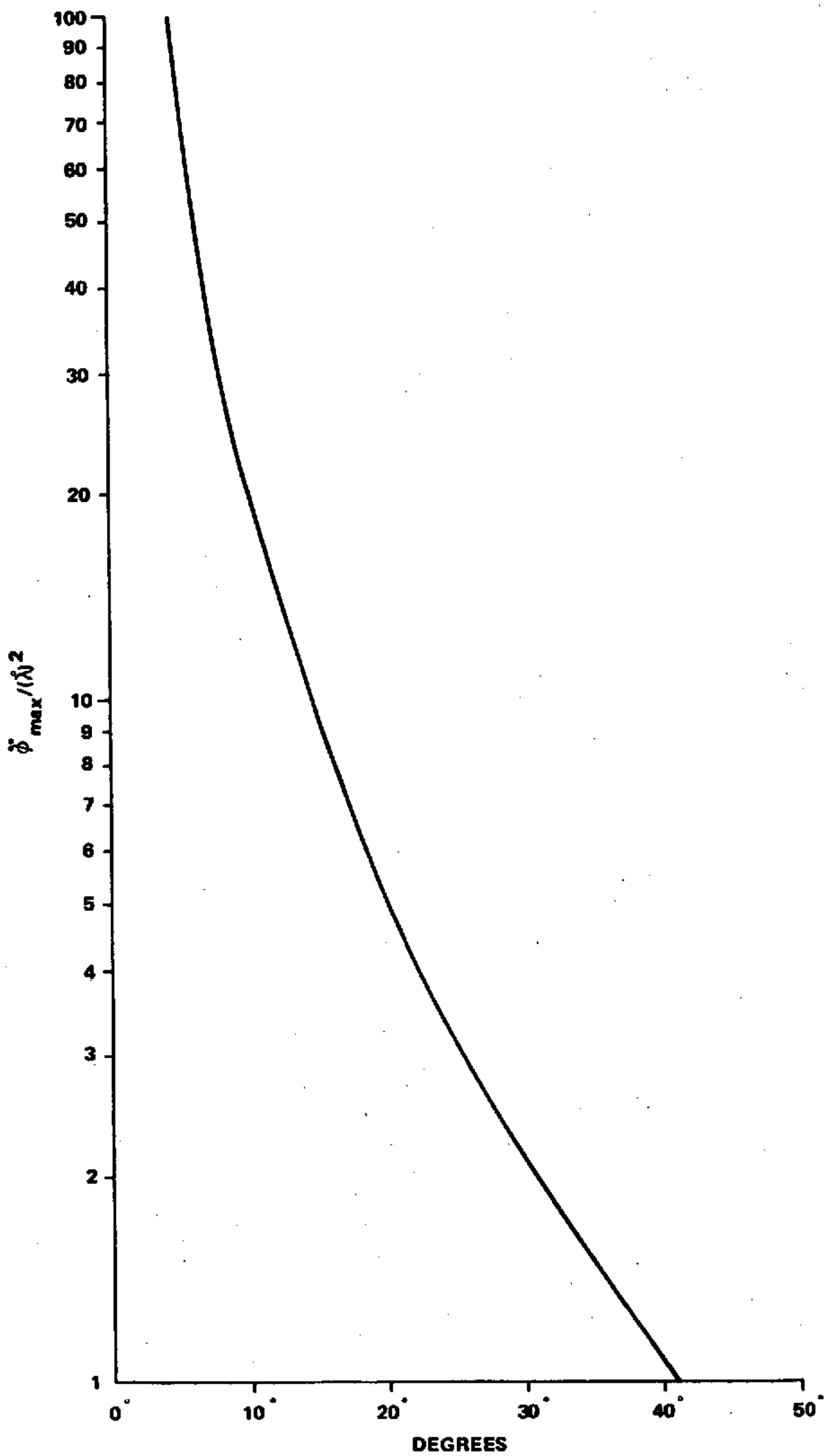

Figure A-3. ACCELERATION MULTIPLICATION FACTOR

Table A-2. ACCELERATION MULTIPLICATION FACTOR

θ_m	$\ddot{\phi}_{max}/(\dot{\lambda})^2$
90°	0
60°	0.2946
45°	0.7872
30°	2.1313
20°	5.102
10°	21.15
5°	85.2
1°	1060.

The power input to the solar panels is proportional to the cosine of the error angle. Values for this angle and the corresponding acceleration factor are tabulated in Table A-3 for various power limits.

Table A-3. POWER LOSS AND ACCELERATION MULTIPLICATION

POWER AVAILABLE (Percent)	θ_e (Degrees)	$\ddot{\phi}/\dot{\lambda}^2$
50	60	0.2946
60	53.13	
70	45.57	
80	36.87	1.38
85	31.79	1.88
90	25.84	2.93
95	18.19	6.25
97	14.07	11.1
99	8.11	29.5

Note that values of power available in the range of 80 to 90 percent appear reasonable, yielding acceleration factors on the order of two.

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Appendix B

TERMINAL RENDEZVOUS ANALYSIS

Appendix B

TERMINAL RENDEZVOUS ANALYSIS

Consider the relative coordinate system defined as on Figure B-1, and centered at the location of the target satellite (in a circular orbit).

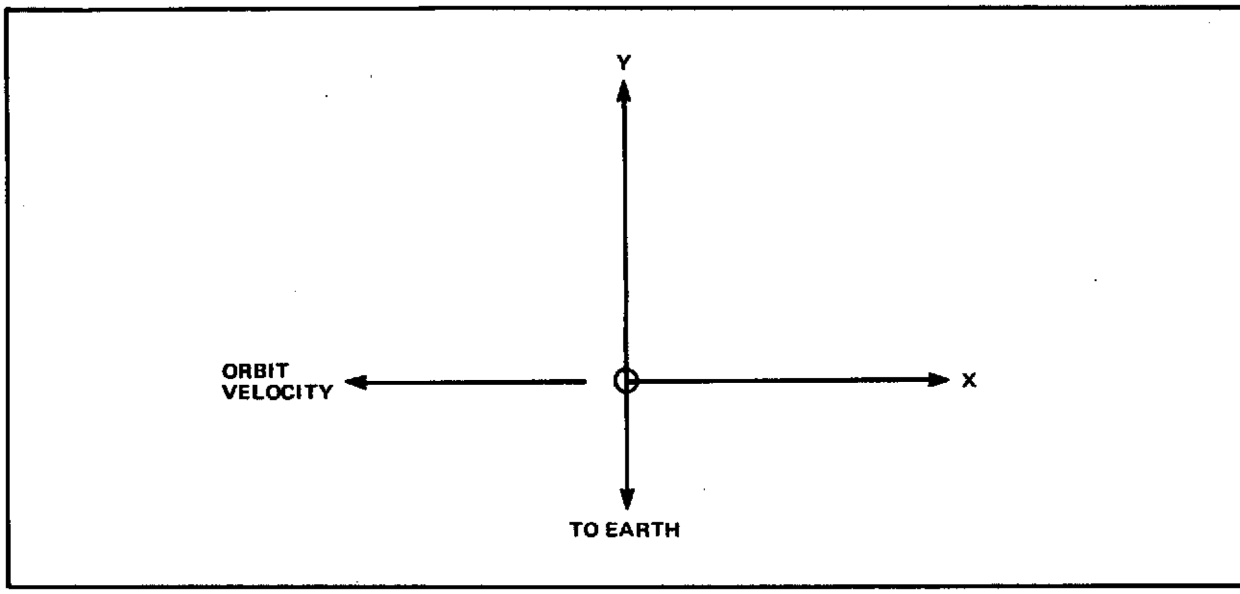

Figure B-1. RELATIVE COORDINATE SYSTEM

The linearized equations of motion for a vehicle in this coordinate system are:

$$\ddot{x} = a_x + 2\,n\,\dot{y}$$
$$\ddot{y} = a_y - 2\,n\,\dot{x} + 3\,n^2\,y \qquad \text{(B-1)}$$

where

n = mean motion of target satellite

and

a_x, a_y = external accelerations applied to vehicle $\underline{a} = \underline{F}/m$.

The particular problem of interest is that of terminating a continuous orbit-raising process by rendezvous from below. This problem is mathematically equivalent to that of departing the target satellite for the earth, as can be seen by changing the signs of x, a_x, and t in Eqs. (B-1). This latter problem is more convenient to investigate since the vehicle can be initialized to zero positions and rates.

In a continuous orbit-raising (lowering) process, the vehicle thrusts horizontally in the negative (positive) x direction. The motion is obtained by setting

$$\begin{aligned} a_x &= a = \text{constant} \\ a_y &= 0 \end{aligned} \tag{B-2}$$

in Eqs. (B-1). A particular solution is

$$\begin{aligned} \ddot{x} &= -3a \\ \ddot{y} &= 0 \end{aligned} \tag{B-3}$$

with first integral

$$\begin{aligned} \dot{x} &= -3at \\ \dot{y} &= -\frac{2a}{n} \quad . \end{aligned} \tag{B-4}$$

In the terminal maneuver, the satellite should begin at rest. After a long time has elapsed, motion of the satellite should be asymptotic to that in Eqs. (B-4). If the complete time history of y(t) were known, the resulting motion could be determined. The method used in this study is to specify, empirically, this function

$$\dot{y} = f(t) \ . \tag{B-5}$$

Almost any function such that

$$\begin{aligned} f(0) &= 0 \\ f(\infty) &= -\frac{2a}{n} \end{aligned} \tag{B-6}$$

can be used, with certain restrictions to be discussed.

From Eqs (B-1), one obtains by differentiation and substitution

$$\dddot{y} = \dot{a}_y - 2\,n\,a_x - n^2\,\dot{y} \ . \tag{B-7}$$

Since y and its derivatives are known functions, a_x is related to a_y through the condition

$$a_x^2 + a_y^2 = a = \text{constant}. \tag{B-8}$$

Eq. (B-7) then represents a quadrature in a_y. Explicitly,

$$\dot{a}_y = \ddot{f}(t) + n^2 f(t) + 2n \sqrt{a^2 - a_y^2} \ . \tag{B-9}$$

The initial value of a_y is given by setting $\dot{x} = \dot{y} = y = 0$ in Eqs. (B-1). One obtains

$$a_{y_o} = \ddot{y}_o = \dot{f}(0). \tag{B-10}$$

The time history of a_y and thus the desired pitch program is given by solving Eq. (B-9), with the initial value given by Eq. (B-10). If the choice of f(t) is such that $|a_y|$ exceeds a at any time, the specified motion cannot be realized and the function chosen is inappropriate.

As mentioned before, any function is satisfactory which satisfies the boundary conditions in Eqs. (B-6) and the above condition on $|a_y|$. The functions investigated in this study were those of the form

$$f(t) = -\frac{2a}{n}(1 - e^{-\lambda t}). \tag{B-11}$$

Since for this function,

$$\dot{f}(0) = \frac{2\ a\lambda}{n} \tag{B-12}$$

the values of λ are restricted to the range

$$0 \leq \lambda \leq \frac{n}{2} \ . \tag{B-13}$$