(NASA-CR-160321) POWER EXTENSION PACKAGE (PEP) SYSTEM DEFINITION EXTENSION, ORBITAL SERVICE MODULE SYSTEMS ANALYSIS STUDY. VOLUME 1: EXECUTIVE SUMMARY (McDonnell-Douglas Astronautics Co.) 28 p

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POWER EXTENSION PACKAGE (PEP)

SYSTEM DEFINITION EXTENSION

Orbital Service Module Systems Analysis Study

VOLUME 1
Executive Summary

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PREFACE

The extension phase of the Orbital Service Module (OSM) Systems Analysis Study was conducted to further identify Power Extension Package (PEP) system concepts which would increase the electrical power and mission-duration capabilities of the Shuttle Orbiter. Use of solar array power to supplement the Orbiter's fuel cell/cryogenic system will double the power available to payloads and more than triple the allowable mission duration, thus greatly improving the Orbiter's capability to support the payload needs of sortie missions (those in which the payload remains in the Orbiter).

To establish the technical and programmatic basis for initiating hardware development, the PEP concept definition has been refined, and the performance capability and the mission utility of a reference design baseline have been examined in depth. Design requirements and support criteria specifications have been documented, and essential implementation plans have been prepared. Supporting trade studies and analyses have been completed.

The study report consists of 12 documents:

- Volume 1 Executive Summary
- Volume 2 PEP Preliminary Design Definition
- Volume 3 PEP Analysis and Tradeoffs
- Volume 4 PEP Functional Specification
- Volume 5 PEP Environmental Specification
- Volume 6 PEP Product Assurance
- Volume 7 PEP Logistics and Training Plan Requirements
- Volume 8 PEP Operations Support
- Volume 9 PEP Design, Development, and Test Plans
- Volume 10 PEP Project Plan
- Volume 11 PEP Cost, Schedules, and Work Breakdown Structure Dictionary
- Volume 12 PEP Data Item Descriptions

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Rockwell International – Orbiter interfaces
Lockheed Missiles and Space – Solar array
TRW Systems Group – Solar array

In addition, Spar Aerospace Products, Ltd., participated as an MDAC subcontractor in defining remote manipulator system modifications and dynamics, and European Space Agency/European Research National Organization provided detail data on Spacelab design/utilization.

MDAC wishes to acknowledge the significant contribution of these organizations in support of the PEP Project.

CONTENTS

Section 1
INTRODUCTION

The first Orbiter launch will usher in a new era of capability for space operations. The Orbiter will provide a large payload-to-orbit capability, relatively low launch cost, and rapid turnaround time, and will allow active scientific participation on these missions.

A review of the planned sortie missions (those in which the payload remains in the Orbiter), primarily Spacelab missions, which constitute a major portion of Orbiter flights, reveals that utilization of the large-payload capability and the opportunity for scientific participation are curtailed in many cases because of payload energy shortages. The shortages involve the power level available to the payload and mission-duration limitations caused by inadequate cryogenic supplies. In some cases, the expected data return for the mission payloads is less than desired. In others, all of the payloads or equipment in support of experiments cannot be usefully placed on board because of the limitations. In others, effectiveness of the missions (completeness of scientific coverage) is severely limited by power-duration shortages. A review of mission requirements by NASA mission planners, including the JSC Science Panel, indicates a need for increased power and duration during the first five Spacelab missions; the need will be greater for later missions.

The Power Extension Package (PEP) has been defined to answer this Space Transportation System (STS) program need. PEP consists of an array deployment assembly (ADA), power regulation and control assembly (PRCA), and the necessary interface and display and control equipment, as shown in Figure 1.

When required for a sortie mission, PEP is easily installed in the Orbiter cargo bay, usually at the forward end of the Orbiter bay above the Spacelab tunnel, but anywhere in the cargo bay if necessary. The ADA consists of two lightweight, foldable solar array wings (and the boxes which contain them) and deployment masts, two diode assembly interconnect boxes, a sun tracker/control/instrumentation assembly, a two-axis gimbal/slip ring assembly, and the remote manipulator system (RMS) grapple fixture. These items are mounted to a support structure that interfaces with the Orbiter. When the operating orbit is reached, the ADA is deployed from the Orbiter by the RMS. The solar array is then extended and oriented toward the sun, which it tracks by using the integral sun sensor/gimbal system. The power generated by the array is carried by cables on the RMS into the cargo bay, where it is processed and distributed by the PRCA to the Orbiter load buses.

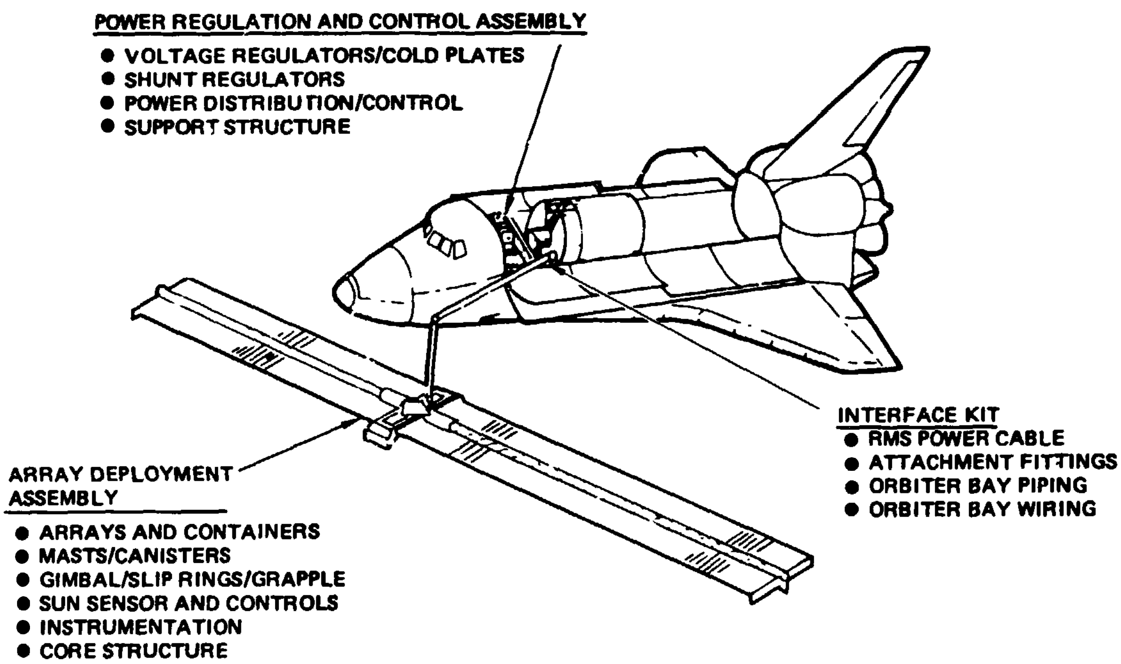


Figure 1. Power Extension Package

The PRCA, which remains in the Orbiter cargo bay, consists of six pulse-width-modulated voltage regulators mounted to three cold plates, three shunt regulators to protect the Orbiter buses from overvoltage, a power distribution and control box, and a multiplexer/demultiplexer (MDM) data bus coupler, all mounted to a second support beam that interfaces with the Orbiter.

After the mission is completed, the array is retracted and the ADA stored in the Orbiter for return to earth.

PEP is compatible with all currently defined missions and payloads and imposes minimal weight and volume penalties on these missions. It can be installed and removed as needed at the launch site within the normal Orbiter turnaround cycle.

Use of PEP reduces fuel cell cryogen consumption, with an attendant increase in mission duration, and increases the level of power available. For a 55-deg inclination orbit, PEP extends the Orbiter from a baseline capability of 7 kW to the payload for 6 days duration to 15 kW to the payload for up to 20 days. Duration further increases up to 48 days at sun-synchronous inclinations. This twofold increase in power and up to eightfold increase in duration is accomplished at low weight and low cost.

The summary benefits offered at a predicted weight of 2,351 lb to PEP users throughout the Orbiter's flight regime are:

- High User Power: 15 kW steady-state
- Long Duration: 20 days at 55 deg, 48 days at 97 deg
- Early Availability: 1983
- Increased Heat-Rejection Capability

PEP has been defined technically and programmatically to a level of detail sufficient to establish its feasibility and cost-effectiveness. The technology on which it is based consists of a modified solar electric propulsion (SEP) array (scheduled for flight test in 1980), standard design regulator and control equipment (a prototype MDAC regulator has been delivered to JSC and tested on the electric power distribution and control (EPDC) simulator, and a minimally modified Orbiter design (defined and documented by RI). To assure technology readiness, the JSC PEP project office is providing Research and Technology Operating Plan (RTOP) funding to permit an immediate start on the following items:

A. Qualification of manufacturing processes and a pilot solar cell production line for wraparound solar cells.

B. Demonstration on the EPDC simulator of the fuel cell/voltage regulator interface and the rotating gimbal slip ring assembly.

C. Dynamic analysis of the PEP/RMS/Orbiter system and requirements development for operational software.

The programmatic analyses indicate that the requirements for entry into Phase C/D have been fulfilled. An ATP for Phase C/D in FY81 would allow IOC to be in the first quarter of 1983–in time to service the energy-short Spacelab missions beginning with Spacelab 6. Beyond that point, the benefits of PEP await only the imaginative utilization of potential users.

This summary document includes a review of the requirements from which PEP was derived, a description of the PEP system, an assessment of its performance capabilities, and a description of features of the recommended PEP Project.

Section 2
NEED FOR INCREASED ENERGY

As noted, analysis of planned early sortie missions indicates the need for a significant increase in the Orbiter electrical energy capability, i.e., the electrical power and mission duration offered to prospective payloads. This fact was established during the basic Orbital Service Module (OSM) Systems Analysis Study and was verified in this extension phase. The power and duration requirements derived in the basic study are shown in Figures 2 and 3, respectively. The requirements were derived for the sortie missions of the NASA STS Mission Model of October 1977. Each sortie mission of the NASA definition was reviewed in the light of the needs of the user rather than the capabilities of the Orbiter. The needs were extracted from agency 5-year plans, user and mission planning documents (i.e., Outlook for Space, SP-386), and communications with individual users. The total power needs, i.e., the sum of payload, Spacelab equipment (1.5 kW for a pallet to 4.2 kW for module combinations), and the 14 kW allocated to the Orbiter, are shown in Figure 2 for each mission scheduled in the first years of the October 1977 model.

As seen, the total requirement varies from 18 to 32 kW for the first 3 years of operation, which indicates a clear need for an increase over the basic Orbiter capability of 21 kW. The 29-kW level was selected as a requirement because it can accommodate nearly 90% of the missions shown and appears to offer a reasonable balance between increased capability, cost, weight, and utilization over all the missions. When this capability is implemented, it is expected that other factors, such as the potential reduction of the Orbiter power consumption and the judicious scheduling of missions, would allow the full accommodation of all mission power requirements.

The companion mission-duration requirements are shown in Figure 3. These also were derived by correlating user duration needs to the scheduled missions in the October 1977 mission model. The duration requirements vary from 5 to 45 days for the first 3 years, significantly beyond the baseline Orbiter capability of 6 days. A nominal design duration of 20 days was selected for a PEP requirement as being responsive to the majority of the mission needs. Subsequent analysis has shown that a system designed to this nominal capability can, in fact, extend operations to 48 days under certain mission conditions without increasing the size of the PEP system.

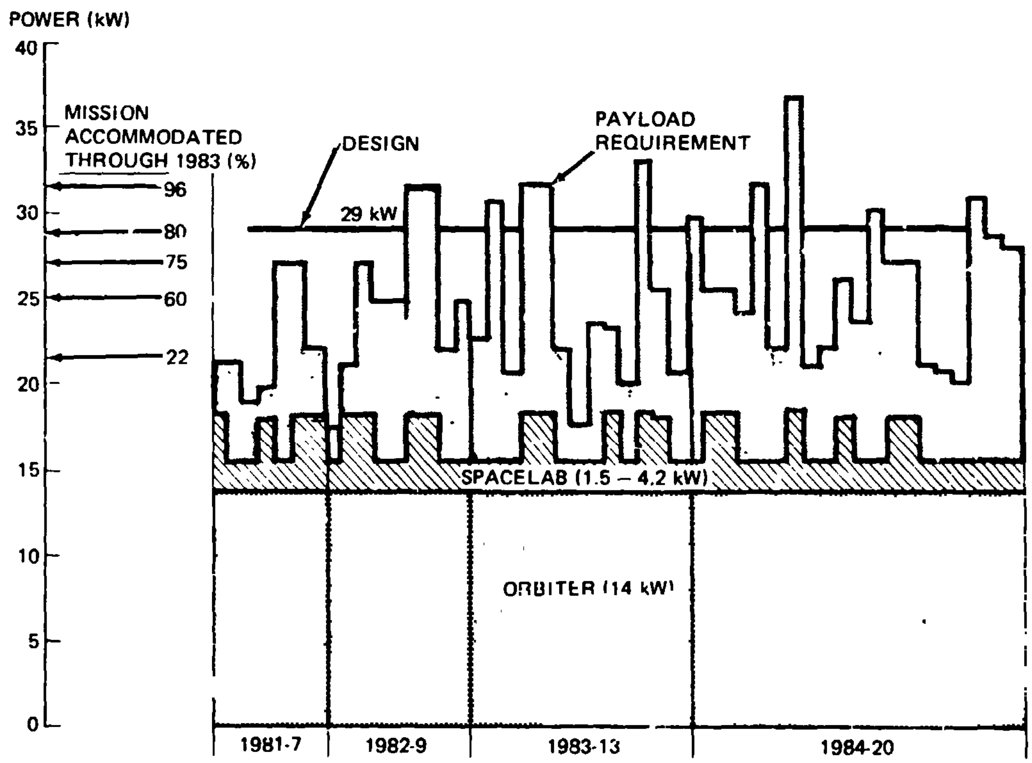


Figure 2. Spacelab Mission Power Requirements

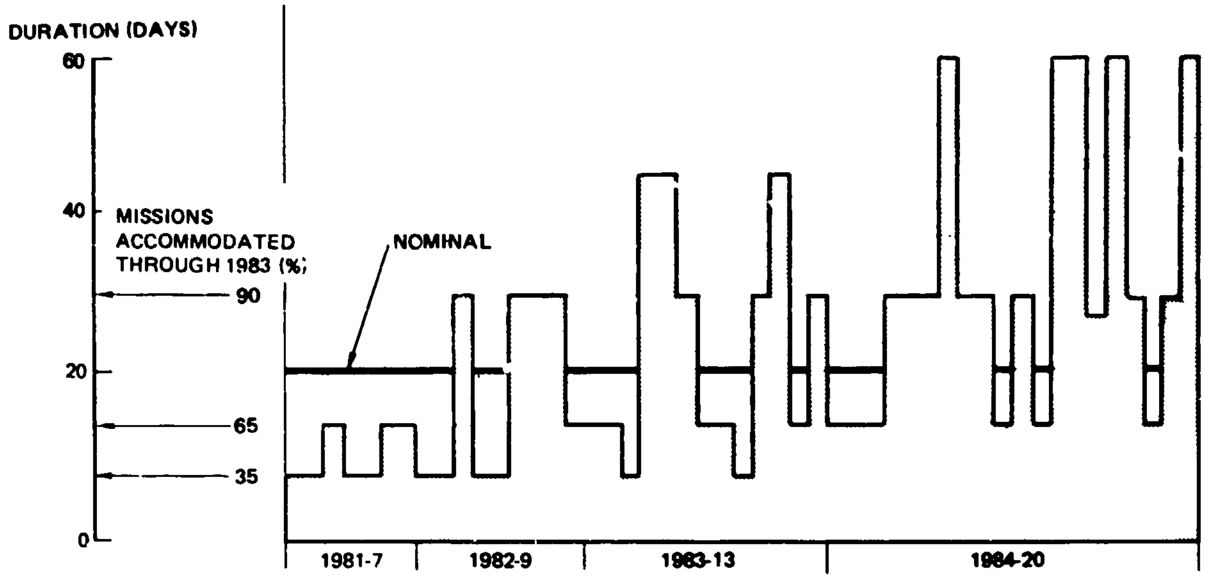


Figure 3. Mission Duration Requirements

The requirements effort of the extension phase of the study included a review of the basic requirements data with specific users, an update of the mission model to account for current planning activities, and a determination of the value of increased energy offered to specific users. The requirements data were reviewed with the mission planners and potential users, shown in Table 1, in addition to the members of the JSC Science Panel. The result was a spectrum of viewpoints from overall agency planners to specific mission advocates. The inputs were used to update the mission model data and to confirm the conclusion that increased Orbiter electrical power capability and mission duration are needed and can be used effectively.

Table 1. User Data Sources

Jesse W. Moore	OSS	Sortie mission model
William C. Snoddy	OSS	Free-flyer mission model
Robert Pace	MSFC	Spacelab 1 and 2 requirements
Dr. Charles Pellerin	OSTA	Spacelab 3 requirements
Carmine De Sanctis	MSFC	Spacelab 1, 2, and 3 interface
Richard F. Hergert	JSC	Spacelab 4 requirements
Robert C. Weaver	GSFC	Spacelab 5 requirements
Andrew J. Stophan	OSS	Mission planning data
Dr. Adrienne F. Timothy	OSS	Mission planning data

A comparison of more recent mission model data from several sources, including Jesse Moore of NASA Headquarters, with the October 1977 mission model is shown in Figure 4. The October 1977 model is used as a basis for comparison

FISCAL YEAR

MODEL	PARAMETERS	81	82	83	84	85	86	87	88	89	90	91	
10-77	INCLINATION, ALTITUDE, SIZE WEIGHT, DURATION	6	8	10	17	20	22	23	22	23	22	23	201
POP 79-1	LAUNCH SITE	1	4	9	9	13	17	14	19	18	17	19	156
13000-0-6P	INCLINATION, ALTITUDE, CONFIGURATION	1	4	2									
J. MOORE		1½	4¼	4¾	6¼								
ORBITER OPTION 1		1	3	5									

Figure 4. Mission Model Data Base (Sortie Missions)

because it is the most complete in terms of data definition and it contains the most ambitious flight schedule. The data shown are for sortie missions only. The comparison mission models show a reduced number of sortie flights, especially for the first few years. The reductions are due to funding and planning constraints and result in a decrease in the number of scheduled flights. The distribution in terms of power, duration, and orbit regime remains unchanged; in fact, there would be more reason to extend the power-duration capability for fewer missions in order to recapture as many of the original objectives as possible.

The orbit regime requirements for the planned sortie missions are shown in Figure 5. The missions are distributed along six inclination locations from 28.5 to 97 deg, with most of them at low inclination. The altitude requirements are also well distributed from 150 to 300 nm. These data indicate that, to accommodate the individuality of each mission or set of payloads, the full low earth orbit spectrum reachable by the Orbiter should be maintained, i.e., 28.5 to 104 deg of inclination and 100 to 600 nm altitude.

Specific Spacelab missions were considered in detail to determine the ability of PEP to serve them. Requirements for the first five Spacelab missions are listed in Table 2, with comments elicited from cognizant NASA personnel for each. First, the wide variety of Orbiter requirements persists–no two of the five are alike. Second, all five missions are either in need of extended capability (primarily duration) to accommodate their basic planned experimentation package or could use it to increase the data return on a given mission opportunity, to increase the payload equipment utilization, or to fully accommodate all of the planned mission equipment.

Spacelab 4 is being planned primarily as a science mission using a long module, with the potential addition of physics/astronomy instruments mounted aft on a pallet if they can be accommodated. The baseline Orbiter capability would limit the duration capability to about 6 days with only life sciences. The desired 10 days cannot be achieved even without the planned physics and astronomy payload. The addition of PEP would easily allow the full accommodation of Spacelab 4 power level and 10-day duration as planned. The longer duration capability offers a further advantage in the collection of additional data as a function of mission duration (Figure 6). This relative duration advantage increases according to the nature of the process measured in a given research area. Some changes occur during launch and show no advantage with duration (i.e., fluid redistribution and vestibular changes). The others exhibit significant data increases on a constant or changing scale with duration. For example, step-change advantages are shown as successive generations of fruit flies develop and are studied.

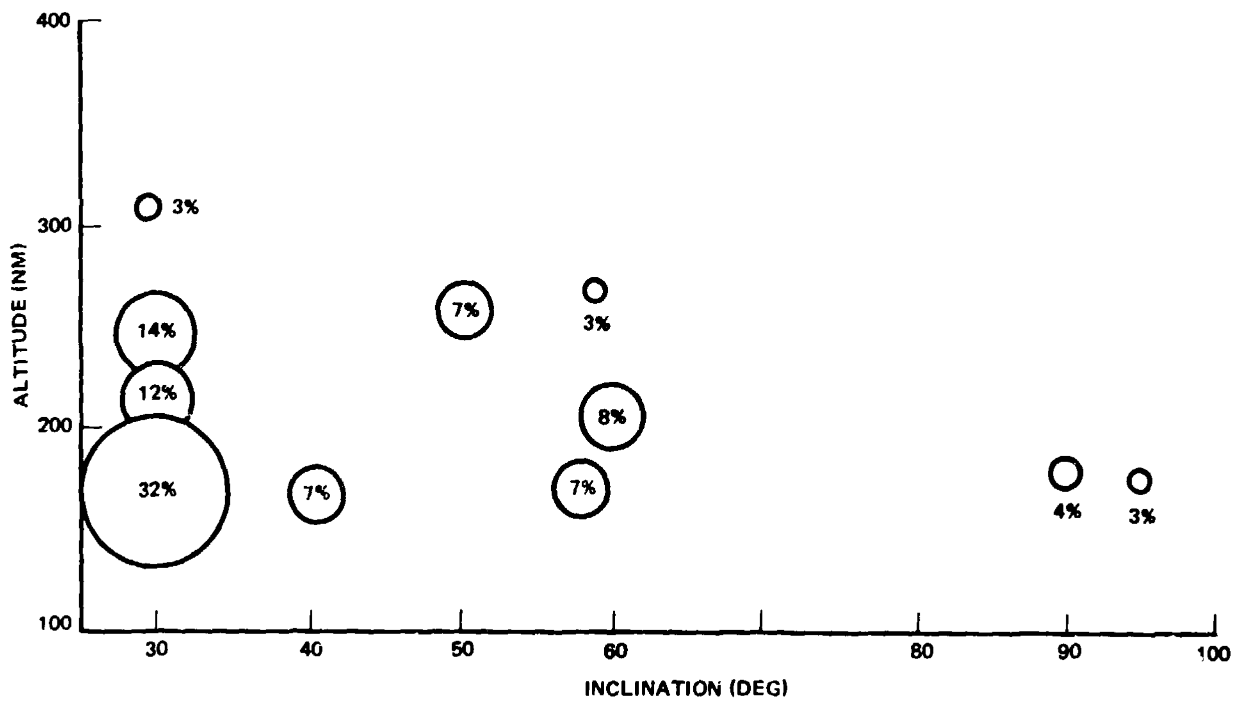


Figure 5. Altitude and Inclination Requirements (Sortie Missions)

Table 2. Early Spacelab Mission Requirments

Spacelab mission no.	Payload	Inclination (deg) and altitude (nm)	Payload power (kW)	Duration (days)	User comments
1	LM and pallet	57/135	6.7	7	Energy short with 4 tank sets
2	3 Pallets, physics and astronomy	57/225	6	9	300 kWh Short with 5 cryo set currently over weight
3	LM and pallet processing, life science Earth observation	57/200	7.7	8	Energy short with 5 cyro sets
4	LM-life science pallet, physics and astronomy	28/160		10	Desire increased power, duration, weight
5	SM + 3 pallets, physics and astronomy	57/216		7	Would like more power and duration

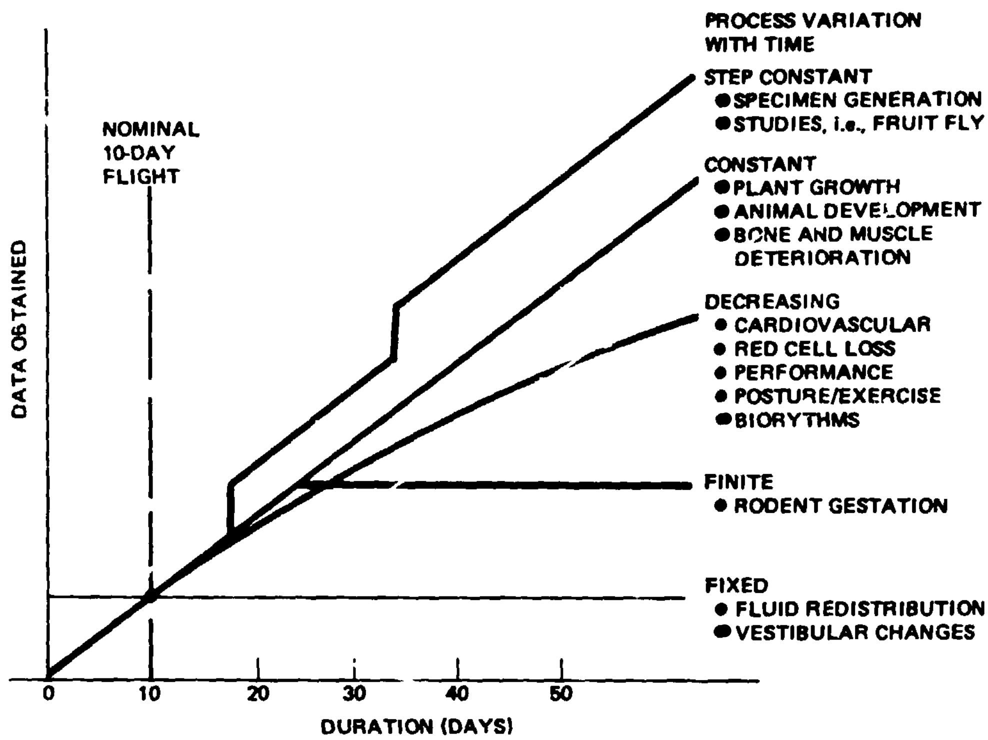


Figure 6. Measurements and/or Data Increase With Time

Quantitative accomplishments that can be made on 7-day and 48-day missions in five of the research areas being pursued on Spacelab 4 are shown in Table 3. For example, at 7 days, bone demineralization is barely detectable; at 48 days, it is a measurable quantity. Similar advantages are shown for the other research areas; all are significantly enhanced in terms of degree of accomplishment by the longer duration capability offered by PEP.

The requirements analysis conducted during the extension study phase has reiterated the need for PEP and again substantiated the basic requirements to which it is designed. These are:

- Sortie mission compatible
  - Module and/or pallet missions
  - Low weight, CG control
- Power: 29 kW (15 user, 14 Orbiter)
- Duration: 20 days nominal at 55 deg; inclination and launch time dependent (based on four cryogenic tank sets)
- Orbit inclination: 28.5 to 104 deg
- Orbit altitude: 100 to 600 nm
- All attitude capability
- Rapid ground turnaround; no serial impact
- Available for early Spacelab missions
- Power type and quality compatible with Orbiter and payloads

Table 3. Life Sciences Duration Requirements, Spacelab 4

Research area	7-day mission	48-day mission
Cardiovascular deconditioning	Adaptive changes still in progress	Stabilizes in less than 42 days; entire mechanism can be studied
Bone demineralization	Detectable only; changes still in progress	Measurements at 14, 28, and 42 days permit correlation with other loss factors
Loss of red blood cell mass	Changes still in progress	Maximum in 20-40 days; can measure during "turnaround" period
Genetics	Less than one fruit fly generation	14 Days for egg to mature adult: 2-3 generations
Morphology and development (e.g., frog)	Development still in progress, 10%	80% Development adequate for definite conclusions on gravity effects

Section 3
PERFORMANCE CAPABILITY

The basic performance capability of PEP in terms of power available to payloads, as a function of mission duration and orbit inclination, is shown in Figure 7; the power supplied to payloads can be up to 15 kW continuous. A payload service of 7 kW, which is equivalent to the baseline Orbiter, can be supplied for 12 days at 28.5 deg. for 20 days at 55 deg, and up to 48 days at 97 deg. The power level increase with inclination is due to the increased amount of sunlight available at higher inclinations. At sun-synchronous inclination (~97 deg), the orbit can be in continuous sunlight where the bulk of the power is supplied by the PEP solar array (up to 26 kW); the Orbiter fuel cells would operate at the 3-kW idle level (1 kW per fuel cell) for a total of 29 kW. At 3 kW, the available cryogens are consumed by the fuel cells in 48 days. The Orbiter capability without PEP (cryo only) is also shown in Figure 7 for comparison purposes. Its nominal mission duration capability at 7 kW is about 6 days.

The PEP system provides a factor of two increase in power to the payload and a factor of up to eight increase in duration when compared with a nominal Orbiter with four cryo tank sets, and does so with less chargeable weight. In addition, PEP has the following advantages:

- PEP has no serial impact on turnaround time; tank set installation or removal incurs a 39-hour penalty.
- PEP can support a given set of program requirements with fewer flights because of its increased duration; this reduces cost.
- PEP procurement costs are more than offset by savings in cryo tank procurement and fuel-cell refurbishment costs during the life of PEP.

The effect of varying the number of Orbiter cryogen tank sets on PEP performance is shown in Figure 8 for an orbit of 55-deg inclination and 220-nm altitude. For reference, note that PEP can deliver 7 kW to the payload for 20 days with four tank sets, 16 days with three tank sets, and 12 days with two tank sets. These numbers are for a solstice launch, which maximizes the sun availability for a given orbit; equinox launch reduces the duration capabilities by about 5 days.

PEP with only two tank sets provides durations of from 6.5 to 12 days, well in excess of the baseline Orbiter with four tank sets (~6 days). This advantage is more pronounced at higher inclinations; PEP can supply 7 kW for 20 days at 97 deg with only two cryo tank sets.

With this capability, an Orbiter with high payload performance could be configured with

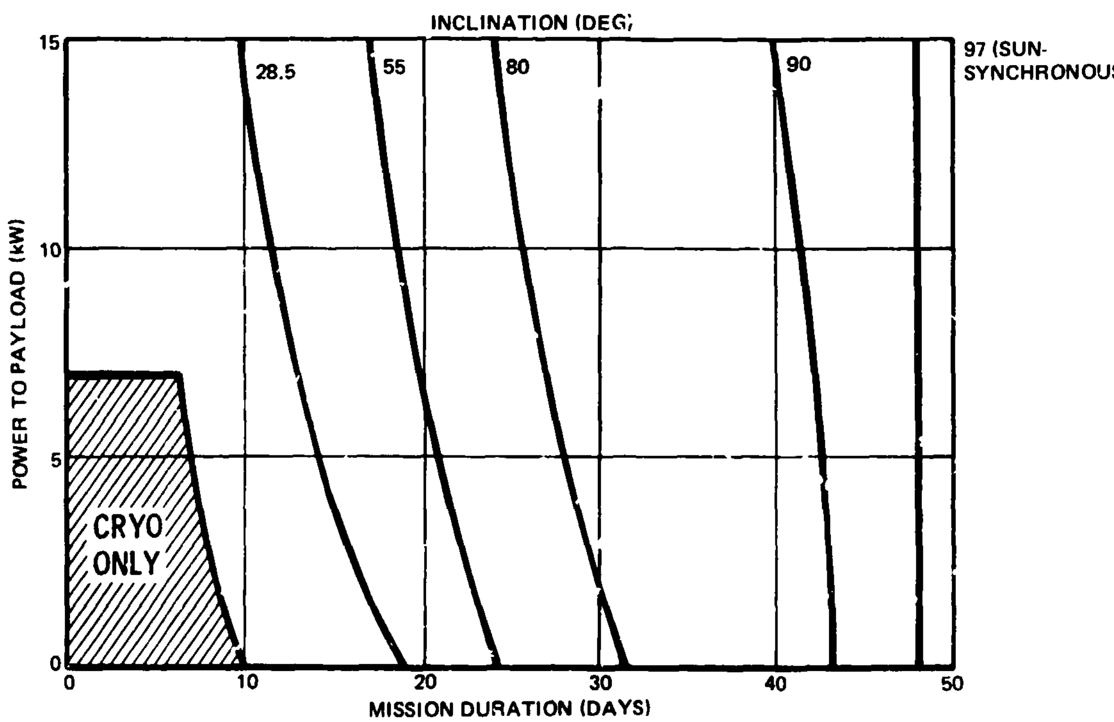


Figure 7 PEP Performance Envelope (4 Cyro Tank Sets, 3-kW Fuel Cell Idle, 220 NM)

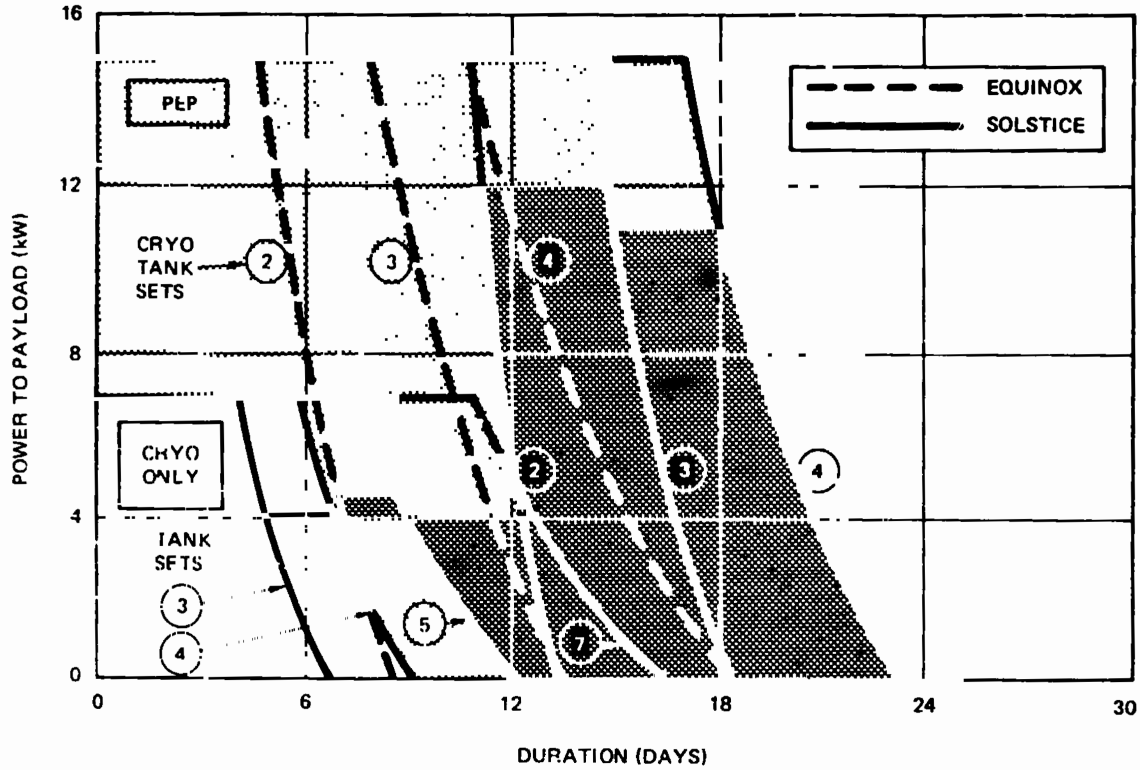


Figure 8. PEP Performance Benefits 55 Deg x 220 NM)

fewer tank sets to accommodate delivery missions of high payload weights. The payload gain with only two tank sets would be about 3,500 lb. This same Orbiter could also be used to satisfy long-duration missions by installing PEP.

The flexibility thus achieved through the use of PEP is expected to enhance the ability of the Orbiter to respond to a wide variety of mission requirements. The current estimated PEP weight of 2,351 lb (2,266 lb of PEP plus 85 lb of attach fittings) is equivalent to 1.2 cryo tank sets.

The nominal PEP performance has been predicated upon an ability to achieve an idle level of 1 kW per fuel cell in the sunlight. The duration sensitivity with respect to idle level varies at 55 deg from about -2 to -4.5 days per kilowatt; the variation is dependent upon power level and launch date.

The PEP performance variation is not a strong function of altitude; at 55 deg, the sensitivity is about 1.5 days per 100 nm of altitude. The payload-lifting capability of the Orbiter also varies slowly with increasing altitude until the Orbiter integral OMS tanks are full. Figure 9 shows the duration versus Western Test Range delivered payload capability of Orbiter with PEP. At sun-synchronous inclination (97 deg), almost 20,000 lb can be maintained on orbit by the 1982-1985

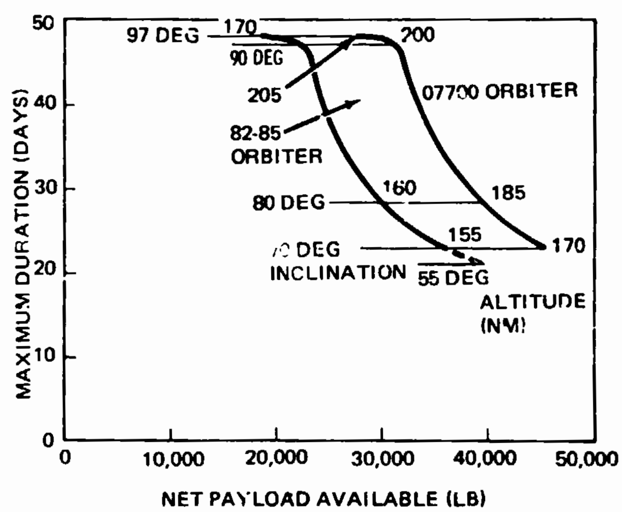


Figure 9. Orbiter/PEP Payload

Orbiter for 48 days; at 55 deg, 42,000 lb for 20 days. Note that the indicated operating altitudes for VAFB serviced inclinations (>55 deg) are low (155 to 170 nm range). Analysis of PEP systems with respect to aerodynamic loads, lifetime, and attitude control indicates that operation below these altitudes is within the inherent capability of PEP.

The time of launch affects the performance of PEP because of the changes in sun angle and resultant time in sunlight per orbit. Figure 10 shows the effect of mission duration capability as a function of launch date for a 220-nm orbit at various inclinations. The reference capability of 20 days at 7 kW and 55-deg inclination occurs near summer and winter solstice. These conditions allow the traverse of the maximum Beta angle (the angle between the sun line and its projection on the orbit plane). The duration capability is reduced to 14 days at 7 kW to the payload and 55-deg inclination at other times of year. The yearly variation is most pronounced for a 90-deg inclination orbit because of the wide range of Beta angle encountered. At sun-synchronous inclination, a full-sun condition can be achieved at 220 nm any time of the year, resulting in a full 48-day capability. These curves are for the condition of launching at the most opportune time of day for each day of the year shown. The minimum Beta line is the duration capability for the worst-case launch condition.

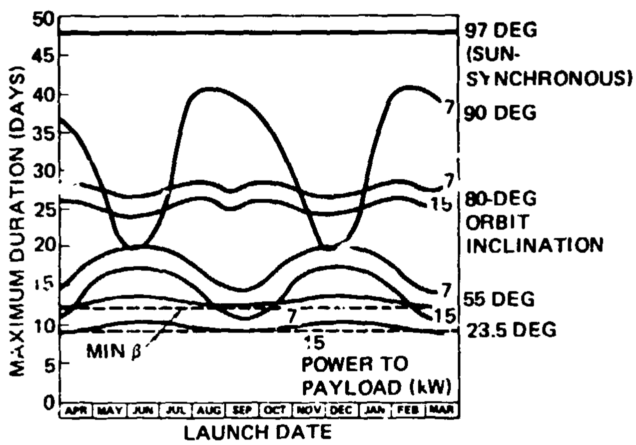


Figure 10. PEP Performance – Launch Date Effects (Altitude = 220 NM)

The launch time of day effect is shown in Figure 11 for several combinations of inclination and launch date. A near-terminator launch, 0600 or 1800 hours, provides the maximum duration capability. The launch window is longest for a morning launch for the conditions shown because of the regression pattern of the orbit.

The allowable launch window is ~2 hours, as compared with a typical 10-minute rendezvous mission launch window. The effect of launch time on sun angle and mission-duration capability is shown in Figure 12 for a 97-deg and 220-nm orbit. An 0600 launch would allow terminator viewing of the earth's surface for a 48-day duration, a noon launch (overhead sun) would allow a 12-day duration, and an 1800 hour launch a 42-day duration (terminator viewing).

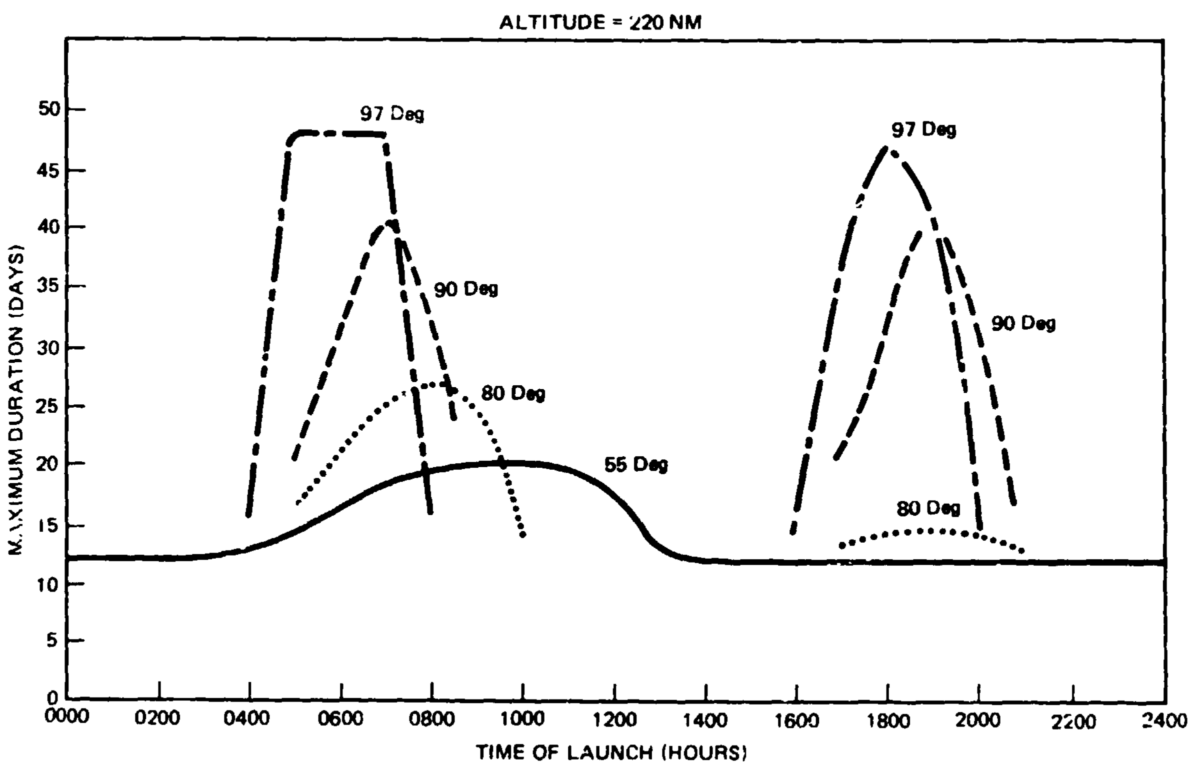


Figure 11. Mission Duration – Launch Window Effect (3-kW Fuel Cell Idle, 4 Cryo Sets, h = 220 NM, 7 kW to Payload)

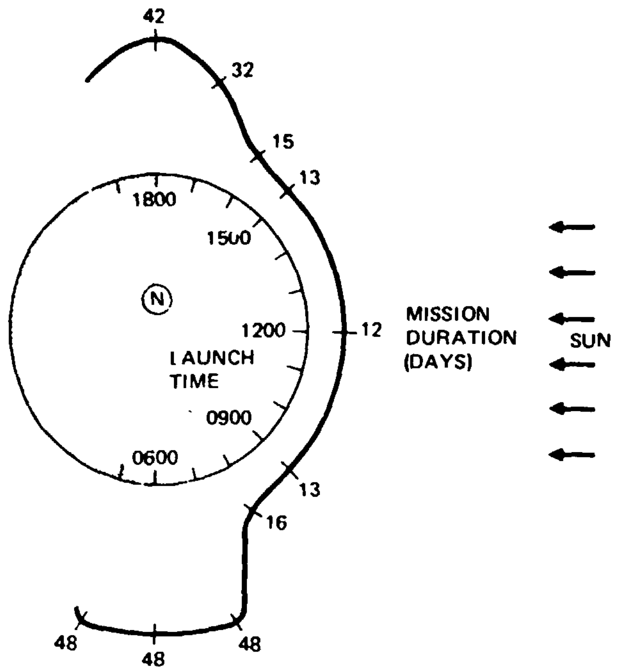


Figure 12. Mission Duration – Sun Illumination (97 Deg x 220 NM Orbit, 7-kW Payload Power, Autumnal Equinox)

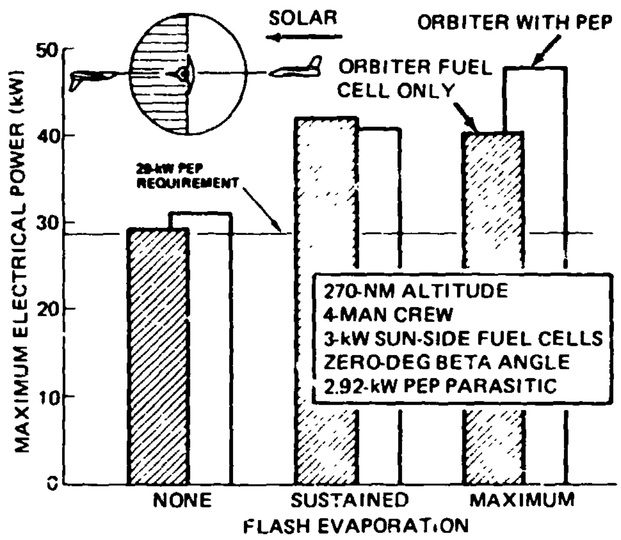


Figure 13. Heat-Rejection Performance – Low-G Sustained Operation

The potential for increasing mission duration by means of elliptical orbits was examined. For a perigee fixed at 100 nm, an increase in the apogee will increase the mission duration capability by about 1 day per increase of 100 nm in apogee.

Heat-rejection performance capability depends upon Orbiter orientation, radiator deployment angle, and degree of flash evaporator system (FES) operation. The least favorable orientations occur when the Orbiter bay is facing a hot environment such as occurs when earth or solar viewing. FES operation is necessary for these orientations to use the maximum capability of PEP.

Favorable orientations, which orient the radiators away from a direct view of earth or sun, result in full or nearly full use of the entire PEP power capability with the FES (Figure 13). Note that the heat-rejection capability with PEP is greater than the baseline Orbiter in all cases. This is because with PEP the level of fuel cell operation is lower, thus reducing fuel cell waste heat. Studies have shown that increasing the radiator deployment angle to 60 deg will significantly improve PEP performance and this Orbiter modification has been baselined.

As presently designed, the European Spacelab can accommodate 12 kW of power even though only 7 kW is available from the Orbiter. European plans for follow-on development include an increase in this capability to 15 kW plus increased thermal control and heat-rejection capacity.

The ability of the Orbiter to maintain its orientation, pointing accuracy, and maneuver capability with PEP has been examined and verified. For example, the acceleration levels that would be experienced in the payload bay on a PEP low-g mission (applicable for materials processing, etc.) are shown in Table 4 and compared to the Orbiter-only case. The long-term limiting factor is the acceleration due to free oscillations (2×10^{-6} g) and is the same for both cases. The effects of crew motion and primary and vernier reaction control system (RCS) operations are similarly intermittent and larger in magnitude than those due to oscillations.

The performance capabilities of PEP have been shown to be dependent upon several mission parameters. The sensitivities to each have been illustrated here to give potential mission planners a measure of the degree of capability and flexibility of PEP. Careful analysis of specific mission requirements, as they mature and become available, will allow the utilization of full PEP capabilities.

Table 4. Acceleration Levels

Disturbance	Acceleration (g's)	
	Orbiter only	Orbiter with PEP
Aerodynamic (~160 nm)	5.5×10^{-7}	1.2×10^{-6}
PEP gimbal drive torque	—	1.8×10^{-6}
Gravity-gradient and aero-free oscillations	2×10^{-6}	2×10^{-6}
Crew motion		
17-lb pushoff	10^{-4}	10^{-4}
2-lb pushoff	10^{-5}	10^{-4}
Vernier RCS	10^{-4} to 10^{-3}	10^{-4} to 10^{-3}
Primary RCS	4×10^{-3} to 4×10^{-2}	4×10^{-3} to 4×10^{-2}

Section 4
DESIGN DEFINITION

The PEP system concept has been detailed to a level sufficient for entry into Phase C/D of the procurement cycle. Highlights of that design definition are summarized in this section.

The two major elements of PEP, the ADA and the PRCA, are integrated into the Orbiter bay as shown in Figure 14. They are normally installed at the forward end of the Orbiter bay above the Spacelab tunnel, but the ADA can be located anywhere in the cargo bay within reach of the RMS.

Figure 15 shows the Orbiter dynamic envelope available at the nominal installation location; this envelope, within which PEP fits, is compatible with both the module configuration and the all-pallet configuration with igloo of Spacelab. The Orbiter/PEP retention provisions are shown in Figure 16. The ADA shares two standard Orbiter bridge fittings with the Spacelab short tunnel and one standard bridge fitting with the Spacelab module; three remotely operated lightweight custom retention latches lock the ADA to the bridge fittings and allow the ADA to be mounted over standard Spacelab pallets. The PRCA is installed forward of the ADA on two custom lightweight bridge fittings which provide clearance for mounting adjacent to the RMS; the PRCA trunnions are locked into a bridge-fitting journal.

On orbit, the ADA is grasped by the RMS, using a special end effector, and is moved outside the Orbiter bay. The ADA is then positioned to allow the mission and payload orientation requirements to be met while allowing solar array alignment normal to the sun line. The selected location could be at the left or right side of the Orbiter, below, or in front, as needed to best satisfy mission objectives. When the optimum position is established by the RMS, the RMS joint brakes are maintained locked throughout the selected orientation sequence, e.g., Y-axis perpendicular to the orbit plane. The deployed solar array is then aligned normal to the sun line by the two PEP gimbals.

The outer Alpha gimbal is placed by the RMS with its axis perpendicular to the orbit plane; this allows the Alpha gimbal 360-deg drive to rotate as needed in the plane of the orbit. The 0 to 90-deg Beta gimbal allows the array to nod toward the sun, accounting for variations in orbit Beta angle. Power is transferred from the array across slip rings on the Alpha gimbal, through an added remotely activated RMS end effector power umbilical connector, and along power cables attached to the RMS, which terminate at

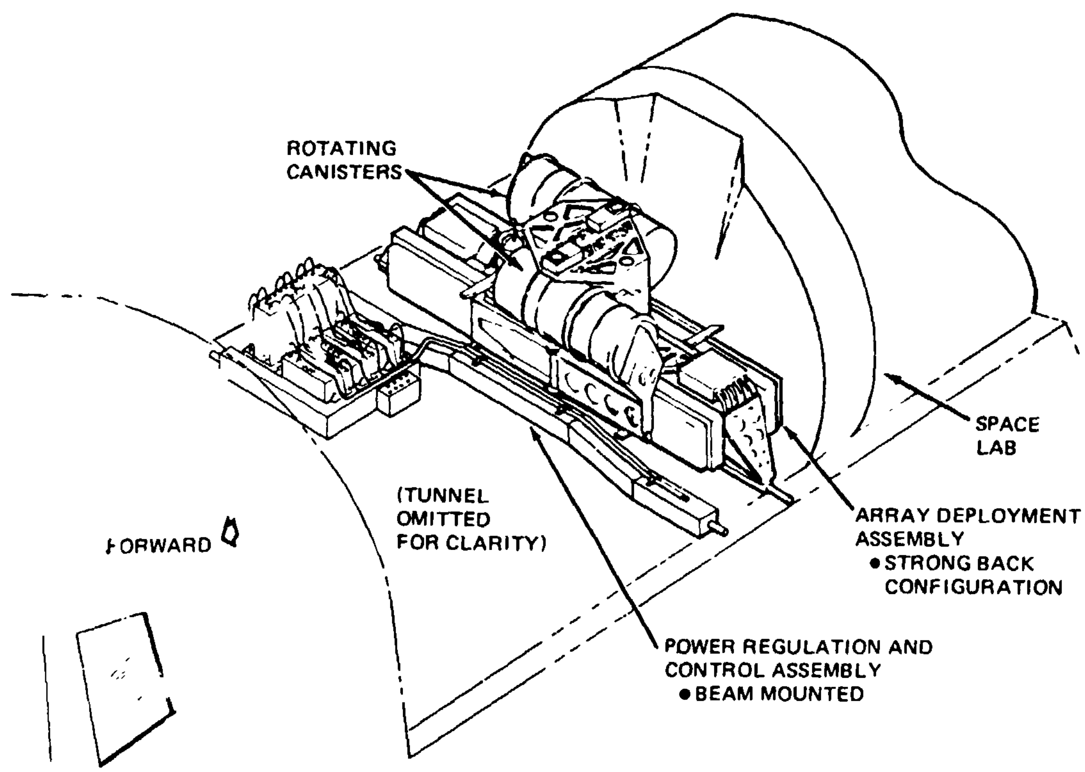
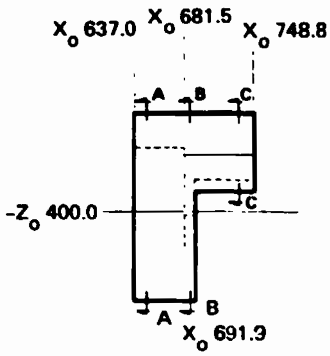


Figure 14. PEP Reference Installation (Two-Beam Configuration)



NOTES:

1. THE ENVELOPE SHOWN DOES NOT INCLUDE TRUNNION INTERFACE WITH THE ORBITER. IN SECT C-C TRUNNION INTERFACE MUST OCCUR ABOVE STATION Z_o 419 OR OUTSIDE OF THE EXTENSION OF THE 90R BELOW STATION Z_o 419.
2. THE LIMIT STATIONS X_o 748.8, X_o 691.9, AND X_o 681.5 ARE CLEARANCES FROM HANDRAILS ON THE SPACELAB TUNNEL AND AIRLOCK, RESPECTIVELY.
3. THE LIMIT STATION Z_o 429 IS BASED ON TUNNEL AND TUNNEL STRUT DEFLECTIONS.
4. LIMIT STATION Z_o 460 IS BASED ON AIRLOCK CLEARANCE.
5. LIMIT STATION X_o 637 IS BASED ON MMU AND EVA ENVELOPE CLEARANCE.
6. LIMIT STATION Z_o 414 BASED ON IGLOO CLEARANCE.

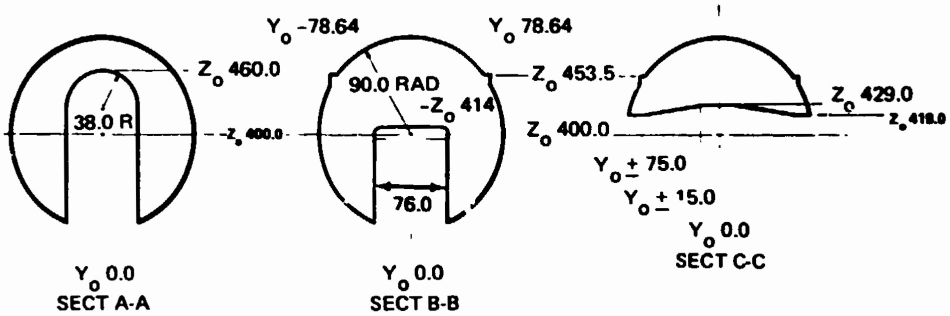


Figure 15. PEP System Stowed, Maximum Dynamic Envelope

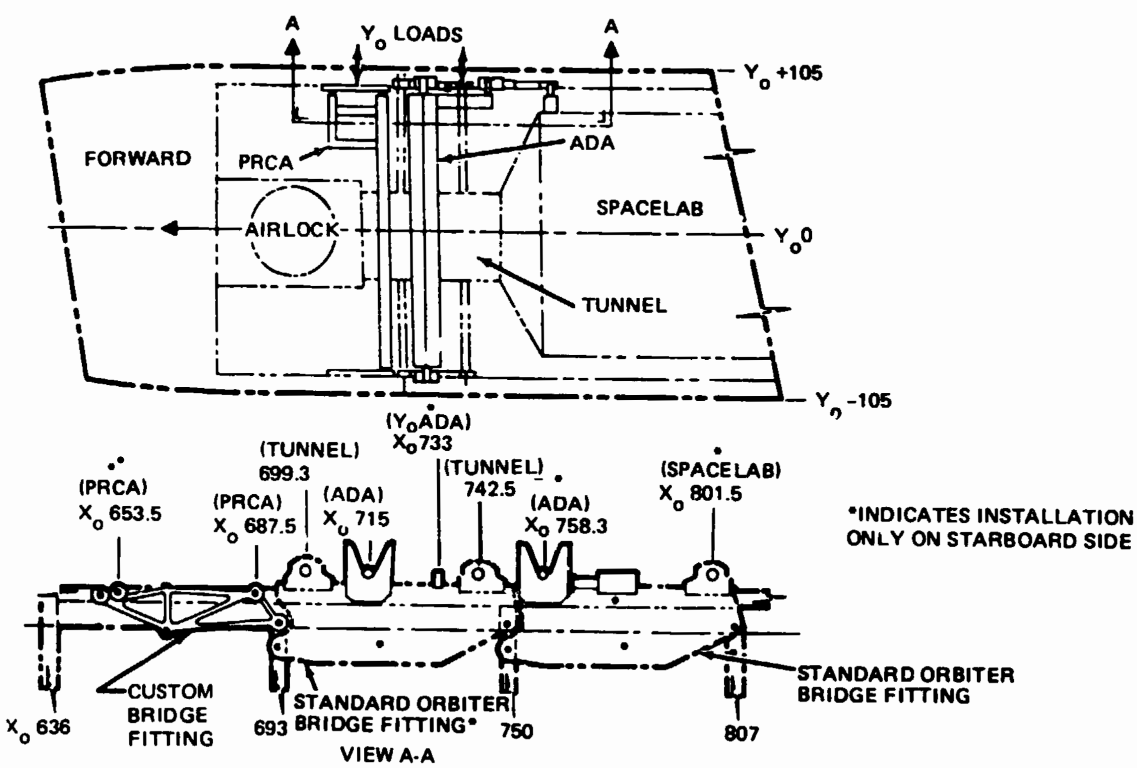


Figure 16. PEP Retention Provisions

an RMS shoulder connector. From the shoulder, the power is transferred by cable to the PRCA, where it is controlled, regulated, and distributed to the Orbiter buses for use by the Orbiter and its payloads. This modification retains the ability of the RMS to serve its intended payloads when PEP is not operational.

An exploded view of the ADA design is shown in Figure 17. The core structure is a box beam that provides for transverse attachment across the Orbiter payload bay. Two solar array wing assemblies are bolted to opposite sides of the beam. The deployment canisters are mounted on top of the beam and undergo a 90-deg rotation prior to deployment. Also mounted to the core structure are the two diode assembly packages, which provide array module isolation and interconnection; the two axis gimbal/slip ring/RMS grapple fixture, which provides array orientation, power transmission, and RMS attachment; the sun sensor and sun sensor processor, which derive control signals for array positioning; and the pointing and control electronics, which drives the gimbal and provides the signal processing to generate information for the Orbiter displays.

The solar array wing shown in Figure 18 is based on the SEP array concept; it consists of 50 hinged panels per wing of 2 x 4 cm solar cells attached to a flexible substrate. Although the current baseline assumes a wraparound cell configuration, both conventional cells and larger cell sizes have also been investigated. Final design selection will occur early in Phase C/D.

The array is deployed and retracted by actuation of the deployable mast shown in Figure 19; the mast consists of a composite triangular truss stored helically wound in a canister. During mast extension and retraction, folding of the array is controlled by guide wires; when fully extended, the 3.84 x 36 m array wing is kept under tension by the mast through negator springs to assure the required flatness.

The mast is deployed from its canister by redundant motors, driving through a gear box, which provides for two-speed operation. During the first 2 feet of mast extension, the canisters are unlatched and auto-rotated at slow speed; after latching in this position, the array is fully deployed at high speed. The sequence is reversed during retraction.

The mast canister assembly is attached to the support beam through a compliant mount; it controls the frequency response of the deployed array to Orbiter-induced loads in two axes (in the array plane and perpendicular to the array plane). The compliant mount is locked out when the array is retracted.

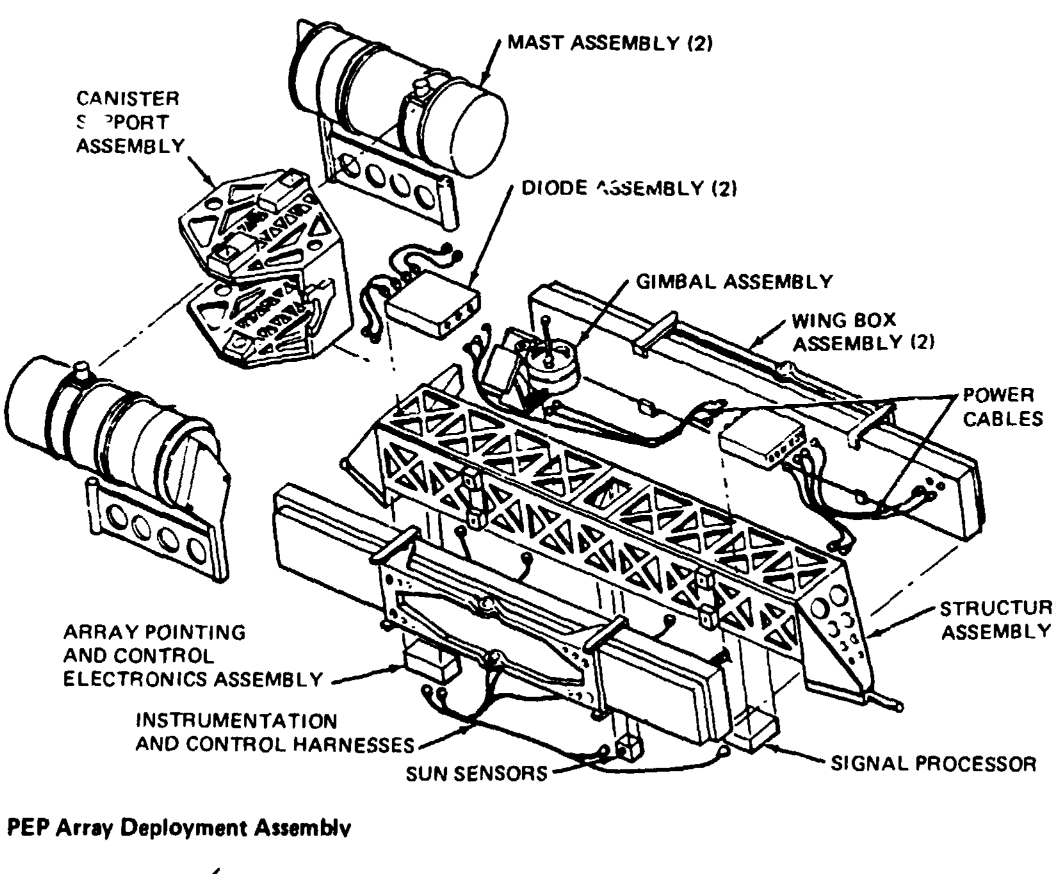
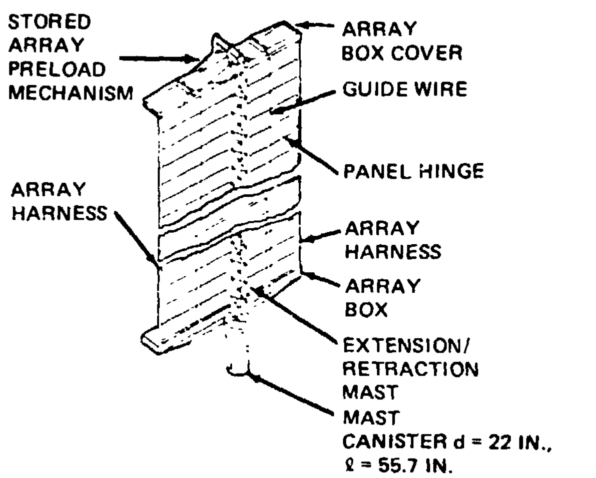


Figure 17. PEP Array Deployment Assembly

When PEP is aboard the Orbiter, the Orbiter digital autopilot software may be required to inhibit the length and/or frequency of VRCS pulses to limit Orbiter rates and VRCS plume loads on the array. In conjunction with the compliant mount, this restriction would assure that the array or mast would not be subjected to excessive loads and that the RMS brake torques would not be exceeded.



DESIGN DATA

• Si CELL	– 8 MILS THICK
	– 2 X 4 cm WRAPAROUND
	– 12.8 EFFICIENCY
• COVER	– 6-MIL MICROSHEET
• SUBSTRATE	– 1/2-MIL KAPTON WITH INTERNAL WIRING AND WELDED INTERCONNECTS
• WING SIZE/AREA	– 3.84 X 37.8 METERS/ 145 METERS^2
	– 50 PANELS PER WING
• OUTPUT OF WING	– 16.4-kWe RAW POWER

Figure 18. PEP Array Wing Characteristics Current Baseline

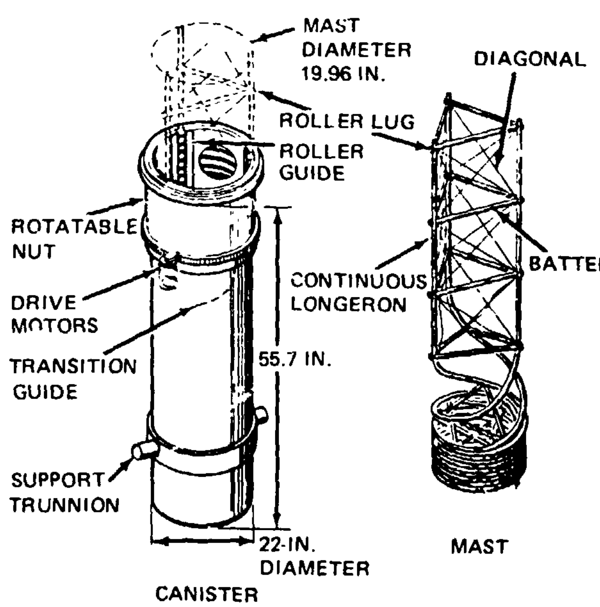


Figure 19. Extension Mast

The gimbal/slip ring/grapple assembly and its relationship to the RMS system are shown in Figure 20. The standard RMS end effector mates physically with the grapple fixture on this assembly and a remotely actuated umbilical mates with the PEP harness installed along the RMS.

All safety-critical latches and actuators that are incorporated in the ADA are capable of EVA manual operation in an emergency.

The avionics equipment on the support beam (sun sensor, sun-sensor signal processor, and the pointing and control electronics assembly) provides the system control interface with the Orbiter through an MDM and data bus coupler assembly mounted on the PRCA. In addition, the Orbiter's multifunction CRT display system (MCDS), the systems management computer, and switches located on the on-orbit station standard switch panel constitute an intrinsic part of the system equipment. The MCDS includes a keyboard through which crew commands are input via the display processor to the general-purpose computer and then relayed via the bus couplers to the MDM for control of in-bay power equipment or transferred to the electronics assembly for ADA control. PEP status data, transferred from these units to the computer, is processed and displayed on the CRT. Figure 21 shows a typical PEP status display.

The PRCA is shown in Figure 22. It consists of six voltage regulators mounted on three Orbiter-style cold plates, three shunt regulators, PRCA, MDM, data bus couplers, and power cables mounted to a beam support structure. As noted earlier, the PRCA remains in the Orbiter payload bay during a mission.

Figure 23 shows the PEP electrical installation in the Orbiter. Power coming down the RMS harness is routed along the PRCA beam to the voltage regulators. Power from the PRCA interfaces with the Orbiter Main A distribution assembly at Station 693 on the port side and with Mains B and C at Station 636 on the starboard side; all three circuit grounds are tied to the Orbiter structure at these interfaces. Power cables to the main distribution assemblies from these interfaces are supplied as kit items for installation below the cargo bay liner.

The PEP electrical system is shown schematically in Figure 24. High-voltage power from the arrays is provided via the RMS harness to the six pulse-width-modulated regulators. These regulators perform two major functions: (1) each contains a microprocessor which continuously moni-

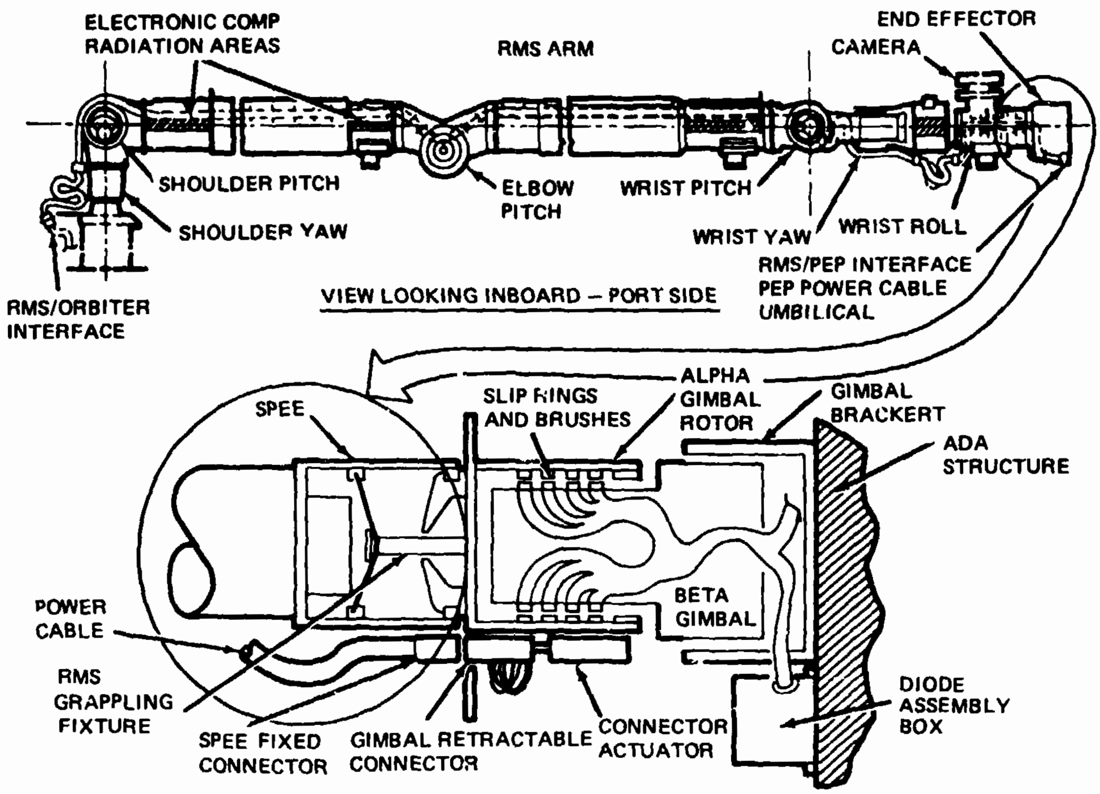


Figure 20. Gimbal/Slip Ring/Grapple Assembly

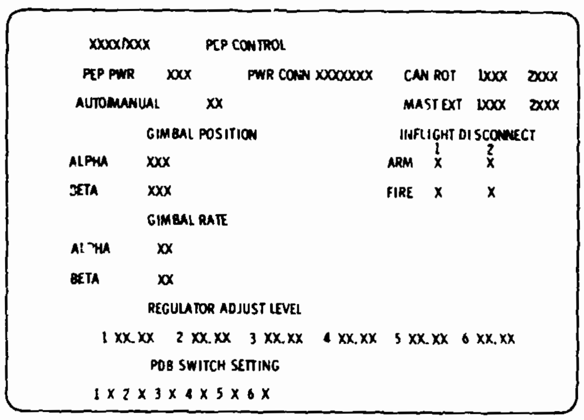


Figure 21. PEP Control Display Format

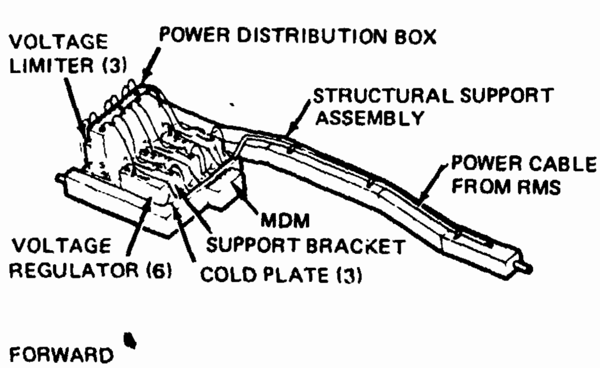


Figure 22. Power Regulation and Control Assembly

tors the array output and supplies control signals to assure that the peak power from the array is available to the system, and (2) they maintain a suitable output voltage (nominally 32.6 V) to keep the fuel cells idling at the 1 kW needed to achieve the specified PEP mission duration.

Each regulator has internal overvoltage and current limiting circuitry and is provided with remote sensing capability. Output power of pairs of regulators are supplied to the three Orbiter main buses.

A development PEP regulator, Figure 25, has been constructed by MDAC using Company funds. The unit shown has been shipped to NASA/JSC for testing on their Orbiter electric power distribution and control simulator and will be used to verify the method of regulating the solar array output power and the Orbiter bus interface.

The voltage regulator cold plates, which are of standard Orbiter design, are each tied into both the primary and secondary Freon 21 loops downstream of the Orbiter aft avionics bays. Quick disconnects and a jumper for use when PEP is not on board support quick PEP installation and removal.

One shunt regulator is tied to each of the three Orbiter buses via the power-distribution

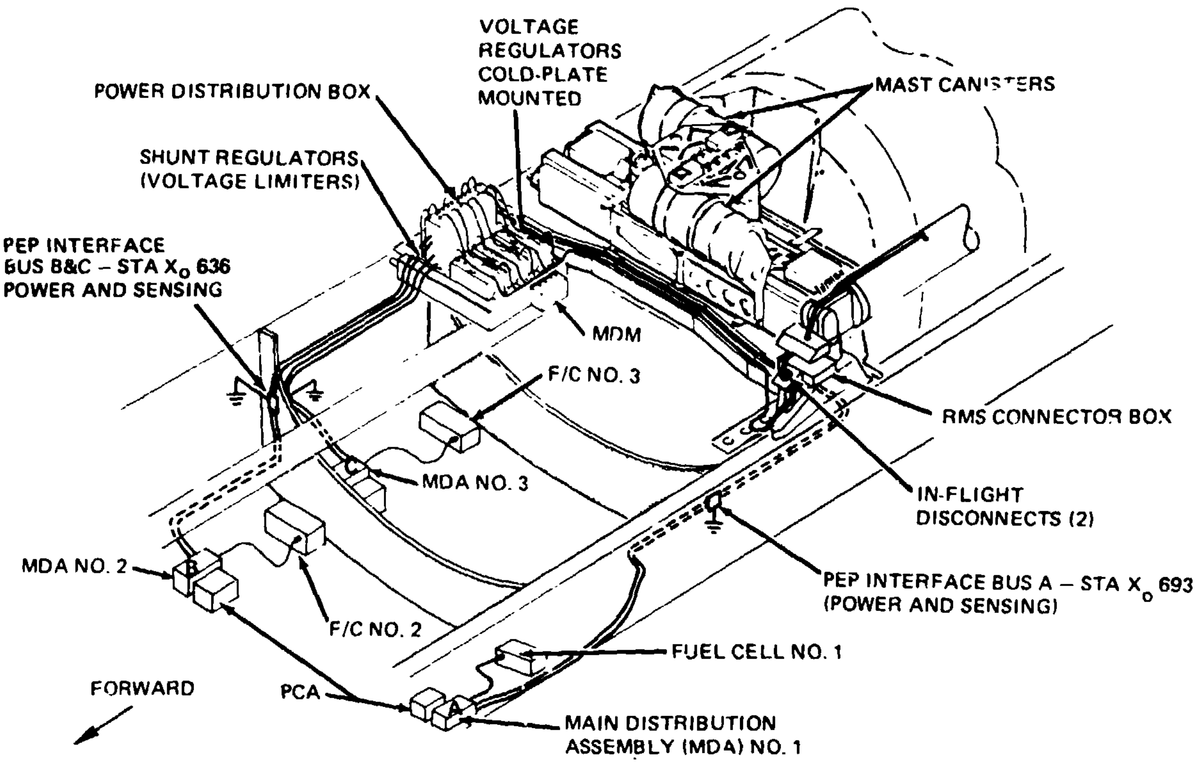


Figure 23. PEP Power Distribution Interfaces

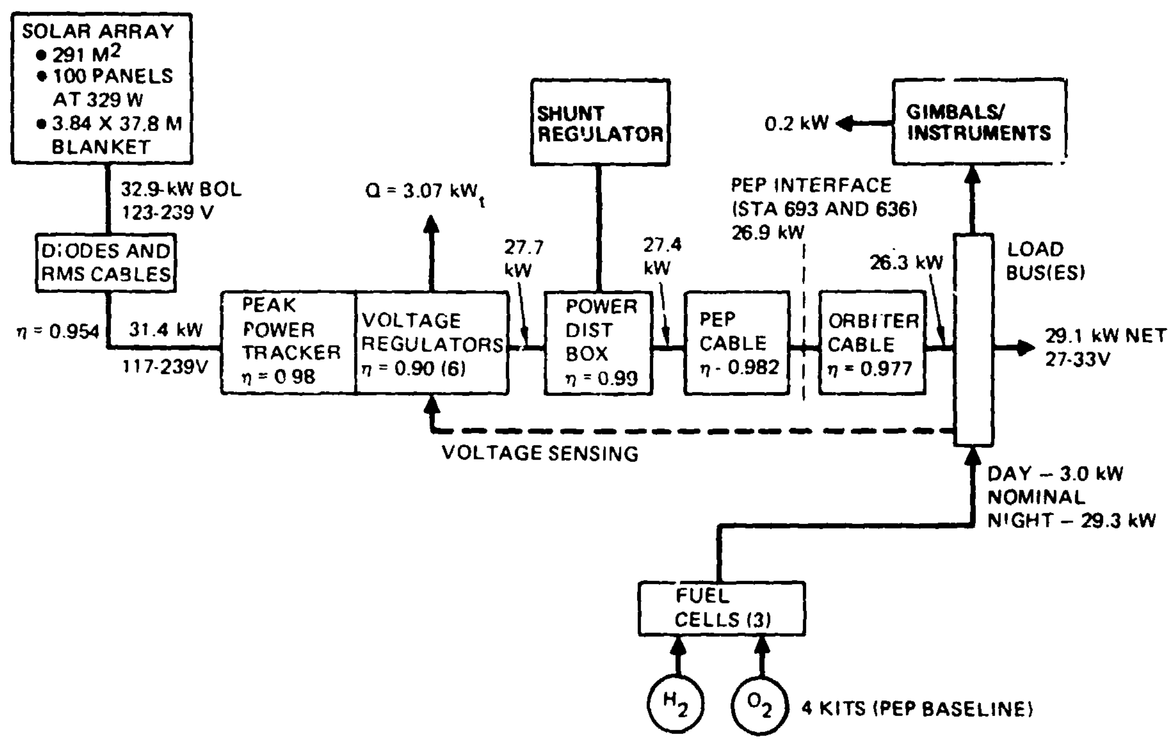


Figure 24. PEP Electrical Power System

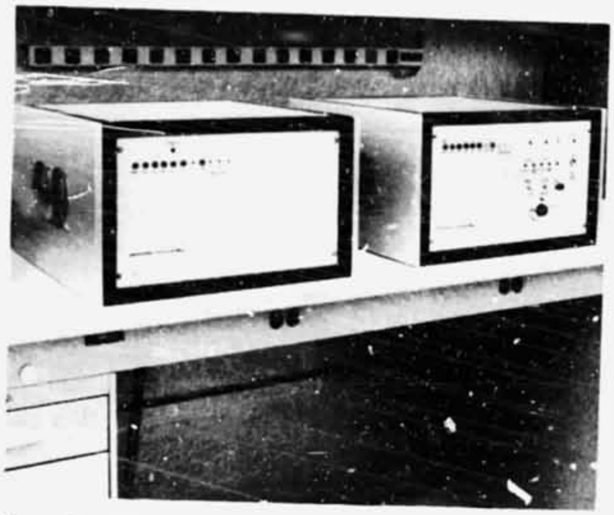


Figure 25. MDAC Prototype Regulator

box; should failure of internal protective circuitry allow a voltage regulator to supply greater than 33 V to the Orbiter bus, the shunt regulator will draw sufficient current to limit the bus voltage until the voltage regulator can be shut down. The shutdown is done automatically within 200 msec; the associated fuel cell will then make up the power lost for the duration of the mission.

The currently estimated PEP system weight summarized in Table 5 is 2,266 lb, with an additional 85 lb needed for retention fittings to attach the PEP system to the Orbiter sill through bridge fittings. (Approaches to meeting the JSC weight goal of 2,010 lb have been identified but not yet incorporated into this definition.) The weight distribution among major elements is: (1) ADA, 1,374 lb, (2) PRCA, 666 lb, and (3) interface kits, 226 lb. The selected PEP concept summarized here is the result of several design analysis and trade study iterations (i.e., installation envelope, array aspect ratio, packaging configuration, regulator alternatives, plume impingement, thermal control, etc.).

Table 5. PEP Weights

Subsystem	Weight (lb)
Solar array	1,031
Structure/mechanical	300
Power distribution and regulation	698
RMS power cable	101
Thermal control	44
Avionics	92
Subtotal	2,266
Retention provision	85
Total	2,351

The feasibility of the PEP design integration with the Orbiter has been verified by layout drawings, analysis, and the definition of all the interfaces between PEP, the Orbiter, and RMS. Twenty-one distinct interface items have been detailed in the areas of electrical, structural/mechanical, avionics, thermal, and crew; these interface definitions resulted from joint MDAC/RI and MDAC/Spar activities. Consistent with the design philosophy of integrating PEP without infringing on the basic capabilities of Orbiter, the modifications required for PEP installation are minimal. They include items in the power distribution system, the coolant loop, data bus connector, physical attachments (electrical harness, plumbing, and equipment mounting), and a power on/off switch located on the aft flight deck (details are in Volume 4). The total Orbiter scar weight is estimated at 46 lb.

Program definition and planning activities in the areas of product assurance, logistics and training; ground and flight operations support; and the design, development, and test baseline are given in Volumes 6, 7, 8, and 9, respectively.

The deployment and mission operations of PEP are monitored and controlled in the on-orbit station on the Orbiter aft flight deck. The deployment sequence is illustrated in Figure 26 and timelined in Figure 27. The RMS grapples the ADA and moves it to the deployment assembly

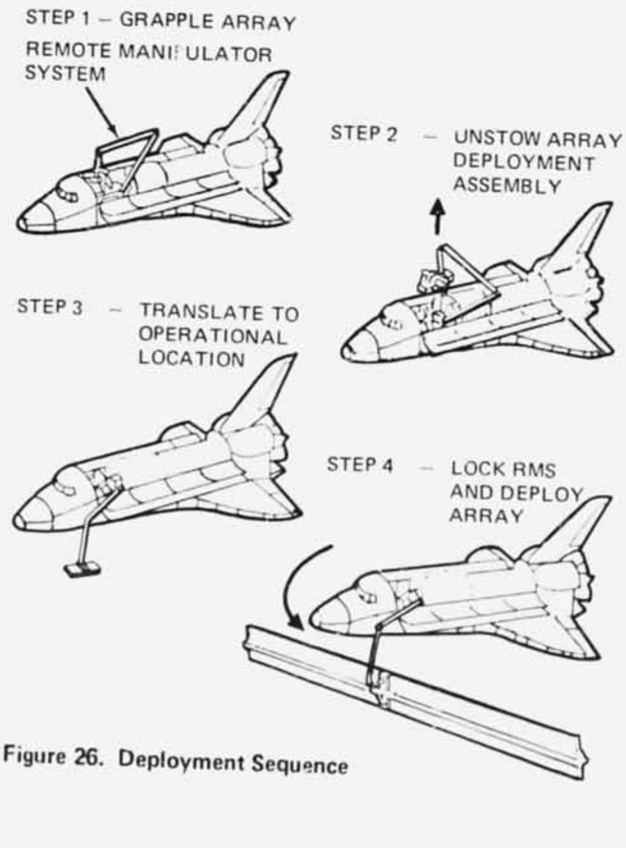


Figure 26. Deployment Sequence

position where the RMS is fixed. The array is then deployed and oriented normal to the sun line. The total elapsed time is 39.1 min. if all tasks are done in series. This time can be reduced to 30.6 min. for a parallel operational alternative.

Several deployment options have been analyzed which address RMS status and checkout, parallel operations, and location preferences. The time required for retraction storage is 36 min., including placing of the attach trunnions back into their latch fittings.

A typical PEP operation on orbit is shown in Figure 28. The Orbiter is oriented with the Y-axis perpendicular to the orbit plane for an earth observation mission (Z-axis aligned along the local vertical). The Alpha gimbal axis is perpendicular to the orbit plane and thus allows relative orbital rotation while maintaining solar alignment of the array. A zero Beta-angle condition is shown; as Beta changes with orbit regression, the operation is the same, with the Beta gimbal slowly adjusting the array to maintain normality to the sun line.

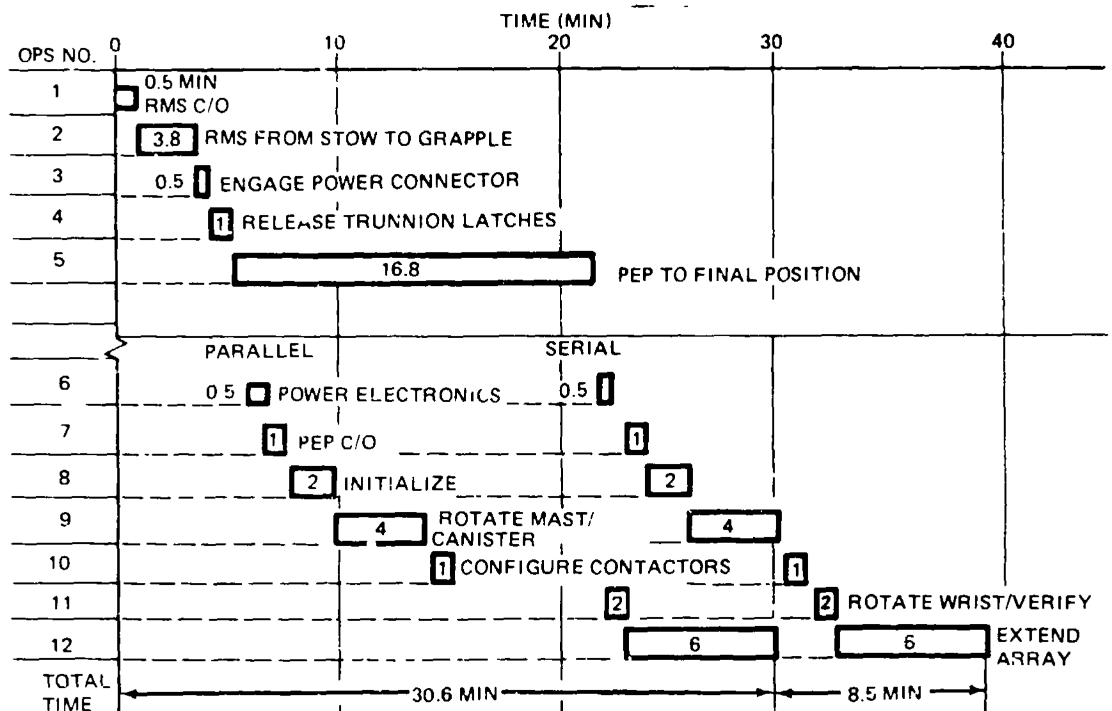


Figure 27. PEP Orbital Deployment Timeline (Warm RMS)

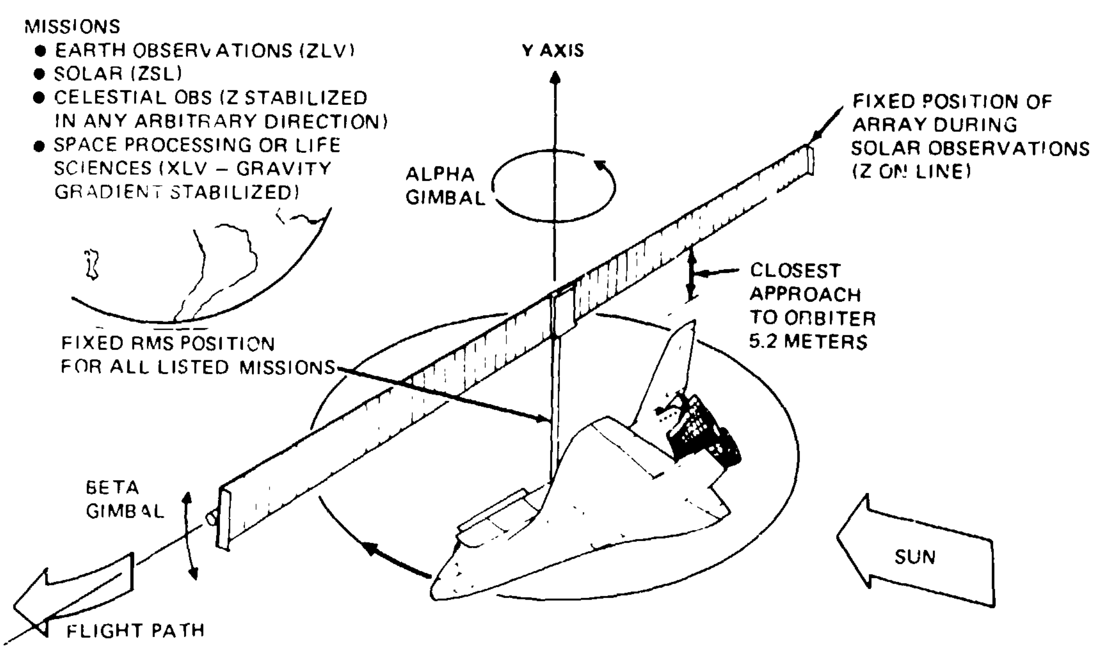


Figure 28. PEP Array Orientation – Y-POP Orbiter

Section 5
PROJECT DEFINITION

Early implementation of the PEP Project is needed to satisfy the energy shortages present on early Spacelab missions and to allow fuller utilization of the planned payloads on these missions in terms of increased data return and/or the addition of more payload equipment than could now be accommodated. Accordingly, the project schedule of Figure 29 has been detailed to allow early implementation of PEP. Key milestones on the schedule that lead to an IOC in the first quarter of 1983 are the Phase C/D RFP release in November 1979 and ATP by October 1980. The first of two PEP units is delivered 30 months after ATP with the qualification flight and IOC in April 1983. The second unit would be delivered in September 1983. This schedule is based upon the parallel development activities of both PEP and the necessary Orbiter accommodations.

The milestones of Preliminary Requirements Review (PRR), Preliminary Design Review (PDR), and Critical Design Review (CDR) will be achieved at 3, 8, and 17 months after ATP. The solar array design activity must begin early and be accomplished concurrently with the system design because it is the pacing development and production item. The pre-Phase C/D study (current study) has identified long-leadtime items and the necessary acquisition steps have been incorporated in the design and development planning and scheduling. This schedule can be implemented with minimal risk by closely monitoring the critical paths or events. Especially critical are the needs to set firm design requirements at 3 months (PRR) and the parallel design and development of the solar arrays and regulation equipment.

The real-year funding required to implement the PEP Project in relation to the PEP procurement schedule is shown in Figure 30. The total of \$87.6 M for delivery of two PEP systems is shown distributed across the major elements of the project. This funding plan reflects a relatively modest FY81 requirement of \$12 M. Yearly funding requirements peak at \$46.8 M in FY82 at a time when the basic Shuttle development funding is offloading.

Three technology items have been identified for early attention to permit the planned schedule to be met. The items, now being pursued as part of JSC RTOP activity, are:

A. Voltage Regulator. JSC will test the MDAC prototype and several other voltage regulators in their EPDC simulator.

B. Gimbal Assembly. Gimbal and slip ring requirements have been identified and preliminary design is in progress. Performance tests of a development unit slip ring assembly are planned by JSC on the EPDC simulator.

C. Solar Cell Assembly. Ongoing contracts are being focused to select cell type and geometry, develop the process steps, and qualify a pilot production line.

Preparations for the PEP Project have been completed to the level of detail needed to enter a Phase C/D development period. Specific project items that have been constructed are the design definitions, specifications, plans, work breakdown structure, master schedule, project cost data, and data requirement lists and data item descriptions. These items are illustrated in Figure 31.

The next step is implementation of the PEP Project. The successful development of this important extension to the STS capability will not only meet the immediate and future needs of Orbiter sortie mission payloads, it will provide a basic new solar array power system capability for application to a variety of future programs.

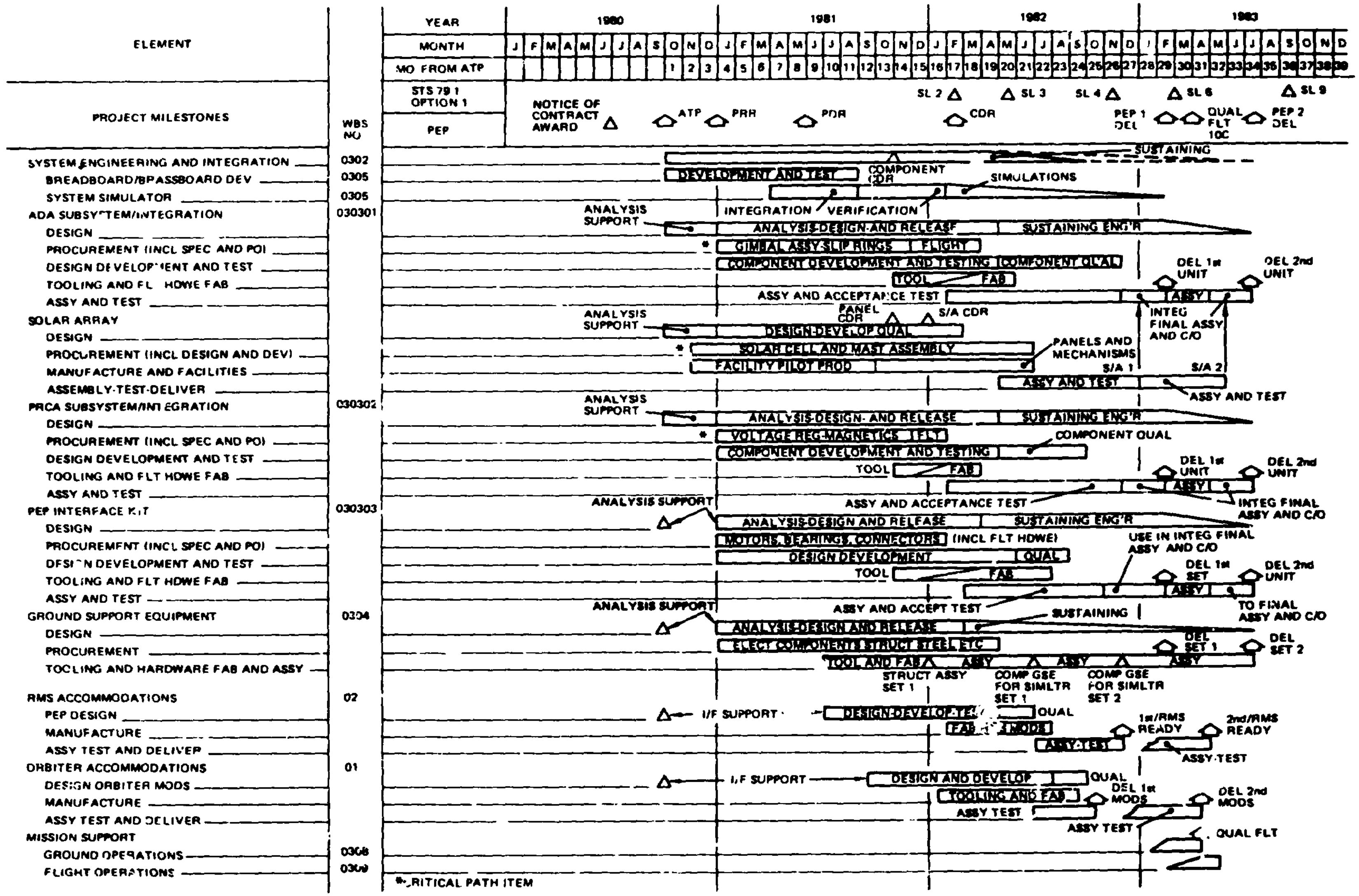
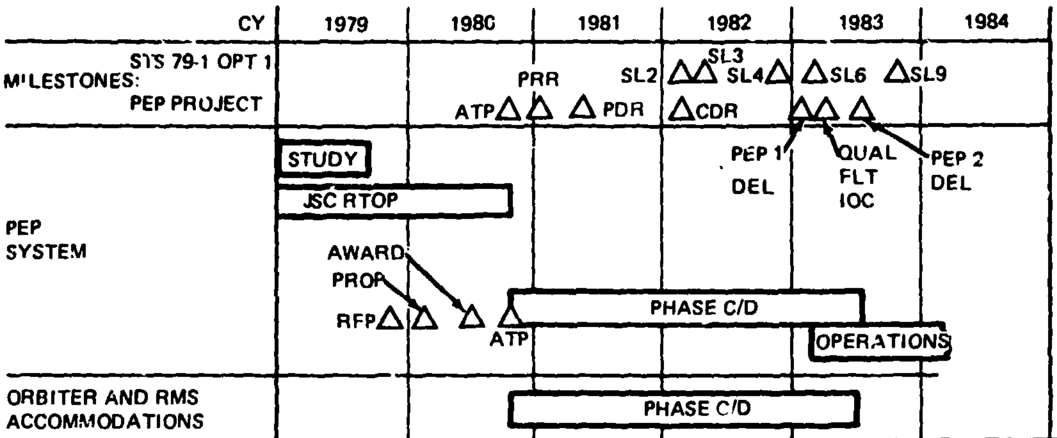


Figure 29. PEP Project Master Schedule



FUNDING RL YR $M	FY 79	FY 80	FY 81	FY 82	FY 83	FY 84	TOTAL
SYS/SUBSYS AND INTEG			5.6	23.3	14.0	–	42.9
SOLAR ARRAY			5.5	21.1	12.2	–	38.8
PEP SYSTEM			11.1	44.4	26.2	–	81.7
ORBITER AND RMS ACCOMMODATIONS			1.0	4.7	1.2	–	6.9
TOTAL PHASE C/D			12.0	49.1	27.4	–	88.6

Figure 30. PEP Project Schedule and Funding (Reference Configuration Planning Baseline)

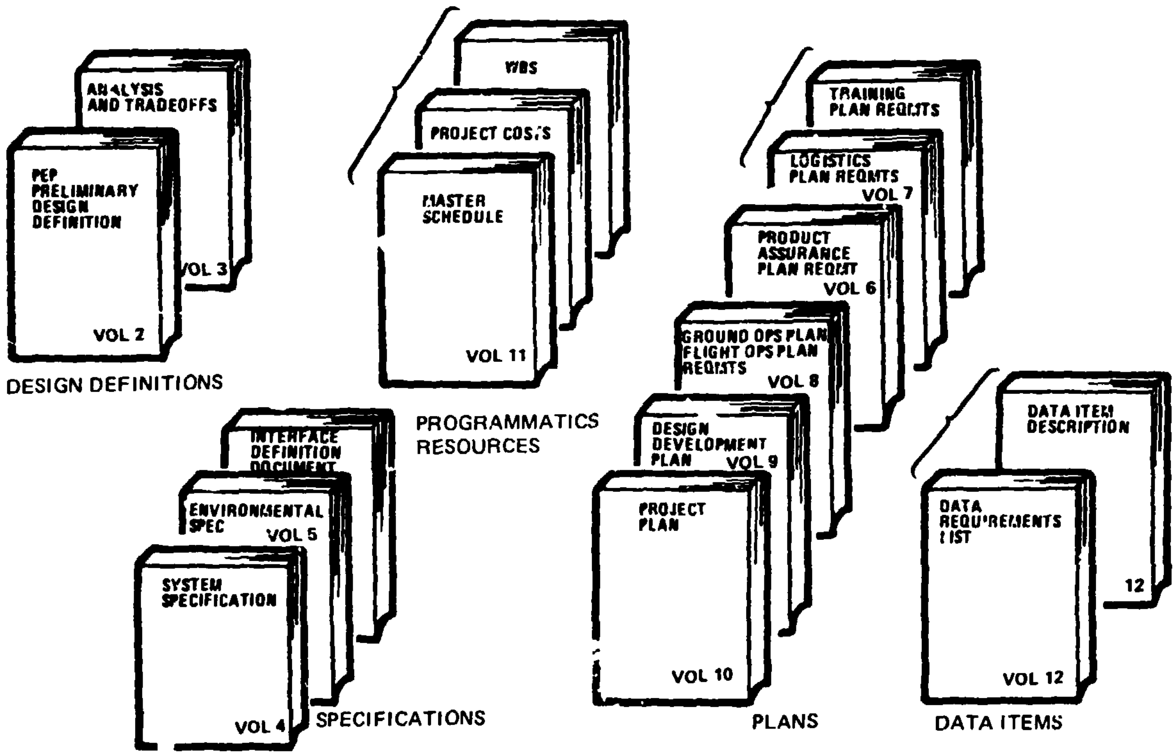


Figure 31. PEP Study Products