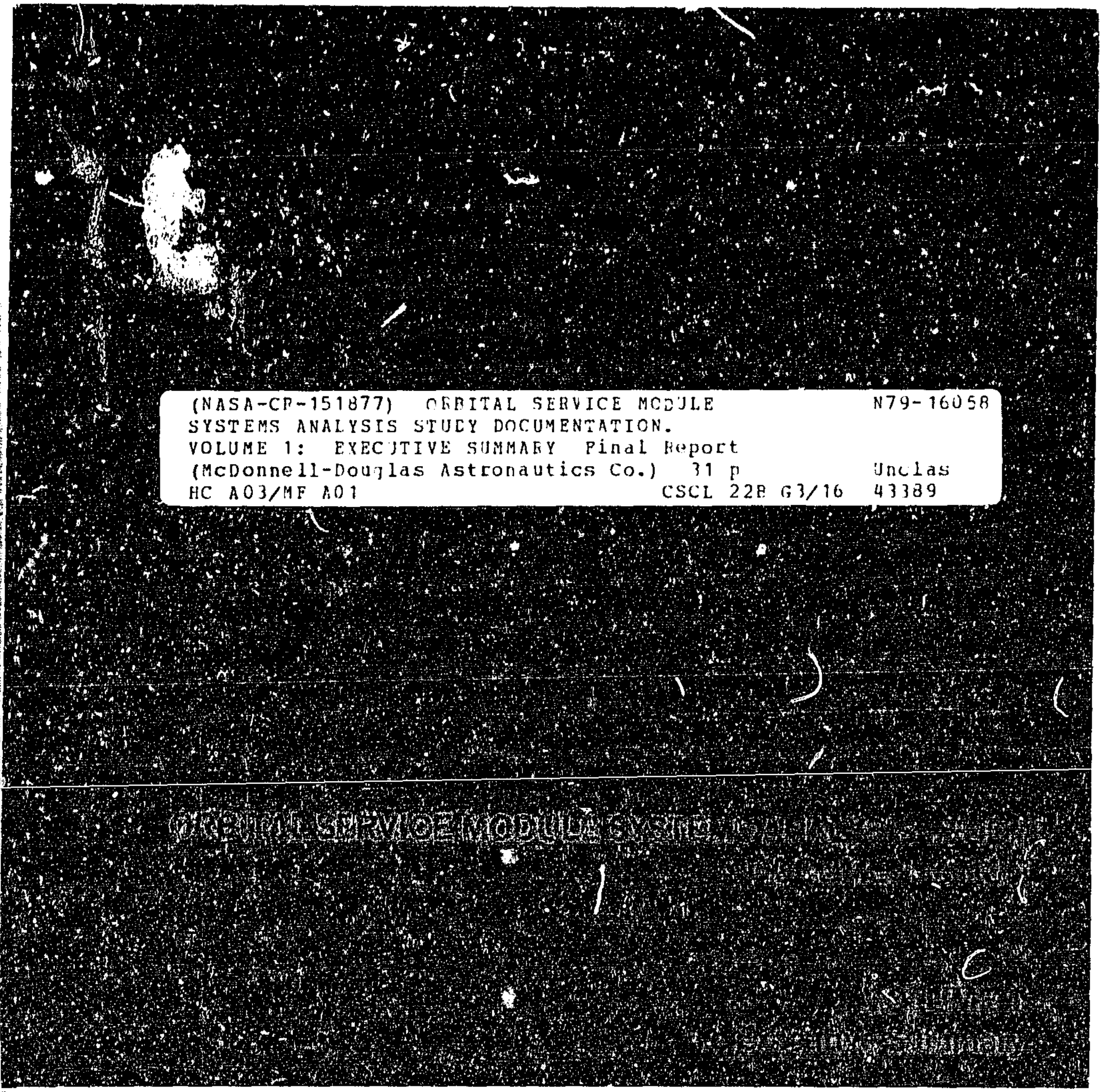

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PREFACE

The purpose of the Orbital Service Module Systems Analysis Study has been to investigate potentially feasible system concepts for providing additional power, thermal control, and attitude control to the baseline Orbiter in order to support a greater variety of space missions and to extend the Orbiter's ability to remain in orbit. The result of these analyses has led to an incremental growth plan that offers the flexibility of adding capability as, and when, it is needed in order to satisfy emerging user requirements.

The study consists of three documents:

Volume 1 Executive Summary
Volume 2 Technical Report
Volume 3 Program Plan

Questions regarding this study activity should be directed to:

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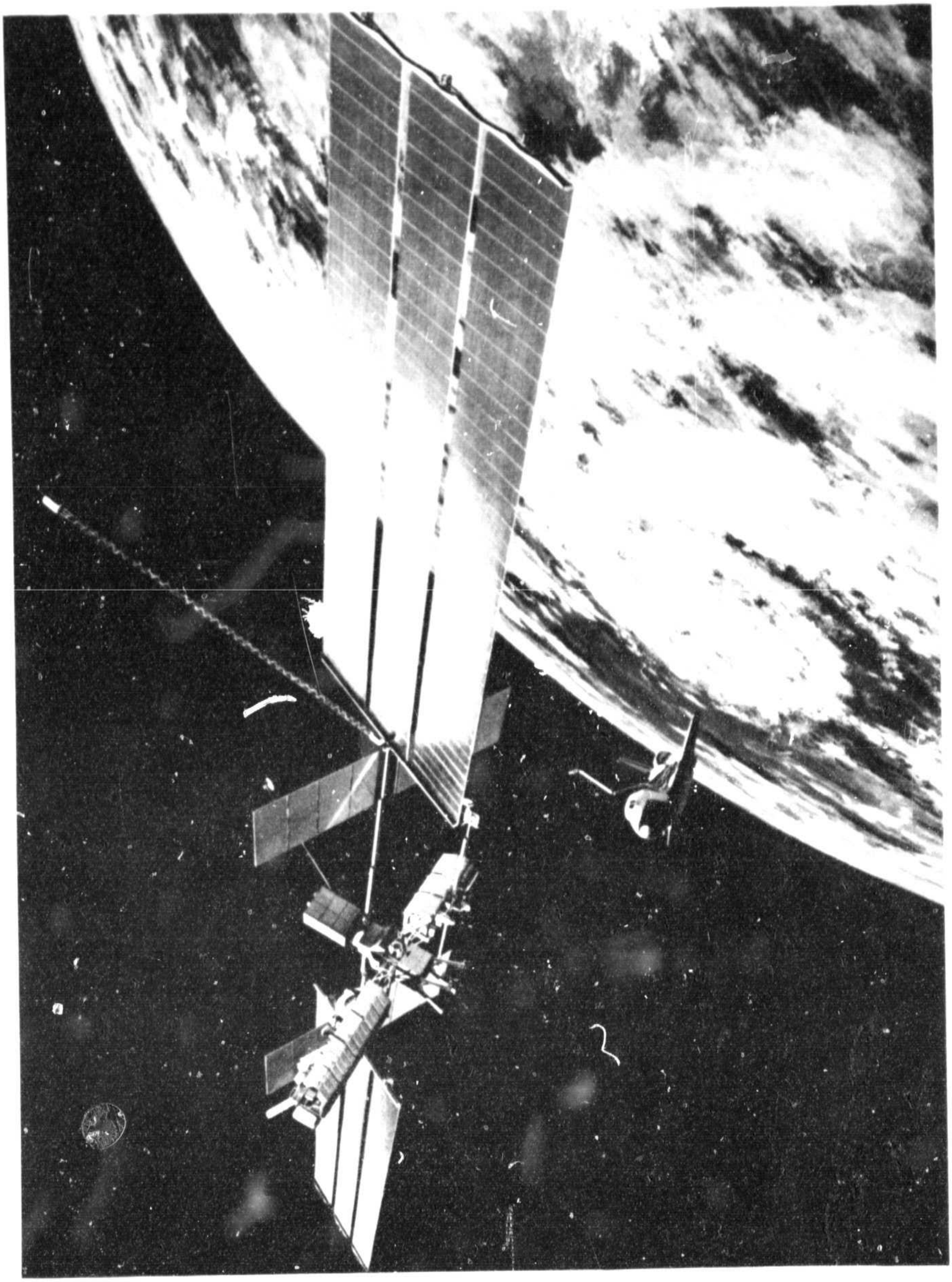

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INTRODUCTION

Payloads planned for use with the Orbiter will be designed to use the wide variety of services it offers. In addition to launch and recovery, this includes services such as power, heat rejection, data acquisition, communications, and attitude orientation. Analysis of potential missions for the 1981-to-1991 time period has indicated that the demand for these services in terms of quantity and the duration of their on-orbit availability is steadily increasing. To meet this increasing demand for orbital services, the capabilities offered by the STS must undergo an evolutionary growth.

The Orbital Service Module Systems Analysis Study examined the likely operational requirements for power and other services expected during the early years of Shuttle operation and attempted to define a realistic evolutionary program of capability growth responsive to the needs of users. In accomplishing this task, the Orbital Service Module (OSM) study addressed and answered the following questions:

- WHAT ARE THE PROJECTED REQUIREMENTS FOR ORBITAL SUPPORT SERVICES THROUGH THE NEXT DECADE?
- WHAT EVOLUTIONARY PATH OF SYSTEM DEVELOPMENT IS MOST RESPONSIVE TO USER NEEDS?
- WHAT INITIAL STEP SHOULD BE TAKEN?
- WHERE DOES THIS INITIAL STEP LEAD?
- WHAT DESIGN CONSIDERATIONS ARE KEY IN PROVIDING FULL SUPPORT CAPABILITY FOR THE PAYLOADS?
- WHAT SYSTEM DESIGN CONCEPTS MEET THE FULL-CAPABILITY REQUIREMENTS?
- HOW COULD THE FULL-CAPABILITY CONCEPT BE DEPLOYED AND WHERE WOULD IT NORMALLY OPERATE?
- WHAT ARE THE BASIC PROGRAMMATIC OPTIONS AVAILABLE FOR THE DEVELOPMENT OF FUTURE SYSTEM CAPABILITY?

In responding to these questions, the study confirmed the need for an immediate increase in the power and duration in orbit offered to payloads operated in the Orbiter-attached or sortie modes. Requirements for more power, and longer duration for payloads operating in future free-flying modes are less definite, but strong evidence supports the need for an autonomous power module capable of supplying multiple services to a variety of free-flying payloads in the post-1984 era. To meet these emerging requirements, this study postulated an initial Power Extension Package (PEP) to augment the power and duration capabilities of the baseline Orbiter. This would be followed, in 1984, by an autonomous power module with increased capability. On the basis of current projections of post-1984 requirements, the autonomous power module should provide a full range of orientation, communications, and heat-rejection capabilities commensurate with multiple payload support at a level of 35 kW or greater. Payload requirements and probable funding constraints need further definition, however, which may dictate the need for supplemental systems, including additional reduced-capability, lower-cost power modules. Accordingly, the study has also examined a small, PEP-sized, free-flying intermediate power module, as well as a somewhat larger power module using four of the PEP-sized solar array wings but providing more limited payload support than does the 35-kW full-capability module.

It is believed that the concepts and design considerations examined in this study and described in the technical report will provide the nation's policy makers with the background they need to formulate a fiscally responsible evolutionary program, supporting space applications, science, and exploration, while taking advantage of the flexibility of the Space Shuttle to reduce the cost of operating in space. Such an evolutionary program will ensure American scientific and technological leadership in space for decades to come.

WHAT ARE THE PROJECTED REQUIREMENTS FOR ORBITAL SUPPORT SERVICES THROUGH THE NEXT DECADE?

The point of departure for the Orbital Service Module Systems Analysis Study summarized in this document was the NASA STS Mission Model of October 1977. This mission model supplied mission, payload, schedule, orbit, and weight data for the early years of operation of the Space Transportation System. Supplemental sources such as the "Outlook for Space," SP-386, January 1976, the "NASA – 5-Year Plan, May 1978 (draft)," and the "Preliminary Definition and Evaluation of Advanced Space Concepts" prepared for the NASA OSTS by the Aerospace Corporation (Aerospace Report No. ATR-78 (7675)-1, June 30, 1978) were used to establish representative mission scenarios. Operational requirements for the period beyond 1984 are less precise because these later missions depend largely upon the results and experiences obtained during the initial years of operation.

The patterns of mission requirements observed in each of the research and applications areas were consistent in that more electrical power and longer stay times in orbit were desired as experience was gained and the missions became more sophisticated (see Figure 1).
To establish the most logical initial performance improvement increment beyond the current

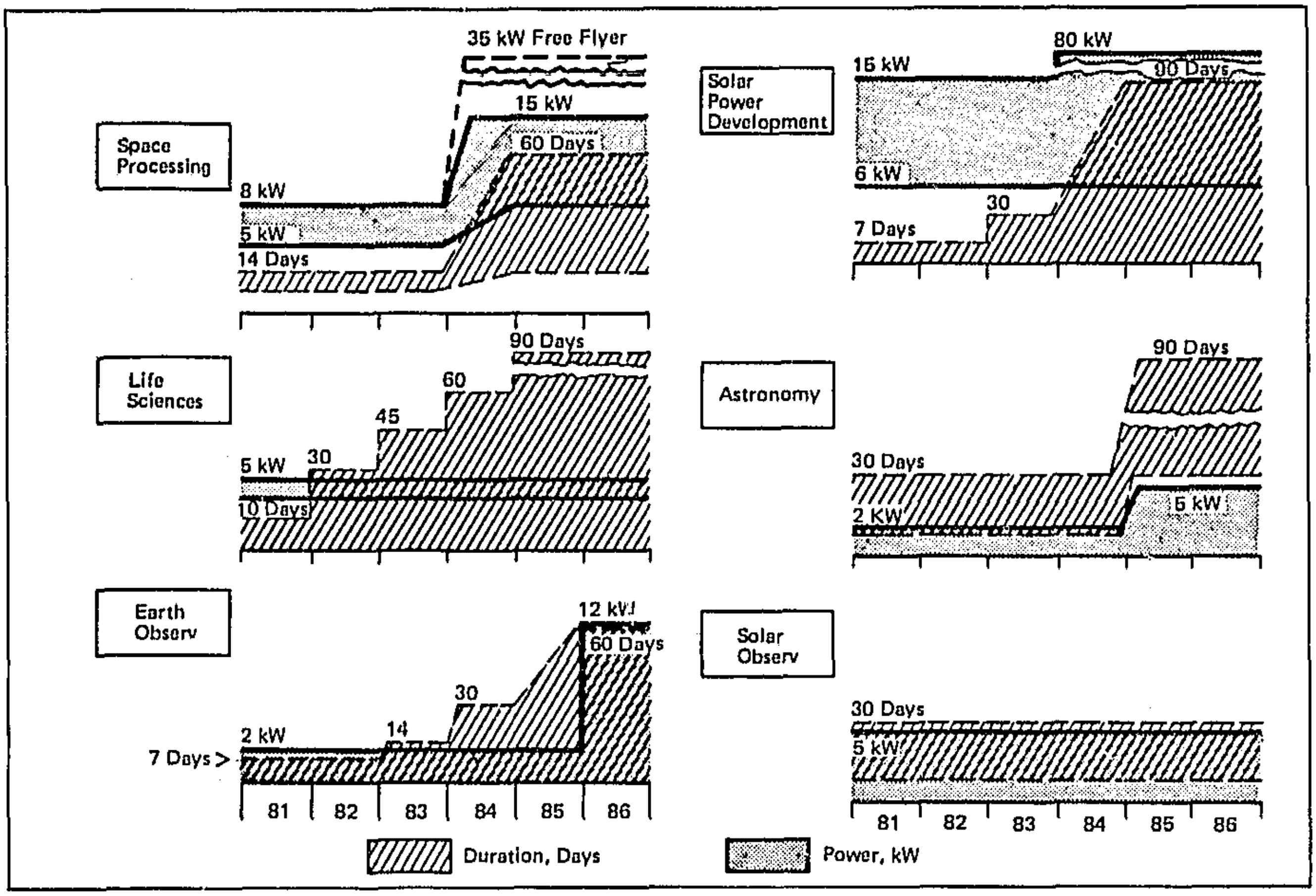


Figure 1. Growth Patterns in Six Research and Applications Areas

Orbiter capabilities, the power and duration requirements as described in the October 1977 STS Mission Model were analyzed for the Spacelab missions scheduled for the 1981-to-1984 time period.

Power requirements for each of the 49 Spacelab missions (1981 through 1984) scheduled in the October 1977 STS model are shown in Figure 2.

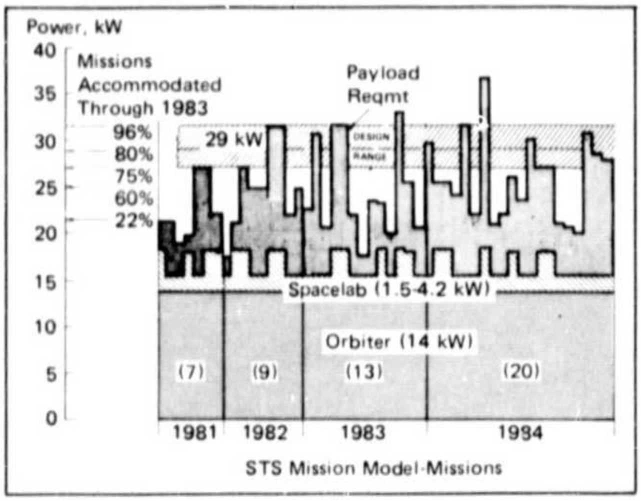


Figure 2. Power Requirements—STS Mission Model (Spacelab Missions)

They consist of 14 kW for Orbiter housekeeping, an assessment of 1.5 to 4.2 kW for Spacelab support (i.e., pallet, igloo, module), plus the power requirements for the complement of payloads carried on each Spacelab mission. The power needs were obtained from direct knowledge of the payload requirement or by correlating the identified payload with projected user requirements obtained from the previously mentioned sources. As seen, the totals vary from 17 kW to 33 kW in the first 3 years. The suggested design range is overlaid on these requirements, capturing between 75% and 95% of the 1981-to-1983 missions. A 29-kW value accommodates 80% of the missions as defined, or 23 of the first 29, a figure that would appear to be a proper balance between increased capability offered and utilization over all the missions. There is reason to believe that, for many missions, Orbiter power consumption may be reduced as a result of flight test experience and judicious selection of mission parameters. This and further refinement and consolidation of the mission objectives may make it possible to accommodate all the requirements within a 29-kW design value.

In Figure 3, the corresponding duration requirements are shown for each of the Spacelab missions scheduled in the STS mission model for the 1981 to 1984 time period. The duration requirements vary from 5 to 45 days for the first 3 years. The design range to accommodate 65% to 90% of the missions is indicated by the shaded area. Note that the requirements for increasing duration after 1983 create a rationale for a free-flying service module that can support payloads independent of an Orbiter.

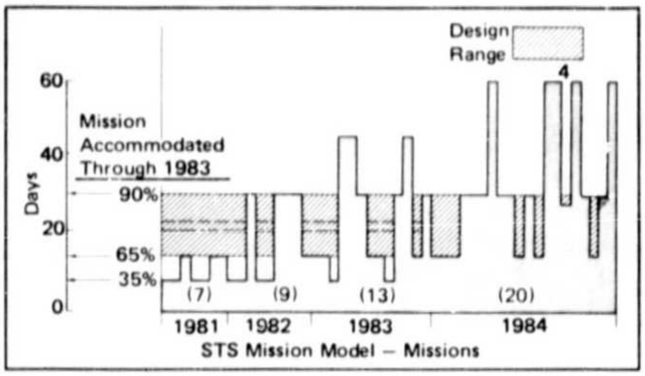


Figure 3. Mission Duration—STS Mission Model (Spacelab Missions)

Table 1 summarizes the user requirements in the 1984 to 1991 time frame for six key areas. Material processing is the dominant user of power, requiring levels ranging from 33 to 36 kW for second-generation research and development up eventually to 100 kW or more for production facilities in the 1990's. This area also requires long-duration operations approaching continuous. Any orbit and orientation would be satisfactory for materials research as long as its g-level requirement of 10^{-5} were not exceeded. On the other hand, life sciences research activities are seen as requiring fairly constant levels of power over the years, but missions of longer and longer duration will be desired to increase the data and observations that are essential to an understanding of the fundamental life processes. Similar patterns are observed in the research areas of earth observations, solar observations, and

Table 1. User Requirement Summary 1984 – 1991

	Materials Processing	Life Sciences	Earth Observ Comm	Astronomy	Solar Observ	SPS
Power, kw	33 – 36	2 – 9	6 – 12	5 – 14	5 – 13	10 – 25
Duration, Days	~ Cont	60 – 120	7 – 120	30 – 80	7 – 86	~ Cont
Inclination						
Any	X	X	– –	– –	–	–
28°	–	–	– X	X	–	X
55°	–	–	X –	X	X	–
Polar	–	–	X –	X	X	–
Orientation						
Any	X	X	–	–	–	–
Solar	–	–	X –	–	X	–
Stellar	–	–	–	X	–	–
Earth	–	–	X X	–	X	X
Stability, Sec	–	–	30	0.1	0.1	0.25 Deg
Crew	No	3 – 4	0 – 4	0 – 4	2 – 4	2 – 3
G-Level, g	10^{-5}	10^{-3}	–	–	–	–

astronomy, and in applications areas, such as communication systems and solar power satellite development.

In addition to missions dedicated to a specific area of research or application, an emerging need is seen for free-flying orbital service platforms providing simultaneous support for multidisciplinary payloads. The development of large "antenna farms" for multiple communications services has been suggested by representatives of the communications industry and NASA is currently investigating the requirements for large geostationary platforms that can support multiple communication and data relay antennas and earth observation sensors. A recent NASA/University Workshop (summer 1978) examined the desirability of a single low-earth-orbit platform that could provide support to multiple scientific users. In each case, the proposed service platform concepts studied were predicated upon the availability of centralized subsystem services.

The primary conclusion reached from the analysis of available mission model data and an analysis of the emerging requirements for orbital service platforms is that an immediate need exists for more power and on-orbit stay time for the Orbiter-attached (sortie mode) payloads. Furthermore, the need continues to increase with time. Free-flying payloads delivered and serviced by the Orbiter can be anticipated in the mid-1980's, and from 1984 on, both sortie mode and free-flying support capabilities will be required. In addition to the need for more power and longer orbital stay times, free-flyers will require extensions of current Orbiter capabilities, such as orientation, heat rejection, data acquisition, and communication. Multiple inclinations are dictated by the mission model data, and these requirements may be accommodated with multiple power modules or with a combination of the power module and Orbiter sortie (power extended) modes of operation. As the mission model data crystallizes, further effort will be needed to refine the definition of mission requirements.

WHAT EVOLUTIONARY PATH OF SYSTEM DEVELOPMENT IS MOST RESPONSIVE TO USER NEEDS?

Granted that the implementation of the NASA STS mission model will require more capability than is currently provided by the baseline Orbiter, the question is how can this capability best be provided?

As a precursor to this funded Phase A study, Johnson Space Center (JSC) established an in-house OSM Program Approach based on incremental growth in orbital services offered by the STS. Beginning with modest improvements to the basic Orbiter, particularly improvements in heat rejection, the JSC concept progresses through an initial solar array power supplement for the Orbiter to a family of free-flying power modules with increasing capability.

The original JSC incremental growth concept, shown graphically in Figure 4, provided the frame of reference for this study. The mission requirements analyses and design concepts developed in this study have verified the need for, and the feasibility of, an incremental growth concept for OSM.

Several scenarios can be postulated for evolving from the baseline Orbiter to one or more orbiting service modules. All scenarios begin with an increase in the basic power and orbit duration offered by the Orbiter to attached (sortie) payloads. The need for this is clearly established by the near-term mission requirements.

The initial increase in capability could be accomplished by extending the cryogenic capacity of the Orbiter or by adding an Orbiter-attached solar array system. This study examined in detail the tradeoff between these alternatives.

Figure 5 plots (in dotted lines) the number of cryogenic tank sets required to provide 7 kW of power for payloads as a function of time in orbit. As an example, 14 cryogenic tank sets would be required to provide 7 kW for 23 days in orbit. Such a solution would reduce the available payload weight of the standard (four cryogenic tank sets) Orbiter by 20,000 lb to less than 9,000 lb.

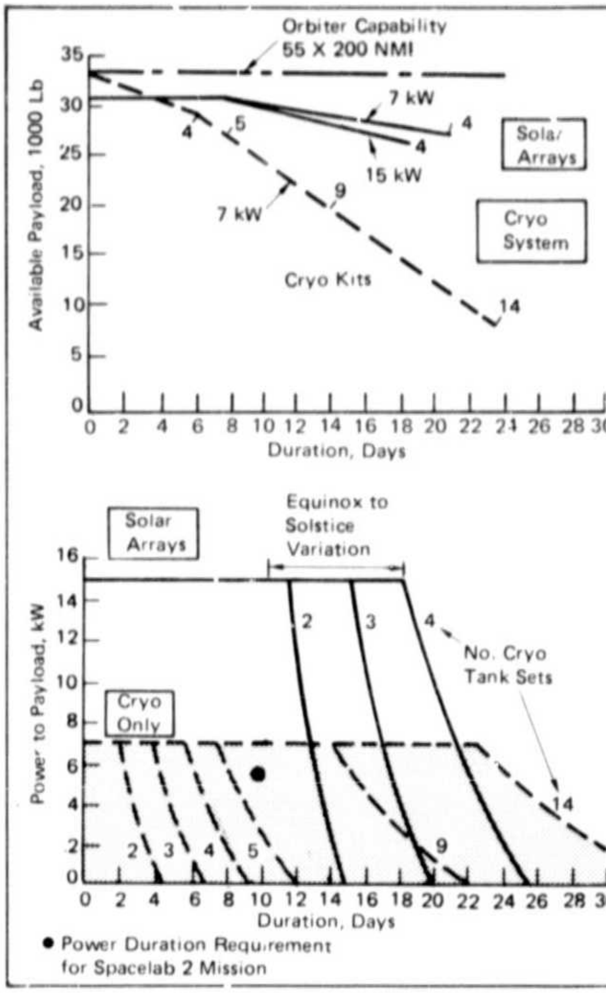


Figure 5. Orbiter Performance Improvement

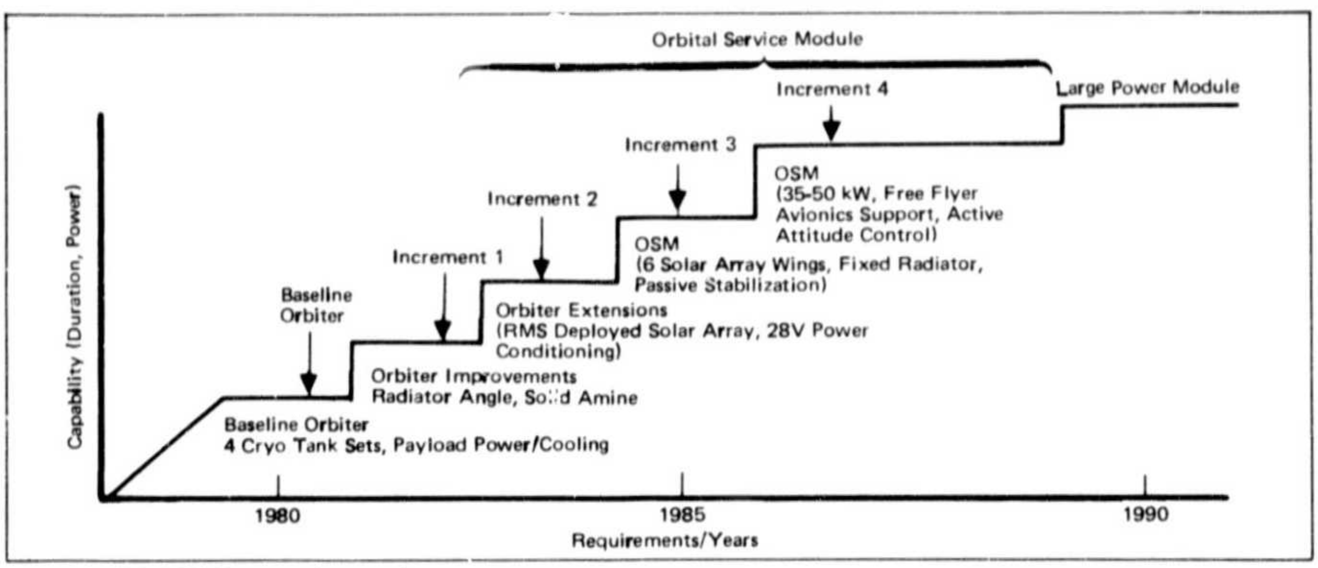


Figure 4. JSC OSM Program Approach

The solid lines in Figure 5 describe the performance envelope of a solar array configuration sized to satisfy the requirements outlined for the 1981-to-1984 time period. The output from solar arrays and the mission duration possible will vary from a minimum at equinox to a maximum at solstice, and the envelopes plotted illustrate the maximum performance capability with solar arrays supplemented by two, three, or four cryogenic tank sets. As a point of reference, the dot on the plot of power vs duration identifies the design requirement of Spacelab 2. As can be seen, the mission would require five cryogenic tank sets to achieve a mission duration of 8 days. To achieve 9 days, the power supplied could be no more than 5 kW. On the other hand, solar panels and only two cryogenic tank sets would be more than adequate to meet the demands of the latter mission. The obvious conclusion from the data plotted is that the way to achieve missions of longer duration with adequate power and still have significant working payload capability is to rely more on solar power. Solar arrays appear to represent the best solution within today's state of the art.

The solar power extension package (PEP) represents the first step in the evolutionary growth pattern. As an extension of the basic Orbiter capability, this step offers increased power and increased duration, while retaining all the other Orbiter resources and services offered to users.

The next step is to extend this capability to an autonomous, continuously orbited service module. Payloads, such as applications modules, would be transported to the orbiting service module by the Orbiter. After supporting initial checkout of the payload, the Orbiter would be free to support other orbital missions or return to the recovery area for the next payload.

As an example of the need for such a service module, materials processing missions planned for the period require essentially continuous operation, but only occasional visits by the Orbiter to retrieve processed materials and set up new or continued processes, because the operations can be controlled and monitored from the ground. Other applications, such as solar and earth observations, also can effectively use an extended-duration mode. Hence, development of an Orbital Service Module (OSM) that can simultaneously support multiple-applications modules independent of the Orbiter will increase the effectiveness of NASA's Orbiter fleet because it will allow more frequent flights with a given fleet size.

The size and functional capability of this next autonomous OSM step will be influenced primarily by user needs and funding constraints. Based on current knowledge of projected mission requirements, this study has identified a "full-capability," multipayload OSM in the 35-kW range as a candidate solution to meet the full range of payload requirements. One such full-capability concept is shown in Figure 6. Also shown is a small, intermediate-capability concept and a larger "limited-capability" concept. Both of these represent size, capability, and cost increments leading toward the full-capability system.

Further analysis of mission requirements and funding constraints is needed to firmly establish the pattern of evolutionary steps leading to full capability. Meanwhile, the study has ascertained that such steps are feasible and that a high degree of element commonality can be preserved.

The steps beyond the PEP need not be firmly defined immediately for two reasons: (1) development of the array (and the manufacturing capability necessary to produce it) significantly reduces lead times required to produce a free-flying OSM, and (2) the addition of the PEP capability to the Orbiter-Spacelab allows an aggressive space program to be pursued in the early 1980's.

WHAT INITIAL STEP SHOULD BE TAKEN?

Based upon the analyses conducted to date, the initial PEP should provide 29 kW, 15 kW of which should be available to the payload. The package should be designed to accommodate mission durations of 19 to 21 days, be capable

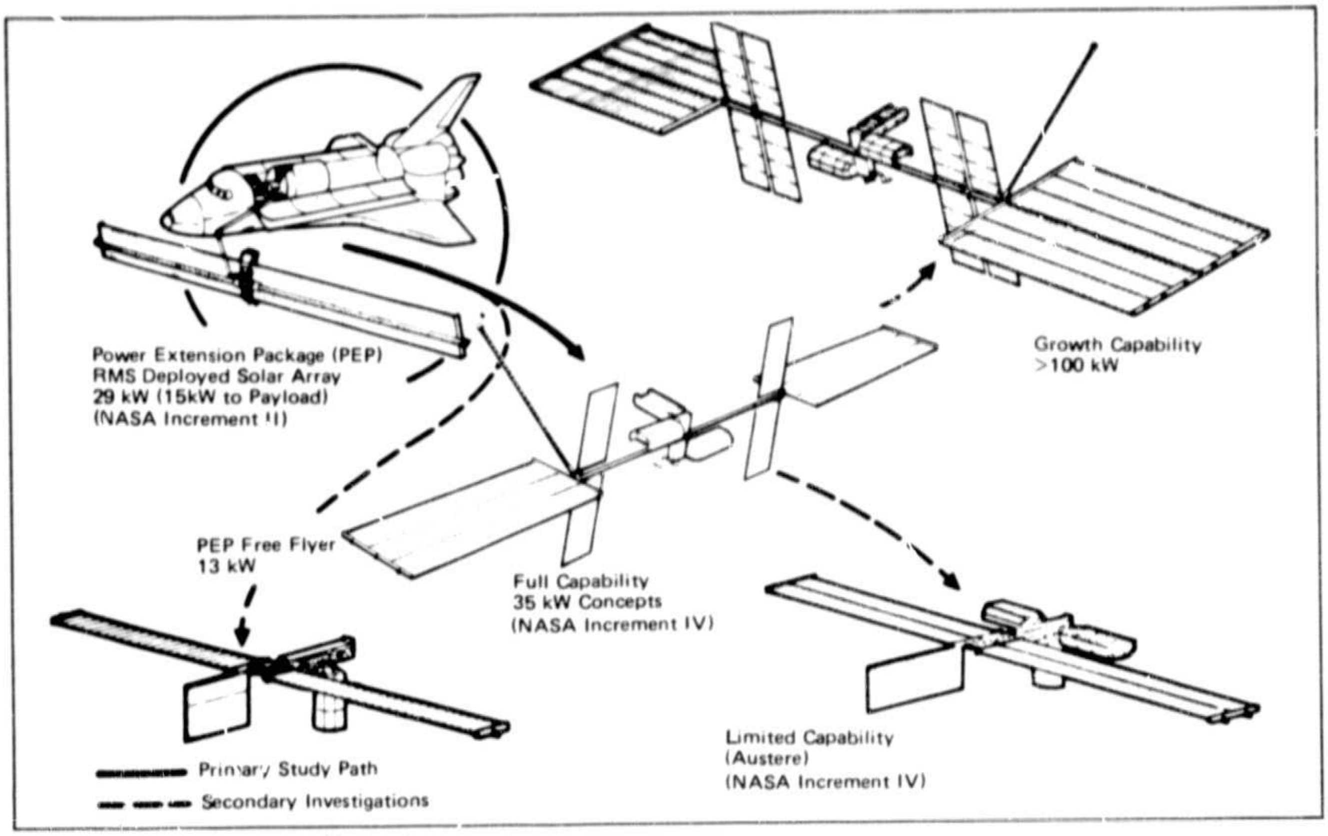


Figure 6. OSM Concepts and Derivatives

of multiple orientations, and be predicated upon existing technology. This suggests the use of solar arrays developed under the Solar Electric Propulsion Stage program and the technology of existing Orbiter systems insofar as possible. The Orbiter remote manipulator system (RMS) offers a highly flexible means for deployment and positioning of the solar arrays. Figure 7 portrays one concept that meets these requirements.

In the PEP concept, the solar arrays provide most of the power (26 kW) while the Orbiter is in the sun, and the standard Orbiter fuel cells provide all the power on the shade side of the orbit. The fuel cells (three are currently used to provide electrical power to the payloads) idle at 3 kW (1 kW each) during the sun-side operation. The combination of solar arrays and fuel cells provide a continuous capability of 29 kW. Thus, the minimum design modifications are required to existing systems and minimum initial (scar) weight to the Orbiter. By use of a voltage regulation subsystem, excellent response to peak loads and load sharing can be obtained. Switching is avoided, and Orbiter load interaction and interference with experiments are minimized.

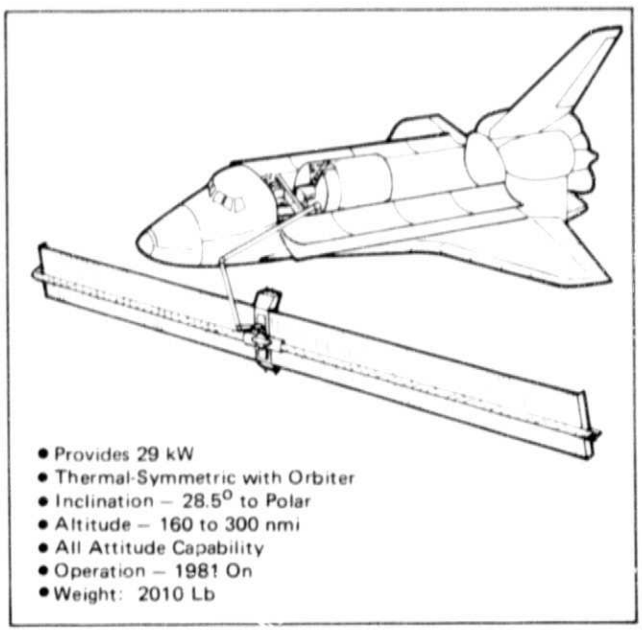


Figure 7. Power Extension Package

Table 2 summarizes the baseline characteristics of the initial PEP. Figure 8 portrays the method of interface with the RMS and the Orbiter. The

Table 2. PEP Baseline Characteristics

Power and Duration:	• 29 kW, 19 Days • 21 kW, 21 Days
Array Size:	• Two SEP Type Wings, 4.0 Meters X 38.3 Meters Each
Storage Location:	• Over Spacelab Short or Long Tunnel • Standard Orbiter Attachment • Aft Location Optional
Deployment:	• Remote Manipulator System (RMS)
Array Rotation:	• Separate Gimbal/Torquer Drive • RMS Inactive Except During Orbiter Maneuvers
Heat Rejection:	• Uses Orbiter Radiators • Flash Evaporator Supplement — for Some Orientations
Output Voltage:	• Per Orbiter Specs

PEP kit stowage in the Orbiter cargo bay resul in no loss of available payload volume. The package easily fits into the forward area between the airlock and the Spacelab, as shown, and is designed to fit anywhere in the cargo bay using standard attach fittings. The two-mast canister for deploying the arrays and the two-blanket boxes are shown in the stowed position in the upper part of Figure 8. The linkages are designed to rotate the canisters 90 degrees when the mast begins to emerge. The array module and the equipment support beam may be easily removed from the Orbiter when they are not needed for a mission or for maintenance. The RMS connection to the solar array is made through a standard grapple connection over the two-axis gimbal system of the array. In the lower and center part of Figure 8, the array is shown in its deployed position, with the two extendable masts deployed from their initial storage canisters. Each wing of the array is 4 meters wide and 38.6 meters long.

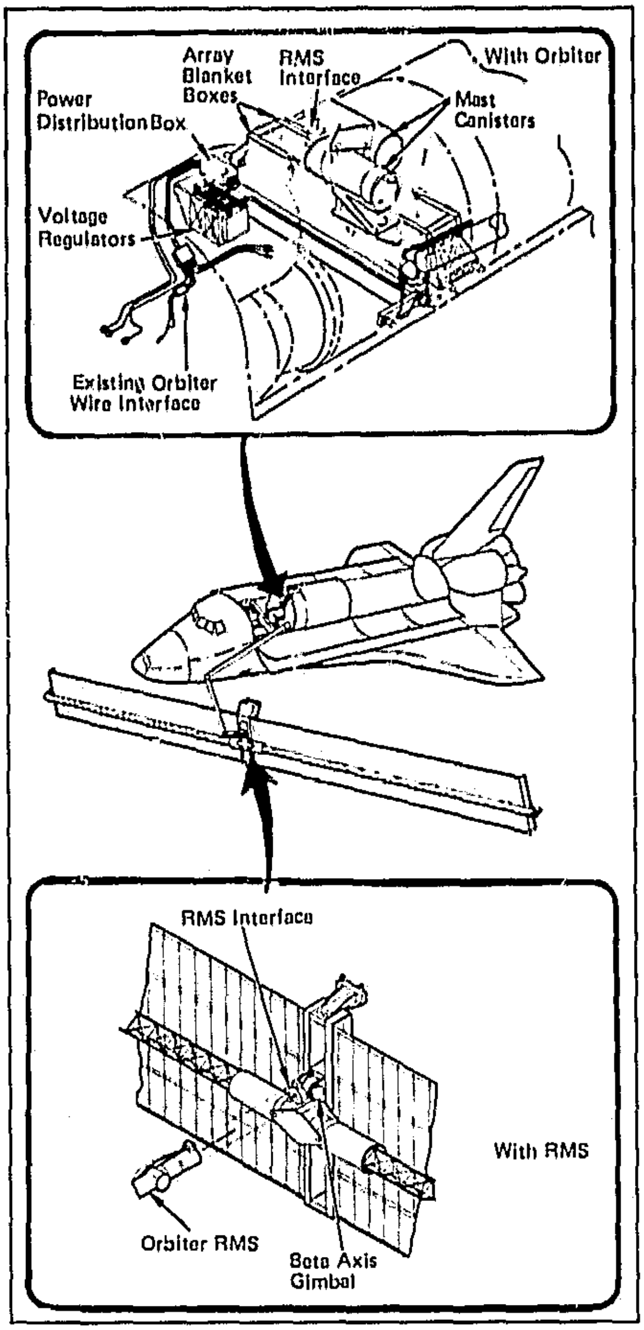


Figure 8. PEP Interfaces

Figure 9 summarizes the performance of the initial PEP system. Each of the three fuel-cell power plants on the basic Orbiter provides 7 kW continuously, or a total of 21 kW. The Orbiter requires 14 kW, leaving 7 kW for payload systems. A review of the 49 Spacelab missions scheduled for the period between 1981 and 1984 in the NASA mission model has indicated the use of Spacelab pallet, igloo, and module configurations; therefore, between 1.5 and 4.2 kW are required for basic Spacelab support during periods of its operation. Thus, the Orbiter with the Spacelab module, under normal operating conditions, would allow about 3 kW for payload operations (see Figure 9). A 29-kW PEP system would provide 14 kW to the Orbiter, 4 kW for the Spacelab module, and 11 kW for payload operations. In brief, the payload requirements for power and duration during the initial years of operation of the STS are fairly well established. The initial PEP as

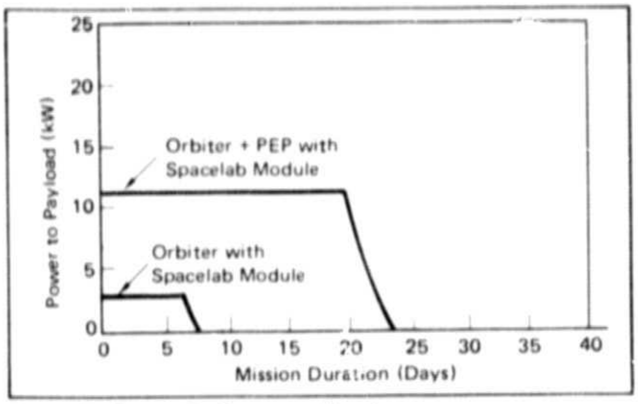


Figure 9. Power Performance Envelopes (Orbiter Plus PEP, 55 Deg X 250 mm)

described will be of enormous value in providing increased power. Advantages include the following:

A. Orbiter turnaround time is less than with an all-cryogenic solution.

B. Payloads can be increased because of the large weight saving of the solar array approach for Spacelab missions as compared with the cryogenic system.

C. Greater flexibility will be provided in overall STS operations planning and scheduling.

No problems of technical feasibility have been identified to date and none are anticipated, inasmuch as the basic system is predicated upon current technology and hardware already under development.

Studies have shown that the PEP concept is compatible with the shared RMS, and is compatible with the basic Orbiter mission in weight, volume, and center of gravity. The development schedule appears to be compatible with the schedule of Spacelab 2, and it provides a discretionary payload advantage of 3,000 lb for that mission. Implementation of PEP allows the flexibility of an Orbiter with only two cryogenic tank sets to accommodate short-duration, high-payload performance missions without PEP; then, with PEP, to offer high power and long-duration capability on other missions. It is NASA's present intention to initiate program activity in FY 79 in order to start actual Phase C/D development in FY 80. On this basis, the initial system could be available to support Spacelab 2.

WHERE DOES THIS INITIAL STEP LEAD?

Beyond the capability in the 1981-to-1984 time period provided by the initial PEP concept, the need for further growth steps can be predicted. Table 3 summarizes the design requirements and principal design considerations for a full-capability power module in the post-1984 time period as derived from the current NASA mission model. Present plans indicate that higher

Table 3. Mission-Derived Design Requirements

Full Capability OSM Function	Increment IV Requirement	Power Module Key Design Considerations
Power, KW	35-40	Power Output
Duration, Days	Continuous	Orientations
Thermal, KW	Symmetric	Gimbal Requirements
Inclination, Deg	28.5,57, Polar (28.5 Nom)	Control Sizing
Altitude, NMI	180 - 300 (220-235 Nom)	Field of View
Operational Time Period	1984 On	Radiator Size, Location
Orientation	All Attitude	Plume Effects
Stability	0.4 Sec - 0.1°	Payload Clearance Envelope
Acceleration Level	10^{-5} G	RMS Capabilities
Berthing/Docking Ports	4-6	
Interface Compatibility		
• Orbiter	Yes	
• Multiple Free Flyers	Yes	
Orbit Keeping Interval	60 Days	
Comm/Data	To 10 MBPS	

power levels (35 to 40 kW) and continuous missions will be required. Mission objectives call for various orbit inclinations and require all attitude orientations, low g levels, and the capability for docking and undocking payloads delivered by succeeding Orbiter flights.

Figure 10 illustrates some of the alternative configuration concepts for an advanced OSM. The full-capability OSM or advanced power module may assume a number of logical geometries. One of the most important considerations is the principal axis orientation of the module with respect to the orbit plane. The orientation selection will have a significant effect on gravity gradient torques and sizing and on the saturation of the control moment gyros (CMG) used for attitude control. The symmetry or asymmetry of the mass distribution with regard to the array axis will have a similar impact on momentum buildup and CMG saturation. Location of berthing ports can also impact control system sizing because the att[illegible]ent of modules will result in various m[illegible]tributions. Optimally, it is desired to maintain all mass centers of grav[illegible] as close to the solar axis as is operationally reasonable.

Three generic types of configurations are shown in Figure 10. The symmetric concept is characterized by a central subsystem core assembly with attached payloads, separated array wings to provide clearance for payload orientation and the Orbiter when berthed, and geometric as well as mass symmetry to minimize control disturbances. The long axis of the solar array is in the orbit plane and perpendicular to the sun line. A five-berthing port cluster provides all attitude capability for the payload modules by providing two gimbals and 2 degrees of freedom relative to the solar array. This concept would also have an Orbiter berthing port on at least one end of the central core structure. The concept would provide full power in any payload orientation. Radiators would be mounted perpendicular to the arrays on the array support booms. The central mass plus deployable ballast could be used to minimize the need for CMG desaturation. The concept would have unpressurized subsystems and berthing accommodations.

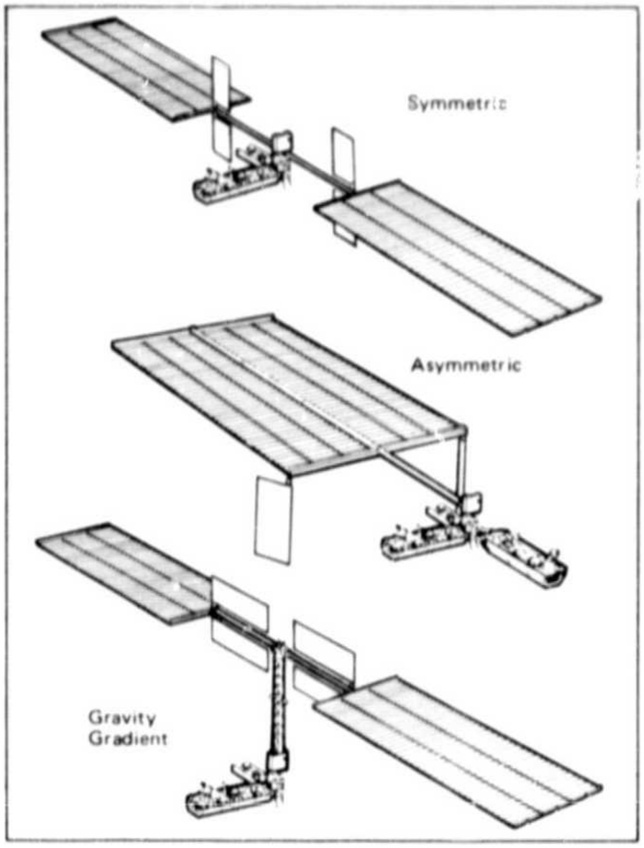


Figure 10. Full Capability-Major Configuration Alternatives

It would be developed from six of the wing and mast assemblies used in the initial PEP design.

The asymmetric concept also would use a multiple-berth, all-attitude payload cluster with one Orbiter berth fixed to the payload-subsystems cluster. This concept would offer the best field of view and tend to minimize stray light reflections on the potential payload sensors, but would have the highest moments and total bias momentum. The asymmetric concept is shown with representative pallet-type payload assemblies, a six-panel wing with a telescoping mast, and array-mounted radiators.

The gravity gradient concept separates the two main mass assemblies (array/radiators and subsystem/payloads) to provide gravity-gradient-stabilized orientation with respect to local vertical, primarily to enhance earth viewing.

These concepts are representative of the configurations examined to derive design characteristics most suitable for a full capability OSM. Comparative analysis was used to segregate the design considerations with most significant impact on performance, cost, and service to the payloads and to provide design guidelines leading to an optimum configuration.

WHAT DESIGN CONSIDERATIONS ARE KEY IN PROVIDING FULL SUPPORT CAPABILITY FOR THE PAYLOADS?

In evolving full-capability support for free-flying payloads, several design considerations were of major concern: (1) to maximize the *field of view* of the payloads; (2) to minimize the effects of *plume impingement*; (3) to provide adequate *payload clearance* for orienting and positioning; (4) to minimize disturbances that affect *control system sizing*; (5) to design for deployment within the reach envelope of the *remote manipulator system*, (6) to extract the maximum *power output* from the costly solar array; and (7) to design for the optimal *radiator area/performance*.

Field-of-View Effects

A computerized analysis was used to assess the field-of-view offered by each of the candidate configurations. Figure 11 illustrates the geometry of a typical earth-viewing situation. In this case, the symmetric type of configuration is shown. The computerized analysis provided several critical characteristics valuable in comparing the field-of-view of each configuration. These included percentage of the hemispherical solid angle instantaneously obscured by the OSM major elements, percentage of the hemispherical field of view subject to obscuration during an orbital angle, shape of the obscurations, and time required for the obscuration to sweep the field-of-view of the observer.

Figure 12 illustrates the observations seen by an observer on the symmetric OSM configuration as he looks toward the nadir, with the OSM traveling in a solar-inertial-orientation and the array axis in the orbit plane. Three glimpses of the obscuration are seen: one radiator as it enters the field of view, the edge-on view of one array wing and radiator as OSM passes the terminator, and a radiator as it leaves the hemispherical field of view.

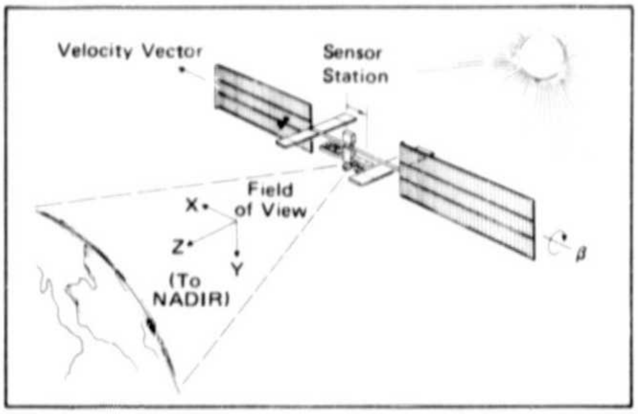


Figure 11. Field of View Effects – Earth Viewing Payload

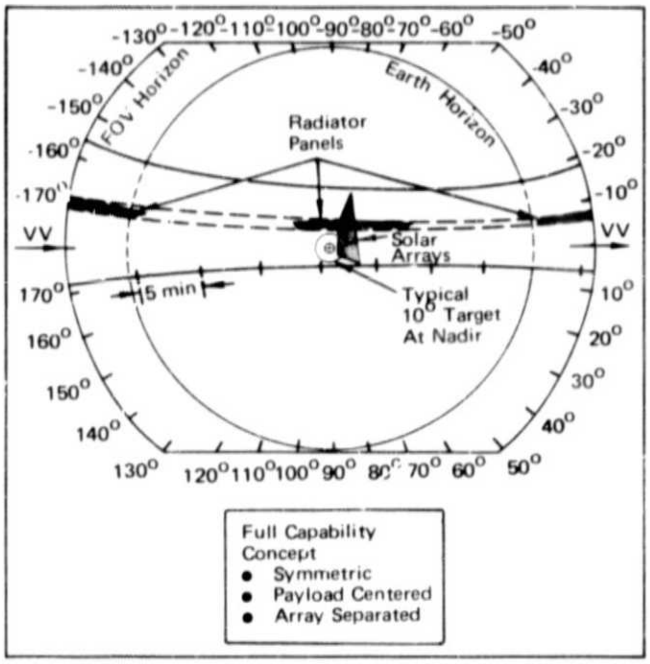
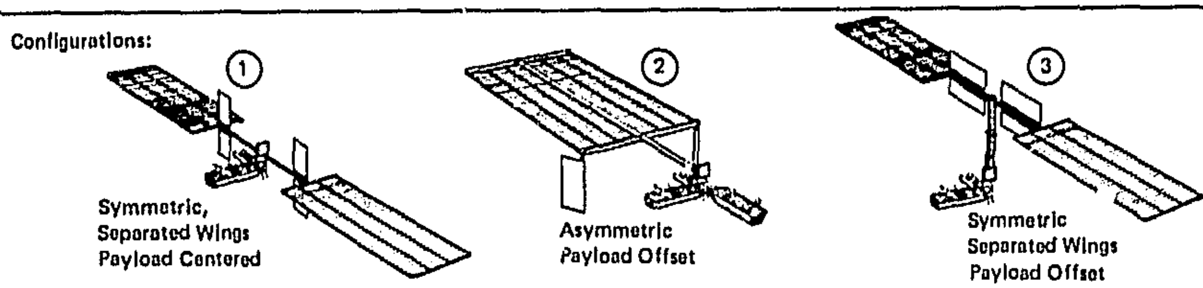


Figure 12. Sensor Field of View - OSM Concept 1 Earth Viewing Payloads

To fully assess each configuration concept, the parameters of interest were varied, including configuration geometry, orientation, location of the observer (sensor) from the center coordinates, and viewing direction.

Figure 13 illustrates the field-of-view study results using the three basic configuration geo-



Configuration/ Orientation	Target	Instantaneous FOV Obscured	FOV Subject To Obscuration
1 Solar Inertial	Earth	0.8% Twice/Orbit	22%
Pop - Local Vert	Earth	0.8% Twice/Orbit (*)	11% (*)
2 Solar Inertial	Earth	1.8% Once/Orbit	55%
Pop - Local Vert	Earth	0.8% Twice/Oribt (*)	7% (*)
3 Pop - Local Vert	Earth	0	0
Pop - Local Vert	Celestial	16%, Once/Orbit	85%

(*) Can be reduced to Zero By Locating Observation Point Farther From Array Centerline.

Figure 13. Field of View Effects Observation Point 4M From Center

metries examined. Two combinations of vehicle orientation and viewing direction are shown for each of the three configurations.

Solar-inertial orientations obviously offer clear fields of view for solar observations and large unobstructed fields for celestial observations. With the array axis in the orbit plane (IOP), the OSM obstructs the field of view during earth observations. With the symmetric design, this occurs twice per orbit, as each half sweeps through the field. With the asymmetric, this occurs once per orbit. By orienting the long axis (Y) of the array across or perpendicular to the orbit plane, and aligning the body with the local vertical, a clear view of the nadir is obtained over the entire orbit. Of course, celestial viewing may be impaired. The extent of this is illustrated by the Concept 3 data.

From these analyses, the symmetric configuration offers reasonable viewing opportunities when operated solar-inertial (array axis IOP) if two gimbals are used to permit payload orientation. The asymmetric concept offers a wider unobstructed view angle and minimizes the probability of reflected radiation entering the field of view. The gravity gradient concept offers excellent earth viewing but has major obstructions for celestial observations.

Plume Impingement

RCS plume effects are an important consideration in configuration design. Plume impingement during Orbiter rendezvous and departure can exert excessive pressure loads on the arrays, and bending loads on their support structure, and can cause contamination and erosion, local heating, acceleration of the OSM away from the arriving Orbiter, and differential loading or turning moments that disturb the OSM orientation. Figure 14 shows typical pressure contours from RCS forward thrusters. Similar patterns result from the aft thrusters. Separation of the array wings helps to minimize these effects. For this reason, separated wings are preferred to joined wings.

In this illustration, outline profiles of three basic concepts -- symmetric, asymmetric and gravity gradient -- are shown. Calculations for the symmetric case indicate an array separation distance of approximately 140 feet is needed. For the same Orbiter approach angle, the other configurations experience higher plume pressure

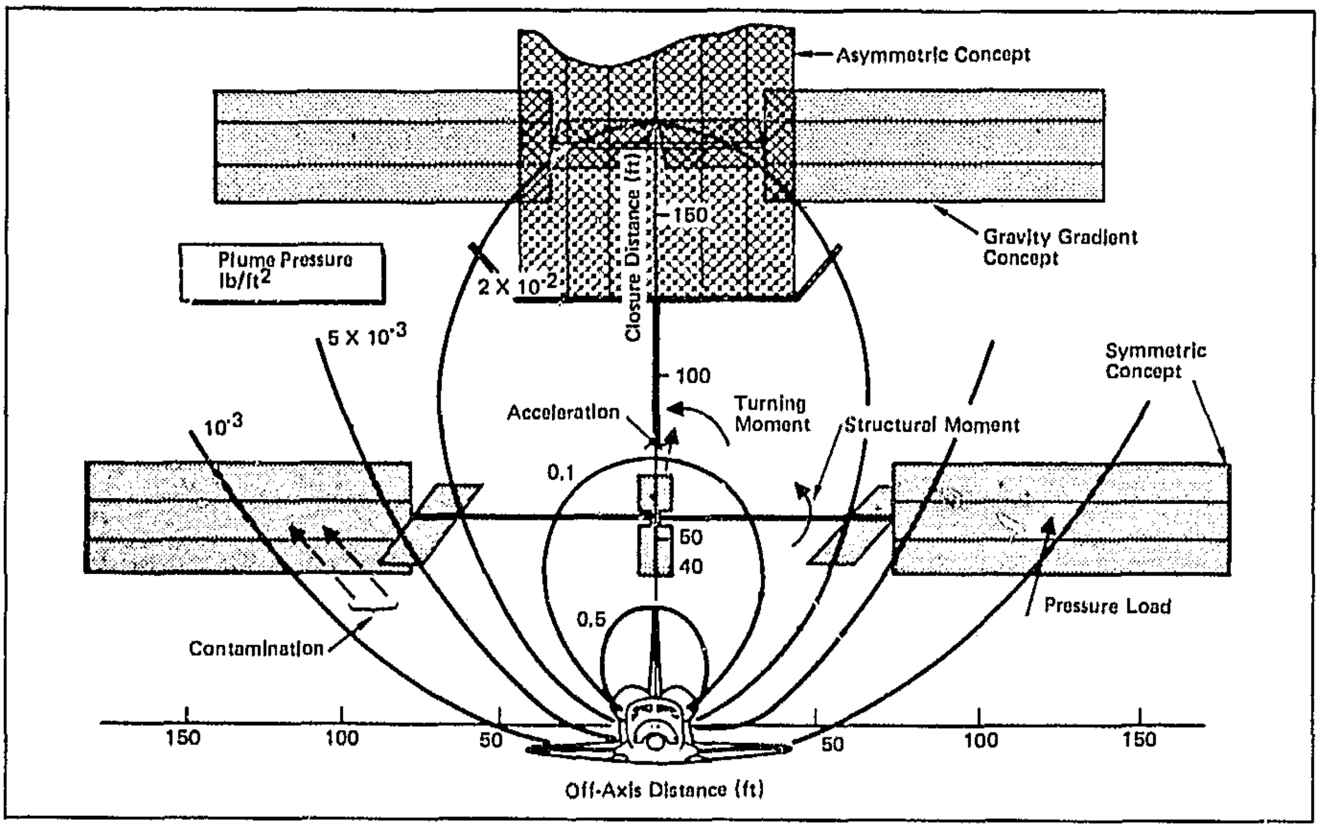


Figure 14. RCS Plume Pressure

unless the separation distance between berthing port and array surface is increased to about 175 feet.

Studies are currently in progress to assess the effects of plume impingement using the X-axis thrusters for rendezvous braking. Although more propellant is consumed, plume loads on the OSM solar arrays are reduced by more than an order of magnitude.

Payload Clearance Envelope

Payload orientation requirements critically impact configuration design. With the array solar oriented, the design should permit orientation of the payload module cluster or individual payloads toward any viewing objective. As illustrated in Figure 15, a clearance envelope requirement of a radius of approximately 20 meters results. This dimension is based on a payload using the maximum length of the Orbiter payload bay. This establishes the minimum separation on the symmetric concept between the solar arrays and between the solar array and the payload cluster on the asymmetric concept.

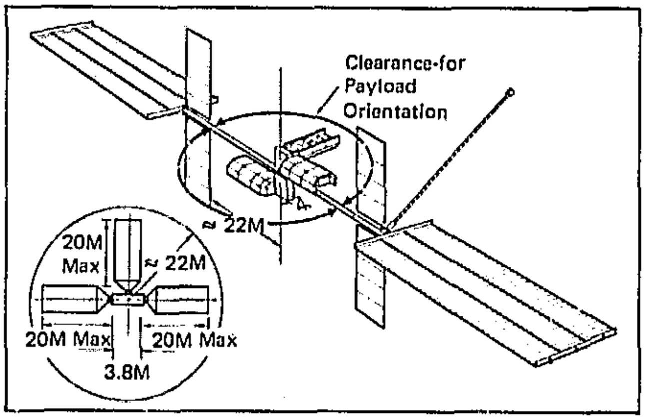


Figure 15. Payload Clearance Envelope

Control Effects

In establishing the design requirements for the control system, the configuration geometry and desired orientation of the OSM are the key determinants.

Figure 16 illustrates three basic starting points for developing operational orientations. The solar inertial (A) places the array perpendicular to the sunline and produces maximum power at all times. However, unless the orientation of the OSM body and payload mass is controlled, the resulting gravity-gradient and aerodynamic torques will be excessive.

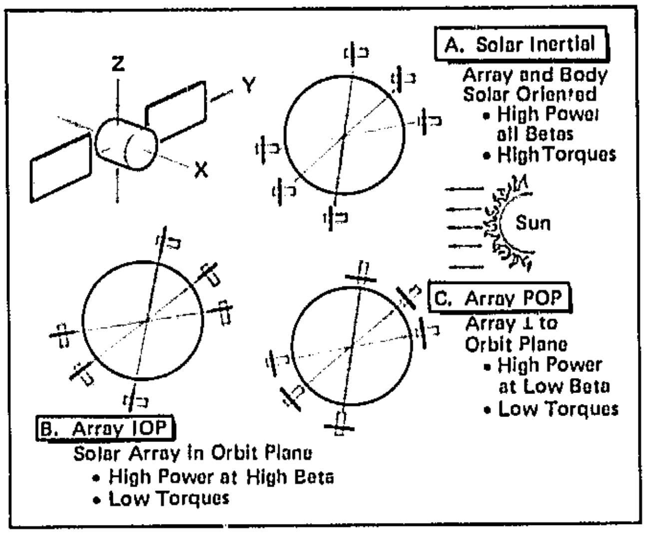


Figure 16. Orientation Comparison

The orientations (B and C) tend to minimize disturbance torques because the dominating element, the solar array, is either in the orbit plane (IOP) or perpendicular to the orbit plane (POP). However, both orientations suffer major power losses as a function of beta angle.

Payload orientation and field of view is a third requirement variable that the selected design must accommodate. Solar inertial (A) is excellent for solar and stellar observations but offers a poor platform for earth observations. Array POP (C), with the body axis aligned along local vertical, or array IOP (B), with the body also aligned to local vertical, provide for good earth viewing but are poor platforms for solar and stellar observations.

To achieve all objectives (full power output, low disturbance torques, and full payload orientation freedom), a design concept is required that includes the best features of all three of these orientations shown.

By aligning the array Y axis in the orbit plane as illustrated in Figure 17 and allowing the array to rotate about Y, the array will be sun-oriented at all times, and maximum power output will be obtained. This is the solar inertial orientation. By using gimbals in the Y axis (as shown on the top of the figure), the body may be rotated relative to the array to enhance payload pointing or to minimize body-induced torques.

By adding a second set of gimbals (as shown on the bottom of the figure), the body and attached payloads can be further oriented to minimize torques or, more importantly, to enhance viewing. With 2 degrees of gimbal freedom relative to the array orientation, any desired payload pointing direction can be achieved. By pointing the X axis perpendicular to the orbit plane, the nadir can be tracked with only a single rotation about X.

This latter orientation and gimbal arrangement is the one selected for the Full-Capability "Reference" Configuration. Viewing in all orientations is good except for orbit conditions wherein one array wing passes through the local vertical during earth terminator observation. When this occurs, the array axis must be shifted away from local vertical to avoid terminator obscuration.

To provide an unobstructed view of the earth's surface, the solar array longitudinal axis may be stabilized perpendicular to the local vertical. This condition is satisfied with the axis either perpendicular to the orbit plane or in the orbit plane (or at any angle between these extremes). With flexibility to the user as a major consideration, our studies have shown that the system should be designed for long-term operation in low-disturbance, high-power orientations, but should have capability for less-efficient, short-term orientations to satisfy user needs. Thus, the long-term IOP-solar orientation is selected, and the POP-local vertical orientation is retained as an alternative for earth-oriented mission emphasis. System mechanization permits other intermediate orientations as mission operations require.

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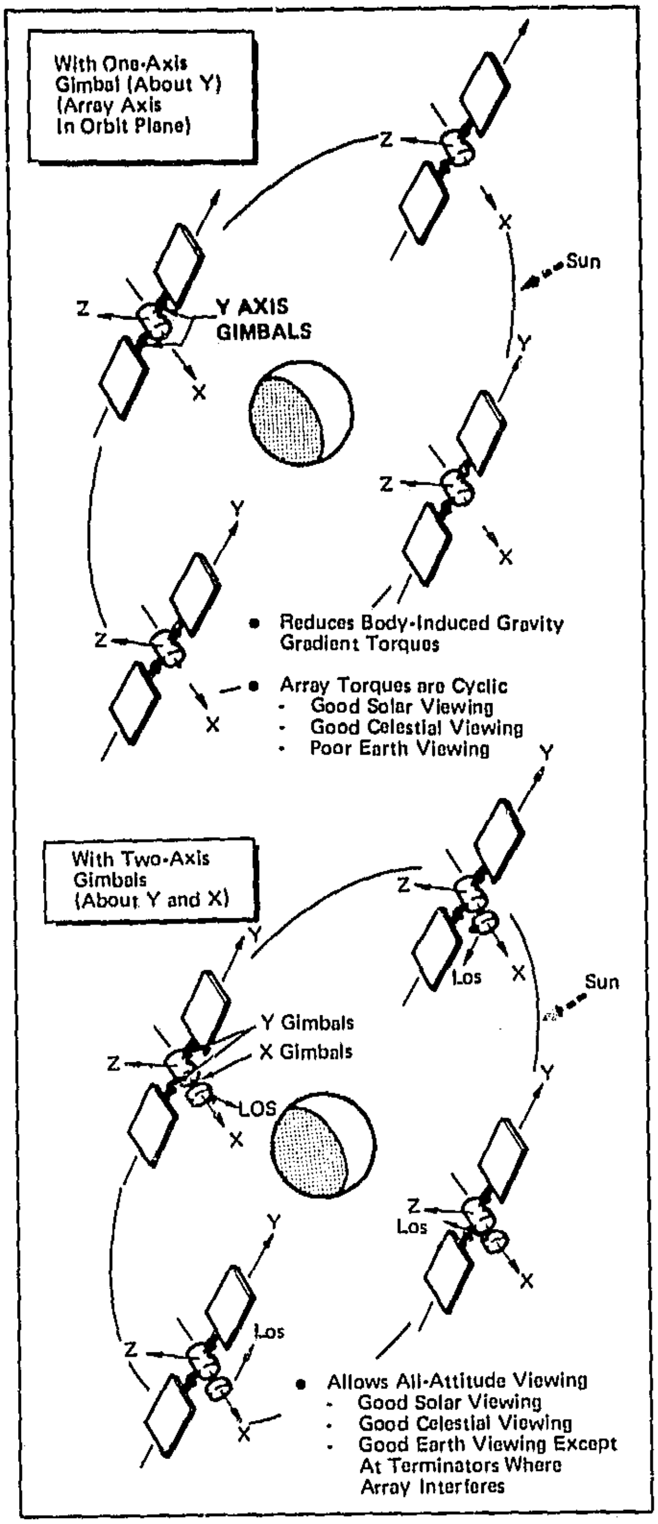

Figure 17. Combined IOP and Solar Inertial

Table 4 compares the control system sizing for the three generic system concepts. The symmetric and asymmetric concepts are compared for the basic IOP-solar inertial orientation (array axis in the orbit plane with the other axes at the inertial angle for worst moment) and POP-local vertical (array axis perpendicular to orbit plane, and other axes aligned to achieve minimum torque about the POP axis). With a mass balance of 2 degrees of freedom, the basic IOP-solar inertial orientation is left with 723 ft-lb-sec per orbit about the POP axis. Several methods of desaturating this momentum as it accumulates are available. Although the sizing for the POP orientation appears small, it provides maximum power only for low beta angles. If the vehicle must be tipped to compensate for beta angle, the balance weight on a 100-foot boom would be 10,000 pounds and 3 degrees of gimbal freedom would have to be added.

The asymmetric concept has three to four times the sizing requirements of the symmetric concept for the basic IOP-solar inertial orientation. The POP-local vertical requirements are similarly large, and a 10,000-lb weight is required to hold the vehicle tilted to compensate for high beta angles.

The gravity-gradient configuration must have at least a 31-meter offset of the solar array to be gravity-gradient stable under the influence of aerodynamic torques with the solar array axis POP. The configuration shown uses a 36-meter offset, allowing an average of a 16-degree tilt of the mast to balance the average aerodynamic moment with gravity gradient. The cyclic momentum is sized to absorb the aerodynamic torques about the average value. The gravity-gradient concept requires an 11,900-pound balance weight on a 100-foot boom to hold the vehicle in the IOP-solar inertial orientation.

Remote Manipulator System (RMS)

The construction and operational characteristics of the RMS are such that care must be taken to ensure proper clearances and rotational freedom for each portion of the RMS arm. A typical payload deployment path is illustrated in Figures 18 and 19. The elbow joint of the RMS arm bends in only one direction, similar to a human elbow joint; therefore, the kinematics of the RMS operation requires a rotation of the arm at the shoulder joint of the RMS to rotate the elbow

Table 4. Control Actuation System Sizing

Concept	Orientation	Bias Momentum (Ft-Lb-Sec/Orbit)	Balance Weight (Lb)	Cyclic Momentum (Ft-Lb-Sec)
1. Symmetric	IOP-Solar Inertial	2555(*) Without Balance Boom	1560(*)	± 3335(*)
		723(*) With 2-DOF Balance Boom		
	POP-Local Vertical	630 Without Balance Boom	1000	630
		450 With 2-DOF Balance Boom		
2. Asymmetric	IOP-Solar Inertial	6275 Without Balance Boom	6080	± 9200
		2590 With 2-DOF Balance Boom		
	POP-Local Vertical	4400 Without Balance Boom	2300	5670
		1905 With 2-DOF Balance Boom		
3. Gravity Gradient (36-Meter Mast)	Array Solar Inertial Mast Tilted 16° From Local Vertical About POP Axis	34,320 Without Mast Tilt or Balance Boom	6352	2160
		440 With Balance Boom and Mast Tilt		
	Array POP — Mast Tilted 16° From Local Vertical About POP Axis	22,150 Without Mast Tilt	1000	1640
		440 With Mast Tilt		

* Worst Case

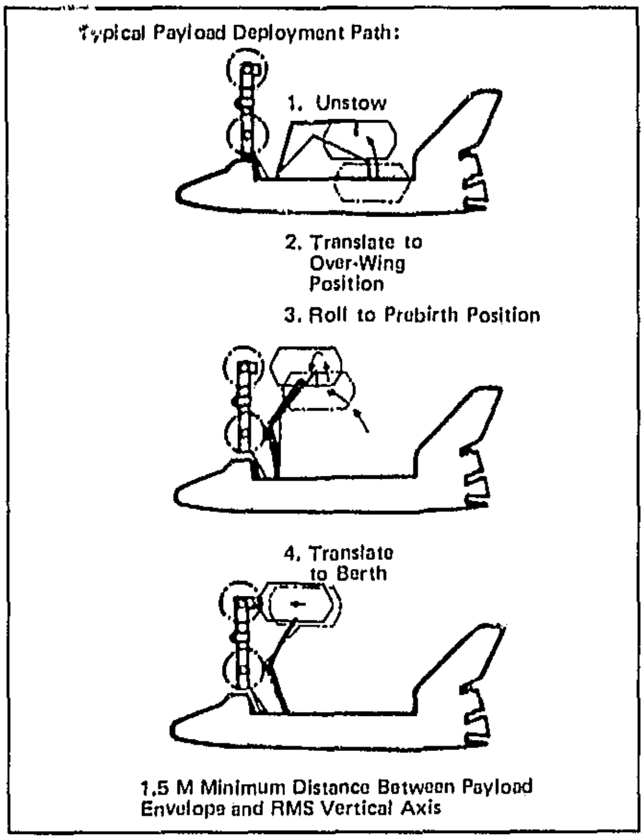

Figure 18. Typical Payload Deployment Path

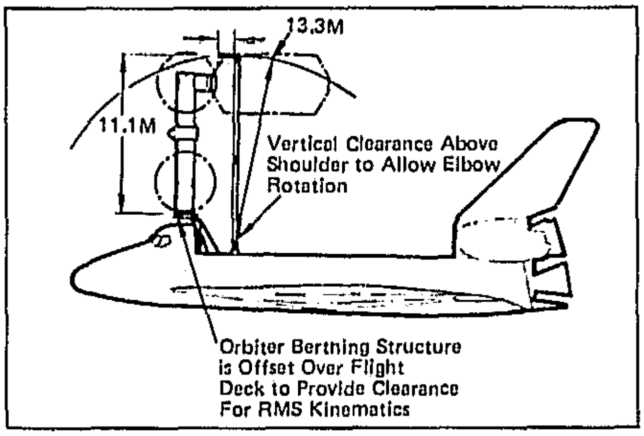

Figure 19. RMS Reach Envelope

flexural direction (i.e., to be able to bend the upper part of the RMS *downward* forward of the shoulder joint). The ability to raise the complete RMS arm in a vertical direction is desirable so as to minimize interference of the arm with surrounding payloads; then the arm may be rotated any amount at the shoulder joint without obstruction. The berthing structure of the power module to the Orbiter has been designed to ensure these clearances.

Power Output

With the power system sized for 35 kW, 28 VDC at beta ≃ zero, a maximum of 41 kW is obtained in a 28.5-degree orbit and 63 kW is obtained in a 55-degree orbit.

The range of the OSM power module's capability is affected by several factors. Power levels delivered to the bus vary with the point in the design lifetime, the voltage level delivered, and the beta angle, which changes throughout the year and depends on orbit inclination and altitude.

As illustrated in Figure 20, the power module rated power is 35 kW at 28 volts after 5 years of on-orbit operation. Immediately after launch, however, a minimum of 42 kW would be delivered at 28 volts, a value of 20% higher than the rated power. The decay of this power to the rated level due to cell radiation effects is shown by the dashed line.

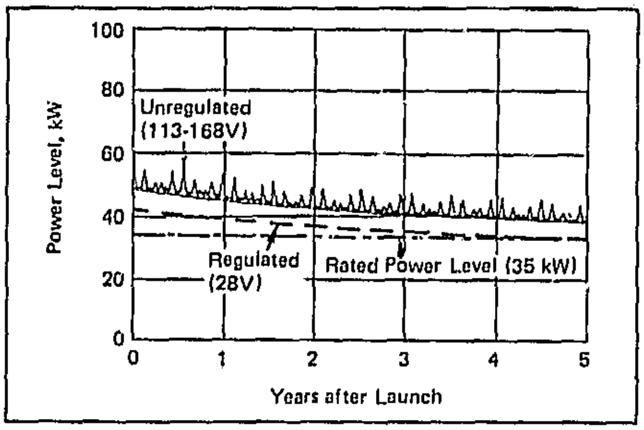

Figure 20. Power Capability – 28.5° Inclination

By changing the voltage from 113 to 168 volts unregulated, a further power increase is available: 48.6 kW beginning of life (BOL), or 39% more than the rated power. The decay of this nominal power level is illustrated by the smooth solid line.

In addition, excursions of beta angle from zero increase the percentage of time spent in sunlight per orbit. This increases the available power above the nominal (beta = zero) case. A 5-year daily power profile of the OSM in a 28.5-degree, 235-nmi orbit, and delivering 113 to 168 volts unregulated to the bus is shown by the spiked solid line. Peak capabilities of 55 kW are achievable at 28.5 degrees.

The power can reach even higher levels (90-kW range) at high beta angles as could occur in a 55-degree orbit where the power module might be in constant sunlight for fairly long periods.

Radiator Design

Another major consideration in configuration design is the optimum location of the thermal control radiators on the OSM. Five locations were considered for each of the three reference design configuration options. Figure 21 illustrates possible radiator locations for Configuration 1. Radiator locations on the other configurations studied were similar to those shown for Configuration 1.

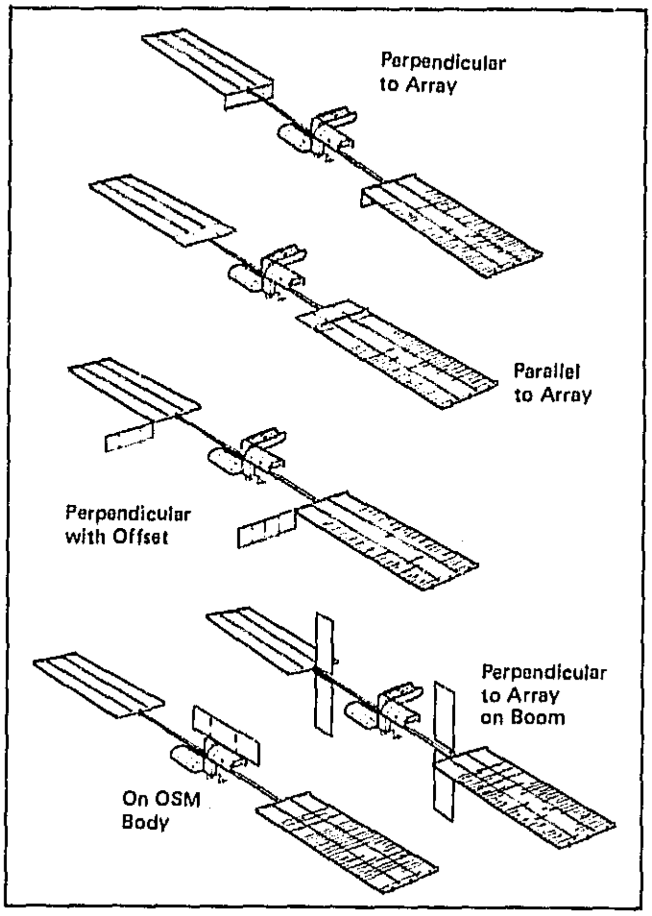

Figure 21. Candidate Radiator Locations (Configuration 1)

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The concept that locates the radiators perpendicular to the solar arrays is mechanically easy to stow and deploy inasmuch as they lie along the array boxes. Their proximity to the arrays, however, results in incident heat on the radiators.

The concept that locates the radiators parallel to the solar arrays is simple from a packaging standpoint; however, radiator area is increased, because the top side of the radiator is exposed to full solar heat during the entire sun side of the orbit.

The concept that locates the radiators on the boom perpendicular to the solar array is complex in regard to stowage and deployment, but incident heat on the radiator is small.

Locating the radiators on the OSM body can eliminate rotating fluid joints, but direct solar impingement can occur for some orientations.

The bar graph in Figure 22 presents the radiator area requirements for the three configuration options and five candidate radiator locations. Radiators located perpendicular with offset and perpendicular on the boom require about 600 ft^2 less area (single side) than if positioned at other locations. The radiator location perpendicular to the array is larger because of the heat energy that impinges on the radiator from the adjacent solar array. Locating the radiator parallel to the array or on the OSM body results in direct solar impingement on the radiator for the design point, thereby increasing the area. Figure 22 also gives the radiator area for each of the three alternative configurations. Areas for Configurations 1 and 3 are nearly the same for all radiator locations, but Configuration 2 is slightly different for most radiator locations.

Figure 23 shows the effect of the thermal environment (sink temperature) on radiator area. The area increases rapidly for sink temperatures above 400°R and goes to infinity as the sink temperature approaches radiator fluid outlet temperature of 500°R.

Other considerations examined in addition to location and radiator area were drag, experiment scan angle, packaging, and complexity. Based upon all considerations, the perpendicular-on-boom arrangement was selected for the full-capability "reference" configuration.

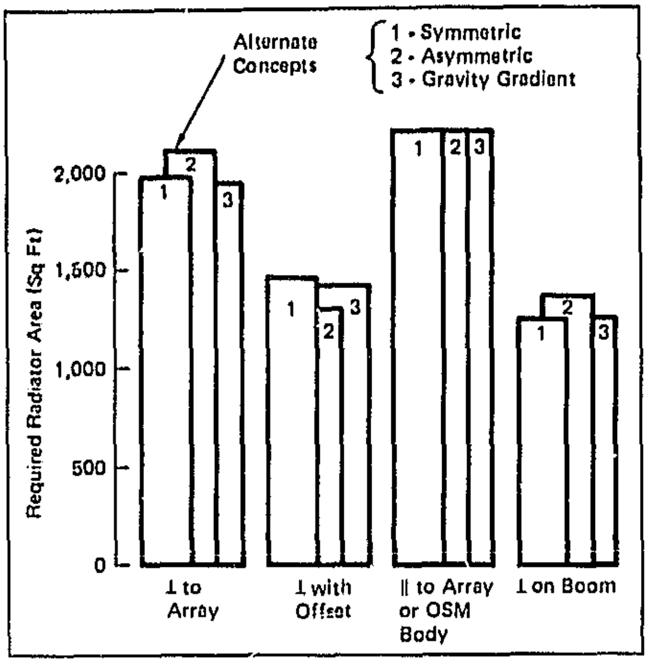

Figure 22. Radiator Performance and Area

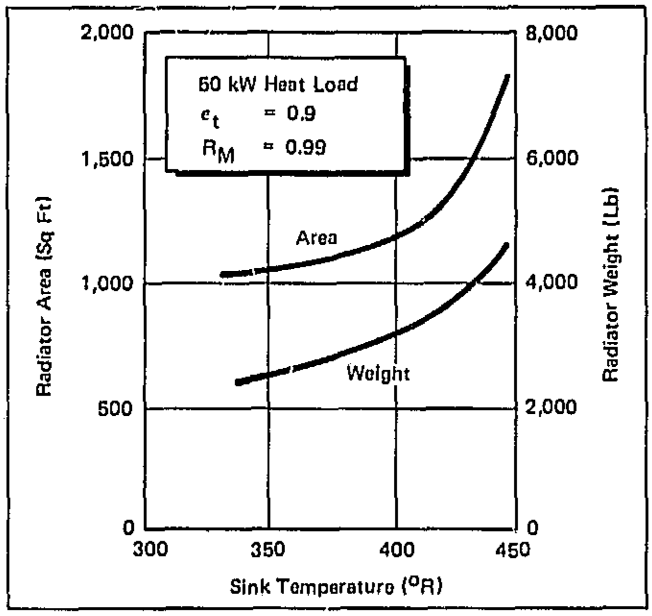

Figure 23. Radiator Performance Area

WHAT SYSTEM DESIGN CONCEPT MEETS THE FULL-CAPABILITY REQUIREMENTS?

Figures 24 and 25 illustrate a full-capability "reference" OSM configuration that is based upon the analyses conducted by the study team, and that combines all the preferred features.

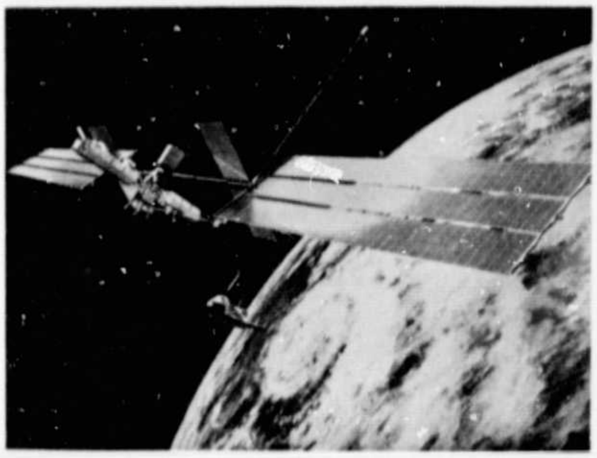

Figure 24. Photo Sample

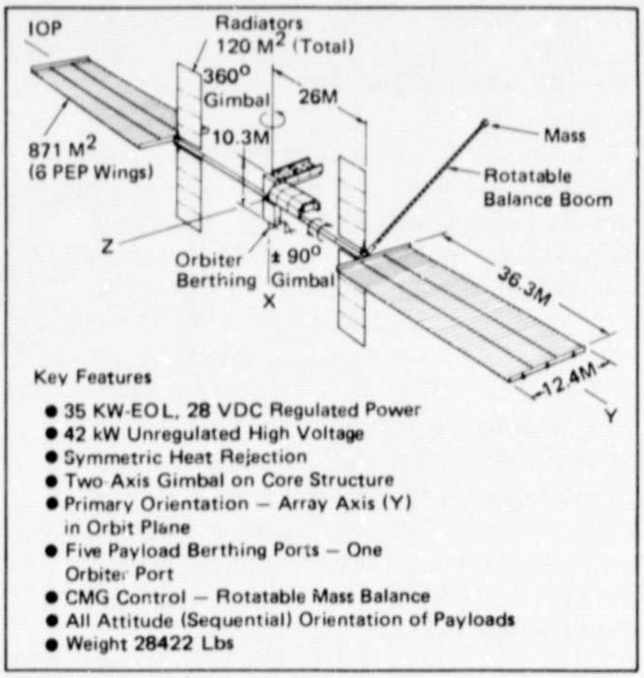

Figure 25. Full Capability Reference Concept

The concept as illustrated can accommodate all requirements identified to date. The symmetric configuration is preferred from the designers' viewpoint. It places the array axis in the orbit plane to minimize bias moments, and the balanced area moments minimize CMG size. Separation between the wings provides for rendezvous and departure plume clearance without the necessity of retracting the array. Separation also increases the payload field of view and allows for a more centralized mass, to minimize bias moments. A two-gimbal system would allow all-attitude payload orientation with full power in all attitudes.

Both IOP and POP orientations can be used, and a mass balance would minimize CMG desaturation. Placing the radiators perpendicular to the solar arrays minimizes the area and weight requirements. By providing for berthing the Orbiter to the OSM core above the Orbiter cabin, the RMS reach on the Orbiter is adequate for payload berthing.

All the configurations studied were compatible with pressurized or unpressurized access, single Orbiter launch, and payload bay stowage of the deployment mechanization.

Total system weight is another important consideration. The available preliminary information indicates that a concept similar to the recommended configuration would be the lightest. Several wing mast options are available for storage flexibility, and the commonality of the solar array blanket with the earlier PEP design is retained. Ultimate determination of the preferred configuration must await further interaction with potential users to make certain that the full spectrum of mission requirements is accommodated.

A more detailed illustration of the central portion of the full-capability "reference" concept is presented in Figure 26. The full-capability power module core contains a two-axis (beta axis and orbit rate axis) gimbal system, five payload berthing ports, one Orbiter berthing port, and subsystem installation areas. The bulk of the OSM equipment is externally mounted on the body structure and covered by hinged thermal/meteoroid protective panels.

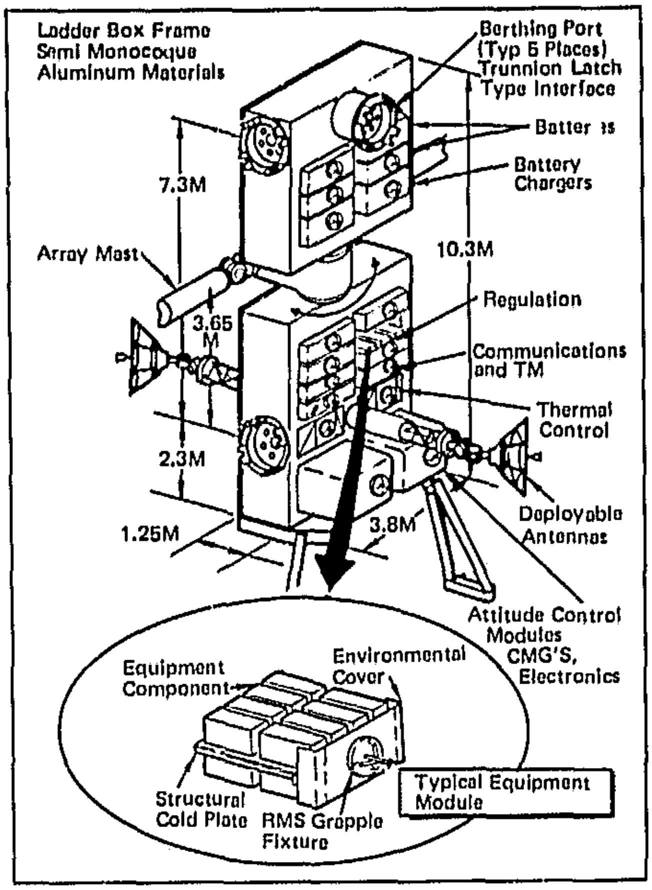

Figure 26. OSM - Full Capability Concept Core Body

All payload berthing ports and all subsystem installations are accessible to RMS reach from the Orbiter's berthed location. The Orbiter berthing structure is offset from the subsystem core's vertical axis to facilitate operation of the RMS arm. The berthing structure possesses standard payload attachment trunnions, which will attach to normal Orbiter payload attachment fittings.

HOW COULD THE FULL-CAPABILITY REFERENCE CONCEPT BE DEPLOYED AND WHERE WOULD IT NORMALLY OPERATE?

Figure 27 shows how the reference configuration would be folded and arranged for boost-phase stowage in the Orbiter payload bay. With forward being to the left in the figure, the power module is arranged to place the main body as well as the installed equipment, e.g., the batteries, toward the aft end of the Orbiter bay for CG control. The array structural boxes are pulled up against the main body without pivoting. The standoff mast is folded double toward the front, and the radiator panels are accordion folded and supported against each other tent-fashion over the module.

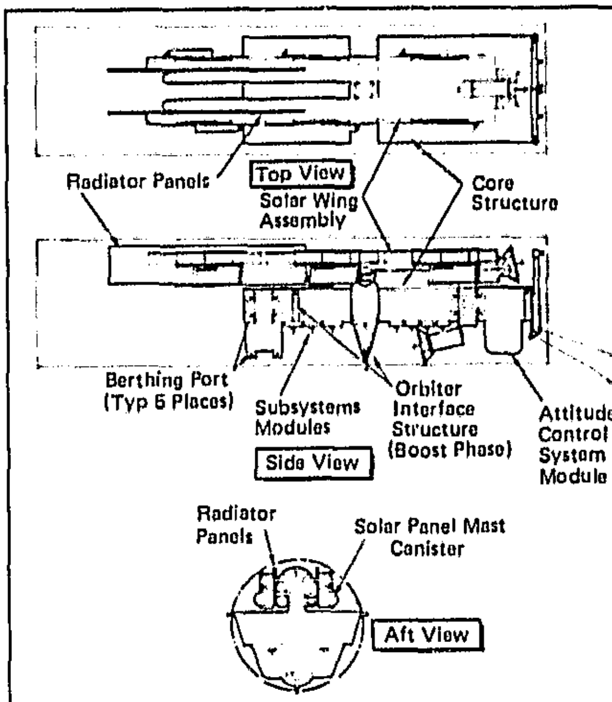

Figure 27. Orbiter Stowage-Full Capability Concept

Figure 28 illustrates the sequence of unstowing and automatically deploying the OSM from the Orbiter through orbit placement. After grappling with the RMS (1), the OSM is translated and pitched to a vertical position, the Orbiter interface legs are deployed, and the OSM is placed in the Orbiter's payload retention fittings (2). After an umbilical connection has been made with the Orbiter at one of the legs, the radiator support restraints are released and the standoff masts are extended (3). The solar array panels are then deployed, systems are checked, and gimbals are exercised (4). When final orbit is achieved, the OSM is activated and released, and the Orbiter departs (5).

Deployment assisted by extravehicular activity (EVA) is somewhat similar to the sequence shown in the figure; however, assembly occurs at various points. In one concept being studied, Steps 1 and 2 are identical, except the equip-

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ment module does not contain booms, radiators, and array assemblies. The EVA crewmen assist in Steps 1 and 2 with manually latching operations and assist the RMS operator. After the equipment module is mounted, the boom segments are moved into place, aligned, and fastened. The segments can be moved by either the RMS or the Manned Maneuvering Unit, but the RMS is preferred to reduce MMU fuel consumption. EVA-assisted deployment continues with the installation of array boxes and radiators. Several approaches are being considered for deploying the arrays, including assembly of array booms, manual deployment, or electromechanical deployment using a portable battery-powered electric motor.

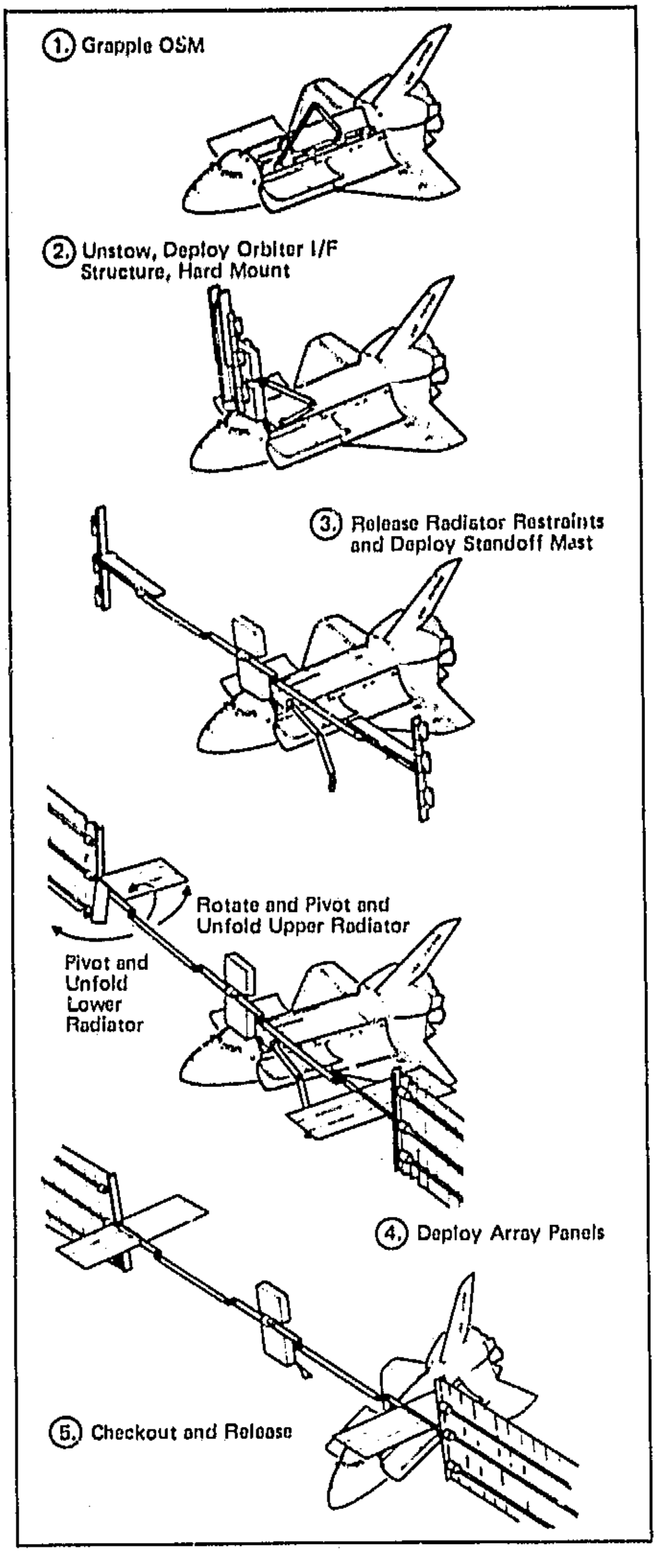


Figure 28. Power Module Deployment

The major factors that would influence the selection of the orbit inclination for an orbiting power module are summarized in Figure 29. User requirements span the 28.5-degree-to-sun-synchronous (~98-degree) region, with emphasis at the extremes. In the October 1977 Traffic Model, 159 (66%) of the missions are to be flown at 28.5 degrees, 30 (12%) at any inclination and 52 (22%) at the polar region. Future support of geosynchronous-bound missions would require a 28.5-degree location.

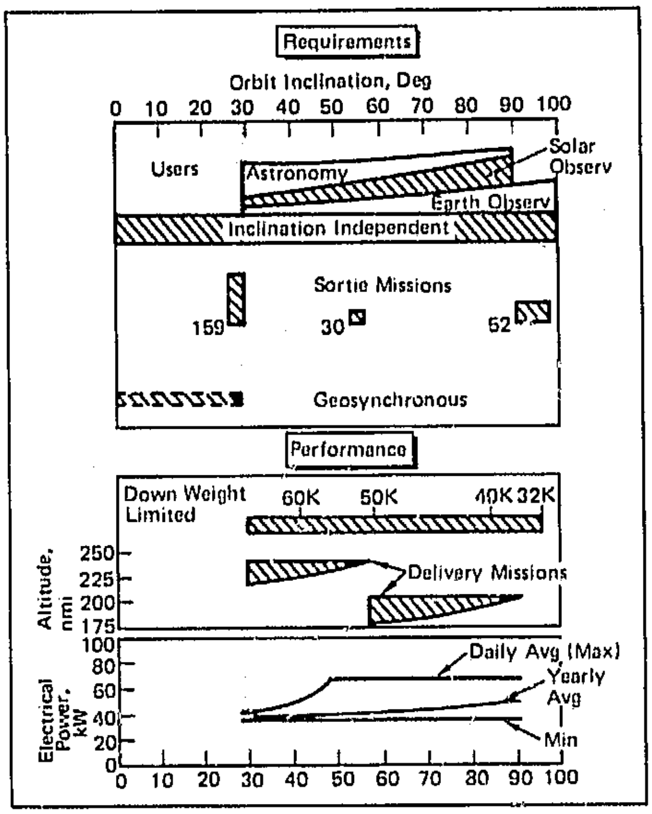


Figure 29. Orbit Inclination Selection

Orbiter performance (payload) is related to inclination in that inclination affects the altitude that can be flown. For down-weight limited missions, the inclination capabilities are as shown (32,000 pounds can be delivered to and returned from orbits up to 95-degree inclinations.) Delivery missions (not constrained to planned landing weight restrictions) are affected by inclination and altitude as shown. More capability is available for the lower-inclination missions.

The electrical performance of the power module is dependent upon beta angle, which is directly related to inclination. More sun time at higher beta angles thus allows higher performance on higher inclination missions.

On the basis of the above data, the power module must be designed to be compatible with the full range of orbit inclinations (28.5 to 98 degrees). This would permit the accommodation of any particular inclination that subsequent analysis and program decisions would dictate. The power module would be nominally positioned at 28.5 degrees to accommodate the widest range of users, to be available to serve and take advantage of the majority of the planned Orbiter flights, and to ensure that the lowest power output condition is adequate.

The orbit altitude range for the power module was selected based on the parameters illustrated in Figure 30. No distinct user requirements were found to directly influence the altitude selection.

Orbit-keeping required (to counter drag) is a strong function of altitude and solar cycle. The amount needed rapidly increases for altitudes below 210 nmi.

Orbiter performance reduces with increasing altitude (shown for integral OMS capability). An altitude of 220 nmi would allow maximum Orbiter capability. The maximum net weight delivered (considering payload delivered minus reboost requirements) would indicate an altitude in the 215-nmi range.

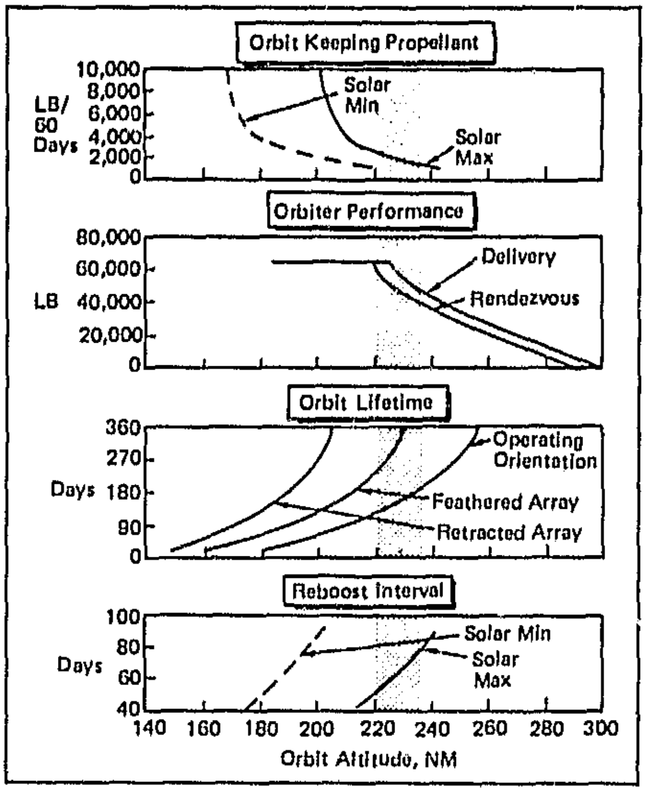


Figure 30. Orbit Altitude Selection

Orbit lifetime parameters are also shown. A contingency lifetime (in the event of Orbiter unavailability) of 6 months to 1 year would seem to be desirable. This would dictate a minimum altitude of 210 nmi (6 months) or 230 nmi (1 year), provided the power module could be flown with the arrays feathered. The capability could be extended by incorporating array retraction capability and/or a contingency reboost system.

A periodic reboost capability of about 20 nmi is compatible with candidate reboost concepts. This would require a reboost interval as shown, i.e., 60 to 90 days for the 215- to 240-nmi range.

A power module operating altitude from 220 to 235 nmi was selected based on these data. This would allow full Orbiter performance capability and require reboost intervals from 85 to 150 days, depending upon solar activity. In addition, a contingency lifetime of 250 days in the feathered condition or over 1 year retracted would be ensured.

WHAT ARE THE BASIC PROGRAMMATIC OPTIONS AVAILABLE FOR THE DEVELOPMENT OF FUTURE SYSTEM CAPABILITY?

It is expected that the nation's funding for new space programs and extensions of existing programs will continue to be limited and highly competitive in the foreseeable future. It is, therefore, essential that proposed new developments clearly reflect the requirements of the user community and that the proposed users represent a real and viable market. We must recognize that payload requirements are just now emerging for the post-1984 period, although the needs for electrical power and increased orbital stay time in the near term appear firm. Design cannot lead requirements but rather must evolve as requirements are established; thus, continuing OSM activity on free-flying power modules, both in the development of user requirements and the definition of power module concepts, should be responsive to realistic user needs. Specification of more advanced systems awaits the development of a consensus of real user needs. When such programs evolve, they should be predicated upon a realistic evolutionary growth, taking advantage of the existing state of knowledge and technology. Each step can then represent a relatively small increment in expense. Decisions to commit to the next step in hardware development can be made in sequential fashion and need not be made until the requirement has been firmly established.

Figure 31 protrays three optional growth paths predicated upon this requirements-oriented evolutionary approach. In each case, the initial step is the development of PEP to meet currently defined requirements.

As experience is developed and the requirements of longer-term payloads are established, one option might be to procure two additional PEP units and develop a power module for use in a 28.5-degree orbit for the 1984 time period and beyond.

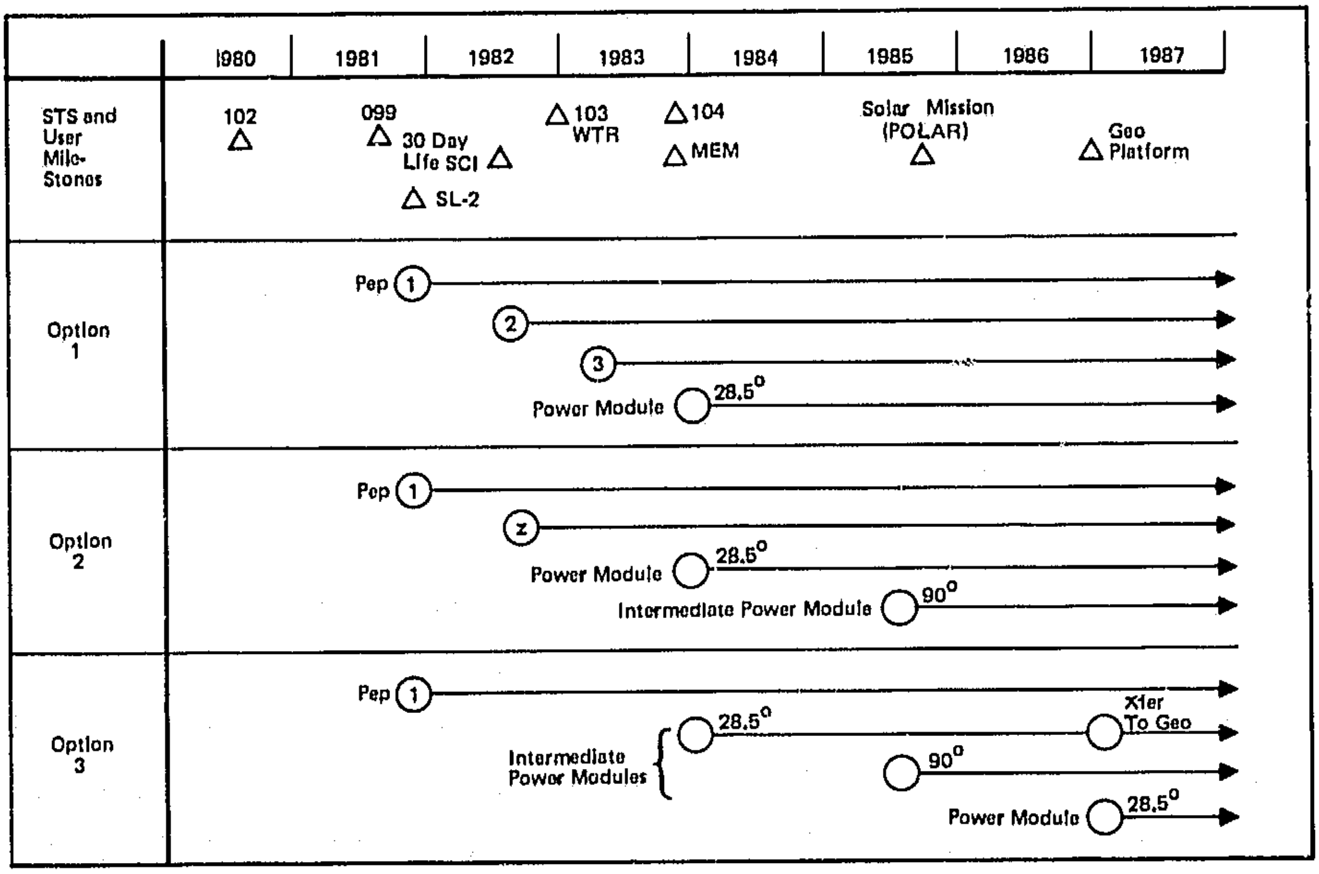


Figure 31. Candidate OSM Program Options

The review of user requirements noted that many projected payloads that can beneficially use long stay times in orbit require relatively low power. These include earth and solar observations, astronomy, and life sciences. Preliminary data suggest that up to three or four payloads in these categories could be supported for about 12 to 13 kW. Additionally, some of these payloads require very high orbital inclinations or altitudes. Thus, a second option would be to procure one additional PEP if the use level so indicates, then proceed in 1984 to a full-power module in a 28.5-degree orbit. This would be followed in the mid-1980's by development of a smaller free-flyer (intermediate-power module) for servicing payloads in polar orbit only.

A third option would be to follow the initial PEP with the development of intermediate-power modules for free-flying payloads in 28.5-degree and polar orbits, deferring the development of the more sophisticated power module until the need was clearly established in the late 1980's.

It is recognized that new mission requirements will continue to evolve as experience in space operations is accumulated. By designing inherent capabilities for growth into the basic OSM, its useful operating life can be extended indefinitely. Variations for the full-capability 35-kW reference module possible in later years might include adding provisions for the simultaneous orientation of multiple payloads along different axes and uprating the system to provide 100 kW of power or more.

The alternative development strategies available to further uprate the full-capability OSM are summarized in Table 5. They include on-orbit uprating techniques as well as the more conventional ground-based and new vehicle design growth techniques. It is important to note that all the alternative approaches can result in total program savings through the use of common evolutionary subsystems if program continuity is maintained. Each, however, has unique advantages and disadvantages.

On-orbit growth can be accomplished by either replication or addition of subsystems. By replication, another identical OSM is constructed and attached to the existing vehicle with a suitable adapter. Addition of subsystems on orbit does require that the provisions for on-orbit modifications be designed into the initial OSM.

Table 5. OSM Uprating Strategies

Uprating Technique	Advantages	Disadvantages
On Orbit		
Replication/Siamese Twin	Low Initial Cost	• Limited Size Flexibility. • Geometry Limitations Introduce Operational Limitations
Addition of Subsystems	Minimum Cost at Time of Uprating	• Initial Cost • Practical Limitations to Size of Growth Increments • Limited Flexibility
Ground-Based Uprating	Low Initial Cost Great Flexibility	• High Cost at Time of Uprating • Large System
New Vehicle Based on Common Subsystem	Low Initial Cost	• Down Time • High Cost at Time of Uprating (Unless Earlier OSM is Still Required)

In review, any one of the uprating strategies may prove to be best and none should involve significant initial costs unless the on-orbit addition of subsystems is carried to an extreme. Since the longer-range mission requirements will probably not be firm when an OSM design study is initiated, preliminary plans, based upon the emerging requirements scenarios, should be formulated for each. When requirements are firm, the optimal development and operation plans can be selected, and the basic design approach can be frozen.

Preliminary analyses suggest a fairly linear relationship between power level required and system cost in the range of power levels considered in the study. Figure 32 summarizes the relative costs that might be expected for the power distribution subsystem and the solar array subsystem.

When programmatic costs are considered, the development and production costs for PEP are about $47 million in 1978 dollars (see Figures 33 and 34) and about $139 million in 1978 dollars for the full-capability reference design power module. These funding requirements may be reduced by taking advantage of the fact that fewer cryogenic tank sets will need to be procured, and common development of solar arrays and masts, power conditioning, and distribution equipment, and gimbal components (see Figure 35) will lead to further reduction of overall program costs. It is estimated in the case of the initial PEP, for example, that the $47 million would in reality reflect only a $21 million net increase, since some $26 million would be related to reduction in cryogenic tank set orders and common equipment development with the power module program. If initiated in FY 80, the development of PEP could be compatible with the capture of the currently scheduled Spacelab 2 mission.

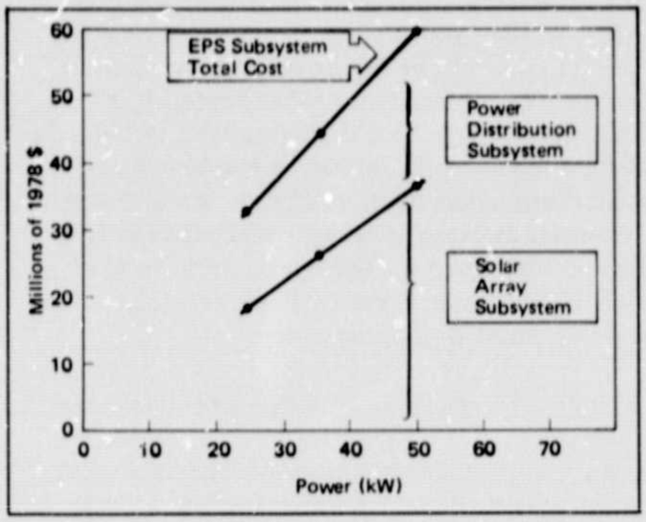
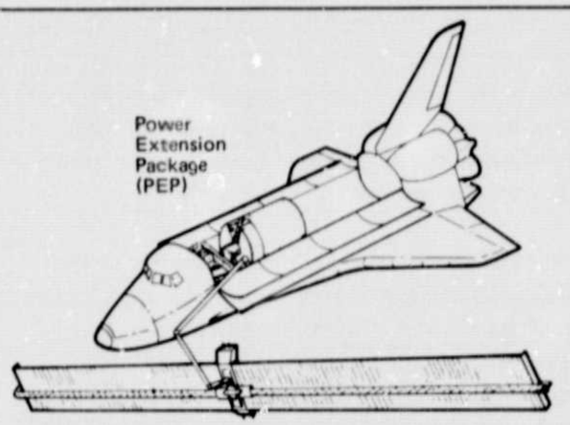
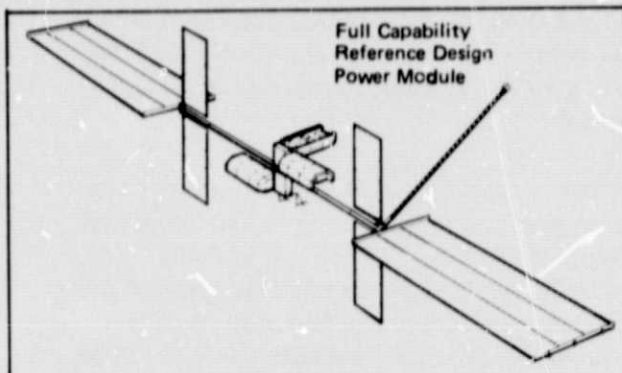


Figure 32. Electrical Power Subsystem Cost Sensitivity



Element	Devel	Prod	Total
Orbiter Mods	3.8	0.7	4.5
Solar Array	7.4	8.5	15.9
Subsystems	(8.9)	(5.5)	(14.4)
Struct/Mech	2.7	0.5	3.2
Power Distr	2.1	1.7	3.8
Thermal	0.2	0.2	0.4
Avionics	1.8	1.8	3.6
Interface Kits	2.1	1.3	3.4
System Level	7.7	2.8	10.5
OPS Supt	0.5	1.2	1.7
Total	28.3	18.7	47.0

Figure 33. Cost – Millions of 1978 Dollars



Element	Devel	Prod	Total
Solar Array	1	26	27
Subsystems	(32)	(30)	(62)
Str/Mech	10	4	14
Power Dist	7	10	17
Thermal	3	3	6
Avionics	6	8	14
Control	6	5	11
System Level	26	24	50
Total	59	80	139

Figure 34. Cost – Millions of 1978 Dollars

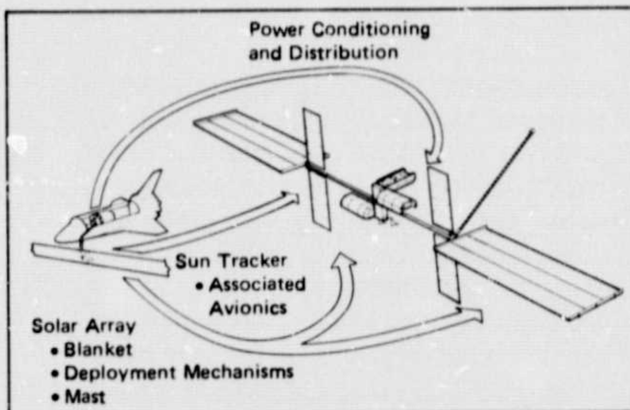


Figure 35. OSM Commonality

SUMMARY AND CONCLUSIONS

The systems concepts developed in this study are logical building blocks in an evolutionary plan to provide more orbital capability (power and mission duration) to potential Space Transportation System users in timely fashion and in the most economical manner possible.

The intent of the OSM evolutionary approach is to ensure good balance in the use of the tremendous flexibility offered by the Orbiter baseline configuration, through provision of payload services such as delivery and return weights, power, cooling, orbit location, attitude control, and duration.

The first step in the OSM program is an Orbiter improvement, one that develops the major components of later orbitally stored systems such as solar arrays, and power conditioning and distribution equipment. The first step provides for power levels of 21 to 29 kW while the OSM is illuminated by the sun. The initial step in the OSM development, the Power Extension Package is deployed by the Remote Manipulator System, thus permitting maximum flexibility in array pointing such that payload and thermal control radiator attitude requirements can be accommodated. The fuel cell and cryogenic system still provide power during dark-side operations. This first Orbital Service Module increment increases the baseline Orbiter capability from about 6-day/21-kW missions to 20-day/29-kW sorties. The array is stored for boost and entry immediately forward of the Spacelab in the space above the tunnel.

As payload power requirements increase beyond the levels provided by the initial program, increment weight and volume considerations make storage of the power system on orbit appear favorable. Thus, the next step is an orbitally stored module that uses solar arrays, power conditioning and distribution equipment, and sun tracker and associated avionics developed during the first program step, together with new subsystems for thermal and attitude control. It can support payload operations in both Shuttle-tended and free-flyer (untended) modes. For operation in a free-flying mode, communications and data avionics subsystems, in addition to those noted, are added to provide all required payload services.

The "full-capability" power module would provide 35 kW indefinitely and sufficient heat dissipation to allow payloads to use the power capability. In this "full-capability" power system, all Orbiter payload operations would be supported completely independent of the Orbiter. It is important to note that the full-capability Orbital Service Module concept can be implemented, with components developed in the initial Power Extension Package at power levels between 12 and 60 kW, without a firm sizing decision for the "full-capability" system being required during this initial development step.

As indicated, the primary design objective in the early stages is to provide flexibility so far as program growth steps are concerned. Hence, exact characteristics in later program increments will not be frozen until the commitment to that step is made. It should also be noted that the Orbital Service Module program concept contains another element of flexibility – the ability to replicate a current or previous increment at any point in the program. Thus, as the pace of orbital operations quickens, additional modules that are essentially tailored to the exact requirements can be procured.

In summary, the payload requirements in the 1981-to-1984 time frame for power and orbital durations are well understood. Furthermore, the key variables and design drivers affecting configuration definition are known. In particular, the interrelationships between array sizing and geometry, control system sizing and payload orientation, and field-of-view requirements have been established. With these relationships quantified, the impact on alternative design concepts that variations in key mission requirements and in funding availability will have can be readily assessed.

For beyond the 1984 time period, the payload requirements are still emerging, and continuing studies to establish requirements for those more advanced payloads are necessary to define in more specific detail their power, orientation field-of-view, and general operational environment.

To date, no technological barriers to the accomplishment of the evolutionary plan have been identified. The initial development of the Power Extension Package concept would resolve all questions on the solar arrays, and the most significant items remaining deal with the specifics of design and development of the mechanisms for the full-capability concepts.