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16. Abstract

An assessment is made of a manned space station operating in the post-1995 period with sufficiently high power demands to require a multi-hundred kilowatt range electrical power system. The nuclear reactor is a competitor for supplying this power level. Load levels were selected at 150kWe and 300kWe. Interactions among the reactor electrical power system, the manned space station, the space transportation system, and the mission were evaluated. The reactor shield and the conversion equipment were assumed to be in different positions with respect to the station; on-board, tethered, and on a free-flyer platform. Mission analyses showed that the free-flyer concept resulted in unacceptable costs and technical problems. The tethered reactor providing power to an electrolyzer for regenerative fuel cells on the space station, results in a minimum weight shield and can be designed to release the reactor power section so that it moves to a high altitude orbit where the decay period is at least 300 years. Placing the reactor on the station on a structural boom is an attractive design, but heavier than the long tethered reactor design because of the shield weight for manned activity near the reactor. When the reactor is placed in the space station, preferably at the C.G., the station dynamics is simplified, but this installation results in the heaviest shield.

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FOREWARD

This contract report was prepared by the authors identified on the title page. Gordon R. Woodcock of Boeing Aerospace Company also provided guidance and contributions to the program with respect to space station mission and operational concepts.

The Boeing contract manager was Sidney W. Silverman and the technical leader was Dr. Harvey J. Willenberg. Information about the nuclear reactor power concepts, hardware, and design and operation was prepared in a subcontract to Boeing by Charles Robertson under the management of William Terrill of General Electric Company at King of Prussia, PA.

At NASA Lewis Research Center, Jack A. Heller was the project manager, who had significant assistance from Joseph J. Nainiger and Dr. John Dunning of the space systems office.

LIST OF ABBREVIATIONS

CDG	Concept Definition Group
CFE	Continuous Flow Electrophoresis Process
DSCG	Directional Solidification Crystal Growth
ECG	Electroepitaxial Crystal Growth
ECLSS	Environmental Control/Life Support System
ED	Engineering Demonstration
EOS	Electrophoresis Operations in Space
EPS	Electrical Power System
EVA	External Vehicular Activity
GaAs	Gallium Arsenide
GEO	Geosynchronous Orbit
IEF	Isoelectric Focusing
LEO	Low Earth Orbit
MPS	Materials Processing In Space
MRWG	Mission Requirements Working Group
OMV	Orbital Maneuvering Vehicle
OTV	Orbital Transfer Vehicle
RFC	Regenerative Fuel Cell
RPS	Reactor Power System
SBOTV	Space Based Orbital Transfer Vehicle
STS	Space Transportation System
VCG	(Chemical) Vapor Transport Growth
VOC	Verification of Concept

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SUMMARY

A space station with a nuclear reactor source of energy will be designed and operated different from one with the energy converted from sunlight. The objective of this study is to determine the applicability of the SP-100 class of nuclear reactor electrical power system to a manned space station which is under consideration by NASA. In the contract technical tasks the missions which can utilize a nuclear electric power system are to be determined. Using typical generic conceptual designs of nuclear reactor power systems and space stations, the assessment was made of the impact of the nuclear power system on the manned space station architecture and operation. In turn, the effect of a manned space station on the nuclear reactor power system was determined.

The scope of the program included placing the nuclear reactor and energy conversion equipment: (a) on the tation similar to a submarine installation, or on a structural beam a distance away from the manned activities and equipment; (b) on a flexible tether; or (c) on a free-flyer platform. To develop a size for use in the analyses, the electrical system power level was selected at 150 kWe and 300 kWe so that several energy conversion methods would be examined.

As a result of the analysis it was found that the nuclear reactor power system could be placed in any of the three positions, but that at the system level the penalties were much greater in the case of the free-flyer than in the other two. The penalty was the large number of shuttle flights required to operate the space station and place the nuclear electrical power system in orbit. In turn, the reactor shield weight was low and the reactor could be operated in a long-lived orbit. With a reactor on the space station or on a flexible tether, the system could be implemented and operated with few shuttle flights.

It was found that as the space station power level grew into the multi-hundred kilowatt range and the area of solar arrays increased accordingly, that the architecture of the space station was affected. The nuclear reactor power system then became an attractive alternate energy source since it did not require solar orientation and it was immune to shadowing by the spacecraft appendages or structures.

1.0 INTRODUCTION

1.1 Program Background

The SP-100 program was initiated in 1983 to establish performance limits and advance technology for 100-kW class of space nuclear power systems in support of military and civilian missions. The first phase of the SP-100 program is the development of rationale for a reactor electrical power system, technology assessment, and conceptual designs of reactor powered conversion subsystems. Identification of military and civilian mission requirements for 100-kW class space nuclear power systems, identification of system concepts that can meet these mission requirements, and resolution of the technological feasibility issues associated with the concept development are the goals of the effort. A determination of the costs and schedule required to proceed with development is also to be made. These activities are proceeding toward a recommendation in mid-1985 as to whether to proceed to the next phase, which would be the design, manufacture, and testing of a developmental ground engineering system.

Potential missions that might benefit from space nuclear reactors fall in four major areas: military missions, planetary missions, manned space station missions, and civilian/commercial missions. This report identifies potential manned space station non-military missions that would benefit from a space nuclear reactor and analyzes nuclear reactor-powered electrical system configurations to accomplish these missions.

1.2 Scope of Work

Current NASA plans for manned space station missions were reviewed to assess the applicability, benefits, and constraints of 100-kWe class of nuclear reactors in supplying space station electrical power. Power requirements were summarized from the NASA Mission Requirements Working Group reports (ref. 1) to determine basic mission needs. Other loads were also examined, including space station housekeeping requirements and various space station growth scenarios. The alternative growth scenarios considered missions that might be enabled by the availability of sufficient power at the space station. These loads are not currently in the reference mission set, and power demand growth might result from aggressive commercial involvement in the space station program.

Additional mission factors that might affect the use of nuclear power were reviewed qualitatively. These factors include atmospheric drag, space station attitude control and flexibility, spacecraft traffic management, and environmental interactions. Power growth scaleability was also considered. Although this contract did not include a tradeoff between solar-powered and nuclear reactor-powered electrical systems, such an analysis was conducted at NASA LeRC.

Safety requirements were reviewed that may apply to space nuclear reactors, in general, and to reactors on or near a manned space station, in particular. Existing requirements were reviewed for their applicability to this system and for consistency. The impacts of safety requirements on a manned space station and its missions were assessed and additional safety requirements were identified.

Fourteen different system configurations were considered for the nuclear reactor power system and space station combination. These configurations fell into three general classes: on-board nuclear reactor power systems, tethered nuclear reactor power systems, and free-flying nuclear reactor power systems. Power transmission by electromagnetic beaming, by conduction, and by fluid transfer was considered. A general screening of these configurations was made to reduce the number to one candidate from each configuration class. This selection was made on the basis of required nuclear reactor power system sizes, traffic constraints, heat removal, and space station/mission impact.

A conceptual design of each of the three candidate configurations was performed to provide quantitative data for a trade study. The trade study compared the three designs on the basis of initial and life-cycle mass and volume in orbit, Space Transportation System logistics, nuclear safety, and orbit mechanics. Space station power levels of 150 kWe and 300 kWe were evaluated. The space station architecture, orbit, and size considered was a generic design which is representative of what might satisfy mission requirements currently planned in the mid to late 1990's. A generic reactor design was used which is close to one option currently being developed elsewhere in the SP-100 program. Several power conversion options were considered, including thermoelectric, thermionic, and three different dynamic cycles (Brayton, Rankine, Stirling). Radiation shield variations treated included four pi, shaped four pi, two pi, and conical (shadow) shields.

Requirements imposed on the nuclear power system due to operation with the manned space station in a low-earth orbit and requirements imposed on the space station and on elements of the Space Transportation System due to the nuclear electric power system were defined.

1.2.1 Contract Statement of Work

The statement of work of the contract is shown in flow diagram form in Figure 1. Essentially, the program dealt with three technical tasks and one task to encompass reporting and presentations. The task titles identify the work. These are:

- Task I: Identify manned missions that would materially benefit and/or be enabled through the use of nuclear power, and their respective power requirements.
- Task II: Assess the impact of the presence of a nuclear electric power system (NEPS) on the architecture of the Space Station with particular emphasis on safety.
- Task III: Prepare three conceptual designs of a NEPS, including on-board, tethered, and free-flyer approaches.
- Task IV: Document the study in a final report.

1.3 Configurations Selected

The trade studies were performed for three system configurations: boom-mounted nuclear reactor power system with electrical transmission lines; tethered nuclear reactor power system with an electrolysis plant which pumps gaseous hydrogen and oxygen through hoses to fuel cells on the space station; and a free-flying nuclear reactor power system with an electrolyzer and liquifier, and transfer of fuel cell reactants with a space-based orbital transfer vehicle. Two power levels were treated: 150 kWe with thermoelectric conversion, and 300 kWe with a Stirling cycle

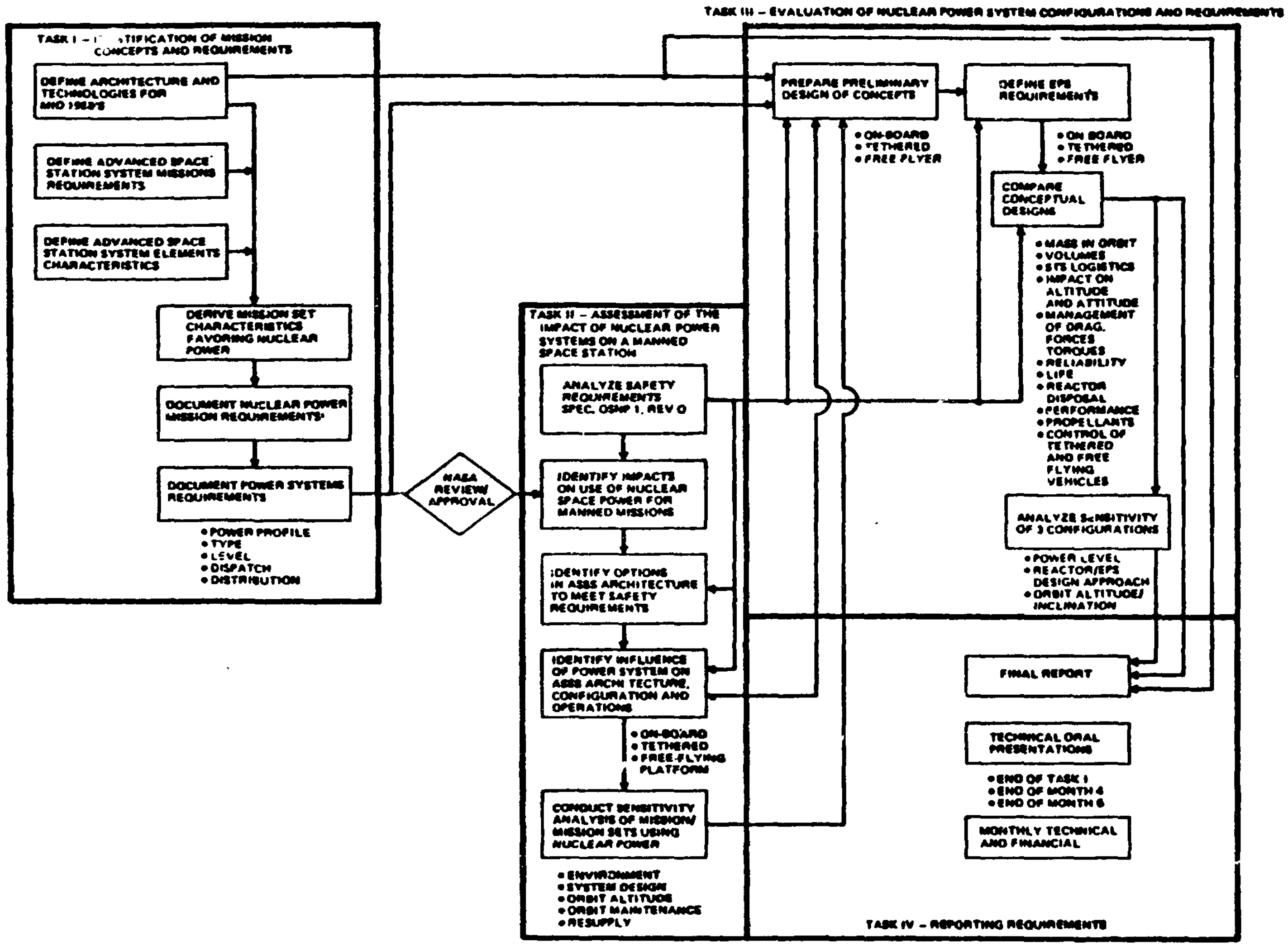

Figure 1: Technical Approach – Functional Flow Diagram of Tasks

heat engine. The loads are space station loads, not generated power levels. The boom-mounted nuclear reactor power system was rated at 153 kWe to compensate for transmission losses and used a shaped four pi shield at the reactor. The tethered nuclear reactor power system was rated at 208 kWe to compensate for electrolysis and fuel cell inefficiencies. A man-rated four pi shield and an instrument-rated conical shield were treated for the tethered reactor. The free-flying reactor required two reactors on the same spacecraft to avoid burnup limitations with the thermoelectric system for the 150 kWe power level. These tandem reactors were each rated at 360 kWe and used one conical (shadow) shield. Although the Stirling cycle conversion could also be considered for the 150 kWe load, the two conversion systems were kept separate so as to evaluate the two concepts. For a 300 kWe load, a single large reactor with a Stirling cycle conversion was evaluated to determine the system parameters.

1.4 Trade Study Parameters

The mass and volume in orbit were determined for each configuration, both for initial operation and for total life cycle operation. Space Transportation System (STS) logistics were then compared. The results are summarized in Figure 2 for 150 kWe and in Figure 3 for 300 kWe at the space station.

From Figures 2 and 3 we see that the boom-mounted nuclear reactor power system requires the fewest shuttle flights for initial operation and essentially no makeup propellant over the ten year lifetime. The boom-mounted and tethered nuclear reactor power systems both require some active form of propulsion to boost the reactor to a long-lived orbit at the end of its useful life, which the free-flyer does not, since it can be operating at any selected altitude. Operating the reactor only in a long-lived orbit, however, requires a large number of shuttle flights to place the initial water/fuel charge in orbit, and even more flights to provide OTV propellant, as well as 22 OTV flights (round trip) for fuel transfer over the ten year lifetime.

From the analysis it was determined that the boom-mounted power system is the optimum for minimum logistics - orbiter flights and total mass in orbit in ten years. If the radiator can be detached, the remainder of the reactor power system with a long tether will, when severed, remain in a 300 year orbit without a supplemental boost.

1.5 General Conclusions

The mission requirements and sensitivity analyses performed for this study indicate that a space nuclear reactor power system is a viable candidate for manned space stations with high power needs. Although no single space station mission was identified which favors nuclear power specifically, mission power requirements of at least 150 kWe or more have been projected for the space station. In this power range, the design of a space station with a solar photovoltaic electrical power system becomes difficult because of the large area of the array and the exclusion volume the array creates because of rotation. These difficulties relate to station dynamics, orientation flexibility, traffic management, energy storage and distribution. Environmental effects involve corona and voltage breakdown due to the ionospheric plasma, and contamination and debris from the space station and vehicular traffic. Other solar power systems with higher efficiency conversion systems may move the point at which nuclear power is used for prime space station power. The atmospheric drag of the reactor-powered systems is extremely low

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	ON-BOARD	TETHER (MAN RATED)	TETHER (INSTRUMENT RATED)	FREE FLYER
INITIAL MASS IN ORBIT	31.5 t	64.7 t	46.9 t	584 t
INITIAL VOLUME IN ORBIT	236 m^3	368 m^3	361 m^3	2790 m^3
STS LOGISTICS FOR IOC NUMBER OF ORBITER FLIGHTS NUMBER OF OTV FLIGHTS NUMBER OF OMV FLIGHTS	 1.3 0 0	 2.6 0 1	 1.9 0 1	 29 2 1
10-YEAR CUMULATIVE MASS	32.0 t	73.2 t	56.5 t	1296 t
10-YEAR CUMULATIVE VOLUME	237 m^3	377 m^3	370 m^3	3560 m^3
CUMULATIVE STS LOGISTICS NUMBER OF ORBITER FLIGHTS NUMBER OF OTV FLIGHTS NUMBER OF OMV FLIGHTS	 1.3 0 1	 2.9 0 2	 2.2 0 2	 68 22 1

Figure 2 System Trade Table. 150kWe at Space Station

	ON-BOARD	TETHER (MAN RATED)	TETHER (INSTRUMENT RATED)	FREE FLYER
INITIAL MASS IN ORBIT	31.1 t	77.8 t	61.1 t	1043 t
INITIAL VOLUME IN ORBIT	241 m^3	428 m^3	421 m^3	4640 m^3
STS LOGISTICS FOR IOC NUMBER OF ORBITER FLIGHTS NUMBER OF OTV FLIGHTS NUMBER OF OMV FLIGHTS	 1.5 0 0	 3.2 0 1	 2.7 0 1	 49.5 5 1
10-YEAR CUMULATIVE MASS	31.9 t	94.2 t	77.7 t	2964 t
10-YEAR CUMULATIVE VOLUME	242 m^3	448 m^3	441 m^3	6752 m^3
CUMULATIVE STS LOGISTICS NUMBER OF ORBITER FLIGHTS NUMBER OF OTV FLIGHTS NUMBER OF OMV FLIGHTS	 1.5 0 1	 3.9 0 2	 3.4 0 2	 128 45 1

Figure 3 *System Trade Table. 300 kWe at Space Station*

when compared with solar-powered alternatives even at the 270 nmi altitude proposed for the space station. In addition, nuclear reactor power system radiators can be oriented into the orbit plane. The compactness of the nuclear reactor system also facilitates assembly of large space structures, by providing accessibility in essentially all directions and because the reactor is insensitive to shadowing by the large structures.

The trade studies showed that the cost of restricting reactor operation to a higher orbit than the space station for maintaining a 300 year decay life is many shuttle flights and many orbital transfer vehicle flights. The boom-mounted nuclear reactor power system, on the other hand, requires only one and a fraction of another shuttle flight for all power subsystems plus space station drag makeup over the entire reactor lifetime. The tethered reactor requires two to three shuttle flights. An orbital boost system is necessary for the boom-mounted and tethered nuclear reactor power systems for end-of-life disposal of the reactor.

Comparison of the various concepts for transferring power from the reactor to the space station revealed that the combination of electrolysis and fuel cells is an attractive option. This combination, which is essentially a separated regenerative fuel cell (RFC) system, uses reactor electrical power to electrolyze water into its constituents which are transported to fuel cells on the space station to be recombined and returned to the reactor as water. The RFC appears to be much more efficient than either microwave or laser power transmission. This system is also attractive because of its synergism with space station life support and propulsion systems because of its use of hydrogen, oxygen, and water. Since the fuel cell reactants and water are easily storable, this also allows the reactor to operate in a base load mode, producing a steady rate of reactants at constant power even while the space station power demand fluctuates. Additionally, for make-up fuel due to leakage and use in other systems, the main source, water, is all that is required. This can be brought up when required and presents no hazard during transportation and storage.

2.0 SPACE STATION MISSION REQUIREMENTS

Space station missions that may benefit from the use of a nuclear reactor power source are the power-intensive materials processing missions and those that require broad accessibility over large volumes. With the Mission Requirements Working Group, NASA maintains a current file of space station mission requirements for planning purposes (ref. 1). The most recently compiled summary projects an average power requirement for 75 kWe at I.O.C. and 150-200 kWe in the mid to late 1990's. Peaking needs and space station housekeeping may raise this need to 150 kWe to perform the missions in the MRWG data set. Additional missions made feasible by the availability of sufficient space station power, or by robust commercial demands, might lead to a requirement for 300-500 kWe in the 21st century. Large space structures assembly missions would benefit from a power source which allows greater accessibility and is less sensitive to traffic, drag, and solar shadowing than large solar arrays.

2.1 Electrical Power Needs

2.1.1 Reference Space Station Mission Set

NASA's Space Station Task Force has compiled a reference file of space station mission descriptions. This reference file was originally established as a validated summary of missions identified by the eight NASA contractors of the Space Station Needs, Attributes, and Architectural Options Study during fiscal year 1983 (ref. 3). The Mission Requirements Working Group (MRWG) reviewed all the contractor mission descriptions in May 1983 to establish a single planning document (ref. 1). This document represents NASA's reference mission set for space station development and planning activities. The MRWG now meets regularly to review and update the Space Station Mission Requirements Report and release this summary report on a monthly basis.

The complete set of missions considered by the MRWG includes candidate missions in three categories: Science and Applications, Commercial Utilization, and Technology Development. These categories are each summarized by a discipline panel which interfaces with its respective user community to analyze and assemble potential missions which could be supported by a space station system. Data included in the monthly Space Station Mission Requirements Report are brief mission descriptions; time phasing; allocation of resources such as power, data transmission rate, crew time, volume, and transportation requirements; and integrated resource requirements.

The complete space station system includes more than just a single, manned facility: it includes a manned facility, free-flying platforms in various orbits, service vehicles, orbital maneuvering vehicles, and launch and resupply vehicles of the Space Transportation System. The current reference mission set for the 1990's considers a single space station in low-inclination, low-earth orbit as its only permanently manned facility. Other facilities that may occasionally be visited by man include platforms in low altitude polar orbit and in geosynchronous orbit.

The average electric power requirements of all the reference payloads in the space station mission set are shown in Figures 4 and 5. The average payload power requirements in the low inclination, manned space station begin with 55 kWe in 1991 and grow steadily to 112 kWe by 1996. The maximum power in the current reference mission set is 123 kWe in 1999. These power requirements are for the

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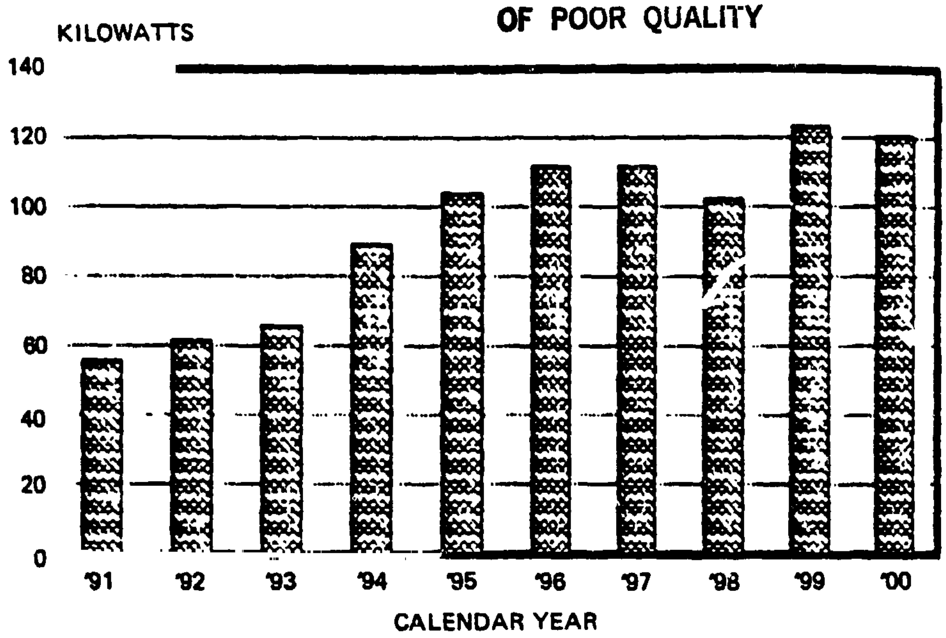


Figure 4   28.5° Station & Platform Mission Power

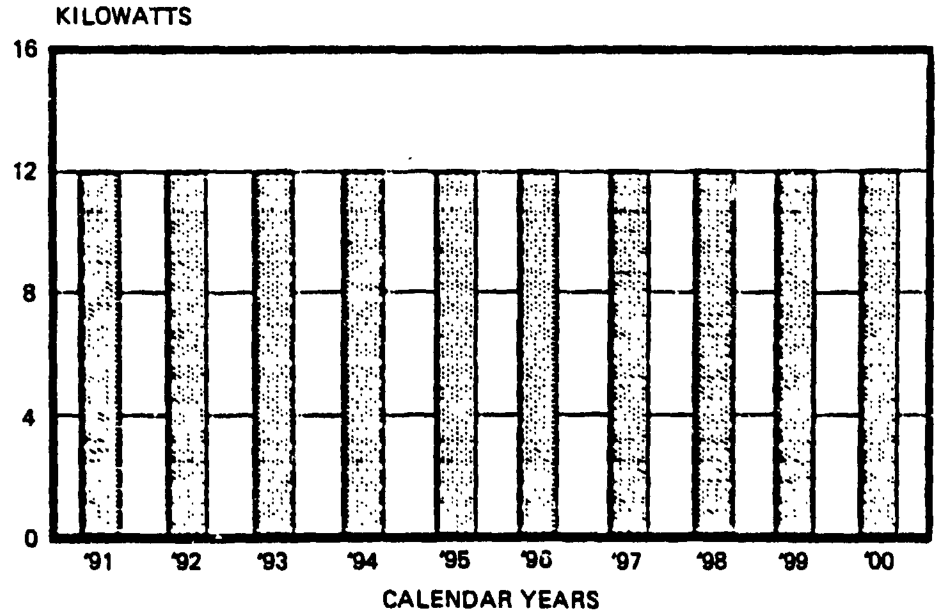
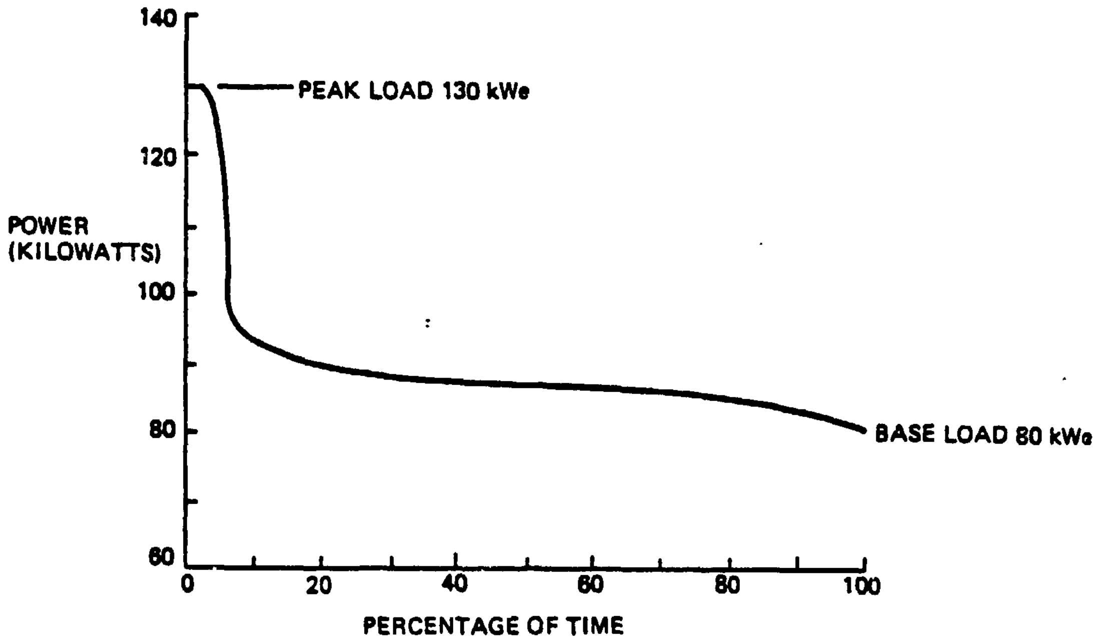


Figure 5   Polar Platform Mission Power

reference missions and describe payload requirements only on the manned space station. They do not include housekeeping loads, nor do they include the power requirements for space transportation.

A typical daily mission load demand curve for the year 1994 is shown in Figure 6. A peaking factor of about 1.5 is evident. The daily duty cycles were prepared without consideration of load-sharing, i.e. no attempt has yet been made to optimize the daily time-sequencing of the various missions to distribute the load. If load-sharing were seriously investigated, it is likely that the peak power of 105-135 kWe in the 1993-95 time frame can be reduced, if no other missions are considered.

A summary of average power requirements for space station missions in the 1994-2000 time period is given in Figure 7. It can be seen from the table that about eighty percent of the power is consumed by materials processing missions. A 20 kWe limit was selected for individual loads as an arbitrary value. The greatest power demand comes from various crystal growth production units, biological processing production units, optical fiber production, and the processing laboratory. Several different methods of crystal growth will be needed for different types of crystals. These missions and their load characteristics are described below.



**Data taken from MRWG report for 1994*

Figure 6 Daily Mission Load Demand Curve

Figure 7 *Space Station Mission Power Requirements*

MISSION TITLE	POWER (kWe)		
	1994	1997	2000
SOLAR OPTICAL TELESCOPE	1.25		
TRANSITION RADIATION & ION CALORIMETER	0.55		
STARLAB	1.66		
HIGH THROUGHPUT MISSION		2	
HIGH ENERGY ISOTOPES		0.3	
PINHOLE/OCCULT		0.5	
ADVANCED SOLAR OBSERVATORY			1.7
ANIMAL AND PLANT VIVARIUM		3.05	3.05
DEDICATED CELSS MODULE			19
CELSS PALLET	0.6	0.6	
LIFE SCIENCES LABORATORY	3	4.5	3
TOTAL SCIENCE AND APPLICATIONS	7	11	27
REMOTE SENSING TEST		8	
COMMUNICATIONS TEST	0.2	0.2	0.2
MPS PROCESSING LABORATORY	20	20	20
EOS PRODUCTION UNIT	15	15	15
ECG PRODUCTION UNIT	20	20	20
IEF PRODUCTION UNIT	4	4	4
DSCG PRODUCTION UNIT	7	7	7
VCG PRODUCTION UNIT	10	10	10
OPTICAL FIBER PRODUCTION UNIT		10.6	10.6
SOLUTION CRYSTAL GROWTH		2	2
IRIDIUM CRUCIBLE		3	3
MERGED TECHNOLOGIES-CATALYSTS			2.5
TOTAL COMMERCIAL	76	100	94
TECHNOLOGY DEVELOPMENT MISSIONS	5.6	1.7	0.65
TOTAL	89	112	122

2.1.2 Crystal Growth

Semiconductor crystal growth missions include electroepitaxial crystal growth (ECG), chemical vapor transport growth (VCG), and directional solidification (DSCG). Electroepitaxial crystal growth of gallium arsenide is illustrated in Figure 8. A saturated solution of a few percent gallium arsenide in gallium is brought into contact with a monocrystalline seed crystal and a polycrystalline source crystal. When electric current is established normal to the seed-solution interface, the arsenic ions migrate toward the seed crystal and crystallize with the solvent on the surface of the seed. As long as the solution remains saturated near the growth region, highly stoichiometric and uniform crystallization occurs by epitaxial growth. The electric current forces transport of the solute to the seed crystal interface so that this region remains saturated as uniform single crystals are grown. The solute concentration is maintained by dissolution, diffusion, and current-driven transport from a polycrystalline source. The process is carried out inside a furnace which maintains precise control of the crystal growth temperature. The temperature range being considered is 800-950°C.

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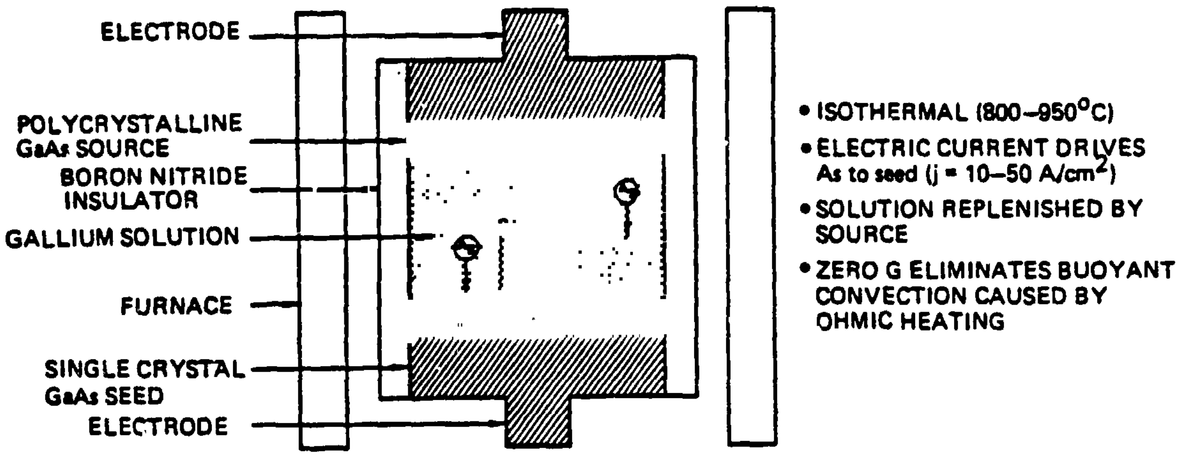

Figure 8 Electroepitaxial Crystal Growth

The electric current densities required for current-controlled electropitaxial growth are in excess of 10 A/cm^2. This current flows through the melt and the crystal. Resistive heating and Peltier cooling at the interface cause temperature gradients in the melt. In a gravitational field these temperature gradients would cause thermal convection, which would destroy the uniformity of the crystal. Thermal convection does not occur under microgravity conditions. Therefore high current densities are allowed in the microgravity environment of space and, since growth rate is linearly proportional to current density, rapid growth of large, uniform, compound semiconductor crystals is allowed.

Microgravity Research Associates, Inc. is developing the electroepitaxial growth process for space-based growth of gallium arsenide (GaAs) crystals. GaAs is a superior semiconductor to silicon in several areas. It has a much higher switching speed, lower power requirements, lower heat loss, and higher temperature resistance. All of these advantages combine to allow gallium arsenide to have a higher circuit element density with greatly enhanced processing speeds and reduced cooling requirements. Furthermore, GaAs is much more resistant to radiation than is silicon, allowing its use in nuclear, space, and military environments where silicon-based semiconductors would quickly degrade. Finally, GaAs emits coherent light, which allows its use in optical processing equipment.

Ground-based GaAs growth experiments have been performed with low current densities and small dimensions to suppress thermal convection, but real experimental verification of the concept must await spaceflight. Microgravity Research Associates has a Joint Endeavor Agreement with NASA to develop the electroepitaxial growth process for gallium arsenide.

The electrical power required for electroepitaxial growth of gallium arsenide depends on both temperature and thickness of the crystal. At 875°C furnace temperature, the energy required to grow 1 cm thick GaAs crystals in five days is about 66 kWh/kg. In addition to the power required for the growth current, an additional power of 40 kWh/kg is required to maintain the furnace temperature for five days. The power load is then 66 kWh/kg of uninterruptible DC electric power at 28 VDC, plus 40 kWh/kg of interruptible power for additional heating.

The estimated market demand for space produced gallium arsenide is shown in Figure 9 (ref. 4). These projections assumed market prices that reflect decreasing production costs resulting from the following flight scenario: in 1990, crystals are grown in the orbiter on six day missions; in 1991, crystals are grown on a free-flying platform which is serviced by the orbiter; in 1992-93, crystals are grown in dedicated modules attached to the space station; beyond 1993, crystals are grown in free-flying platforms serviced from the space station. The average power required to satisfy these demand projections is shown in Figure 10. In the mid-1990's, the power required for electroepitaxial growth of gallium arsenide is likely to be in the range of 10-30 kWe. By the year 2000, this value is expected to rise to as high as 100 kWe. In the MRWG reference set, a ground rule was observed which limited the space station-supplied power for any single commercial mission to 20 kWe. For this study we did not observe the 20 kWe limit. It is implicit in this assumption that commercial space station users requiring more power would either provide their own power source on the space station or would move their processing system to a free-flying platform.

Another method which is currently used for growing semiconductor crystals involves transport of the crystalline elements from a source to a growth crystal in the vapor phase, as depicted in Figure 11. To grow crystals with this chemical vapor transport method, a polycrystalline source of material is heated in the presence of a gaseous transport agent. A chemical reaction between the source and the transport agent results in exclusively gaseous products which are removed from the source. The growth crystal is located at the other end of the growth ampoule, and is maintained at a lower temperature than the source material. The gaseous products are transported down the temperature gradient to the growth crystal, where they undergo the reverse chemical process and condense into the original chemical product, in monocrystalline form.

The crystal uniformity of the product reflects the uniformity of the vapor phase. In full gravitational fields, thermal convection of the vapor disturbs the uniform flow and limits the crystal perfection. Thermal convection does not occur to a significant degree in microgravity, so more uniform crystals can be made at high growth rates.

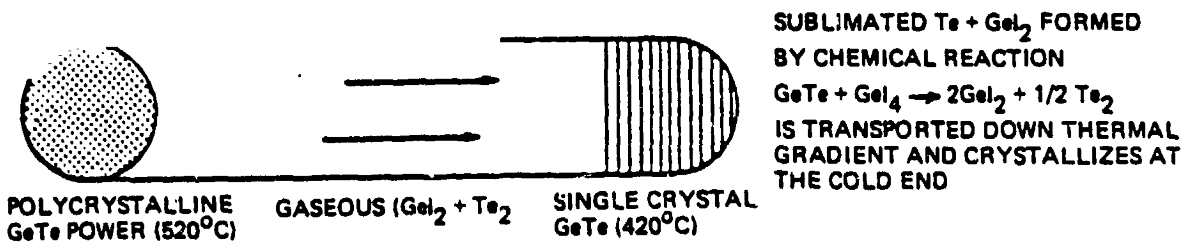

Figure 11 Chemical Vapor Transport Epitaxial Crystal Growth

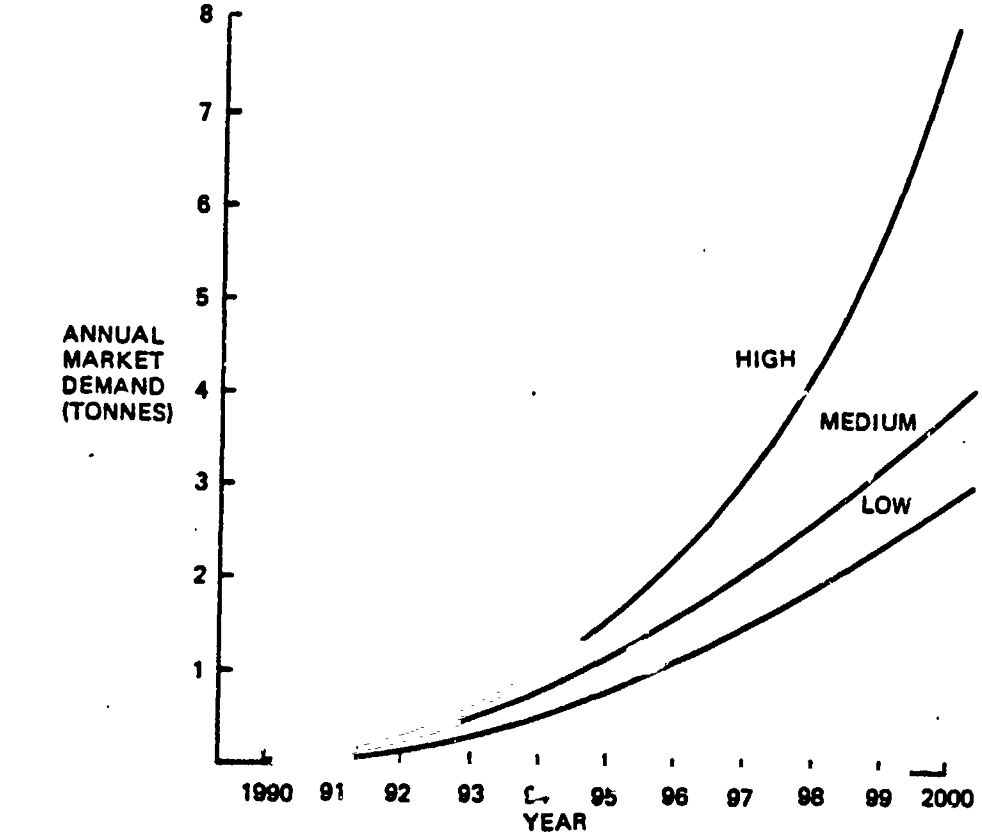

Figure 9 Space-Produced Gallium Arsenide Market Projections

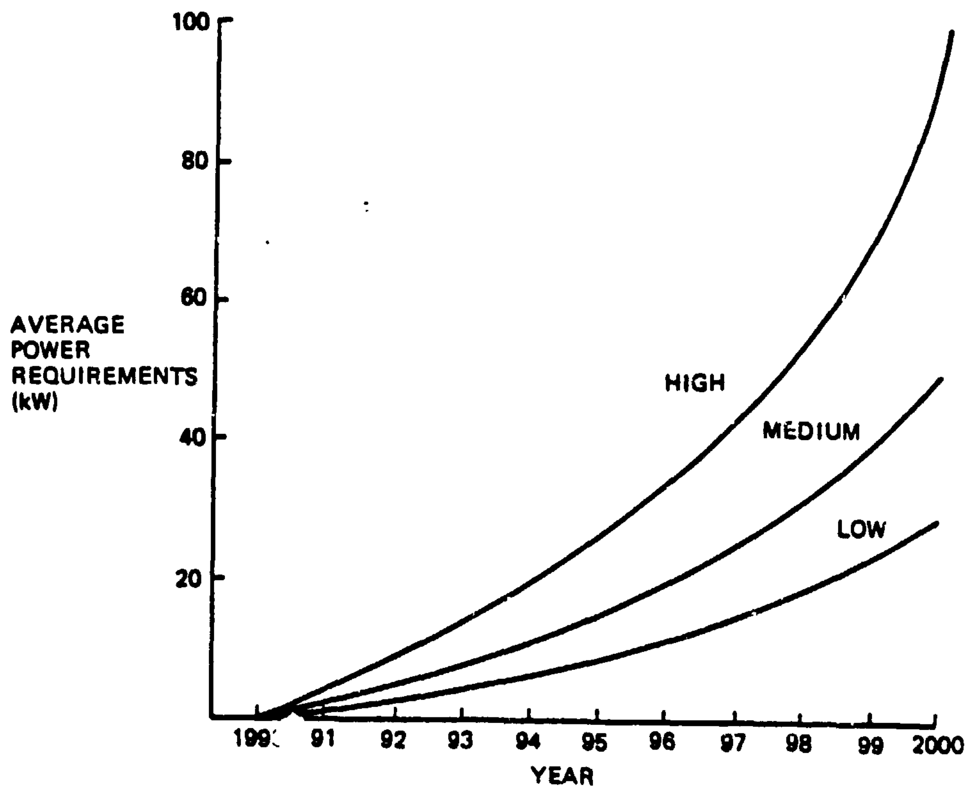

Figure 10 Average Power Required for Gallium Arsenide Production

This process is being developed at Rensselaer Polytechnic Institute for growth of large, monocrystalline, compound semiconductors. Previous experiments have been performed with germanium and group VI compounds (ref. 5). Future plans call for vapor phase growth of large, ternary semiconducting compounds, such as $Hg_xCd_{1-x}Te$. These could be used as infrared detectors, with a response function which can be selected by choosing x.

A series of crystal vapor growth experiments are scheduled on the MEA-A facility aboard the orbiter. The first flight was on STS-6. The crystal material on this flight was germanium-selenium. All subsequent flights, beginning in August 1984, are planned with mercury-cadmium-telluride.

Directional solidification crystal growth techniques can be applied to semiconductors or metals. In general, the material to be crystallized is melted in a crucible within a furnace. The furnace is designed with a temperature profile which encompasses temperatures above and below the melting point. Crystal growth occurs at the cooler end of the crucible. The crucible is either stationary or it is slowly pulled out of the furnace, down the thermal gradient. In either configuration, crystal growth proceeds as a result of heat transfer from the melt.

A steady power of 10 kWe has been assumed for chemical vapor transport production in the mid 1990's, and 7 kWe for directional solidification. This power is required primarily as heat. It can, in general, be interrupted for periods on the order of minutes. It can be either DC, or AC at any frequency.

2.1.3 Biological Materials Processing

The usefulness of a wide range of biological materials depends on the degree to which they can be concentrated and purified. Current processes for separating these materials are often limited by convection. The materials are purified by flow processes in aqueous solution. The sharpness of the flow patterns is degraded by thermal and buoyancy-driven convective forces. This lack of resolution limits the purity of the separation products. Elimination of convective forces can greatly enhance the sharpness with which different materials can be separated, as well as increasing the concentration of the product. The improved separation of pharmaceuticals that can be achieved in space offers a near-term commercial product of space-based materials processing. Two processes have been considered for this application: continuous flow electrophoresis and isoelectric focusing.

Biological materials, such as proteins, enzymes, and cells often have a surface electric charge distribution which causes them to respond to an electric field. When placed in a fluid medium with an electric potential difference between two ends, these materials will be transported toward one end. The speed with which the materials travels varies according to the charge distribution. Different materials have different mobilities. This difference in mobility allows different materials to be separated according to their electric charge distribution.

Figure 12 shows conceptually how a continuous flow electrophoresis apparatus works. A liquid buffer solution is located between two electrodes. A potential difference between the electrodes establishes an electric field in the solution. Those components of the material to be separated which have the highest electric mobility move the fastest to one electrode. After some time in the field, the various components of the material are separated. On Earth, buoyancy-driven convection caused by concentration differences and by density changes due to

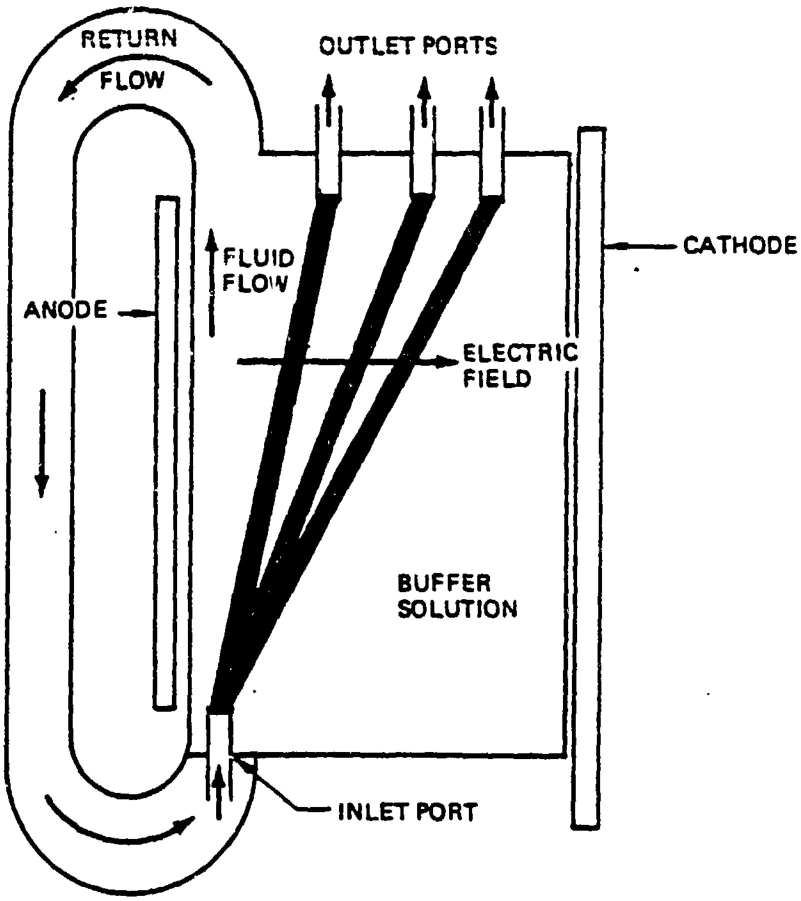

Figure 12 Continuous Flow Electrophoresis

Joule heating limits the size of the sample which can be separated. Typically, samples of less than 0.1 ml are separated on a porous gel plate by a batch process where the samples are frozen into place after some period in the chamber. The small size of the apparatus limits the separation resolution, while convection considerations limit the possible purity of the product.

Elimination of gravity-driven convection in space allows a continuous flow electrophoresis process (CFE) in which the buffer solution flows continually perpendicular to the electric field, while the sample is also added continuously to the processing chamber. Much larger volumes of sample material can be separated in this manner with much larger processing chambers. As shown in the figure, the product materials are collected in distinct collection vials. The continuous processing, larger volume, longer time in the electric field, and lack of convection allows much higher materials throughout, higher yield from a given quantity of sample material, finer separations, and higher purity of product material than can be achieved on Earth.

A similar technique which has been proposed for space-based separation of biological materials is isoelectric focusing. The buffer solution establishes a pH gradient when the electric field is imposed. Since the mobility of the material to be separated varies as the pH of the buffer, the sample material moves in the direction of the gradient to a particular value of the pH: the isoelectric point. The products are well-focused within the pH gradient and then collected as in continuous flow electrophoresis. This method may have potential for even finer resolution than electrophoresis, but has not yet been demonstrated in space. Since the pH environment of isoelectric focusing is extreme, it is not suitable for processing of living cells. Three experiments are scheduled for hormone purification in the orbiter mid-deck in 1984.

The requirements for space station accommodation of biological separations processes are generally less than for semiconductor growth, especially since the temperatures are very low. Power requirements especially are eased. The need for refrigeration to preserve samples imposes special thermal control constraints.

The continuous flow electrophoresis process is currently the most advanced MPS program. The first successful results were obtained in the Apollo-Soyuz spacecraft. In this experiment, in which material was separated by a batch process in a small column and frozen in place, the degree of separation of human fetal kidney cells was higher than any previous results. As a result, distinctions were identified between three different cell types that had not previously been identified. The success of the Apollo-Soyuz experiment, an extensive ground-based research program, and especially the STS-4 mission has resulted in optimism that the concept is understood well enough to do engineering design. McDonnell-Douglas Astronautics and Ortho Pharmaceuticals Division of Johnson and Johnson have a Joint Endeavor Agreement with NASA (ref. 6). The developers now plan to fly a production prototype electrophoresis unit in 1985, which will produce the first commercial product. The development program apparently calls for a fleet of unmanned, shuttle-tended free-flyers beginning in 1986. Although specific plans are proprietary, it seems reasonable to project attaching these automated factories to a space station when one becomes available. This would allow the use of space station power and control systems, as well as facilitating materials storage, delivery, and retrieval by operating through a single central base. Present planning are for CFES III electrophoresis experiment to go on shuttle flight 41-D (to be launched in 1984) attended by Mrs. C. Walker (of McDonnell Douglas Co.) in space.

A large number of biological products has been proposed for space station separation. A recent forecast of the potential for space-produced pharmaceuticals indicated the numbers of patients could be helped annually, shown in Figure 13 (ref. 7).

The power required to process sufficient material by electrophoresis to satisfy this demand in the mid-1990's has been estimated to be 15kWe. Isoelectric focussing of additional materials is assumed to add another 4kWe. This power is primarily that required to maintain the electric field in the apparatus, which must be DC power. Other power consumers are for refrigeration of the biological materials and for fluid pumping. These uses are likely to be for AC power.

FIGURE 13: ESTIMATED ELECTROPHORESIS PRODUCT DEMAND

Bioproduct	Annual Patient Load
Pancreatic beta cells	3.2 million
Epidermal growth factor	1.1 million
Human growth hormone	0.85 million
Antitrypsin	0.5 million
Interferon	20 million

2.1.4 Glasses and Fibers

The reduced gravity in orbit allows materials to be processed in a container-free environment. Fluids in microgravity conditions form large globules that "float" in space without spilling or breaking up. This allows the possibility of melting and resolidifying materials without the materials ever contacting the container walls while in the molten state. This property might be useful for a variety of material classes, the most hopeful class being high quality and unique glasses.

There are two features of glass processing that make the containerless processing available in microgravity especially attractive. First, the high melting points of most glasses make them extremely reactive in the molten state. The chemical reactivity causes molten glass to interact with the container walls, resulting in impurity introduction into the melt. In gravitational processing, these impurities are unavoidable. Since the optical and mechanical properties of glass are very sensitive to impurity levels, chemical reaction with the crucible often seriously degrades the glass quality. Containerless processing should eliminate impurity generation by this mechanism and allow more perfect optical properties and stronger glasses.

Second, glasses are distinguished from metals and other solids by their lack of crystalline structure. Under gravitational conditions, molten glass as it cools tends to solidify around nucleation sites at the crucible walls, because the walls are cooler than the interior of the melt and the impurity level is higher there. Crystals tend to grow around these nucleation sites, resulting in a higher degree of crystalline structure than is desirable. In a containerless environment, a higher degree of supercooling is possible without the onset of heterogeneous nucleation, thus allowing a lower level of crystalline structure and therefore more ideal glassy properties. Homogeneous nucleation also allows the processing of glasses with different chemical mixes than are possible on earth. So containerless processing allows more ideal glassy properties and should allow unique glasses to form which cannot be duplicated on earth.

A facility for processing glass in space will be dominated by the furnace. The furnace has two primary functions: a programmable power supply for heating and a positioning control system for holding the melt in place. The material sample would likely be heated by absorption of some sort of electromagnetic radiation:

most likely in the microwave or infrared frequency range. Other heating mechanisms that have been proposed include electron beam impingement and solar concentrators. Although the melting temperature of most of the candidate glasses is very high, the actual heating power load can be quite low because containerless processing eliminates conductive and convective heat losses. A kilogram sized specimen of silica glass can be melted in a half hour with about a 1 kW heating source. Heat losses can be further minimized by using infrared reflecting walls. A glass specimen would be heated to a few degrees superheat and then rapidly cooled to promote homogeneous nucleation.

The heated samples tend to drift in space due to orbital dynamics and g jitter if they are not actively held in place. They can be positioned by several means. They can be attached to a sting which holds them in place by surface tension. This method may result in heterogeneous nucleation and conductive heat loss to the sting. If the samples can be allowed to come in contact with a cover gas, they can be held in place by acoustic pressure driven by loudspeakers in the walls of the chamber. Truly containerless processing in a vacuum can be achieved by positioning the sample with either electromagnetic or electrostatic forces.

Uses for space-processed glasses will likely be restricted to those for which high purity is essential. These might include optical fibers with high transmissivity which would require fewer repeaters than current systems, and allow faster, more efficient data transmission. Optical fibers are finding increasing use where faster, more compact data transmission is desired and where resistance to electromagnetic interference is demanded.

Specialty glasses which might benefit from space processing include optical filters, where particular spectral bands are to be suppressed, and lead glasses, as for viewing radioactive substances. Another common use is for laser host materials, such as neodymium-doped YAG. It seems likely that many more applications would develop once space-based processing demonstrates the formation of glass forms that could not be reproduced on earth. Optical glasses for lenses and mirrors might be processed in space with very low crystallization, which would provide higher quality image processing.

No containerless processing of glasses has yet been done by the United States in space. Projections of future demand have been based on expected properties of containerless-processed glass - not on actual experimental results. Until experiments have been completed and the results evaluated, it will be difficult to foresee a commercial market. A more plausible development scenario would start with at least a four or five year experimental research program where different glass materials would be formed by different cooling process and examined. An orderly research program would establish the properties of different materials, the effects of experimental conditions such as sample temperatures, cooling rates, and positioning methods, and the efficiency of a variety of heating and cooling techniques. Once this basic research program has advanced our knowledge of space-processed glasses and their fabrication techniques, it may be easier to identify commercial markets with some understanding of costs and benefits.

Assuming a basic research program begins soon and is supported at a reasonable level on a continuous basis, commercial production might begin by 1990. An estimate of the rate of production of space-produced glass was based on a recent estimate of market projections of optical fiber components through 1990, and on the assumptions that space processing will offer significant improvements in fiber quality. An estimate of the production rate of space-processed optical fibers is shown in Figure 14.

Without a specific process for producing containerless-processed optical fibers, it is difficult to project space station accommodations. Reasonably consistent assumptions can be made, however, to make rough order-of-magnitude estimates. Here we assume that a single containerless glass furnace has an electric heater and a radiofrequency positioning system. We further assume that a molten source of glass has a mass on the order of 1-3 kilograms and that this source is fed continuously and 100 fibers can be pulled simultaneously from the source and stored on spools. If the pullers draw 10 m diameter fibers at 1 m/s with an 80% availability, then a single such furnace would produce 435 kg/yr while consuming about 50 Watts of electric power. Assuming that the glass is packaged into the orbiter with a 1.5:1 structure ratio, the demanded throughput by 2000 requires 340 furnaces.

With these somewhat speculative estimates, the electrical power demand can be scaled. The power grows steadily with calender time, reaching about 5-6 kWe by 1994-96 and 17 kWe by 2000. This power can be AC or DC, but interruptible only for short periods.

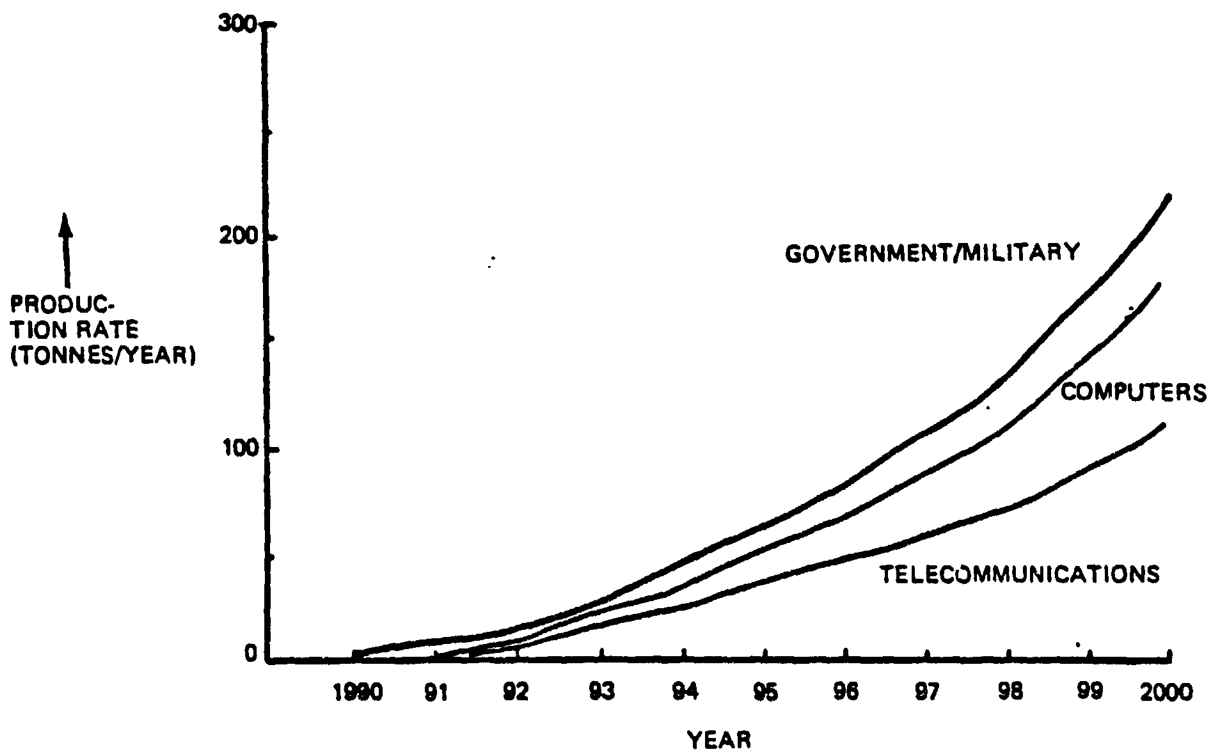

Figure 14 Estimated Space-Processed Optical Fiber Production Rate

2.1.5 Materials Science Laboratory

Materials Processing in Space (MPS) is at an early phase of evolutionary development. At the time of a manned space station MPS will be at verification of concept (VOC) and engineering demonstration (ED) phases. These phases will have had extensive ground-based research and (in many cases) shuttle/Spacelab investigation of concept (IOC) phases as precursors. In the period before the deployment of the space station there must have been at least 5 years of continuous commitment to IOC at a level of at least $30 million per year. Such a committment would be adequate to support about 50 ground-based (IOC) research endeavors during one year. In the early years, from these 50 experiments, about 10 may be selected for flight and accommodation on the space station. In addition, a modification to these 10 experiments, or 10 new experiments, would take place each year. Should the MPS facility aboard the space station be made available to international participants, there would probably be a doubling of experimental activity.

The most important advantages of the space station over the shuttle/Spacelab are that it would provide experimentation facilities with much more power and greatly extended time in space. The space station would also allow the presence of the human experimenter in space. Human experimenters are essential in the early phases of MPS development because it is only after a process has been reduced to a routine that automated manufacture can be considered.

The space station as a facility for MPS can be a national (or international) laboratory for continued research and development in materials that exploit the unique low-gravity environment of space. The early configuration and capabilities will be determined by the prior commitment to MPS and the experience gained through use of the shuttle/Spacelab. Because this commitment may justify only about 10 VOC and/or ED endeavors per year, the start-up of MPS activities aboard the space station should be designed accordingly. Success, even at this modest activity level however, will stimulate construction of new facilities as needed. The industrial infrastructure and technological capabilities that produced the space station should be adequate to meet the requirements for future expansion of MPS activities.

The following is a list of candidate, multi-purpose MPS experiment systems which may require accommodation in a space station materials science laboratory.

- Solidification Experiment Processing System
- High Gradient Furnace Processing System
- Electromagnetic Containerless Processing System
- Isoelectric Focusing Separation System
- Float Zone Processing System
- Acoustic Containerless Processing System
- Electrostatic Containerless Processing System
- Solution Crystal Growth Processing System
- Vapor Crystal Growth Processing System
- Bioprocessing Systems
- Fluid Science Facility
- Combustion Science Facility
- Extraterrestrial Materials Processing Demonstrations

The limits to the availability of power (about 1.5 kWe continuous) on the shuttle/Spacelab is currently one of the most constraining influences on MPS. Experiments with high melting point materials (most notably the electronic materials with a high commercial value) will dominate the power requirements of the MPS facility. A float-zone processing experiment is an example of an experiment requiring a large amount of power when designed to allow free, 360° access to instrument observation of a molten zone. In this case, about 16 kWe are required for the heat source to process a 5 cm diameter sample of silicon (1410° C melting point temperature) using an incandescent source with focusing reflective optics. The power required to process samples having different sizes and melting points is proportional to the square of the sample diameter and approximately proportional to its absolute melting point temperature.

An example of an intermediate power requirement for the processing of electronic materials would involve the use of an insulated high gradient (250 K/cm) furnace. In this case, the insulating enclosure of the furnace reduces the required power considerably. To process a 5 cm diameter sample of an electronic material with a melting point of 1400°C in such a furnace would require a power source of approximately 1 kWe. This power requirement is nearly proportional to the square of the sample diameter and the design temperature gradient of the furnace and only weakly dependent on the melting point temperature of the sample.

All experiments have a minimum power requirement needed for experiment manipulators, data handling and display, controls, and instruments. A reasonable estimate of these requirements, based on Spacelab experiment requirements, is approximately 0.5 kWe per experiment system. Most experiments at room temperature do not appreciably exceed this minimum power requirement. The use of 1.0 kWe per room temperature experiment can reasonably be assumed.

Within these technical guidelines one can estimate the power requirements for an early version of a materials science laboratory based on the capabilities identified in Figure 15.

The power required for the early versions of a materials science laboratory will be dominated by the processing needs of electronic materials. These needs are dependent on and adjustable to, the size of the sample to be processed and, to a lesser degree, on the processing temperature.

Larger size samples for experiments may be required to demonstrate the validity of the process on a pilot plant scale. This increase in scale will require one or two orders of magnitude greater power as success of experimentation beyond the ED phase dictates.

FIGURE 15: ESTIMATED POWER REQUIREMENTS FOR EARLY VERSIONS OF MSL

Experimental Mutli-Purpose Experiment Facilities	Processing Temperature (C)	Sample Diameter (cm)	Power Heating Source (kWe)	Other (kWe)
one (1) high-power electronic materials processing	1500	5	16	0.5
four (4) intermediate power electronics materials processing	1500	5	4	2.0
five (5) room temperature (pharmaceutical and other)	25	-	-	5.0
Subtotal: 10 multipurpose experiment systems			20	7.5
Approximate Total : 30 kWe				

2.1.6 Need For Uncommitted Power

The preceding materials processing missions represent potential commercial enterprises at this time. It is the nature of commercial enterprise during the early investigation of concept phase that specific market areas and magnitudes are speculative estimates. The NASA reference mission set is a reasonable scenario of mission development based on current knowledge of projected product demand, current understanding of microgravity processes, and estimates of future development of competing processes. It is possible, even likely, that the actual commercial materials processing mission set in the 1990s will be significantly different than currently projected; some of the missions projected today will not become commercial successes and other missions not yet conceived will be successful ventures.

Although specific details of the commercial MPS mission set cannot be confidently predicted, the NASA reference set represents a reasonable projection based on current knowledge and credible assumptions. A number of generalizations can be made of likely market trends. For instance, it is clear that any commercial process must develop through the various phases discussed above. Any new process under development will likely consume at least one kilowatt during the verification of concept phase and more during the engineering demonstration phase, eventually growing at 2-10 kWe/yr once a commercial market is established. If new materials and processes are to be developed, then sufficient power must be made available to progress through the pilot/prototype phases. If a number of processes are to be developed as the commercial opportunities are seized, then a surplus of power must be available to test these processes. This means that, whatever power level is required to satisfy the demand of existing missions, additional power is necessary if new commercial missions are to be developed.

If the space station power level is determined by assessing the requirements of predefined missions, then the commercial MPS market volume will become a self-fulfilling prophecy. There will always be just enough power available to satisfy the requirements of the planned mission set, and never enough to develop new missions. Although this might be a satisfactory approach to fulfilling the requirements of a pre-planned program, it would not satisfy a market-driven commercial demand. The commercial attractivness of materials processing in space therefore depends to a large extent on the availability of ample power beyond that which is committed to established missions.

2.1.7 Housekeeping Power

The previous electrical power requirements are all for mission needs. A manned space station has certain power requirements beyond mission needs just to maintain utility, or housekeeping, functions. These functions include life support system operations, lighting, communications/telemetry, thermal control, and data management. These power requirements have been investigated in a previous Boeing study of a Space Operations Center (ref. 22). A breakdown of the requirements is given in Figure 16. The reference configuration is for an eight-person crew with a highly closed environmental control/life support system.

An investigation of the various contractors' reports of the NASA Space Station Needs, Attributes, and Architectural Options study revealed a remarkably close agreement for those contractors that estimated housekeeping demand. The estimated requirements scale with crew size when the life support system is open. Comparing the various studies, the housekeeping power requirement is 3.70 ± 0.56 kW/crew-member. Figure 17 shows how the housekeeping demand is likely to change with time. The bottom curve shows the power demand with an open ECLSS and a minimum crew size-about 20-30 kWe.

The middle curve shows the requirements for the CDG reference crew size, which starts at eight people in 1991 and grows to ten in 2000. There are considerations to vary crew size over different numbers. Assuming a space station which grows in scope and crew size, it is likely that the ECLSS loop will gradually become closed in the late 1990's as our familiarity with space station operation grows and the ECLSS technology advances. The top curve in Figure 17 shows the housekeeping power requirements as the space station crew size grows to 16 and advanced ECLSS closure is developed. By the year 2000, the housekeeping power requirements may grow to 75 kWe. This value must be added to the mission power requirements to determine the total space station power needs.

2.1.8 Electrolysis Propellant Production

When space-based orbital transfer vehicles (OTVs) are added to the space station system, they will likely use H_2/O_2 propellants. Extreme caution must be taken in launching these propellants in the space shuttle. The safety hazards associated with transporting liquid hydrogen and oxygen complicate the launch integration and necessitate massive tankers. The tanks must be much more massive and complex for liquid hydrogen and oxygen than for water. The launch safety can be greatly simplified and the actual launch mass lightened if the propellant is transported from earth to the space station as liquid water and thus electrolyzed to its H_2/O_2 constituents at the space station. If sufficient power were available for electrolysis for OTV propellant production, then this might become a desirable mission.

LOAD	REFERENCE CONFIGURATION				INTERMITTENT POWER	
	SUNLIGHT		OCCULTED***			
	AC	DC	AC	DC	AC	DC
• LIFE SUPPORT	7,119W	10,209W	6,565W	2,610W	3,850W	3,750W
• COMMUNICATIONS/TELEMETRY	–	9,370W	–	9,270W		
• DATA MANAGEMENT SYSTEM	–	1,000W	–	1,000W		
• PROPULSION SYSTEM (HEATERS)	–	200W	–	200W		
• THERMAL CONTROL SYSTEM	300W	2,000W	300W	2,000W		
• ATTITUDE CONTROL SYSTEM	–	250W	–	250W		
• ELECTRICAL POWER SYSTEM LOADS	12,500W	4,500W	12,500W	4,500W		
* (BATTERY RECHARGE FOR HOUSEKEEPING LOAD)	–	(29,900W)	–	–		
	19,919W	27,522 (57,429W)	19,365W	19,830W	3,850W	3,750W

*Reference Only

Figure 16 *Electrical Load Summary for Reference 8-Person Station*

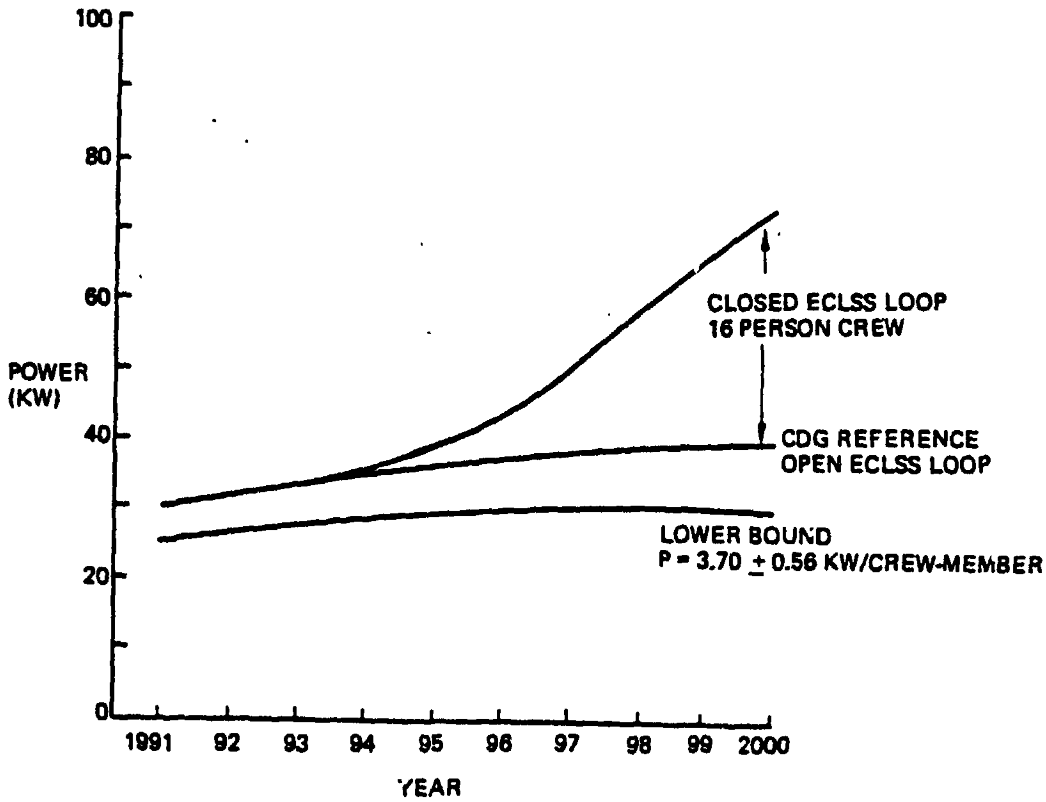

Figure 17 *Housekeeping Power Requirements*

The propellant required for one round trip of an OTV from a low-earth space station to geosynchronous orbit will be about 20,000 kg. The electrical power required for electrolysis of one kilogram of water is about 4 kWh. By the year 2000, the projected traffic for a space-based OTV is about six round trip flights per year. If all the propellant for these flights is transported in the orbiter as water and electrolyzed on the space station in low earth orbit, then an additional 55 kWe of power would be required at the space station. It is unlikely that this mission would be implemented if the additional power were to come from solar panels, but if the incremental power were easily available from a reactor, then it might become a viable approach.

2.1.9 Summary of Power Requirements

The electrical power requirements are summarized in Figure 18. The requirements are divided into three categories: mission loads, housekeeping loads, and electrolysis power for propellant production. Curve A represents the NASA standard reference mission set. This is the summary of average power required for missions currently planned in the NASA space station program.

The minimum space station power capability must include more power than just the average required for missions. It must include housekeeping loads and some excess for peak loads. Curve B represents the minimum capacity of a space station which could satisfy the MRWG reference mission set. It assumes a 30 percent peaking factor on the mission requirements and an open ECLSS loop which uses 3.7 kW per crew member. This minimum power scenario begins with 90 kW in 1991 and grows steadily to 200 kW by 2000. Thus, a space station will require at least 200 kW just to satisfy the MRWG reference mission set with minimum housekeeping for a 12 person crew.

If the commercial materials processing volume were driven primarily by market considerations rather than being constrained to available power levels, it is likely that higher power would be demanded. Curve C represents the power requirements in a market-driven scenario. It was assumed in projecting this curve that each new material being developed follows a market demand growth like that of Figure 9. It was also assumed that three processes are eventually commercialized for crystal growth: electroepitaxy, chemical vapor transport, and directional solidification; and that one new product is developed for each process every three years. Curve C also includes pharmaceutical processing by electrophoresis for six products and isoelectric focusing for one more product by the year 2000, and optical fiber production as shown in Figure 14. An open ECLSS loop and a 30 percent peaking factor was assumed for Curve C. This market-driven scenario leads to rapid power demand growth in the late 1990's, reaching 330 kWe by 2000.

As the space station evolves and the crew size grows, a permanent manned presence will require increasing closure of the environmental control and life support subsystems (ECLSS). Partial closure of the space station ECLSS involves recycling waste water and atmospheric components, including carbon dioxide and nitrogen. Curve D depicts a space station power growth profile which reflects an open ECLSS loop for the initial space station and gradual upgrading to a partially closed loop by the year 2000.

Finally, Curve E represents the power requirements if water electrolysis in space is used for production of propellant for a space-based orbital transfer vehicle. The projection assumes two flights per year by 1997 and six flights per year by 2000.

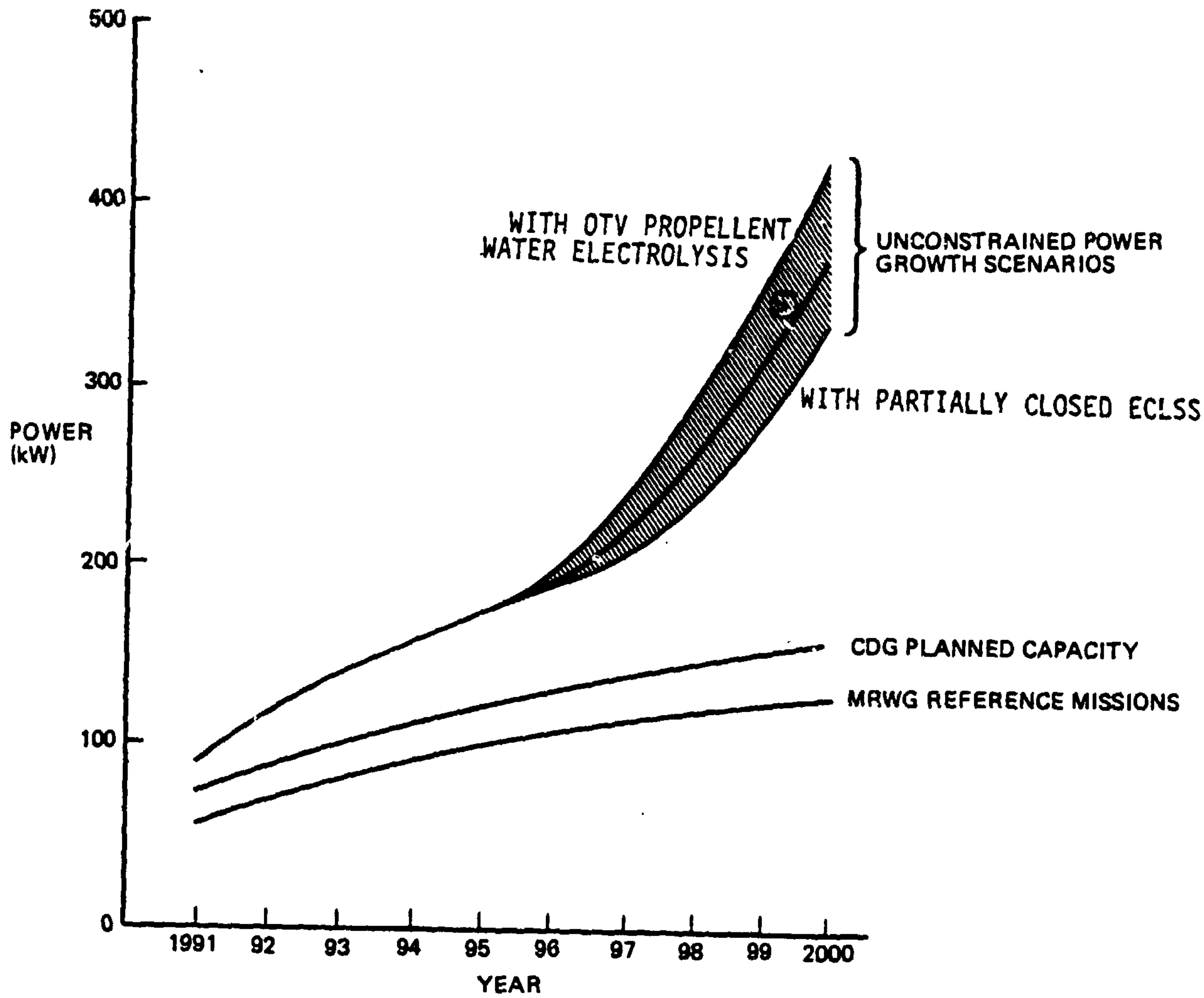

Figure 18 *Summary of Space Station Power Requirements*

2.1.10 Emergency Power/Safe Haven Power

A reliable source of auxiliary power, but not a load to be supplied by the nuclear reactor power, must also be provided on the space station in case of failure of the primary power source. This auxiliary supply should be capable of sustaining life support and communications operations under emergency conditions. A space station ground operations requirement being considered is that an emergency shuttle flight could always be available for a launch to the space station for a rescue mission within 21 days, maximum. The average power required during such emergency conditions will likely depend on specifics of space station design and crew size, but 10 kWe for an 8 person crew seems to be a reasonable estimate. The emergency power supply should therefore be capable of highly reliable supply of 10kWe of electrical power for a period of 21 days. If more than the normal crew size is to be accommodated, such as at crew change, then higher power levels will be required. This power includes life support, communications, and any power which would be required for reactor restart or removal following emergency shutdown. Some versions of the emergency power/safe haven power supplies have separate energy storage which can be dedicated when main power is lost. Other concepts show the power for emergency or safe haven being drawn from the main power system.

2.2 Additional Mission Factors

The magnitude of the electrical power load is not the only factor that tends to favor a nuclear power source for space station missions over a solar system. Other factors are summarized in Figure 19.

FIGURE 19: NUCLEAR POWER SYSTEM BENEFITS

- System scaleability to high power levels
- Power generation system size independent of orbit
- Potential elimination/minimization of exclusion volumes
- Insensitivity to shadowing from large structures
- Low drag (altitude dependent)
- Controllability
- Insensitivity to natural environment

Since a solar power source can only generate power while in sunlight, the arrays must be sized also to supply energy storage for nightside use. The energy storage will be either a battery, regenerative fuel cell, or flywheel. The solar array, power conversion, and power conditioning equipment are sized for energy storage recharge, the load, and losses. Thus, a solar system with a 100 kWe mission load might have to generate at least 200 kWe on the sunlit side of the orbit. This is not true of a nuclear power system, where the power level does not include orbital considerations, i.e. recharge of energy storage. If load leveling is used, energy storage would be included and recharge of the energy storage would be a load on the reactor.

Large solar arrays pose a problem with restriction of available volumes. Regardless of space station orientation, the arrays remain in a fixed inertial orientation, i.e. they always point toward the sun even as the space station revolves around the earth. The volumes that they sweep out during their rotation relative to the space station are much larger than the station itself. These volumes are not accessible for time periods larger than one orbit period. Thus, they cannot be used for large structure assembly, as shown in Figure 20. Other space station architectures might mitigate this problem, but the large sweep volume of the arrays will always result in large exclusion zones. The effect of this exclusion volume will vary with space station design and the relation of the volume to the working area around the station.

The impact of solar array sweep volumes on large space structure assembly is more graphically illustrated in Figures 20a-d. In the first, Figure 20a, it is apparent that orbiter traffic is curtailed by the presence of the arrays. The orbiter must maneuver close to the space station without damaging the arrays or contaminating them with maneuvering propellant. For the split array configuration shown here, the maximum "throat" size of any structure is determined by the clearance between the arrays, as shown in Figure 20b.

This maximum throat width is determined when the initial space station is built, and will likely never increase even if the power system is upgraded. Figure 20c illustrates the sizes being considered for space station assembly missions. The mission depicted here is a passive microwave radiometer scheduled for 1992-95 deployment in the reference mission set.

Figure 20d illustrates another problem with large structure assembly that is alleviated by using a reactor. That is the issue of shadowing. Space station orientation is severely limited by large structures to avoid shadowing of the arrays. This may prove difficult to avoid in situations like that shown here. A reactor would alleviate this problem, since it is independent of the sun.

The large size of the solar arrays necessary for power levels in excess of 75-100 kWe poses other problems as well. The preferred space station altitude has been largely determined by consideration of atmospheric drag due to large arrays. Figure 21 shows the orbit decay time of a space station placed in a 400 km orbit and in a 525 km orbit with no stationkeeping (ref. 8). The orbit makeup propellant required to maintain a space station with an eight-person crew and 75 kWe power level in a 500 km, 28.5° inclination orbit is 4000-5400 kg/year if N_2H_4 is used for orbital makeup propulsion, and 2600 kg/year if cryogenic propulsion is used. The drag makeup requirement for an attached reactor of the same power is 40 kg/year.

Finally, the compactness, radiation resistance, and independence of optical quality of a reactor power supply makes it less sensitive to the natural environment than solar arrays. The compactness reduces ram/wake and plasma effects at space station altitudes, including high voltage induced over broad exposed areas. The natural radiation environment has little effect on reactor operation, although a reactor must nevertheless be oversized at beginning-of-life because of fuel burnup.

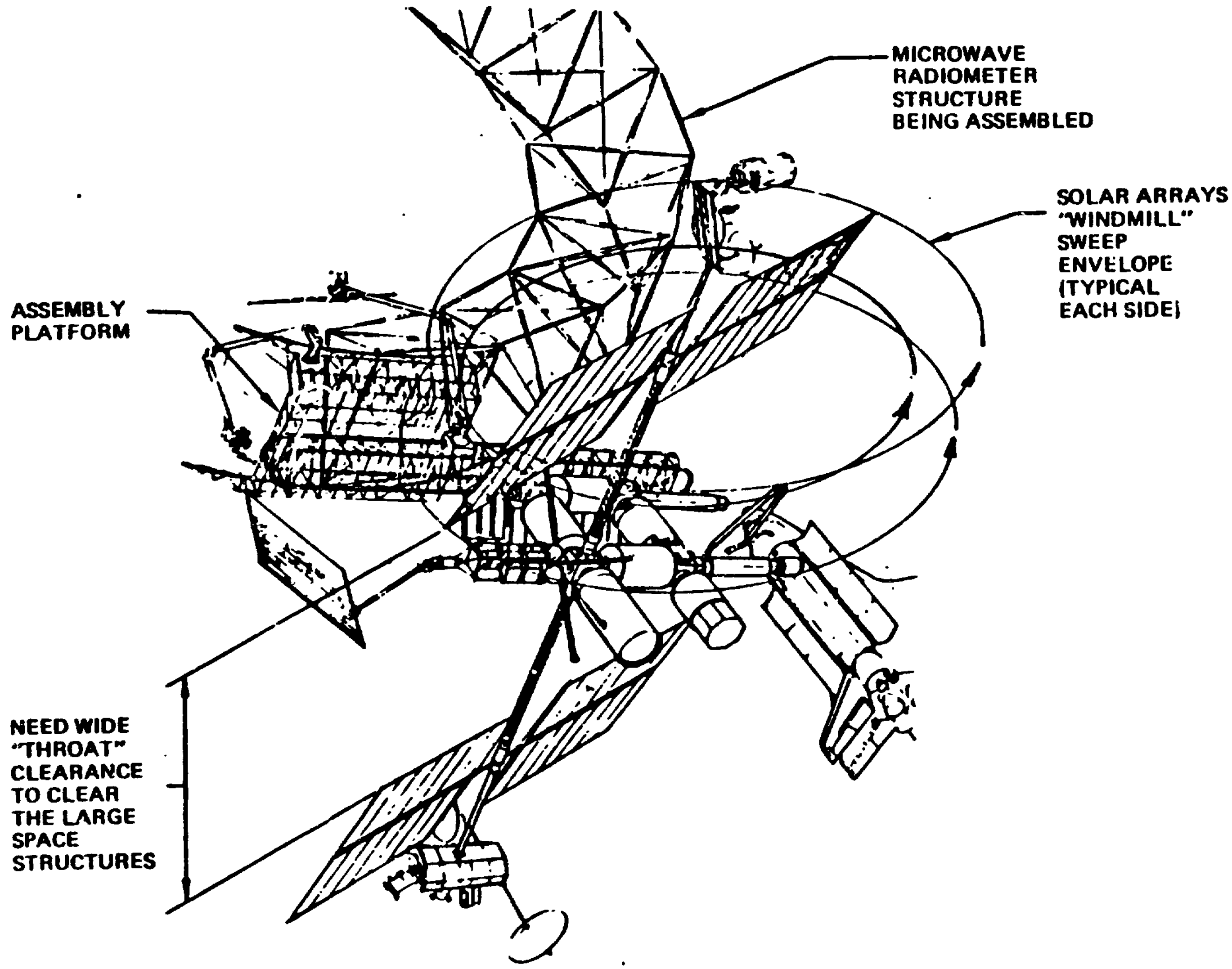

Figure 20 *Large Space Structures Assembly Operations*

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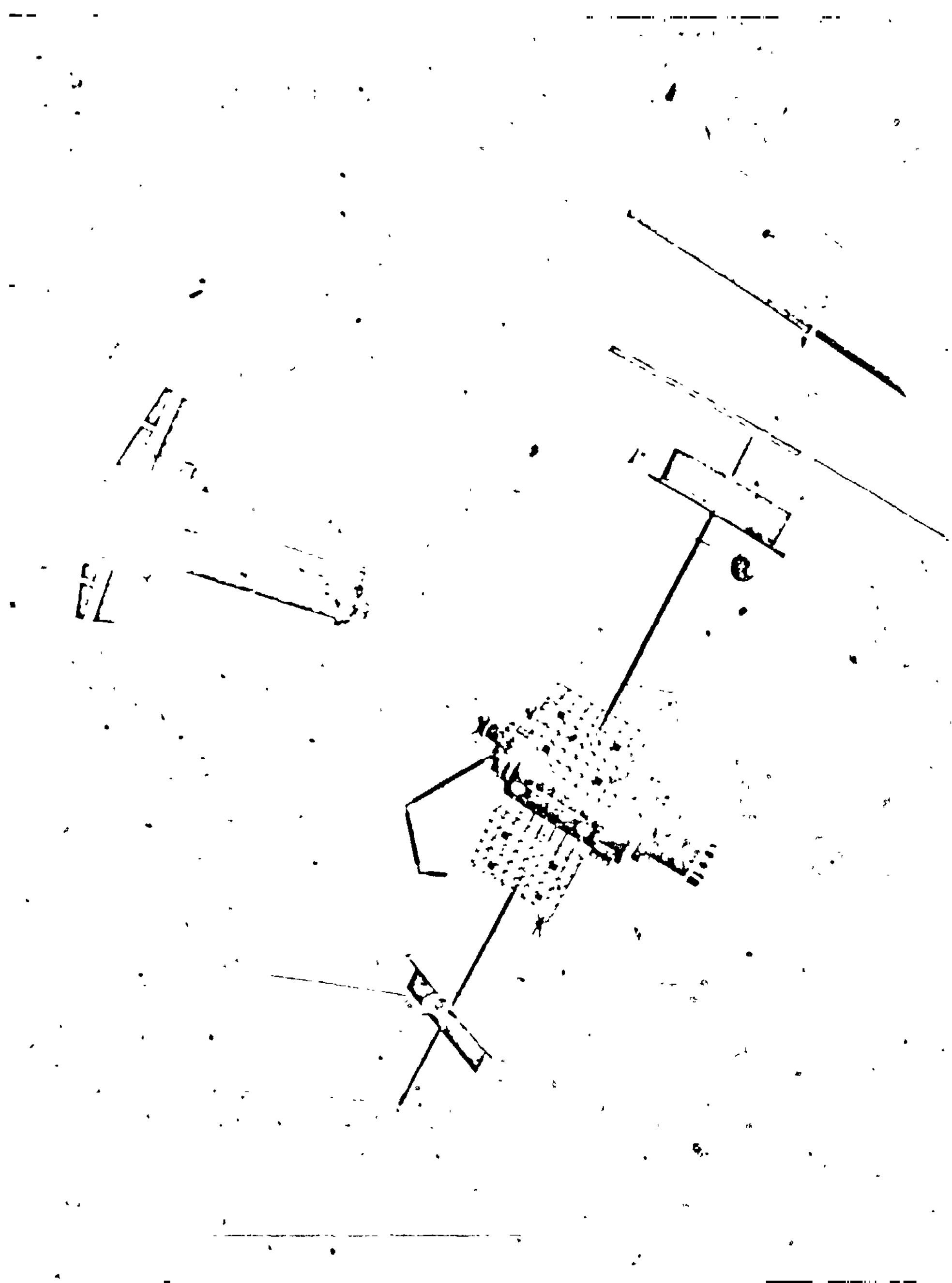

Figure 20a Orbiter Approaching Space Station

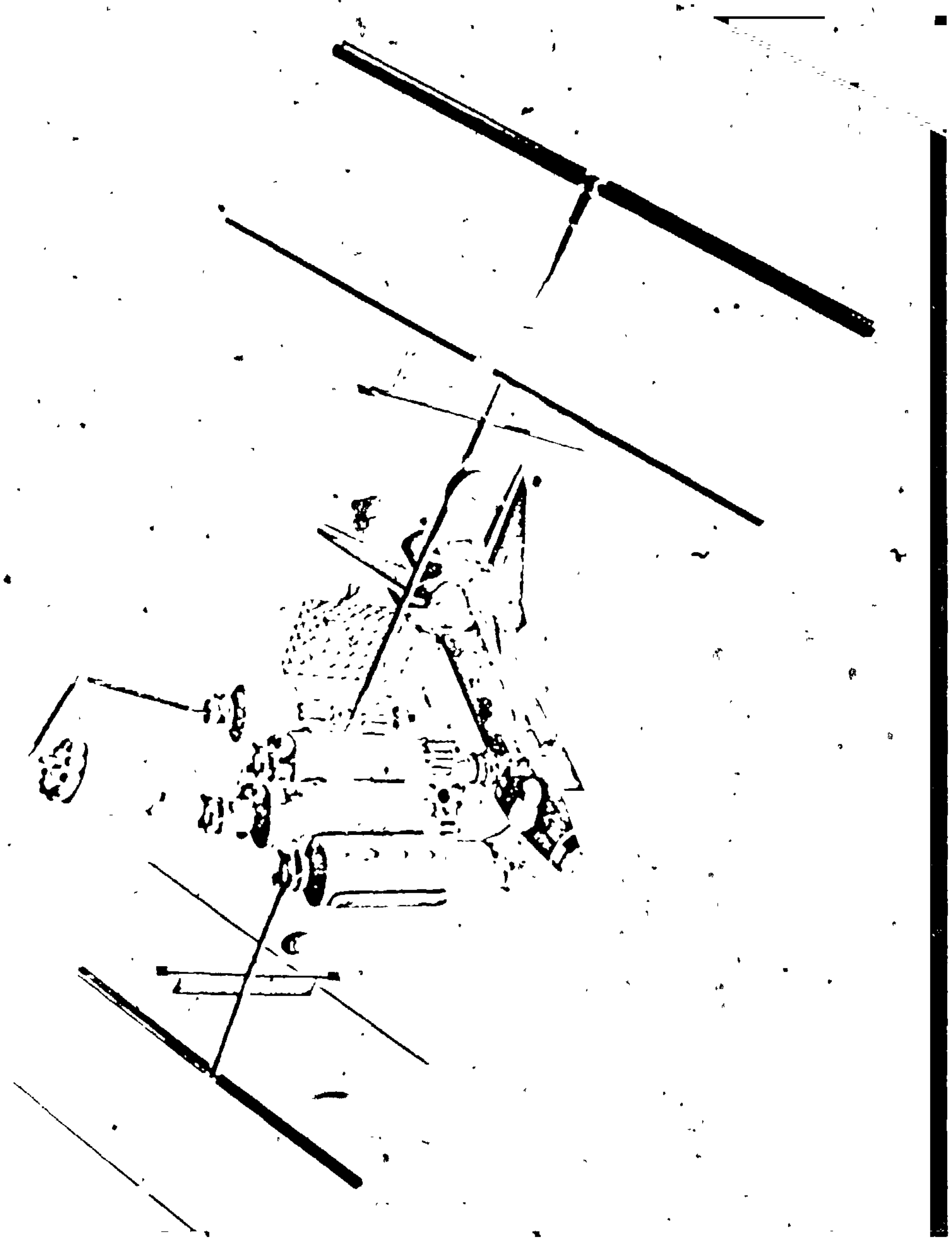

Figure 20b Orbiter Docked At Space Station

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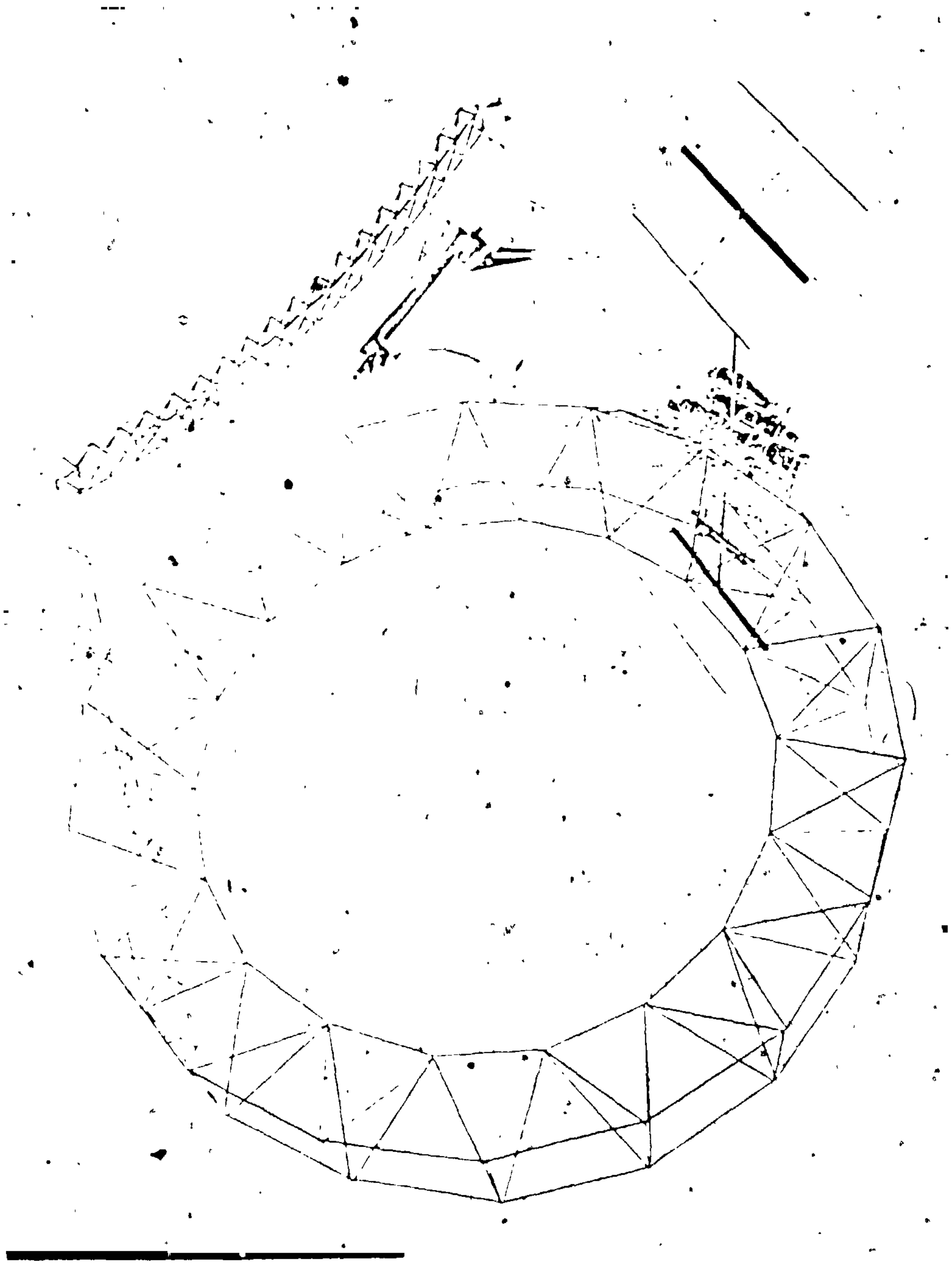

Figure 20c   Large Assembly Under Construction

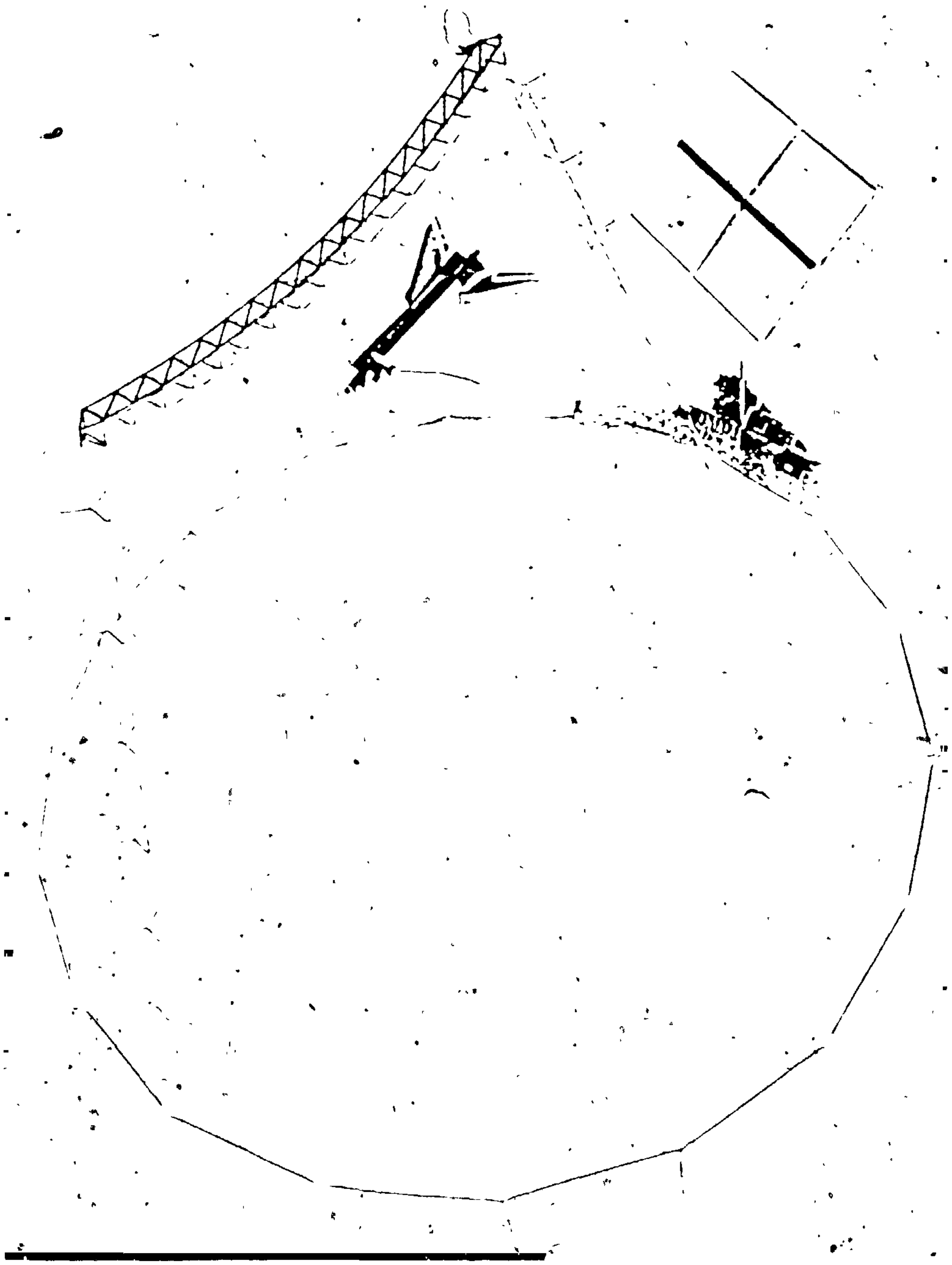

Figure 20d Array Shadowing From Large Structure

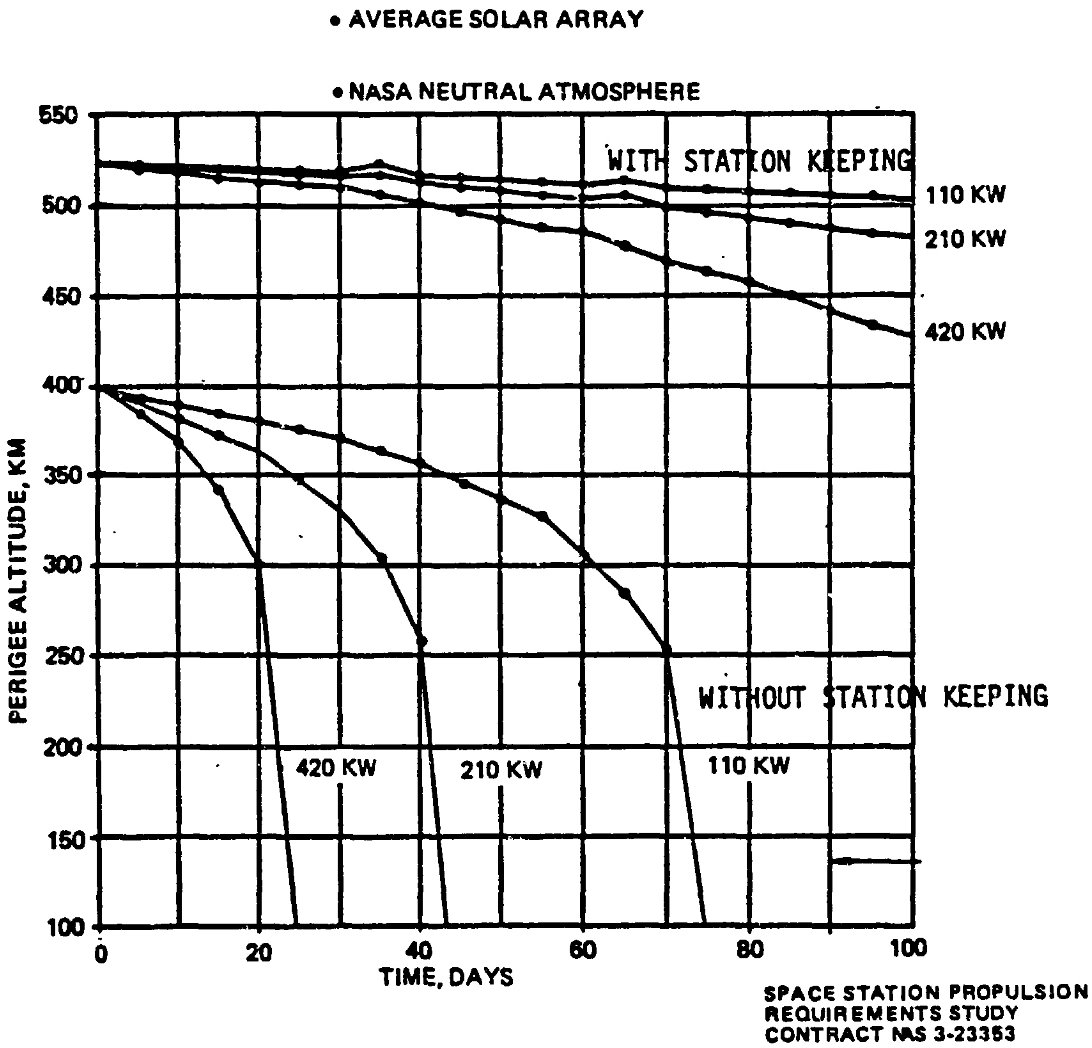

Figure 21 Space Station Orbit Decay Time

3.0 SPACE STATION SYSTEM ARCHITECTURE

The space station architecture and design requirements have yet to be defined. Several general architectural options are still under consideration, and preliminary space station design is expected to begin in late 1984. The space station system, however, will have a number of elements in addition to the manned space station facility, including vehicles and platforms. These elements and their anticipated capabilities in the mid to late 1990's are summarized below. Although the nuclear reactor electrical system will not be involved with all the elements of the space station system, all the elements are described in order to provide a complete picture so that the analyses of on-board, tethered, and free-flyer vehicles with reactors can be made and the system interrelationships can be evaluated.

3.1 Summary of Major Elements

The major elements of the complete space station system are depicted in Figure 22. The items inside the shaded circle are considered space station elements. The reference space station system consists of the manned space station, unmanned platforms, an orbital maneuvering vehicle (OMV), and an orbital transfer vehicle (OTV). This system is expected to be in place by the mid-1990's - the time frame being considered for early reactor operation in this study. Additional elements being considered for a growth space station system in the late 1990's include a geosynchronous manned space station, a manned polar platform, a manned orbital maneuvering vehicle, and manned orbital transfer vehicle, and shuttle-derived cargo launch vehicles. Each of these elements are described below.

3.2 STS Capabilities

For the reference space station of the early to mid 1990's, the launch vehicle from earth to low earth orbit will be the space shuttle, or STS. The STS consists of the two solid rocket boosters, the external tank, and the orbiter. All payloads to orbit must fit within the constraints of the orbiter cargo bay. The dimensions of the cargo bay which are available for payload are 18.3 m long and 4.4 m diameter. The launch capabilities from Kennedy Space Center are shown in Figure 23 (ref. 9). Currently, the orbiter can place 19,500 kg in a 500 km high, circular orbit with 28.5° inclination. With improvements planned for the main engines, by the time a space station is in orbit, the orbiter payload mass is expected to be at least 25,000 kg. This latter value has been used in this report as the STS launch capability.

3.3 Space Station

3.3.1 Space Station Modules

Although a space station design has not yet been prepared, it is expected to contain the following types of modules: resource modules, living quarters, laboratory modules, logistics modules, servicing structure, and a manipulator system. It may also contain multiple berthing adapter modules, if the specific design warrants it.

The resource module provides the space station electrical power, communications, data processing, attitude control, and orbit reboost. This module also has a thermal control/radiator system to reject heat from the resource module equipment and possible equipment in other station modules. This resource module includes built-in capability (scar) for future power and thermal control growth.

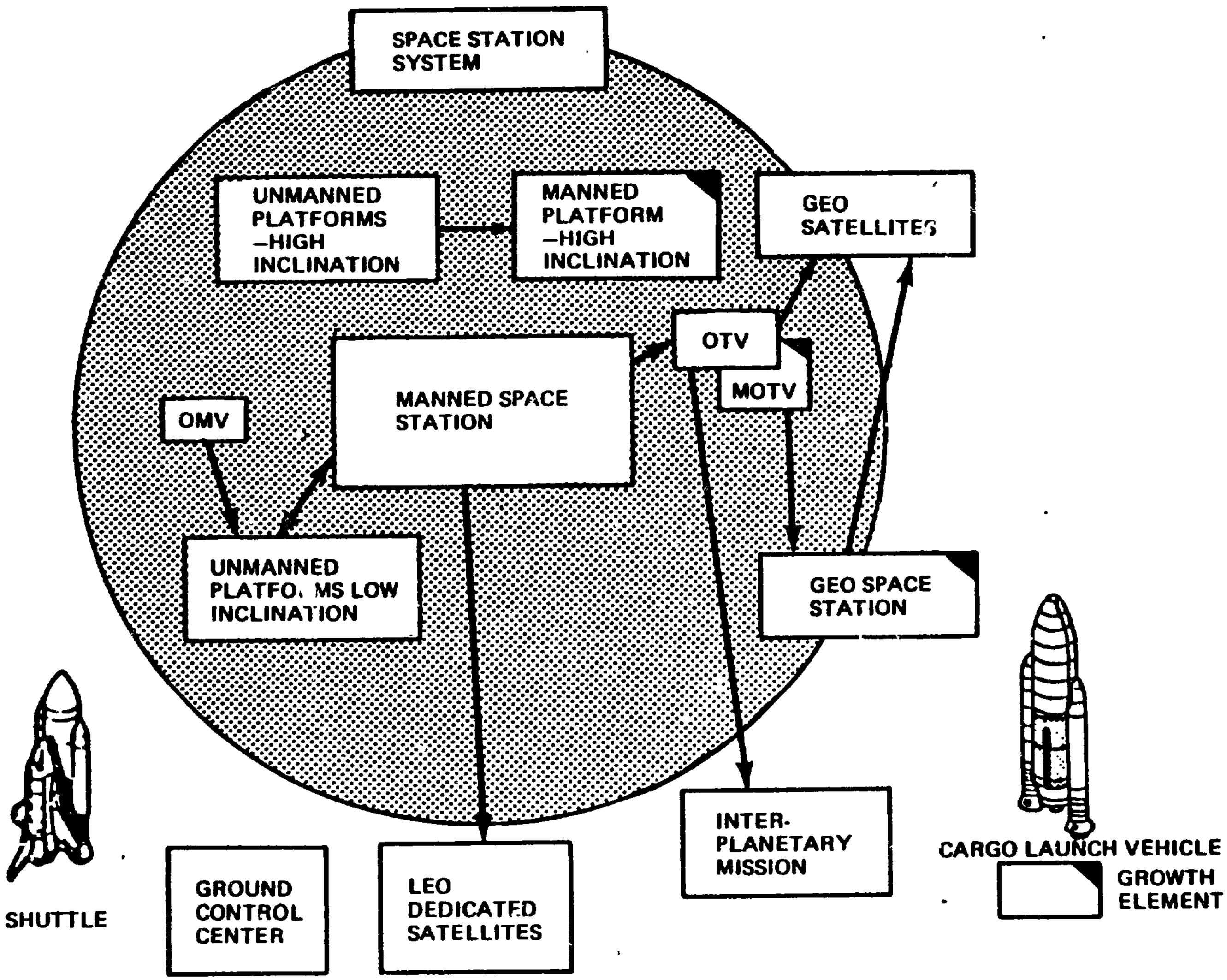

Figure 22. *Space Station System Elements*

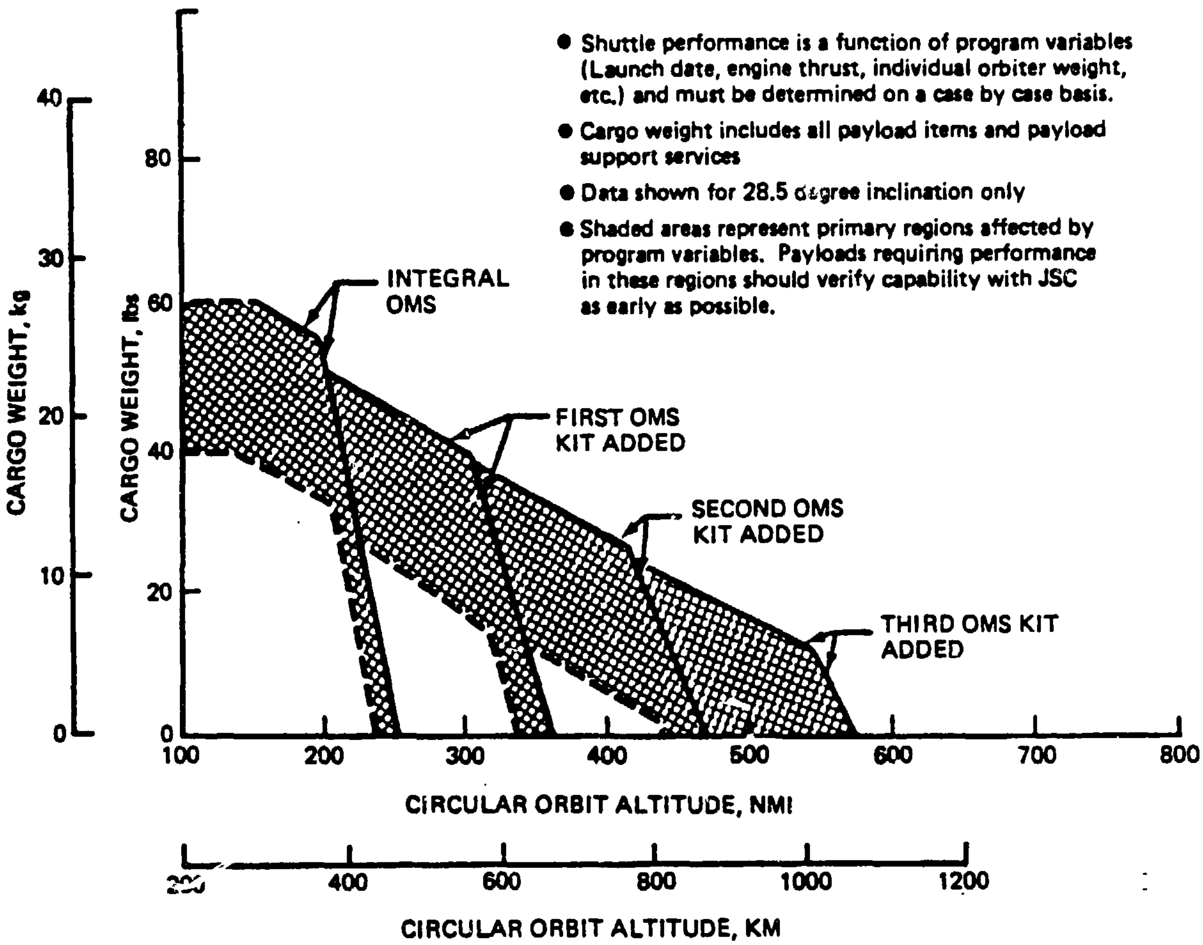

Figure 23 Near-Term STS Payload Mass to Circular Orbit—KSC Launch, Delivery Only

This module may be unpressurized. Power generation equipment, radiators, antennas, and attitude control thrusters will be attached.

The living quarters module primarily provides for those crew support functions that can be classified as "off duty" in nature: private crew quarters, galley/wardroom, health maintenance (exercise) facility, etc. The private crew quarters will provide the vibration and acoustic isolation necessary for restful sleep and private relaxation. Each compartment will contain sleeping provisions, an audio/video tape center, writing desk, personal storage, and lighting and ventilation. The galley/wardroom will contain the food storage and preparation equipment and dining area. This area will also serve as a community room for group R&R and meetings. Environmental control and life support subsystem (ECLSS) equipment will be located in the module to provide humidity, thermal control, and atmospheric revitalization. Sufficient food, water, oxygen and nitrogen air revitalization, waste management, and medical supplies will be stored (internal and external) and provided to meet a 21-day safe haven requirement.

The laboratory module(s) will accommodate scientific, life-science, materials processing, and technology research. It provides the facilities to the user to conduct space research in a zero-g environment including on-orbit accommodation, utilities, and manned presence. The space station provides for the laboratory module the following features: power for the users, dissipation of excess thermal energy, communications within the station and to ground, and the necessary microprocessing capability for computation and data handling of research data. Research units such as racks, instruments, or processing equipment are transported to the space station in the logistics module via the shuttle. Crew members replace research units with the new units, and the original units are returned to ground via the logistics module.

The logistics module is used to resupply the space station with all required consumable items. A logistics module is nominally scheduled to be transported to the space station every ninety days and there exchanged with the previous, depleted logistics module which has been loaded with wastes and parts to be returned to earth.

The servicing structure will provide the facilities for performing satellite servicing, OTV servicing, OMV servicing, satellite assembly, and (perhaps) service as a location for mounting mission payloads. To provide for these operations, the servicing structure would be equipped with hangars, storage platforms, propellant storage tanks, and servicing fixtures. The hangars are unpressurized protective enclosures for storing satellites and vehicles (OMVs and OTVs). The storage platforms would provide a surrogate payload bay for storing the contents of an orbiter delivery. The propellant storage tanks would provide storage for cryogenic, liquid, and gaseous propellants that would be required for vehicles stationed at the space station and for satellites which are assembled and serviced at the station. The servicing fixtures are mechanisms for holding and positioning satellites while they are assembled and serviced.

The manipulator system will be a relocatable element (i.e., either it can be moved from berthing port to berthing port or it will be mounted on a track system). It will be used to offload the orbiter, move elements from place to place on the station and assist in satellite assembly and servicing operations. It will have the capability of being operated remotely from a work station inside the station and from a manned work station on the end of the manipulator.

3.3.2 Space Station Configuration Options

In order to conduct trade studies of reactor-powered space station configurations, it was necessary to select a reference space station concept. At the time of this study, there is a wide variety of station architectural concepts still being considered by the Concept Definition Group (CDG). Figure 24 shows some of these concepts.

The concept selected as the reference space station for this nuclear space station study is of a class of concepts referred to as "gravity gradient concepts", related to the power tower geometry of Figure 24. The gravity gradient stabilized space station concept was selected because it was judged to be the concept most amenable to adding a nuclear reactor system after the initial station starting as a solar powered station. This judgement was based on looking ahead at the potential reactor and radiator configuration, and their integration requirements. This selection did not result from a thorough evaluation of the configuration options and is not to be construed as a recommendation for the space station design; it merely serves the purpose of assigning a common data base for a comparison of the various reactor-space station configurations. As will become apparent in Section 8 below, this selection did not bias the results of the trade studies.

3.3.3 Reference Space Station Design Data

The specific configuration used here for reference space station design data is shown in Figure 25. The parameters were derived from numerous space station studies at Boeing and other aerospace companies. The mass, volume, and frontal area normal to the orbit plane data for this configuration are summarized in Figure 26. The resource module mass has been reduced from that shown in earlier studies by the mass of the solar arrays, gimbals, energy storage, and radiators.

The orbiter is assumed to be docked to the space station for 14 days out of every 90-day cycle. Its mass and frontal area are included in space station global properties as a weighted average. The OTVs are assumed to be docked at the space station for 76% of the time. The total space station properties are shown in Figure 27, for an initial space station (early 1990's) and a growth station (late 1990's). For this study, a growth space station with a crew of eight was assumed, since several studies used this crew size. The power generation subsystem was excluded from these data.

The space station orbit used in this study is a 500 km high circular orbit with an inclination of 28.5°. A nominal atmosphere in this orbit, valid for average drag calculations over a complete 11-year solar cycle, has a density of 3.18×10^{-13} kg/m^3. This results in a drag force for the initial space station of 4.83 mN for the initial space station, and 11.2 mN for the growth station.

3.4 Unmanned Platforms

The space station system will also include several unmanned platforms which serve as mounting bases and resource modules for unmanned payloads. These platforms will be capable of being placed in low earth orbit at any inclination, and will be serviced from the orbiter or from the space station, with the OMV when necessary. Current plans call for one unmanned platform co-orbiting with the manned space station in a 28.5° inclination and another in a high inclination circular orbit with an altitude of 700-750 km.

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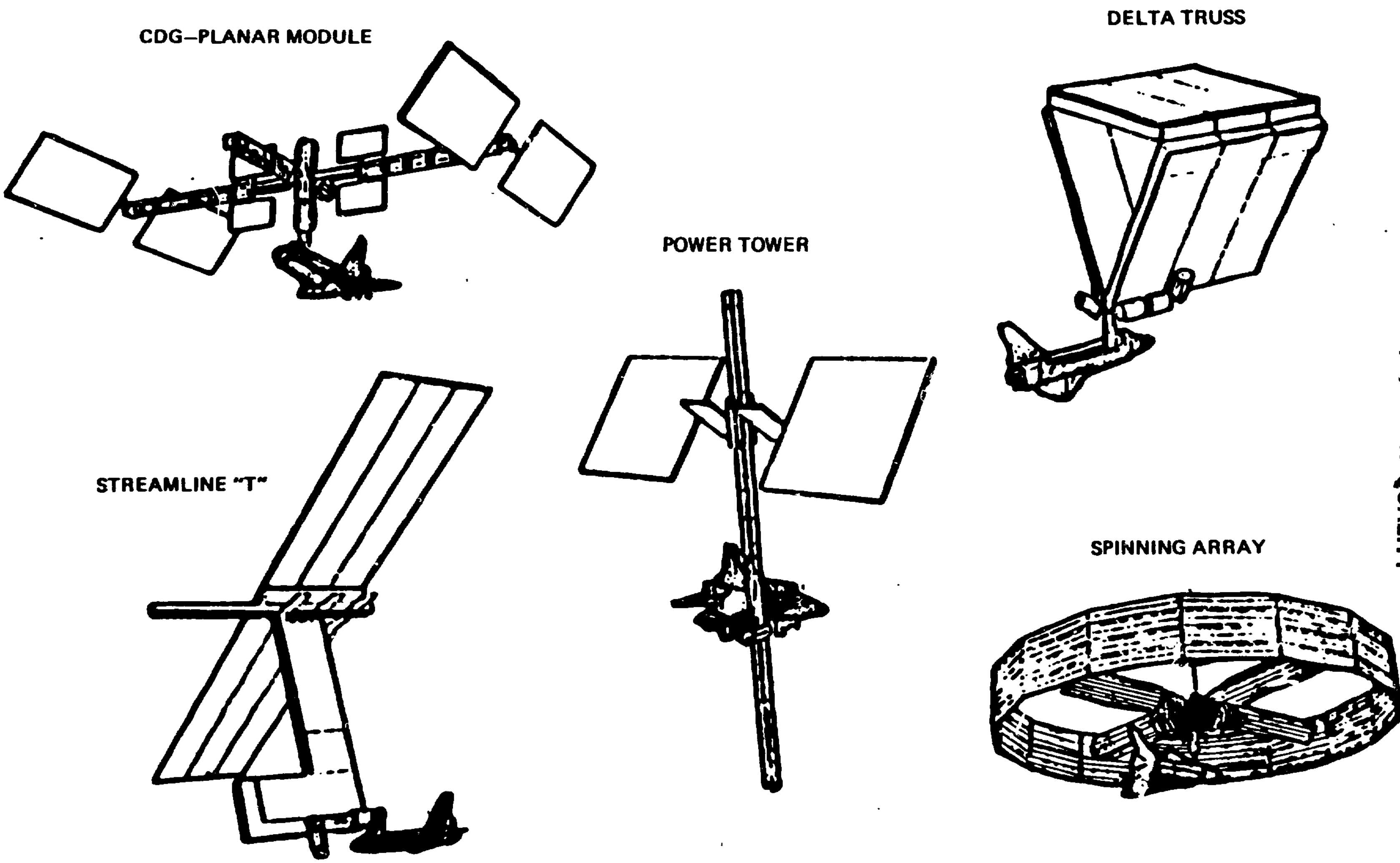

Figure 24 Basic Space Station Geometries

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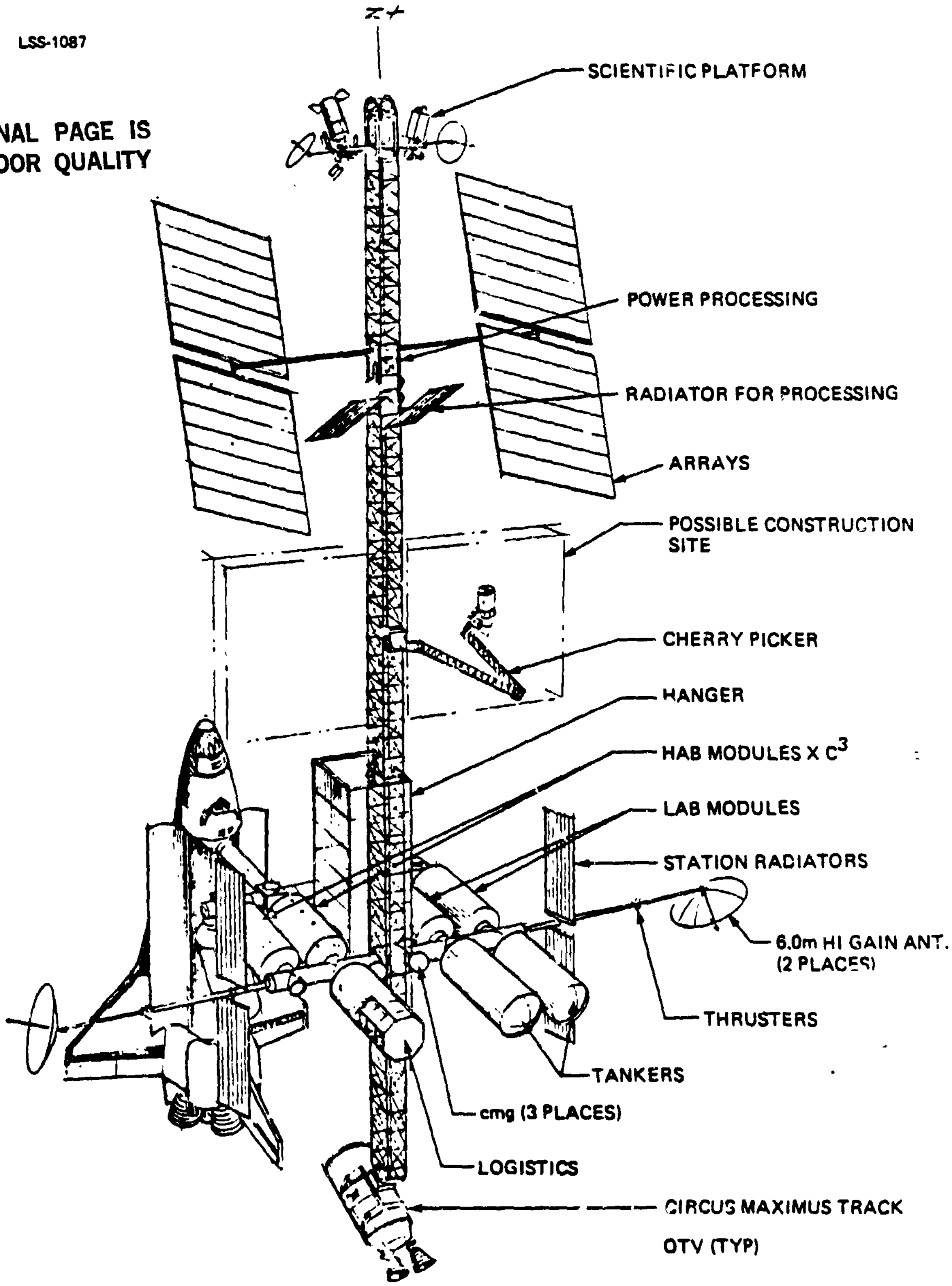

Figure 25 Gravity Gradient Stabilized Space Station Concept

Figure 26 *Reference Space Station Mass and Dimensions*

Module	Mass (kg)	Volume (m^3)	Frontal Area (m^2)	Number of Units IOC	Number of Units Growth
Lab Module	12,200	110	31.7	2	6
Lab Module	7,900	110	31.7	1	2
Resource Module	13,200	110	31.7	1	2
Logistics Module	30,000	110	31.7	1	1
OTV	37,700	190	55.1	0	2
OMV	4,500	20	6.4	1	1
Service Structure	4,500	—	—	1	2
OTV Tank Farm	50,000	—	—	0	1

Figure 27 *Total Space Station Reference Data*

	Mass (kg)	Volume (m^3)	Frontal Area (m^2)
Initial Space Station	93,100	510	210
Growth Space Station	265,450	1610	485

3.5 Orbital Maneuvering Vehicle (OMV)

The OMV is a general-purpose, remotely controlled, free-flying vehicle capable of performing a variety of missions in space. One configuration is illustrated in Figure 28. The initial OMV will be delivered to orbit by the shuttle, controlled from the shuttle, and returned to earth in the shuttle after performing its mission(s). Current NASA planning includes a new start in 1985 for the basic OMV so that the vehicle will be operational in the early 1990s. The plan also includes expanding the capability of the vehicle to remain in orbit for an extended period of time by adding a solar array power source and by enabling it to be refueled in orbit. With these capabilities, it can be left in orbit and need not be returned in the shuttle. OMV capabilities to deliver and retrieve payloads in higher altitude coplanar orbits are shown in Figure 29. For this study, it is important to note that the OMV should be capable of placing an 18,000 kg payload in a 1200 km orbit, when deployed from the space station. or to place and return a 7000 kg payload to/from the same orbit.

3.6 Orbital Transfer Vehicle (OTV)

The orbital transfer vehicle considered in this system analysis will be a cryogenic reusable upper stage vehicle that will be stationed at the space station starting in the mid-1990's. There are a variety of OTV concepts being considered, so it is too early to define its characteristics. The studies by Boeing and Martin Marietta, initiated in July, 1984, will result in conceptual designs and predicted performance. Its primary mission will be to launch satellites from the space station to geosynchronous orbit. One potential configuration is illustrated in Figure 30 (ref. 10). This design would be capable of transferring a round trip payload of 5,000 kg between the space station and geosynchronous orbit.

A space-based OTV might also be used to transfer payloads to low earth orbits which are out of plane with the space station and require large Δv from the station. Another application might be to place massive payloads into low Δv orbits, such as to transfer heavy fuel tank payloads between the space station and other low earth orbits. This mission will be discussed in Section 8.8 as a means of placing a reactor in a somewhat higher orbit than the space station.

3.7 Advanced Space Station System Elements

At the present time, the Space Station program concept projects, by growth, beyond the year 2000. Boeing, and others, have looked at advanced space station system concepts that go beyond the first station. These advanced concepts include the following: geosynchronous space station, polar orbit space station, manned OMV and OTV. and heavy lift launch vehicles. Each of these is described below.

3.7.1 Geosynchronous Orbit Space Station

A space station in geosynchronous orbit (GEO) may become an economic venture in the future. The primary mission for a GEO space station would be communications satellites and platform servicing. The primary operational mode would be satellite servicing by a manned servicing vehicle that would be permanently stationed at the GEO space station. A secondary mode would be retrieving smaller satellites to the space station for servicing and then returning them to their orbital slot. A secondary mission would be life sciences research (primarily medical research looking at the long-term exposure of the crew to the GEO radiation environment).

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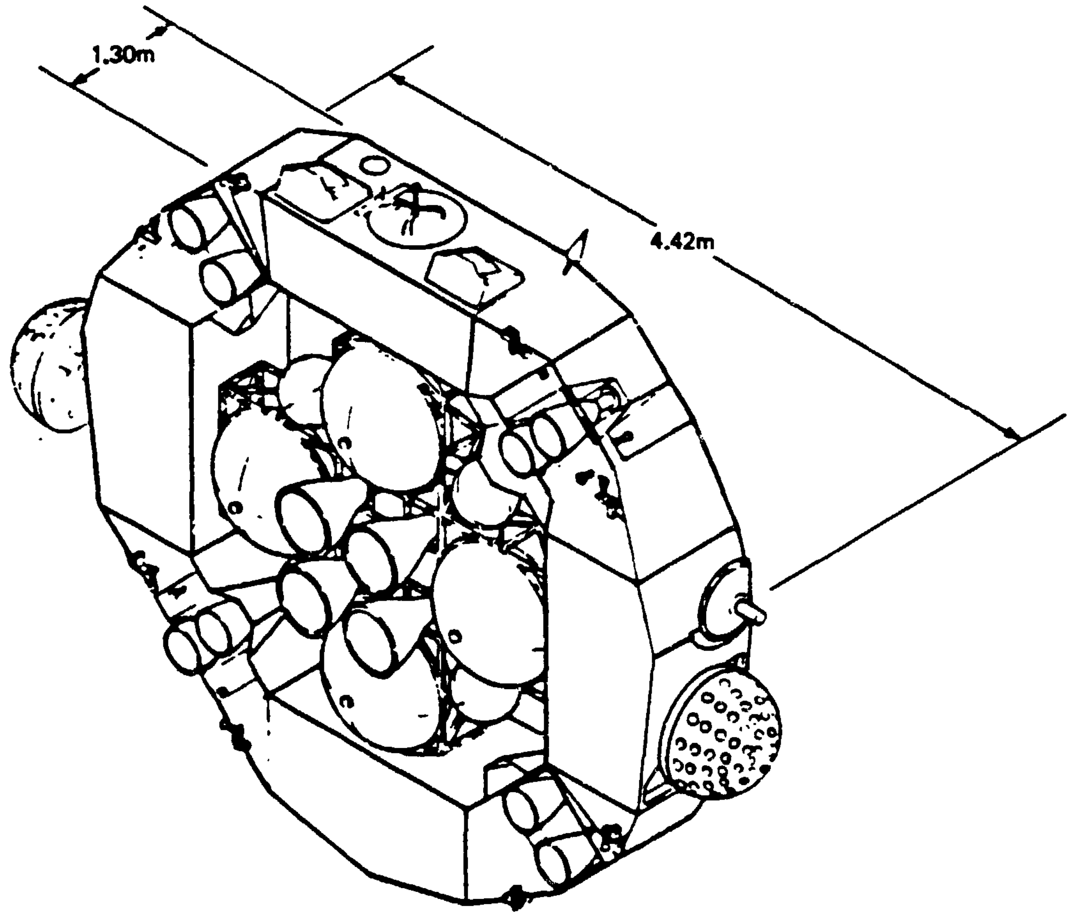

Figure 28 Orbital Maneuvering Vehicle (OMV)

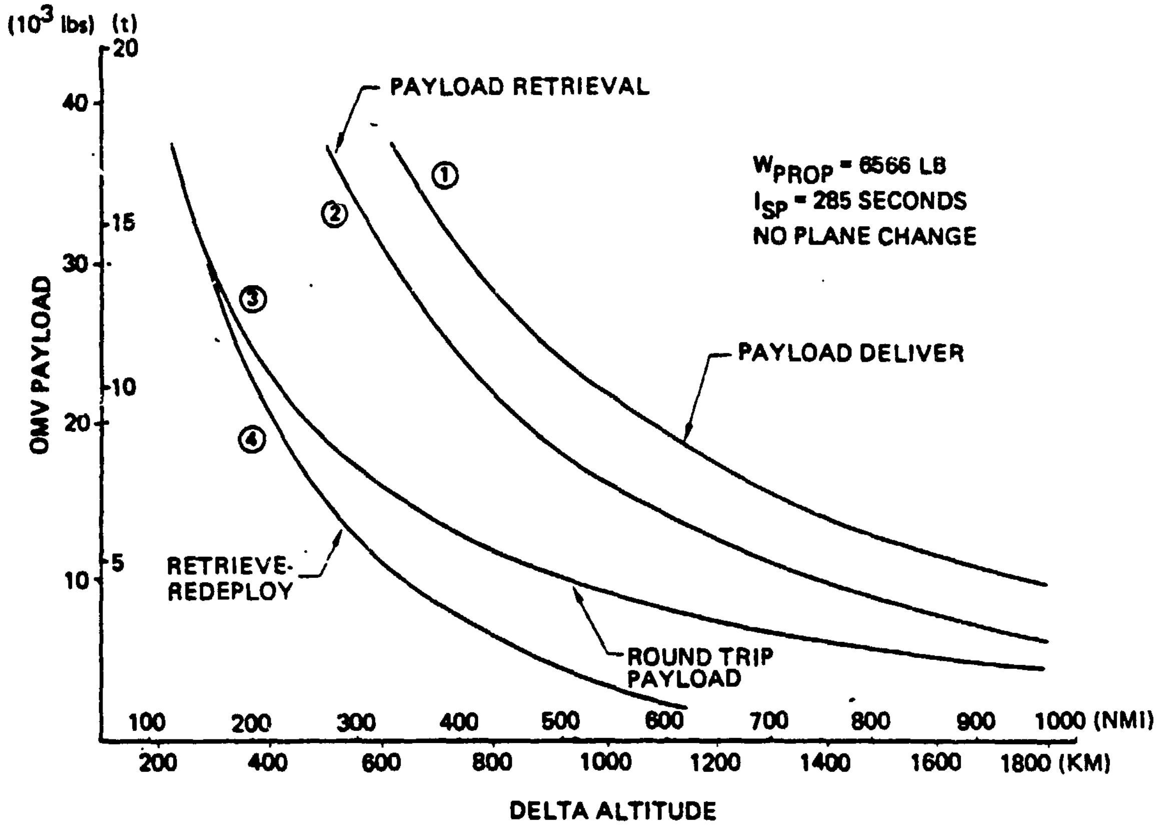

Figure 29 *OMV Performance*

A GEO space station could also be used as a platform for civilian and military earth and space observation instruments that would be enhanced by the presence of a crew. Figure 31 shows one GEO space station that was developed for an earlier study (ref. 11).

3.7.2 Polar Orbit Space Station

In the current NASA planning, an unmanned polar orbit platform has been identified as being required to fulfill earth-observation and space physics mission requirements. In Boeing's space station needs studies (ref 3) was shown an economic benefit of having a manned space station principally because the manned presence would provide the capability of keeping the instruments serviced on a daily basis. Figure 32 shows a concept for a polar space station that was defined in this study. The concept would require approximately 20 kWe of electrical power to serve the missions and housekeeping requirements.

3.7.3 Manned Orbital Maneuvering Vehicle

In the current NASA Space Station planning, an unmanned OMV is defined. However, Boeing (and others) has identified requirements for a manned OMV to enhance the satellite servicing capabilities. This manned OMV would be stationed at the GEO space station. Figure 33 shows a concept for this manned OMV. This concept is sized for a crew of two.

3.7.4 Manned Orbital Transfer Vehicle

In the current NASA space station planning, an unmanned OTV is defined. However, Boeing (and others) has identified requirements for a manned OTV to enhance the capability for servicing satellites located at GEO. This manned OTV would also be stationed at the GEO space station. Figure 34 shows a concept for this manned OTV. The crew cabin would be the same as that used for the manned OMV.

3.7.5 Heavy Lift Launch Vehicle

NASA and the military have sponsored studies of future requirements for unmanned HLLV's. Figure 35 shows a potential development schedule for some of the HLLV concepts that have been defined by Boeing. The first one in particular, referred to below as a cargo launch vehicle (CLV), would utilize the existing STS solid rocket boosters on an unmanned launch vehicle to increase the payload to orbit capabilities by a factor of three. This unmanned, shuttle-derived vehicle would remove the orbiter and modify the existing external tank (ET) by placing the payload in the nose of the ET.

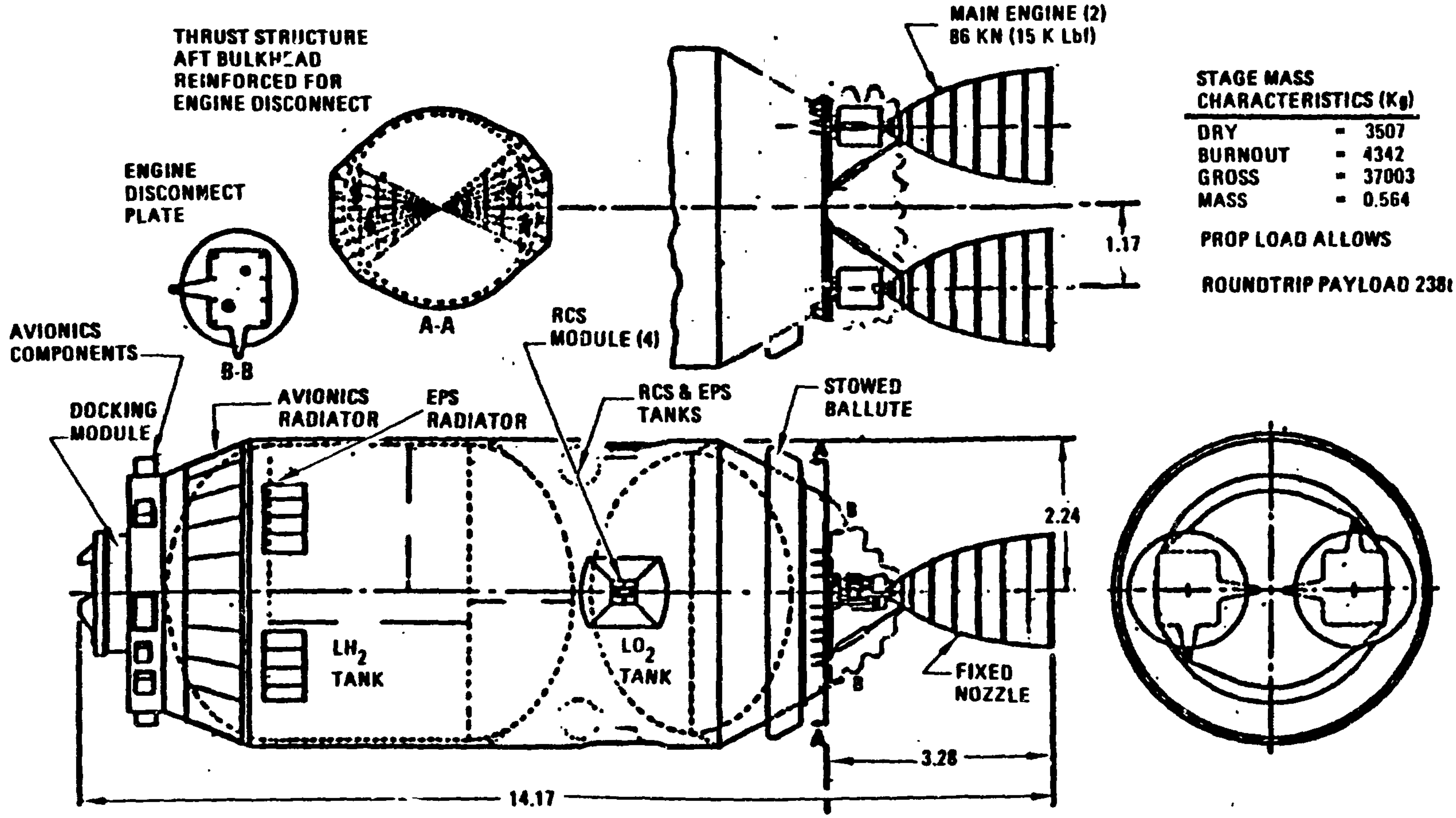

Figure 30 Orbital Transfer Vehicle

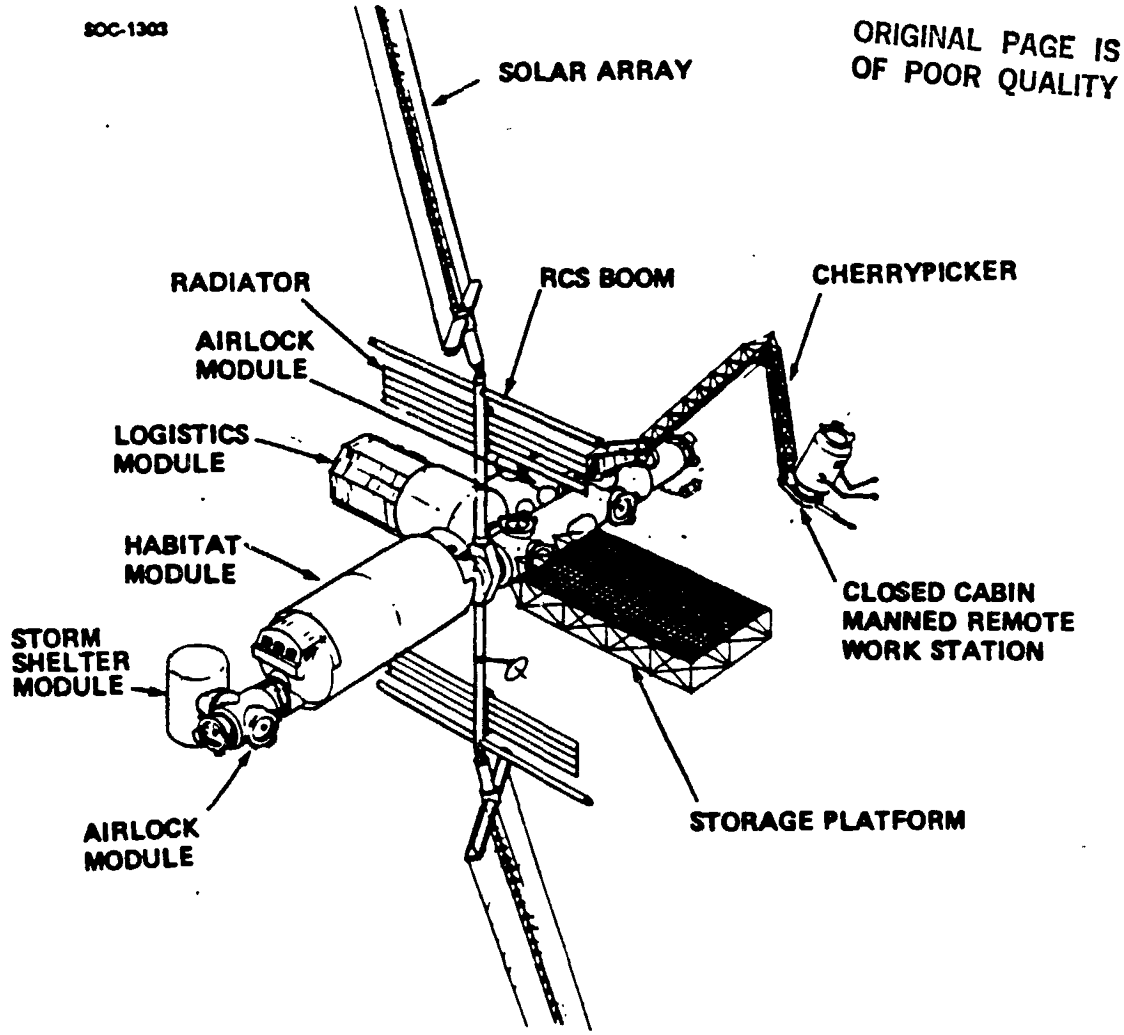

Figure 31 GEO SOC Concept No. 2

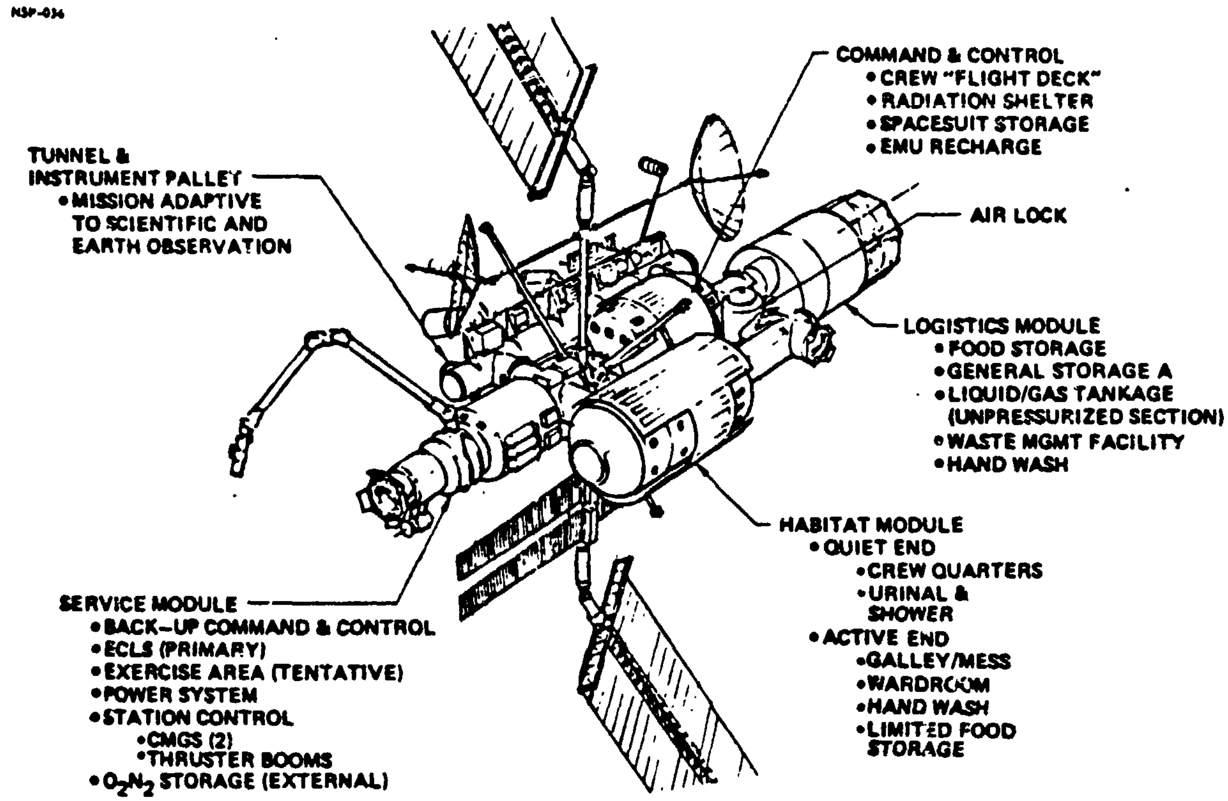

Figure 32 Polar Space Station Concept

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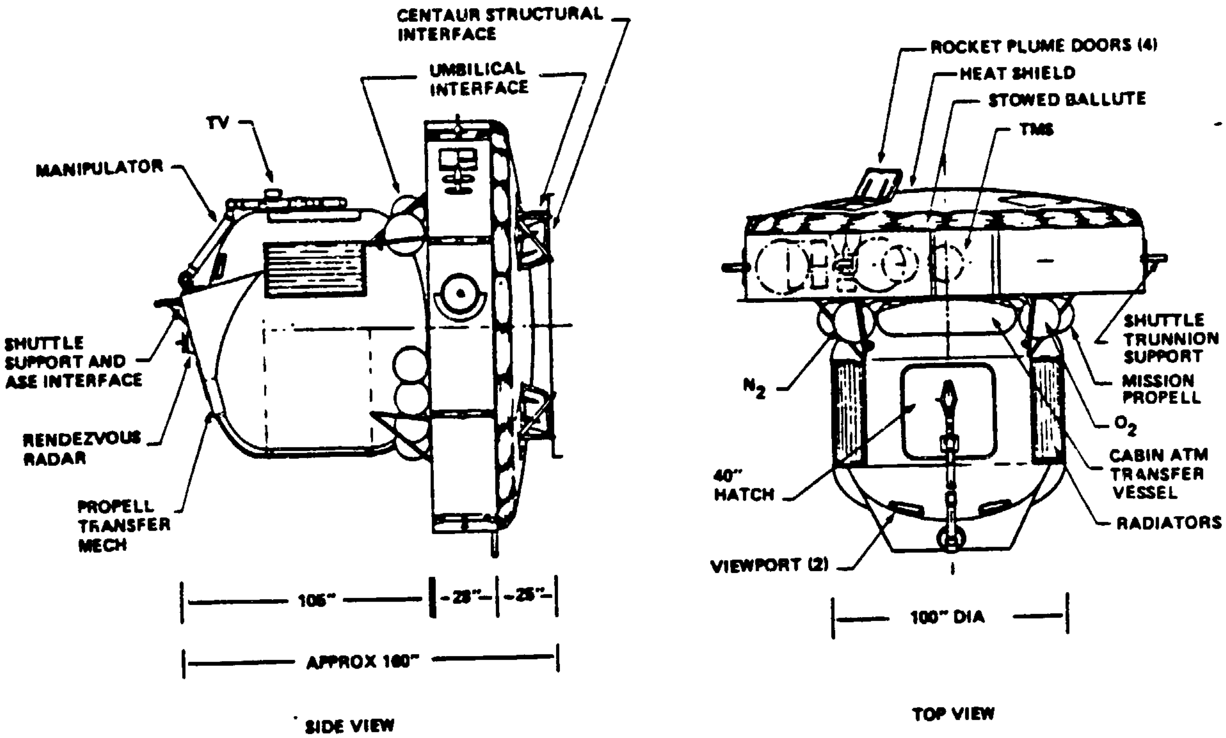

Figure 33 *Manned OMV Concept*

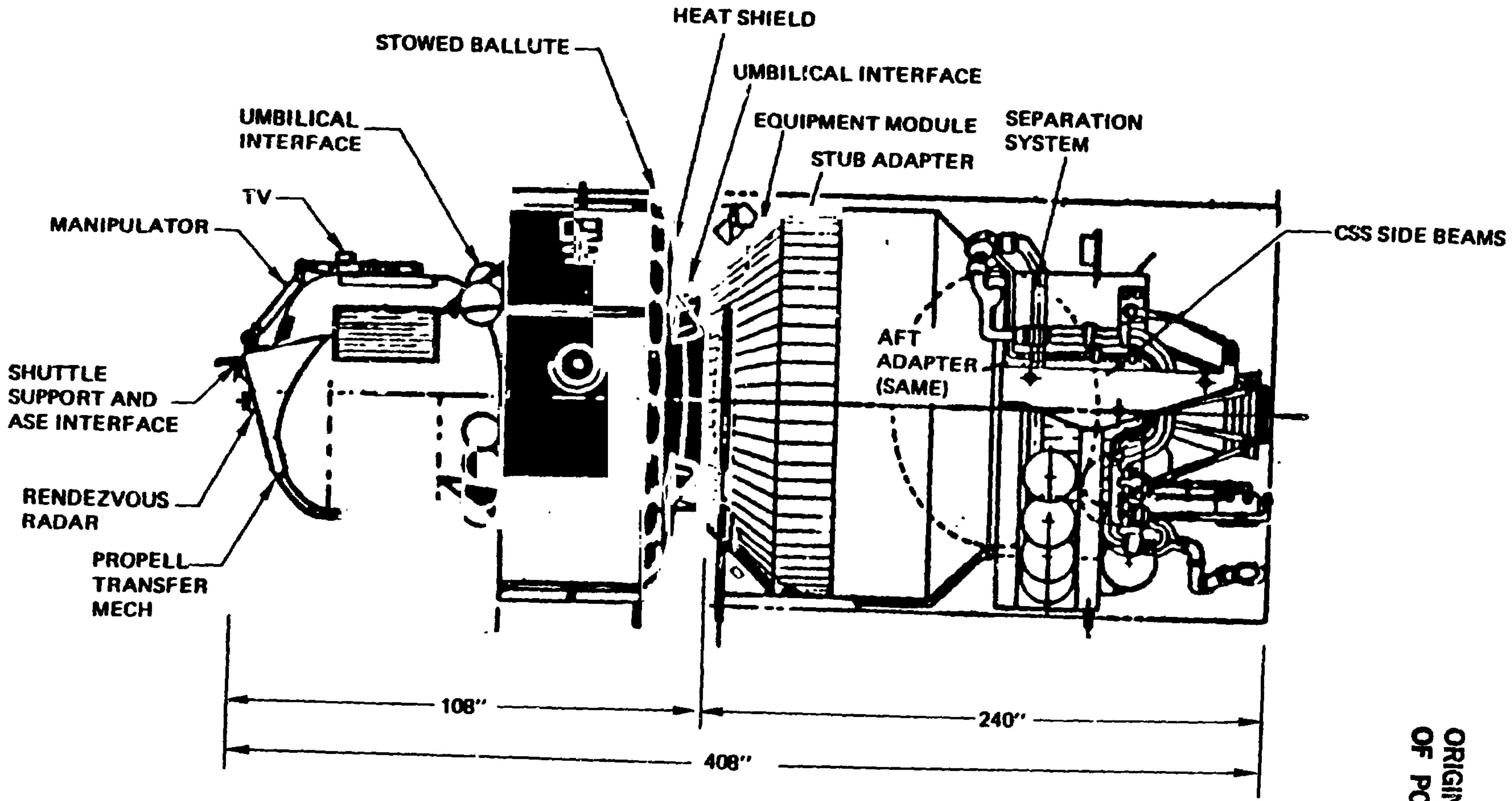

Figure 34 *Manned Orbital Transfer Vehicle*

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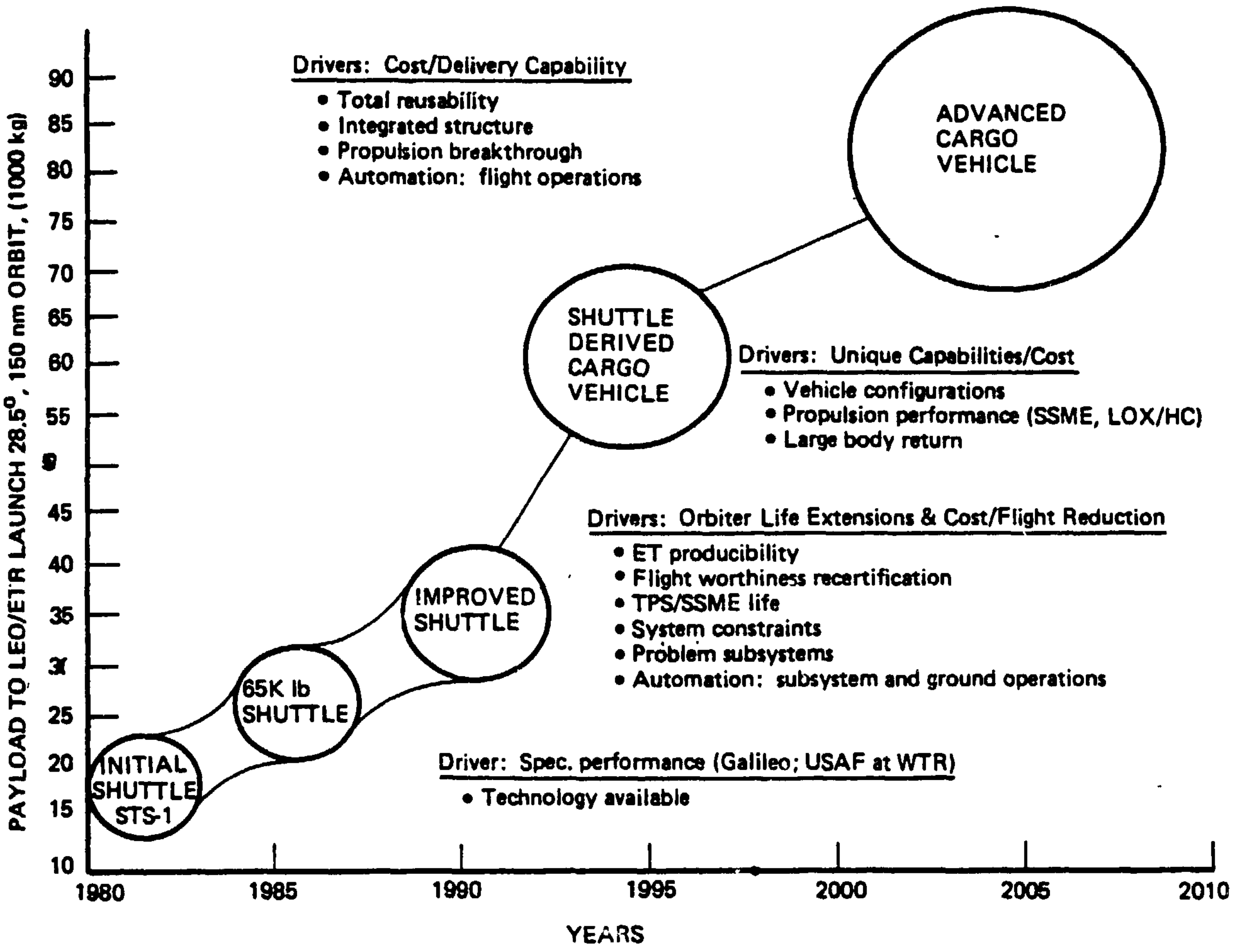

Figure 35 *Advanced Launch Vehicle Projections (from NASA Space Systems Technology Model Volume IIA, January 1984)*

4.0 SAFETY REQUIREMENTS

There is a large body of safety requirements that regulate terrestrial nuclear power generation and spaceflight separately, but little that relates directly to the operation of nuclear reactors in space. This section reviews existing regulations that may be used as a guide to establishing safety requirements for nuclear reactors as a source of electrical power for a manned space station. Radiation guidelines and the natural radiation environment are discussed. Other safety requirements that are not radiation-related are also discussed. The present designs of the SP-100 reactors are not addressing manned applications. Hence, the regulations related to SP-100 will not be applicable for a manned space station. Safety documentation for the space station application will have to be developed. In manned application, safety for the human is paramount.

4.1 Summary of Applicable Documents

The documents listed in Figure 36 have been reviewed to determine their applicability to the present study of a nuclear reactor for a manned space station.

FIGURE 36: NUCLEAR SAFETY REQUIREMENTS DOCUMENTS

OSNP-1, (SP-100, currently being revised) Nuclear Safety Criteria and Specifications for Space Nuclear Reactors

NHB1700.7A, Safety Policy and Requirements for Payloads Using the Space Transportation System (STS)

JSC11123, Space Transportation System Payload Safety Guidelines Handbook

10CFR20, Standards for Protection Against Radiation

10CFR50, Appendix A, General Design Criteria for Nuclear Powerplants

DOE Order 5480.1A, Environmental Protection, Safety, and Health Protection Programs for DOE Operations

OSNP-1, Part B in particular defines nuclear safety specifications for the SP-100 program. This was written primarily for unmanned reactor systems. It is currently undergoing a major revision, with the revised criteria to be approved soon. NHB1700.7A describes safety policy and requirements that apply to all payloads which utilize the shuttle system. JSC11123 is a handbook that describes potential hazards based on accumulated spaceflight experience and provides design guidelines to facilitate compliance with STS policy as stated in NHB1700.7A. Neither of the STS documents are restricted to nuclear safety alone.

10CFR20 and 10CFR50, Appendix A are federal regulations. The former applies to any radioactive source and sets limits on the allowable concentration of each of the radionuclides in the environment. This would be particularly relevant in case of a launch or reentry accident in which radioactive material might be released in the atmosphere or in the ocean. The latter regulates the design of nuclear powerplants in general and does not particularly apply to space reactors.

DOE Order 5480.1A provides for protection of the public and government property against accidental loss and damage due to DOE operations. This is particularly relevant for the SP-100 program, in which the space reactor will be treated as a DOE operation. The order encompasses environmental, safety, and health protection.

4.2 OSNP-1

OSNP-1 is divided into two parts: Part A describes nuclear safety criteria for space nuclear reactors and Part B describes design specifications to ensure compliance with the policy stated in Part A. Part A describes radiological safety policy:

> "The policy of the United States for all U.S. nuclear power sources in space is to ensure that the probability of release of radioactive material and the amounts released are such that an undue risk is not presented, considering the benefits of the mission."

This policy is implemented by defining a dose limitation philosophy and by establishing safety design reporting procedures. Part A specifies that a Preliminary, Updated, and Final Safety Analysis Report be prepared at designated stages of reactor design and submitted to DOE for approval and that the reports be reviewed by the Interagency Nuclear Safety Review Panel.

The dose limitation philosophy established in OSNP-1, Part A follows these three principles:

- o No practice shall be adopted unless its introduction produces a positive net benefit;
- o All exposures should be kept as low as reasonably achievable, economic, technical, and social factors being taken into account;
- o The risk to individuals shall not exceed the limits specified for the appropriate circumstances.

Safety criteria are also established in Part A for radiation protection in case of accidents. The following criteria are to be used to assess the consequences of a hypothetical accident:

- o For individuals, a specific analysis shall be prepared to demonstrate that the accidental dose is as low as reasonably achievable.
- o For populations, a specific analysis shall be prepared to demonstrate that the risk index is as low as reasonably achievable.

The risk index is defined as the sum of the individual accident probabilities times the associated integrated population dose.

Nuclear safety requirements for a manned space station system must address radiation protection of the general population while the reactor is in the atmosphere, and radiation protection of space station personnel while the reactor is in orbit. The safety impacts on system design affect the launch vehicle, reactor, and space station during the different mission phases. The mission phases can be separated into ground handling, launch and ascent, reentry, nominal mission, mission emergency, and end of life.

Reactor design requirements for the SP-100 reactor program are specified in OSNP-1, Part B. The following safety design requirements are specified:

- The reactor shall be designed to remain subcritical if immersed in water or other fluids (such as rocket propellants) to which it may be exposed.
- The reactor shall have a significantly effective negative power coefficient of reactivity.
- The reactor shall be designed so that no credible launch pad accident, range safety destruct actions, ascent abort or reentry from space resulting in Earth impact could result in a critical or supercritical geometry.
- The reactor shall not be operated (except for zero power testing yielding negligible radioactivity at the time of launch) until a stable orbit or flight path is achieved and must have a reboost capability from low-earth orbit if it is operated in that orbit.
- Two independent systems shall be provided to reduce reactivity to a subcritical state. They shall not be subject to common cause failure.
- The reactor shall be designed to ensure that there is an independent shutdown heat removal system or independent heat removal paths within the heat transport system to provide decay heat removal.
- The unirradiated fuel shall pose no significant environmental hazard.

These design requirements impose certain functional requirements on the reactor system, including:

Reactor Control System - Positive coded telemetry shall be required to start up the reactor. Thus, the reactor shall have a reactor control system which, in addition to the design requirement of being capable of initiating reactor start-up, shall be capable of controlling power escalation to full power level and of reducing power to a full shutdown mode. The system shall be capable of being operated in a directed positive shutdown mode and of restarting the reactor following a shutdown.

Reactor Protection System - A reactor protection system which includes two independent systems not subject to common cause failure to reduce reactivity to a subcritical state shall be provided. This system, which is one of the engineered safety features, shall be capable of sensing conditions which would call for reactor shutdown and of automatically shutting the reactor down, with restart capability. Except for the neutron absorber/reflector elements and their activators, the protection system shall be independent from the control system. Conditions calling for automatic reactor shutdown by the protection system are:

- Failure of the reactor control system
- Failure of the spacecraft attitude control system
- Exceeding of fuel design temperature limits
- Failure of Earth/spacecraft reactor control and/or safety systems communications system.

The reactor protection system shall have fault detection sensors and shall be designed to be able to be tested while the reactor is operating (without actually shutting the reactor down). It shall be capable of performing this function, assuming a single system failure, and shall not be subject to common cause failures with systems/conditions upon which it is called to activate in case of their failure. The reactor protection system shall be designed such that any failure of the system puts the reactor in a safe condition in a reasonably short time. The reactor systems contractor shall specify in the detailed technical safety specification (subject to DOE approval), the minimum shutdown reactivity during:

- o assembly/testing
- o ground handling/storage
- o transportation
- o launch

Reentry Core Dispersal Capability - For short-lived orbit missions, a reentry core dispersal capability shall be provided. This capability will ensure that the reactor core and activated structural elements will be dispersed such that the Nuclear Safety Criteria for Space Nuclear Fission Reactors (Part A) are not exceeded. This capability shall be provided such that the reentry environment of the reactor separate from the launch vehicle will initiate and cause the required dispersal with no active systems required to operate.

Reactor Control and Safety Systems Communication System - A reactor control and safety systems communication system shall be provided such that reactor, reactor control system, power conversion system, and safety systems status may be monitored and controlled. Two independent systems shall be provided.

Instrumentation System - An instrumentation system shall be provided which provides, through the reactor control and safety systems communication system, signals to allow continuous determination of:

- o Reactor power level (and rate of change)
- o Fuel temperature
- o Control/reflector element positions

Status of:

- o Reactor control system
- o Reactor protection system
- o Power conversion system
- o Spacecraft attitude control system
- o Orbital altitude boost system (short-lived orbit missions only)
- o Independent electrical power source

Core Cooling System - Following a shutdown the core cooling system shall be capable of providing adequate heat removal from the core with sufficiently independent shutdown heat removal paths or with an independent shutdown heat removal system to prevent the fuel temperature limit from being exceeded and safety systems from being inoperable.

Functional requirements can be derived from OSNP-1 and imposed on the space station system. These include:

Orbital Altitude Boost System - An orbital altitude boost system shall be provided by the mission agency (short-lived orbit missions only). This system shall be capable of boosting the reactor to a high altitude orbit following completion of the mission or upon mission failure such that the radiation protection safety criteria of Section 5.2 of Part A (of OSNP-1) are met. Typically, an orbital lifetime of at least 300 years will be required.

Spacecraft Attitude Control System - A spacecraft attitude control system shall be provided on the spacecraft by the spacecraft system contractor such that spacecraft attitude can be controlled to permit communication with the reactor control and safety related systems and (for short-lived missions) to permit the orbital altitude boost system to perform its function.

Independent Electrical Power - An independent source of electrical power shall be provided such that the reactor control system, the reactor protection system, and the reactor control and safety systems can operate independent of reactor operating mode or reactor power conversion system operation. This secondary source of power shall be capable of providing power for safety related systems operation and reactor restart capability for a minimum of 24 hours following failure of the reactor power conversion system during the spacecraft mission lifetime.

4.3 Long-Lived Orbits

Functional requirements stated in OSNP-1, Part B, included a requirement for an orbital boost system that would place the reactor in a high orbit at end of life or upon mission failure, assuming it was operated in a short-lived orbit. An orbit lifetime of 300 years is suggested. Figure 37 shows the required orbit altitude for a 300 year orbit as a function of reactor power system ballistic coefficient. Atmospheric conditions were taken in a 28° inclination orbit, averaging the 11-year solar activity cycle from 1994 until 2294. The ballistic coefficient relates to the mass-to-drag ratio of the reactor power system. The range of ballistic coefficients considered in this report is from 16 gm/cm^2 for a free-flyer reactor with its associated radiators and fuel processing equipment, to 150 gm/cm^2 for a man-rated 150 kWe reactor, shield, and power conversion system alone, without radiator. Thus, a free flyer with $m/C_DA=16$ gm/cm^2 must operate in a 780 km orbit if it is to have a 300 year orbital lifetime, while a reactor with $m/C_DA=150$ gm/cm^2 must be boosted to a 600 km circular orbit if the radiator is first removed.

4.4 Radiation Protection

The radiation protection guides for space flight as established by NASA with the advice of the Radiological Advisory Panel, Space Science Board, National Academy of Sciences were proposed (ref. 12) with the following conditions:

> "They are proposed on the assumptions that (a) they are to be used only for current space-mission and vehicle-design studies; (b) space missions of the next 10 to 20 years will be high-risk operations, and the radiation hazard should be considered realistically and in perspective with other inherent risks; (c) they will be subject to review and revision as additional pertinent information becomes available and before application to actual operations; (d) an active career in earth-orbital operations can be terminated at the end of any specific mission; (e) the number of people involved will be small and most will be in the older-than-30 age group; (f) participants will be highly

motivated volunteers well informed about the nature and extent of the radiation risk; and (g) the agencies concerned appreciate the desirability of keeping exposure as low as practicable by appropriate engineering and operational considerations."

The astronaut radiation exposure limits given in Figure 38 (ref. 13) are based on a five year career. These limits may be lowered by the mid-to-late 1990's. Protons and electrons trapped in the VanAllen belts produce strong radiation fields in the near-earth environment.

Radiation dose rate in the Van Allen belts varies with the position in the belts which are not symmetrical (ref. 14). The South Atlantic Anomaly, extending from 0 to 60 degrees west longitude and 20 to 50 degrees south latitude, has trapped protons of greater than 30 MeV intensities. These trapped protons are at a concentration between 100 to 200 miles altitude which is equivalent to those found at 800 miles altitude elsewhere.

The space station orbit is designated in Figure 40 at 28.5° inclination and 270 nautical miles. An unshielded astronaut would receive about 2 rads/day. Over 90% of this dose is received during the 5% of the time in which the astronauts pass through the South Atlantic Anomaly. Aluminum shielding of 3.3 mm thickness reduces this dose by a factor of 100, and each additional millimeter of aluminum reduces the dose by another factor of 2.5. Thus, an aluminum shield of 5.0 mm thickness would reduce the 90-day dose from 200 rads to 0.4 rads.

4.5 Payload Safety Guidelines Handbook

This handbook (ref. 13) has been prepared to assist STS payload developers to achieve compliance with payload safety policy and requirements as defined in NHB1700.7A (ref. 15). It contains a summary of hazards of which the payload developer should be cognizant and suggested guidelines to be considered in the design and operation of STS payloads to eliminate or control these hazards. The hazards summary and guidelines are grouped into fifteen generic subsystems. These guidelines incorporate system safety experience accumulated by NASA, military, and aerospace industry sources or manned and unmanned spacecraft and aircraft.

Although practically all the subsystems relate to the current study, the subsystems of particular importance are radiation, electrical, and cryogenics. A brief summary of the guidelines that should be considered specifically for nuclear-powered, manned space stations follows. For more detailed guidelines, the reader is referred to the handbook.

4.5.1 Radiation

Liquid metal heat transfer loops should be designed for safe handling and freedom from leakage. Liquid metal coolants can be avoided by using a Brayton cycle or organic Rankine cycle when feasible. If liquid metals are used, then double-walled containment must be provided throughout the coolant loop, and leak detection must be provided during all operational phases of the mission. The liquid metal coolant loops should not require breaking or opening during orbital operations because of the high temperatures involved.

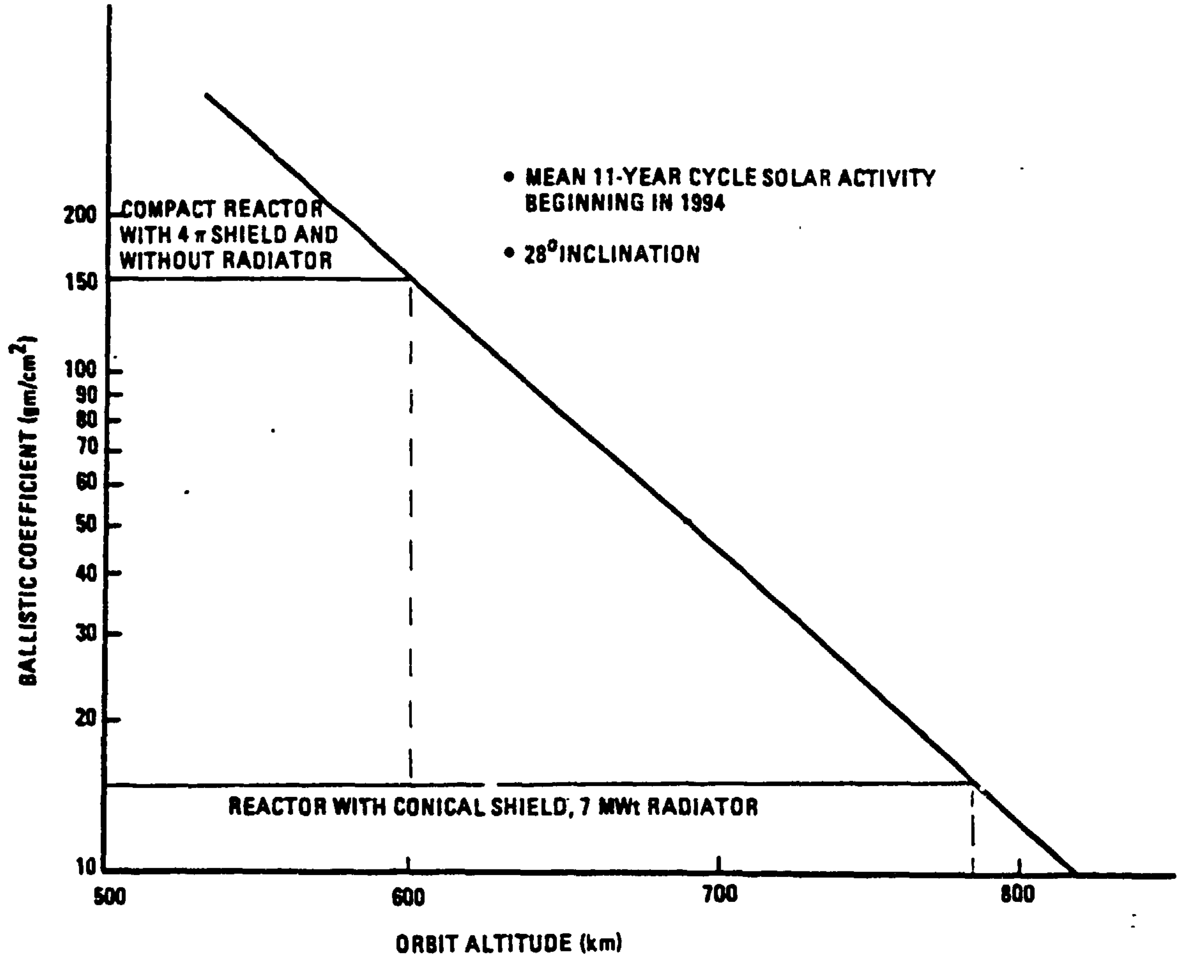

Figure 37 *300-Year Orbit Altitude*

TOTAL ALLOWABLE RADIATION LIMITS FOR THE CREW FROM ALL SOURCES
DESIGN GOALS LIKELY TO BE MUCH LOWER

CONSTRAINTS	REM*		
	BONE MARROW (5 CM)	SKIN (0.01 MM)	EYE (3 CM)
1-YEAR AVERAGE DAILY RATE	0.2	0.6	0.3
30-DAY MAXIMUM	25	75	37
QUARTERLY MAXIMUM	35	105	52
YEARLY MAXIMUM	75	225	112
CAREER	400	1200	600

*from JSC 11123, established by the Radiation Safety Panel for Manned Spacecraft

Figure 38 *Radiation Exposure Limits for Humans*

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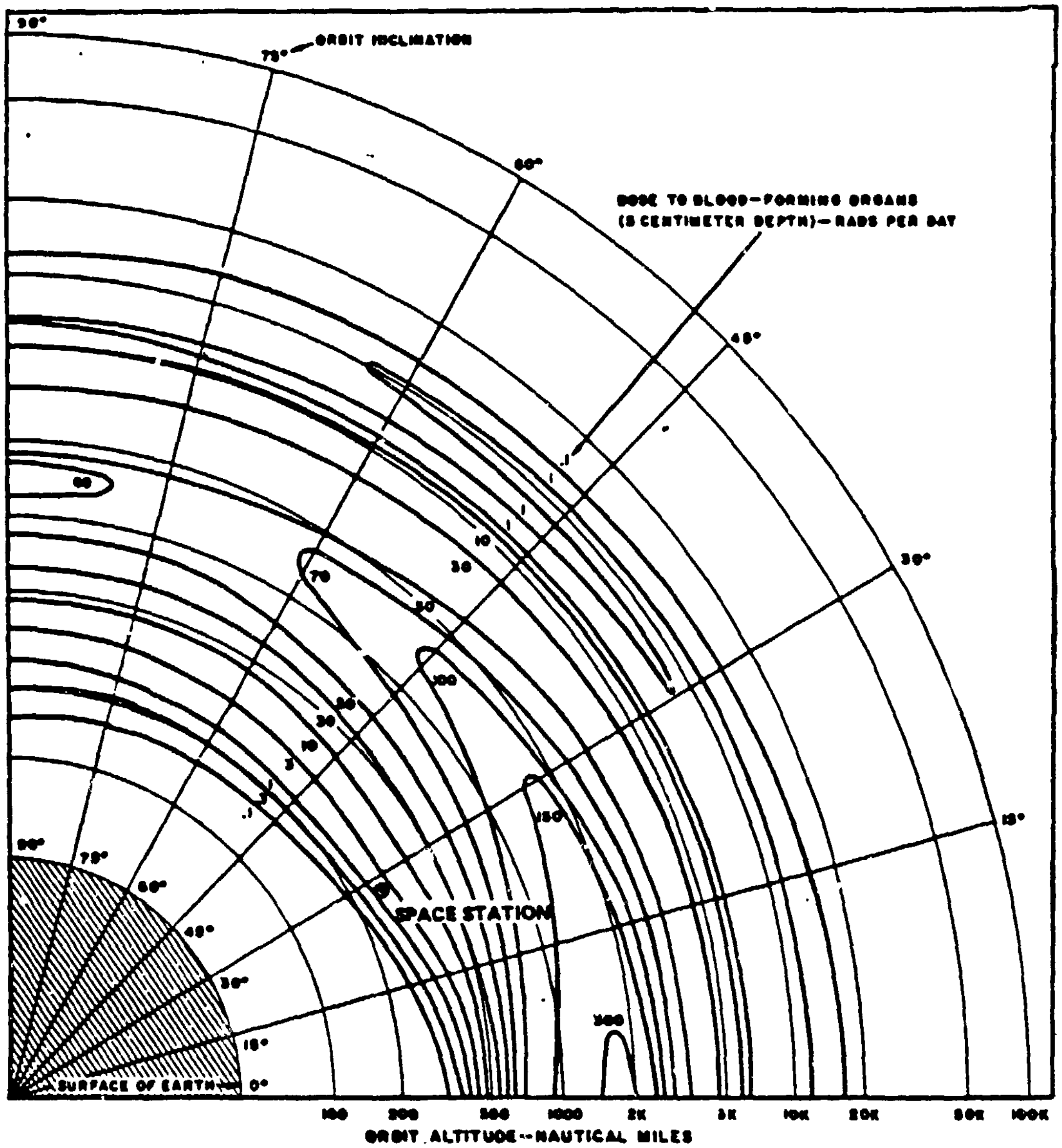

Figure 40 *Daily Dose to Blood Forming Organs*

Adequate blast overpressure, fragmentation, and fireball protection should be provided to assure containment of all radioactive material if an accident occurs.

Reactor systems should be instrumented to allow ground and flight crews to monitor radiation levels of critical elements and temperature and pressure of the primary heat transfer loop to detect leakage and thermal performance.

A redundant lockout circuit should be provided to prevent inadvertent activation of a nuclear system.

A redundant automatic means of reactor shutdown should be provided to control operation under all contingencies.

Equipment should be provided for locating radioactive material which has been inadvertently released in a manned area or module.

Tracking and recovery-locating devices should be provided on nuclear payloads to facilitate land or water recovery if the payload is jettisoned.

A reactor jettison system capability should be provided with all nuclear payloads to protect the crew and vehicle. Safe procedures should also be established for the disposal of radioactive waste or radiation-contaminated material.

A positive and permanent shutdown system should be provided for malfunctioning reactors and for reactors which have completed their missions.

Design of nuclear hardware should include intact reentry and impact to protect the general public from potentially dangerous radiation. This guideline appears to be inconsistent with the OSNP-1 requirement 3.4.3, which states that, for short-lived orbit missions, a reentry core dispersal capability shall be provided.

To prevent possible radiation leakage danger to ground and flight crews, nuclear reactor power modules should be designed for transportation to space while in a preoperational mode. The reactor should not be activated while in immediate proximity to the orbiter. This latter guideline might be reconsidered if the reactor operates at the space station, since frequent orbiter traffic is likely.

To protect the flight crew, on-orbit nuclear cargo transfer operations should not require extravehicular activities. This implies that all activities involved with reactor transfer from the orbiter to its operational location and condition on the space station must be performed with remote manipulators. It is not clear if a preoperational reactor should be considered nuclear cargo in this context.

4.5.2 Electrical

Guidelines in the electrical subsystems section of the Handbook describe hazards which could result in electrical shock or burns. They also cover indirect hazards such as battery explosions caused by internal short or excessive heating caused by regulator failure. Other indirect hazards relate to fire resulting from circuit breakers, toxic outgassing and fires related to high temperatures, and EMI-induced signal errors such as false resets and timing starts. This section discusses design guidelines for connectors, batteries, circuit breakers, cables and wiring, control circuits, power system, and parts/components/elements.

4.5.3 Cryogenics

Some of the systems which will be considered below utilize cryogenic storage of fuel cell reactants, generally in large quantities. The cryogenics section of the Handbook describes hazards involved with personnel exposure to extreme cold, combustion of cryogenic fluids, low temperature environment effects on mechanisms and structural materials, and the need for pressure relief valves.

5.0 ELECTRICAL POWER SYSTEM

The electrical power system with the nuclear reactor as an energy source consists of the elements shown in the functional diagram of Figure 41. To size the electrical power system, the loads and the requirements must be established.

5.1 Electrical Power System Requirements

5.1.1 Space Station

Altitude:	270 Nm, 500 Km, Low Earth orbit	
Inclination:	28.5°	
Life:	Station	30 years
	EPS	10 years

5.1.2 Reactor

The reactor will be of the SP-100 class in the power range of 50 to multihundred kilowatts. For the manned space station the reactor will be man-rated. A reactor on a free-flying vehicle need not be man-rated when the vehicle is at a safe distance from the space station. Reactor control will be provided from an independent power source and must be available at all times with a reliability of 0.99999.

Shielding shall be provided to protect the equipment and persons in the space station to the value of 5.72 mrem/hr. Shielding shall also protect the astronauts on the side away from the space station to the value of 200 mrems at 30 meters.

5.1.3 Operational Environment

The space environment for the reactor and its power system shall be defined by the position of the reactor with respect to the space station. At the space station operational altitude and orbit inclination, the plasma environment is shown in Figure 42. This environment shall be used to guide the selection of EPS equipment operating voltages. The energy conversion and the power conditioning equipment which operate at or near the reactor and the shield must be capable of operating in the defined radiation and thermal environment. Precise values of the environment parameters can only be defined for each conversion system concept selected and where they are located with respect to the shield and the reactor.

Included in the operational environment will be the forces caused by any dynamic machinery - turbines, pumps, reciprocating engines. These forces must be balanced by opposing forces so that the net force will be essentially zero. In the event of a failure of any dynamic equipment, the counter-motion equipment shall be shut down when the failed equipment is shut down. Thermal energy of the power system radiators will provide a thermal environment which must be integrated with the space station. EVA traffic around the radiators must also be controlled. The meteoroid environment can be protected against by shielding radiators with bumper shields and redundant tubes in fluid cooling loops. The heavy equipment-reactor and shield will not be damaged except from large pieces of meteoroid; for which there may not be any protection. To date, no such accidents have been recorded.

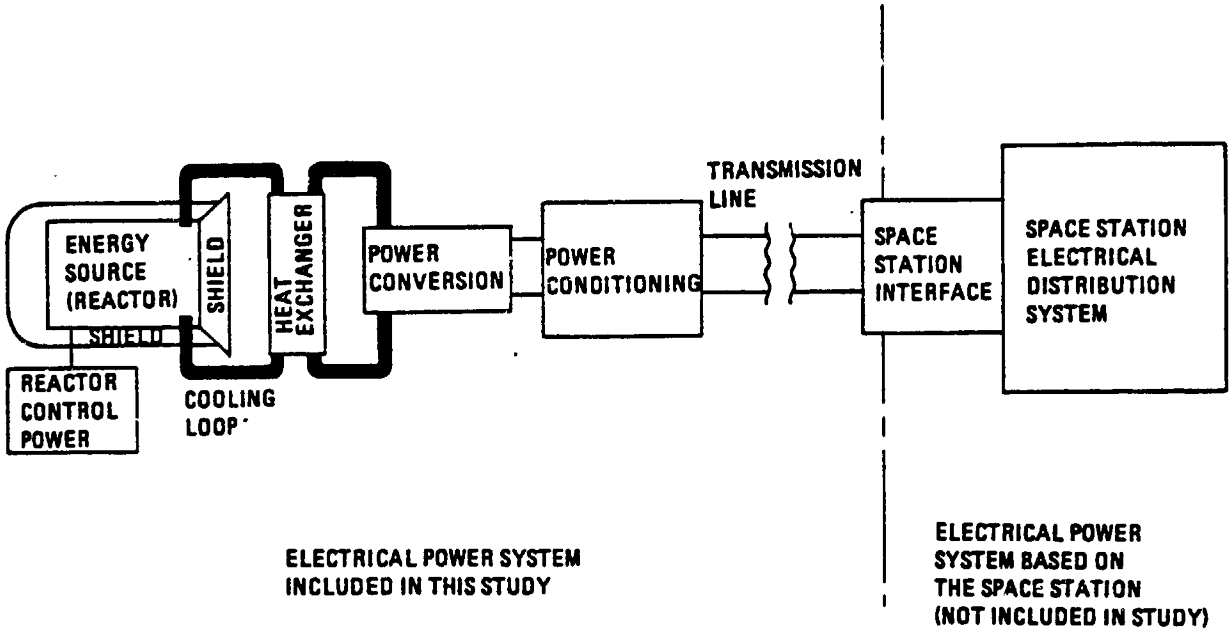

Figure 41 Electrical Power System Functional Diagram

5.2 Power Conversion System Options

Figure 43 shows schematically the conversion systems which are candidates for converting the reactor thermal energy. The concepts which can be selected are dependent upon the power level of the system. For the power level up to about 150 kWe, thermoelectric conversion is an outstanding candidate. At high power levels the low conversion efficiency results in large areas of radiators. Hence, at the higher power levels higher efficiency conversion is required. Other conversion methods can also be used in this range and for the higher power ranges.

5.3 Transition From Solar to Nuclear Power

An assumption is made that the initial space station will be powered by a solar array (photovoltaic or dynamic engine) with electrochemical energy storage because of the date of the operational space station (1992), which requires that the date for freezing the technologies will be 1987. It is unlikely that any nuclear reactor power system can be available and demonstrated/qualified by 1987.

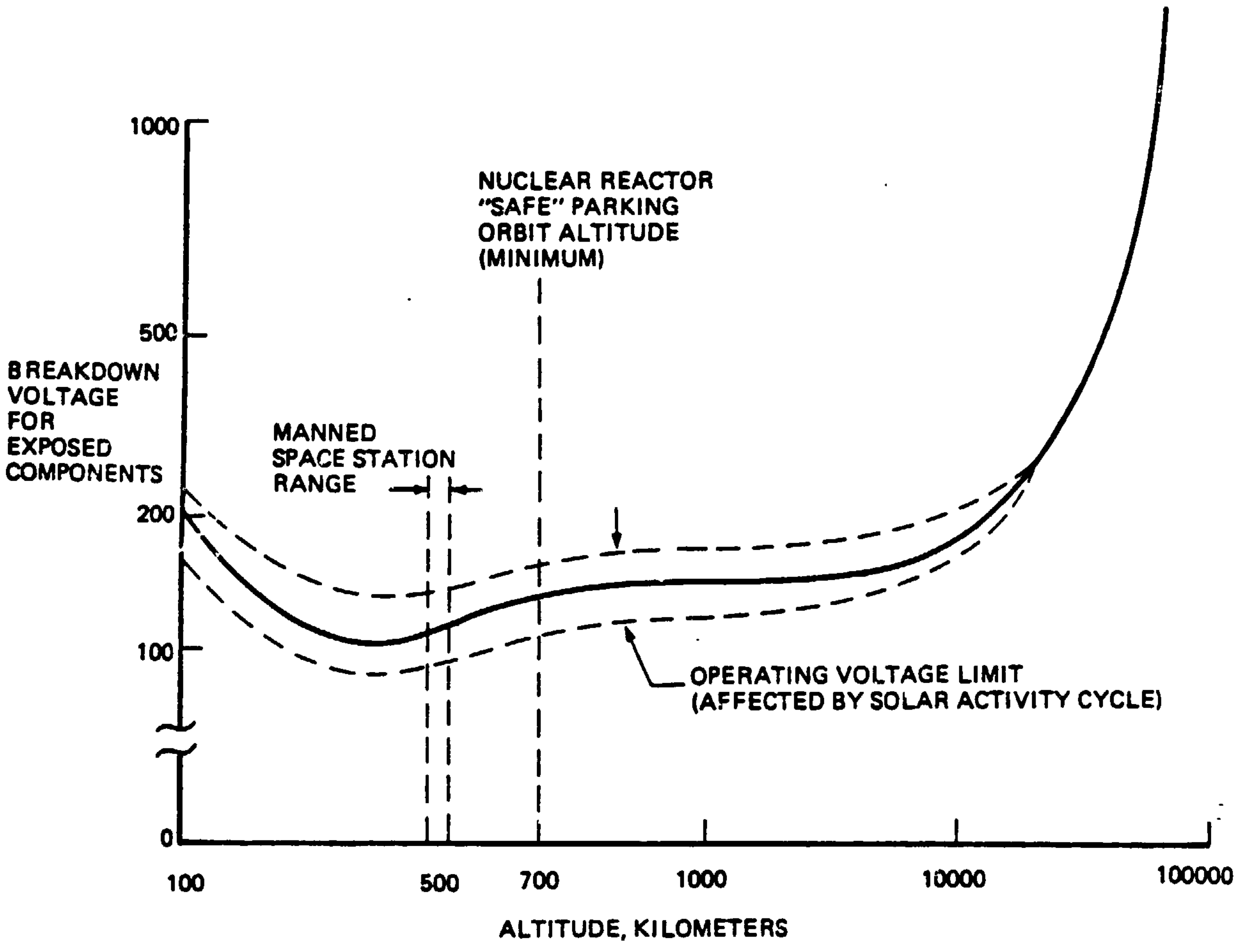

Figure 42 Effects of Plasma on Exposed High-Voltage Components

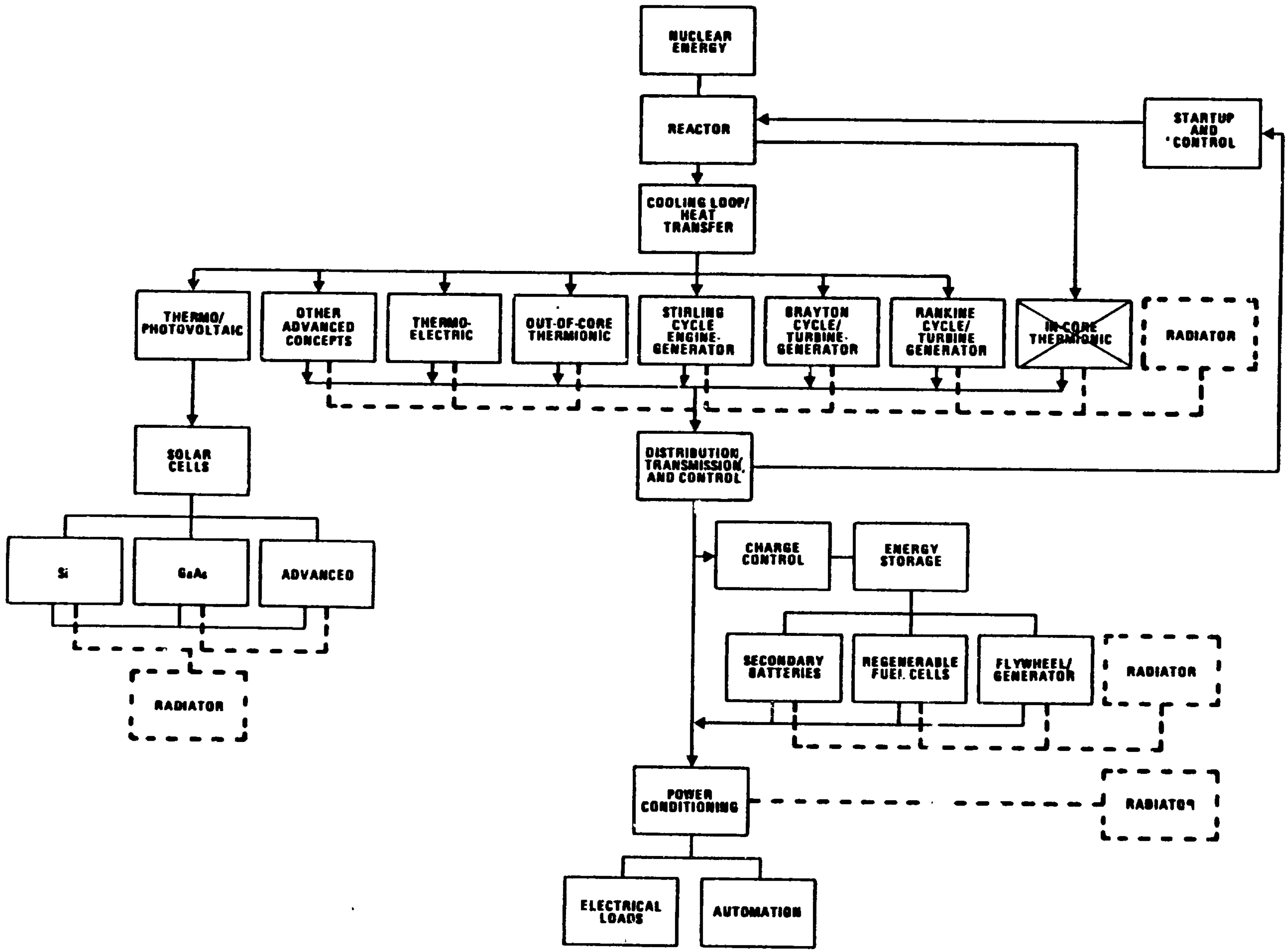

Figure 43 Electrical Power Subsystem Candidates

The photovoltaic solar array will generate DC power, which must then be transmitted to the space station over a transmission line. In most cases the transmission line will be quite long, 70 to 100 meters, so that a high voltage will have to be generated for the transmission line. Because of the Earth's plasma, the voltage of the solar array will have to be kept below the plasma breakdown voltage unless all exposed connections can be sealed from the plasma.

As for the transmission line, it can be operated at high voltage DC and/or AC. Since we are not selecting a point design, we will assume that we can have either, or both. DC power lines will be 2-wire, insulated cables. AC power lines will most likely be 3-phase, 4-wire, Y-connected, to minimize current in the lines. The precise voltage limit is indeterminate at this time, but the plasma environment and integrity of the insulation materials will determine the upper limit voltage on an exposed solar array. For the AC transmission line to the space station modules, the upper voltage limit will be the peak of the sine wave (1.414 x r.m.s. value). High voltages will be required in order to minimize losses in the transmission line between the conversion equipment and the user loads.

In the space station the distribution will be some form of AC and/or DC. At this time the frequency and the voltage are indeterminate and are to be selected in the space station phase B studies. The requirement will be to select a distribution configuration which will be compatible with any other power system configuration when a reactor power system will be installed on the growth space station. Growth from the IOC station to later advanced stations is to be considered in the original design.

For the high-voltage transmission line, the requirement shall be to install two or more sets of 4-wire cabling with each wire rated for three times the maximum operating current, so as to accommodate starting currents and fault currents. The number of sets of installed cables shall be two more than the quantity selected for the initial quantity of generation equipment.

For the installation of the reactor, the requirement imposed on the space station will be to provide the structural equipment for mounting or tethering the reactor/shield and providing the interface link. The analyses in Section 8 show that the shield mass dominates the nuclear reactor powered electrical system mass.

6.0 NUCLEAR REACTOR-POWERED SPACE STATION CONFIGURATION OPTIONS

An advanced space station with electrical power needs greater than 100-150 kWe can benefit from a nuclear reactor power source. The configuration options for including a reactor as the power subsystem for a space station are considered in this section. The options include attaching the reactor rigidly to the space station; attaching the reactor to a flexible tether; or locating the reactor on a free-flying spacecraft removed from the space station. Power transmission can be achieved by electrical conduction, by fuel transport, or by electromagnetic beaming. Fourteen different configuration options have been investigated, and three were selected for more detailed trade studies.

6.1 Complete Option Tree

The configuration options can be categorized according to the location of the reactor and the mechanism by which power is transmitted to the space station. The complete option tree considered in this study is shown in Figure 44. The reactor, shield, power conversion system, and radiator designs were considered on a generic basis in this preliminary assessment, i.e. general system advantages and disadvantages were investigated rather than focussing on subsystem designs. The power level was also not defined in this systems assessment, but a value between 100 kWe and 300 kWe was implicitly assumed.

Reactors which are rigidly attached to the space station can be located either in the space station itself or on a boom. The space station electrical power can be drawn from the nuclear power system either directly through conducting wires or indirectly as in a regenerative fuel cell. The indirect case consists of an electrolysis plant located with the reactor and fuel cells located with the space station. Fuel cell reactants would be pumped through fluid pipes located in the boom structure to the fuel cells, with a return line pumping the reaction product back to the electrolysis plant.

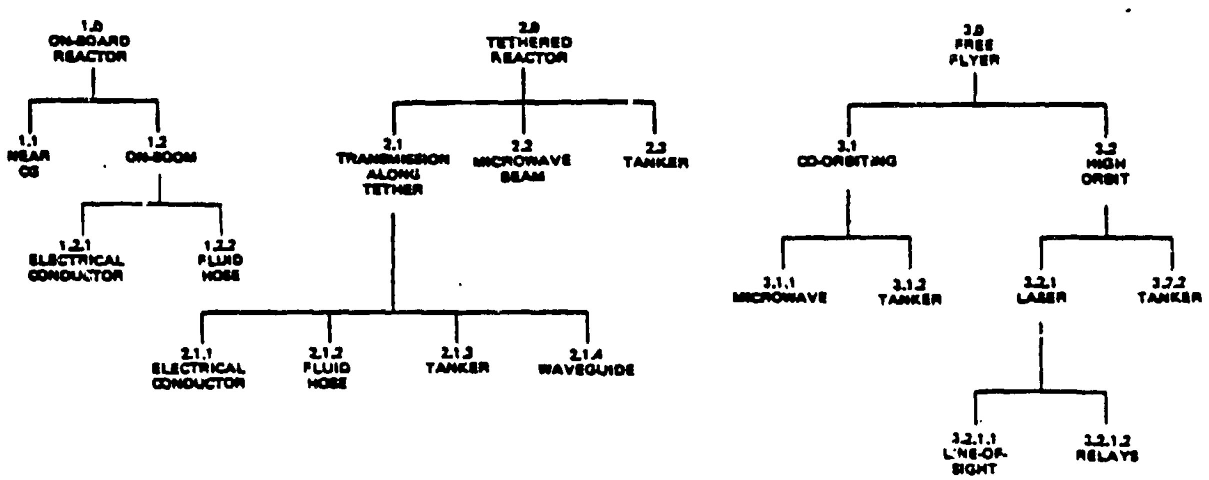

Figure 44 Reactor-Space Station Configuration Options

Reactors which are attached to the space station with a long, flexible connection are tethered reactors. Power transmission can be either along the tether or through free space. The tether can be electrically conducting with reactor-generated electricity transmitted through the tether. With an electrolysis plant located with the reactor, the reactants can be transferred either continuously through flexible hoses or with a batch process using tanks transported either along the tether or through space by an OMV or an OTV. Electricity generated at the reactor can also be converted to electromagnetic energy and beamed to the space station, either through free space or through a waveguide.

Free-flying reactors can be either in the same orbit (co-orbiting) as the space station or in a higher orbit. If the reactor is co-orbiting, power can be transferred either by microwave beaming or by fuel tankers. If the reactor is in a higher orbit, then the reactor and space station are not always nearby each other. A tanker can still be used to transport fuel cell reactants and reaction products back and forth, but for electromagnetic beaming the vehicles must be in line with each other. Laser power transmission can be used in conjunction with an energy storage device or with relay satellites. If beaming is only done when the reactor and space station are in direct line of sight, then energy must be stored for use when the earth occults the direct path. With a sufficient number of relay satellites, light can be beamed continuously from the reactor to the space station.

6.2 On-Board Reactor Power System Selection

The most straightforward configuration option is that where the electrical power is generated near where it is to be distributed. This option places the reactor on the space station with sufficient shielding to protect space station personnel to safe radiation dose rates for continuous residence at the space station. This option is referred to as the submarine option because of its similarity to a nuclear power plant on a manned submarine.

This concept is illustrated in Figure 45. The reactor, power conversion system, and shield are located on the space station, with the radiator somewhat removed from the station. A 100 kWe space reactor will reject from 300 to 2000 kW of heat at temperatures in excess of 550 K, with details dependent on the power conversion efficiency. This heat will be rejected by radiation from 40-80 m^2 of radiators. To prevent excess heating of the space station and astronauts performing extravehicular activities, the radiator must be removed to some distance from the space station, probably 30 meters or more away.

This approach requires pumping of a heat transfer fluid from the reactor to the separated radiator. It also has the heaviest shield mass of all the options considered, since the shielding must protect people residing in close proximity on a continuous basis. The shield mass required to reduce the radiation dose rate to 5.7 mrem/hour at a distance of 3 meters from the reactor in all directions is about 35-45 tonnes, depending on power level and conversion system.

On the positive side, this approach allows the most flexibility in space station attitude, since there is no dominant inertia moment. It has a very high ballistic coefficient, requiring a minimum of orbital makeup propulsion. This configuration also allows a wide envelope of EVA operations except near the radiator. This envelope is not constrained in any direction by ionizing radiation levels. The only EVA constraint is that due to thermal exposure near the radiators. This option

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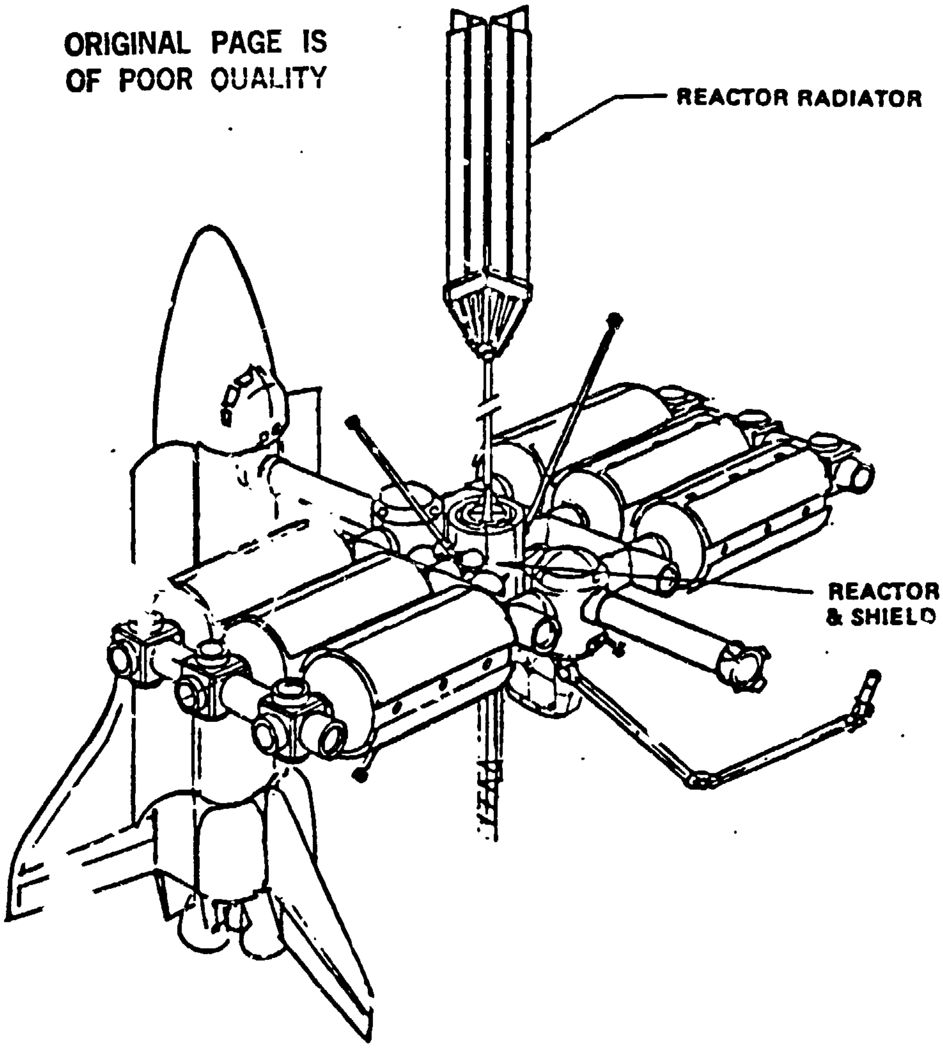

Figure 45 *Reactor Near Space Station CG* ('Submarine' Concept)

provides good opportunity for manned maintenance of the reactor system if that requirement is desired.

The shield mass can be reduced significantly by placing the reactor on a boom attached to the space station, as shown in Figure 46. Rather than shielding for continuous manned presence close to the reactor in all directions, this option requires shielding for limited exposure periods in all directions near the reactor and continuous exposure only on the space station, which is removed to some distance away from the reactor. This reduces the shield mass depending on distance, power level, and conversion system. This option also places the radiator near the heat source, reducing the need to pump heat.

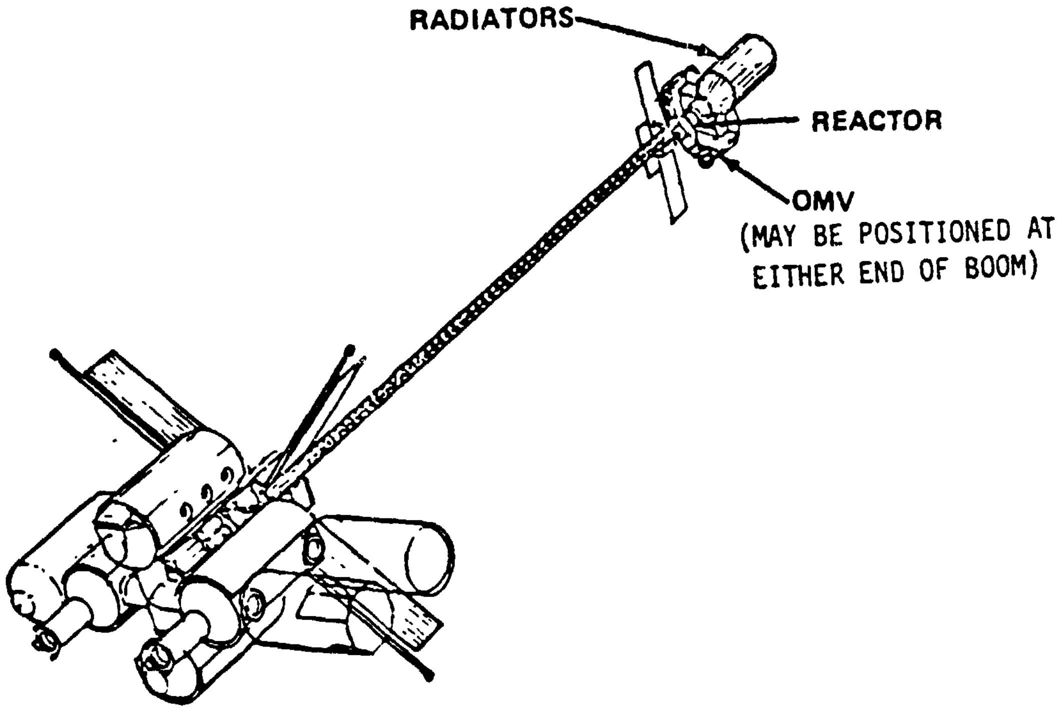

Figure 46 Reactor on Boom

The power conversion system is located at the reactor end of the boom. Electricity can be used to electrolyze fuel cell reactants, which are then pumped to fuel cells on the space station, or transmitted directly to the space station electrical power distribution system through conducting wires. If the fuel cell option is used, the reactor must provide enough electrical power to make up for electrolysis, pumping, and fuel cell inefficiencies, i.e.

$$P_0 = \frac{P_e}{\eta_{e\ell}\ \eta_{Fc}} + P_{pumping}$$

where P_e is the electrical power demand at the space station,

$P_{pumping}$ is the power required to pump the fuel,

P_0 is the reactor electrical output power,

$\eta_{e\ell}$ is the electrolysis efficiency, and

η_{Fc} is the fuel cell efficiency.

Current capabilities for a regenerative fuel cell system efficiency are 60-66% (ref. 16) for a hydrogen/oxygen system. Hydrogen/halogen regenerative fuel cell systems might boost the system efficiency still further, perhaps to 80-90%. Ignoring pumping losses, the reactor output must be 11-67% higher than space station demand for the fuel cell option. Electrical conduction, however, would require an excess power of less than 5%.

The boom-mounted reactor option achieves its mass savings by restricting personnel access around the reactor, allowing lighter shielding. Exclusion zones are therefore introduced within which EVA and orbiter traffic is restricted to limited duration. This might affect reactor maintainability if the design requires manned maintenance. The boom-mounted reactor also introduces a preferred space station attitude. The large mass at the end of a long boom will drive the system toward a highly stable gravity gradient mode.

The relative advantages and disadvantages of the three on-board reactor configuration options are summarized in Figure 47. The submarine option, with the reactor on the space station, requires the heaviest shield and must utilize some means of transporting waste heat to the remote radiator. The boom-mounted systems have lighter shields and do not require remote heat rejection. Of the two boom-mounted systems, the conducting transmission line option requires the least reactor power, so it was judged preferable to the fluid pipe option. Since the technology for electrical transmission is more straightforward than that for heat transfer over 60-70 meters length in space, the conducting transmission line was selected as the preferred option for detailed trade studies.

Figure 47 *On-Board Reactor Options*

Configuration	Advantages	Disadvantages
1.1 Submarine	• Good attitude flexibility • Manned maintainability • Good traffic accessibility • Minimum transmission distance	• 35-45 tonne shield mass • Need for separated radiator
1.2.1 Conducting Transmission Lines	• 12-20 tonne shield mass • Radiator near heat source	• Traffic restrictions • Attitude limitations • Need for power transmission
1.2.2 Fluid Pipe	• 12-20 tonne shield mass • Radiator near heat source	• Traffic restrictions • Attitude limitations • 11-67% power losses

6.3 Tethered Reactor Power System Selection

The tethered reactor power system option is shown generically in Figure 48. The reactor is located at one end of a long, flexible tether and the space station is located at the other end. Dynamic stability considerations will likely dictate that the tether be vertical, with the reactor above the space station. Power transmission is by conduction, fuel transfer, or electromagnetic beaming.

The advantages and disadvantages of tethered reactors in general vis-a-vis on-board reactors are summarized in Figure 49. The shield mass will be discussed in detail in Section 6.4 below. As expected, the mass decreases steadily with tether length. For a man-rated shield, this shield mass decreases only asymptotically

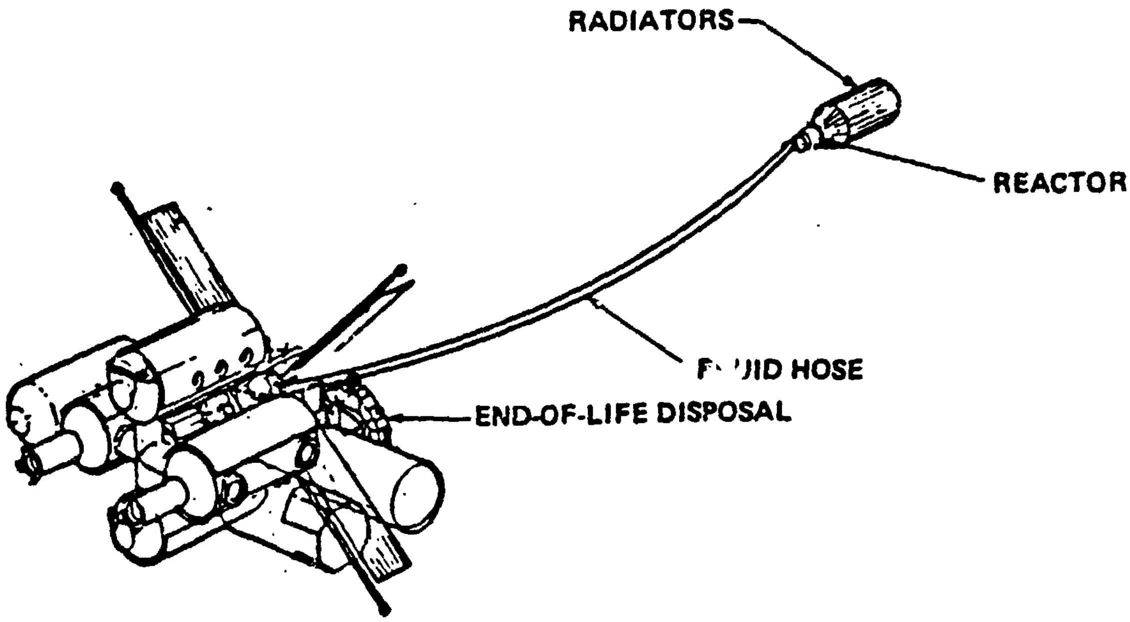

Figure 48 *Tethered Reactor*

Fig· ·· 49 *Tethered Reactor* Characteristics

Advantages	Disadvantages
• Decreased shield mass	• Shield mass asymptotic beyond 60 meters
• Distant radiator	• Gravitational forces ~ Θ (0.05 μg/m)
• Reduced traffic constraints	• Single point failure from tether break
• Obviates perigee kick stage for long tethers	

beyond 60 meters. That is, the man-rated shield mass decreases to an asymptotic value with distance which it reaches at about 60 meters and no benefit can be gained in reducing the shield mass by increasing the tether length beyond this distance.

The next threshold for tether length occurs at about 25-30 kilometers. This occurs when the gravity gradient induced tension in the tether would impart an impulse to the reactor if the tether were severed. This impulse places the reactor in a transfer orbit with the same perigee as the initial reactor circular orbit and an apogee which is seven times farther from the system center of gravity than the perigee. For example, if the space station were originally in a 500 km high circular orbit and the tether length were 30 km with a system center of gravity at 505 km, a tether break would transfer the reactor from its original 530 km circular orbit to an elliptical orbit with perigee at 530 km and apogee at 680 km. Thus, the tension in the tether could obviate the need for a perigee kick stage when placing the reactor in a long-lived orbit.

A long tether with a length of more than a kilometer would simplify the concern of thermal radiation to the space station, orbiter, or EVA astronauts. Traffic constraints would also be relaxed, since the space station and most traffic patterns are out of the ionizing and thermal radiation fields.

One drawback to a long tether is the acceleration on the space station. The reactor-space station system will travel in an orbit corresponding to a balance of forces at the system center of gravity. Pseudogravitational forces will occur at all other locations, since the gravitational and centrifugal forces will not cancel. The strength of these forces is about 0.5 x 10^{-6} m/s^2 for each meter away from the center of gravity, or 50 micro-g's per kilometer. Many of the reference space station missions require microgravity conditions for materials processing. While it is not yet well-known what level of acceleration is acceptable to complete these missions, it is believed that g-levels between 10^{-5}g and 10^{-3}g may be required. If the missions are sensitive to 10^{-5}g (10^{-4} m/s^2), then a tether length of 200 meters or more would perturb the mission. If 10^{-3}g is allowed, then 20 kilometer tethers would be acceptable.

Power from the reactor can be transmitted to the space station by a conducting tether. This is essentially the same configuration as that selected above for further analysis as an on-board reactor, and will not be considered again here since we wish to select one of each type of concept for analysis.

The reactor power system can also have an electrolysis plant associated with it, and fuel cells on the space station. This is shown schematically in Figures 50 and 51. the fuel cell reactants have a high systems synergism if H_2-O_2 fuel cells are used: the reactor would then provide oxygen for life support systems, hydrogen for stationkeeping propellant, and perhaps fuel for cryogenic orbital transfer vehicles. In this study the analysis did not include additional power for serving the integrated systems. The reactant flow rate required for 100 kWe is 8-10 gm/s for fuel cell efficiencies of 60-80%. Assuming a regenerative fuel cell system efficiency of 60-75%, the reactor output power must be 1/3 to 2/3 higher than the load required at the space station.

The reactants can be carried to the space station either through hoses on a continuous basis, with hydrogen and oxygen gas flowing to the space station and water returning to the reactor, or with free-flying tankers carrying liquid fuels

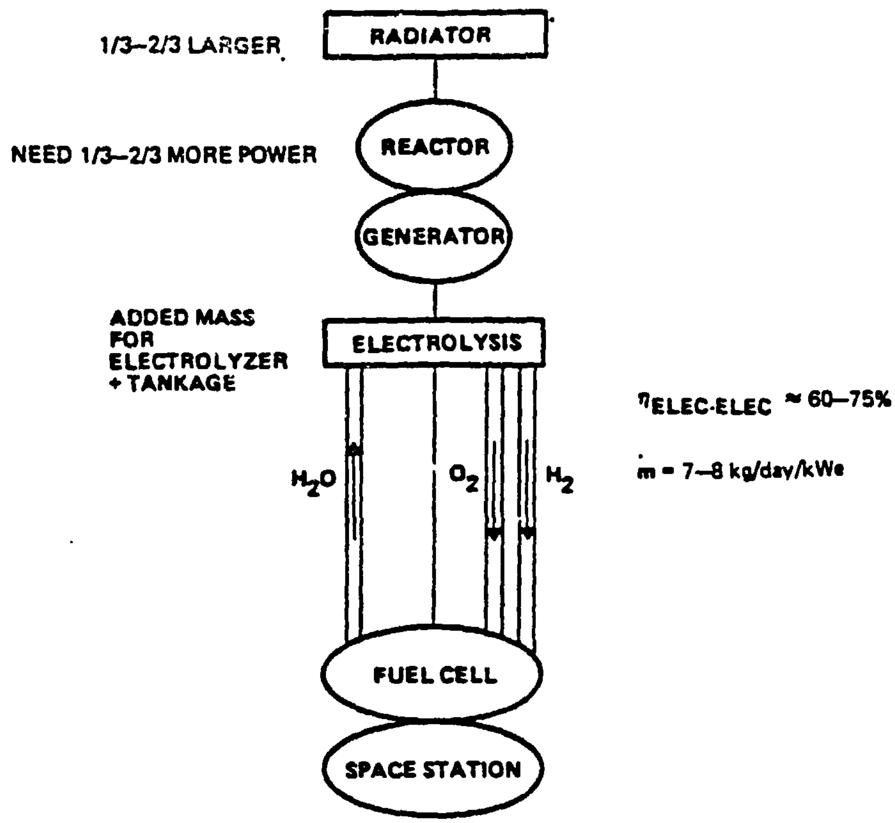

Figure 50 *Fluid Hose Tether*

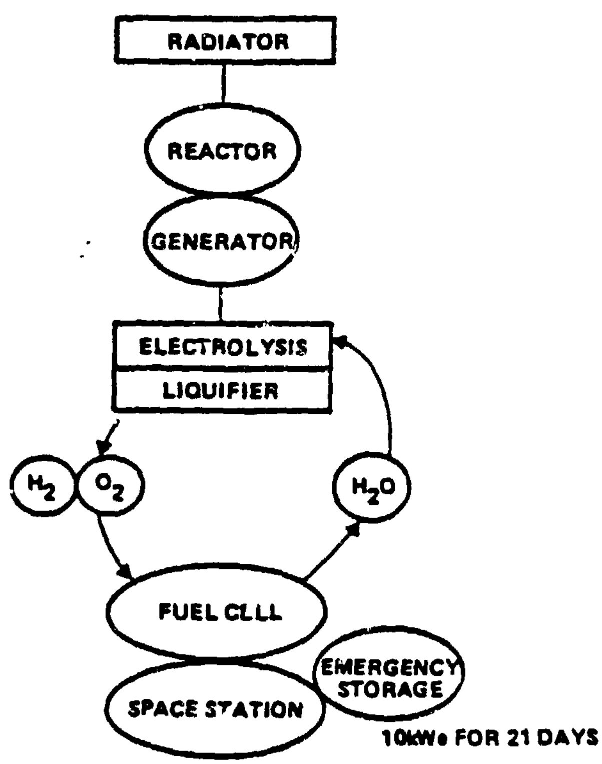

Figure 51 *Tethered Tanker*

both ways. The tanker option requires a liquefaction plant at the reactor and involves substantial traffic of the fuel tankers, while the hose option requires neither. The hoses might be more vulnerable to meteoroid impact. In each case, additional mass must be placed in orbit for the electrolyzer, fuel cells, and hoses or tankers.

A variation on this option is one where the fuel is carried in tanks which travel along the tether. This is referred to as the clothesline option, with the hydrogen and oxygen moving along the tether to the space station and the water moving to the reactor. This is essentially the same as the tanker option, except that the tankers are attached to the tether.

Power can also be transmitted to the space station in the form of microwave beams. Figures 52 and 53 illustrate two forms of microwave beaming. A klystron at the reactor generates microwaves which are transmitted to the space station, either through a waveguide or through free space. In the free space beaming case, the output power required at the reactor is:

$$Po = \frac{Pe}{\eta_{dc-rf}\ \eta_{trans}\ \eta_{rf-dc}}$$

where η_{dc-rf} is the klystron efficiency, including the waveguide feed and the near-field losses,

η_{trans} is the transmission efficiency, accounting for path loss and spillover,

η_{rf-dc} is the antenna efficiency, including rf-dc conversion and rectenna heating losses.

In the waveguide case, the reactor output power is:

$$Po = \frac{10^{\mu L}}{\eta_{dc-rf}\ \eta_{rf-dc}}\ Pe$$

where μ is the waveguide attenuation coefficient,

L is the waveguide length.

Klystron efficiencies of 35% have been measured, and optimistic projections estimate efficiencies of 70% may be possible. Antenna efficiencies are likely to be about 50%, but optimistic projections may place this value as high as 90%. Waveguide attenuation coefficients between 0.01 and 0.03 decibels per meter seem likely, while beam losses of 20-40% are reasonable for S-band (3GHz) transmission through kilometer distances. Thus, the range of reactor output required to deliver 100 kWe to the space station for the free space microwave beaming option is 200-950 kWe. For a one kilometer waveguide tether, reactor outputs of 1.6 MWe-600 MWe are required.

The relative advantages and disadvantages of the five tethered reactor configuration options are summarized in Figure 54. The microwave options require considerable excess reactor power to compensate for system losses: at least a factor of three for klystron and antenna efficiency and another factor of ten for waveguide attenuation. The two tanker options require a liquefaction plant and tanker vehicles. The liquefaction plant would be a major power drain and would require a substantial development program to enhance the system lifetime.

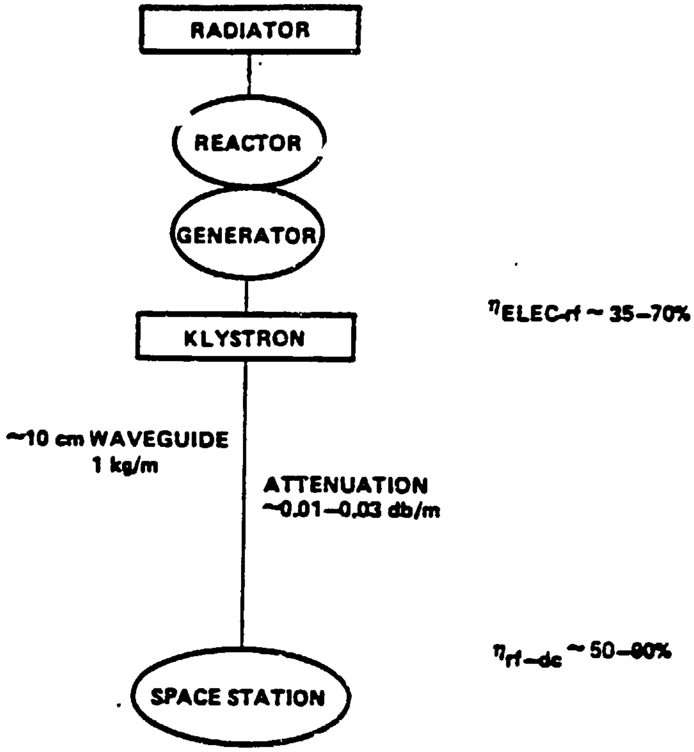

Figure 52 *Waveguide Tether*

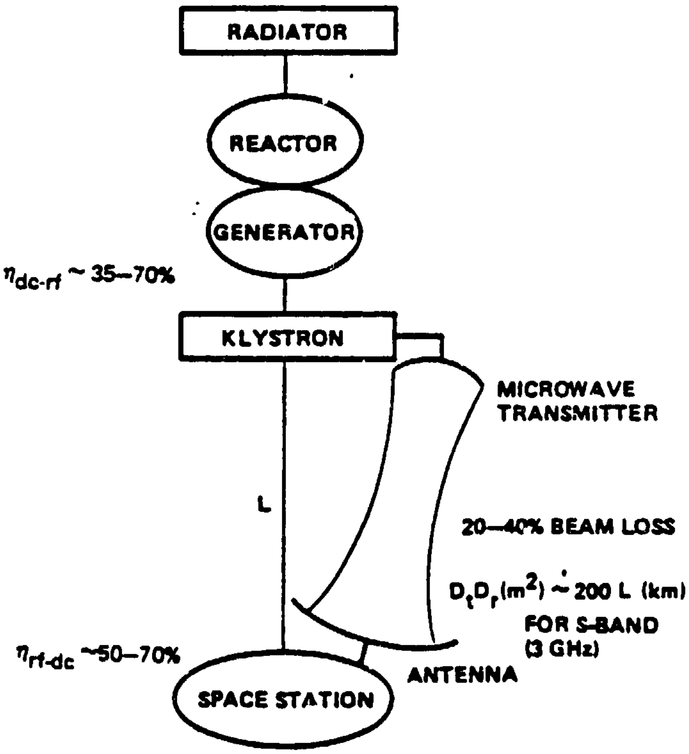

Figure 53 *Microwave Beam Tether*

Figure 54 *Tethered Reactor Options*

Configuration	Advantages	Disadvantages
2.1.2 Fluid Hose	• Synergism with ECLSS • Synergism with orbit propulsion • Provides OTV fuel • Continuous fuel • Reactor power ~ 1.4 P_e	• Meteoroid vulnerability • Hose mass
2.1.3 Clothesline Tanker	• Synergism with ECLSS • Synergism with orbit propulsion • Provides OTV fuel • Light tether	• Need for liquifaction plant • Need for tanker vehicles • Tether stability
2.1.4 Waveguide	• Microwave safety	• Waveguide mass • Reactor power ≥ 30 P_e
2.2 Microwave Beam	• Light tether	• Antenna size • Microwave safety • Reactor power ≥ 3 P_e
2.3 Tanker	• Synergism with ECLSS • Synergism with orbit propulsion • Provides OTV fuel • Light tether	• Need for liquifaction plant • Need for tanker vehicles • Tanker traffic

Tanker traffic would occupy crew time for docking and fluids transfer, which would impact mission effectiveness.

The fluid hose tether provides continuous fuel in an untended mode of operation and involves no cryogenic fuel. It might be vulnerable to meteoroid impact and requires an ability to locate and repair leaks. Of all the tether options, the fluid hose requires the least reactor power. The fact that the fluid hose/fuel cell options provide high system synergism and fair system efficiency led to the selection of these over the microwave options. The fact that the fluid hose provides fuel on a continuous basis and does not require a liquefaction plant or crew-monitored traffic management led to the selection of the fluid hose as the most favored tether option for detailed trade studies. The tether mass will be analyzed as part of the trade study in Section 8.7.

6.4 Free-Flyer Reactor Power System Selection

A free-flyer reactor option is shown generically in Figure 55. The reactor, power conversion system, shield, and radiator are located on a free flying platform, which can either be co-orbiting with the space station or in a higher orbit. Power transmission is either by fuel tankers or by electromagnetic beaming.

The advantages and disadvantages of free flyer reactors in general, vis-a-vis on-board or tethered reactors, are summarized in Figure 56. The principal advantages are the very low shield mass required if no manned presence is allowed in the vicinity, and the possibility of obviating the need for an orbital booster at end of life by placing the reactor in a long-lived orbit to begin with. Since the reactor is on a free flying platform, the platform must have all of the systems required of an independent spacecraft, including attitude control, communications, and propulsion subsystems. Furthermore, the savings in shield mass come at the expense of precluding manned access. This means that no manned operations are allowed on or near the reactor, no traffic is allowed nearby, and no manned maintenance is permitted.

The reactor can be co-orbiting with the space station, flying in formation at some distance in front of the station, so that the space station, because of its greater mass will not catch up with the reactor power system. In this arrangement, power can be transmitted either by microwave beaming or by fuel tankers. These configurations are shown schematically in Figures 57 and 58. They are analogous to tethered reactor options described above, with many of the same features. The reactor would be located in front of the space station because of its higher ballistic coefficient. This is for safety reasons: if orbit stationkeeping capability were lost, the higher drag of the space station would make it slow down more and separate it from the reactor. This separation distance is shown in Figure 59. At an altitude of 500 km, without orbit makeup propulsion, the reactor would move about 30 kilometers away from the space station in the first day and continue to accelerate away from the space station, to 2000 kilometers within about one week. This illustrates an inherent safety feature of the coorbiting free flyer, but also illustrates the need for orbit makeup propulsion.

Placing the reactor in a high orbit allows selection of a long-lived orbit for all reactor operations. The greatest drawback to this approach results from orbital dynamics considerations. The higher orbit has a longer period, so the reactor and space station do not maintain a constant phase relationship and, further, differential nodal regression moves the two spacecraft out of plane with each other. A 500 km circular orbit has a period of 5672 seconds and a 700 km circular orbit has a period of 5921 seconds. Thus, a reactor that is initially 200 km directly over a space station at 500 km will trail 1800 km behind the space station after only one orbit. The differential node regression rate for these two orbits, at an inclination of 28.5°, is 0.68°/day. Thus, a reactor that is initially in plane with the space station at a 200 km higher altitude will drift out of plane at this rate and not be coplanar again for 530 days.

Power transmission from a high orbit reactor to the space station can be accomplished by electromagnetic beaming or by tankers. Because of the distances between the reactor and the space station, microwave beaming would require enormous optics. Diffraction phenomena dictate the minimum optics sizes which can be used for electromagnetic beam transmission. The ideal limit to the receiver diameter D is:

$$D_r = \frac{2\lambda R}{D_t}$$

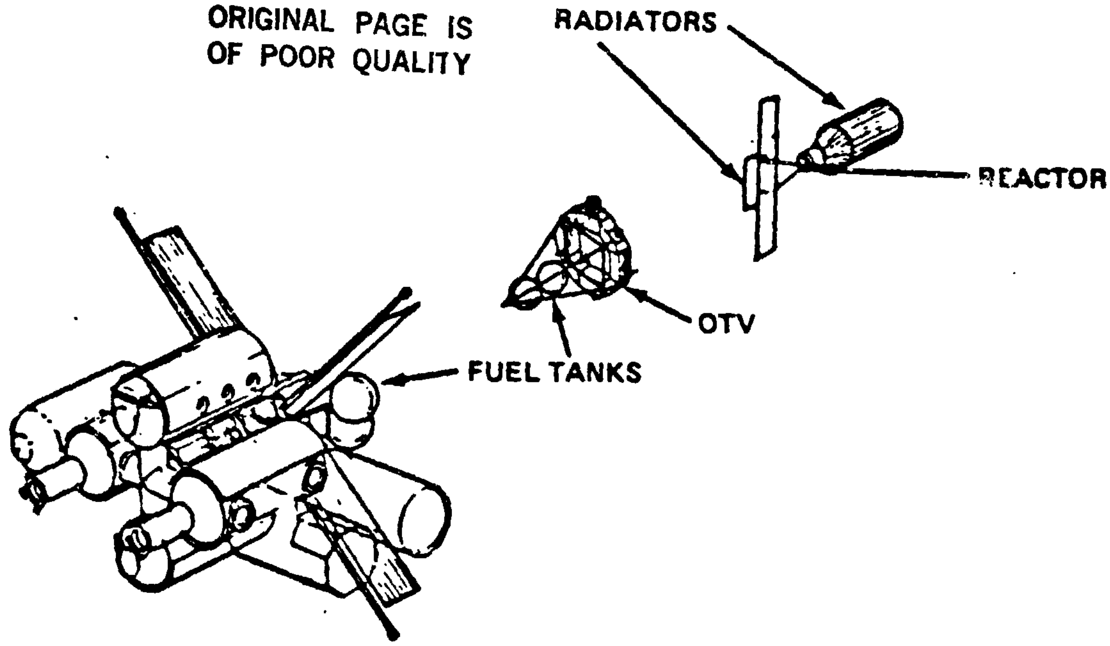

Figure 55 Free-Flyer Reactor

Figure 56 *Free Flyer Reactor Attributes*

Advantages	Disadvantages
• Light shield	• Requires independent spacecraft systems • Attitude control • Communications • Propulsion
• Long-lived orbit	• Requires power transmission • No manned maintenance/operations

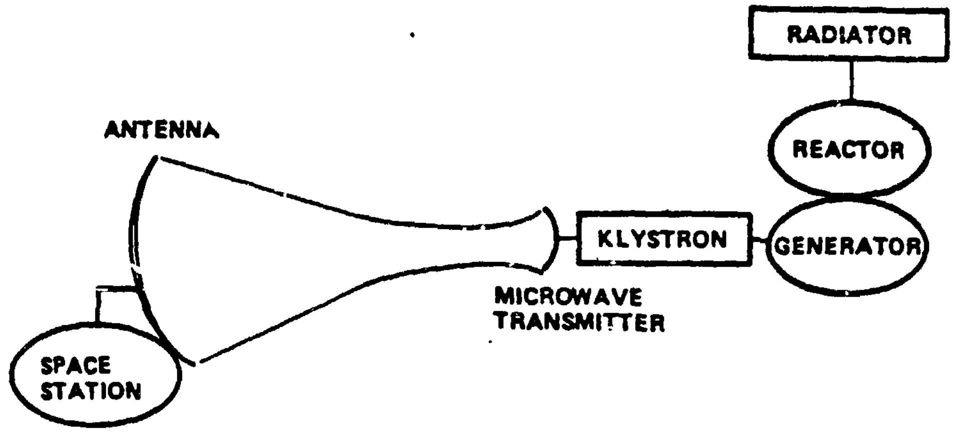

Figure 57 *Coorbiting Microwave Free Flyer*

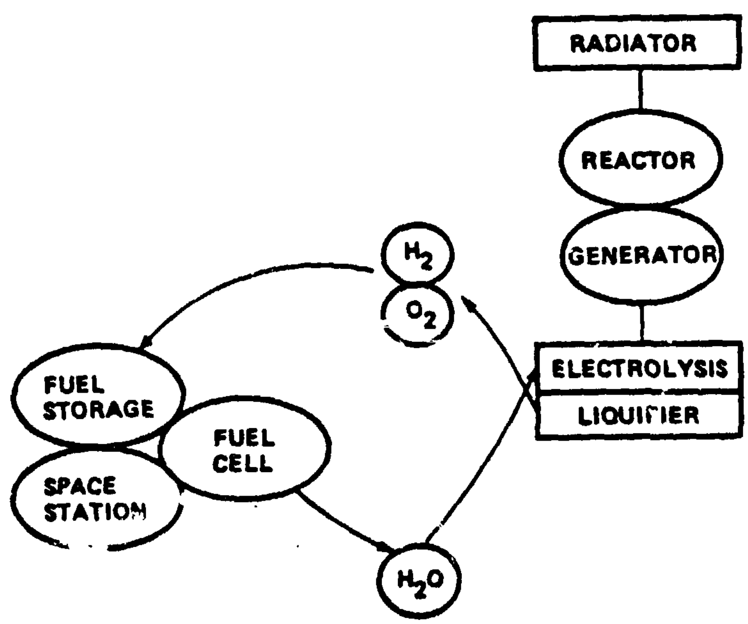

Figure 58 *Coorbiting Tanker*

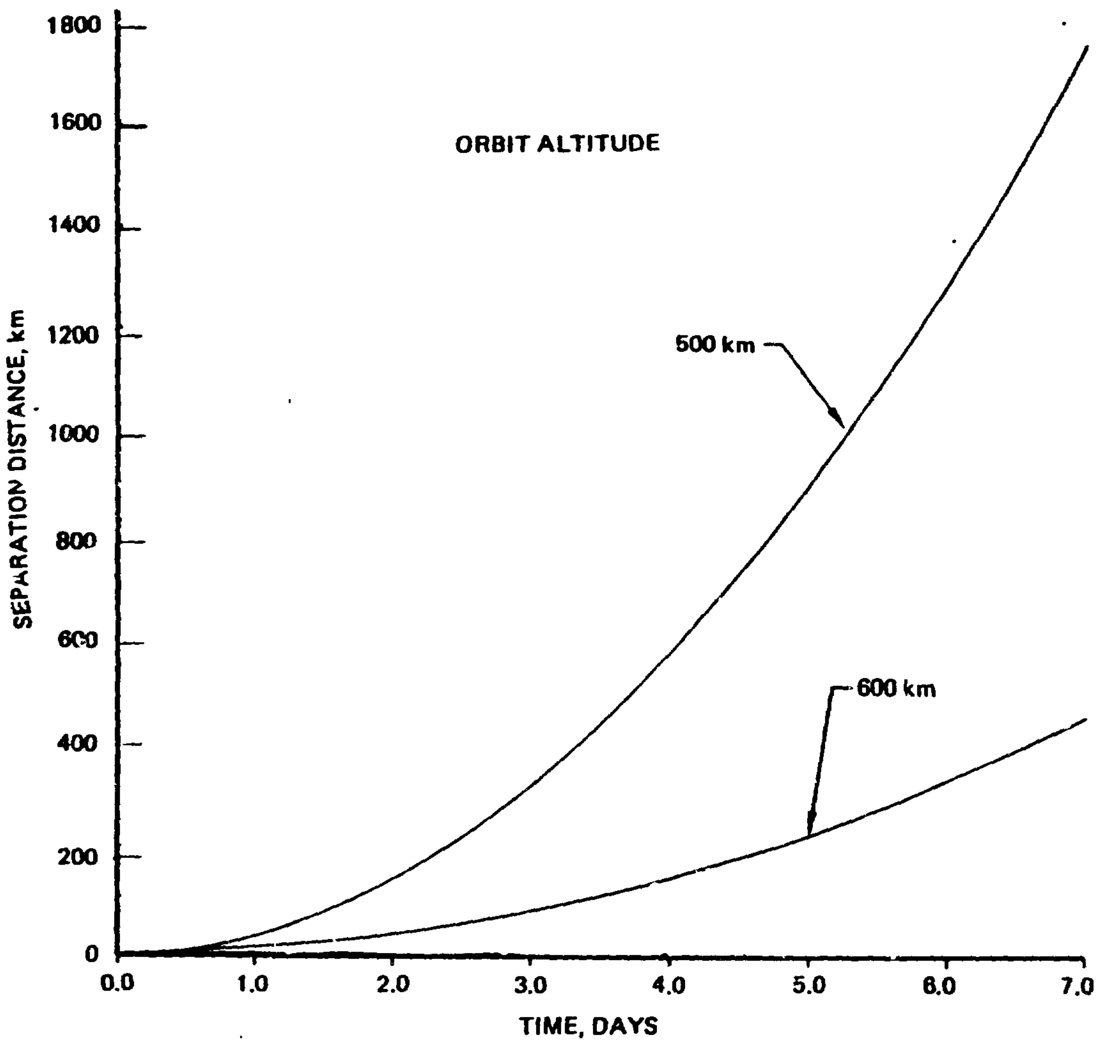

Figure 59 Separation Distance Between Space Station and Free-Flying Power Source

where λ is the beam frequency,

R is the distance from the transmitter to the receiver,

D_t is the transmission aperture diameter.

Microwave wavelengths are 1-10 cm. This means that the minimum size for microwave antennae and transmitters must be about 100-300m if beaming is done only when the spacecraft are close, and 300-1000m if beaming is done whenever they are in line of sight. This large size, and the impact it would have on space station missions and design, dictate the use of shorter wavelengths. A laser with a wavelength of 0.5 m would require transmitter/receiver apertures of several meters. Laser transmission was considered as the only feasible spectral band for this study.

The laser can only [illegible] power to the space station directly when the station is not occulted by the earth, i.e. only when the direct line of sight from the reactor to the space station does not intersect the earth. If the two spacecraft are in plane at 500 km and 700 km, the viewing opportunities last for 9.0 hours out of every 37.3 hours, as shown in Figure 60. In this figure, the bars represent times when the space station is visible from the reactor. The rest of the time, they are occulted by the earth. Figure 61 shows the azimuth and range when the two are in direct line of sight. These figures become more complex when the spacecraft are not in coplanar orbits.

The space station requires a steady power source. Laser power transmission therefore requires either high power during periods of good viewing, combined with energy storage on the space station, or a system of laser relay satellites. The latter was not investigated in this study because it would require a large number of mirrors with extremely high pointing precision. Figure 62 illustrates the former concept. The reactor output power must be sufficient to compensate for laser and receiver inefficiencies. It must satisfy:

$$Po = \frac{Pe}{\eta_{e-L}\ \eta_{L-e}\ VF}$$

where η_{e-L} is the electric-to-light laser efficiency,

η_{L-e} is the light-to-electric efficiency of the receiver,

VF is the view factor.

Typical high power continuous wave laser efficiencies are under 1%. The theoretically achievable efficiency might be as high as 40%. Current photovoltaic receiver efficiencies are around 15-18%, but some concentrator designers predict efficiencies up to 35%. For a 100 kWe space station, the reactor output power must therefore be around 1 GWe [illegible]h current technology, and even with optimistic technology advances the reactor output power must be about 3-20 MWe.

The high orbit free flyer reactor option is shown in Figure 63. The reactor has an electrolysis and liquefaction plant, and the space station has a fuel cell. Liquid hydrogen and oxygen are stored in tanks at the reactor which are carried to the space station with an orbital transfer vehicle. The OTV carries water to the reactor on the return trip. Since the propellant required for large plane changes is

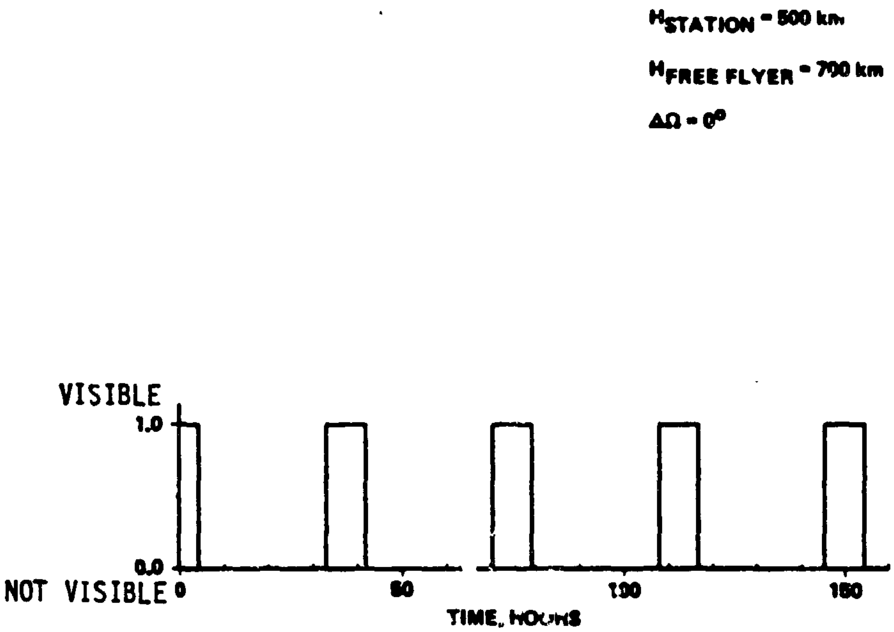

Figure 60 High Orbit Free Flyer Viewing Factor

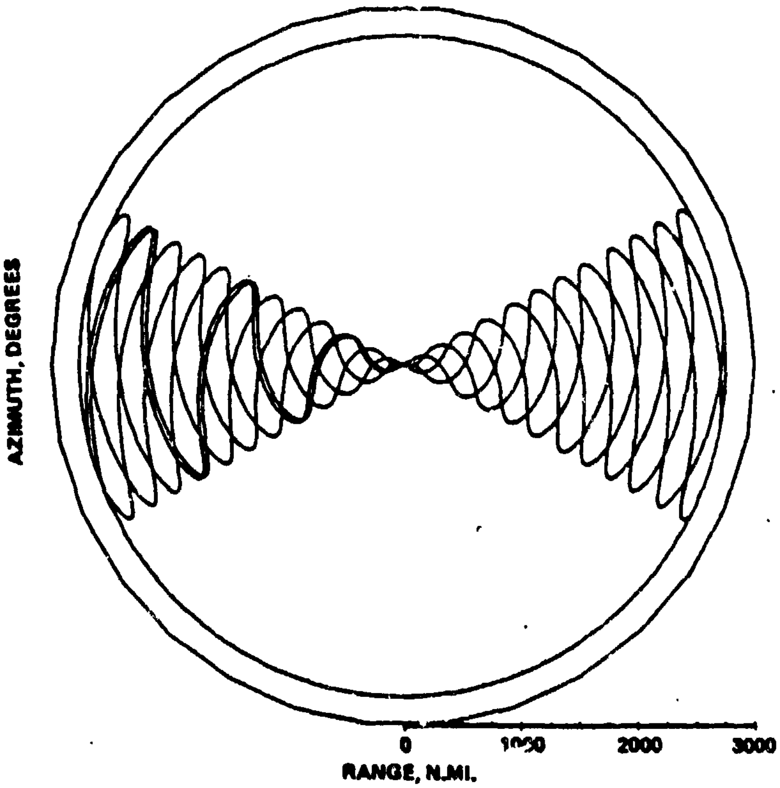

Figure 61 Relative Free Flyer Position

c - 2

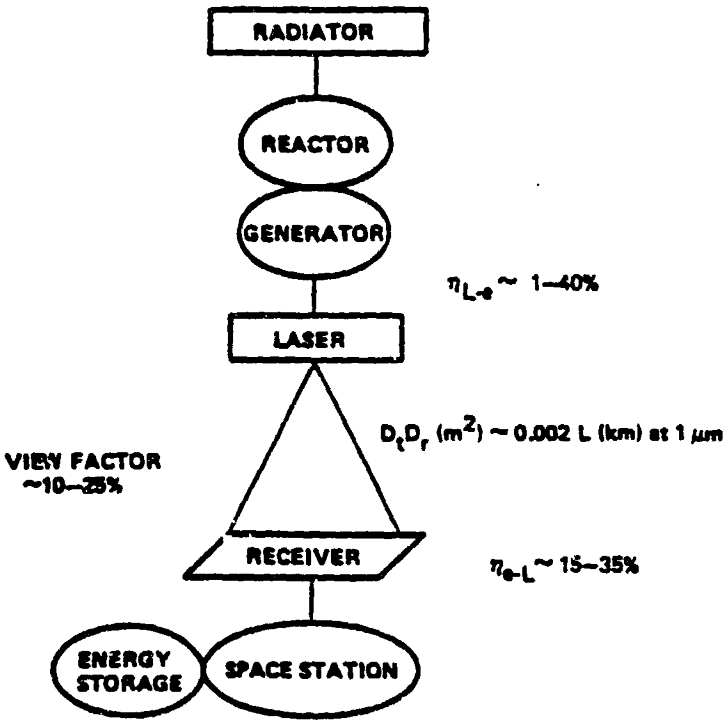

Figure 62 Line-of-Sight Laser Free-Flyer Reactor

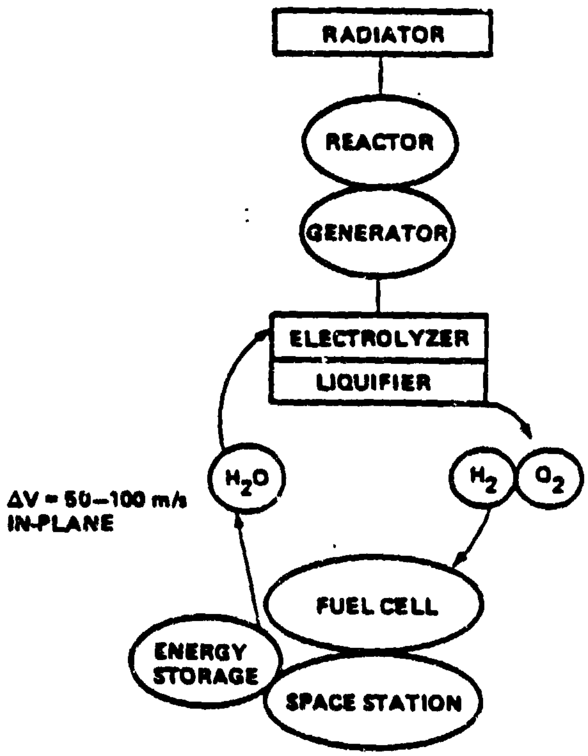

Figure 63 High Orbit Free-Flyer Tanker

very large, the frequency with which the reactor and space station become coplanar determines the launch frequency and therefore the storage and payload mass. The orbital transfer V and the payload mass determines the propellant requirements. These will be discussed quantitatively in Section 8.7.

The advantages and disadvantages of each of the free flyer options are summarized in Figure 64. The coorbiting options offer the same features as their tethered analogs, with the added complexities associated with the independent spacecraft systems, i.e. the tethered microwave option seems more attractive than the coorbiting microwave option and the coorbiting tanker offers no advantages over the tethered tanker. For the high orbit options, the lasers require extremely high reactor power. Therefore, the high orbit free flyer tanker was selected as the most attractive free flyer option.

6.5 Mission Sensitivity

One configuration from each general class was selected for a detailed trade study. These are indicated in Figure 65: one on-board reactor option, one tethered reactor option, and one free flyer option. The on-board option consists of a boom-mounted reactor with conducting wires to the space station. The tethered and free flyer reactors use reactor power to electrolyze water to H_2 and O_2 which are used in fuel cells at the station; the tether supplies continuous fuel and the free flyer delivers it in discrete batches. Two of the options place the reactor in a short-lived orbit and the free flyer is in a long-lived orbit. The microwave and laser beam options were not considered further, primarily because of the excess reactor power required to compensate for beam generation and conversion inefficiencies.

Any design configuration selected must be capable of performing the missions described in Section 2, above. It must provide power to the space station in the 100-300 kWe range, largely for materials processing missions. It must permit good viewing for astronomy and earth observation missions. It must allow extravehicular activity with considerable freedom of motion for space-based assembly and maintenance. It should not be subject to large accelerations which would jeopardize microgravity quality for materials processing. The constraints imposed on vehicular traffic and the frequency and magnitude of docking disturbances should be minimal. Finally, since crew size is limited and their time is valuable for mission performance, the power system should not require large amounts of crew time.

Figure 66 summarizes the potential impacts of the nuclear power system on space station attributes affecting mission performance. All three systems are readily scaleable in power level in the range from 100 to 300 kWe and above. All three have low drag and minimum control complexities when compared with solar power systems. The nuclear reactor power systems do not interfere with viewing in any direction, except for the boom-mounted reactor which only blocks a very small area ($\sim 10^{-3}$ steradians, mostly due to the radiator). The free flyer reactor allows the space station complete flexibility to select the attitude, while the tether and boom-mounted configurations will have a strong gravity gradient moment.

The boom-mounted and tethered reactor shields are designed to reduce radiation levels at the space station to acceptable levels for habitation with higher radiation dose levels for short periods in areas away from the space station. This will introduce some restrictions to traffic near the reactor, but these are minimal by design. The thermal output of the radiator at high temperature will limit

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Figure 64 *Free-Flyer Reactor Options*

Configuration		Advantages	Disadvantages
3.1.1	Co-orbiting Microwave Beaming	• Fixed reactor-station orientation • Fixed spacing with RCS	• Microwave safety • Viewing interference • Differential drag • Antenna size • Reactor power >3 P_e • End of life disposal
3.1.2	Co-orbiting Tanker	• Low ΔV • Flexible station attitude • System synergism	• Tanker traffic • Need for liquifaction plant • Need for tanker vehicles
3.2.1.1	Line-of-Sight Lasers	• Inherently safe disposal • No fuel storage	• Need for energy storage • Pointing complexity • Laser hazards • Reactor power >20 MWe
3.2.1.2	Lasers With Relays	• Inherently safe disposal • No fuel storage • No energy storage	• System complexities • Spacecraft number
3.2.2	High Orbit Tanker	• Inherently safe disposal • System synergism • Moderate system efficiency	• Orbit complexity • Propellant losses • Infrequent launch opportunities • Need for liquifaction plan • Need for tanker vehicles

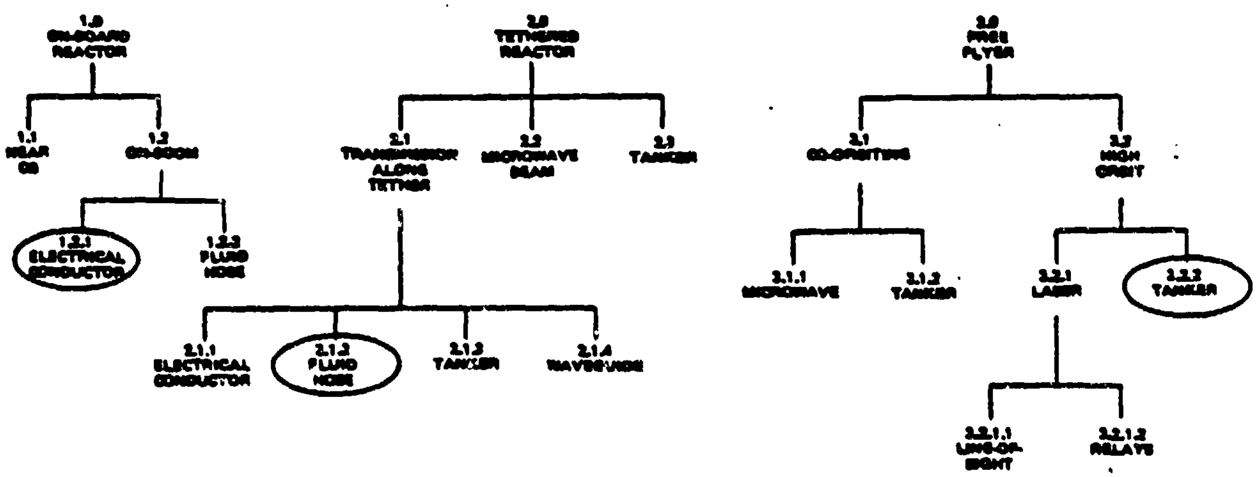


Figure 65 Reactor-Space Station Configuration Selection

extravehicular access in the vicinity, which constrains extravehicular activity within some exclusion volume near the boom mounted reactor. There are no EVA constraints on the space station missions in the free flyer configurations, and only the obvious traffic restriction of avoiding collision with the tether in the tethered reactor case.

Microgravity quality is very high in both the boom-mounted and free flyer cases. The tethered reactor would induce an acceleration on the space station on the order of 0.01 m/s^2 (1 milli-g). This might adversely impact materials processing missions. If it does, then other designs would have to be considered, such as locating materials processing facilities at the center of gravity along the tether or providing a counter balance below the space station to move the center of gravity down to the station.

The free flyer reactor will have tanker traffic between the reactor and the space station ferrying fuel back and forth. This will cause docking disturbances and will occupy crew time in rendezvous, docking, fuel transfer, and return. There will also be additional orbiter traffic and docking to provide OTV propellant for the tanker. The OTV will also likely require periodic servicing at the space station, further occupying crew time. There will be no docking disturbances or crew time associated with the boom-mounted or tethered reactor options.

Potential Impact of Nuclear Power System	Nuclear Space Station Configuration Concept		
	On-Board	Tethered	Free Flyer
Power scaleability	Yes	Yes	Yes
Drag	Low	Low	Low
View blockage	Low	None	None
Attitude flexibility	Preferred attitude	Preferred attitude	Flexible
Traffic constraints	Minimal	Minimal	None
EVA constraints	Some	None	None
Microgravity quality	High	Medium	High
Docking disturbances	None	None	Some
Crew time	None	None	Some

Figure 66 *Summary of Potential Impacts on Missions*

7.0 REACTOR AND ELECTRICAL POWER SYSTEMS

This section of the report summarizes the evaluations performed on the reactor and electric power generation subsystems in assessing the applicability of nuclear power systems for manned space stations. The section is organized with initial subsections discussing the major components of a nuclear power system, namely, the reactor, the shield, and the power conversion subsystems. These are followed by a presentation of parametric weight data for a range of power levels, shield configurations and power conversion subsystem options. Data and sketches are presented for specific power system-space station arrangements and for alternate higher powered system arrangements. A separate report section discusses system operational and reliability characteristics in general terms. The final section summarizes the conclusions reached on the characteristics of nuclear power systems for the space station application.

The goals of the study were to examine three operational locations for the nuclear power system: on-board the space station or closely attached to it with a boom, tethered but at a long distance (~30 kilometers) from the station, and as a free flyer. To aid in this examination, parametric mass information and layout sketches of power systems operating in the power range of 100 to 300 kWe of net electrical power and using dynamic and static conversion cycles were to be determined. An underlying assumption in the studies was that the reactor and power conversion systems would be close derivatives of SP-100 technology since the development of a different nuclear power system specifically for the space station application is unlikely to happen. Two specific power systems were assumed for the comparisons of the boom-mounted, tethered and free flying configurations: a baseline thermoelectric power system developing 150 kWe of power, and a 300 kWe system utilizing an advanced dynamic conversion cycle with essentially the same reactor. Operational characteristics were to be determined as well as dimensional and mass estimates.

The study approach included on evaluating the relative mass characteristics of alternative nuclear power system concepts. Since the shield component is the major weight item in manned applications of nuclear power systems, design aspects that significantly influenced the shield design and mass were of primary concern. Those aspects include the geometric configuration of the shield, the allowable dose rates from the reactor, and the size and efficiency of the power conversion concept. Four shield geometries were originally considered, designated as four-pi, two-pi, conical, and shaped four-pi configurations. The form of the four geometries is shown in the sketches in Figure 67 which also depicts a general, close-coupled arrangement of the power system and space base. An indication of the relative masses of the shield geometries and their relative areas of dose protection can be envisioned from Figure 67.

Dose rate limitations from the reactor were assumed at two locations: at the space station proper and at a 30 meter distance from the four-pi shield. The reactor dose requirements were assumed to be one half of the total allowable dose.

The effects of power conversion type on shield size and weight were determined as a function of three conversion systems; one dynamic, one static, and a baseline thermoelectric conversion system. The generic dynamic and static conversion systems were synthesized from characteristics of specific systems. A discussion of the power systems evaluated and their characteristics of interest is given in Section 7.2.

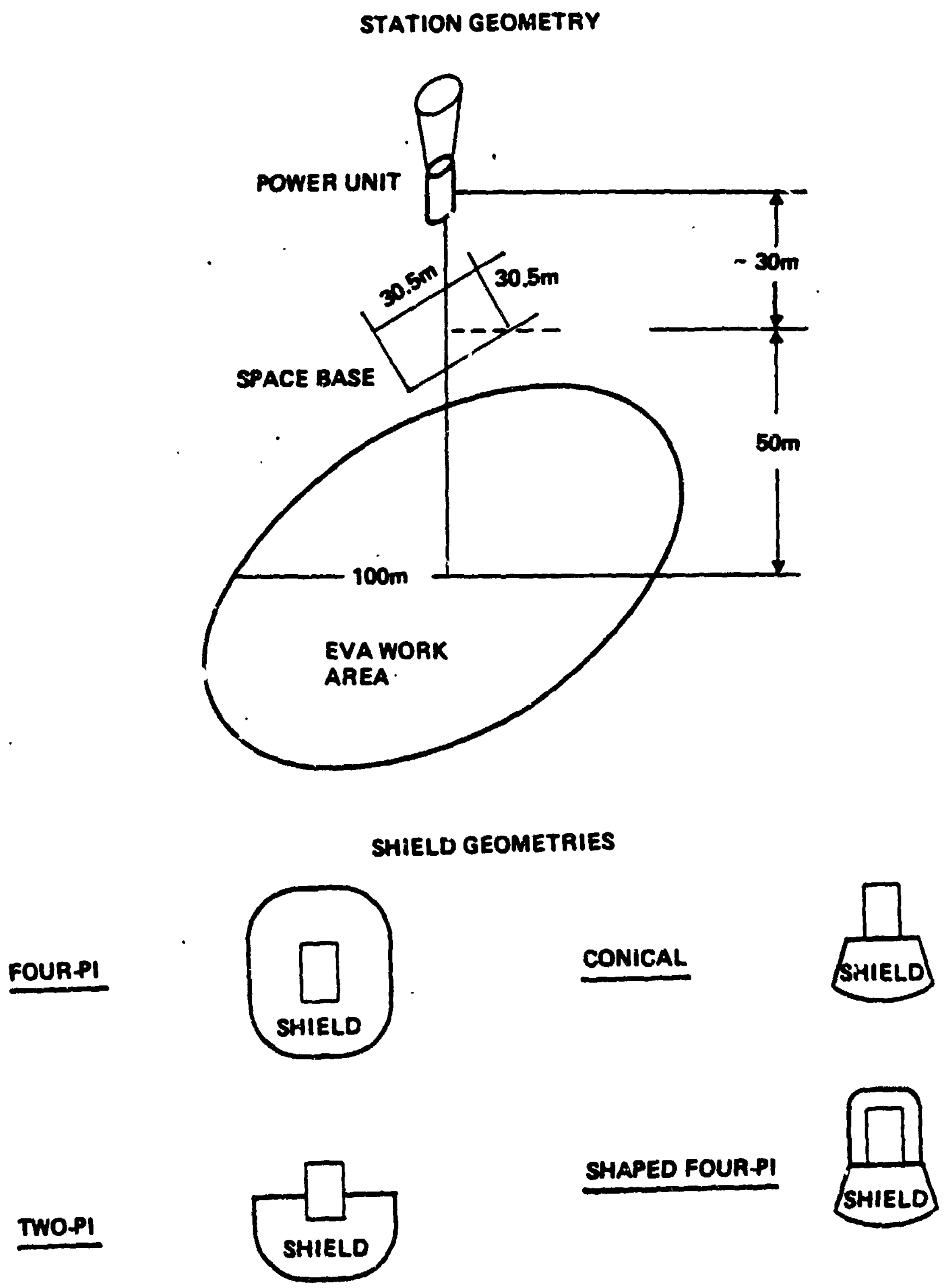


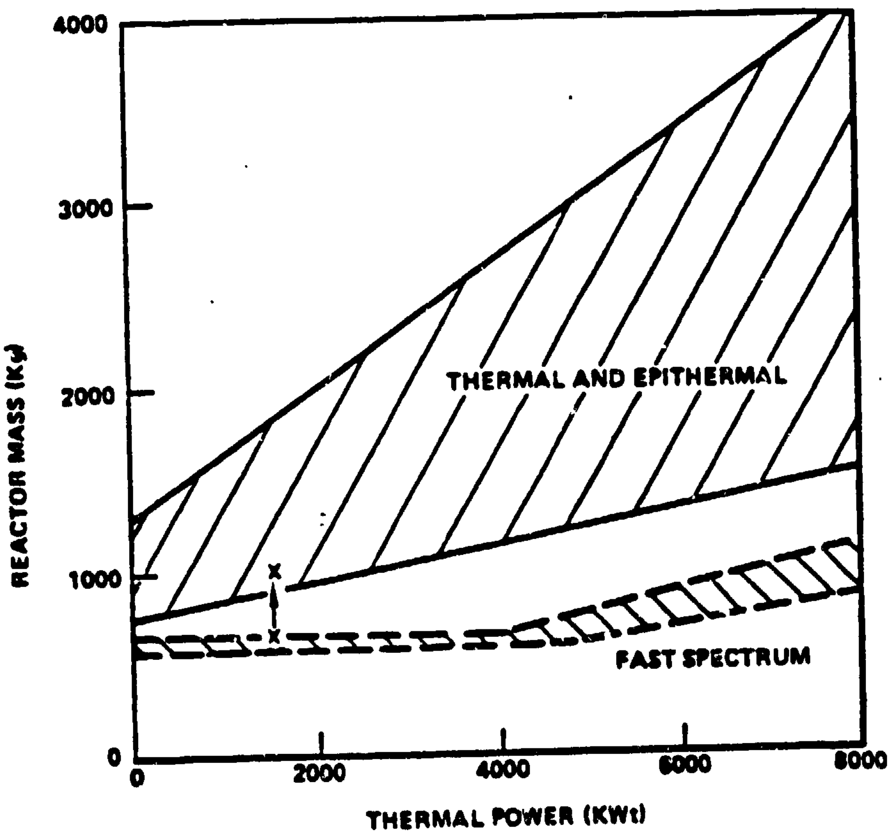
Figure 67 Shield Concepts

7.1 Reactor Options

One option in the selection of a reactor for a space nuclear power system is the neutron energy spectrum of the operating reactor. A fast neutron spectrum reactor was assumed for this study because of its significant advantage in size and weight for both the reactor and its associated shield. The mass advantage is illustrated in Figure 68 which compares the mass of generic reactors having fast, epithermal or thermal neutron energy spectra. The figure also shows that a single fast reactor core design of constant mass can satisfy thermal energy requirements up to 4 megawatts in contrast to the increasing mass required for the other reactor types over the same power range.

Under the generic designation of fast spectrum reactors various reactor concepts exist which for this discussion have been categorized as follows:

- o Liquid metal cooled
- o Gas cooled
- o Heat pipe cooled
- o In-core thermionic



X
CORRESPONDS TO 400 Kg REACTOR MASS INCREASE
X

Figure 68 Reactor Subsystem Mass of Generic Reactor Types Versus Thermal Power

All of these concepts can provide the required thermal energy and all can be designed to meet the safety requirements and the required lifetime capability. But significant differences exist in their design and operating characteristics which provide a basis of selection for a given application. The in-core thermionic converter can be integrated in the reactor core and its size and mass vary with output power. In-core thermionic conversion is still in the early stages of development and has not demonstrated adequate life in operation.

Heat pipe cooled, fast spectrum reactors have been proposed but none have been built to date. The size and mass of this type of reactor is also dependent on the thermal output requirements because of the void space introduced into the reactor with the addition of heat pipes.

Gas cooled reactors are ranked low in terms of general application because they are most practical when integrated with a gas dynamic power conversion cycle such as a Brayton unit. Even that combination potentially introduces radioactivity into the power conversion devices, thus limiting or precluding maintenance activity and requiring more shielding for manned systems. Another significant disadvantage is the system's vulnerability to single point failure of the working gas (reactor coolant) containment.

Liquid metal cooled reactors are considered to be the most desirable type for space based applications. They are relatively small and compact because of the high heat transfer rate capability of liquid metal. The fission products are contained so that equipment or personnel are unaffected. They are easily integrated with the common power conversion cycles, and as mentioned previously, one design can accommodate up to 4 megawatts of thermal energy. Most significantly, the liquid metal cooled reactor is further advanced in technology development because of the large amount of data and understanding achieved in the Liquid Metal Fast Breeder Reactor programs. Consequently, a reactor with the characteristics described in Figure 69 was assumed for this study. Although the characteristics listed are general in nature, the size and mass values are taken from General Electric's study of SP-100 application.

Figure 69 *Reactor Description*

- **Characteristics**
  - **Fast spectrum**
  - **Pin fuel**
  - **Refractory alloy clad and structure**
  - **Liquid metal heat transfer**
  - **Reflector drum control**
  - **Multiple pumped primary loops**
- **Size – approximate envelope – 57 cm**
- **OD, 74 cm length**
- **Mass – 500 – 600 kg**
- **Power range – to about 4000 kWt**

7.2 Power Conversion System Options

There are a number of power conversion options available for a nuclear space power system that can be grouped into two general classes, dynamic and static. It was desired to evaluate the characteristics of these two classes, and select one as typical for the later comparisons of power system location and arrangement with the space station.

Figure 70 enumerates the individual power conversion cycles considered and lists the principal attributes of each. Three dynamic cycles: Brayton, Stirling, and potassium Rankine; and three static devices: baseline thermoelectric, advanced thermoelectric, and thermionic were investigated. Characteristics of the in-core thermionic arrangement were assumed to typify thermionic performance in general.

Although the inert gas, closed cycle arrangement required for a space-based Brayton unit is not fully developed as yet, it is similar in component design and operation with common gas turbine machinery. Hence, it is considered to be close to the state-of-the-art. Brayton units, like most dynamic cycles, have high conversion efficiencies although the Brayton performance is dependent on relatively low heat rejection temperatures and corresponding large heat rejection areas. Recuperated Brayton units are relatively bulky but can be built with large output power capacity per unit.

Stirling engines have very good performance potential in terms of efficiency and mass but need substantial development for long life, unattended space applications. They are probably limited in unit output power capacity so a number of units would have to be operated in parallel in order to achieve 100-300 kWe total output power.

Potassium Rankine cycles have high heat rejection temperatures which, coupled with high conversion efficiency, results in the smallest heat rejection area requirement of all of the conversion cycles. Another of its positive attributes is a large output power capacity per unit which would allow growth in total station power without the complexity of integrating many conversion units with a single reactor. Significant advances in potassium Rankine technology development occurred in the 1960's and 70's but the development was suspended and would have to be recovered or repeated (ref. 17).

Thermoelectrics are the most common heat-to-electric conversion devices used for space applications. They are very reliable because of very large numbers of individual units which also allow a wide range of voltage-current output characteristics. Baseline thermoelectric materials have been used successfully in many diverse applications, mostly powered by radioisotope heat sources. Their main disadvantage is a relatively low conversion efficiency, although a high heat rejection temperature offsets the low efficiency effect on heat rejection areas.

Advanced thermoelectric materials currently under development may eventually result in higher conversion efficiencies than presently attainable.

Thermionic conversion is potentially more efficient than thermoelectric conversion because of higher hot-side operating temperatures. Although its technology development has been pursued for a number of years, more development is needed before a practical unit can be readied for a space application.

Figure 70 *Power Conversion Systems Studied*

Dynamic Systems	Static Systems
Brayton • Close to state-of-the-art • Growth potential • Range of efficiencies for specific applications	**Baseline Thermoelectric** • Close to state-of-the-art • Highly flexible arrangements possible • Highly reliable and redundant • Relatively low efficiency
Stirling • Attractive specific mass and radiator area • Growth potential • Range of efficiencies for specific applications • Needs substantial development	**Advanced Thermoelectric** • Graceful growth from baseline T/E • Better efficiency than baseline • Technology needs development
Potassium Rankine • Most attractive specific radiator area • High growth potential • Needs recovery of prior technology and development	**In-Core Thermionic** • Attractive static system • Moderate efficiencies • Good specific radiator area • Needs technology development

- These are systems considered for SP-100 studies
- They have quite different technology readiness status and performance

Representative characteristics of the individual power conversion cycles in terms of mass and efficiency are summarized in Figure 71. The dynamic systems have much higher conversion efficiencies (and correspondingly, lower reactor power requirements) and slightly larger masses than the static systems. The characteristics of the individual cycles were used to determine "average" characteristics of the dynamic and static classes as shown by the last two lines in the figure, labeled generic dynamic and generic advanced static, respectively.

The thermoelectric is a little larger in mass than a dynamic system or an advanced static system. Thus, any system arrangement that is practical with the baseline thermoelectrics would also be practical with all other power conversion cycles. For that reason, the power conversion characteristics described in Figure 72 were assumed for all subsequent power system evaluations.

Figure 71 *Power Conversion Cycle Characteristics*

SYSTEM DESCRIPTION	EFF. %	HEAT REJ. TEMP, K PRIMARY	SECOND.	PERCENT* SEC. HEAT REJ.	MASS, KG @100kWe	@200kWe	@300kWe	REACTOR POWER, kWTH @100kWe	@200kWe	@300kWe
BRAYTON	22	550	343	10	1500	2200	2700	455	909	1364
STIRLING	25	640	343	10	1600	2300	2600	400	800	1200
BASELINE THERMOELECTRIC	5.1	850	343	2.5	1300	2000	2700	1961	3922	5882
ADVANCED THERMOELECTRIC	9	850	343	2.5	1300	2000	2700	1111	2222	3333
THERMIONIC	10	850	343	20	1500	1600	1700	1000	2000	3000
POTASSIUM RANKINE	20	790	650	10	2000	2900	3393	500	1000	1500
GENERIC DYNAMIC	22.2	NA	NA		1550	2250	2650	450	900	1350
GENERIC ADV. STATIC	9.5	NA	NA		1400	1800	2200	1050	2100	3150

*PERCENT OF ELECTRICAL POWER OUTPUT

Figure 72 *Space Power System* Characteristics

- Use baseline thermoelectric system
- Radiation coupled modules with heat pipe heat transfer
- Most conservative choice from a size and mass standpoint
- Assumed parameters
  - 5.1 percent overall efficiency
  - 850K primary heat rejection temperature
  - 2.5 percent of net power rejected at 343K
  - Mass and reactor power dependent on output power (See Table 7-2)

7.3 Shield Options

7.3.1 Materials

Lithium hydride is assumed to be the neutron shielding material in this study. The selection of lithium hydride is supported by a wealth of engineering data and fabrication experience for instrument rated shields. Space nuclear reactor shields using lithium hydride for the SNAP-2, 4, 8, and 10A were designed and built in the 1960's. The SNAP-10A was successfully flown and operated in 1965. The SNAP contractor developed the procedures and facilities for fabricating the shields up to 96 inches in diameter by 80 inches thick.

Lithium hydride is brittle below about 530 K, and has very little strength. It must be clad (usually with stainless steel) to minimize hydrogen loss at operating temperatures and large bodies of lithium hydride are best formed by casting. It is compacted and cast within a stainless steel honeycomb (or mesh, in tight corners) to provide the strength to survive launch loads. This also assures that continuous axial cracks do not develop which would result in neutron streaming and degraded shielding capability.

Although the need for alternate neutron shield technologies is not expected, both titanium hydride and zirconium hydride can be considered candidate materials. However, both will result in heavier neutron shields than lithium hydride for the same hydrogen density. Titanium hydride is also less stable at operating temperature than lithium hydride.

Tungsten appears to be superior to depleted uranium for gamma shielding in space nuclear reactor systems. Although both have attractive high densities and melting temperatures and can be readily fabricated, the depleted uranium has two disadvantages:

1. It is toxic and slightly radioactive which can complicate fabrication and introduce unnecessary nuclear safety complications.

2. The residual U-235 isotope not removed during the enrichment process which produces the depleted uranium will undergo fission during space reactor operation. This introduces a low level fission radiation source within the shield, which increases both internal heating and shield size and weight.

Both of these problems are eliminated by the selection of tungsten.

7.3.2 Dose Rate Requirements

The reactor shield designs are based on two separate dose rate limitations to the inhabitants of the space station,

(a) A dose rate of 5.72 mrem/hr at any location in the space station.

(b) A fly-by dose rate of 200 mrem/hr at a distance of 30 meters from the reactor.

The dose rate at the space station is actually one-half the total allowable rate of 35 rem/quarter, with the remaining exposure assumed to occur from natural terrestrial and extra-terrestrial sources.

7.3.3 Shield Configuration

The basic shield geometries considered in this study are shown in Figure 67. The designations used to describe the shield geometries indicate the solid angle of shielded volume as measured from the reactor midplane. The four-pi configuration would be used if the power system were within or close to inhabited areas of the station. The thickness of four-pi shields usually allows normal human activity immediately adjacent to the shield (similar to submarine reactors).

Two-pi and conical shields protect only limited areas in a given direction from the reactor. Their principal advantage is obviously the much lighter shield masses. They would be used only at long distances from the space station and in regions remote from approach routes to the station. The main use of these partial shield configurations is for unmanned power system applications.

The shaped four-pi configuration saves shield weight in those applications in which complete radiation protection is required for a limited area only but some less stringent protection is needed over all other areas. A prime example of such an application is a boom-mounted location of a reactor power system on the space base. The thick, base section of the shaped four-pi shield protects the space station proper while the thin side and end sections of the shield screen the EVA work areas and shuttle approach paths to the station.

Figure 73 illustrates schematically a divided shield configuration concept designed to minimize radiation from primary loop activation or fission products. Heat exchange equipment in the gallery space between the primary and secondary shield sections transfers thermal energy from the highly radioactive reactor coolant to a secondary coolant loop which then transports the energy to the power conversion equipment. Induced radioactivity in the secondary coolant loop is sufficiently reduced below that of the primary loop to permit limited maintenance activity on the power conversion equipment. The split shield arrangement also provides a potential method of replacing the reactor by separating it from the rest of the system at the heat exchanger in the gallery location.

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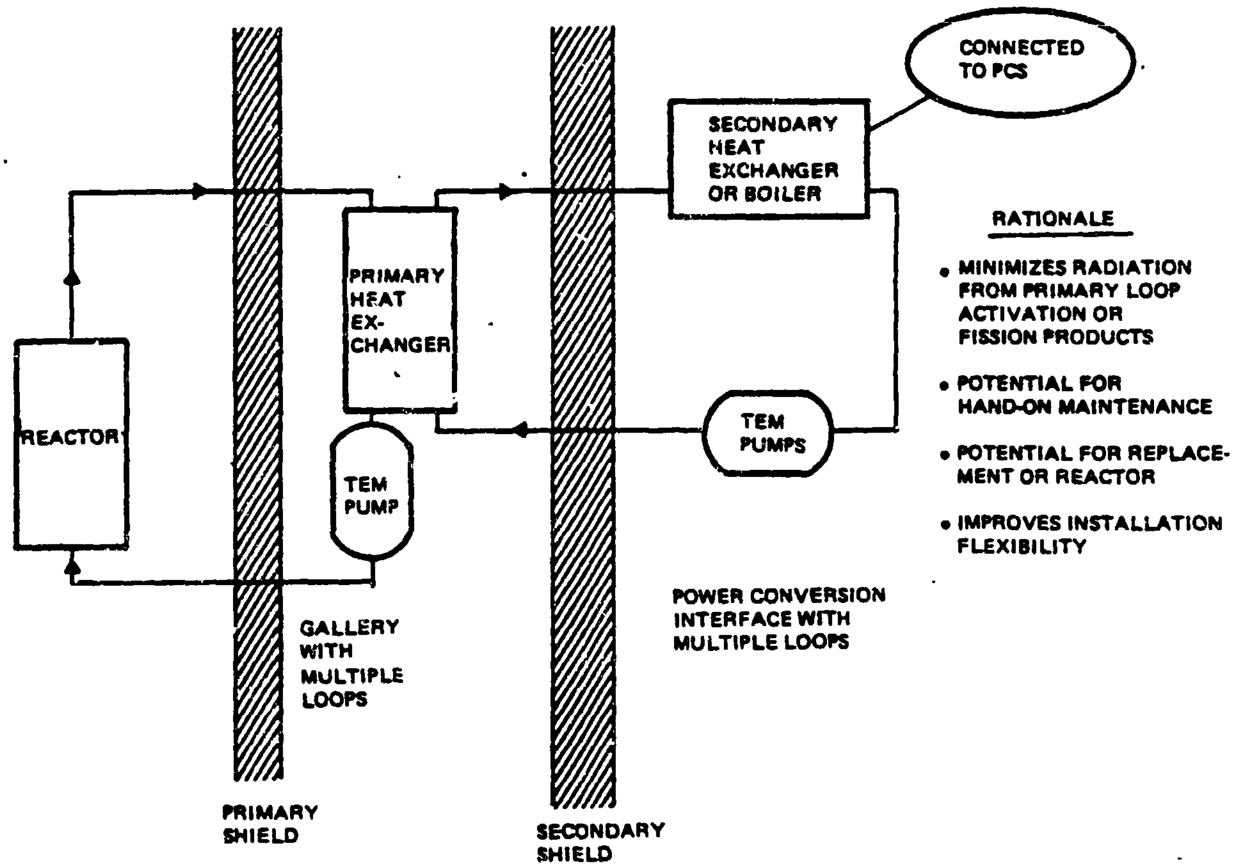
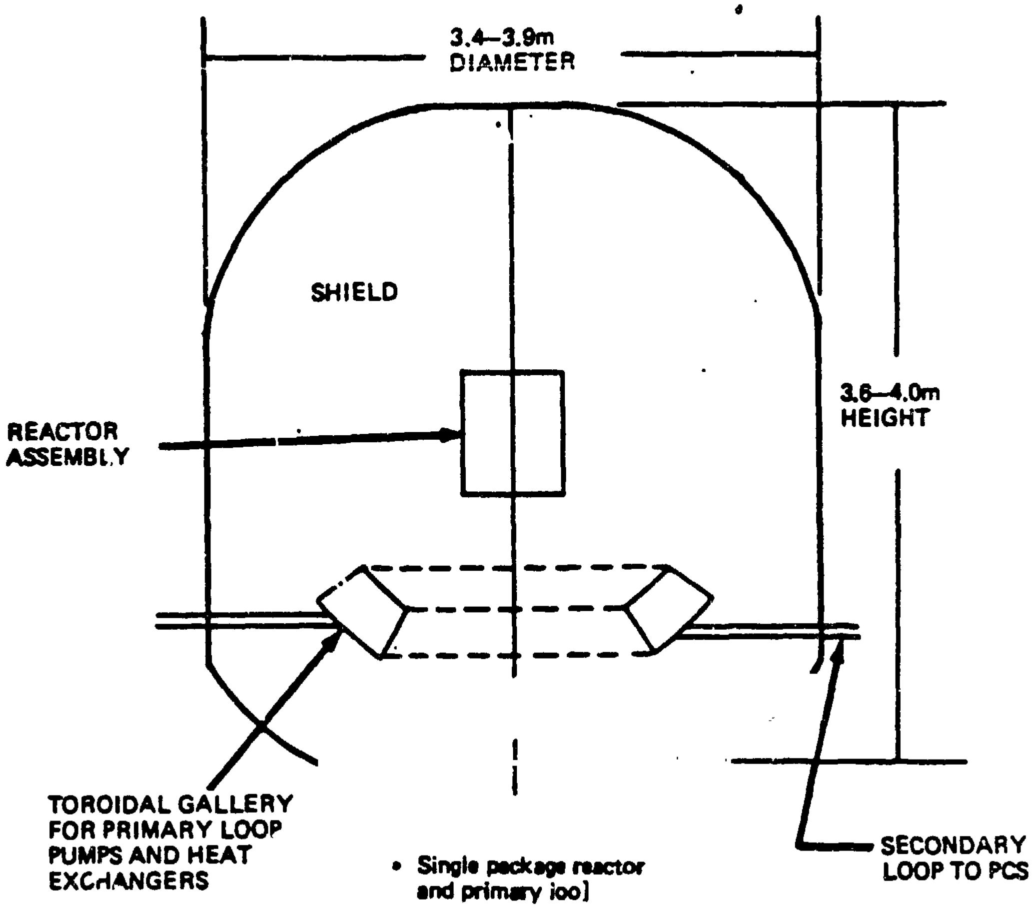


Figure 73 Divided Shield Concept for Manned Applications

Three shield layouts that demonstrate the use of the divided shield concept are presented on Figures 74, 75, and 76. A design for an on-board power system, similar to a nuclear submarine installation, is shown in Figure 74. The reactor and primary cooling loop are permanently encased within a four-pi shield structure which is sized to permit normal activity in the immediate vicinity of the power system. In the strict sense, the shield is not divided but rather is a single unit with an internal void space used as the heat exchange gallery. The reactor-shield assembly is integral and inseparable from the space station structure.

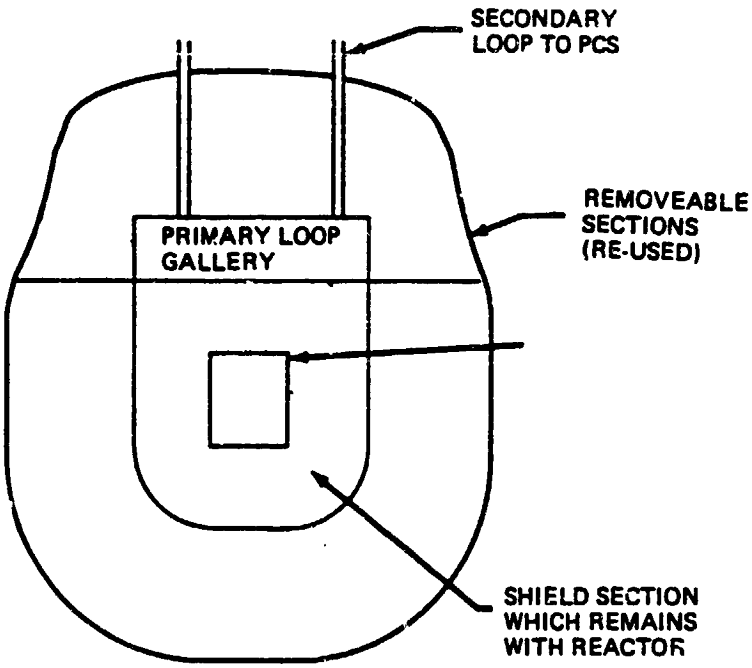
A modification of the station integrated arrangement is the replaceable reactor configuration shown in Figure 75. The shield is split into two basic sections, one which always is attached to the reactor and one which stays with the space station. The station section may be further divided into two or more parts to facilitate the reactor removal and/or power conversion equipment replacement.

A split shield layout employed with a shaped four-pi shield, shown in Figure 76, is illustrative of a boom-mounted power system in which the power conversion and/or the reactor subsystem could be replaced, if so designed at the outset.



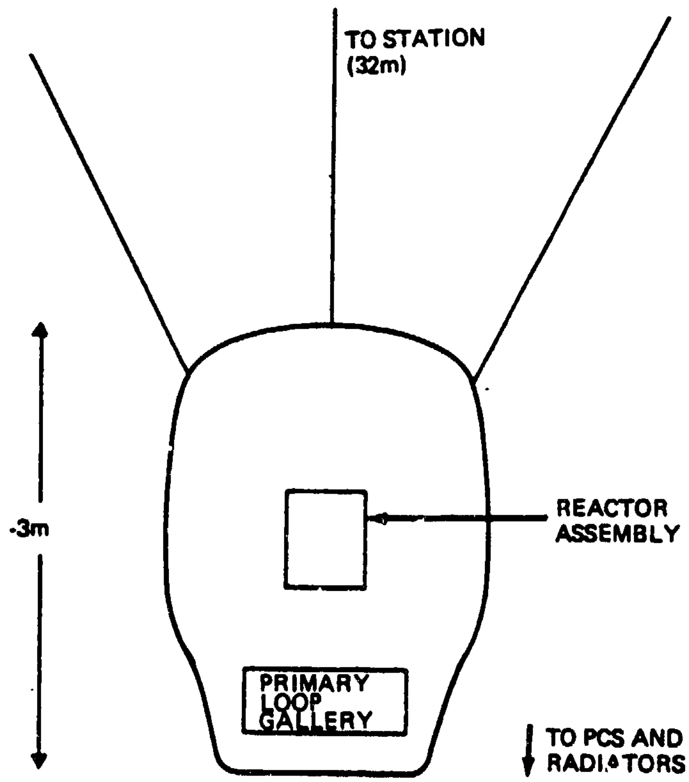
- Single package reactor and primary loo]
- Shield consists of layers of LiH and Tungsten
- Sized for 5.72 mrem/hr on/or near surface
- PCS location is flexible

Figure 74 Shield Layout, Four-Pi Integrated On-Board System



- Multiple package reactor shield and primary loop
- Reactor can be removed and replaced
- Primary loop gallery may be serviceable
- Fixed shield and top sections stay with station
- PCS can optionally be replaced with reactor

Figure 75 Shield Layout, Four-Pi Replaceable On-Board System



- Separation distance—32m
- Cone half-angle—31°
- 4-π dose rate 200 mrem/hr @ 30m
- PCS and radiators located away from station
- Design can be made modular

Figure 76 Shield Layout, Four-Pi Shaped Shield

7.4 Parametric Data

Power system mass estimates, including the reactor, shield and power conversion components but not the heat rejection radiator, were determined as a function of separation distance from the space station for a set of parameters which included shield geometry, output power level and type of power conversion. This section will present cross plots of the parametric data to summarize and illustrate interesting conclusions.

Figure 77 shows the variation in mass of an on-board power system having a four-pi shield geometry when the required dose rate of 5.7 mrem/hr is assumed to occur at a distance from the reactor designated by the abscissa value of the figure. The power conversion system is assumed to be a dynamic system. The curves show a total system mass of almost fifty thousand kilograms for a dose design point distance of two meters, and a mass of about thirty thousand kilograms when the dose design point is moved twenty meters from the reactor. Thus, significant reductions in power system mass can be achieved if an exclusion zone is established around an on-board system. The data of Figure 77 also show that an almost constant difference of five thousand kilograms exists between 100 kWe and 300 kWe power systems regardless of separation distance.

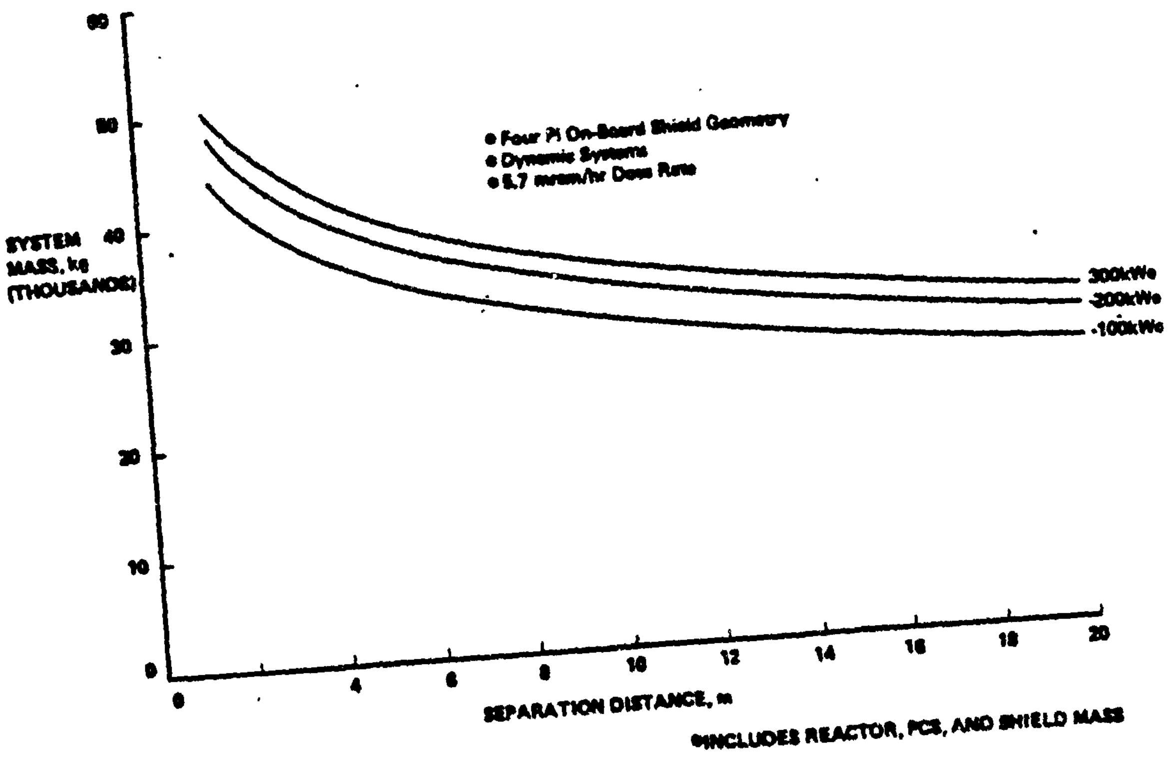


Figure 77 Effect of Output Power and Reactor Distance on Power System Mass

The variation of power system mass with type of power conversion system and at constant output power is illustrated in Figure 78. The trends and weights of the power system mass are similar to those shown in the previous figure. Power systems with dynamic power conversion are the lightest while those with baseline thermoelectric conversion are the heaviest. A difference of four or five thousand kilograms exists between systems of the two power conversion types independent of separation distance.

The data of Figure 79 are similar to that of Figure 78 except they apply to a shaped four-pi shield instead of an on-board four-pi geometry. The analysis was extended to a much longer separation distance because the shaped four-pi configuration is mainly applicable at the longer distances. The results of Figure 79 show that beyond twenty meters, separation distance has negligible effect on power system mass. The results also mirror those of Figure 78 in that power systems employing baseline thermoelectrics are about four to five thousand kilograms heavier than dynamic power conversion systems.

An example of the effect of shield geometry on power system mass is presented in Figure 80. In contrast to the previous figures, a very large difference in mass is apparent between the three curves of the figure. For separation distances greater than twenty meters, a conical shield is significantly lighter than the two-pi configuration which, in turn, is very much lighter than the shaped four-pi shield. The design dose rate for each of the curves of Figure 80 is 5.7 mrem/hr at the space station. However, the shaped four-pi geometry has side and end shielding that allows flyby operations within thirty meters of the reactor while the complete lack of side shielding in the other two geometries precludes such flyby approaches. Thus, practical limitations on the use of the conical and two-pi geometries offset their respective mass advantages.

7.5 Specific Electrical Power Systems

Based on the results of the parametric shield mass calculations and subsequent spacraft tradeoff studies, three configurations of electric power system-space station integration were identified for continued evaluation:

(a) A boom-mounted power system located 70 meters from the station

(b) Power systems tethered to but operating at a distance of about 30 kilometers from the station

(c) A free-flying power system operating in an orbit 700 kilometers higher than the space station

For each of these configurations, a baseline power requirement of 150 kWe net to the station was established, assumed to be provided by the baseline thermoelectric power conversion cycle. A duplicate set of evaluations was performed for each of the power system-space station configurations assuming a net power requirement of 300 kWe supplied by an advanced dynamic conversion cycle. Discussions for all of the configurations are contained in the succeeding paragraphs.

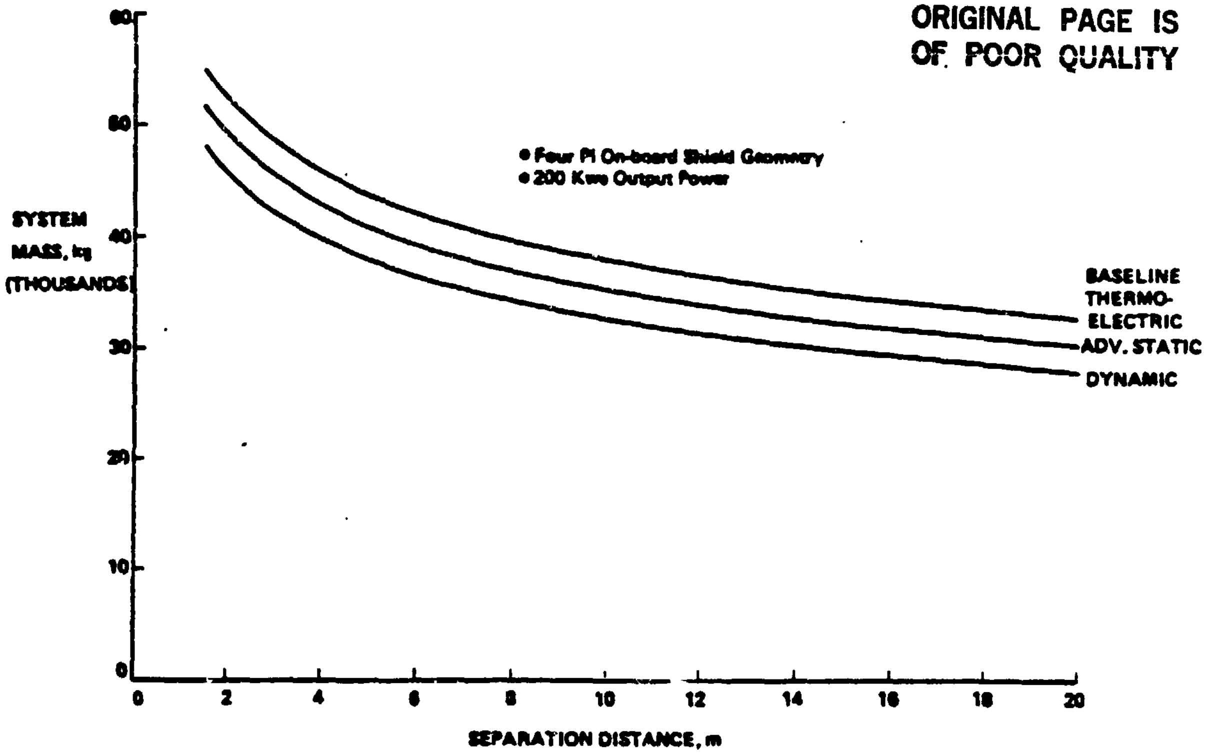


Figure 78 Effect of Power Conversion Type on Power System Mass with a Four-Pi Shield

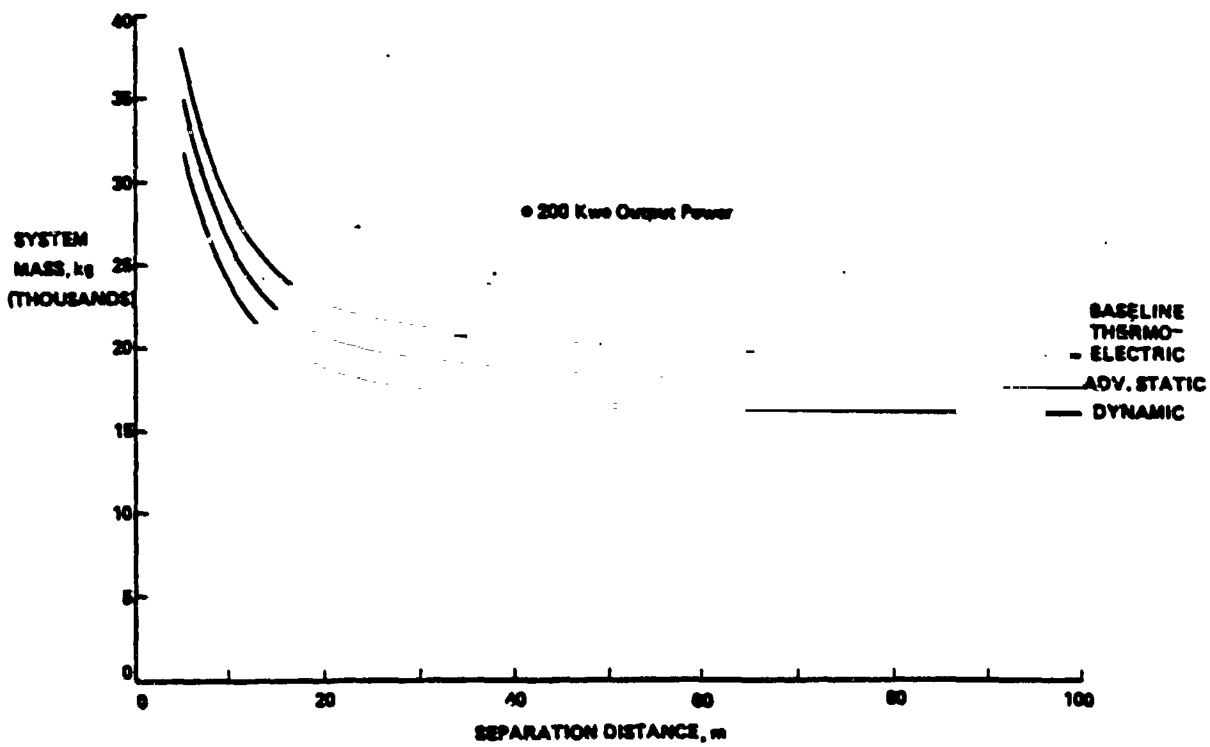


Figure 79 Comparison of Power System Mass for Various Power Conversion Types with a Shaped Four-Pi Shield

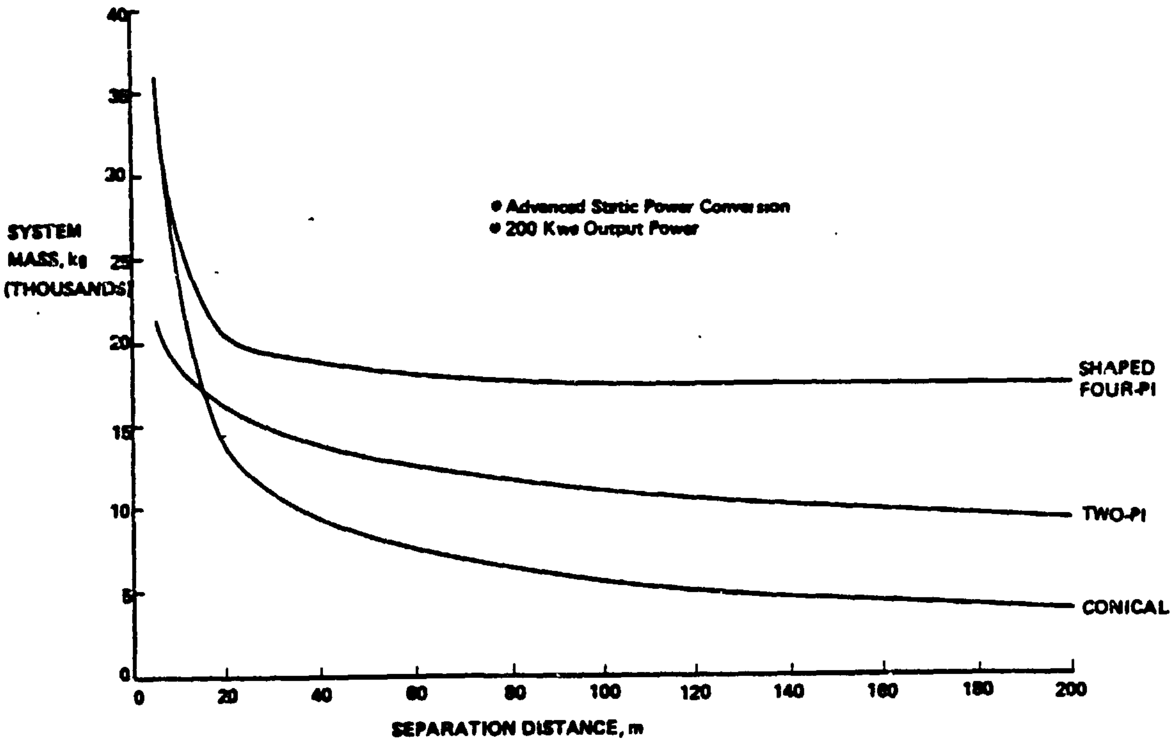


Figure 80 Effect of Shield Geometry on Power System Mass

7.5.1 Boom-Mounted Configuration

A layout of the 150 kWe boom-mounted reactor power system is presented in Figure 81. In this configuration, the electrical power is delivered to the space station via aluminum cables. The 70 meter boom length was established by determining a minimum overall weight of reactor power system plus cabling as a function of boom length. With shorter booms, the increase in shield weight would be greater than the weight savings in the conducting cable, with reverse conditions occurring with booms longer than 70 meters.

As shown in Figure 81, a shield half angle of 22.4° protects the space station at the 70 meter separation distance. The shield thickness within this angle is sized to achieve the 5.7 mrem/hr design dose rate in the station. The shaped four-pi shield geometry is slightly thinner on the sides and on the end away from the station to meet the flyby dose rate of 200 mrem/hr. The primary and secondary loop bays are located on the far side of the shield from the station in separate shield sections which, in theory, can be disconnected to provide for replacement of either the reactor assembly or the power conversion unit. The latter component, though not shown in Figure 81, is physically attached to the shield section containing the secondary loop bay.

The shield mass for the layout shown on Figure 81 is estimated to be 18,000 kilograms with the mass of the reactor plus power conversion an additional 1,650 kilograms. A summary of the boom-mounted system characteristics, as well as the characteristics of the other configurations, are listed in Figure 82.

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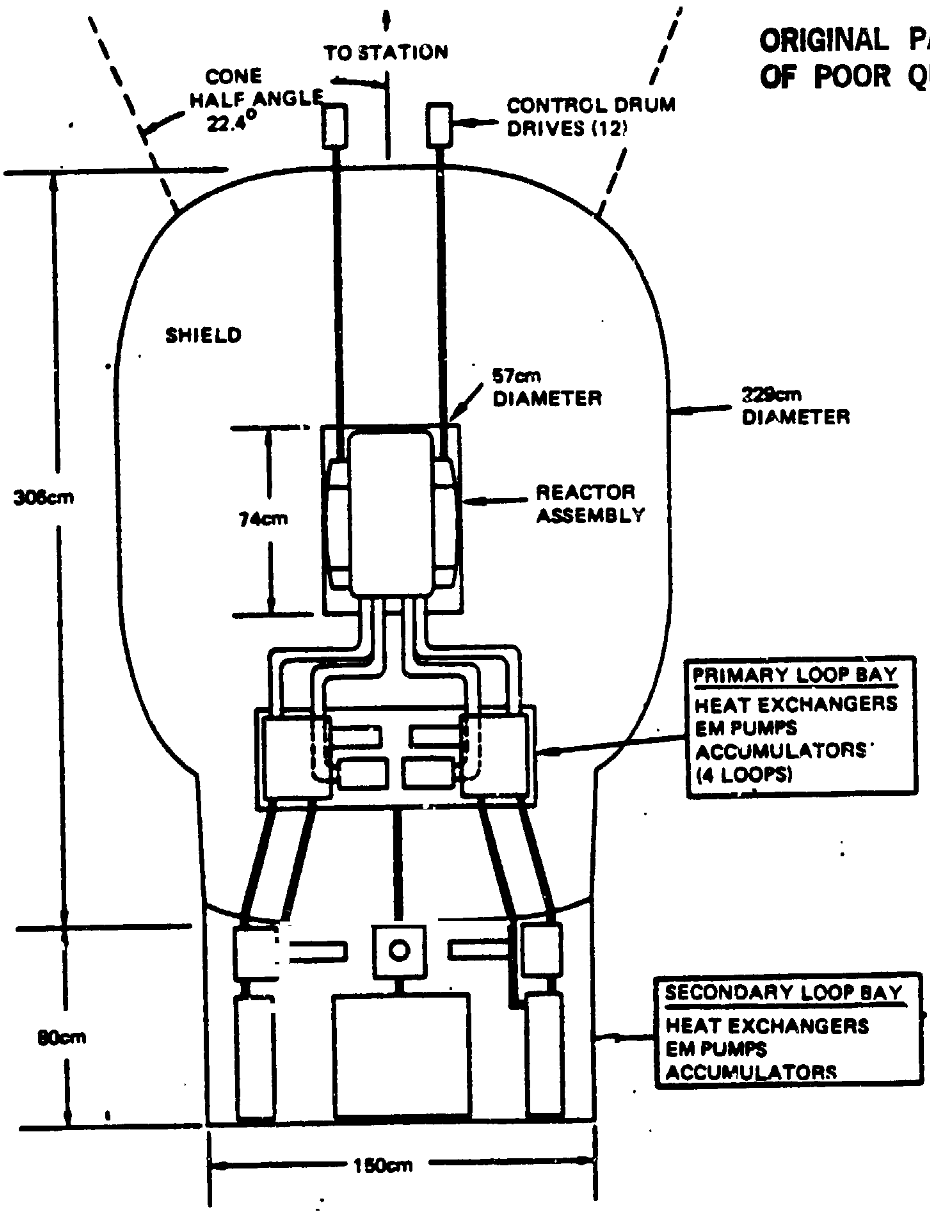

Figure 81 Boom-Mounted Configuration

Figure 82 *Characteristics of Selected Power System Configurations*

ITEM	BOOM MOUNTED SYSTEM	TETHERED SYSTEMS (MAN RATED)	TETHERED SYSTEMS (INSTRUMENT RATED)	FREE-FLYER SYSTEM
METHOD OF POWER TRANSMISSION	METAL CONDUCTORS	ELECTROLYSIS PLANT	ELECTROLYSIS PLANT	ELECTROLYSIS PLANT
PCS TYPE	BASELINE T/E	BASELINE T/E	BASELINE T/E	BASELINE T/E
ELECTRIC POWER REQ. NET TO STATION	150 kWe	150 kWe	150 kWe	150 kWe
NET PCS OUTPUT	153 kWe	200 kWe	200 kWe	300 kWe
SEPARATION DISTANCE (TO STATION OR PAYLOAD	70 m	30 km	25 m	25 m
SHIELD TYPE	SHAPED FOUR-PI (MAN RATED)	FOUR-PI (MAN RATED)*	CONICAL (INSTRUMENT RATED)	CONICAL (INSTRUMENT RATED)
DOSE RATES USED AT STATION OR PAYLOAD	5.7 mrem/hr	5.7 mrem/hr	14.4 rem/hr	14.4 rem/hr
FLY-BY	200 mrem/hr @ 30 m	200 mrem/hr @ 30 m	UNSPECIFIED	UNSPECIFIED
MASS (REACTOR & PCS)	1050 kg	2100 kg	2100 kg	3700 kg
VOLUME (REACTOR & PCS)	2.1 m^3	2.1 m^3	1.6 m^3	2.8 m^3
MASS (SHIELD)	10000 kg	18700 kg*	890 kg	2300 kg
VOLUME (SHIELD)	7.1 m^3	7.4 m^3	0.5 m^3	1.3 m^3
SYSTEM DRAG AREA	~0.1 m^2	~0.1 m^2	1.8 m^2	2.8 m^2

*MAN-RATED FOR MANNED MAINTENANCE AFTER REACTOR SHUTDOWN AND FLY-BY

7.5.2 Tethered Configuration

The layouts of two distinctly different tethered configurations are shown in Figures 83 and 84; a man-rated system in the former figure and an instrument-rated system in the latter. In both systems, the electrical power generated is used in an electrolysis plant to decompose water into its hydrogen and oxygen components. The gases are then pumped through hoses which are grouped together to form the umbilical tether between the reactor power system and the station. At the station, the hydrogen and oxygen are recombined in the fuel cell to generate electrical power when needed, with the reformed water returned to the power system electrolyzer through tether ducts to complete the cycle. The power system is held by the tether under tension approximately 30 kilometers distance away from the space station in a higher altitude.

The man-rated tethered configuration shown in Figure 83 has a four-pi shield geometry that is sized to allow flyby approaches during power system operation, and manned maintenance in the power conversion unit and electrolysis plant after reactor shutdown. The primary and secondary loop design in corresponding shield sections is identical in function to the boom-mounted design of Figure 81. The tethered systems must generate higher gross power outputs than the boom-mounted system because of the power required to run the electrolysis plant. This higher gross power is reflected in the slightly heavier masses for the shield and power conversion components as shown in Figure 82.

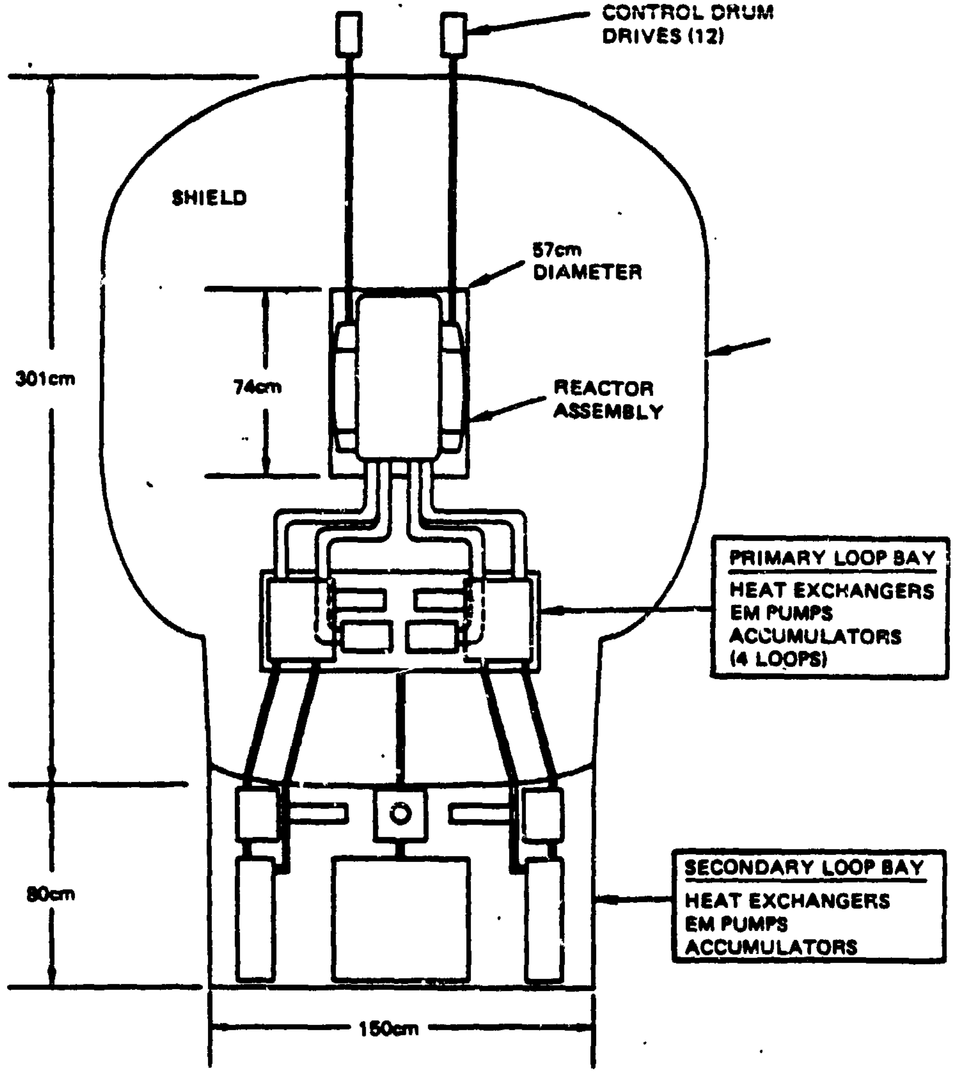

Figure 83 Tethered Configuration (Man Rated)

The instrument-rated tethered configuration has a shield mass that is only five percent of the mass of the man-rated configuration (see Figure 82). This is due to its conical shield geometry with a cone half angle of only 17°, as shown in Figure 84, and a much higher allowable dose rate. This type of shadow shield precludes manned approach to the power system so the dose rate is determined by tolerances of instrumentation located approximately twenty-five meters away from the shield. Thus, a very substantial savings in power system mass is realized for the disadvantage of excluding manned maintenance of the power converter and electrolysis plant.

7.5.3 Free-Flyer Configuration

The free-flyer configuration is similar in shape and concept to the instrument-rated tethered system in that a conical shadow shield geometry is used to protect an unmanned power conversion system and electrolysis plant. A much larger electrolysis plant, hence more electrical power, is required in the free-flying configuration to provide propulsive fuel for a tanker which transports liquified oxygen and hydrogen fuels from the power system to the space station. The required electrical power of 360 kWe exceeds the capacity of a power system using a single 4 megawatt thermal reactor with thermoelectric conversion. Consequently, two of these 4 MWt reactors, placed end-to-end in tandem arrangement, were assumed as the nuclear energy source for the system, even though a redesigned, single reactor of greater thermal energy output might result in a smaller mass for the overall system. The conical shield for the free-flying configuration is considerably larger than the conical shield of the tethered configuration because of the two reactor arrangement, the higher total reactor power level and the larger area payload (power conversion subsystem plus electrolysis/liquefaction plant) to be protected.

7.5.4 300 kWe Dynamic Systems

The configuration evaluations described in the three preceding report sections for 150 kWe net power output to the space station were repeated for assumed net power output requirements of 300 kWe. To produce this quantity of power it was considered necessary to assume the use of more efficient advanced dynamic power conversion as represented by Stirling engine systems. Layouts for the boom-mounted, tethered, and free-flying configurations were not prepared because they would be very similar to those shown in Figures 81, 83, and 84. The main differences would be the substitution of four Stirling engine components for the heat exchangers in the secondary loop bays of Figures 81 and 83, and slightly smaller shields due to higher conversion efficiencies.

A summary of the Stirling engine system characteristics is presented in Figure 85. Comparison of these characteristics with those of Figure 82 show slightly lighter shield masses for the Stirling engine configurations even though the net power to the station is doubled.

7.6 System Design and Operational Considerations

Some design and operational aspects of the reactor power system were examined in a preliminary manner for this study. The areas considered include startup and shutdown operations, maintenance and repair possibilities, safety aspects of the design and operation, and reliability/lifetime concerns. The following paragraphs discuss these various considerations.

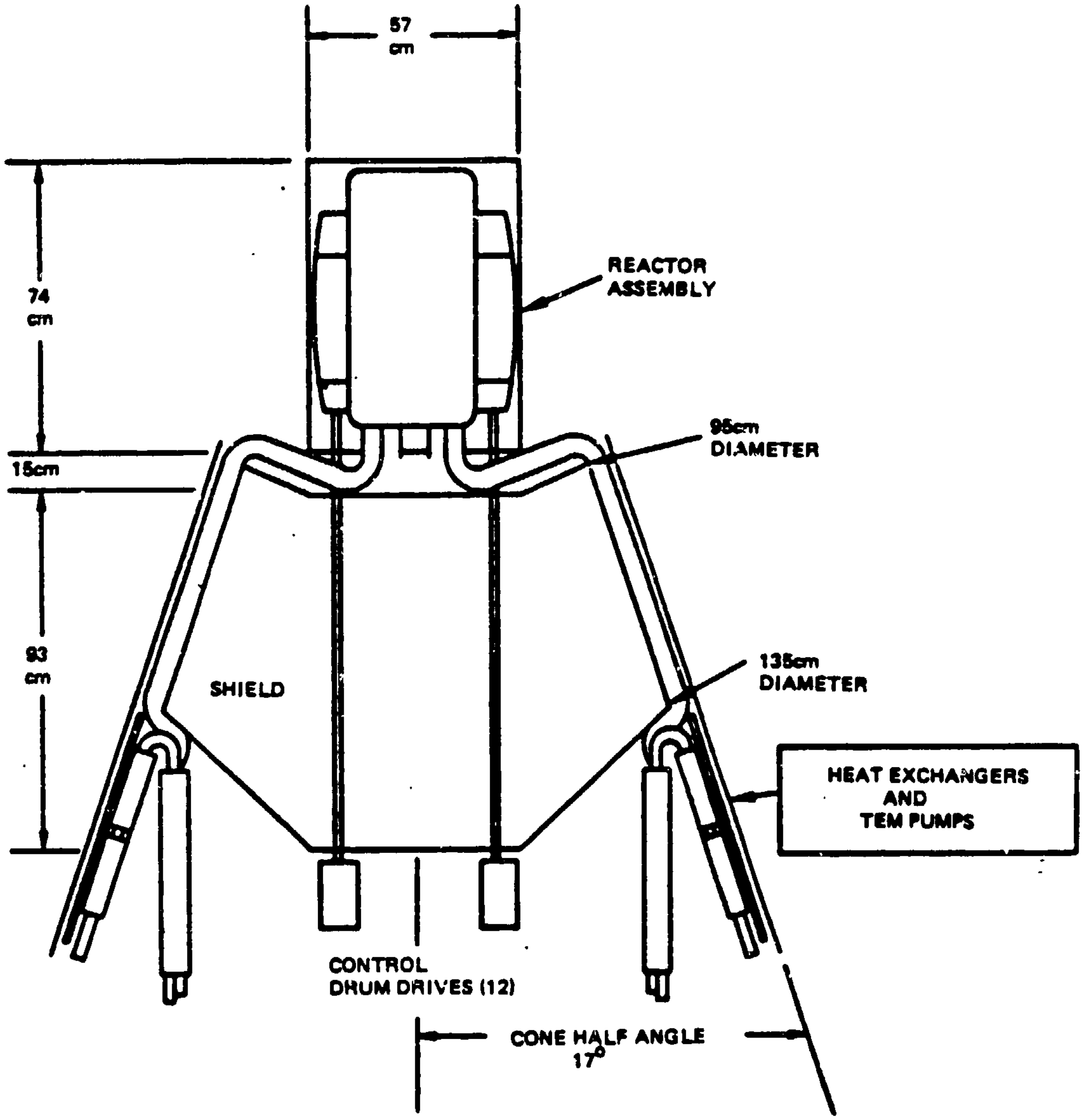

Figure 84 Tethered Configuration (Instrument Rated)

Figure 85 *Characteristics of Selected Power System Configurations Using Higher Power Stirling Engine Conversion*

ITEM	BOOM MOUNTED SYSTEM	TETHERED SYSTEMS (MAN RATED)	TETHERED SYSTEMS (INSTRUMENT RATED)	FREE-FLYER SYSTEM
METHOD OF POWER TRANSMISSION	METAL CONDUCTORS	ELECTROLYSIS PLANT	ELECTROLYSIS PLANT	ELECTROLYSIS PLANT
PCS TYPE	STIRLING ENGINE	STIRLING ENGINE	STIRLING ENGINE	STIRLING ENGINE
ELECTRIC POWER REQ. NET TO STATION NET PCS OUTPUT	 300 kWe 306 kWe	 300 kWe 416 kWe	 300 kWe 416 kWe	 300 kWe 720 kWe
SEPARATION DISTANCE (TO STATION OR PAYLOAD	55 m	30 km	25 m	25 m
SHIELD TYPE	SHAPED FOUR-PI (MAN RATED)	FOUR-PI (MAN RATED)	CONICAL (INSTRUMENT RATED)	CONICAL (INSTRUMENT RATED)
DOSE RATES USED AT STATION OR PAYLOAD FLY-BY	5.7 mrem/hr 200 mrem/hr @ 30 m	5.7 mrem/hr 200 mrem/hr @ 30 m	14.4 rem/hr UNSPECIFIED	14.4 rem/hr UNSPECIFIED
MASS (REACTOR & PCS)	2800 kg	2900 kg	2900 kg	3500 kg
MASS (SHIELD)	18600 kg	17100 kg	5 610 kg	780 kg

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7.6.1 System Startup, Shutdown, and Disposal

Procedures for the startup, shutdown, and eventual disposal of the reactor power system are summarized in Figure 86. A prerequisite for safe, successful initiation of reactor operation in orbit are a number of prelaunch operations which will be powered from the previously existing power system in the station - the energy storage from the solar photovoltaic power system or a separate energy storage source. These include a check of the nuclear criticality of the reactor measured at essentially zero power (approximately 1 Watt thermal) output, and a calibration of the nuclear worth of each of the control drums. The first check assures thermal power generation will be achieved, and the second check assures that control margin exists for all credible orbit operations and calibrates control system trim for automatic startup.

The power system is launched and mated with the space station in a cold, inert state with control drums mechanically locked in the shutdown position. After a complete checkout of the electrical power generation circuitry and the power system control circuitry, the reactor control drums are unlatched and the nuclear reaction initiated. The reaction rate is held at low power and temperature until the lithium coolant in the primary and secondary loops is melted. The reactor is then slowly ramped in power to achieve near operating temperatures, lithium coolant flow is established in both the primary and secondary loops, and the reactor power increased to operational levels. In the event of a restart, the procedure will have to be reprogrammed since the original conditions for a new reactor will not be present. It is presently envisioned that the power system will operate continuously at full output level with any power in excess of the station demand being radiated to space via a shunt resistance. This mode of operation eliminates thermal transients in the power system and any attendant effects on system life and reliability.

Figure 86 *Startup and Shutdown Considerations for Boom and Tethered Configurations*

- **Startup Sequence**
  - o Complete ground checkout including:
    - Zero-power criticality
    - Control drum calibration
  - Mated to station in a cold, shutdown condition
  - Complete electrical and control system checkout
  - Ramp to operating temperature at low power
  - Increase to operating power at nearly constant temperature
  - Operate at constant reactor power using shunt load
- **Shutdown**
  - Normal – controlled shutdown
  - Scram – emergency shutdown
- **Reactor Removal & Transfer to a Safe Orbit**
  - Shutdown reactor
  - Allow to cool
  - Break connections at secondary loop
  - Attach aftercooling equipment
  - Attach transfer booster and boost to long-lived orbit

Two shutdown procedures are shown on Figure 86: a normal, controlled shutdown, and an emergency "scram" shutdown. The latter procedure would be available for man-rated configurations to deal with potentially catastrophic accidents. The "scram" procedure might be an irreversible action that would preclude subsequent restart of the power system.

Disposal of the reactor at the end of its useful life would be accomplished by the sequence listed in Figure 86. The reactor would be shut down and allowed to cool both thermally and radioactively. If the power conversion subsystem were to be reused, it would be separated from the reactor at the secondary loop bay and replaced with a temporary aftercooling subsystem, if aftercooling is still needed. A booster vehicle would be attached to the primary shield section containing the reactor and the reactor disconnected from the station. The reactor section would then be boosted to a safe orbit and abandoned.

7.6.2 Maintenance and Repair Considerations

Instrument-rated shield geometry systems employing shadow type shields of limited solid angle must be designed for long life without planned maintenance or repair because their nuclear environment precludes manned activity in the immediate vicinity of the system. However, man-rated systems with some form of four-pi shielding geometry offers the potential for limited maintenance and repair activities. The specific shield arrangements shown in Sections 7.5.1 and 7.5.2 contain extra gamma shielding which would allow limited contact maintenance or repair operations on the secondary heat transfer loop components and power conversion modules after a system shutdown. Twelve days after a reactor shutdown, the dose rate at the secondary loop bay would be approximately 100 mrem/hr, which would allow a person to work at that location for 2 to 3 hours per day. Approximately seventy-three days after system shutdown, the dose rate would decrease to 30 mrem/hr at that location which would permit an 8-hour working period per day. The use of heavier gamma shield sections would allow the maintenance/repair activities to commence sooner or extend for longer work periods per day. Considerably more detailed evaluations are needed to determine the usefulness of limited maintenance and repair and the extent of those activities. Of prime importance is the effect that maintainability would have on the overall reliability of the power system. This evaluation would have to be made in conjunction with parallel studies to identify practical maintenance activities and component candidates for periodic repair or replacement.

7.6.3 Safety Aspects

The safety aspects of any nuclear reactor powered system are always of prime concern, and the space application of that type of system creates a number of potentially hazardous conditions not present in ground based applications. A number of those conditions and potential solutions have been identified but considerably more analysis and development will be required. Some of the areas of concern that have been identified are:

1) Nuclear subcriticality during postulated accidents.
2) Reactor core dispersion upon re-entry into the earth's atmosphere.
3) Containment of liquid metals during system operations.
4) Safe control during operations.
5) Achievement of safe disposal of the reactor.
6) Provision for safe haven power if reactor power system fails.

Achieving and/or maintaining nuclear subcriticality following accidents during prelaunch, launch, and operational phases of the mission is an essential safety consideration. The control drums of the reactor will be mechanically locked in the shutdown position for all operations leading to reactor startup in orbit. Additional control system interlocks will be designed into the system for safety purposes. In the event of a launch abort that results in the water immersion of the reactor, a separate removable safety plug of nuclear "poison" material may be built into the reactor to offset the positive nuclear effect of the water on the reactor criticality. Also, a "scram" capability may be designed into the control drum actuators, or into separate scram control rods, to quickly shut down an operating reactor located o the space station in the event of an accident. For any safety operations, the powe will be provided from an emergency power system.

A safety condition that must be addressed is the accidental re-entry of a nuclearly "hot" reactor into the atmosphere. Present ruling requires that the reactor be passively disassembled into small discrete sections that burn up in the atmosphere before impact on the surface of the earth. With a four-pi shield geometry enveloping the reactor, removal of the shield must be achieved before the reactor can be dispersed. Analyses will have to be performed to determine if the shield removal will occur naturally in its passage through the atmosphere or if separation capability must be designed into the component. A design can be made in which the shield will separate from the reactor.

Before operation of the power system, the lithium coolant is cold and solid. After operation has been achieved, the molten lithium must be contained in the event of a break in the coolant circuits, because of its very corrosive capability and its radioactivity in the reactor loop. In man-rated four-pi shield geometries, the enveloping configuration of the shield could provide a secondary containment structure for the primary cooling loop, if so designed. However, the secondary coolant loop penetrates the shield volume, thus a fracture of this loop would release liquid metal to the surroundings. Design studies are required to determine the necessity, and methods, of liquid metal containment from normally exposed coolant circuits.

Safe operation of the reactor power system will require a fail-resistant control system. This will probably be accomplished by fully redundant electric and electronic control circuitry, a built-in capability for manual override of the controls, and an independent, uninterruptible power supply for all components of the control system. Such an independent power supply can be an advanced high energy density lithium battery.

7.6.4 Reliability and Lifetime Considerations

The first space reactor power system development is expected to be the SP-100 application which has an eventual reliability requirement of 95% for seven years of full power operation. It will have a design life requirement of ten years and a full power lifetime requirement of seven years. It is assumed that the SP-100 will meet its life and reliability goals and that a man-rated reactor power system for the space station application will be based on the SP-100 design. At the present time there is no reason to expect significantly shorter life or lower reliability for the manned space station version of the system. However, there are some differences in the design and application of the space station version that will have to be evaluated. They include:

1) The reliability of additional safety features included in the man-rated power system and their effect on overall system reliability

2) The effect on overall system reliability of provisions made in the design to provide maintainability, if present

3) A free-flying configuration for the reactor power system which would have an instrument-rated conical shield geometry, but with provision for some flyby activity by manned vehicles.

The system mass of man-rated reactor power systems is dominated by the shield component, and the bulk of that shield mass is needed to provide flyby capability in any direction from the reactor. The shield mass for a man-rated system is approximately twenty metric tonnes, which provides possible capability for limited contact maintenance on particular components of the power system.

The shield mass of the free-flying power system configuration is only three metric tonnes, a very significant reduction compared to the man-rated systems of the other two configurations. This relatively small mass accrues from the use of an instrument-rated, conical shield geometry in the free-flying arrangement.

8.0 SPACE STATION TRADE STUDIES

Of fourteen space station-reactor configuration options assessed, three were selected on first order merit for further quantitative evaluation. These three include one reactor which is rigidly attached to the space station, one which is attached to a long tether, and one which is a free-flyer. Conceptual designs were developed for each configuration to identify the major elements and to enable a quantitative estimate of mass, volume, and orbital drag area. A comparison of configurations on the basis of mass and volume in orbit, and STS logistics was made. The comparison included initial operations and life cycle logistics. Power levels considered were 150 kWe and 300 kWe net power to the space station.

8.1 System Selection Recapitulation

8.1.1 Configurations

Three configuration options were selected for further analysis. These options were: 1) reactor attached to a boom on the space station with electrical power transmitted through conducting cables to the station; 2) reactor and electrolyzer attached to a long tether, with oxygen and hydrogen flowing through the tether to a fuel cell on the space station and water returning to the reactor; and 3) reactor and electrolyzer on a free-flyer in a higher orbit than the space station, with reactants transferred to and from the space station with ar orbital transfer vehicle. The major elements of each option are illustrated in Figure 87 through 89.

The boom-mounted reactor configuration is shown in Figure 87. The reactor is mounted at the end of a boom which is attached to the space station. The power conversion system, shield, and radiators are at the reactor end of the boom. Power transmission from the power conversion system to the space station is achieved with conducting cables within the boom. An OMV or OTV is provided at the space station which is available at any time for boosting the reactor to a high orbit. An emergency power system is also available at the space station to provide critical power to the station during power failures.

Figure 88 shows a tethered reactor system in which the reactor is at one end of a long, flexible tether and the space station is at the opposite end. The power conversion system, shield, and radiators are at the reactor end of the tether. Power transmission to the space station is achieved by moving gaseous hydrogen and oxygen, produced by the electrolyzer, to tanks at the reactor. From the tank storage, the gases are pumped through hoses in the tether to holding tanks at the station. Electrical power is produced at the space station with fuel cells. The water produced by the fule cell is then returned to the electrolyzer through hoses in the tether. At the space station a small vehicle is provided for boosting the reactor to a high, long-lived orbit.

The free-flyer reactor system is shown in Figure 89. The reactor, power conversion system, shield, and radiators are on a free-flying spacecraft in a long-lived orbit, while the space station is at a lower orbit. Water is electrolyzed and the hydrogen and oxygen are liquefied at the reactor to minimize transportion volumes. Electricity is produced at the space station with fuel cells. An orbital transfer vehicle is maintained in a hangar at the space station. When the free-flyer and station orbits are in phase, the OTV leaves the space station with a full charge of water plus sufficient H_2-O_2 propellant for a round trip, carries the water to the reactor, and returns to the station with a full charge of separated fuel cell

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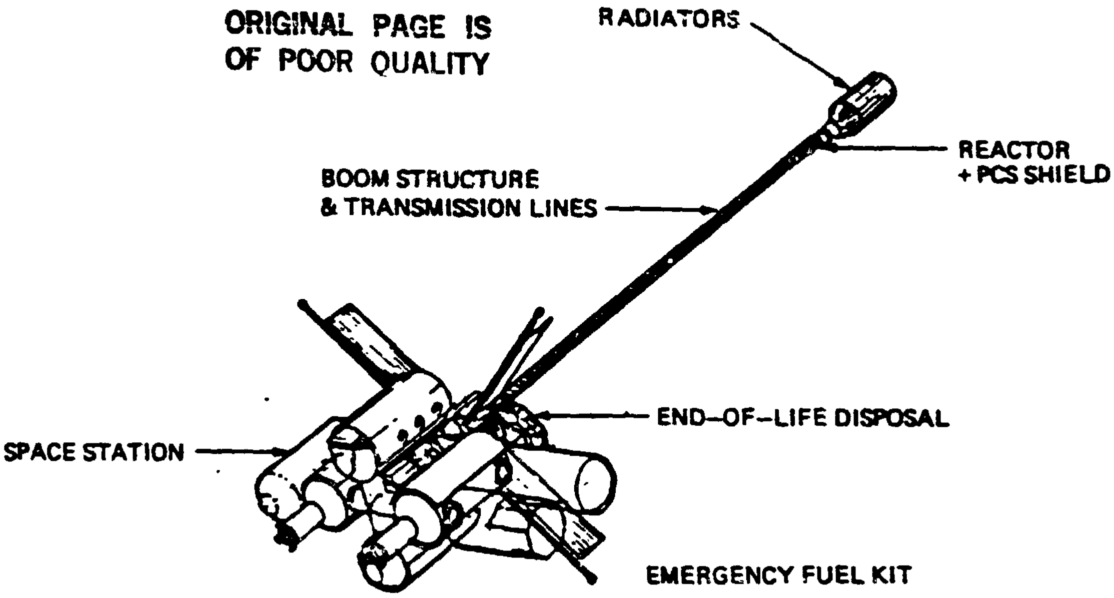

Figure 87 *Boom-Mounted Reactor System Elements*

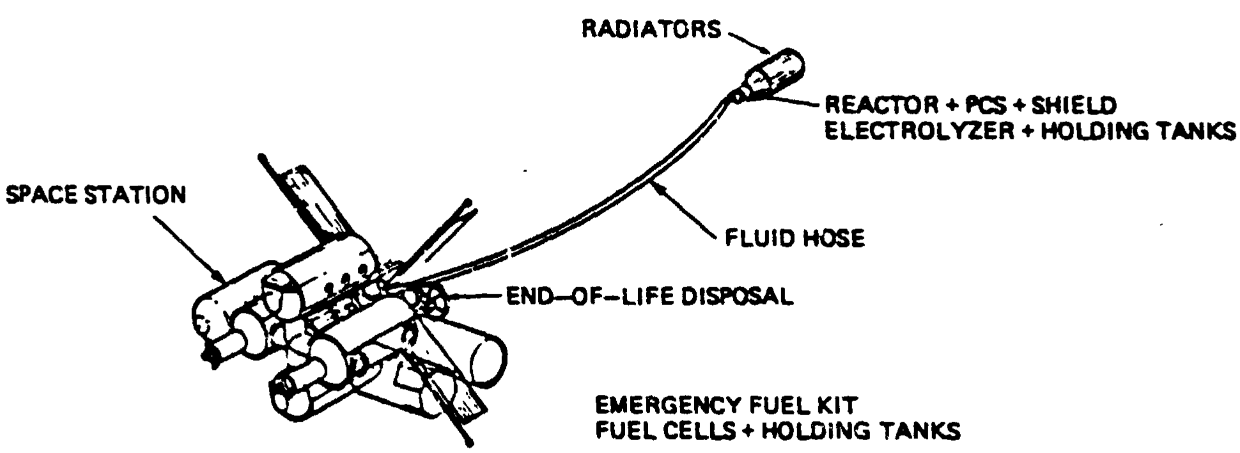

Figure 88 *Tethered Reactor System Elements*

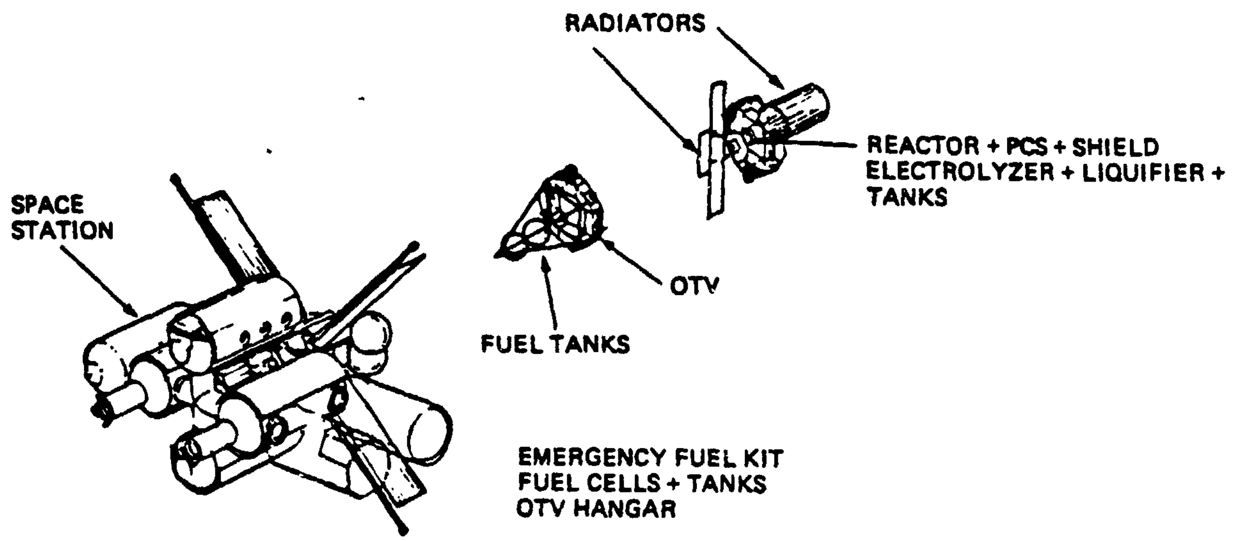

Figure 89 Free Flyer Reactor System Elements

reactants. The reactants are stored as liquid oxygen and liquid hydrogen, so that the free-flyer also has a liquefier, a low temperature radiator, and storage tanks. Fuel cells, an emergency power system, and an OTV hangar are stored at the space station. An analysis was made to determine the logistics and the approximate number of shuttle launches required to support this part of the mission.

8.1.2 Net Power to Station

The three configurations were analyzed with the same net electrical power at the space station. Since power losses are different in the three cases, then the electrical output of the reactor is different in each case. For the boom-mounted option, a transmission line was selected with resistive losses of 2% of the net electrical power. Other losses are assumed negligible, so the net electrical output of the reactor is 153 kWe for the boom-mounted case. In the tethered reactor case, an electrolyzer efficiency of 90% and a fuel cell conversion efficiency of 80% results in a required reactor electrical power of 208 kWe. (See sections 8.3 and 8.4 below). The required reactor electrical output for the free-flyer is 360 kWe, as described in Section 8.8.4. The reactor output for 150 kWe net to the station is summarized in Figure 90. Trade studies were also performed for a net power of 300 kWe at the station, with reactor powers doubled over the 150 kWe case.

FIGURE 90: NET REACTOR ELECTRICAL POWER

Configuration	Reactor Output
Boom-mounted	153 kWe
Tether	208 kWe
Free Flyer	360 kWe

8.1.3 Power Conversion System

As described in Section 7.2, several power conversion system options were investigated to determine the interrelationships with the Space Station. For the trade studies at 150 kWe, a baseline thermoelectric system was selected for comparison. This was considered to be the most conservative choice from a size and mass viewpoint. The overall power conversion efficiency using radiation-coupled modules with heat pipe heat transfer was assumed to be 5.1%. Waste heat is rejected at two different temperatures: 97.5% of the heat is rejected at 850 K, while the remaining 2.5% is rejected at 343 K.

A more advanced system was considered for a later space station option. This system used a Stirling cycle heat engine with a net electric power to the space station of 300 kWe. This system was investigated to provide a comparison with higher power and with advanced dynamic conversion systems. The power conversion efficiency was assumed to be 25%, with 96.7% of the waste heat rejected at a temperature of 640 K and 3.3% rejected at 343 K.

8.1.4 Reactor System

The reactor system selected for the trade studies is that described in Section 7.1. It is basically a fast reactor with pin-type fuel elements, refractory alloy cladding and structures, and drum-type reflectors for reactivity control. Heat removal from the reactor core is by multiple pumped liquid metal loops. This design is evolving from the SP-100 project with a design thermal power of 4000 kWt. Due to burnup limitations with this design, the reactor system assessed for the thermoelectric free-flyer was actually two reactors in tandem. This approach was taken to allow a burnup in excess of 40 MW-years for the existing design.

8.1.5 Shield

The shield design was described in Section 7.3. For the boom mounted reactor case, a shaped four-pi shield design was used, with an allowable continuous dose rate at the space station of 12.5 rem per quarter from the reactor alone of 5.7 mrem/hr, and a maximum dose rate of 200 mrem/hr at a distance of 30 meters in any direction from the reactor. The former dose rate represents the allowable continuous dose to people residing on the space station for half a year, while the latter dose rate represents occasional doses of short duration (1-2 hours) during orbiter or EVA maneuvering. The shield design consisted of lithium hydride for neutron shielding, encapsulated in multiple layers of tungsten for gamma shielding.

For the tethered reactor option, two shield types were considered. A man-rated shield was considered to allow manned traffic in the vicinity of the reactor. This was a four pi shield with a dose rate of 200 mrem/hr at 30 meters from the reactor during operation. This shield might also allow limited manned maintenance after reactor shutdown. A conical shield was also considered which is not man-rated. The dose rate for this instrument-rated shield is 14.4 rem/hr at a distance of 25 meters from the reactor, behind the shield. A conical instrument-rated shield was also assessed for the free-flyer case.

8.2 Radiator Parametrics

The reactor radiator sizes were determined parametrically for all the power conversion systems. One-sided cylindrical radiators and two sided flat plate radiators were considered. The results are summarized in Figures 91 and 92. The radiators summarized in these tables are assumed to be "white", meaning the solar spectrum absorptivity is 0.2, and the emissivity is 0.9. This sizing was done for a reference 100 kWe, based on optimization of a simple model to achieve minimum mass. Meteoroid protection is not included in the model, since it is beyond the scope of the contract. The scaling with power levels greater than 100 kWe is assumed to be linear with electrical power. Cylindrical one-sided radiators were treated in the trade studies.

8.3 Fuel Cell Scaling Parameters

STS fuel cell data were used as the basis for fuel cell scaling parameters. Hydrogen and oxygen are electrochemically converted to water to produce electrical power. The maximum continuous power output of these units in the orbiter is 7 kW at a flowrate of 2.6 kg/hr, with an efficiency of 61%. The O_2/H_2 ratio is 8:1. These units weigh 91 kg with a volume of 0.134 m^3. With periodic servicing, these units may accumulate 5,000 hours of on-line service.

Figure 91 *Cylindrical One-Sided Radiator Size for 100 kWe*

	Primary Radiator				Secondary Radiator			
System Type	Thermal Power (kW)	Rejection Temperature (K)	Area (m^2)	Mass (kg)	Thermal Power (kW)	Rejection Temperature (K)	Area (m^2)	Mass (kg)
Brayton	345	550	116.0	415	10	343	28.1	65
Stirling	290	640	52.0	213	10	343	28.1	65
Baseline Thermoelectric	1858	850	104.0	549	3	343	8.4	20
Advanced Thermoelectric	1009	850	56.4	298	3	343	8.4	20
Thermionic	880	850	49.1	260	20	343	56.0	129
Potassium Rankine	390	790	29.5	146	10	650	1.7	6.8

Figure 92 *Two-Sided Flat Plate Radiator Size for 100 kWe*

	Primary Radiator				Secondary Radiator			
System Type	Thermal Power (kW)	Rejection Temperature (K)	Area (m^2)	Mass (kg)	Thermal Power (kW)	Rejection Temperature (K)	Area (m^2)	Mass (kg)
Brayton	345	550	57.9	255	10	343	15.3	44
Stirline	290	640	25.8	130	10	343	15.3	44
Baseline Thermoelectric	1858	850	51.3	330	3	343	4.6	13
Advanced Thermoelectric	1009	850	27.9	180	3	343	4.6	13
Thermionic	880	850	24.2	156	20	343	30.7	88
Potassium Rankine	390	790	14.5	88	10	650	0.3	4.1

This study assumes fuel cell efficiency will reach 80% by the mid-90's and that that reliability and lifetime can be increased to meet the 10-year program requirement. A factor of 1.5 was applied to the STS fuel cell mass and volume data to account for spares, and for electrical, fluid, and structural support equipment. The fuel cell scaling parameters used are listed in Figure 93.

8.4 Electrolyzer Scaling Parameters

The scaling parameters used for the electrolysis systems were obtained from a survey of regenerative fuel cell data (Ref. 17). The study used solid polymer electrolyte units for space-based water reduction of propellants. Some of the advantages cited include: 1) minimum power required per unit of gas generated and 2) 10-20 year life potential. Although the flowrates in the referenced study were significantly higher (7700 lb/hr vs. 80-240 lb/hr) the scaling parameters were assumed to be approximately linear with flowrate. The post-processing reactant gas de-humidifier was discussed separately. Because of its small relative size in this study, the de-humidifier parameters have been included in the electrolyzer scaling parameters. These parameters also include water cooling of the electrolyzers, and the water circulation system. Figure 94 lists this study's electrolysis scaling parameters.

8.5 Emergency Fuel Kit

Since the emergency power system can have its own power or it could derive its power from the prime power supply, an emergency power system was devised for the case where a separate power supply did not exist. The same basic emergency fuel kit was used for all three configurations. Fuel was stored at ambient temperature (300K) in glass-wrapped tanks. Ten kWe average power was assumed for emergency operations for 21 days (5040 kWhe). Water is vented overboard. Station radiators were used. The fuel cells were assumed to be 80% efficient. This subsystem weighs 5.8 t and has a volume of 23 m^3.

8.6 Boom-mounted Reactor Configuration

The boom-mounted reactor produces 3000 kW of thermal power with a conversion efficiency of 5.1%, resulting in 153 kW of electrical power and 2847 kW of waste heat. The optimum boom length is determined by adding the reactor shield mass to the mass of the transmission wires and boom structure. The shield is a man-rated shield, taken here to be a shaped four pi configuration. The shield mass was shown in Figure 7-11 as a function of distance from the station. The wire mass can be calculated as $M = 2n\,\rho\eta\,\frac{P}{V\Delta V}\,\ell^2$, where

- n is the number of wire pairs ($n = 2$ taken here),
- ρ = mass density of the conductor ($\rho = 2700$ kg/m^3 for aluminum),
- η = electrical resistivity ($\eta = 2.94 \times 10^{-8}$ Ω-m for Al at 50°C),
- P = electrical power ($P = 150$ kW taken here),
- V = voltage ($V = 150$ V taken here),
- ΔV = voltage drop through the line ($\Delta V = 3$ V taken here),
- ℓ is the boom length.

For two pairs of aluminum wires carrying 150 kWe at 150 Volts with a 3 Volt potential difference, the wire mass is $M = 0.122\ \ell^2$, with 15% of the wire mass added for insulation. The boom mass is taken to be 1.5 kg/m, or 1 lbm/ft. The combined reactor, PCS, shield, wire, and boom mass is shown in Figure 95 as a

Specific mass	20.4 kg/kWe
Specific volume	0.030 m^3/kWe
Reactant specific energy	3.52 kWh/kg
Electrical power conversion efficiency	80%

Figure 93 *Fuel Cell Scaling Factors*

Mass	30 kg/(kg/hr water processed)
Volume	0.0129 m^3/(kg/hr water processed)
Electrical power	4.84 kWe/(kg/hr water processed)
Conversion efficiency	90%
Heat rejection rate	1.46 kWt/(kg/hr water processed)
Cell pressure	2.76 MPa
Cell temperature	422 K

Figure 94 *Electrolyzer Scaling Parameters*

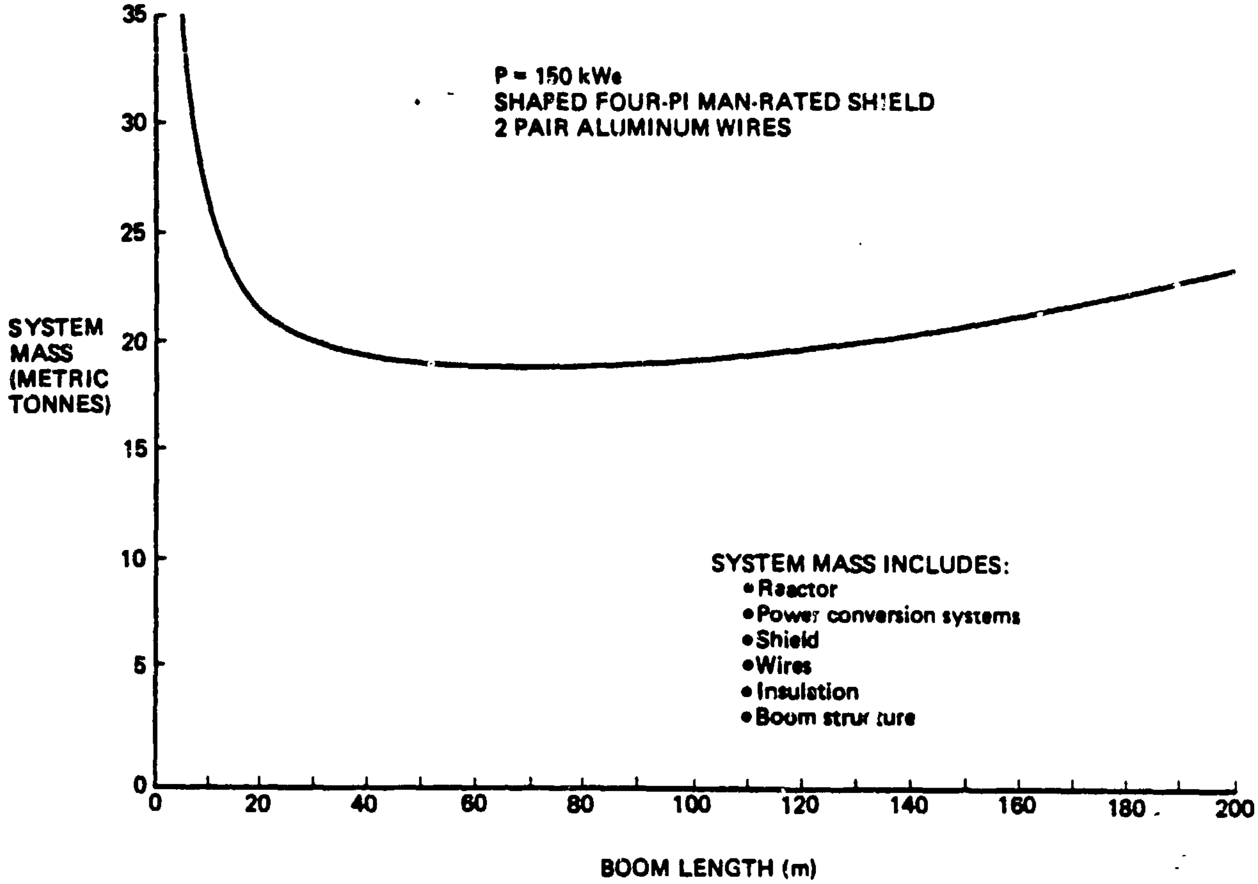

Figure 95 System Mass versus Boom Length

function of boom length for a 153 kWe reactor. Inspection of this figure shows a broad minimum in the system mass between about 50 and 80 meters. This occurs when the shield mass asymptotically approaches its minimum because the fly-by dose dominates, resulting in more of a uniform four-pi shield than one which is weighted toward the space station, while the wire mass is increasing by the square of the boom length. The absolute minimum occurs at about 70 meters separation distance.

The boom-mounted reactor configuration system masses are shown in Figure 96. The mass of the 153 kWe reactor with power conversion system is 1650 kg, and the volume is 2.1m^3. A shaped four-pi shield a 70 meters separation distance weighs 18,000 kg and occupies 7.1m^3 volume. The primary radiator, which rejects 2843 kWt at 850 K, weighs 840 kg. The secondary radiator, which rejects 5 kWt at 343 K, weighs 31 kg. A cylindrical, one-sided radiator was assumed in this study. The radiator volume is 164m^3. The total drag area for the reactor/shield/radiator configuration, is 54m^2, of which 48.2m^2 is due to the primary radiator.

The transmission lines are composed of two pairs of aluminum wires with a diameter of 3.0 cm and length of 70 meters. The mass is 687 kg with insulation. The volume is 0.2m^3 and the drag area is 8.4m^2. The boom mass is 105 kg.

Two additional subsystems are required at the space station for a boom-mounted reactor power supply. The first is a propulsion stage to boost the reactor and shield to a long-lived orbit at the end of its useful lifetime or following a serious accident. An orbital maneuvering vehicle (OMV) is capable of performing this mission and will be available when the space station is operational, so it was used in this study as a reference booster. In actual practice, a smaller booster with lesser capabilities would also be adequate for the end of life disposal mission, but OMV specifications are in existence and it seems reasonable that available equipment would be used. A dedicated OMV is not necessary, but operations requirements may dictate that one be present in a state of operational readiness in case of a serious reactor accident. The space station also has an emergency power system, to be used during any reactor shutdown as described in Section 8.5. This kit is a fuel cell set which provides 10 kWe for 21 days. The fuel cells' mass is 203 kg, the fuel supply mass is 1420 kg, and the tank mass is 4170 kg. The volume of the fuel tanks is 22.6m^3, and that of the fuel cells is 0.3m^3.

The total system mass is 31,530 kg for initial operation and occupies a volume of 236m^3. The launch vehicle to the space station will be the space shuttle. Launch capability of the STS to a 500 km circular orbit with an inclination of 28.5° is currently 19.5 tonnes, (STS Users Handbook). With an augmented thruster system, this is expected to reach 25.0 tonnes within a few years. The available volume in the cargo bay is 300m^3.

An STS manifest plan is shown in Figure 97. The reactor, power conversion system, shield, radiator, boom, and transmission lines can be launched to the space station in a single shuttle flight. The payload mass would be 21.2 tonnes, and the volume would be 193m^3, resulting in a load factor of 0.85. The orbital maneuvering vehicle and emergency power kit would be launched on another flight, with a payload mass of 10.3 tonnes and a volume of 43m^3, resulting in a load factor of 0.41. The total number of shuttle flights to place the operational system at the space station is therefore 1.26.

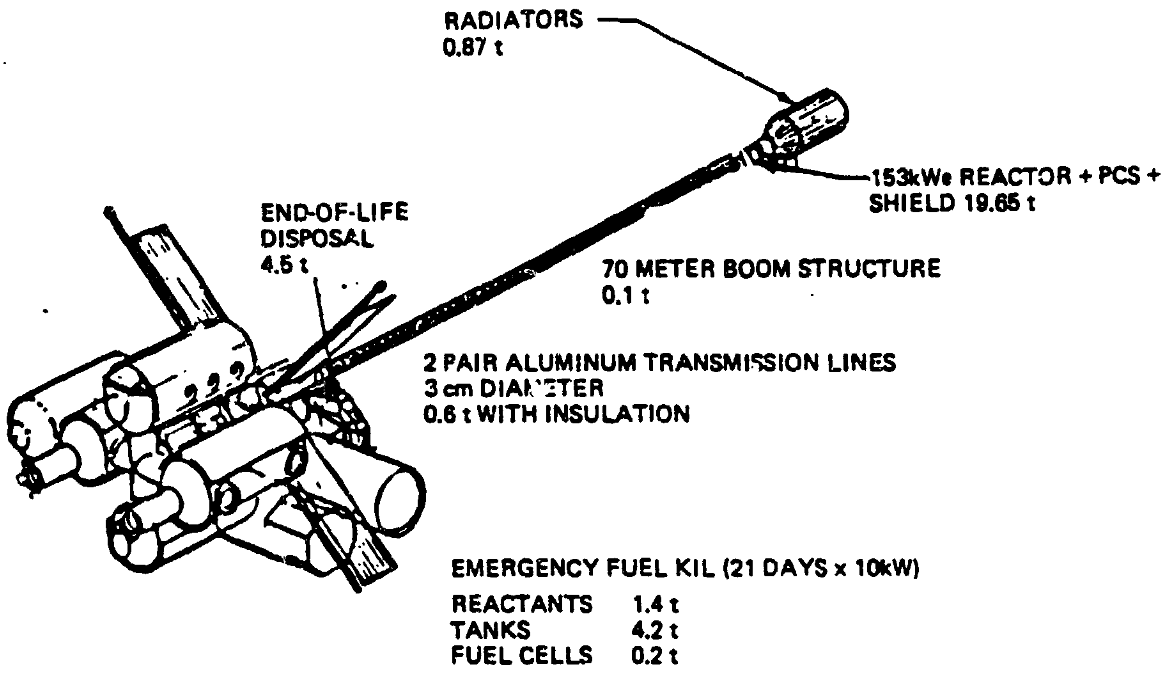

Figure 96 Mass Statement for 150 kWe Boom-Mounted Reactor Configuration

FLIGHT NUMBER	CONTENTS	MASS (t)	VOLUME (m^3)
I	REACTOR, PCS, SHIELD, RADIATOR	20.5	173
	BOOM + TRANSMISSION LINES	0.7	20
II	OMV	4.5	20
	EMERGENCY POWER KIT	5.8	23
RECURRING (10 YEAR LIFE CYCLE)	DRAG MAKEUP PROPELLANT (INCLUDING STATION)	0.47	1.1

I_{SP} = 360 SEC

AUGMENTED STS LIMITS TO 500 KM ORBIT	25.0	300
CURRENT LIMITS	19.5	

TOTAL FLIGHTS	
IOC	1.3
RECURRING	0.02

Figure 97 STS Manifest Plan for Boom-Mounted Reactor System (150 kWe)

If the STS is used to deliver the reactor power system, the center-of-gravity limitations of the orbiter must be taken into account. In order to allow for abort landings with the payload onboard, the center of gravity of the payload is constrained to be ne-- the center of lift of the orbiter wings (Figure 98). The majority of the mass of the payload is in the reactor and its radiation shield, and they are placed near this location. (Figure 99). The radiator extends forward from this. The support boom and power line are collapsed and carried inside the radiator during launch. They are unpacked and attached on orbit. The center of gravity of the system is located at about station X=1067 inches, which is approximately the center of the preferred CG region.

The weight statement for the mission is shown in Figure 100.

Figure 100.
MASS STATEMENT FOR SINGLE STS LAUNCH OF REACTOR SYSTEMS

Reactor & Shield	19,650 kg
Radiator	860
Boom & Transmission	700
Support Cradle	3,500
Total	24,710 kg

The space station drag area is taken here to be $196m^2$. This represents a station with a crew of 8-12, designed for space construction. Such a station, with 6 laboratory modules, 2 habitation modules, 2 resource modules, 1 logistics module, 2 orbital transfer vehicles, and 1 orbital maneuvering vehicle, has a frontal area of $180m^2$. With the orbiter docked, this area is $286m^2$. Assuming the orbiter is docked for 14 days out of every 90 day resupply period yields a time-averaged drag area of $210m^2$. The rest of the system adds another $62m^2$. This results in a total drag force of 0.0055 N in a nominal atmosphere at 500 km. Assuming a cryogenic thruster with a specific impulse of 360 seconds, the orbital makeup propellant requirement is 46.5 kg/year, or 465 kg total propellant mass over a ten year lifetime. This will not have an impact on STS logistics.
The technology required to implement the boom mounted reactor option is under development and expected to be in place by the start of the space station growth design. The two items which need to be developed specifically for this option is an end-of-life disposal system, i.e. a booster vehicle, and a man-rated reactor system.

8.7 Tethered Reactor Configuration

The tethered reactor produces 4,085 kW of thermal power with a conversion efficiency of 5.1%, resulting in 208 kW of electrical power at the reactor and 3,877 kW of waste heat. Two shield configurations were considered: a four-pi man-rated shield which limits the dose rate to 200 mrem/hr at a distance of 30 meters in all directions, and a conical, instrument-rated shield which is designed for 14.4 rem/hr at the power system control electronics. The heavier, man-rated shield would allow manned traffic near the reactor and would allow some limited contact maintenance of power subsystems after reactor shutdown. The instrument-rated shield would not allow any manned access near the reactor.

The tether is taken to be 30 kilometers long to take advantage of the tension for orbital boost at end of life. With an 80% fuel cell conversion efficiency, and 0.85

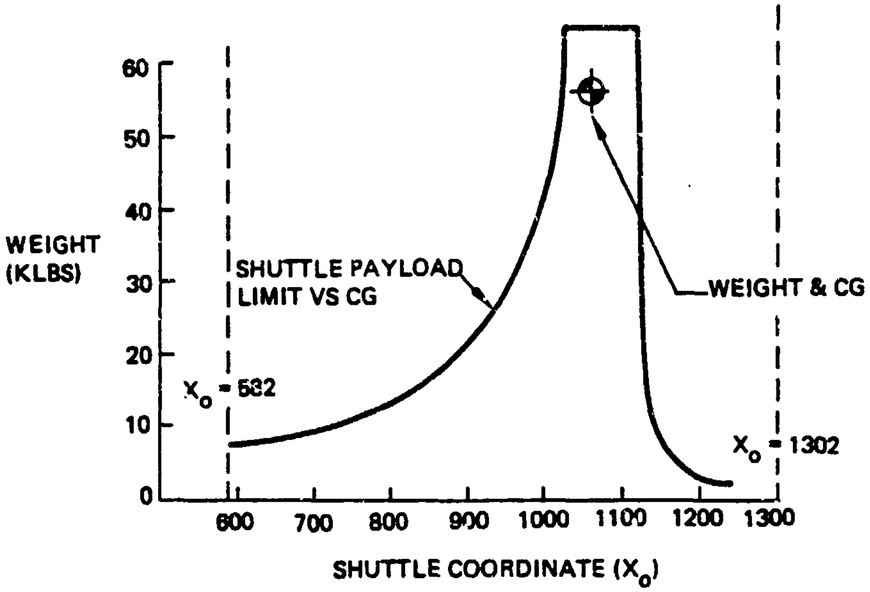

Figure 98 *Single Launch Weight Distribution* (150 kWe)

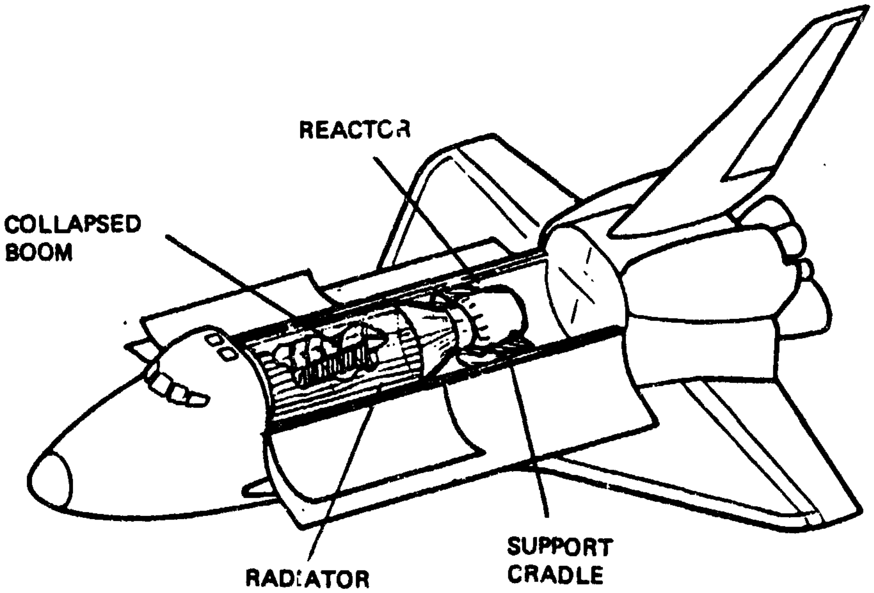

Figure 99 *Single Launch Cargo Bay Manifest* lbs of H_2 & O_2 per KWH, the space station needs a fuel flow rate of 42.6 kg/hr. The water volumetric flow rate from the station to the reactor is therefore 0.0426 m^3/hr, or 1.18 x 10^{-5} m^3/s. The volumetric flow rates of hydrogen and oxygen from the reactor to the space station are 0.169 m^3/s and 0.0084 m^3/s, respectively, at a static pressure of 100 kPa (1 atm) and a temperature of 313 K. Three water lines, three oxygen lines, and three hydrogen lines are bundled together for system redundancy and fail safe operation. The hoses are made with 5-mil thick mylar walls and the bundle is wrapped with 15 layers of multi-layer insulation to prevent water freezing.

The water hoses have a diameter of 1.06 cm. This allows laminar flow with a velocity of 4.5 cm/s and a frictional pressure drop of 250 kPa. The pressure safety margin is about 15. The gas hoses have a diameter of 6.0 cm. The hydrogen flow velocity is 2.0 m/s with a pressure drop of 2.8 kPa, and the oxygen flow velocity is 1.0 m/s with a pressure drop of 12.5 kPa. The pressure safety margin is about three.

The tethered reactor configuration system masses are shown in Figure 101. The mass of the 208 kWe reactor with power conversion system is 2100 kg and the volume is 2.1 m^3. The man-rated shield mass is 18,700 kg and the volume is 7.4 m^3, while the instrument-rated shield is 890 kg and 0.5 m^3. The primary radiator, which rejects 3871 kWt at 850 K, weighs 1140 kg. The secondary radiator rejects 5 kWt and weighs 31 kg. The radiator volume is 245 m^3. The electrolyzer mass is 1600 kg, with a 25% excess capacity, and occupies 0.7 m^3.

Fuel generated at the reactor is pumped into holding tanks. Fuel is drawn from these tanks into the tether hoses, and flows to tanks on the space station. At each end of the tether are holding tanks containing oxygen and hydrogen gases and water. A twelve-hour fuel supply at full power, or 1800 kWh reserve, is stored at each end. The full tanks weigh 2100 kg, including structure and pumps, and occupy 38 m^3.

The space station has an emergency power system (safe haven power) for use when the reactor is not operating or if the fuel is not flowing. This consists of sufficient fuel and storage tanks for 10 kWe operation for 21 days. Together, the emergency fuel kit weighs 5800 kg and occupies 23 m^3. The main fuel cells which supply 150 kWe to the space station, weigh 3700 kg, with a twenty percent excess capacity. An orbital boost vehicle can be stored at the space station for reactor boost to a long lived orbit, if required.

The tether itself consists of nine mylar hoses (3 each of the H_2, O_2, and H_2O for reliability) wrapped in a bundle of multi-layer insulation. The empty tether weighs 15,000 kg. The hoses must be filled before operation commences. The mass of water in the hoses is 7940 kg.

The total system mass is 64,700 kg for initial operation and occupies a volume of 369 m^3 in its stored configuration. An STS manifest plan is shown in Figure 102. The reactor, power conversion system, shield, radiator, and electrolyzer can be launched to the space station on a single shuttle flight, with a load factor of 0.95. A second flight would carry the tether in a collapsed configuration, the fuel cells, and both sets of holding tanks, for a load factor of 0.92. A third flight would take the emergency power kit, water, and an orbital transfer vehicle, for a load factor of 0.73. The total number of STS flights to place the complete system in its initial operation condition is less than three: a full cargo bay equivalent of 2.60. The

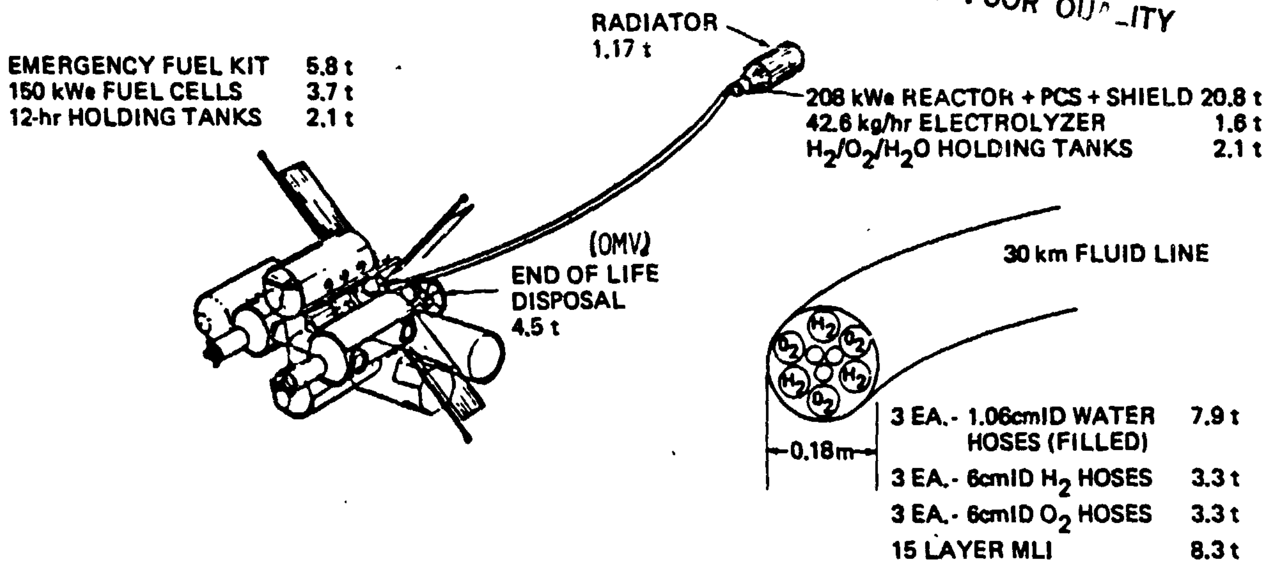

Figure 101 *Mass Statement for 150 kWe Tethered Reactor Configuration*

FLIGHT NUMBER	CONTENTS	MASS (t)	VOLUME (m^3)
I	REACTOR, PCS, SHIELD, RADIATOR	22.0* (4.16)	256* (248)
	ELECTROLYZER	1.6	0.7
II	HOSES (COLLAPSED)	15.0	13
	FUEL CELLS	3.7	6
	HOLDING TANKS	4.2	41
III	OMV	4.5	20
	EMERGENCY POWER KIT	5.8	23
	WATER	7.9	9.5
RECURRING (10 YEAR LIFE CYCLE)	DRAG MAKEUP PROPELLANT (INCLUDING STATION)	8.6	9.0

*INCLUDES MAN-RATED SHIELD. FIGURES IN PARENTHESES REFER TO INSTRUMENT-RATED SHIELD.

I_{SP} = 360 SEC

TOTAL FLIGHTS	
IOC	2.6*(1.9)
RECURRING	0.34

Figure 102 *STS Manifest Plan for Tethered Reactor System* numbers in parentheses in Figure 102 are for an instrument-rated shield. The full orbiter equivalent is 1.9 if equipment can be manifested inside the radiator envelope.

As shown in Section 8.6, the time-averaged space station drag area is 210 m^2. The frontal area of the tether is 5400 m^2, while the atmospheric density decreases by one-third. The drag force on the system is 0.096 N. The orbital makeup propellant required is 855 kg/year, for a ten year cumulative total of 8550 kg, or 0.34 equivalent orbiter payloads. Ninety-four percent of this drag is due to the frontal area of the tethered hose bundle itself.

Most of the technology required to implement the tethered reactor option is currently under development. The fuel system is essentially a regenerative fuel cell with the electrolyzer and fuel cell separated. The space-based electrolysis process needs to be developed for this program if it is not already developed by the start of space station operation. Tether dynamics needs to be understood. Micrometeoroid protection may be necessary for the hoses and some form of leak detection and repair technology would need to be incorporated. As with the boom-mounted reactor option, an end-of-life disposal system is required.

8.8 Free-Flying Reactor Configuration

8.8.1 Introduction

The basic free-flying reactor configuration was discussed above in 8.1.1. A requirement of a 300 year safe orbit for this configuration was imposed. This constraint implied a minimum 600-700 km orbit for the free-flyer. Because of the long period between co-planar free-flyer/station orbits, the main system driver was the amount of reactant required in orbit to fulfill the total power x time requirement. This requirement further impacted the fuel phase and the transfer vehicle selected.

Figure 103 describes the basic free-flyer scenario. The reactor and electrolysis/liquefaction plant process water and store the reactants until the reactor and station are co-planar. A Space Based OTV (SBOTV) flies from the station to the free-flyer with a fully loaded water tank and empty LH_2 and LO_2 tanks, drops these tanks off, and retrieves the empty water tank and full LH_2 and LO_2 tanks for return to the station. During the resupply mission - which would take less than 1 day - the station operates in a minimum power mode (10 kW) using the emergency backup kit for fuel. Makeup fuel to replace SBOTV propellant, boiloff, and other losses are brought to the station from earth in the form of water, and then flown to the free-flyer for processing on its next resupply mission.

8.8.2 Orbital Considerations

For purposes of this study, the space station was assumed to be in a circular orbit at 500 km altitude and 28.5° inclination. Resupply orbit transfer requirements and strategies with the reactor in a circular orbit at altitudes of 600, 700, 900, and 1200 km and 28.5° inclination were analyzed.

Gravitational pertubations due to the earth's oblateness cause changes in orbit elements over time. In this study the change of greatest interest was the drift in

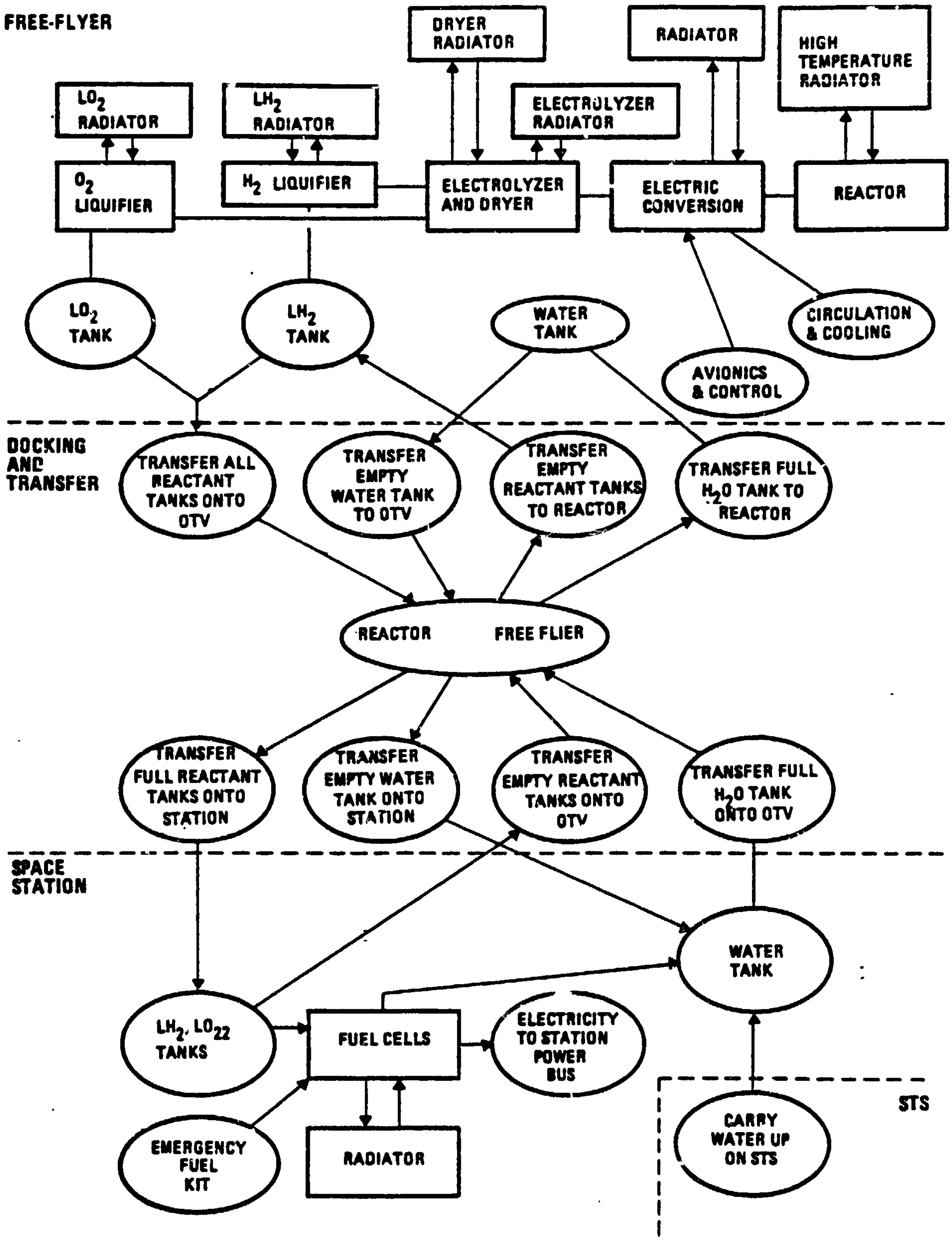

Figure 103 *Free-Flying Reactor Scenario Block Diagram* the longitude of the ascending node (Ω) due to first order secular perturbations. This rate of drift is:

$$\Delta\dot{\Omega} = -\frac{3}{2}\,\frac{J_2\,\dot{\theta}}{(1-e^2)^2}\left(\frac{R_e}{a}\right)^2 \cos i$$

where Ω = rate of change of longitude of ascending node,
J_2 = zonal harmonic coefficient (= 1.08263×10^{-3}),
$\dot{\theta}$ = orbital mean motion,
e = orbit eccentricity,
R_e = Earth radius,
a = orbit semi-major axis,
i = orbit inclination.

Since both orbits are circular and at the same inclination, the difference in drift rates is a function of altitude difference. The differential ascending node drift means that most of the time an orbit transfer between the space station and reactor would involve a plane change, which is expensive in terms of ΔV, and therefore propellant consumption. Figures 104 and 105 show the resulting ΔV requirements. In Figure 104 the requirements are shown as a function of the difference in ascending nodes. In, Figure 105, the drift rates have been taken into account to show the ΔV requirements as a function of time since the orbits were coplanar rather than of the difference in longitude of the ascending nodes. It can be seen in both figures that the ΔV requirements become prohibitive when the ascending nodes are more than about 25° apart. From Figure 105, it can be seen that the orbits are close to coplanar every 180 days when the reactor is at 1200 km altitude, but this occurs only every 1090 days when the reactor is at 600 km altitude. This suggests that the launch windows for fuel transfer occur only once every 0.5-3 years. This and the power demand determine the tanks payload size.

One way to reduce the long time between launch windows is to use phasing orbits. By using an intermediate, transfer orbit with a different ascending node drift rate, the transfer can be achieved in less time than it would take for the station and reactor orbits to become coplanar and for less ΔV than a direct transfer. To transfer from the space station to the reactor, three burns are needed. In the first, the transfer orbit apogee is raised to the selected altitude (higher than the reactor altitude). The perigee is then raised to the reactor altitude. After the phasing (transfer) orbit ascending node and reactor orbit node coincide, the final burn is made to circularize at the reactor altitude. The transit times for the four reactor altitudes and a variety of phasing orbit apogees are shown in Figures 106a through d along with the time required for the station and reactor orbit ascending nodes to coincide. The corresponding ΔVs are shown in Figure 107.

It can be seen from Figures 106 that the transit time using phasing orbits are usually less than the time needed for the station and reactor orbits to become coplanar. The ΔV requirements are less than for most direct transfers between non-coplanar orbits since all of the phasing orbit burns are in-plane.

In most cases shown, the difference in the semi-major axes of the phasing orbit and the reactor orbit is greater than the difference between the semi-major axes of the space station and reactor orbits and the ascending node relative drift rate is correspondingly greater. In addition, since the phasing orbit's semi-major axis is greater than the reactor orbit's, the relative drift rate is in the opposite direction from that of the reactor and station orbits.

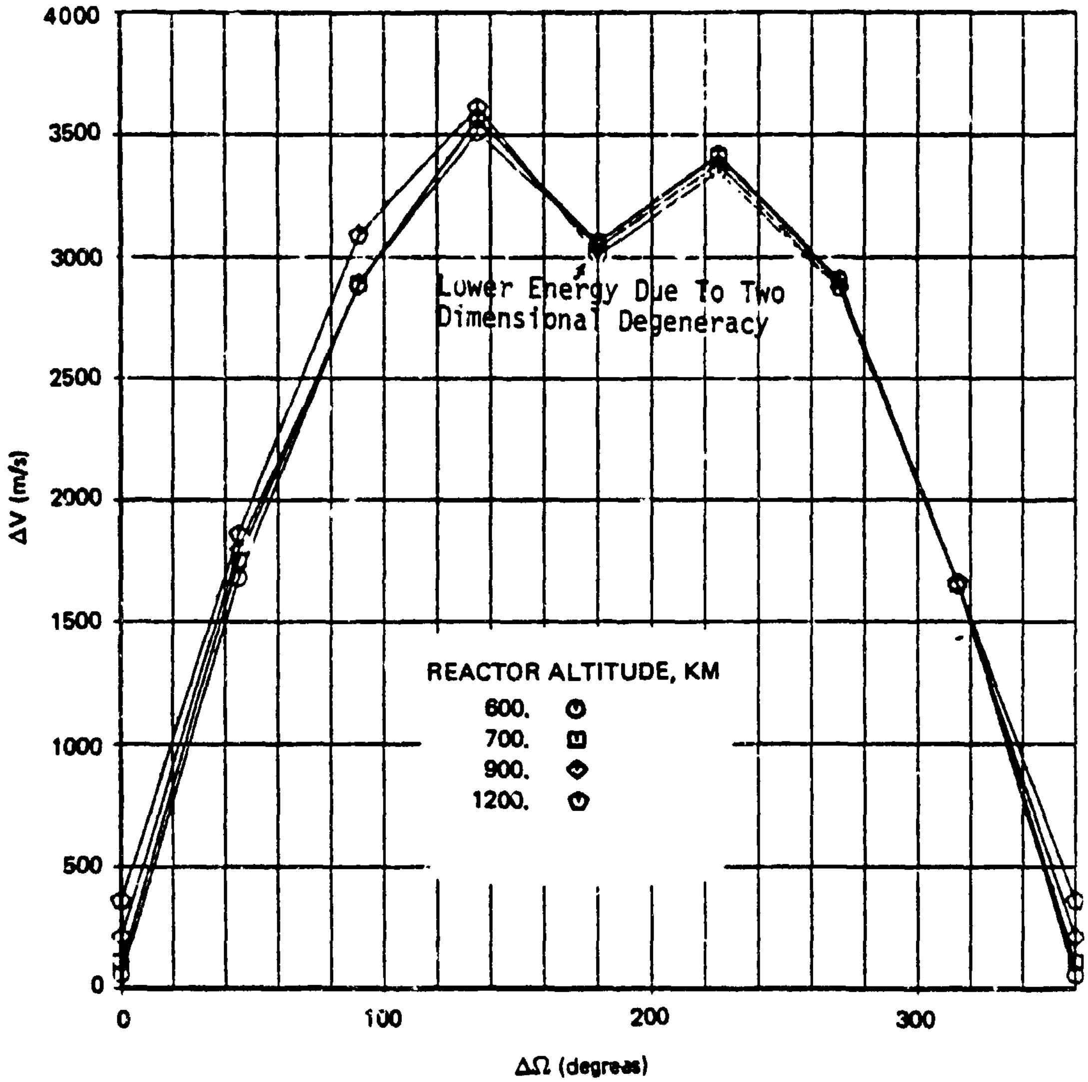

Figure 104 *Orbit Transfer Delta V Requirements as a Function of Orbit Plane Difference and Reactor Altitude*

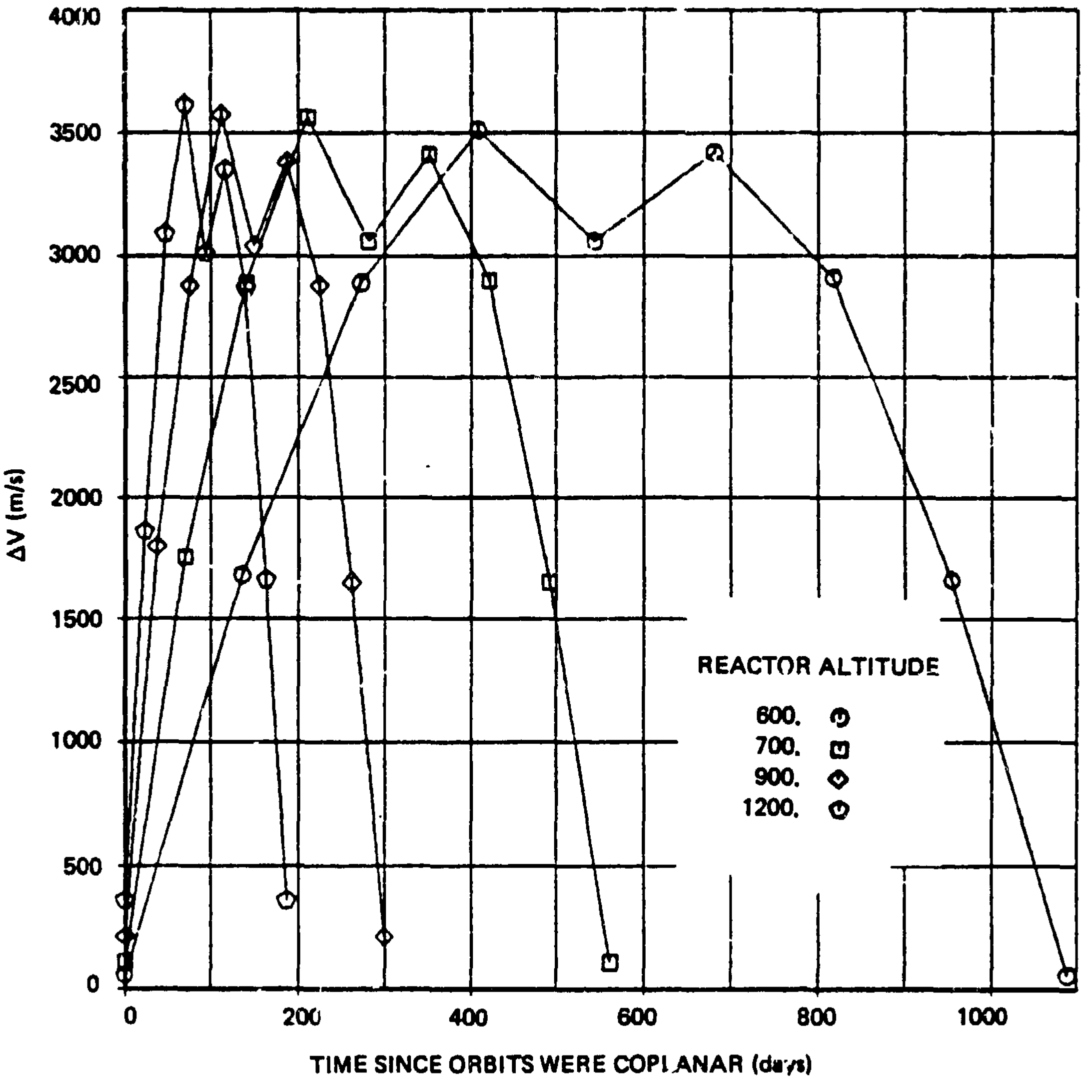

Figure 105 *Orbit Transfer Delta V Requirements as a Function of Time Since Orbits Were Coplanar and Reactor Altitude*

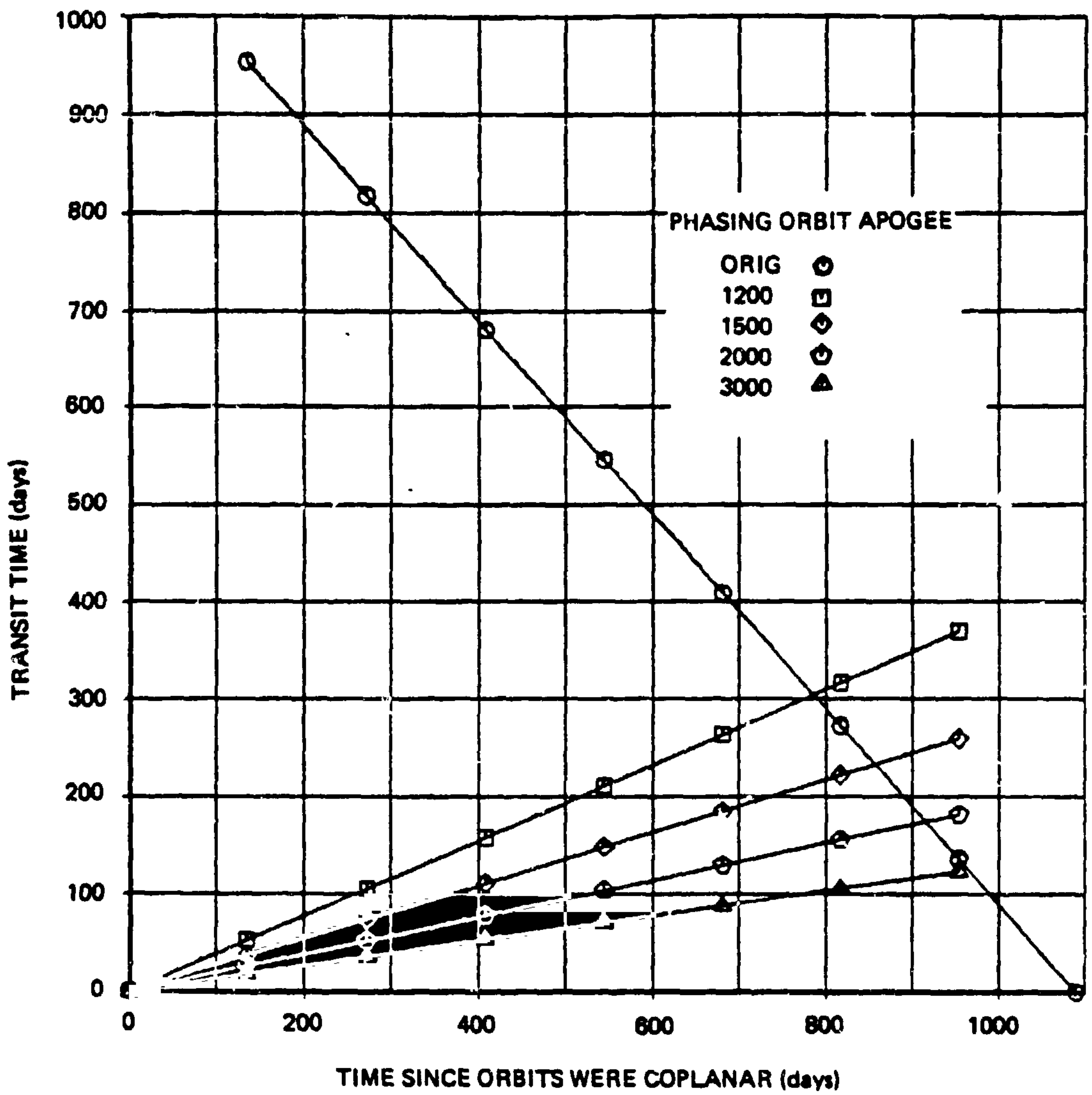

Figure 106a *Orbit Transfer Transfer Times With Phasing Orbit--Final Orbit Altitude 600 Kilometers*

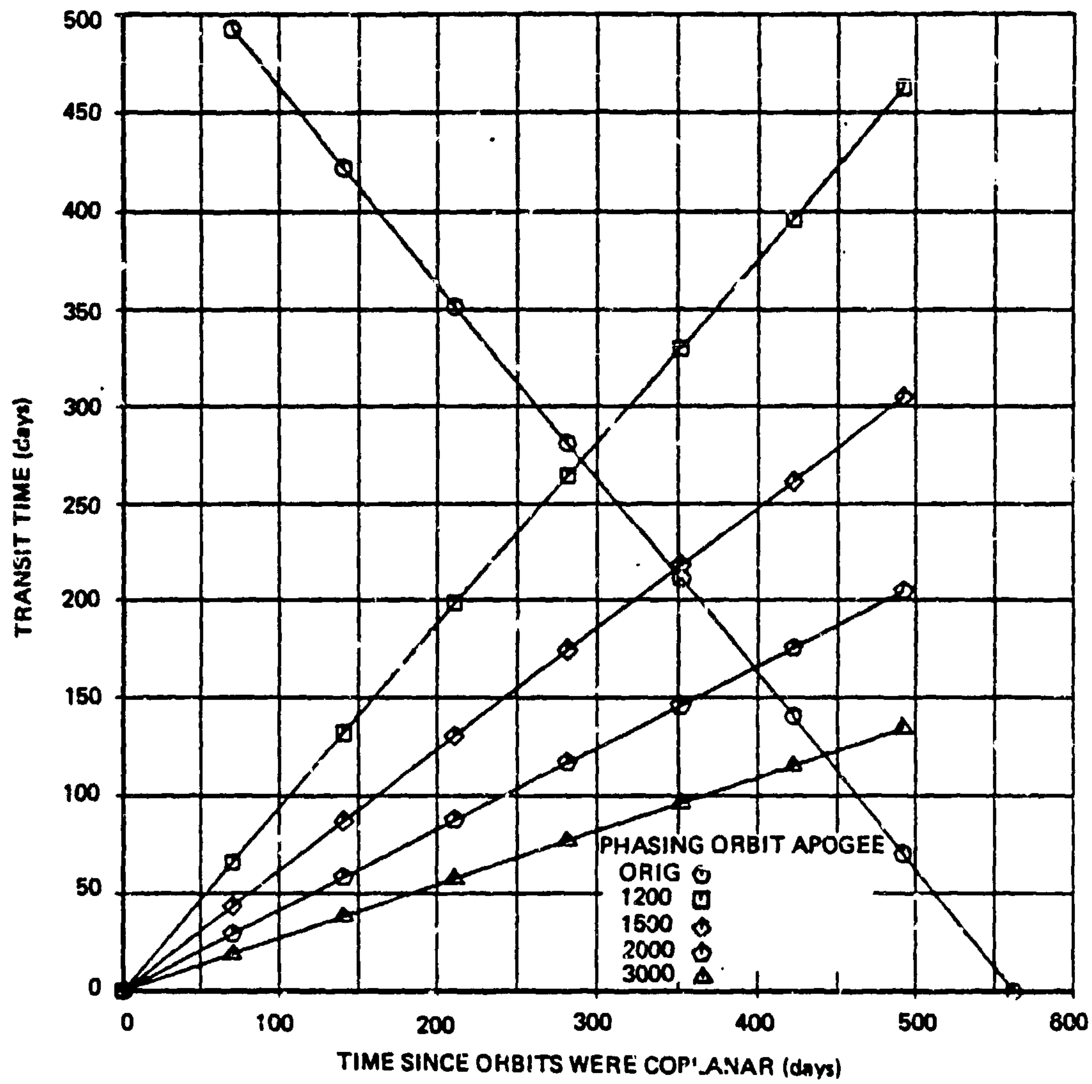

Figure 106b *Orbit Transfer Transfer Times With Phasing Orbit—Final Orbit Altitude 700 Kilometers*

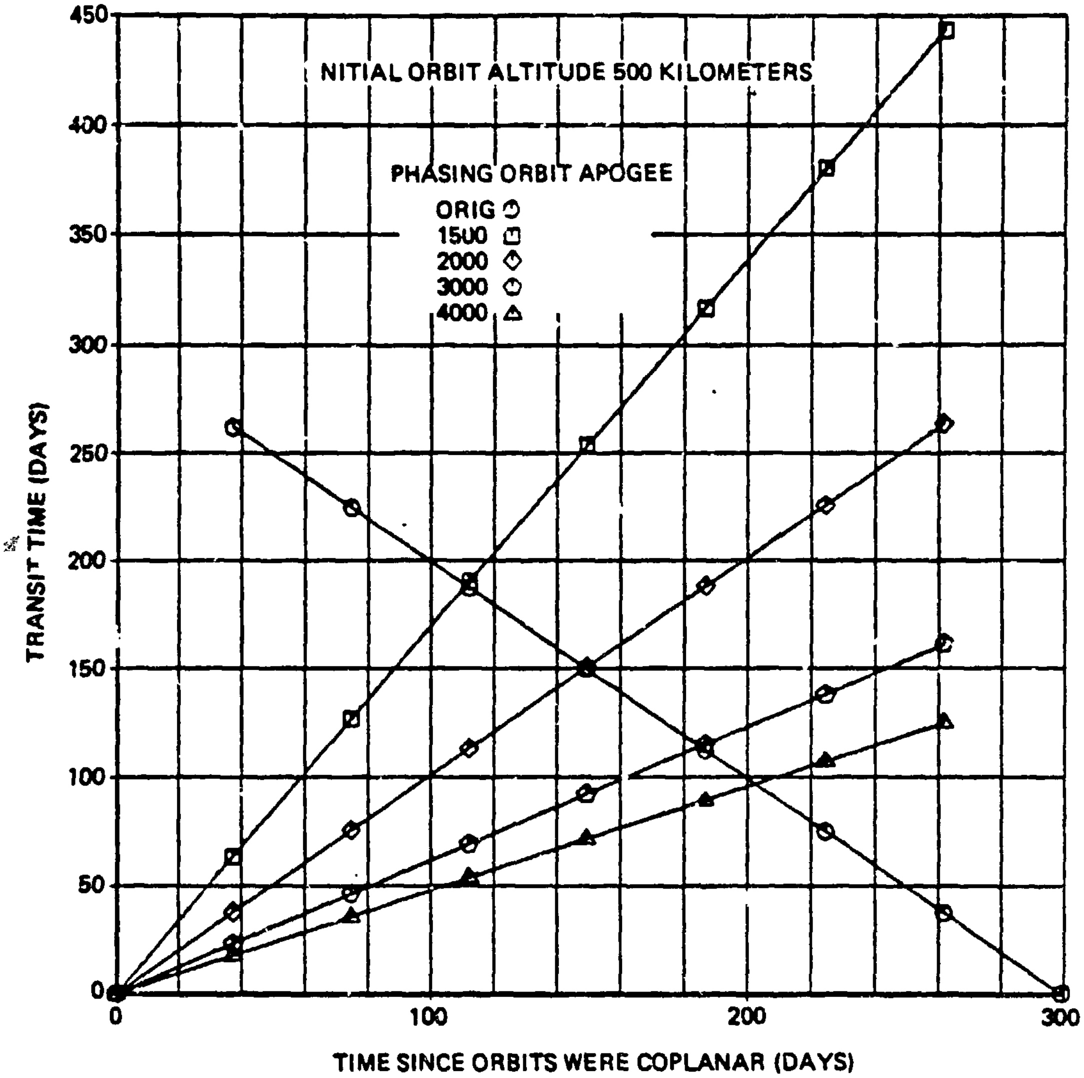

Figure 106c *Orbit Transfer Transfer Times With Phasing Orbit—Final Orbit Altitude 900 Kilometers*

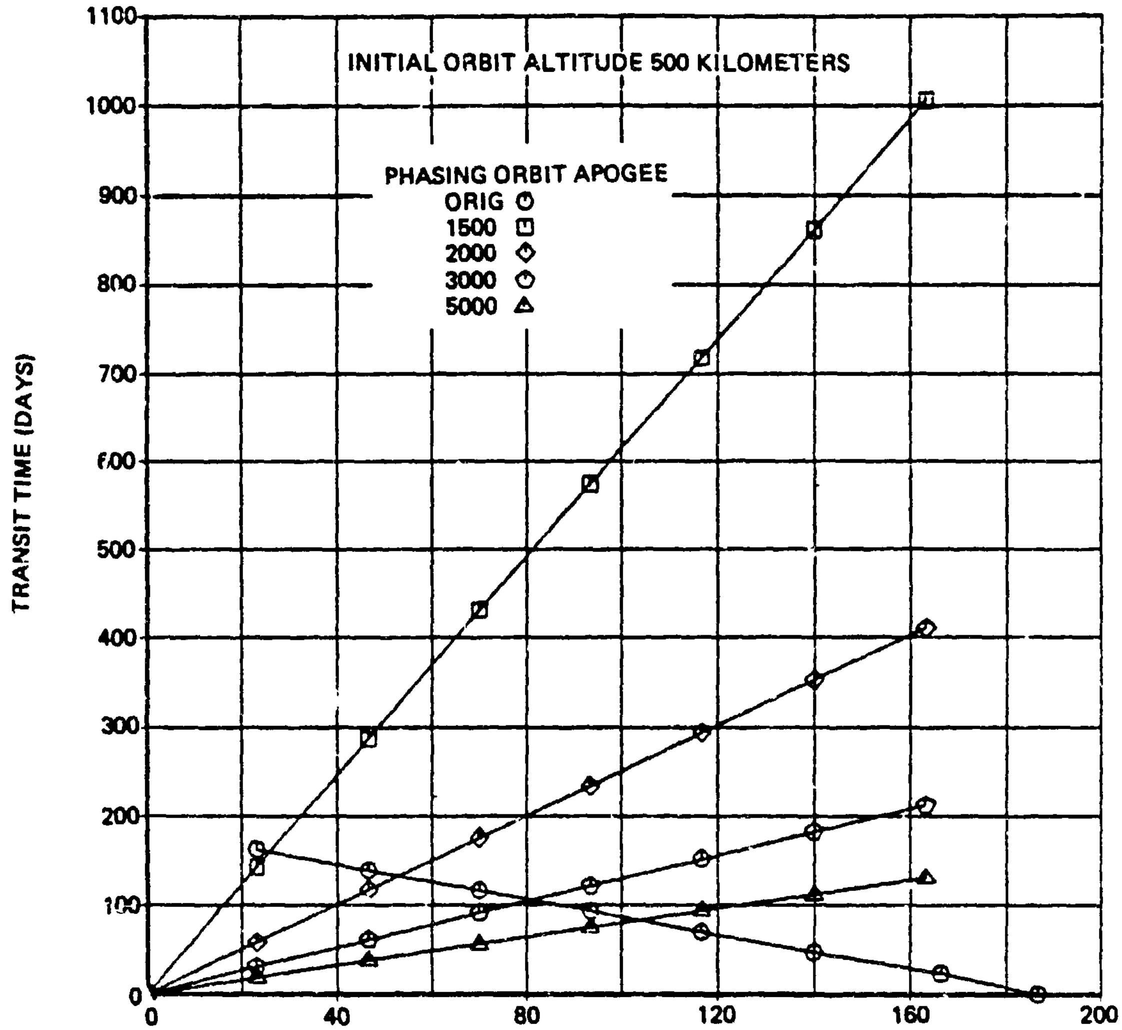

Figure 106d *Orbit Transfer Transfer Times With Phasing Orbit—Final Orbit Altitude 1200 Kilometers*

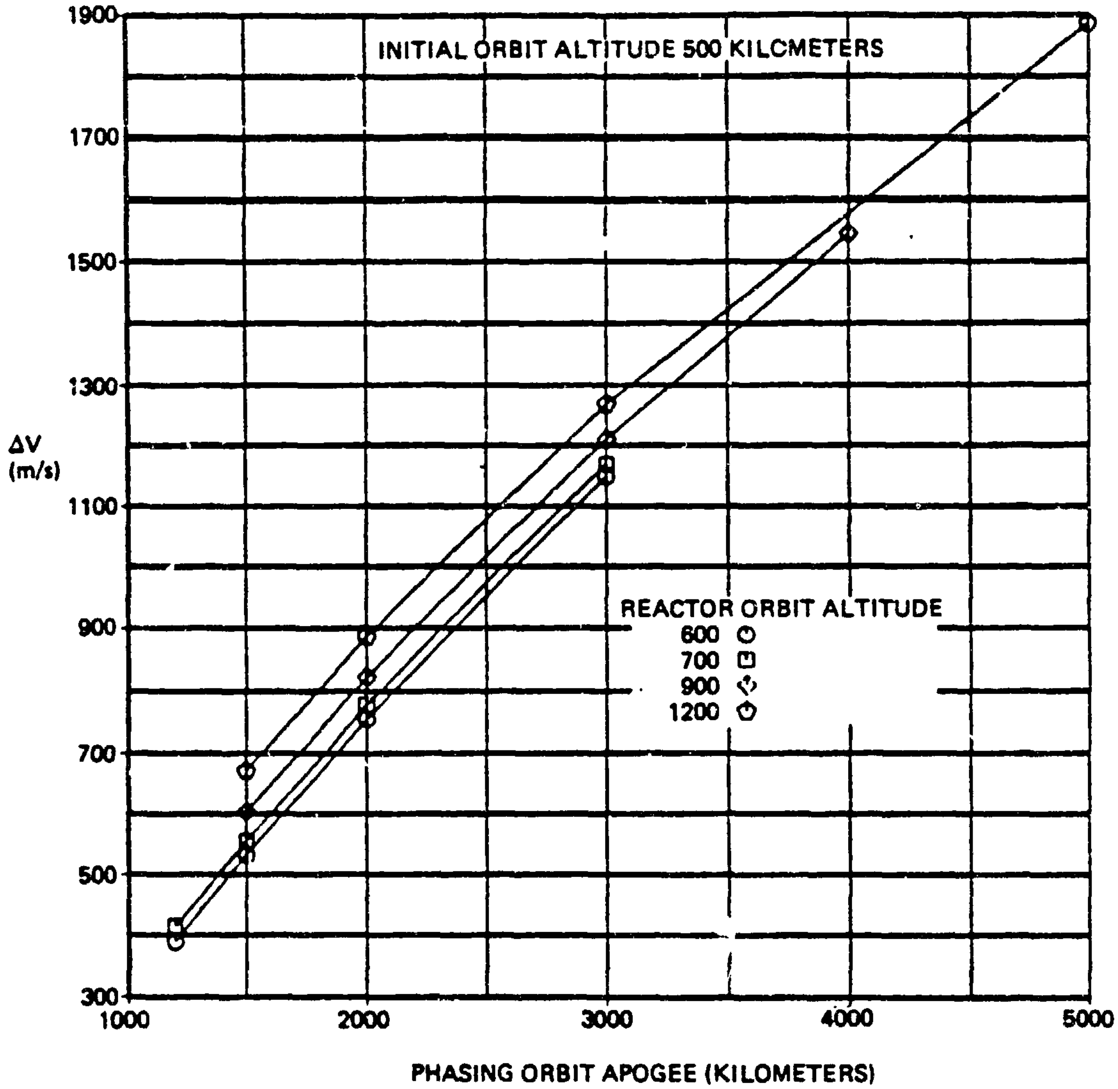

Figure 107 *Transfer Orbit Delta V Requirements With Phasing Orbit*

As an example of a resupply strategy using phasing orbits, the case where the space station was in a 500 km altitude orbit, the reactor was in a 700 km orbit, and the phasing orbit apogee was 1500 km (Figure 106b) was studied. A tanker launched when the two orbits were coplanar would arrive the same day. The next tanker could be launched after 30 days, with a transit time of 19 days, thus arriving 49 days after the last one. This could be continued every 30 days for 360 days, until the transit time via the phasing orbit became greater than the time until the station and reactor were coplanar again. Meanwhile, the tankers would be arriving at the reactor every 49 days.

In all cases using phasing orbits, the ΔV requirement is larger than the ΔV required for a direct Hohmann transfer orbit between the space station and reactor. Thus, the increased launch frequency can be achieved at the expense of increased OTV propellant requirements, increased reactor power, and increased OTV fleet size.

Although the minimum ΔVs for coplanar orbit transfers are relatively small, the long periods between them imply very large reactant payloads for the high station electric power requirement. Reducing the period between transfers increases the minimum ΔV. A compromise between these conflicting factors was made in selecting a 1200 km circular orbit. In this configuration, a transfer is performed every 180 days with ΔV = 325 m/s. Over the 10 year system life, 20 flights are necessary. Even for the 150 kWe station requirement, each fuel payload is 230 t.

8.8.3 Fuel Phase Selection

Three different options were considered for reactant storage and transfer: 1) gas, 2) supercritical, and 3) liquid. Figure 108 lists the advantages and disadvantages of each. Tank mass and volume were evaluated for each option assuming a 100 kWe station requirement with 30 day resupply at 65% total system efficiency (72,000 kWh). Figure 109 presents the tank mass and volume ratios relative to liquid tanks. These ratios increase with increasing reactant capacity.

The gas storage option assumed glass-wrapped tanks at ambient temperature (300K), pressurized to 3000 psia, with a factor of safety on pressure of 2.0.

The supercritical reactant option assumed H_2 at 33 K and 15 atmospheres, with a compressibility factor of 0.45 in titanium tanks, and O_2 at 160 K and 58 atmospheres, compressibility = 0.4 in aluminum tanks. These tanks were single-walled with multi-layered insulation passive thermal control. Their factor of safety was 2.5.

The liquid storage tanks were assumed to be 7.6% of reactant mass for H_2 and 0.58% for O_2. These tanks were single walled, with multilayer insulation, and had low boiloff rates. The last two tanker options would be launched dry, with water launched separately. Although the liquid storage option requires the most technology development, the savings in mass and volume - and therefore STS and OTV logistics -were overriding factors and the liquid storage option was selected for the reference design.

8.8.4 Liquefier Characteristics

Production of 207,000 kg of reactants (23000 kg of LH_2 and 184000 kg of LO_2) is required during a six-month tank refill cycle. The liquefaction process assumed for hydrogen and oxygen reactants is shown in Figure 110. Beginning with reactants

Figure 108 *Fuel Cell Scaling Factors*

	Gas	Supercritical	Liquid
Tank weight	High	Medium	Low
Tank volume	High	Medium	Low
Cost	Low	Medium	High
Reliability	High	Medium	Low
Safety	High pressure concern	Control temperature, pressure	Control temperature, pressure most dangerous
Refrigeration	No	Yes	Yes
Liquifaction	No	No	Yes
Compressor	Yes	No	No
Time	Fast	Medium	Slow

Figure 109 *Mass and Volume Ratios of Gas, Supercritical and Liquid Reactant Tanks Relative to Liquid Tanks*

Tank	Gas	Supercritical	Liquid
Mass Hydrogen	100	8	1
Oxygen	64	17.5	1
Volume Hydrogen	4	3	1
Oxygen	3.5	2.5	1

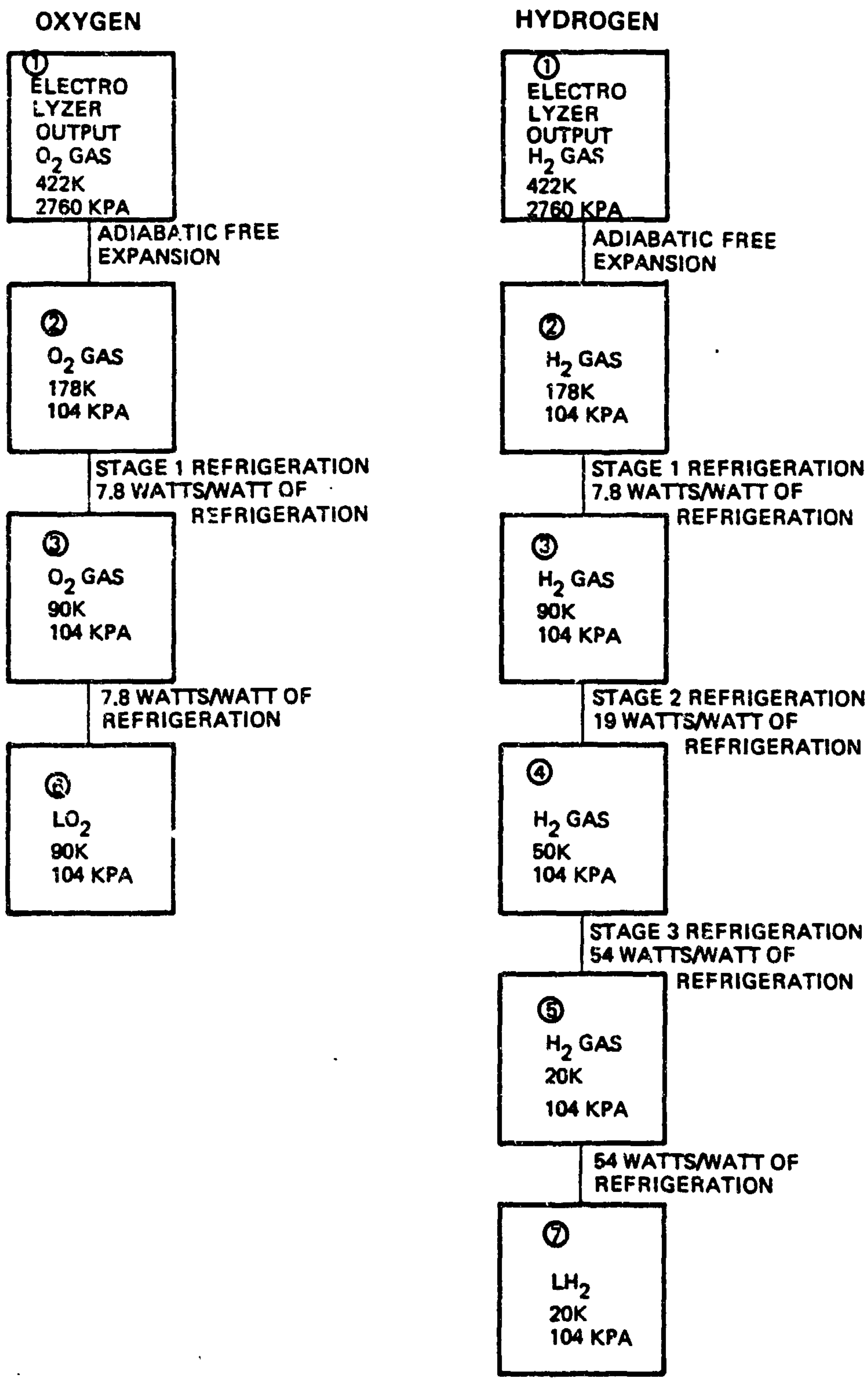

Figure 110 Reactant Liquifaction Process

from the electrolyzer at 2760 kPa and 422 K; a free expansion to 104 kPa and 178 K is employed. A single stage cryogenic refrigerator then cools the oxygen to the 90 K boiling point and accomplishes the liquefaction. Three stages of cryogenic refrigeration are required to produce the liquid hydrogen: these operate at 90 K, 50 K, and 20 K.

The current state-of-the-art in cryogenic refrigeration for space applications is such that small units producing a few Watts of refrigeration at temperatures from 10 to 100 K can operate with a life of approximately one year. Development within the industry is proceeding to extend life and capacity for machines using various thermodynamic cycles and mechanical design approaches. It is believed that cryo-refrigeration technology in the multi-kilowatt range will be available within an acceptable time frame to support a space station program. Figures 111a, b, and c show projected cryo-refrigerator performance data with assumed design points indicated for the three stages upon which this study is based. These data are based on projected 1995 cryo-refrigerator performance per Reference 18. Figure 112 shows the corresponding refrigeration power computations for each stage and total power computations.

The data of Reference 19 were employed as representative of the latest technology in low-loss on-orbit cryogenic storage. The LO_2/LH_2 storage configuration utilizes vapor-cooled-shields (VCS) in both tanks using boil-off from the hydrogen tank. A thermodynamic vent system (TVS) is employed in the hydrogen tank which utilizes a para-to-ortho conversion to maximize refrigeration effect obtained from the hydrogen boiloff. A total hydrogen boiloff of 4100 kg occurs during the 6-month fill cycle; LO_2 boiloff is negligible due to the VCS and TVS operation.

8.8.5 Processor Low Temperature Radiators

To efficiently radiate heat from the processor subsystems (electrolyzer, dehumidifier, LH_2 liquefier, LO_2 liquefier), these subsystems radiators were assumed to radiate at different temperatures. The effective sink temperature was assumed to be 175K. Figure 113 lists the radiator mass and volumes.

8.8.6 Space-Based OTV Performance

The selection of a free-flyer orbit altitude of 1200 km meant that a ΔV of 325 m/s was required with in-plane refueling flights every six months. Large, heavy payloads were required to meet the 100-500 kWe continuous station demand for six month intervals. Because of these heavy payloads, an orbit transfer vehicle (OTV) was considered. Over the ten year system life, 20 OTV flights were necessary for resupply alone so a dedicated Space-Based OTV (SBOTV) was selected (Ref. 1). A difference from the reference 1 design used was that a ballute was not necessary for orbit reduction, a comforting thought for a cryogenic payload. An 8/1 LO_2/LH_2 ratio was assumed with an I_{sp} of 480 seconds, the dry OTV mass was 5500 kg.

The SBOTV is stored at the station between missions. Before leaving the station, the SBOTV tanks are filled and the payload attached. The OTV tanks are not filled again until the next mission. As described below, the payload primarily consists of empty LH_2/LO_2 tanks (except residuals) and a full water tank. The OTV flies to the free-flyer, attaches the tank kit, and returns to the station with a second tank kit with full LH_2/LO_2 tanks and an empty H_2O tank. Water to replace propellant

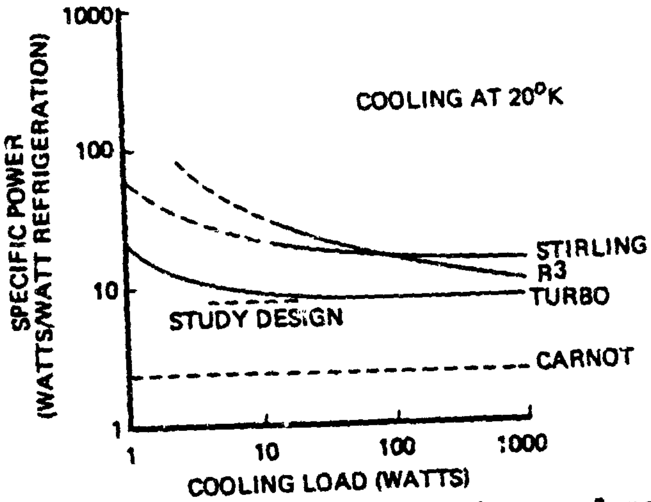

Figure 111a. Projected 1995 Refrigerator Performance Summary–20°K

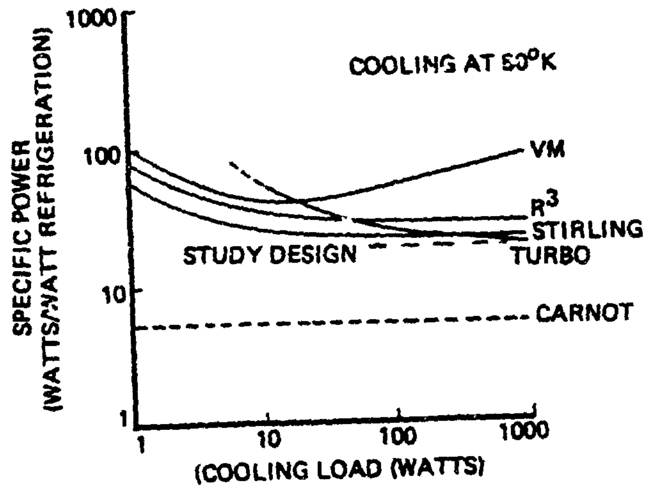

Figure 111b. Projected 1995 Refrigerator Performance Summary–50°K

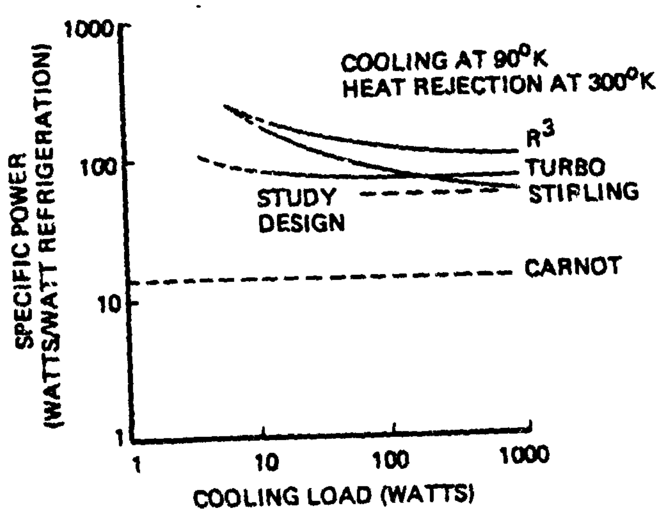

Figure 111c. Projected 1995 Refrigerator Performance Summary–90°K

Figure 112 *Reactant Liquifaction Power Requirements*

		Oxygen					Hydrogen				
Process *	ΔT Kelvin	Cp vapor J/kg-°K	h_{fg} J/kg	Refr. required J/kg	Power Factor	Power required J/kg	Cp vapor J/kg-°K	h_{fg} J/kg	Refr required J/kg	Power Factor	Power Required J/kg
① → ②	244	—	—	—	—	0	—	—	—	—	0
② → ③	88	921	—	81,400	7.8	0.630×10^6	12,980	—	1.39×10^6	7.8	8.884×10^6
③ → ④	40	—	—	—	—	—	11,306	—	0.451×10^6	19	8.569×10^6
④ → ⑤	30	—	—	—	—	—	10,467	—	0.314×10^6	54	16.956×10^6
⑤ → ⑦	0 (Liquifies)	—	—	—	—	—	—	0.447×10^6	0.447×10^6	54	24.138×10^6
③ → ⑥	0 (Liquifies)	—	211,700	211,700	7.8	1.651×10^6	—	—	—	—	—
Total						2.281×10^6					58.55×10^6

*See Figure 8-16

and other losses is launched from earth, transferred to the H_2O tank, flown to the free-flyer, processed, and flown back to the station as payload.

To eliminate losses which would occur during propellant transfer from tank-to-tank, the OTV does not contain a dedicated set of transfer tanks. Rather, it delivers tanks "kits" from the station-to-reactor and reactor-to-station which "plug-in" to the systems. An automated mating capability is required at the reactor.

The station power requirement evaluated was 150 kWe. Fuel cell efficiency was taken as 80%. Boiloff was assumed to be 6%/6 months (4% H_2, 2% O_2). Enough water was flown up, processed, and returned as payload to fulfill the next mission's propellant requirements. Figure 114 lists payload characteristics.

For the 150 kWe power option, 34 t of propellant is necessary for a roundtrip mission. This is very nearly the reference 1 maximum propellant capability.

8.8.7 Reactor Power Flow Distribution

The free-flyer electric power distribution is described in Figure 115. The electrolyzer requires a total of 262 kWe. This produces a reactant flow rate equivalent energy of 240 kW. A price of 113 kW is paid for liquefying the reactants. The circulation and cooling, avionics, and control power requirements were scaled (linearly) from an earlier study.

8.8.8 Mass/Volume Statement - Initial Operating Conditions

Figure 116 describes the initial configuration of the system and identifies the mass of each major subsystem. Figure 117 lists the mass and volume of these and other subsystems. The dominant element is clearly the reactant/water mass.

8.8.9 Makeup Fuel Requirements

Over 34 t of propellant are required by the SBOTV for each roundtrip resupply mission. This loss, and others, are made up by launching water from earth to the station on the STS, and from the station to the reactor on the SBOTV. Other losses include boiloff and drag makeup fuel.

Drag makeup was evaluated for the contribution the elements of this study would have in addition to the basic station. In the 150 kWe case, 95.5 m^2 were added to the station cross-section resulting in a 10 year drag makeup requirement of 136 kg with a ballistic coefficient (M/C_DA) of 1406 kg/m^2, an atmospheric density of 3.18×10^{-13} kg/m^3 at 500 km, and an I_{sp} of 450 sec. Drag at the free-flyer is negligible.

Boiloff was assumed to be 4%/6 months for LH_2 and 2%/6 months for LO_2. Advanced tank design would be required to result in these low rates. Figure 118 summarizes the total makeup fuel requirement for the two options studied.

8.8.10 Logistics

The mass and volume of payload elements necessary for initial operating conditions (IOC) are tabulated in Figure 119. These values were used to evaluate SB OTV launch requirements from the station to the reactor. Shuttle launch capability was

Figure 113 . *Low Temperature Radiator Characteristics*

Subsystem	Radiator Temp (K)	Radiator Power (KW)	Area (m^2)	Mass (t)
Electrolyzer	283	62	341	1.0
Dehumidifier	275	18	113	0.3
LH_2 Liquifier	300	192	807	2.3
LO_2 Liquifier	335	64	165	4.7

Figure 114 *Tanker Payload Mass*

	Mass (tons)
H_2O or H_2 & O_2	230
H_2O tank (1% including structure)	2.3
LH_2 tank (10% including structure)	2.5
LO_2 tank (2% including structure)	4.1

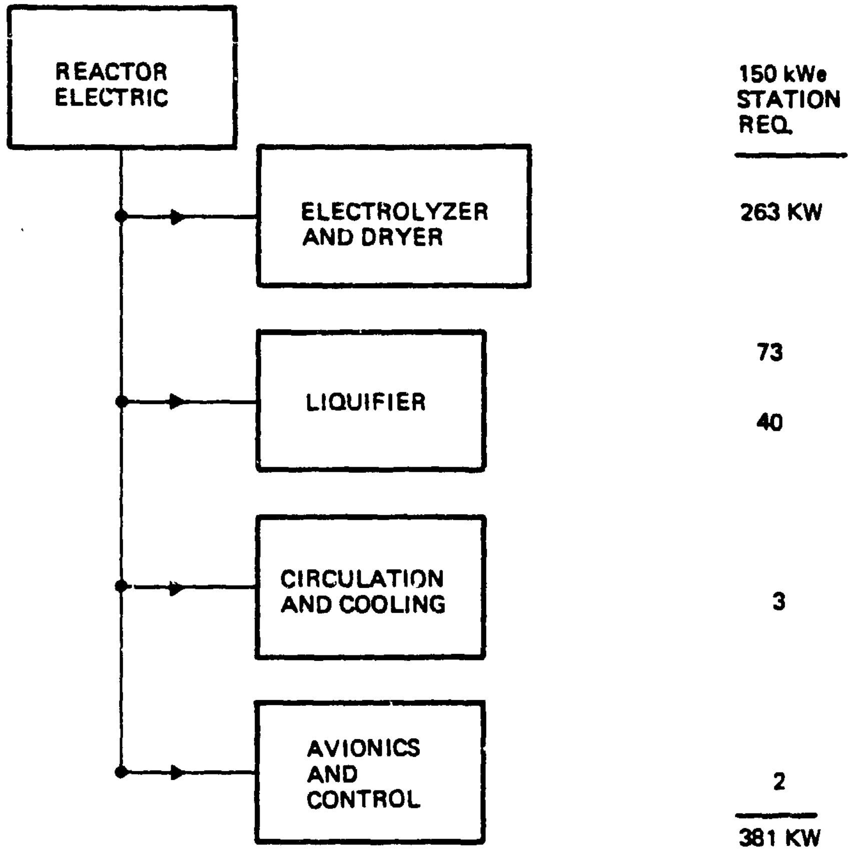

Figure 115 Free Flyer Reactor Power Flow Distribution

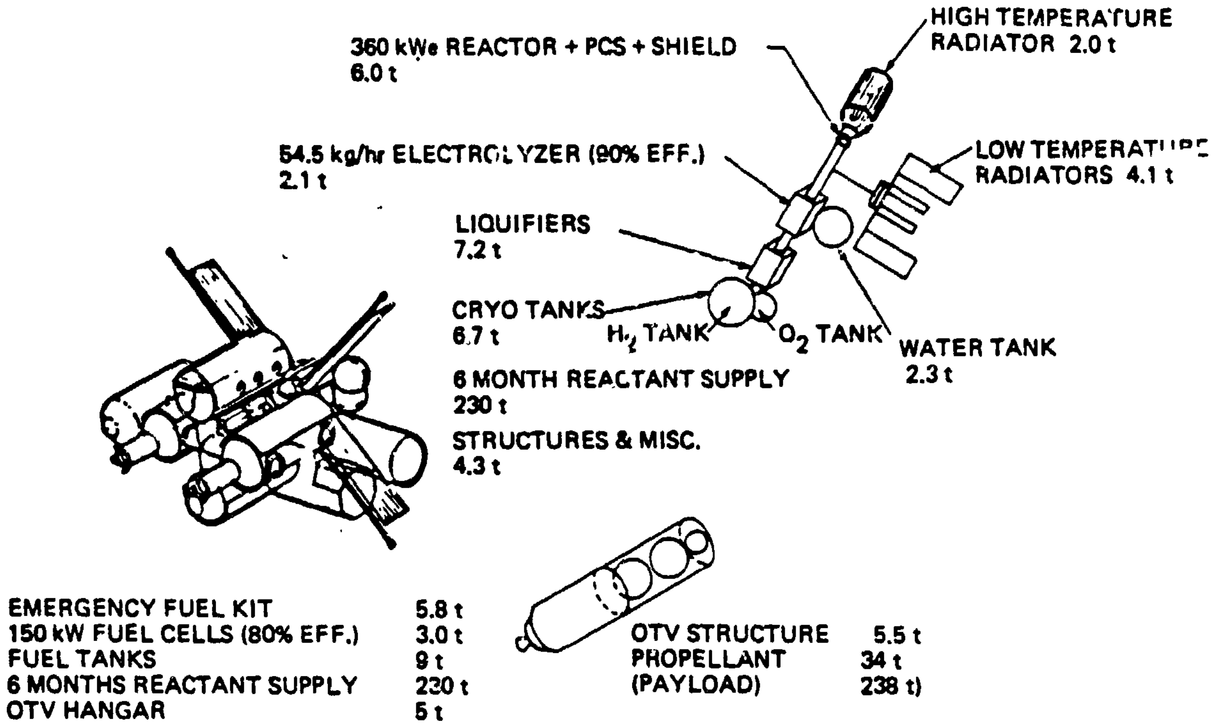

Figure 116 Mass of 150 kW Free Flying Reactor

	150 kWe		300 kWe	
Element	Mass (t)	Volume (m^3)/ Area (m^2)	Mass (t)	Volume/ Area
At Station				
LH_2 & LO_2 or H_2O	230	—	460	—
Tanks				
LH_2	2.6	366	5.2	732
LO_2	4.1	229	8.2	458
H_2O	2.3	230	4.6	460
Fuel Cells	3.0	5	6.1	9
OTV Hangar	5.0	283	10.0	566
Miscellaneous	1.3	3	2.0	6.5
SBOTV	5.5	207	11.0	414
At Reactor				
Electrolyzer/Dryer	2.1	5.25	2.8	6.8
Liquifier				
H_2	5.0	12	6.0	24
O_2	2.25	3	2.2	4.5
Low Temp Radiators				
Electrolyzer	1	341 m^2	2	682 m^2
Dryer	0.3	113 m^2	0.6	226 m^2
H_2 Liquifier	2.3	807 m^2	4.6	1614 m^2
O_2 Liquifier	0.5	165 m^2	1.0	330 m^2
Structure	3.0	—	4.5	—
Miscellaneous	1.3	3	2.0	4.5
LH_2 & LO_2 or H_2O	230	—	460	—
Tanks				
LH_2	2.6	366	5.2	732
LO_2	4.1	229	8.2	458
H_2O	2.3	230	4.6	460

Figure 117: Mass And Volume of Fuel Cell System

Figure 118 *Makeup Fuel Requirements for Free-Flyer Tanker*

	150 kWe per 6 months (t)	10 years (t)
Propellant	34	680
Boiloff	13.8	276
Drag Makeup	0.007	0.1
Total	(78	956

Figure 119 *Free-Flyer Logistics—Initial Operating Conditions*

Element	STS	SBOTV
H_2O	18.4	
Tanks		
LH_2	2.8	2
LO_2	2	
H_2O	2	
Reactor + PCS + Shield	0.24	—
Fuel Cells	0.15	—
Radiator	0.15	—
OTV Hangar	0.5	—
Miscellaneous	0.2	—
SBOTV	1	—
Electrolyzer/Dryer	0.1	
Liquifier		
H_2	0.2	
O_2	0.1	0.1
Free-Flyer Low Temp Radiators	1	
Structure	0.2	
Total Flights	28.6	2.1

assumed to be a maximum of 25 t or 300 m^3 to a 500 km station. Load factors were evaluated by determining whether the mass or volume of an element "filled" the shuttle's payload bay. For example, water was mass limited while the tanks were volume limited. SBOTV loads were determined by mass only. Figure 120 lists the number of launches required for each element for initial operating conditions.

To replace propellant and other losses, water must be launched from earth to the station. Based on the load factors described above, two STS flights are required every six months for water delivery and 40 STS flights are required over the 10 year system life. Figure 120 lists the lifetime logistics summary for the free-flyer configuration.

8.8.11 Reduced Logistics Alternatives

Because of the magnitude of the reactant requirement discussed above, multiple free-flyer configurations were considered. One configuration considered was to place N (two or more) free-flyers in orbits of equal altitude but separated by 360/N degrees. As a point design five equally spaced free-flyers in 700 km orbits was selected with a 150 kWe station power requirement. The time between transfer opportunities with a free-flyer was 102 days (510 days between transfer with any given free-flyer). This reduced the V required by a factor of 2.7. Although this approach reduced the number of cumulative STS flights by 18, it increased the number of OTV flights by 23. Figure 121 lists STS logistics for this configuration. Because this configuration did not significantly alter the logistics of the free-flyer, multiple reactor configurations were not further considered.

The possibility of using an unmanned, shuttle-derived launch vehicle with greater payload capacity than the STS was considered. This launch vehicle is shown in Figure 122. The Unmanned Launch Vehicle (ULV) uses STS propulsion, but not an orbiter. The engines and avionics are contained in a recoverable module under the External Tank. The payload is positioned above the tank. The baseline two-engine configuration has a payload volume 27 m long and 7.6 m in diameter. This is three times the STS cargo volume. The payload capability is 54 tonnes to 500 km circular orbit, which is 2.16 times the STS payload projected to that altitude.

The larger available volume allows all flights to be mass-limited rather than volume-limited. In the 150 kWe case, the number of launches is reduced from 29 to 10.5 for the initial operation. The 10 year cumulative launches are reduced from 68 to 24.

The cost per flight of the ULV is estimated to be $66 million, compared to $100 million for the STS. Thus the cost advantage for the ULV is more than four to one.

Note that the number of launches for the other options would also be reduced by the ULV. The on-board, man-rated tether, and instrumented tether options would require 0.6, 1.6, and 1.2 ULV flights respectively.

Three alternatives for transporting reactant from the reactor to the space station are chemical, resistojet, and ion propulsion OTV's. For comparison purposes, the payload requires more V than an impulsive burn. Figure 123 gives details of the three options.

Figure 120 *Lifetime Logistics for Free-Flyer Reactor*

	IOC	Life	Lifetime Total
STS	29	40	69
SBOTV	2	20	22

Figure 121 *Logistics for Five Free-Flyer Reactors*

Logistics for IOC	
Number of STS flights	36
Number of OTV flights	6–10
Number of OMV flights	5
Cumulative logistics	
Number of STS	50
Number of OTV	40–50
Number of OMV	5

Figure 123 *Alternative Propulsion Modes*

		Propulsion Mode		
Element	Unit	Chemical	Resistojet	Ion
Propelaint	—	Oxygen-Hydrogen	Hydrogen	Argon
Payload mass	tons	238	238	238
Total mass	tons	284	284	284
Specific impulse	seconds	480	850	2500
Mission ΔV	m/s	600	848	848
Mass raito tio	m_i/m_f	1.136	1.1071	1.0352
Propellant mass	tons	34	27.5	9.65
Propulsion specific power	W/kg	25830	60	28.5
Propulsion total power	MW	310	1.11	1.035
Thrust	N	132000	222	26
Burn time	days	.014	8.5	105

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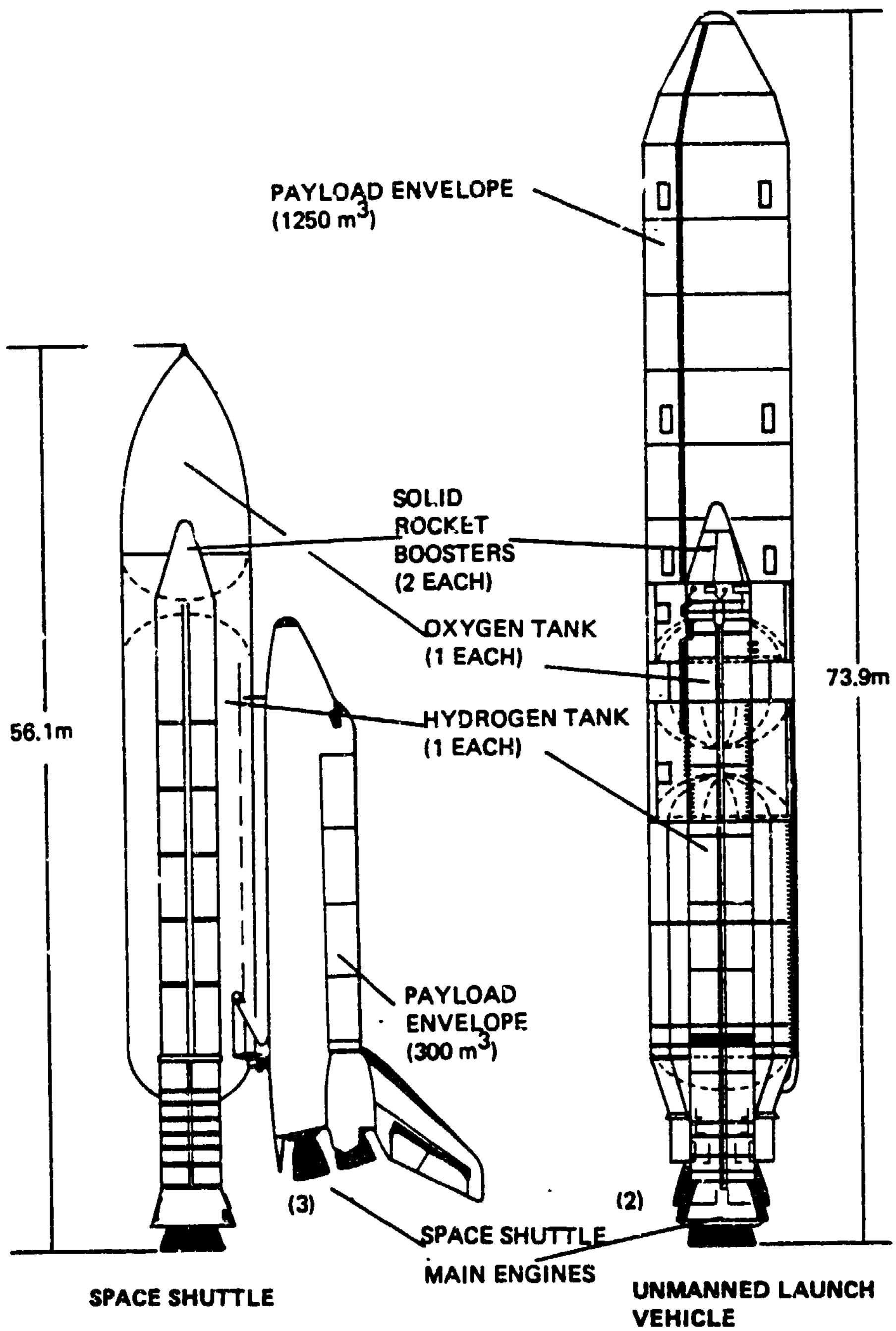

Figure 122 Unmanned Launch Vehicle—STS Derived

8.9 System Trade Summary

The results of the trade studies described in Sections 8.6-8.8 above are summarized in Figure 124. Because of the large tanker size and startup fuel inventory, the free-flyer reactor option requires 29 orbiter flights to get the system started. Once the system starts up, this option still requires two round trip OTV flights per year for fuel transfer plus four STS flights per year to provide propellant to the OTV. Selecting this option would result in considerable space station traffic and essentially a dedicated orbiter and orbital transfer vehicle for fuel production and delivery. There are alternatives available for reducing the number of shuttle flights, but these involve either a new launch vehicle or more OTV flights and do not change the basic conclusions - that the free-flyer reactor involves many flights on a continuing basis.

The boom-mounted reactor system requires 1.3 STS flights to place all the equipment in orbit. This includes not only the power generation equipment, but also the end of life booster vehicle and an emergency power kit. An orbital maneuvering vehicle is already being planned as a part of the space station system, so it can be expected to be in place by the mid-1990's. It also seems reasonable that some form of emergency power kit will be available during the early space station period. As Figure 99 shows, without these two items the entire boom-mounted reactor power system can be delivered to the space station in a single STS launch. The additional payload delivery to orbit for drag makeup over the reactor lifetime is trivial for the boom-mounted reactor case, even including that required for the space station itself.

The tethered reactor requirements lie between those of the other two cases. A tethered reactor system with an instrument-rated shield can be delivered in about two full STS flights and one OMV flight, with another 855 kg/year of water to be used for orbit makeup propellant. A man-rated shield would increase the mass by 17,800 kg, or 0.7 full orbiter bay equivalents, for the possibility of some manned maintenance of the power generation system.

8.10 High Power Scaling

The data collected for this study have been generated parametrically. When a specific power level was needed, such as to generate the mass and volume data of the last four subsections of this report, a space station power of 150 kWe was selected. To investigate the three system configurations at a higher power level, the same trade studies were performed at 300 kWe. The results are shown in Figure 125.

The power range of the reactor core used for this study extends to about 4 MWt. At 5.1% efficiency, which is projected for current thermoelectric conversion technology, this limits the electrical output of a single reactor to about 200 kWe. In the free-flyer case at 150 kWe, this necessitated treating the reactor as a pair of baseline reactors on a single spacecraft. For the higher power levels considered here, a higher efficiency power conversion system was selected. This was taken to be a Stirling cycle for this report, although a Brayton cycle should yield very similar results. With a power conversion efficiency of 25%, the reference reactor with a Stirling engine is then able to provide 1000 kWe, instead of the 200 kWe provided by the baseline thermoelectric system.

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	ON-BOARD	TETHER (MAN RATED)	TETHER (INSTRUMENT RATED)	FREE FLYER
INITIAL MASS IN ORBIT	31.5 t	64.7 t	46.9 t	564 t
INITIAL VOLUME IN ORBIT	236 m^3	368 m^3	361 m^3	2790 m^3
STS LOGISTICS FOR IOC				
NUMBER OF ORBITER FLIGHTS	1.3	2.6	1.9	29
NUMBER OF OTV FLIGHTS	0	0	0	2
NUMBER OF OMV FLIGHTS	0	1	1	1
10-YEAR CUMULATIVE MASS	32.0 t	73.2 t	55.5 t	1296 t
10-YEAR CUMULATIVE VOLUME	237 m^3	377 m^3	370 m^3	3560 m^3
CUMULATIVE STS LOGISTICS				
NUMBER OF ORBITER FLIGHTS	1.3	2.9	2.2	68
NUMBER OF OTV FLIGHTS	0	0	0	22
NUMBER OF OMV FLIGHTS	1	2	2	1

Figure 124 System Trade Table. 150kWe at Space Station

	ON-BOARD	TETHER (MAN RATED)	TETHER (INSTRUMENT RATED)	FREE FLYER
INITIAL MASS IN ORBIT	31.1 t	77.8 t	61.1 t	1043 t
INITIAL VOLUME IN ORBIT	241 m^3	428 m^3	421 m^3	4640 m^3
STS LOGISTICS FOR IOC				
NUMBER OF ORBITER FLIGHTS	1.5	3.2	2.7	49.5
NUMBER OF OTV FLIGHTS	0	0	0	5
NUMBER OF OMV FLIGHTS	0	1	1	1
10-YEAR CUMULATIVE MASS	31.9 t	94.2 t	77.7 t	2964 t
10-YEAR CUMULATIVE VOLUME	242 m^3	448 m^3	441 m^3	6752 m^3
CUMULATIVE STS LOGISTICS				
NUMBER OF ORBITER FLIGHTS	1.5	3.9	3.4	126
NUMBER OF OTV FLIGHTS	0	0	0	45
NUMBER OF OMV FLIGHTS	1	2	2	1

Figure 125 *System Trade Table. 300 kWe at Space Station*

The boom-mounted reactor supplies 306 kWe to the transmission lines. These lines are assumed to be two sets of four wires each, supplying three-phase alternating current power at 115 volts. The optimum boom length is 55 meters, with the complete transmission line set weighing 650 kg. The mass of the reactor and power conversion system is 2600 kg and the shaped four-pi shield mass is 16,500 kg. Even though the 300 kWe reactor is closer to the space station (55m) than the 150 kWe reactor (70m), the shield mass of the high power case is somewhat lower. This is a consequence of the higher efficiency assumed for the high power case; 153 kWe is generated from 3000 kWt at 5.1% efficiency, while 306 kWe is generated from only 1224 kWt at 25% efficiency.

The radiator mass is 850 kg for the boom-mounted reactor. This is for a primary radiator that rejects 887 kWt at 640 K and a secondary that radiates 31 kWt at 343 K. The boom structure weighs 160 kg.

The total mass in orbit for the 300 kWe boom-mounted reactor is 31,100 kg. As mentioned above, the reactor, power conversion system, man-rated shield, radiator, transmission lines, and boom structure requires one full shuttle launch. The emergency power kit and OMV requires 41% of a second flight.

The 300 kWe tethered reactor power output is 416 kWe, or 1664 kWt. The instrument-rated shield mass is 5610 kg and the man-rated shield mass is 17,100 kg. The radiators weigh 890 kg for the 640 K primary and 270 kg for the 343 K secondary. The reactor and power conversion system mass is 2900 kg.

Since the output power is twice that of the reference case, the reactant flow rate is doubled. Assuming linear scaling of the electrolyzer fuel cell, and fuel storage tank masses, these quantities are doubled. The water hose diameter is 1.26 cm, so the water in three hoses weighs 11,200 kg.

The total mass for the 300 kWe tethered reactor with a man-rated shield is 77,600 kg, which requires 3.2 shuttle launches. For an instrument-rated shield, these numbers are 61,100 kg and 2.7 shuttle launches, respectively.

The free-flyer reactor with a Stirling cycle requires 2880 kWt to produce 720 kWe at the reactor for 300 kWe of space station output. The reactor and power conversion system mass is 3500 kg and the shield mass is 780 kg. For each fuel delivery, two orbital transfer vehicles with the same specifications as those described in Section 8.8 are used. The total number of shuttle launches is about fifty for initial startup, and over one hundred over the ten-year lifetime. The total number of OTV flights is 45. The traffic rate is four orbital transfer vehicle round trips and 7.65 orbiter flights per year.

9.0 SYSTEM REQUIREMENTS

The study/analysis has resulted in establishing requirements for the nuclear electric power system, the space station, and the space transportation system. Since the electrical power system with the reactor as the energy source was to be considered with any candidate conversion device, requirements must therefore be general enough to encompass any of the conversion systems with any of the reactor placement configurations. The requirements for the various combinations and configurations will be grouped as much as possible so as to identify and treat all of them.

9.1 Nuclear Electric Power System Requirements

Figure 126 shows the tabulation of requirements for the nuclear electric system for the three conceptual designs:

- o On the space station
- o On a tether to the space station
- o On a free-flyer spacecraft

The grouping shows the static conversion systems and the dynamic conversion systems. Transmission line requirements will depend upon whether the line is DC or AC and whether the conversion equipment output is DC or AC. The static conversion equipment output is DC and power conditioning will have to be supplied to convert to AC or to another DC voltage level. Dynamic conversion equipment output is AC and can be designed for some levels and frequencies compatible with the transmission line. For high frequency, power conditioning will be required since the machines generally do not operate at the speeds required.

- o For Brayton cycle or Rankine cycle:
  Generator RPM = (Frequency in Hz) x 60 for a single pole pair (equivalent)
  At 20,000 Hz, the generator rotational speed is:
  1,200,000 RPM for 1 pole pair
  120,000 RPM for 10 pole pairs

- o For Stirling cycle:
  Linear generators, which would be paired with a Stirling (reciprocating) engine, are limited in frequency. To attain higher frequencies with the Stirling engine, the reciprocating motion would have to be converted to rotary motion to drive a conventional circular generator. An alternative would be to convert the frequency to a higher one with power conditioning.

The electrical power system reactor, shield, and conversion equipment is to be designed for placement in a long life orbit after damage or expended life. Provision is to be made for severing the reactive components from the non-radiative portions at a convenient interface. The replacement equipment shall be reconnected at the interface.

To provide monitoring and control of the reactor, a separate and self-contained power supply shall be provided. This power supply shall be highly reliable and shall provide that any malfunction of the reactor or conversion equipment shall be fail-safe. The power supply may be an energy storage device with recharge capability,

Configuration (Placement)	Conversion: Static	Conversion: Dynamic	Transmission	Thermal Control
On Space Station				
At CG	Full 4-PI shield for 5.72 mrem/hr			
(With RFC)		Condition to DC from AC, or generate DC for electrolysis	Provide voltage and frequency directly at distribution voltage and frequency (No transmission line)	Provide active cooling loop from conversion equipment to high temperature radiators. Place radiators away from station equipmnte, traffic, and EVA areas
		Provide counter-motion to cancel torque or vibration		
On structural boom	Shaped 4-PI shield for 5.7 mrem/hr toward station and 200 rem at 30 meters away from station		Provide high-voltage DC or AC transmission line, compatible with earth's plasma effect	Provide radiators at conversion equipment
		Provide counter-motion to cancel torque or vibration		Provide radiators at conversion equipment
(With RFC) Electrical Transmission		Condition to DC from AC or generate DC for electrolysis		
(With RFC) $H_2O/H_2/O_2$ Transmission			Provide H_2O and H_2 and O_2 lines to fuel cells in station. Electrolyzer near reactor (No electrical transmission line)	Provide radiators at conversion equipment and for electrolyzer on boom

Figure 126 *Nuclear Electric System Requirements*

Configuration (Pacement)	Conversion: Static	Conversion: Dynamic	Transmission	Thermal Control
On Tether (With RFC power system in station) Electrical transmission to station	Shaped 4-PI shield for 5.7 mrem/hr toward station and 200 mrem/hr at 30 meters on side away from station. For very long tethers (e.g., 35 km) shielding will be designed for 200 rem/hr at 30 meters on all sides		Provide high-voltage DC or AC transmission line is to be compatible with earth's plasma	Provide radiator at conversion
	DC output	Provide counter-motion to cancel torque or vibration Condition AC to DC or generate DC for RFC electrolysis		
(With RFC power system in station) Non-electrical, gaseous/fluid transmission line			Provide hoses for water flow from space station to electrolyzer at end of tether, and gaseous H_2 and O_2 returning to station	

Configuration (Placement)	Conversion: Static	Conversion: Dynamic	Transmission	Thermal Control
On Free-Flyer (With RFC power system in station) Tanker transportation of liquids between LEO station and free-flyer at high altitude orbit	Minimal shield for reactor; instrument rated.		With tanker transportation of liquids between LEO station and high altitude free-flyer. H_2 and O_2 are to be liquified at electrolyzer station, for transporation to space station	Provide radiator at conversion equipment
	DC output	Provide counter-motion to cancel torque or vibration		
		Provide DC for RFC electrolysis.		
	Size reactor and conversion equipment to operate electrolyzer and refrigerator to liquify H_2 and O_2			

Figure 126 *Nuclear Electric System Requirements (Cont'd)* and shall be rated for 21 days of operation. An alternative power supply may be an APU.

Safety Requirements - The nuclear reactor, shield, and conversion equipment shall meet safety requirements given in guidelines and nuclear safety requirements documents. A list of recommended documents and guidelines is given in Figure 127.

Environment - The nuclear electric power system shall operate in all the environments associated with the altitude and orbits specified. Of special interest are the specific items such as the Earth's plasma, radiation, micrometeoroids, and contamination due to exhausts, outgassing, and ejecta.

To survive the launch environment the nuclear electric system must be mounted and supported to protect tubing and thin structures from vibration.

9.2 Space Station Requirements Due To Nuclear Electric Power System

The space station requirements due to the nuclear electric power system are involved with the design and operation of the space station.

9.2.1 Mounting and Installation

Provision is to be made for mounting the nuclear reactor, shield, conversion equipment, radiators, power equipment, and the control and monitoring components. Also included is the emergency power system and mounting, so as not to compromise the design for reentry in emergency conditions or after a cooling period.

Installation Options

A) On space stations - A boom is to be provided to support the reactor, conversion equipment, shield, power conditioning, radiator, and transmission line when the reactor is mounted on the station. The mount and structure are to be designed so as to facilitate maintenance, repair, and replacement. Provision is to be included for disposal of the reactor to a high altitude orbit.

B) On tether - A tether is to be provided to support the reactor, conversion equipment, shield, power conditioning, radiator, and transmission line. For electrical transmission, the tether shall be the electrical cable assembly which can support the operational loads. For non-electrical transmission, the tether shall be suitable hoses for transporting water to the electrolyzer and gas hoses for transporting H_2 and O_2 to the station. Reinforcements may be used, if required. The tethered platform which contains the electrical power system components shall also have redundant attitude control and communication/telemetry equipment which is powered by the on-board electrical system. Provision shall be made in the design for mounting a propulsion engine for parking the reactor at a high altitude. This shall be redundant so as to be reliable.

C) On a free-flyer - The free-flyer vehicle shall include mounting of the electrical power system components, and shall have on-board propulsion attitude control, and communications. These systems shall be redundant so as to provide reliable operation.

- OSNP-1, Nuclear Safety Criteria and Specifications for Space Nuclear Reactors

- 10CFR20, Standards for Protection Against Radiation

- 10CFR50, App. A, General Design Criteria for Nuclear Powerplants

- DOE Order 5480.1A, Environmental Protection, Safety, and Health Protection Programs for DOE Operations

- NHB 1700.7A, Safety Policy and Requirements for Payloads Using the Space Transportation System (STS)

- JSC 11123, Space Transportation System Payload Safety Guidelines Handbook

Figure 127 Nuclear Safety Requirements Documents

9.2.2 Mission Operation

Since the mission operation is controlled by the space station operators and ground control, the traffic control near a reactor, the conversion equipment, and the radiators will be under the operator's jurisdiction. It will be required that traffic near the harmful components be monitored carefully since there are no lane markers in space. The flight operation personnel will control traffic so that vehicles do not venture too close to the reactor for any excessive length of time. It will be the responsibility of the operators to comply strictly with safety regulations.

Maintenance and repair are the responsibility of the space station maintenance crew. Persons servicing the nuclear system shall be schooled and trained for familiarity with nuclear technology and the safety regulations. In the event of an accident to the reactor in which a dangerous level of radiation is expected, provision will be made for a "radiation-proof" shelter in which personnel can isolate themselves safely for an emergency period.

The control of remote equipment on tethered platforms or on free-flyer will be under control of the space station traffic controllers. Requirements applicable to the nuclear electrical power system will be prepared for and submitted to the traffic controllers. Orientation of spacecraft radiators will be controlled by the mission operators who orient the space station. Communication with the remote spacecraft will be controlled by the traffic controllers.

When station keeping or attitude control is required, it is the responsibility of the space station operating personnel to provide it.

Installation of EPS equipment will be the responsibility of the space station maintenance personnel. Provision will be made for the installation of additional or different electrical equipment throughout the vehicle. Growth will be provided for where possible. Cabling will be fastened securely to the cable run structure so as to prevent buckling or loosening of the structural members under fault conditions. Interface connections specified by the electrical power designers will be provided.

9.3 STS Requirements Due To Nuclear Electric Power System

STS requirements due to the nuclear electric power system are:

- To provide an uncontaminated environment free of debris which can affect the electrical system.
- To provide orbital vehicles for placing the reactor, shield, and conversion equipment into position on the boom or tether.
- To provide an OMV for moving the reactor and shield into a high, long-life orbit after the end-of-life or in the event of a malfunction.
- To provide an OTV for transporting liquids between the space station and the free-flyer vehicle.
- To place the free-flyer in its specified orbit.

10.0 CONCLUSIONS

This study identifies the applicability of a nuclear reactor electrical power system to the space station. At power levels above 100 kWe, and primarily at the levels between 150 and 300 kWe, the nuclear reactor power system has a competitive position because it offers the following:

Energy density which remains high as the power levels increase. Since there is no requirement to provide for recharging of batteries, fuel cells, flywheels, or thermal salts as for solar panel oriented systems, the system weight position improves when compared with solar powered systems.

10.1 Reactor Viability As A Source Of High Power

When analyzing the loads, we found that loads larger than the original NASA Standard Reference Set were feasible. Electrical loads can be identified to as high as 400 kWe. Recent analyses by NASA and industry showed that the housekeeping loads have grown to 40-50 kWe and additional high-power loads have been uncovered. From this we can assume that the space station loads will be greater than we were originally given in the mission set. Historically, loads always increase when the preliminary design becomes a detailed design and the loads are examined in depth.

10.1.1 Mission Power Requirements

Because it is a compact energy source, the nuclear reactor is an excellent candidate for high power applications. As the loads grow, the reactor/shield/conversion package remains essentially at the same level of weight within a specific range. In manned space stations the total spacecraft loads are higher because the life support system power level is substantial.

Analysis of the mission loads shows that there are numerous scientific and commercial loads which require high power levels. The commercial loads are characterized by growth with time as the market develops and the product sales increase. In Section 2.1 the details of the loads were defined.

10.1.2 Benefits

With a nuclear reactor energy source the high power requirement can be fulfilled with minimal penalty for drag and orientation. The thermal radiators can be streamlined into the orbit plane so as to reduce drag, thus saving station keeping (orbit maintenance) fuel. Because of its independence from the sun, the lack of a need to recharge a thermal storage element will reduce the power system to the size of the load requirement, plus losses and degradation.

For those loads which require heat directly, the thermal interface can be established so that the heat is not derived from heat-to electricity-to heat.

10.2 Regenerative Fuel Cell Subsystem Attractiveness

Of the options examined, the regenerative fuel cell is an attractive candidate because it avoids some problems when it is linked directly to the reactor power. In that option, the reactor will power the electrolyzer, which separates the H_2 and O_2

from the water, the fuel cell output. It is much simpler to transmit the liquid and gases than to cope with an electrical power transmission line operating in the hostile environment of the Earth's plasma and contamination from outgassing, exhaust, and debris.

The fuel cell is an excellent candidate for energy storage as well as a prime power converter. It's efficiency is high, has operational background, is relatively reliable because it has been improved over the years of use, first as an open-cycle cell, then as a closed-cycle system in the Apollo program.

10.3 Evaluation of Options

Three reactor/space station configurations were evaluated and the economics parameters show that:

- o The reactor on the structural boom requires the fewest STS flights.
- o With the reactor operating in a 300-year orbit requires 68 shuttle flights, for a 150 kWe power load. The quantity of flights is influenced by the provision for fuel for the tanker vehicle.
- o When the reactor is on a fixed boom or on a tether, the reactor is close enough to the space station and at a low altitude where it requires boosting to higher, long-life orbit.

11.0 RECOMMENDATIONS

A. It is recommended that an analysis be made of concepts which allow for space station designs to provide for growth in the power system with time. The on-board distribution system should be designed to accommodate AC although the early station may revert to DC. To change from solar generated power to nuclear generated power maximum use should be made of the original on-board electrical power system equipment, and how the original equipment can be enlarged by adding modules.

The design and operation should be examined to determine how the original solar array can be disposed of or stowed when the growth power system is substituted. If stowed, it can be the emergency power system.

B. Since SP-100 reactor design is not directed to manned space stations, the reactor and shield technology will have to be researched to develop concepts of disposal and reentry. Venting will have to be changed for the man-rated design. This will affect the reentry package design.

C. A more detailed study with quantitative data should be made of some of the bypassed options. Because of the absence of the electrical transmission line, the EPS design would be simple provided that there is a simple way to transfer the heat to the thermal control radiators.

D. The placement of the reactor in the high-altitude orbit should be given a further analysis for reduction of the quantity of trips by increasing the number of tankers, and increasing the number of free-flyers with reactors, electrolyzers, and refrigerators.

With a larger transport vehicle the quantity of operational flights also could be reduced. Combining the larger tankers with the increased number of free-flyers might decrease the number of shuttle launches in a 10-year cycle to an acceptable quantity.

A number of shuttle flights are involved with providing water as makeup for the system when the tanker uses a large quantity of fuel. An analysis can be made of the logistics of transporting the water to orbit on a space-available basis. This would reduce the assigned cost for the power system.

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