NASA Conference Publication 2058

Future Orbital Power Systems Technology Requirements

A symposium held at
Lewis Research Center
Cleveland, Ohio
May 31 and June 1, 1978

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National Aeronautics
and Space Administration

Scientific and Technical
Information Office

1978

PREFACE

With the advent of the space shuttle as a transportation system, NASA is actively involved in program planning for missions requiring several orders of magnitude more energy in orbit than in the past. Therefore, a two-day symposium was held at the NASA Lewis Research Center, Cleveland, Ohio, to review the technology requirements for future orbital power systems.

The purpose of this meeting was to give leaders from government and industry a broad view of current government-supported technology efforts and future program plans in space power. It provided a forum for discussion, through workshops, to comment on current and planned programs and to identify opportunities for technology investment. To lay the foundation for the discussions, survey papers were presented to review the technology status and the planned programs.

Workshop groups were small, yet they contained more than enough expertise for lively and rewarding discussions. The free and informal exchange of ideas and opinions made the meeting highly successful.

CONTENTS

OAST SPACE POWER TECHNOLOGY PROGRAM

Jerome P. Mullin
NASA Headquarters

The NASA Space Power Technology Program is aimed at providing a sound technological basis for future space electrical power systems. While future needs for electrical energy in space cannot be known with certainty, an analysis of programmatic trends and opportunities now under study identify two classes of need. The first is for very high performance systems to support electric propulsion and ambitious geosynchronous missions. The second is for very high power levels at low cost to support the Shuttle-based habitation and use of near-Earth space. In this paper the current R&T base program is first described, then special attention is directed toward outlining a new system technology initiative specifically oriented toward providing the utility power plant technology base for semi-permanent Earth orbital facilities expected to be needed in the middle to late 1980's.

The R&T base program involves five areas of research: Photovoltaic Energy Conversion, Chemical Energy Conversion and Storage, Thermal-to-Electric Conversion, Environmental Interactions, and Power Systems Management and Distribution. The general objectives and planned direction of efforts in each of these areas is summarized below and in Figures 1 through 15.

In the area of Photovoltaic Energy Conversion, the aim is to improve conversion efficiency, reduce mass, reduce cost, and increase operating life of photovoltaic converters and arrays. Emphasis is being placed on very efficient thin solar cells, lightweight blankets, radiation resistance, low cost and advanced cells, and both planar and concentrator array designs.

In Chemical Energy Conversion and Storage, the objective is to achieve improved energy density, life, operational capability, and cost of space battery and fuel cell systems. Research is being done to increase energy density and life of nickel cadmium batteries and to increase their capacity; to achieve high energy density probe and lander batteries; and to evaluate new electrochemical concepts for very high energy density. In addition, effort is being initiated to evolve the fuel cell-electrolyzer concept for high capacity low-Earth-orbit energy storage applications.

Thermal-to-Electric Conversion efforts aim at technologies which can be used with either nuclear or solar heat sources and focus on achieving acceptable efficiencies in thermoelectric and thermionic converters and on evaluating Brayton systems for low and high power application. In addition, some work on ancillary equipment necessary for system feasibility is carried out.

Research is also undertaken to insure that future power systems can adequately cope with the space environment. This work includes both ground and flight efforts on spacecraft charging and on high voltage - space plasma interactions.

Finally, work aimed at providing the power system management and distribution basis of future systems is undertaken. This effort includes basic high power component, circuit and subsystem research, automated management and ground and flight systems investigations.

It is concluded that execution of this R&T base program will increase the range of future mission opportunities that can be accommodated at acceptable levels of cost and risk. However, the pressing near term high power needs for Earth orbital systems are very great, and the impact of this R&T base program on them at present resource levels will necessarily be limited to a few key technologies. Expansion of technology effort in this area can be accommodated only by reducing efforts aimed at high performance and longer term space power needs or by seeking a focused augmentation of resources.

A specific initiative aimed at this class of needs is outlined in Figures 16 through 29 and is presented for planning purposes only in an effort to illustrate the type of technology preparation that is viewed as being reasonably consistent with orbital energy needs of the near future. The need for this initiative stems from the projected growth in space energy demand and historical evidence suggesting that past experience has been both limited in quantity and costly. The technology initiative is presented in two phases. The first treats solar power generation, bulk energy storage, and power management and distribution. The second phase deals with thermal management and with space to space transmission of power.

The expected return from the type of technology program outlined here is the provision of enabling technology for a class of space powerplants in the multi-hundred kilowatt power range. We can expect to see dramatic reductions over projected capital and operating costs and begin to see new operating concepts involving maintainability, automation, the remote transmission of space power and the beginnings of truly integrated systems operation in space.

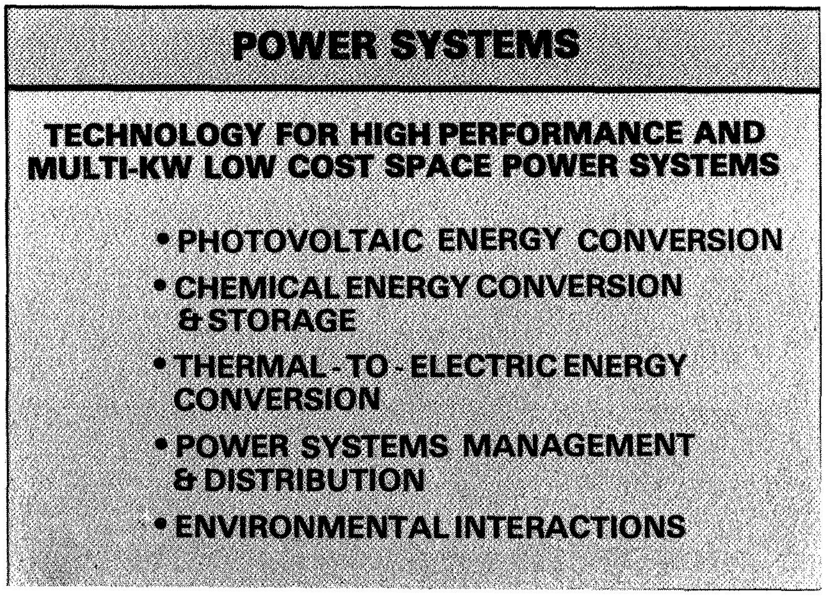


Figure 1.

CURRENT POWER PROGRAM

PHOTOVOLTAIC ENERGY CONVERSION

- *SOLAR CELLS*
- LOW COST BLANKETS & ARRAYS
- LIGHT WEIGHT BLANKETS & ARRAYS

CHEMICAL ENERGY CONVERSION & STORAGE

- NiCd BATTERY
- ADVANCED BATTERIES
- HIGH CAPACITY ENERGY STORAGE

THERMAL TO ELECTRIC CONVERSION

- THERMIONICS & THERMOELECTRICS
- BRAYTON
- THERMAL-TO-ELECTRIC POWER SYSTEMS *TECHNOLOGY*

ENVIRONMENTAL INTERACTIONS

- HIGH VOLTABE/SPACE PLASMA
- SPACECRAFT CHARGING

POWER MANAGEMENT & DISTRIBUTION

- HIGH POWER COMPONENTS
- CIRCUITS & SUBSYSTEMS
- AUTOMATED POWER SYSTEMS MGMT.
- POWER SYSTEMS & REQUIREMENTS

FLIGHT EXPERIMENTS

- SEPS ARRAY
- 5 LDEF EXPERIMENTS
- SOLAR CELL CALIBRATION FACILITY

Figure 2.

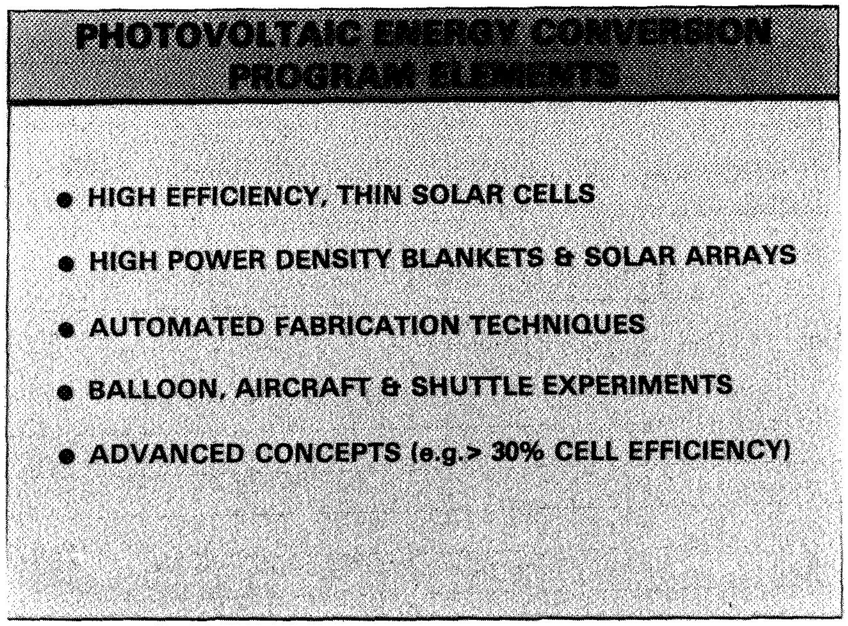


Figure 3.

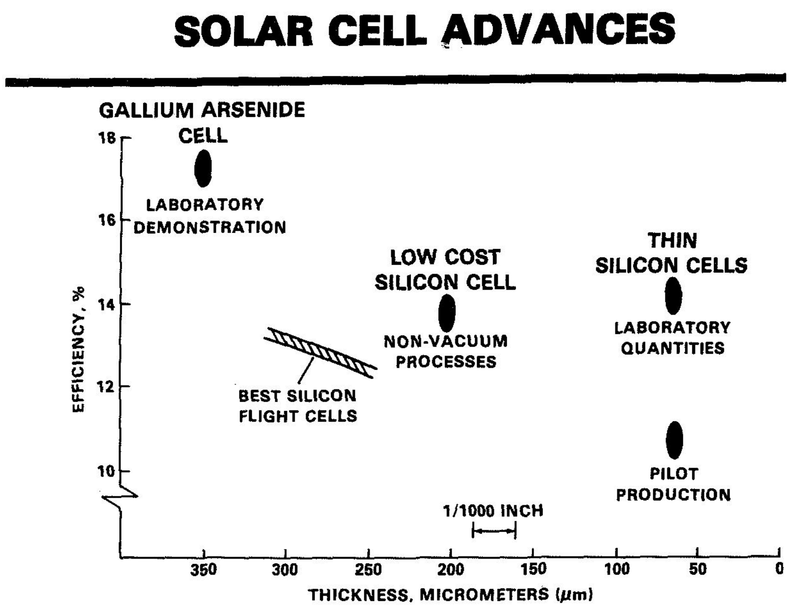


Figure 4.

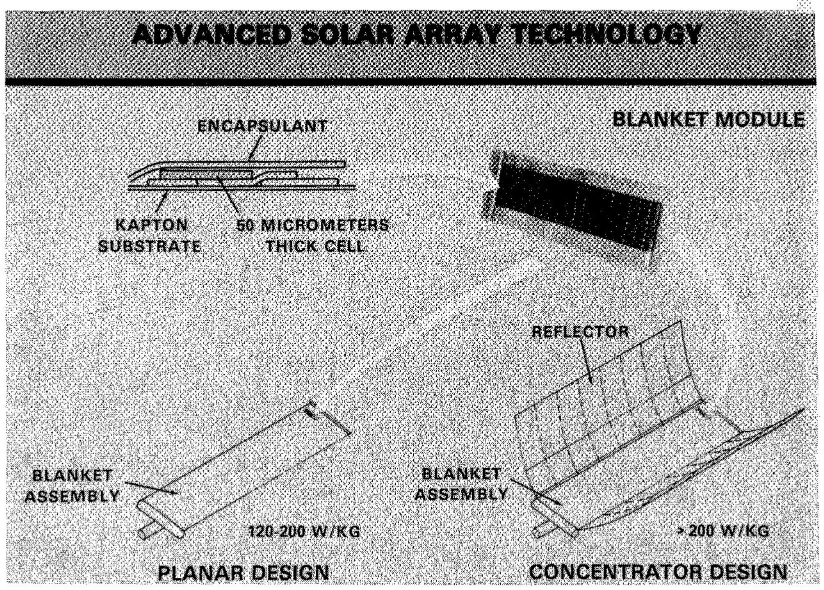


Figure 5.

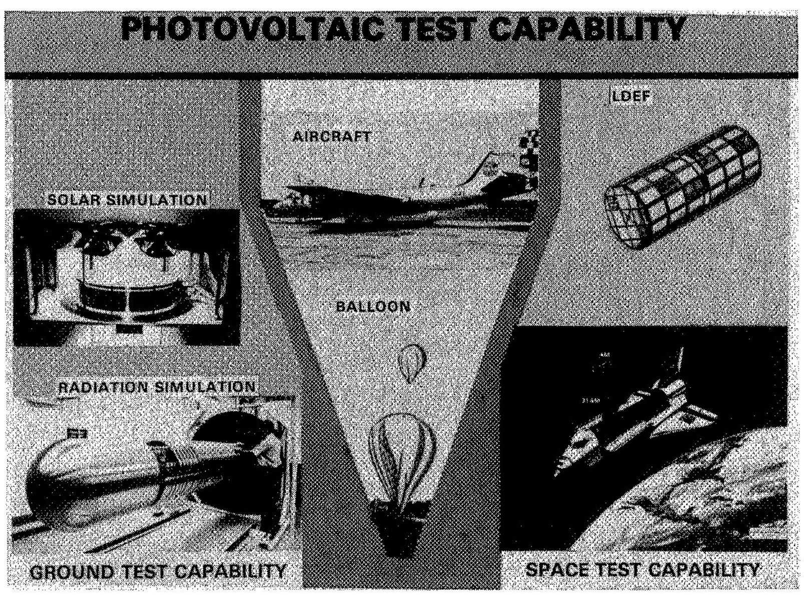


Figure 6.

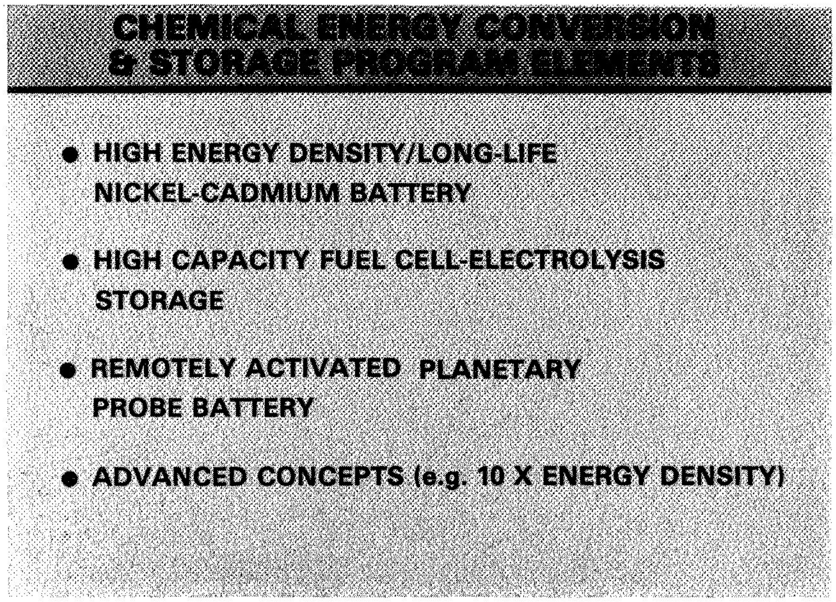


Figure 7.

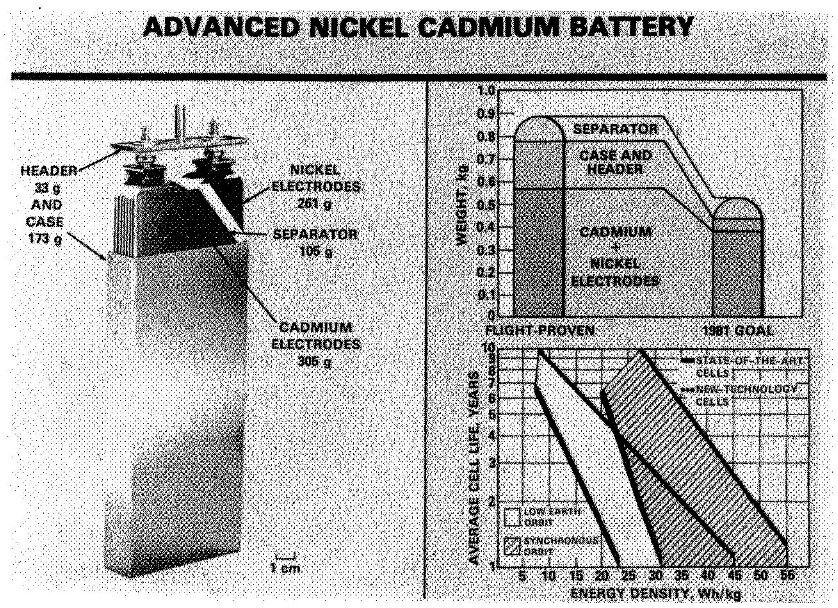


Figure 8.

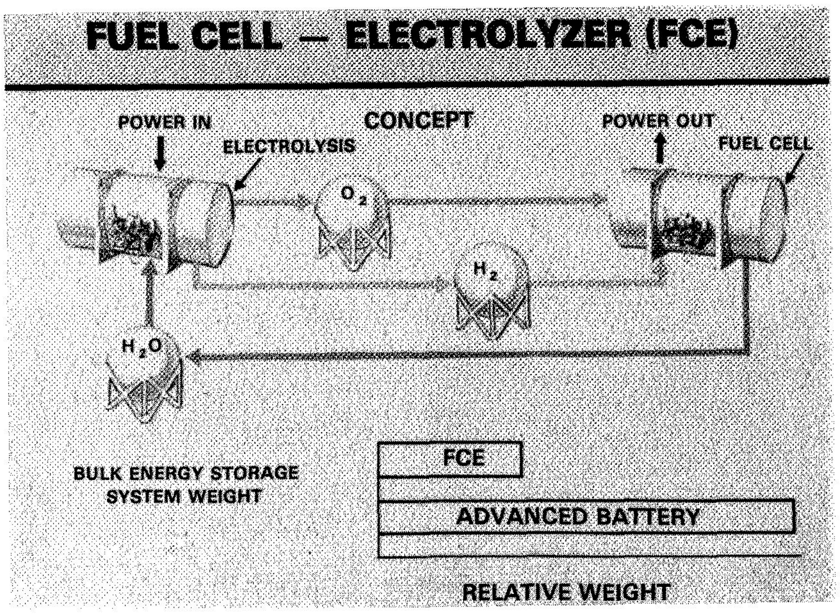


Figure 9.

THERMAL-TO-ELECTRIC ENERGY CONVERSION PROGRAM ELEMENTS

- THERMOELECTRIC ELEMENT EVALUATION
- BRAYTON POWER SYSTEMS
- HIGH TEMPERATURE THERMIONIC CONVERTER/HEAT PIPE
- HIGH TEMPERATURE MATERIALS
- ADVANCED CONCEPTS (e.g. > 15% THERMOELECTRICS & SOLAR BOILER/MHD)

Figure 10.

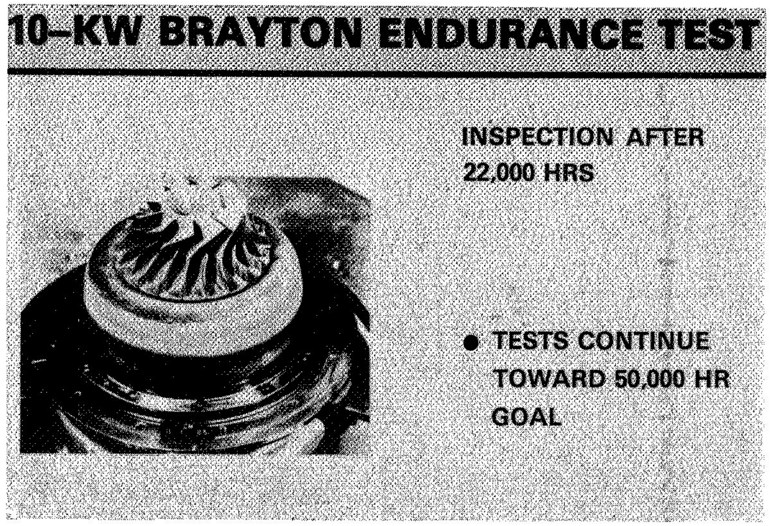


Figure 11.

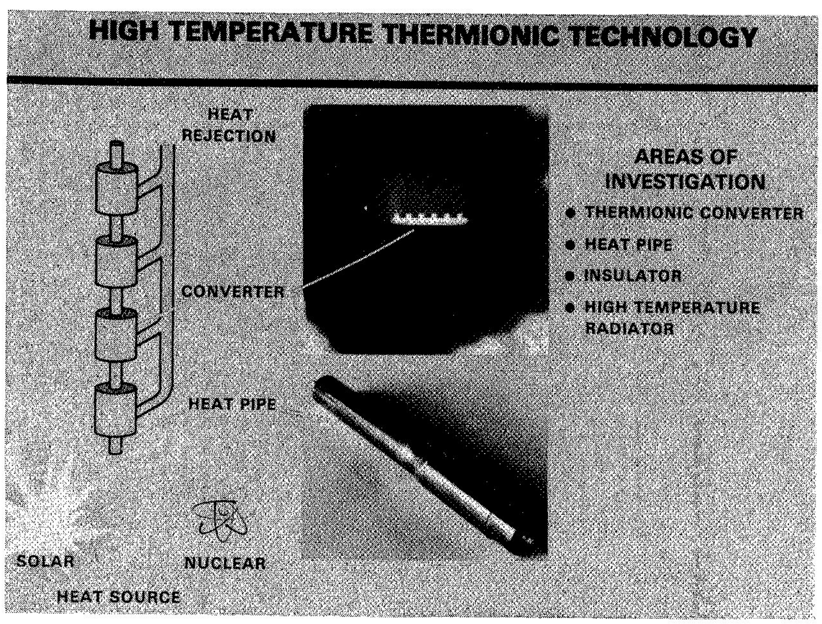


Figure 12.

ENVIRONMENTAL INTERACTIONS PROGRAM ELEMENTS

- JOINT AF/NASA SPACECRAFT CHARGING
- HIGH VOLTAGE/SPACE PLASMA
- SPACE EXPERIMENTS

Figure 13.

POWER SYSTEMS, MANAGEMENT & DISTRIBUTION PROGRAM ELEMENTS

- AUTOMATED POWER SYSTEMS MANAGEMENT
- HIGH VOLTAGE, HIGH POWER COMPONENTS & DISTRIBUTION
- USE OF TERRESTRIAL & AIRCRAFT COMPONENTS
- LASER POWER TRANSMISSION

Figure 14.

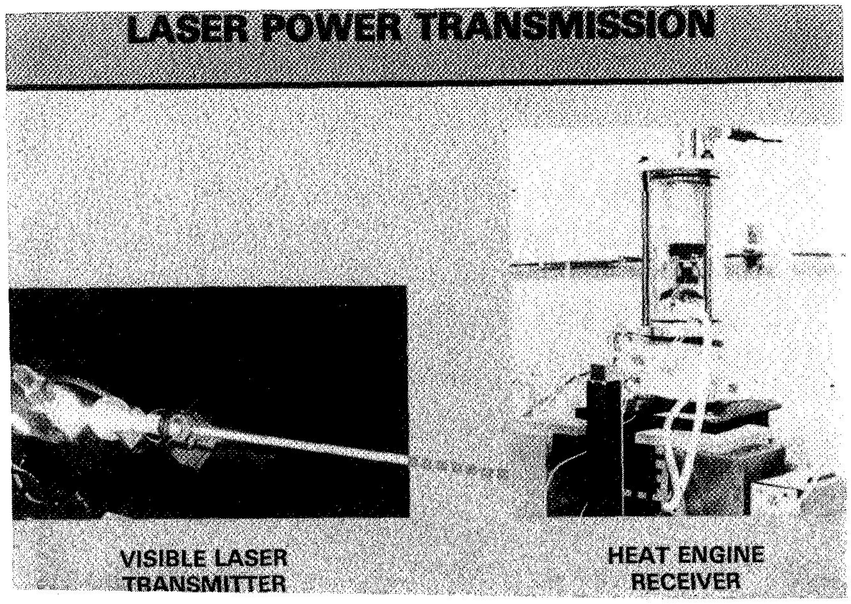


Figure 15.

LOW COST ORBITAL ENERGY SYSTEMS TECHNOLOGY

OBJECTIVES:

- PROVIDE THE UTILITY POWER PLANT TECHNOLOGY BASE FOR SEMI-PERMANENT EARTH ORBITAL FACILITIES NEEDED IN THE MIDDLE TO LATE 1980.s.

BENEFITS:

- REDUCE CAPITAL & OPERATING COST OF SPACE ENERGY SYSTEMS BY 10 TO 100X
- REDUCE RISK BY EXTENDING SPACE ENERGY EXPERIENCE BASE TO 100X LARGER SYSTEMS
- OPEN NEW OPERATING CONCEPTS & ACHIEVABLE ECONOMIES OF SCALE WITH TRANSMISSION
- REDUCE ENVIRONMENTALLY IMPOSED OPERATIONAL CONSTRAINTS
- TECHNOLOGY BASIS FOR COMBINED SYSTEMS
  - EC/LS
  - HEAT REJECTION
  - ORBITAL PROPELLANT PRODUCTION

APPLICABLE TO SPS

Figure 16.

SPACE ENERGY DEMAND

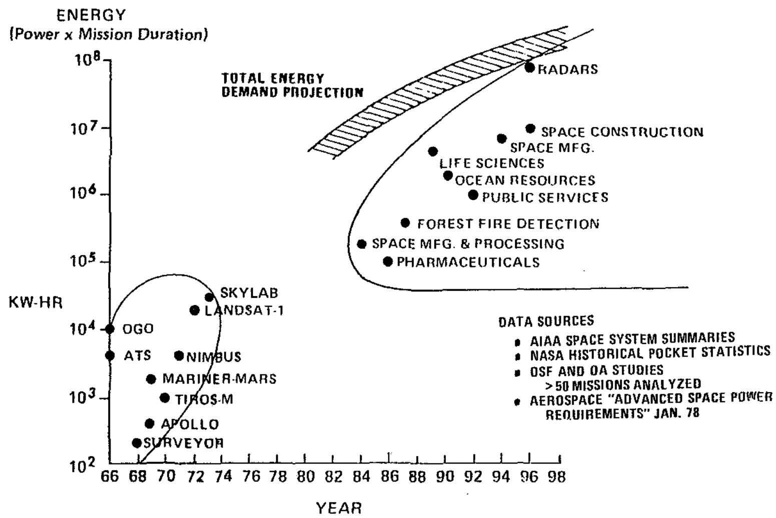


Figure 17.

LOW COST ORBITAL ENERGY SYSTEMS TECHNOLOGY

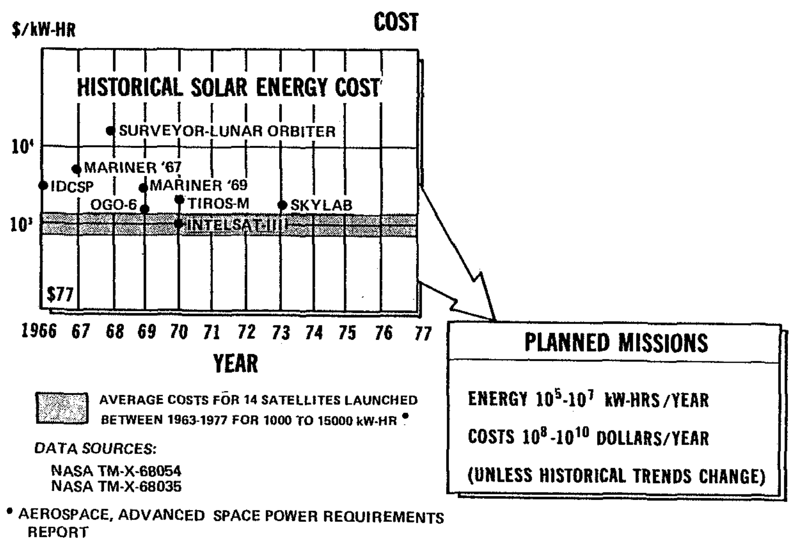


Figure 18.

MULTI-KW ORBITAL POWER PLANT

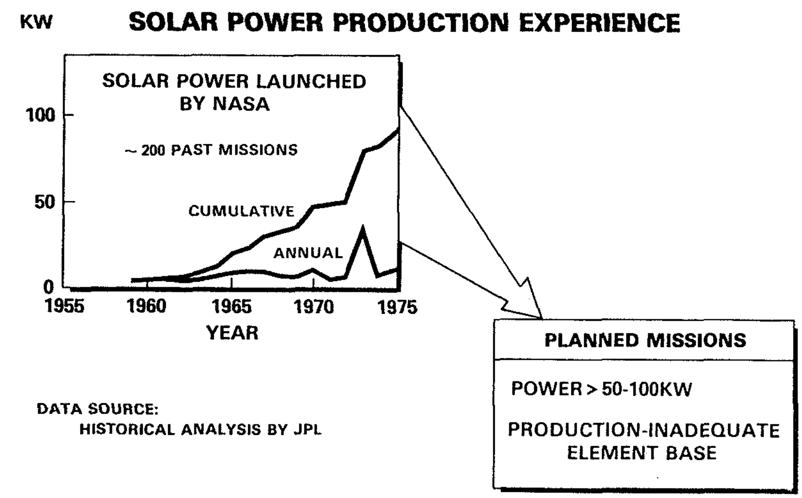


Figure 19.

LOW COST ORBITAL ENERGY SYSTEMS TECHNOLOGY

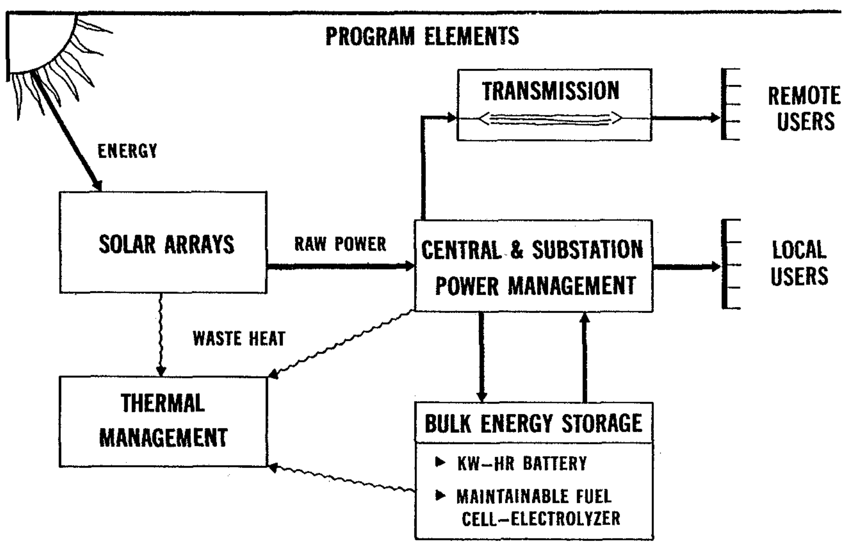


Figure 20.

LOW COST ORBITAL ENERGY SYSTEMS TECHNOLOGY

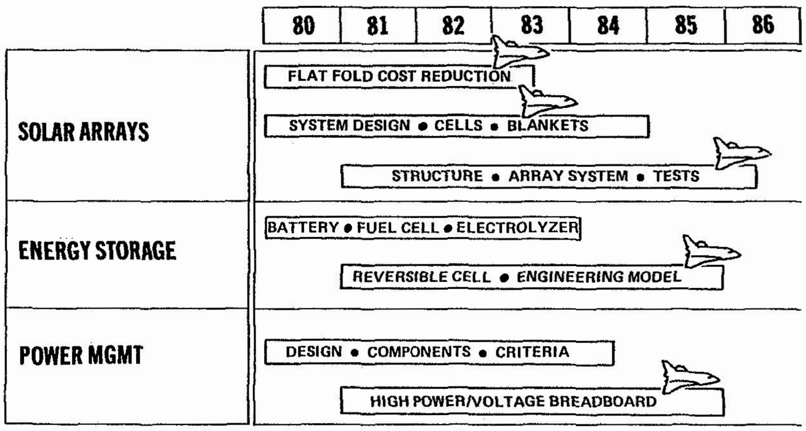
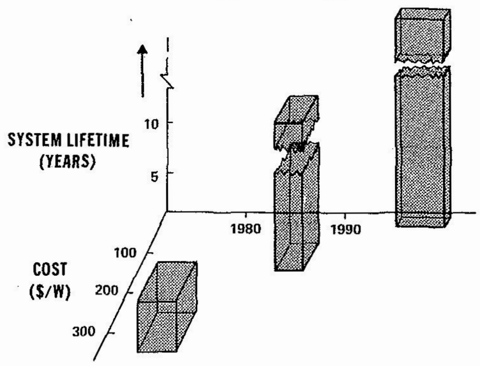


Figure 21.

SOLAR ARRAYS



NEAR TERM

REDUCE CAPITAL COST OF SEP FROM $300/W TO $100/W

FAR TERM

REDUCE CAPITAL COST OF BLANKET TO $10/W

- AUTOMATED ASSEMBLY
- LARGE AREA CELLS

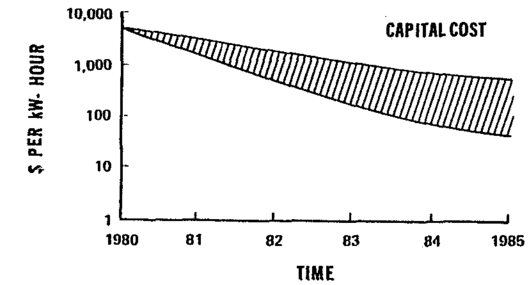
REDUCE ENERGY COST FROM $1000/kW-HR TO $100/kW-HR

- ERECTABLE STRUCTURES
- ON ORBIT MAINTENANCE

Figure 22.

BULK ENERGY STORAGE

OBJECTIVE: REDUCE CAPITAL AND OPERATING COST* OF ENERGY STORAGE BY 10 TO 100 TIMES



>100 A-H TORROIDAL NiCd BATTERY

- PARTS REDUCTION
- HIGHER THERMAL LIMIT

ADVANCED BATTERIES

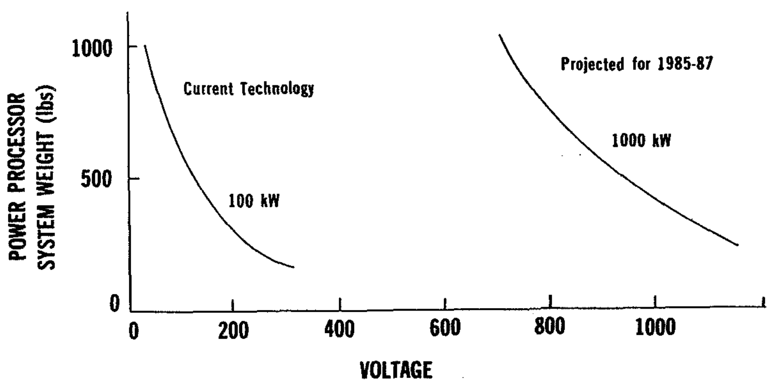
FUEL CELL/ELECTROLYZER

- MAINTAINABLE/INDEFINITE LIFE
- LAUNCH WEIGHT SAVING, 200,000 kg/500 kW
- REVERSIBLE CELL FEASIBILITY

*PRESENT NiCd CAPITAL COST ~$8000/kW-HR

Figure 23.

POWER MANAGEMENT DISTRIBUTION AND CONTROL



COMPONENTS – HIGH POWER SWITCH, TRANSFORMER, CAPACITOR

CIRCUITS – HV CONVERTER

SYSTEM – 100 kW BREADBOARD, AUTOMATION

ENVIRONMENT – DESIGN CRITERIA FOR HV SPACE PLASMA INTERACTIONS

Figure 24.

LOW COST ORBITAL ENERGY SYSTEMS TECHNOLOGY

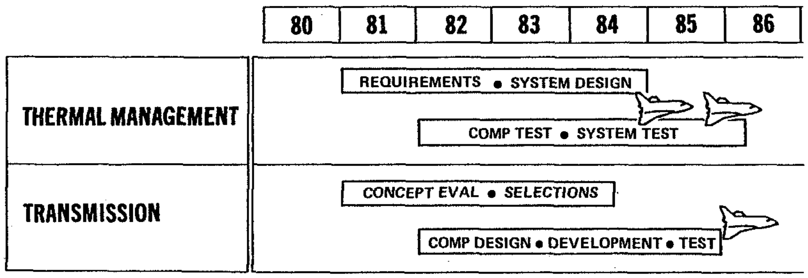


Figure 25.

THERMAL MANAGEMENT

OBJECTIVE: TECHNOLOGY TO ENABLE A SYSTEMS APPROACH TO THERMAL MANAGEMENT

ACQUISITION (Equipment Cooling)

- HIGH DENSITY COLD PLATE
- ADVANCED INTEGRATION TECHNIQUES

TRANSPORT (Circulating Systems & Heat Pipes)

- HIGH CAPACITY HEAT PIPES
- THERMAL UMBILICAL

DISSIPATION (Radiators & Coatings)

- STABLE LOW α/ϵ COATINGS
- ERECTABLE RADIATOR MODULE
- MAINTENANCE METHODS

Figure 26.

POWER TRANSMISSION

OBJECTIVE: DEMONSTRATE FEASIBILITY OF CENTRAL POWER SOURCES FOR SPACE USE

- TRANSMITTER/RECEIVER COMPONENT EXPERIMENTS
- SYSTEM CONCEPTS
- QUANTIFY COSTS & BENEFITS OF ORBIT TO ORBIT ENERGY TRANFER
    - ELIMINATE INDIVIDUAL GENERATION & STORAGE
    - ELIMINATE DRAG & GRAVITY GRADIENT TORQUES
    - PROVIDE MANEUVER ENERGY
- SUBSYSTEM & SYSTEM DEVELOPMENT & DEMONSTRATION

Figure 27.

OAST SYSTEM TECHNOLOGY PLANNING

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NASA Headquarters

INTRODUCTION AND SUMMARY

The space program is twenty years of age. Today we employ space projects effectively in some limited areas such as communications and weather forecasting. Opportunities appear to be opening for space systems to serve society in resource management, disaster warning, electronic mail, electronic business and banking, teleconferencing, broadcasting, distressed vehicle location, air-traffic control, zero-gravity and vacuum-produced equipment, and probably energy delivered from space. All of these programs and many others can be described as technologically feasible.

The future will see space platforms which are tens, hundreds, and eventually thousands of meters in size, using and producing kilowatts and megawatts of power, processing data at rates that could handle the contents of the Library of Congress each second--and all, of economic necessity, constructed at very low unit costs. Today's spacecraft are measured in sizes of a few meters, using at most a few hundred watts of power for transmitting and handling a tenth to a hundredth of a percent of forecast data rates, all painstakingly crafted at relatively high unit costs.

Preparing for the space program transition from the demonstration-oriented era to a cost-effective operational era presents extensive and challenging technology goals. Accordingly, the NASA Office of Aeronautics and Space Technology has developed a planning model for space technology consisting of a Space Systems Technology Model, technology surveys, and technology forecasts. The Technology Model describes candidate space missions through the year 2000 and identifies their technology requirements. The technology surveys and technology forecasts provide, respectively, data on the current and estimates of the projected status of relevant technologies. These tools are used to further the understanding of the activities and resources required to ensure the timely development of technological capabilities.

Basic electronics progress serves as the driver to future programs in that payloads are primarily comprised of sensors, data processors, and transceivers. The revolutionary growth that we have witnessed in electronics technology in recent years (Fig. 1) is evidenced not only in performance but also in reduction of cost and increase in reliability. All of these capabilities are progressing at the remarkable exponential rate of doubling every one to three years. This growth is reflected in virtually any measure of performance, cost, or reliability.

Expanded payload capability is stimulating the entire aerospace industry to conceive and advocate ambitious future program concepts. To support this expanding payload capability, comparable advances must be developed in the supporting technologies of power, structures, control, and transportation. Key technological forecasts for missions that "drive" technology requirements are summarized in Fig. 2. It is apparent from this figure that the future needs of technology growth are remarkably uniform and demanding. The coming twenty years must provide growth of three to four orders of magnitude whether we are concerned with the volume of transportation to Earth orbit, the handling and processing of data quantities, or the requirements of power and size of the spacecraft systems.

Technological capabilities for future space systems have been forecast to expand at an exponential rate. The realization of such technological advances will enable future space systems with enormous capability for providing benefits to serve vital national needs. At today's costs for generating these capabilities, the programs being considered would exceed reasonable budget levels. The costs illustrated in Fig. 3 for transportation to low Earth orbit, for handling data, for generating electrical energy, and for constructing spacecraft vehicles must be reduced substantially. With forecasts of capability increases of three to four orders of magnitude in the next twenty years, unit costs of accomplishment must be reduced by orders of magnitude. Although a corresponding 10^3 or 10^4 drop in unit cost is probably not realizable, a 10^1 or 10^2 reduction could help keep future programs, and their benefits, within reach.

THE TECHNOLOGY MODEL AND FORECASTS

The space missions or systems in the Technology Model, both near and far term, are divided by their area of application into four OAST Space Themes: Exploration, Global Services, Utilization of the Space Environment, and Transportation. The near-term missions are derived primarily from the current NASA five-year planning document. The far-term missions are derived from advanced studies conducted by OAST and other NASA offices. Figures 4 and 5 list the missions and systems of the Technology Model.

For each system in the Technology Model, the primary and secondary technologies that will enable or substantially enhance that system are identified, and where possible, the required level of achievement is noted.

As companion to the Space Systems Technology Model, a technology forecast handbook is being prepared. This will be a reference document containing historic trends and projections of each "capability measure," based on best available data from many sources.

OAST SPACE R&T PROGRAM PLAN

The OAST space technology plans are structured into major thrusts of

- Information Systems
- Spacecraft Systems

- Transportation Systems
- Power Systems

These interdisciplinary groupings provide a focus to technology activities, allowing long- and short-term goal orientation and intermediate milestone identification.

The Research and Technology (R&T) Base Program is the mainstream of technology program activity. While R&T base is the resource for bringing technology to a level of readiness for transfer to planned programs, technology readiness for program application often requires flight validation. This need may be satisfied by either aircraft or space demonstration, with the Spacelab available as a qualification test platform. Shuttle and Spacelab also provide a new and valuable "real space" environmental test facility in support of research and development. OAST technical planning avails itself of this capability.

Details of the FY 79-83 OAST Space R&T Program follow. For each planning thrust, a brief description of the elements is presented, long-range plans with expected benefits are given, and indications of the applicability of these space systems technologies to other NASA programs are presented.

Information Systems

Major elements of the information systems technology effort, summarized in Fig. 6 are instrument pointing, sensing and data acquisition, data processing, communications, data reduction, and data distribution. Long-range plans for this program include both augmentation of the base program in selected areas having potentially high payoffs in future mission applications or significant deficiencies in current activities, and intensified initiatives in certain specific technologies (Fig. 7) having direct benefit to planned and proposed NASA missions. Augmented programs in microwave radiometry and IR detectors will permit development of new and improved concepts for space sensors. Complementary to, and directly supporting, the sensor development activity is a program augmentation in instrument-pointing system technology to provide precise pointing and stabilization of sensor and experiment platforms. To build a strong technology base in advanced communications systems and services, program augmentations are planned in the areas of X-band power amplifiers, multibeam antennas, and data compression. The overall objectives of this effort are to reduce the time and cost required for the collection, processing, and dissemination of space-generated data by a factor of 100 to 1000 over a 10-year period.

Phased programs in NASA End-to-End Data Systems (NEEDS) and Efficient Sensing Systems (ESS) illustrated in Fig. 7 are planned to improve the efficiency and effectiveness of NASA information systems.

The Information Systems Technology Program has the potential to provide enabling and enhancing technologies to numerous possible NASA programs. The NEEDS Program could significantly reduce the cost of future SEASAT, LANDSAT, Shuttle, Global Earth Resources, and Environmental Monitoring Programs. The

ESS Program would optimize the data collecting capabilities of proposed TIROS, Environmental Monitoring, STORMSAT, and LANDSAT missions.

The NASA End-to-End Data Systems Program will build on ongoing critical technology developments by first providing a technology base for Real-Time Data Management and, in subsequent years, developing the technology for Low-Cost Data Distribution. The FY 79 system technology emphasis on Real-Time Management has two major thrusts: development of on-board data reduction technology followed by a Shuttle demonstration, and development of technology to expedite user access to space-generated data including a ground demonstration.

In the area of Efficient Sensing Systems, the overall objective is to expand usable data-gathering capability by a factor of 10. The first phase of this new initiative will focus on development of both high-resolution linear array infrared detectors for terrestrial observations and linear arrays of microwave radiometers to improve the spatial resolution of oceanic and ground monitoring. To implement these systems, a precision platform and tracking system will be developed to perform high-resolution imaging and spectroscopy experiments of planetary surfaces, atmospheres, and satellites. In subsequent years, the program will build on the augmented technology base to provide enhanced environmental monitoring systems, precision pointing capability, and multimission sensing technology.

Spacecraft Systems

Elements of this thrust include structures, assembly, guidance and control, materials, thermal control, on-board propulsion, and planetary entry. An immediate major objective is to provide the technology in structures, materials, assembly, and controls for economical large-area space structures. Objectives to be addressed later in the 5-year period include development of analytical methods for nonlinear large deflection; automated operations including techniques for the use of teleoperators, free-flying robotic manipulators; and development of technology relating to the use of extraterrestrial resources for the construction of future space systems.

Future needs in communications, Earth resources, and space industrialization will require spacecraft of several hundred meters to several kilometers in size compared to our current experience with spacecraft of several tens of meters. This represents a technology challenge beyond putting more of the same types of structure in orbit. Large structures are more flexible, thus requiring greater structural efficiency (stiffness and strength per unit mass). More sophisticated, distributed controls are required for both pointing and figure control. In addition, large structures must be assembled in space using manipulators and teleoperators not currently existing.

Figure 8 depicts the technology elements of this thrust leading to efficient large spacecraft. A current program has laid the groundwork for large space structure concepts. The proposed system technology augmentation--Large Space Systems Technology--beginning in FY 81, will define representative systems as focal points in order to establish structures, controls, and assembly technology requirements. Later phases would focus on technology test and verification activities of the two major structural categories--antennas and platforms. Additional new programs are planned for a new nonlinear deflection analysis capability, automated space operations (function, pointing, transmission, maintenance), and a technology program proposed to develop the ore processing procedures for extraterrestrial materials for use in space construction.

Figure 8 depicts the ongoing program and proposed augmentations. The augmentations include two new R&T base efforts: long-life composites and free-flying robotic manipulators. The first is needed to provide the design base for what is expected to be the principal structural material for spacecraft. The second will provide the technology needed for assembly, maintenance, and other future space operations.

The basic entry technology R&T program which is contained in the Spacecraft Systems thrust develops the gas dynamic, aerothermodynamic, and flight mechanics technology base required to improve entry spacecraft design, safety, reliability, and efficient aerodynamic operation for Earth orbital and planetary exploration missions. The near-term program establishes the technology base to assure survival and reliable performance of outer planet probes.

In the long term, the entry technology program establishes the aerothermodynamic technology and configurational design concepts required to achieve significant improvements in operational efficiency, safety, reliability, and economy for space transportation systems operational in the 1990s.

The primary thrust of the OAST spacecraft systems program is to provide technology readiness for the middle to late 1980s suitable for Earth communications, Earth observations, and space platforms; and deep space communications and astronomy in the 1990s (Fig. 9). However, the program will begin in the early 1980s to provide usable output suitable for supporting potential communications and Earth resources sensing mission, space construction base, and missions in radio astronomy and deep space communications.

Transportation Systems

Technology for launch vehicles and orbital transfer vehicles includes efforts in chemical propulsion, low-thrust propulsion, structures and materials, thermal protection systems, aerothermodynamics, and zero-gravity experiments (Fig. 10). Several objectives need to be addressed if desired technology advances are to be achieved. A continuing objective is to develop low-thrust propulsion for orbit-to-orbit cargo delivery and interplanetary transfer of scientific payloads. Other objectives are to advance chemical propulsion, materials and structures, and thermal protection systems technologies that will lead to fully reusable, much longer life vehicles that require minimum servicing and maintenance between flights.

The chemical propulsion objective is to provide a technology base for future large-scale, Earth-to-orbit propulsion systems including long-life, minimum maintenance reusable propulsion systems. Advanced structural concepts and materials for use in future transportation systems include fully reusable, low-maintenance structures capable of withstanding high temperature and composite structural elements to reduce vehicle weight. The continuing objective to develop low-thrust transportation for orbit-to-orbit and interplanetary service includes ion thrust systems, electromagnetic mass drivers, and magnetoplasmadynamic accelerations.

Another continuing objective is to demonstrate propulsion system concepts (such as long-life, highly flexible systems) suitable for a late-1980s reusable space-based orbital transfer vehicle.

A later specific objective is to provide a technology base for large-scale reusable propulsion systems for Earth-to-orbit vehicles, including minimum serviceability, low recurring costs, oxygen/hydrogen, oxygen/high-density-fuel engines, and high-performance, lightweight dual-fuel systems; and to conduct flight experiments to develop the technology for propellant management in zero-gravity environment.

A specific objective starting in FY 83 is to develop advanced, low-maintenance structural concepts; materials capable of withstanding high temperatures; and lightweight composite structural elements to reduce vehicle weight.

Although the transportation systems technology program is aimed at a future low-cost, high-capability space transportation system family of vehicles (Fig. 11), it will also provide potential enabling/enhancing technologies to such NASA programs as Shuttle/IUS improvement/growth, Shuttle derivatives, and high-energy planetary missions.

Power Systems

This technology program seeks to advance our capability to generate, store, process, and distribute electrical energy for use in space systems. Advances over current levels of technology are required to fully realize the advantages of the high performance needed for electric propulsion and to effectively use near-Earth space. As indicated in Fig. 12, the base program provides technology for both high-performance and multikilowatt low-cost future power requirements via the conduct of research in solar cells and arrays, batteries and fuel cells, thermo-electrics, Brayton cycle, thermionics, power management, and advanced concepts such as laser transmission. Augmentation of this program aimed at the increased performance and power level requirements anticipated in the 1980s and beyond appears to have high potential payoff and hence is planned.

Space energy costs have been very high, in the range of several thousand dollars per kilowatt-hour for past systems compared to terrestrial costs of a few cents per kilowatt-hour. As can be seen from Fig. 13, the cost of space energy has remained relatively constant for over a decade, so the cost reductions indicated for future potential missions represent a very important technology challenge and opportunity. Additional technologies which are significant for some of the largest power-using and -producing missions are large-structure construction and low-cost Earth-to-orbit transportation (described earlier).

Future missions are expected to have energy requirements 100 times or more greater than past missions; hence, investments should be made now in technology aimed at reducing costs if such future missions are to be kept within reasonable cost bounds. Space solar power installed in the past cumulates to less than 100 kW since the beginning of the Space Program, and we are faced with power demands on single missions under discussion for the 1980s which approximate that cumulative level and can anticipate growth to the megawatt range in the 1990s.

Figure 14 shows actual average power for some prior NASA missions and projected average power for some missions from the Technology Model. There is generally a smooth increase in power level when both actual and projected missions are considered, except for the SPS, which is orders of magnitude above the other missions. However, SPS is not a power user but a power producer and as such perhaps should not be expected to fit the trend of power-using systems. Similarly, the large power module also falls above the power levels for power user missions of the mid-1980s.

Such large increases in power levels also require technology advances in related space systems. Some which are prominent in the Technology Model are power storage and lifetime, heat pipes and heat rejection, automated power conditioning and power management, lightweight power system materials, and thermionics. Perhaps the most important technology need is to reduce significantly the cost of space energy.

The orbital power program is aimed both at reducing cost and at providing the technological basis for future high-power orbital systems. This program seeks to attack the critical problems of low-cost generation, maintainable bulk energy storage, large-scale thermal and power management, as well as to seek the economies of scale of central power through evolution of the enabling technology of power transmission.

As suggested in Fig. 15, significant program outputs that are projected are applicable to such near-term NASA programs as JOP, comet rendezvous, OTV, space processing, and public service COMSAT. Far-term goals, however, are to realize the combined advantages of high performance and low cost in enabling systems of the future, such as SPS and NEP.

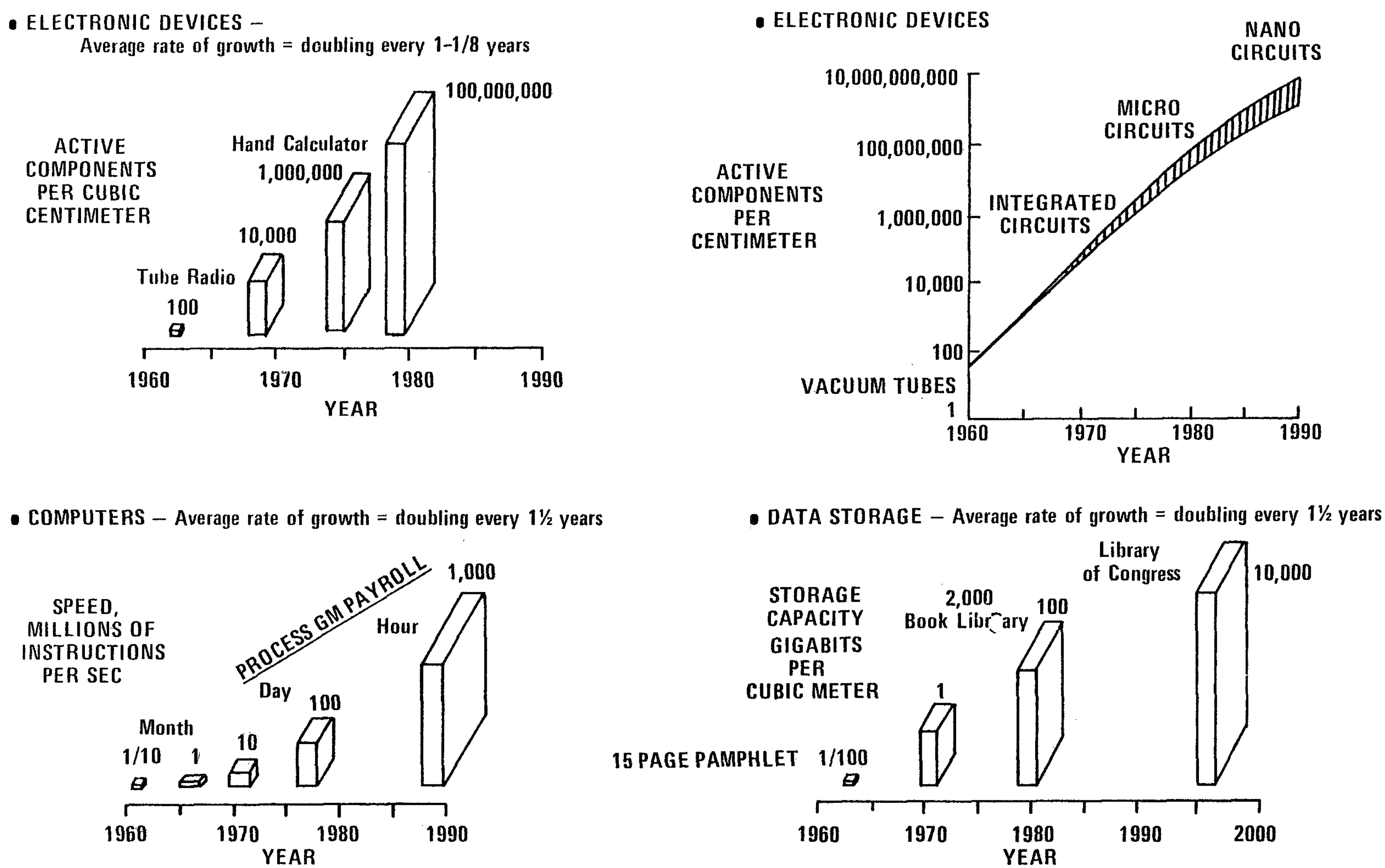


Figure 1. Basic Technology Growth

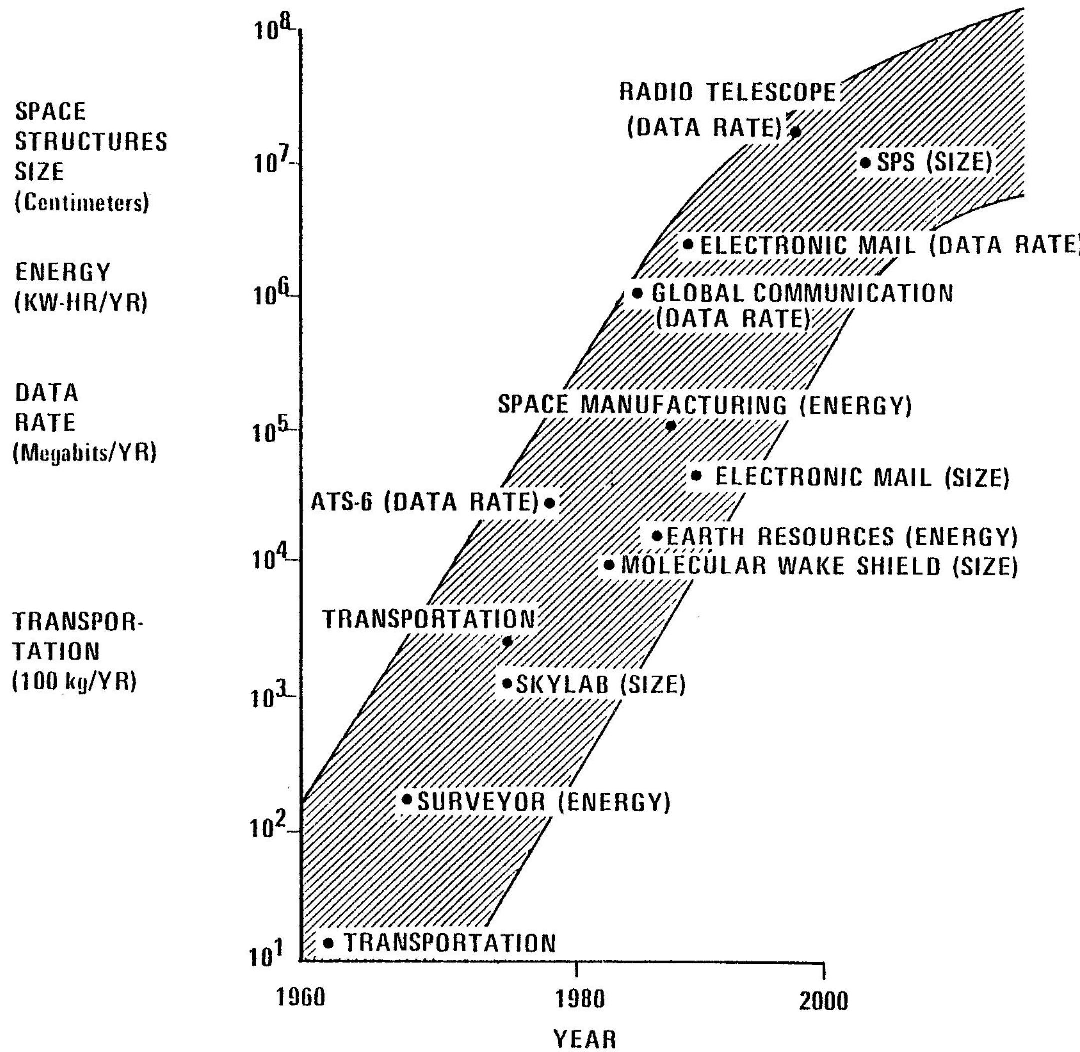


Figure 2. The R&T Challenge

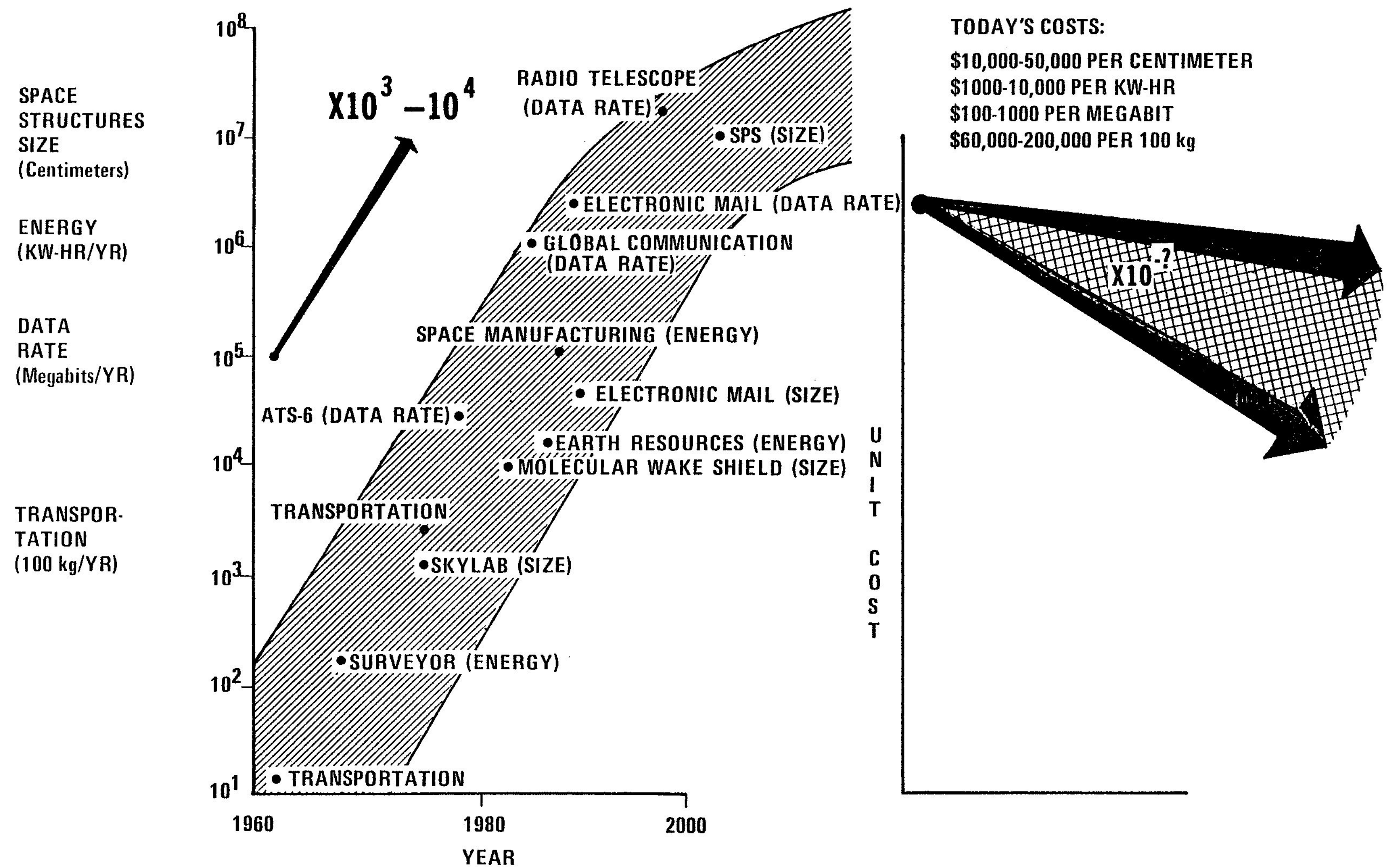


Figure 3. The R&T Challenge and Opportunity

VOLUME I: NEAR-TERM MISSIONS

EXPLORATION OF THE UNIVERSE

1. SOLAR MESOSPHERE EXPLORER (1981)
2. BIOMEDICAL SPACELAB EXPERIMENTS FACILITY (1981-83)
3. GALILEO–JUPITER ORBITER/PROBE (1982)
4. SOLAR POLAR MISSION (1983)
5. VENUS ORBITAL IMAGING RADAR (1983)
6. SPACELAB MULTIUSER INSTRUMENT PROGRAM (1983-84)
7. BIOMATERIALS PROCESSING LABORATORY (1983-84)
8. GAMMA RAY OBSERVATORY (1984)
9. SYNOPTIC TROPOSPHERE AND TERRASPACE ENVIRONMENT SATELLITE (1984-85)
10. COMET RENDEZVOUS (1985)
11. ORIGIN OF PLASMAS IN THE EARTH'S NEIGHBORHOOD (1986)
12. SOLAR PROBE (1986)
13. ADVANCED X-RAY ASTROPHYSICAL FACILITY (1986)
14. X-RAY OBSERVATORY (1986)
15. COSMIC BACKGROUND EXPLORER (1986)
16. SATURN ORBITER DUAL PROBE (1987)
17. COSMIC RAY OBSERVATORY (1987)
18. PINHOLE SATELLITE (1987)
19. MARS SAMPLE RETURN PROGRAM (1988)
20. LARGE AREA MODULAR ARRAY (1988)
21. AUTOMATED MOBILE LUNAR SURFACE SURVEY (1988-89)
22. EXTREME ULTRAVIOLET EXPLORER (1988-90)

Figure 4. OAST Space Systems Technology Model

VOLUME I: NEAR-TERM MISSIONS (CONT.)

GLOBAL SERVICES

23. SEASAT FOLLOW-ON (1982)
24. ENVIRONMENTAL MONITORING SATELLITE (1982-83)
25. GEODETIC SURVEY SATELLITE (1984)
26. TIROS-O (1984)
27. SOIL MOISTURE SATELLITE (1985)
28. STORMSAT-A (1985)
29. GLOBAL COMMUNICATIONS LAND MOBILE SERVICES (1986)
30. PUBLIC SERVICES COMMUNICATIONS SATELLITE (1986)
31. GEOSTATIONARY PLATFORM (1987-88)

UTILIZATION OF THE SPACE ENVIRONMENT

32. ADVANCED SPACELAB PROCESSING PAYLOADS (1982-83)
33. TELEOPERATOR ORBITER BAY EXPERIMENT (1983)
34. MOLECULAR WAKE SHIELD (1984)
35. SPACE MANUFACTURING MODULE (1984-85)
36. SPACE HEALTH CARE PROGRAM (1986)
37. LARGE POWER MODULE (1986-88)

SPACE TRANSPORTATION SYSTEMS

38. SPIN-STABILIZED UPPER STAGE (1984)
39. SOLAR ELECTRIC PROPULSION STAGE (1985)
40. ORBITAL TRANSFER VEHICLE (1988)

Figure 4 (Cont.). OAST Space Systems Technology Model

VOLUME II: FAR-TERM MISSIONS

EXPLORATION OF THE UNIVERSE

1. LARGE EARTH ORBITAL SOLAR OBSERVATORY (1992-95)
2. ASTROPHYSICS SPACE LABORATORY (1992-95)
3. ATMOSPHERIC PHYSICS LABORATORY (1993-95)
4. SPACE-BASED RADIO TELESCOPES (19[illegible]-2000)
5. AUTOMATED PLANETARY STATION (2000-2000+)

GLOBAL SERVICES

6. GLOBAL COMMUNICATIONS SYSTEM (1987-89)
7. GLOBAL CROP INVENTORY AND PRODUCTION FORECASTING SYSTEM (1987-89)
8. HIGH-RESOLUTION SEA SURVEY SYSTEM (1987-89)
9. DISASTER WARNING SYSTEM (1988-90)
10. EARTH ENERGY BUDGET MONITORING SYSTEM (1989-91)
11. LARGE-SCALE ALL WEATHER SURVEY SYSTEM (1993-96)
12. GEOLOGICAL MAPPING SYSTEM (1993-96)
13. GLOBAL NAVIGATION SYSTEM (1995-97)
14. SATELLITE POWER SYSTEM (2000–2000+)

UTILIZATION OF THE SPACE ENVIRONMENT

15. AUTOMATED PRECURSOR PROCESSOR (1990-92)
16. NUCLEAR WASTE DISPOSAL SYSTEM (1995-97)
17. TELEOPERATOR VEHICLE SYSTEM (1995-97)
18. LUNAR BASE AND AUTOMATED PRECURSOR SYSTEMS (1995–2000+)
19. SPACE STATION (2000–2000+)

SPACE TRANSPORTATION SYSTEMS

20. PRIORITY ORBITAL TRANSFER VEHICLE (1990-92)
21. CARGO ORBITAL TRANSFER VEHICLE (1990-92)
22. HIGH ENERGY ORBITAL TRANSFER VEHICLE (1991-93)
23. PRIORITY LAUNCH VEHICLE (1990-92)
24. HEAVY-LIFT LAUNCH VEHICLE (1995-2000)

Figure 5. OAST Space Systems Technology Model

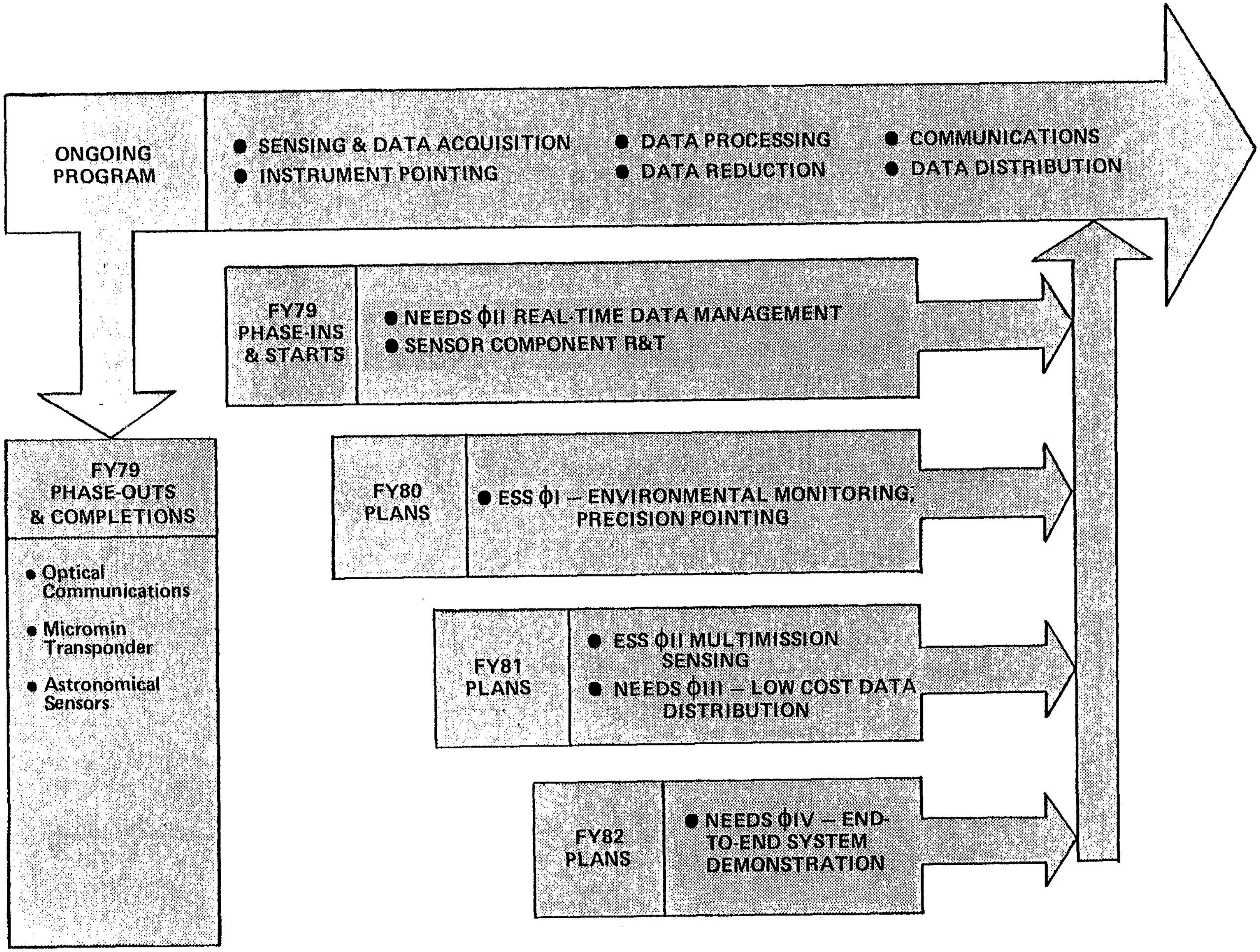


Figure 6. Information Systems Technology — 5 Year Plan

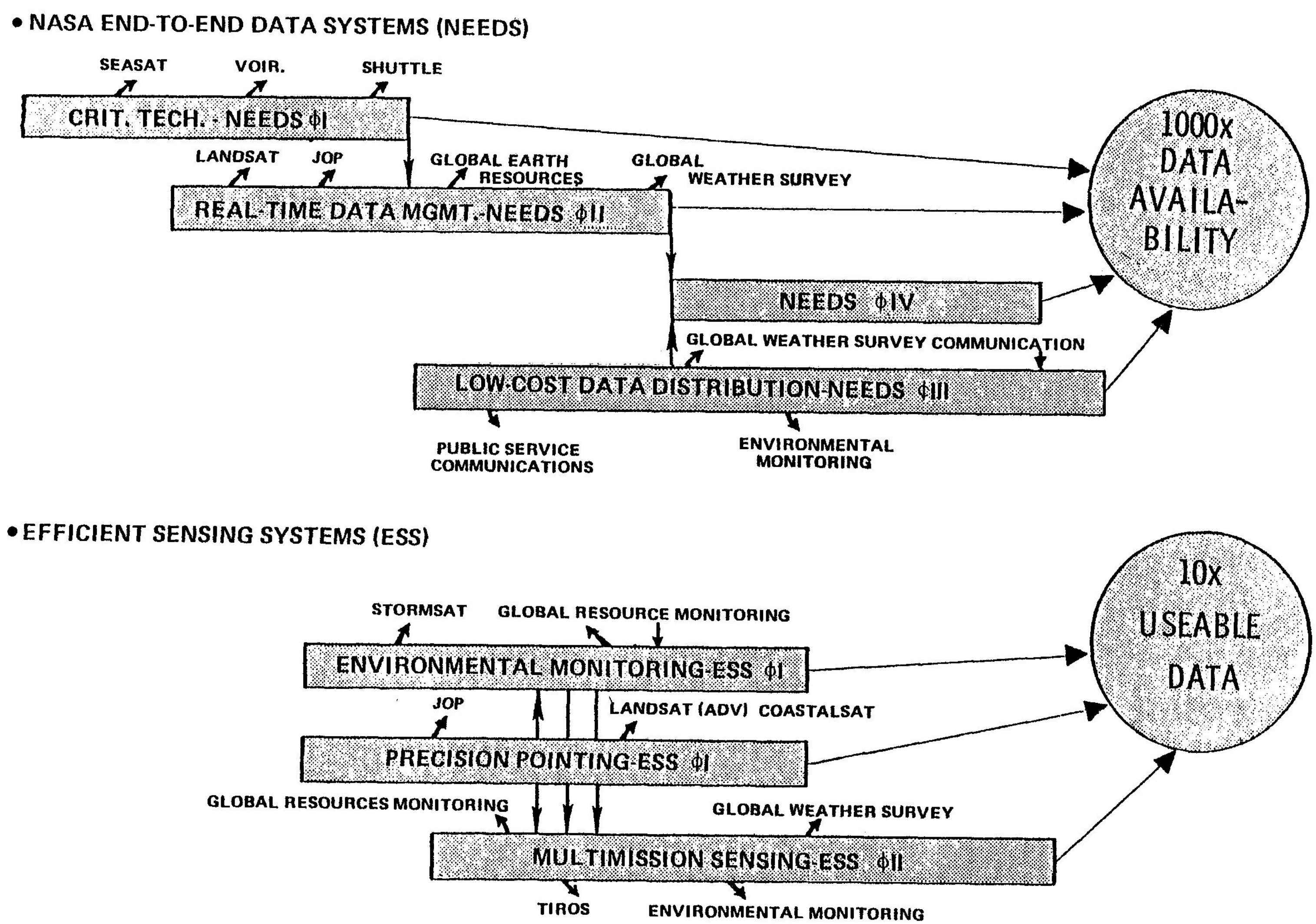


Figure 7. Information Systems Technology

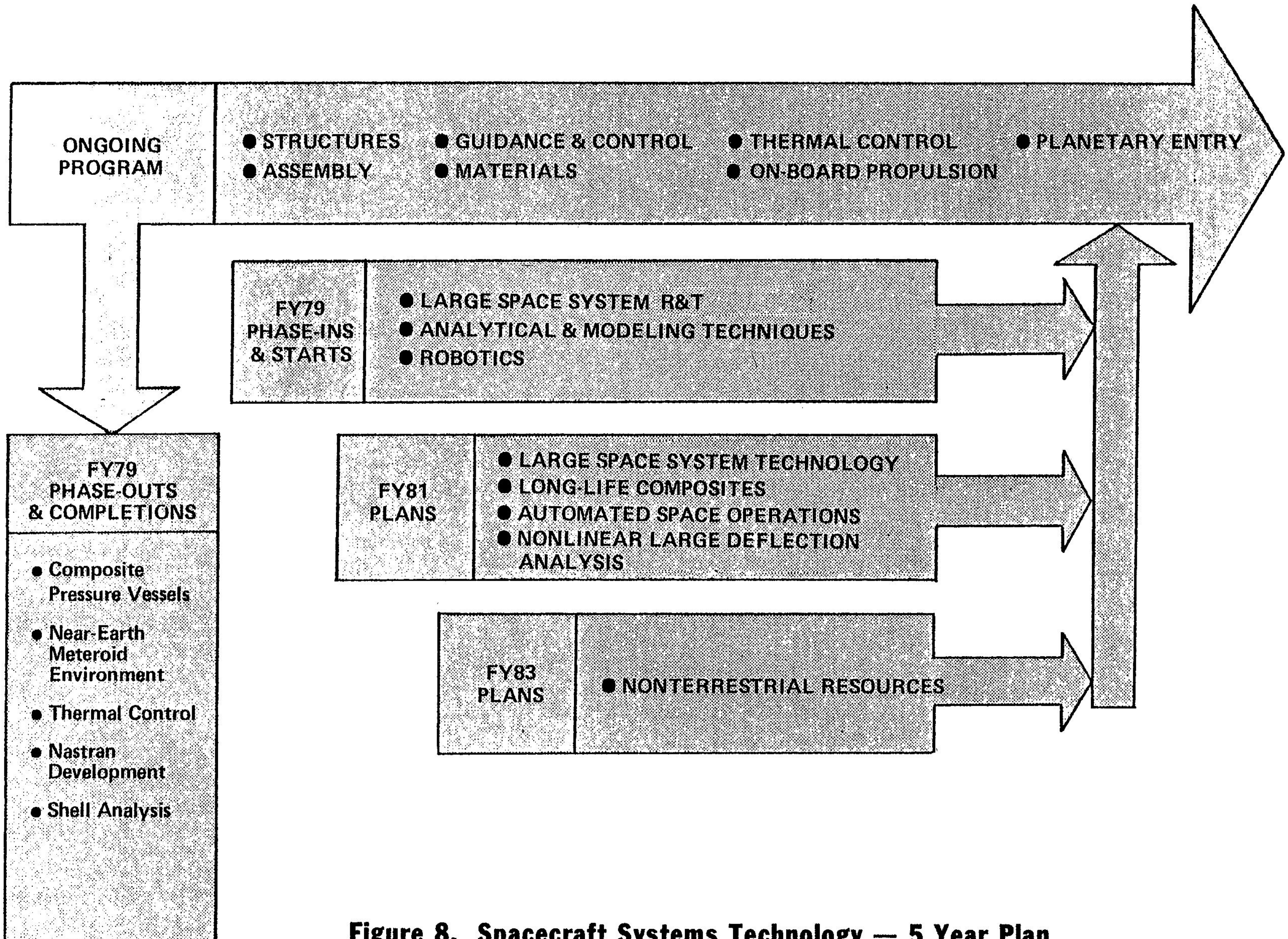


Figure 8. Spacecraft Systems Technology — 5 Year Plan

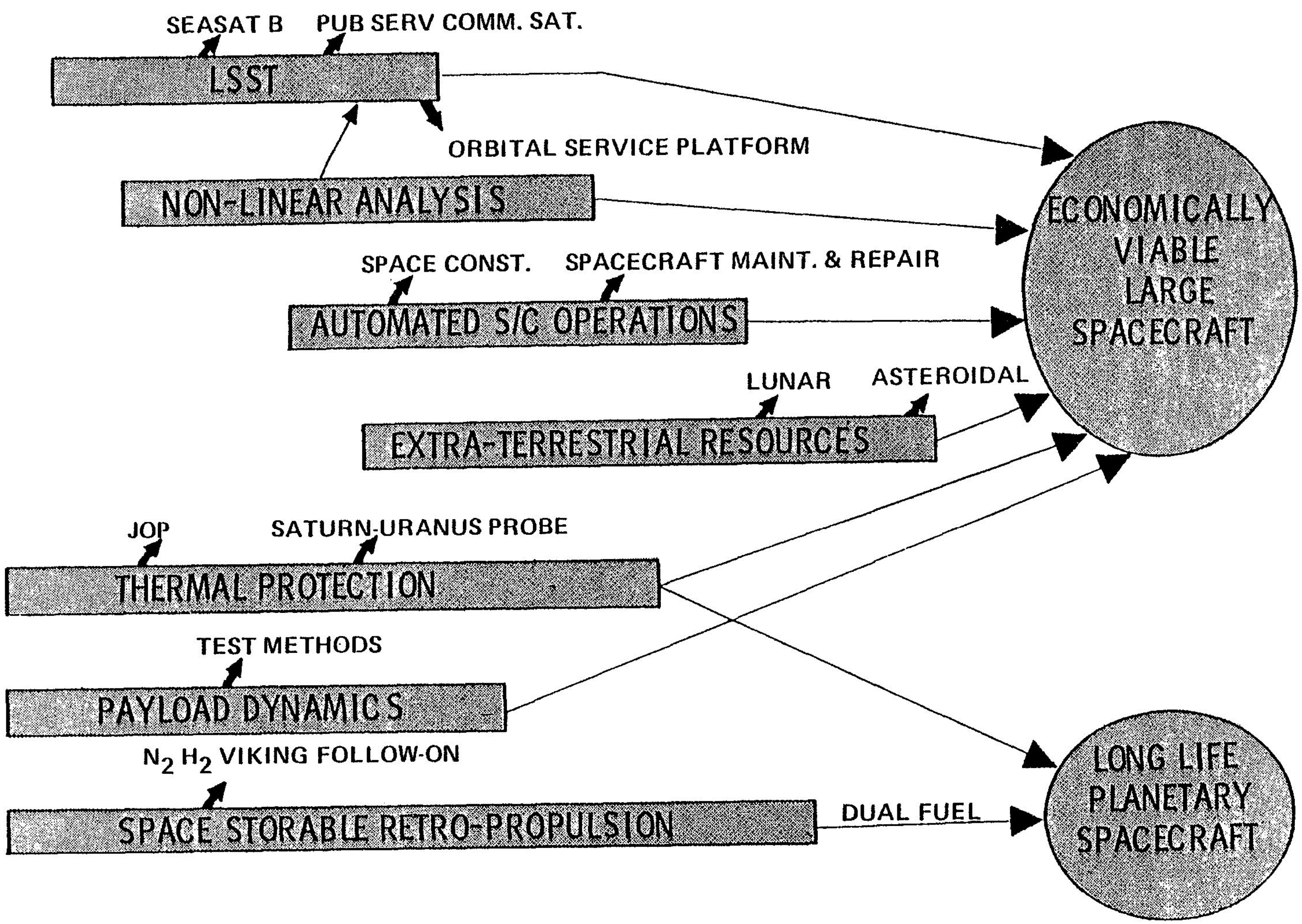


Figure 9. Technology for Spacecraft Systems

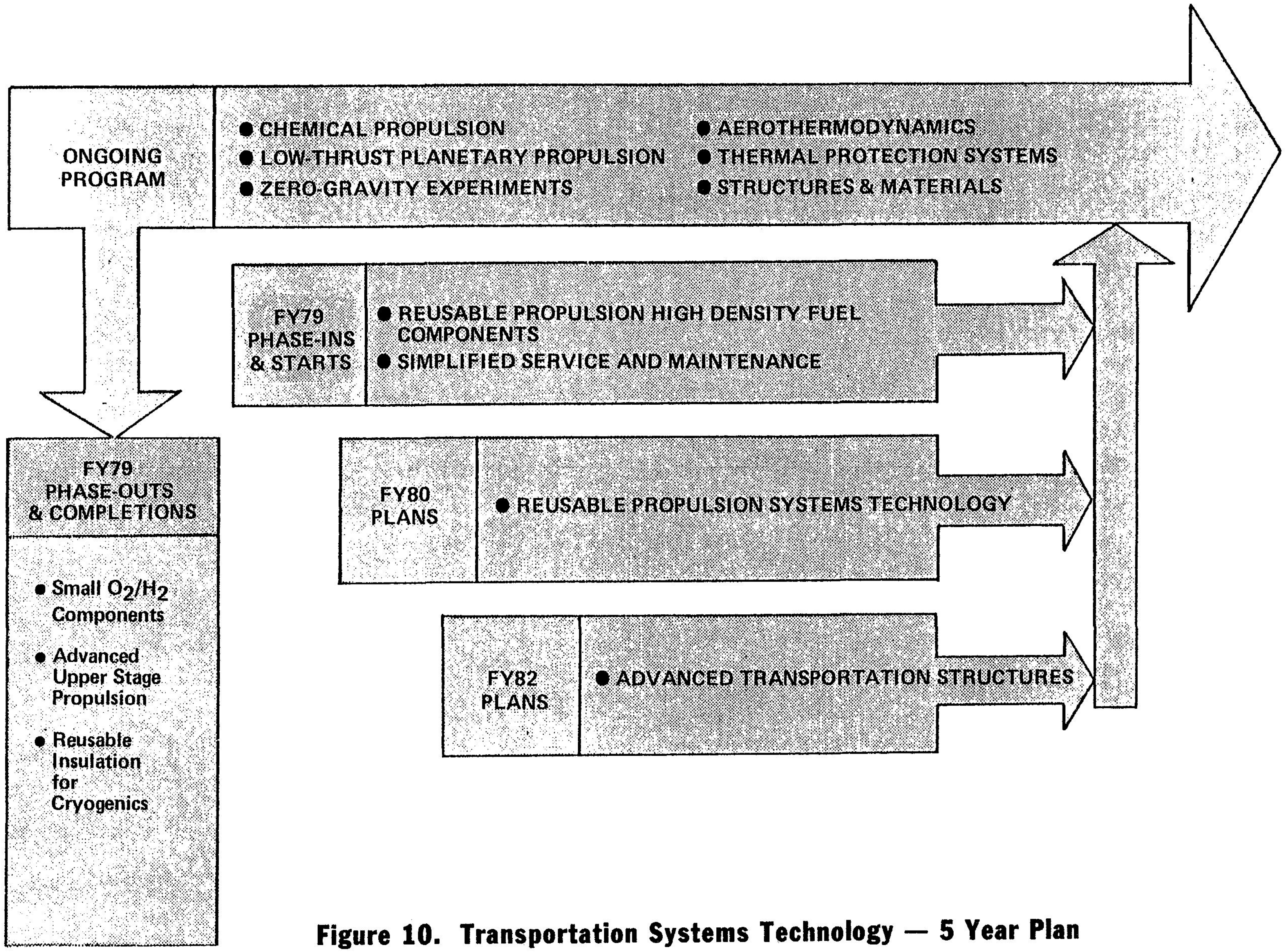


Figure 10. Transportation Systems Technology — 5 Year Plan

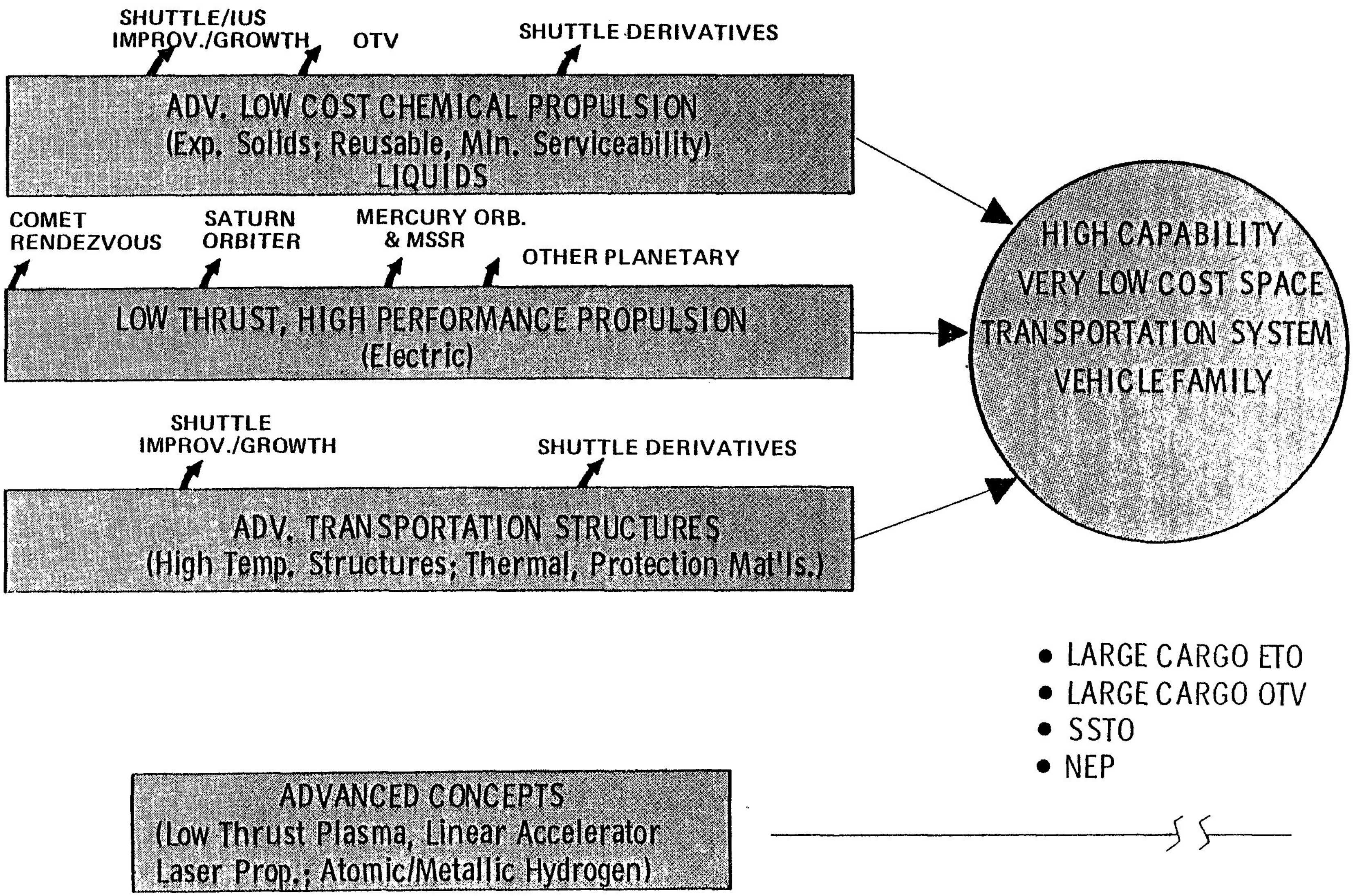


Figure 11. Transportation Systems Technology

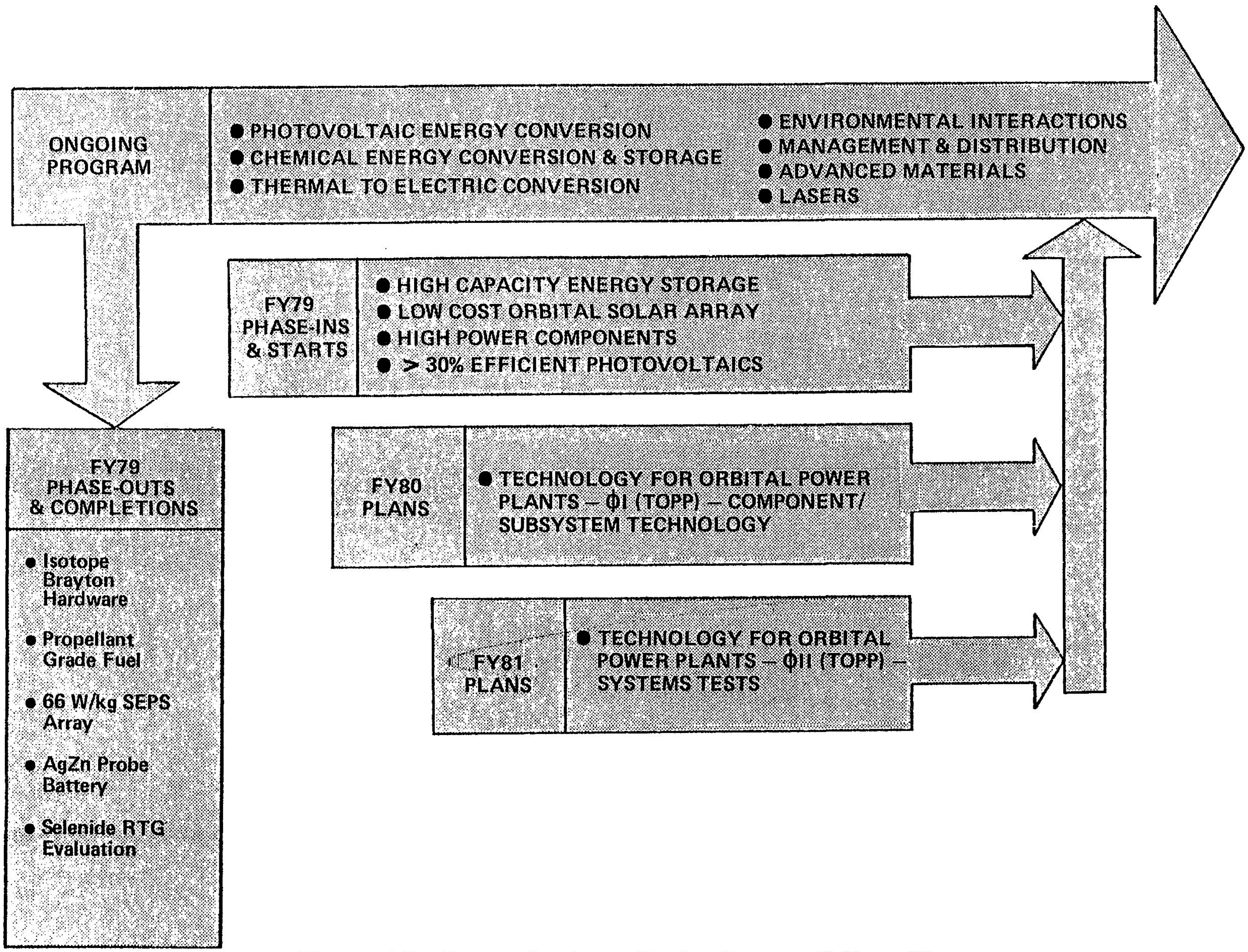
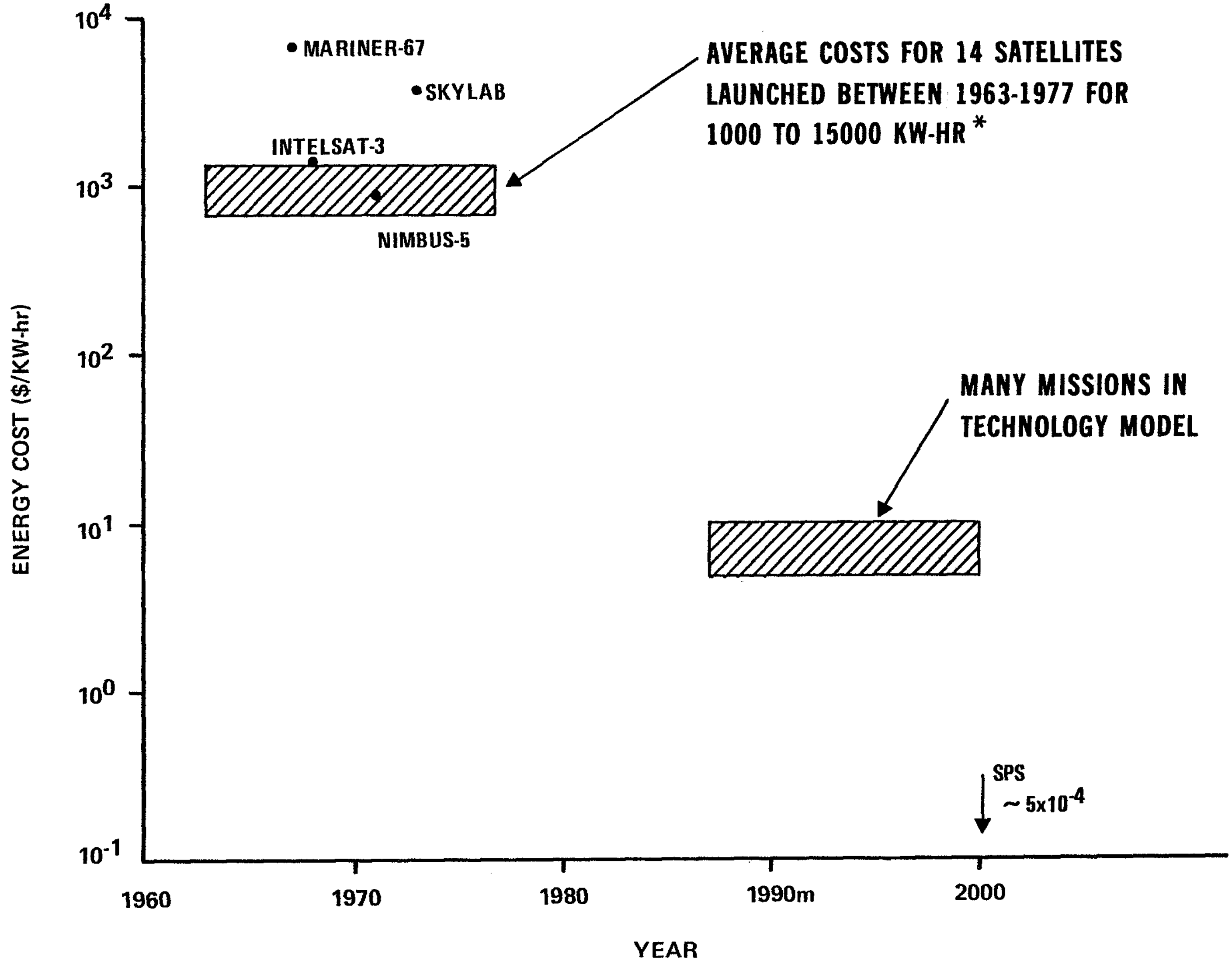


Figure 12. Power Systems Technology — 5 Year Plan



*AEROSPACE, ADVANCED SPACE POWER REQUIREMENTS REPORT

Figure 13. Space Power Costs

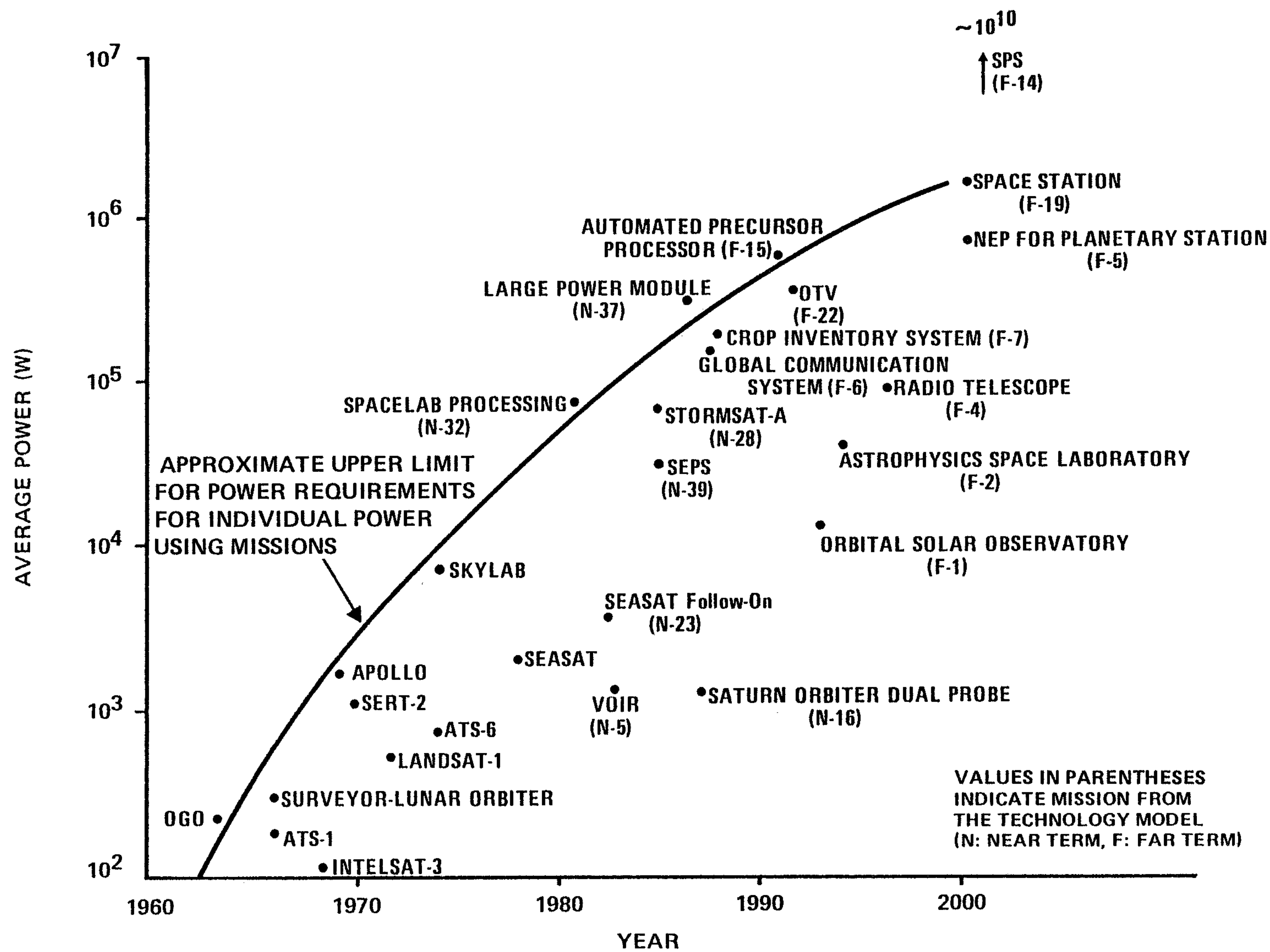


Figure 14. Mission Power Requirements

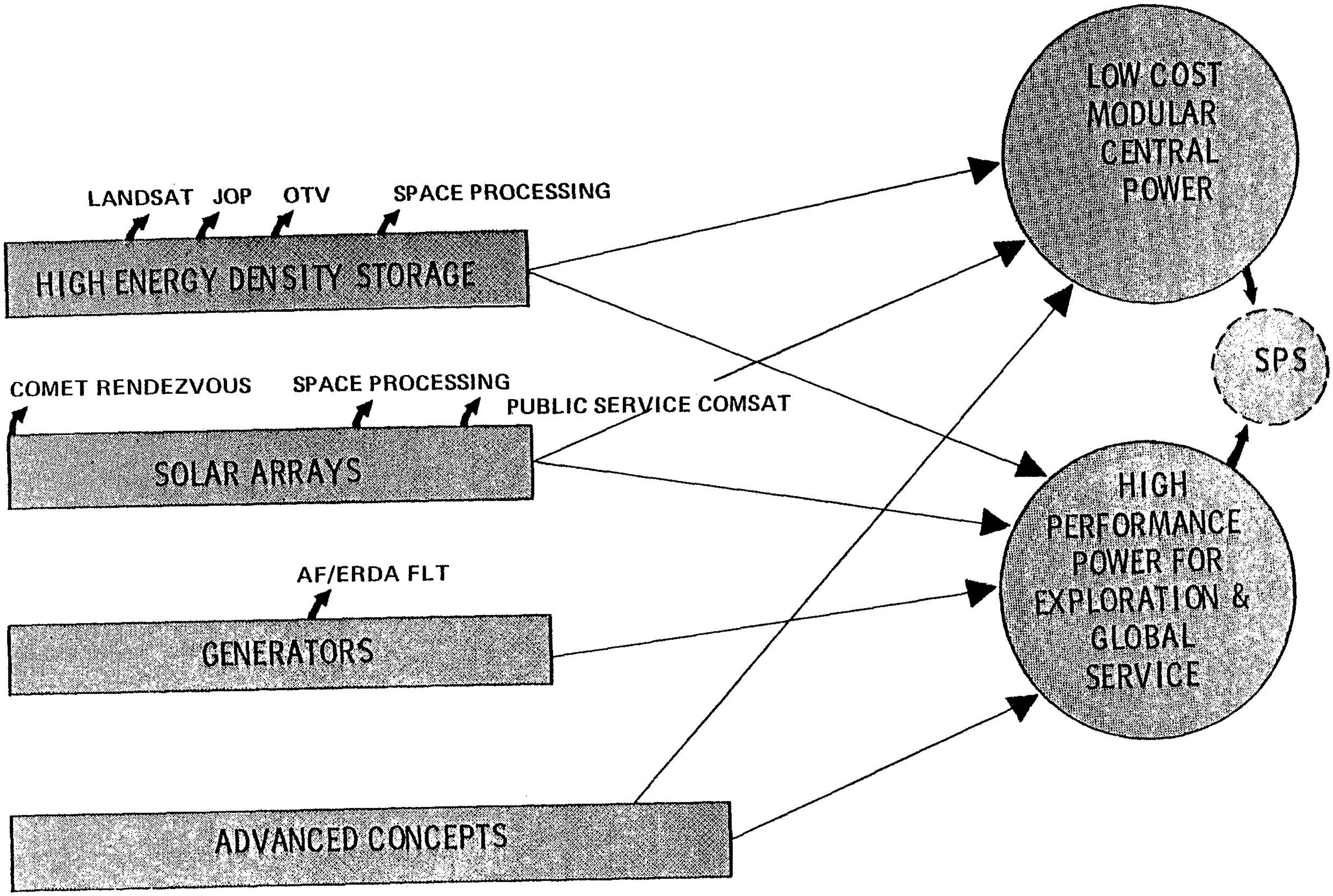


Figure 15. Power Systems Technology

HISTORICAL AND PROJECTED POWER REQUIREMENTS

Malcolm G. Wolfe

The Aerospace Corporation

SUMMARY

Since the inception of the U. S. national space program, power level requirements have been increasing steadily at about 100 watts per year for both civilian and military satellites. The demand could be expected to increase at about the same rate; however recent shuttle and shuttle follow-on planning activities (ref. 1, 2, 3) have introduced the eventual need for very large, multipurpose space platforms to be deployed. This would result in a step function in individual satellite power level requirements, demands for higher total energy requirements, and the need for different approaches to designing power systems for indefinite lifetime operation and periodic servicing and maintenance. Some of the proposed multipurpose space platforms could require power levels of over 200 kW. If the SPS (Satellite Power Station) is implemented then, of course, another massive step function would occur in space power requirements.

INTRODUCTION

Historical data can be extrapolated to provide a prediction of the future with a high probability of success in many situations and an examination of historical space power characteristics shows a steady evolutionary change. However, a radical change is about to occur in the method of access to space. The Space Shuttle will provide economical transportation and increased flexibility with the availability of man in low earth orbit, if required, in the early 1980s. In the late 1980s the capability of the IUS (Inertial Upper Stage) and SSUS (Spinning Solid Upper Stage) to transfer space systems to high orbit will be amplified by the development of the OTV (Orbit Transfer Vehicle) which will eventually permit man to become an intrinsic part of space operations out to geosynchronous orbit and beyond.

In order to fully exploit space and the flexible operational capabilities of the STS (Space Transportation System) and its derivatives planning studies (ref. 1, 2, 3) have examined the potential of very large multipurpose systems having indefinite lifetimes, which require deployment and/or assembly on orbit (and therefore the need for orbital space assembly facilities with their own power supplies), periodic servicing (either automated or manned) and possibly manned residence for extended periods of time. The space power requirements are likely to be quite different to the requirements of conventional single-purpose satellites.

HISTORICAL SPACE POWER TRENDS

PRIME POWER REQUIREMENTS

Using Refs. 4 through 7, a survey was made of satellites launched or planned to be launched during the 1959-1979 time period, together with their user group function, power system type, and prime power requirements. Scatter diagrams of power versus launch date for each user group were prepared (the details are discussed in Ref. 8) and are shown in Figs. 1 through 4. A trend line of 100 watts per year is shown which appears to represent the rate of increase over the time period examined. A general problem solving computer program (GYPSY) was used to perform a regression analysis of the historical prime power requirements data. A total of 175 launches were used, including 96 NASA, 44 DoD and 35 civil data points. The best fit to all data was found to be:

$$\mathrm{Ln}P = A + BM + CM^2 + DM^3$$

where: P = Prime power in watts
M = Number of months after June 1959

and the coefficients are as follows:

	A	B	C	D
NASA	6.41	-0.0186	6×10^{-5}	5×10^{-8}
DoD	6.9	-0.06	0.0005	-10×10^{-5}
Civil	5.4	-0.05	6×10^{-4}	-2×10^{-6}
All	6.5	-0.0377	-0.00029	-6×10^{-7}

Computer plots of the output are shown in Figs. 5 through 8.

POWER SYSTEM COSTS

Background. For a number of years the Aerospace Corporation has collected satellite and launch vehicle hardware costs on ongoing programs from government and private industry sources and incorporated them into a computerized cost data bank. This data bank has a number of uses, including being used as a base for developing future systems non-recurring and recurring costs, and is being constantly expanded. It has been found expedient to organize the data to suit the accounting procedures of industry as far as possible and the format used for documentation is illustrated in Table 1.

Cost Analysis. Historical electric power subsystem costs were analyzed for the years 1963 through 1977 and the percentage distribution by major component is listed in Table 2. The electrical subsystem cost per kilowatt-hour as a function of year of first flight is given in Fig. 9 and as a function of kilowatt-hour in Fig. 10. The data is scattered but some trends can be postulated. The ground rules used to develop the costs are listed in Ref. 8.

FUTURE SPACE POWER REQUIREMENTS

Two approaches were used in Ref. 8 to develop future space power requirements. One approach emphasizes a future in which large multipurpose, multi-user satellites will be the objective of early development and deployment; the other approach emphasizes a future in which many dedicated, single-user satellites will be deployed in the near and mid term, with large multipurpose satellites not being introduced until the far term. As far as total power requirements are concerned, the two approaches lead to more or less the same conclusions since, in general, the accumulation of several initiatives on one space platform results in a corresponding accumulation of total power. Where differences will occur, however, is in such areas as the need for supporting and folding large solar array blankets and the establishment of servicing and maintenance policies, and the establishment of policies for the design, development and deployment of remote space power modules. If remote space power modules are used to supply power to other satellites via laser or microwave links, consideration must be given to whether they have to supply a multitude of low-powered satellites or a small number of high-powered satellites.

MISSION/TRAFFIC MODEL APPROACH

Mission models and, from these, traffic models were synthesized to correspond to the average yearly budgets illustrated in Table 3. An iterative process was used to match the budgetary goals with specific mission/traffic models. The details of the procedure and the ground rules used are described in Ref. 8. Mission/Traffic models were developed to satisfy low and high average budgetary levels for the following mission categories*:

1. NASA Observation
2. NASA Communication
3. NASA Support
4. NASA Scientific
5. NASA Planetary
6. DoD Surveillance
7. DoD Communication
8. DoD Navigation and Meteorology
9. DoD Weaponry
10. Non-NASA/Non-DoD Communication
11. Non-NASA/Non-DoD Observation
12. Non-NASA/Non-DoD Support
13. Non-NASA/Non-DoD Scientific

The traffic models illustrated in Ref. 8 have no official approval, either of NASA or of DoD, and are intended to be representative only. Nevertheless, the component parts have been extracted from published documents in most cases and serve to provide a reasonable representation of the future.

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* The mission categories are themselves divided into groups of missions which have functional similarities.

The power requirements derived in the study described in Ref. 8 are summarized in Table 4 and Figs. 11 and 12. It should be noted that contributions from the SPS program are not included since they would tend to obscure the total picture.

ADVANCED SYSTEM SCENARIO APPROACH

Background. A very large number of future initiatives have been identified for both NASA and DoD and in order to handle the literally hundreds of known initiatives a rationale was established (Ref. 2) for categorizing the initiatives into five generic categories or eleven groups, as follows:

Category	Initiative Group
Information	1. Public Service Systems Using Microwave Multibeam Antennas
	2. Public Service Systems Using Long Microwave Antennas
	3. Active/Passive Radar and Power Distribution Systems
	4. Observation and Designation Systems Using Optics at Low Altitude
	5. High Altitude Navigation, Location, and Relay Systems
	6. Observation Systems Using Synchronous Altitude Optics
Processing	7. Space Processing and Manufacturing
Energy	8. Large Scale, High Energy, Far-Term Systems
Science	9. National Operations Facilities
	10. Scientific and Research Experiments
Planetary	11. Planetary

The generic groups attempt to subsume each of the identified initiatives and are intended to be broad enough that other initiatives yet to be identified will be likely to fall within one of the groups. A natural progressive increase in capability can be postulated for each of the eleven groups, exemplified by the deployment of a series of space systems over a period of time, with each system having a considerable increase in capability over its predecessor (but not necessarily replacing its predecessor). The increase in capability and the time period between each launch impacts the needs for technology advancements, the launch vehicle and support facility needs, and the overall space program funding requirements.

The development plan for each group provides the development required to satisfy the initiatives contained within that group. An orderly step-by-step technology program is the primary determinant of the number of time-phased steps in each of the development plans. Each step is intended to culminate in demonstrated flight hardware capable of operational use; however, the operational option may not be exercised.

In the construction of the development plans it was found expedient to lump the low and high altitude optical concepts (Groups 4 and 6) together and also to combine the scientific and research experiments (Group 11) with the national operations facilities required to operate them (Group 9).

The construction of development plans in this manner provides maximum flexibility for dealing with an indeterminant future for the following reasons:

1. Each development plan is not linked to a single initiative, the need for which may change radically during the development time period.

2. The decision as to which initiative to promote can be delayed until late in the development schedule.

3. The unexpected need for crash programs is minimized.

Power Level Requirements. The development plans and estimates of the resulting prime power requirements are illustrated in Figs. 13 through 20. In general, the required power levels increase monotonically within each generic group. An optimistic and conservative schedule is approximated for each operational capability step. Representative initiatives are listed and coded to indicate their source as follows:

(OFS) = The NASA "Outlook for Space" study (Ref. 9)
(5-YP) = The NASA Five-Year Plan (Refs. 10 and 11)
(A) = The Aerospace Corporation "Advanced Space Systems Concepts and Their Orbital Support Needs (1980-2000)" Study (Ref. 1)

Power vs Time Requirements. Figs. 21 through 28 show the power requirements for each initiative group as a function of time. Of the two solid plots, one represents an ambitious, well-funded, overall NASA space program, and one represents a more conservative approach where procurement of major systems is delayed approximately a further seven years. (The seven-year cycle was selected in a relatively arbitrary manner. However, it represents an estimate of the average time necessary to procure a major advanced space system, from initial go-ahead to IOC.) The dashed plot, in each case, indicates a stretched-out program in which each development program commences at approximately the same time as the optimistic program, but the procurement of major line items is spread over a longer period of time.

Results. The data included in Figs. 21 through 28 can be used in a number of ways. One use is to perform a rough rank ordering of the power requirements of the initiative groups. This provides information to determine which initiative groups can be "captured" by a given space power development plan at a specific point in time. In general, the initiative group development plans are divided into a number of steps or subgroups providing the option of not consummating all of the possible steps. Table 5 lists the subgroups of each initiative group in power demand rank order. It lists also the approximate IOC dates for an optimistic, well-funded NASA space plan, a more conservatively funded plan, and a stretched-out plan. The table demonstrates the power levels necessary to capture individual initiative group and subgroup developments.

Table 6 lists the power demands (in rank order) of initiative subgroups as a function of approximate IOC date. The utility of the table is to demonstrate which subgroups or development plan steps can be captured by a given space power capability in a given year. For instance, a 10 kW space power capability achieved in 1988 would capture Subgroups 5/2, 9&11/2, and 4&6/3 in the case of an optimistic space plan, but not be required until 1996 to capture the same subgroups if a conservative space plan were to be implemented. The data can be used as a tool for space planning in two ways:

1. If a projection is made of the space power technology capability at a given time in the future, the subgroups of initiatives that the projected technology will be able to "capture" is determinable.

2. If a projection is made of the total space system capability (the specific initiative subgroups implemented) at a given time in the future, the space power technology capability that will be required is determinable.

With the aid of information on expected advancements in space power technology, an assessment can be made as to whether those planned advancements will meet the requirements objectives. If not, then the plans can be modified to attempt to meet those objectives.

CONCLUDING REMARKS

If national space planning embarks on a policy of deploying large multipurpose satellites the needs of DoD and the civil sector will not, in general, drive space power requirements since they will be trailing NASA needs. Present NASA space planning policy does appear to be leaning towards the eventual implementation of a few very large multipurpose satellites which can be serviced on orbit and have indefinite lifetimes. The rationale for such a policy is that it makes maximum use of the unique capabilities of the Space Shuttle and leads as rapidly as possible to the exploitation of space for the immediate benefit of mankind. The large multipurpose satellites can be designed to service vast numbers of different users equipped with small, cheap user terminals. Some of the possible uses are personal communications, electronic mail, educational, and health and welfare TV, and personal navigation. The implication is that NASA will not be restricted to its traditional R&D role but will show leadership to commercial and private users by participating in commercial applications in certain areas.

The planning policy outlined above would result in the need for such space facilities as the Space Construction Base and the increasing participation of man beyond low earth orbit. The large satellites may be self-powered or may receive their power from separate space (the Space Power Module) or ground-based power plants.

DoD needs are somewhat different. The implementation of a few large undefended multipurpose satellites makes the space system fleet more vulnerable to enemy attack. The alternatives are either to provide active defense systems or to orbit a larger number of smaller satellites. The emphasis on survivability and anonymity in the case of DoD systems means that the DoD criteria for selection of space power system, subsystems and components may be different than the NASA criteria. For instance, at high power levels the DoD is more likely to select a more compact system than a solar cell/battery system with its large radar cross section. Solar cell design would also have to consider the susceptibility of solar cells to, for instance, continuous-wave lasers.

At this time, official DoD planning shows a less intense drive towards large multipurpose satellites than NASA planning. Nevertheless, DoD is presently initiating a well-funded study on the orbital assembly of large spacecraft and a few high-powered systems are already described in DoD planning documents. In addition, during the studies conducted by Aerospace for NASA in recent years, a large number of DoD initiatives were identified which require high power. Many public sector initiatives have a parallel military application and DoD space power technology requirements, in many ways, parallel the needs of NASA.

In the civil sector, the U.S.'s lead in the commercial application of space is partly based on satisfying individual users by providing relatively small, reliable, cheap satellites that can be clearly identified with a specific customer. It is not clear that foreign countries will be willing to relinquish the prestige associated with having their own satellite or be willing or able to fund their own large multipurpose satellites. The utility and economic benefits of such systems will have to be clearly demonstrated, either by NASA or by domestic civil users, before they are accepted by foreign users. This will probably result, in the near term, in a greater tendency for foreign users to lease time on U.S. satellites rather than to purchase their own multipurpose systems.

It is concluded that within the context of the above arguments, the demands by civil users on space power requirements and technology can be subsumed within those of NASA. There are some differences between the power levels and the technology requirements of NASA and DoD in the near term but these are likely to be less apparent in the far term.

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Table 1. Satellite Power System Cost Summary Format

SATELLITE _______

__ Mo., Des. Life, ____ W, BOL Pwr, ____ W, Avg Pwr,

First Launch 19__

Cost Category \ Item	Solar Array (____ sq ft)	Battery (__A-H)	Power Control Unit	Converters	Wiring	Drive	Total
Non-recurring							
Design Engrg.							
Test & Eval.							
Recurring (5 Sat.)							
Syst. Engrg.							
Production							
Total (1977 $)							
Average (5 Sat.)							

Subsystem Weight/Satellite Weight

Cost/lb. (kg)

Cost/ft^2(m^2)

Cost/A-H

Cost/kW-H

Table 2. Satellite Electrical Power Cost Percentage Distribution by Major Components

Year of 1st Launch	Solar Array	Batteries	PCU Plus Converters	Wiring	Array Drives
1963	43.3	16.7	37.0	2.9	-
1964	23.5	22.6	15.8	23.6	14.6
1967	34.2	9.6	45.8	10.3	-
1967	21.6	10.9	23.1	-	44.4
1969	62.5	9.0	15.9	12.6	-
1970	46.2	13.2	32.2	8.5	-
1970	9.3	11.1	9.2	22.4	48.0
1971	46.0	12.1	28.9	13.0	-
1971	21.4	19.3	32.1	27.1	-
1974	26.9	8.9	26.5	37.8	-
1974	34.2	15.9	33.6	16.3	-
1975	23.3	12.1	36.7	28.0	-
1975	18.4	14.7	43.3	23.6	-
1977	10.8	9.9	41.6	9.4	28.4

Table 3. Assumed Average Yearly Budget Goals for 1980-1995

ORGANIZATION	CONSERVATIVE BUDGET ($B)	OPTIMISTIC BUDGET ($B)
NASA		
Institutional	2.0	2.0
Transportation	1.0	2.0
Programs	1.0	2.0
Total	4.0	6.0
DoD Programs	0.7	1.5
Civil (Non-NASA, Non-DoD Programs)	0.5	1.0

Notes:

(1) Budgets are in 1977 dollars.
(2) Budgets are averages and therefore peak budgets will exceed these values in certain years.

Table 4. Energy Demand (1981-1995)

ITEM	CALENDAR YEAR																	15 yr Total (1981-95)	15 yr Average (1981-95)
	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	1992	1993	1994	1995		
NOMINAL BUDGET																			
NASA	4.1	31.5	89.5	156.4	203.5	259.1	267.2	604.4	693.9	693.3	2649.6	2825.6	2754.3	2776.2	2870.4	3010.6	6842.5	26696.5	1779.8
DoD	-	85.7	174.7	198.7	364.3	411.0	527.0	1059.5	1226.7	1313.6	1451.4	1924.7	1928.6	2326.9	2949.7	3312.1	3319.0	22487.9	1499.2
CIVIL	52.7	137.4	201.6	237.0	309.3	321.2	364.0	288.8	409.0	451.9	465.0	512.8	617.9	594.8	534.2	463.3	434.4	6257.9	417.2
NASA & DoD & CIVIL	56.8	254.6	465.8	459.2	877.7	991.3	1158.2	1952.7	2329.6	2458.8	4566.0	5263.1	5300.8	5697.7	6354.3	6786.0	10595.9	55442.3	3696.2
OPTIMISTIC BUDGET																			
NASA	4.1	31.5	89.5	432.2	558.1	613.7	2790.3	2976.0	3144.3	3143.7	3917.1	3896.0	5007.3	2312.8	2801.1	2862.5	3370.6	37919.3	2528.0
DoD	-	85.7	174.7	198.7	443.1	450.4	1039.6	1288.3	1613.0	3166.6	3848.2	4392.7	6769.5	7325.2	6525.0	10344.7	10305.3	57884.9	3859.0
CIVIL	65.0	149.7	251.4	309.8	465.5	547.7	675.5	655.8	815.8	892.0	922.0	944.5	1074.0	1085.7	1111.1	1089.8	1107.4	11948.0	796.5
NASA & DoD & CIVIL	69.1	266.9	515.6	940.7	1466.7	1611.8	4505.4	4920.1	5573.1	7202.2	8687.3	9233.2	12850.8	10723.7	10437.2	14297.0	14783.3	107752.2	7183.5

Table 5. Initiative Group Rank Ordering

INITIATIVE		IOC DATE			
Group/ Subgroup	Title	Optimistic Program	Stretched Program	Conservative Program	Power Level
2/1	PUBLIC SERVICE SYSTEMS USING LONG MICROWAVE STATIONKEPT ANTENNAS - I	1983	1983	1990	1.0 kW
3/1	POWER DISTRIBUTION SYSTEMS AND ACTIVE/PASSIVE RADAR - I	1982	1982	1989	1.0 kW
2/2	PUBLIC SERVICE SYSTEMS USING LONG MICROWAVE STATIONKEPT ANTENNAS - II	1987	1991	1994	1.3 kW
5/1	HIGH ALTITUDE NAVIGATION, LOCATION, AND RELAY SYSTEM - I	1983	1983	1990	1.7 kW
2/3	PUBLIC SERVICE SYSTEMS USING LONG MICROWAVE STATIONKEPT ANTENNAS - III	1992	1999	1999	2.0 kW
4 & 6/1	OPTICAL OBSERVATION, DESIGNATION, AND MEASUREMENT - I	1982	1982	1989	2.0 kW
9 & 11/1	SCIENTIFIC/RESEARCH EXPERIMENTS AND NATIONAL FACILITIES - I	1984	1984	1991	2.0 kW
5/2	HIGH ALTITUDE NAVIGATION, LOCATION, AND RELAY SYSTEM - II	1988	1992	1995	2.2 kW
5/3	HIGH ALTITUDE NAVIGATION, LOCATION, AND RELAY SYSTEM - III	1994	2001	2001	3.0 kW
1/1	SERVICE PLATFORMS USING MICROWAVE MULTIBEAM ANTENNAS - I	1983	1983	1990	4.0 kW
3/2	POWER DISTRIBUTION SYSTEMS AND ACTIVE/PASSIVE RADAR - II	1986	1993	1993	5.0 kW
4 & 6/2	OPTICAL OBSERVATION, DESIGNATION, AND MEASUREMENT - II	1986	1988	1993	5.0 kW
9 & 11/2	SCIENTIFIC/RESEARCH EXPERIMENTS AND NATIONAL FACILITIES - II	1988	1991	1995	5.0 kW
4 & 6/3	OPTICAL OBSERVATION, DESIGNATION, AND MEASUREMENT - III	1990	1994	1997	10.0 kW
7/1	SPACE PROCESSING AND MANUFACTURING - I	1983	1983	1990	10.0 kW
9 & 11/3	SCIENTIFIC/RESEARCH EXPERIMENTS AND NATIONAL FACILITIES - III	1993	2000	2000	10.0 kW
4 & 6/4	OPTICAL OBSERVATION, DESIGNATION, AND MEASUREMENT - IV	1995	2002	2002	20.0 kW
1/2	SERVICE PLATFORMS USING MICROWAVE MULTIBEAM ANTENNAS - II	1987	1990	1994	25.0 kW
8/1	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - I	1982	1982	1989	25.0 kW
3/3	POWER DISTRIBUTION SYSTEMS AND ACTIVE/PASSIVE RADAR - III	1990	1997	1997	50.0 kW
7/2	SPACE PROCESSING AND MANUFACTURING - II	1988	1992	1995	50.0 kW
7/3	SPACE PROCESSING AND MANUFACTURING - III	1993	2000	2000	100.0 kW
1/3	SERVICE PLATFORMS USING MICROWAVE MULTIBEAM ANTENNAS - III	1993	2000	2000	100.0 kW
8/2	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - II	1984	1986	1990	210.0 kW
3/4	POWER DISTRIBUTION SYSTEMS AND ACTIVE/PASSIVE RADAR - IV	1994	2001	2001	300.0 kW
8/3	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - III	1987	1990	1993	2.0 MW
8/4	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - IV	1992	1996	1999	15.0 MW
8/5	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - V	1996	2000	2003	1.0 GW
8/6	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - VI	2000	2004	2007	15.0 GW

Table 6. Initiative Subgroup Power Demand vs IOC Date

OPTIMISTIC PROGRAM IOC											
1982-1984		1985-1987		1988-1991		1992-1994		1995-1997		1998-2000	
CONSERVATIVE PROGRAM IOC											
1990-1992		1993-1995		1996-1998		1999-2001		2002-2004		2005-2007	
Subgroup	Power	Subgroup	Power	Subgroup	Power	Subgroup	Power	Subgroup	Power	Subgroup	Power
2/1	1.0 kW	2/2	1.3 kW	5/2	2.2 kW	2/3	2.0 kW	4 & 6/4	20 kW	8/6	15 GW
3/1	1.0 kW	3/2	5.0 kW	9 & 11/2	5.0 kW	5/3	3.0 kW	8/5	1 GW		
5/1	1.7 kW	4 & 6 /2	5.0 kW	4 & 6/3	10.0 kW	9 & 11/3	10.0 kW				
4 & 6/1	2.0 kW	1/2	25.0 kW	3/3	50.0 kW	1/3	100.0 kW				
9 & 11/1	2.0 kW			7/2	50.0 kW	8/2	210.0 kW				
1/1	4.0 kW			8/3	2.0 MW	3/4	300.0 kW				
7/1	10.0 kW					8/4	15.0 MW				
8/1	25.0 kW										

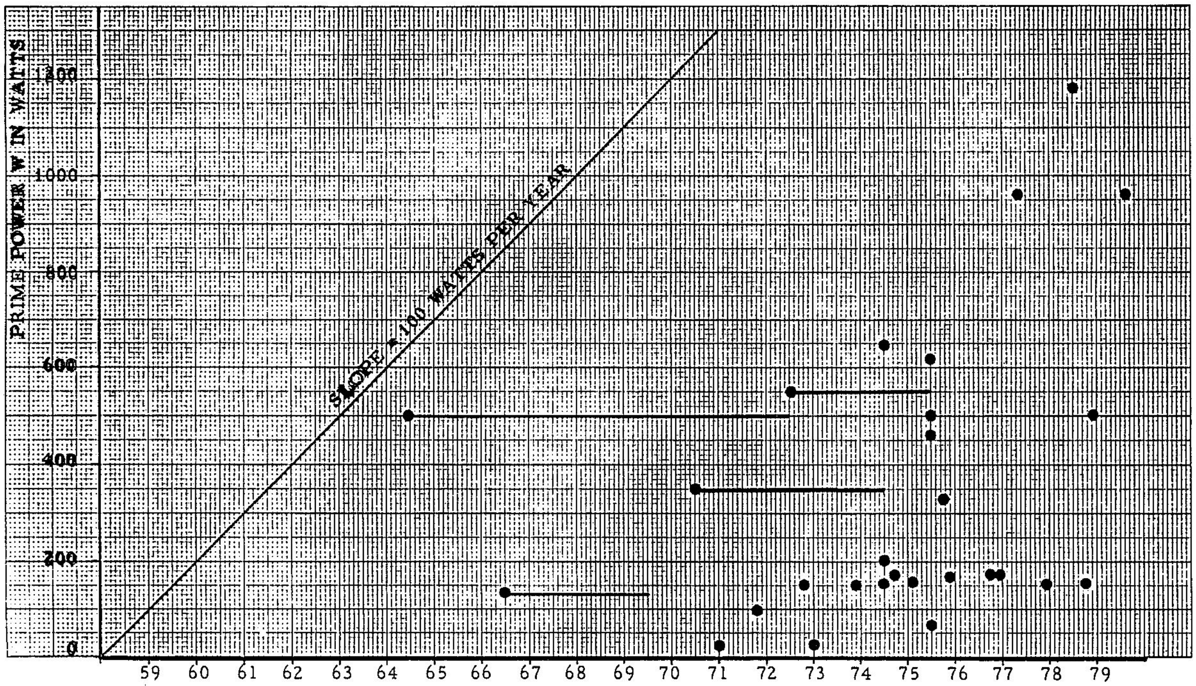


Figure 1. NASA Satellites Prime Power Trend, 1959-1979

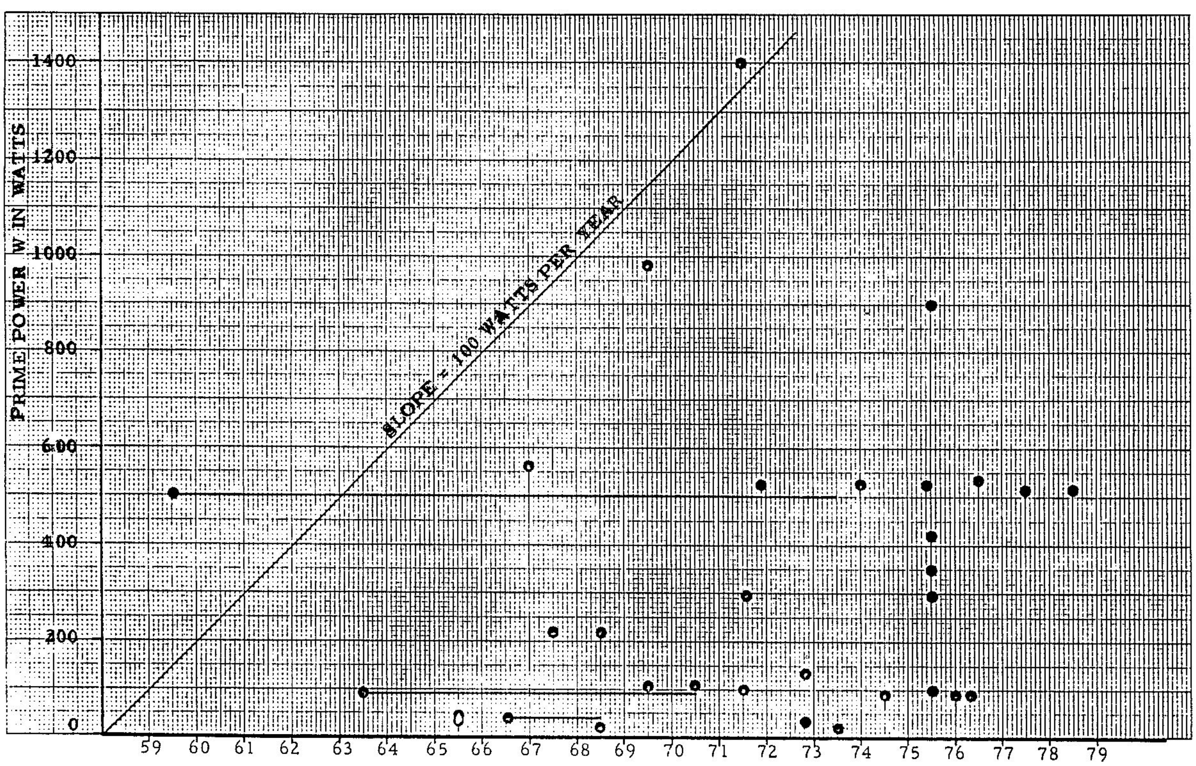


Figure 2. DoD Satellites Prime Power Trend, 1959-1979

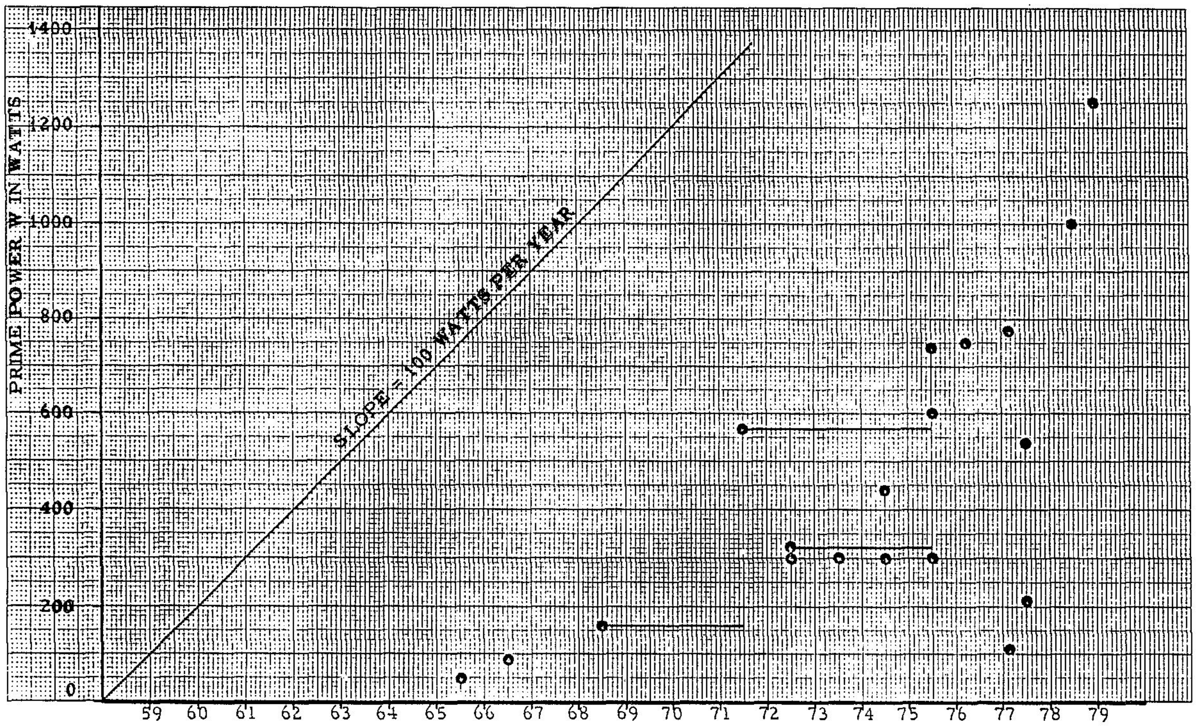


Figure 3. Civil Satellites Prime Power Trend, 1959-1979

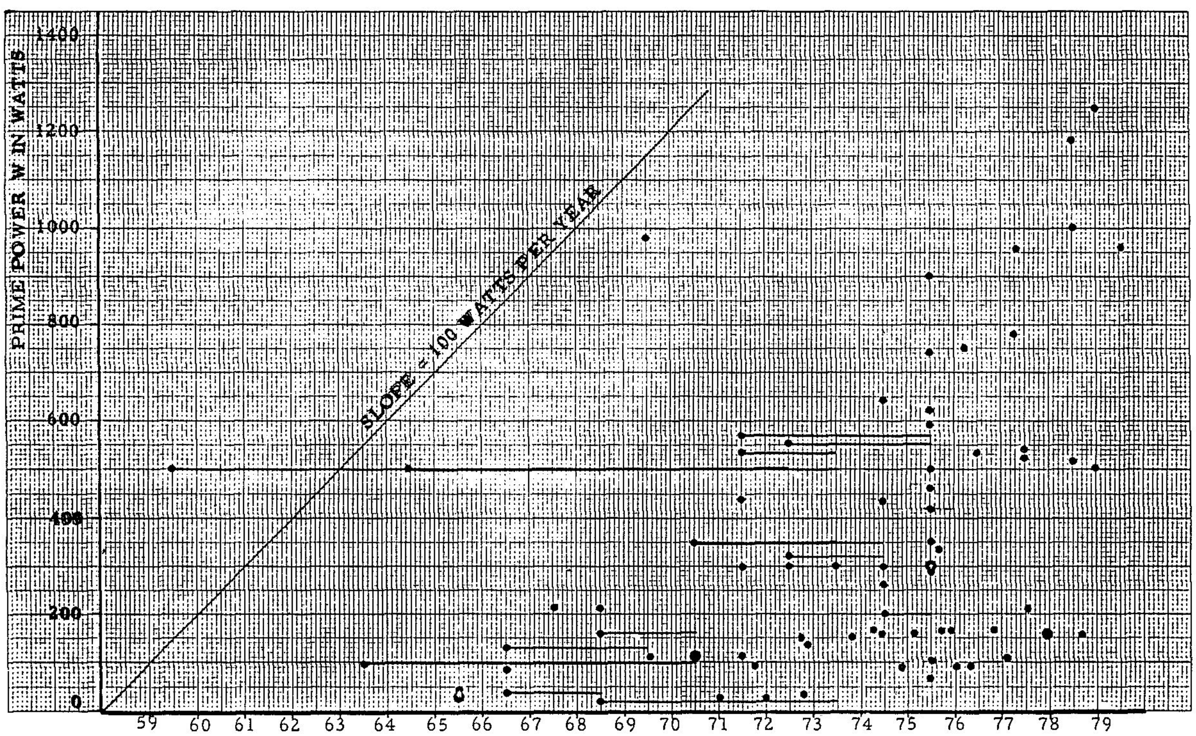


Figure 4. All Satellites Prime Power Trend, 1959-1979

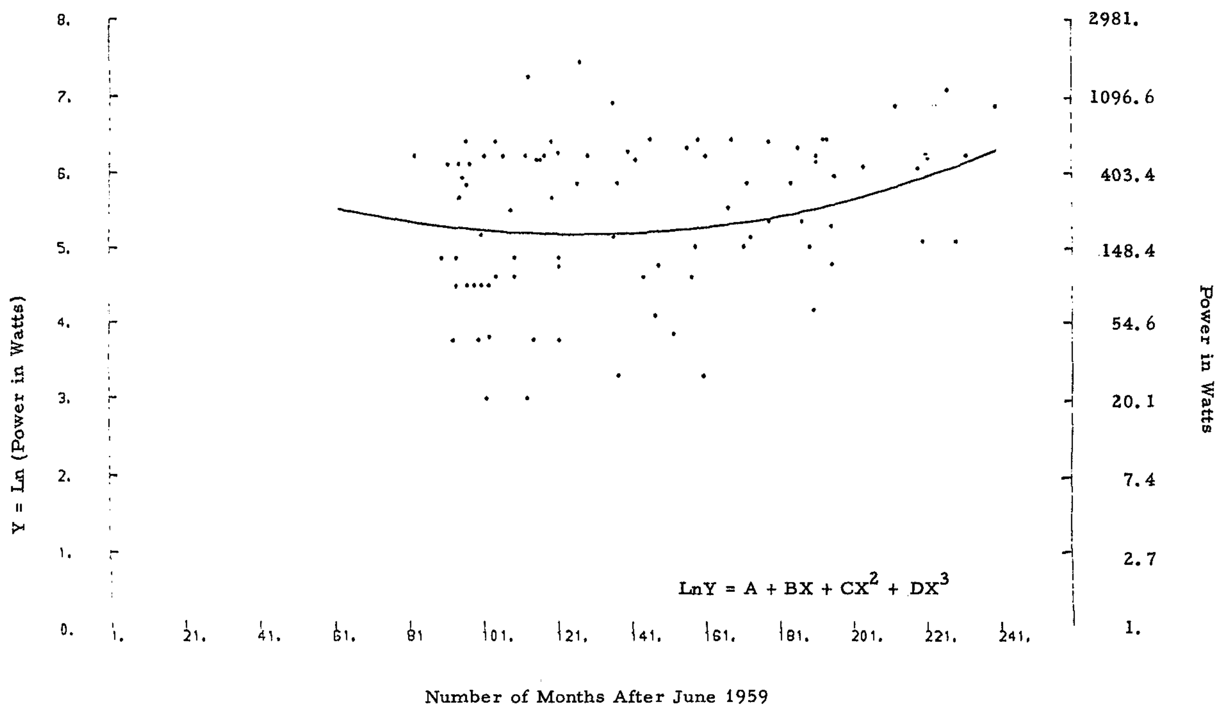


Figure 5. Satellite Prime Power Regression Analysis - NASA

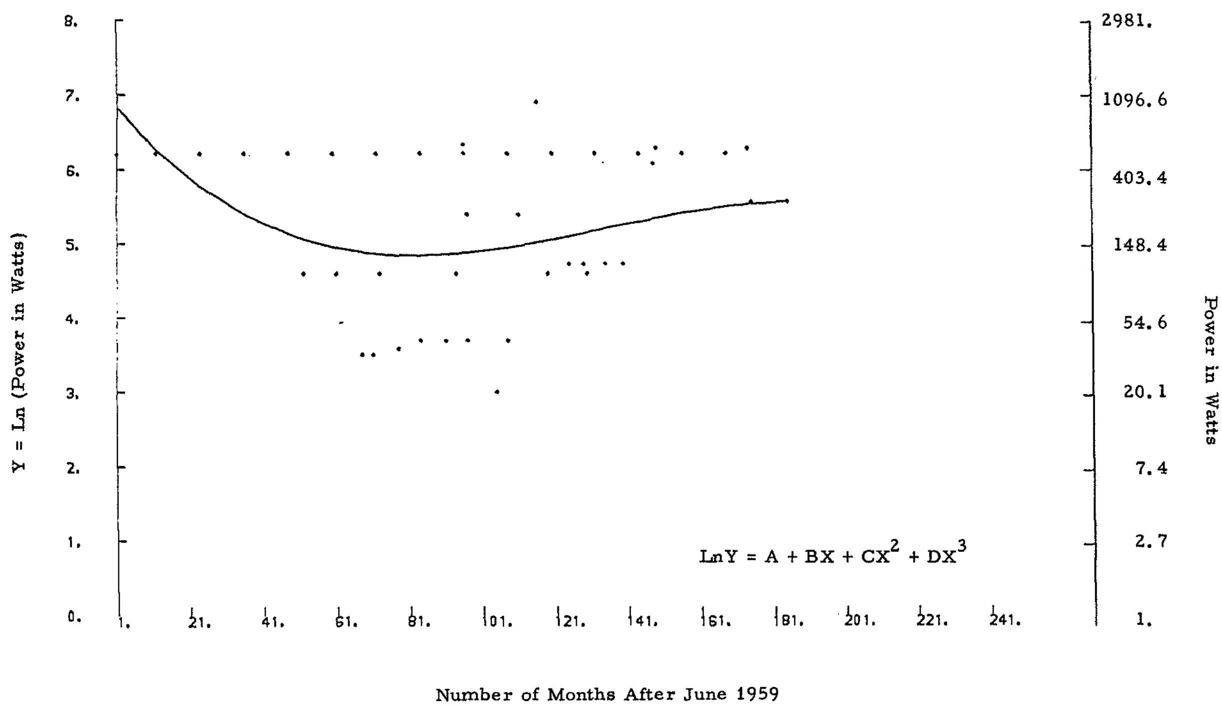


Figure 6. Satellite Prime Power Regression Analysis - DoD

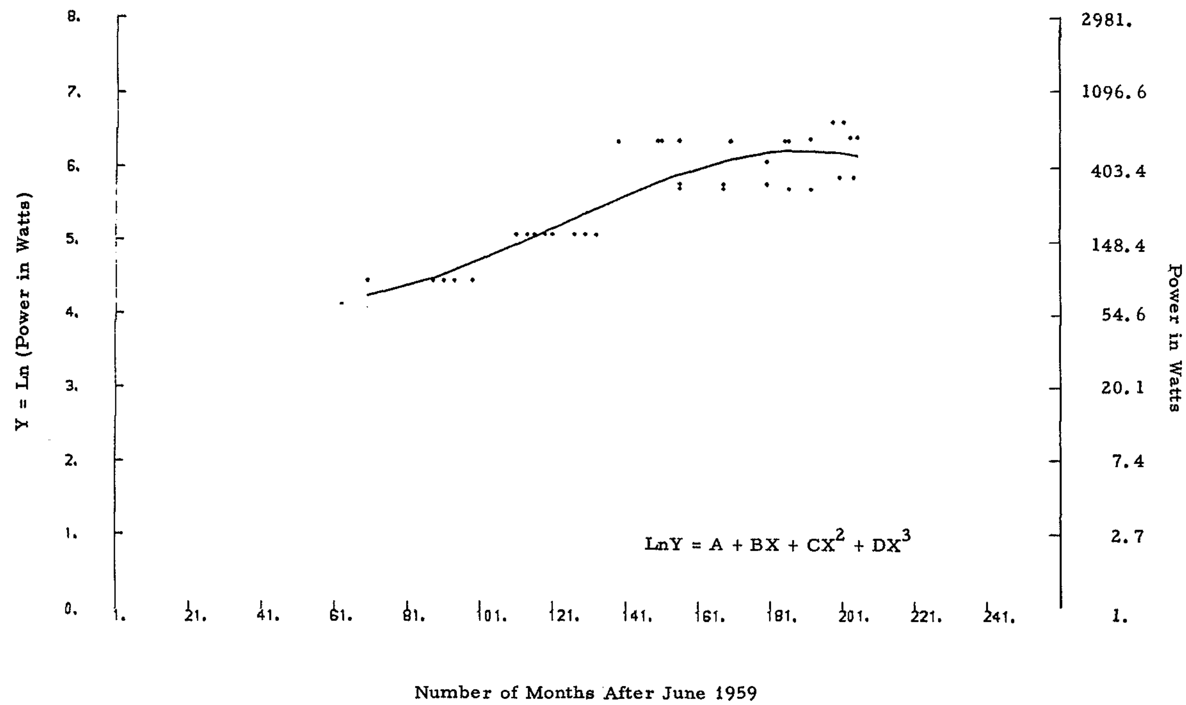


Figure 7. Satellite Prime Power Regression Analysis - Civil

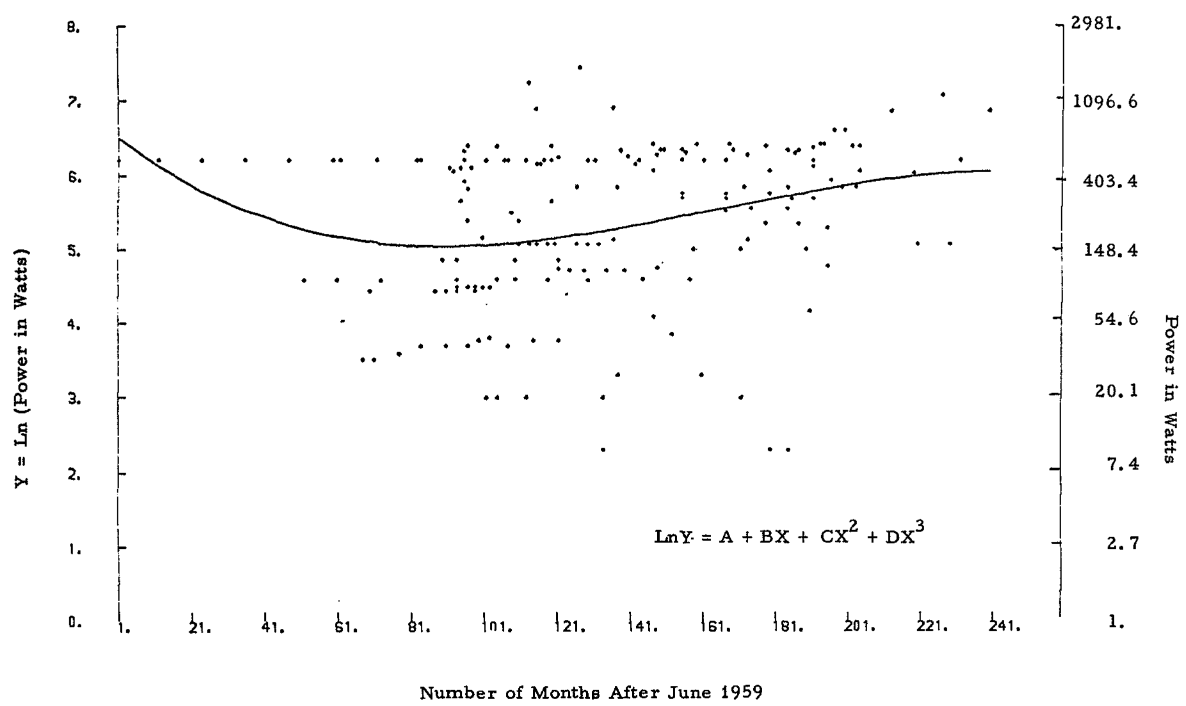


Figure 8. Satellite Prime Power Regression Analysis - All Launches

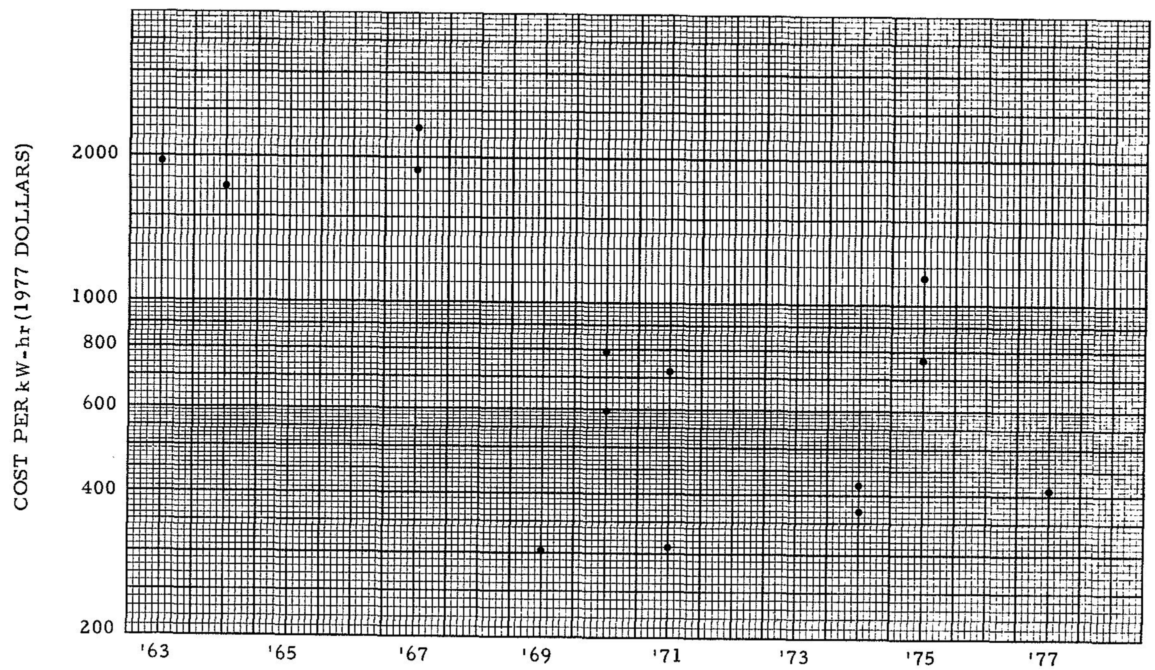


Figure 9. Electrical Subsystem Cost per kW-hr vs Year of First Flight

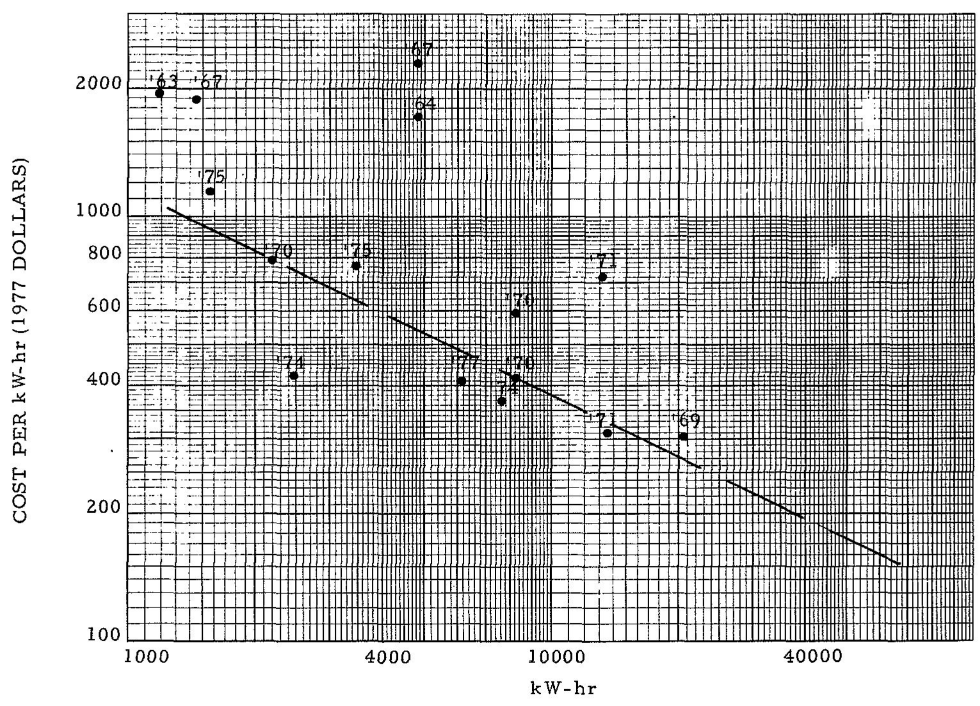


Figure 10. Electrical Subsystem Cost per kW-hr vs kW-hr

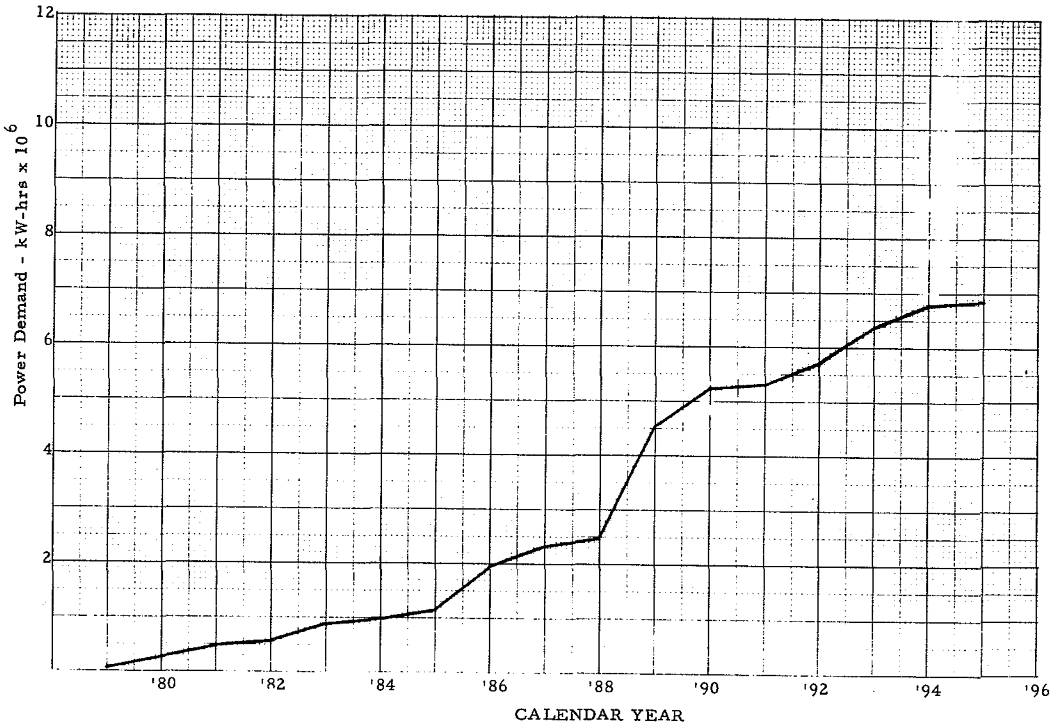


Figure 11. Total Space Energy Demand, 1985-1995 - Nominal Budget

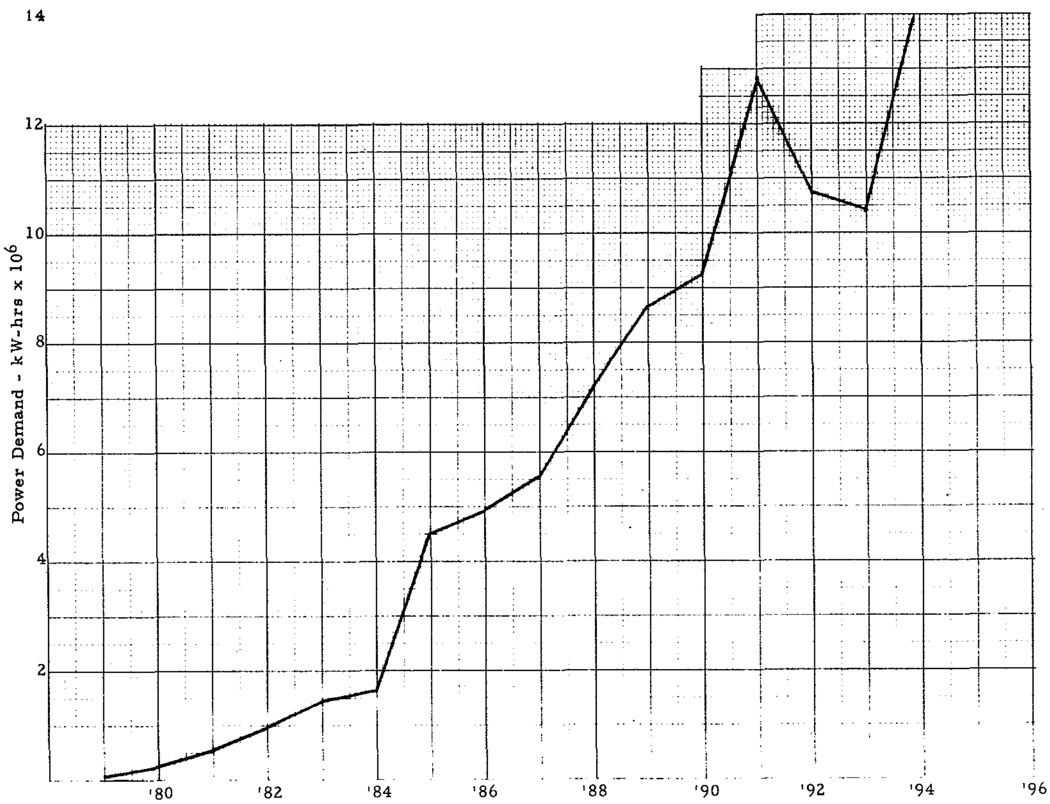


Figure 12. Total Space Energy Demand, 1985-1995 - Optimistic Budget

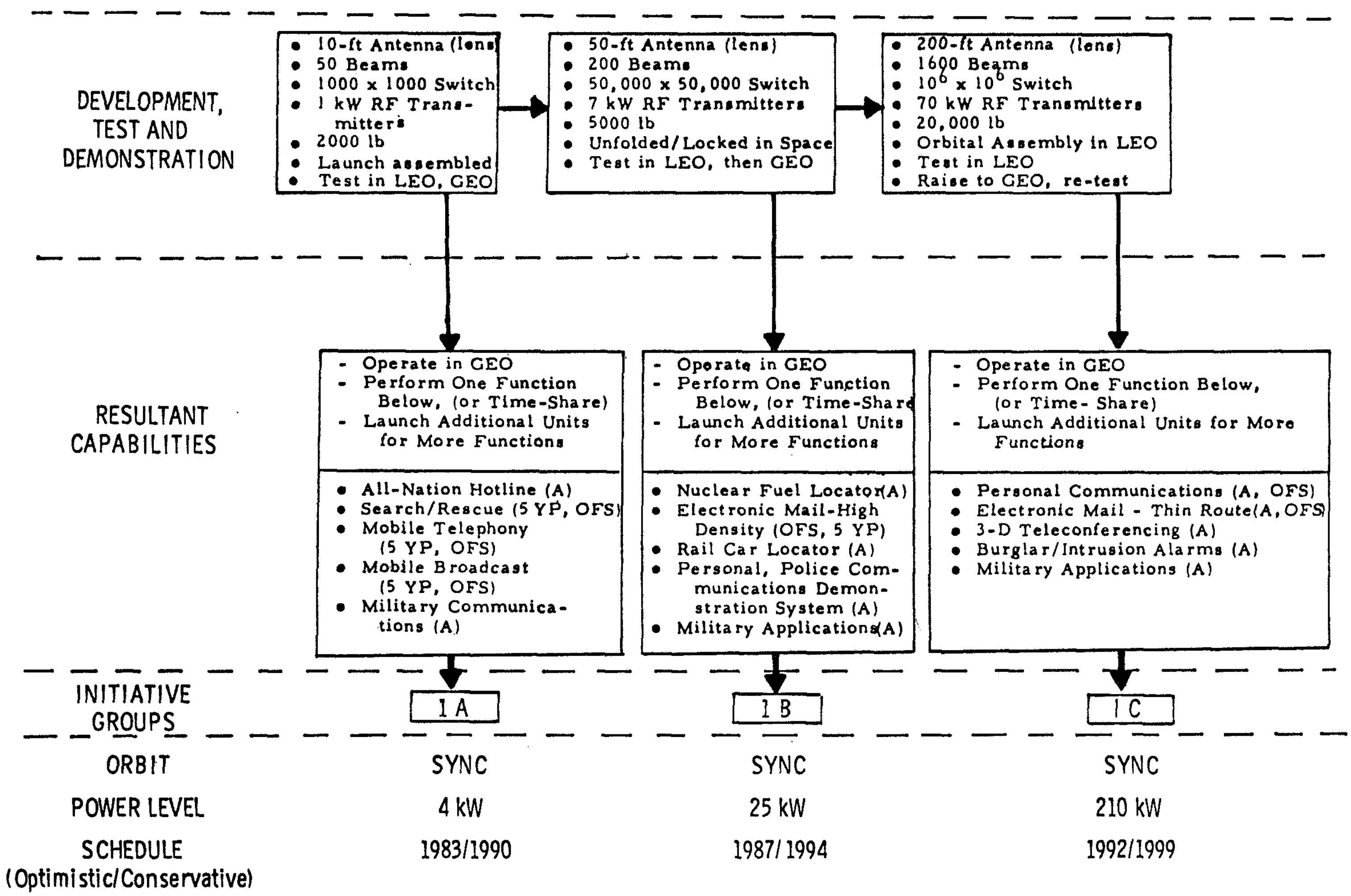


Figure 13. Group 1 Initiatives

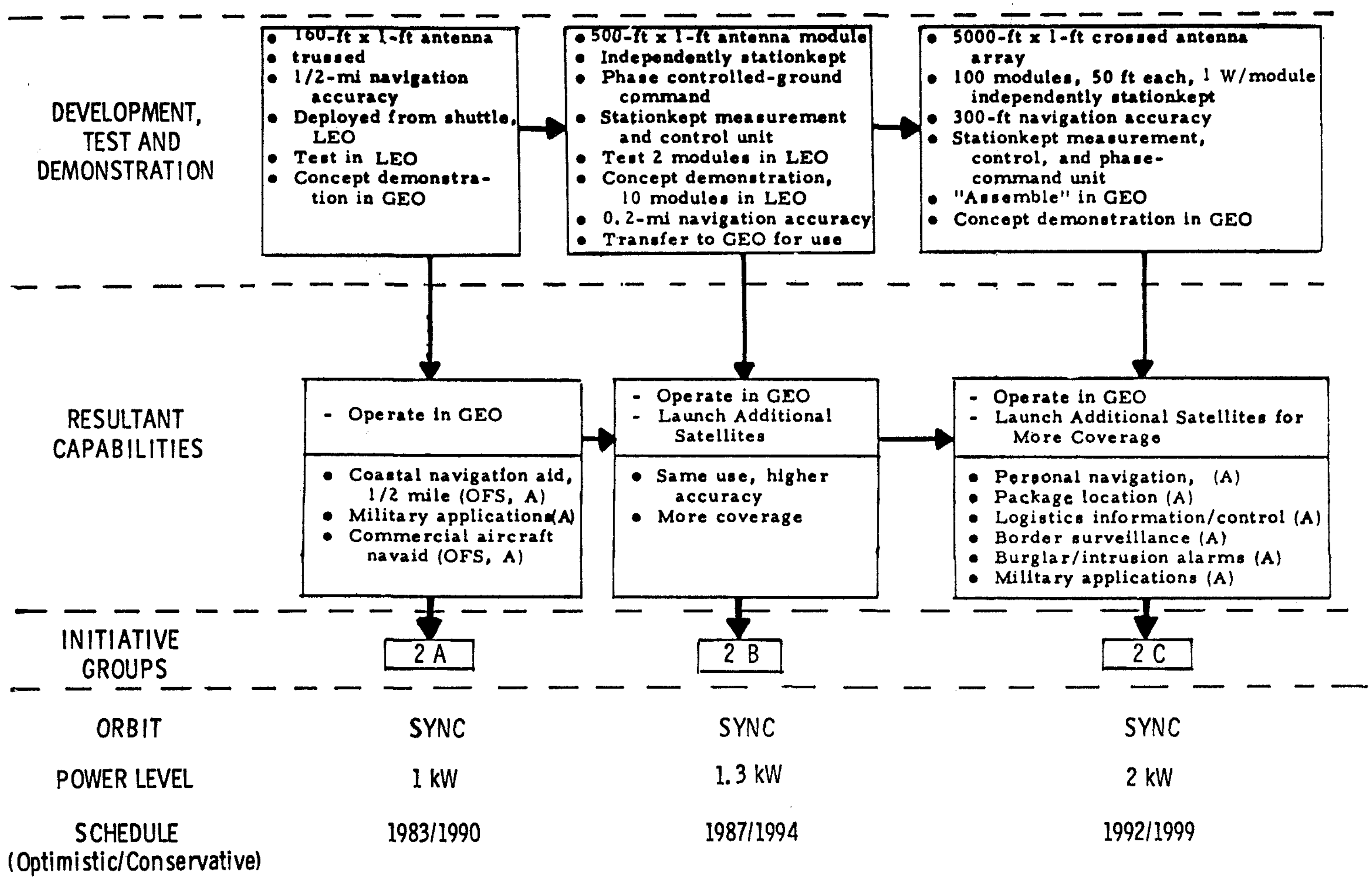


Figure 14. Group 2 Initiatives

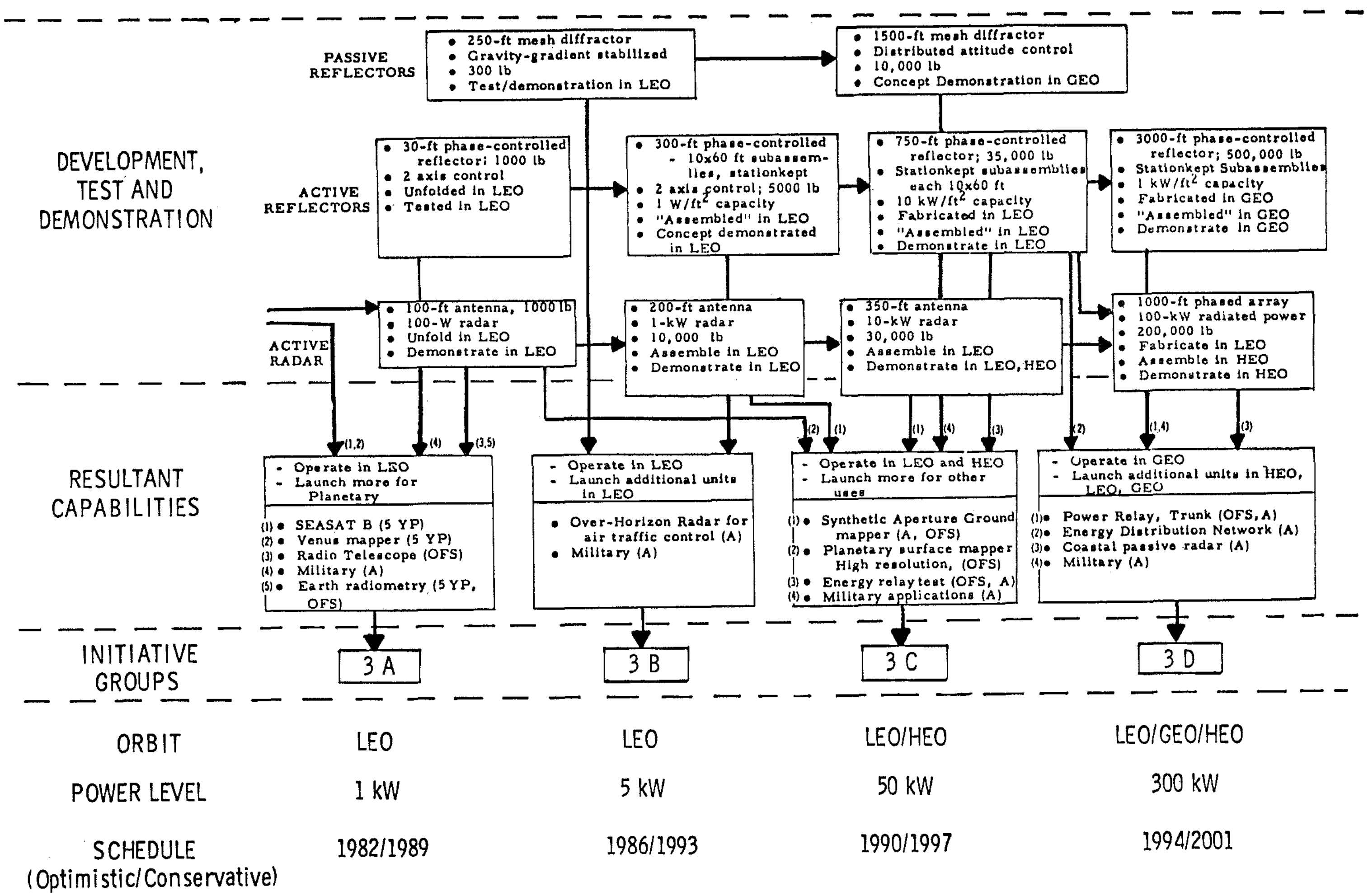


Figure 15. Group 3 Initiatives

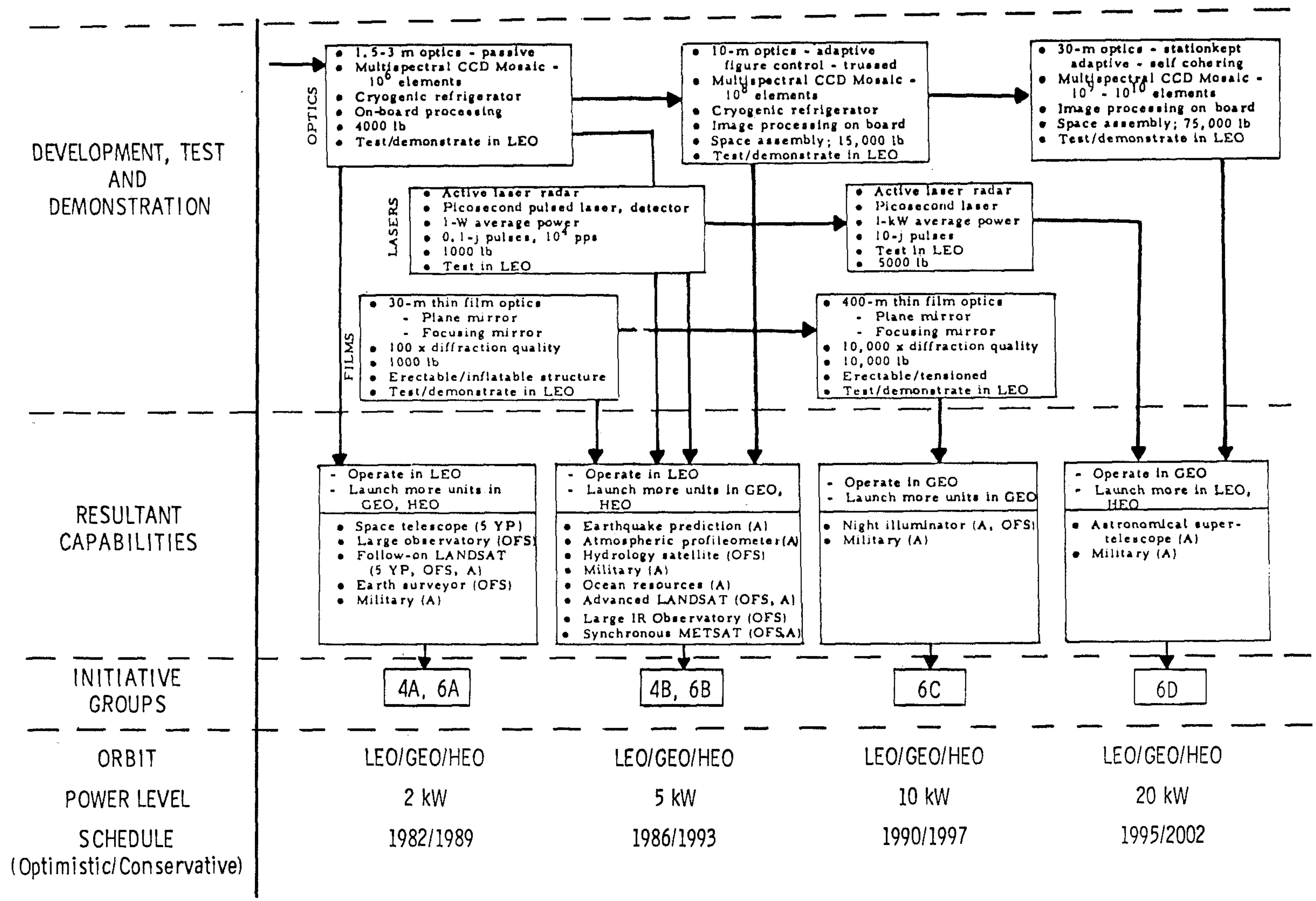


Figure 16. Group 4 and 6 Initiatives

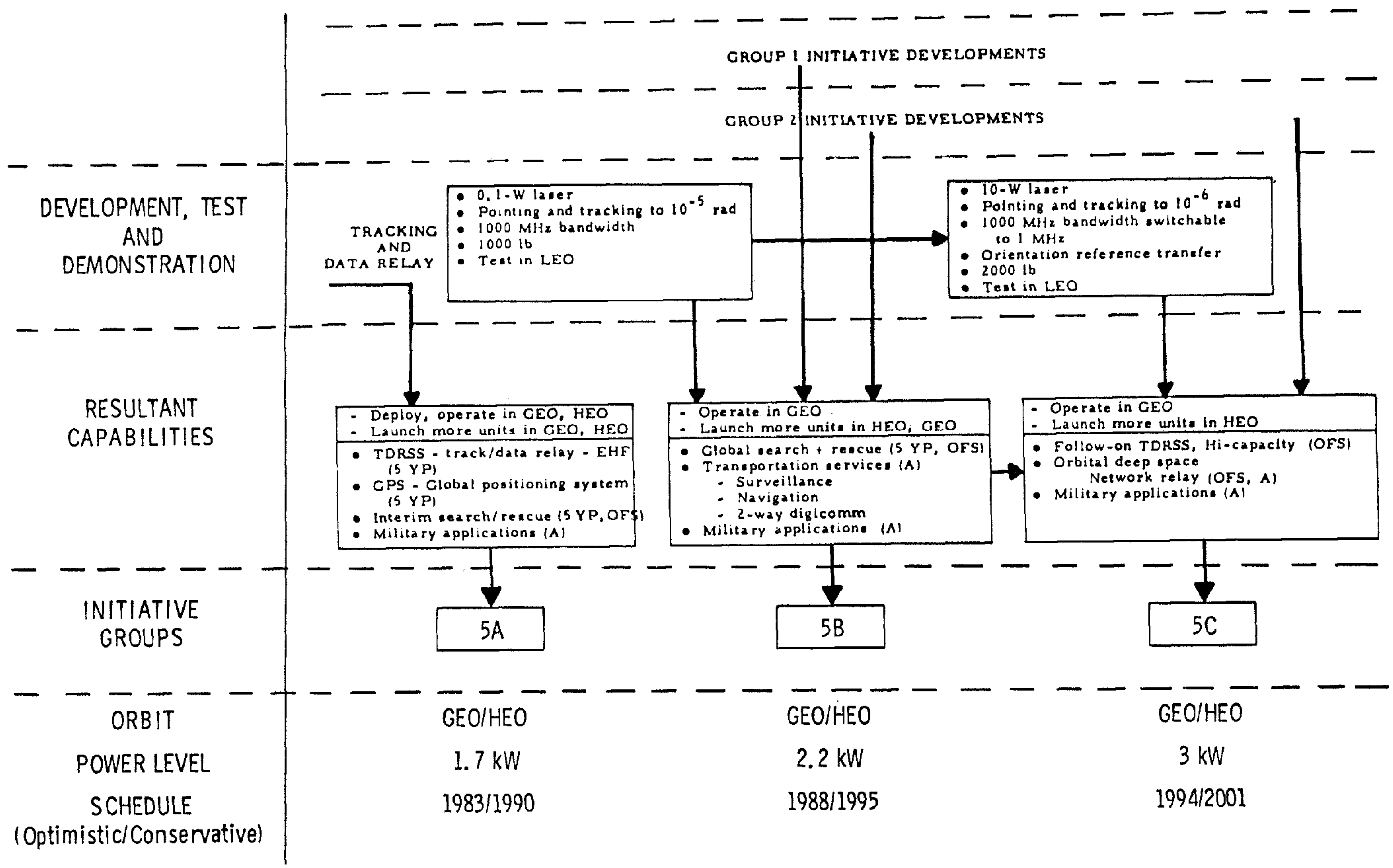


Figure 17. Group 5 Initiatives

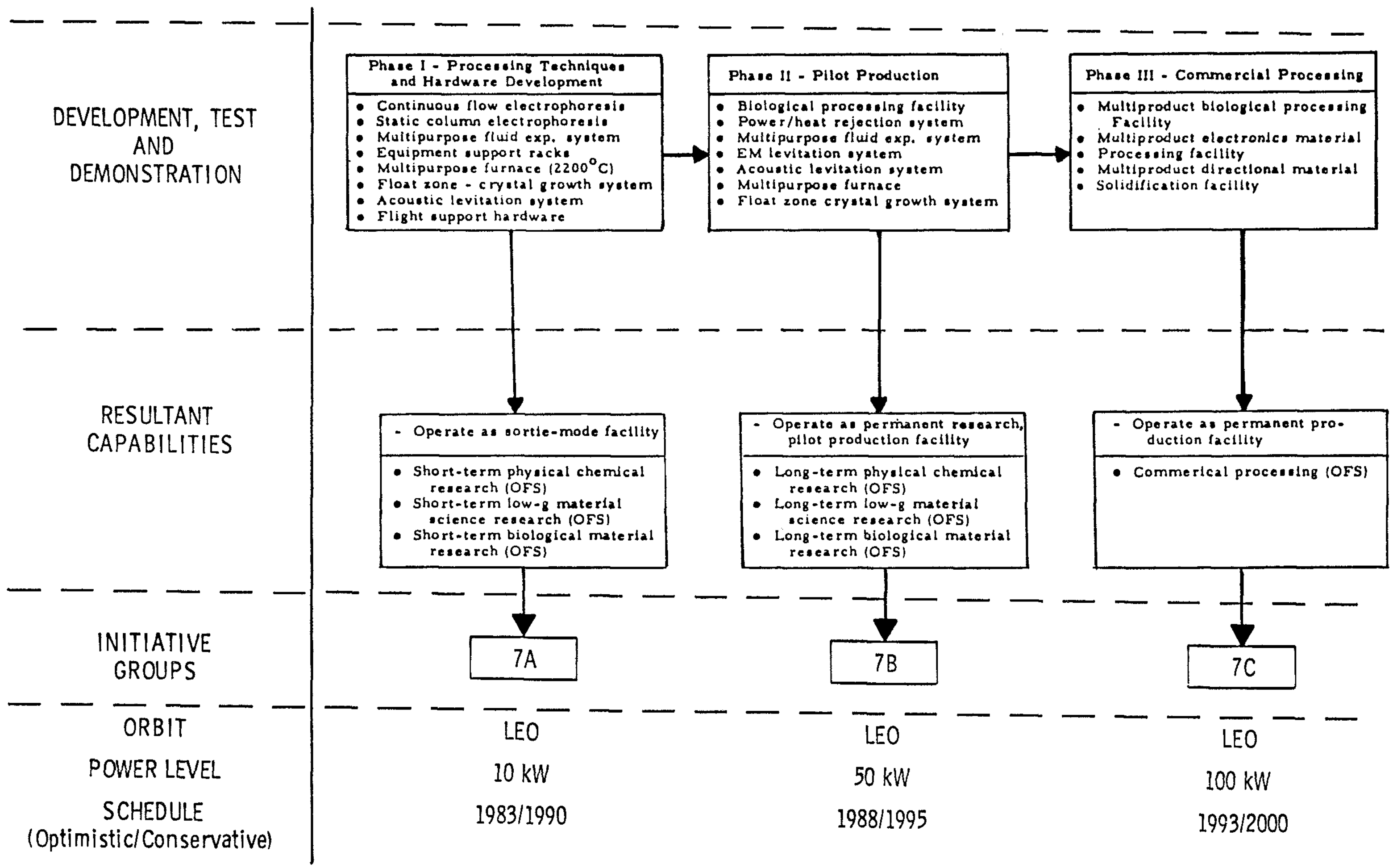


Figure 18. Group 7 Initiatives

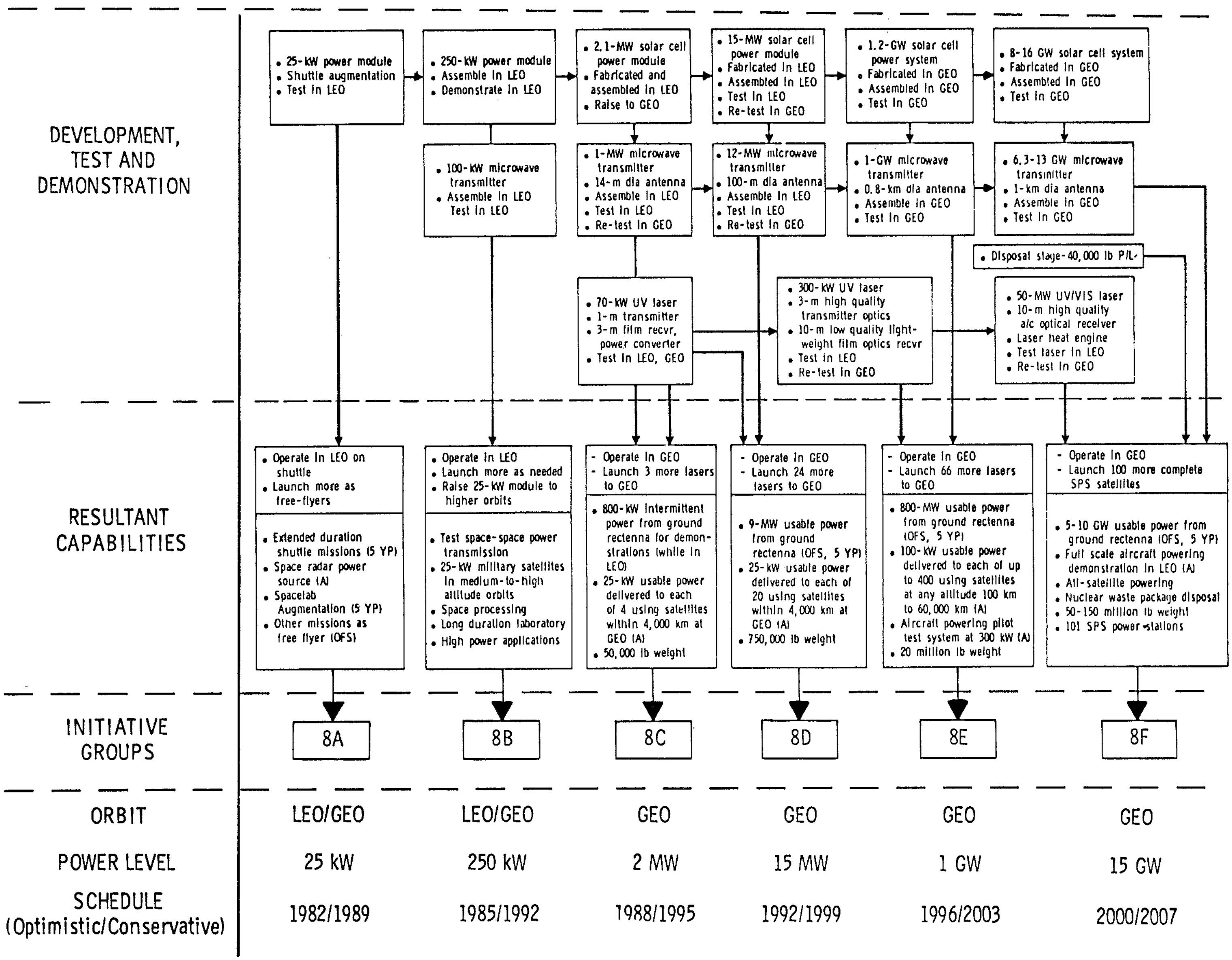


Figure 19. Group 8 Initiatives

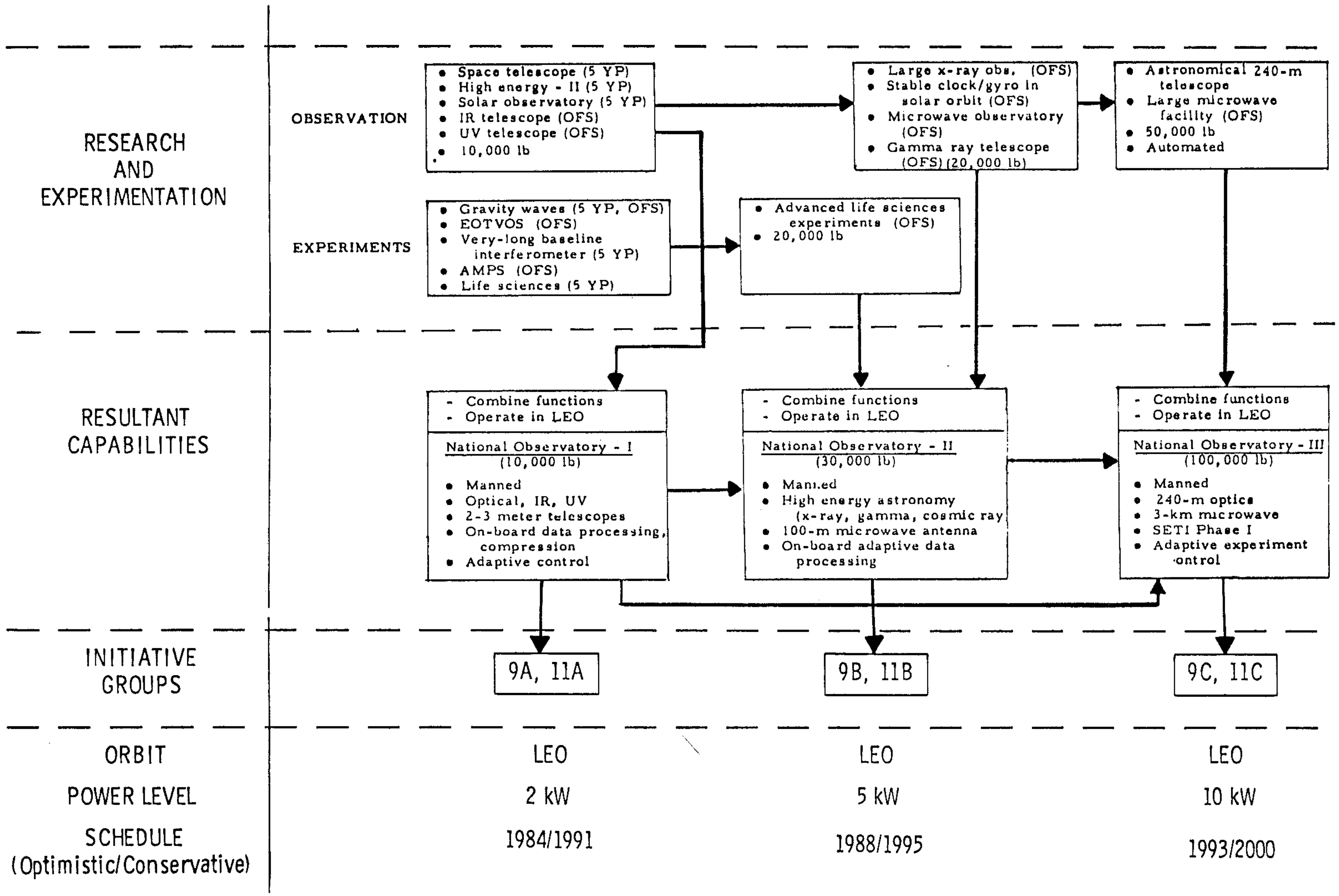


Figure 20. Group 9 and 11 Initiatives

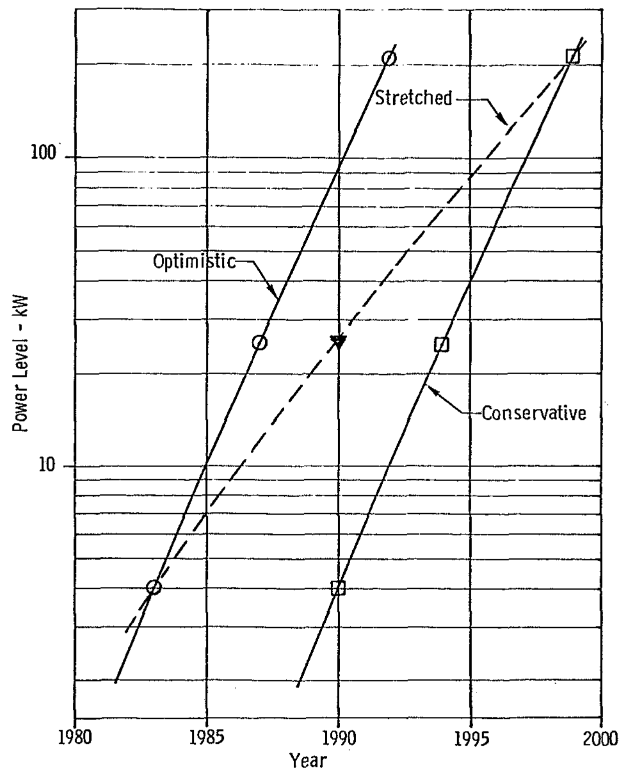


Figure 21. Power Requirements - Group 1 Initiatives

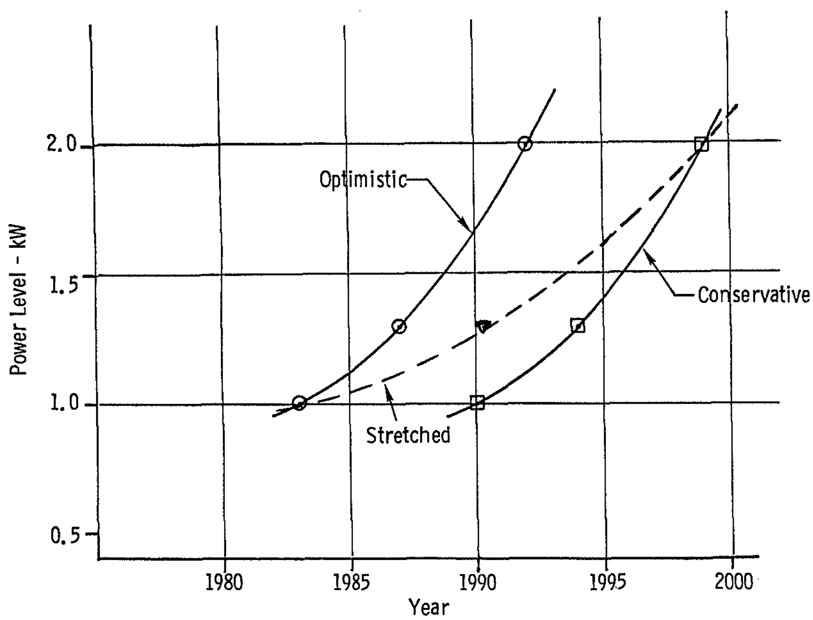


Figure 22. Power Requirements - Group 2 Initiatives

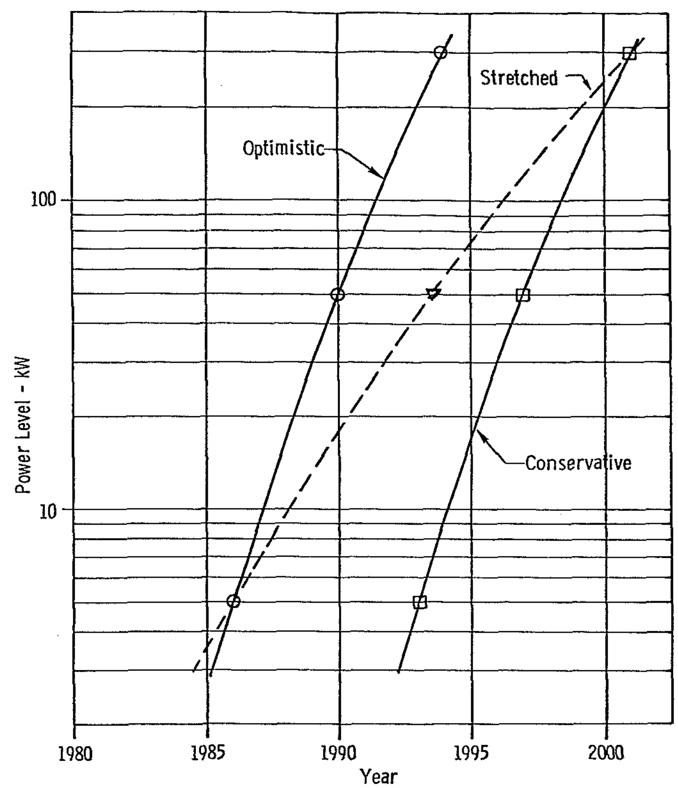


Figure 23. Power Requirements - Group 3 Initiatives

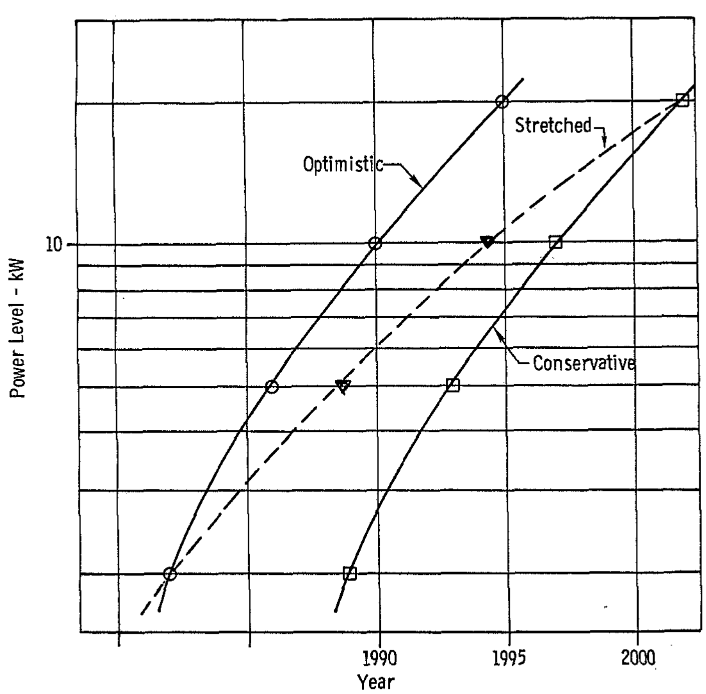


Figure 24. Power Requirements - Group 4 and 6 Initiatives

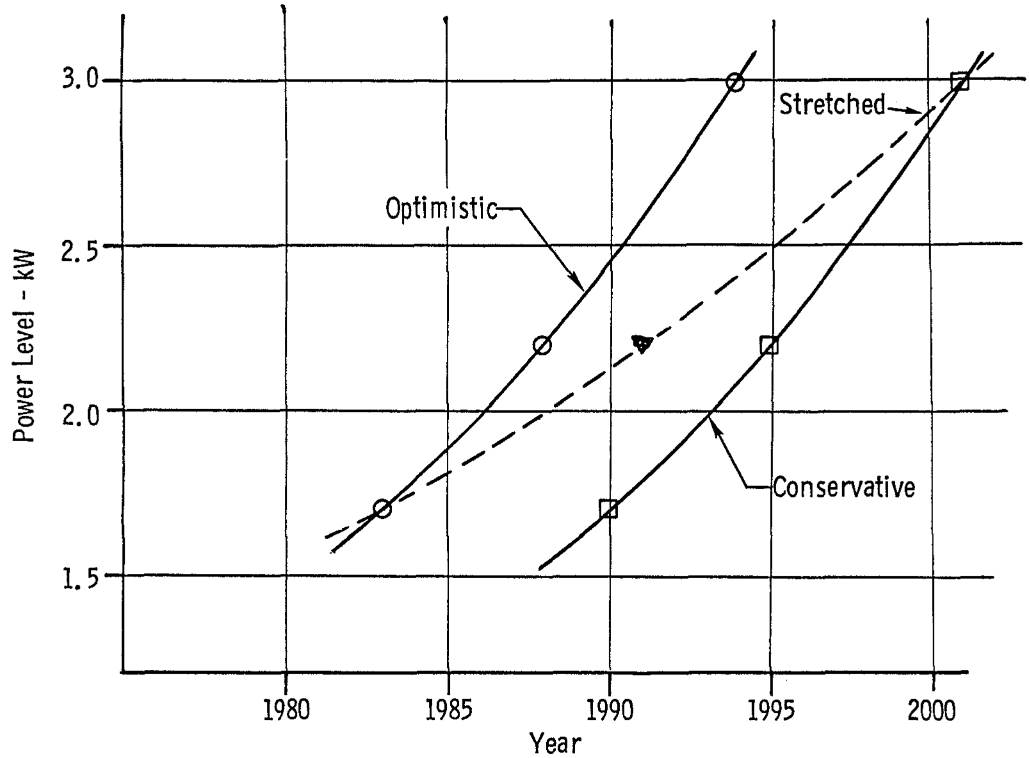


Figure 25. Power Requirements - Group 5 Initiatives

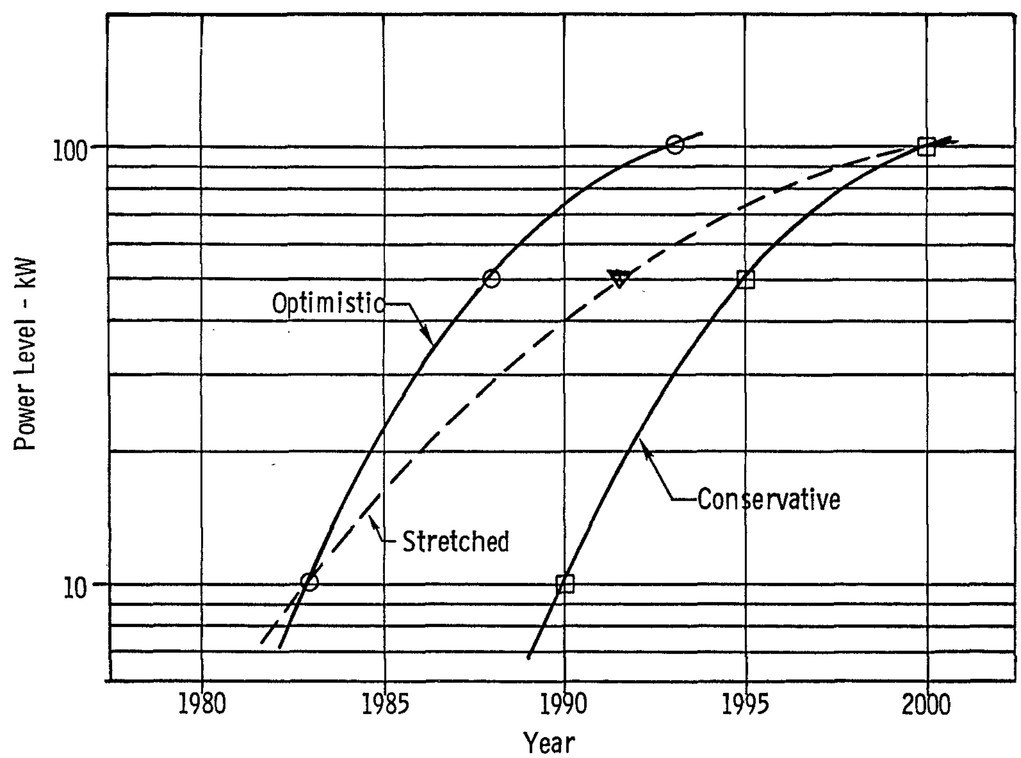


Figure 26. Power Requirements - Group 7 Initiatives

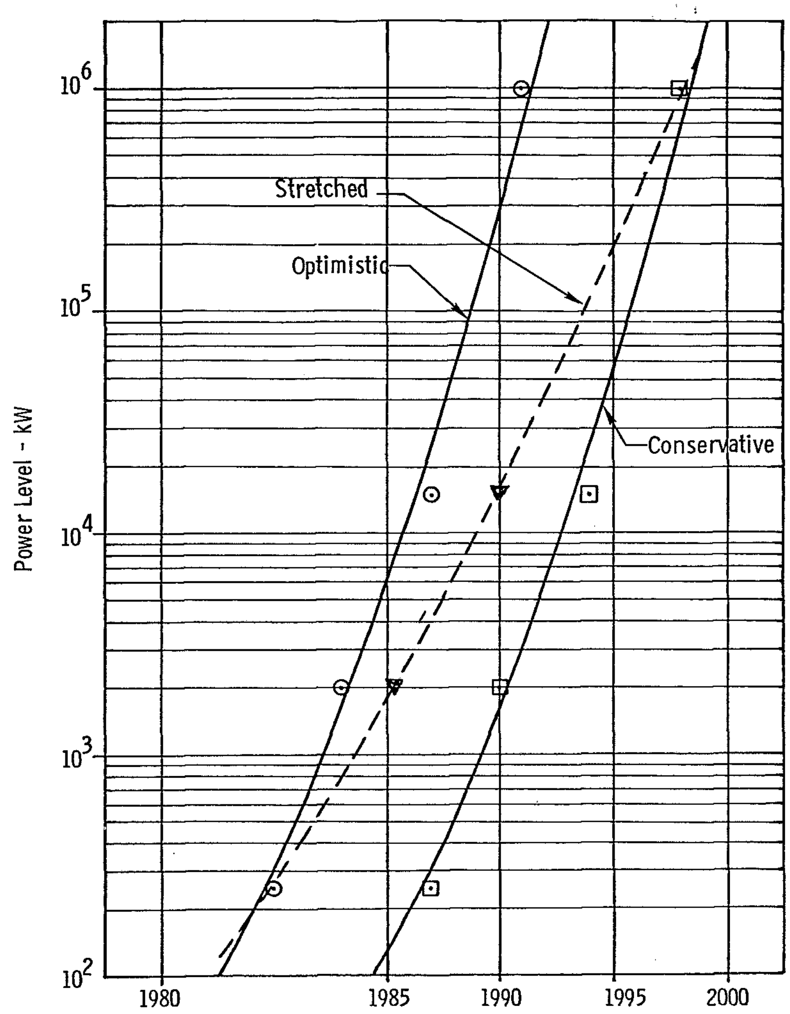


Figure 27. Power Requirements - Group 8 Initiatives

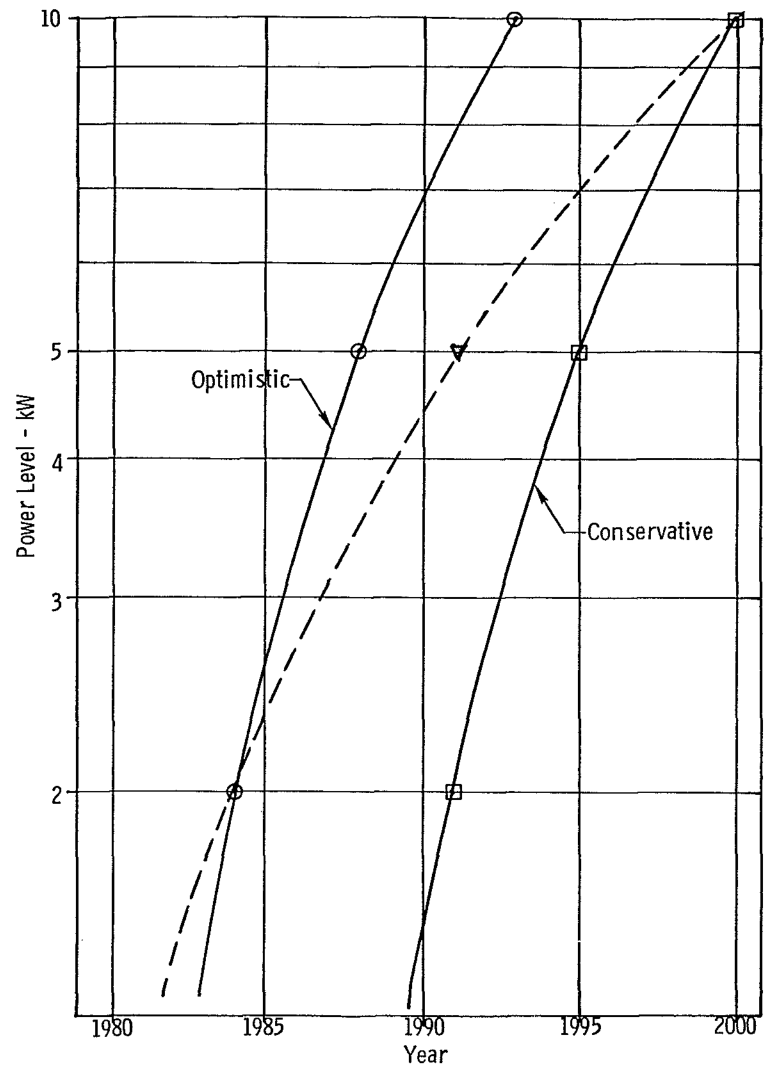


Figure 28. Power Requirements - Group 9 and 11 Initiatives

OVERVIEW OF OFFICE OF SPACE TRANSPORTATION SYSTEMS

FUTURE PLANNING

Melvyn Savage and J. William Haughey
NASA Headquarters

SUMMARY

This paper summarizes the Space Transportation Systems' key milestones as well as the future planning of the Office of Space Transportation Systems. It includes a brief description and identification of candidate new starts with target development initiation and first flight dates.

INTRODUCTION

With the Space Shuttle era about to commence, how to obtain the most benefits from these new OSTS systems is of primary interest to OSTS planners. As a result of evaluating approved systems capabilities, the new opportunities of the Space Shuttle era, and the capabilities need to take advantage of these new opportunities, the Advanced Programs Division has identified eight near-term candidate new-start activities. This paper begins with a brief summary of the Space Transportation Systems' schedule but primarily addresses these new start candidates.

MISSION OF THE OSTS

The mission of the Office of Space Transportation Systems (OSTS) includes the definition, design, development, production, operations, and management of the Space Transportation System. The present thrust of the OSTS mission is to effectively exploit the capabilities of the Space Transportation System. Specific goals of the OSTS are enumerated in Figure 1.

TARGET MILESTONES OF THE OSTS

The approved space transportation target milestones are summarized in Figure 2. As indicated, the first manned orbital flight is targeted for mid-1979, with the first operational flight planned for a year later. By late 1983, the fourth orbiter should have been delivered. Spacelab first delivery is planned for early 1980. The decade of the eighties is aimed at effectively operating the newly developed reusable Shuttle system. It marks a major step forward in reducing the costs of transporting men and material to and from space.

SPACE SYSTEMS ENGINEERING

Advanced Programs is responsible for the definition of OSTS new-start candidate programs. Until the Shuttle is operational, future space systems engineering will focus on systems enhancing the usefulness and effectiveness of the Shuttle. As a consequence, studies and engineering support associated with permanently manned space stations in low Earth orbit and geosynchronous orbit and other ambitious missions have been deferred.

The present focus of space systems engineering is summarized in Figure 3. It includes extending and enhancing the mission duration of the Orbiter and Spacelab up to 90 days, providing supplementary on-orbit electrical power for experiments and payloads, enabling orbiter-tended space construction and assembly of large space structures, and studying the requirements and concepts for transportation of larger payloads from low to high Earth orbit.

SPACE SYSTEMS ENGINEERING PROGRAM SCHEDULE

The Advanced Programs planned near-term new-start candidates--initiation of Phase C/D ($\triangle$) and initial operations capability ($\blacktriangle$)-- are presented in Figure 4. As indicated, definition system and subsystem studies and engineering support activities for these candidates are planned. These are our currently identified flight systems development candidates for initiation in the FY 80-82 time period. We will, of course, continue to study concepts and systems together with the other NASA Program Offices that could become new-start candidates in the FY 81-84 time period. Examples are geosynchronous platforms, low Earth orbit construction platforms, a manufacturing experimentation module to be attached to a free-flying power module, solar terrestrial observatory platforms, a life science experimentation module, and a solar power satellite demonstration platform.

SKYLAB REBOOST/DEORBIT MISSION

For several years we have been studying Teleoperator Retrieval System concepts. With the decision to proceed with a project to either preserve Skylab in orbit or provide for a controlled reentry, various concepts to accomplish this mission were evaluated. A system called the Teleoperator Retrieval System (TRS) was selected. This reusable system, of course, has significant potential for a variety of other missions. For example, it can be used to retrieve satellites and stabilize them for Shuttle pickup. It can also be used for stand-off space operations close to the Shuttle itself. The TRS is shown in Figure 5. It will be available in late 1979 for delivery to orbit by the Shuttle. It will be carried to Earth orbit in the payload bay of the Shuttle in an early Orbital Flight Test (OFT) flight. When the Shuttle is in position, the TRS will be ejected from the Shuttle's payload bay and flown to the Skylab. Then, controlled by a video link from the Shuttle, it will dock with Skylab and, if the reboost mission is selected, boost Skylab to an altitude of approximately 220 n. mi. The TRS will then undock and remain in orbit to be retrieved on a later Shuttle flight for subsequent use. The deboost-mission profile is similar except that after separation from the TRS, Skylab will go to a planned impact in a broad ocean area. The choice between reboost and deboost need not be made until late in the development program for the TRS.

25-KW POWER/SERVICES MODULE

Additional on-orbit electrical power and duration, beyond that available from the baseline Shuttle orbiter, is required (Figure 6). Two competitive technical approaches for provision of additional on-orbit needs are being evaluated by NASA Headquarters.

The Marshall Space Flight Center (MSFC) approach (Figure 7) is based upon proceeding directly to develop a 25-KW Power Module that is based on utilizing available subsystems or subsystems already in development. The module provides additional on-orbit electrical power, heat rejection, active stabilization, and can be operated in a sortie mode with the Shuttle and Spacelab. It can also be used as a free-flyer to support palletized payloads.

The Johnson Space Center (JSC) technical approach (Figure 8) is based on an incremental phased approach, to provide augmented power, heat rejection and duration capability for Shuttle missions. The JSC initial increment is a remote manipulator system deployed solar array carried up and down on each mission that augments the Shuttle fuel cells on the sun side of each orbit. It allows variations in fuel cell output to tailor missions to particular power and duration requirements. The second increment then involves developing a free-flyer module to handle those requirements that cannot be accommodated with the deployed array initial increment.

KEY FACTORS IN CONCEPT SELECTION

Significant factors driving the on-orbit augmentation concept selection include: Users requirements (i.e., power, heat rejection, duration and when required), the missions and operations system flexibility, the need and timing for use, the orbit inclination and duration flexibility, the relative growth potential, and the costs versus benefits accrued.

These and other relevant factors are presently being evaluated with near-term concept selection intended to allow development initiation in FY 80.

SPACE TETHER

The Space Tether (Figure 9) can provide a needed operational capability that is presently not available. Scientific measurements of Earth from Earth's upper atmosphere cannot presently be made continuously or efficiently. Satellites decay in orbit in the upper atmosphere and deorbit in a matter of hours. Sounding rockets give only short-duration readings as they pass up and down through the upper atmosphere. The Tethered Satellite System (Space Tether) offers a means for performing a wide variety of relatively long-duration scientific and operational missions in the upper atmosphere. The instruments capable of performing Earth-dynamics or atmospheric and space plasma physics measurements can be suspended by the Space Tether from the payload bay of the Space Shuttle and trolled through the Earth's upper atmosphere at altitudes of approximately 120 kilometers. Two Phase B definition studies are presently under way and should be completed in early 1979. A verification flight should be accomplished in 1982 and initial operational capability (IOC) is planned for 1983.

SPACE FABRICATION DEMONSTRATION

In-orbit fabrication of structural elements offers promise as a method for efficiently packaging and transporting to space the materials for large space structures and as a concept for highly automated construction (Figure 10). Full development of a space fabrication capability will require an orbital demonstration to proof test a space fabrication machine, handling fixtures, and structural assembly concepts. The demonstration will have to include transporting structural materials into space, fabricating structural elements or trusses using the fabrication machine, assembling the structural elements into a useful space configuration, and mounting sensors for test and applications purposes.

In FY 1979 and FY 1980 the Grumman-built beam builder will be at the Marshall Space Flight Center for test, evaluation, and demonstration. In addition, some study effort will be initiated to evaluate modification of the beam builder to fabricate beams using composite materials. Finally, a study will be accomplished to define a flight demonstration of space fabricating equipment.

A space fabrication and erection test flight and a space construction platform will provide engineering support for NASA missions involving large space systems. In the early 1980's relatively small platforms can obtain scientific and global type information while communications would be advanced by developing a narrowband technology antenna. In the mid-1980's medium-sized platforms at mid latitudes and small polar platforms would provide increased scientific capability, and a large power module would be available for a range of activities. The late 1980's and early 1990's might see the development of small geosynchronous platforms, space science laboratories, cryogenic telescopes, global services platforms, narrowband communications satellites, and an SPS test article. Late in the 1990's would possibly see the placing in orbit of SETI antenna, and solar power satellites.

SATELLITE PLACEMENT, RETRIEVAL, MAINTENANCE, AND REPAIR

The Shuttle system being developed will provide new capabilities for satellite placement, retrieval, maintenance and repair that will be very useful to Shuttle payload users in the 1980's. The systems or systems combinations that will provide this capability include the Orbiter/Remote Manipulator System (RMS), Orbiter/RMS/Extravehicular Mobility Unit (EMU), Orbiter/RMS/EMU -- Manned Maneuvering Unit (MMU), Orbiter/EMU-MMU, Orbiter/RMS/Teleoperator Retrieval System (TRS), and the Orbiter/TRS combination.

In FY 79 we will be studying system requirements and the capabilities of the above-mentioned systems to establish how currently planned capabilities can be exploited and what new capabilities will be required. Operations at "stand-off" distances from the Orbiter in low Earth orbit of 800-1600 KM will be studied to determine how to augment the TRS capabilities. We will also continue to investigate maintenance and repair activities both in low Earth orbit and geosynchronous orbit. Our studies and engineering support activities are directed to determining what capabilities should be provided in the 1983-85 time frame. The current target is a hardware start of some type in FY 1981. (See Figure 11.)

SOLAR ELECTRIC PROPULSION STAGE

NASA will use the Inertial Upper Stage (IUS) being developed by the U. S. Air Force for certain high-energy missions, including some automated planetary missions. However, studies have shown that certain planetary missions cannot be performed without assistance from an additional class of propulsive vehicle. A Solar Electric Propulsion Stage (SEPS) is one concept that has been under study for several years. Another alternative is to integrate the thrustors and solar arrays into the planetary spacecraft because the total system in a sense operates as a spacecraft for mission durations of several years. Development of a SEPS module is included in our present five-year plan as a FY 81 new start with an IOC capability of 1985. (See Figure 12.)

SKYLAB REHABILITATION

The large living quarters and crew accommodations aboard Skylab can be effective adjuncts to Shuttle-Spacelab long-duration missions. Figure 13 is an artist's sketch of Skylab being utilized with a power module to support a large space structure, space fabrication, and assembly demonstration mission. Studies are under way to identify reactivation requirements for Skylab onboard systems, subsystems, and experiments and to identify additional uses and benefits associated with rehabilitation and reuse. Representative candidate missions include the following areas of investigation:

- Degradation of materials and equipment from long-term space exposure
- Space construction engineering
- Space processing
- Bio-Science
- Communications
- Earth and space sciences

SPACE POWER SYSTEMS ENGINEERING

Studies have indicated that in the mid 80's there will be requirements for hundreds of kilowatts in orbit in order to satisfy a wide range of future new space opportunities. Figure 14A describes key missions that will require power in the 100's of KW level. These include such things as construction of large structures, materials processing, communications, solar power technology development, as well as scientific and application missions.

The most appropriate power level to be developed after the 25-KW Power Module is not known at this time. Figure 14B shows a space construction platform. Attached to and forming a part of this construction platform is a 250-KW power array that could be used in space construction operations as well as in technology demonstrations. Figure 14C shows another concept of a multi-hundred-kilowatt power module being constructed. This construction approach, of course, would be important in demonstrating space construction technology applicability to a wide range of future large structures including solar power satellites.

Recent activity involving the Offices of Space Science, Space and Terrestrial Applications, Aeronautics and Space Technology, and Space Transportation Systems has resulted in identifying potential large space systems as shown in Figure 14D. This chart shows that power levels in the multi-hundred kilowatts are likely to be required by the 1986 time period.

In responding to anticipated future needs, OSTS has been investigating how to best provide power modules of the 100's of KW's size. We are targeting for an operational capability in the 1986-1987 time period. Two systems concepts studies are presently under way (Figure 14E). These studies are investigating power modules that would have the potential of providing hundreds of kilowatt power in orbit for such space operations as materials processing, space construction, advanced communications systems, and other future applications and scientific projects. The JSC/MDAC study could be described as a "clean sheet" modular approach since it is not committed to using any existing available hardware. The MSFC/LMSC study is based on evolving from the 25-KW Power Module in an orderly and timely fashion.

During the FY 79-80 time period conceptual studies and preliminary design activities will be continued. Better definition of user requirements will be accomplished by working directly with the other program offices as they shape their long range plans. These efforts will establish the power level and best concepts for the multi-hundred KW system.

The power-related and space construction technology efforts of the next 5-6 years are important because they will most assuredly influence the capabilities and systems that are selected for this large power module. OAST future new initiatives, as well as some of the R&T base work in both the power and large structures area, are being phased to support technology needs for this system.

ORBITAL TRANSFER VEHICLE

The Space Transportation System being developed restricts manned operations to low earth orbit. Higher energy unmanned missions are constrained by the capabilities of the currently approved inertial upper stage and the spinning satellite upper stages; therefore, space systems engineering projects involving construction of large structures and assembly of large power modules will of necessity be concentrated in low Earth orbit through the mid 80's. Our planning indicates a need for an orbital transfer vehicle (OTV) having manned geosynchronous capabilities by the late 80's. Such an OTV might also be needed for such potential missions as disposal of nuclear wastes, demonstration of space power technology, and maintenance and repair of geosynchronous large space platforms. The OTV (Figure 15) is planned to be an FY 1982 development initiation.

SPACE SOLAR POWER SYSTEM DEMONSTRATION

(MULTI-MEGAWATT SYSTEMS)

The development of a large power module in low Earth orbit constitutes a desirable first step in the evaluation of the technologies necessary to the space solar power concepts. The solar energy satellite test article could utilize a large power module, a phased array transmitting antenna and a maneuverable space rectenna to conduct selected microwave tests.

Based partially on the SPS demonstration information, a commitment might be forthcoming to develop multi-gigawatt systems capable of supplying a large percentage of the national electric grid total power. The multi-gigawatt system operating from geosynchronous orbit could be operational in the late 1990's.

MATERIALS EXPERIMENTATION MODULE

A manned materials processing module which could be flown in the late 1980's in conjunction with a 250-KW Power Module is shown in Figure 16. High-value products such as semiconductor materials, optical materials and high temperature materials such as turbine blades could be produced on an economical scale. These processes will require major dedicated facilities in space that will utilize hundreds of kilowatts of power. Initially, the Space Shuttle orbiter, with additional power and on-orbit time, will facilitate the operation necessary to prove out the processing concepts and actually make some marketable materials. Beyond that, small Shuttle-tended free flyers and materials processing modules attached to a space station will make larger quantities of high-value products for earth markets.

CONCLUDING REMARKS

Eight new programs (new-start candidates) responsive to near-term future opportunities and requirements have been reviewed. These new starts extend and enhance the capabilities of the Shuttle orbiter and Spacelab, provide supplementary on-orbit electrical power for experiments and payloads, enable Shuttle Orbiter-tended space construction and the usage of the products thereof, and allow the transportation of larger payloads between low and high Earth orbit. More specific details on many of these candidates will be presented in other papers at this symposium.

OFFICE OF SPACE TRANSPORTATION SYSTEMS MISSION

- TO PROVIDE EASY, LOW COST ACCESS TO, FROM, AND WITHIN SPACE FOR PAYLOADS AND SYSTEMS DEVELOPED BY NASA AND OTHER USERS
- TO DEVELOP MORE EFFECTIVE CAPABILITIES FOR HUMANS TO LIVE, WORK, AND CONDUCT EXPERIMENTS IN SPACE FOR EXTENDED PERIODS OF TIME
- TO DEVELOP FLIGHT SYSTEMS THAT WILL ENHANCE THE STS' UNIQUE CAPABILITIES AND GREATLY EXPAND ITS USEFULNESS
- TO PLAN AND CONDUCT SPACE OPERATIONS

Figure 1.

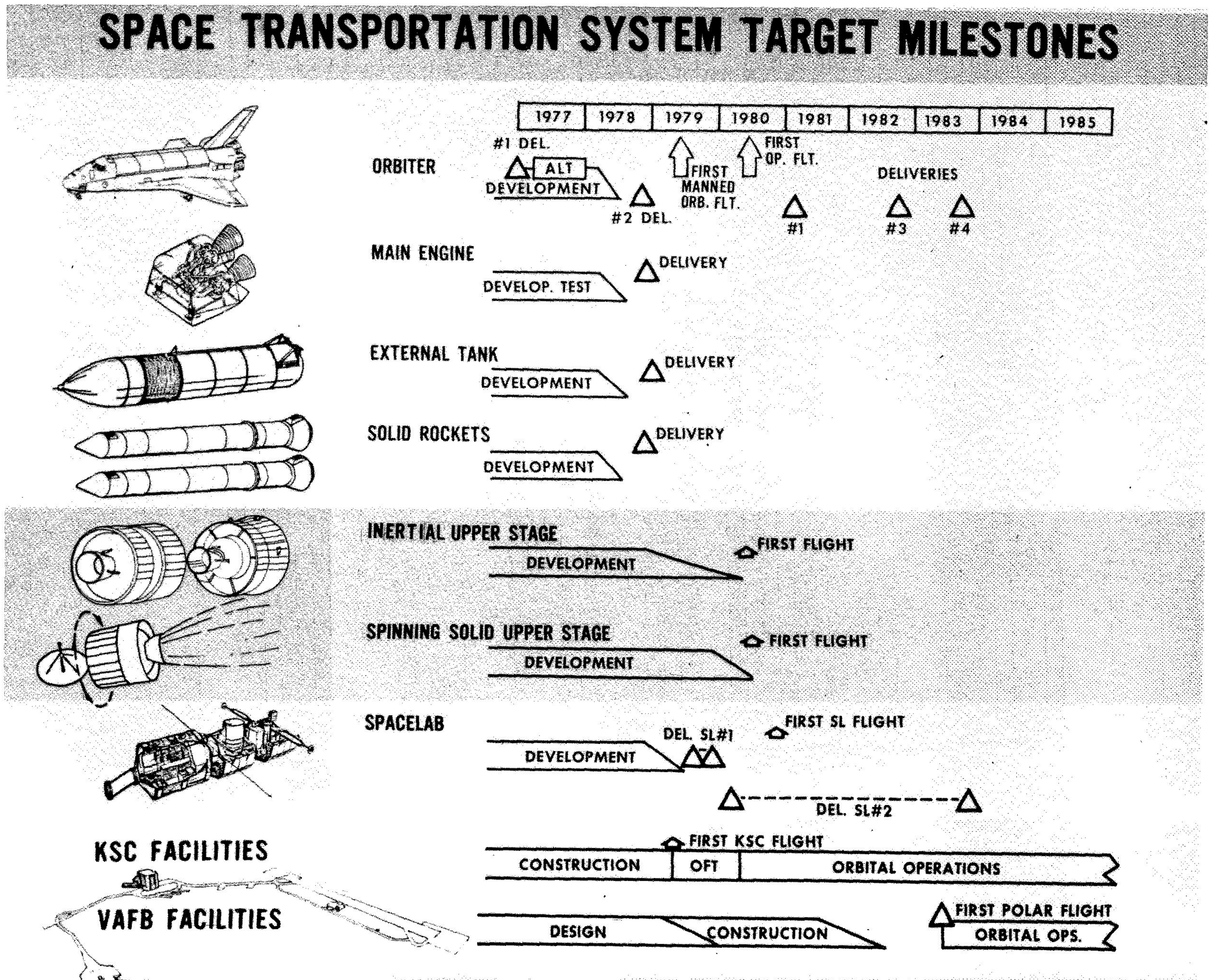

SPACE TRANSPORTATION SYSTEM TARGET MILESTONES
1977
1978
1979
1980
1981
1982
1983
1984
1985
ORBITER
#1 DEL.
ALT
DEVELOPMENT
#2 DEL.
FIRST MANNED ORB. FLT.
FIRST OP. FLT.
DELIVERIES
#1
#3
#4
MAIN ENGINE
DEVELOP. TEST
DELIVERY
EXTERNAL TANK
DEVELOPMENT
DELIVERY
SOLID ROCKETS
DEVELOPMENT
DELIVERY
INERTIAL UPPER STAGE
DEVELOPMENT
FIRST FLIGHT
SPINNING SOLID UPPER STAGE
DEVELOPMENT
FIRST FLIGHT
SPACELAB
DEVELOPMENT
DEL. SL#1
FIRST SL FLIGHT
DEL. SL#2
KSC FACILITIES
FIRST KSC FLIGHT
CONSTRUCTION
OFT
ORBITAL OPERATIONS
VAFB FACILITIES
DESIGN
CONSTRUCTION
FIRST POLAR FLIGHT
ORBITAL OPS.

Figure 2

SPACE SYSTEMS ENGINEERING

FOCUS:

- TO INCREASE THE SHUTTLE'S AND SPACELAB'S MISSION DURATION FROM THE INITIAL SEVEN TO TEN DAYS TO 60 TO 90 DAYS SO THAT VALUABLE EXPERIMENTS AND MISSIONS THAT WOULD OTHERWISE NOT BE POSSIBLE CAN BE ACCOMPLISHED.

- TO PROVIDE SUPPLEMENTARY POWER FOR EXPERIMENTS AND MISSIONS WHOSE POWER REQUIREMENTS EXCEED THE SHUTTLE'S POWER-GENERATING ABILITY.

- TO ENABLE THE SHUTTLE TO SUPPORT THE ASSEMBLY AND CONSTRUCTION IN SPACE OF THE LARGE STRUCTURES REQUIRED TO MEET PREDICTED NEEDS FOR COMMUNICATIONS, FOR THE SENSING OF VARIOUS ASPECTS OF THE EARTH'S ENVIRONMENT AND SURFACE, FOR THE PROCESSING OF MATERIALS, AND FOR THE GENERATION OF POWER IN SPACE.

- TO TRANSPORT LARGE PAYLOADS SUCH AS THE COMMUNICATIONS, EARTH SENSING, AND SPACE POWER SYSTEMS MENTIONED ABOVE FROM LOW-EARTH ORBIT TO GEOSYNCHRONOUS ORBIT AND RETURN.

Figure 3.

SPACE SYSTEMS ENGINEERING PROGRAM SCHEDULE

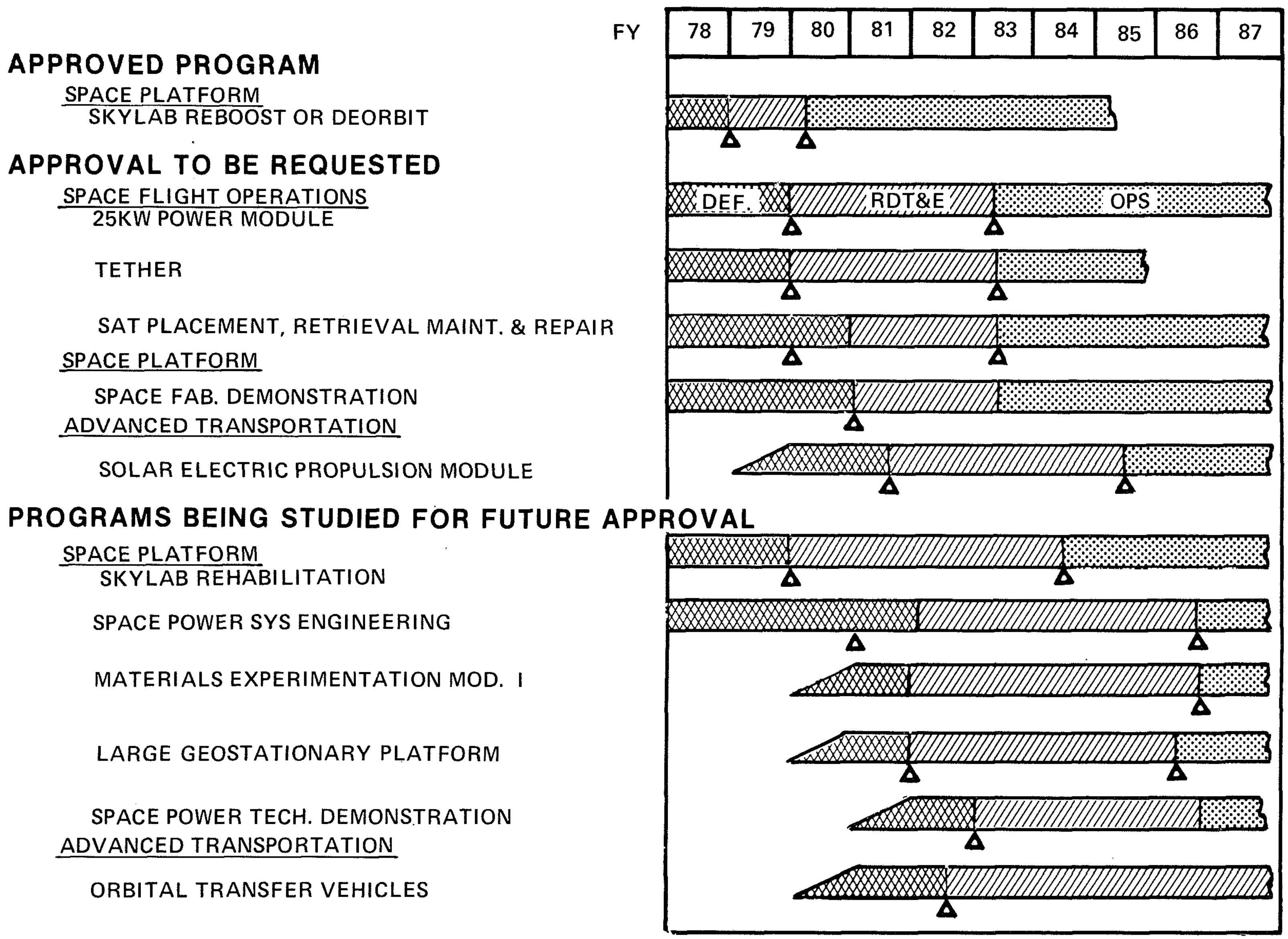


Figure 4.

NASA HQ MT78-1833 (1)
REV. 5-24-78

Skylab Boost Mission Configuration

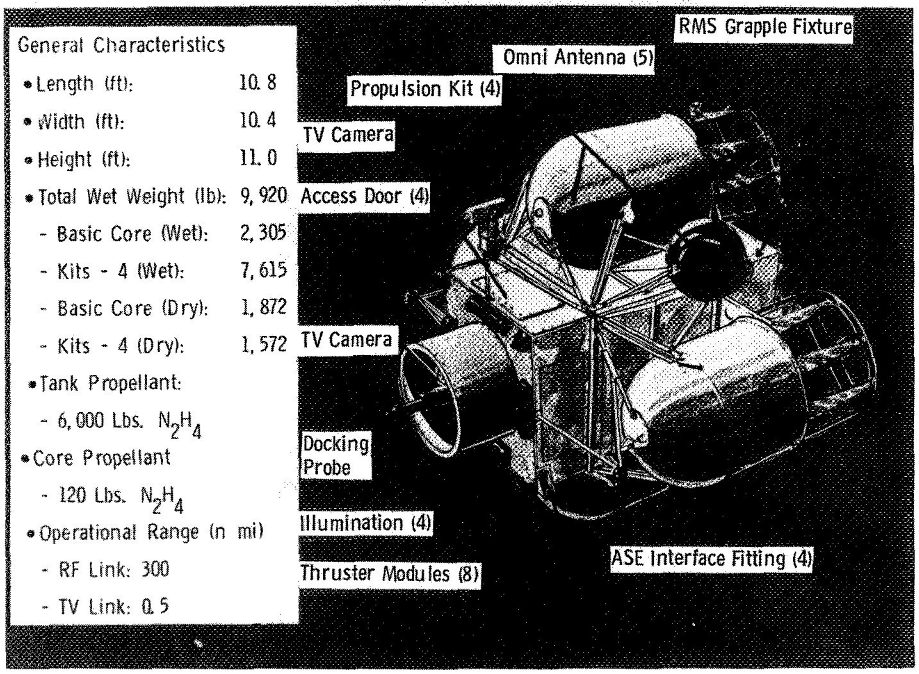


Figure 5.

USER REQUIREMENTS

25 KW POWER MODULE

	POWER/ENERGY REQUIREMENTS	DURATION REQUIREMENT
o SPACE PROCESSING	6-7 KW 12-20 KW	7-10 DAYS* 30 DAYS*
o EARTH OBSERVATIONS/ COMMUNICATIONS	7-15 KW	30 DAYS MINIMUM
o LIFE SCIENCES	NEED 870 WATTS MORE THAN ORBITER CAPABILITY	30 DAYS
o SPACE PHYSICS AND ASTRONOMY	SOLAR TERRESTRIAL OBSERVATORY, PHYSICS AND ASTRONOMY MISSIONS 7-15 KW	30 DAYS MINIMUM*
o TECHNOLOGY	SEVERAL POTENTIAL PAYLOADS W/LARGE ENERGY REQUIREMENTS	15-30 DAYS
o GENERAL	POWER SOURCE FOR SKYLAB REHABILITATION USES	15-30 DAYS

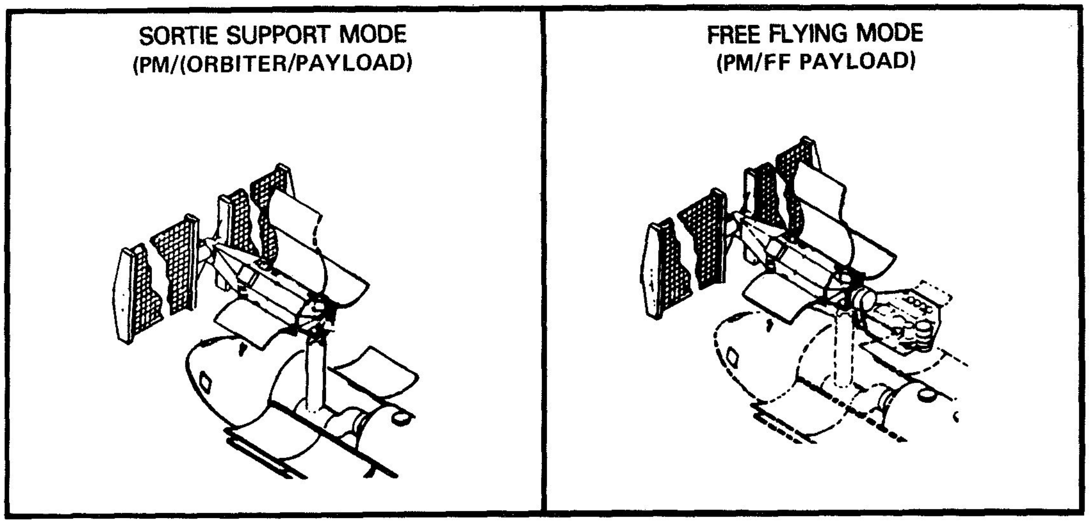
* FREE FLYER CAPABILITY HIGHLY DESIRED

Figure 6.

OSTS FIVE YEAR PLAN

MSFC - 25 KW POWER MODULE

25 KW POWER MODULE PRIMARY OPERATIONAL MODES



CURRENT STATUS

- IN-HOUSE EFFORTS DEFINING CONCEPT SYSTEMS TRADES BASED ON PRELIMINARY CONCEPT
- SCIENCE AND APPLICATIONS USER REQUIREMENTS BEING DEFINED
- CONTRACTED 25 KW POWER MODULE EVOLUTION STUDY, UNDERWAY
- ESA-SPACELAB INFORMATION EXCHANGED

FEATURES

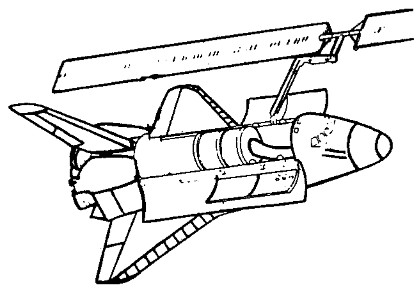
- ORBITALLY STORED
- INITIAL USE CONFIGURATION ACCOMMODATES FREE-FLYERS
- ORBITER EXTENSION MODS UTILIZED
- PROVIDES ACTIVE ATTITUDE CONTROL
- PROVIDES SUPPLEMENTARY COOLING CAPABILITY

Figure 7.

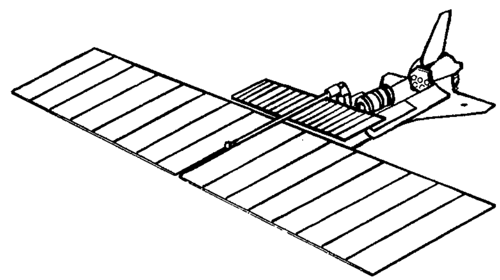
NASA HQ MT78-1838 (1)
4-27-78

OSTS FIVE YEAR PLAN
JSC - ORBITAL SERVICE MODULE

INITIAL CAPABILITY
RMS DEPLOYED ARRAY



FREE FLYING MODE
(OSM/FF PAYLOAD)



FEATURES

- PHASED INCREMENTAL BUILD-UP
- INITIAL CAPABILITY NOT ORBITALLY STORED
- EARLY AVAILABILITY
- ORBITER EXTENSION MODS UTILIZED
- FOLLOW-ON CAPABILITY PROVIDES ATTITUDE STABILIZATION
- POWER AND HEAT REJECTION BALANCED

CURRENT STATUS

IN HOUSE EFFORTS FOCUSED ON USER REQUIREMENTS AND PRELIMINARY SYSTEMS ANALYSIS AND DEFINITION

CONTRACTED STUDY FOR "CLEAN SHEET" GROWTH DEFINITION

LIMITED SYSTEM AND SUBSYSTEM STUDIES

NASA HQ MT78-1834 (1)
4-27-78

Figure 8.

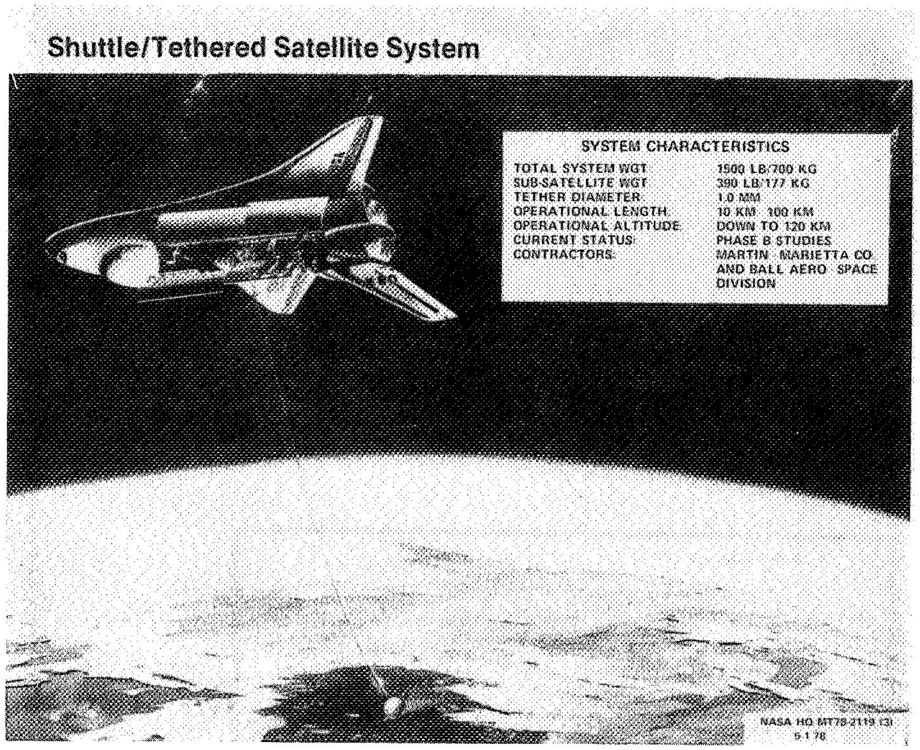


Figure 9.

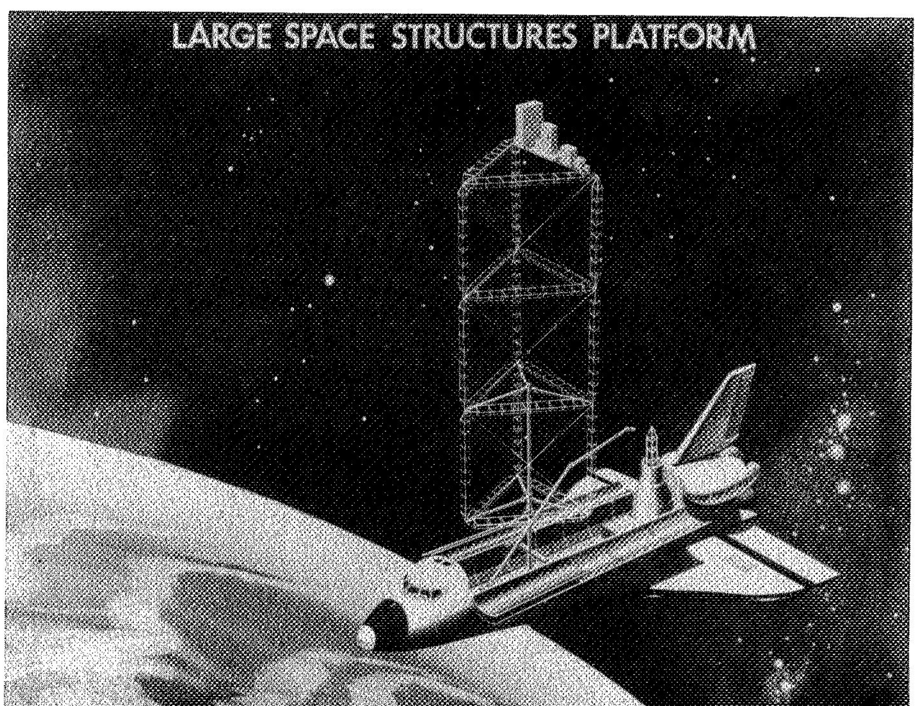


Figure 10.

SATELLITE PLACEMENT, RETRIEVAL, MAINTENANCE, AND REPAIR

OBJECTIVE

- o DETERMINE USAGE OF PLANNED CAPABILITIES
- o DEFINE SYSTEMS REQUIRED IN 1983-1985 TIME PERIOD INCLUDING OPERATIONS 800 -- 1600 KILOMETERS FROM ORBITER

CAPABILITIES TO BE EXPLOITED

- o PLANNED CAPABILITIES
  - o ORBITER
  - o REMOTE MANIPULATOR SYSTEM
  - o EXTRAVEHICULAR MOBILITY UNIT
  - o TELEOPERATOR RETRIEVAL SYSTEM
  - o MANNED MANEUVERING UNIT
- o NEW CAPABILITIES

	MISSION CASES TO BE STUDIED	
	LOW EARTH ORBIT	GEOSYNCHRONOUS EARTH ORBIT
PLACEMENT	X	
RETRIEVAL	X	
MAINTENANCE AND REPAIR	X	X

Figure 11.

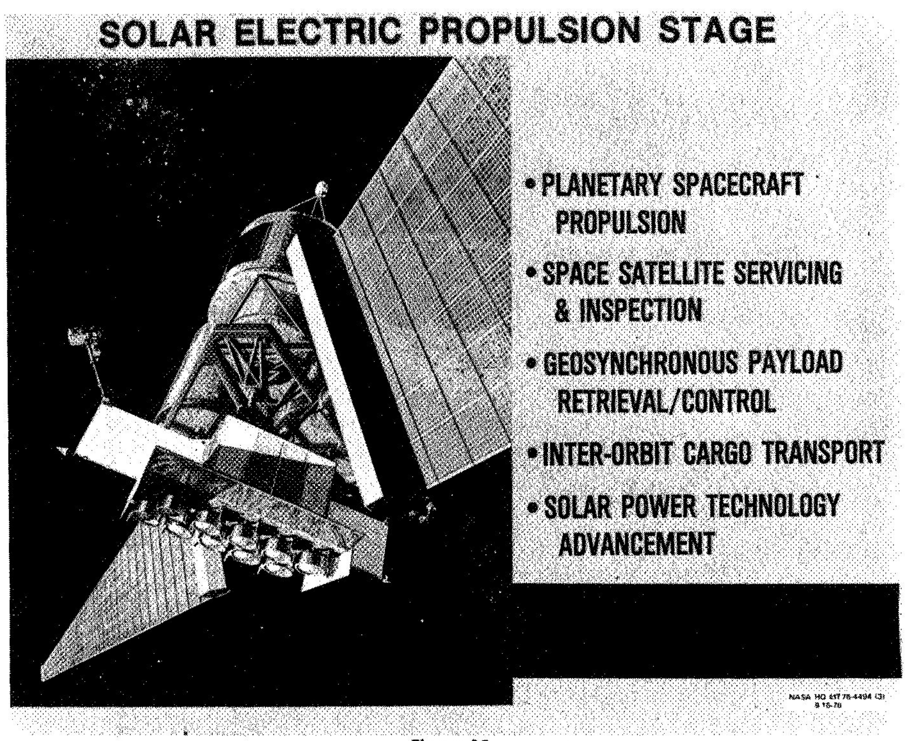


Figure 12.

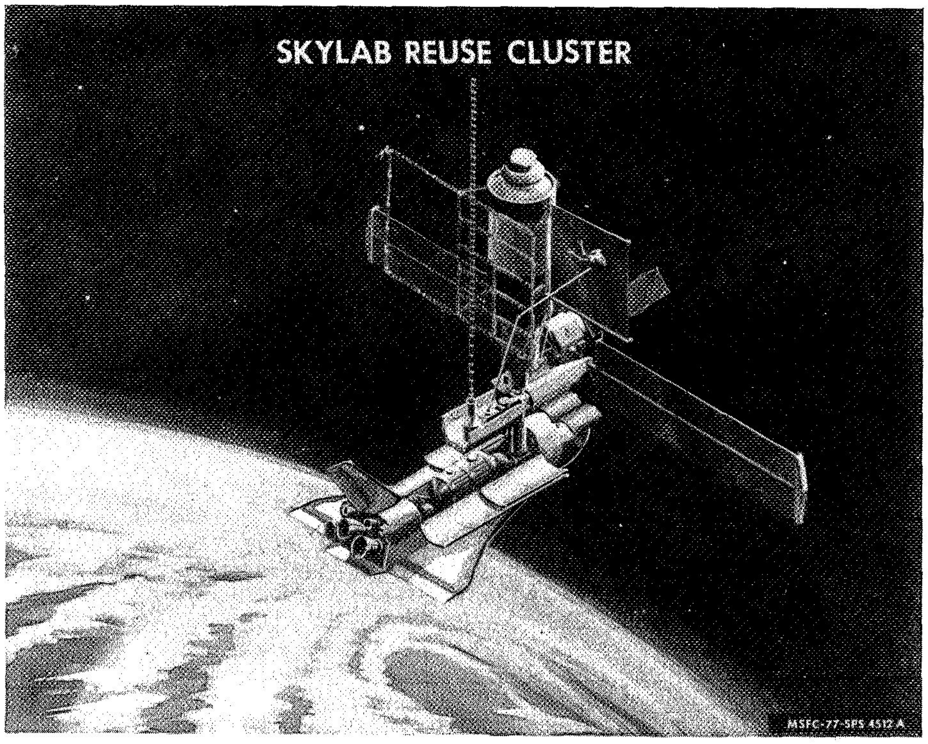


Figure 13.

MULTI-HUNDRED KW POWER REQUIREMENTS

REQUIREMENTS FOR 100's OF KW's IN ORBIT LIKELY FOR NEW SPACE OPPORTUNITIES:

- o CONSTRUCTION AND USE OF LARGE STRUCTURES SYSTEMS
- o MATERIALS/INDUSTRIAL PROCESSES
- o NEW CAPABILITIES IN COMMUNICATIONS
- o SPACE PLATFORMS FOR SCIENCE AND APPLICATIONS
- o TECHNOLOGY DEVELOPMENT FOR SOLAR POWER SATELLITES
- o ADVANCED SPACE PROPULSION

Figure 14A.

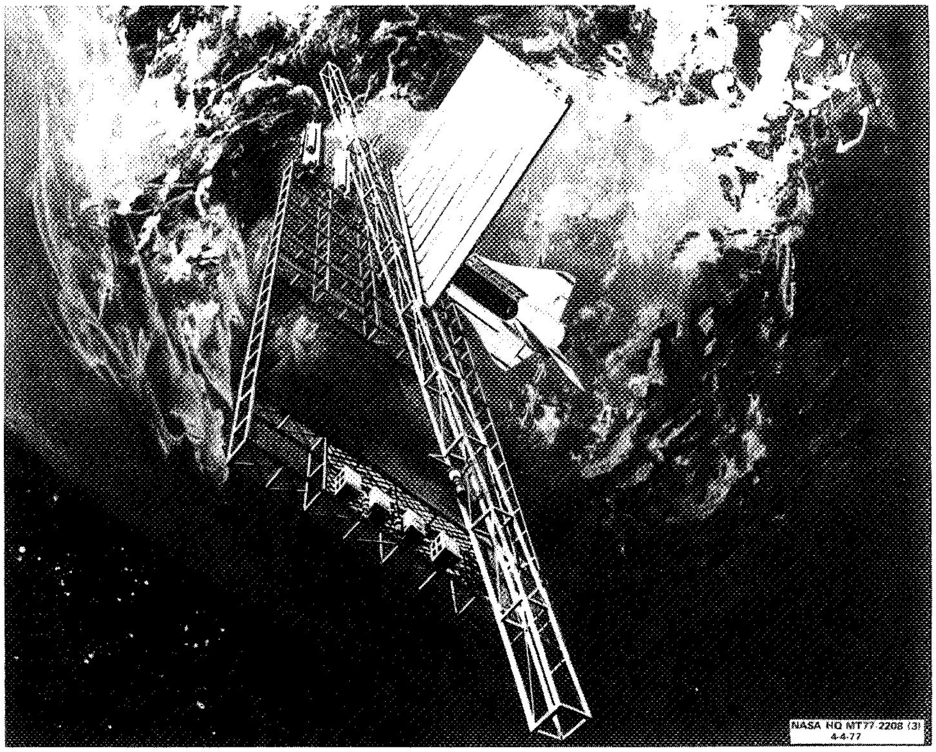


Figure 14B.

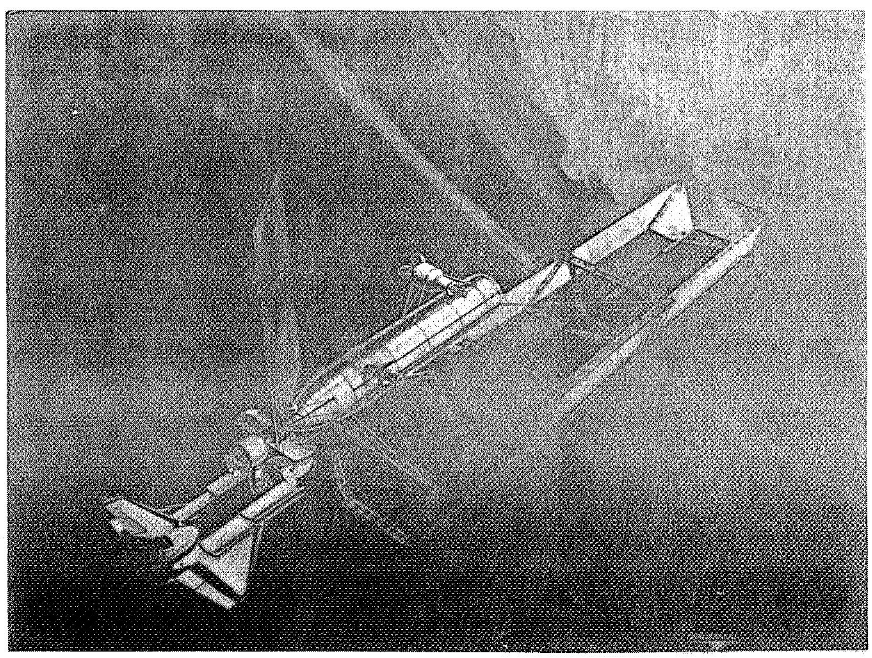


Figure 14C.

NASA MISSIONS INVOLVING
LARGE SPACE SYSTEMS

	80 — 84	84 — 86	86 — 90	90 — 2000
SCIENCE	SMALL PLATFORM (MID LATITUDE) 25KW	25KW MEDIUM PLATFORM (MID LATITUDE); SMALL PLATFORM (POLAR)	SMALL GEO PLATFORM 25KW; 25-250KW SPACE SCIENCE LABORATORY; VLBI; GRAVITY WAVE/ PINHOLE SATELLITE	SETI; RECEIVING LABORATORY 25KW 250KW
GLOBAL INFORMATION	SMALL PLATFORM 25KW		GLOBAL SERVICES→ PLATFORMS; CRYO TELESCOPE	
COMMUNICATIONS	NARROWBAND TECHNOLOGY ANTENNA		50-100KW NARROWBAND COMMUNICATIONS SATELLITE	ODSRS
SPACE PROCESSING	25KW	UNMANNED MEM 100KW	25KW GEOSTATIONARY ←PLATFORM	MANNED MEM 500KW
ENERGY		100,S KW LARGE POWER MODULE	SPS TEST ARTICLE 500W-2MW	SPS 5-10GW
ENGINEERING SUPPORT	SPACE FAB. & ERECTION TEST FLIGHT 25KW	SPACE CONSTRUCTION PLATFORM 250KW		

Figure 14D.

PLANNING FOR MULTI-HUNDRED KW POWER MODULE

CURRENT ACTIVITIES: TWO SYSTEMS CONCEPTS STUDIES UNDER WAY
- JSC/MDAC - "CLEAN SHEET" MODULAR APPROACH
- MSFC/LMSC - EVOLUTION FROM 25 KW POWER MODULE

OUTPUT: USER REQUIREMENTS AND SYSTEMS CAPABILITIES
SYSTEMS CONCEPTS, SCHEDULES, COST
TECHNOLOGY DRIVERS

FY 79 AND 80: CONTINUE CONCEPTS STUDIES AND DEVELOPMENT OF USER REQUIREMENTS

FY 81: INITIATE PRELIMINARY DESIGN COMPETITION (PHASE B - 2 CONTRACTORS)

FY 82: COMPLETE PRELIMINARY DESIGN AND SELECT DEVELOPMENT CONTRACTOR

FY 84: SYSTEM DESIGN FREEZE

FY 86 - 87: FIRST FLIGHT OF MULTI-HUNDRED KW PLATFORM

Figure 14E.

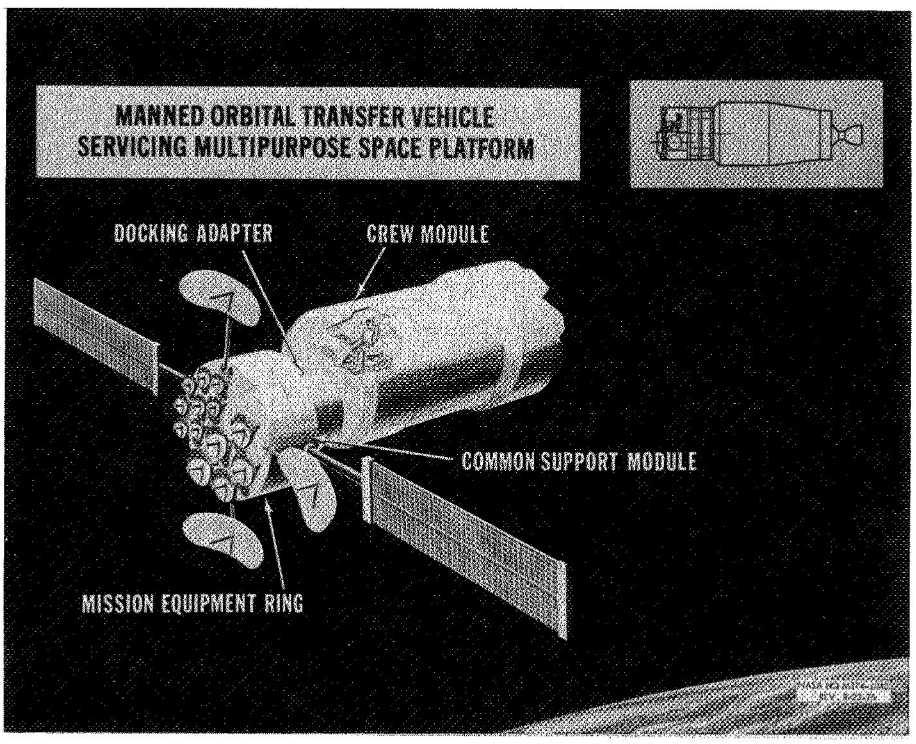


Figure 15.

MATERIALS EXPERIMENTATION MODULE

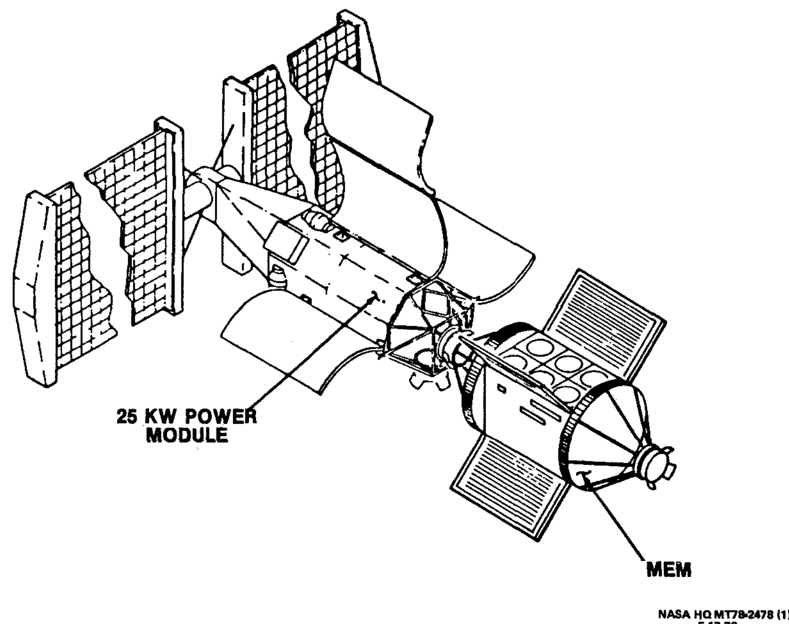


Figure 16.

MILITARY NEEDS FOR ORBITAL POWER

L. D. Massie, R. R. Barthelemy and E. T. Mahefkey
Air Force Aero Propulsion Laboratory
Wright-Patterson AFB

SUMMARY

Results of the DOD/ERDA (now Department of Energy) Space Power Study completed in October 1977 show a trend toward higher power levels for future DOD space missions. Consequently, the major new thrusts of Air Force Advanced Technology Plans center on the development of military solar power systems which will extend capabilities to the 10-50 KW_e power range for new classes of missions while maintaining technology applicability to the .5 to 10 KW_e present mission class. Plans call for technology level, subsystems level, and integrated power system level efforts. These efforts will emphasize performance, reliability, autonomy and survivability.

INTRODUCTION

The objective of the Air Force Space Power Advanced Development Program (Project 682J) is to develop and demonstrate space power system technology to meet increasing power needs of Air Force Satellites in the post-1978 time period.

Past 682J efforts which have successfully transitioned to operational applications include the Flexible Rolled-Up Solar Array (FRUSA), Hardened Array Solar Power System (HASPS), and the 2 KW_e Long Life Battery (LLB). The 1.5 KW_e FRUSA was successfully flown on Space Test Program Vehicle 71-2. The HASPS has been selected for meeting the 7.3 KW_e prime power requirement of the SIRE (Space Infrared Experiment) P80-2 mission. The 2 KW_e LLB effort provided the baseline technology from which the 14.3 watt• hrs/Kg (6.5 watt• hrs/Lb) FLTSATCOM battery evolved.

The present FY78 efforts in Project 682J include the multiple phase Task 682J04 High Efficiency Solar Panel (HESP), Task 682J05 Nickel-Hydrogen Battery (Ni/H2), a recently completed Task 682J06 Gallium Arsenide Solar Concentrator Hardness Study (Concentrator) and a February 1978 Task 682J07 new-start Nuclear Dynamic Power System Applications/Integration Study.

In view of findings and conclusions of the DOD/ERDA Space Power Study Report, the Technology Program Plan for Space Power Advanced Development was recently modified to address the projected trend toward higher power levels and an enhanced threat environment for military satellites. The new planning initiatives include a Task 682J08 High Voltage High Power (HVHP)

System; Task 682J09 High Energy Density Rechargeable Battery (HEDRB); Task 682J10 Fault Tolerant Power System (FTPS); Task 682J11 Thermal Energy Storage Subsystem (TESS) and Task 682J12 Cascaded Solar Cell Development Program.

STATUS OF PRESENT EFFORTS

Task 682J04.- HESP Phase I has been completed with the demonstration of silicon solar cells having efficiencies of 14 percent. HESP Phase II has been initiated with the objective of demonstrating 16 percent efficient silicon cells and improved experimental quantities of 16 percent efficient gallium arsenide cells. Under HESP Phase II, recent silicon and radiation-hardened gallium arsenide developmental cells delivered to AFAPL have demonstrated efficiencies as high as 15.5 and 17 percent respectively. Radiation resistance of gallium arsenide cells has been improved to the point where some cells are superior to high output silicon cells at 1-Mev electron fluence levels as high as 5×10^{15} e/cm^2 as shown in Figure 1. As shown in Figure 2, temperature coefficients of gallium arsenide cells are clearly superior to those of silicon, making gallium arsenide cells attractive for Concentrating Photovoltaic Power System Concepts. Gallium arsenide cells have favorable values of solar absorptance ($\alpha = .78$) as compared to silicon ($\alpha = .85$, smooth surface cells; $\alpha = .94$, textured surface cells). Both HESP textured silicon cells and gallium arsenide cells being flown as experiments on the NTS-II (Navigation Technology Satellite) are performing well after more than 223 days in orbit. Both cell types are to be included in the forthcoming DIABLO HAWK underground nuclear test.

Task 682J05.- In the area of rechargeable batteries, Ni/H_2 cell design has been completed and the initial group of twenty-six (26) test cells are being manufactured for evaluation. In addition NASA Marshall Space Flight Center has provided funds for two Ni/H_2 cells for independent test and evaluation. Twenty-four (24) cells will be built in the spring of 1978 for an Industry Dispersal Program under which independent industry tests will be conducted. The Naval Research Laboratory has expressed interest in utilizing the Air Force developed Ni/H_2 cells on NTS-III scheduled for launch in 1981. Negotiation of a Memorandum of Agreement between the Air Force and Navy pertaining to the NTS-III cells is anticipated during the second quarter of 1978. It is presently estimated that a total of forty-eight, 35-ampere-hour cells would be required for meeting NTS-III requirements. Sixteen of the forty-eight cells would be assembled into a high performance battery, integrated, and utilized as the NTS-III energy storage subsystem. A successful orbital test of early vintage Ni/H_2 cells has already been conducted on a SAMSO Special Projects Vehicle. The advanced development program Ni/H_2 cells currently being fabricated are expected to be vastly superior to these early vintage cells already flown in space in terms of cycle life, depth of discharge and energy density. Figure 3 is a schematic of the 50-ampere-hr Ni/H_2 cell design. Figure 4 is a photograph of the 21-cell Ni/H_2 battery flown on the SAMSO/SP Vehicle. Table I is a preliminary weight breakdown for a 1.15-KW-Hr Ni/H_2 Battery based upon a Hughes Aircraft Company conceptual design.

Task 682J06.- A gallium arsenide cell concentrating photovoltaic concept recently studied under a Contract with Rockwell International is considered to be a promising technology option for advanced laser threat hardness. The concept utilizes Cassegrainian optics in conjunction with a Winston collector to focus sunlight on a single high efficiency gallium arsenide cell at a concentration ratio of about 500 to 1. Cell operating temperature is maintained at approximately 120°C by an integral, distributed heat pipe radiator. The final study report (AFAPL-TR-78-30) pertaining to this concept will be distributed in June 1978.

Task 682J07.- The recently initiated Nuclear Dynamic Power System Applications/Integration Study will provide the analysis, design, and spaceflight integration considerations needed to assure a successful space demonstration of a 1.3-KW_e radioisotope-fueled dynamic power system. An important part of this study will deal with analyzing future special purpose Air Force missions which will benefit from this technology. In addition, the program will provide for (a) analysis of requirements for integration and orbital operations, (b) evaluation of nuclear and laser hardness, and (c) develop preparatory information needed for the assessment of safety and environmental impacts. The overall program is to assure Air Force applicability of the DOE Nuclear Dynamic Power System Technology.

FUTURE PLANS - APPROVED PROGRAM

Future space power advanced development plans, within the approved program, are primarily extensions of present efforts and include the following: (a) Gallium Arsenide Solar Panel work which is directed toward panel design, fabrication and spaceflight qualification; (b) HESP Phases III and IV which are for advanced cell production demonstration, flight experiment design, experiment spaceflight qualification and orbital flight test; (c) completion of single-cell Ni/H_2 efforts through orbital flight test and implementation of a Common Pressure Vessel Ni/H_2 program; (d) implementation of a Concentrating Photovoltaic Power System hardware build and evaluation effort; and (e) completion of the Nuclear Power Supply Study in support of space payload AFAPL 601 - Nuclear Dynamic Power System Flight Experiment.

Figure 5 is a Milestone Chart which encompasses present efforts and future plans within the approved program. The chart shows anticipated technology advances and when they are expected to occur, based upon present budget allocations. For example, 16 percent efficient space-qualified solar cell assemblies are expected in 1980. Demonstration of advanced solar cells through a 20,000-cell flight test is expected by 1983. Conclusive demonstration of single-cell Ni/H_2 batteries through flight test on NTS-3 should occur in 1981. Flight test of a Nuclear Dynamic Power System is scheduled for 1983.

Overall results of these efforts, compared to conventional technology, will double the end-of-life power per unit area of solar arrays, more than double the useable energy density of spacecraft energy storage subsystems, and make new power technology options such as Concentrating Photovoltaic and Nuclear Dynamic Power Systems available for special purpose DOD satellite applications.

RESULTS OF DOD/ERDA SPACE POWER STUDY

A DOD/ERDA Space Power Study was conducted during the period from February 1976 to May 1977. A preliminary report summarizing the results of this study was issued in October 1977. The objectives of the study were to identify future DOD space power requirements and recommend appropriate nuclear and non-nuclear technology development programs needed to ensure that future power requirements can be met. Study participants included SAMSO, AFAPL, ERDA (DOE), LASL, and industry. Study tasks included (a) future requirements through the year 2000, (b) nuclear and non-nuclear technology projections through the year 2000, (c) matching power systems and requirements, (d) spacecraft point designs for one navigation satellite and two surveillance satellites, and (e) recommendations.

Results of the study indicate that the majority of future single spacecraft power requirements will be in the .5 to 10 KW_e power range. However, the study also identified a significant number of potential missions with power requirements in the 10 to 100 KW_e range and beyond as shown in Figure 6. Most of the high power requirements tend to be in the surveillance, space defense systems, ECM resistant communications, and offensive systems areas. Requirements exist for electrical, thermal, and pulse power with some potential missions requiring a combination of all three power forms.

Table II presents the general findings of the study dealing with matching of power systems to mission power requirements. The approved model and extended mission model scenarios cover the .5 to 100 KW_e and beyond power regime. Solar array/battery power systems are the number one choice in the .5 to 5 KW_e range. Within this range there are special purpose isotope applications for missions where precise attitude control and stabilization, extreme hardness, and maneuvering capabilities are needed. In the 5 to 25 KW_e range, solar array/battery power systems are the first choice. There are no isotope applications in this range because of the high cost of the radioisotope fuel. Either solar array/battery power systems or a reactor, if it were available, could fulfill needs in the 25 to 50 KW_e range. A reactor would be the number one choice for power requirements beyond 50 KW_e, with solar array/battery systems being feasible in this range.

FUTURE PLANS-LABORATORY RECOMMENDED PROGRAM

Future potential mission requirements as identified in the DOD/ERDA Space Power Study have been factored into future technology program planning for space power advanced development. The Technology Program Plan was recently revised to reflect need for the following new initiatives:

Task 682J08 - High Voltage High Power System
Task 682J09 - High Energy Density Rechargeable Battery
Task 682J10 - Fault Tolerant Power System
Task 682J11 - Thermal Energy Storage Subsystem
Task 682J12 - Cascaded Solar Cells

Task 682J08 is to develop a strong technology base for a DOD Power Module (10 to 50 KW_e) with a counter-measures capability; Task 682J09 will develop a rechargeable electrochemical battery capable of 66 watt- hrs/Kg (30 watt - hrs/Lb) for low earth orbit (LEO) and 110 watt -hrs/Kg (50 watt - hrs/Lb) for geostationary (GEO) orbit applications; Task 682J10 will demonstrate an autonomous power system where all elements of the system are controlled by local microprocessors in conjunction with a power system microcomputer; Task 682J11 will develop a 110 watt -hr/Kg (50 watt -hr/Lb) thermal energy storage subsystem for Vuilleumier (VM) cryocoolers for surveillance applications; Task 682J12 will develop 25-35 percent efficient monolithic cascaded multiple bandgap solar cells.

Anticipated results of the AFAPL Research and Exploratory Development Programs during the next 2 to 5 years were considered in developing the revised Technology Program Plan for Advanced Development.

Figure 7 is a Milestone chart which pertains to the laboratory-recommended new initiative program plans. The chart shows the expected technology advances and when they are expected to occur if the programs are approved and budgeted. Overall results of these efforts, compared to conventional technology, would triple the end-of-life power per unit area of solar arrays; more than quadruple the useable energy density of spacecraft energy storage subsystems; and provide non-nuclear, autonomous, survivable power system options to fulfill potential high power advanced mission needs. Figure 8 shows the overall Air Force past, present, and future advanced development program response to Space Power Technology. Implementation of the various tasks under the project results in technology options which are mission enhancing and mission enabling. Examples of mission enhancing tasks are the Ni/H_2 Battery and solar cell efficiency improvements which transition directly to system application upon qualification and production demonstration. The Hardened Array Solar Power System (HASPS) is an example of a mission enabling task in that the feasibility of the SIRE P80-2 mission would be questionable without the HASPS technology option. Also, the recommended HVHP task is mission enabling; that is, unless the technology is developed, certain future high power missions cannot be undertaken.

CONCLUDING REMARKS

a. The trend toward military space power requirements in the 10 - 100 KW_e range is valid, based upon the probable needs for advanced surveillance, ECM resistant communications, space-based radar, and space defense missions.

b. Performance enhancements in solar cell efficiency and battery energy density and lifetime are of major importance to spacecraft designers.

c. Design to performance, survivability and reliability/autonomy mandates are important military satellite power system requirements.

d. Advanced solar arrays and batteries will continue to be the predominant power system choice for future Air Force satellites in the foreseeable future. Nuclear power system options should be maintained for specialized missions requiring very high levels of hardness and orbit predictability.

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1. AFAPL-TR-77-36 - "High Efficiency Solar Panel (HESP)"; Spectrolab, Inc., July 1977.

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3. AFAPL-TR-78-30 - "Gallium Arsenide Solar Concentrator Hardness Study"; Rockwell International Corporation, March 1978.

4. AFAPL-TR-76-100 - "Investigation of GaAs Solar Cell Potential Performance and Cost"; AF Aero Propulsion Laboratory, February 1977.

5. AFAPL-TR-77-90 - "Failure Mechanisms in Nickel-Hydrogen Cells"; Hughes Aircraft Company, December 1977.

6. AFAPL-TR-77-89 - "Nickel-Hydrogen Battery Advanced Development Program"; Hughes Aircraft Company, December 1977.

7. Personal Communications with Lt Cecil R. Stuerke, AFAPL/POE-2, WPAFB, OH 45433. Air Force focal point for GaAs Solar Cell advanced development.

8. Personal Communications with Mr. Don R. Warnock, AFAPL/POE-1, WPAFB, OH 45433. Air Force focal point for Nickel-Hydrogen Cell advanced development.

TABLE I - 50 AMPERE HOUR Ni/H_2 BATTERY WEIGHT ESTIMATE

ITEM	WEIGHT (KG)	WEIGHT (LBS)
24 CELLS @ 1.29	30.96	68.25
24 THERMAL SHUNTS @ .027	.65	1.43
48 HEAT PIPES @ .052	2.50	5.51
24 RADIATORS @ .086	2.06	4.54
BATTERY CELL HDW.	1.27	2.80
ELECTRICAL & PROTECTIVE	5.45	12.01
	42.89	94.55

$$\text{USEABLE ENERGY DENSITY} = \frac{C\ V_C^{*}\ N_C\ DOD^{**}}{W} = 26.8\ \frac{\text{WATT} \cdot \text{HRS}}{\text{KG}}\ \left(12.2\ \frac{\text{WATT} \cdot \text{HRS}}{\text{LB}}\right)$$

* NOMINAL CELL VOLTAGE = 1.2 VOLTS

** DEPTH OF DISCHARGE = 80%

TABLE II - MATCHING POWER SYSTEMS TO MISSION REQUIREMENTS

SCENARIO: .5 - 100 KW_E & BEYOND POWER RANGE

- 0 SOLAR # 1 IN THE .5 - 5 KW_E RANGE; SPECIAL PURPOSE ISOTOPE APPLICATIONS IN THIS RANGE
- 0 SOLAR # 1 IN THE 5 - 25 KW_E RANGE; NO ISOTOPE APPLICATIONS IN THIS RANGE
- 0 25 - 50 KW_E - EITHER SOLAR OR REACTOR (IF AVAILABLE) IN THIS RANGE
- 0 IF AVAILABLE, REACTOR # 1 IN THE 50 - 100 KW_E RANGE (AND BEYOND); SOLAR FEASIBLE

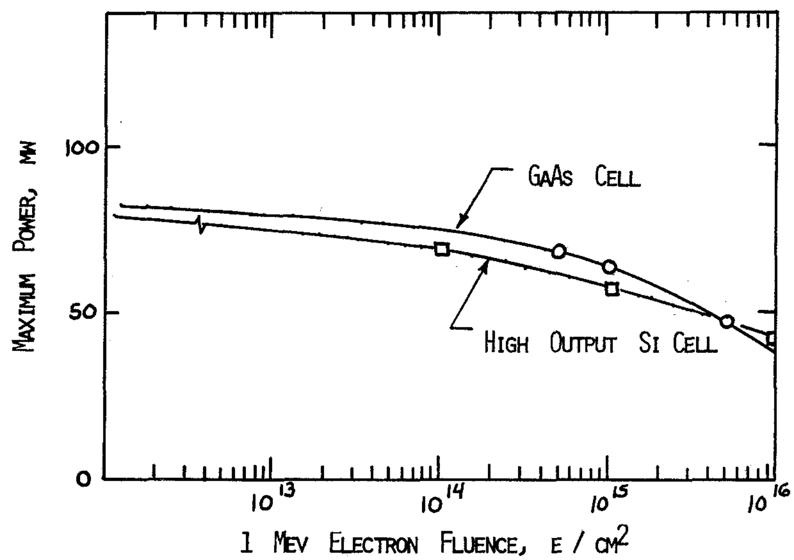

FIGURE 1. IMPROVED RADIATION TOLERANCE OF RECENT GaAs SOLAR CELLS

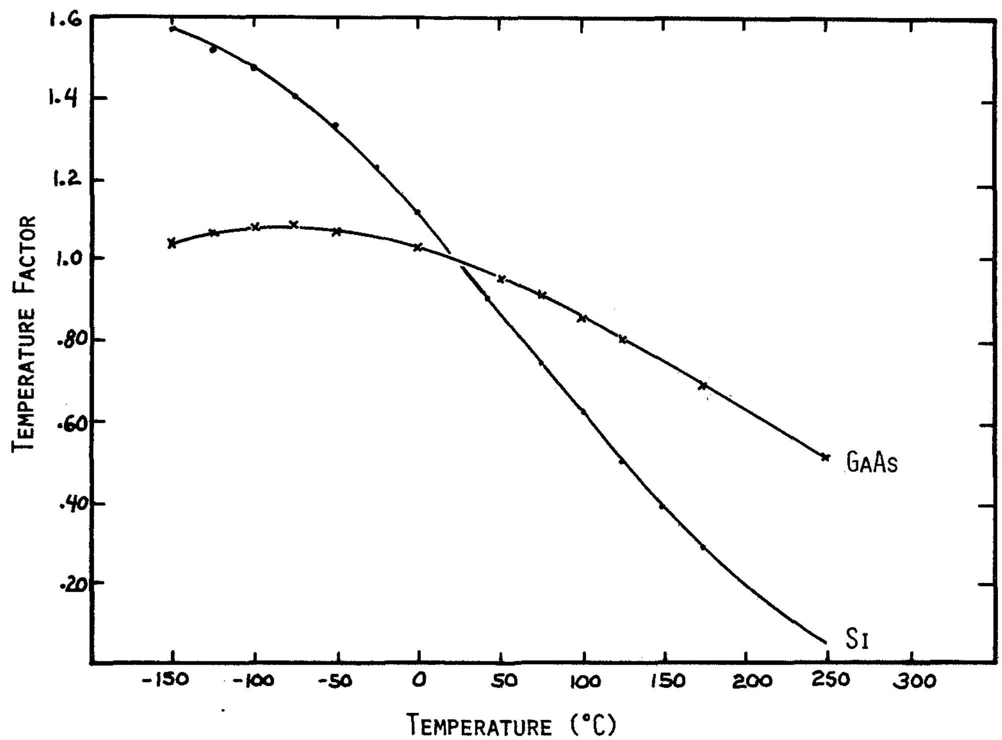

FIGURE 2. TEMPERATURE SENSITIVITY OF GaAs AND Si SOLAR CELLS

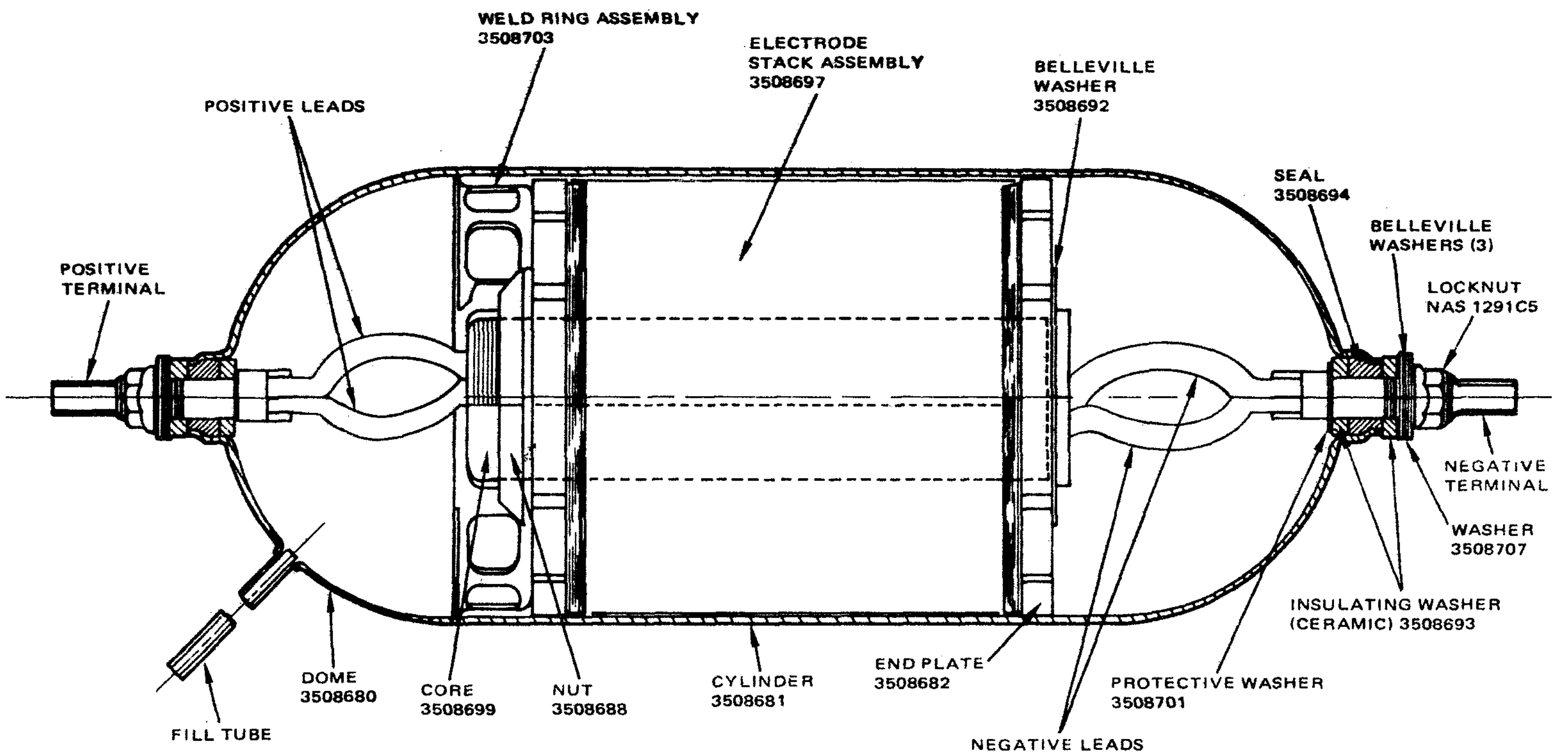

FIGURE 3. SCHEMATIC OF 50 AMPERE HOUR NICKEL-HYDROGEN CELL

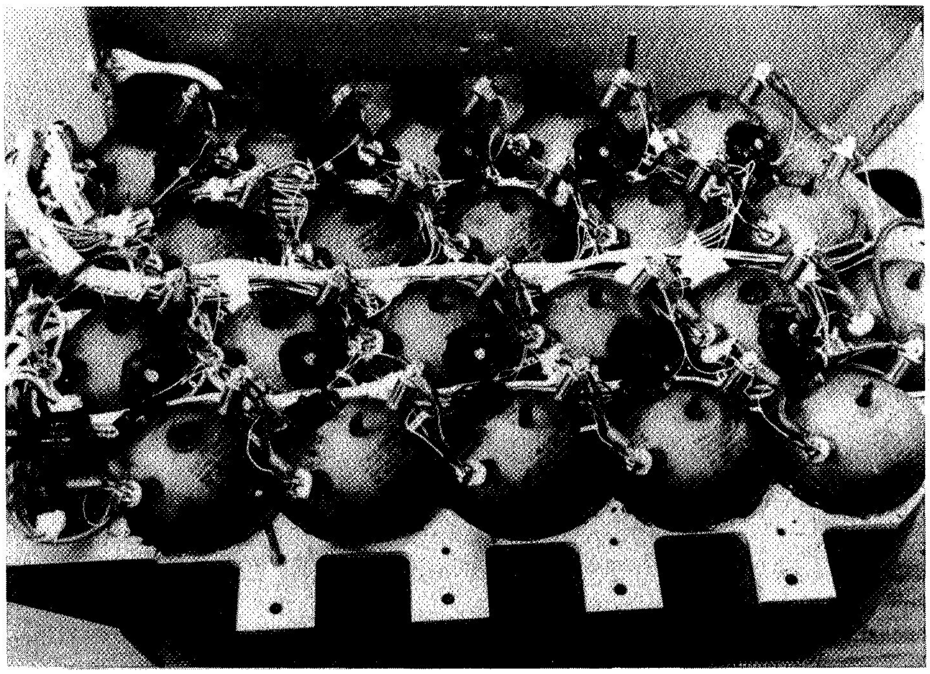

FIGURE 4. PHOTOGRAPH OF AIR FORCE NICKEL-HYDROGEN BATTERY SPACE EXPERIMENT

TASK/SHORT TITLE	FY78	FY79	FY80	FY81	FY82	FY83
682J04/HESP			1		2	3
682J05/Ni-H_2		4		5	6	7
682J06/CONCENTRATOR	8				9	10
682J07/NUCLEAR POWER SUPPLY STUDY		11		12		13

GOALS

1. 16% HARDENED SOLAR CELLS
2. 16% HARDENED SOLAR PANEL
3. FLT EXP.-AFAPL 501
4. SINGLE CELL Ni-H_2 9 WH/LB LEO 16 WH/LB GEO
5. NTS-3 FLIGHT
6. CPV Ni-H_2 12 WH/LB LEO 20 WH/LB GEO
7. Ni-H_2 LEO FLT. EXP-AFAPL 503
8. HARDNESS STUDY
9. SYSTEM DESIGN
10. FLIGHT EXP.
11. INTEGRATION STUDY
12. SAFETY
13. NUCLEAR DPS FLT. EXP.-AFAPL 601

FIGURE 5. MILESTONE CHART FOR ADVANCED SPACE POWER SUPPLY TECHNOLOGY (SAMSO/AFAPL APPROVED)

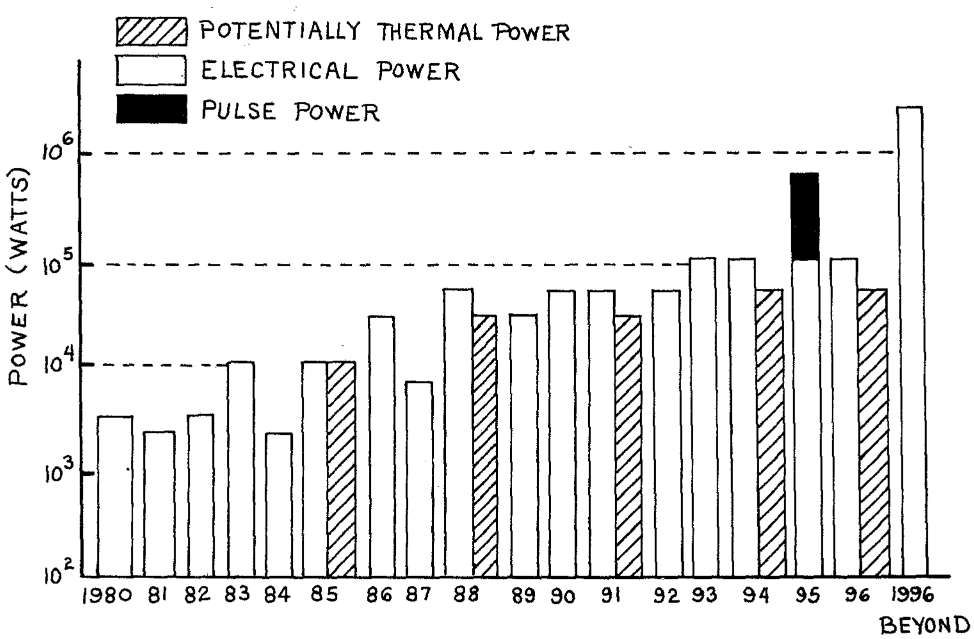

FIGURE 6. MAXIMUM SINGLE-SPACECRAFT POWER REQUIREMENTS BY YEAR

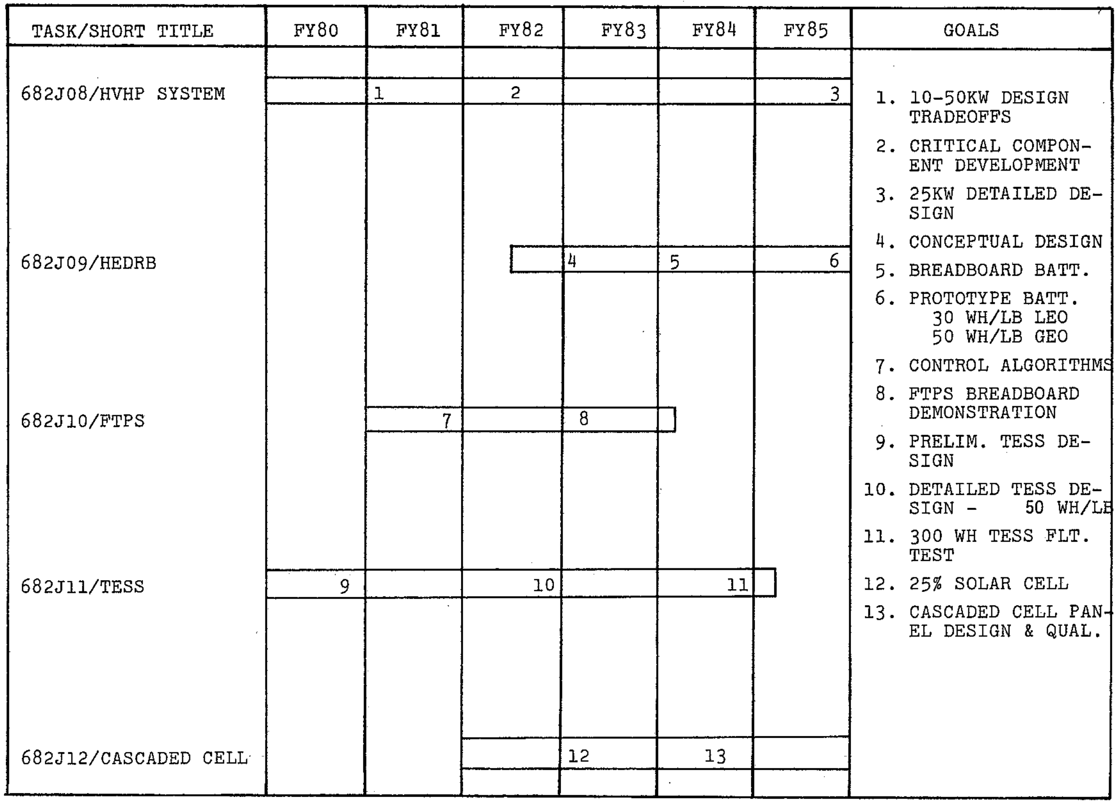

FIGURE 7. MILESTONE CHART FOR ADVANCED SPACE POWER SUPPLY TECHNOLOGY (AFAPL RECOMMENDED NEW INITIATIVES)

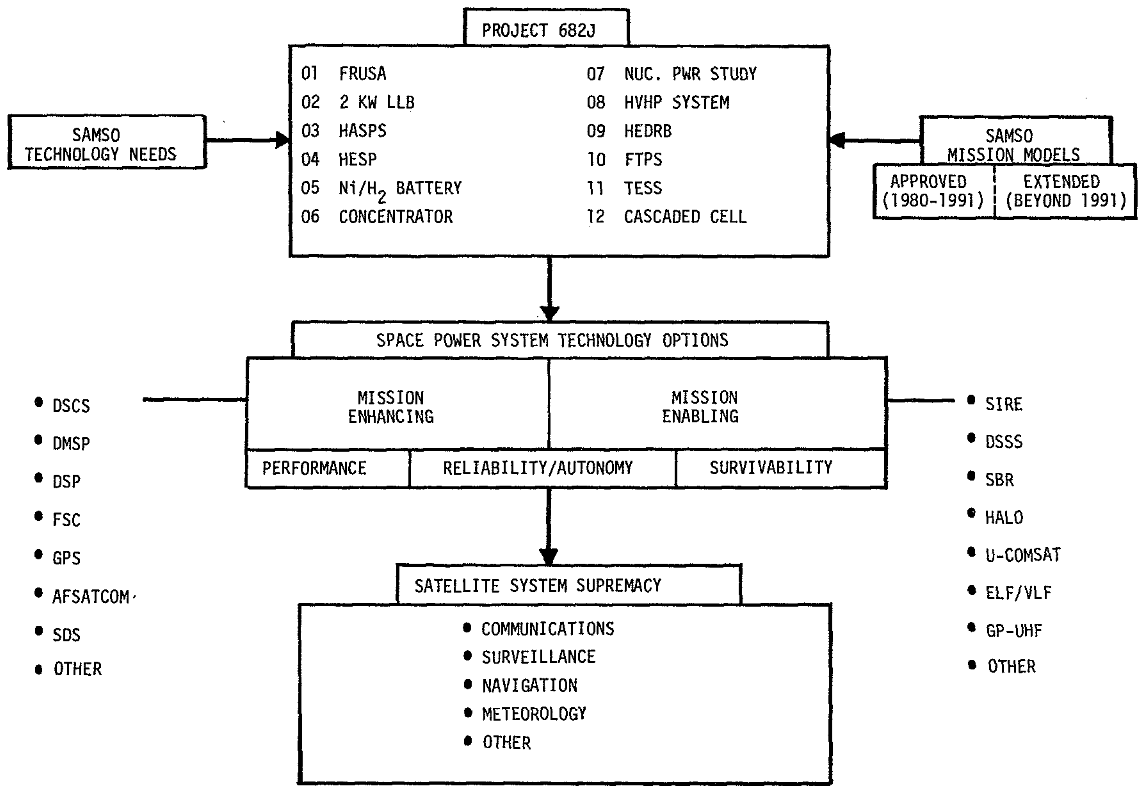

FIGURE 8. SPACE POWER ADVANCED DEVELOPMENT PROGRAM RESPONSE TO POWER REQUIREMENTS

SATELLITE POWER SYSTEMS PROGRAM

Ralph I. LaRock
NASA Headquarters

The Department of Energy and the National Aeronautics and Space Administration are engaged in an intensive three-year analysis to determine what course of action the Federal government should pursue relative to this nonconventional energy system.

Opinions concerning the potential viability of the concept now cover a wide emotional spectrum which ranges from very negative to highly enthusiastic. In reality, this diversity of opinion merely reflects the uncertainty which surrounds the technical feasibility and operational practicability of the idea. Accordingly, we are now in the process of developing an information base which will be sufficient by 1980 to support a decision on whether or not to proceed with the next phase of the program.

The current program plan which was approved by the Administration in February of this year is, with the exception of microwave effects, entirely analytical. We must, therefore, realize that this effort is very unlikely to achieve a firm recommendation to implement the concept. Rather, if no insurmountable barriers are found, recommendations directed to laboratory experimentation and field and space testing are likely to result.

The program is now organized as shown on Figure 1. DOE has overall program coordination responsibility which is assigned to an SPS project office under the Director of Energy Research. NASA is responsible for the definition of the overall systems concept and all technology which is involved. The Office of Energy Programs, NASA Headquarters, manages the effort and is supported by Marshall Space Flight Center and Johnson Space Center. The remainder of the program is managed by the DOE with the SPS project office and the Assistant Secretary for Environment sharing programmatic responsibility. Participating DOE laboratories include Battelle Pacific Northwest, Los Alamos Scientific Laboratories and the Argonne National Laboratory.

The SPS Working Group assists the DOE coordinator and is composed of senior project personnel from both Agencies. The objective is to insure that the results of work performed by the various participating organizations are integrated to achieve scheduled program milestones.

The major milestones are shown by Figure 2 and relevant activities for each fiscal year are listed in Figure 3. It should be noted that, along with the baseline concept selection milestone in Oct. 1978, initial recommendations for an experimental research plan will also be completed. We anticipate that the initial plan will be directed mainly to definition of experiments which should start in 1980 and which address highly critical program issues. In addition, an outline of other experimental research projects which can begin in subsequent years and which will be needed to achieve full technology readiness for SPS will be prepared.

Our System Definition Centers, JSC and MSFC, are now working to evolve a consolidated recommendation for a baseline SPS concept. Preliminary recommendations based upon independent assessment by each Center of various candidate SPS concepts were presented in January of this year. As was expected there were some significant differences as well as many areas of agreement. The differences are now in process of resolution by way of a MSFC/JSC working group. The essential elements of the initial recommendations made by the Center are shown by Figures 4 and 5.

It is important to note that the baseline system approach is expected to continue to change with time as we become more knowledgable of the specific problems to be resolved and as our technological capability evolves. However, it is important to establish and maintain a baseline to guide the combined efforts of the DOE and NASA as the program progresses.

Program funding by Agency management responsibility is shown by Figure 6. It is anticipated that if no absolute barriers to the concept are identified by 1980, that additional funding for further field test work could be made available by the Administration.

The technological challenge presented by the SPS is well recognized by all who are familiar with the size and complexity of the system. However, the overall system problem is only partially technical - in fact, the most difficult issues to resolve will probably lie in the environmental effects and international areas. Accordingly, it will be mandatory that NASA continue to work closely with the DOE as we join forces to assess all aspects of the problem to gain the understanding which is so vitally needed to guide our future programmatic effort.

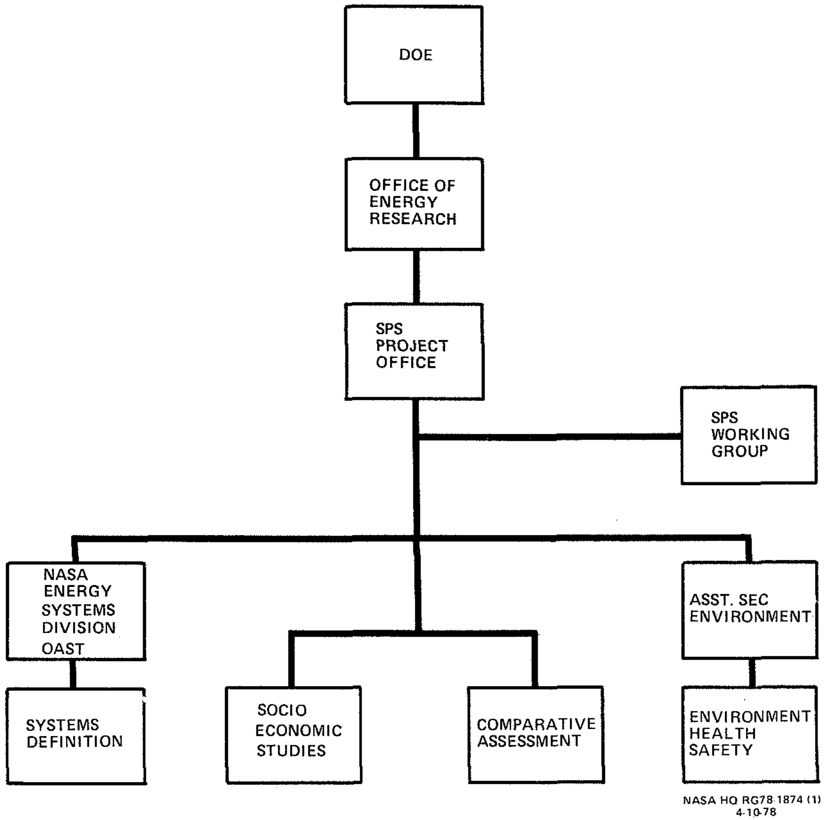

Figure 1.

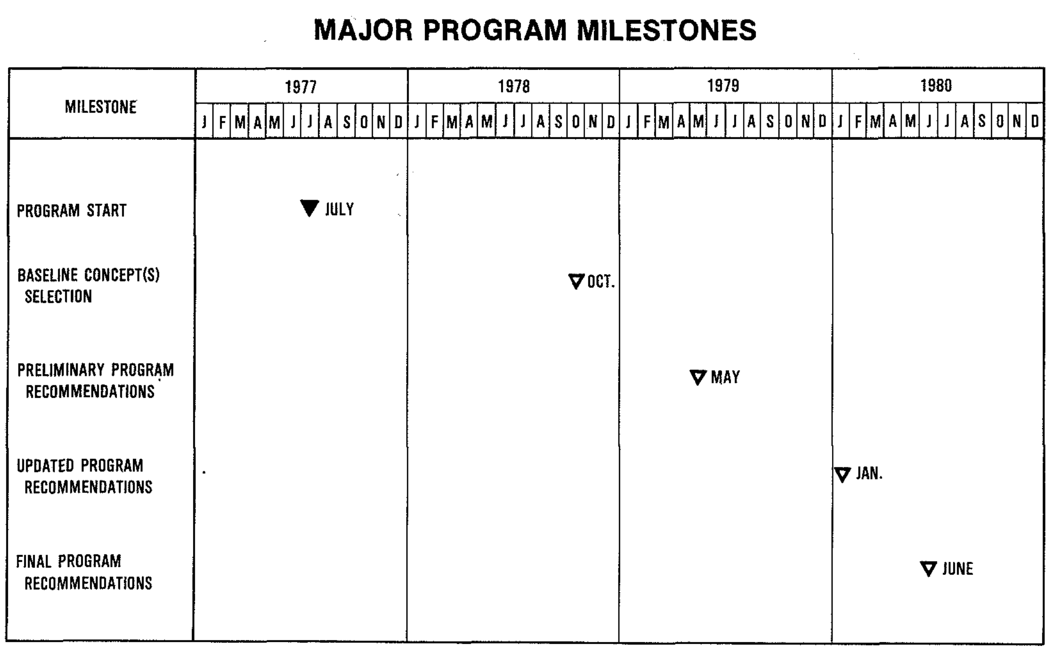

Figure 2.

NASA HQ RG78-1560 (1)
3-10-78

ACTIVITY SCHEDULE FOR SPS PROGRAM

YEAR	NASA MANAGED	DOE MANAGED
1977	CONCEPTUAL SYSTEM (S) DEFINED TECHNICAL REQUIREMENTS IDENTIFIED SUBSYSTEM TRADEOFFS COMPLETED	INITIAL GUIDELINES DEFINED INITIAL METHODOLOGY DETERMINATION
1978	BASELINE CONCEPT(S) RECOMMENDATION PRELIMINARY SYSTEMS INTEGRATION BASELINE CONCEPT(S) SELECTION * EXPERIMENTAL RESEARCH PLAN	INITIAL ENVIRON. IMPACT ASSESSMENTS INTERIM METHODOLOGY DETERMINATION PRELIMINARY ENV. IMPACT ASSESSMENT BASELINE CONCEPT(S) SELECTION * INTERIM METHODOLOGY UPDATE
1979	BASELINE SYSTEMS INTEGRATION PRELIMINARY PROGRAM RECOMMENDATIONS * FINAL SYSTEMS INTEGRATION TECH. STATUS ASSESSMENT	BASELINE ENVIRON. IMPACT ASSESSMENT PRELIMINARY PROGRAM RECOMMENDATIONS * FINAL ENVIRON. IMPACT ASSESSMENT FINAL COMPARATIVE ASSESSMENT FINAL METHODOLOGY
1980	UPDATED PROGRAM RECOMMENDATIONS * STUDY INTEGRATION TECHNOLOGY ADVANCEMENT PLAN	UPDATED PROGRAM RECOMMENDATIONS * STUDY INTEGRATION FINAL PROGRAM RECOMMENDATIONS *

(* MAJOR MILESTONES)

NASA HQ RG78-1207 (1)
1-24-78

Figure 3.

SPS PRELIMINARY BASELINE COMPARISON

	MSFC	JSC
POWER GENERATION CAPABILITY	5GW	10 GW
OVERALL DIMENSIONS (Km)	3.8x21	5.3x24
POWER CONVERSION-PHOTOVOLTAIC	GaAs (CR=2) (0.2 MILLS)	SILICON (CR =1) (2. MILLS)
STRUCTURE MATERIAL	ALUMINUM	GRAPHITE COMPOSITE
CONSTRUCTION LOCATION	GEO	LEO
TRANSPORTATION		
• EARTH-TO-LEO -CARGO (PAYLOAD)	HTO WINGED 1 or 2-STAGE (91,000 K g)	VTO WINGED 2-STAGE (424,000 K g)
-PERSONNEL (NO)	MODIFIED SHUTTLE (75)	MODIFIED SHUTTLE (75)
• LEO-TO-GEO -CARGO	DEDICATED ELECT. OTV	SELF-POWERED $^1/_8$ SPS MODULES
-PERSONNEL (NO)	2-STAGE LOX/LH_2 (75)	2-STAGE LOX/LH_2 (75)
MICROWAVE POWER TRANSMISSION		
• NO. OF ANTENNAS	1	2
• ANTENNA POINTING/CONTROL	MOTOR DRIVE	CMGs
• DC-RF CONVERTER	KLYSTRON	KLYSTRON
• FREQUENCY (GHZ)	2.45	2.45
• RECTENNA DIMENSIONS (Km)	10x13	9.4x13
• RECTENNA POWER DENSITY (mw/cm²)		
CENTER	23	23
EDGE	1	1

Figure 4.

SPS PRELIMINARY BASELINE COMPARISON

MASS STATEMENT (10^6 KG)	5 GW	10 GW
COLLECTOR ARRAY (DRY)	13.9	51.8
ANTENNA SYSTEM	14.2	25.2
TOTAL SPS DRY WEIGHT	28.1	77.0
TOTAL SPS DRY WEIGHT WITH 30% GROWTH	36.5	100.1
COST (10^9 1977 $'s)		
COST TO PLACE FIRST SPS (INCLUDES DDT & E)	66	87
AVERAGE UNIT SYSTEM COST	14	23

Figure 5.

SATELLITE POWER SYSTEM

PROGRAM DEFINITION PLAN (FUNDING BY AGENCY MANAGEMENT RESPONSIBILITY)

	FY 77	FY 78	FY 79	FY 80	TOTAL
DOE					
ENVIRONMENTAL FACTORS	220	1,940	2,050	1,740	5,950
COMPARATIVE ASSESSMENT	95	376	754	565	1,790
SOCIO-ECONOMIC ASSESSMENT	164	537	537	322	1,560
TOTAL DOE	479	2,853	3,341	2,627	9,300
NASA					
SYSTEMS DEFINITION	1,800	1,700	1,300	800	5,600
SPACE RELATED TECHNOLOGY	700	0	0	0	700
TOTAL NASA	2,500	1,700	1,300	800	6,300
TOTAL DOE/NASA	2,979	4,553	4,641	3,427	15,600

NASA HQ RG78-1076 (1)
1-24-78

Figure 6.

ALTERNATIVE POWER-GENERATION SYSTEMS

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INTRODUCTION

At present, Earth-orbital power systems consist almost exclusively of photovoltaic arrays and batteries. Because the characteristics of this class of power system are both well known and gradually improving through evolution, mission planners are on familiar ground in selecting photovoltaic power systems. The photovoltaic system, of course, requires orientation of a solar array of large area toward the Sun. This array obscures the field of view, adds to atmospheric drag in low orbit, and could possibly interfere with rendezvous or with departure from an orbiting spacecraft. The performance of the photovoltaic array also degrades as a result of radiation damage, and the batteries used for energy storage are of limited life in low Earth orbit.

Thermal space power systems have very different characteristics. Chiefly, they are more compact, of long life, and far less susceptible to radiation damage than photovoltaic systems. Those power systems that obtain their thermal input from nuclear heat sources can produce power whether in sunlight or shade and without the orientation toward the Sun required by the solar arrays.

Like the photovoltaic power systems, the thermal power systems are also evolving; but, unlike the photovoltaic systems, they have had comparatively little use in space. This paper surveys the present state of the art of thermal power systems. Because of the great potential variety of thermal power systems, the heat sources, the power-conversion systems, and the integration of thermal power systems with missions are treated sequentially.

SOLAR HEAT SOURCES

The Sun emits radiant energy equivalent to that from a blackbody at about 5800 K. At the Earth's distance from the Sun, the Sun's thermal flux is 1400 watts per square meter. Because a paraboloidal mirror and solar heat receiver can collect at least 80 percent of this energy, solar mirrors can provide 10 times the heat per unit of collector area that is obtainable from photovoltaic arrays in combination with a resistance heater. Thus, solar mirrors have a great size advantage over solar-cell arrays if the energy sought is heat.

Inasmuch as a power-conversion efficiency of 0.30 is readily achievable, an electric power output of about 350 watts per square meter is potentially obtainable from a solar thermal power system during full-sun operation - about three times the power from arrays of solar cells. The technologies that can provide these high levels of either power or heat are thus of considerable interest.

A perfect paraboloidal mirror can produce a small image of the Sun, the image size being determined by the mirror's focal length and the Sun's apparent radius of 4.8 milliradians (16 arc-min). On the other hand, a real mirror will have surface inaccuracies and will therefore produce a larger image. The resulting high flux of solar energy can be focused on an aperture in a heat-receiving cavity (a hohlraum, fig. 1), and the thermal energy collected by the heat receiver can be used directly or can be converted to electric power in a thermal power system. The hot cavity will radiate heat through the cavity's aperture as would a black surface at the mean radiant temperature within the cavity. Because geometrical errors in the mirror's surface increase image size and thereby require a larger aperture, these surface errors result in an increase in the thermal power lost by radiation from within the cavity. By exploring the relation among mirror-surface error, aperture size, and radiation loss through the aperture, the efficiency achievable by solar heat-collection systems can be assessed.

For specificity in the following discussion, a mirror is assumed to be a paraboloid of revolution, to have a diameter of 30.5 meters (100 ft), and to be so oriented in space that both the Sun and the aperture of the solar heat receiver are centrally positioned on the axis of the paraboloid. A mirror of this size intercepts roughly 1 megawatt of sunlight. Further, a ray from the mirror lip to the focus is taken to form an angle of 45° with the mirror axis. Under these conditions, focal length is 18.4 meters (60.4 ft) and the f-number of the optical system is f/0.6.

Each area element of the mirror surface forms a circular image of the Sun of 8.6-centimeter radius at the image plane. For surface elements correctly oriented, the Sun's image formed by each element is centered on the mirror's axis. Accordingly, an error in orientation of a given surface element displaces the Sun's image formed by that element radially from its nominal location on the mirror's axis. For analysis, the surface errors were assumed to have a Gaussian distribution, that is,

$$p = \sqrt{\frac{2}{\pi}} \exp\left[-\frac{1}{2}\left(\frac{\varepsilon}{\sigma}\right)^2\right]$$

where

p probability density for a given error

ε surface error

σ standard error

For various radial positions on the image plane, the flux from various elements of the mirror surface was integrated over this probability distribution. The results are given in figure 2.

The perfect image (zero error) has a flux cutoff at the image radius of 8.6 centimeters. As the standard error of the mirror surface increases, the image is spread out over progressively larger areas and peak flux decreases. For a conservative reflectivity of 0.9, peak flux is 4000 watts per square centimeter with zero error and about 3000 watts per square centimeter if the standard error is 2 milliradians (7 arc-min).

For various radial positions on the image plane, the values of flux ϕ in figure 2 were multiplied by $2\pi r$ and replotted in figure 3 in order to make graphic the selection of optimum aperture size. The significance of the ordinate in figure 3 stems from the following relation:

$$P = \int_0^R 2\pi r\phi \, dr$$

where

P thermal power entering aperture

R radius of aperture

r radius on image plane

Thus for any given aperture radius R, the area under any given solid line from 0 to R represents the solar power entering the aperture. In turn, the area under the same curve for all aperture radii greater than R represents the solar power striking the aperture plate and therefore lost by not entering the receiver cavity.

The heat radiated from the aperture itself is shown by the dashed lines in figure 3 for two values of cavity temperature. The value of 1200 K is characteristic of the maximum temperature of a number of power-conversion systems, and 1800 K is approximately the melting point of iron and thus is representative of high-temperature processing in space. The values of radiation from the aperture have been increased by 60 percent above the values for a blackbody in order to account for the thermal radiation from the aperture during the shadow portions as well as the sunlit portions of a low orbit about the Earth. Nominal values of 60 minutes of sunlight and 36 minutes of shadow were assumed. Thus, for both the solid and dashed lines the area under each line is proportional to the energies - for an entire orbit - that enter the aperture, that are reradiated through the aperture, or that are deflected by the aperture plate. The specific areas are identified in figure 4. For any given cavity temperature and given error in mirror surface, the image radius at which the dashed line crosses the solid line is the optimum aperture size. At radii smaller than the optimum, the solar flux exceeds the energy reradiated and at larger radii the reverse prevails.

For various given mirror errors, the net energy captured was integrated from zero to the optimum aperture radius. The resulting collection efficiencies are shown in figure 5. Collection efficiency is the ratio of the net energy captured to the solar energy incident upon the mirror; heat losses from the outer surface of the receiver were neglected. At low errors, efficiency asymptotically approaches the value of 0.9 assigned to mirror reflectivity. For a cavity temperature of 1200 K, collection efficiency is nearly constant for surface errors less than 1 milliradian (3 arc-min) and drops slowly to 0.8 for a standard error of 6 milliradians (21 arc-min). For a cavity temperature of 1800 K, collection efficiency is above 0.75 for mirror errors below 2 milliradians (7 arc-min). Thus, overall collection efficiencies over 0.80 are achievable, even at cavity temperatures as high as 1800 K, if only mirrors can be made with sufficient accuracy (1.5 mrad, or 5 arc-min).

Figure 6 shows a mirror 6 meters (20 ft) in diameter that was made by NASA Lewis of magnesium and in 12 sectors. Each sector was given its parabolic shape by creep-forming it over a heated, machined aluminum die. For each sector a plate of magnesium 2.5 centimeters thick was milled on the back in order to produce flanges along each edge and a roughly rectangular grid of ribs. The front surface and the ribs were all approximately 1.5 millimeters (60 mils) thick. After creep-forming, each sector was spray coated with epoxy. The surface tension of the epoxy produced a glossy surface onto which aluminum was deposited by vaporization in a vacuum. After the sectors were bolted together into a paraboloidal mirror, the mirror surface was inspected for accuracy by using the optical-inspection machine in figure 7. The standard deviation of the errors was about 1 milliradian (4 arc-min). The distribution of errors was also very close to a Gaussian curve, as had been assumed in analyzing the effects of mirror error on performance. This mirror weighed about 5 kilograms per square meter (1 lb/ft^2).

A mirror 1.8 meters (6 ft) in diameter was also made by NASA Lewis from 0.4-millimeter- (17-mil-) thick magnesium sheet, also by creep-forming the sectors over a heated aluminum form (fig. 8). The sectors were joined by slotted splines and epoxy (ref. 1). Total weight of the mirror was 1.6 kilograms per square meter (0.32 lb/ft^2), but its surface accuracy was not measured.

A comparable mirror was manufactured by TRW from aluminum sheet 0.4 millimeter (20 mils) thick by stretch-forming the sectors over a mandrel (ref. 2). Eight sectors and a rear supporting torus were bonded together into a paraboloid 1.5 meters (5 ft) in diameter. Just as for the Lewis mirrors, the front surface was coated with epoxy and aluminized. The standard deviation of errors in the mirror surface was 0.3 milliradian (1 arc-min). In full sunlight, such a mirror can supply over 900 watts per kilogram.

Thus, lightweight mirrors of sufficient accuracy for efficient collection of solar thermal energy (fig. 5), even if temperatures of about 1800 K are sought, have been built and tested on Earth. Thermal power outputs in excess of 1100 watts per square meter are achievable, a value 10 times the output presently attainable from solar arrays. Although these accurate mirrors have been assembled on Earth, large mirrors would require assembly or erection in space and this remains to be demonstrated. The heat from such mirrors can be used for power generation and/or space processing. For example, a paraboloid 100 meters in diameter appears potentially capable of supplying 5 megawatts of average thermal power in low Earth orbit. In most instances, the attainable temperature will be limited by the materials of the solar heat receiver rather than by the attainable accuracy of the concentrator.

NUCLEAR HEAT SOURCES

Either nuclear reactors or radioisotopes can also provide heat for direct use or for power generation. In terms of adaptation to the mission, nuclear energy sources are very different from solar sources. They are very compact and, since they operate with complete independence from the Sun, they permit operation in any Earth orbit without the constraint of orientation toward the Sun. In turn, operations in space can be significantly simplified because the field of view is not obscured, because rendezvous is simpler, and especially because the Earth (or any other celestial body) can be continuously observed without the constraint of also orienting an array of solar cells toward the Sun.

Life, cost, and nuclear-radiation shielding are all significant factors in nuclear heat sources. Plutonium-238 is the accepted radioisotope for spaceflights. Because its half-life is 87 years, the thermal output declines less than 8 percent in 10 years. Thus, the life of the radioisotope does not limit mission duration, in any practical sense. On the other hand, plutonium-238 costs about $650 per watt of heat produced at the time the radioisotope capsules are manufactured. If radioisotopic decay is included, unit cost is roughly $700 per thermal watt produced after 10 years. Obviously, this unit cost results in overall costs of $700 000 per thermal kilowatt and $700 million per thermal megawatt. Also, the total quantity of radioisotope that can be readily produced in a year's time is limited (ref. 3). These factors of unit cost and availability will make radioisotopic heat sources up to a few tens of kilowatts reasonable, but larger heat sources less reasonable. The technology for such radioisotopic heat sources is nearly all available, and a number of radioisotope power supplies have already been flown. The multihundred-watt capsules each produce 2400 watts of heat and operate at about 1100° C.

For nuclear reactors, reactor life is a design variable and very long lives (decades) are achievable. Basically, as heat is continually produced by the reactor, its fuel wears out. Two factors account for this wearing out:

(1) As uranium is progressively consumed, the reactor's ability to remain critical and to sustain a chain reaction declines.

(2) The fuel swells because of accumulating radiation damage to the fuel structure and because of accumulating fission products (2 product atoms for each uranium atom fissioned).

Within given limits on these two design variables, the reactor can be designed for almost any reasonable energy output, simply by incorporating enough fuel into the reactor and making the reactor large enough. Within the limit on energy output of a given reactor, power can be traded for life, and conversely.

As shown by reference 4, reactor weight increases fairly slowly if greater energy output (or longer life) is sought. Within a given family of reactors designed for the same operating temperature and a 7-year life, reactor weight is essentially constant for thermal powers from 16 to 200 kWt and increases only one-third as fast as thermal power from 200 to 1000 kWt. At the 1000-kWt level, reactor weight is estimated to be 360 kilograms. Similary, reactor cost will also change only slowly with power and life.

Reactor shield weight varies greatly with mission-related factors. For unmanned spacecraft, rather thin shields just between the reactor and payload (shadow shields) can be used and might weigh only a few hundred kilograms. On the other hand, even a shadow shield for manned flight might weigh 10 tons because of the low dose-rate limits specified for human beings. Such a shadow shield would prevent man's intrusion into the unshielded zone. Although uniform shielding all around the reactor (4π shielding) would give great operational freedom about the reactor, shield weight would then increase to perhaps 70 tons. Various shield weights between these limits can be achieved by compromising man's operational freedom about the reactor (chiefly with respect to solid angle) and by tailoring the reactor's shield design to fit these operational constraints. Thus, the reactor shield selected for manned flight will require detailed consideration of the relation between shield design and man's activities about the spacecraft. In contrast with shielding for reactors, radioisotopic heat sources using plutonium-238 require only minor shielding.

The radioisotope is most useful at low powers (below perhaps a few tens of kWt), and the reactor for high powers. The reasons for this stem from the facts (1) that the radioisotope, with its comparatively high unit cost, increases in cost in direct proportion to thermal power and (2) that the reactor and its shield increase only slowly in weight and cost as required thermal power increases. Thus, at high thermal powers, reactors would be the preferable nuclear heat source.

POWER-CONVERSION SYSTEMS

Some overall characteristics of thermal power-conversion systems are summarized in table I. Thermoelectric power systems have already flown as radioisotope thermoelectric generators (RTG's) on several long-lived spacecraft. To date, these power systems have produced powers up to 150 watts and had overall efficiencies of about 0.06. With modest advances in technology, efficiency up to perhaps 0.10 appears achievable. The RTG's are highly developed, rugged, inert in terms of interaction with a mission, and long lived. For long-duration missions that exploit the Space Transportation System, RTG's should be considered for both emergency power and free fliers.

Figure 9 shows, at the left, two multihundred-watt (MHW) RTG's mounted atop a Lincoln Experimental Satellite. The two RTG's produced 250 watts of electric power from two MHW heat sources. These same two MHW capsules are also intended for use in the mini-Brayton concept shown at the right. The Brayton conversion system, with its higher conversion efficiency, would produce

1300 watts from these same two highly developed heat sources. A 10-kilowatt version of such a Brayton power-conversion system has been under test for several years (fig. 10). The measured efficiency of this power-conversion system is 0.29 (fig. 11), but heat losses from a nuclear heat source might lower overall system efficiency to 0.27 or 0.28. Substituting already developed components would raise power-conversion-system efficiency to about 0.32 (ref. 4).

The main rotating component of this engine has a compressor, a turbine, and a generator on a single shaft supported by two gas-lubricated journal bearings and a double-acting thrust bearing (figs. 12 and 13). This rotating component has completed 36 000 hours of testing, and system performance has been stable over this period. Testing will continue toward a goal of 50 000 hours.

Organic Rankine systems for use in space have been investigated for about the past 15 years, and one power-conversion system operated for 8000 hours. Efficiency of 0.15 has been demonstrated, and 0.18 is projected for the future.

Thermionic converters have been investigated for generation of space power for about two decades. One converter operated stably for over 45 000 hours. Current concepts for thermionic powerplants (ref. 5) incorporate thermionic converters operating at 1650 K and having efficiencies of about 0.15. In reference 5, converter output at 0.15 efficiency is 500 kilowatts of unregulated power at 54 volts dc and about 9300 amperes. Power conditioning and regulation reduce this output to 343 kilowatts, for an overall efficiency of 0.10. Research on thermionic diodes (ref. 6) has shown that at higher temperatures (1800 to 2000 K) and high power densities (20 to 30 W/cm^2), converter efficiency can be raised to 0.3. In turn, overall system efficiency might then be about 0.2. As shown by reference 7, thermionic and reactor-Brayton powerplants for unmanned flight have about the same specific weight (g/W). However, exploitation of the higher power densities and higher operating temperatures of reference 6 will reduce thermionic weight significantly.

On the other hand, high operating temperatures cause more swelling of the reactor fuel or, alternatively, require reduced energy production from a given mass of fuel. In reference 7, UC-ZrC fuel is used for thermoelectric and Brayton power conversion within its swelling limits. Inasmuch as fuel swelling dominates reactor design in the megawatt range, the thermionic reactor could not use the UC-ZrC fuel within reasonable swelling limits. The solution was to switch to the more advanced Mo-UO_2 fuel and to increase reactor size. A larger reactor means increased shield weight, a crucial problem for manned flight inasmuch as shield weight dominates powerplant weight.

For a given reactor and shield operating at a given reactor temperature, a given amount of thermal energy can be produced over the mission. This energy might be used at a high rate but then only for a limited period. High power-conversion efficiency is an advantage because it would permit either the highest electrical power or the longest mission duration from a given reactor and shield. This is one of the outstanding characteristics of the Brayton power-conversion system. Among the power-conversion systems investigated for use in space, the Brayton system could also produce the largest electric power from any given heat source - be it solar mirror, radioisotope, or nuclear reactor.

MISSION INTEGRATION

Thermal power systems offer the opportunity for integrating the power system with the mission in ways very different from those offered by a photovoltaic power system. Factors of possible impact are as follows: (1) heat for processing in space, (2) heat for life support, (3) refrigeration of cryosensors, and (4) laser power. The Brayton power-conversion system is used as an example of what is achievable, chiefly because possible application of this system has received more attention.

Heat for Processing in Space

If heat for processing is needed at high temperature, some heat can be drawn directly from the power system's primary heat source. For nuclear heat sources, temperatures up to that of the heat source itself are usable. If still higher temperatures are needed, a solar mirror can readily achieve 2000 K with good efficiency (fig. 5). A solar mirror can provide about 1100 watts per square meter when in sunlight, about 10 times the value presently achievable with arrays of solar cells.

All the thermal power systems reject waste heat that might be useful to a mission. In a Brayton system optimized for high efficiency, this heat might be available in a fluid heated to 175° to 200° C (350° to 400° F). Higher temperatures are achievable with modest reductions in power-generation efficiency.

Heat for Life Support

Similarly, supplying heat for life support was investigated (ref. 8). In particular, adaptation of the Brayton cycle so as to provide heat at the required temperature was studied.

Consider now the problem of supplying equal amounts of energy in electrical and thermal forms, and compare area requirements for a photovoltaic and a solar-Brayton system in low Earth orbit (60 min of sunlight and 30 min of shade). Consider that the solar-Brayton system will continuously supply 10 kilowatts of electric power and 10 kilowatts of otherwise-rejected heat. For a collection efficiency of 0.8 and a conversion efficiency of 0.25, 54 square meters is required - corresponding to an average of 370 watts per square meter for a full sun-shade orbit. If for the photovoltaic system the combined efficiency of power processing and of battery charge and discharge is 0.7, if all the heat is produced during only the sunlit portion of the orbit, and if output of the photovoltaic array is taken as 140 watts per square meter (13 W/ft^2), 230 square meters is required - corresponding to an average output of 87 watts per square meter for a full sun-shade orbit. Thus, the photovoltaic array would require over four times the collector area of the solar-Brayton system. Requiring even modest amounts of heat at moderate temperature thus favors the thermal power systems.

Cryogenic Cooling

If a given spacecraft requires cryogenic cooling of, for example, infrared sensors for a long time, two reasonable choices are the Vuilleumier (VM) cooler and a Brayton system that is adapted for refrigeration as well as its usual function of producing power. In reference 9, the adapted Brayton system was analyzed and compared with the VM cooler.

The adapted Brayton cycle is shown in figure 14. The compressed gas is divided into two streams: one for power generation and the other for refrigeration. The refrigeration stream is then cooled in a radiator to the temperature at the compressor inlet. This stream is then further cooled in a recuperating heat exchanger. This compressed, cooled gas is then expanded in a turbine whose energy extraction further cools the gas and whose power output augments that of the Brayton power system. The cold gas at the turbine discharge provides the cryogenic cooling and is then reheated in the recuperating heat exchanger almost back to the compressor inlet temperature.

For this cooling application, the Brayton system is operated on neon gas, which liquefies at approximately 27 K. For the analysis in reference 9, cooling by the Brayton system was limited to 50 K in order to avoid any liquefaction and thereby to simplify the performance calculations.

A cooling load of 40 thermal watts at 50 K was selected. For this cooling load, a VM cooler continuously requires about 120 watts of electric power and 3200 watts of heat. A photovoltaic array is assumed to produce electric power for both the power and heat demands. The orbital period is taken as 90 minutes and the sunlit portion as 60 minutes. Charge-discharge efficiency of the batteries is taken as 0.7 and the array output as 140 watts per square meter. For these conditions, the required array area is 41 square meters. Reference 9 shows that the refrigeration load reduces the electric power output of the Brayton system by 700 watts. If the efficiency of the Brayton power-conversion system is taken as 0.3, the collection efficiency as 0.8, and the sunlit period as 60 minutes in a 90-minute orbit, the added collector area required in order to regain the 700 watts of electric power is 3.1 square meters. Thus, the photovoltaic system requires 13 times as much collection area as the Brayton system.

Laser Power Transmission

A concept for generation of a gas-laser beam by adaptation of a Brayton power system was analyzed in reference 10. The concept is shown schematically in figure 15. Gas for operation of the laser is first compressed and then heated in the recuperator and in the nuclear reactor. After rapid expansion in a supersonic nozzle, the gas in temporary disequilibrium emits its beam of laser power. The resulting high-velocity stream is diffused in order that much of its kinetic energy might be recovered. The resulting stream of still-hot gas then passes through a turbine that drives the compressor and an alternator.

SUMMARY OF RESULTS

In considering space power generation systems as alternatives to photovoltaic systems, the following conclusions have been reached:

1. Radioisotope thermoelectric generators (RTG's) are highly developed and available for orbital application. Their inherently low powers make them most appropriate for special applications such as emergency power or free fliers.

2. Solar paraboloidal mirrors are suitable for supplying heat for either power generation or space processing. Mirror accuracy already demonstrated is sufficient for temperatures required to melt iron (1800 K). However, the means for assembling or erecting these mirrors in space are not yet developed. Collector areas required to supply process heat are only 1/10 those required by arrays of solar cells.

3. Dynamic power systems can use heat from either solar or nuclear sources. The highest efficiency and longest life have been demonstrated by the Brayton system, which has so far attained efficiency of over 25 percent and life in excess of 4 years.

4. The thermal power systems provide unusual opportunities in mission integration. A given solar mirror might drive a thermal power system as well as provide high-temperature heat for space processing. Otherwise-wasted heat from power generation can be used for life support, and the Brayton system can also be adapted for cryo-cooling of infrared sensors. If intense beams are needed from gas-dynamic lasers, the Brayton system might not only provide the hot, pressurized gas for the laser but also produce electric power from the hot gas stream exhausted by the laser.

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TABLE I. - ENERGY CONVERSION CHARACTERISTICS

Concept	Present efficiency	Future efficiency	Demonstrated endurance, hr
Thermoelectric	0.06	0.10	(a)
Organic Rankine	.15	.18	8 000
Brayton	.25	.32	36 000
Thermionic	.10	.20	45 000

[a]Spaceflight.

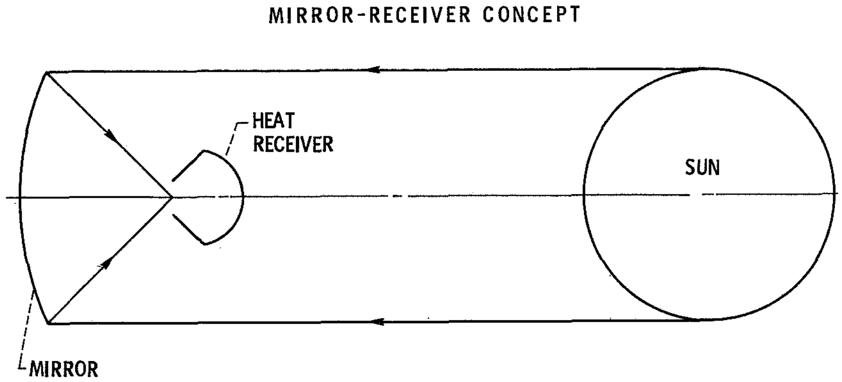

Figure 1.

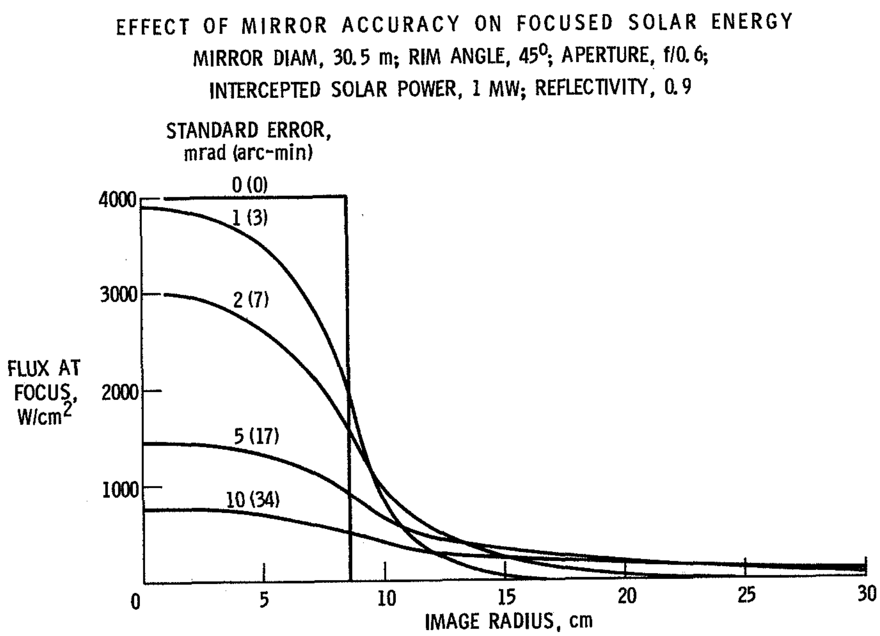

Figure 2.

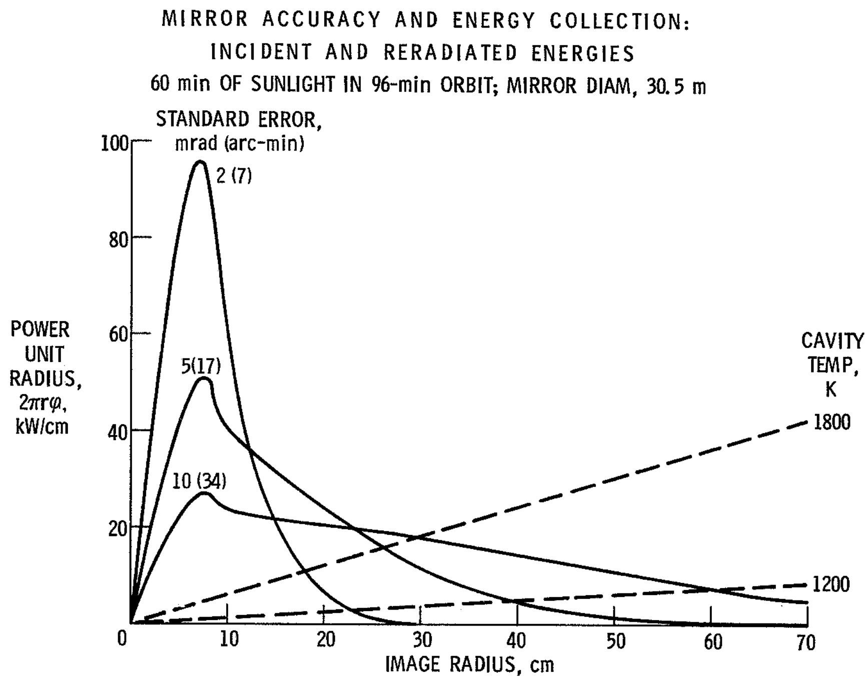

Figure 3.

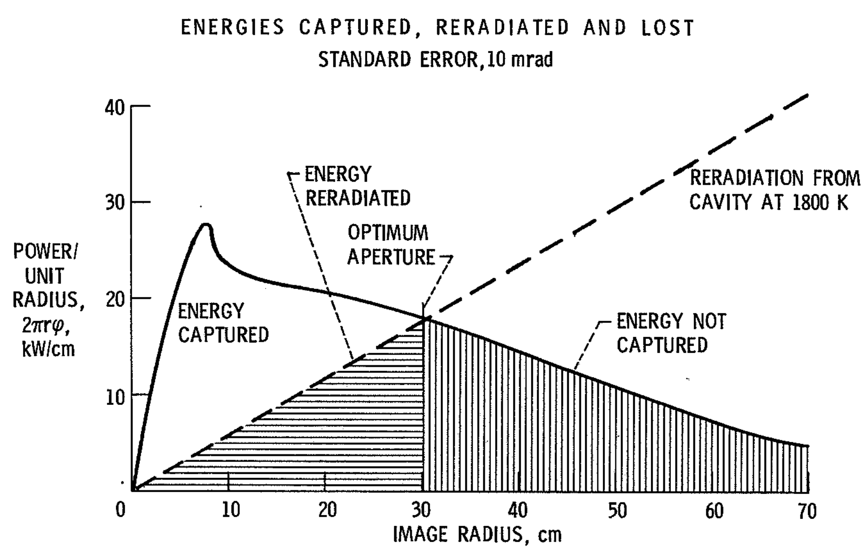

Figure 4.

EFFICIENCY OF COLLECTING SOLAR ENERGY

ORBIT, 96 min; SUNLIT PORTION, 60 min; REFLECTIVITY, 0.9

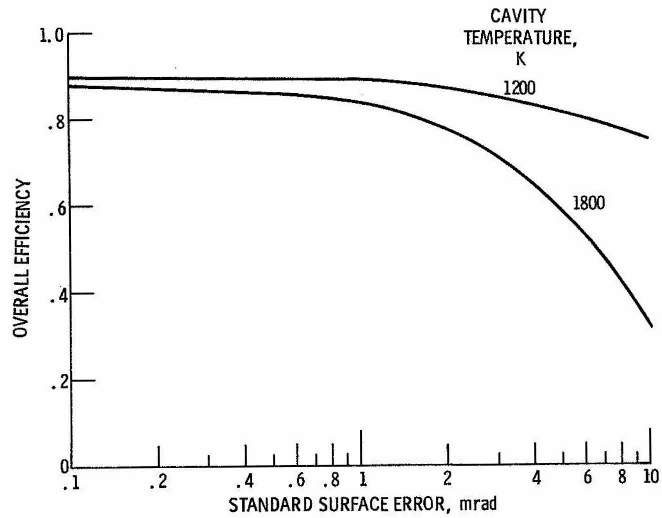

Figure 5.

SOLAR MIRROR - 20 ft IN DIAM

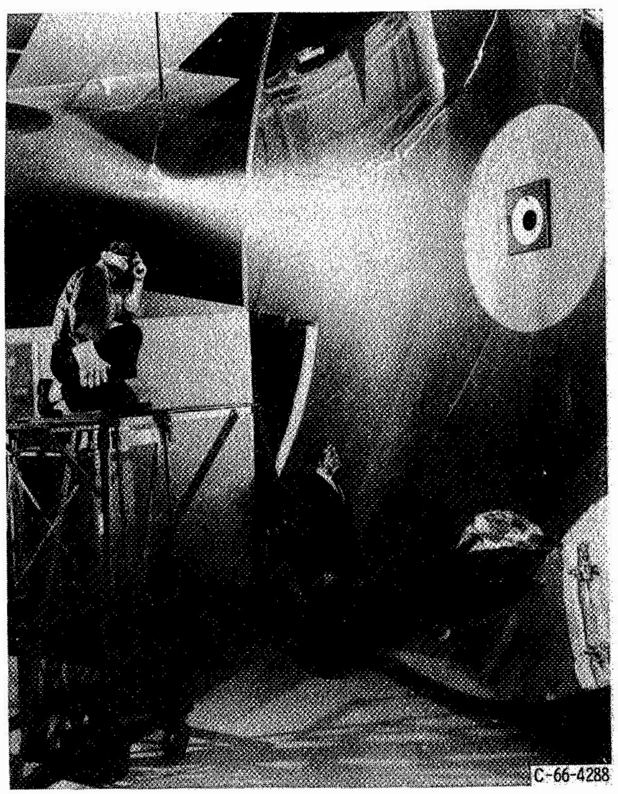

Figure 6.

MIRROR INSPECTION MACHINE

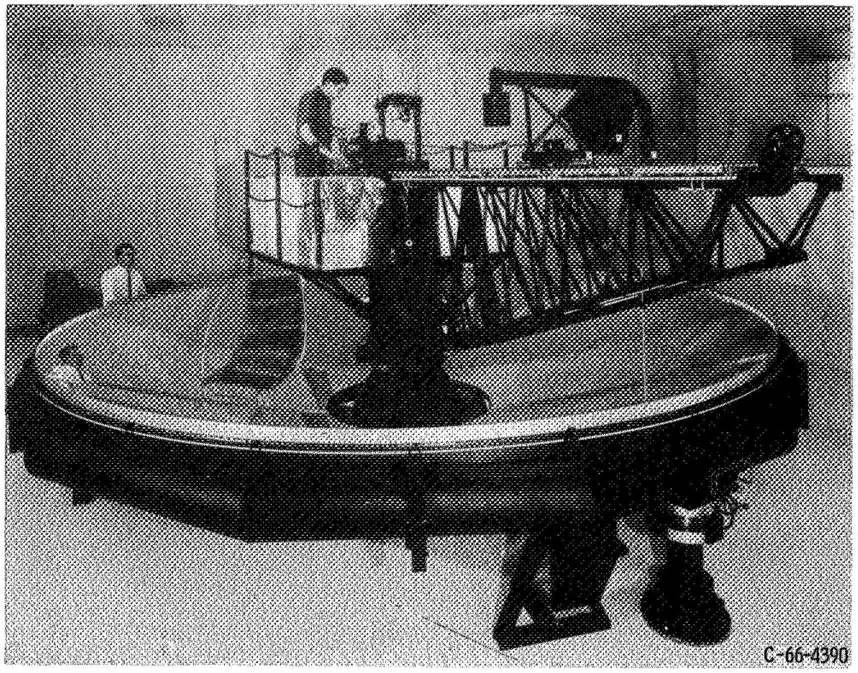

Figure 7.

SOLAR MIRROR - 6 ft IN DIAM

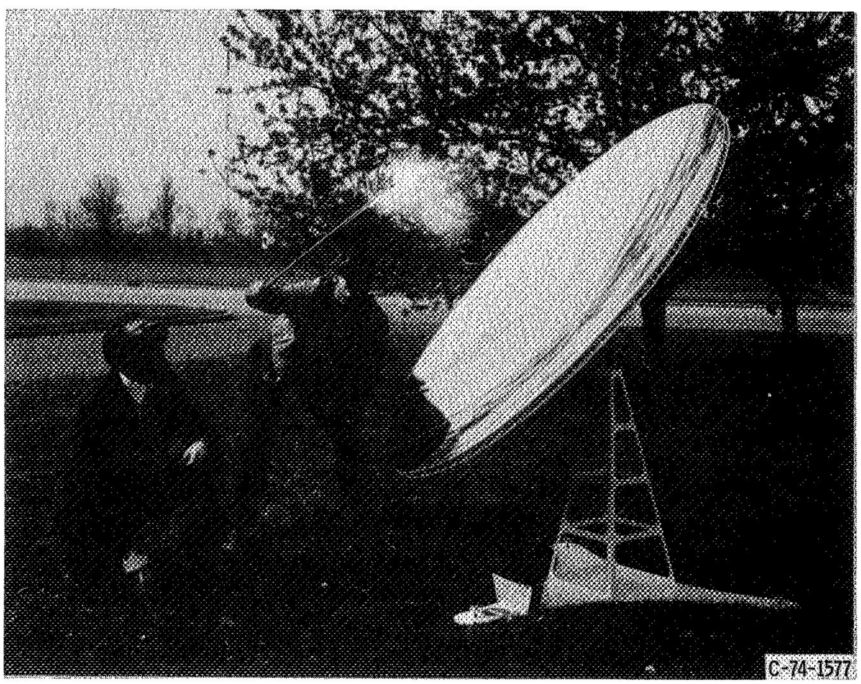

Figure 8.

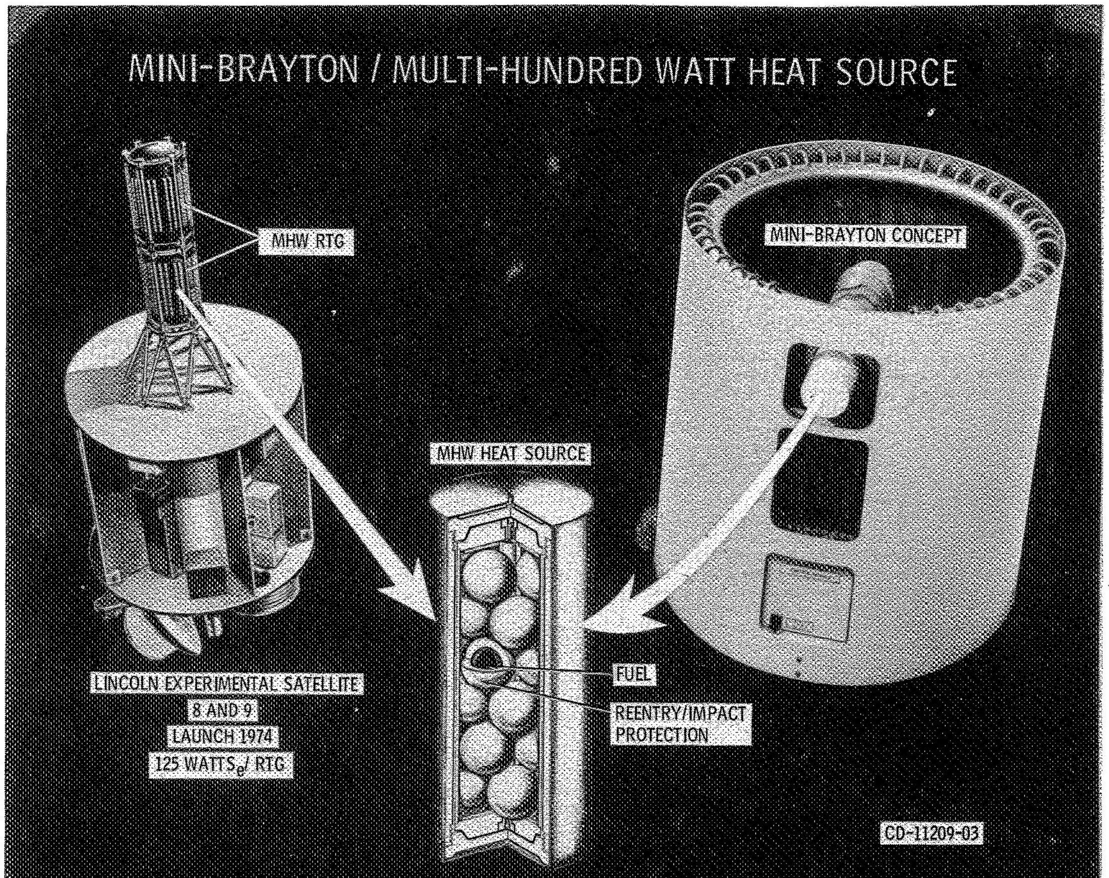

Figure 9.

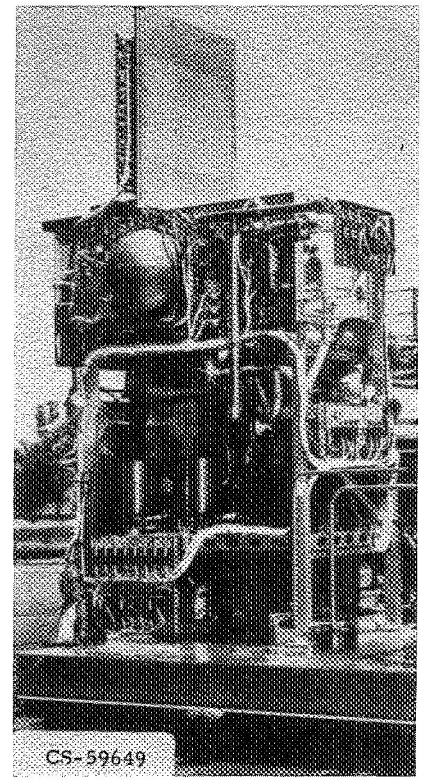

Figure 10.

BRAYTON ENGINE PERFORMANCE

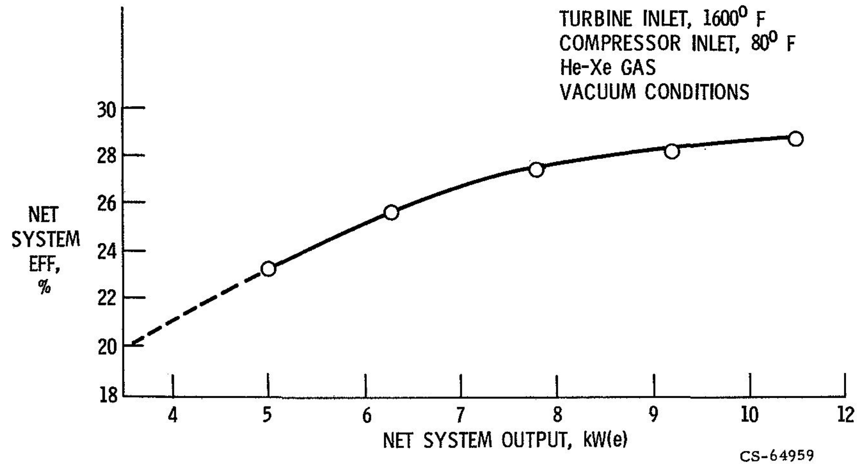

Figure 11.

BRAYTON ROTATING UNIT

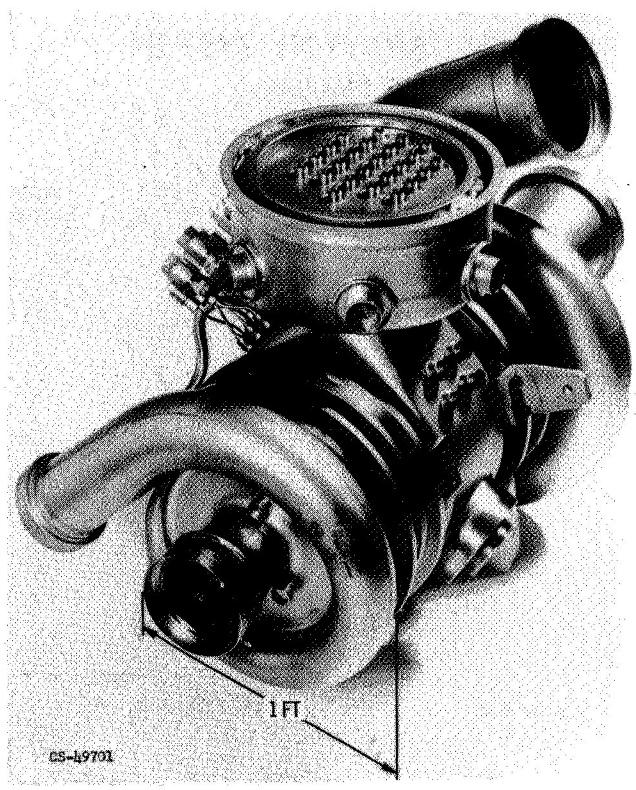

Figure 12.

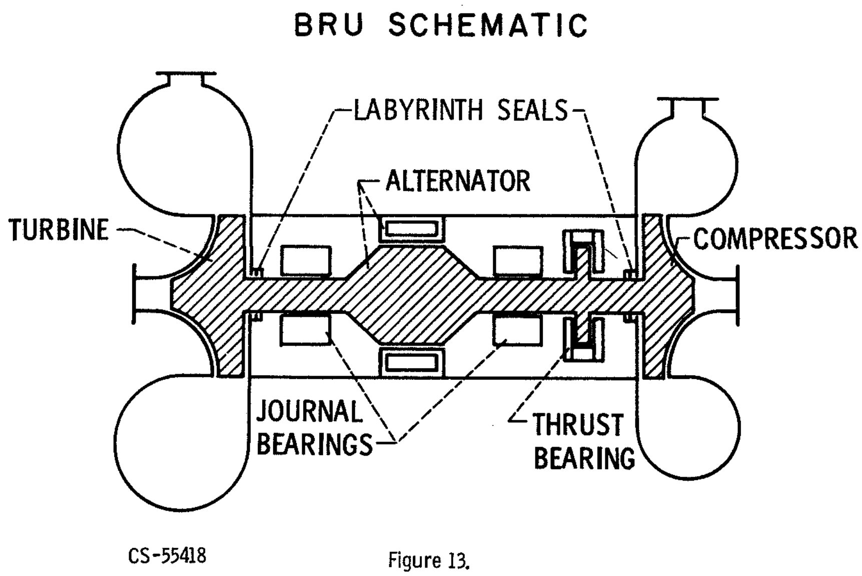

Figure 13.

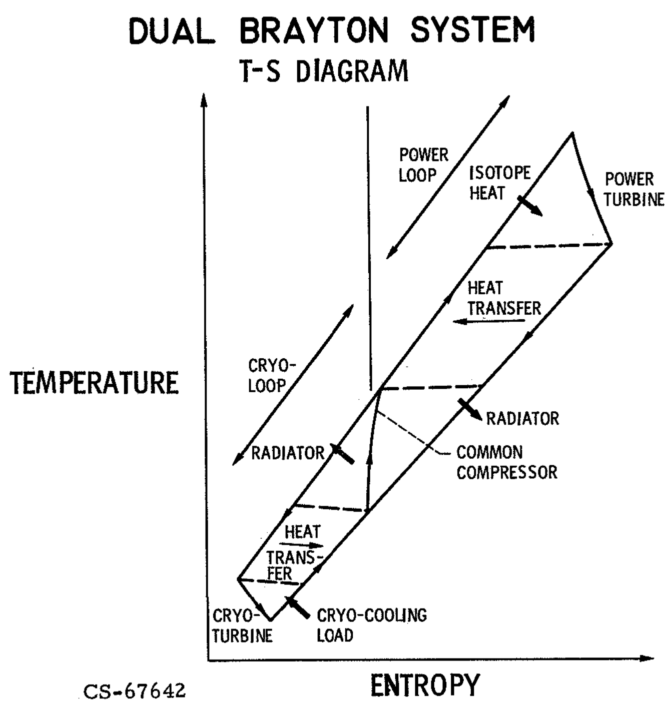

Figure 14.

CONCEPT FOR A LASER–BRAYTON POWER SYSTEM

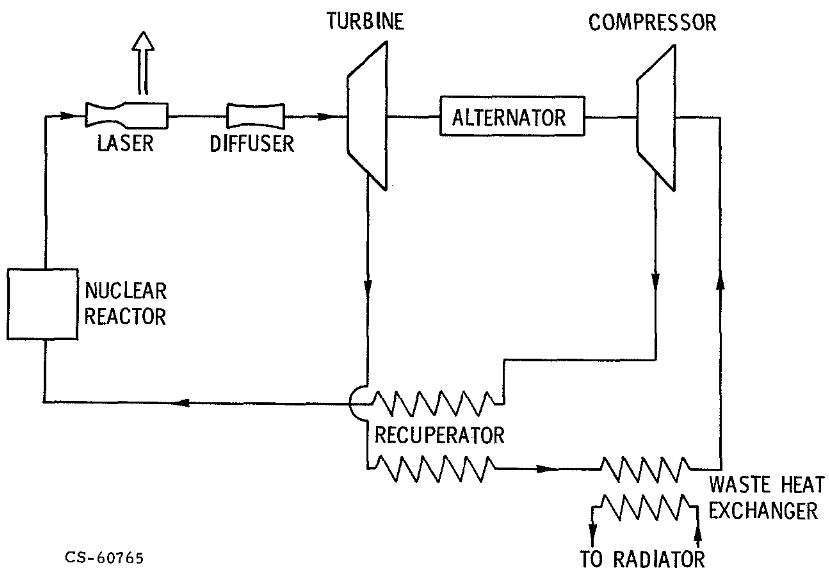

Figure 15.

A BRIEF SURVEY OF THE SOLAR CELL STATE-OF-THE-ART

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INTRODUCTION

This is a brief survey of the space solar cell state-of-the-art at the present time. Modern high performance cells made for space are discussed and the major recent developments that are expected to influence what solar cells will be available in five years or so are described.

MODERN SILICON SOLAR CELLS

The modern solar cell era started in 1972 when the COMSAT Corporation announced the violet cell with an efficiency exceeding 13% AM0 (reference 1). For nearly a decade prior to that the efficiency level for silicon solar cells had reached a plateau of 10 to 11%. A number of further innovations have been made since 1972. Modern cells in commercial production and in use or selected for flight use incorporate various combinations of these improvements.

Figure 1 illustrates the major features available. The cells are generally 0.2 to 0.3 mm (8 to 12 mils) in thickness and have a nominal base resistivity of 2 or 10 ohm-cm. Most have a smooth front surface, as depicted in the right-hand portion of figure 1, and have a shallow junction in the range of 0.10 to 0.15 μm in depth. The shallow junction increases the short-circuit current about 10% and improves radiation resistance. The top contact grid fingers are more closely spaced to compensate for the higher sheet resistance of the top N layer due to the shallow junction. So as not to increase the shadowing, grid fingers are now much narrower. Photoresist masks or bimetallic shadow masks are used to make fingers less than 0.025 mm wide and the shadowed area is reduced by 3 to 5%.

Tantalum pentoxide has now replaced silicon monoxide as the antireflection coating because its index of refraction provides a better optical coupling with the cover cement. The improvement in current is about 7%. Multiple layer antireflection (MLAR) coatings are now also becoming available on cells. They can lower the reflectivity and increase current another 3% or more. The shallow junction, tantalum oxide and thin photoresist fingers are the main features of the violet cell.

Another means employed to reduce surface reflections is the textured surface. The front surface is etched chemically to yield a random arrangement of small ($\sim$ 5 μm) pyramids which trap the light and also refract the light entering the cell so that it has a longer path length within the cell. The textured front surface, also with a tantalum oxide antireflection coating, increases the current about 7%. However, because the rough surface also has a low reflectivity for infrared light, the textured surface increases the operating temperature of the cell. An increase in cell temperature reduces voltage and hence power output. The advantage of the increased current is reduced or nullified by the reduced voltage.

At this time it appears the textured surface is most important for thin cells, especially when used in conjunction with a back surface reflector. The back surface reflector is a layer of reflecting metal, usually aluminum, that provides for internal reflection of light that would otherwise be absorbed at the rear contact. The back surface reflector reduces cell operating temperature by reflecting the unuseable red light from the back surface and out the front.

The back surface field is a heavily doped P^+ region at the back surface. Aluminum is usually employed as the dopant for the P^+ region. The back surface field increases the open-circuit voltage to 0.6 V or higher, independent of thickness and base resistivity. It also increases the current about 2%. The advantage of the back surface field is lost after sufficient electron irradiation. For example a back surface field cell 0.2 mm in thickness loses its advantage over a non-field cell of the same thickness after a fluence of about 5×10^{14} 1-MeV electrons/cm^2 (reference 2).

Modern cells are available in quite a variety of combinations of these features with efficiencies ranging from 11.8 to 14.8% AMO. They can be classed into two categories as shown in table I, hybrid and violet-type cells and back surface field cells. The back surface field cells generally have higher initial performance but there is overlap in the performance of these groups because of the different combinations of features available.

The cost of the cells are dependent on the specific details of a particular purchase (specifications, schedule, etc.) as well as cell type. A rough generalization (to within $\pm$ 10%) can be made, however--namely that the cell cost is about $100 per watt at beginning of life. For some missions the power requirements are heaviest early in the mission and back surface field cells may be cost effective. For other missions end-of-life power requirements dictate the array size and the non-field cells would be the economical choice.

RECENT SOLAR CELL R&D ADVANCEMENTS

Research is continuing on raising the efficiency of silicon solar cells. The open-circuit voltage is the parameter limiting the efficiency. Theory indicates that an open-circuit voltage approaching 0.70 V and an efficiency in the range of 18 to 19% AMO are possible if the N^+ region of the cell can be improved (references 3 and 4). Figure 2 shows as a function of base doping level the predicted open-circuit voltage and the voltage actually achieved with conventional N-P junctions. The data points are for base resistivities of 0.01, 0.1, 1, and 10 ohm-cm. Until recently the voltage has been limited to about 0.6 V. Also shown in figure 2 is a recent, yet unpublished, result by Lindholm at the University of Florida. By employing an N^+-N-P-P^+ structure Lindholm achieved a voltage of 0.64 V. More importantly his measurements indicate that in his device the voltage was not limited by the N or N^+ region but by the P region, which is amenable to improvement.

A recent spectacular achievement in silicon cell technology is the ultra thin cell. The key step in achieving a practical cell 0.05 mm (2 mils) in thickness is the use of an alkaline etch that very uniformly reduces the cell thickness (reference 5). The status of the thin cell activity at Solarex is summarized in table II. Pilot production of 2x2 cm cells is underway with efficiencies as high as 14% AMO; large cells, 5x5 cm, are in development with efficiency as high as 11% AMO.

Thin cell development is being supported at Spectrolab also, and the status is summarized in table III. This effort is in the laboratory development phase and cells with efficiencies to 15% AMO have been made. Some of these cells, which have back surface fields, were irradiated at JPL and exhibited radiation damage comparable to non-BSF cells for a 1-MeV electron fluence of 10^{15}. This result conforms to expectations that thin BSF cells should maintain their advantage out to high fluences (reference 6).

Wraparound contact cells have both contacts on the rear of the cell and thereby offer important advantages in cell interconnection and array assembly. Two general types are illustrated in figure 3. In one type the junction and N region are wrapped around the edge of the cell to the rear. With the wraparound junction approach it has been found that shallow junctions could not be used because of shorting through the junction at the cell edge. The efficiency is limited thereby to about 11.5% AMO (reference 7). The other approach shown in figure 3 employs an insulator around the edge and avoids the junction shorting problem. A shallow junction can be used. However, insulating layers applied by vacuum evaporation have pinholes that allow shorting of the N contact metallization to the P base region.

A method for applying a wraparound glass insulator layer by screen printing and firing was developed during a program to develop techniques for low cost fabrication of space-quality solar cells. In this program the main interest was on methods that would be easily mechanized or automated, especially methods that do not require use of vacuum chambers. This work was extended to include wraparound contacts. Table IV lists the main processes. The metallization steps utilized screen printing and the antireflection coating was applied by spinning-on and firing a commercially available preparation to yield a silicon oxide-titanium oxide coating. Junction diffusion was by heating of a spin-on source of dopant commonly used in the semiconductor device industry. Fifteen hundred cells were made in the contractor's terrestrial cell production facilities with an average efficiency of 10.9%.

High efficiency wraparound contact cells are now under development and the processes selected for their fabrication are listed in table V. Screen printing was found superior to vacuum evaporation for the application of the aluminum for the back surface field and the glass wraparound insulator. Efficiencies for a few cells have been over 15% AMO (reference 7). Pilot production with a goal of 14.5% average efficiency is planned.

The nonreflecting vertical-junction silicon solar cell which was conceived to increase radiation resistance is fulfilling its promise. The cell is made with a profusion of thin deep grooves in the top surface of the cell (figure 4). The junction follows the surface of the grooves and a greater portion of the electrons and holes are generated near the junction than in a planar cell, resulting in less sensitivity to carrier lifetime reduction by radiation damage. In the present program at Solarex the grooves are chemically etched into the surface of the aligned 110 silicon wafer through an oxide mask. Cells have been made in the laboratory with efficiencies as high as 14%. The vertical-junction cell has been found to degrade at about one half the rate of planar cells under 1-MeV electron irradiation (references 8 and 9).

It has long been recognized that gallium arsenide solar cells have the potential for higher efficiency, higher temperature operation, and better radiation resistance than silicon cells. However, results with gallium arsenide were not good until Hovel and Woodall (reference 10) introduced the gallium arsenide cell with a gallium aluminum arsenide window, which is illustrated in figure 5. The clear window is epitaxially grown on the gallium arsenide and eliminates carrier recombination at the gallium arsenide surface that had caused poor performance in early non-window cells. The performance achieved in space-program-supported gallium arsenide R&D activities is summarized in table VI (references 11 and 12). The best cells from terrestrial programs, whose efficiencies are reported for a terrestrial sunlight spectrum and sometimes with concentration, are estimated to have AMO efficiencies comparable to the space cells. Efficiencies above 18% AMO have been achieved, but it has been found that higher radiation resistance and higher end-of-life efficiency is achieved by using a smaller junction depth and window thickness. The beginning-of-life efficiency for the more resistant cells is in the 16-17% AMO range. The radiation damage resistance for the thin window and junction cells is significantly better than for silicon cells.

Individual glass covers are customarily bonded to solar cells to protect them from the electrons and protons in space. Fused silica microsheet, and cerium-doped microsheet are commonly used. They are stable and well proven but are expensive (very roughly 1/3 the cost of a cell). The covers are bonded to the cells with a silicone adhesive, the best of which are darkened slightly by UV light. Coatings are sometimes applied to the covers to filter out the UV and protect the adhesive.

FEP-Teflon sheet which has high resistance to UV darkening has been adopted as the cover glass adhesive on the Solar Maximum Mission to save costs. The material cost is low, a UV filter on the cover is not required, and the application and cleanup labor is reduced. The glass cover is applied by heat and pressure bonding of the sandwich of FEP sheet between the cell and cover. FEP cemented covers have been successfully tested in flight experiments on the ATS-6 and NTS-2 satellites.

Borosilicate glass has a thermal expansion coefficient closely matching that of silicon. Such glass can be bonded directly to the silicon cell by electrostatic bonding. The bond is made under pressure at elevated temperature with an electrostatic field between the cell and cover (reference 13). The SPIRE Corporation under Air Force support is investigating how to adapt the process to the modern, high performance cells.

Plastic materials have been investigated as cover materials that are less expensive and/or easier to apply than glass. Heat-bonded FEP Teflon covers were found to embrittle and crack, allowing proton damage to the cells in the ATS-6 flight experiment. Preliminary investigations at Lewis indicate adhesive bonding of FEP covers may eliminate cracking but the process requires further development.

Other polymeric materials that can be applied by spraying, dipping or spinning are also being investigated. Such coatings would be especially suitable for thin cells. The materials include FEP, silicones and polyimides. The coatings investigated so far have been darkened by UV, some severely. These coatings require further development before they would be acceptable for use on space cells.

The Air Force and NASA are continuing to support improvements in space solar cells. The general goals include improved efficiency, radiation resistance, lower weight and lower cost. The major ongoing solar cell R&D programs and their targets are listed in table VII for the Air Force and table VIII for NASA.

CONCLUSIONS

The following conclusions were reached from this brief survey of the solar cell state-of-the-art:

1. High performance silicon solar cells with a wide variety of features and efficiency to nearly 15% AMO are commercially available and are being utilized in flight programs.

2. Silicon cells as thin as 0.05 mm (2 mils) with high efficiency (14% AMO) and radiation resistance are nearing readiness.

3. Wraparound contacts can be applied to silicon cells 0.2 mm (8 mil) thick without compromising performance.

4. R&D programs are continuing to yield more efficient and radiation resistant silicon solar cells.

5. Gallium arsenide cells with high efficiency and radiation resistance have been made in laboratory facilities.

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TABLE I. - MODERN SILICON SOLAR CELLS

BEGINNING OF LIFE OUTPUT, AMO 28^{o} C

	POWER FOR 2 x 4 CM	EFFICIENCY
HYBRID AND VIOLET-TYPE CELLS	128-148 MW	11.8-13.7%
BACK SURFACE FIELD CELLS	136-160 MW	12.6-14.8%

COST

$90-$110 PER WATT, BOL

TABLE II. - ULTRA THIN SILICON SOLAR CELLS

SOLAREX/JPL

DESCRIPTION

ETCHED TO FINAL THICKNESS
0.05 MM THICK
SHALLOW JUNCTION
UNTEXTURED
PARTIALLY REFLECTING BACK SURFACE FIELD

STATUS

2 x 2 CM CELLS

IN PILOT PRODUCTION, 2000 CELLS DELIVERED TO JPL.
CURRENT PRODUCTION CELLS GIVE 65-74 MW (12 - 14% AMO).

5 x 5 CM CELLS

IN LAB DEVELOPMENT.
150 CELLS DELIVERED.
BEST EFFICIENCY ABOUT 11%.

TABLE III. - ULTRA THIN SILICON SOLAR CELLS

SPECTROLAB/JPL

DESCRIPTION

- 2 x 2 CM ETCHED TO FINAL THICKNESS
- 0.05 MM THICK
- SHALLOW JUNCTION
- TEXTURED
- PRINTED AL PASTE BSF
- AL BACK SURFACE REFLECTOR

STATUS

- IN LAB DEVELOPMENT
- BEST CELLS GIVE > 80 mW (14 - 15% AMO)
- EXHIBIT LOW RADIATION DAMAGE -- ONLY 17% LOSS AFTER 10^{15} 1 MeV ELECTRON FLUENCE, COMPARABLE TO NON-BSF

TABLE IV. - NON VACUUM PROCESSES FOR POTENTIALLY LOW COST SOLAR CELLS

SPECTROLAB/LeRC

DESCRIPTION

SURFACE TREATMENT	NaOH TEXTURING ETCH
JUNCTION DIFFUSION SOURCE	SPIN-ON DOPANT
CONTACTS	SCREEN-PRINTED Ag
ANTIREFLECTION COATING	SPIN-ON SiO_2-TiO_2
BACK SURFACE FIELD	SCREEN-PRINTED AL
INSULATOR FOR WRAPAROUND CONTACTS	SCREEN-PRINTED GLASS

STATUS

1500 CELLS MADE IN TERRESTRIAL CELL PRODUCTION FACILITIES	
AVERAGE EFFICIENCY, AMO	10.9%

TABLE V. - HIGH EFFICIENCY WRAPAROUND CONTACT SOLAR CELL PROCESSES AND STATUS

	SPECTROLAB/LeRC
DESCRIPTION	
SURFACE TREATMENT	NaOH TEXTURING ETCH
JUNCTION DIFFUSION SOURCE	GASEOUS DOPANT
CONTACTS	EVAPORATED CrPdAg
ANTIREFLECTION COATING	EVAPORATED Ta_2O_5
BACK SURFACE FIELD	SCREEN-PRINTED Al
INSULATOR FOR WRAPAROUND CONTACTS	SCREEN-PRINTED GLASS
STATUS	
LAB R&D NEARING COMPLETION	
MAXIMUM EFFICIENCY ACHIEVED	15.2%
EFFICIENCY GOAL FOR PILOT PRODUCTION	14.5% AVG.

TABLE VI. - GaAlAs-GaAs SOLAR CELL PERFORMANCE

EFFICIENCY

- HUGHES/AFAPL
  - CELL SIZE: 2 x 2 CM
  - EFFICIENCY: 16 - 17% AMO
- IBM/LaRC
  - CELL SIZE: 0.1 CM^2
  - EFFICIENCY: 18.5% AMO

RADIATION DAMAGE RESISTANCE

- EOL AND BOL EFFICIENCIES CAN BE TRADED OFF BY VARYING THICKNESS OF WINDOW AND JUNCTION DEPTH.
- DAMAGE RESISTANCE WITH OPTIMUM WINDOW AND JUNCTION IS SIGNIFICANTLY BETTER THAN FOR SILICON

TABLE VII. - MAJOR ONGOING SOLAR CELL R&D PROGRAMS —

AIR FORCE

ACTIVITY	TARGET	
NON-REFLECTING VERTICAL JUNCTION SILICON CELL	15% BOL, 12% @ 5 x 10^{15}	1978
HIGH EFFICIENCY SOLAR PANEL PROGRAM-PHASE II-Si	16% BOL, RAD. RES.	1979
SILICON CELL OPTIMIZATION	18% BOL, RAD. RES.	1981
EXTENSION OF ELECTROSTATIC BONDING TECHNOLOGY		
PULSED LASER HARDENING		
HIGH EFFICIENCY SOLAR PANEL PROGRAM-PHASE II GaAs	18% BOL, RAD. RES.	1980
MULTIBANDGAP SOLAR CELLS	25% BOL	1982

TABLE VIII. - MAJOR ONGOING SOLAR CELL R&D PROGRAMS —

NASA

ACTIVITY	TARGET		CENTER
HIGH EFFICIENCY SILICON CELL	18% BOL	1980	LeRC
INCREASED RADIATION RESISTANCE FOR HIGH EFFICIENCY SILICON CELLS	< 15% DEGRAD. AFTER 10 Y IN GEO	1982	LeRC
ULTRA THIN SILICON CELLS AND COVERS			
FRONT AND BACK CONTACT CELLS	13% BOL, 2 x 2 PILOT	1979	JPL
BACK SURFACE CONTACT CELLS	14% BOL	1980	LeRC
HIGH EFFICIENCY WRAPAROUND CONTACT SILICON CELL	14.5% AVG, BOL PILOT	1979	LeRC
LOW COST SILICON CELL TECHNOLOGY	$5/W TECH. READY	1980	LeRC
GALLIUM ARSENIDE CELL RESEARCH	< 25% RAD. DAM. AFTER 30Y IN GEO	1980	LaRC

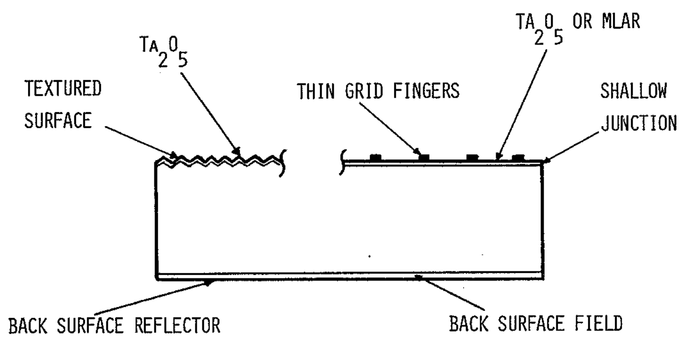

FIGURE 1. - FEATURES OF MODERN SILICON SOLAR CELLS.

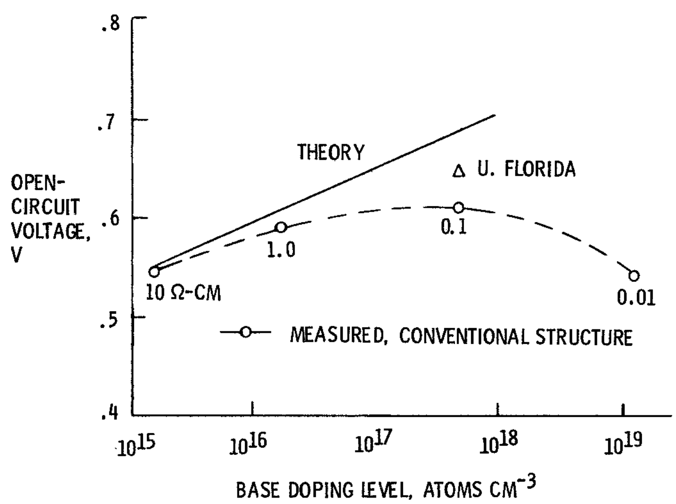

Figure 2. - Experimental and predicted open-circuit voltage dependence on base doping levels.

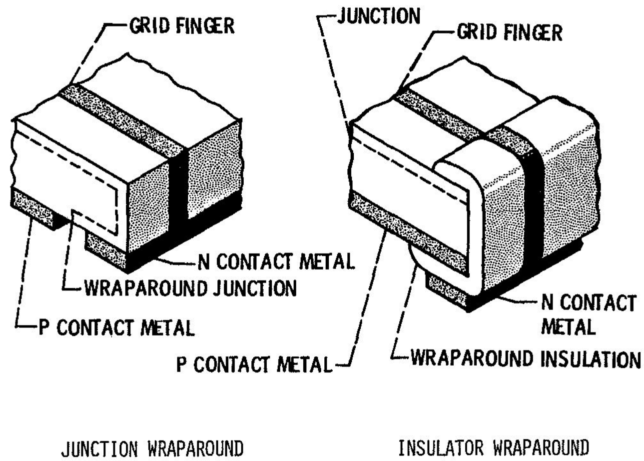

FIGURE 3. - TYPES OF WRAPAROUND CONTACT SOLAR CELLS. CLOSE-UP VIEW OF CELL CORNER.

SPONSOR: AFAPL

CONTRACTOR: SOLAREX

STATUS: LABORATORY R&D
BOL EFFICIENCY 14% AMO
DEGRADES AT HALF THE RATE OF PLANAR SILICON CELLS UNDER 1 MEV ELECTRON IRRADIATION

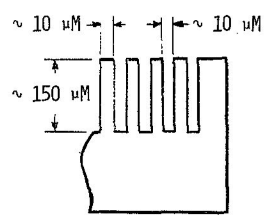

FIGURE 4. - NONREFLECTING VERTICAL-JUNCTION SILICON SOLAR CELL.

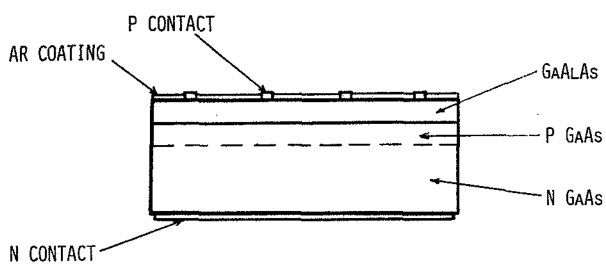

FIGURE 5. - DIAGRAM OF A GaAlAs-GaAs SOLAR CELL.

SOLAR ARRAY SYSTEMS

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SUMMARY

This paper discusses the recent past, present state-of-the-art, and future needs in the area of large photovoltaic solar arrays. In the past most attention was focused upon performance whereas in the future most of the effort should go into cost reduction. Suggestions are made regarding possible approaches to reducing cost such as on-orbit maintenance, extended lifetime, solar concentrators, and high-voltage modular concepts.

INTRODUCTION

Since the beginning of the Space age, solar photovoltaics have been the dominant source of space power. Their operation is well understood, they are not size limited and they allow considerable flexibility in design. With increased electrical load, additional arrays can normally be added. This is a definite advantage over nuclear or solar dynamic options. Weight, cost and size have always been important for solar array designers. However, in the past, because of the relatively small electrical load requirements, array sizes have ranged in the neighborhood of a few hundred watts to a few tens of kilowatts.

In general, the space budget in the 60's was of such magnitude that array cost was not an overriding consideration, and the fact that arrays were not generally large meant that weight and size were not terribly constraining. As the trend toward missions with larger electrical load requirements continues, arrays must be larger and the situation is changing. As larger arrays are required, they represent a larger part of the cost and weight of the mission; therefore, techniques are required to decrease weight and cost. One of the first significant advances in this area was made by the Air Force with the design and fabrication of FRUSA (Flexible Rolled Up Solar Array), a window-shade type array which represented a significant savings in weight of solar arrays and which could be modularized to form large arrays. FRUSA was followed by HASPS (Hardened Solar Power System) Solar Array, which although developed by the Air Force primarily to be hardened against radiation for military purposes, also represents an advance over FRUSA technology.

Array technology advancement in NASA in recent years has centered around the 66-W/kg SEPS array and 200-W/kg high performance array. These two array developments were specifically directed at multi-kW applications. Therefore, cost, weight and size are important features which have received considerable attention. The most visible contribution has probably been in weight reduction. Future solar arrays will be larger than ever to meet missions such as large power stations, expanded public service platforms, space construction base, space processing, etc. The power requirements of these types of missions range up to several hundreds of kilowatts, making the array an even more significant part of the total spacecraft than ever before. In many cases it will be the largest part; therefore, the economic viability of a mission will depend to a large part upon the cost of the solar array. This means that ways must be found to build larger arrays for lower cost/ benefit ratios. It also means that such approaches as on-orbit maintainability must be identified to allow reduction in life-cycle costs so that the technology investment may have a very high payback in terms of overall benefit. With this reasoning as the basis for the need for an advancement in the state-of-the-art of Solar Arrays, the new-initiative program alluded to in this paper was begun.

SOLAR ARRAY STATE-OF-THE-ART

Development of large solar arrays over the past decade has taken place within both the Air Force and NASA. As we investigate the current state-of-the-art of solar array technology available from these developments, we find it to be conspicuously all planar. The high state of development of 1-Sun silicon solar cells is at least one reason for this situation. However, in retrospect it appears that more development effort should have been invested in concentrators of some sort in view of their intuitively obvious advantages in deep space application. However, with the new high-flux cells being developed and increased concern for specific economic performance over specific weight performance, the concentrators will undoubtedly be given more attention in the future.

As we examine current state-of-the-art arrays, we turn first to the FRUSA (Flexible Rolled Up Solar Array) as a revolutionary departure from the prior approach to solar arrays. It was unique at the time of design in both the deployment and retraction subsystem and in its lightweight substrate design. The FRUSA Array shown in Figure 1 consists of two flexible panels 1.68m wide and 4.88m long which roll up on a common 20cm diameter storage drum. It also has a two-axis sun acquisition and tracking orientation mechanism. Deployment is accomplished by extendable metallic booms driven out by electric drive monitors. These booms collapse to roll up but spring out and become rigid when deployed. The basic array is rated at 1.5 kW, with growth to 3 kW accomplished easily by adding another flexible array to one arm of the orientation mechanism. The array specific power rating is 22 W/kg without the orientation mechanism. The array utilizes 2 x 2 cm N/P silicon cells 8 mils thick. They are covered with 6-mil microsheet coverslides and are connected 81 cells in series by 222 in parallel on each panel, using a bus system which is fabricated from a copper/kapton laminate. The cells are attached to the interconnects by solder. The substrate consists of a 0.001-inch kapton H-film bonded to 0.001-inch fiberglas. The array was flown in 1971 in a 430-N.M. polar orbit and successfully completed a 6-month flight test validating the concepts and techniques used.

The HASPS (Hardened Solar Power System) Solar Array is a technology extension of the FRUSA with radiation hardening being the prime motive. A slight weight penalty is paid by HASPS over FRUSA by the required modifications for radiation hardening. This array uses 2 x 2 cm cells 8 mils thick of both conventional N/P and lithium-doped variety. The cells have aluminum contacts and 6-mil fused silica covers without the magnesium fluoride coating. Conductors and interconnects are also aluminum, and cell contact to interconnect bonding is accomplished by ultrasonic welding. Each of the HASPS solar panels measures 4.42m x 2.29m when fully deployed, and the first flight of this type array will be in the early 1980's on a SIRE (Space Infrared Radiation Experiment) spacecraft (see Figure 2). It will be in a sun-synchronous orbit and the array will consist of 4 panels packaged in two drums with a total of 81,000 cells to provide an array power of 7.3 kW.

Within NASA the largest array developed and flown was the 21-kW Skylab array. This array consisted of 6 wings utilizing honeycomb rigid panels and standard N/P silicon cells. Data taken in April 1978 indicates that since its launch in 1973, the electrical performance has degraded only slightly, probably less than 10%. This array was rather heavy in comparison to the lightweight flexible designs, but it has produced probably the cheapest energy yet delivered in space at something just over $1000 per kilowatt hour.

Aside from Skylab, the NASA solar array development in recent years has centered primarily around electric propulsion and deep space application. An example is the 200-W/kg Solar Array. An artist's concept of this array is shown in Figure 3. The solar array blanket design uses 2 x 2 cm solar cells 50u thick (2 mils). The cells are interconnected with Invar interconnects. Attachment to the interconnects is by welding. The welded assemblies are bonded to the Kapton substrate (1 mil) and the modules are encapsulated with RTV 655 plastic encapsulant 3 mils thick. The 80-cell modules which have been fabricated and tested to prove the blanket technology weigh 12.64 grams. The beginning-of-life power output at AMO, 28°C is 4.72 watts for a specific power of 373 W/kg.

The SEPS solar array shown in Figure 4 was developed specifically for the Solar Electric Propulsion stage concept. A single SEPS array wing is rated at 12.5 kW and measures 4m x 32m. It will use a quarter of a million N on P solar cells. The cell blanket consists of a printed circuit flexible solar array substrate which is a lamination of two sheets of 1/2-mil kapton with 1/2 mil of high temperature polyester adhesive. The interconnect is etched 1-oz. copper. Parallel gap welding is used to bond the 2 x 4 cm solar cell assemblies to the substrate. A cell assembly is composed of an 8-mil wraparound contact, 2-ohm-cm base resistivity solar cell with a 6-mil fused silica cover with ceria-stabilized glass as an alternate. An electrical module is 306 cells in series and 5 cells in parallel (1530 cells) and there are 82 modules in an array wing (2 modules/panel). The V_{mp} at 1 A.U., 55°C is 125 VDC. The harness is a flat conductor cable assembly attached to the two long edges of the wing on the back of the blanket. The conductors are 3-mil-thick aluminum of widths in the range of 0.050 to 0.25 in. to control voltage drops. The SEPS specific power rating is 66 W/kg.

Zero-g deployment of a portion of the SEPS array has been accomplished by way of KC-135 flights. Future plans for this technology include a Shuttle flight test on an early mission (Figure 5), hopefully on one of the first test flights. This array is also the baseline for the NASA 25-kW Power Module project and is being considered for use on the ENCKE Comet Rendezvous Solar Electric Propulsion program.

TECHNOLOGY NEEDS

Having briefly reviewed the state-of-the-art in solar arrays, the question which naturally arises is "What next?" In what areas are advancements in array technology needed? A partial answer to this question can be found by an examination of the types of missions which are to one extent or the other being considered for the future. This provides only a partial answer to the question for two reasons: (1) At any given time it is very difficult to compile with confidence an "official" listing of future space missions. This is a problem inherent in the operation of a Government Agency whose budget is directly dependent upon Congressional decisions; (2) The technology itself can be the driver. That is, missions which were not previously possible can be made possible by technology advancements, particularly in an area as important to a mission as space power. To a lesser extent, technology advancements can also change a mission or class of mission which were previously marginal or undesirable into very practical missions. An example is seen in the area of many science-oriented missions which are relatively short in duration ranging up to perhaps 5 to 8 years maximum. These missions must therefore pay the penalty of an expensive short-life system. Technology advances particularly in the area of cost reduction could change this situation for the better, making such missions more viable.

Although we must understand the above limitations to determine the direction which future array technology should take, this should not prevent us from utilizing what information we do have. For instance, the Space Shuttle era allows us to expand our thinking into other areas. We no longer need to think of a solar array as inaccessible after launch. The availability of the Shuttle allows us to consider some sort of on-orbit maintainability. This might take any one of a number of forms, for instance replacement could be accomplished at a component (cell, etc.), module, panel or array level. Considerations in the determination of the direction and/or desirability of on-orbit maintainability would involve as a minimum the following: (1) The desirability to limit the number of Shuttle launches required for replacement; (2) the desirability to limit the number and complexity of astronaut operations; (3) the desirability to limit replacement to only that fraction of the system which is life limited; (4) How will an on-orbit maintainability requirement impact the original array design? The above are only a few of the areas of consideration which will need to be investigated relative to on-orbit maintainability. Logic would seem to dictate that replacement should be made at the highest possible level; however, the overriding consideration is reduction in life cycle costs, and the area of on-orbit maintainability should be thoroughly investigated to determine the benefit in reducing array costs.

Advances in solar cells themselves provide us with more options than were prevously available. A good example is the recent coming to the forefront of cells such as GaAs whose capability to operate at reasonable efficiencies at high temperatures make concentrator arrays look more promising.

Along with the opportunities offered by the advent of high-flux cells and concentrators, however, come some problems. Since high temperatures are generated in concentrator array systems, materials to withstand high temperatures as well as methods for taking excess heat from the cell will be needed.

Another array-allied area which has impact upon array design is in power conditioning. One area needing investigation is chopping the array DC power into square-wave AC power for ease of voltage transformation and transmission. Also, an extensive investigation into allowable maximum operating voltages and optimum operating voltages is needed. Lower converter voltages may be necessary to keep down corona losses whereas higher voltages for transmission may be desirable.

From the present vantage point, one thing seems clear when future arrays are considered: they will be bigger. In the past, array advancements were dominated by weight reduction; in the future, they must be dominated by reduction in cost. The current costs of $1,000 to $10,000 per kilowatt hour of energy in space are unacceptable for future arrays, probably by an order of magnitude. The increase in overall array size and the present trend toward restrictively low budgets underscore the need to direct future Solar Array Technology toward the goal of lower life cycle costs.

Acknowledgements: Assistance in the preparation of this paper by Mr. Ernest Costogue of Jet Propulsion Laboratory and Mr. Dave Massey of Wright-Patterson Air Force Base is gratefully acknowledged.

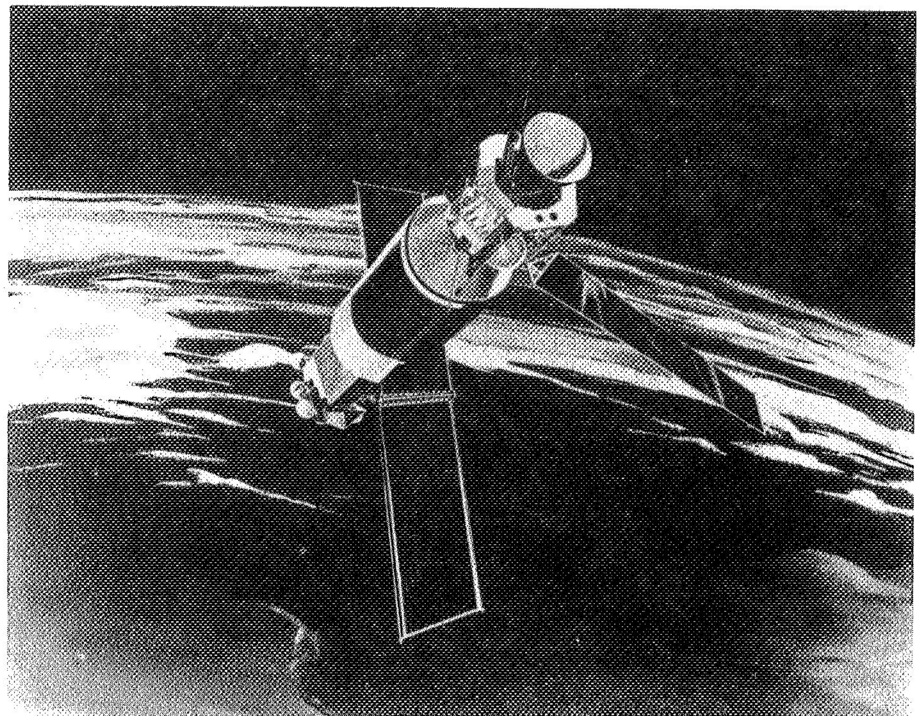


FIGURE 1. FRUSA SOLAR ARRAY

HARDENED SOLAR POWER SYSTEM (HASPS)

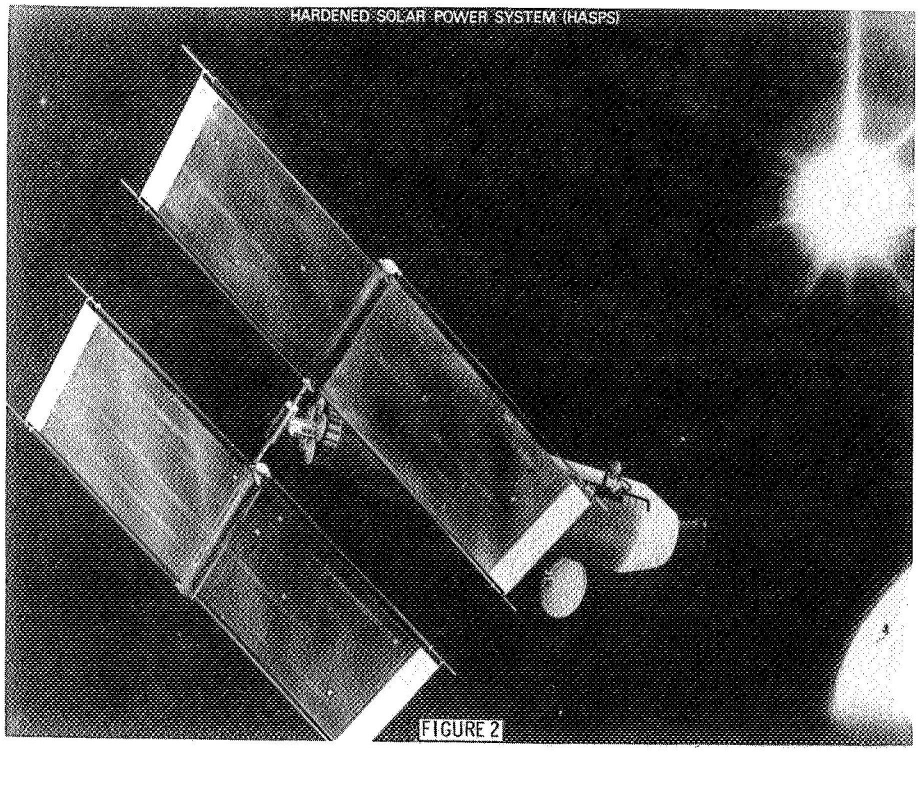
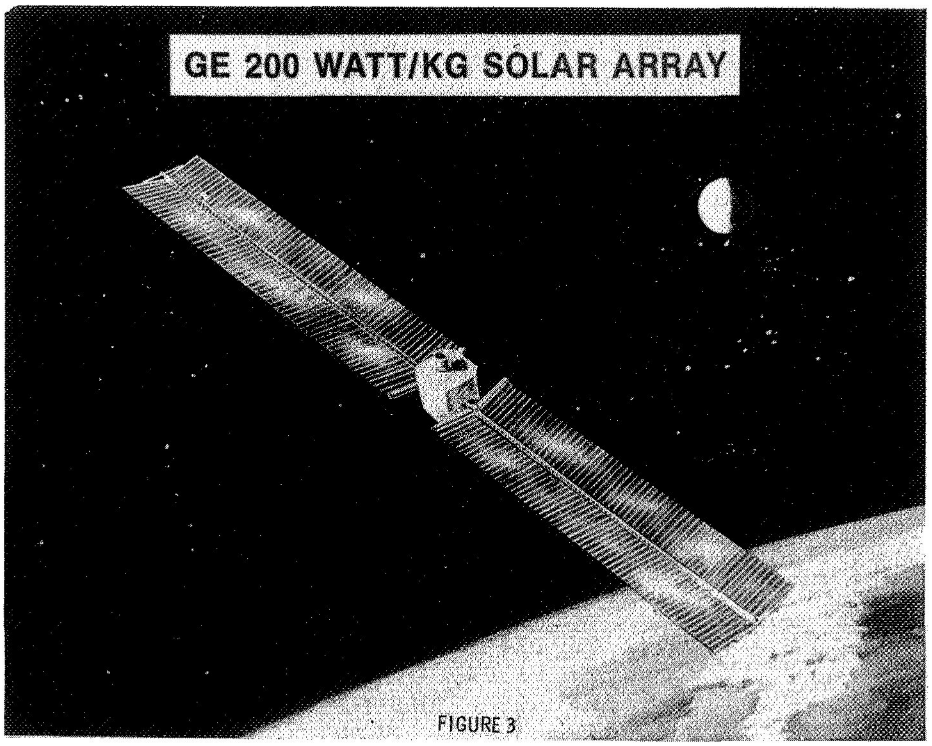


FIGURE 2


GE 200 WATT/KG SOLAR ARRAY
FIGURE 3

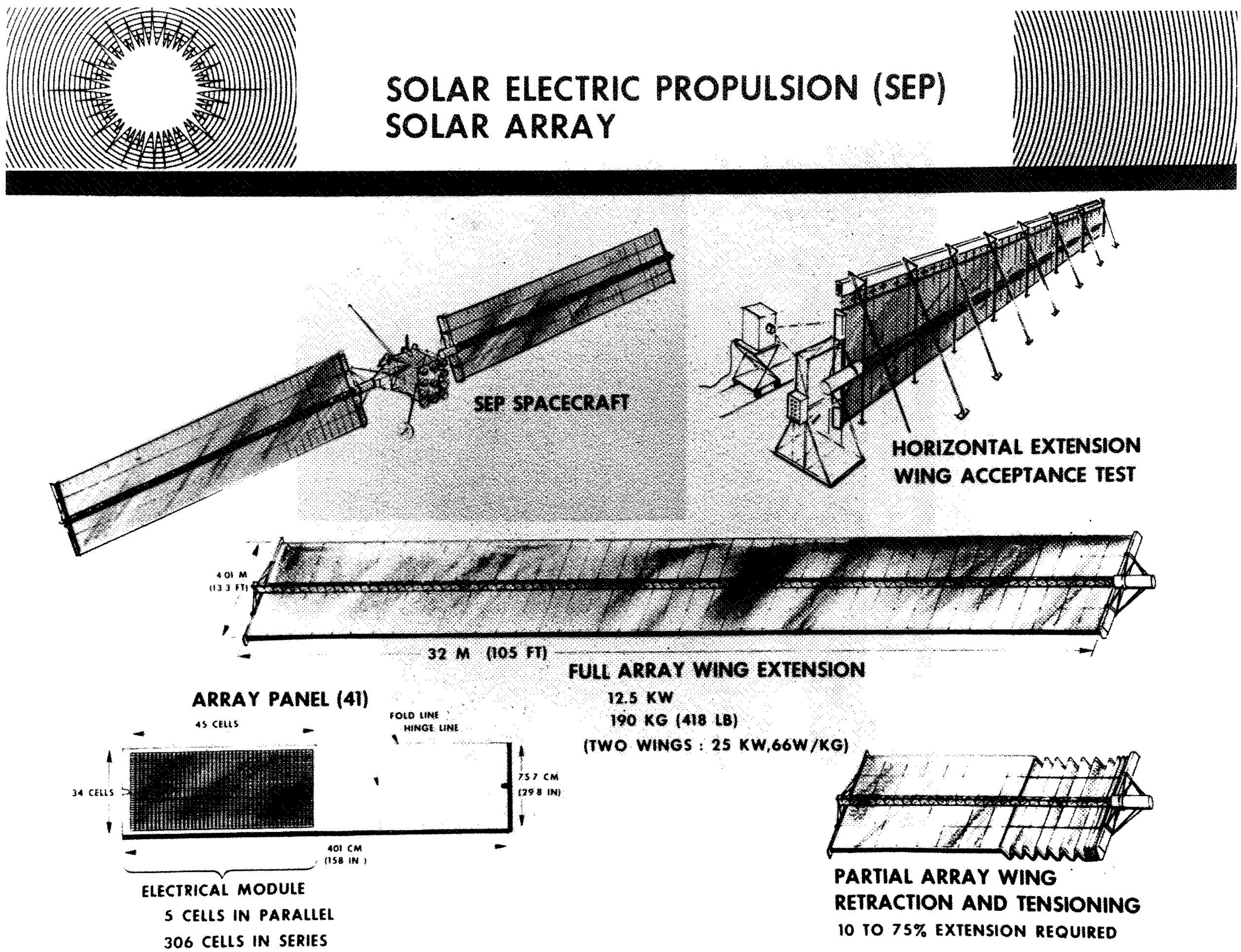

SOLAR ELECTRIC PROPULSION (SEP)
SOLAR ARRAY
SEP SPACECRAFT
HORIZONTAL EXTENSION
WING ACCEPTANCE TEST
4.01 M
(13.3 FT)
32 M (105 FT)
FULL ARRAY WING EXTENSION
12.5 KW
190 KG (418 LB)
(TWO WINGS : 25 KW,66W/KG)
ARRAY PANEL (41)
45 CELLS
FOLD LINE
HINGE LINE
34 CELLS
757 CM
(29.8 IN)
401 CM
(158 IN)
ELECTRICAL MODULE
5 CELLS IN PARALLEL
306 CELLS IN SERIES
PARTIAL ARRAY WING
RETRACTION AND TENSIONING
10 TO 75% EXTENSION REQUIRED

FIGURE 4

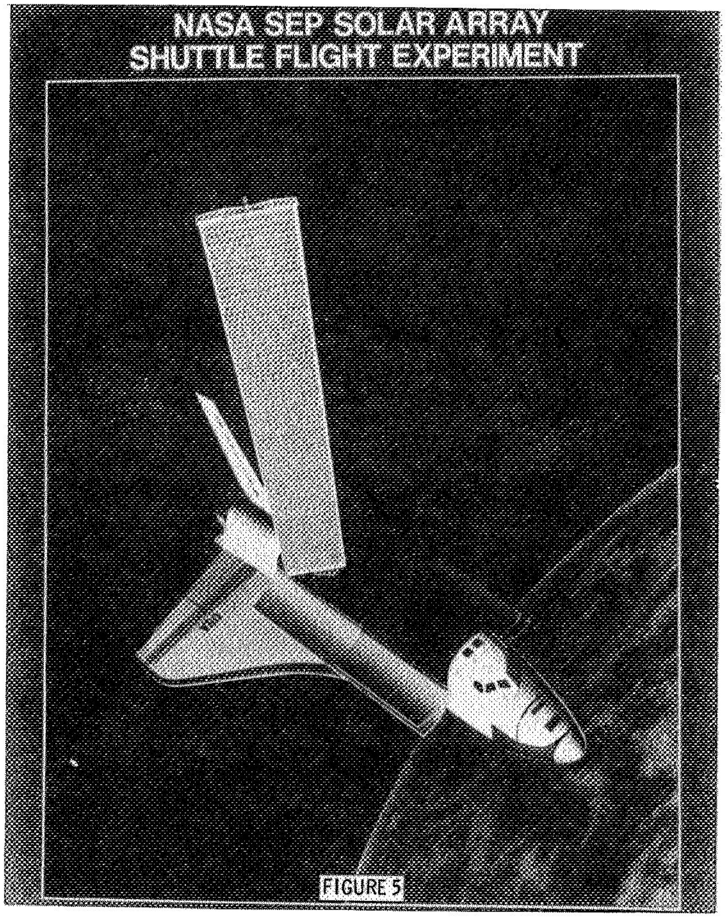


FIGURE 5

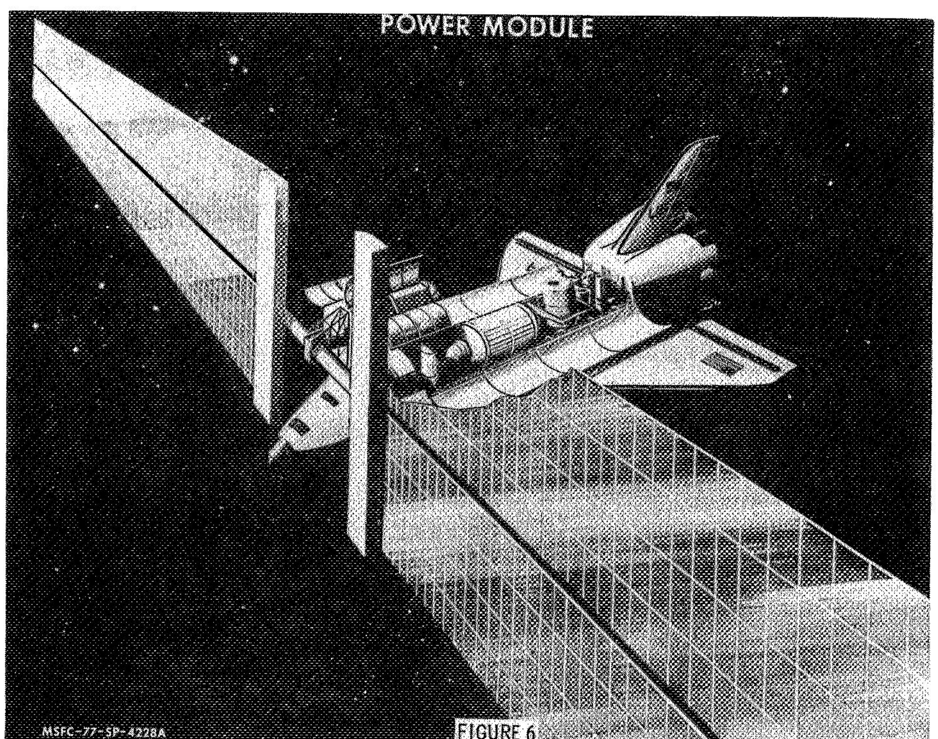


FIGURE 6

TECHNOLOGY STATUS — BATTERIES AND FUEL CELLS

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SUMMARY

The current status of research and development programs on batteries and fuel cells and the technology goals being pursued are discussed. Emphasis is placed upon those technologies relevant to earth orbital electric energy storage applications.

INTRODUCTION

Discussions of orbital energy storage usually have concentrated upon the details of the nickel-cadmium battery system. Even though tried and true, this system has limited capability and we must look to alternatives, especially with the growth in projected power levels and life for orbiting systems. This paper attempts to give a brief overview of the electrical storage technologies that are essential to those missions. The technology, the approaches being taken, and their current status are summarized in the figures, which are copies of the VU-graphs used in the oral presentation. Rechargeable battery technology will be reviewed first, followed by fuel cell technology. Fuel cells form one part of the fuel cell-electrolyzer system, which has promise for very large orbital storage application. The paper which follows this one will discuss these particular systems in more detail.

SECONDARY-BATTERY TECHNOLOGIES

The technology objectives for secondary batteries for orbital applications are long cycle and calendar life, high energy density, efficiency, reliability, and low cost. Advances are achieved through technology programs in cell components (separators, electrodes, etc.), materials (which, common to many energy technologies, are often critical to advancement), thermal management, designs, operating techniques (to provide optimal conditions and reconditioning possibilities) and test-evaluation procedures which permit valid, rapid verification of new concepts. Entirely new systems must be explored and defined to continue the evolutionary process. Kerr and Pickett (ref. 1) recently examined space battery technology for the 1980's. Figure 1 summarizes the current performance of orbital storage batteries against the demanding requirements of typical low earth orbit (LEO) and synchronous (SYNC) orbit applications. The nickel cadmium system is used. Further improvement towards the "ultimate" nickel-cadmium battery is being sought in the joint Jet Propulsion Laboratory-NASA Lewis program with the cooperation of Goddard Space Flight Center and the Air Force. This program has been described (ref. 2) and is summarized in figure 2. A prototype cell is to be demonstrated by October 1981. Multikilowatt applications require larger cell sizes in excess of 100 ampere hours (AH) that can be actively cooled. As shown in figure 2, a new toroidal construction is the subject of a current feasibility study. The projected energy density and life for the advanced nickel-cadmium technology in LEO and SYNC in comparison with the state-of-the-art are shown in figure 3.

A higher energy alkaline battery, the long-life rechargeable silver-zinc system made possible by the inorganic/organic (I/O) separator (ref. 3), is summarized in figure 4. It should be noted here that operating procedures can influence the operational life exemplified by the increased life for discharged versus float stand in a real-time SYNC orbit test. Single-cell protection is also beneficial and now is ready for application.

One of the most important new batteries for space application is nickel-hydrogen, being developed primarily by Comsat (for SYNC) and the Air Force (for SYNC and LEO). Figure 5 provides a summary. It tolerates deep discharge and has intrinsically long-cycle-life capability. Electrolyte management in LEO has been a difficult problem, now apparently under control, but test data are very limited and the full energy-density potential has not been demonstrated. Recent reports (refs. 1,4) provide details of the technology.

Another metal-hydrogen system tolerant to deep discharge, namely silver-hydrogen, looks promising for long-life SYNC application. Figure 6 provides a summary. Its success is dependent upon the NASA I/O separator. Wet stand tests are just beginning, but based upon studies of the silver electrodes in many sealed silver-zinc cells, only about 3 percent loss of capacity should be expected per year in silver-hydrogen. It should be noted that in silver-hydrogen, the silver electrode remains intact and stable, quite different from silver-zinc. To obtain the energy-density goals it has been necessary to increase the utilization of the silver electrode to 75 percent. This has been achieved. The European Space Agency has a prime interest in this system.

The promises of substantially increased energy-density systems in the >100-watt-hour-per-pound range have motivated much research and development over the past 20 years. Based on alkali-metal negatives, these systems are currently receiving major attention by the Department of Energy (DOE) and the Electric Power Research Institute (EPRI) for electric vehicle and load-leveling applications, particularly the high-temperature lithium-iron sulfide and sodium-sulfur systems. The potential application of these systems to space has been discussed (refs. 1,5). More recent data (refs. 6,7,8) are summarized in figure 7. Generally, little experience is available in multicell batteries with these systems in the United States, though European groups seem to be advanced. The cycle life and demonstrated cell energy density are well below the technical goals at present. With the level of effort these systems are receiving, the technology should have an adequate chance to prove itself over the next few years if it is ever going to do so. If it does, space applications may benefit.

The ambient to 150° C systems depend upon either solid lithium or liquid sodium contained in the sodium-ion-conducting ceramic, beta alumina. The former negative electrode does not seem to hold much promise for long cycle life needed for space, the latter does. Positives considered for combination with these negatives are shown in figure 7. The most exciting prospects appear to lie with the layered compounds that intercalate large concentrations of alkali-metal ion with no structural change. Investigations are well along based on lithium; work is just beginning with the sodium system (ref. 9).

In conclusion, nickel-cadmium improvements will maintain its preeminent position. Metal-hydrogen cells are here for SYNC orbit evaluation. Higher energy systems still are in their infancy but some may emerge within 3-5 years. Battery component technology and exploratory work must continue in order to meet the increasing demands of orbital electrical storage requirements.

FUEL CELLS FOR SPACE

The technology objectives for space fuel cells are long life, high specific power, reliability, maintainability and low cost. Technology programs lead to advances through cell components, materials and catalysts, thermal management, designs, controls and ancillaries. New approaches and systems concepts can make important contributions. Figure 8 summarizes the major NASA fuel cell applications, starting with Apollo to the Shuttle Orbiter and looking to the future. Two technologies, the solid polymer electrolyte (SPE) acid system and the matrix aqueous alkaline system, based on hydrogen and oxygen reactants, are available (ref. 10). Over the years since Apollo, the alkaline technology has experienced order-of-magnitude improvement in specific power, cost and system endurance. Figure 9 compares cell weights. The so-called lightweight technology represents the next step beyond the Orbiter and is based on the compact lightweight constructional approach shown in figure 10 (ref. 11). It is fair to characterize the alkaline system as having high performance with limited life and the SPE acid system as having modest performance with long life. Technology efforts have focused on the deficiencies and today the two technologies are converging (ref. 12). This is illustrated by the two plots in figure 11. The current space fuel cell efforts are summarized in figure 12.

For completeness, because of the very large technology and engineering development and demonstration efforts of DOE, EPRI, and the Gas Research Institute (GRI), the emerging fuel cell technologies (ref. 13) for commercial application in the time frames indicated are shown in figure 13 with a few pertinent comments regarding their space applicability. There seems to be little benefit associated with using phosphoric acid systems in space, aside from a potential cost advantage but at a performance penalty relative to the space fuel cells. The high efficiencies and high reject temperatures of molten carbonate and solid oxide systems may offer future benefits for space when the technologies mature.

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ORBIT	REQUIREMENT	CURRENT PERFORMANCE
LOW EARTH (LEO)	~6000 CYCLES/YEAR ~35 MIN. DISCHARGE ~55 MIN. CHARGE	Ni-Cd, 2-4 WHR/LB. < 30,000 CYCLES < 5 YEARS
SYNCHRONOUS (SYNC)	~84 CYCLES/YR (2 ECLIPSES) WET STAND MAX. 1.2 HR. DISCHARGE ~6.8 HR. CHARGE	Ni-Cd, 5 WHR/LB. ~ 300 CYCLES 3.5 YEARS

Figure 1. - Current orbital storage batteries.

NICKEL-CADMIUM GOALS	APPROACH/STATUS
13 WHR/LB, 30,000 CYCLES 5 YRS (LEO) 13 WHR/LB, 900 CYCLES 10 YRS (SYNC)	o LIGHTWEIGHT COMPONENTS: CASE-HEADERS, NON-SINTERED AND POROUS PLASTIC PLATED PLAQUE, ELECTROCHEMICALLY IMPREGNATED PLAQUE, OPTIMIZE PRECHARGE AND REDUCE XS NEGATIVE MASS o NEW SEPARATORS: INORGANIC/ORGANIC (NEW TYPE), PBI, AMERACE o DEEP DISCHARGE RECONDITION: HIGH RATE RECOMBINATION OF H_2? o PROTOTYPE CELL DEMO BY 9/81 o MUCH PROGRESS MADE IN REDUCING WEIGHT FOR APPLICATIONS, E.G. NATO III
MULTIKILOWATT, >100 AH, LOW COST MAINTAINABLE (LEO)	o NEW TOROIDAL CONSTRUCTION FOR HEAT MANAGEMENT, REDUCED PARTS, SIMPLE ASSEMBLY o FEASIBILITY STUDY BEGINNING WITH FABRICATION OF MODEL CELL

Figure 2. - Secondary-battery technology - nickel cadmium.

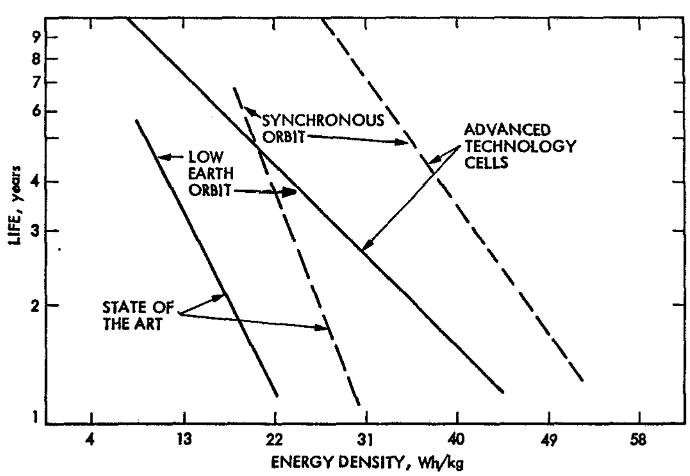


Figure 3. - Life as a function of energy density (assumes optimum temperature of 0° to 10° C).

SILVER-ZINC

GOAL	APPROACH/STATUS
24 WHR/LB, 450 CYCLES 5 YRS (SYNC)	o SEALED 40 AH; 12 AH CELLS DEVELOPED o INORGANIC/ORGANIC SEPARATOR (I/O) o 24 WHR/LB ACHIEVED IN PRODUCTION CELLS o REAL TIME SYNC TESTS: 60% DOD MAXIMUM 5 ECLIPSES (2.5 YRS) 210 CYC. FLOAT STAND 9 ECLIPSES (4.5 YRS) 380 CYC. DISCHARGED STAND o 40% DOD 1 CYCLE/DAY 1.2 HR DISCHARGE 450 CYCLES o SINGLE CELL PROTECTOR TECHNOLOGY DEVELOPED/TESTED ON 28V, 40 AH BATTERY: 60% IMPROVEMENT IN BATTERY LIFE OVER BATTERY LEVEL CONTROL; GENERALLY APPLICABLE AND AVAILABLE.

Figure 4. - Secondary-battery technology - silver-zinc.

NICKEL-HYDROGEN

GOALS	APPROACH/STATUS
22-25 WHR/LB, 900 CYCLES 10 YRS (SYNC)	o "PINEAPPLE SLICE" STACK CONSTRUCTION o NYLON, POLYPROPYLENE AND INORGANIC SEPARATORS USED
6000 CYCLES, 1 YR AND 30,000 CYCLES, 5 YRS (LEO)	o IMPROVED THERMAL, OXYGEN AND ELECTROLYTE MANAGEMENT NEEDED IN LEO o VOLUME ~1.5 - 2 X NI-CD o TEST DATA LIMITED, ~15 WHR/LB >650 CYCLES ACHIEVED IN LAB SYNC TEST o TECHNOLOGY AVAILABLE FOR EVALUATION o FLIGHT TEST OF NON-OPTIMIZED PROTOTYPE ON NTS-2 (8 WHR/LB) - RESULTS GOOD TO DATE, GOOD TEST FOR SYNC o AF FLIGHT TEST FOR LEO, PIGGY BACK EXPT. o COULD HAVE COMMERCIAL APPLICATION THUS REDUCED COST

Figure 5. - Secondary-battery technology - nickel-hydrogen.

SILVER-HYDROGEN

GOALS	APPROACH/STATUS
30 WHR/LB. 900 CYCLES >10 YRS (SYNC)	o STACK CONSTRUCTION - SLICES OR ROLLED, NOVEL USE OF HEAT PIPE CONSIDERED o I/O SEPARATOR, OTHERS UNSUITABLE
1 YR, HIGH RATE (LEO)	o STRESS ELECTROLYTE MANAGEMENT TECHNIQUES - ELECTROLYTE RESERVOIR PRINCIPLES o ~75% UTILIZATION AG ELECTRODE ACHIEVED o 20 AH, 30 WHR/LB CELLS >900 CYCLES @ 75% DEPTH 1.2 HR DISCHARGE, WET STAND TESTS BEGUN o SILVER ELECTRODE REMAINS STABLE INTACT; (DIFFERENT THAN AG-ZN), NO H_2 ELECTRODE POISONING o 50 AH >35 WHR/LB AEROSPACE WEIGHT CELLS READY 1979; USER EVALUATION NEEDED

Figure 6. - Secondary-battery technology - silver-hydrogen.

HIGH ENERGY DENSITY SYSTEMS $>$100 WHR/LB

- o ALKALI METAL NEGATIVES (LI, NA)
- o AEROSPACE/DOD GOALS PUSHED EARLY TECHNOLOGY WORK BEGINNING IN LATE 50s ⟶
- o ERDA/DOE - EPRI PROGRAMS NOW VERY LARGE FOR COMMERCIAL ELECTRIC VEHICLE AND LOAD-LEVELING APPLICATIONS (\$ ~10M/YR)

A. HIGH TEMPERATURE (350°-450° C)

POSSIBLE BENEFIT FOR SPACE: EVENTUAL COST \$40/KWHR, REJECT TEMP., HIGHER ENERGY DENSITY

1. LITHIUM $\binom{AL}{SI}$ - IRON SULFIDE

GOAL: 85 WHR/LB, >1000 CYCLES
~10 YR

STATUS:
- o 50 AH CELLS <35 WHR/LB
  >500 CYCLES, 4 HR RATE
- o 2 CELL BATTERY <20 WHR/LB
  ~600 CYCLES
- o LAB CELLS >1000 CYCLES;
  60 WHR/LB, 33 CYCLES
- o LARGE BATTERY TEST IN
  VEHICLE LATE 1978

2. SODIUM-SULFUR WITH SOLID ELECTROLYTE

GOAL: 100-120 WHR/LB, 2500 CYCLES
(>10 YR) - HIGH ENERGY CELL

STATUS:

BETA-ALUMINA TYPE

75 AH CELLS: 55 WHR/LB; 170 CYCLES
MFG 1. ⊕ ELECTRODE, CONTAINER PROBLEMS
MFG 2. ⊕ PROBLEM SOLVED; 300 CYCLES
CHROME-ALLOY MILD STEEL CONTAINER
FOREIGN TECHNOLOGY MORE ADVANCED

GLASS CAPILLARY FIBER TYPE

6AH CELLS: 60 WHR/LB; 217 CYCLES
GOOD TUBESHEET - FIBER INTERFACE
MANUFACTURABILITY GOOD
~\$25/KWHR
SMALL CELLS 3800 CYCLES

B. ORGANIC ELECTROLYTE (AMBIENT TO 150° C)

- o LITHIUM (SOLID) OR SODIUM (LIQUID)/SOLID ELECTROLYTE (>100° C) FOR NEGATIVE ELECTRODES
  - LITHIUM RECHARGEABILITY LIMITED TO ~100 CYCLES AT PRESENT; SOME REPORTS OF BREAKTHROUGH TO 1000s OF CYCLES
  - SODIUM (LIQ)//BETA ALUMINA - MANY THOUSANDS OF DEEP CYCLES DEMONSTRATED
- o FOR POSITIVE ELECTRODES
  DISSOLVED SULFUR OR SOLID INSOLUBLE TRANSITION METAL DICHALCOGENIDE OPERATING IN ORGANIC ELECTROLYTES ARE MOST PROMISING CONCEPTS FOR GOOD RATES AND LIFE
  - o STABILITY OF CONCENTRATED ALKALI POLYSULFIDES IN ORGANICS APPEAR LIMITED FOR LONG LIFE APPLICATIONS
  - o TRANSITION METAL DICHALCOGENIDES HAVE LAYERED STRUCTURES ACCEPTING LARGE CONCENTRATION OF ALKALI IONS WITHOUT STRUCTURAL CHANGE IN WHICH ALKALI IONS DIFFUSE RAPIDLY. EXCITING OPPORTUNITIES; A FEW EXAMPLES.

Li / VARIOUS ELECTROLYTES / Li_xTiS_2 $<100\ \frac{WHR}{LB}$ >250 CYCLES 10 MA/CM^2

o MAJOR DEVELOPMENT EFFORT IN PROGRESS

Li / ELECTROLYTE / $Li_xCr_yV_zS_2$ $\approx 115\ \frac{WHR}{LB}$ RATE?

Na_L / BETA ALUMINA / ORGANIC ELECTROLYTE / $Na_xNi\ PS_3$ $\approx 125\ \frac{WHR}{LB}$ PRACTICAL ESTIMATE

o EXPLORATORY WORK BEGINNING

Figure 7. - Secondary-battery technology - high-energy-density systems.

PAST:

BIOSATELLITE (SPE)
GEMINI (SPE)
APOLLO (1.5 KW, 1962, ALKALINE)

PRESENT:

SHUTTLE ORBITER (12 KW, 1973, ALKALINE)

FUTURE:

ORBITAL TRANSFER VEHICLE (OTV)
SPACE BASE - SOLAR PHOTOVOLTAIC ENERGY STORAGE (WITH ELECTROLYZER)
SHUTTLE DERIVATIVES

Figure 8. - Major NASA fuel cell applications.

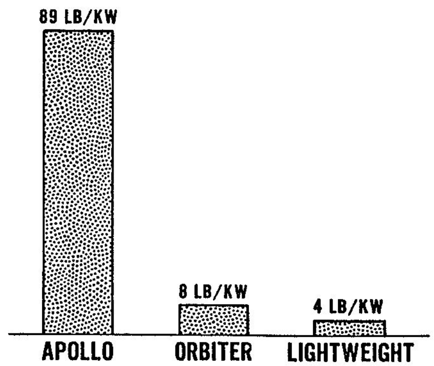


Figure 9. - Cell weight comparison.

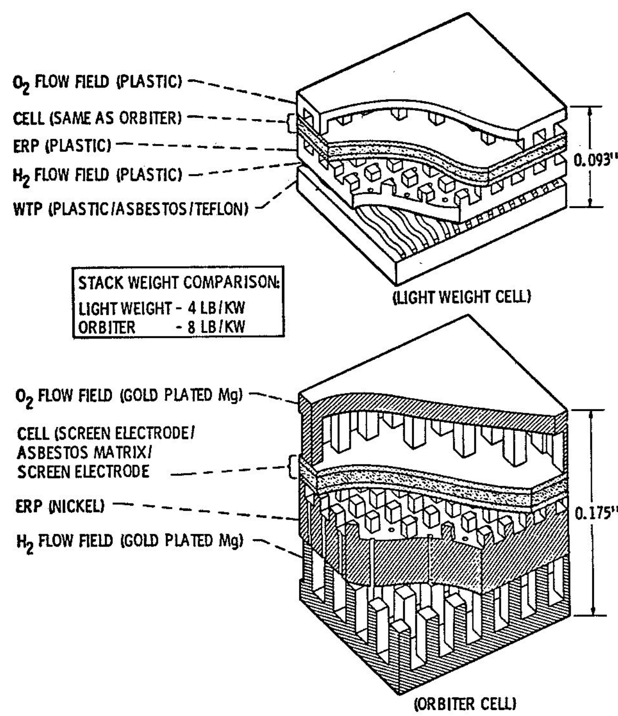
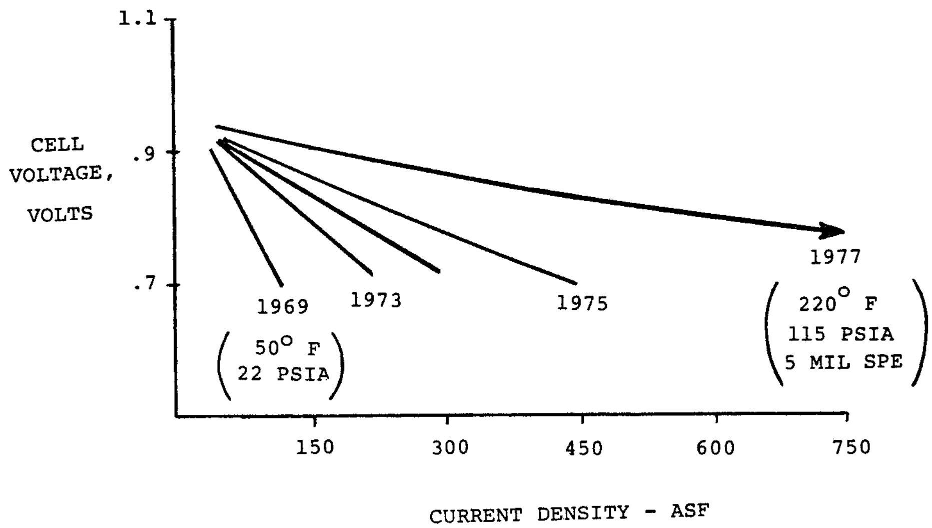
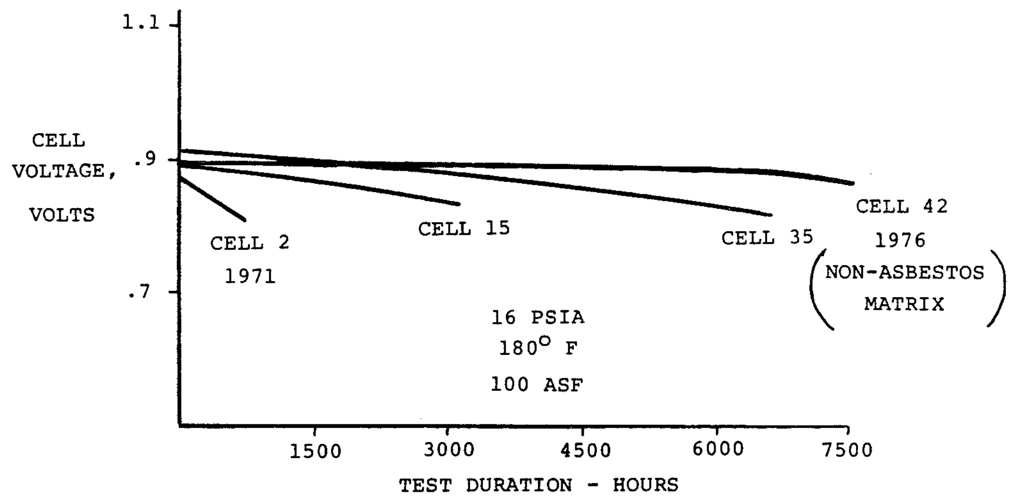


Figure 10. - Structural comparison: lightweight fuel cell versus orbiter fuel cell.



(a) Acid SPE fuel cells.



(b) Lightweight alkaline fuel cells.

Figure 11. - Improvements in performance of acid SPE and lightweight alkaline fuel cells.

ALKALINE - H_2/O_2

- o 20 LB/KW, 10,000 HR @ 200 ASF, 185° F USING PROPELLANT GRADE REACTANTS DEMONSTRATED AT CELL LEVEL
  - POWER PLANT DEVELOPMENT BEGUN
- o 20 LB/KW, >40,000 HR, 0.91V @ 200 ASF, 180° F CELL DEVELOPMENT BEGUN
  - LIGHTWEIGHT OXIDATION-RESISTANT MATERIALS DEFINED
  - NEW ELECTROCATALYSTS INCL. SUPPORTED COMMERCIAL TYPE
  - NEW PBI MATRIX
- o 6 LB/KW, >3000 HR, 0.9V @ 1000 ASF, 250-300° F USING PROPELLANT GRADE REACTANTS
  - AS ABOVE, SAME MATERIALS REQUIREMENTS
  - RESEARCH CELL ACHIEVED >0.90V @ 1000 ASF FOR 16 HRS
  - HIGH CURRENT DENSITY REDUCES COST

ACID (SPE) - H_2/O_2

- o 15-20 LB/KW, >40,000 HR, 0.95 @ 120 ASF, 180° -220° F, BREADBOARD SYSTEM BEING BUILT FOR TESTING
  - HUMIDIFICATION OF REACTANTS IS KEY TO PERFORMANCE
  - ENDURANCED ACHIEVED IN CELLS

Figure 12. - Space fuel cell technology thrust.

COMMERCIAL SYSTEMS - LOW COST ~$300/KW (DOE-EPRI-GRI, ~$40M/YR AND GROWING)

PHOSPHORIC ACID (1980-1985)

- LIQUID SEALS A PROBLEM FOR SPACE
- INEFFICIENT RELATIVE TO ALK. OR SPE; 190° C
- ENDURANCE @ 300 ASF SUFFICIENT FOR 40,000 HR SYSTEM (PROVIDING ACID INVENTORY CAN BE MAINTAINED)

MOLTEN CARBONATE (1990)

- 650° C TO OPERATE; EFFICIENT
- CELL ENDURANCE DEMONSTRATED TO 13,000 HRS @ 100 ASF
- HIGH REJECT TEMP.
- MAINTAINING TEMP. IN SPACE?

SOLID OXIDE (2000)

- 1000° C OPERATE; EFFICIENT
- CELL ENDURANCE DEMONSTRATED TO ~40,000 HRS @ 120 ASF
- CELL INTERCONNECT PROBLEM SOLVED
- ALL SOLID STATE SYSTEM
- HIGH REJECT TEMP.
- MAINTAINING TEMP. IN SPACE?

Figure 13. - Possible future fuel cell technology opportunities in space.

TECHNOLOGY STATUS –

FUEL CELLS AND ELECTROLYSIS CELLS

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SUMMARY

Electrochemical technology as it applies to fuel cells and electrolysis cells has been active for sixteen years. This activity has provided the basis for electrical power systems for three successful space flight programs plus a fourth program which is in the final stages of vehicle development testing. The technology has matured from less than 100 hours test operational capability in the beginning to systems operational capability of several thousand hours. Future applications for this technology include large orbital energy conversion and storage facilities in the multi-hundred kilowatt range.

INTRODUCTION

Since the selection of H_2/O_2 fuel cells as the primary electrical power systems for the Gemini and Apollo programs in March 1962, phenomenal strides have been made in the technology. For example, a comparison of the Apollo fuel cell with the Shuttle fuel cell at baselining shows the Shuttle unit at equivalent weight produces eight times as much power for six times as long and is at least an order of magnitude easier to start and stop. Since Shuttle baselining in 1973, additional capability has been achieved in this technology.

There is little doubt that, except for the space program, fuel cells would still be more or less a laboratory curiosity, an oddity seemingly holding the potential of alleviating many energy conversion limitations. The urgency of the space program and particularly the limited lift capability of rockets being dealt with at the time, provided the impetus required to launch the vigorous technology and development programs that brought the concept into fruition. Except for fuel cell electrical power, the Gemini flights using batteries would have been limited to about four days. The Apollo mission as performed would not have been possible on a battery system, even with the 3.4 $\overline{M}$ kg (7.5 $\overline{M}$ lbs) thrust of the Saturn V launch vehicles.

This paper will present the status of the baselined Shuttle fuel cell as well as the acid membrane fuel cell and space-oriented water electrolysis technologies. The more recent advances in the alkaline fuel cell technology area are the subject of a companion paper. A preliminary plan for the focusing of these technologies towards regenerative energy storage applications in the multi-hundred kilowatt range will also be discussed.

BACKGROUND

Figure 1 graphically illustrates the Agency background in fuel cell development and applications.

In 1962, contracts were awarded for development of fuel cell electrical power systems for both the Gemini and Apollo programs. At the time of baselining these fuel cells, very limited test operating experience existed.

Gemini

Figure 2 is a picture of the Gemini fuel cell.

One of the fundamental objectives of the Gemini program was to gain long-duration experience in space as a precursor to the Apollo mission. To perform missions beyond approximately four days required a more energy-dense electrical power system than the conventional battery system. Thus the driver that would bring fuel cells into practical reality appeared. The ion-exchange membrane concept indicated the lowest potential weight of all concepts considered, and it operated at relatively low temperature [237-310°K (75-100°F)] and pressure [138 kPa (20 psia)], conditions quite compatible with the low-Earth-orbit environment. This program spanned four years with the flight program completed in February 1966.

Development of the Gemini fuel cell (fig. 2) was highly constrained by the rigorous mission schedule. The technology was in its embryonic stages and the time for the technical thoroughness required for methodical development was an unavailable commodity. Because of this, compromises relative to specification performance were dictated. While all of the electrical requirements of all flights were met, a higher degree of energy austerity was exercised than planned. But the most significant compromise was the non-usability of the fuel cell product water. This was due to a membrane degradation phenomenon which contributed impurities to the product water rendering it unacceptable under the rigid dietary limitations of the astronauts. This degradation was the principal performance and life-limiting issue of this fuel cell and was not understood until after the Gemini program.

Apollo

In its initial conception the Apollo Command/Service vehicle was to land on the surface of the moon. This required the power system to reject heat to a 394°K (250°F) environment which dictates an operating temperature exceeding 394°K. This requirement was a significant issue in selection of the Bacon fuel cell, which as modified operated at approximately 505°K (450°F) and 414 kPa (60 psia).

Basic development of the Apollo fuel cell (fig. 1) occurred between 1962 and 1966. It was incrementally "qualified" in 1966 for Earth-orbital manned operations, called "Block I" configuration, to 400 hours, becoming the first subsystem to be qualified for Apollo. It was subsequently qualified after additional component development for the lunar missions, called the "Block II" configuration, also to 400 hours. A "delta qual" program was conducted on the Block II configuration to 1000 hours in 1969, toward support of extended duration missions then in the planning stage using Apollo vehicles.

The fuel cell contract was completed in December 1969 with the delivery of the last of approximately 100 units to the prime contractor. At the completion of the Skylab program in 1974, Skylab IV was utilizing fuel cells with approximately five years of shelf life.

There were no major failures of fuel cells on the Apollo flights. A malfunction in the hydrogen pump electrical circuit on one module of Apollo 10 led to isolation of that fuel cell, resulting in a subsequent thermal oscillation in one of the other two modules because of the increased electrical load. These were the most severe fuel cell anomalies experienced in the entire flight program.

The success of the Gemini and Apollo programs, coupled with the technological base developed in support of those programs, has firmly established the fuel cell energy conversion concept as a viable special duty power system.

CAPILLARY MATRIX TECHNOLOGY

Because of developmental problems encountered in the early stages of both Gemini and Apollo fuel cell programs and the uncertainty of meeting program schedules with qualified hardware, backup technology development was initiated in 1964, which was identified as a "Multi-Mission Fuel Cell Development Program." In addition to the backup applications, there were other applications envisioned requiring advanced technology such as Earth-orbital long-duration space stations and extended missions up to 90 days using Apollo vehicles. The alkaline capillary matrix concept, which was first conceived and demonstrated by Mond and Langer (ref. 1) in the late 1800's, was selected for this development. Principal features leading to this selection were relatively mild temperature [355°K (180°F)], flexible operating pressure, liquid electrolyte over a broad temperature range which simplified activation, superior efficiency (over acidic), and most important, electrolyte retention by an inert capillary matrix of asbestos.

Due to the adequate and timely resolution of the problems encountered in the Gemini and Apollo programs, this technology never achieved the status or configuration of a viable alternate. It did, however, achieve baseline status in 1968/69 for both the A.F. Manned Orbital Laboratory and the "wet workshop" concept of the AAP (Apollo Applications Program). The A.F. program was canceled in May 1969. The AAP configuration was then changed to the "dry workshop" concept, which eliminated the requirement for a 2500-hour fuel cell and resulted in the cancellation of the fuel cell development program in June 1969. Figure 4 shows a final version of this effort. After cancellation, the level of effort was then cut back to technology advancement status.

The Shuttle program was taking form in the late '60's and it soon became evident that a fuel cell with even greater capability than the AAP fuel cell was needed. Thus in July 1970, as a result of a competitive solicitation, two contracts were awarded for development of the technology necessary to meet the somewhat indefinite requirements of the Shuttle: the acidic SPE (solid polymer electrolyte) fuel cell and the alkaline capillary matrix fuel cell. The approach was simply to initiate a technology race under an environment not constrained by programmatic issues, and the superior technology at the time of baselining would be selected. The programs were conducted in parallel toward similar specifications and equally funded at approximately $500K each per year for 3-1/2 years. Both technologies demonstrated 2500 hours of uninterrupted operation as a technology readiness performance test and were thus essentially declared technological equals by the prime Shuttle contractor. In August 1973, the capillary matrix fuel cell was baselined for the Shuttle.

Shuttle Fuel Cell

Figure 5 represents the baselined version and characteristics of the Shuttle fuel cell module. Figure 6 represents the cross section and gives the characteristics of the cell.

The configuration at baselining consisted of the following:

- 3, 32-cell stacks in electrical parallel
- Pt/Pd catalyzed electrodes
- 0.5 mm (.020") reconstituted asbestos matrix
- 32% (nominal) KOH electrolyte
- Au-plated Mg separator plates
- Fiberglass epoxy frame material

The operating conditions of the baselined configuration were

- 355°K (180°F) nominal
- 4 atmospheres pressure
- 44-311 mA/cm^2 (40-300 ASF)
- 27.5-32.5 volts
- 7 kW nominal power rating

Several contributions to this configuration were derived from the capillary matrix technology programs. Among the most important are

a. Reconstituted asbestos matrix electrolyte holder. The physical properties of the commercial grade matrix material were found to be of such irregular proportions that positive and uniform separation of electrodes was unreliable and that capillarity was low (low electrolyte retention ability) due to irregularly distributed, coarse fibers. Reprocessing of the commercial asbestos greatly improved these properties.

b. Matrix thickness. Earlier technology effort established a solid data base for matrix thickness of 0.75 mm (0.030"). The capillary matrix fuel cell technology contractor attempted to use 0.25 mm (0.010") thickness in order to maximize performance. When difficulties were encountered with this thickness, a compromise was made between the previous data base and the 0.25 mm (0.010") thick matrix and 0.50 mm (0.020") thickness was adopted.

c. Separator plate composition, surface finish and protective plating. Magnesium plate stock of high-purity alloy was a prerequisite to achieve the quality finish that was necessary to accept the protective gold plating with minimal imperfections.

d. Reactant flow distribution and pressure control. Reactant labyrinth flow and porting design improvements were developed to assure uniform flow across the cell and also cell to cell from the manifold.

A comparison of the Shuttle fuel cell, which was baselined in 1973, with the Apollo fuel cell, which was baselined in 1962, shows that for essentially equivalent weight, the Shuttle fuel cell produces 8 times as much power (nominally) for 6 times as long and requires only 15 minutes to start up versus 24 hours for Apollo, and shutdown is instantaneous, versus 17 hours for Apollo.

Subsequent to baselining and as a result of electrode catalyst technology effort sponsored by LeRC (Lewis Research Center), the cathode electrode catalyst was changed from a Pt/Pd mix to one of Au/Pt alloy. This change resulted in approximately 50% increase in power capability and led to a reduction of one stack per module, which yielded a 15% [55 kg (120 lbs) total] weight reduction for the power system in the spacecraft.

SOLID POLYMER ELECTROLYTE TECHNOLOGY

Figure 7 illustrates the current version and characteristics of the SPE fuel cell stack. Figure 8 illustrates a cross section of the current cell configuration and its characteristics. Figure 9 illustrates the advanced cell configuration juxtaposed for comparison with the current cell configuration.

Shortly after the completion of the Gemini fuel cell program, the principal life-limiting degradation phenomenon of the ion-exchange membrane was identified. The membrane was synthesized from a ring-structured monomer (styrene) which was attacked chemically during operation by outgassing species of the canister foam filler material. The propagation of ruptured and cleaved rings slowly deteriorated the membrane until reactant gas crossover occurred, resulting in localized catalytic H_2 and O_2 ignition, causing failure of the fuel cell. In addition, the deteriorated membrane species contaminated the product water rendering it unusable. These phenomena led to the search for and development of chemically stable, ionically conductive, polymeric materials suitable for electrolytic membrane fabrication. A sulfonated fluorocarbon with a chemical structure and stability similar to duPont Teflon eventually emerged as an acceptable material. In 1969, the Biosatellite was flown with a primate aboard which consumed the product water produced by a fuel cell utilizing this new material for over 30 days.

During the pre-Shuttle technology program of the early '70's, a series of failures occurred which threatened the objective of a long-life (2500-5000 hr), reliable SPE fuel cell. The operating conditions required [355°K, 414 kPa, 108-430 mA/cm^2 (180°F, 60 psia, 100-400 ASF)] imposed stresses on the membrane which resulted in failures seemingly similar to the Gemini failures. Through an Agency-wide "Task Force" type effort, the nature of the failures was sorted out and found to be initiated by excessive drying of the membrane by incoming reactants followed by an oxidative attack in the dehydrated zone by an intermediate, short-lived specie of reaction. Understanding the cause (but not the exact mechanism) of the failures led to a chemical treatment of the membrane to buffer the chemical attack and a design feature to eliminate the dehydrating phenomenon. This was a signal success which was verified by a 2500-hour test. This test time demonstrated a factor of 4 improvement from previously achievable results at similar stress levels.

With this milestone of achievement secured, the major limitation in operational life of the SPE fuel cell was resolved. While the membrane is not totally inert, the degradation product, F^-, (in the form of HF) is identifiable in the product water and a measure of its concentration provides a barometer for predicting useful life. Testing of latest configuration cells is producing product water with 50 to 150 PPB (parts per billion) F^-. The longest test run of any fuel cell of record anywhere is continuing beyond 48,000 hours with essentially invariant performance and producing water with 1000-2000 PPB F^-. It seems reasonable to infer, then, that the latest cells producing 50-150 PPB F^- will run 10 to 20 times as long as the cells producing 1000-2000 PPB F^- based upon membrane degradation alone. Using the 48,000-hour test as a base, the technology is now indicating a potential of operating with an indefinite life capability.

Figure 10 illustrates the performance trend of the SPE fuel cell associated with the various applicable programs from the Gemini program to the present technology program.

CONTINUING TECHNOLOGY

Since the 1973 competition for the Shuttle fuel cell program the capillary matrix technology has been advanced under the sponsorship of LeRC, and JSC has sponsored the continued advancements in the SPE technology. These continuing efforts have been justified on several accounts. Among these are satisfying the ever-increasing power demands of the Shuttle; increasing hardware life capability; and maintaining technological growth capability toward meeting the requirements of the more ambitious space missions in planning such as space construction and manufacturing operations, support of solar power satellite buildup, etc.

ELECTROLYSIS TECHNOLOGY

Figure 11 illustrates the cross section and associated reactions of an electrolysis cell along with a similar illustration of a fuel cell cross section for comparison. Also given are principal elements of the technology status. The SPE is arbitrarily used for this illustration; the alkaline cell would be similar. It is readily apparent that the fuel cell and electrolysis cell technologies are fundamentally one and the same.

Electrolysis technology supported by the Agency and managed by the Crew Systems Division of JSC has been focused on the application of the SPE fuel cell technology toward the on-board generation of H_2 and O_2 from supplied and reclaimed water for life support functions. This effort was initiated under contract in 1970 for a six-man, closed-cycle system. Following that effort a contract was awarded for the fabrication of a closed-cycle, 9 kg/day (20 lb/day) O_2 generating system called, "Space Station Prototype." This effort was completed with delivery of the system in September 1973.

In addition to the Agency support of electrolysis technology, the A.F. has supported development of high-pressure technology to produce propellants for attitude and station-keeping control in synchronous orbit. The Navy has also sponsored development of high-pressure systems toward the life-support requirements of a 140-man crew on nuclear submarines. And finally, the A.F. and Navy have jointly sponsored development of oxygen concentrator systems (a hybrid fuel cell/electrolysis cell) for use on high-flying aircraft.

Cell sizes deriving from the JSC programs range from 7.6 cm (3 in.) diameter to 16.25 cm (6.4 in.) diameter. Other government applications have utilized cells of 21.8 cm (8.6 in.) diameter and commercial applications for bulk hydrogen generation are presently working toward an initial scaled up size of 0.23 m^2 (2.5 ft^2).

Operating conditions for electrolysis cells can be quite variable over a range of temperature from 305°-428°K (90°-300°F), up to 21 MPa (3000 psia), and current density to approximately 1870 mA/cm^2 (2000 ASF). Operating voltages vary according to conditions. An illustration of cell voltage variation versus current is given in figure 12 for both electrolysis cells and fuel cells.

Electrical power systems trade and design studies have long recognized the potential of this technology in a regenerative configuration with fuel cells and photovoltaic cells as an energy conversion and storage system toward support of the longer duration, high-power missions. Electrolysis technology developers have also recognized this potential, but the driving motivation to stimulate development has not been present. With the space program outlook for the '80's and '90's and in light of the trade studies already conducted, the motivation is now available to justify initiation of the long-range technology development activity which is required to bring the concept into reality.

The principal electrolysis cell technologies currently available are direct derivatives of the fuel cell technologies which have been developed under NASA sponsorship. The acidic SPE electrolysis concept derives from the Gemini/Shuttle technology development and presently exhibits superior performance efficiency. The alkaline capillary matrix electrolysis concept derives directly from the capillary matrix fuel cell technology which was begun in the mid '60's. As a fuel cell, this concept exhibits superior efficiency to the SPE fuel cell. Thus a regenerative energy conversion and storage system with maximized performance efficiency based upon present capability and ignoring interaction phenomena of dissimilar species (acidic/alkaline) would be composed of acidic SPE electrolysis cells and alkaline capillary matrix fuel cells (fig. 13 and fig. 14). However, continued research in the catalyst, electrode, and electrolyte area could profoundly influence optimization configurations and thus it is premature to make such commitments until the superior technologies emerge through further development.

REGENERATIVE ENERGY STORAGE TRADE STUDIES

Since the early '60's several definition studies for conceptual designs of long-duration space vehicles accommodating up to 150 men have been conducted. These studies have included several power systems using either solar or nuclear energy. Indepth studies were conducted from 1969 to 1972 by Rockwell/Space Division under contract to JSC (ref. 2) and MDAC (McDonnell Douglas Astronautics Company) under contract to MSFC (Marshall Space Flight Center) (ref. 3). Both contractors baselined a photovoltaic/NiCd regenerative power system in the Phase B system definition reports. Then, under an add-on task to the Rockwell/Space Division contract, an RFC (regenerative fuel cell) trade was conducted in light of the SA/NiCd system previously baselined. Results of both trades are given in reference 2. Results of that trade are summarized in figure 15. The RFC system was baselined on the basis of that trade study which was oriented toward a 10-year life including maintenance and replacement, as required.

Following these results, two contracts were awarded to conduct indepth RFC system analyses and component designs, results of which were published as design data handbooks (ref. 4 and ref. 5). These studies honed the initial trades to identify the marginal technology areas, sensitivities, maintenance considerations, etc.

The latest power system trade study for orbiting vehicles was conducted by MDAC in 1977 (ref. 6). The results of this study are summarized in figure 16. The study included NiCd batteries, NiH_2 batteries, energy wheels, and RFC's for energy storage. The RFC system shows a weight savings of 65% compared with the NiCd system.

As a result of these study activities, a large reference base has been established for the concept of the regenerative fuel cell system as a viable energy conversion and storage concept for long-duration orbital operations.

Future mission plans for long-duration orbital operations consistently identify the viability of a fuel cell/electrolysis/photovoltaic system. The fuel cell is already firmly established as the primary power system for the Shuttle and is not likely to be superseded for this generation of logistics vehicle. Solar array augmentation for extending the Earth orbital stay time of the Shuttle is under consideration. Any logical extension of the Shuttle capabilities beyond this level will require a dedicated space power system with heavy dependence on the photovoltaic systems. The RFC fits well with the low Earth orbital requirements for energy storage.

While the RFC technology is justified solely on supplemental power system energy storage applications, it also offers many future options such as certain life support functions and propellant processing. The only resupply requirement for the logistics vehicle is water.

ENERGY CONVERSION/STORAGE TECHNOLOGY PLAN

Future missions of the variety discussed in the previous section could occur in the mid-to-late '80's. Therefore, in keeping with the practice begun and the success realized with the fuel cell technology programs preceding the Shuttle, a preliminary technology plan has been prepared that is designed to bring the technology of regenerative electrochemical cells to a state of readiness for application to those programs.

The initial task of the plan provides for a state-of-the-art assessment during the first year to be conducted by the Agency and by contract. This effort will be a thorough analysis of all aspects of the technology in order to firmly establish a set of guidelines and goals, and to define priorities required to improve the technology to the capability required over the following six years. The technology will be oriented toward establishment of a regenerative orbiting power platform of up to 500 kW_e rating. Using this as a tentative goal, a preliminary set of requirements will be derived in order to permit the technology programs to be conducted on representatively sized cells, stacks, modules, etc., and to determine flow rates, thermal loads, stress levels, and other factors which are required to conduct a well-ordered technology effort. These specifications will be updated as necessary based upon inputs from other programs dealing with requirements and sensitivities analyses of ongoing orbital operations planning.

The major thrust of the technology program will be focused on the electrochemical aspects; i.e., the electrolyte, electrode and catalyst environment because that is where the reactions occur and that is where any improvements in efficiency and life will be achieved. The merits of dual-mode cells will be evaluated during the first three years of the program. Even if final system designs should favor dedicated modules there appear to be advantages to cell commonality from a cost, manufacturing, and inventory standpoint.

A continuing cell and stack task is identified to evaluate and prove out all effort deriving from the electrochemical tasks. At this level, engineering activity becomes visible as related to the cell and stack sizing and design requirements.

At the module level, components become a factor in technology improvement, especially in the area of maintainability.

Early in the technology program an interim breadboard test will be conducted using 5-7 kW_e modules of both fuel cell and electrolysis cells, operating in both the dedicated and reversible modes. Finally, a field technology readiness demonstration test will be conducted on engineering model modules of representative sizes in 1985.

CONCLUDING REMARKS

Over the last 16 years approximately $170 $\bar{M}$ has been expended through NASA to develop a strong capability in fuel cell electrochemical technology which is also intimately related to electrolysis technology. This expenditure has provided for the technology advancements as well as development and hardware costs for three flight programs with a fourth program in active development. In the beginning it was difficult to achieve 100 hours of test operation. Routine test durations of 5000 hours have been achieved on test hardware, and isolated one-of-a-kind tests have approached 50,000 hours of test operation. Specific weight [kg/kW (lbs/kW)] has been reduced by an order of magnitude, and a factor-of-two further reduction appears to be feasible. These advancements are graphically illustrated in figure 17 for both the alkaline and SPE technologies. Further weight reduction to less than 5 kg/kW (10 lbs/kW) and life exceeding 100,000 hours appear now to be feasible.

A direct outgrowtn of the development of fuel cells for space is the development toward terrestrial applications. The space program proved the utility of the concept and laid the foundation; this is the foundation upon which future utilization of fuel cells will be built.

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5. Lockheed Missiles and Space Division, "Design Data Handbook for Regenerative Fuel Cell Study," LMSC-D159786, NASA Contract NAS9-12508, Nov. 1972.

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Figure 1. Fuel Cell Programs

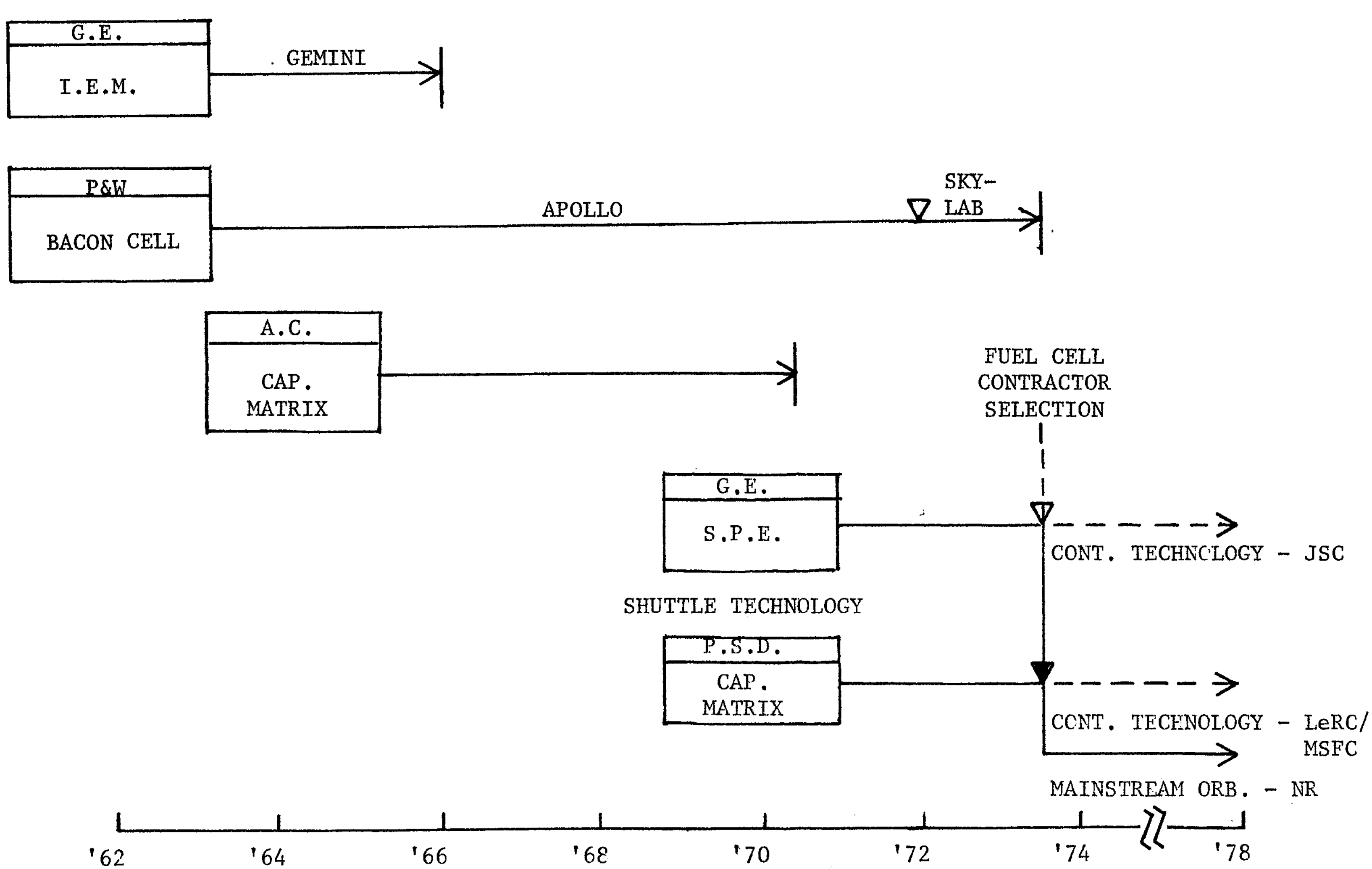


Figure 2. Gemini Fuel Cell

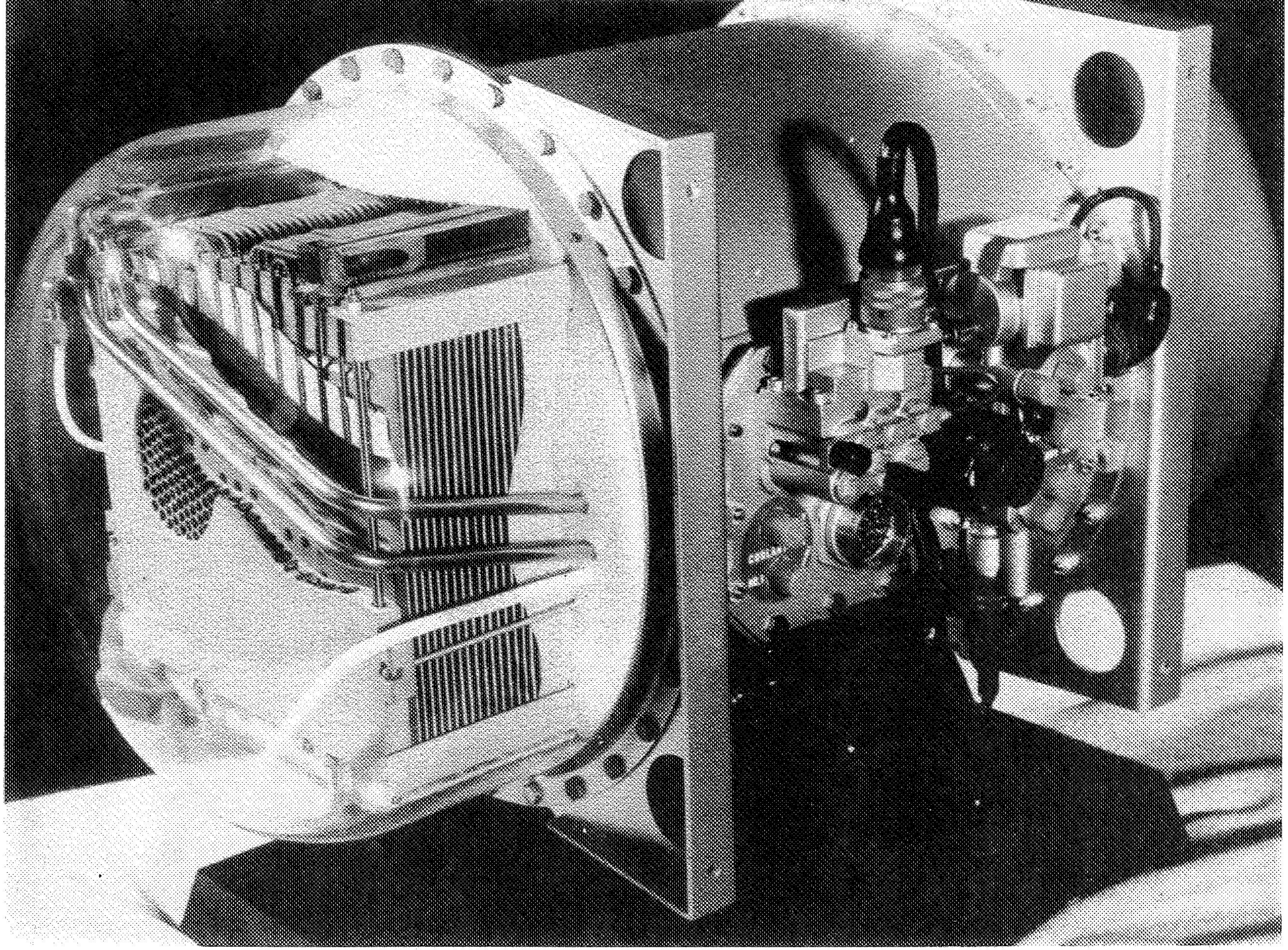
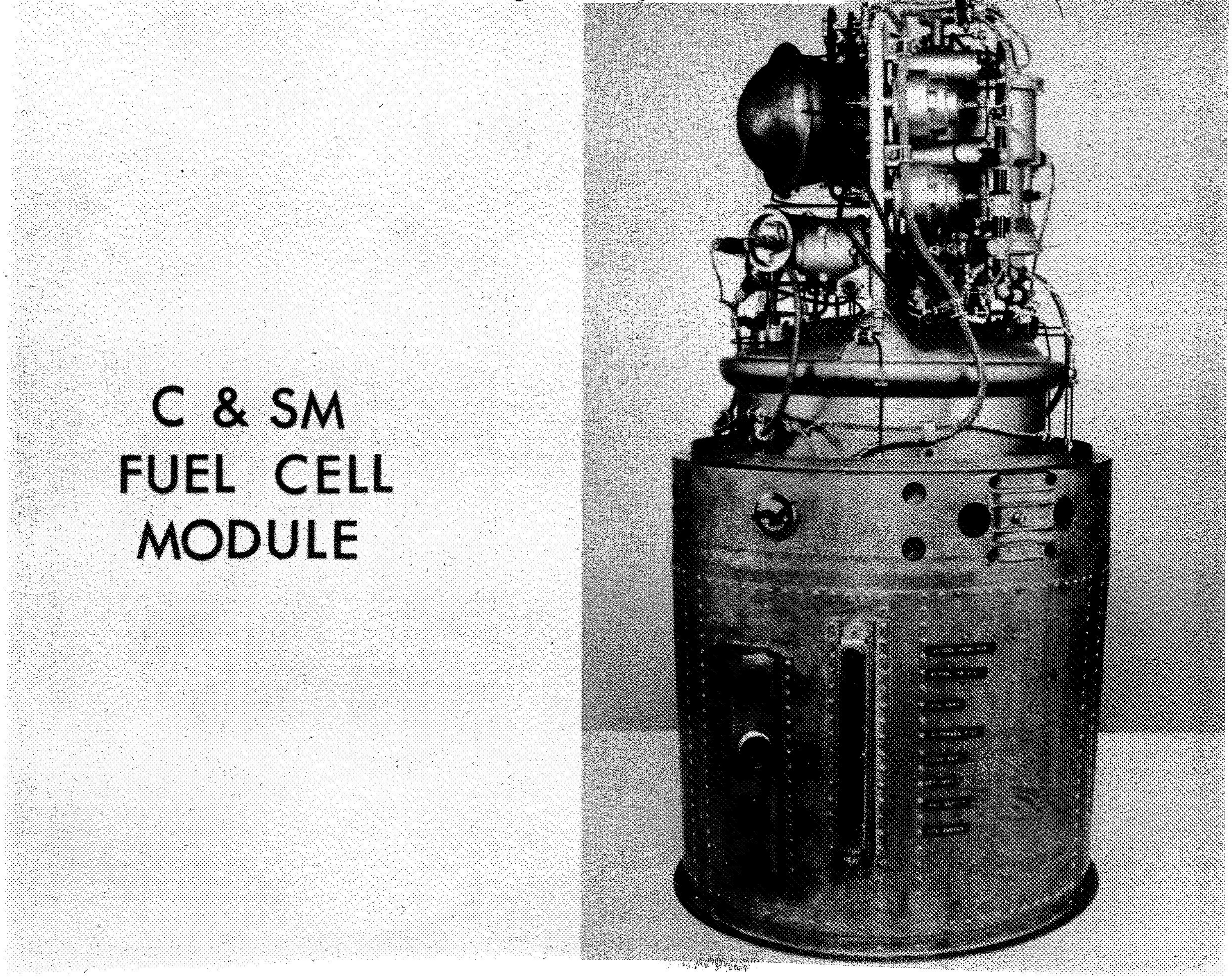


Figure 3. Apollo Fuel Cell



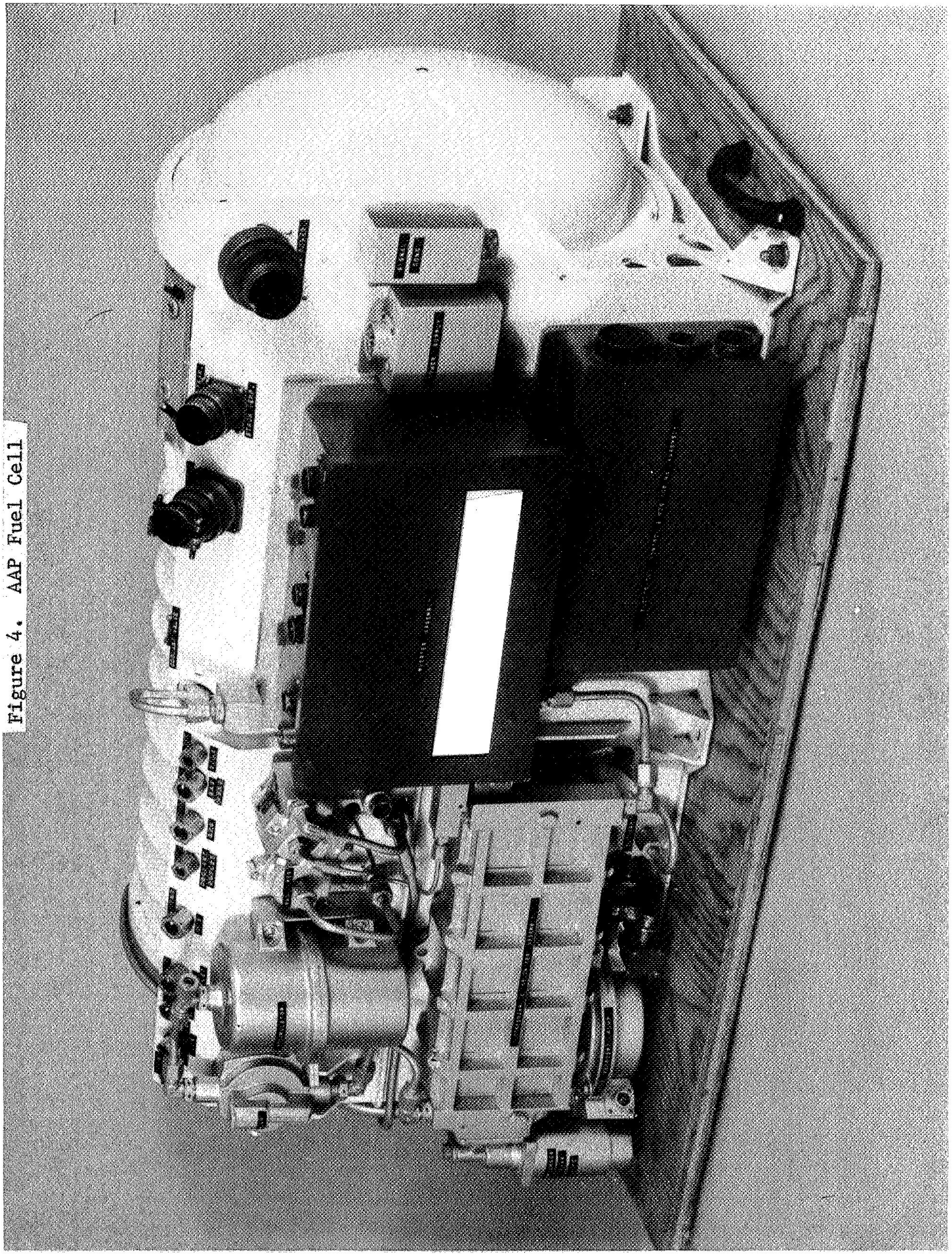
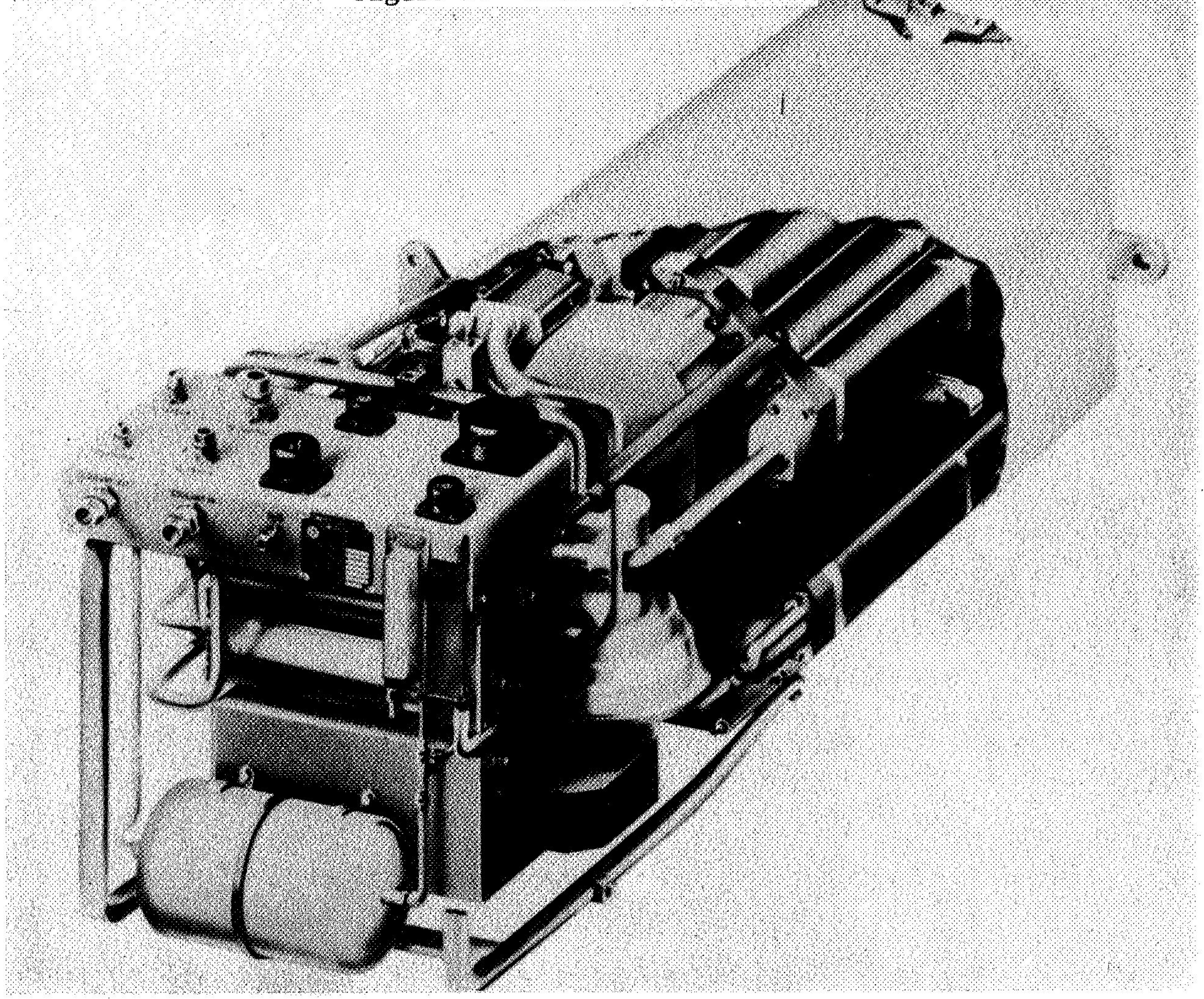


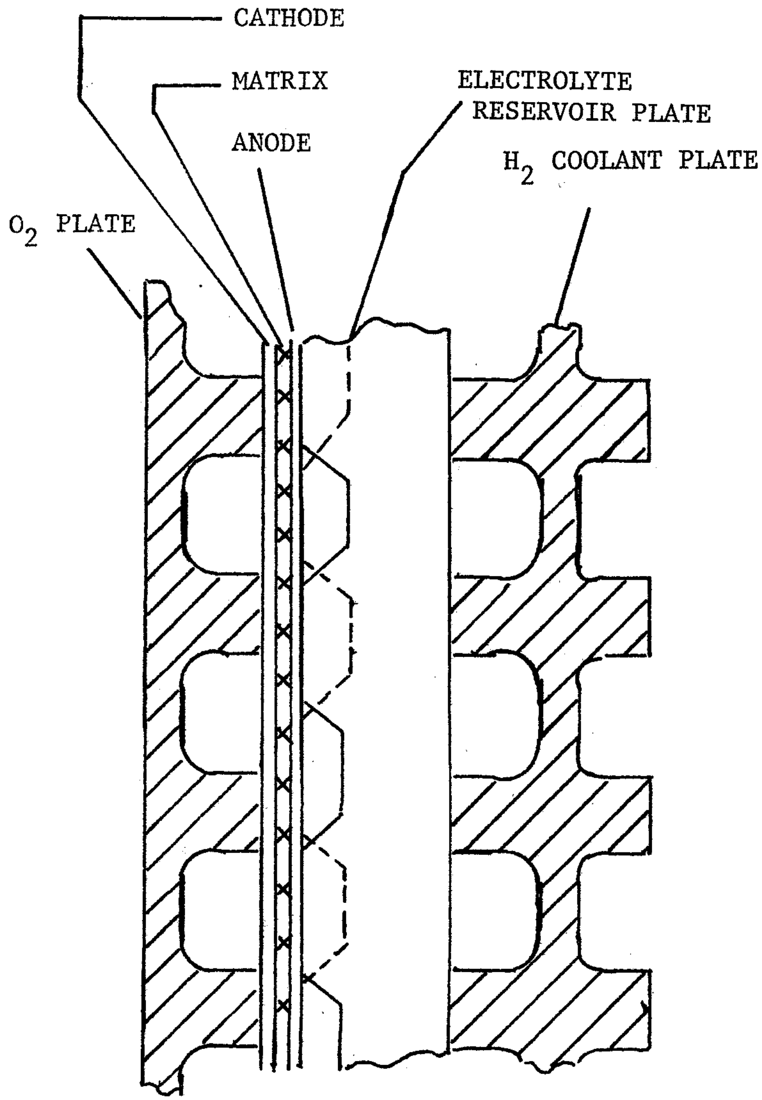
Figure 4. AAP Fuel Cell

Figure 5. Shuttle Fuel Cell



SPECIFICATIONS

POWER :	2 kW, min. (32.5v) 4.5 kW, avg. (2500 hrs) 7.0 kW, nom. 10 kW, 1 hr. emerg. 12 kW, 15 min.	OP. TEMP. :	355-383^{o}K (180-230^{o}F)
		OP. PRESS. :	414 kPa (60 psia)
		LIFE :	2500 hr. @ avg. pwr. 5000 hr. w/maint.
VOLTAGE:	27.5v - 32.5v	WEIGHT :	91 kg (200 lb.)



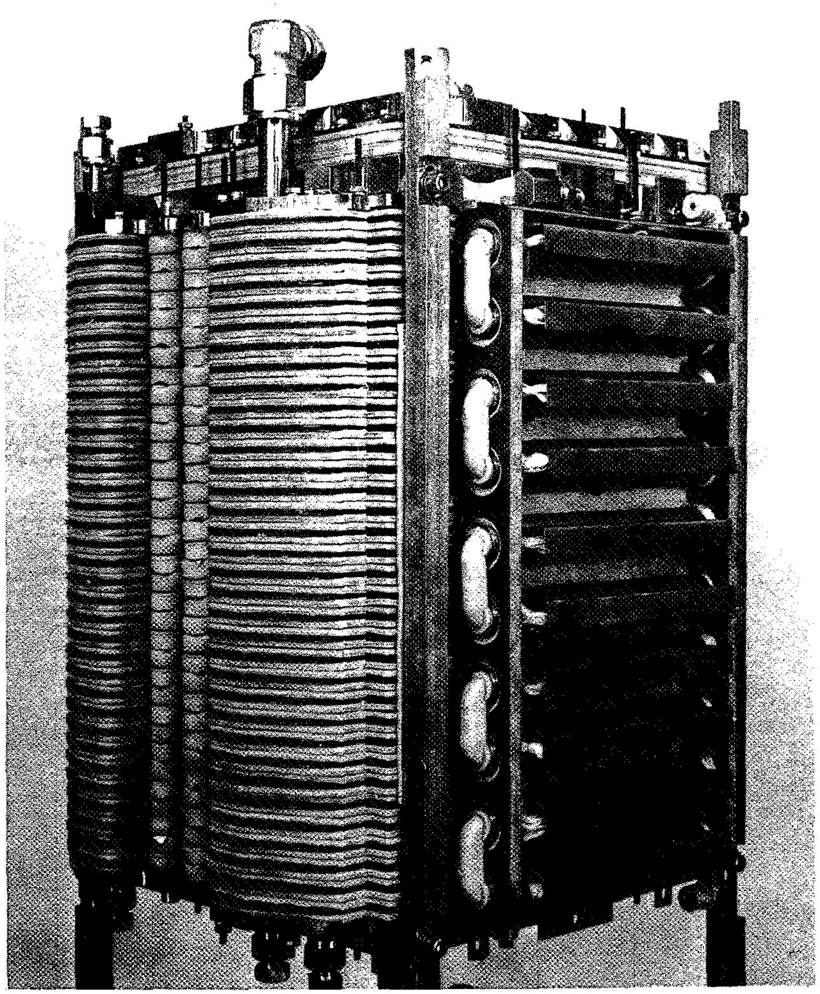
Cell Cross Section

CELL DESCRIPTION

CURRENT COLLECTION	:	**Bi-Polar**
ACTIVE AREA	:	0.047 m^2 (0.508 $ft.^2$)
THICKNESS	:	7.6 mm (0.3 in.)
O_2 PLATE	:	Au-Plated Mg
CATHODE	:	100-Mesh Au Screen
	:	10 gms/cm^2, Au-Pt Alloy
MATRIX	:	0.5 mm (.020 in.) Recon.Asb. 32% (wt.) KOH
ANODE	:	100-Mesh Ag Screen 10 gms/cm^2 Pt
ERP	:	Sintered Ni Powder
H_2 COOLANT PLATE	:	Au-Plated Mg
COOLANT	:	FC-40
CELLS/STACK	:	32, Electrical Series
STACKS/MODULE	:	2, Electrical Parallel

Figure 6. Shuttle Fuel Cells

Figure 7. SPE Fuel Cell

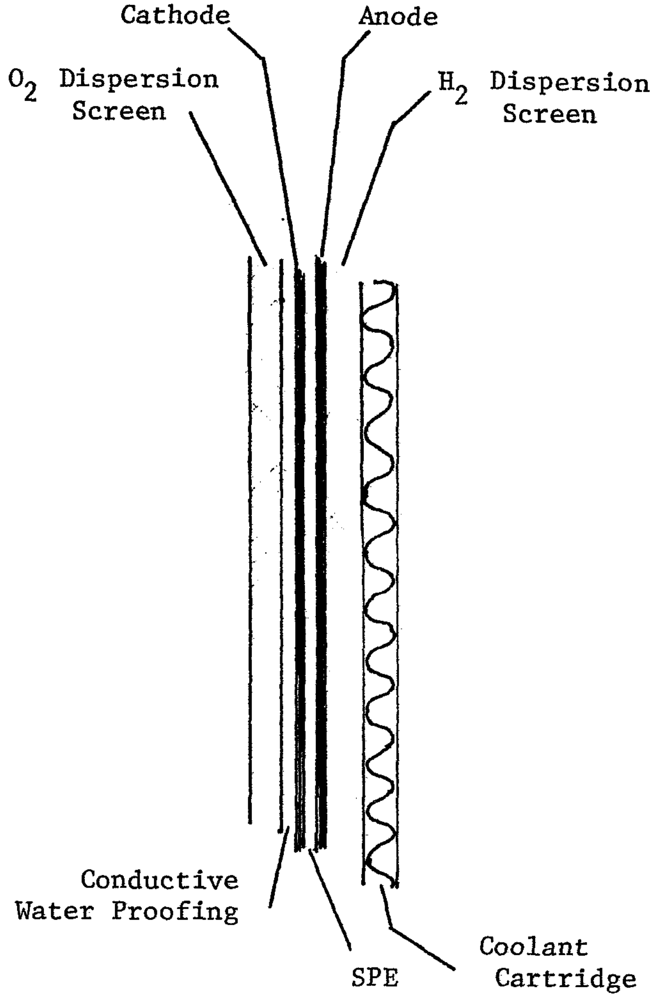


SPE REACTOR STACK

TECHNOLOGY STATUS

POWER	:	Design Option (36 cells, 14 kW)
VOLTAGE	:	('73) 0.75-0.9v; 200-50 mA/cm^2 (ASF) ('75) 0.70-0.9v; 500-100 mA/cm^2 (ASF) ('77) 0.75-0.95v; 1000-100 mA/cm^2 (ASF)
CELL AREA	:	0.035 m^2 (0.375 ft^2)(Back-to-back) 0.065 m^2 (0.7 ft^2)(Bi-Polar) 0.102 m^2 (1.1 ft^2)(Bi-Polar; Optimum)
PERFORMANCE	:	Invariant (<1 μv/cell-hr)
OP. TEMP.	:	355-428°K (180-300°F)
OP. PRESS	:	414-793 kPa (60-115 psia)
LIFE	:	('73) >5000 Hours (module) ('78) >48,000 Hours (short stack)
WEIGHT	:	('73) 9 kg/kW (20 lbs/kW)

Figure 8. SPE Technology

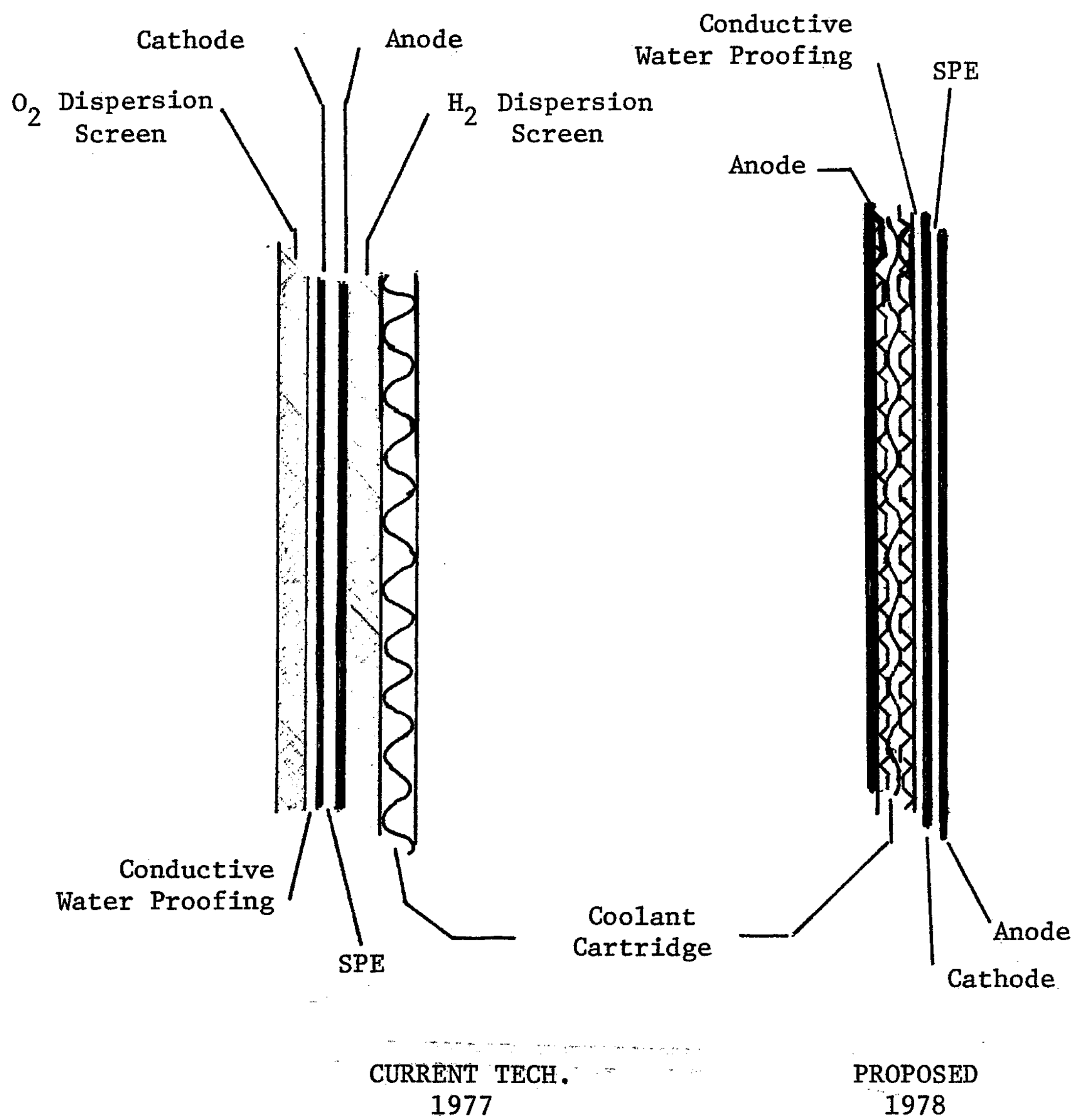


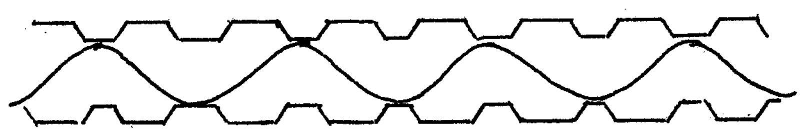
Cell Cross Section

CELL DESCRIPTION

CURRENT COLLECTION	:	Bi-Polar
ACTIVE AREA	:	0.102 m^2 (1.1 $ft.^2$)
THICKNESS	:	1.9 mm (0.075 in.)
O_2 SCREEN	:	0.38 mm (0.015 in.) Nb
COND. WATER PROOFING	:	Graphite/Teflon admixture
CATHODE	:	4 mg/cm^2 Pt Powder
SOLID POLYMER ELECT.	:	0.127 mm (0.005 in.) sulfonated fluorocarbon
ANODE	:	4 mg/cm^2, Pt Powder
H_2 SCREEN	:	0.38 mm (0.015 in.) Nb
COOL. CARTRIDGE	:	0.89 mm (0.035 in.) thick Nb
COOLANT	:	H_2O

Figure 9. General Electric
SPE Fuel Design





COOLANT CARTRIDGE
1978

Figure 10. SPE Fuel Cell Performance

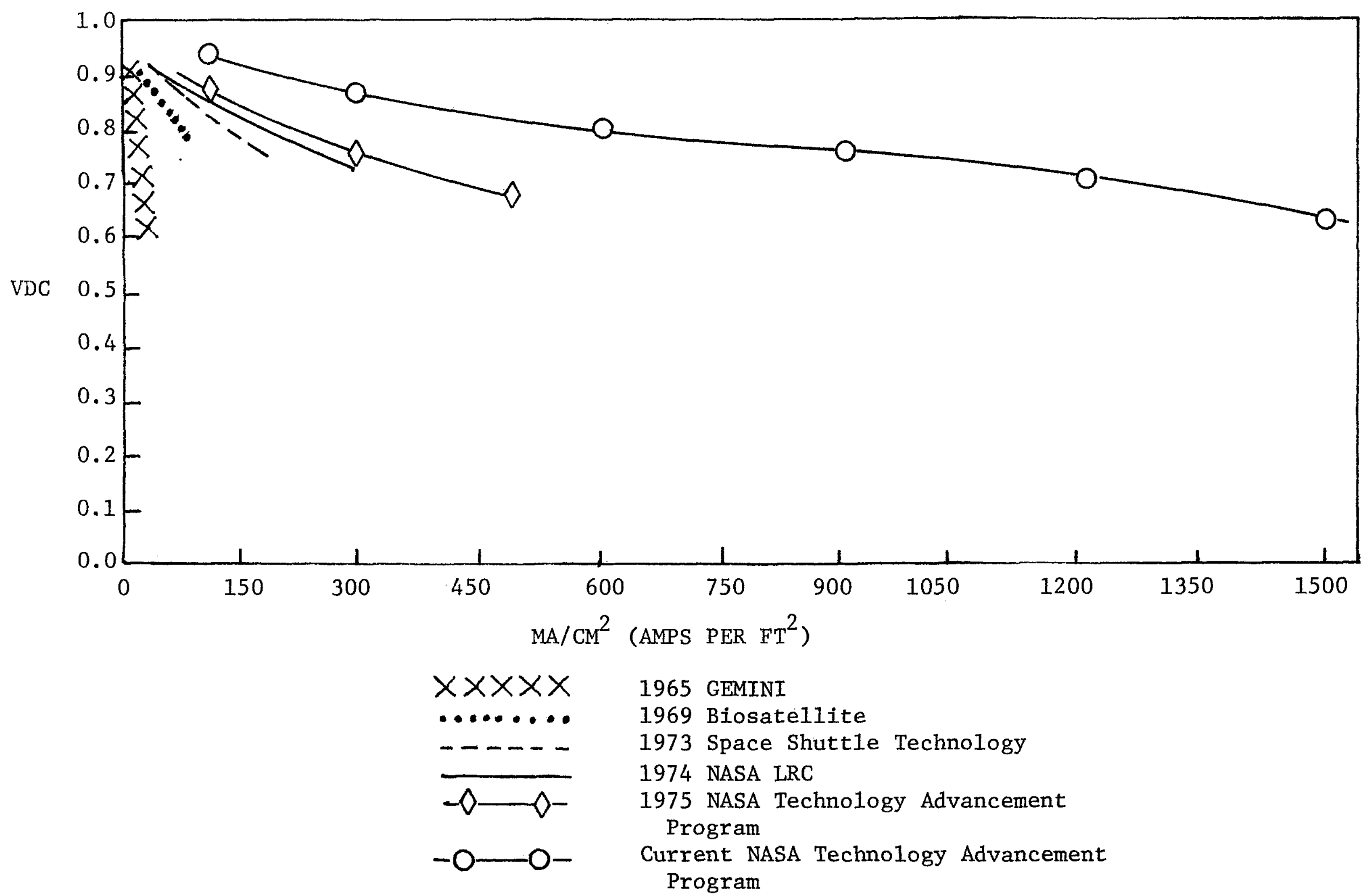
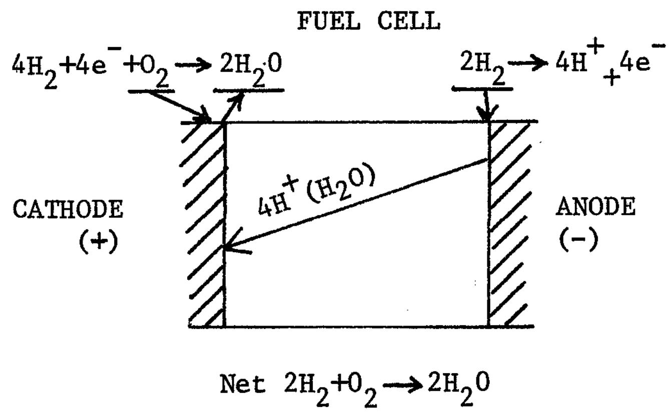
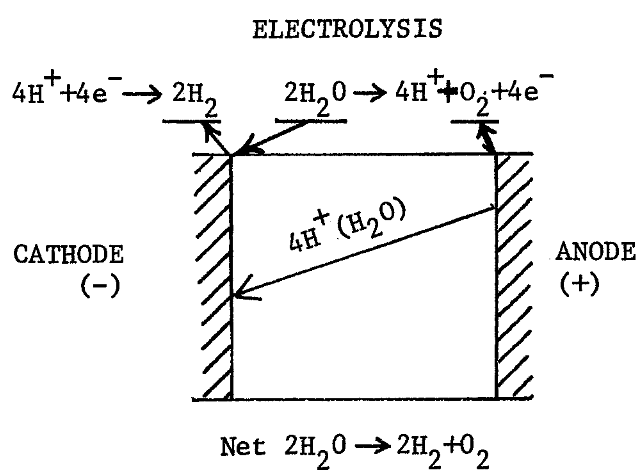


Figure 11. Electrolysis Cell





SPE

ELECTROLYSIS STATUS

Operating Temperature - 300°-425°K (90°-300°F)
Pressure - 21 MPa (3000 psia)
Δ Pressure - to 3.5 MPa (500 psid)

Voltage, cell - 1.5-2.2v (@ 0-2,000 mA/cm^2 [ASF])

Test Experience
Cell - Hi C.D.- 20,000 Hrs
Lo C.D.- 45,000 Hrs
Module - 78 cells - 5,000 Hrs

Figure 12. Voltage - Current Trend for H_2/O_2 Electrochemical Cells

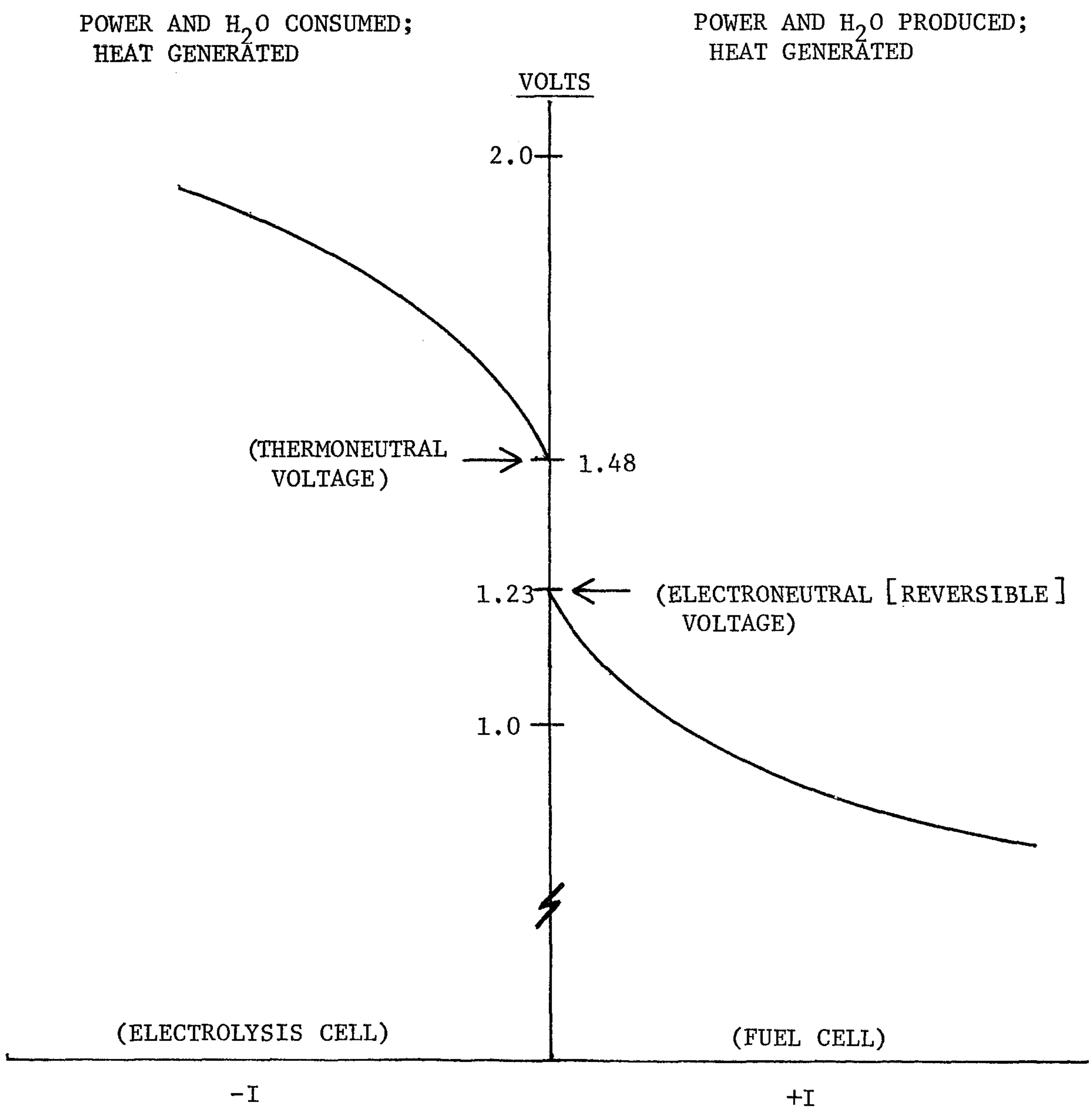


Figure 13. Water Electrolysis Performance

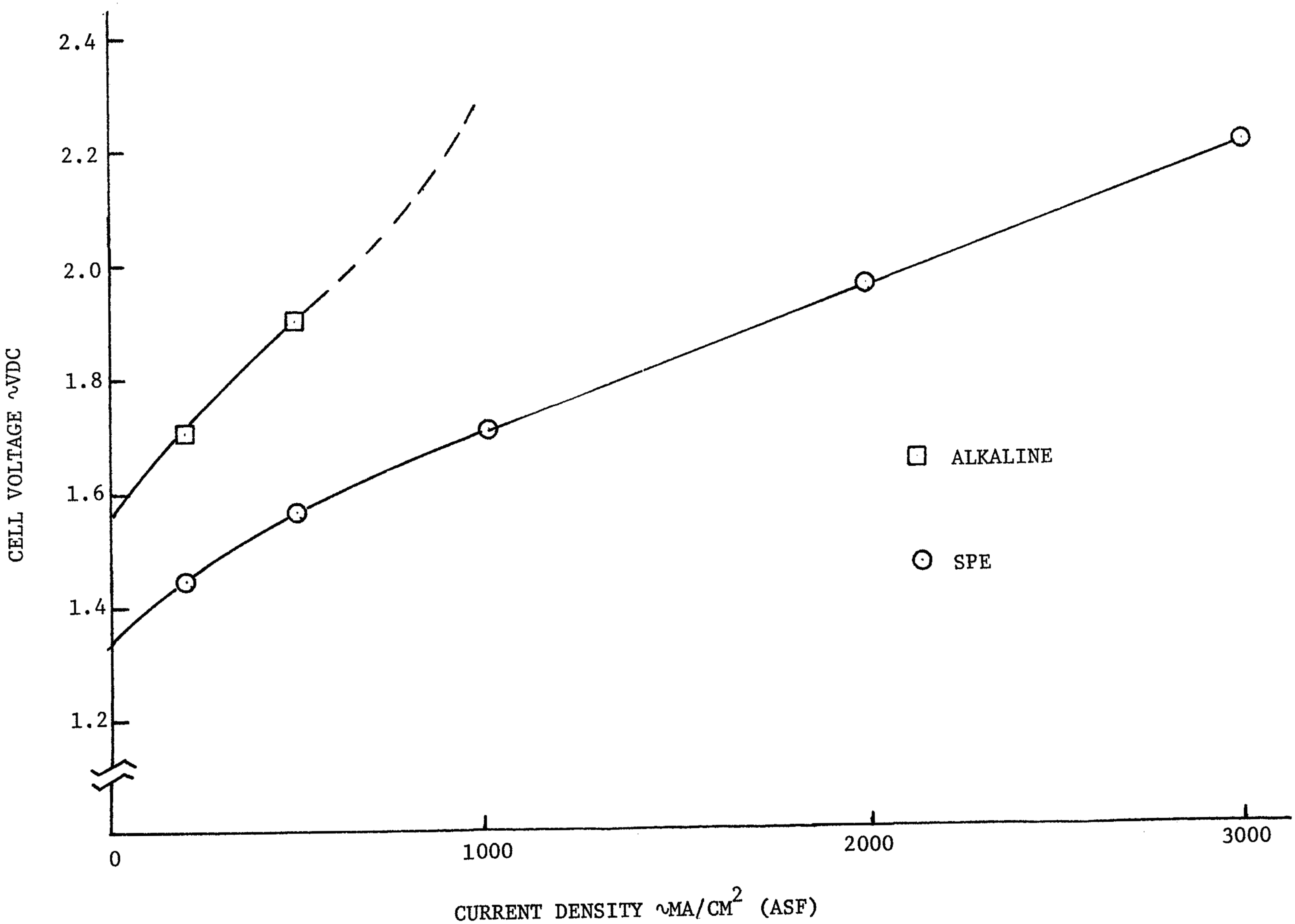


Figure 14. H_2-O_2 Fuel Cell Performance

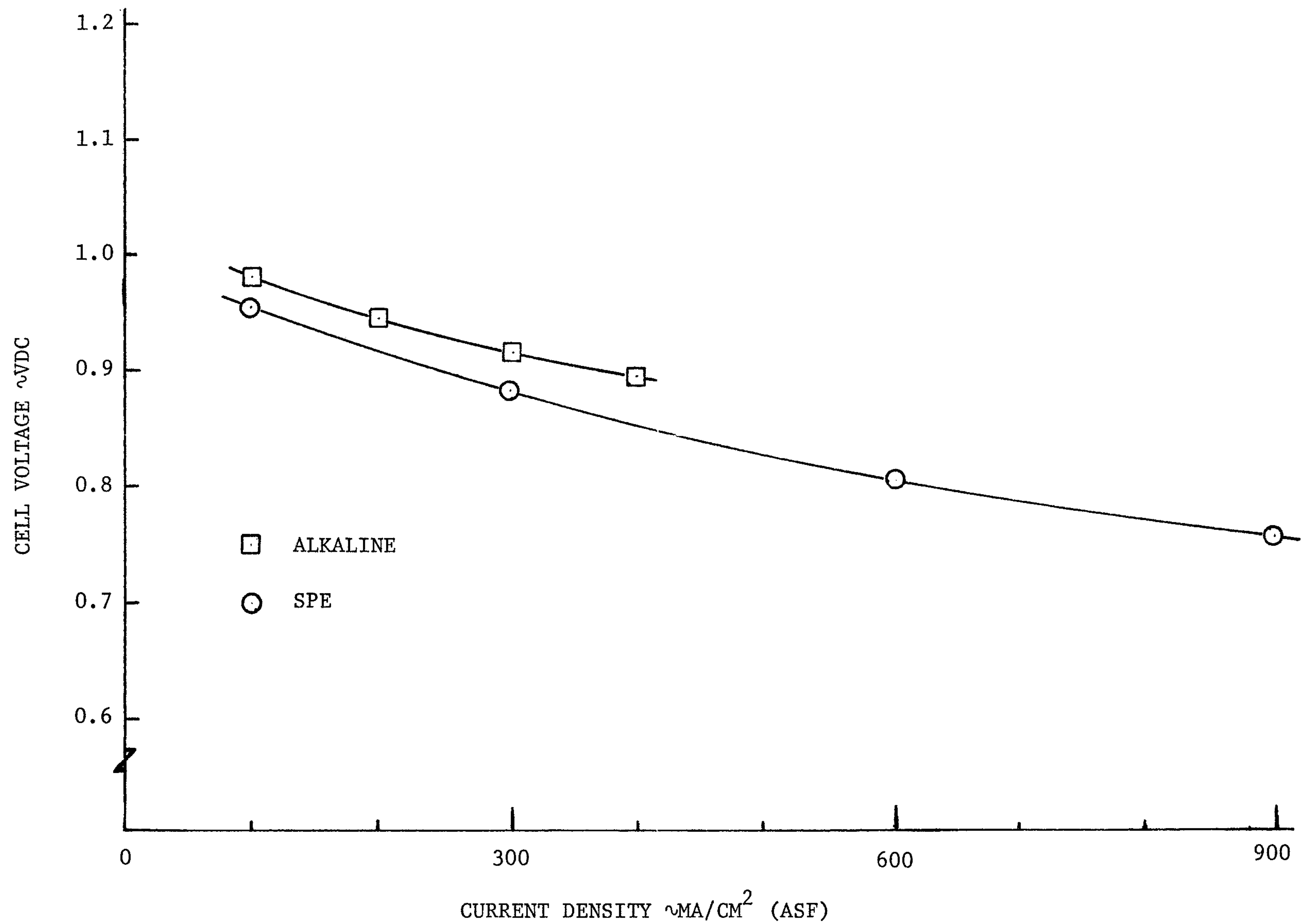


Figure 15. Fuel Cell, Electrolysis Regenerative Energy Storage Comparisons to Battery Energy Storage

Evaluation Criteria	Fuel Cells, Electrolysis Regener, Energy Storage	Battery Energy Storage
Thermal Control	Single Temperature Development (4.8 $\overline{M}$ less development cost)	Two loop development Dual temperature ranges
Charge-Discharge Efficiency	0.525	0.625 (higher efficiency)
Solar Array Area Requirement	22 m^2 (240 ft^2) less [700 m^2 (7540 ft^2) SA] (based on 24-hour cycle)	723 m^2 (7780 ft^2) solar array (based on per orbit cycling)
Secondary Power Requirement (emergency, buildup)	Utilize energy storage F C's	Adds F C's to energy storage assembly (battery capacity inadequate)
ISS Interface	Four equivalent subassemblies	32 equivalent sub-assemblies (more complex)
Launch Weight	1280 kg (2817 lbs)	4169 kg (9172 lbs) (heavy!)
Cost, $		
Development	14.7 $\overline{M}$ (assumes shared development)	13.7 $\overline{M}$ (includes secondary power)
Hardware	5.3	7.5 (includes secondary power)
Operations	7.9	10.0 (includes launch
Overall (IOC + 5-Yr Ops)	27.7	32.2 [$114/kg]($250/lb)
Sensitivities:	• Fuel cell lifetimes • Amount of shared development of electrolysis & fuel cells • 24-hr cycling	• Voltage degradation • Charge scheme-- available energy & charge time constraints

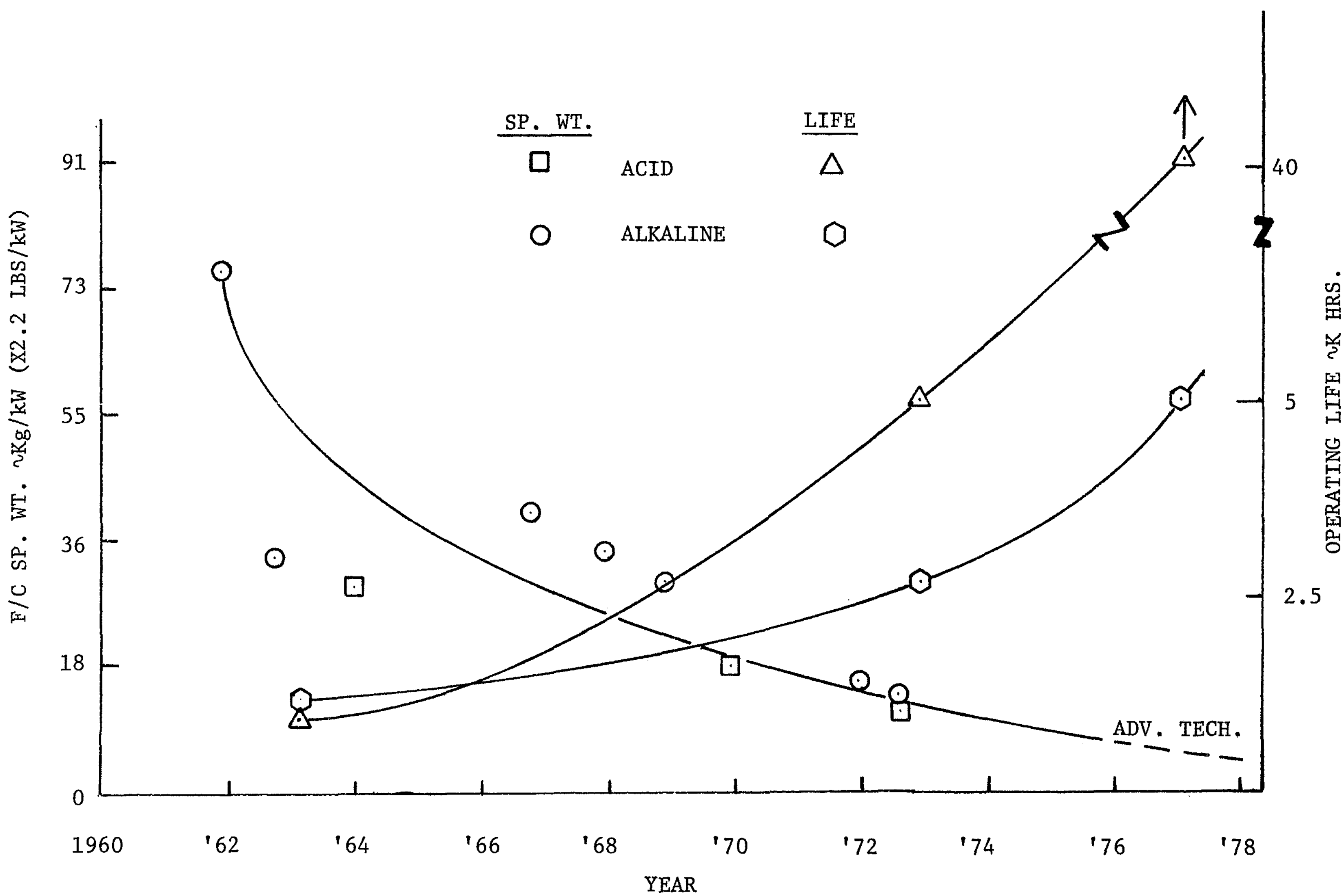
Figure 16. Fuel Cell/Electrolysis Cell Energy Storage*
MDAC Summary

	NiCd	RFC
SOLAR ARRAY AREA, m^2 (ft^2)	2,407 (25,910)	2,587 (27,847)
STORAGE EFFICIENCY, %	62.0	54.1
DEPTH OF DISCHARGE, %	14.5	33.0
ENERGY DENSITY, Wh/kg (Wh/lb)	3.93/27.08 (1) (8.66/59.6)	25.0/75.1 (1) (55/165.2)
PEAK LOAD CAPABILITY	∿ 10X	∿ 4X
LAUNCH WEIGHT, kg (lbs)	34,763 (76479)	16,083 (35,383)
RESUPPLY WEIGHT, kg (lbs)(10 yrs)	21,000 (46,200)	2,994 (6,587)
TOTAL 10 YR WEIGHT	55,763 (122,679)	19,077 (41,970)

*100 kW_e Average at inverter output; baseline power platform

(1) usable/absolute

Figure 17. H_2-O_2 Fuel Cells Specific Weight and Operating Life versus Calendar Time



POWER MANAGEMENT AND CONTROL FOR SPACE SYSTEMS

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SUMMARY

Power management and control technology for the large, high-power spacecraft of the 1980's is discussed. System weight optimizations that indicate a need for higher bus voltages are shown. Environmental interactions that are practical limits for the maximum potential on exposed surfaces are shown. A dual-voltage system is proposed that would provide the weight savings of a high-voltage distribution system and take into account the potential environmental interactions. The technology development of new components and circuits is also discussed.

INTRODUCTION

Future use of space for large-scale manufacturing and construction, materials processing, and medical and scientific research will require large quantities of bulk electrical energy. These uses will also impose a series of new diverse requirements on electrical power systems in terms of lower weight, increased life, reduced cost, improved efficiency, and ruggedness. These requirements dictate the development of a high-voltage, utility-type power system and power system control. The technology development proposed is the first phase toward realizing such a power system.

SPACE POWER SYSTEM ELEMENTS

Even though proposed space power systems may vary from a few hundred watts to gigawatts, all will share the same generic system elements:

(1) Power source: Power sources fall into two major classes, solar and nuclear. A technology ready for near-term application in the mid and late 1980's will be the flexible-bodied deployable solar array.

(2) Rotary power transfer: Solar power systems require constant orientation to maintain optimum sun angle. Nuclear systems require thermal control systems pointed toward deep space. In Earth orbital applications, the power source must rotate every 90 minutes. To carry the power across this interface requires a rotary coupling device with low friction, low loss, and long life characteristics.

The simplest, most direct approach is to carry the power across the interface by means of cables. This method requires rapid reverse drive of the solar array to unwrap the cables periodically. Slip rings have been used extensively for power transfer. However, technology adequate for the high powers and high voltages being considered has not yet been developed. The development of on-array conversion systems that create high-frequency alternating current could enable power to be transferred across the interface by noncontact means, by using either inductive or capacitive couplers.

(3) Transmission: As space systems grow in physical size, the design of the power distribution system, in addition to power level, will require serious attention in order to keep line losses to an acceptable level. Distribution and conversion at the load sites will require the development of high power, high-voltage line switches to provide load disconnect capability. Conversion at the load site will require switches, transformers, and other components suitable for operation from the distribution bus.

POWER MANAGEMENT AND CONTROL TECHNOLOGY

Transmission System Characteristics

In the large space systems now under study, power must be distributed from the source over long distances. Thus, careful consideration will first be given to optimizing the distribution system to minimize weight and power losses.

Environmental Interactions

Operation of power systems in the plasma environment of space could present design limitations that must be carefully considered before the system design is finalized.

Power Management Philosophy and Approach

Trade-offs between the optimum distribution system and the limitations imposed by the environment, coupled with the characteristics of the source and the requirements of the load, will factor into the power management philosophy that will be adopted.

Choices will be made between direct- or alternating-current systems, the distribution voltages, the branching philosophy with its attendant switching, and load control. The power management will be controlled with a microprocessor to insure maximum use of the available power, to govern load sharing on a priority basis, and to monitor system performance to prevent catastrophic failures.

The development of a lightweight, efficient multikilowatt power system will require new circuit and component technology. High-power space systems will undoubtedly increase the transmission and distribution voltage levels and require higher voltage switchgear and remote power controllers. To achieve lightweight conversion, it will be necessary to develop new circuit concepts that use higher frequencies. The recent developments of fast high power semiconductors switches and high energy density capacitors provide for significant weight savings.

TRANSMISSION LINE OPTIMIZATION

In an electrical power distribution system, the transmission line contributes to the total system weight in three ways. The first is the weight of the transmission line itself. This is given by

$$W_{TL} = 2dA\ell$$

where

W_{TL} transmission line weight, g

d density of transmission line material, g/cm^3

A cross-sectional area of transmission line, cm^2

ℓ transmission line length, cm

The factor of 2 arises from the need to have a return. If the spacecraft can be used as a return, the weight is only half as much. The analysis in this section does not include the case of spacecraft electrical return.

The second contribution to the weight of the power system, due to the transmission line, is the extra power generation system weight ΔW_{PG} required to make up for power loss in the transmission line:

$$\Delta W_{PG} = \frac{2I^2\rho\ell\alpha_{PG}}{A}$$

where

I transmission line current, A

ρ transmission line resistivity, Ω-cm

α_{PG} marginal specific weight of power generating system, g/W

The third contribution to the weight of the power system, due to the transmission line, is the extra heat rejection system weight ΔW_{HR} needed to handle the heat produced by losses in the transmission line:

$$\Delta W_{HR} = \frac{2I^2\rho\ell\alpha_{HR}}{A}$$

where

α_{HR} marginal specific weight of heat rejection system, g/W

The total weight penalty that can be allotted to the transmission line $\Delta W'_{TL}$ is

$$\Delta W'_{TL} = W_{TL} + \Delta W_{PG} + \Delta W_{HR}$$

$$= 2dA\ell + \frac{2I^2\rho\ell}{A}(\alpha_{PG} + \alpha_{HR})$$

To find the minimum transmission line penalty, set $d\,\Delta W'_{TL}/dA = 0$. This gives an optimum current density of

$$J = \frac{I}{A} = \sqrt{\frac{d}{\rho(\alpha_{PG} + \alpha_{HR})}}$$

and an optimum conductor area of

$$A_{opt} = I\sqrt{\frac{\rho(\alpha_{PG} + \alpha_{HR})}{d}}$$

The weight of the optimum direct-current transmission line in terms of the power transmitted is given as

$$W_{TL,opt} = \frac{2P\ell}{V}\sqrt{\rho d(\alpha_{HR} + \alpha_{PG})}$$

where

P power, kW

ℓ transmission line length, cm

ρ resistivity of transmission line, g/cm^3

α_{HR} specific mass of heat rejection system, kg/kW

α_{PG} specific mass of power generation system, kg/kW

V line voltage, V

When transmission line materials are chosen, the figure of merit used for both minimum transmission line weight and transmission line weight penalty is the product ρd. Figure 1, which compares some of the better known conductors, shows that aluminum is about a factor of 2 better than copper and sodium is nearly another factor of 2 better than aluminum. The optimum direct-current transmission line weight, in terms of power output P and load voltage V is then

$$W_{TL,opt} = \frac{2P\ell}{V} \sqrt{\rho d(\alpha_{PG} + \alpha_{HR})}$$

and the optimum transmission line penalty is twice this.

For a three-phase transmission line, where V is the line-to-neutral rms voltage and the neutral carries no current and can be considered much smaller than the other lines, the total transmission line weight is $3dA\ell$. For this case, the optimum transmission line weight W_{TL} is

$$W_{TL(3\phi,\text{line-to-neutral voltage,no neutral})} = \frac{P\ell}{V} \sqrt{\rho d(\alpha_{PG} + \alpha_{HR})}$$

The total transmission line weight penalty is twice the transmission line weight, as in the direct-current case.

Depending on what is the limiting factor, voltage-to-neutral, line-to-line voltage, rms voltage, or peak voltage may be used in the weight comparison. The effects of these parameters can be given in a factor F_V for the type of voltage distribution, where

$$W_{TL} = \frac{F_V P\ell}{V} \sqrt{\rho d(\alpha_{PG} + \alpha_{HR})}$$

Table I shows the factor F_V for different types of power distribution, with the transmission line weight being minimized for that type of distribution.

The quantity $P\ell/V\sqrt{\rho d(\alpha_{PG} + \alpha_{HR})}$ is plotted in figure 2 for $\rho d = 7.6\times10^{-6}$ for an aluminum transmission line and $\alpha_{PG} + \alpha_{HR} = 30$ kg/kW. Other values of $P\ell$, V, ρ, d, and $\alpha_{PG} + \alpha_{HR}$ may be obtained by appropriate scaling.

Tentative conclusions for minimum transmission line weight penalty are

(1) Use as high a distribution voltage as possible, subject to constraints on insulation converter weights, etc.

(2) Use aluminum (or in an advanced system, sodium) transmission lines rather than copper.

(3) Use three-phase alternating-current, double-voltage alternating-current ($\pm$V), double-voltage direct-current distribution, or a spacecraft return system.

SPACECRAFT-ENVIRONMENTAL INTERACTIONS

Power systems must be very large to generate the multikilowatts needed for future missions, and these systems must operate in the charged-particle environment of space. This requirement introduces the topic of interactions between the large power system and charged-particle space environments.

A pictorial representation of a large space system is shown in figure 3. This illustrates the categories of the interactions of concern. The first category is called "spacecraft passive" - interactions that arise from the environment acting on the spacecraft. The geomagnetic substorm flux of particles occurring at geosynchronous altitude typifies this category of interaction. This flux charges spacecraft surfaces to a value where discharges can occur and cause electronic anomalies. This spacecraft charging interaction has been the subject of intense investigation for the past 3 years.

The second category of interactions, called "spacecraft active," is a new concept and is the principal interaction of concern here. This category is illustrated by the current loop through the environment. The power system will float electrically at some voltage relative to the plasma potential. The exposed interconnects that are positive will collect electrons, and the negative interconnects will collect ions. This collection can be considered to be a current loop through the environment that is parallel to the power system. It is parasitic and represents power losses. This loss depends on plasma density and operating voltages. This interaction must be understood and technology developed to minimize losses.

Other illustrations of this second category of interactions are ion thruster efflux interactions (if these systems use ion thrusters) and electromagnetic interactions. The ion thruster efflux can enhance the plasma density and increase the power system interactions. The electromagnetic interactions are torques introduced by the motion of this large, differentially charged body moving through the Earth's magnetic field.

Solar Array - Environmental Interactions

A ground test investigation of solar array - environmental interactions has been under way at the Lewis Research Center for the past several years. Typical results of testing solar array segments in simulated low-Earth-orbit plasma environments is shown in figure 4. The test is run in the dark with voltages relative to ground applied to the segment by a power supply. The current is measured by a sensor floating in the high-voltage line. For positive voltages up to 200 volts the electron current collection appears to be proportional to the interconnect area, the plasma density and particle temperature, and a reduced value of the operational voltage (about 10 percent). At about 200 volts there is a transition in the current collection from interconnect area to whole panel area collection. At about 500 volts this transition is complete and the electron current collection is proportional to the panel area, the plasma density, and the operational voltage. For negative voltages between 0 and -10 volts there is combined electron and ion current collection. For negative voltages between -10 and about -700 volts the ion current collection seems to be proportional to the interconnect area, the plasma properties, and a reduced value of operational voltages (about 10 percent). At above -700 volts there is a transition to arcing. This arcing is believed to be caused by field emission from the interconnect to the environment. It collapses the operational voltage to essentially zero for the duration of the arc ($\sim\mu$sec).

Spacecraft-Environment Interaction Technology

A spacecraft-environment technology program must be established to understand the interactions, to evaluate the impact on system performance, and to develop design guidelines and recommend practices to minimize the interactions. First, the power losses through the environment must be evaluated as a function of operational voltages. For the large power system, efficiency can be improved and weight reduced if the operational voltage can be increased from the currently proposed 200 volts to a few kilovolts. However, increasing the voltage also increases the environmental losses. Hence, it would be beneficial if, at higher operational voltages, losses could be reduced to a tolerable level.

Ground testing of small solar array segments operating at elevated voltages in simulated space plasma conditions has indicated that arcing can occur on negative voltage areas of the array. At plasma conditions corresponding to about 900 kilometers altitude, this arcing occurs at about -800 volts. Space flight testing has verified this phenomenon. This arcing tendency must be eliminated in any future high-voltage space power system.

The effect of charge deposition on (or in) the materials used on these large space power systems must be understood if the proposed 20- to 30-year operational life is to be realized. This deposition can result in material degradation over long mission life. A charged vehicle can enhance contamination by electrostatically attracting charged particles back to the surfaces and thus reducing transparency of the solar cell covers. The charged-particle environment can be enhanced by ionizing outgassing molecules or by products from arcing.

Both the operational hazards from parasitic currents and arcing and the long-term effects of system degradation must be understood and controlled.

Dual-Voltage System

A high-power system configuration that would provide the weight savings of the optimized distribution system and take into account the potential environmental interactions is a dual-voltage system. The solar array would generate power at 200 to 300 volts that would be upconverted to voltages of about 1000 volts or greater directly on the solar array. The advantages of the upconversion would be lower weight of the rotary power transfer device and lower I^2R losses. The upconverter would be lightweight, efficient, and not regulated. Such a converter could easily be implemented as a voltage multiplier for direct-current distribution or as a series-resonant inverter for alternating-current distribution with a rotating transformer. As the capacity of future power systems increases, system optimizations will probably demand that solar array voltages increase significantly beyond 200 volts. Solutions to the environmental interaction problem will make possible this higher power transmission and distribution.

The high-voltage distribution system would provide power to the user site and be downconverted to the individual load requirements.

HIGH-POWER-CONVERSION TECHNOLOGY

A major portion of any space power system will be the inversion and conversion equipment. To meet the proposed space power system weight goals, it will be necessary to reduce the specific weight of high-power-conversion equipment to less than 1, and probably less than 1/2, kilogram per kilowatt. To achieve these goals, the circuit and component technology has been directed toward higher frequencies. Higher frequency circuits allow a significant reduction in component weight and especially in the magnetics, which usually account for approximately 50 percent of the power conversion weight. Efficiencies must be maintained high so the component weight reduction will not be significantly offset by an increased thermal control mass.

Three basic types of converters and inverters are presently under research and technology development at Lewis: the capacitance diode voltage multiplier (CDVM), the series-resonant inverter (SRI), and the conventional converter. A CDVM transfers energy from its source to the load by capacitance rather than by magnetic coupling and consequently does not require the weight-intensive power transformer. The CDVM uses a higher switching frequency, which reduces the circuit capacitance and the filter inductance. By eliminating the transformer, using a high switching frequency, and incorporating high-energy-density capacitors, the mass and dissipative losses of the CDVM are significantly reduced over present-day power processors. The efficiency of this type of converter has been demonstrated to be typically 95 percent.

The series-resonant inverter using thyristor switches has been developed to a technology readiness status and was chosen as the baseline 3-kilowatt power processor unit for the 30-centimeter mercury ion thruster system. An in-house study has shown that the SRI technology could be extended to meet the potential requirements of magnetically isolated inverters and converters used in large space power systems. The new series-resonant inverter would be a modular unit whose power output would be determined by the transmission and distribution system selected but that could provide outputs of 20 to 25 kilowatts. The unit would use bipolar transistors presently under development and would switch at frequencies greater than 30 kilohertz. The technology of heat-pipe-cooled magnetics presently under development at the 3-kilowatt level would also be used.

Since frequency plays such an important role in establishing the minimum size and weight of electrical equipment, a technology development effort has been started on a conventional converter with switching frequencies of about 100 kilohertz.

POWER COMPONENT TECHNOLOGY

To reach the performance levels required for power distribution and control in space - in terms of weight, efficiency, life, and reliability - considerable component technology development is required. Some of the more crucial areas for development are described here.

Rotary Transfer Devices

Different parts of a spacecraft often have different orientation requirements. An example is a solar-array-powered spacecraft, where the solar array must face the Sun and the antennas for communications and control must face the Earth. This multiple orientation requirement means that a rotary joint or joints are required in the spacecraft. These joints are available for low power but need development in the high-power ($\geq$100 kW) range. Lewis has a current program to do this.

Fast Switches, Magnetics, and Capacitors

In a high-performance space power distribution system, voltage conversion - such as from a low- or medium-voltage solar array to a high-voltage communications tube - is often required. In the past, voltage conversion equipment has been heavy, with a higher than desired power loss. Lighter weight magnetic devices and capacitors as well as switches to operate at higher frequencies are needed to make lighter weight, higher efficiency converters and ultimately to achieve lower power costs in space. Component development is being done at Lewis on transformers, low- and high-voltage capacitors, diodes, power transistors, and silicon-controlled rectifiers.

Materials

The dielectric materials needed are of the bulk insulation type and the conformal coating, or potting, type. One of the most useful is parylene, which is coated and polymerized from a gas and has exceptional penetrating power. Parylene is extremely inert and free from pinholes. It is being investigated at Lewis for a wide variety of applications.

High-Power, Efficient Switchgear

A missing element for higher power electric distribution in space is the necessary switchgear. Much work has been done at 28 volts, and some at 120 volts. More work is needed for the higher voltage distribution systems (300 V, 1000 V, and up) that will be needed to handle the higher powers for future space applications. Lewis is presently investigating 120-volt dc, 300-volt dc, and 1000-volt dc solid-state switchgear.

CONCLUDING REMARKS

Future use of space for large-scale manufacturing and construction, materials processing, and medical and scientific research will require large quantities of electrical energy. With the growth to multikilowatt power levels, new approaches must be developed to handle the transmission, distribution, conversion, and control of such high power in space.

Preliminary system optimizations strongly favor the high-voltage utility-type power management and control system. A system weight optimization of the transmission and distribution system was performed at three normalized power levels. The results clearly indicate the substantial benefits that can be achieved with high-voltage distribution.

An important consideration in the high-voltage power system optimization will be the limitations imposed by environmental interactions. Higher voltage reduces I^2R losses but increases environmental losses.

One possible high-power system that would provide optimum transmission and distribution and whose design would take into account the potential environmental interactions is a dual-voltage configuration. Such a system would generate solar array power at 200 to 300 volts, would be upconverted to 1000 volts for transmission across the rotary device, and would be distributed to the user at high voltage. Downconversion would be provided to the individual load requirements. This approach is compatible for the near term only; higher power systems will require that the technology of handling high voltage in this environment be understood.

The development of a high-voltage, lightweight, efficient multikilowatt power system will require new circuit and component technologies to reach the performance levels required. The component technology must be developed for efficient rotary power devices; high-speed, high-power semiconductor switches; high-energy-density magnetics and capacitors; and high-power switchgear. To achieve lightweight conversion, it will be necessary to develop new circuit concepts that use higher frequencies.

TABLE I. - VOLTAGE DISTRIBUTION FACTOR

Type of distribution and method of specifying voltage	Voltage distribution factor, F_V
dc, 2 wire	2.00
dc, 2 wire, ±V, balanced	1.00
dc, 3 wire, ±V, balanced	1.225, ($\sqrt{3/2}$)
Single-phase ac, 2 wire, rms	2.000
Single-phase ac, 2 wire, ±V, rms, balanced	1.000
Single-phase ac, 3 wire, ±V, rms, balanced	1.225, ($\sqrt{3/2}$)
Single-phase ac, 2 wire, pk-pk	2.828, ($2\sqrt{2}$)
Single-phase ac, 3 wire, ±V, pk-pk, balanced	1.732, ($\sqrt{3}$)
Three-phase ac, 3 wire, V_{LTN}, rms, balanced	1.000
Three-phase ac, 4 wire, V_{LTN}, rms, balanced	1.155, ($2/\sqrt{3}$)
Three-phase ac, 3 wire, V_{LTN}, pk-pk, balanced	1.414, ($\sqrt{2}$)
Three-phase ac, 4 wire, V_{LTN}, pk-pk, balanced	1.633, ($4/\sqrt{6}$)
Three-phase ac, 3 wire, V_{LTL}, rms, balanced	1.732, ($\sqrt{3}$)
Three-phase ac, 4 wire, V_{LTL}, rms, balanced	2.000
Three-phase ac, 3 wire, V_{LTL}, pk-pk, balanced	2.449, ($\sqrt{6}$)
Three-phase ac, 4 wire, V_{LTL}, pk-pk, balanced	2.828, ($2\sqrt{2}$)

TRANSMISSION LINE MATERIALS

MATERIAL	DENSITY g/cm^3	RESISTIVITY Ωcm	ρd
COPPER	8.96	1.72×10^{-6}	15.5×10^{-6}
SILVER	10.50	1.59×10^{-6}	16.70×10^{-6}
GOLD	19.32	2.44×10^{-6}	47.1×10^{-6}
ALUMINUM	2.6989	2.82×10^{-6}	7.62×10^{-6}
SODIUM	0.97	4.3×10^{-6}	4.17×10^{-6}
BERYLLIUM	1.85	4.0×10^{-6}	7.4×10^{-6}

Figure 1.

RANSMISSION AND DISTRIBUTION SYSTEM MASS VERSUS VOLTAGE

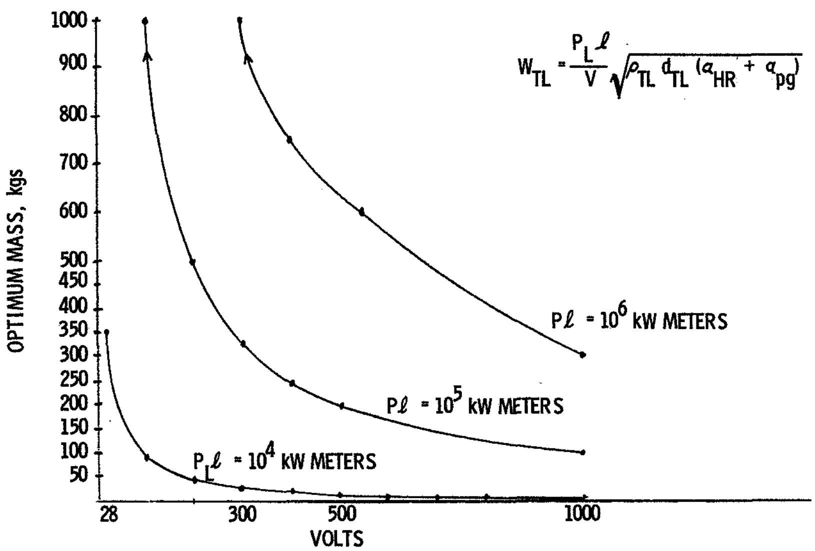

Figure 2.

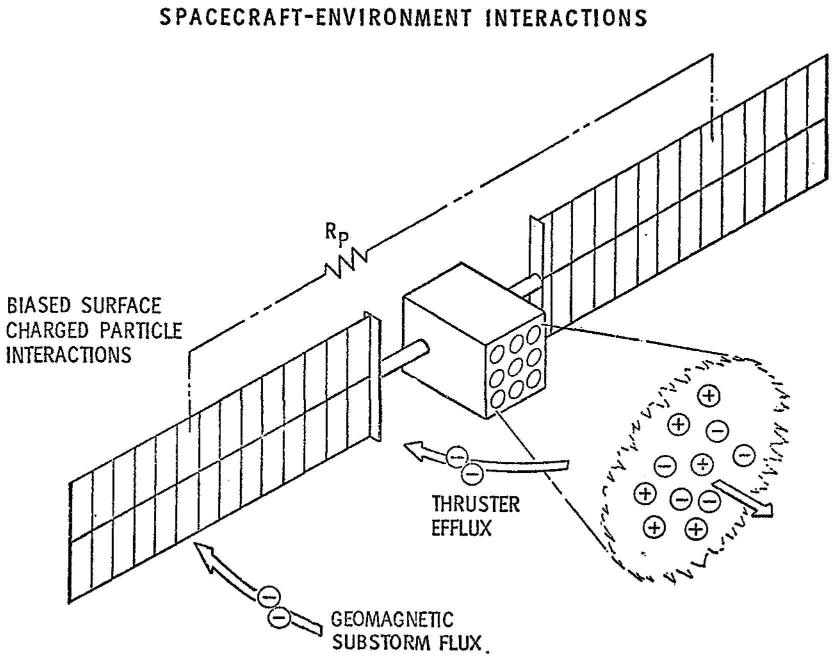

Figure 3.

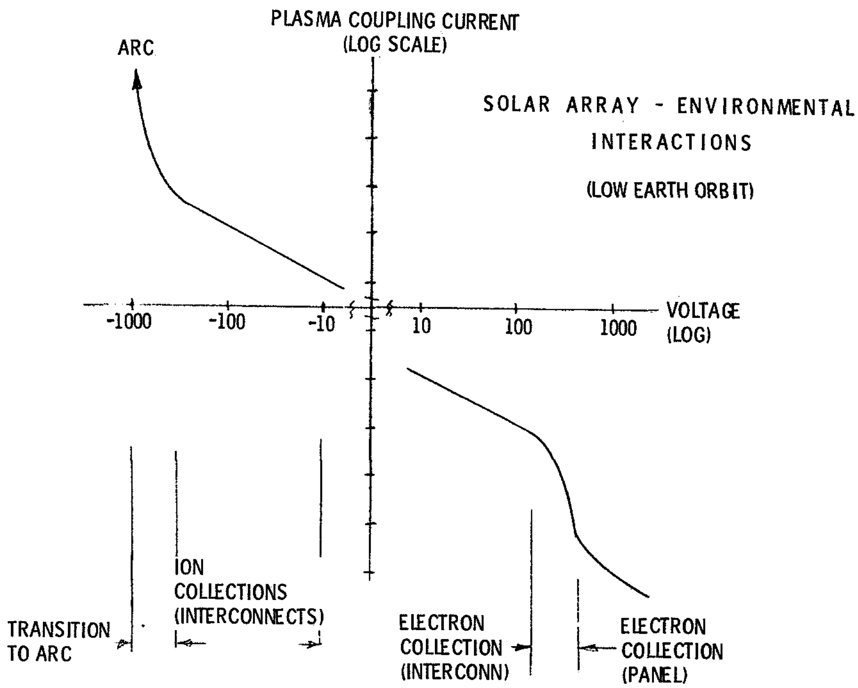

Figure 4.

LASER POWER TRANSMISSION FOR SPACE POWER AND PROPULSION

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SUMMARY

A review is being made of the state of development of major components and subsystems required for ground-to-space, space-to-space, or space-to-ground laser power transmission for electric or thermal power or propulsion. System characteristics are being evaluated from an applications viewpoint, and major problem areas are being identified. The objective is to identify a rewarding first application of lasers for space power and propulsion. An evolution of laser power transmission capabilities over the next 20 years is projected. Supporting technology requirements are to be identified, priorities set, and continued developments coordinated with other government agencies. This paper is an early status report of this work.

INTRODUCTION

A myriad of proposals have been made on the use of lasers in future space systems. Each shows the characteristics and advantages of such systems. It has been difficult to make an economic case for space-to-space power transmission for electric or thermal power generation as compared with conventional power systems on each satellite. Results appear more promising for laser propulsion. The possibility has been suggested that laser power transmission for propulsion may justify the laser system major components, which in turn could allow the laser energy electric or thermal power conversion system to be more competitive. It is expected that DOD work in this area will reduce the nonrecurring and recurring costs of NASA-proposed systems, but the degree cannot be quantified until a better definition of the NASA program is available.

PRIOR WORK

Prior work at NASA Centers, in industry, and at universities has addressed both major components and potential system application. NASA centers involved in lasers for power transmission have principally been Lewis, Ames, JPL, and Langley. The Lewis work is phasing out, and MSFC is picking up parts of this work. Industry participation has included LMSC, AVCO, BDM, Math Sciences Northwest, Westinghouse, Ball Brothers, and others. Universities involved are Washington, California, Pennsylvania, Pacific, Ohio State, Stanford, MIT, and others.

APPLICATIONS

Systems work at Lewis and their contracts with LMSC and others have addressed laser power transmission both for use in generating electricity on multiple receiving satellites and for use in an orbit-raising propulsion system at the receiving end. This work used a space-located laser and relay system. AVCO has proposed a GW ground-based laser for booster operation and for ground-to-GEO transfer. The University of Washington has proposed space-based lasers and adaptive optic relays for aircraft propulsion. Ball Brothers, BDM, Schaffer Associates and others have proposed a GEO-to-ground transmission by laser power instead of by microwave as presently being used in the satellite power system studies.

MAJOR TROUBLE SPOTS FROM APPLICATIONS VIEWPOINT

- Laser
  - Low ($\leq$100 K) heat rejection temperature
  - Complex and high-power laser supporting subsystems
  - Low-efficiency use of solar energy (for space located)
  - Hazards of a pointing error
- Laser energy collection and conversion
  - Large and heavy heat rejection radiators for heat engines
  - Conventional photovoltaics response not matched by current high-power laser output spectrum
  - High-energy density required for thermal electronic laser energy conversion (TELEC)
- Laser propulsion
  - Acquisition and tracking over wide range of laser-propulsion unit altitudes and ranges
- Laser transmission for electric or thermal energy use at receiving satellite
  - Economics dictate multiple use of source laser
  - Noncontinuous transmission requires high-charge-rate storage system on receiving satellite
  - Unless laser power source benefits in scaling to its large power capacity, in location, or other factors, its power source will be $\sim$5 times or more larger than the total of conventional dedicated satellite power systems.

PROPOSED DEVELOPMENT SCENARIO

Figure 1 illustrates a development and implementation program for the next 20 years. Ground-based lasers are proposed for the first applications. The reasons are that a minimum of space-based laser problems would have to be solved. Emphasis would be concentrated on the receiving and conversion equipment. It is anticipated that large amounts of DOD-sponsored technology would be exploited. Many of the adaptive optics problems encountered to make atmospheric transmission without too much beam divergence would be common to later requirements of the space-based relay system. Early ground deployment would not have to address the hazards of a loss of pointing as far as ground damage is concerned. Safety of operations could be demonstrated.

Between 1990-2000, space-based laser relays could be deployed to redirect the ground-based laser beam to the desired point of application. Later in the decade, low- and medium-power space-based laser systems would be deployed. The lightweight optics systems would be qualified in the previously deployed laser relays, and the laser power system and heat rejection systems would receive the required emphasis. If a large deviation from DOD-type lasers was required, the change and economics could more credibly be justified from previous applications experience with ground-based systems.

Post-2000 deployment of space-based GW laser systems is projected. Earlier deployment of GW laser booster systems is projected. NASA planned future activities are

- Electron storage ring laser
- Solar pumped laser
- Laser-to-electric energy conversion
- Laser-augmented chemical propulsion
- Laser propulsion
- Visible lasers
- System studies

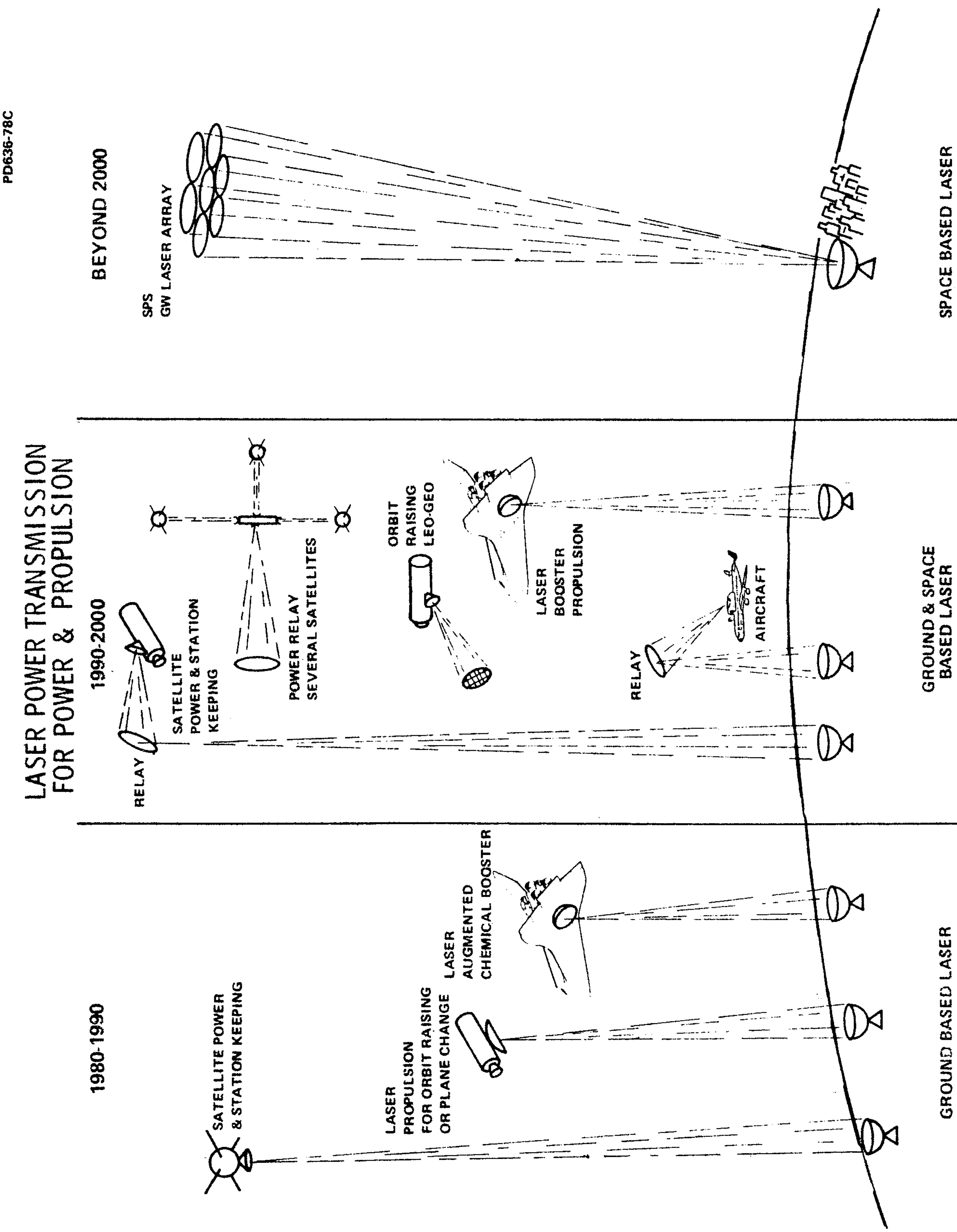
PD636-78C
LASER POWER TRANSMISSION
FOR POWER & PROPULSION
1980-1990
1990-2000
BEYOND 2000
SATELLITE POWER
& STATION KEEPING
LASER
PROPULSION
FOR ORBIT RAISING
OR PLANE CHANGE
LASER
AUGMENTED
CHEMICAL BOOSTER
GROUND BASED LASER
RELAY
SATELLITE
POWER & STATION
KEEPING
POWER RELAY
SEVERAL SATELLITES
ORBIT
RAISING
LEO-GEO
LASER
BOOSTER
PROPULSION
RELAY
AIRCRAFT
GROUND & SPACE
BASED LASER
SPS
GW LASER ARRAY
SPACE BASED LASER

Figure 1.

SPACECRAFT ACTIVE THERMAL CONTROL TECHNOLOGY STATUS

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SUMMARY

The primary means for rejecting heat from manned spacecraft while on-orbit has been through a space radiator system which is mounted on the vehicle and which rejects heat from a fluid circulating through it by radiation to the space environment. The Shuttle Orbiter heat rejection system exemplifies this existing state-of-the-art. Radiator systems for all foreseeable future space missions will need to be compactly stored during launch and subsequently deployed in orbit. In addition for orbital power system missions, they will need to operate for time periods over wider heat load ranges, and possibly at temperature levels which considerably exceed the life capabilities of existing fluid circulating systems. Therefore, the overall goal and objective of technology development effort has been to develop radiator heat rejection systems that meet these basic requirements.

Four separate advanced space radiator concepts have been pursued in an integrated effort to develop multi-mission use, low-cost heat rejection systems which can overcome the limitations of current radiator systems and meet orbital power system type mission requirements. The first approach that has been pursued is a wide-heat-load-range, modularized space radiator system. The modular radiator system has been designed to satisfy wide heat load ranges by use of controlled fluid stagnation. The stagnation control method eliminates the usual radiator fluid freezing point operational limit by providing controlled freezing and thawing of the radiator fluid. The second approach that has been pursued is a spacecraft heat rejection subsystem that can be easily deployed in orbit in order to minimize the vehicle integration requirements of providing heat rejection to future spacecraft. The subsystem is designed as a compact SHRM (self-contained heat rejection module) which provides sufficient flexibility within its design to accommodate a wide variation in spacecraft heat loads and cooling temperature requirements. The third approach pursued also provides heat rejection capability without being dependent on vehicle area. It is a lightweight, flexible fin radiator system which can be deployed and thus is not a "slave" to vehicle configuration since it can be compactly packaged and attached to a vehicle port. The technology established for development of flexible deployable systems using plastic films was extended to develop a deployable radiator which uses a flexible, highly conducting composite material (i.e., teflon film with silver wire mesh). The fourth approach that has been pursued provides a radiator which does not require a circulating coolant on the radiator panel and thus particularly applies to very long duration missions where long life reliability is an overriding design parameter. This radiator panel concept uses heat pipes, which minimizes uses of high-cost, low-reliability mechanical-dynamic components and maximizes

meteoroid protection. The heat pipe radiator system has been designed to use modular "building blocks" to satisfy the varying heat rejection requirements of future spacecraft.

INTRODUCTION

For a self-contained spacecraft, there are only two fundamental techniques available for actively controlling the dissipation of waste energy from the vehicle: (1) emit the energy in the form of thermal radiation and (2) reject the energy to some form of mass which can be jettisoned overboard. The second technique is useful for short missions and for supplemental and emergency uses on long missions, but weight penalties preclude its use as the primary method of heat rejection on long missions. The expulsion of mass in the form of water vapor has been used as the sole means of actively controlling heat rejection on the relatively short missions of the Mercury and Lunar Module vehicles. This is also the method employed by the space suit systems used for extravehicular activity. For the longer Gemini, Apollo Command/Service Module, Skylab, and Shuttle Orbiter missions a space radiator is used as the primary active method of heat rejection with water evaporators used only for supplemental and re-entry heat rejection.

In order for future heat rejection systems to have the relatively universal applicability necessary, the system must be designed to overcome several current radiator design limitations. Specifically, the maximum heat rejection capability for current systems is limited by several factors, including the severity of the external thermal environment, the temperature of the internal spacecraft heat sources, the availability of radiator surface area on the vehicle, the reliability of a circulating fluid system, micrometeoroid protection requirements, and available surface coatings. Thus, the primary technical objective of development activity has been to develop a low-cost space radiator system that can overcome one or all of these limitations. In addition, it is necessary to develop improved radiator control techniques that can allow the system to operate over a wide heat load range.

The primary goal of active thermal control development activity has been to develop a radiator system approach which is not integral with the spacecraft skin, and thus, can be separately developed and manufactured. The independent development approach has significant potential to reduce spacecraft development costs by (1) minimizing development and certification testing required by each different space mission, (2) providing longer production runs, (3) simplifying integration between the heat rejection system and the vehicle, and (4) providing for the reuse of heat rejection systems which are returned from orbit. This minimum-cost concept, in conjunction with the required technical improvements, can provide Orbiter payload heat rejection, as well as heat rejection necessary for spacecraft operating for very long duration missions, such as the orbital power module. Thus, the current development activity has applicability to a very broad range of future possible missions and could result in significant overall cost savings during spacecraft development and operations. The following discussion will briefly describe the four separate advanced space radiator concepts that have been pursued in an integrated effort to develop multi-mission-use, low-cost heat rejection systems which can overcome the limitations of current radiator systems. These concepts were not considered to be necessarily competitive alternatives, but unique design approaches which have the combined capability to meet a wide range of specific advanced mission requirements. Also, in order to establish a firm background to compare the the advanced space radiator concepts, the Orbiter active thermal control system will be briefly described.

SHUTTLE ORBITER ACTIVE THERMAL CONTROL

The Orbiter heat rejection system exemplifies the existing state-of-the-art in thermal management (reference 1). The Orbiter ATCS consists of two simultaneously operating coolant loops, using Freon-21, which transport heat from the Orbiter subsystems and payloads through liquid heat exchangers and pin-fin coldplates to the heat sinks. The locations of the major ATCS components are widely distributed throughout the Orbiter (see figure 1).

During on-orbit operations, heat rejection is accomplished primarily by the space radiators (see figure 2), supplemented by water evaporation. Use of water is required because of the limited radiator area available. The radiators are designed to reject heat in all orbiter attitudes. However, even with the best available surface coating and use of all available area, there are some attitude and heat load combinations where the environmental absorbed heat (solar, albedo, and Earth emission) on the radiators prevents the cooling of the Freon-21 returning from the panels to the required return temperature. Water evaporation is automatically activated to cool the Freon-21 to the required return temperature under this maximum load condition. In addition to maximum heat load limitations use of parallel tube flow in the radiator panels (I-tube panel) requires that a minimum heat load be applied in some attitudes to avoid freezing the coolant in the panels. A temperature control assembly controls flow through a variable position flow control valve which maintains the mixed radiator outlet to the required set point temperature by mixing hot bypass flow with cold flow from the radiators (see figure 3).

The Orbiter radiator heat rejection system has up to 8 radiator panels attached to the inside of the PBD (payload bay doors). The two forward panels on each side of the vehicle are deployed away from the doors to increase the surface area available for heat rejection. As previously mentioned, Freon-21 flows through two independent radiator coolant loops. The four radiator panels in loop 1 are installed on the left side of the Orbiter. The four panels in loop 2 are installed on the right side of the Orbiter. Since the forward panels reject heat from both sides of the panel, they are designed with flow tubes attached to each face sheet. There are 68 tubes in the forward panel, 34 on each face sheet. The aft panels remain attached to the aft doors; consequently, they radiate from the upper surface only, and thus are designed with 26 tubes attached only to the upper face sheet.

The radiator panels are constructed of aluminum honeycomb bonded to .0043 cm (0.011 inch) aluminum face sheets with metlbond 329-7 adhesive. High density honeycomb core is used at hardpoints. Aluminum tubes are imbedded in the honeycomb and bonded into the structure to provide parallel Freon-21 flow paths within each panel. The radiating surfaces are coated with a silver teflon coating which provides a low absorbtance of solar flux (α = 0.10) and a high thermal emittance (ε = 0.76). The coating is applied in 10.2-cm (4-inch) wide strips and bonded to the aluminum with a "permacel" adhesive. Heat rejection is effected by transmitting sensible heat from the fluid to the aluminum tubes by convective heat transfer, then conducting it to the radiating surface where it is radiated to space. Flex hoses traveling in hose reel assemblies that can accommodate the open and closed positions of the payload bay doors are used to transfer the Freon-21 to the radiators.

ADVANCED HEAT REJECTION DEVELOPMENT STATUS

Modular Wide-Heat-Load Fluid Radiators

For early spacecraft with missions of sufficient length to require a space radiator, the limited mission objectives and operations restricted the required operating heat load range. It has been recognized that for large Earth-orbiting vehicles, the heat rejection system would need to accommodate a much wider range of operating conditions. Some of the sources for this increased range requirement are normal operations with varying experiment payloads and heat sources, planned maintenance activities, and planned quiescent periods.

A fluid space radiator with a fixed area exposed to space has three limitations on the range of heat loads over which the panel can operate: high load, low load, and transient response . The maximum heat rejection of the panel is limited by the panel area, the radiant environment, and the temperature at which the heat transport fluid receives the waste heat. Unless a refrigeration scheme is used, the radiator must operate at a temperature below the temperature of the equipment rejecting heat to the fluid. Since the radiator system must be sized for the high load conditions, and panel area required is relatively insensitive to the low load control technique used, the effort involved with extending radiator heat load range concentrates on minimum load requirements and transient response capabilities. The minimum heat rejection of a panel is limited by the freezing point of the heat transfer fluid and the control technique used. Variations in heat rejection may be limited by the transient response of the system to a change in heat load.

The modular wide-heat-load-range fluid radiator developed (reference 2) achieves heat load control by varying the flow split between a "prime" and "bank" circuit as shown for a typical panel arrangement on figure 4. The flow split can be controlled by a valve which senses the mixed outlet of the prime and main circuits and compares it to a desired set point temperature. During periods of low load, the majority of the flow is routed to the prime tube of the panel and the bank is allowed to stagnate (freeze), thus reducing the effective panel area. As the load increases, more flow is routed to the bank, and the panel begins to destagnate (thaw) from the inside out (i.e., the shortest tubes destagnate first).

The selection of Freon-21 as the fluid for the wide-heat-load-range radiator is based on the following: (a) broad temperature range between freezing and boiling points with operation at a reasonable pressure in the 4°C to 38°C range, (b) good pumping power and heat transfer characteristics, (c) low viscosity at temperatures just above the freezing point, and (d) a sharp well-defined freezing point. With these characteristics, as soon as a tube thaws, flow quickly redistributes itself to provide a balanced share of the flow in the bank of parallel flow tubes.

The three modular panel configurations that have been tested are shown on figure 5: triangular, U-tube rectangular, L-tube rectangular. The triangular panel test provided the fundamental characteristics of the design and led to the U-tube design. A system of eight U-tube panels has been tested to (1) prove the modular design concept by demonstrating the panel flexibility and "building block" approach of the system design, and (2) demonstrate system performance over a full range of heat loads, environments and flow configurations. The U-tube radiator panels tested consisted of eight 1.82 m x 3.66 m (6 ft x 12 ft) flat panels. Each panel consisted of extruded tubes welded to .008 cm (0.02 inch) aluminum sheet on 15.24 cm (6.0 inch) centers in a U-shaped pattern (see figure 4). The "U" shaped flow passages (tubes) on each panel include flow control orifices at the inlet of each tube to maintain the proper flow distribution among the tubes. The wide heat load range is obtained by routing the majority of the flow to either the innermost prime tube or the bank of remaining tubes, thereby changing the panel radiation effectiveness.

The U-tube radiator tests encompassed a full range of external thermal environments, vehicle internal heat load generations, and radiator panel plumbing arrangements. In addition, various radiator control temperatures were evaluated, as well as radiation from one and both sides of the panels. Under all test conditions, the radiator system rejected the proper heat load and maintained the control temperature within expected tolerances. At low load and cold external environments, the radiator panels flow stagnated as designed and subsequently recovered the capability to reject high heat loads under conditions where the imposed heat load rate of change was several times faster than expected for a typical vehicle (Orbiter). The assessment of the different plumbing arrangements, which encompassed flow arrangements from all eight panels in parallel and all eight panels in series to several combinations in between those extremes, demonstrated the complete modularity of radiator panels. No flow distribution or flow instability problems were encountered under any test condition, which included freeze/thaw cycles under transient environment asymmetries and transient heat loads. A high to low heat load ratio of 50:1 was demonstrated for these panels.

Modular radiator panels containing a bank of L-shaped tubes (all manifolded together) have also been tested (see figure 6). The wide-heat-load-range capability is obtained on this panel by varying the flow between the radiator tubes and a bypass line, instead of varying the flow between a prime tube and a separate tube bank. At low heat loads, all the radiator tubes receive less flow, causing the flow in the bank of tubes to successively stagnate by freezing (from the longest to the shortest tube) and thus progressively reduce the overall radiator effectiveness. The innermost tube and bypass valve are sized such that the innermost tube never stagnates, even when the rest of the tubes are effectively bypassed. This approach has been termed inherent stagnation. The inherent-stagnation design eliminates the requirement for additional panel supply and return lines that would be required by a separate "prime" tube, thus providing design simplification and weight savings. If the radiator is exposed to the worst cold environment, the Freon in over half of the bank of tubes freezes. The continuous flow provided to the innermost tube is sufficient to insure that the stagnant radiator tubes can be thawed as the heat load increases. Both of the L-tube panels tested exhibited good heat rejection characteristics with high fin effectiveness and tube to face sheet conductance. An average fin effectiveness of 0.96 was measured. Low load stagnation/destagnation operation was demonstrated with the inherent stagnation method.

In summary, a wide-load-range fluid radiator concept has been developed. The key factor in the extremely fast transient response of this design is the combination of panel design and fluid selection. The proper selection of fin thickness, tube spacing, and manifold design complement the selection of Freon-21 which has an extremely high viscosity at temperatures just above its freezing point. As soon as a tube thaws out, the panel flow pattern is re-established to provide an even share of the flow to the tube. This results in a rapid rise in the temperature of the tube with a correspondingly high temperature difference between this tube and the adjacent frozen tube. The high temperature difference provides a high potential for heat transfer to thaw out the next tube. This phenomenon is repeated as each tube thaws out sequentially.

SHRM (Self-Contained Heat Rejection Module)

The self-contained heat rejection module (SHRM) has been conceived for use on future spacecraft that will be carried into orbit by the Shuttle. The SHRM (see figure 7) is a separate module which contains the necessary equipment to effect heat rejection in orbital environments. The basic goal of the SHRM program was to develop the technology for and to demonstrate for the first time a full-scale heat rejection system that contains deployable radiators and integral flow control equipment (reference 3). Fluid swivels provide fluid transfer between the deployable radiator panels since compactness of volume and envelope was a design requirement. Heat transfer to the SHRM from the heat source is through a contact heat exchanger. The contact heat exchanger permits thermal coupling or uncoupling of the SHRM to a heat generating payload by a mechanical joint rather than by fluid interconnection. This will facilitate system installation since neither the SHRM or the payload fluid system needs to be broken into and reserviced. A high degree of flexibility for multiple mission support was achieved by incorporating a dual-mode system approach. The "dual mode" concept refers to a spacecraft heat rejection system which operates as a conventional, low-temperature, liquid phase radiator system during periods where minimal or nominal heat rejection is required. During operations involving severe external environments or high power requirements, a vapor compression system is automatically switched on to a refrigeration/high-temperature radiator mode.

A schematic of the system is shown in Figure 8, with its two independent parts, a high-temperature radiator system and a dual-mode refrigeration/radiator system. This independent system approach has three distinct advantages: (1) separate controls can be used for each system, (2) only half of the system can be used for some applications (i.e., for high-temperature applications the liquid radiator would be used; for low-return-temperature applications only the refrigeration unit would be used), and (3) parallel flow radiator panels can be used on the condensing radiator and wide heat load tube layouts as discussed in the prior section of this paper can be used on the liquid phase radiators.

Several approaches for deploying the SHRM radiators were considered including foldout hinged panels, telescoping devices, mechanical linkages, rotating panels, and scissor-type deployment mechanisms. The scissor mechanism was finally selected because of the existence of a qualified and proven deployment mechanism used for the Apollo Telescope Mount solar cell array deployment system. One of these units was obtained from NASA-MSFC, and radiator panels were placed on it in lieu of solar cells.

With any concept of deployable radiators some form of relative motion between panels must be accommodated in the plumbing system. For a scissor mechanism, rotary motion is required. Several concepts were considered, including flex hoses, coiled tubes, and fluid swivel fittings. The decision to utilize the qualified ATM solar array deployment mechanism which has a very restrictive space envelope influenced the choice toward the fluid swivel fitting. An Aeroquip Omniseal was selected for the fluid swivel, because it uses teflon for sealing, which is compatible with Freon-21, and a stainless steel spring to provide resilience at low temperature. A swivel fitting was designed around this seal (see figure 9). The fitting has been successfully tested at -140°K with no detectable leakage for both static and dynamic conditions in a vacuum environment.

Several approaches were considered for the contact heat exchanger including flat coldplates, irregular coldplates with sawtooth or pin surfaces, heat pipes, and stacked coldplates. The stacked coldplate approach was selected because of its favorable envelope requirements and the relatively small contact force required to achieve the necessary heat transfer. In this design the two sides of the heat exchanger are formed by coldplates which are connected to a common manifold (see figure 10). The contact heat exchanger is mated by sliding the two sides together in a manner similar to operation of a radio tuner. Bolts are used to apply pressure to the two sides of the contact heat exchanger to provide a pressure of up to 2000 KN/m^2. This pressure will provide contact conductance coefficients of 4600 $J/s\ m^2\ K$ when an interstitial filler of conductive silicone grease is used.

The four SHRM panels were constructed of aluminum tube extrusions seam welded to 0.00102-m (0.040-in.) aluminum sheets at 0.16 m (6.3 in.) internals. The radiator panel size to fit on the ATM frame was 2.57 m (101.25 in.) by 2.37 m (93.25 in.). These panels provide a total radiating area of 48.7 m^2 (524.5 ft^2) since they radiate from both sides.

Extensive thermal vacuum testing of the SHRM has been completed (reference 4). The overall objective of achieving the first full-scale demonstration of a deployable radiator system and mapping the heat rejection capacity of the first dual-mode radiator refrigeration system have been accomplished during this testing. Multiple thermal vacuum deployments and retractions were successfully conducted and maps of the heat rejection capacity in both modes were generated for two difference return temperatures, 2°C and -12°C (35 and +10°F). These maps indicated distinct operation ranges for the two modes as a function of heat load and thermal environment. Component evaluations based on the test results indicated all the components performed as expected, except for one of the eight fluid swivels. A redesign of this component will be necessary for use in a dual-mode system; however, it is completely acceptable for use in a pumped liquid system which contains no refrigerant oil.

Flexible Deployable Space Radiator

Flexible radiator systems utilize panels made of composite flexible fin material to reject heat and can be "rolled" up, folded, or compacted during storage and deployed for orbital operation. Because of their flexibility, these radiators are easily adapted to an existing vehicle since they can be stowed in compact units which are not susceptible to damage by dynamic loads during launch. Since flexible radiators do not require extensive structural support, they are inherently lighter in weight than rigid panels. Also, the same flexible radiator design can be used in several different missions so that developmental and integration costs are reduced.

Two designs have completed the feasibility demonstration phase of development: a soft-tube concept which unrolls to deploy and a hard-tube concept which deploys into a cylinder shape using the spring force of helically wound aluminum transport fluid tubes (reference 5). Transport fluid temperature control is by either a bypass system like that previously discussed for the rigid panels or by partial extension to regulate the radiating area.

Soft-Tube Concept Description: A typical soft-tube flexible radiator system is shown in figure 11. The radiator panels incorporate flexible tubing to allow the "wings" to be rolled and unrolled from a cylindrical storage drum. Panel size, arrangement and single or multiple panel configurations are dependent on heat load, vehicle interface and storage requirements. Radiation is from both sides of the panels.

The composite flexible radiator panel fin material and soft-tube arrangements are shown in figure 11. The composite has outer layers of teflon which provide structural strength and resistance to ultraviolet degradation and also a high radiating surface emittance combined with a low solar absorbtance. A highly conductive wire mesh is fusion bonded to the interior surface of one layer of the teflon to provide a high lateral conductance. Silver metal can be vapor deposited on the inner surfaces to reflect incident solar radiation. The silver/teflon layers are then adhesively or fusion bonded to the tubes in a sandwich construction. The transport fluid tubing diameter and spacing on the panels were selected to provide minimum system weight including the effects of pumping power penalty and structural mass for protection from meteoroid penetration. The resulting radiating fin effectiveness is in excess of 0.85. The baseline design uses a transport fluid (coolanol 15) which has long term compatibility with the flexible tubing and results in an operating temperature range from about -29°C (-20°F) to 85°C (185°F).

Deployment forces for the soft-tube flexible radiator system are provided by a nitrogen gas pressurant which inflates two manifolds, one on either edge of the panel, causing the panel to unroll from the storage drum. Panel retraction forces are provided by flat, preloaded "watchsprings" which are incorporated into the gas deployment manifolds. Heavier deployment/retraction mechanisms such as the Storable Tubular Extendable Member (STEM) may be substituted for gas inflation manifold deployment where precise positioning of the deployed panel is desired.

A soft-tube article measuring 1 m x 1.8 m (40 in. x 72 in.) was fabricated. Tests were conducted in thermal vacuum conditions at equivalent radiating sink temperatures ranging from -18°C (0°F) to -190°C (-310°F) with coolanol 15 transport fluid inlet temperatures from 32°C (90°F) to 71°C (160°F). The test verified heat rejection capability and demonstrated the design temperature distribution through the tube wall, glue line and composite radiating fin. Repeated deployment and retraction under thermal vacuum conditions verified the gas deployment system and the mechanical integrity of the construction. Testing at partially deployed positions showed that heat rejection may be controlled by this technique. Subsequently, a full-scale prototype wing of the soft-tube concept has been fabricated and is currently under test. Both gas pressurization/watchspring and STEM deployment/retraction approaches will be tested.

Hard-Tube Concept Description: One hard-tube flexible radiator concept that has been fabricated and tested incorporates aluminum tubes with a flexible composite fin material. This typical hard-tube system is shown in figure 12. The cylindrical panel configuration incorporates the aluminum tubes in a helical pattern so that the panel can be compressed for storage. The composite flexible radiator fin material and tube configuration arrangement described above and in figure 11 for the soft-tube design is identical for the hard-tube design except the tubes are aluminum. The aluminum tubes allow for greater meteoroid protection, a wider fluid temperature range of -96°C (-140°F) to 149°C (300°F) and greater fluid system operating pressures. As with the soft-tube concept, the tube spacing and diameter were selected to provide a minimum weight system. Overall radiating fin effectiveness is again in excess of 0.85.

The helically coiled aluminum tubes provide the forces necessary for deployment of the hard-tube concept. A motor-driven cable or boom compresses the coil to retract the system. As with the soft-tube design, a STEM may also be utilized when precise positioning of the panel is required.

A hard-tube test article was fabricated, which measures 0.71 m diameter by 1.14 m long (28 in. x 45 in.). Thermal vacuum tests were conducted with Freon-21 fluid inlet temperatures ranging from 16°C (60°F) to 71°C (160°F). The tests verified deployment, heat rejection, temperature distribution, mechanical integrity, and the capability to regulate heat rejection by partial deployment.

Subsequent work is now underway to fabricate and test a full-scale prototype wing of a hard-tube flexible radiator panel designed for long duration mission applications. It is constructed with steel transport tubing and metal bellows manifolds to accommodate Freon-21 transport fluid. The metal bellows will allow this hard-tube concept to be rolled and unrolled from a cylindrical storage drum as previously discussed for the soft-tube system (see figure 11). Expanded silver metal and teflon will be fusion bonded to the transport tubing to form the radiator fin. Micrometeoroid barriers are being designed for the manifolds and transport tubing. The deployment system will employ a Storable Tubular Extended Member (STEM) and a spring-loaded storage drum.

Modular Heat Pipe Radiator

As previously discussed, current manned spacecraft reject their waste heat by mechanically pumping fluid through a space radiator system which radiates the heat to space. As such, reliability is relatively low since system operation is vulnerable to failure from a single meteoroid penetration of a radiator fluid tube. High reliability for long duration missions can be achieved, but the resulting space radiator system is generally heavy because of the required redundant plumbing, pumping, and valving hardware. Heat pipes offer an attractive alternative for eliminating many of the single point failures in a space radiator system. The development effort pursued uses a radiator panel concept which utilizes multiple heat pipes. Therefore, the loss of a single heat pipe is not catastrophic and meteoroid protection is maximized.

The basic heat pipe radiator concept couples a fluid heat source to a radiative heat sink through an intermediate array of heat pipes, which are designed to maximize heat rejection per unit of radiator system wet-weight. The panel has the capability of being thawed from a frozen state without the benefit of a warm environment. This permits the panel to freeze during low load conditions and results in a wider operating range between maximum and minimum loads.

The first heat pipe radiator panel tested consisted of six L-shaped ammonia feeder heat pipes welded to the condenser section of a variable conductance heat pipe (VCHP) header (reference 6). The evaporator section of the header was attached to a finned fluid heat exchanger and the six feeder pipes were bonded to a 1.2 by 2.4-m (4 x 8 ft) radiating fin. Although the VCHP header performed below its design capacity, other test results were encouraging. The operational feasibility of a heat pipe-to-fluid heat exchanger was established, and the panel feeder heat pipes were very effective in isothermalizing the radiating fin.

Subsequently, a prototype modular heat pipe radiator panel was designed and fabricated. This flight-weight panel is a segment of a multi-panel system concept that consists of individual radiator modules that can be grouped in building-block fashion to satisfy a given heat rejection requirement. The ultimate success of this type of system would result in many significant advantages to future spacecraft including reduced development/test costs, wide flexibility of application, and manufacturing economies.

The prototype heat pipe radiator panel concept is illustrated in figure 13. Each of the panel feeder heat pipes is an identical sub-module of the panel and comes attached to its own radiator fin and fluid header sections. Thus, any desired panel area can be formed by simply piecing the required number of heat pipe sub-modules together, with the header tubes lap welded and the radiator fins spot welded to one another. The feeder heat pipes are purely isothermalizers and as such can be either longitudinally grooved pipes or artery designs. The former is simpler, but the latter type (a spiral artery) was used in the prototype since they are much less sensitive to adverse tilt during ground tests. Another advantage is their higher transport capacity with ammonia, the selected working fluid (254 W-m versus 76 W-m). This permits longer condensers and hence, radiator fin lengths to be used in the panel design, which results in fewer heat pipe sections. The artery pipes, with their fine circumferential grooves, also have higher evaporator film coefficients (1.4 versus 0.7 W/cm^2 deg C), which increases the effectiveness of the heat exchanger section and results in reduced fluid to panel temperature drops. The prototype panel radiator area is 6.3 m^2 (68 ft^2). It contains 11 ammonia spiral artery heat pipe segments, spaced every 28 cm (11 in.), and is designed to reject 2200 W.

An important consideration in the design of the prototype heat pipe panel was the ability of the ammonia heat pipes to be thawed from a completely frozen state in a zero absorbed heat environment. Such a requirement could result when a spacecraft sustains a dormant operating mode coincident with a very cold environment, then resumes normal operation while still in the same environment. The test results from the first radiator panel test showed that thawing was always promoted by raising the environment above the ammonia freeze point, but the test results were inconclusive as to whether the frozen ammonia pipes could repeatably extract enough energy from the warm fluid stream to thaw themselves in a cold environment. However, thawing can be assured by maintaining a high-temperature boundary along a frozen condenser section and relying on cross-fin conduction to supply the needed energy to thaw the first pipe. The other heat pipes would then be sequentially thawed in a similar manner. Therefore, a low-freezing-point heat pipe has been included on the prototype panel as one of the feeder pipes to insure that at least this one pipe would remain operational in the coldest cases. The general requirements for the low-freezing-point heat pipe are (1) it must have a relatively poor coupling to the fluid when the ammonia heat pipes are frozen, the inlet temperature low, and the environment cold, in order to minimize panel heat losses and promote good high load/low load ratio; (2) it must also have a good enough coupling to the fluid to maintain the minimum required boundary temperature when inlet temperature and flow rates are raised. Propane was selected for the low-freezing-point heat pipe, because it meets these requirements with a very low freezing point, -187°C (-305°F).

Thermal vacuum testing of the prototype modular heat pipe radiator panel has verified its design (reference 7). Two separate test series were run; first, normal mode performance and then freeze/thaw performance. For the normal mode, steady-state performance maps were obtained; panel heat rejection and temperature profiles were measured for various combinations of absorbed environment, inlet temperature, and flowrate. The main objective of the freeze/thaw tests was to determine if a frozen panel could be thawed in a zero absorbed environment by simply increasing the fluid inlet temperature. The maximum heat rejection recorded for the heat pipe radiator in a near-zero environment was about 2800 W. On a unit basis, this is 420 W/m^2 (39 W/ft^2). Two of the eleven ammonia heat pipes were less effective than the others, since they frequently operated at lower temperatures than the surrounding pipes. Near its capacity limit, the prototype panel had two operating modes, depending on the stability of the fluid inlet conditions. The arteries can be either fully primed with working fluid or partially deprimed. The former provides a 2800-W capacity, and the latter 2200 W. When subjected to cycling inlet conditions, the heat rejection peaks and valleys generally lay between the primed and unprimed steady-state limits. Most of the data indicated a fully primed condition. The low-freezing-point propane heat pipe worked as designed. The frozen panel was successfully thawed in a near-zero environment by increasing the inlet temperature along a controlled ramp.

Two additional prototype heat pipe radiator panels have been fabricated and are included in a three-panel system level thermal vacuum test currently in progress. These three panels will be arranged in various radiator system configurations. Different combinations of the three panels in series and parallel arrangements are being tested to evaluate system interaction. In addition, three smaller single heat pipe radiators are being tested to investigate design improvements in the thermal interface between the heat pipe and the radiator fin (see figure 14).

CONCLUDING REMARKS

Long-term orbital applications in which large amounts of electrical power are generated and utilized will require waste heat rejection beyond the capabilities of existing radiator systems. The optimum, minimum-cost technique of rejecting heat for such applications can be developed based on judicious application and extension of the radiator technology developed over the last 10 years in the areas of radiator deployment methods, flexible/lightweight radiator fins, heat pipe radiators, fluid swivels, and heat rejection control techniques.

The orbital power systems effort must begin with system level heat rejection trades to apply the key techniques that have been developed into an optimum integrated thermal management system. The design effort should directly compare pumped fluid, heat pipe, rigid vs. flexible fins and other appropriate radiator system concepts for the specific orbital power systems mission. Techniques for system level reliability improvements (isolated flow loops, replaceable heat pipes, etc.) must be developed. The effort should integrate such concepts as heat pipe radiators, flexible fin materials, non-metallic tubes (with thin gage metallic tube liners), micrometeoroid barriers, and materials which are not degraded by extremely long exposure to the ultra-violet spectrum. Also, advanced temperature control schemes for providing constant system outlet temperatures over a wide band of heat loads as appropriate for the orbital power system should be an integral part of the effort. The radiator system concept developed must achieve long-life by remaining operational in the micrometeoroid environment of space through on-orbit refurbishment and special design/construction features. The system should accommodate the large size requirement by deployment from a compactly stowed volume.

Panel element tests should be conducted to evaluate fabricability and performance. A representative portion of the full-scale system including the deployment technique should be fabricated and tested to confirm the final design concept. Finally, a flight demonstration program should be established for evaluation of the detail approaches to insure that the real problem areas, such as (a) articulating fluid lines, (b) maintaining flow distribution in large multi-panel systems, (c) maintenance-tolerant designs for in-space repair/replacement, (d) deployment and initial coolant servicing design, (e) temperature control scheme for large surface area radiators, and (f) surface property maintenance on-orbit, have been successfully solved.

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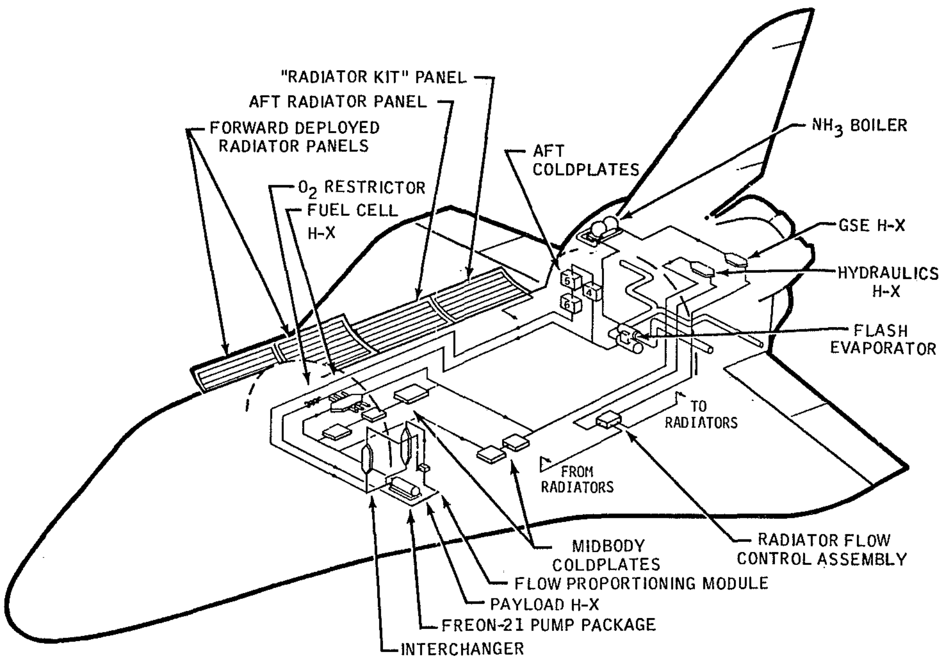


Figure 1. - Shuttle Orbiter active thermal control subsystem.

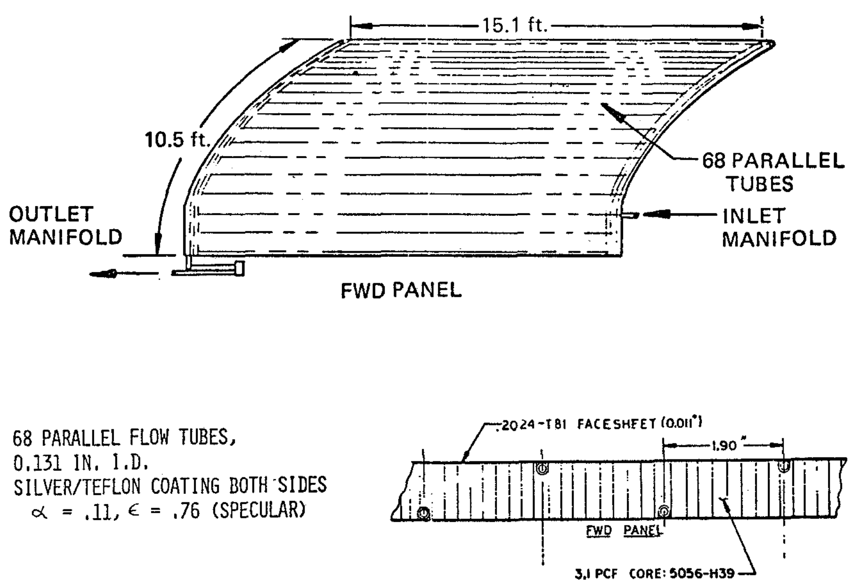


Figure 2. - Orbiter radiator panel physical characteristics.

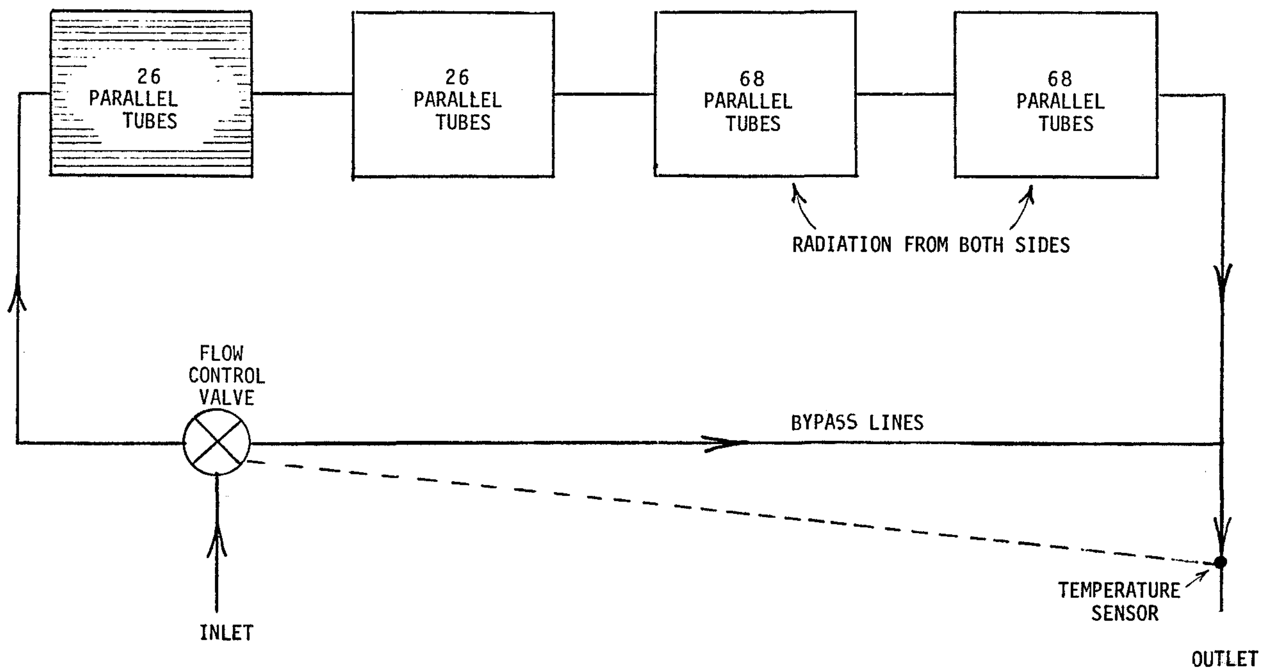


Figure 3. - Orbiter radiator system temperature control.

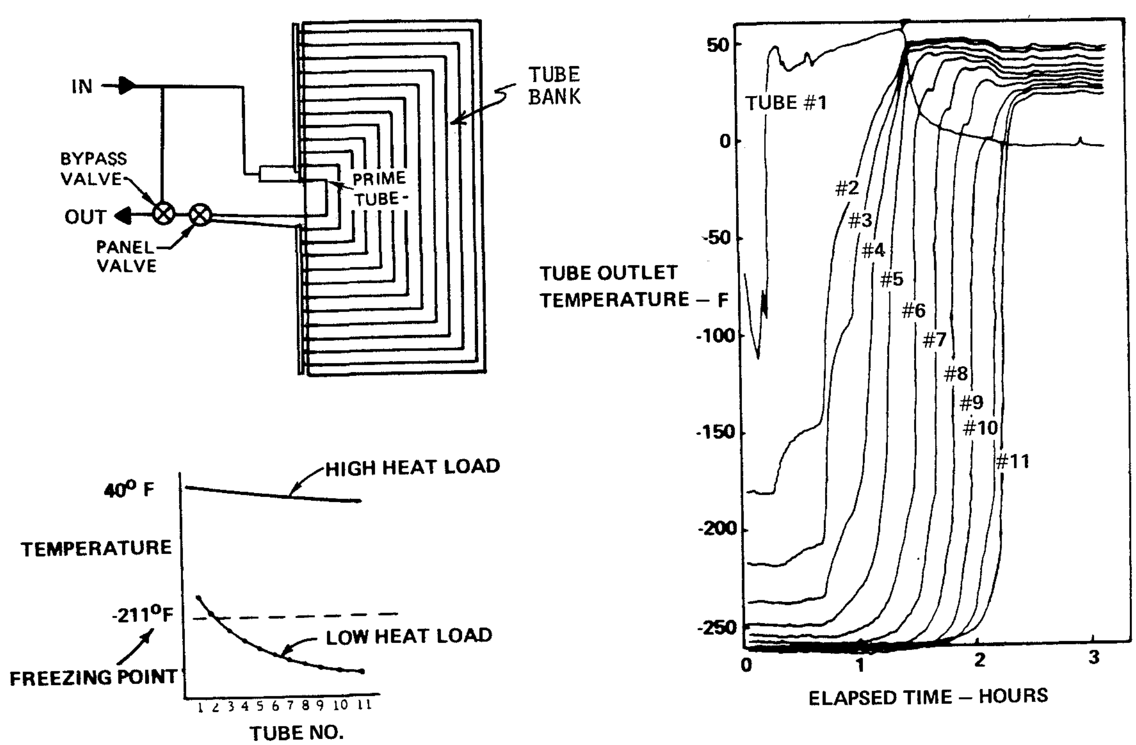


Figure 4. - U-tube wide-heat-load-range fluid radiator.

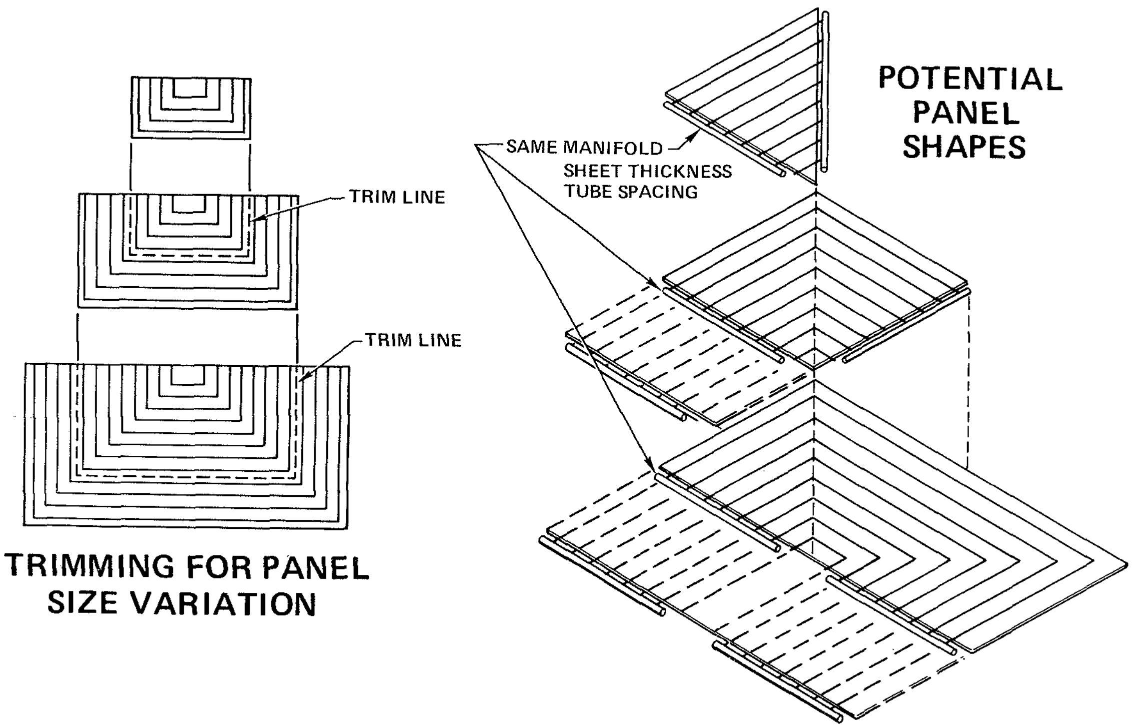


Figure 5. - Wide-heat-load-range fluid radiator modularity.

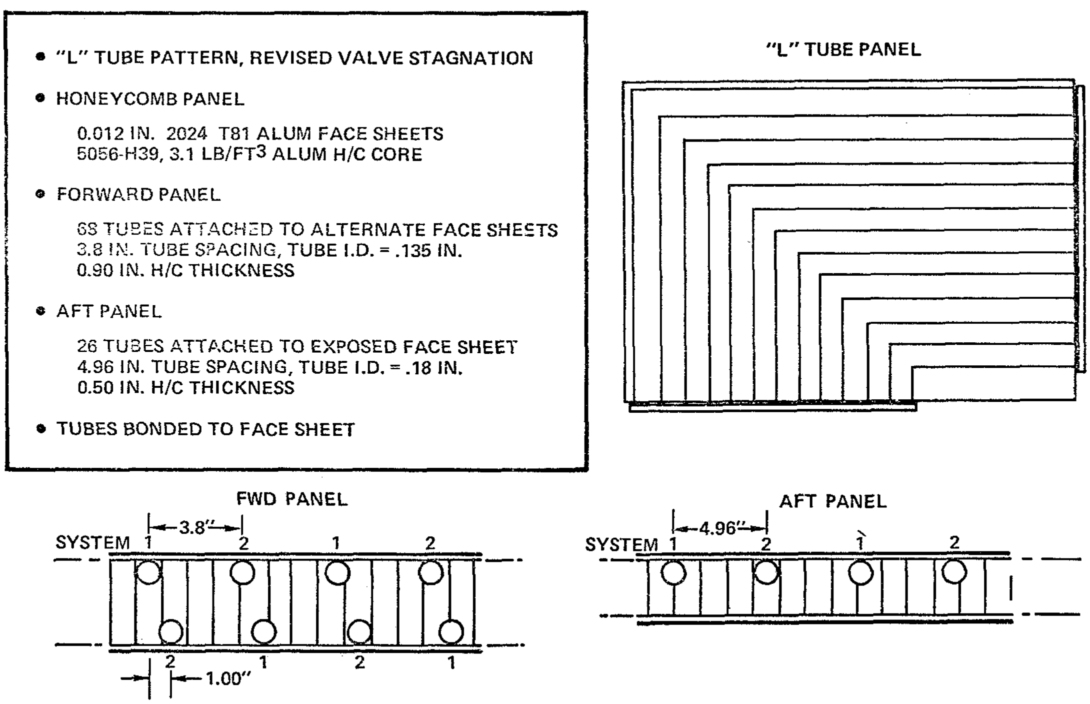


Figure 6. - L-tube wide-heat-load-range radiator configuration.

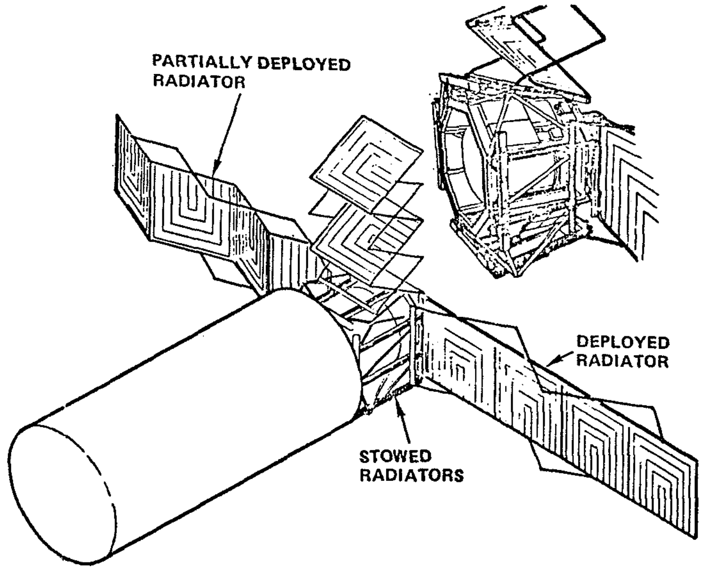


Figure 7. - Self-contained heat rejection module (SHRM).

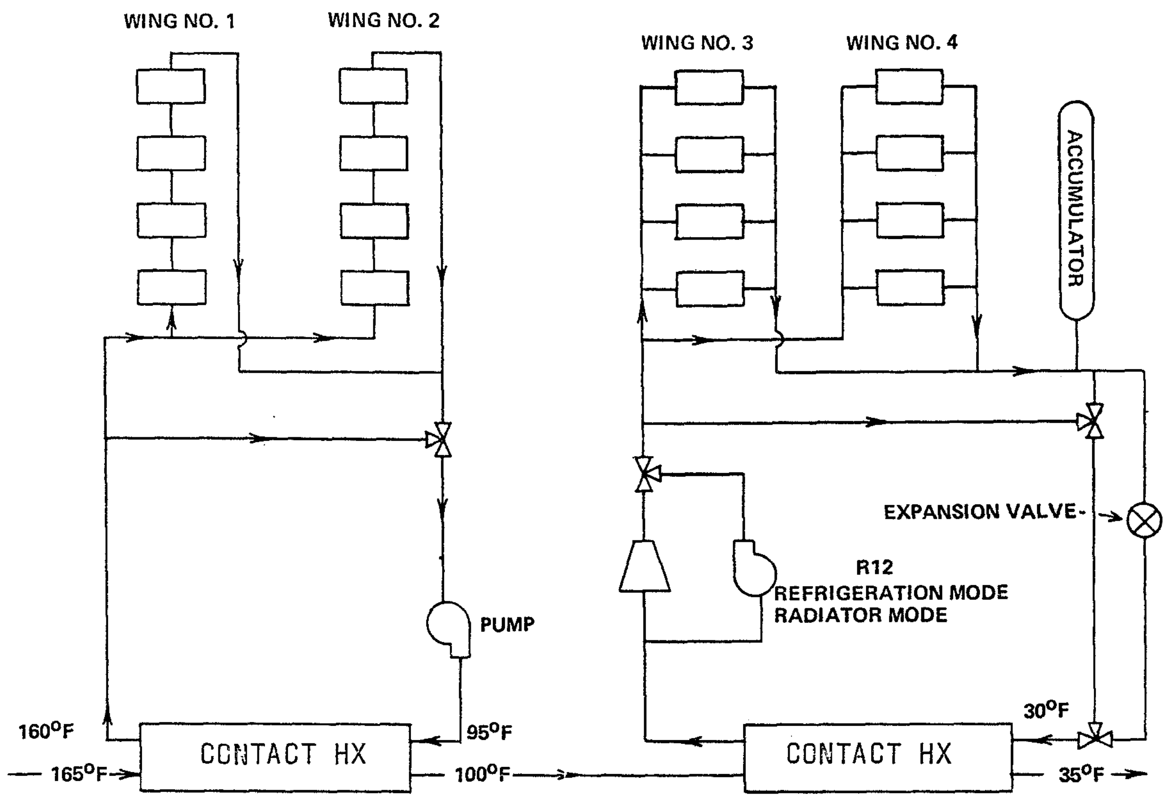


Figure 8. - Self-contained heat rejection module (SHRM) flow schematic.

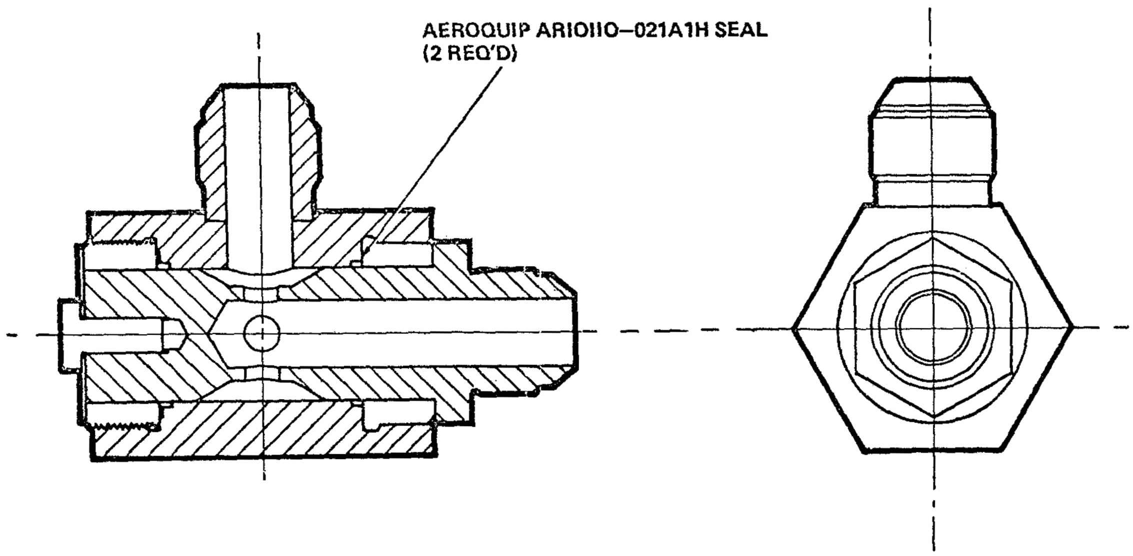


Figure 9. - Fluid swivel for deployable space radiators.

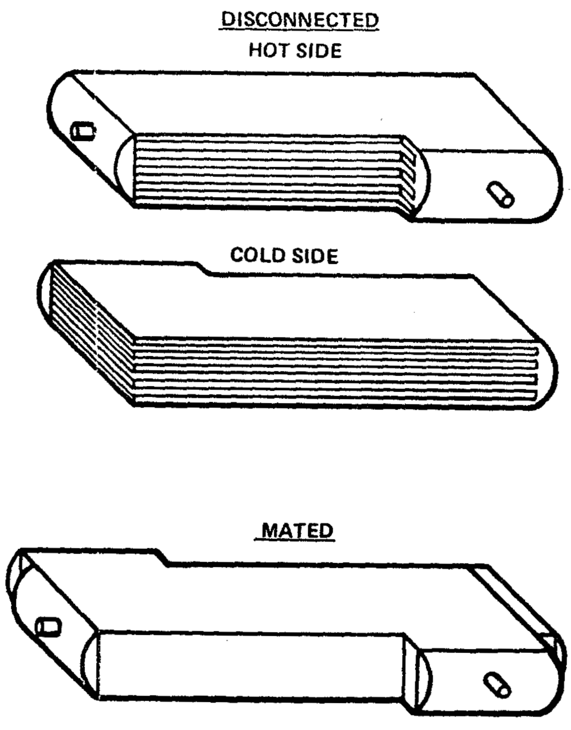


Figure 10. - Contact heat exchangers.

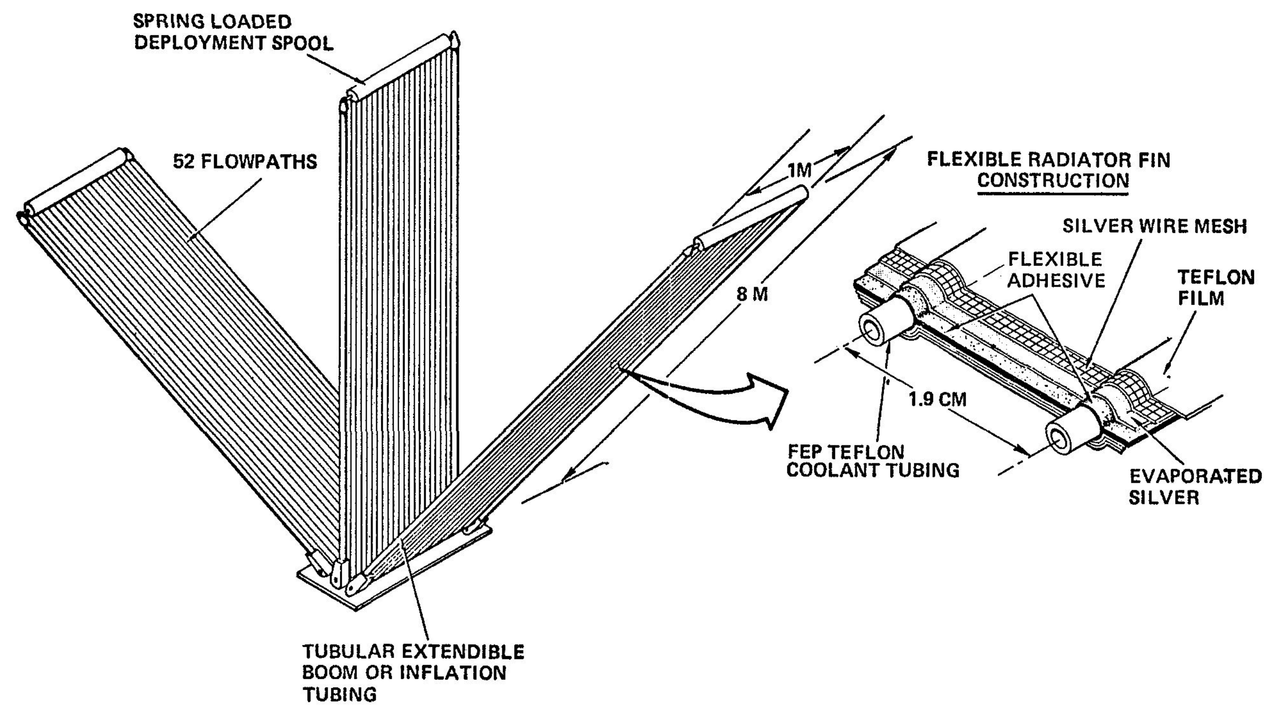


Figure 11. - Flexible-fin/deployable space radiator.

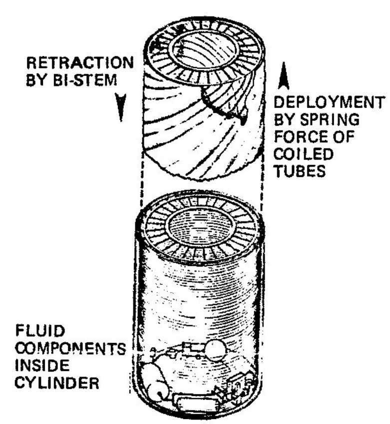


Figure 12. - Hard-tube flexible-fin/
deployable space radiator.

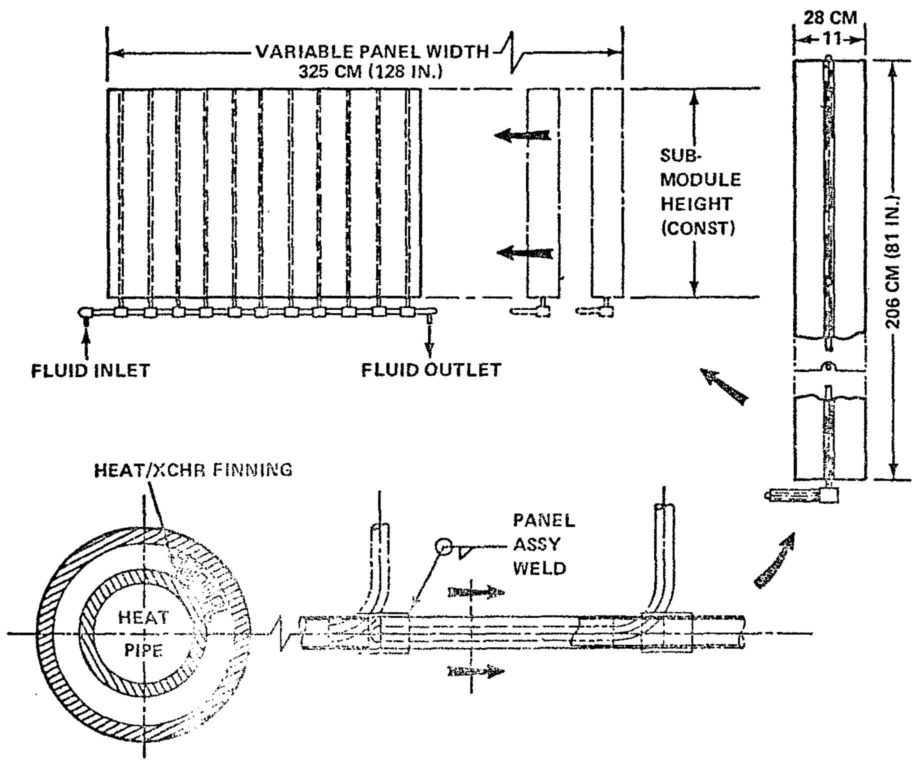


Figure 13. - Modular-heat-pipe radiator.

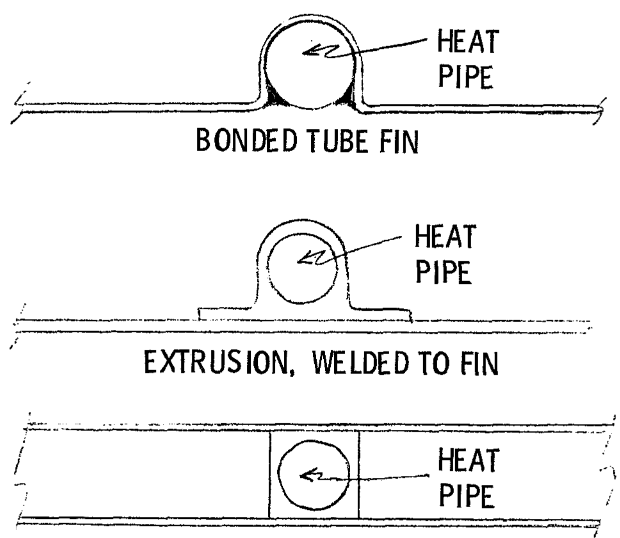


Figure 14. - Heat-pipe/radiator fin interface techniques tested.

POWER MODULES AND PROJECTED POWER SYSTEMS EVOLUTION

Lott W. Brantley
NASA Marshall Space Flight Center

SUMMARY

Photovoltaic, solar thermal, and nuclear power systems are being considered to supply future Earth orbital electrical power requirements. A growth scenario from a 25-kW Power Module in the early Shuttle era to the 5- to 10-GW Satellite Power System in the year 2000 is presented. Photovoltaic systems are presently baselined in this evolution. The Photovoltaic Power System and subsystem growth projections, consistent with this scenario, have been developed and are summarized.

INTRODUCTION

The Space Shuttle and Spacelab Systems are expected to open the door to low-cost space transportation and experimentation. This together with the requirement for low-cost orbital operations is expected to result in the increase in scope, size, and consolidation of Earth orbital operations. These large concentrations of space activity will result in large centralized power modules. The early time frame is driven by NASA support requirements. Later requirements may be dominated by the needs of the industrial or space commercialization sectors. The Satellite Power System in the scenario presented here is envisioned as such a venture.

SYMBOLS AND ACRONYMS

kW	kilowatts of power (electrical or thermal, depending on context)
kW_t	kilowatts of thermal power
Wh	watthours of energy
PM	power module
SCB	space construction base
SPS	Satellite Power System
Si	silicon solar cells
GaAlAs	gallium arsenide solar cells with a gallium aluminum arsenide window
LEO	low Earth orbit
GEO	geosynchronous Earth orbit
CMG	control moment gyro
STS	Space Transportation System
FF	free flyer
Pk	peak power
EOL	end-of-life power

SEPS solar electric propulsion stage

ECS environmental control system

25-kW POWER MODULE

Out of a 1977 space construction base study to be described later, in particular, the Power Systems Special Emphasis Task,* came the requirement for 25 kW average electrical power during the man-tended mode, the first few years of operation. From this and particular space-processing Shuttle payloads came the requirement for an early 1980's 25-kW Power Module that could support Shuttle-orbited payloads to extend the on-orbit time of the Shuttle and payload, to provide higher average power for a nominal 7-day mission, or to support a shuttle-delivered free-flyer payload.

The 25-kW Power Module (fig. 1) is designed to provide 25 kW average power in a 235-n mi, 50^{o}-inclination low Earth orbit; therefore, the solar array is oversized to compensate for the orbit dark time and energy storage is provided. A control moment gyro (CMG) system provides stabilization and maneuvers for the Power Module and Power Module/Orbiter Configuration. A heat rejection system dissipates waste heat in excess of that which can be rejected by the Shuttle.

The Power Module is designed for several operational modes (fig. 2). Power levels, heat rejection, and mission time are varied to support a power module/orbiter/payload sortie of up to 60 days, a high-power pallet-deployed sortie of up to 7 days or a free-flying power module mission of months or years duration. The power module is electrically and mechanically compatible with the Shuttle Orbiter. It is designed for Shuttle delivery to orbit, on-orbit maintenance, or return by Shuttle to Earth.

Presently, a power module evolution study is under way. Emphasis is on near-term steps, identifying driver missions, defining growth through modular steps, and/or modification to the baseline 25-kW Power Module. Figure 3 illustrates modification and modular growth to the 100-kW level and support of an on-orbit-assembled large multihundred-kilowatt power module.

LARGE POWER MODULE

In 1977, study efforts emphasized a permanent manned space construction base, and the special emphasis task of that study compared a solar photovoltaic, a solar thermal, and a nuclear reactor power system in the hundred-kilowatt range. The mission power requirements are illustrated in figure 4. Figures 5 to 7 illustrate the power systems compared and their interface with the space construction base. Figure 8 gives a quantitative comparison of competing system characteristics. It was concluded that either system could be built to meet the system requirements by the projected mission launch date ('83-'86). However, both of the other systems have a lower development status than photovoltaics. Since they did not offer a significant or mission-required improvement and their total cost was significantly higher, photovoltaics was selected as the baseline

*This task was initiated by OAST and supported also with a DOE nuclear power system definition study.

system. Indirect political and economic pressures are resulting in a widening technical gap between photovoltaic and competing systems. Reduced technology funding and increasing concern over launch, operation, and disposal of nuclear sources from Earth orbit together with continuing emphasis and success with photovoltaic systems have resulted in photovoltaics displacing the nuclear systems as a baseline to the hundreds-of-kilowatts power level. An earlier space base study had baselined a reactor-Brayton system for use at this power level. From a cost viewpoint, it is interesting to note that the recurring cost of nuclear systems is equal to or lower than all competing systems. Also, not surprising is that photovoltaics show much lower nonrecurring costs. The implication is that a space program with many like units in the power range required could show economic benefit from the nuclear option over all but the GaAlAs photovoltaic system.

The photovoltaic system at this power level and higher must contend with a high drag, gravity gradient torques, and an increasing energy storage system complexity and heat rejection impact on the ECS system.

SATELLITE POWER SYSTEM

POWER TECHNOLOGY GROWTH PROJECTIONS

Figure 9 illustrates a 5-GW (on Earth) Satellite Power System being considered for the year 2000. Photovoltaic collectors are used to power a microwave transmitter to illuminate rectenna receiving sites on the ground, 7 to 8 GW are produced by the solar array at GEO. Figure 10 illustrates the power growth projection consistent with this technology. A factor of greater than 3 increase every year is required for 20 years to achieve this goal. Figures 11 and 12 show the technology advances required in specific cost, specific weight, array area, conditioning and distribution voltages, etc., that must be achieved versus time and power levels, respectively.

Power conversion cost/watt is expected to experience the reduction due to large terrestrial system buys that result in automated, high, continuous production facilities. The space system is expected to require an order of magnitude less material than an equivalent ground-based system. The required reduction in W/kg is to be achieved by going to on-orbit fabrication and assembly of support structures, reuse of deployment and storage equipment, increasing cell efficiency with thinner cell stacks, and use of thin-film concentrators.

Higher Wh/kg energy storage will be achieved by using electrolysis-fuel cell maintainable systems and molten salt or metal batteries. Power conditioning increases in W/kg are to be achieved by going to higher voltages. Power distribution voltages are seen to be pushed to that allowable in space plasma and under space charging conditions. A limit of about 2 kV DC in LEO and 40 kV DC at GEO is projected based on limited measured data now available in this area.

Power transmission by hardwire is projected until circumstances require electromagnetic transmission. Current studies are evaluating ground-to-space, space-to-space, and space-to-ground transmission by laser or microwave for electric power, thermal power, or propulsion.

Power levels to hundreds of kilowatts are expected in LEO. In a SPS development scenario, multimegawatt and multigigawatt systems are expected in GEO. In LEO, large multimegawatt systems are expected to have more problems with drag, gravity gradient torques, array distribution voltage limits, and large energy storage systems - with attendant large heat rejection systems.

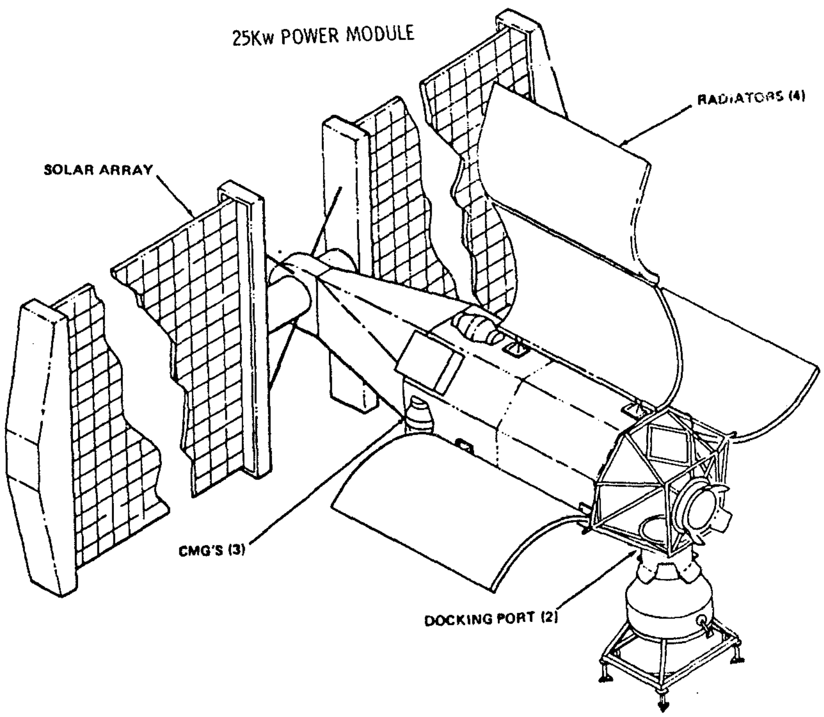
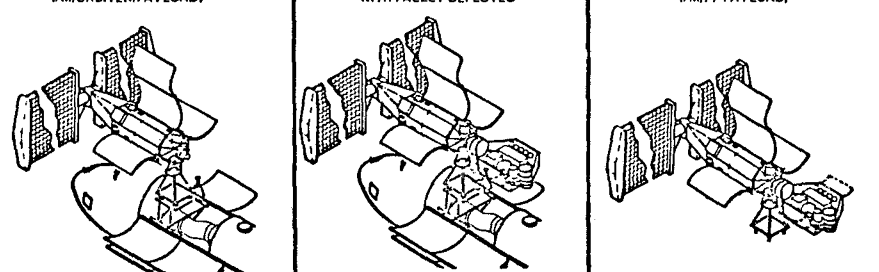


Figure 1.

BASELINE POWER MODULE PRIMARY OPERATIONAL MODES



RESOURCES AVAILABLE TO PAYLOAD	SORTIE SUPPORT MODE (PM/ORBITER/PAYLOAD)	SORTIE SUPPORT MODE WITH PALLET DEPLOYED	FREE FLYING MODE (PM/FF PAYLOAD)
POWER	11KW	32KW	25KW
HEAT REJ.*	17.5KW–19.5KW	17.6KW–19.6KW	12KW–14KW
STAB & CONT.	CMG	CMG	CMG
ORB DURATION	60 DAYS	7 DAYS	INDEFINITE

*HEAT REJECTION INCREASES WITH A REDUCTION IN 25KW PM ELECTRICAL LOAD AND BETA ANGLE. 8.5 KW_t FROM ORBITER (SPEC) INCLUDED HERE.

Figure 2.

25 KW POWER MODULE GROWTH POTENTIAL

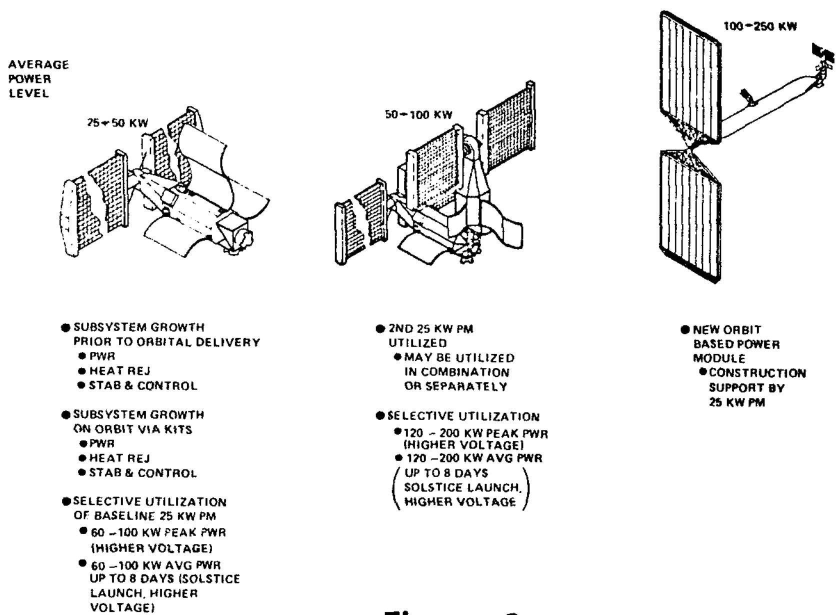


Figure 3.

BASELINE LEO MISSION POWER REQUIREMENTS (APRIL '77)

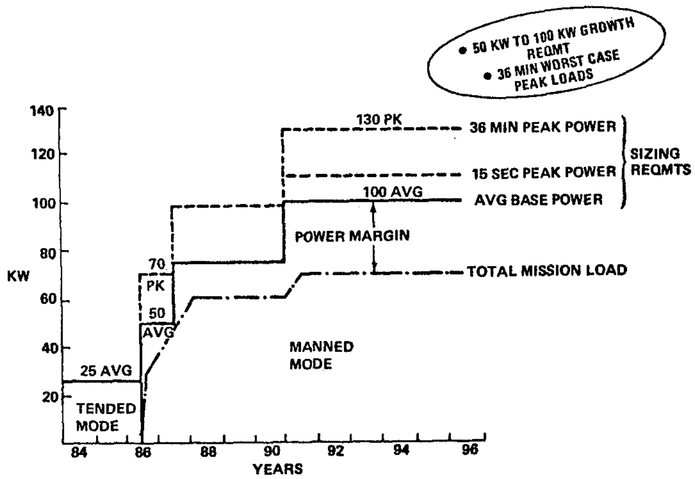


Figure 4.

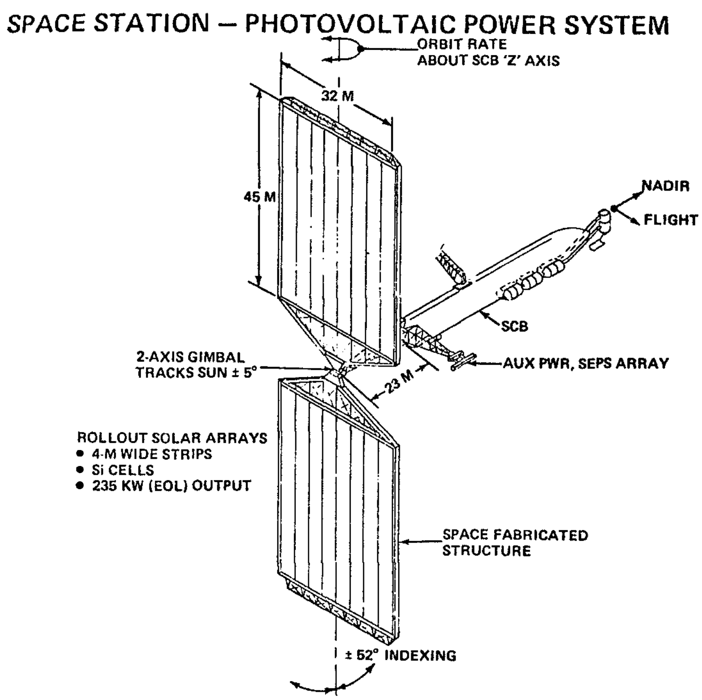


Figure 5.

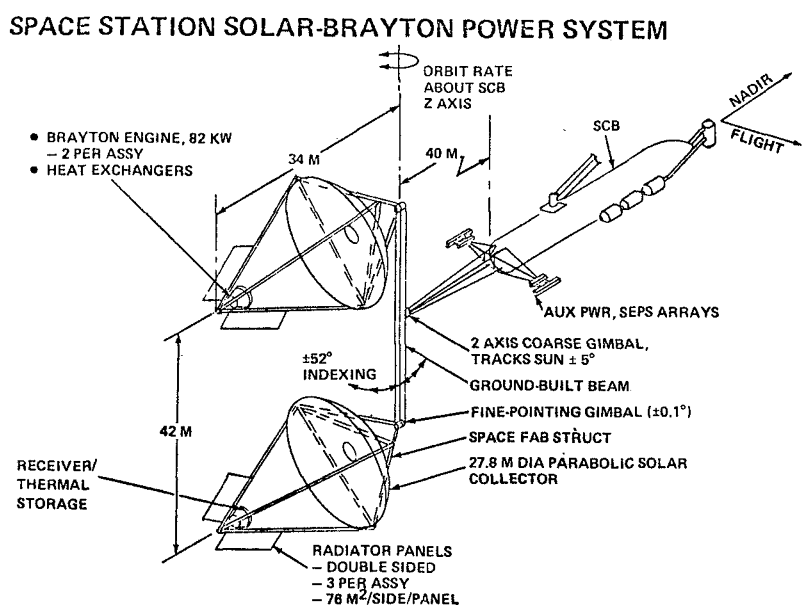


Figure 6.

SPACE STATION REACTOR-BRAYTON POWER SYSTEM

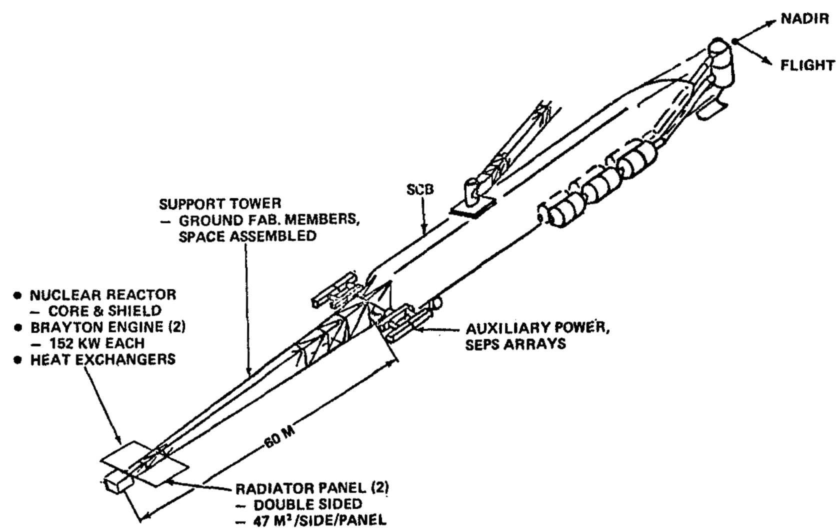


Figure 7.

POWER SYSTEM SUMMARY COMPARISON

	PHOTOVOLTAIC		SOLAR-THERMAL		NUCLEAR-REACTOR		
	Si	GaAlAs	BRAYTON	THERMIONIC	BRAYTON	THERMIONIC	THERMOELECT
CAPABILITY, KW – EARLY LOADS – ADV LOADS	70 130	70 130	70 140	70 140	130 130	130 130	130 130
COST ('77 $M) – TOTAL PROGRAM – DDT&E	254 97	203 91	344 182	411 236	410 295	502 390	459 343
SAFETY HAZARDS	HI-PRESS NiH_2		HOT SURFACES, LOSE SUN POINTING		HOT SURFACES, NUCLEAR RADIATION & DISPOSAL		
PERFORMANCE MASS (EARLY/ADV)10^3KG GROWTH POTENTIAL	9.1/14.2 GOOD	10.8/18.4 GOOD	12.8/21.6 POOR	12.8/19.9 POOR	24.6 GOOD	26.3 GOOD	26.3 GOOD
DESIGN COMPLEXITY	HI-DRAG, ECS IMPACT		MED DRAG, TIGHT PTG, MANY COMPONENTS		LOW DRAG, SPEC HDLG EQUIP.		
OPERATIONS COMPLEXITY	HI-VIEW FACTOR BLOCK, EASY MAINT		LONG BUILDUP, MOD VIEW FACT. BLOCK		NO BLOCKAGE BUT SPEC MAINT & DISPOSAL PROCED		
DEVELOPMENT RISK	LOW	MED	MED	HIGH	HIGH	HIGHEST	HIGH
PROGRAM FLEXIBILITY	GOOD POTENTIAL				TRANSPORT LIMITED		

Figure 8.

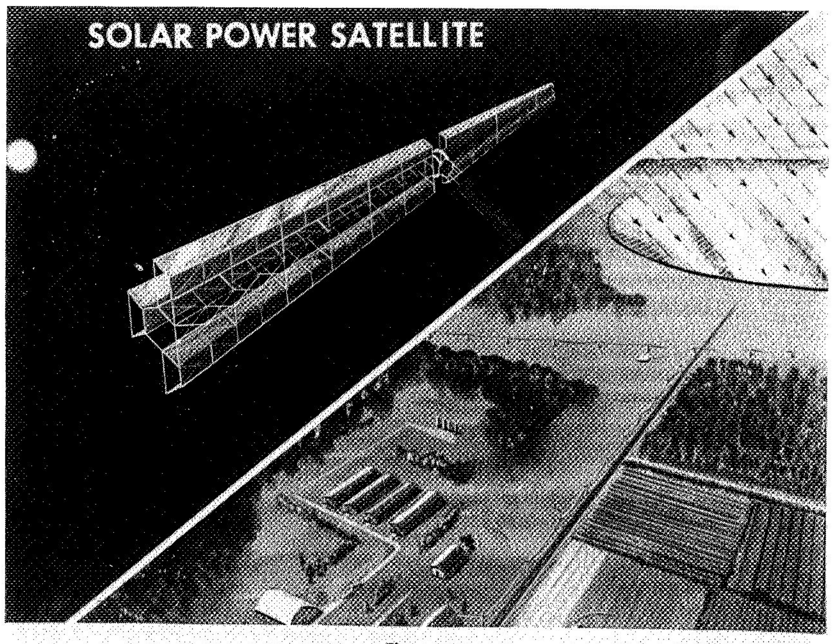


Figure 9.

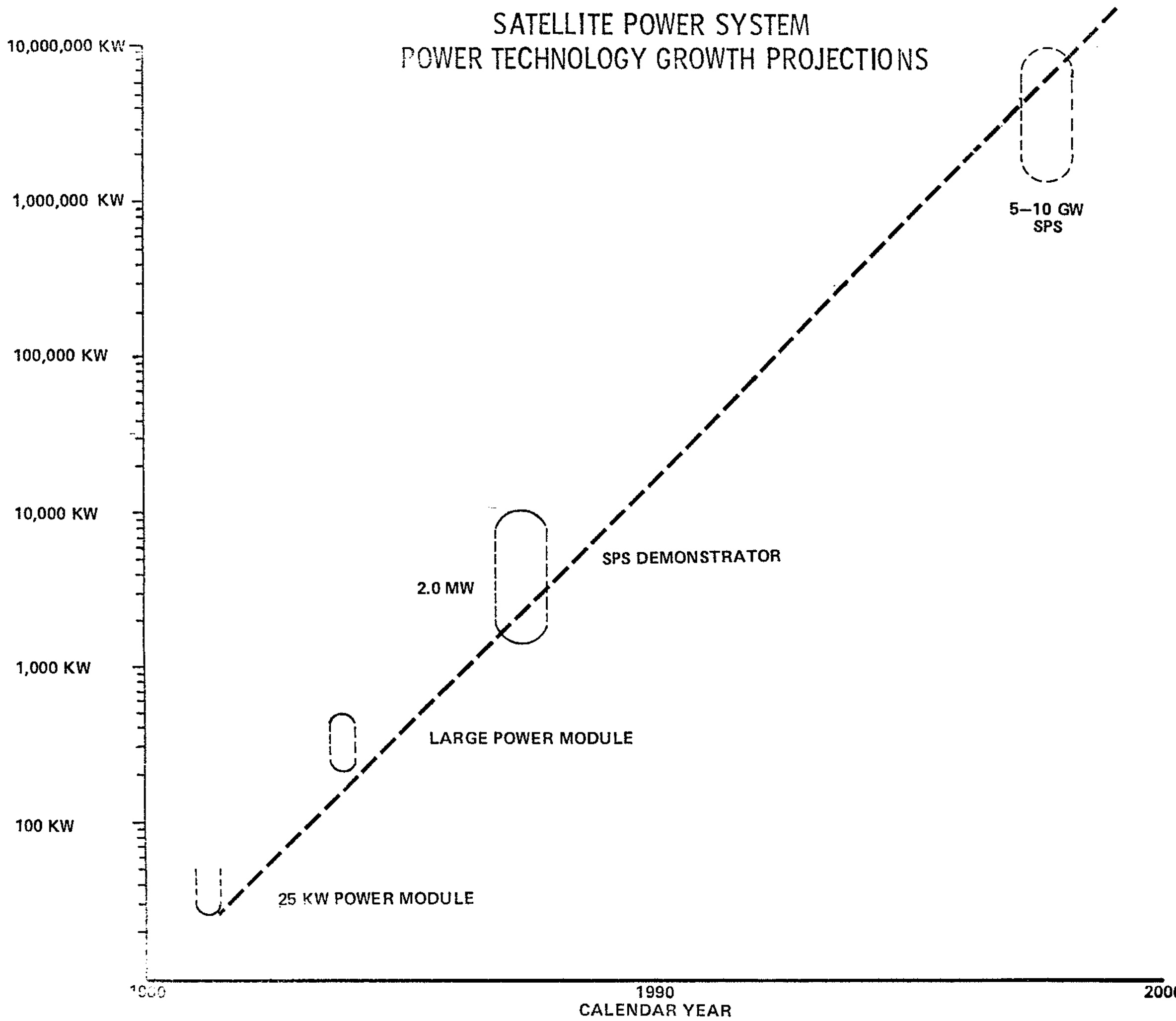

SATELLITE POWER SYSTEM
POWER TECHNOLOGY GROWTH PROJECTIONS
10,000,000 KW
1,000,000 KW
100,000 KW
10,000 KW
1,000 KW
100 KW
5–10 GW
SPS
SPS DEMONSTRATOR
2.0 MW
LARGE POWER MODULE
25 KW POWER MODULE
1990
2000
CALENDAR YEAR

Figure 10.

TECHNOLOGY ADVANCEMENT REQUIRED

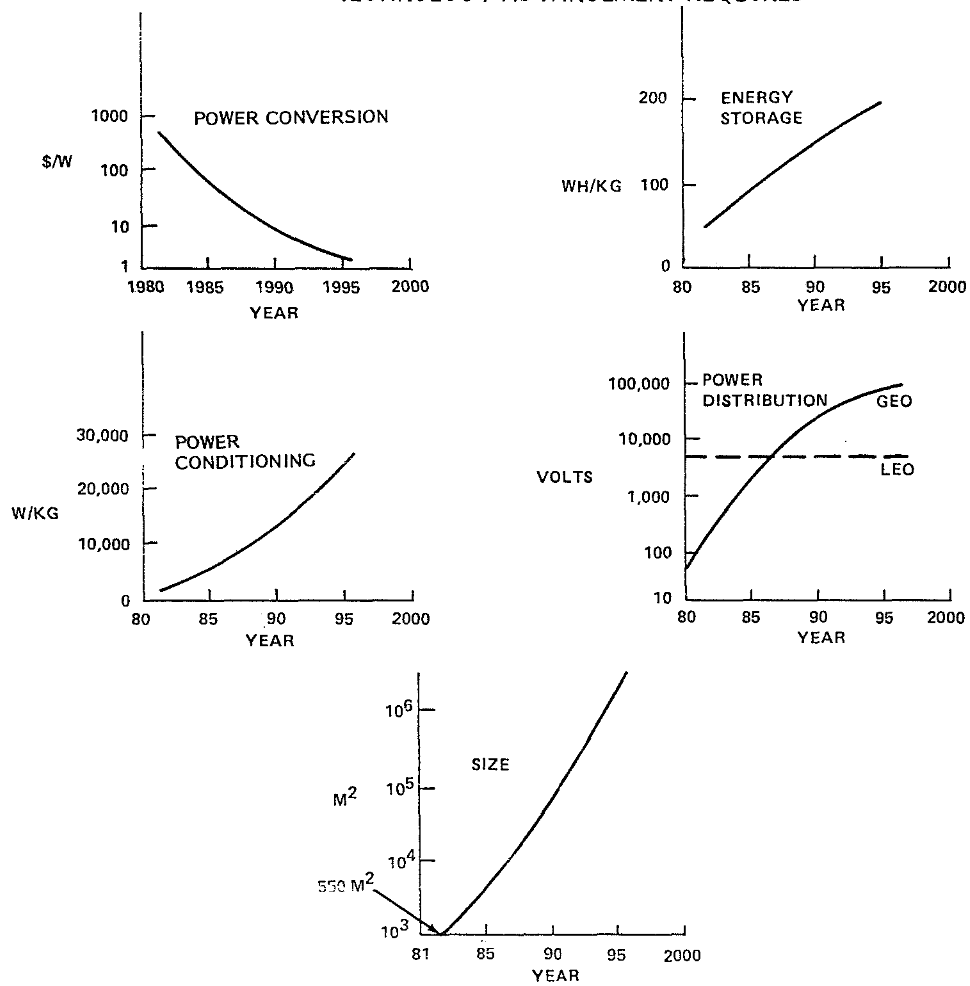


Figure 11.

POWER SYSTEM EVOLUTIONARY
TECHNOLOGY REQUIRED

	TENS OF KILOWATTS	HUNDREDS OF KILOWATTS	MULTI-MEGAWATT	MULTI-GIGAWATT
POWER CONVERSION	\$300/WATT 66 W/KG	\$30/WATT 150 W/KG	\$10/WATT 300 W/KG	\$0.75/WATT 600 W/KG
ENERGY STORAGE	KWH; 44 WH/KG	KWH; 80 WH/KG	MWH; 150 WH/KG	MWH; 200 WH/KG
POWER CONDITIONING	800 W/KG	5000 W/KG	10,000 W/KG	26,000 W/KG
POWER DISTRIBUTION	115V DC	2K VDC	12K VDC	40K VDC
POWER TRANSMISSION	HARDWIRE	HARDWIRE	HARDWIRE LASER	HARDWIRE LASER RF
ORBIT	LEO	LEO/GEO	GEO	GEO
CONSTRUCTION	DEPLOYABLE	ON-ORBIT ASSY(?)	ON-ORBIT ASSY	ON-ORBIT ASSY
SIZE	550 M^2	4200 M^2	30,000 M^2	30 KM^2

Figure 12.

JSC SPACE BASE/POWER MODULE STUDIES

Jerry W. Craig
NASA Johnson Space Center

Studies of on-orbit systems have shown that users of the Shuttle system will require increased electrical energy and associated services. In particular, users of the Orbiter/Spacelab combination will require both higher electrical power and longer duration than is available with the current baseline system. Additionally, since operations costs (and user charges) increase slowly with duration, the economics of this system are more attractive to all users if its duration is extended beyond the baseline 5 to 10 days. Present Orbiter/Spacelab mission capability is primarily constrained by the hydrogen and oxygen available to generate power in the Orbiter fuel cells. It is also necessary to assure that considerable attitude or pointing flexibility is retained to assure efficient operation of the Orbiter radiator cooling system. Beyond these early limitations, it is foreseen that orbital operations will eventually need even greater quantities of the basic space utilities: electrical power, heat rejection, and attitude control. Such operations, forecasted for the mid to late 1980's, will be best accommodated by a module stored in orbit that can furnish these to a docked Orbiter/Spacelab or other vehicles.

The JSC approach to provision of the requisite services is the Orbital Service Module concept. The Orbital Service Module represents a concept for an evolutionary program which will provide this increasing level of utilities service. Continuous matching of capability to real user needs, while avoiding the pitfalls usually associated with prediction of long-range requirements, is a primary objective of this approach. Thus, the program is structured as a series of evolutionary steps or increments. Since each increment is, in itself, a nominal uprating of existing capability, lead times are relatively short and an OSM program commitment need not be made until user requirements are firm. As a result, annual funding (including that for initial increments required by Spacelab operations in the early 1980's) is considerably less than that needed to produce a full-capability power module.

The Orbiter baseline configuration offers tremendous operational flexibility. The initial step in the OSM approach is to assure good balance in the use of this flexibility in provision of payload services such as delivery and return weights, power, cooling, orbit location, attitude control, and duration. This is done through a large solar array deployed and positioned by the Remote Manipulator System. Power is routed to the Orbiter by a cable strapped to the RMS, where it is conditioned and placed on the Orbiter and payload buses. Fuel cells still provide power during night operation. (See figs. 1 to 5.)

In order to properly size and plan the various increments, mission requirements must be derived. This was accomplished by analysis of the STS 10-77 traffic model. Results indicate the Power Extension Package (PEP) (first step in the incremental growth of services) should be sized for a 29-kW power level, and the free-flying module (second and third steps) to provide 35 kW average power (fig. 6). These results are tentative, and additional study and user interaction will be needed to properly size the free-flying module. Figure 7 shows that Spacelab missions to many inclinations and altitudes will use the PEP and/or power module. Also note that the PEP permits sharing of Spacelab with delivery missions to 28° orbits. Most deliveries of SSUS payloads do not use the full Orbiter payload potential, therefore pallets with PEP can be co-manifested on these flights. This sharing will permit large cost savings to the user as he will then pay only a portion of the total Shuttle flight cost.

The requirements analysis results are summarized in figure 8. Note that PEP will meet all requirements through 1984. The free-flying power module will be needed as the users' free-flying payloads are developed and become available in the 1983-84 time frame. Figure 9 shows that PEP will provide 29 kW for 20 days or 21 kW for 30 days. The free-flying module will provide 35 kW indefinitely.

Figures 10 to 13 describe the PEP hardware configuration, its installation in the Orbiter payload bay, and the operational deployment sequence. Note the PEP takes virtually no usable payload bay volume. Figures 14 and 15 describe Orbiter thermal control modifications and capabilities associated with PEP. Power levels up to the full 29 kW provided by PEP (15 kW to payload) can be accommodated by the thermal control system. Figure 16 shows the PEP weight. Figure 17 is an artist's concept of the initial free-flying power module (Increment III - This module is passively stabilized and contains relatively little avionics. It will provide power and cooling to such free-flying payloads as the materials experimentation module at a minimum cost.) Figure 18 shows the relative capabilities of PEP and the free flyer. Figure 19 shows the initial deployment sequence of the free flyer. Figure 20 shows the actively stabilized free-flying power module. The CMG and avionics pod can be added to the passively stabilized free flyer (fig. 18) after it is already placed in orbit. This configuration can also support free-flying manned modules when they are needed to relieve constraints on Orbiter on-orbit stay time. Figure 21 shows the free-flyer weight estimates. Figure 22 emphasizes the potential commonality of the PEP and free flyers. Figure 23 reveals the JSC baseline program plan and funding. Because of the commonality of PEP and free-flyer development, the net development cost of PEP is approximately $20 to $25 million.

This incremental approach also permits great flexibility in the spread of funding for the program. Note the PEP will be available to support even early Spacelab missions. This early availability of increased power and duration will save up to $0.5B in operations cost during the first 2 to 3 years of operation (as compared to similar operations using cryo kits). It also precludes the need to develop energy-conservative payload hardware.

In summary, the JSC incremental growth approach maximizes the use of the Shuttle investment, provides early services when they are needed, and permits the free-flying power module to be optimized to payload requirements as they emerge.

OSM PROGRAM RATIONALE

- MISSION ANALYSIS VERIFIES POWER AND HEAT REJECTION CAPABILITY CRITICAL TO EXPLOITATION OF FULL STS POTENTIAL FOR ORBITER ATTACHED PAYLOADS
- ORBITER FLEET SIZE AND TURNAROUND CONSIDERATIONS DICTATE FREE FLYER SUPPORT CAPABILITY NEEDED IN 1984-86 TIME FRAME
- OSM CONCEPT OFFERS INCREMENTAL GROWTH FROM THE BASELINE ORBITER
  - USE FULL ORBITER MISSION FLEXIBILITY
  - MOST COST EFFECTIVE SUPPORT OF EARLY PAYLOADS
  - EACH STEP IS BUILDING BLOCK FOR FUTURE EVOLUTION
- FREE-FLYER SUPPORT CAPABILITY OPTIMIZED TO USER REQUIREMENTS AND SCHEDULE; MINIMUM OVERALL COST

Figure 1.

INCREMENTAL GROWTH CONCEPT

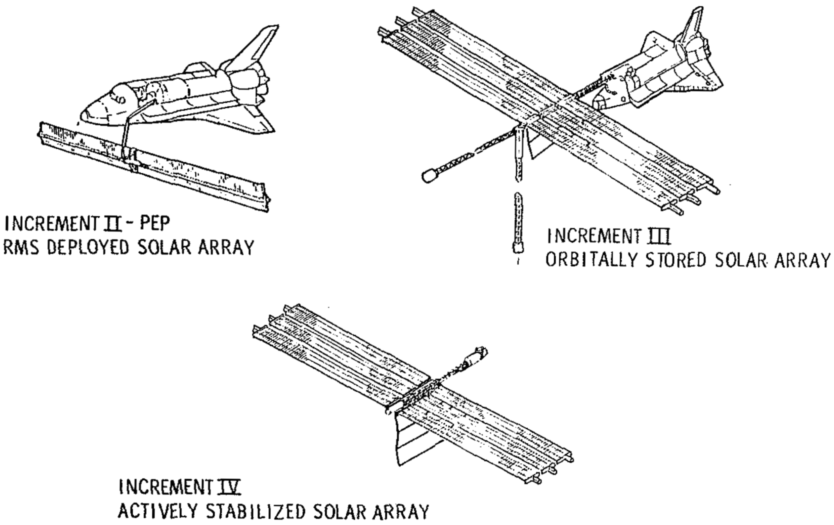


Figure 2.

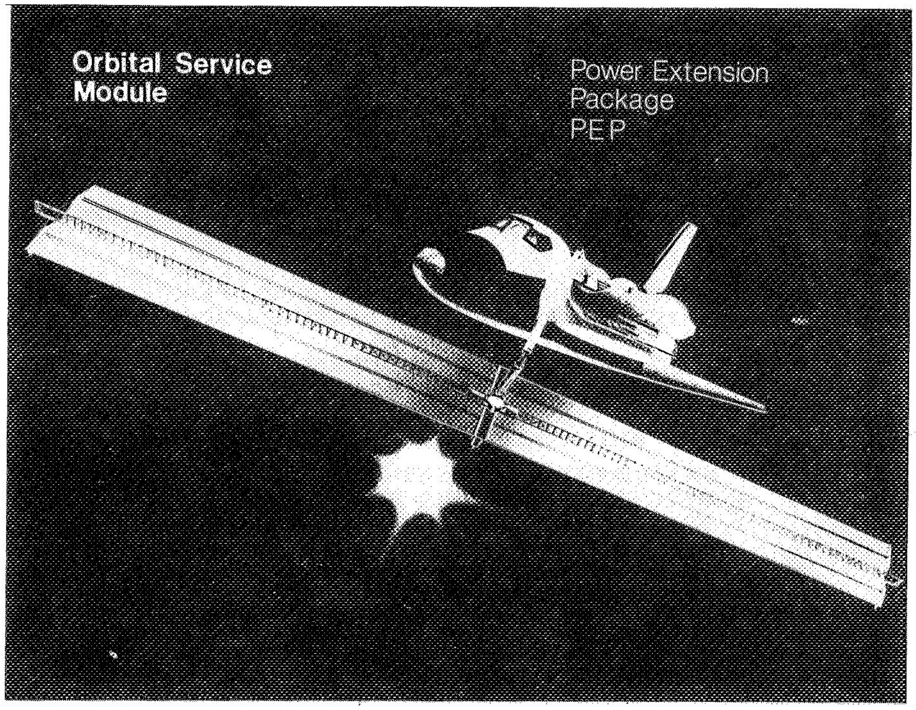


Figure 3.

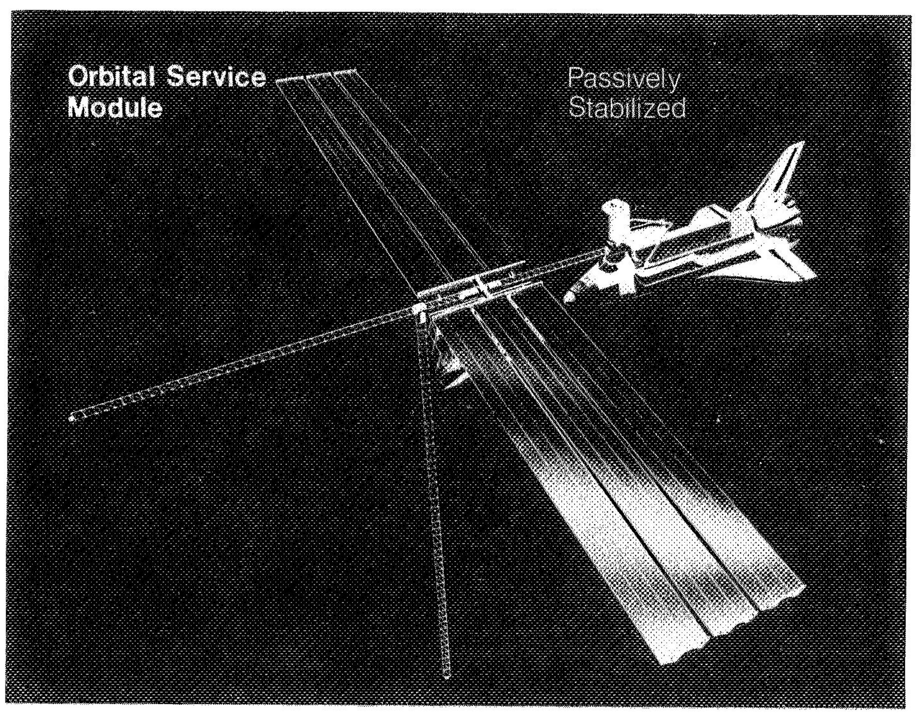


Figure 4.

NASA-S-78-11648

OSM REQUIREMENTS DERIVATION

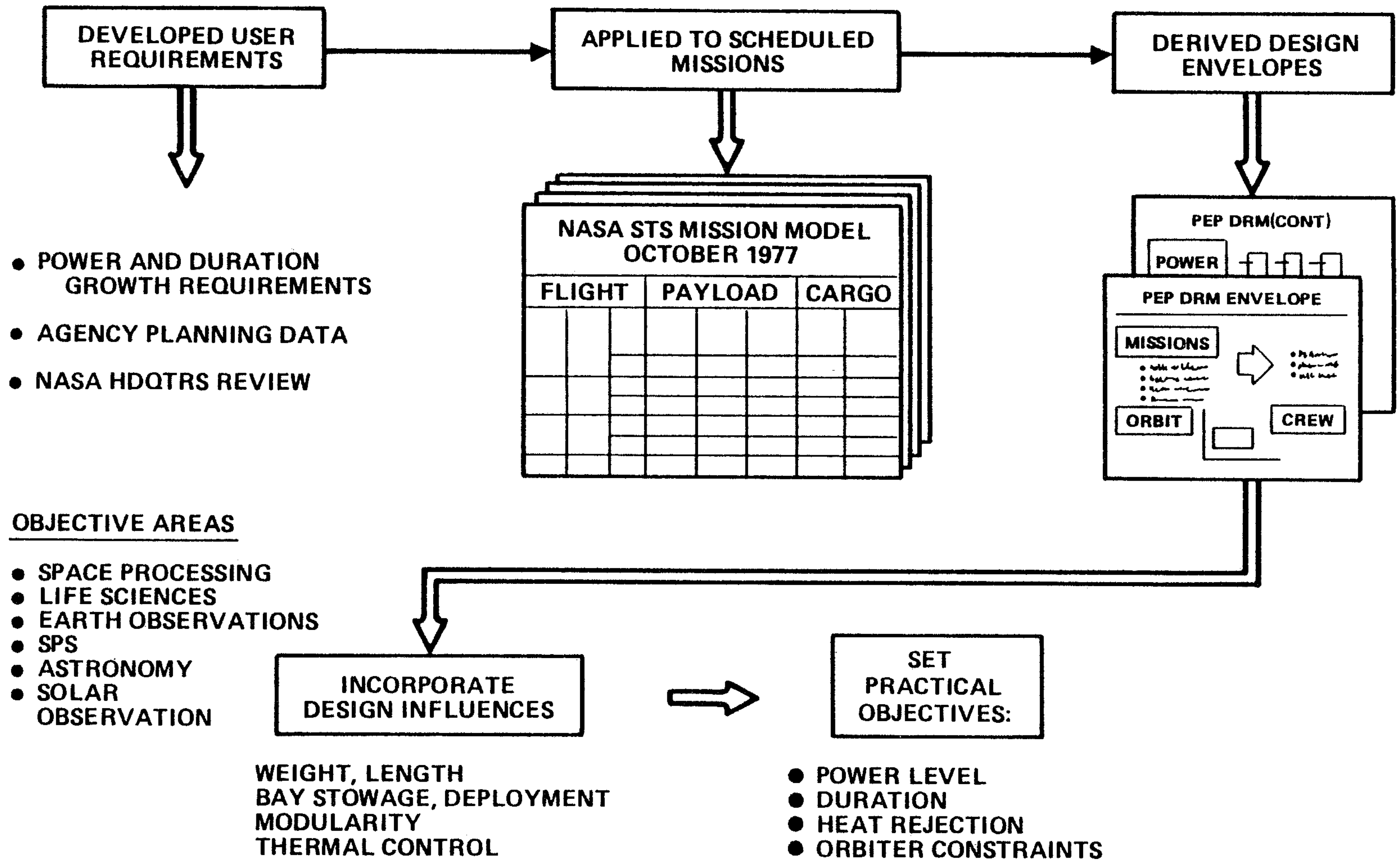


Figure 5.

NASA-S-78-11640

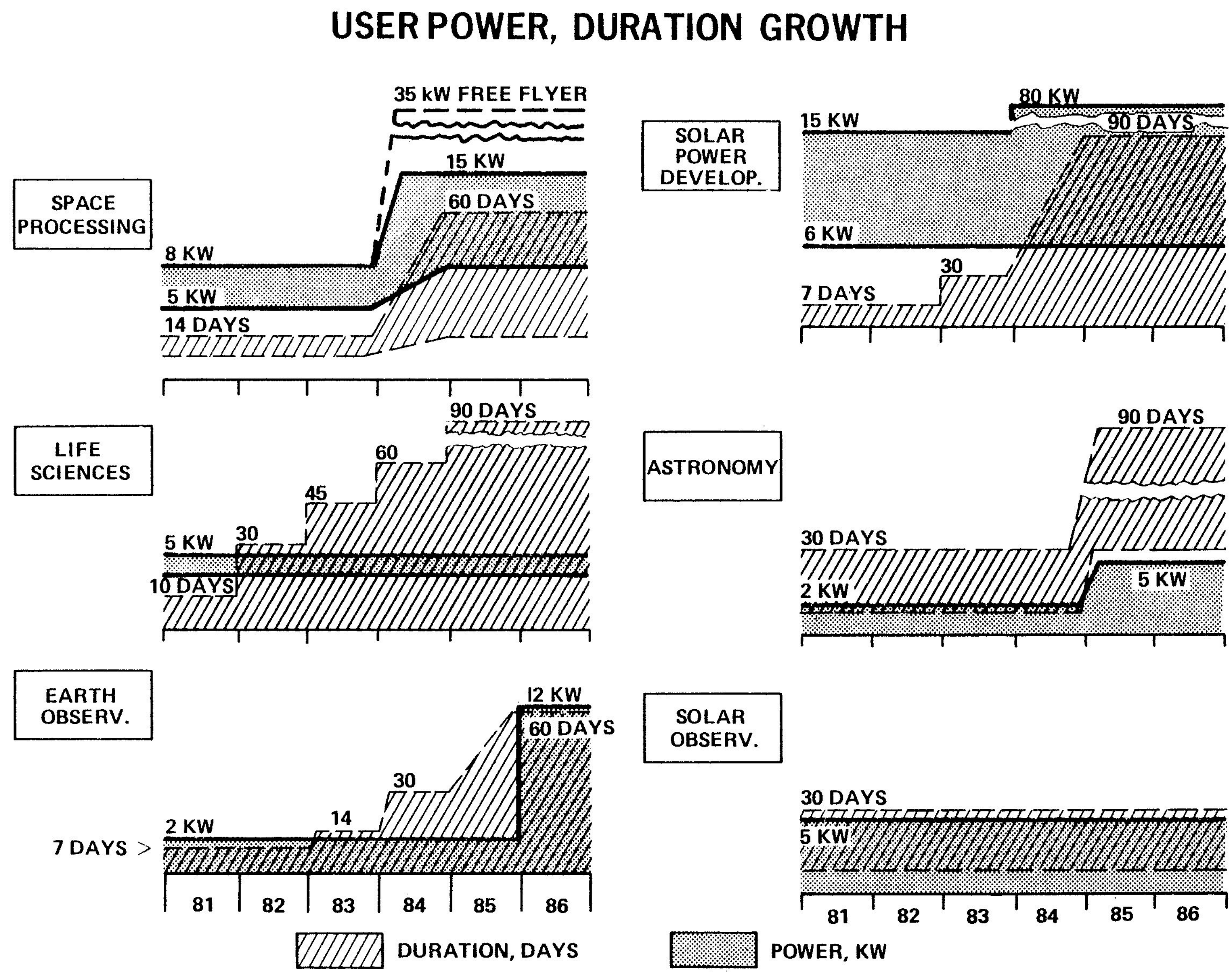
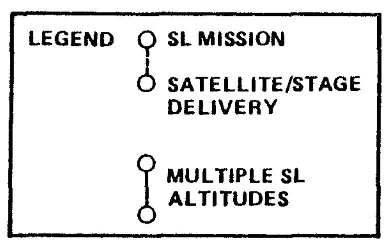


Figure 6.

NASA-S-78-11625 A

ORBIT INCLINATION/ ALTITUDE REQUIREMENTS STS 10-77 MODEL-SL



⊗ >7 kW USER POWER
● >7 DAYS DURATION

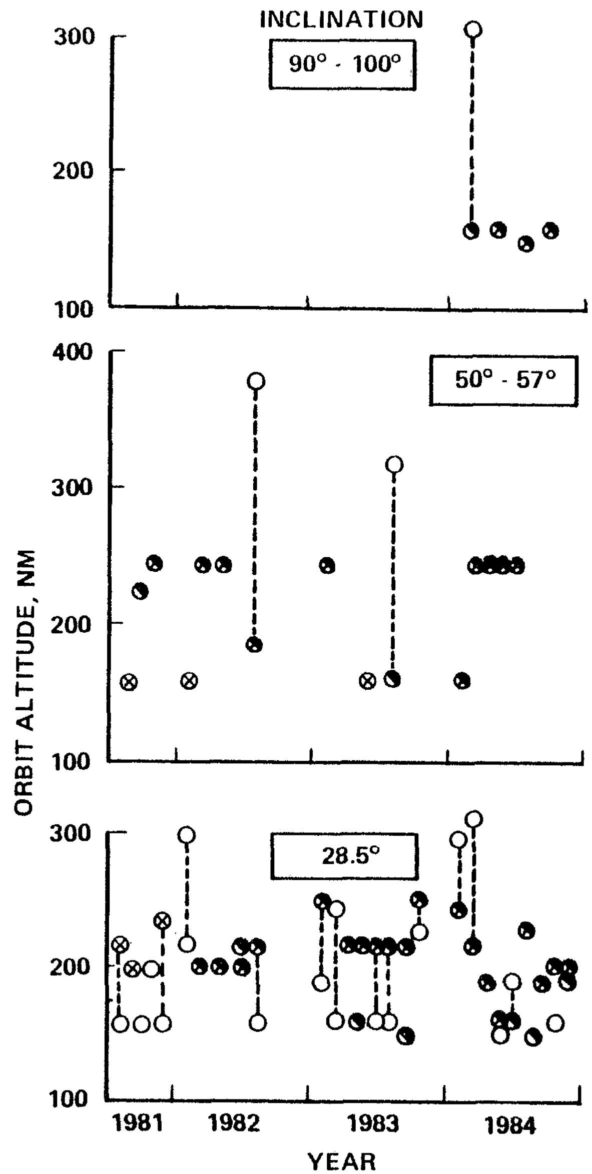


Figure 7.

NASA-S-78-11652

REQUIREMENTS ANALYSIS RESULTS

- EARLY MISSION REQUIREMENTS (1981-1983) EXCEED BASELINE ORBITER CAPABILITY

- POWER RANGE	24 TO 32 KW
- DURATION RANGE	14 TO 30 DAYS

- REQUIREMENTS INCREASE AFTER 1983

	1984		1990
- POWER	35 KW	→	100 KW
- DURATION	60 DAYS	→	CONTINUOUS
- FREE-FLYER MISSIONS			

- MULTIPLE ORBIT REQUIREMENTS EXIST EARLY AND CONTINUE

	1981-1983	>1983
- INCLINATION	28.5° & 55°	28.5°, 55° AND POLAR
- ALTITUDE	300 TO 600 KM	

- ALL ORIENTATION CAPABILITY REQUIRED
  - EARTH, SOLAR, INERTIAL, GRAVITY GRADIENT

Figure 8.

NASA-S-78-11631A

POWER PERFORMANCE ENVELOPES — ORBITER AND OSM

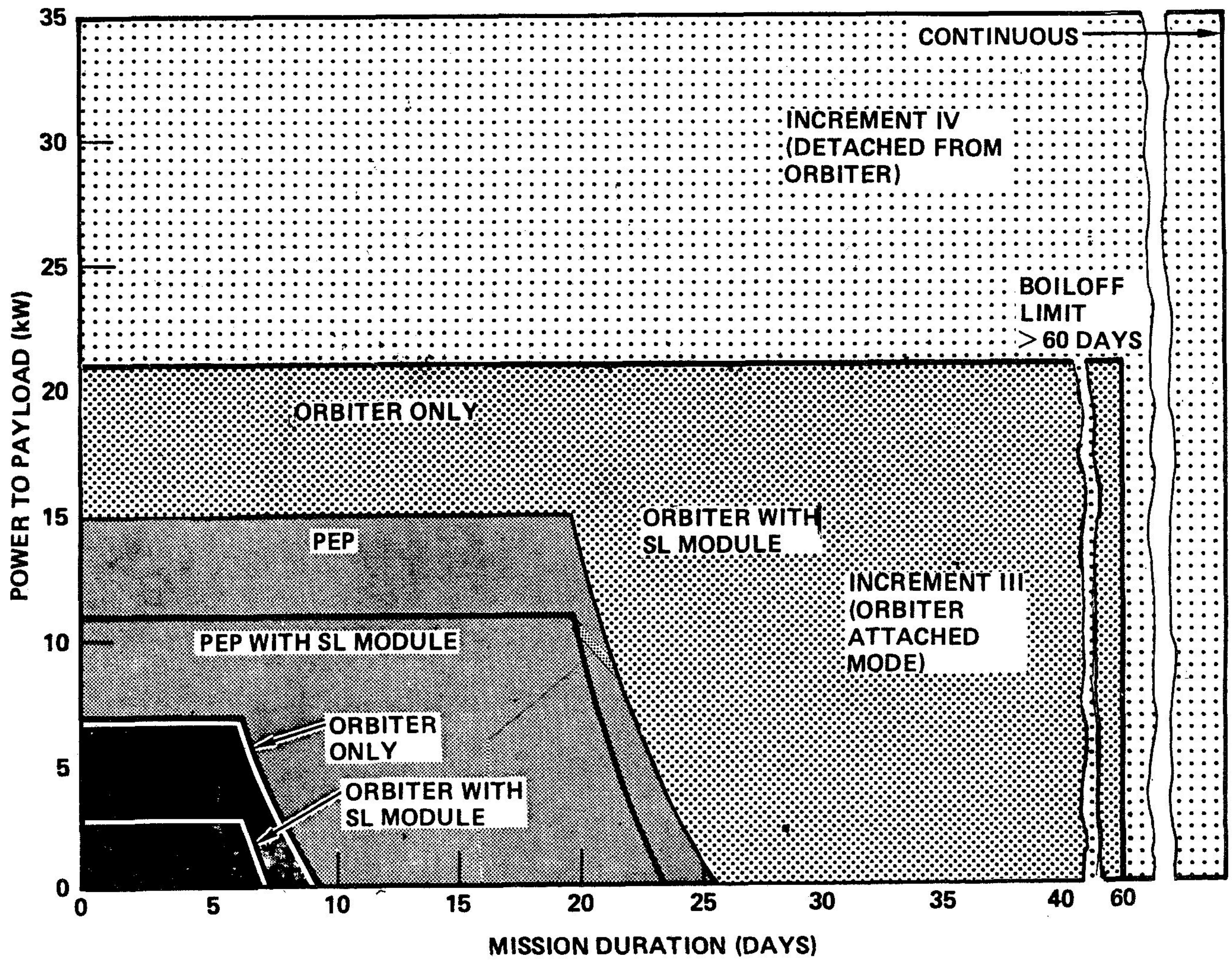


Figure 9.

NASA-S-78-11643

PEP 32 KW SOLAR ARRAY
(ONE 16 KW WING ILLUSTRATED)

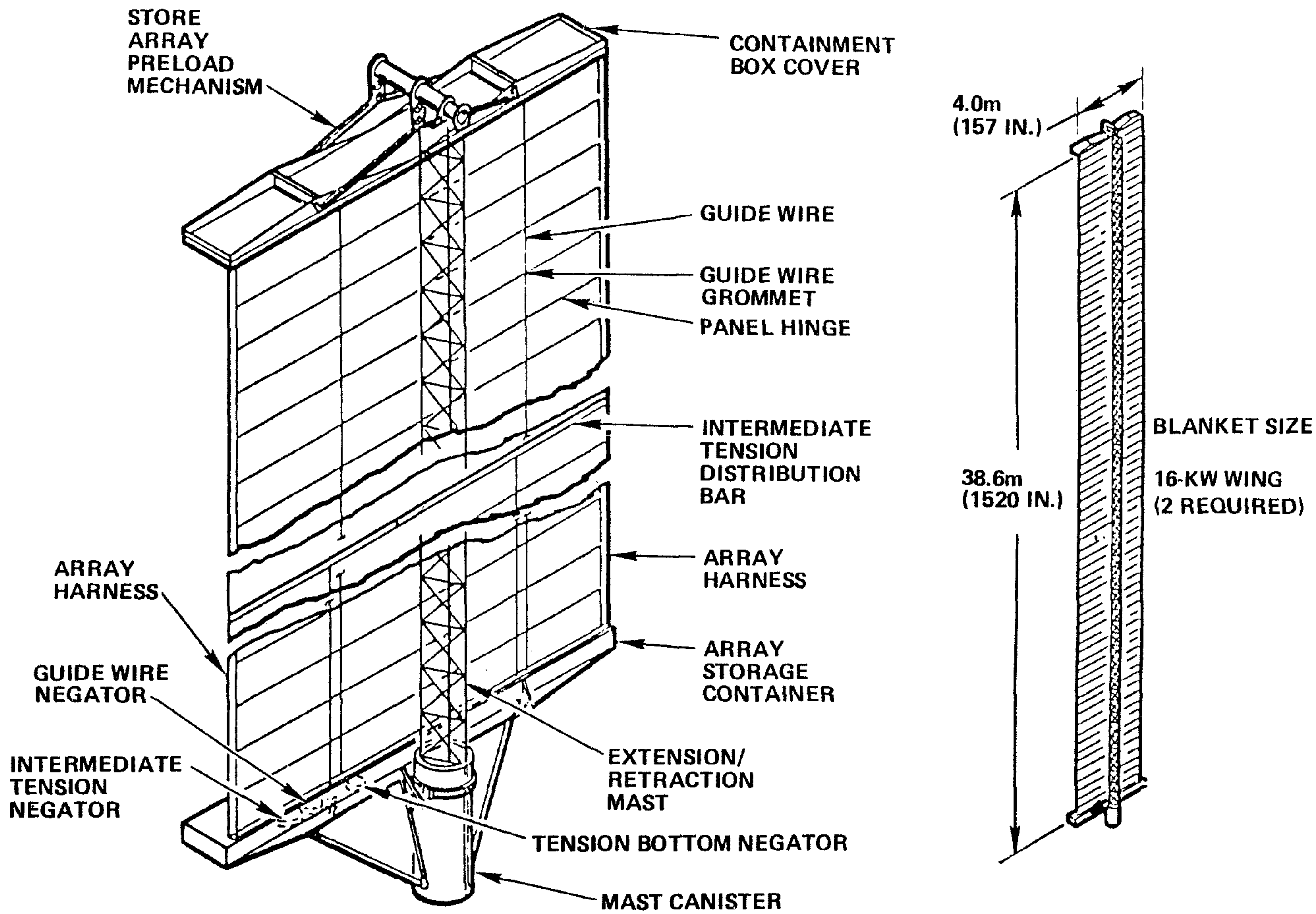


Figure 10.

NASA-S-78-11627

POWER EXTENSION PACKAGE (PEP)

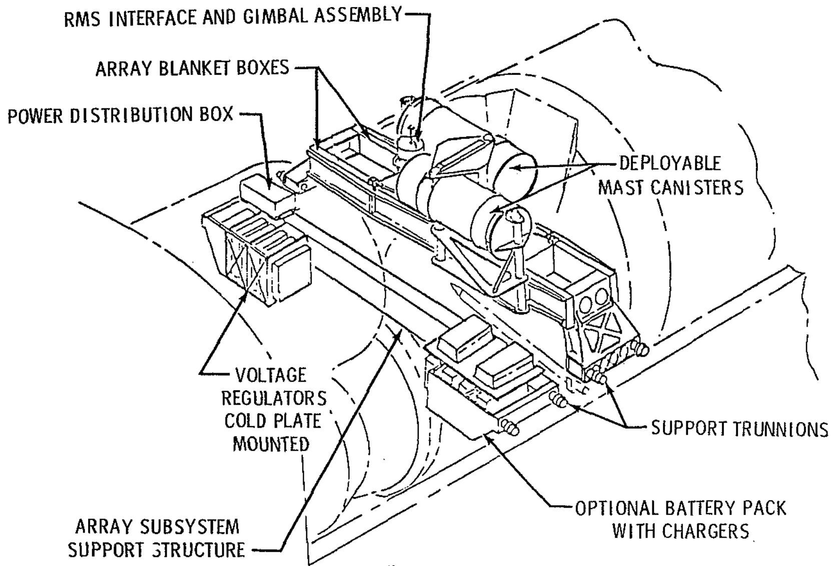


Figure 11.

NASA-S-78-11644A

SOLAR ARRAY RMS INTERFACE

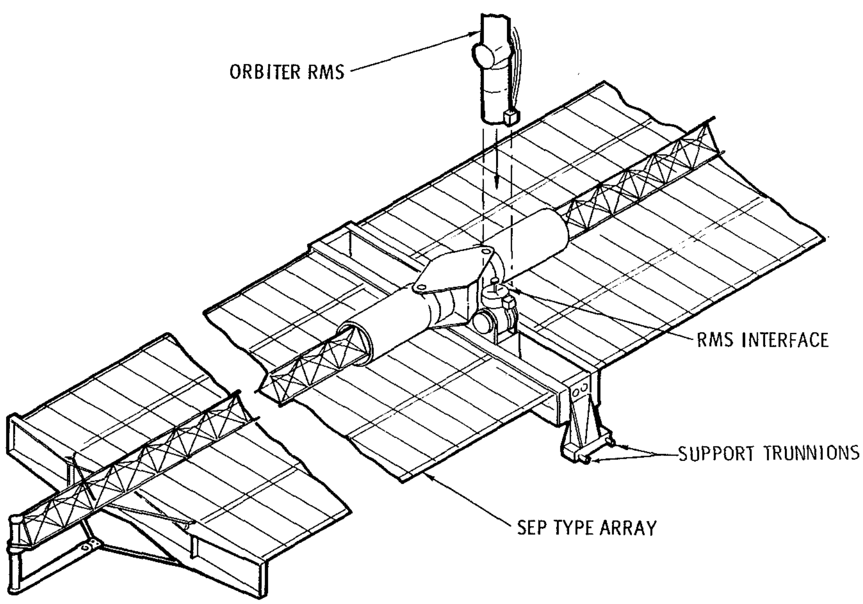


Figure 12.

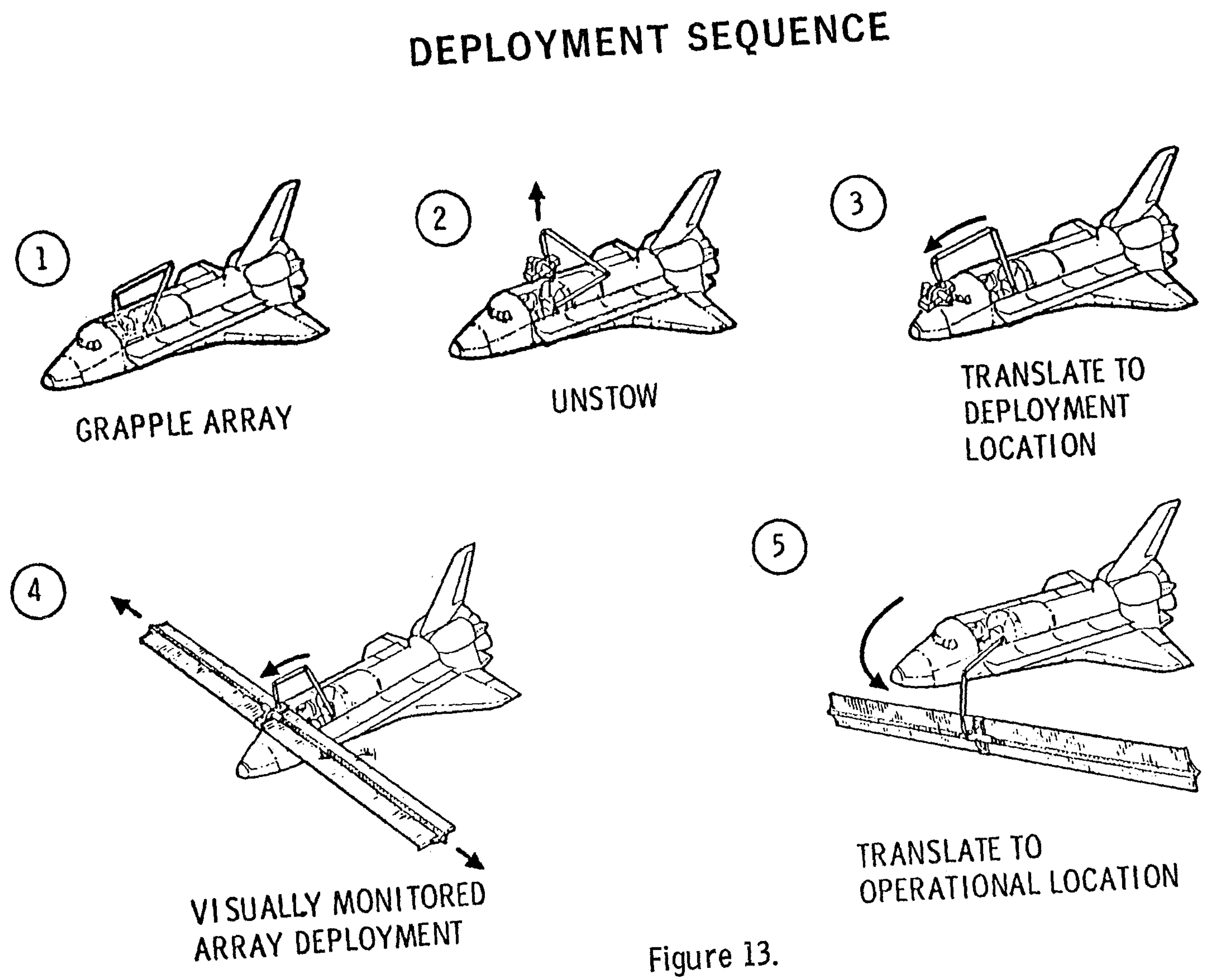

DEPLOYMENT SEQUENCE
1
GRAPPLE ARRAY
2
UNSTOW
3
TRANSLATE TO
DEPLOYMENT
LOCATION
4
VISUALLY MONITORED
ARRAY DEPLOYMENT
5
TRANSLATE TO
OPERATIONAL LOCATION

Figure 13.

NASA-S-78-11835

TYPICAL PEP ON-ORBIT HEAT REJECTION CAPABILITY

EFFECT OF PLANNING PARAMETERS IN EARTH VIEWING ATTITUDE

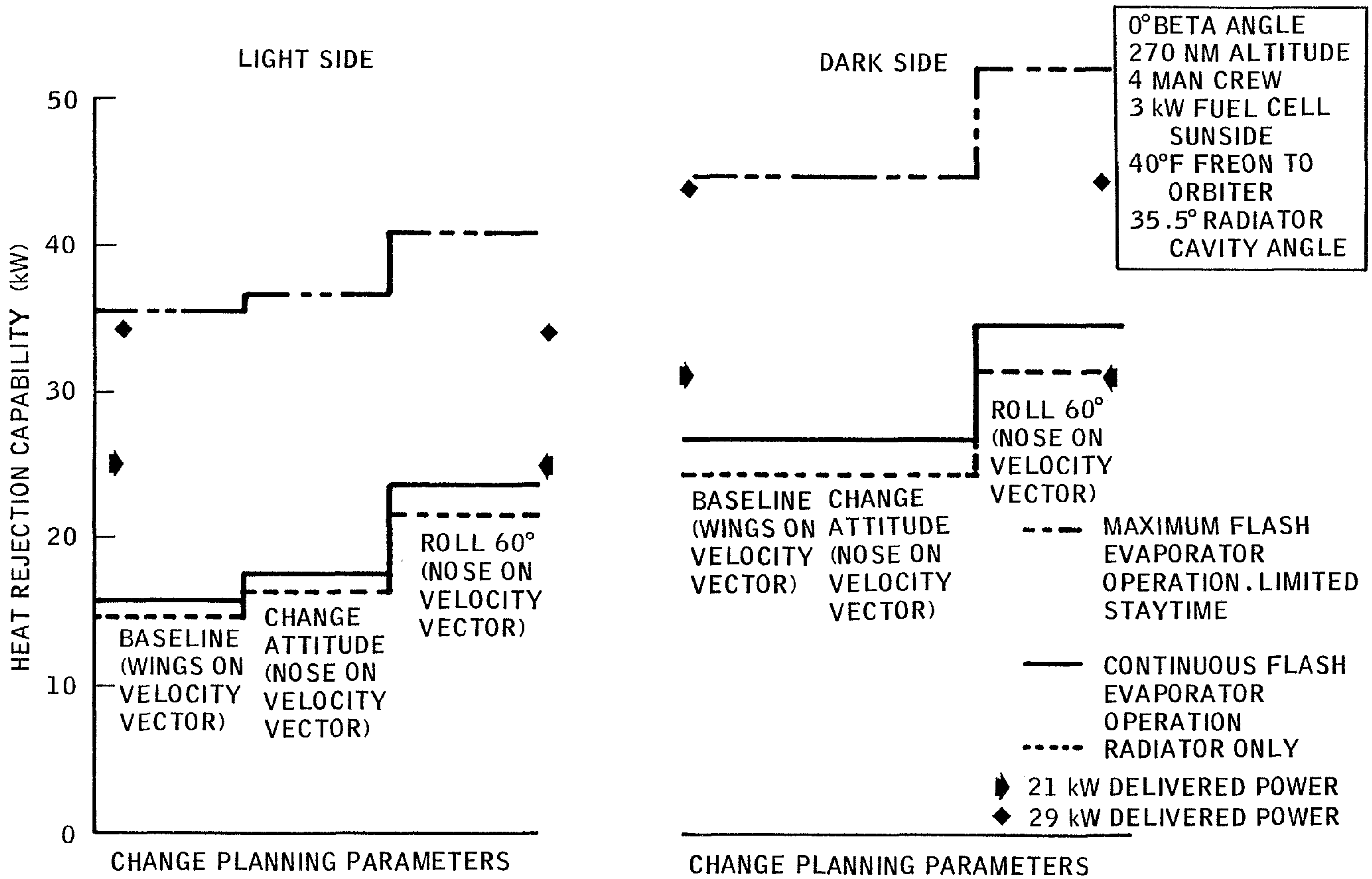


Figure 14.

NASA-S-78-11683A

PEP ACTIVE THERMAL CONTROL FEATURES

- ORBITER PROVIDES HEAT REJECTION
  - RADIATOR CAVITY INCREASED TO 60°
  - USE PAYLOAD PLANNING VARIABLES
- PAYLOAD COOLING PROVIDED BY ORBITER PAYLOAD HEAT EXCHANGER
- PEP POWER CONDITIONING EQUIPMENT COOLED BY ORBITER AFT COLDPLATE COOLANT LOOPS
- SOLID AMINE FOR CO_2 AND HUMIDITY CONTROL

Figure 15.

NASA-S-78-11832

PEP SYSTEM WEIGHT SUMMARY

	WEIGHT - LBS	
PEP	2494	
SOLAR ARRAY		1392
STRUCTURE SUPPORT		199
POWER DISTRIBUTION AND CONTROL		561
THERMAL CONTROL		88
CONTROL ELECTRONICS		254
PAYLOAD RETENTION FITTINGS	408	
CO_2 REMOVAL	-253	
LiOH		-654
SOLID AMINE (ENTRY)		401
TOTAL	2649	

Figure 16.

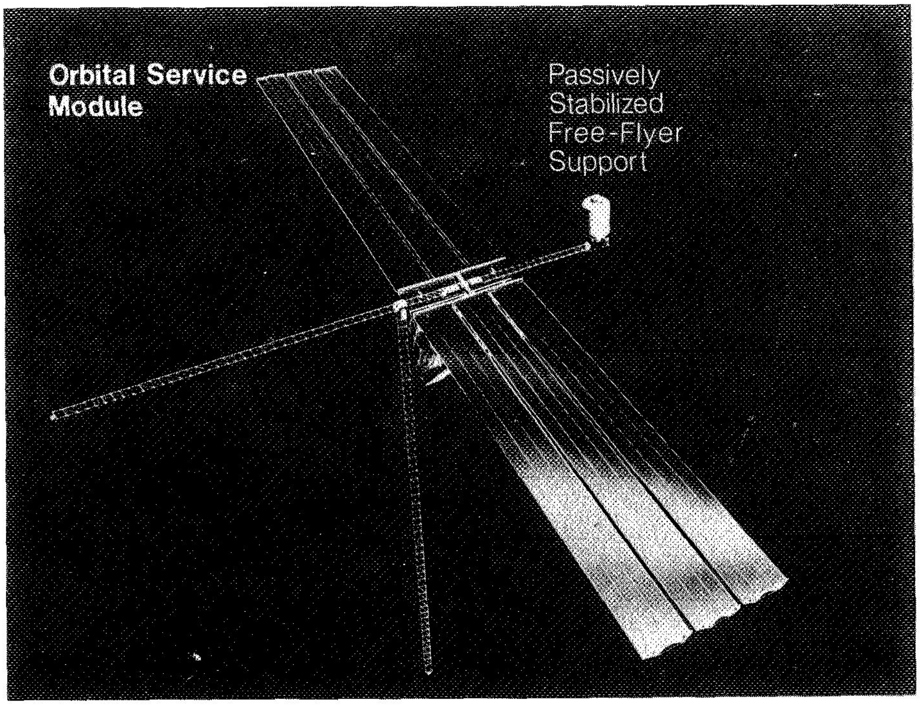


Figure 17.

NASA-S-78-11760A

OSM EVOLUTION

	BASELINE ORBITER	INCREMENT II (PEP)	INCREMENT III	INCREMENT IV
POWER SYSTEM	FUEL CELL	SOLAR CELL/ FUEL CELL	SOLAR CELL/ BATTERY	SOLAR CELL/ BATTERY
POWER AVAILABLE TO PAYLOAD	7 KW	15 KW	21 KW (1) 35 KW (2)	35 KW
TOTAL POWER OUTPUT	21 KW	29 KW	35 KW	35 KW
DURATION - DAYS	6-1/2	30 (3)	60 (4) TO CONTINUOUS	60 (4) TO CONTINUOUS
SOLAR COLLECTION AREA M^2	N/A	309	1000	1000
STABILIZATION	ORB RCS	ORB RCS	GRAVITY GRADIENT	CMG'S - ALL ATTITUDE
FREE FLYER SUPPORT	N/A	NONE	DOCKING MODULE LIMITED COMM. AND DATA	ADDS WIDE BAND DATA
HEAT DISSIPATION	ORB ONLY	ORB ONLY	ORBITER/OSM (1) OSM ONLY (2)	SYMMETRIC OSM

(1) ATTACHED TO ORBITER
(2) DETACHED FROM ORBITER
(3) AT 21 KW WITH OPTIONAL BATTERY PACK
(4) BOIL-OFF LIMITED ORBITER

Figure 18.

NASA-S-78-11646

OSM INCREMENT III DEPLOYMENT

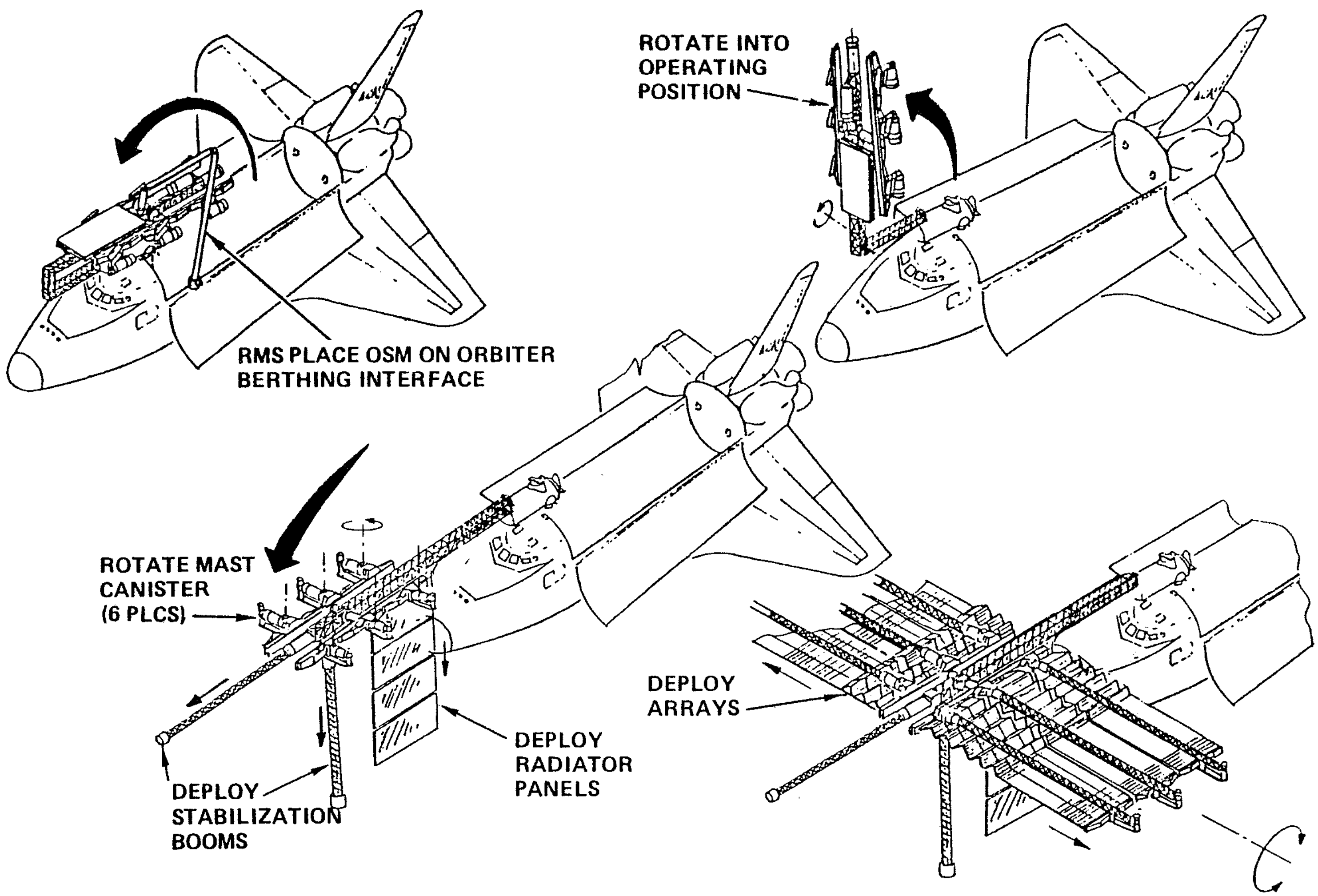


Figure 19.

NASA-S-78-11766

INCREMENT IV CONFIGURATION ACTIVELY STABILIZED

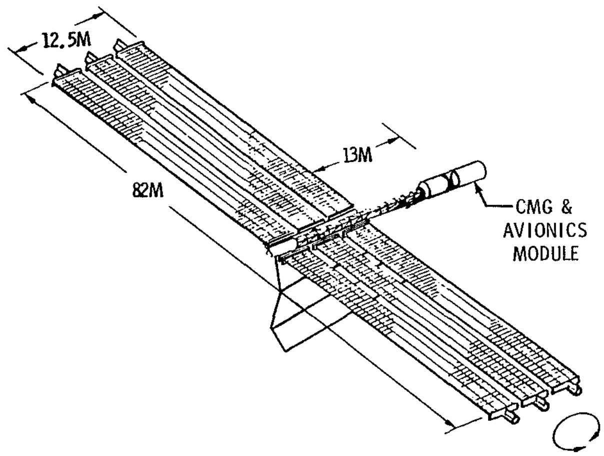


Figure 20.

NASA-S-78-11833

OSM WEIGHT ESTIMATES

EQUIPMENT	INCREMENT III	INCREMENT IV
SOLAR ARRAY	4,176	4,176
STRUCTURE SUPPORT	2,848	3,148
COUNTER BALANCE AND SUPPORT	4,792	—
POWER DISTRIBUTION AND CONTROL	11,139	11,239
THERMAL CONTROL	2,100	3,000
ATTITUDE CONTROL AND CONTROL ELECTRONICS	520	2,623
TOTALS - LBS	25,575	24,186

Figure 21.

NASA-S-78-11665

SYSTEM COMMONALITY

	INCREMENT II	INCREMENTS III/IV
ARRAY		
NUMBER OF WINGS	2	6
WIDTH OF WING (M)	4	4
LENGTH OF WING (M)	38.6	37.8
PANELS PER WING	51	50
POWER RATING PER WING (KW)	16	13
BATTERY		
BATTERY CELL RATING (AMP HR)	65*	65
CELLS PER MODULE	28*	24
BATTERY MODULES	4*	54
NUMBER OF BATTERIES	2*	18
NUMBER OF BATTERY CHARGERS	2*	9
CHARGER POWER RATING (KW)	4.38*	4.38
REGULATORS		
NUMBER OF REGULATORS	6	9
REGULATOR PEAK POWER (KW)	6	6

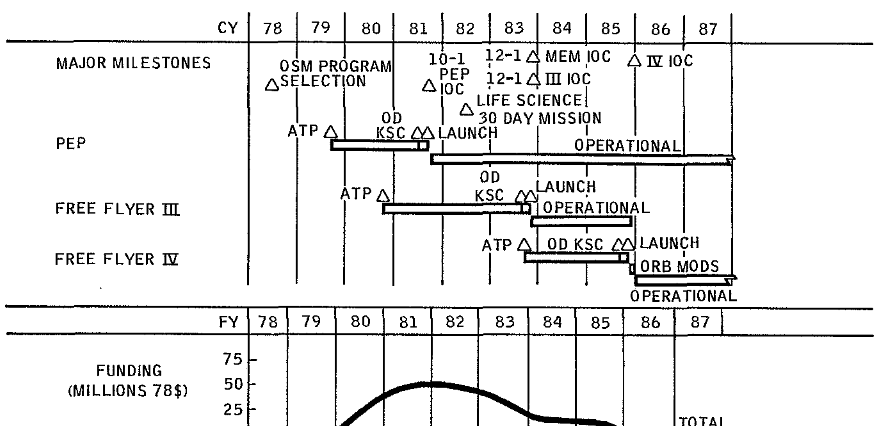
*OPTIONAL BATTERY KIT

Figure 22.

NASA-S-78-11708B

JSC BASELINE PROGRAM PLAN AND FUNDING

PEP IOC 1981, III IOC 1983, IV IOC 1985



	78	79	80	81	82	83	84	85	86	TOTAL
PEP			20	25						45
III				26	50	31				107
Δ III-IV							14	14		28
(78$) TOTAL			20	51	50	31	14	14		180
(REAL YEAR$) TOTAL			23	62	66	43	21	22		237

Figure 23.

AN ECONOMICAL APPROACH TO SPACE POWER SYSTEMS

Fred Teren
NASA Lewis Research Center

A continuing concern of NASA is the high cost of its future space programs and, in particular, the cost of power and energy in support of space programs projected for the mid-1980's and beyond. Figure 1 illustrates projected energy demand for all NASA, DOD and Civil missions for the time span 1981 to 1995. It can be seen that the projected energy demand increases by about a factor of ten between 1981 and 1990. Typical energy costs have ranged from about $300 to about $2000 per kW-hr, with an average of about $800/kW-hr for long-duration missions. At these levels, the cost of the required energy would be several billion dollars per year by about 1985 and might constrain the numbers and types of programs NASA will be able to carry out. Thus, it is important that NASA find ways to reduce the cost of space power systems. One way to accomplish this is to start with the traditional space-qualified systems approach and look for ways to reduce costs through modifications in procedures and changes in components. Non-recurring costs can be reduced by elimination of custom and unique designs for each mission and by volume buys to take advantage of mass production techniques. Also, unnecessary documents and specifications can be eliminated, and ways can be found to simplify the component parts. This approach is being pursued extensively in the NASA space power systems program.

The advent of the Space Transportation System (i.e. Space Shuttle) will affect the cost of space power systems in several ways. The most obvious effect will be a substantial reduction in the cost of transporting systems from Earth to low Earth orbit (LEO). A less obvious but potentially very important effect is that a new approach to accomplishing mission reliability is made possible. Historically, payload costs have typically been driven by requirements to assure that highly complex, advanced technology, irretrievable, weight-critical, and schedule-critical vehicles and spacecraft performed to full requirements on their first and only flight.

In order to establish that these systems have a high probability of performing for the mission duration, reliability requirements were introduced at the onset of the program and integrated into nearly every aspect of the development. They include such practices as conservative designs, redundancies, use of high reliability components, clean room facilities, established fabrication techniques, quality control, inspections comprehensive testing, etc.

With the introduction of Shuttle it is possible to either service space systems in orbit, replace modules in space, or return faulty units to Earth for repair. Failure of a component or system in space need not imply mission failure. Thus, mission reliability can be achieved through a combination of component reliability, maintenance and servicing. This is what we refer to as the commercial approach to reliability. There are, of course, many reliable low cost commercial systems operating today which give dependable performance because they can be serviced. These systems, which were not designed for space application, include aircraft, automobiles, appliances, radios, television and large power generating equipment. With Shuttle making space servicing practicable, it is of interest to study the possible application of a low cost commercial approach to the design and fabrication of reliable space systems.

Inherently related to the use of a commercial approach is the use of commercial components (i.e. components used in terrestrial and/or aircraft equipment). Relative to space-qualified components, commercial components typically are produced in larger numbers and at lower cost but may be less reliable. Thus, the use of commercial practices and components represents a tradeoff between cost, reliability, and servicing, in which low cost is emphasized and reliability is achieved through servicing and maintenance.

The program objectives are therefore to demonstrate the applicability of a commercial approach and commercial components to the development of a low cost photovoltaic space power system. Once this has been achieved, the knowledge gained and procedures established can be applied to other space systems as well.

Before undertaking a substantial effort to identify and apply commercial practices to the development of space power systems, the potential for substantial cost reduction should be established first. In order to do this, mission and system requirements must first be specified, so that a possible power system and its components may be identified. The power system's proposed mission is a circular 200-nautical-mile-altitude Earth orbit having inclination of 28 degrees. The system must supply 2 kilowatts of electrical power continuously to the load and is specified to consist of a solar array power source and batteries for energy storage. Excess energy from the array is stored during periods of sunlight and transmitted from storage to the load during solar occulation.

A comparison of costs of space-qualified and commercially available equipment is shown in table I for the principal categories of a space power system which is designed to meet the above specified requirements. A 5-kW solar array is required to deliver 2 kW average power to the load. The space-qualified solar array blanket is assumed to cost 1.5 million dollars, or $300/watt. Approximately one kW-hr of energy storage is required, and NiCd batteries are used for this purpose. Power processing includes a battery charger and bus regulator, and mechanical systems include a solar array drive and deployment mechanism. It is assumed that the commercial system will utilize industrial type components such as terrestrial solar arrays and avionics batteries and power converters. It can be seen that over half the cost of the space-qualified system is the cost of the solar array. On the other hand, a cost of $30/watt is typical for a terrestrial array. Substantial cost savings are also possible if commercial equipment is used instead of space-qualified equipment in the other categories.

Overall, nearly an order of magnitude cost reduction is estimated to be possible - $390,000 for the commercial system vs. $2,650,000 for the space-qualified system. It is recognized, however, that the quoted costs for the commercial equipment are based on the use of the system in a terrestrial environment. Some modifications would surely be required if this equipment were to be used in the space environment, and these modifications would inevitably increase cost. However, these cost increases are expected to be small compared to the potential cost saving of over two million dollars. On the basis of these preliminary cost estimates, further pursuit of the commercial approach is justified.

Our approach to the development of an economical approach to space power systems is to conduct two programs, which we call ECOP (Economical Orbital Power) and SPEX (Space Power Experiment). The objective of ECOP is to demonstrate the applicability of a commercial approach to the development of a low cost photovoltaic space power system. The objective of SPEX is to demonstrate the application of industrial hardware for space power systems.

The ECOP program starts with studies and leads eventually to the design, fabrication and testing of a 2-kW space power system. The studies will define and compare commercial and space-qualified approaches to the design, fabrication and testing of a photovoltaic space power system and estimate the cost which would result for each approach. The specific power system type to be considered is a photovoltaic system, with rechargeable batteries for energy storage. Batteries, rather than fuel cells, are specified for energy storage because of the availability of both space-qualified and industrial types. The system is to operate in a low Earth orbit and is to provide 2 kW of average power to the load. A contract has recently been initiated with Solarex to study the commercial approach to space power system development. A second contract is anticipated, to conceptually design and estimate the cost of a space power system using the traditional space-qualified approach. The Solarex contract will examine the commercial approach in detail, including the approach to design, fabrication, documentation and R&QA. Considered in the study will be costs, manpower, methods, practices and procedures involved for a complete cycle of a new product from conceptual design to a finished fabricated product. These studies will estimate the cost of space power systems developed through the use of commercial and space-qualified approaches. This will establish with more assurance the potential for cost reduction of space power systems through the use of a commercial approach.

The next step in the ECOP program is to design and fabricate, under contract, a 2-kW photovoltaic space power system, using a commercial approach as defined in the earlier study. By conducting this program as a contracted effort, the cost of commercial power systems will be firmly established. The choice of a 2-kW power level allows possible use of the system by free-flyer experiments and allows a system to be developed at low cost which is still large enough to supply information about the cost of future multikilowatt systems. Potential users will be contacted during the design phase, and an appropriate test program for the power system will be evolved through consideration of user needs.

Concurrent with the above program, Lewis is conducting an effort to design, build and flight test a small (less than 100 watt) photovoltaic space power system. This program, called SPEX (Space Power Experiment), will demonstrate the application of industrial hardware for space power systems. Lewis engineers will define the system, select and purchase commercially available components, integrate the system and define and conduct a limited test program.

The SPEX experiment is a low cost, solar array - battery power system. The power system consists of terrestrial solar arrays and an avionics battery and dc to dc power converter. The battery charge scheme is based on the capability of the battery to accept a low rate overcharge for an indefinite time period so no battery charge is required.

All costs will be accounted for in the SPEX program and compared with predictions made by the ECOP studies. The SPEX power system is scheduled to be flight tested on the long-duration experiment facility (LDEF).

TABLE I

Cost of space power systems using space-qualified and commercial components. System operates in LEO, delivers 2 kW average.

Category	Space Qualified	Commercial
Solar Array	$ 1.5M	$ 150K
Batteries	180K	5K
Power Processing	360K	10K
Mechanical Systems	225K	75K
Systems Integration and Qualification Testing	400K	150K
	$ 2.65M	$ 390K

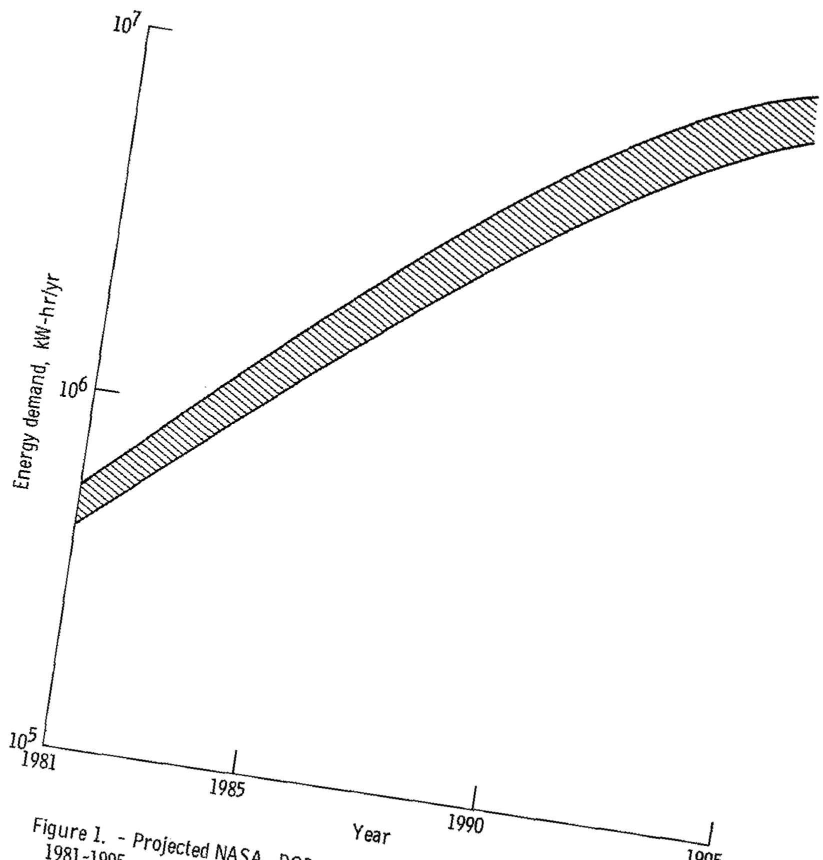


Figure 1. - Projected NASA, DOD, and civil space energy demand for the years 1981-1995.

PHOTOVOLTAIC POWER SYSTEMS WORKSHOP

Cochairmen: Harrison J. Killian
Aerospace Corp.

and Ronald W. Given
Lockheed Missiles & Space Co., Inc.

The photovoltaic power systems workshop was divided into two groups. Their discussions concerned apparent deficiencies in NASA planning and technology development relating to a standard power module (25-35 kW) and to future photovoltaic power systems in general. Corollary conclusions and recommendations, although not stated, are believed to be obvious from the discussions.

The concerns of the workshop are listed in order of importance. It is significant that the two groups, functioning independently, reached similar positions.

A. Have adequate system studies been done -

1. To establish guidelines for standardization?

There is a need to develop design guidelines for commonality, modularity, materials, design options, etc., all of which are important to standardization and low cost. These guidelines should evolve in harmony from an identification of the technical problems associated with both near-term and far-term power systems, that is, from appropriate system studies.

2. To establish a design that is flexible -

a. For adjusting to various user needs?

Many future missions are being identified that would use large amounts of power. These missions involve diverse functions and diverse orbits, that is, diverse needs as regards a power system. A power module which hopes to "capture" many, if not most, of these missions should be sufficiently flexible in design to satisfy these diverse needs without inflicting undue penalties on any user.

b. For incorporating technology advances?

Many improvements in component and design technology can be postulated during the hopefully long life cycle of a standard power module. New users will want these improvements. The basic module design should be capable of accommodating improvements with a minimum of rework and new development.

c. For growing to larger size?

A power module design capable of growing in output would seem to be axiomatic with high mission capture rate over a long life cycle. As above, these higher output versions should be obtainable with a minimum of rework and new development.

3. To establish that the initial design (25-35 kW) is the right direction to evolve to the next generation (100-300 kW)?

It was indicated in the conference that the first power module should be an evolutionary precursor to the larger power systems which would follow. In order to achieve this, studies of the larger, more far-term systems should be performed.

B. Should a standard power system module be developed separately from a standard spacecraft?

If each subsystem of a complete spacecraft were developed separately as a standard module, the composite result could be chaotic. It seems possible that the necessary compatibility among subsystems, aimed of course at best meeting user needs, might best be achieved by development of a standard spacecraft.

C. Have proper approaches to cost reduction been identified?

1. What are the major cost elements?

Most of the cost attributable to a power system after it is in orbit is believed to be of nonhardware origin. Thus it may be that efforts to reduce cost could be more productive if they focused on nonhardware as well as hardware costs. Typical nonhardware costs are engineering and testing.

2. What are the relationships between component reliability, reliable design, and low cost?

Relaxation of component reliability requirements may be possible through fault-tolerant power system design. If so, how much relaxation might be possible and what is the cost reduction payoff? Certainly, "infant mortality" and generic failure requirements cannot be relaxed. Similar statements/questions could be posed regarding a maintainable design.

3. Can a large investment in development yield low recurring and life cycle costs?

A highly sophisticated design, thoroughly wrung out during development, might be able to utilize less reliable (less expensive) components and still achieve long power system life, yielding overall a low life-cycle cost.

D. Is energy storage avoidance being considered adequately?

Energy storage is the greatest technology hurdle for large power systems. High-energy-density batteries or other energy storage devices are a must if energy storage is to be included in large power systems without an exorbitant weight penalty. The history of high-energy-density battery development gives little cause for optimism about future prospects. System designs which avoid or minimize energy storage needs may be highly desirable.

E. Is attitude control being considered adequately?

Attitude control (including configuration management) is an important feasibility issue with large solar power systems. Solar array flexing is a major concern in this area.

F. Are thermal effects of heat rejection on solar array configuration stability being adequately considered?

Thermal control and heat rejection is a major design area. A good deal of emphasis in this area was evident and is considered appropriate. However, are the thermal effects on solar array configuration stability being adequately considered?

G. Is assembly of large power systems in space being considered adequately?

Large power systems (above about 50 kW in size) may have to be put in orbit by using more than a single launch and then assembled. The LSST (large space structures technology?) study purportedly is concerned with large solar array assembly in space but not with large solar power system assembly in space.

H. Is terrestrial photovoltaic work being factored into space power systems for possible payoff?

It is probable that the large amounts of money being spent to develop low-cost photovoltaic systems for terrestrial power will have some payoff for space use. It also seems probable that the differences between space and terrestrial requirements will eventually result in the terrestrial solar cells not being directly applicable to space. Thus, efforts should be undertaken to channel promising terrestrial developments into space-type solar cells.

SOLAR CELL WORKSHOP

Chairman, Eugene L. Ralph
Spectrolab, Inc.

The workshop addressed three issues in respect to the NASA solar cell technology requirements for future orbital missions. First, we identified technology areas that were considered most significant and indicated what deficiencies and concerns we had with each area. Second, we made recommendations of what tasks should be undertaken to reduce the costs and risks of future orbital power systems. Third, we made an attempt to identify the lowest priority items in the present program in terms of content and timing.

TECHNOLOGY DEFICIENCIES AND CONCERNS

Three technology areas were identified as being most significant, and the concerns for each are listed along with some conclusions. The three most significant areas were radiation resistance, manufacturing capability, and cost reduction. The comments made for each area are listed here:

(1) Radiation resistance: This area was considered more important than efficiency. Of particular concern was end-of-mission-life efficiency. It was noted that military requirements are often quite different than NASA's. The conclusions were that we still have many approaches available to improve end-of-life output, that good radiation resistance and high efficiency are compatible objectives, that both material properties and impurity control are major factors to be better understood, and that other materials such as gallium arsenide and amorphous silicon provide new opportunities for progress.

(2) Manufacturing capability: The concern here was the capability and availability of new improved solar cells such as thin 2-mil cells and wraparound cells. The comment was made that users must provide the incentive for this area by deciding to use new technology. There is a need for tooling buildup and pilot production of the new technology, and this takes a lot of time and money. It was felt there was not sufficient backing of the manufacturing programs to meet the time scales projected. Also there was a feeling that the qualification and integration steps for achieving technology readiness were uncertain. The conclusions were that more stimulation of thin-2-mil-cell and wraparound-cell manufacturing capability is needed, that sustained comitments are needed, that there is no assurance these new technologies will be available when needed, that long-range plans and expected commitments are not sufficient assurance to manufacturers, and that large surprise program requirements (such as comet ion drive) could disrupt industry and cause problems.

(3) Cost reduction: This was stated to be "a can of worms." High-volume production does not seem to be justified by near-term program plans. The ter-

restrial cost-reduction program will have a very limited impact over the next 5 years. Terrestrial and space technology may even be incompatible. But, over the long run, they will probably be compatible and may merge together. The conclusions were that the approach should be to reduce the cost of high-quality cells rather than to increase the performance and reliability of a low-cost cell, that near-term cost reductions are a problem because of low-volume production, that the long-term cost-reduction goals can probably be met if the volume projections are correct and terrestrial technology merges with space technology, and that GaAs cell cost reduction is a major problem.

(4) Other concerns: Efficiency is a major cost driver on the complete system. Silicon cells are well developed, with open-circuit voltage the last hurdle. GaAs cells are closer to the theoretical limit. Other approaches that will lead to 25 percent or greater efficiency are a major problem. The conclusions were that 25 to 30 percent efficiency would not be needed in the next 10 years, that we must continue to build a good research base for future thrusts, that there is no need to accelerate in this area but we should continue as we are, and that lack of basic knowledge is a serious deficiency.

Process technology is primarily concerned with contacting methods although they are not now a problem. Welding technology is not well advanced and implemented, especially on very thin cells. Thin cover glasses are also not readily available. The nonglass cover technology is not a necessity but is highly desirable since glass problems are not fully known. Texturized surface technology is in pretty good shape, but absorptivity control still needs improvement.

RECOMMENDATIONS

Two recommendations made by the workshop related to activity that was needed beyond that presently being done.

(1) Pilot demonstration: It was recommended that we start demonstrating manufacturing capability of new cell technology on a large-scale pilot line basis. In particular, thin cells (2 mil), wraparound cells, and thin glass covers (2 mil) are important technologies that should be brought to manufacturing readiness as soon as possible.

(2) GaAs cell technology: It was recommended that high-efficiency, radiation-resistant GaAs cell technology be accelerated and brought to readiness. In particular, emphasis should be placed on contact metallization, manufacturability, material availability, thin cells, and the use of concentration.

LOW-PRIORITY AREAS

The workshop believes that the present program is a bare-minimum effort with no obvious areas that are unimportant, considering the very ambitious large-scale missions being projected for the future. However, if a priority rating were to be applied to the solar cell technology development program, the lowest priority areas would be the development of concentrator cells and GaAs cells. Both of these technologies are felt to be less important in the near term and primarily to provide advancements that could be used 5 to 10 years from now.

SOLAR ARRAY WORKSHOP

Paul Goldsmith
TRW Defense & Space Systems Group

The solar array workshop, which was attended by 20 people, began with a review of the needs and objectives in this area as presented by the various government representatives during the preceding sessions. The major problem noted with respect to needs was the potentially conflicting requirements of low cost and low weight. Since the importance of weight and cost and relationship between them are strongly mission dependent, the workshop concluded that the requirements of military missions in synchronous orbit could be quite different from the requirements of NASA low-orbit missions and that an assignment of specific technology deficiencies could only be related to specific mission classes.

TECHNOLOGY DEFICIENCIES

The major overall problem areas identified during the workshop were as follows:

(1) Lack of an overall program technology plan for specific mission classes

(2) Funding not compatible with technology requirements

(3) Dependency on results from DOE terrestrial programs, which may not produce usable results

The specific technology deficiencies identified were as follows:

(1) The overall problems of using solar arrays at voltages of hundreds of volts and higher are not understood and have not been amply demonstrated. Problems include not only a definitive understanding of plasma effects versus voltage, but also system level problems with higher voltage, including load switching and voltage regulation.

(2) The space application of concentrators is not well understood. Problems include

(a) Design optimization for cost or weight reductions

(b) Applicability of high concentration ratios

(c) Lifetime characteristics

(d) Packaging and deployment characteristics

(e) Heat rejection techniques

(f) Orientation and spacecraft interaction characteristics

(g) Applicability to different mission classes

(3) The relationship between solar cell stack parameters and mission weight and cost is not well understood and is mission dependent. The following approaches all have weight and cost implications whose benefits vary greatly between low-orbit and synchronous-orbit missions as well as between conventional and shuttle launches:

(a) Increase cell stack efficiency

(b) Reduce cell stack costs at the expense of efficiency

(c) Reduce cell stack weight at the expense of efficiency and cost

(4) The potential role and benefits of gallium arsenide cell technology integrated into space arrays are not clear. Problem areas include:

(a) What efficiency and cost goals are required to permit the economical use of GaAs in planar as well as concentrator arrays?

(b) Does the space utilization of GaAs arrays depend upon a terrestrial market?

ADEQUACY OF CURRENT AND PROPOSED PROGRAMS

The workshop's comments on current and proposed programs were directly related to and integrated into our discussion of problems and deficiencies as follows:

(1) Military synchronous-orbit missions are presently very close to being weight constrained because of the IUS weight restrictions. It is not clear that certain missions well above 2 kW can be launched without significant weight reductions, especially in the power subsystem area. The need for improved performance, higher efficiency cell technology, higher voltage, hardness, and higher energy density batteries was identified, but a quantitative assessment of specific needs and performance improvements was not made. It was recommended that this be done in each of the power system technology areas so that specific goals could be established for driving the technology.

(2) The needs for NASA higher power, low-orbit missions such as the power module were discussed and, except for the comments made in the deficiency section of this report, the workshop agreed with the NASA plans for proposed programs. These were understood to have included

(a) Concentrator versus planar studies

(b) Concepts for on-orbit maintainability

(c) Heat rejection techniques

(d) Techniques for solar array stationkeeping and pointing

The group concluded that in addition to these programs, systematic studies should include the importance of weight and volume in these missions, and the interrelationship between cost, weight, and volume parameters.

ADDITIONAL TASKS

Implicit in the discussion were recommendations of additional tasks which should be undertaken. In addition to these, the working group provided the following ideas and comments:

(1) Inflatable arrays

(2) Spectrum selection to increase efficiency

(3) Solar cell annealing techniques

(4) Reduce cell operating temperature

(5) Interconnect designs for long-life operation

(6) Rollup array backup for PEP and/or power module usage

(7) Accelerate work in polymer coatings for cells

(8) On-array power conditioning

(9) Techniques for converting array power to ac

(10) Accelerate development of low-weight and low-cost arrays by evaluating alternative solar array module approaches which could lead to significant improvements both in manufacturability and in weight reduction

SOLAR ARRAYS

<u>NEEDS</u> - LOW COST AND LOW WEIGHT

- o PROBLEMS OF PRIORITY
- o MISSION DEPENDENCY

<u>GENERAL DEFICIENCIES</u>

- o LACK OF OVERALL PROGRAM TECHNOLOGY PLAN
- o DEPENDENCY ON OTHER AGENCIES
- o FUNDING NOT COMPATIBLE WITH PROGRAM R&D'S

<u>SPECIFICS</u>

- o NEED FOR EARLY GaAs SYSTEM VERIFICATION
- o MAJOR WORK IN HIGH VOLTAGE TECHNOLOGY REQUIRED
- o NEED TO INTEGRATE CONTROLS, STRUCTURES AND POWER
- o DID NOT SEE ADVANCED TECHNOLOGY IN LIGHTWEIGHT STRUCTURES, ANNEALING CONCEPTS, RADIATION HARDENING, SPECTRUM SHIFTING, ETC.
- o "CLEAN SHEET" APPROACH IN MODULE DESIGN
- o DID NOT SEE AGGRESSIVE APPROACHES TO COVER PROBLEM
- o ROLE OF CONCENTRATORS NOT CLEAR

<u>NEEDS</u> - COST/WGT

<u>DEFICIENCIES</u>

- o HIGH VOLTAGE
- o CELL STACK OPTIMIZ
- o APPLICABILITY OF CONC.
- o CONTROLS/STRUCTURES
- o GaAs VERIFICATION

<u>IDEAS</u>

- o "CLEAN SHEET" APPROACH
- o ANNEALING
- o CELL OPERATING TEMP.
- o INTERCONNECT LIFE
- o INFLATABLE CONCEPTS

BATTERY WORKSHOP

Cochairmen: Richard H. Sparks
TRW Defense & Space Systems

and Floyd E. Ford
NASA Goddard Space Flight Center

The battery workshop was attended by 18 people from industry and government. Review of the requirements for energy storage and the plans for battery development was vigorous, with widespread participation. The workshop followed a debate format, with the objective of recommending improvements to the development plans presented by NASA and the Air Force. The issues addressed were

(1) Significant technology deficiencies which can be identified

(2) Adequacy of current and proposed programs to resolve the technology deficiencies identified

(3) Additional tasks which should be undertaken, including benefits and timing

(4) Lowest priority items in the presently planned program, both in content and in timing

The workshop was limited to 2 hours, which necessarily restricted the depth of our review; however, a general consensus was reached by both workshop subgroups. The main theme of their conclusions was that the power system trade-off studies for large power systems (25 kW and larger) have not been adequately performed:

(1) Early 1970's battery technology is generally compared with projected 1980's fuel cell capability.

(2) Small-battery data are compared with a single large-fuel-cell-system data base.

(3) Cost trade-offs do not include redundancy and scaling factors for larger battery and fuel cell systems.

(4) Effects of bus voltage on the energy-storage-system concept have not been identified.

The consensus of the workshop was that the battery development program is underscoped because the trade-off studies have not adequately considered battery advanced technology capability and relatively low cost at the <u>system level</u>; hence, the priority associated with advanced battery development is too low.

The most significant technology deficiencies identified by the workshop were as follows:

(1) Battery life development goals do not meet mission-planning goals and ground rules:

(a) Low-Earth-orbit missions are being planned based on a 10-year life. Battery development goals in the present programs are 5 years. A 10-year life requirement is needed.

(b) Geosynchronous-orbit missions are being planned with lifes ranging from 7 to 15 years. Battery development goals are 10 years. A 15-year life requirement may be needed.

(2) Low-cost batteries are widely discussed; however, mission requirements are based on low-cost battery systems which include total system reliability, life, and maintenance costs. The low-cost battery concept is not totally consistent with low-cost battery system concepts:

(a) Battery costs are already relatively low; but implementation costs within the aerospace software and hardware systems are high. Implementation cost reductions cited in the workshop are

- To increase battery life to reduce replenishment costs
- To reduce battery redundancy weight and complexity to save initial costs and launch costs

(b) Batteries are small (less than 60 Ah), necessitating the use of multiple battery assemblies and control electronics for 25- to 100-kW systems. Cost data presented at the workshop show a significant cost leverage by reducing the number of components in the battery system. A 100- to 500-Ah battery size capability is needed.

(c) Parts screening costs have significant leverage on battery costs; however, without screening of parts made with commercial processes, reliability and hence maintenance cost are impacted strongly. Refined screening methods commensurate with the maintenance cost models being planned should be developed.

(d) Battery system redundancy is very costly both in weight and in electronics complexity. A low-cost, low-weight redundancy concept is not available and needs development.

(3) Deep-discharge, long-life applications for batteries are not well developed. Operating conventional batteries at deep depths of discharge (DOD) is the best single way to reduce energy-storage-system specific weight. Present nickel-cadmium battery studies are based on 15-percent DOD for 3.5- to 5-year low-Earth-orbit missions. Increasing DOD to 60 percent for a 10-year life is a major improvement which should be developed to meet the mission goals presented:

(a) Nickel-cadmium batteries can be operated at deeper depths of discharge (up to 85 percent for some applications) by using new operating methods and newly developed plate and separator processes. The nickel-cadmium battery should be developed for a 10-year life at greater than 20-percent DOD.

(b) Nickel-hydrogen batteries are being developed by the Air Force for a 1-year life at 80-percent DOD and for a 5-year life at 60-percent DOD in low Earth orbits. The capability should be improved to 60-percent DOD for 10 years, and a low-Earth-orbit flight experiment should be flown at 60-percent DOD to validate the system in space.

(4) Peak loads requiring load-leveling battery systems were shown by both Air Force and NASA mission models in 1987-1988. No technology is available to support load-leveling missions within reasonable weight constraints. This mission requirement should be immediately translated into energy-storage-system requirements for further development.

(5) Large bulk-energy-storage battery systems are not available for 100-kW and larger systems in the 1990's. Work in sodium-sulfur, advanced lithium, and large nickel-hydrogen systems needs to be done to identify potential capabilities and to develop a large bulk-energy-storage system for the larger space platforms.

Other technology deficiencies were cited by the workshop but had lower priorities than the preceding items.

During the workshop discussions, the group commented on current and proposed NASA and Air Force programs. The comments are summarized as follows:

(1) Overall funding for battery systems is too low to meet the large-power-system program goals. NASA has underestimated the potential of batteries for large space systems by using overly conservative battery performance data for system trade-off studies, thus causing a low priority to be given to advanced battery development. Emphasis on battery development should be increased.

(2) A 100-Ah cell program start in 1978 is needed; however, the scope of the planned feasibility program should be expanded to encompass a cell size range of 100 to 1000 Ah to provide a more useful data base for large-power-system designs.

(3) Very limited studies of nickel electrodes and nickel-cadmium cell designs for long life are planned and funded. A much more extensive concentration on electrochemically impregnated and other more physically stable electrodes should be initiated and directed toward 10-year life cycle service, including

(a) More fundamental studies of deep-discharge cycle phenomena

(b) More investigation of operating methods which do not overstress the electrode structure

The payoff will be for both NiCd and NiH_2 batteries. Other comments on existing programs were varied and, in general, fell within the scope of those summarized.

The workshop worked toward a summary of additional tasks needed to meet the NASA and Air Force mission objectives. Several additional tasks were identified:

(1) The first recommendation was to perform updated spacecraft system-level trade-off studies comparing existing and future battery systems, including fuel cell systems. The studies should use projected 1980-82 capability and 1985-87 capability and should determine sensitivity to different types of batteries and fuel cells. Weight, cost, redundancy, life, and control-system-complexity sensitivity variables should be included:

(a) System cost and weight trade-offs should be performed for candidate energy storage systems.

(b) System redundancy and refurbishment trade-offs should be performed for candidate energy storage systems.

The workshop noted that maintenance of heavy components in space through refurbishment may not be practical and will be very expensive.

(2) The second recommendation was to initiate development of substantially larger battery cells with active cooling provisions which can be adapted to large space systems. Development should be centered on a 250- to 500-Ah cell or battery design, scalable from 100 to 1000 Ah for system design flexibility.

(3) The third recommendation was to initiate a near-term NASA program to implement the nickel-hydrogen battery system into NASA systems studies. An early low-Earth-orbit flight experiment should be performed at deep depths of discharge (40 to 60 percent) to assess the technology capability and to determine refinements needed for larger space cell development.

(4) The fourth recommendation was to increase funding to cover a broader battery technology program for large long-life systems, including

(a) 10-year Low-Earth- and geosynchronous-orbit system life

(b) Lower system costs by eliminating system complexity:

- Fewer batteries
- Low-cost redundancy concept
- Minimum replenishment

(c) Increased monitoring of DOE technology developments to select timely technology spinouts for space applications (Lithium, sodium, and other high specific energy system developments should be monitored closely.)

(d) Investigation of technology required to support high-voltage power systems

The workshop encountered difficulty in identifying the three lowest priority technology items in the NASA plan. The work planned is narrow and needs to be widened. However, priority recommendations were developed to be offered in assisting future plans as follows:

(1) Increasing the specific energy of NiCd cells should receive much lower priority than increasing the life and utilization of stored energy at deeper depths of discharge. The largest overall system cost and weight savings for the larger space systems planned will result from doubling the battery life and increasing depth of discharge to 40 to 60 percent. Small increases in specific energy are of little overall system value.

(2) Higher priority should be placed on following DOE high-energy-density systems work, including

(a) Close surveillance

(b) Identifying early spinout for space to meet 1990-2000 goals

A low priority should be placed on starting new developments paralleling DOE work until a good basis for spinout is established.

(3) Highest priorities should be applied to long-life battery systems which can deliver a high percentage of stored energy to the power system, in contrast to short-life scientific system battery requirements. Space system cost-effectiveness models presented show that short-life energy storage systems are extremely expensive.

These workshop conclusions and recommendations were presented before the entire symposium. No disagreement was received from the floor. The workshop summaries were therefore documented in this report with minimum changes for clarification.

FUEL CELL/ELECTROLYZER WORKSHOP

Chairman, James K. Stedman
United Technologies Corp.

The fuel cell/electrolyzer workshop was primarily concerned with determining the studies and early development testing required to define, demonstrate, and improve the characteristics of fuel cell/electrolyzer energy storage technology. The goal of this effort is to insure that the system is ready to begin engineering development in the 1985 time period. The consensus was that the highest priority task is to expand the endurance data base for both fuel cell and electrolyzer technology and to do it on cells and at test conditions most representative of the energy storage mission. This implies that tests should be conducted on multicell stacks of larger-area flightweight cells. The cells should incorporate design features and operate at conditions that favor long life, such as thicker membranes or separators and lower operating temperature. Both fuel cells and electrolysis cells should be cycled between a very low load and the design load to a low-Earth-orbit profile.

A second high-priority task is to conduct a conceptual design study of a typical, multikilowatt satellite installation, including the necessary trade studies to optimize the design. This conceptual design would study such basic design options as the degree and level of planned maintenance (whether on a cell, substack, component, or module basis) and would be influenced by such vehicle and mission trade factors as replacement interval versus net weight and cost, efficiency versus weight, and weight/cost versus heat rejection temperature. This study is required early in the program to set specific technology goals for the electrochemical performance; to determine the importance of a maintainable packaging concept; and to point out nonelectrochemical areas of the system design, such as controls and circulators, where early development may have a large payoff in reducing total program costs and risk.

The third high-priority task is to demonstrate operation of a complete fuel cell/electrolysis system in a breadboard unit that would include reactant control and storage, pressure balancing, heat removal, and load control. Early breadboarding of powerplant functions has historically been valuable in pinpointing unforeseen problem areas and deficiencies in subsystem concepts. This test would best be conducted by mating the two candidate electrolyte technologies, acid and alkaline, to demonstrate that the systems are compatible and that proper interfacing can be maintained. The consensus of the workshop was that maintaining this compatibility would be no problem because of the demonstrated stability of the electrolytes being considered.

In addition to these three priority areas, the following specific technology deficiencies were mentioned at the workshop:

(1) The possibility of utilizing the same cell and cell stack for both fuel cell and electrolysis functions has not been determined. This commonality

may have weight and program cost advantages if both units are more or less identical.

(2) No high-power electrolysis units have been designed for space nor has large multicell stack endurance testing been completed.

(3) A mathematical model for cell performance and degradation with time is required for both the fuel cell and electrolysis module performance in order to better project the limited performance data to be obtained to the desired life goals of 40 000 or more hours.

(4) To increase cell endurance capability, work on cell materials and electrode catalyst should be increased for both the fuel cell and electrolyzer technologies.

The fuel cell electrolysis program as proposed by H. McBryar of the NASA Johnson Space Center at this conference appears to be essentially adequate to provide desired technical readiness in 1985. However, it is recommended that two tasks be added to this program:

(1) A conceptual installation design for a multikilowatt satellite should be started in 1979.

(2) During the 1982-84 time period the breadboard program should be continued and upgraded with the design improvements as they become available. The control, instrumentation, and maintainability features required should be emphasized.

Research-and-development funding during 1978-79 was judged inadequate to provide the necessary basic technology for the development activities planned for the early 1980's. For example, there is no current, and only minor planned, R&D effort on the electrolyzer for power generation. In addition, the current and planned R&D effort of fuel cells does not provide for enough testing to insure that the desired endurance data base is obtained. It was also judged that the planned system technology funding is low in light of the ambitious life goals and degree of demonstration desired for the concept.

In summary, the group was enthusiastic about the fuel cell/electrolyzer concept for low-cost orbital energy storage and felt that the weight, cost, and flexibility advantages evident in the industry studies can be realized through a well-planned and well-executed technology program beginning in the next fiscal year.

POWER MANAGEMENT WORKSHOP

Cochairmen: Robert E. Corbett
Lockheed Missiles & Space Co., Inc.

and Sidney W. Silverman
Boeing Aerospace Co.

It was the objective of the power management workshop, which was attended by 25 people, to review the NASA OAST space power technology program and especially the proposed new initiatives and to critique the program in view of power technology needs of planned space station and power station applications. As directed by the conference organizers, the discussions were conducted in accordance with the following problem statements:

(1) Identify the three most significant technology deficiencies.

(2) Comment on the adequacy of current and proposed programs to meet these deficiencies.

(3) Recommend additional tasks which might be undertaken to reduce the cost and risk of future orbital energy systems.

(4) Identify the three lowest priority items in terms of content or timing.

Discussions were started with a keynote from the workshop chairman, who posed several problems in connection with the planned development of large space power systems. The most relevant and interesting of these is the technology transition: Given the state of the art in spacecraft power conditioning equipment and given the fact that at some significantly higher power level there is some different technology which is optimal for that power level, when and how is this technology transition made with the evolution in power system size? In the presentation sessions which preceded the workshops, little was said about the power management requirements and state of the art considering its relative importance in power system development. One presenter suggested that on-array power conditioning, dc-to-ac conversion equipment, and solar array pointing systems are some of the power electronics needs of large space power systems; another presenter proposed 30- to 100-kHz inverters and lightweight power transmission lines as important technology areas. Generally these proposals were not supported by results of systems level studies presented at the symposium, so that a great deal was left unsaid. The power system concept, topology, power distribution, grounding, and isolation are important system characteristics which strongly affect the power conditioning equipment and component requirements.

With this as a keynote, there followed a general discussion and gathering of ideas from the participants with the following points being made: The thrust of advanced component development must be generic in that, in the early stages, component development usually precedes the identification of an application. Power switches, for example, are always needed both for power conversion and for switchgear applications. A very successful example of this is the remote power controller technology which has been in development for many years but will be used on P80-2 (SIRE) and the 25-kW Power Module which are both very near-term power system applications. In spite of the usefulness of generic component development, many participants expressed the view that power systems level studies were needed to guide future work. An example of this need that was discussed to some depth was the great variety of options in future power system design: various voltage levels of higher voltage dc and the ac distribution approach. There did not seem to be any agreement on the advantage of ac distribution, but all agreed that too little study had been devoted to the subject to support any particular decision.

Following this general discussion, our effort turned to obtaining a consensus of views on the workshop problem statements. The charts presented in the final plenary session are given as figures 1 to 4, corresponding to the workshop problem statements.

TECHNOLOGY DEFICIENCIES

There was considerable agreement on the power management technology deficiencies, with switchgear for high-power systems having the highest priority. There was also uniform agreement that the component work and the spacecraft system work done to date had not been linked by adequate power system studies. High-power components were also identified as one of the top three items. This should include not only the capacitors, magnetic components, and switches used in power conversion, but also the connectors and other parts used in the distribution system.

ADEQUACY OF PROGRAMS

Throughout our discussions there was a need to distinguish between near term and far term, and these terms themselves were not well defined. Inasmuch as the 25-kW Power Module has not yet been developed, many participants viewed even this very near-term program as a beneficiary of current technology efforts. The strongest view expressed about the current program is that there are no studies on ac or dc systems which bear upon the space systems under study and which would furnish data on conversion and distribution efficiencies, environmental problems, safety, etc., for either the most immediate programs (power module) or for the most distant (e.g., SPS).

RECOMMENDED TASKS

There were many ideas and considerably less agreement on what new tasks were needed, and priorities were a matter of personal viewpoint and the nature of one's affiliation. The greatest agreement was on system concept trades: that ac and the various dc voltage level options should be given a detailed comparison.

A typical area where there was disagreement on priority was the proposal of a user load study to better define the output interface for high-power systems. Some felt that loads would always be diverse, just as they have been to date, and that such a study would be useless. There was fairly uniform agreement on the need for more definitive information on environmental effects and safety considerations for high-power systems.

LOWEST PRIORITY ELEMENTS

Because of the limited funding level for the current technology program and the stated need to have a certain amount of generic component research, there was no strong feeling about what items should be deleted from the technology program. There was agreement that system studies must have priority now in view of the lack of them to date and their potentially strong impact on component development and laboratory demonstrations.

TECHNOLOGY DEFICIENCIES

HIGH VOLTAGE SWITCHGEAR — RELAYING, PROTECTION

LACK OF INFORMATION TO CONDUCT SYSTEM LEVEL TRADES

- AC VS DC (FAR)
- DC VOLTAGE LEVEL SELECTION (NEAR)

HIGH VOLTAGE/HIGH POWER COMPONENTS

- HIGH VOLTAGE CAPACITORS
- MAGNETIC COMPONENT CHARACTERIZATION AND LIGHTWEIGHT TECHNIQUES
- FAST SWITCHING TRANSISTORS

ADEQUACY OF PROGRAMS

FOR NEAR TERM: PROBLEMS ARE SOLVABLE BUT NOT AT CURRENT LEVELS--E.G. REMOTE POWER CONTROLLER IS NOT FULLY DEVELOPED

SYSTEM STUDIES AND DATA BASE TO ACCOMPLISH THEM ARE LACKING

FOR FAR TERM: FUNDING LEVELS INADEQUATE TO ADDRESS NEW SYSTEM AND COMPONENT DESIGN PROBLEMS

RECOMMENDED TASKS

- USER STUDY DEFINE LOAD REQUIREMENTS
- SYSTEM LEVEL STUDIES: TO EVALUATE AC, HVDC OPTIONS
  TO EVAI UATE GROUNDING, EMC, AND ENVIRONMENTAL INTERACTIONS
- ELECTRONIC PACKAGING STUDIES EMPHASIZING THERMAL PROBLEM (NEAR)
- SETUP OF A STANDARDIZATION COMMITTEE FOR POWER SYSTEM SPECIFICATION AND INTERFACES
- EVALUATION OF SAFETY PROBLEM AND PRACTICES
- TEST TECHNIQUES FOR SCREENING OF LOW COST COMPONENTS
- DEVELOPMENT OF SENSING AND DETECTION GEAR FOR AUTOMATED CONTROL
- IMPROVED INTER-COMPANY/AGENCY COMMUNICATION

LOWEST PRIORITY ITEMS

- DEVELOPMENT OF ADVANCED COMPONENTS TO EXTREME ENVIRONMENT
- THERMAL CONTROL/PACKAGING
  - NEAR TERM - IMPORTANT FOR POWER MODULE
  - FAR TERM - LOWEST PRIORITY

GENERALLY, DEVELOPMENT OF COMPONENTS AND CIRCUITS FOR HIGH POWER (FAR TERM) IS LOWEST PRIORITY UNTIL SYSTEM LEVEL STUDIES AND BOX CONCEPTS ARE COMPLETE

LASER/MICROWAVE TRANSMISSION WORKSHOP

Chairman, Wayne S. Jones
Lockheed Missiles and Space Co., Inc.

The laser/microwave energy transmission workshop, held during the Orbital Power Systems Symposium on 1 June 1978, discussed the most significant technology deficiencies and the adequacy of on-going and proposed programs and recommended additional tasks which might be undertaken to reduce cost and risk. The personnel attending the workshop represented both government and industry and each one actively participated in the discussions.

The current primary modes of orbital energy transmission are microwaves and laser beams. The microwave transmission system concept - namely, the Solar Power Satellite (SPS) - is a far more mature concept than laser beam transmission. Many millions of dollars have been expended developing the SPS concept, whereas, less than \$200,000 has been expended on laser beam transmission; therefore, the adequacies of on-going and planned programs, the technology deficiencies, and the recommended additional tasks had to be considered separately.

Table 1 lists the significant technology deficiencies. With the microwave system, only space-to-ground energy has been considered, primarily because of the large wavelengths (10 to 12.5 cm) considered to date. Shorter wavelengths (mm's) could prove to be beneficial and cost effective for space-to-space transmission even though lasers with much shorter wavelengths did not prove to be cost effective. The demonstration of high-powered phased arrays and their scalability to the sizes required for SPS is also lacking. While this area has received some study, the workshop felt a deficiency existed. Environmental and safety problems have also been addressed and certainly the planned program includes considerable funding, particularly in the environmental area. However, the technology requirements and status to resolve the problems have not been clearly defined.

TABLE 1. - SIGNIFICANT TECHNOLOGY DEFICIENCIES

<u>Microwave</u>

- Shorter wavelengths for space-to-space
- High-power phased array scalability demonstration
- Technology to resolve environmental and safety problems

<u>Laser</u>

- Research & development of laser devices/energy converters
- Phase locking techniques

Because of the infancy of the laser energy transmission concept, many technology deficiencies are to be expected. A few of the technology areas are large, lightweight, adaptive optics; pointing and tracking; laser devices; and energy converters. Some of these areas are undoubtedly being addressed by DOD and may be directly applicable to NASA and DOE missions. As a result, the workshop decided to limit the technology deficiency considerations to basically the laser device and energy converters that may be unique to NASA/DOE requirements. Lasers for orbital energy transmission need to be long-life, closed-cycle devices - characteristics not necessarily required by DOD. Many lasers are currently in various states of development and it is not clear that any particular one is going to emerge as "the" laser. Carbon dioxide electric discharge lasers (CO_2 EDL's) are currently considered to be scalable to multimegawatt power levels; however, the electrical-to-laser efficiency leaves something to be desired plus the fact that shorter wavelengths will relieve optics diameter requirements. Two relatively new laser concepts (solar pumped, and free electron) may offer more potential to increase overall system efficiency than other devices further along in development. Energy conversion systems used in conjunction with lasers also should be investigated in significantly more depth. Among these are photovoltaics, thermionic, thermoelectric, and heat engines in conjunction with an energy exchanger. Phase locking to obtain multimegawatt laser power levels is a must. Single lasers have physical limitations relative to size and volume so that phase-locking techniques must be used to avoid wavefront interferences which cause the laser beam to spread beyond desirable limits.

Table 2 lists areas in which the adequacy of on-going and proposed programs should be discussed. For microwave transmission, workshop personnel questioned the adequacy of development testing and technology necessary to define costs and satisfy environmental and safety questions. Relative to laser beam transmission, there was an emphatic concensus that a continuing end-to-end systems analysis was inadequate. Current studies have shown that laser energy transmission has certain advantages that microwave transmission does not have, and the advantages may more than outweigh the lower electrical to laser efficiencies. Many trade-offs need to be made to optimize the overall system for a credible evaluation. The current laser programs within NASA seem to be splintered and without focus. Planned programs do not seem to exist nor is there any degree of probability that they will occur. The workshop personnel felt that close coordination between NASA and DOD should be established to avoid duplication and that a well-planned program should be laid out.

TABLE 2. - ADEQUACY OF ON-GOING AND PROPOSED PROGRAMS

Microwave

- Technology inadequate to define costs and satisfy safety and environment effects
- Inadequate development testing

Lasers

- Inadequate continuing end-to-end system analysis
- Splintered effort within NASA
- NASA/DOD coordination

Table 3 lists some additional tasks that could help to reduce cost and risk. The microwave concept is fairly well funded, and the new tasks may possibly be accomplished with some change in emphasis without significant additional funding. The laser energy transmission concept needs a series of studies to bring this infant concept to a point where credible evaluations can be made to determine its utility for various missions and the possible synergistic effects that could be realized. Candidate concepts for the first application should be evaluated and a logical "road map" should be established to fit within budget and schedule constraints. Technology requirements should be established for projected scenarios and the status of these technologies determined, with initiation of research and development programs to satisfy the technology requirements in a timely manner.

TABLE 3. - RECOMMENDED ADDITIONAL TASKS TO REDUCE COST AND RISK

Microwave

- Increased scope of system, application, safety, and environmental impact studies
- Subscale testing

Laser

- Mission/first application/supporting technology research

We who participated in the laser/microwave transmission workshop thank the National Aeronautics and Space Administration for permitting us to express our views for their consideration in future plans for orbital power systems. We feel the symposium was very worthwhile from many standpoints and would like to see similar symposiums in the future.

THERMAL MANAGEMENT WORKSHOP

Chairman, Roy L. Cox
Vought Corp.

It was evident during the workshop that the development of future orbital power systems will require increased emphasis on thermal management and the integration of thermal concepts into the overall system design in its earliest stages. From the workshop evaluation of orbital-power-system thermal management needs and technology status, three critical areas have been identified where current technology is deficient and where the first priority should be placed on technology development:

(1) Thermal interfaces (acquisition and transport)

(2) Large deployable/constructable radiators

(3) Long-life thermal systems

Subdivisions of these three areas are given in figure 1. From that figure it can be observed that future trends are toward higher energy density equipment and subsystem cooling requirements; efficient transport of thermal energy across both moving and disconnectable static joints; modularity to minimize cost, allow growth, and permit replacement or maintenance for long life; and basic life improvement through design and materials selection.

At the time of the symposium it was found that a well-defined thermal management program had not yet been formulated to support orbital-power-system needs. The following task guidelines to fulfill the most urgent needs were established during the workshop session:

(1) Define and evaluate specific techniques to provide interfacing thermal control of heat generating components and subsystems.

(2) Establish, evaluate and compare heat rejection system alternatives.

(3) Identify and investigate specific critical long-life problem areas, based on these evaluations.

Subsequent to the symposium a specific, preliminary, recommended program was developed, with the discrete objectives and benefits listed in figure 2. Although this program is keyed to the 1986-1987 initial operational capability (IOC) of the NASA large power module, elements of the program have been designed to address (to a limited extent) needs for higher temperature heat acquisition and rejection - which apply, for example, to solar concentrators and high-temperature equipment cooling (such as energy conversion devices). Also, the program addresses needs specific to very large systems, such as the solar power satellite, where space manufacture and/or assembly may be required to obtain feasible subsystem costs. Figure 3 lists program outputs and figure 4 gives the milestone schedule. An estimate of resource requirements has been made and is given in table 1. From the assessment of needs and development lead times it is clear that an FY '79 start is required to support the 1986-1987 IOC.

The workshop was also asked to identify lower priority technology deficiencies. Since this is an extremely difficult undertaking considering the small amount of workshop and symposium time and information available to conduct assessments, we, instead, listed items that were not included within the scope of the program. These are given in figure 5. Upon more detailed investigation according to the previously stated guidelines and during systems trades and concept studies, it may be found that some of these omissions must be incorporated. For that purpose the table 1 resource chart includes a 20 percent growth allowance.

TABLE 1 - RESOURCES ($M)

FY	'79	'80	'81	'82	'83	'84	'85	CUM
LARGE CONSTRUCTABLE/DEPLOYABLE RADIATORS	.30	.66	.96	1.20	1.56	.42	--	5.10
LONG LIFE THERMAL SYSTEMS	.18	.42	.54	.84	.54	2.04	2.04	6.60
THERMAL ACQUISITION/TRANSPORT	.30	.54	.44	.22	.18	--	--	1.68
VERY LARGE RADIATOR SYSTEMS	--	--	--	.12	.42	1.08	1.20	2.82
TOTAL	.78	1.62	1.94	2.38	2.70	3.54	3.24	16.2

FIGURE 1

-THERMAL MANAGEMENT TECHNOLOGY DEFICIENCIES

- FIRST PRIORITY THRUSTS

THERMAL INTERFACES

- HIGH HEAT FLUX COOLING OF POWER GENERATION SYSTEMS SUCH AS CONCENTRATORS
- COOLING HIGH VOLTAGE SYSTEMS
- HEAT TRANSPORT ACROSS JOINTS (FLUID, CONTACT, GIMBALS)

LARGE DEPLOYABLE/CONSTRUCTABLE RADIATORS

- LIGHTWEIGHT (ESPECIALLY GEOSYNCHRONOUS)
- MINIMUM LAUNCH VOLUME
- MODULAR FOR GROWTH
- SPACE ASSEMBLY/REPLACEMENT
- TECHNIQUES FOR MINIMIZATION OF DEPLOYED AREA

LONG LIFE THERMAL SYSTEMS

- MINIMUM COMPLEXITY THROUGH LARGER SUBSYSTEMS
- FLUID COMPATIBILITY AND HIGH TEMPERATURE MATERIALS
- MICROMETEOROID COUNTERMEASURES
- COMPATIBILITY WITH SPACE PLASMA ENVIRONMENT AND NATURAL RADIATION
  - THERMAL COATINGS
- MAINTAINABILITY

FIGURE 2

THERMAL MANAGEMENT TECHNOLOGY PROGRAM

OBJECTIVES

- DEVELOP THE TECHNOLOGY NECESSARY FOR HEAT REJECTION FROM LARGE SPACE POWER SYSTEMS
- EXTEND THE ORBITAL LIFETIME CAPABILITY OF THERMAL MANAGEMENT SYSTEMS TO 5-30 YEARS
- PROVIDE THE TECHNOLOGY NECESSARY FOR HIGH ENERGY DENSITY HEAT TRANSFER AND TRANSPORT
- REDUCE THE COST OF VERY LARGE SCALE HEAT REJECTION SYSTEMS BY ORDERS OF MAGNITUDE

BENEFITS

- REDUCED COST THROUGH MODULARITY, SYSTEMS LEVEL APPROACH, INTEGRATION WITH OTHER SUB-SYSTEMS, REDUCED WEIGHT AND VOLUME
- REDUCED RISK BY EXPANDING EXPERIENCE BASE IN LARGE, LONG LIFE, AND HIGH TEMPERATURE SYSTEMS
- COMMENSURATE THERMAL MANAGEMENT TECHNOLOGY WITH FUTURE POWER GENERATION CAPABILITIES
- STEPWISE GROWTH CAPABILITY
- REDUCED ADVERSE ENVIRONMENTAL INTERACTIONS/DESIGN CONSTRAINTS
- TECHNOLOGY BASIS FOR HIGH TEMPERATURE AND HIGH VOLTAGE POWER SYSTEM THERMAL CONTROL
- EXTENDED LIFETIME THROUGH MAINTAINABILITY/REPLACEABILITY, MATERIALS COMPATIBILITY, MICROMETEOROID COUNTERMEASURES
- FIRM BASIS FOR SCALE-UP TO VERY LARGE SOLAR POWER SATELLITE

FIGURE 3

OUTPUT

LARGE CONSTRUCTABLE/DEPLOYABLE RADIATOR SYSTEM

- SYSTEM LEVEL TRADES INVOLVING ORBITAL POWER SYSTEMS TO ESTABLISH EVOLUTION PATH
- BREADBOARD INTEGRATED RADIATOR/POWER SYSTEM
- BREADBOARD FLIGHT-REPRESENTATIVE VAPOR-COMPRESSION/RADIATOR SYSTEM
- BREADBOARD ENVIRONMENT ORIENTATION SYSTEM FOR RADIATOR
- REPRESENTATIVE FLIGHT CONSTRUCTIBLE/DEPLOYABLE RADIATOR DEVELOPMENT
- LIGHTWEIGHT RADIATING FIN MATERIAL HIGH MANUFACTURING VOLUME PROTOTYPE
- MAINTENANCE/ASSEMBLY DEMONSTRATION IN VACUUM CHAMBER

LONG LIFE THERMAL SYSTEMS

- BREADBOARD HIGH CAPACITY HEAT PIPE & FLIGHT TEST (MINIMIZE OVERALL SYSTEM COMPLEXITY)
- LONG LIFE FLUID LOOP MATERIALS AND COMPONENT DEMONSTRATION /SYSTEMS TEST
- DEVELOPMENT AND DEMONSTRATION OF LONG LIFE THERMAL COATINGS
- THERMAL COATING REPAIR/CLEANING TECHNIQUE DEVELOPMENT AND DEMONSTRATION

THERMAL INTERFACES (ACQUISITION/TRANSPORT)

- FLIGHT REPRESENTATIVE CONTACT HEAT EXCHANGER AND QUICK DISCONNECT DEVELOPMENT AND VACUUM CHAMBER DEMONSTRATION
- FLIGHT REPRESENTATIVE FLUID SWIVEL AND FLEXIBLE FLUID JOINT DEVELOPMENT AND TEST
- BREADBOARD HIGH EFFICIENCY HEAT PIPE EVAPORATOR FOR CONCENTRATOR AND OTHER HIGH DENSITY POWER SYSTEM COOLING
- HIGH VOLTAGE DIELECTRIC HEAT TRANSFER INTERFACES
- BREADBOARD THERMAL TRANSPORT SYSTEM DEMONSTRATION FOR HIGH EFFICIENCY AND LONG DISTANCES (THERMAL UMBILICAL)
- HEAT PIPE INTERFACES FOR MODULAR ASSEMBLY TO HEAT SOURCES AND REPLACEMENT

VERY LARGE SPACE MANUFACTURED OR CONSTRUCTABLE RADIATOR SYSTEMS

- DESIGNS FOR LOW COST AUTOMATED SPACE MANUFACTURE/ASSEMBLY
- GROUND DEMONSTRATION UNIT
- SHUTTLE FLIGHT DEMONSTRATION EXPERIMENT

FIGURE 4

MILESTONES

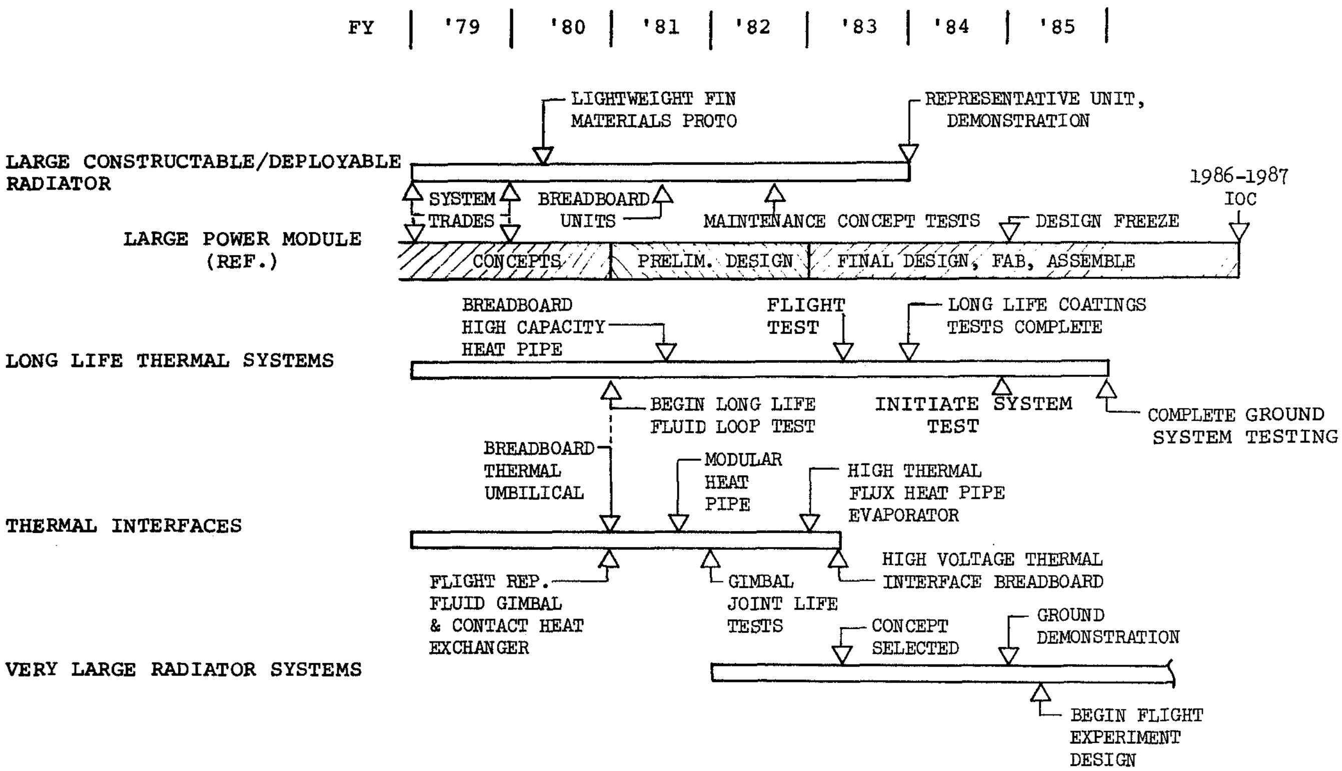


FIGURE 5

THERMAL MANAGEMENT TECHNOLOGY DEFICIENCIES - ADDITIONAL ITEMS

- THERMAL DISTORTION AND HEAT REJECTION BY LARGE STRUCTURES
- CONTAMINANT INTERACTIONS
- ANALYTICAL AND GROUND TEST METHODOLOGY TO VERIFY THERMAL MANAGEMENT DESIGNS
- TWO-PHASE FLUID BEHAVIOR INCLUDING ZERO-g EFFECTS AND FREEZING CHARACTERISTICS
- ADVANCED THERMAL STORAGE MATERIALS
- COMBINED INSULATION/PHASE CHANGE FAILURE MODE PROTECTION FOR ISOTOPE POWER SYSTEMS
- VERY HIGH TEMPERATURE AND NUCLEAR POWER SYSTEM COOLING.

NUCLEAR POWER SYSTEMS WORKSHOP

Chairman, William A. Ranken
Los Alamos Scientific Laboratory

Discussions in the nuclear power systems workshop generally followed the outline proposed by the workshop organizers of identifying technology deficiencies, assessing current programs in terms of meeting these deficiencies, recommending tasks crucial to the development of future orbital energy systems, and identifying low-priority items in the current or planned program.

TECHNOLOGY DEFICIENCIES

In considering the adequacy of technology for space nuclear power, it became necessary to broaden the scope of the discussion because the workshop felt the major deficiency in this area was the lack of a well-defined United States policy on the need and desirability of nuclear power in space. It was felt that this lack was amply demonstrated by the conflicting official statements released in reaction to the reentry of the nuclear-reactor-powered Russian satellite, Cosmos 954. The workshop was apprised that the Cosmos incident has led to U.S. activity in the United Nations to establish an international policy with regard to space nuclear power, and hence it would seem that a national position is currently being formulated.

At the present time the field of nuclear space power can be fairly said to include several variations of isotope and reactor heat source technology as well as four heat-to-electricity conversion methods. If there is a single general trend that applies to the various combinations of heat sources and conversion methods, it is the one toward higher source temperature and higher sink temperature - and consequently lighter weight systems. For this reason the workshop felt that high-temperature-materials data was of prime importance to the design and fabrication of advanced space nuclear power systems. Information on strength, creep, toughness, corrosion, joining, and coating is generally inadequate for materials suitable for reactor fuel, cladding, and structure; high-temperature thermoelectric systems; thermionic converters; insulation for both thermoelectric and thermionic systems; Brayton systems ducts and turbines; and isotope packaging.

Another area where the workshop felt technology development is definitely needed is heat rejection at elevated temperatures. Lightweight and dependable radiators are required for all the nuclear power system concepts, and very little work has been done to date.

PROGRAM ASSESSMENT

With regard to the adequacy of current programs to meet these technology deficiencies, the workshop felt that insufficient effort is being devoted to obtaining the high-temperature-materials information needed for any of the advanced nuclear systems. It noted a similar lack of attention to the development of heat rejection systems capable of operating at elevated temperature.

More generally, it was thought that too little effort is being devoted to all aspects of the development of conversion methods applicable to reactor or isotope heat sources. The thermionic conversion effort seems minimal and was noted to be decreasing in terms of constant-value dollars. The thermoelectric effort was described as very inadequate in view of the impressive potential of high-temperature thermoelectric materials. The funding picture for Brayton and Rankine systems appears to be in a highly recessive state. A summary statement of the workshop's feeling is that the overall U.S. program is grossly inadequate to defend the national interest with regard to space nuclear technology. Adequacy with regard to the reactor portion of the space power program was thought to be attainable with a research and technology development program funded at a level of about $10 million per year.

TASK RECOMMENDATIONS

The workshop was aware of the interest within the Department of Energy (DOE) in providing reactor technology for both orbital power supply and nuclear-electric propulsion applications. It felt that close cooperation, coordination, and mutual support between NASA, the Department of Defense, and DOE is necessary to build a national program adequate to meet future space power needs. These needs were recognized to be in a continuously evolving state requiring periodic mission definition and analysis efforts that are fully integrated with power systems analysis work.

It was the workshop's consensus that NASA should take the lead in establishing an adequate research and technology program for advanced space nuclear power systems, particularly in conversion system development, heat rejection system work (with emphasis on elevated temperatures and heat-pipe technology), and high-temperature-materials research and characterization.

LOW-PRIORITY TASKS

The workshop observed that little of what is considered high-priority work is currently being undertaken and that the number and level of programs that are in progress are too low to make the selection of low-priority tasks meaningful.

WORKSHOP ON ENVIRONMENTAL INTERACTIONS WITH LARGE ORBITAL POWER SYSTEMS

Chairman, Alan Rosen
TRW Defense & Space Systems Group

The environmental interactions workshop identified over 20 separate and distinct effects of the environment on large orbital power systems. Table 1 lists the problems in the categories of plasma interactions, radiation interactions, and other environmental effects. Opposite each problem is a brief description of its potential impact on large orbital power systems.

These problems are at various stages of our understanding and our ability to cope with them. Some are easily handled within the technology, but others represent obstacles that are seemingly insurmountable. For example, it is not clear that it will be possible to avoid shorting the power subsystem through the plasma if it is operated at high voltages ($\sim$40 kV) and low altitudes (a few hundred nautical miles). In the area of plasma interactions, we found many unanswered environmental interaction questions that have not been investigated or carefully evaluated for their effect on large orbital power systems. At this time it is not clear that it will be possible to establish an optimum voltage as a function of altitude or that all possible plasma leakage mechanisms have been identified. The plasma interaction area is characterized by many problems that are beyond present technology (in our ability to cope with them). Therefore this area was identified as requiring immediate attention and work in order to determine the feasibility, and to optimize the design, of proposed concepts for large orbital power systems. Table 2 lists current programs in the area of environmental interactions with space systems. The comments opposite each of the current programs indicate that these programs are not applicable to large orbital power systems. Table 3 lists environmental interaction tasks that are specifically oriented toward large orbital power system technology. The tasks in table 3 are arranged so that highest priority tasks are presented first and then tasks of successively lower priority. The effects of these tasks on optimizing the design and cost of large orbital power systems are given opposite each task description. It is important to stress that the lowest priority tasks are important elements in the design and development of large orbital power systems but can be undertaken within the existing technology and were therefore not categorized as "technology drivers." The highest priority tasks are technology drivers. They affect the feasibility of the proposed alternative concepts of orbital power systems and will be design drivers during the design development of any of the concepts selected for further study.

Table 1

LARGE ORBITAL POWER SYSTEMS
ENVIRONMENT INTERACTION PROBLEMS

PROBLEM	IMPACT
PLASMA INTERACTIONS	
Current Leakage - Pinhole Effects Sparking Leakage Effects Enhanced Leakage at Low Altitude Leakage through Ion Engine/Change Exchange High Voltages Induced by Ambient Plasmas Plasma Grounding vs Single Point vs Multiple Grounds	Power losses through the plasmas.
Environmentally Induced Sparking Structural Replacement Current Resonances Plasma Instabilities and Enhanced Wipe-Out Effects Plasma Structural Heating Problems Contamination: Vacuum Deposition of Arcing By-Products	EMI and possible burn-out of I.C. components. Degradation of thermal control system. Power losses.
Long Term Differential Voltage Stress—Degradation of Thin Sheets Long Term Plasma Charge Collection Enhancement (Pinholes/Micrometeoroids)	Degradation and lifetime curtailment.
RADIATION INTERACTION	
Radiation Belt Effects Solar and Primary Cosmic-Ray Effects	Solar cell degradation damage and lifetime curtailment of electronic subsystems. Hazard to manned missions and EVA.
OTHER ENVIRONMENTAL EFFECTS	
Microwave-Ionosphere Interactions	Power losses.
Voltage Differentials during Mating and EVA	Hazard during rendezvous.
Environment Modification Problems	Sweeping out trapped radiation belts and ambient plasma.
Ambient Magnetic Torquing Magnetic Plasma Torquing Effects	Degradation of attitude control system.
Micrometeoroid Impacts	Damage and degradation of system.

Table 2

ENVIRONMENTAL INTERACTIONS—ONGOING PROGRAMS

PROGRAM	COMMENTS
NASA/LeRC SPACECRAFT CHARGING	
High Voltage—Plasma Effects Investigations	• Oriented towards smaller structures
• Leakage Currents • Arcing • Special Configurations • Analytical Models	• Oriented towards basic phenomenology • Inadequately funded to support orbital power system
AF/NASA JOINT SPACECRAFT CHARGING PROGRAM	
• Material Development	• Oriented towards relatively small orbital spacecraft
• Material Characterizations • SCATHA	• Oriented towards geosynchronous orbit only
NASA/MSFC WAKE SHIELD STUDIES	
• High Vacuum Technology • Wake/Sheath Studies • Magnetic Effects	• Not directly applicable to orbital power systems

Table 3

ENVIRONMENTAL INTERACTIONS—TASKS TO REDUCE COST/RISK

PRIMARY TASK: Investigate plasma interactions with large space structures as a function of spacecraft and plasma parameters—with the objective of generating design guidelines and recommended practices.

TIMING: This task must be undertaken concurrently with conceptual design studies, and continue through the system design phase.

TASK	BENEFIT AND/OR IMPACT ON ORBITAL POWER SYSTEM
INVESTIGATE PLASMA INTERACTION WITH LARGE SPACE STRUCTURES	Develop design guidelines and recommended practices
• Sparking Power Loss	Determine optimum voltage for efficient operation
• Leakage through Plasma	Determine acceptable configurations—impacts weight, efficiency of heat rejection
• Long Term Survivability Contamination, Aging	Determines lifetime of mission
• Electromagnetic Effects	Impacts attitude control, weight, regarding requirements
ENVIRONMENTAL MODIFICATION STUDIES	Environmental impact effects
RADIATION EFFECTS	Lifetime of solar arrays—lifetime of electronic subsystems—design of radiation shield
MICROMETEOROID ENVIRONMENT	Lifetime of system in orbit

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15. Supplementary Notes

16. Abstract

A two-day symposium on the technology requirements for future orbital power systems was held on May 31 and June 1, 1978, at the Lewis Research Center in Cleveland, Ohio. Papers were presented by NASA headquarters, NASA Centers, an NASA contractor, and a Department of Defense representative in three areas: government program plans for space electric power, the status of present applicable technology, and a review of current and recent system studies. After the presentations, workshops led by industry representatives were held in 10 technology disciplines and systems areas to discuss government program plans, technology deficiencies, the adequacy of the programs to resolve these deficiencies, and recommendations for additional tasks that might reduce the testing of future orbital energy systems and the risks involved. Workshop chairmen presented the conclusions of their workshops at a plenary session. The proceedings contains the papers presented as well as the conclusions of each workshop.

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