3.6 TBR

GRC
~~W/FA~~
SQT-JM

JPL PUBLICATION 78-7

Space Power Systems Technology Enablement Study

(NASA-CR-155804) SPACE POWER SYSTEMS TECHNOLOGY ENABLEMENT STUDY (Jet Propulsion Lab.) 135 p HC A07/MF A01 CSCL 22A

N78-19186

Unclas
G3/16 15170

National Aeronautics and Space Administration

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California 91103

JPL PUBLICATION 78-7

Space Power Systems Technology Enablement Study

L. D. Smith
J. W. Stearns

February 15, 1978

National Aeronautics and
Space Administration

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California 91103

Prepared Under Contract No NAS 7-100
National Aeronautics and Space Administration

Preface

The work described in this report was performed by the Control and Energy Conversion Division of the Jet Propulsion Laboratory.

This report completes the work requirements under NASA RTOP 790-40-19, "Technology Enablement - Space Power Systems", sponsored by OAST-Code RX, Space System Studies.

ABSTRACT

A study has been performed of the power systems technologies which are either enabling or enhancing for future missions requiring a few kilowatts or less and using the Shuttle. The U. S. Space Transportation System (STS), when properly utilized, will greatly expand our capabilities in space. The required advances in space power systems to support those capabilities have been systematically determined together with missions that are enabled and enhanced by the technologies identified. The analysis procedure employed benefit/cost/risk analyses to identify high payoff technologies and technological priorities.

CONTENTS

Figures

Tables

SECTION I

SUMMARY

The objective of this Technology Enablement study at JPL was to identify and assess long-life space power system technology, determining the elements that will enable or significantly enhance potential automated NASA missions with small- to moderate-scale power needs using Shuttle. The studies included missions and technologies through the 1980s and 1990s. The study plan compares mission needs with technology programs, resulting in a set of new technology candidates that were then ranked according to benefit/cost/risk incentives. Large Earth-orbital power and space industrialization, being studied elsewhere, was excluded from this study.

A. MISSIONS

Table 1-1 is a summary of expected automated missions for STS synthesized from several studies. Following the standard format for planetary missions, the listing indicates whether the missions are considered to be reconnaissance, exploration, intensive study, or utilization. The time span for accomplishing the missions is also indicated for each of the mission types, above the mission destination. Lunar and inner planet missions are considered together, outer planet and solar escape missions are considered commonly, and Earth orbit and solar missions are identified together.

Power system requirements for the set of missions in Table 1-1 were reviewed against existing development programs, and the missions for which technology is becoming available were then eliminated. These eliminated missions are shown enclosed in the cross-hatched boxes in the table. The missions not eliminated require additional "enabling" technology, and are therefore considered as technology drivers.

B. TECHNOLOGY CANDIDATES AND RANKING

The NASA R&T programs in space power were reviewed against the driver missions defined above to obtain a list of enabling technology candidates. Other possibilities thought to be feasible in meeting mission needs were added to this list. The listings were then collected under two basic groups. The first of these groups was also divided into two subsets.

(1) New Technology Readiness program candidates were listed where the technologies and applications were well-defined and where development program costs and schedules could be reasonably identified. Those technologies directly applicable to space electric power needs were listed first. Technologies that were more-or-less tangential to power needs were listed last.

Table 1-1. Expected Mission for STS (Automated)

Mission	Reconnaissance	Exploration	Intensive Study	Utilization
Lunar and inner planets	1980-1982 Venus	1984-1990 Mercury orbit Mars penetrator Venus orbit/ lander	1985-2000 Lunar polar orbit Lunar Laboratory Mars Rover and SSR Mercury Rover and SSR Venus Rover and SSR	1992-2000 Lunar base[a]
Outer planets and solar escape	1982-1988 Saturn/Uranus Jupiter/Neptune Solar escape Comets and Asteroids	1982-1998 Jupiter orbit Saturn orbit Pluto orbit Asteroid SSR Solar escape Comet SR	1993-1998 Orbit, landers, Rovers, SSR at Jupiter, Saturn, Uranus, and Neptune	1993-2000 Nuclear waste disposal
Earth orbit and solar		1979-1985 Space telescope Solar max. Solar probe Land/sea observations High energy observations Climate/weather Communication Radio astronomy Extra ecliptic	1984-1993 Large telescope Solar observatory Spacelab[a] Earth survey Atmospheric physics Gravity/magnetics Large antennas Power transmission	1980-2000[b] Earth resources Earth observations Communications Navigation Climate/weather Space power Industrialization Colonization

[hatched box] denotes mission covered by present programs

[a]Automated technology with manned systems.

[b]Separate studies being accomplished.

(2) New R&T Base program candidates were listed when there were less-specific applications goals and technology development was less certain. Consideration should be given at some later date to the transfer of program elements to technology readiness programs.

Benefit/cost/risk analysis was made on the first group of technologies. This was the principal input for establishing technology priorities, although other discriminators were also used. For the R&T program candidates, a set of qualitative discriminators was used to set priorities.

The two groups of technology candidates are listed in Table 1-2 in order of priority assigned. For more detailed information the reader should refer to Section VII of this report or to the B/C/R analysis of Appendix D. All candidates of group one appear to have an exceptionally high payoff and should be supported if at all possible.

C. CONCLUSIONS

The majority of NASA missions through 1985 are presently covered by technology readiness programs. A commendable R&T job is being done by NASA in this regard. No one has critically reviewed those programs as part of this study to guarantee their adequacy, but it was assumed that there is sufficient dialogue between NASA Headquarters and the cognizant NASA centers to assure the timely achievement of goals. Technology readiness programs might be further evaluated to consider if modifications would allow better utilization of the volume and mass capabilities of the Shuttle. Relaxation of mass constraints and repackaging might produce very significant cost savings.

Advancing the power technology beyond 1985 is closely correlated with the evolution of the STS. Cost effectiveness in the application of new technology will be associated with larger payloads, higher energy missions, increasing mission lifetimes and on-orbit servicing potential. Technology enablement in space power systems will be necessary if NASA space programs are aimed at expanding mission capabilities but with little expansion of NASA resources. Except for some early planetary missions that remain mass limited, low cost mission concepts become practical with greatly-relaxed mass and power constraints. By application of these, and also perhaps applying advanced power systems methods to systems development, as recommended by the NASA/RP RTAC (Ref. 1-1), it appears that an order-of-magnitude power increase for Shuttle-launched payloads can be accomplished with very small cost increase.

Highest priority has been given to gallium arsenide solar arrays. This technology with its preliminary demonstration of radiation-resistant characteristics (Ref. 1-2) is needed for long-life, low cost photovoltaics in geosynchronous Earth orbit, where solar flare degradation would be very significant. But more than this, radiation hardened arrays will be needed at the earliest possible date to enable the electric cargo orbit transfer vehicle (COTV) with its long dwell time in the Van Allen belts. This technology will enable much larger payloads per Shuttle launch to geosynchronous orbit at reduced cost. Such a system will give the STS an overwhelming market advantage over all expendable launch vehicles.

Table 1-2. Technology Candidates and Ranking

Priority	Technology Readiness Candidates		R&T Base Candidates
	A. Space Power Needs	B. Tangential Needs	
1	GaAs Photovoltaics	Solar Furnace (Replaces 50 kWe Supply)	Primary Fused Salt Battery (with indigenous materials)
2	Radioisotope Thermionic Generator or Advanced RTG	Heat Shield/Heat Pipes	High Temperature Power Components (Inner Planets)
3	Systems with Heat Shield/ Heat Pipes	Fiber Optics Cabling (GSE Requirement)	Radiation Hard Power Components (Outer Planet Radiation Belts)
4	Laser Power Beaming	Stirling Refrigerator (~10 year cryogenics)	Flywheel Energy Storage For Long Life, Many Duty Cycles

78-7

REFERENCES

1-1. Report on the Status and Prospects of the NASA Space Power and Propulsion Research and Technology Program, Ad Hoc Working Group on Space Power and Propulsion, Research and Technology Advisory Council, Committee on Energy Technology and Space Propulsion, National Aeronautics and Space Administration, May 30, 1975.

1-2. Letter, R. Stirn, JPL, to Randolph, L.P., NASA Code RR, July 22, 1977, Ref. 346-59-RS:vf (JPL internal document).

SECTION II

INTRODUCTION

Space power systems technology has a vital part in future missions planning. Low cost, high performance (light weight), and long lifetime can contribute to major advances in our space program if proper planning can be accomplished. Large power systems for electric propulsion are already making an impact in mission planning. Limitations on payload mass and power, it appears, can be rolled back significantly for both Earth orbit and planetary missions (Reference 2-1 through 2-3). Power technology enablement studies reported herein have been carried far enough to give considerable credence to this approach. As a result, we might suggest that more aggressive mission planning be considered.

The scope of this study was such that only order-of-magnitude estimates of performance and cost could be made. Yet, for a first-level screening of space power technologies this is probably adequate. Any further effort should be constrained to focused research and technology programs, for which a prioritization has been recommended in this report. No in-depth cost studies could be accomplished within this scope, but to the extent possible expert consultation was solicited as needed.

This study was performed in calendar year 1977. It therefore has a built-in obsolescence that the reader must recognize. Technology and mission planning activities are proceeding in parallel with this work, and cannot be entirely factored into the study. Additional hardware design such as the Spinning Solid Upper Stage (SSUS) design and development was finalized during this study period. New concepts of reliability are in process of being developed for Shuttle, the implications of which have not been carefully analyzed as yet. And there is major uncertainty in the mission set beyond 1985.

The need for advancing the space power technology for spacecraft is closely dependent on the evolution of the U.S. Space Transportation System (STS). That evolution as assumed for this Space Power study is shown in Figure 2-1. The STS is initially based on Shuttle as a launch vehicle (Reference 2-4). Future STS analysis studies by Boeing Aerospace Company for NASA/JSC (Reference 2-5) indicate additional desired expansion in the 1990s to a heavy lift launch vehicle (HLLV), while the technology model by General Research Corporation (Reference 2-6) also proposes a single stage to orbit (SSTO) vehicle. Beyond launch, the interim upper stage (IUS) (Reference 2-7) and spin stabilized upper stage (SSUS) for Shuttle will eventually be replaced by a set of highthrust and low-thrust vehicles for Earth orbit transfer, lunar transfer, and Earth escape and return (Reference 2-8). Preliminary planning of propulsion technologies, based on space mission estimates for the next 20 years, has been in process for several years. This evolution of the STS was used in the development of the mission sets in Section III.

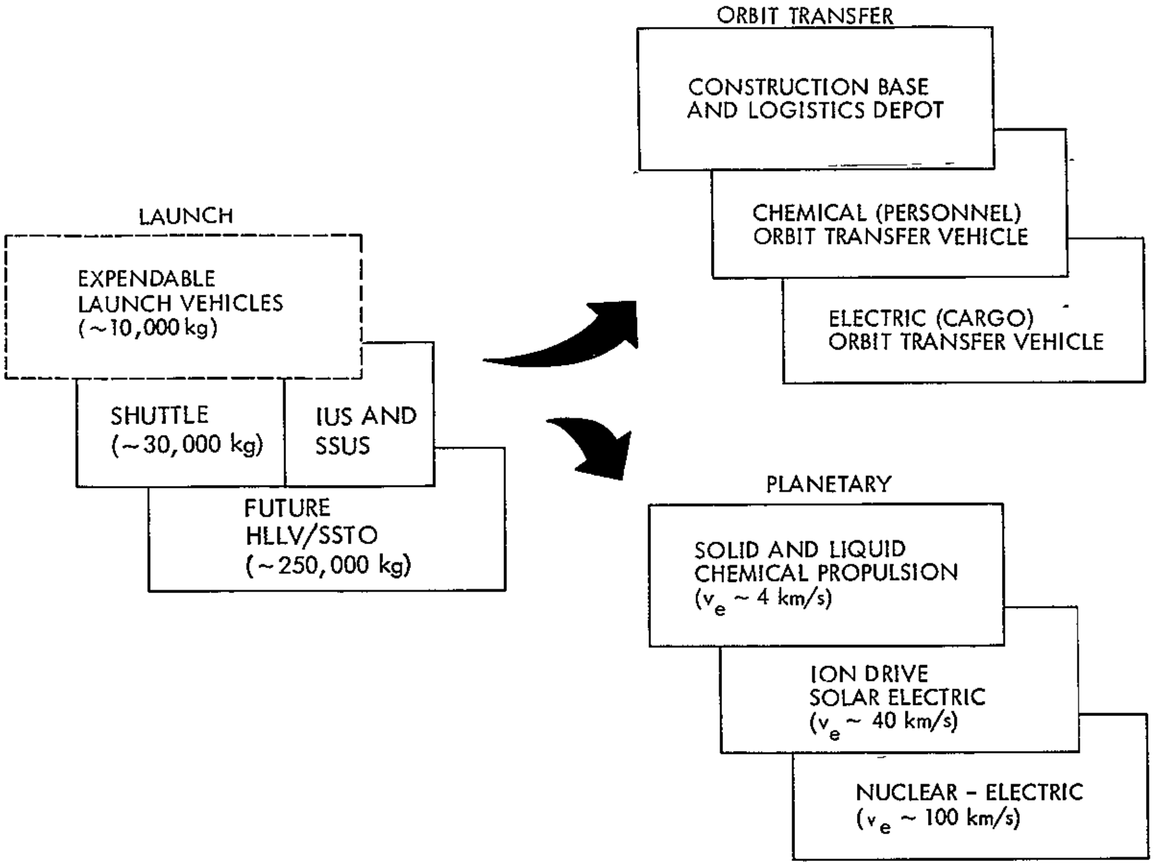


Figure 2-1. Space Transporation System Expected Morphology and Evolution

A. OBJECTIVES

The objective of the Space Power Systems Technology Enablement study at JPL was to identify and assess long-life space power system technology, determining the elements that would enable or significantly enhance potential automated NASA missions with small- to moderate-scale power needs using the Shuttle. This specifically excluded multi-megawatt requirements such as those of the Space Power Satellite or space industrialization needs. Missions and technologies through the 1980s and 1990s were identified. Principal emphasis was given to power levels from 1 watt to a few kilowatts. However, there were some areas where integration with larger electric propulsion supplies was obviously needed.

B. STUDY APPROACH

As shown in Figure 2-2, a mission model was generated that would provide a maximum reasonable exercise of power systems technology concepts. Derivation of this all-inclusive mission set is detailed in Section III of this report. The ongoing "technology readiness" programs within NASA's space power R & T are discussed in Section IV of this

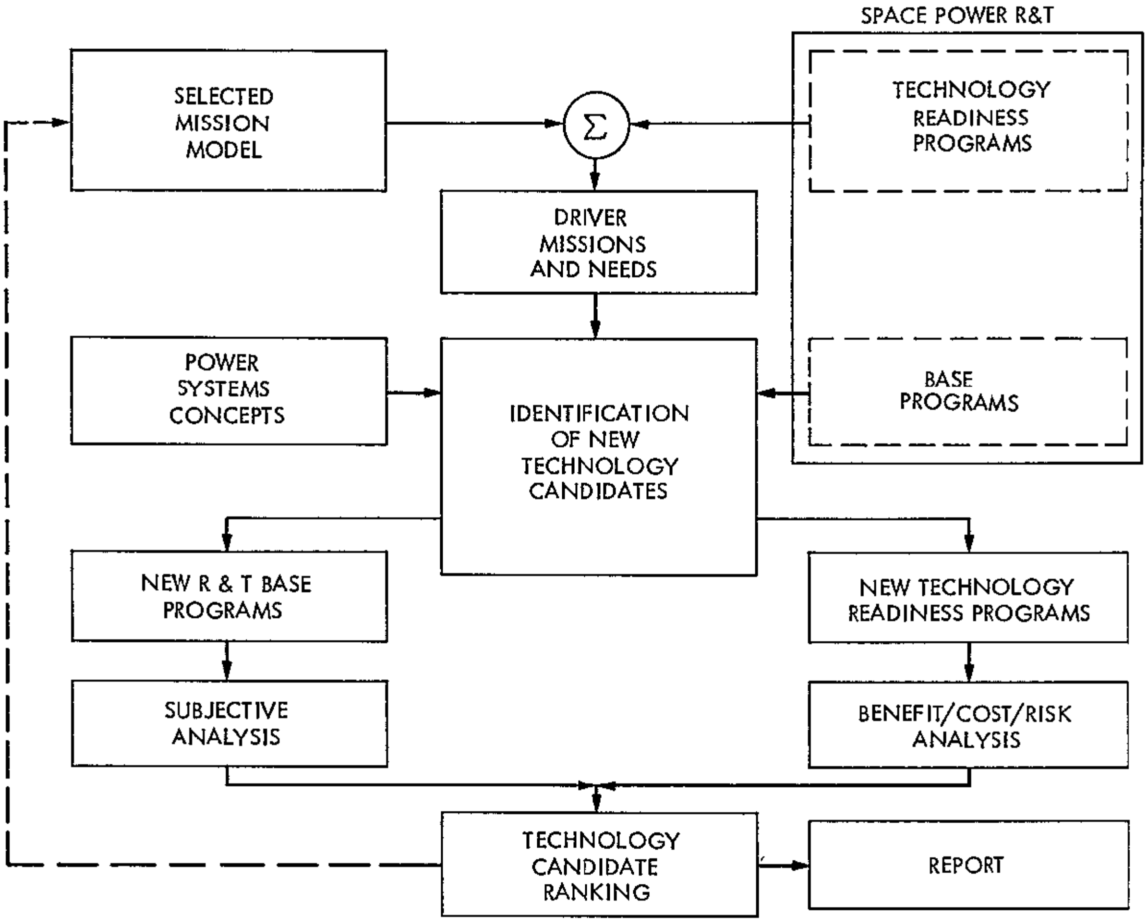


Figure 2-2. Space Power Systems Technology Enablement Approach

report, and are specifically defined as those R & T programs scheduled for demonstrated flight readiness by a particular date. Where technology readiness programs would not fulfill the needs of the mission model, "driver missions" are identified. These missions, which require technology advance, are discussed in Section V of this report, together with the identification of new technology candidates that should meet the needs of the driver missions. The technology candidates evolved from an iterative consideration of the R & T Base programs (programs that are exploratory in nature, investigating basic phenomena and principles for future space applications), power systems conceptualization and the driver missions' needs and schedules. The candidates were found to fall into two basic groups: new technology readiness programs and new R & T Base programs (more-specifically focused). Group (1) was also divided into two subsets.

(1) New Technology Readiness program candidates were listed where the technologies and their applications were well-defined and where development programs could be reasonably

identified in terms of program costs and schedules. Only part of the candidates were directly applicable to space electric power needs and these were listed as belonging to a subset (a). The remainder of the technology readiness candidates were identified as being more or less tangential to our power needs and these were listed in a subset (b).

(2) New R & T Base program candidates were listed where it was thought to be premature to identify new technology readiness programs. Technical feasibility and system concepts were not certain and applications could not be adequately defined. Better focus of R & T programs was therefore suggested, with consideration to be given at some later date to the transfer of program elements into technology readiness programs.

Sections VI and VII of this report discuss the benefit/cost/risk analysis and the application of other discriminators to the technology candidates in order to provide a ranking or prioritization. Study conclusions are also provided in Section VII.

C. SCHEDULE

This study was performed within a 10-month period in calendar year 1977. The milestone and task schedule for the final effort is shown in Figure 2-3. The tasks shown relate to the study approach as discussed above. This schedule allows for possible impacting of NASA FY 1980 planning.

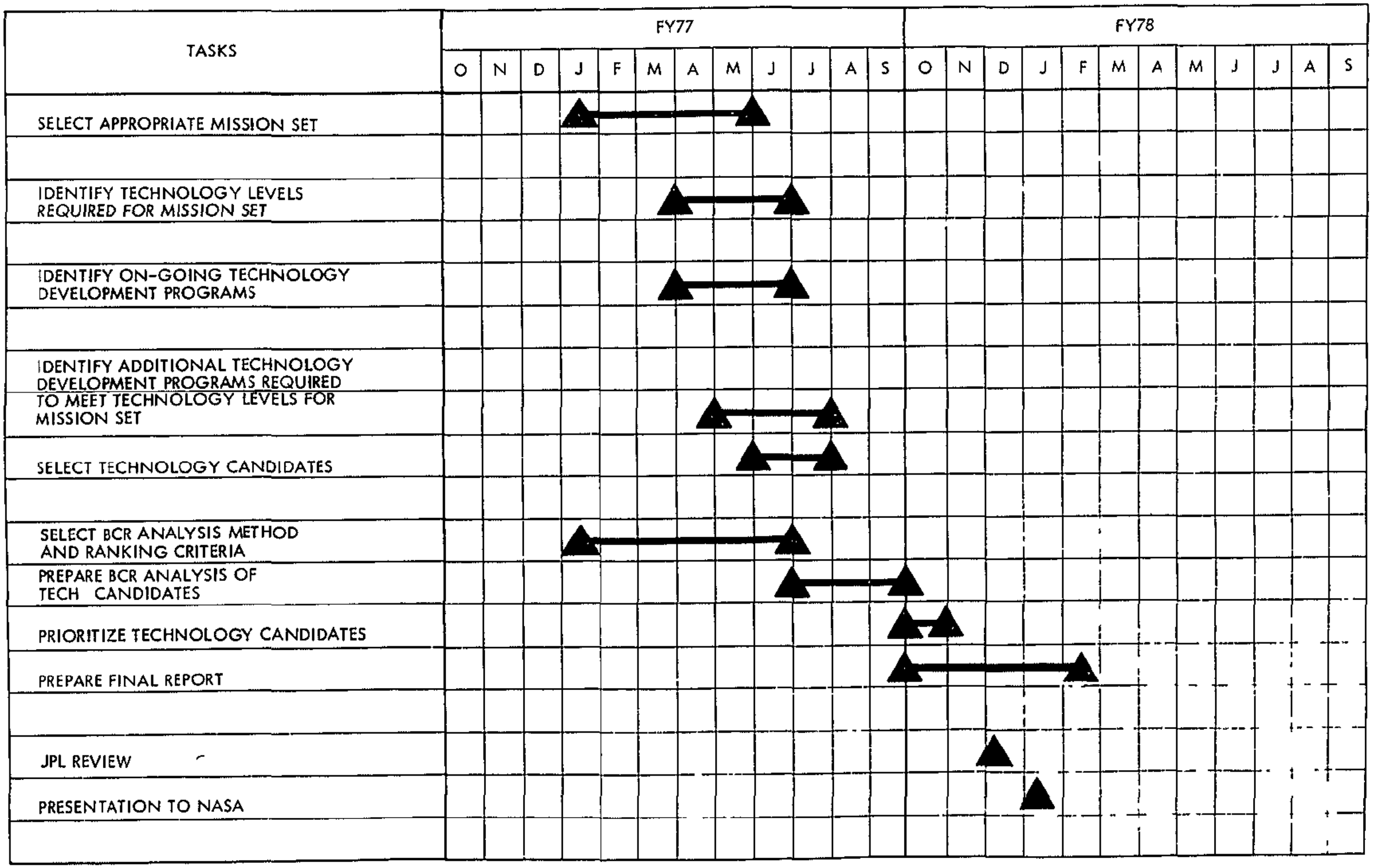


Figure 2-3. Milestone Schedule for Space Power Systems Technology Enablement Studies.

REFERENCES

2-1. Friedman, L., and Ivie, C.V., Automated Planetary Space Station at Jupiter, JPL Internal Document 760-157, May 1977.

2-2. Stearns, J.W., Large-Payload Earth-Orbit Transportation With Electric Propulsion, JPL Technical Memorandum 33-793, September 1976.

2-3. General Electric Co., Final Review of Conceptual Design Study for a 200 W/kg Solar Array for Space Applications, Contract No. 954393, Report No. 200 W/kg 1.77-030, January 1977.

2-4. Space Shuttle Requirements Document, NASA-JSC 07700, latest revision.

2-5. Boeing Aerospace Company, Future Space Transportation Systems Analysis Study, Report D180-20242-1, December 1976.

2-6. General Research Corporation, OAST Space Systems Technology Model, private communication, May 1977.

2-7. Interim Upper Stage System Design Review, Boeing Aerospace Company, June 7-10, 1977.

2-8. NASA 5-year Plan, National Aeronautics and Space Administration, Washington, D.C., 1977.

SECTION III

SELECTED MISSION MODEL

The initial task of this study (Figure 2-2), the selection of a useful mission model against which to evaluate technology readiness programs, was an essential and difficult task because of the traditional uncertainties in such models. Studies to provide improved modeling are continuing within NASA (Ref. 2-8).

This Space Power Systems Technology Enablement study takes advantage of ongoing mission studies, but is not considered to be limited by them. Quite obviously, advanced missions studies must make gross assumptions about the availability of technology, while technology studies must simultaneously make gross assumptions about missions. Both types of studies then need continuous iteration to converge to a set of planned NASA programs. Actual programs and schedules will be further dependent on resources availability which, in turn, depends upon establishing an advocacy with the Congress and the public.

We can expect that growth of resources for space will be very slow in the areas for which this Technology Enablement study applies. Large Earth orbital systems of greatest appeal are the subject of separate study. Extensive scientific investigations in space are needed first to lay the foundation for future exploitation programs, but establishing a broad advocacy is very difficult. The overriding problem that has to be solved, therefore, is how to achieve greater scientific capability in space with very little increase of resources.

With the advent of Shuttle, there are many new possibilities for scientific growth within a fixed budget. But it is going to require technological ingenuity to extract the maximum capability from the STS. If we were simply to adapt previous spacecraft technology to the Shuttle we would find that costs are not significantly different (Ref. 3-1). In parallel with establishing a mission model, some preliminary study was made of cost-effective utilization of the STS. Detail is provided in Appendix A. Although planning for the early 1980's is presently becoming quite firm, missions for the mid to late 1980's should be able to take advantage of cost-effective approaches.

To adequately exercise potential technology options in this enablement study, we observed that there were other areas besides the power systems area that require expansion of technology:

(1) Advanced propulsion

(2) On-board intelligence

(3) Long mission lifetime

(4) Modularity and redundancy

(5) Communications (planetary)
  (a) distance
  (b) data rate
  (c) interference

(6) Environment and its control

(7) Systems integration

As we saw potential limitations in these areas, it was necessary to assume that technology in these discipline areas would proceed so as not to preclude any of the recommendations of this study.

Based primarily on literature survey, a maximum listing of automated missions was derived (Figure 3-1) taking into account the above assumptions and unconstrained budget. Most of the missions planned for the early 1980s have technology programs already established. Missions for the mid to late 1980s will allow extension of present technologies, while missions for the 1990s will allow time for major technology revisions. Those advanced technology studies that are proposed for Spacelab are also shown in the mission listing. The X-X indicates the uncertainty in flight date.

A. LUNAR AND PLANETARY MISSIONS

NASA's lunar and planetary programs have been organizationally focused at JPL. Science is under the cognizance of NASA's Office of Space Science, and potential missions to the moon and planets have been cataloged in some detail (References 2-6 and 3-2 through 3-7).

Reconnaissance missions at the moon and inner planets have been completed, with the exception of a probe to the surface of Venus now proposed for the early 1980s. More advanced exploratory missions have thus far been accomplished only at the moon (Surveyor) and Mars (Viking). Thus, considerable exploratory work still remains at Venus and Mercury and the outer planets for the mid to late 1980s. With the exception of the now-delayed Lunar Polar Orbiter to complete the mapping of lunar resources, intensive studies on the moon have been accomplished. No intensive study work, however, has yet been accomplished at the planets. That activity will be accomplished from the mid 1980s through the 1990s. Some utilization experiments also may be expected on the lunar surface during the 1990s.

3-3

(a) MISSIONS (LUNAR AND INNER PLANETS)	LAUNCH DATE	1980	81	82	83	84	85	86	87	88	89	1990	91	92	93	94	95	96	97	98	99	2000
RECONNAISSANCE																						
VENUS PROBE		X		X																		
EXPLORATION																						
MARS PENETRATORS						X					X											
MERCURY ORBITER						X	X															
VENUS ORBITER IMAGING RADAR					X				X													
VENUS BUOYANT STATION							X															
VENUS LARGE LANDER								X			X											
INTENSIVE STUDY																						
LUNAR POLAR ORBITER		X		X																		
MARS ROVER-ORBITER						X		X														
AUTOMATED LUNAR LABORATORY					X			X														
MARS SURFACE SAMPLE RETURN										X					X							
VENUS ROVER-ORBITER												X	X									
MERCURY LANDER/ROVER/ORBITER												X				X						
VENUS SURFACE SAMPLE RETURN															X			X				
MERCURY SURFACE SAMPLE RETURN																X					X	
UTILIZATION																						
AUTOMATED LUNAR BASE													X			X						
LUNAR SOIL PROCESSING EXPERIMENTS															X		X					
LUNAR BASE OPERATIONS (MANNED SUPPORT)																	X			X		

Figure 3-1a. Selected Overall Mission Set: Lunar and Inner Planets

(b) MISSIONS (OUTER PLANETS AND SOLAR ESCAPE)	LAUNCH DATE	1980	81	82	83	84	85	86	87	88	89	1990	91	92	93	94	95	96	97	98	99	2000
RECONNAISSANCE																						
SEP SATURN/URANUS/TITAN PROBE				X	—	—	X															
SEP URANUS/NEPTUNE FLYBY								X														
SEP JUPITER/PLUTO FLYBY										X	X											
SOLAR SYSTEM ESCAPE					X	—	X															
SEP COMET RENDEZVOUS/FLYBY				X	X																	
SEP ASTERIOD FLYBY							X	—	X													
EXPLORATION																						
JUPITER ORBITER/PROBE				X																		
SEP JUPITER SATELLITE TOUR/G. LANDER					X																	
SEP SATURN ORBITER									X													
NEP PLUTO ORBITER/PROBE																				X		
NEP EXTRA-SOLAR MULTI-PROBE																		X	—	—	—	X
SEP/NEP MULTI ASTEROID RENDEZVOUS/SAMPLE RETURN															X							
SEP COMET RENDEZVOUS/SAMPLE RETURN										X	—	X										
INTENSIVE STUDY (NEP) (1)[a]																						
JUPITER AND SATELLITES AND SSR															X	—	—	—	—	—	—	→
SATURN AND SATELLITES AND SSR																	X	—	—	—	—	→
URANUS AND SATELLITES AND SSR																			X	—	—	→
NEPTUNE AND SATELLITES AND SSR																					X	→
UTILIZATION																						
(16) NEP NUCLEAR WASTE DISPOSAL															X	—	—	—	—	—	—	→

[a]NUMBERS REFER TO GENERAL RESEARCH CORP. MODEL

Figure 3-1b. Selected Overall Mission Set: Outer Planets and Solar Escape

(c) MISSIONS (EARTH ORBIT AND SOLAR)	LAUNCH DATE	1979	1980	81	82	83	84	85	86	87	88	89	1990	91	92	93	94	95	96	97	98	99	2000
EXPLORATION																							
SPACE TELESCOPE			X	—	X																		
SOLAR MAXIMUM MISSION*		X																					
EXPLORERS*		←	—	—	—	ALL	YEARS	—	—	—	—	—	→										
GRAVITY PROBE			X			X																	
SEASAT B*					X	X																	
LANDSAT C, D*				X	—	X																	
HIGH ENERGY OBSERVATORY A, B					X	—	—	X															
TIROS O (PROTOTYPE)						X																	
NIMBUS*		X																					
STORMSAT			X	—	—	X																	
TRAFFIC MANAGEMENT (EXPERIMENTAL)		X																					
DISASTER WARNING				X	—	X																	
(EXPERIMENTAL TECHNOLOGY SATS)		←	—	—	—	—	FREQUENT	—	—	—	—	—	→										
SOLAR PROBE							X																
OUT OF ECLIPTIC PROBE						X																	
ODSRS (EXPERIMENTAL)					X																		
RADIO ASTRONOMY (LBI)					X																		
CLIMATE SATELLITE A, B*					X	—	—	X															

*EXPENDABLE LAUNCH VEHICLE

Figure 3-1c. Selected Overall Mission Set: Earth Orbit and Solar

(c) MISSIONS (EARTH ORBIT AND SOLAR)	LAUNCH DATE	1979	1980	81	82	83	84	85	86	87	88	89	1990	91	92	93	94	95	96	97	98	99	2000
INTENSIVE STUDY																							
(5) [a]LARGE RADIO TELESCOPE											X	—	—	—	—	X							
(3) VERY LARGE SPACE TELESCOPE								X	—	—	—	—	X										
LARGE IR TELESCOPE ON SPACELAB						←	—	—	—	ALL YEARS	—	—	—	→									
(2) LARGE SOLAR OBSERVATORY								X		X													
RADIO ASTRONOMY EXPERIMENTS (VLBI)							X																
SPACELAB SYSTEMS (BESS, AMPS, HIGH ENERGY, etc.)				←	—	—	—	—	ALL YEARS	—	—	—	—	→									
EARTH SURVEY SATELLITE											X												
GRAV SAT A, B							X	—	X														
MAGNETOSPHERIC MULTIPROBES								X		X													
(4) ATMOSPHERIC PHYSICS LABORATORY							X	—	—	—	—	—	—	X									
(7) SEASAT C								X															
ADVANCED OUT OF ECLIPTIC SPACECRAFT									X														
ADVANCED TECHNOLOGY STUDIES ON SPACELAB							X	—	—	—	—	X											
SPACE PROCESSING																							
DATA PROCESSING																							
DISASTER AVOIDANCE CONCEPTS																							
IR CONTAMINATION EXPERIMENTS																							
CRYOGENIC IR TELESCOPE FACILITY										X													
LONG-TERM SOLAR MONITORING S/C									←	—	—	—	—	—	SEVERAL	—	—	—	→				
ADVANCED ODSRS								X															
(1, 22) NEP CARGO ORBIT TRANSFER VEHICLE (WITH LOGISTICS DEPOT)													X										

a. NUMBERS REFER TO GENERAL RESEARCH CORP. MODEL

Figure 3-1c. (Continuation 1)

(c) MISSIONS (EARTH ORBIT AND SOLAR)	LAUNCH DATE	1979	1980	81	82	83	84	85	86	87	88	89	1990	91	92	93	94	95	96	97	98	99	2000
UTILIZATION																							
EARTH RESOURCES SATELLITES					X	—	—	—	—	—	—	—	—	—	→								
(8, 9, 12)[a] ADVANCED EARTH RESOURCES SYSTEMS									X	—	—	—	—	—	—	—	—	—	→				
COMMUNICATIONS SATELLITES		—	—	—	—	—	—	—	—	—	—	—	—	→									
LARGE COMMUNICATIONS SATELLITES								X	—	—	—	—	—	—	—	—	—	—	—	—	→		
(10) PUBLIC SERVICE COMMUNICATIONS SATELLITES											X	—	X	—	—	—	—	—	→				
(EMERGENCY, EDUCATION, TELEPHONE, etc.)																							
AIR TRAFFIC MANAGEMENT SYSTEM			X	—	X	—	—	—	—	—	—	—	—	—	→								
(11) NAVIGATION/COLLISION AVOIDANCE															X	—	—	—	—	—	—	—	→
POLLUTION/DISEASE CONTROL									X	—	—	—	—	—	—	—	—	—	→				
(13) DISASTER WARNING SYSTEMS					X	—	—	—	—	—	—	—	—	—	→								
METEOROLOGICAL SATELLITES		—	—	—	—	—	—	—	—	—	—	—	—	—	→								
(6) STORM INTELLIGENCE NETWORK											X	—	—	—	—	—	—	—	—	—	—	—	→
CLIMATE PREDICTION/MODIFICATION										X	—	X	—	—	—	—	—	—	—	—	—	—	→
(15) LARGE SPACE BASE OPERATIONS													X	—	—	—	—	—	—	—	—	—	→
STABLE CLOCK GYRO IN SOLAR ORBIT														X									
(14) SPACE POWER AND RELAY SYSTEM									X	—	—	—	—	—	—	—	—	—	—	—	—	—	→
(17) TELEOPERATOR VEHICLE SYSTEM									X	—	—	—	—	—	—	—	—	—	—	—	—	—	→
COAST SATELLITE								X	—	—	X												
ICE SATELLITE (SEP)								X	—	—	X												

(a) NUMBERS REFER TO GENERAL RESEARCH CORP. MODEL

Figure 3-1c. (Continuation 2)

The mission model selected for this technology enablement study has assumed a full set of exploratory and intensive study missions at each planet of the Solar System. This includes orbiters, landers, atmospheric probes and surface penetrators, rovers and sample return vehicles. At the inner planets we generally expected to accomplish each of these elements individually. However, intensive study at the outer planets is currently expected to be accomplished by an automated planetary space station carrying multiple payloads (Ref. 2-1). Free fliers within the atmospheres of any of the outer planets were not addressed because of lack of feasibility of propulsion for those missions. Throughout the selected model, maximum inheritance of technology is anticipated in order to obtain maximum amortization of new technology.

B. EARTH ORBIT AND SOLAR MISSIONS

Several NASA mission models had to be studied in arriving at the desired Earth orbit and solar mission model for this study because mission focus was not as well established as for the planetary missions. Principal use was made of the Outlook For Space (Reference 3-6), A Forecast of Space Technology (Reference 3-7), the Shuttle Payload Data Analysis (SPDA) at NASA-MSFC and Convair (Reference 3-8), and the 1973 "official" NASA Payload Model and its subsequent revisions (Reference 3-2). The Technology Payload Model (TPM) at NASA-JSC (Reference 3-9) and the Composite NASA Payload Model by Convair (Reference 3-10) were also used. There is also the Outside User Payload Model at Battelle Memorial Institute (Reference 3-11) covering non-NASA, non-DOD missions. The DOD missions are generally classified and are not specifically included in our modeling effort. However, coordination with DOD's Aerospace Corporation was maintained. Results of our modeling were then coordinated with a more detailed "Technology Model" by General Research Corporation in McLean, VA (Reference 2-3).

Earth orbit and solar reconnaissance missions have long since been completed. Exploration missions will essentially be completed in the early 1980s, with intensive study missions continuing at least to the early 1990s. Utilization programs have already started and should continue to build at a rapid pace until at least the mid 1990s. A cursory look at the utilization programs (other than industrialization) indicates technology needs are probably already covered by the scientific missions as defined above except, perhaps, additional emphasis on low cost. Low cost will therefore be considered as a special part of the benefit/cost/risk analysis. Study is made of the three-way tradeoff among the elements of performance, lifetime/reliability, and cost.

In Figure 3-1c, exploratory missions of the early 1980s will be flights that utilize near-term technology. Many of them are on expendable launch vehicles rather than Shuttle because of the early launch date or because of the high cost of "transition" hardware with Shuttle launch. By the mid 1980s, however, more advanced, large systems with significant automation are an anticipated requirement for intensive study missions of Figure 3-1c. These systems will make more effective use of Shuttle and provide a multiplicity of functions. As technologies become available, intensive study missions will be carried forward into utilization missions where appropriate.

The various systems may be launched into low Earth orbits, sun-synchronous or geosynchronous orbits. They may be equatorial, polar, or of intermediate inclination (Reference 3-12). Some of these orbits may be quite energy consuming, requiring upper stage and spacecraft propulsion. For the mid-1980s and 1990s, an electric orbit transfer vehicle is needed that will minimize transport cost and deliver larger payload than with chemical propulsion.

Although the missions of interest are the automated spacecraft, Spacelab advanced technology studies also were included because the technologies may also apply to automated spacecraft. There is no attempt in this study to impact the Spacelab power subsystems.

REFERENCES

3-1. Space Shuttle "Shuttlizing" Payloads, Engineering 805.13, UCLA Extension Short Course, October 17-21, 1977.

3-2. Interim NASA Payload Model for Planning Purposes, March 1976.

3-3. Atkins, K. L., et al., Mission Summaries prepared for 1976 Summer Meeting of Committee on Planetary and Lunar Exploration, JPL Internal Document No. 662-25, September 1976.

3-4. Saturn/Uranus Atmospheric Entry Probe Mission Spacecraft System Definition Study, Final Report, TRW Systems Group, Contract No. NAS2-7297, July 1973.

3-5. Report on Space Science 1975, National Academy of Sciences, Washington, D.C., 1976.

3-6. Outlook for Space, NASA SP-386, January 1976.

3-7. A Forecast of Space Technology 1980-2000, NASA SP-387, January 1976.

3-8. Beranek, R., Payload Descriptions, Levels A and B Data, NASA/MSFC Program Development, Payload Studies Office, July 1975 (preliminary).

3-9. Alvarez, J. C., et al., Development of the Technology Payload Model, NASA/JSC 11310, May 1976.

3-10. Peterson, J., Composite NASA Payload Model, 1980-1991, General Dynamics Convair Division, October 1975.

3-11. Buckeye, D. L., Outside User Payload Model, BMI Memorandum No. BMI-NLVP-IM-76-13, October 1976.

3-12. Nagler, R. G., "Surveying the Earth's Environment from Space: Spectral, Areal, Temporal Coverage Trends," Paper 77-1585, AIAA Conference on Satellite Applications to Marine Technology, Nov. 15, 1977, New Orleans, La.

SECTION IV

TECHNOLOGY READINESS PROGRAMS

A literature search was conducted on the present technology programs in space power (Reference 4-1) in the areas of performance listed in Table 4-1. Much of this technology is mature, being applied to spacecraft of the present or near-future time frame. Additional space power R & T is being conducted in technology readiness programs, that is, programs having flight readiness dates scheduled. The remainder of the space power R & T is identified as R & T Base programs. The most general definition of R & T Base (Reference 4-2) is "any activity providing an end product which does not involve aircraft or spacecraft demonstrations of components or systems." More specifically, the R & T Base usually refers to technology programs that stop short of flight readiness. It is out of this R & T Base that we expect to establish the enabling technologies of the future, and they will be dealt with in the next section.

Present space power technology readiness programs captured by literature search (Reference 4-1) are shown in Table 4-2. Technology ready dates emphasize the planned missions of the early 1980s. Undoubtedly, follow-on work will be needed in many of these areas for later missions, but such work is expected generally to be categorized as enhancing rather than enabling technology.

It was not the intent of this technology enablement study to evaluate the technology readiness programs; resources did not permit it. These technologies, nevertheless, do represent a significant input to our study in that the majority of NASA missions through 1985 are covered by them. We have not questioned or documented the adequacy of these programs, but we assume that there is sufficient dialogue between NASA headquarters and the cognizant NASA centers to assure the timely achievement of goals. Future enhancement of the technologies might be considered with respect to cost-effective use of STS. Potential capabilities for relaxation of mass constraints and repackaging might produce very significant cost savings for missions beyond the early 1980's.

In addition to programs on a component and subsystem scale, technology readiness programs exist on a vehicle system scale. Much of the system scale activity is funded outside of OAST. A survey of such programs has produced the listing in Table 4-3.

Of particular interest is the multimission modular spacecraft (MMS) being developed by NASA/GSFC (Reference 4-3.). This is a standardized, modularized system that is expected to provide up to 2 kWe of Earth orbital spacecraft power. It is expected to be a versatile system with some elements of automated power management capability. Preliminary studies are now considering repackaging for Shuttle rather than for an expendable launch vehicle for which it is presently configured. Missions that are expected to be covered by MMS are shown in Table 4-4.

Table 4-1. Technology Evaluation Areas

(1) Solar power systems

- (a) Solar-thermal
- (b) Photovoltaic
- (c) Integrated power processing

(2) Nuclear power systems

- (a) Radioisotope
- (b) Reactor

(3) Thermal-to-electric conversion

- (a) Brayton cycle
- (b) Rankine cycle
- (c) Stirling cycle
- (d) Thermoelectric
- (e) Thermionic

(4) Energy storage

- (a) Batteries
- (b) Flywheels
- (c) Fuel cells
- (d) Thermal

(5) Modular power processing

(6) Power components

- (a) Solid state switching
- (b) DC-DC converters
- (c) Transformers/Inverters
- (d) Main bus power regulators
- (e) Overload protection
- (f) High-voltage power diodes
- (g) High-voltage power thyristors

(7) System Management

- (a) Power system modeling
- (b) Automated power management
- (c) Standardization technology

(8) Environment and reliability

- (a) Long-life technology
- (b) High-temperature technology
- (c) Radiation hardening technology
- (d) Heat pipe technology

Table 4-2. Present Space Power Technology Readiness Programs

Technology	Ready Date
Power Conversion	
Lightweight photovoltaics	1980
Selenide RTG (SIG)	1980
BIPS/KIPS isotope dynamics	1981
Thermionics	1990
Solar thermoelectric generator (STG)	1982
Energy Storage	
Batteries (NiCd, NiH, AgH)	1980
Fuel cells	1978
Lithium batteries	1982
Power processing and management	
Modular regulators	1978
Pulsed battery chargers	1978
Solid State switching and protection	1982
Multi-phase inverters	1978
Special-purpose transformers	1977
Micrologic/processor	1979
Automated power systems management	1981

Table 4-3. Ongoing Spacecraft Systems Technology Readiness Programs

(1) Ongoing Programs

- (a) Jupiter Orbiter/Probe (JPL/ARC)
- (b) Venus Orbiter Imaging Radar (JPL)
- (c) Multimission Modular Spacecraft (GSFC)
- (d) Ion Drive (JPL/LeRC)
- (e) Mars Penetrator (ARC)
- (f) Atmospheric Entry Probes (ARC)

(2) Planned Programs

- (a) Mars Surface Sample Return (JPL)

Table 4-4. Space Missions Supported by MMS Technology (Reference 3-2)

Space Science

(1) Explorers

(a) GRE
(b) IUE
(c) IXE

(2) Solar maximum (SMM)

(3) X-ray and gamma ray flare

(4) HEAO-BLOCK II

(5) Soft X-ray telescope

(6) Radio LBI

(7) Relativity

Applications

(1) EOS A-D thematic mapper (TM) and S-band MSS or Landsat

(2) Geopause

(3) Stormsat

(4) Environmental monitoring

(5) Technology demonstration satellite (TDS)

(6) Seasat-B

(7) SEOS (Synchronous EOS)

(8) LEST (1-1/2 m dia)

(9) Tiros-0

Life Science

(1) Biomedical experiment scientific satellite (BESS).

The "Ion Drive", or solar electric propulsion, vehicle system is the principal program that utilizes the lightweight solar array technology of Table 4-2 (Reference 4-4). Work was initially aimed at a 1982-1983 launch toward Comet Halley. More recent studies now include several comet and asteroid reconnaissance missions and an exploratory orbiter at Saturn.

The Mars penetrator and atmospheric entry probes (Jupiter and Uranus) are small, short- to medium-lived systems. They are being studied by Ames Research Center for withstanding difficult environmental conditions (Reference 4-5 and 4-6).

When the above technologies are compared against the power requirements for the missions of Figure 3-1, a large number of missions are found to be covered. This is shown in Table 4-5. The missions here have been summarized in the standard matrix defined by Figure 2-3. The time span for accomplishing each category of mission is also indicated above each matrix element. The missions for which technology or technology readiness programs exist are shown in the cross-hatched boxes in the table. The missions not eliminated require additional "enabling" technology, and are therefore considered as technology drivers. The Earth-orbit and solar utilization missions are, by study definition, eliminated from consideration and to be considered in a separate study by NASA.

Table 4-5. Expected Missions for STS (Automated)

Mission	Reconnaissance	Exploration	Intensive Study	Utilization
Lunar and inner planets	1980-1982 ~~Venus~~	1984-1990 Mercury orbit ~~Mars penetrator~~ Venus orbit/ lander	1985-2000 ~~Lunar solar orbit~~ Lunar Laboratory ~~Mars Rover and SSR~~ Mercury Rover and SSR Venus Rover and SSR	1992-2000 Lunar base[a]
Outer planets and solar escape	1982-1988 Saturn/Uranus Jupiter/Neptune ~~Solar escape~~ ~~Comets and Asteroids~~	1982-1998 ~~Jupiter orbit~~ ~~Saturn orbit~~ Pluto orbit Asteroid SSR Solar escape Comet SR	1993-1998 Orbit, landers, Rovers, SSR at Jupiter, Saturn, Uranus, and Neptune	1993-2000 Nuclear waste disposal
Earth orbit and solar		1979-1985 ~~Space telescope~~ ~~Solar max.~~ Solar probe Land/sea observations ~~High energy observations~~ ~~Climate/weather~~ ~~Communication~~ ~~Radio astronomy~~ ~~Extra ecliptic~~	1984-1993 Large telescope ~~Solar observatory~~ Spacelab[a] Earth survey ~~Atmospheric physics~~ ~~Gravity/magnetics~~ Large antennas ~~Power transmission~~	1980-2000[b] ~~Earth resources~~ ~~Earth observations~~ ~~Communications~~ ~~Navigation~~ ~~Climate/weather~~ ~~Space power~~ ~~Industrialization~~ ~~Colonization~~

[hatched box] denotes mission covered by present programs

[a]Automated technology with manned systems.

[b]Separate studies being accomplished.

REFERENCES

4-1. Sheldon, E.L., and C.K. Sterkin, NASA/RECON Computer Search, MR6927P through MR6955P, May-June, 1977, internal JPL document.

4-2. Sadin, S.R., "OAST Space Theme Workshop", NASA TMX-3486, April 1977.

4-3. Execution Phase Project Plan for Multimission Modular Spacecraft (MMS), Revision 2, NASA/GFSC, July 1976.

4-4. 200 Watts/Kilogram Solar Array Conceptual Approach Study, Phase II, Assessment Report for Proof-of-Concept Experiments and Halley's Comet Concentrator Array, General Electric Space Division, Valley Forge Space Center (performed under Contract 954393 for the Jet Propulsion Laboratory, Pasadena, Calif.), July 8, 1977.

4-5. Mars Surface Penetrator System Description, prepared for Mars Science Working Group (MSWG) 1984, NASA/ARC May 1977 (preliminary).

4-6. Saturn/Uranus Atmospheric Entry Probe, Final Report, McDonnell Douglas Corporation Report #MDCE0870, July 1973.

SECTION V

DRIVER MISSIONS AND IDENTIFICATION OF TECHNOLOGY CANDIDATES

Those missions estimated to be drivers for the technology were identified in the previous section of the report and were indicated in Table 4-5. This section treats the work that was accomplished in the identification of the enabling technology candidates to meet the needs of the driver missions within the time frame indicated.

For each driver mission, mission descriptions were prepared, and are contained in Appendix B of this report. Spacecraft power requirements were estimated and the specific features of the missions that require new technology were identified. A search of the R&T Base programs of NASA and DoD was then made (References 4-2, 5-1), and a list of the maximum number of technology candidates was made. More detail was obtained on each candidate where possible, and a limited synthesis of power subsystem concepts was made. A further evaluation was made of the effort being expended in each technology area, estimating the progress expected as a function of time if the R&T program were left unperturbed. Where progress appeared adequate, this was so documented and the candidate was then treated as if it were a technology readiness effort. The remaining candidates, whose unperturbed programs fell short of mission schedule needs, were then indicated as requiring program augmentation. This final set of technology candidates was carried forward for further analysis and is reported in the next section of the report. Technology descriptions are provided in Appendix C.

Table 5-1 is a summary chart showing (1) the driver missions, (2) the features of the mission that are technology drivers, (3) a list of power technology candidates for meeting the mission needs, (4) whether those candidates are technology readiness type (also whether direct power application or tangential need) or R&T Base type, and (5) the estimated electric power level required by the mission. The driver missions, their features and estimated power level are discussed below, followed by discussion of the identified technology candidates.

A. DRIVER MISSIONS

The high-energy trajectory requirements for Mercury and the outer planets dictate electric propulsion for heliocentric transfer for all except reconnaissance missions (Reference 3-2). According to our estimates, solar electric propulsion will be used beginning in the 1980's, and nuclear electric propulsion will be available in the 1990's.

For lunar and inner planetary missions, Mars appears to be the only body for which technology readiness programs are now approaching adequacy for intensive study. At Mercury, its close proximity to the Sun introduces problems with high temperature direct insolation. Mercury's elliptical orbit also brings it within the solar corona periodically. Orbiters, landers, rovers, etc. at Mercury are expected to require power levels of

Table 5-1a. Potential Driver Missions and Technology Candidates: Lunar and Inner Planet Missions

MISSION (LUNAR AND INNER PLANETS)	FEATURE DRIVING THE TECHNOLOGY	POWER TECHNOLOGY CANDIDATE	CANDIDATE TYPE (SEE NOTE)	POWER LEVEL (kWe)
MERCURY	HIGH TEMPERATURE,	HIGH TEMPERATURE COMPONENTS	2	1-5
ORBITER (SEP)	SOLAR CORONA	ELECTROMAGNETIC SHIELDING,		
LANDER/ROVER/SSR		LASER POWER BEAMING,	1 - a	
		RADIOISOTOPE THERMIONICS,	1 - a	
		ADVANCED RTG,	1 - a	
		GaAs SOLAR CELLS	1 - a	
VENUS	HIGH TEMPERATURE,	RADIOISOTOPE THERMIONICS,	1 - a	2-5
LANDER/ROVER/SSR	HIGH PRESSURE,	ADVANCED RTG,	1 - a	
BUOYANT STATION	CO_2 ATMOSPHERE	STIRLING REFRIGERATION,	1 - b	
		FUSED SALT BATTERIES	2	
AUTOMATED LUNAR BASE	LONG LUNAR NIGHT,	NUCLEAR-ELECTRIC POWER,	"	~500
	POWER NETWORK	LASER POWER BEAMING,	1 - a	
		GaAs SOLAR CELLS	1 - a	
		FLYWHEEL ENERGY STORAGE	2	

NOTE: CANDIDATE TYPE — DEFINITION

CANDIDATE TYPE	DEFINITION
1 - a	TECHNOLOGY READINESS WITH DIRECT POWER APPLICATION
1 - b	TECHNOLOGY READINESS WITH TANGENTIAL APPLICATION
2	R AND T BASE
	NO PROGRAM AUGMENTATION NEEDED

Table 5-1b. Potential Driver Missions and Technology Candidates: Outer Planet and Solar Escape Missions

MISSION (OUTER PLANETS AND SOLAR ESCAPE)	FEATURE DRIVING THE TECHNOLOGY	POWER TECHNOLOGY CANDIDATE	CANDIDATE TYPE (SEE NOTE)	POWER LEVEL (kWe)
URANUS/NEPTUNE/ SOLAR ESCAPE (SEP)	LONG RADIO DISTANCE, LONG FLIGHT TIME	RADIOISOTOPE THERMIONICS, ADVANCED RTG	1 - a 1 - a	~0.5
PLUTO ORBITER/PROBE/ SOLAR ESCAPE (NEP)	NEP INTERFACE, LONG RADIO DISTANCE	NUCLEAR-ELECTRIC POWER, RADIOISOTOPE THERMIONICS, ADVANCED RTG, INDUCTIVE ENERGY STORAGE	* 1 - a 1 - a *	1-50
NUCLEAR WASTE DISPOSAL (NEP)	EARTH ORBIT LOGISTICS, NUCLEAR SAFETY, RENDEZVOUS AND DOCKING	NUCLEAR-ELECTRIC POWER, RADIATION HARD COMPONENTS, INDUCTIVE ENERGY STORAGE	* 2 *	1-20
AUTOMATED PLANETARY STATION ORBITAL MAPPERS, PROCESSORS, BEAMING, PROBES, PENETRATORS, LANDERS, ROVERS, SSR	EXTREME ENVIRONMENTS, JUPITER RADIATION, SATURN/URANUS RINGS, NEPTUNE REMOTENESS, RENDEZVOUS AND DOCKING	NUCLEAR-ELECTRIC POWER, RADIATION HARD COMPONENTS, LASER POWER BEAMING, RADIOISOTOPE THERMIONICS, ADVANCED RTG, INDUCTIVE ENERGY STORAGE	* 2 1 - a 1 - a 1 - a *	10-400

NOTE:

CANDIDATE TYPE	DEFINITION
1 - a	TECHNOLOGY READINESS WITH DIRECT POWER APPLICATION
1 - b	TECHNOLOGY READINESS WITH TANGENTIAL APPLICATION
2	R AND T BASE
*	NO PROGRAM AUGMENTATION NEEDED

Table 5-1c. Potential Driver Missions and Technology Candidates: Earth Orbit and Solar Missions

MISSION (EARTH ORBIT AND SOLAR)	FEATURE DRIVING THE TECHNOLOGY	POWER TECHNOLOGY CANDIDATE	CANDIDATE TYPE (SEE NOTE)	POWER LEVEL (kWe)
SOLAR PROBE/ ORBITER (0 01 AU)	HIGH SOLAR IRRADIANCE (14 MWt/m^2), SOLAR CORONA	HIGH TEMPERATURE COMPONENTS SYSTEM WITH HEAT SHIELD, HEAT SHIELD/HEAT PIPES, ELECTROMAGNETIC SHIELDING, STIRLING REFRIGERATION	2 1 - a 1 - b * 1 - b	0.5-50
EARTH OBSERVATION AND RADIO ASTRONOMY	CRYOGENIC COOLING OF SENSORS (10 YRS, 2 W, 5^0K)	STIRLING REFRIGERATION, GaAs SOLAR CELLS	1 - b 1 - a	1-10
ELECTRIC ORBIT TRANSFER VEHICLE	PROTON BELT DWELL (MULTI-ROUND TRIP)	GaAs, SOLAR CELLS RADIATION HARD COMPONENTS, ELECTROMAGNETIC SHIELDING, FLYWHEEL ENERGY STORAGE, INDUCTIVE ENERGY STORAGE	1 - a 2 * 2 *	100-1,000
SPACELAB PROCESSING	HIGH TEMPERATURE FURNACE (>2800^0K, 6 kWt)	SOLAR FURNACE (~15 kWt), THERMIONIC BOTTOMING	1 - b *	REPLACES 50 kWe

NOTE,

CANDIDATE TYPE	DEFINITION
1 - a	TECHNOLOGY READINESS WITH DIRECT POWER APPLICATION
1 - b	TECHNOLOGY READINESS WITH TANGENTIAL APPLICATION
2	R AND T BASE
*	NO PROGRAM AUGMENTATION NEEDED

1-5 kWe, particularly with the needs for thermal conditioning and EMI control. At Venus the CO_2 atmosphere introduces a high temperature, high pressure environment. Cooling requirements of Venus landers, rovers, and sample return vehicles will boost power needs into the 2-5 kWe range. At the moon, as lunar base activities begin they should be initially automated and later manned. Large power distribution networks will be built up for lunar operations, with nuclear power (500 kWe) for the long lunar nights. Sometime in the future we can also expect a thorium/U-233 breeder reactor on the moon, utilizing the thorium available there.

Outer planet missions are expected to require long-lifetime (10 to 15 years), autonomous, nuclear-powered orbital laboratory operations for the 1990's. Probes, landers, rovers, etc., also will be carried. Another mission would provide for sample return to Earth orbit. Earlier, less-demanding missions would be accomplished with solar electric propulsion. Jupiter radiation environment, Saturn and Uranus rings, and Neptune remoteness appear to be major technology drivers. Scientific probes and nuclear waste disposal missions are also expected, propelled to solar system escape. If high data rate is required at the extremes of the solar system, large power for communication will be available from the nuclear-electric powerplant. The Ion Drive vehicle also needs a radioisotope supply at approximately 0.5 kWe. The NEP vehicles have large nuclear power on board at levels up to 400 kWe.

The Earth orbit and solar missions that are expected to be drivers of the technology also have some very significant needs. The solar probe, which is to fly to within 0.01 AU of the sun, has to face problems of high temperature and of solar corona interactions. A further complication will be communication with Earth from a background temperature (worst case) of 15,000°K. For a small-dish (Viking) communication rate at one kilobit/second, 3 kW of RF transmitted power is needed.

Earth observation and radio astronomy spacecraft carry equipment that requires significant cryogenic cooling. Present concepts propose electric power to energize these coolers. Study might be made to determine if thermal power alternatives can be made available. One requirement of the Spacelab, the high-temperature furnace for space processing, also has problems with electric power energizing that exceeds Shuttle power system and cooling system capabilities (Reference 5-2). Technology for a solar furnace should probably be considered for this application, since the end requirement is already thermal.

The electric orbit transfer vehicle (OTV), although not an actual mission, provides mission support. Because it will operate for considerable periods of time in the proton belts of Earth, radiation-resistant equipment will be needed, and shielding of payloads is essential.

Based on our study guidelines, Earth orbital utilization missions of Figure 4-1c are not entered as technology drivers in this study. Their principal enablement requirement is large, low-cost power for industrial capabilities and communications. It was felt that, by including these missions separately, a better focus on low-power technology could be made in this study.

B. ENABLING TECHNOLOGY CANDIDATES

In support of the driver missions needs of Table 5-1, power technology candidates have been identified and will now be discussed. As noted, there are two types of technology readiness program candidates: those that have direct application to electric power and those that relate tangentially to space power technology. The R&T Base program candidate is another candidate type. Another special candidate type, for which no program augmentation appear presently needed, is denoted by an asterisk. This candidate is already in the R&T Base but should not be overlooked for consideration for future technology readiness programs. It is not necessary in our study to carry this last type beyond this point.

Technology Readiness and R&T Base power systems candidates fall roughly into three basic categories, as shown in Table 5-2: high temperature subsystems, radiation resistant subsystems, and energy storage. An additional item, fiber optics cabling, was found to be important in ground support equipment, and is included in our listing for completeness. It was also noted that there are elements of research that are needed now in the space power R&T Base for potential missions beyond the year 2000. These "reseach horizons" are discussed in Appendix E.

1. High Temperature Subsystems

For spacecraft that travel in toward the sun, temperature problems increase. And even in Earth orbit, spacecraft component heating and cooling requirements (such as for propellant, infrared sensors and masers) may call for unique thermal design. In general, high temperature components (item 1.a) may be of direct value to within approximately 0.2-0.3 AU of the sun. Inside this distance, heat shields are necessary. High temperature component technology appears presently to be quite limited, particularly in the control and power processing area. Major power systems improvements would be possible if heat rejection temperatures could be increasd by as little as 100^{o}K. R&T Base efforts focused in this direction are recommended.

The radioisotope thermionic generator (RTIG) technology and advanced RTG development (item 1.b,c) will provide extended-life landed missions (6 months or more) at Venus. Lightweight, high-temperature capability appears feasible with present breakthroughs in high-temperature ceramics and refractory-metal heat pipes. Concentrator heat pipes with, perhaps, cermet fuel (plutonium) is proposed (Reference 5-3). High temperature heat rejection (~800K) and argon pressurization will make the RTIG compatible with Venus surface environment. A preliminary comparison of the RTIG with the selenide thermoelectric systems for general space applications also looks extremely good. The technology is therefore considered to be a strong candidate for early development, perhaps in a joint effort by NASA and the Dept. of Energy. Modular design would allow power levels from approximately 100 We to 10 kWe.

Table 5-2. Space Power Technology Candidates

Candidate	Type
1. High Temperature Subsystems	
(a) High temperature components	2
(b) Radioisotope thermionic generators (RTIG)	1-a
(c) Advanced RTG's	1-a
(d) Systems with heat shields	1-a
(e) Indigenous material batteries (fused salt)	2
(f) Heat shields/heat pipe radiators	2
(g) Stirling cycle cryogenic coolers	1-6
(h) Solar furnace (15 kWt replaces 50 kWe source)	1-6
2. Radiation Resistant Subsystems	
(a) Gallium arsenide photovoltaics	1-a
(b) Radiation hardened components	2
(c) Electromagnetic shielding	*
(d) Nuclear-electric/thermionics	*
(e) Laser power beaming	1-a
3. Energy Storage	
(a) Flywheels	2
(b) Inductive storage for pulsed power	*
4. Ground Support Equipment	
(a) Fiber optics cabling	1-b

Note:

Candidate Type	Definition
1-a	technology readiness with direct power application
1-b	technology readiness with tangential application R&T Base
2	R&T Base
*	no program augmentation needed

For operation within 0.01 AU, steady-state operation appears possible with a polished tungsten heat shield in combination with a tungsten/lithium heat pipe radiator (item 1.f). Solar insolation is approximately 15 MWt/m^2, and the tungsten system would operate at 2100-2200°K. Waste heat will be used to operate thermionic and/or thermoelectric power converters to generate electrical power for the mission. Spacecraft power level is expected to be quite high for this mission (up to 50 kWe) for reasons of thermal control of the spacecraft, communication power, and electromagnetic shielding within the solar corona. A great deal of systems integration work is anticipated. Large power is available from the heat shield/heat pipe subsystem. The only power system alternative for this mission is the RTG or RTIG at power levels up to perhaps 10 kWe, and they would severely restrict the mission.

Fused salt batteries (item 1.e) that would utilize the indigenous CO_2 atmosphere of Venus are in very early research (Reference 5-4). Careful evaluation is still underway and the timescale for development is not firm. State-of-the-art level 4 by 1984 under an R&T base program seems desirable.

Refrigeration systems are needed for cryogenic cooling of active microwave masers and far-infrared sensors (item 1.g). Large Earth-orbital spacecraft of the future may require several kilowatts of thermal or electrical power for cryogenic pumping, while planetary systems may have smaller requirements. Detailing of requirements has been reported by Aerospace Corporation (Reference 5-1). The free-piston Stirling engine is now coming into focus for the multikilowatt requirements, while the Vuilleumier (VM) refrigerator would be desirable at sub-kilowatt power levels. A Stirling refrigerator will also be needed for spacecraft cooling on the solar probe and for the Venus surface vehicles.

The solar furnace, (item 1.h), of interest to future space manufacturing, is included here because of its potential for thermionic power conversion bottoming. Mirror technology was developed in 1970 for solar-thermionic studies and is applicable here (Reference 5-5). Spacelab experiments have been proposed for high-temperature (>2800 K) space processing (Reference 5-2) of ceramics and glasses that require power levels up to 50 kWe. Maximum Spacelab power is 14 kWe, both from the standpoint of power input and Spacelab cooling capacity. Induction coil and laser heating sources, at 10 to 20% efficiency, require 40 to 50 kWe to develop 6 kWt of heating power. Developing a solar furnace at perhaps 40 to 50% efficiency (concentration ratio 6000) probably would not exceed Spacelab cooling capacity and would also unload the electrical power supply. Another possibility is to use the waste heat from the furnace to produce 1 to 2 kWe of electric power by means of small thermionic power converters appropriately located.

2. Radiation-Resistant Subsystems

Several driver missions potentially require operation in high-energy particle environments. Most effective use of electric propulsion, for instance, requires Earth-orbit spiral through the proton belts, while operations at the outer planets (especially Jupiter) also have very large radiation belts. Solar winds and near-sun operations within the solar corona produce significant problems. Nuclear power sources also generate high energy particles, and even the galactic radiation is significant for very long spacecraft lifetimes. The options are:

(1) to select hardware to operate in the environment, (2) to shield the equipment from the environment, or (3) provide some combination of both.

Photovoltaic components and, to a lesser extent, semiconductors are very susceptible to radiation damage. It is possible that silicon cells on photovoltaic solar arrays could be replaced by less-sensitive gallium-arsenide devices (items 2.a). Such systems require major development efforts to obtain high performance. GaAs thin film photovoltaics are moving toward advanced development programs in FY 1978. Electron radiation hardening has already been demonstrated (Reference 1-2). For a proton radiation environment, the thin-film GaAs cell does not have the junction problem of AlGaAs and should therefore be more radiation resistant. This must be traded against lower beginning-of-life efficiency.

Hardened components (items 2.b) are a primary effort with DOD, (Reference 5-6) although a small effort has been in progress at JPL because of the large radiation field of Jupiter. Availability of better radiation hardened components would be a definite asset in spacecraft power design.

Massive shielding is not a desirable option, except where nuclear power supplies are involved. Electromagnetic shielding, however, (item 2.c) becomes feasible where high current levels are available. (Electrostatic shielding is not attractive for several reasons: it potentially interferes with spacecraft charge control and with scientific instrumentation, and it introduces very high power losses because of unavoidable leakage currents.) Electromagnetic fields can apparently be quite easily established around a spacecraft to provide protection against charged particle radiation. Preliminary estimates indicate current levels on the order of 100A can be effective. Since the basic requirement is only a system integration effort, no R&T program has been identified at this time.

Nuclear-electric powerplants (item 2.d) may also be considered alternatively to photovoltaic supplies because they are already radiation hardened. Radiation shielding is provided between the power-plant and the rest of the spacecraft. Either static power conversion (thermionics) or dynamic systems may be utilized, although thermionics presently appear to offer high performance. Use of nuclear-electric power for propulsion is not a part of this study, but there are also other system implications for the nuclear-electric supplies. Outer-planet intensive study missions of Figure 3-1b propose arrival at the destination planet with a 400- to 500-kWe powerplant in addition to 5 to 10 metric tons of net spacecraft (Reference 2-1). Therefore, comparative investigations must be made of large power options such as high-resolution radar mapping, laser power beaming, major on-board data processing and storage, wide-bandwidth science measurements (multi-spectral equipment), widebandwidth communication to Earth, large-power robotics, in situ materials processing, etc. It is not within the scope of this study to treat these large-power options.

Laser power beaming (item 2.3) has significant possibilities for outer planet exploration with nuclear power plants. The laser beam can be utilized with a photovoltaic collector for electric power, it can be used for surface illumination, and it can be used for spacecraft guidance and navigation.

A copper-halide electrodynamic laser development (Ref. 5-7) in the visible spectrum is presently being planned with a 300 W beam power. A scale up to a beam power of 5-10 kW at approximately 2% operating efficiency is possible with a 300-500 kWe power input. Radiator temperature of approximately 900 K is well matched to the thermionic heat rejection technology now developing.

At the receiver of the laser beam, gallium-aluminum-arsenide photovoltaics can now be tailored to specific laser wavelengths, producing power conversion efficiency of 40% (Ref. 5-8). An orbiting spacecraft with a laser beam can thus produce 2-3 kWe at a landed vehicle on the surface of an outer plant satellite. Preliminary estimates show an order of magnitude cost improvement over radioisotope power supplies, although the impact of beam position control has not been totally considered.

Metal vapor lasers have their power concentrated at just a few calibrated wavelengths. Use of video cameras designed for these wavelengths can also make use of the illumination for visible mapping that would otherwise be impossible.

3. Energy Storage

In the technology readiness programs, covered in Section IV of this report, the lithium battery technology is expected to be flight ready by 1982. Its potential for producing a secondary battery at 220 W-h/kg goes far toward answering the energy storage needs of spacecraft. It has, however, a rather low limit of charge/discharge cycles ($\sim 10^3$). Low orbits and polar orbits at Earth require a much greater number of cycles. Flywheel energy storage therefore may meet the need for long life with a large number of duty cycles (item 3.a). Kevlar and quartz fibers for rotor materials need development if high performance is to be achieved. System problems are also quite severe at this time. Further study should also be given to combine flywheel energy storage with attitude control (Reference 5-9). At zero gravity and the vacuum of space, energy densities of 200-300 W-h/kg may eventually be achievable.

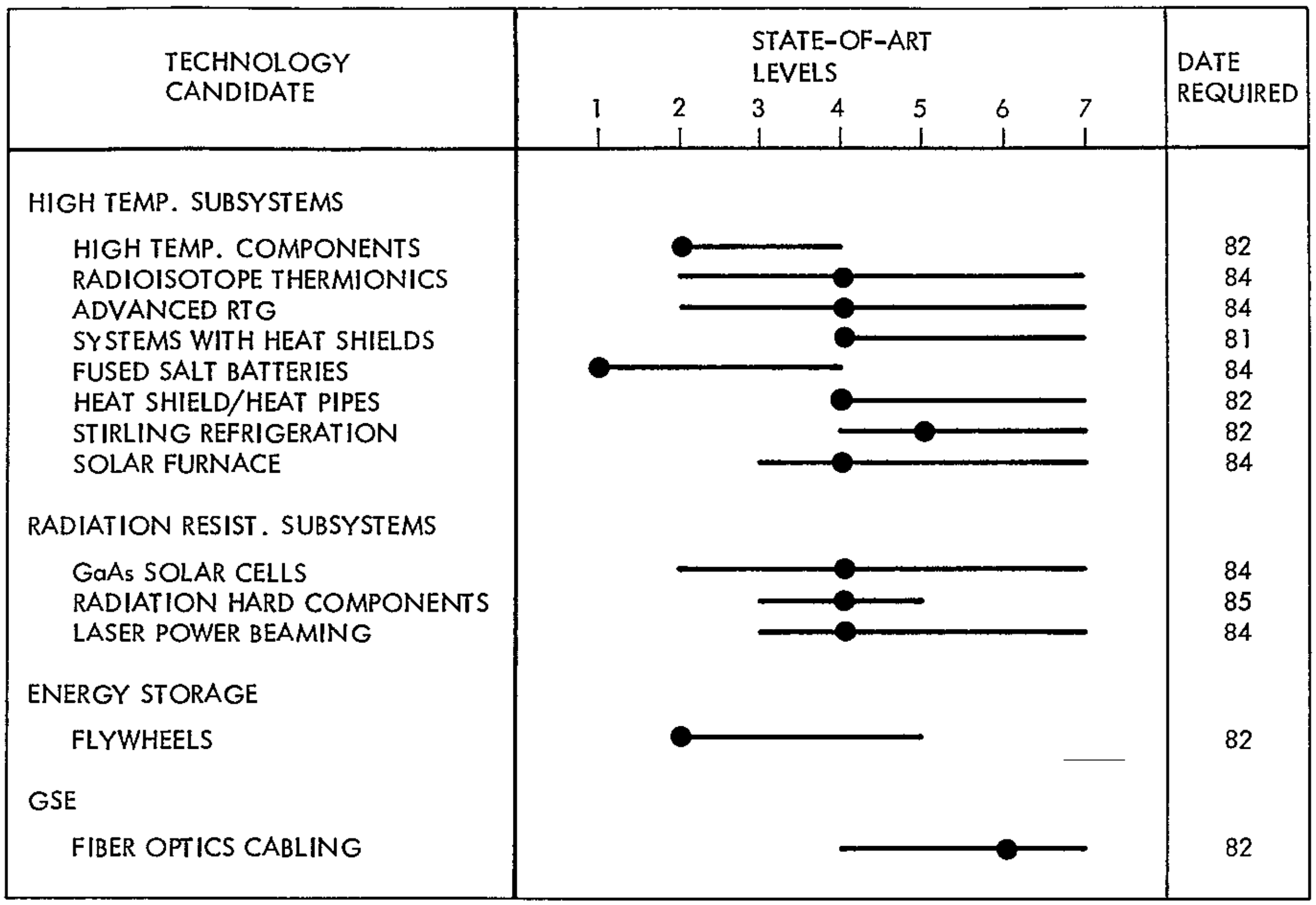
Inductive energy storage (item 3.b) is potentially desirable for large pulsed power applications. Large plasma devices such as lasers and magnetoplasmadynamic (MPD) thrusters will probably need this type of storage, and wider application may also be expected. Present efforts at Princeton University (Reference 5-10) are considered satisfactory, with feasibility now dependent on development of highpower switching.

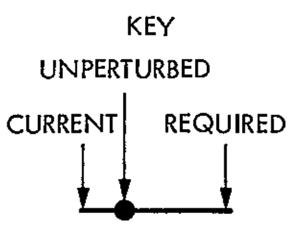
Fiber optics for cabling (item 4.a) has been identified as an important development need for GSE (Reference 5-11). Electromagnetic interference (EMI) problems in GSE cabling can be radically reduced by using fiber optics for signal carriers. EMI caused a loss of the battery and a delay of Viking launch in August of 1975. Fiber optics would probably have eliminated this failure. Although a small program, we would recommend proceeding with it as expeditiously as possible.

C. TECHNOLOGY LEVELS

Schedule implications and state-of-the-art technology levels for the candidates of Table 5-2 are shown in Figure 5-1. Standard nomenclature is used here (References 5-12, 4-2). The present technology state of the art is shown by the position of the left end of the horizontal bar. The required technology level is then shown by the right end of the bar, with the date required shown in the righthand column. A bubble is shown on each bar, indicating the expected technology level (by the date shown) if the program is left unperturbed. Only the programs of Table 5-2 that require augmentation are shown here.

Technology candidates that were identified for new technology readiness programs are shown in Figure 5-1 as going to state-of-the-art level 7. The technology candidates identified for new R&T base programs are shown as going to state-of-the-art level 4 or 5, at which time they should be evaluated for initiation of technology readiness programs. There may be some question whether level 7 (engineering model test in space) is actually needed. Such fine tuning of programs was not evaluated in this study. As a general rule, test in space is considered important to flight readiness status and was so specified. The resulting new technology readiness program details and schedules, including cost and risk estimates, are included in Appendix C. These estimates, together with benefit descriptions, then become the basis for benefit/cost/risk assessment.





STATE OF ART LEVEL DEFINITIONS

1. BASIC PRINCIPLES OBSERVED AND REPORTED.
2 CONCEPTUAL DESIGN FORMULATED.
3. CONCEPTUAL DESIGN TESTED ANALYTICALLY OR EXPERIMENTALLY.
4. CRITICAL FUNCTION/CHARACTERISTIC DEMONSTRATED.
5. COMPONENT/BREADBOARD TESTED IN RELEVANT ENVIRONMENT.
6.. PROTOTYPE/ENGINEERING MODEL TESTED IN RELEVANT ENVIRONMENT
7. ENGINEERING MODEL TESTED IN SPACE.

Figure 5-1. Technology Candidate State-of-the-Art Summary

REFERENCES

5-1. Aerospace Corp., Advanced Mission Analysis Directorate, Advanced Orbital Systems Division, Technology Assessment and New Opportunities Study 2.3, Final Report No. ART-76(7374), December 1976 (Classified).

5-2. Space Processing Payload Equipment Study, NASA-CR-150070, TRW Defense and Space Systems Group, July 1976.

5-3. Prickett, W. Z., G. E. Proposal N-30120, private communication, August 1977.

5-4. Kushida, R., Utilization of Planetary Atmosphere for Power and Propulsion, NASA RTOP Summary, RTOP 506-21-43, 1976.

5-5. Pichel, M., Private Communication, Pichel Industries, Inc. Temecula, CA, October 1977.

5-6. J. H. VanOsdol, Editor, "Design and Integration of Zirconium Hydride Reactor Power Systems", Rockwell International Report No. AI-AEC-13072, (1965).

5-7. Chen, C. J., A. M. Bhanji, and G. R. Russell, "Long Duration, High Efficiency Operation of a Continuously Pulsed Copper Laser Utilizing Copper Bromide as the Lasant", submitted to Appl. Phys. Lett., December 1977.

5-8. Stirn, R. J., Private Communication, October 1977.

5-9. Notti, J. E., et al., Integrated Power/Attitude Control System (IPACS) Study, Vol. 1: Feasibility Studies, NASA CR 2382, April 1974.

5-10. Princeton University, Aerospace and Mechanical Sciences Department, Inductive Storage for Quasi-Steady MPD Thrusters, JPL Contract No. 954351, monthly progress letters.

5-11. Callisen, H. H., "Description and Interface Performance Test Results of the Automated Control and Test System (ACTS)," JPL Report No. 900-816, August 1977 (JPL internal document).

5-12. Ikerd, H. M., "Payloads Technology Space Testing Needs," Preliminary Briefing, General Dynamics Convair Report No. FT-WP-002, August 1975.

SECTION VI

BENEFIT/COST ANALYSIS AND OTHER DISCRIMINATORS

This report section discusses the benefits expected, the bases for analysis and the individual results of the analyses of the identified technology candidates. Comparison of candidates and their relative ranking are discussed in Section VII.

A. TECHNOLOGY DISCRIMINATORS

Benefit/cost/risk analysis has been used in this study as a primary discriminator in prioritizing the technology readiness needs for future missions. A block diagram of the process used is shown in Figure 6-1. Independent input variables to the analysis are mission and system requirements and constraints, technology candidate definitions, program schedules and costing factors. From these inputs it is then possible to generate the benefits available from new technology, convert these benefits into dollar value, and compare benefit dollars against program cost dollars. Sensitivity of program risks against program schedule is also analyzed and converted into program cost.

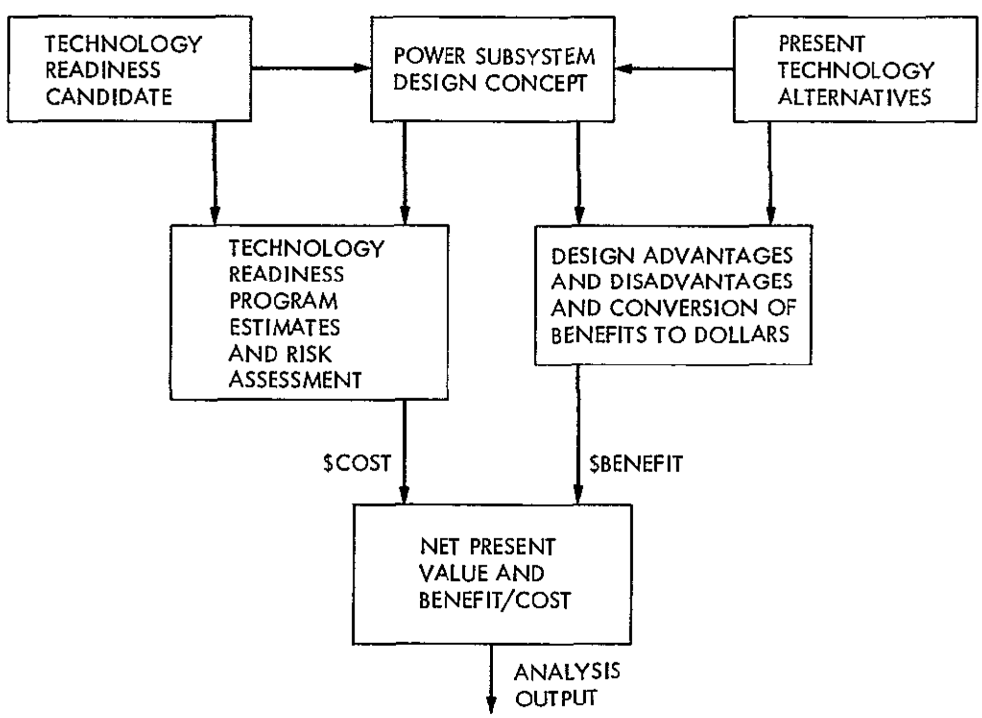


Figure 6-1. Benefit/Cost/Risk Analysis Process

In addition to the B/C/R analysis, there are several other factors that enter into a decision. These are:

(1) Value added by the mission that is enabled.

(2) Amortization of the development over more than one mission.

  (a) Enabling for more than one flight?

  (b) Enhancing for other applications?

(3) Estimation of applicability to other needs.

  (a) Large-power Earth orbit spinoff?

  (b) Terrestrial power spinoff?

For the technologies that have been included for the R&T Base programs, some of the above may be useful. But the principal sorting of R&T Base programs will probably be on the basis of applications potential as follows:

(1) Are applications enabling or enhancing?

(2) Are applications near-term or far-term?

(3) Is the technical complexity of the approach low or high?

Mission timing and spacecraft interfaces were handled directly in the B/C/R analysis. Timing of development implies technology risk as a function of schedule, and was readily parameterized. Spacecraft interfaces were in some instances difficult to specify, but assumptions are made in terms of required hardware and risk that can be treated quantitatively.

High power requirements and low-cost hardware generally track together. This has been recognized in NASA large power technology planning (Reference 6-1) and will be treated in more detail in other NASA studies. However, even within NASA's scientific programs, at relatively low power levels, this problem must also be addressed. There is an earnest desire to greatly expand our space exploration within essentially a fixed budget. The way this could be accomplished is to provide new approaches to mission optimization. One approach, covered under Appendix A, is to reconfigure size, mass and power for minimum cost. The other approach is to increase data return as discussed in item B of this section.

There is a three-way tradeoff to be made among (1) performance, (2) cost and (3) reliability/lifetime. Each time that new technology is introduced, a new optimization is required. The Advanced Propulsion Comparisons (APC) study (Reference 6-7) was a first attempt to look at low-cost missions on a major scale. Larger weight and power availability were shown to be of significant value to low-cost missions. Another possibility, refurbishment of Earth-orbital spacecraft, was controversial in respect to cost savings. This question of refurbishment by on-orbit servicing is evaluated further in Appendix A.

Technology improvement almost always can be related to increased payload capability, improved reliability/lifetime, and lower production cost. The latter is usually important to utilization or industrialization, which is the subject of a separate study. Reliability and lifetime were discussed in the previous section. Only the increased payload capability is discussed here. Assuming that increased power capacity is also available with the increased payload, a large number of options are of real interest:

(1) Reliability and lifetime can be increased (through added redundancy or other technology improvement) and additional propellant can be incorporated to perform additional missions in sequence with the same spacecraft equipment. This adds very little to mission cost, but greatly increases data return.

(2) Equipment cost may be reduced by relaxing size, mass and power specifications. This is a direct cost saving.

(3) Additional science equipment can be provided on a given mission. Since hardware usually represents only 25% of a space program cost, additional science can be very cost effective. Replication of equipment (such as multiple probes) would be even less costly.

(4) For many missions, communication is presently power limited as well as mass limited. Increased communication power may prove extremely fruitful to increase benefit/cost in such cases.

(5) Multi-mission applications may become available because of increased performance reserves. Off-optimum missions can be considered without major system redevelopment, thus eliminating one of the primary costs of present space exploration programs.

(6) Broad-spectrum adaptive science subsystems, previously not considered because of large mass and power demands, can provide mission enhancement. These subsystems would also be valuable from a multi-mission savings viewpoint.

A comparison of planned Shuttle-launched payloads against present spacecraft systems mass shows a near-term growth factor of 1.5 to 2, using the current lightweight-technology of expendable launch systems (Reference 3-8). In addition, however, low-cost concepts of previous studies for DOD (Reference 6-3) suggest another factor-of-two growth to minimize cost. Equipment non-recurring costs are reduced by approximately a factor of two and recurring costs are reduced by about a third. It is our judgement that, to support the larger Shuttle-launched spacecraft, we may expect auxiliary power to increase by about an order of magnitude. Cost of such power increase should rise less than a factor of two.

Our approach for this study is to provide options in technology that support an order-of-magnitude improvement of data "value." If the exponential relationship discussed next in this report holds between data value and quantity of data, this corresponds to a factor of 22,000 in data increase, a number not inconsistent with the Outlook For Space studies (Reference 3-6).

B. CONVERSION OF BENEFITS TO DOLLARS

The areas of analysis for the conversion of benefits to dollars are shown in Table 6-1. Each power subsystem concept and technology candidate is evaluated on this basis. Any benefits derived from other discriminators are to be added as appropriate.

The real benefits of new technology are convertible to budget savings. The savings, however, may occur in several ways. The most direct process is to develop lower cost hardware to replace current hardware, thus reducing mission cost. A less-direct process is to increase the return on the mission with little or no increase of mission cost. A third process, significant to the enabling technology study, is to define new missions not economically feasible with current technology developments. In this last case, benefits may have to be defined out of "whole cloth" and validity may be difficult to establish.

The items of Table 6-1 have been identified as the major areas in which benefits of new technology will be quantified. They fall into two categories: (1) direct benefits and (2) benefits referred to a decreasing marginal utility of data. The latter recognizes a classical, exponentially decreasing value associated with increasing quantity of data from a given mission. It is estimated that such an exponential relationship holds for the present study. Considerable discussion has taken place with science personnel at JPL to evaluate the items shown. Weighting factors, w, will be applied according to the general formula

$$w = A - Be^{-Cx}$$

where x is to be related to a present-technology value of unity. Such a relationship comes about because this study is concerned almost entirely with scientific missions in space, where the principal end product is data. If we make the reasonable assumption that NASA's current space program has been fairly well optimized, varying the data rate from space will probably produce an exponential relationship to the value of the space program. The constant C relates the defined benefit area to data rate. The constants A and B establish the fixed and data-related values for the technology. At present, only relative values have been considered, so that A takes on the value of unity and $B \leq 1.0$. On this basis, the constants of conversion are shown in Table 6-2.

Table 6-1. Areas of Analysis for Conversion of Benefits to Dollars

---

(A) Direct Benefits

- (1) Equipment cost savings
- (2) Launch cost savings
- (3) Cost avoidance
- (4) Standardized design savings

(B) Decreasing Marginal Utility of Data

- (1) Cost breakthrough technology
  - (a) New materials
  - (b) Automated Production
  - (c) Reduced Complexity
  - (d) Automated operations
- (2) Potential for on-board data processing
  - (a) Increased data value
  - (b) Telemetry savings
  - (c) Ground operations savings
- (3) Increased lifetime
- (4) Increased payload mass available
- (5) Wider temperature range/improved control
- (6) Risk reduction

The last item of Table 6-2, involving risk reduction, does not seem to be directly related to mission data. Instead, it recognizes concepts (probably schedule-related) for reducing non-recurring cost during technology development. Such items are generally classified as cost avoidance, and may even show up as benefits in the R&T base programs rather than as mission savings.

The benefit/cost/risk analysis details in the Appendix D correspond to the items in Tables 6-1 and 6-2. Summary curves for the four technology readiness candidates analyzed are shown in Figure 6-2 (a through d). Those technology items that are truly enabling ($B/C = \infty$) are not included here, although detail is shown in the Appendix D.

Table 6-2. Constants For Conversion of Benefits to Dollars

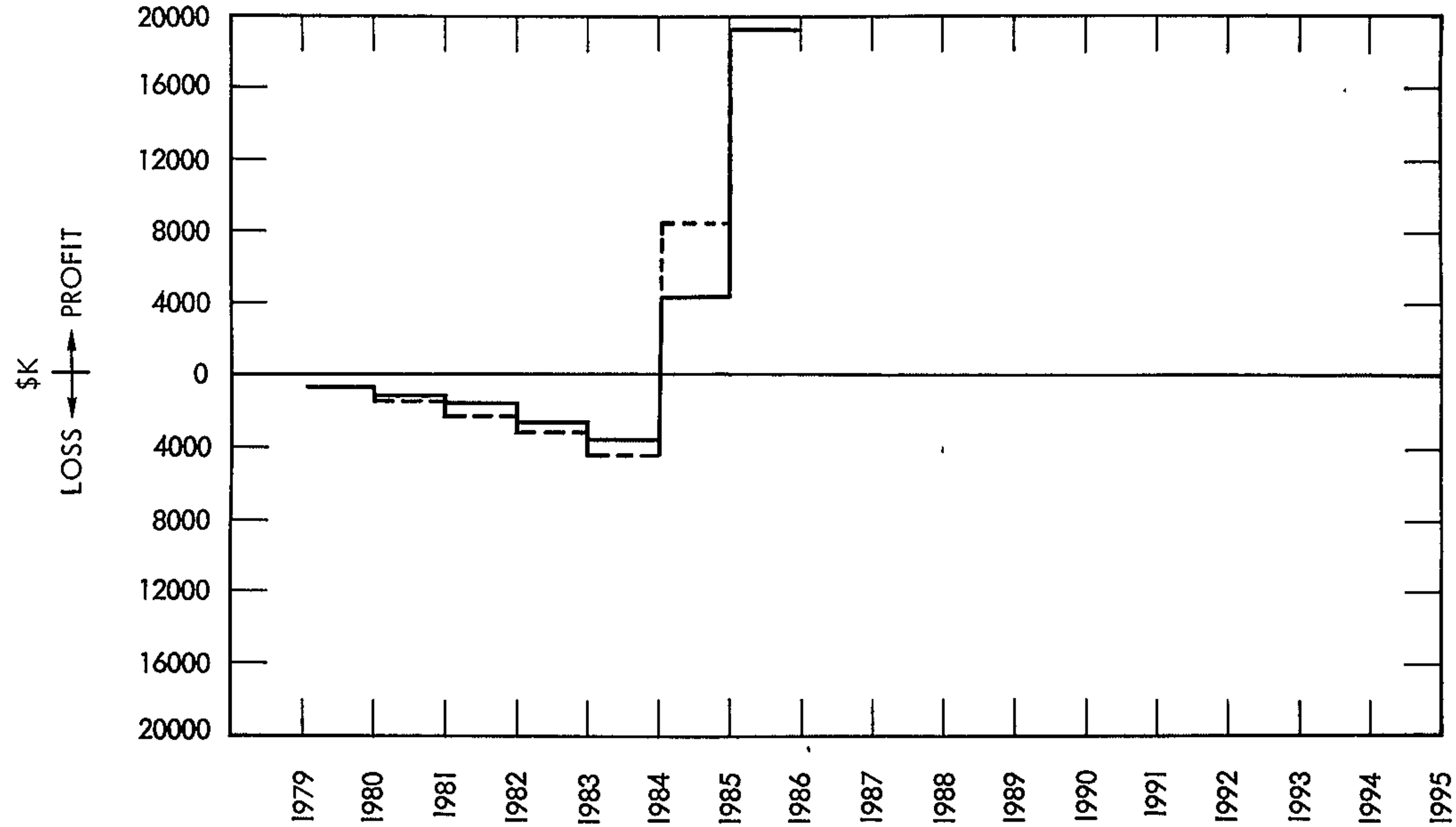
Area of Analysis (From Table 6-1-B)	Estimated Constants ($w = A - Be^{-Cx}$)		
	A	B	C
(1)	1	1.0	0.5
(2)	1	1.0	10.0
(3)	1	1.0	1.0
(4)	1	1.0	1.0
(5)	1	1.0	0.5
(6)	1	0	-

REFERENCES

6-1. Mullin, J. P. and L. B. Holcomb, "Status of the NASA Space Power Program", AIAA Paper No. 77-505, March 1977.

6-2. "Advanced Propulsion Comparison Study", edited by I. L. Helms, Space Nuclear Systems Division of USAEC and NASA, Washington D. C., April 1973.

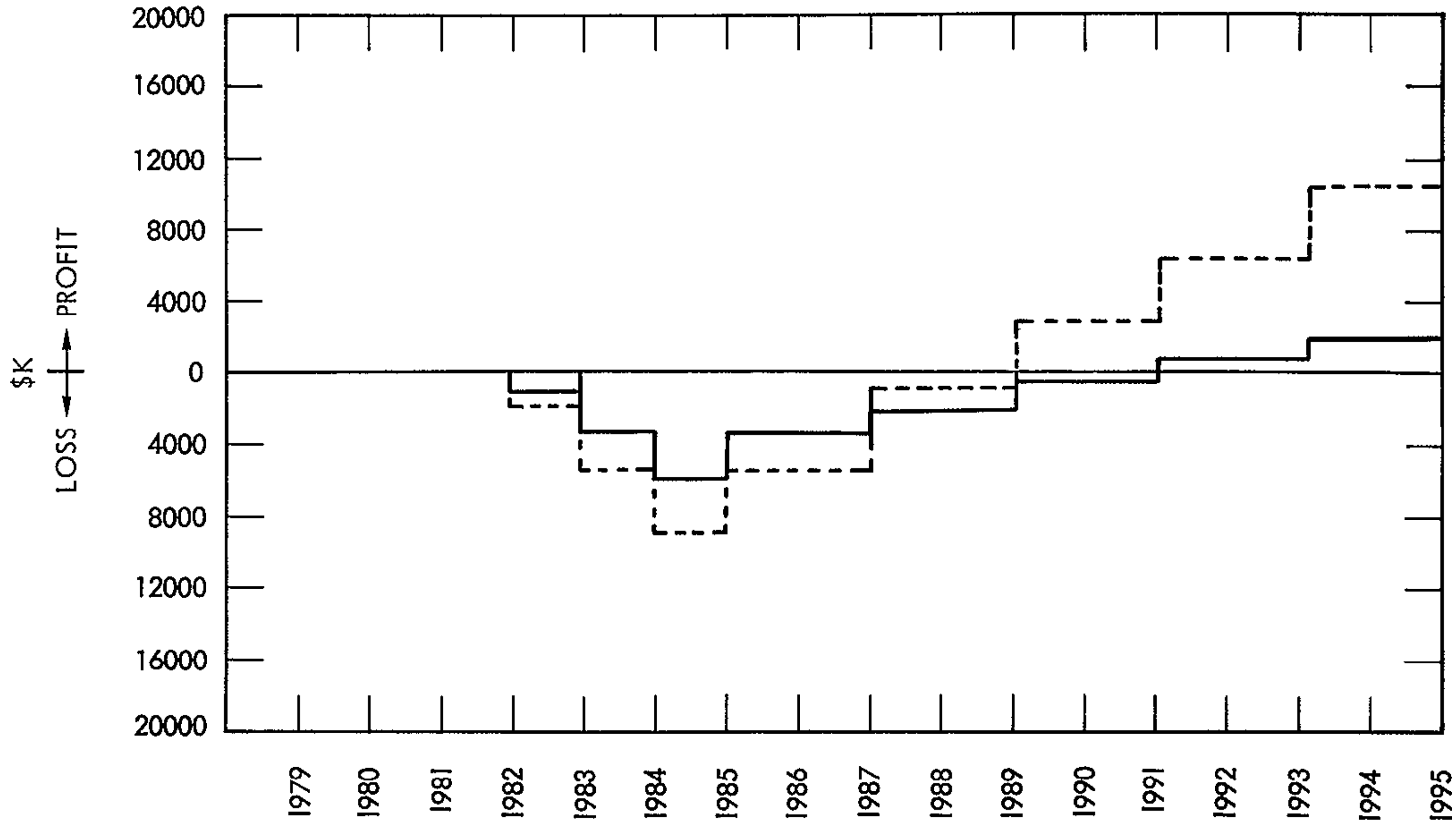
6-3. "Design Guide for Low Cost Standardized Payloads", Lockheed Missiles and Space Company, Report No. LMSC-D154696, April 1972.



KEY		PAYOFF YEAR
1979 dollars, 10% discount rate:	———————	1984
Undiscounted dollars:	— — — — —	1984

BENEFIT/COST RATIO	
1 Mission	3.8
4 Missions	13.4
7 Missions	20.5

Figure 6-2a. Benefit/Cost Analysis, GaAs Solar Cells



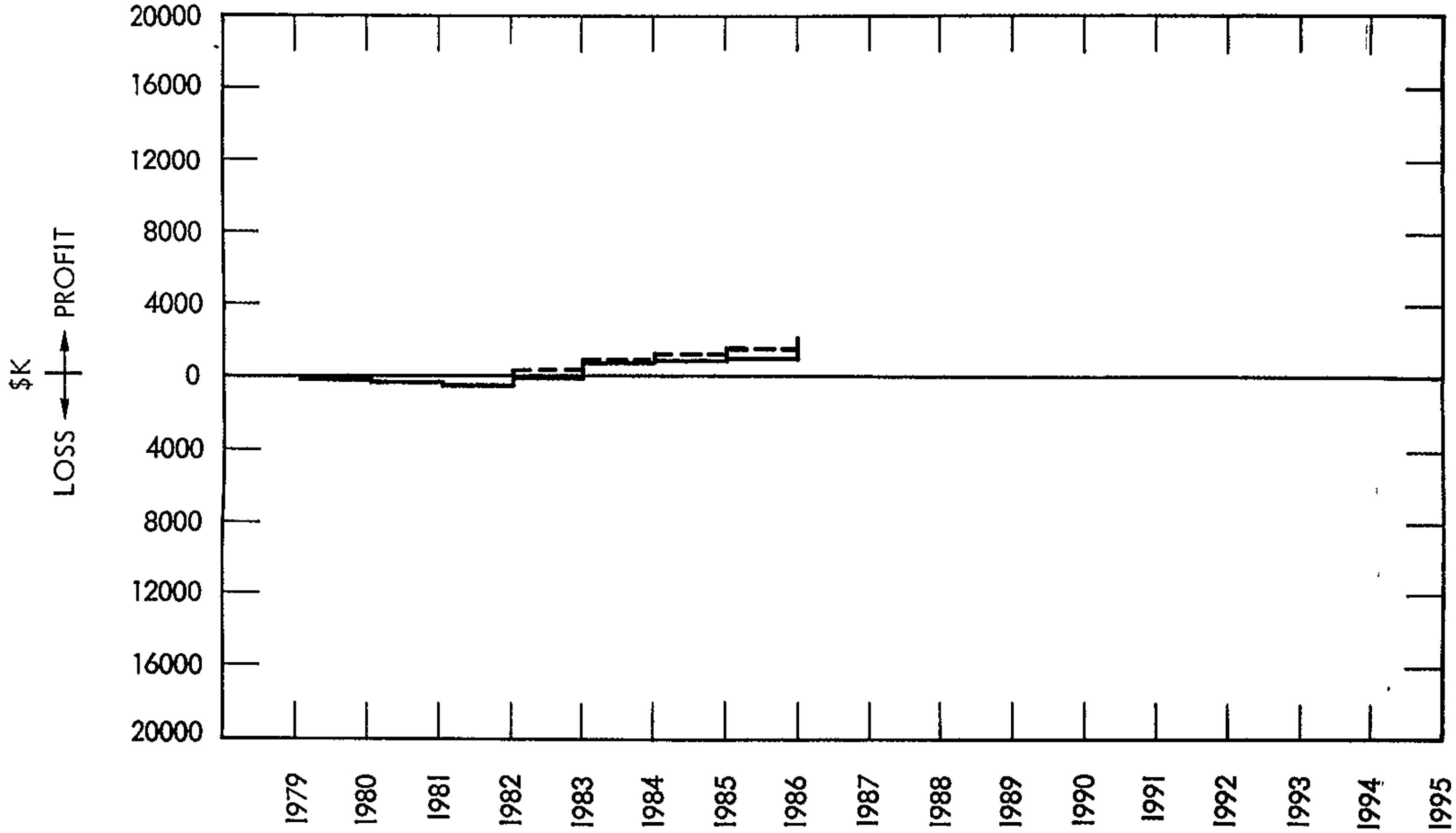
KEY

	PAYOFF YEAR
1979 dollars, 10% discount rate: ———	1991
Undiscounted dollars: — — —	1989

BENEFIT/COST RATIO

1 Mission	0.4
4 Missions	1.1
7 Missions	1.5

Figure 6-2b. Benefit/Cost Analysis, Laser Power Beaming



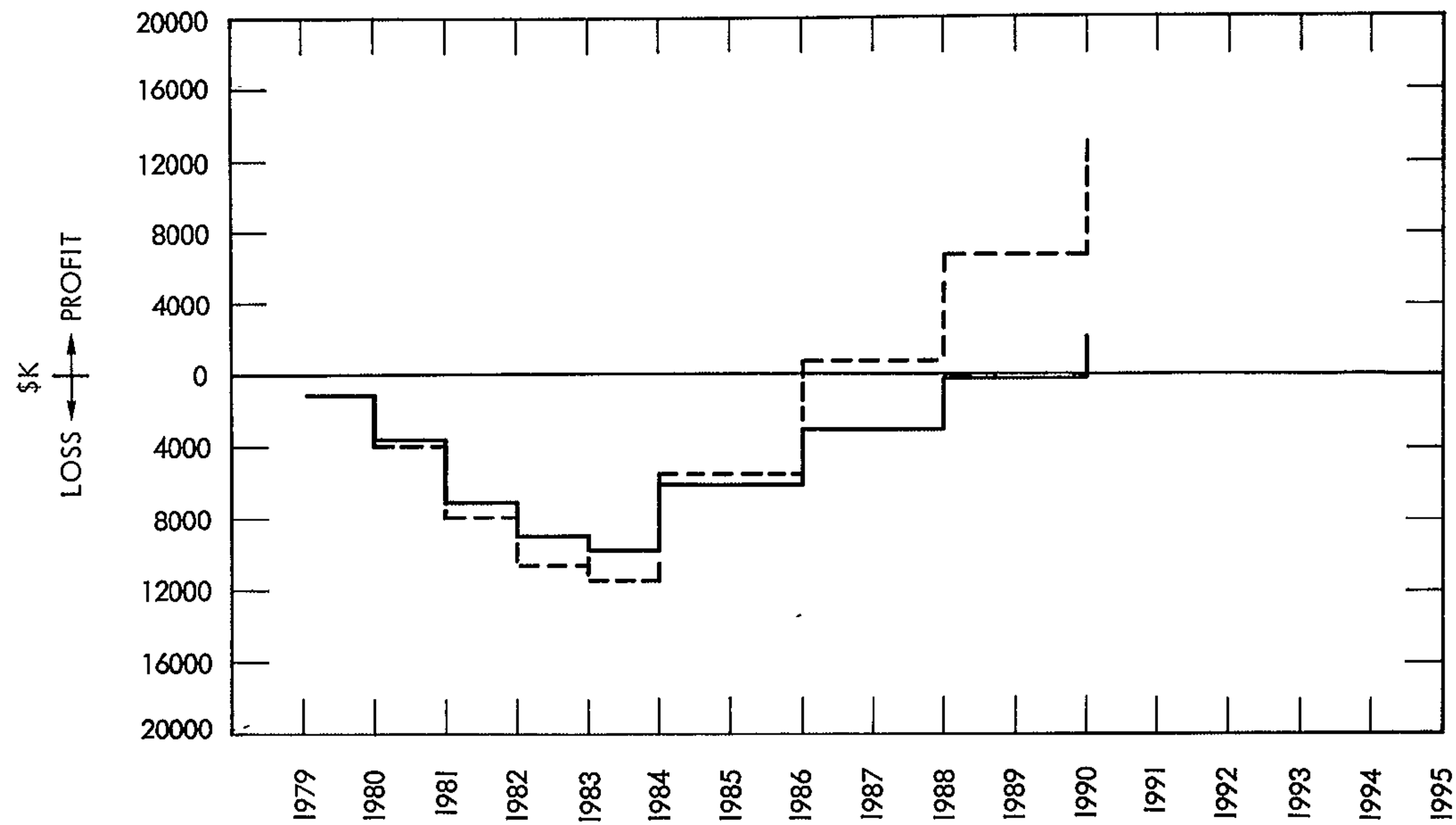
KEY

	PAYOFF YEAR
1979 dollars, 10% discount rate: ———————	1983
Undiscounted dollars: — — — —	1982

BENEFIT/COST RATIO

1 Mission	1.0
4 Missions	3.3
7 Missions	5.1

Figure 6-2c. Benefit/Cost Analysis, Fiber Optics Cabling



KEY		PAYOFF YEAR
1979 dollars, 10% discount rate:	————	1990
Undiscounted dollars:	— — — —	1986

BENEFIT/COST RATIO	
1 Mission	0.4
4 Missions	1.2
7 Missions	1.7

Figure 6-2d. Benefit/Cost Analysis, Stirling Cryogenic Cooling

78-7

SECTION VII

TECHNOLOGY RANKING AND CONCLUSIONS

This report section presents the results of the Technology Enablement study for space power. Comparisons are made of the technology candidates on the basis of the analyses completed in the previous section. Relative ranking is established for the candidates and conclusions are drawn.

Dealing, as we are, primarily with enabling technologies, it is difficult to establish a basis for cost comparison with other technologies. For missions that are truly enabling, cost comparison cannot be made. Also, for the R&T Base candidates any benefit/cost ratio would be zero. Only four of the candidates have a finite benefit/cost ratio, therefore. Tables 7-1, 7-2, and 7-3 show the ranking for the three types of technology candidates identified. Results of the benefit/cost analysis are summarized and the other discriminators listed.

A. NEW TECHNOLOGY READINESS PROGRAMS

1. Direct Application to Space Power Needs.

The new Technology Readiness programs that are directly applicable to space electric power are evaluated and prioritized in Table 7-1. It should be noted that large industrial-type power technology is not included. Neither is nuclear reactor power for nuclear electric propulsion.

GaAs solar arrays are virtually enabling to the cargo orbit transfer vehicle (COTV) with long dwell time in the Van Allen belts. The potential importance of this technology is therefore very great. Auxiliary space power will benefit by much lower degradation from solar flare activity, but separately would not support the high priority of GaAs. The assumption made is that traffic buildup in space will be sufficient by 1988 to support an operational fleet of three COTV's. Spinoff for large power applications also appears possible.

Radioisotope systems with high heat rejection temperature are enabling for long-life operations on the surface of Venus. Either thermionic or advanced thermoelectrics can be used. In addition, these high temperature supplies would provide significant enhancement over standard RTG supplies.

Thermal power conversion systems for use with the heat shield of the solar probe are enabling for that mission. Power levels up to 50 kWe may be necessary, eliminating RTG supplies. The technology however has very limited application.

Laser power beaming provides several operations at the outer planets. It would be used with nuclear reactor power systems. Many of these operations can also be provided at the inner planets, utilizing the large solar electric supplies of the Ion Drive vehicles. Benefit/cost break-even is estimated at 4 flights.

Table 7-1. New Technology Readiness Program Technology Candidate Recommendations (Technologies Directly Applicable to Space Power Needs)

						Spinoff	
Priority	Candidate Technology	Flights	B/C Ratio	Qty Enabled Flights	Qty Enhanced Flights	Earth Orbit Power	Terr. Power
1	GaAs Photovoltaics	1	3.8	>10	>10	Yes	Yes
		4	13.4				
		7	20.5				
2	Radioisotope Thermionic Generator or Advanced RTG		∞	>2	>10		
3	Systems With Heat Shield/ Heat Pipes		∞	>1			
4	Laser Power beaming	1	0.4		5-10	Yes	
		4	1.1				
		7	1.5				

2. Applications Tangential to Space Power Needs.

The new Technology Readiness programs that are more-or-less tangential to the needs of space power are shown in Table 7-2. As such they are of secondary interest in a space power study. Their importance to meeting mission needs, however, cannot be denied and they represent significant interfaces with space power. Ranking has been established primarily on the basis of benefit/cost ratio.

The candidates of Table 7-2 were discussed in Section V of the report. Further detail is provided in Appendix C.

B. NEW R&T BASE PROGRAM TECHNOLOGY CANDIDATES

The new R&T base program technology candidates are evaluated and prioritized in Table 7-3. The selected technologies are broad enough that they all appear to be applicable to multiple missions at the present time. In prioritizing these candidates, near-term applications are more important than far term, enabling technology is more important than enhancing, and low complexity is more important than high complexity. Within this matrix, any finer degree of prioritization is somewhat arbitrary.

The item 4 on the list involves advanced development on high density energy storage. As space power levels increase, this technology may become very important. It should be cross-checked against technology recommendations of other studies on large Earth-orbital power systems. There is no certainty, however, that flywheel technology can be developed in the near term.

C. CONCLUSIONS

NASA's present space power technology readiness programs are satisfactory for the missions planned through 1985. Since these programs are reviewed at least annually with respect to technical content and schedule, a critical review in this study is not considered necessary. Further review might be considered to assure that the technology is truly Shuttle-oriented, making best use of the volume and mass capabilities available. Relaxation of mass constraints and repackaging may produce significant cost savings.

Beyond 1985, much appears to depend upon the further evolution of the STS. Radiation-resistant GaAs solar arrays will be an important contribution. Large payload capability at low transport cost will lead to cost effectiveness in the application of new technology to higher energy missions and increasing mission lifetimes. NASA space programs are expected to be aimed at expanding mission capabilities but with little expansion of NASA resources. In such case, further advances in space power systems will be necessary, providing much larger power levels at much lower specific cost. Low cost mission concepts then become practical with greatly relaxed mass and power constraints. This is a radical departure from previous concepts of spacecraft optimization.

Table 7-2. New Technology Readiness Program Technology Candidate Recommendations
(Technologies Tangentially Applicable to Space Power Needs)

Priority	Candidate Technology	Flights	B/C Ratio	Qty Enabled Flights	Qty Enhanced Flights	Spinoff: Earth Orbit Power	Spinoff: Terr. Power
1	Solar Furnace (Replaces 50 kWe Supply)		∞	1-3		Yes	Yes
2	Heat Shield/Heat Pipes		∞	1-2			
3	Fiber Optics Cabling (GSE Requirement)	1	1.0		>10		
		4	3.3				
		7	5.1				
4	Stirling Refrigerator (10 year cryogenics)	1	0.4		>10	Yes	
		4	1.2				
		7	1.7				

Table 7-3. R&T Program Technology Candidate Recommendations

Priority	Candidate Technology	Type		Application		Complexity	
		Enable	Enhance	Near	Far	Low	High
1	Primary fused salt battery (with indigenous materials)	X			X	X	
2	High temp. power components for inner-planet operations		X	X			X
3	Rad. hard. power components for operation in outer-planet radiation belts		X	X			X
4	Flywheel energy storage for long life, many duty cycles[a]		X		X		X

[a]Potentially important to large power systems.

Space power systems advances are therefore needed in two directions. First, there is an apparent need to increase spacecraft power level by approximately an order of magnitude with very little cost increase. Low cost techniques for which size and mass (and efficiency) are no longer primary constraints should be able to meet this goal. Secondly, as more ambitious missions are opened up by the STS, space power systems enabling technology is needed in a number of specific areas as shown in Table 7-1, 7-2 and 7-3. Cost effectiveness has been identified for the Technology Readiness candidates, and the incentives have been considered for the R&T Base candidates. It is this second direction that has been primarily evaluated in this Technology Enablement study, but the first, although only exercising sound systems practices, should not by any means be under-emphasized.

APPENDIX A

STS UTILIZATION, LIFETIME/RELIABILITY AND ON-ORBIT SERVICING

Understanding the cost-effective use of an evolving Space Transportation System (STS) was found to be essential to development of a mission model and technology recommendations. The STS, we found, requires a revolutionary change in spacecraft concepts if full advantage is to be taken of its capabilities. In addition, however, early evolution of electric propulsion and the cargo orbit transfer vehicle is important so that transport costs from low Earth orbit (LEO) to geosynchronous equatorial orbit (GEO) can be reduced. Thus, where payloads of the early 1980's will look much like those of expendable launch vehicles, we expect the payloads of the late 1980's to be much larger and heavier and to take a radical design departure from present spacecraft. Competition between Shuttle and foreign launch vehicles is expected to be intense during the early "transition" phase, but by carefully planned growth technology the STS should eventually dominate. The question of the ultimate development of a "heavy lift launch vehicle" (HLLV) or "large lift vehicle" (LLV) was found to be unimportant to this Technology Enablement study. An excellent presentation of capabilities and possible alternative approaches to space missions is now available in a short course coordinated by "Hap" Hazard, Navy Space Systems Activity, SAMSO, Los Angeles, CA (Reference A-1).

Our studies here have taken a brief look at alternatives in order to outline a growth pattern for missions with STS. A further, more-detailed analysis, using Operations Research techniques, may be in order within NASA. The STS should be able to compete in a world marketplace against other transportation systems under development. The most vulnerable market is the large number of payloads going to GEO. A second market is the Earth observation satellite traffic in polar or near-polar orbits. A transportation system that cannot capture these markets will be relegated to low-density traffic.

A space freighter type of operation has been envisioned with the STS, but is not totally achievable with the technology of the early 1980s. It appears that a priority development effort is needed for a cargo orbit transfer vehicle by the mid-1980s using electric propulsion. Prior to completion of such development, maximum integration of spacecraft with IUS and SSUS appears vital. Relaxed requirements in spacecraft mass, volume and power level are important to improving the cost advantages of STS. If possible, Shuttle modifications should be considered to improve mass and center-of-gravity capabilities.

A. TRANSITION PAYLOADS

Payloads capable of being launched by either expendable launch vehicles or Shuttle cannot take full advantage of Shuttle potential. The mass and envelope restrictions of expendable launch move toward relatively-high-cost spacecraft design. There are, however, some offsetting concepts in development at the present time that will help reduce cost for these missions of the early 1980s:

(1) The NASA Low Cost Systems Office is coordinating establishment of a line of standardized Sub-Systems to be made available to all Spacecraft and Space Systems designers. The potential for cost savings through Standardized Components/Subsystems should be carefully considered.

(2) The Multimission Modular Spacecraft (MMS) is being developed by NASA/Goddard for flights commencing in FY 1980 (Reference A-2). Four basic spacecraft modules are available with an adapter structure: a communication and data handling subsystem, a power subsystem, an attitude control subsystem, and a signal conditioning and control unit. Propulsion, batteries and solar arrays are not presently included. The MMS is another "standardization" approach to spacecraft. On-orbit servicing with Shuttle has been studied.

(3) The Interim Upper Stage (IUS) potential as a spacecraft bus is currently being evaluated by Boeing (Reference A-3). This concept is discussed in more detail below.

(4) The Spinning Solid Upper Stage (SSUS) is a smaller, spinning propulsion package as an alternative to IUS (Reference A-4). As with the other concepts, this is a standardized piece of hardware to minimize cost.

B. UPPER STAGE INTEGRATION

A review of the potential and planned missions of the early 1980's has shown the importance of integration between the spacecraft and the upper stage systems. The Agena, as an upper stage for expendable launch vehicles, fulfilled this requirement nicely. Its liquid-propellant rocket system was an important feature of the versatility required in the integration process, and has been lost in the IUS and SSUS concepts. Power, communications, guidance and control, and structures were all provided with long lived spacecraft applications in mind.

The IUS, as developed for DOD applications, is very limited. Propulsion packages are single burn, with just a few configurations available. The avionics equipment is very sophisticated, but covers only the minimum launch requirements. It was apparently designed for a very specific set of missions. Other potential users find great difficulty in applying IUS to their needs in a cost-effective manner. Mission planners have therefore avoided IUS as much as possible.

Were it possible to redesign the IUS, considerable improvement could be made. In the propulsion area, greater versatility in velocity package selection could be accomplished. Structure could be provided for mounting sensors on the IUS. By integration of the IUS avionics with the spacecraft functions, replacing primary batteries by secondary batteries, adding solar arrays and a set of momentum wheels, $2 million savings of redundant equipment alone can be made. Communication requirements of many NASA users also are not being met by IUS, which is a difficult NASA-DOD coordination problem that has already been recognized.

Whether there is value in re-engineering the IUS to accomplish integration with spacecraft is very doubtful. If in fact the STS can be planned to provide a versatile cargo orbit transfer vehicle (COTV) for the mid- to late-1980's, any continuing need for IUS will shortly disappear. A liquid-propellant stage might be more productive, particularly if it were aimed at future needs for a personnel orbit transfer vehicle (POTV).

The SSUS takes a different approach from that of the IUS toward integration with spacecraft. Here the concept is to provide almost a minimum-cost velocity package that a spacecraft designer can now integrate specifically. The fixed, single-burn characteristics of the solid rocket motor make this a reasonable concept. The SSUS is a small package that will provide equivalent payload capability on Shuttle to the Atlas (A) and Delta (D) expendable launch vehicles. It is therefore easily identified with the "transition" payload conditions.

The primary problem with the velocity-package concept is that every spacecraft is a different design. The SSUS must therefore be integrated with spacecraft over and over again, which is a costly process. Designers, in fact, are already thinking in terms of more optimum alternatives than SSUS for their spacecraft designs (Reference A-1). Thus the solid-propellant velocity-package market issue is wide open as of this time. The concept, however, is a significant contribution toward the optimization of Shuttle payloads, as will be discussed next.

C. SHUTTLE-OPTIMIZED PAYLOADS

Launch by Shuttle is significantly different than by expendable vehicle launch. The primary payload constraint is length. Mass and diameter are greatly relaxed. Once this is recognized, Shuttle-optimized payload design is directly attainable. A recent NASA study by Hughes Aircraft Company (Reference A-5) has developed the concept of an optimized spinning spacecraft going to GEO. The spacecraft fabrication would use low-cost materials and components (heavier than for the usual weight-limited systems) in a maximum-diameter annulus, thus yielding highest stability for a spinning satellite design. The perigee-kick and apogee-kick propulsion packages were nested inside the spacecraft annulus. Short spacecraft length is a definite asset to Shuttle packaging, and in addition the lower-cost spacecraft more than offsets any increased transport cost.

Total program cost for spacecrafts, exclusive of primary propulsion, may run \$5K-\$100K per kilogram. In addition, delivery cost to GEO, via Shuttle and solid-rocket upper stages, will be approximately \$4,000/kg. This means that spacecraft equipment mass should be increased if, by so doing, the equipment cost saving is greater than \$4,000/kg. For an electric COTV, discussed below, where GEO delivery cost with Shuttle is estimated at under \$1,200/kg, the incentive to reduce spacecraft costs by increasing mass is three times greater than with solid-rocket upper stages.

For spacecraft that are less than 4.5 m long, stowage crosswise in the Shuttle cargo bay may utilize minimum Shuttle bay length. EVA assembly in low Earth orbit (LEO) of two or more spacecraft subassemblies may also improve cargo bay length utilization. This latter is also valuable for adjustment to meet Shuttle center-of-gravity constraints more adequately. In general, cost per unit length of ~\$1,100,000/m is more constraining than the cost per unit mass of ~\$700/kg (due east launch from ETR). If spacecraft are to be stowed axially in the Shuttle bay, either small diameter spacecraft ($D < 1.7$ m) or large diameter spacecraft ($D > 3.5$ m) will be able to provide reasonable packing densities ($\rho/\rho_0 > 0.6$).

Carrying propellants within the spacecraft envelope (other than hydrogen) is an excellent way to improve Shuttle payload density. However, large chemical rocket nozzles typical of high-performance propulsion will reduce payload density. Combining several small spacecraft programs into a single, large spacecraft may also be able to improve packaging density.

Potential for additional volume for Shuttle payloads is also being identified. A 3-meter "stretch" section for Shuttle appears easily feasible. External "strap-on" volume could be designed for very long payload envelopes, but would require relatively expensive aerodynamic containers. There are also small volumes potentially available for fluids in the space below the Shuttle cargo bay. Denser packaging, with integrated propulsion, however, is generally considered as a more-desirable approach to cost-effective payloads on Shuttle.

Transportation cost planning is presently based on 75 percent Shuttle bay utilization. The expected problem, apparently, is an inability to exactly match cargo bay capacity to the scheduled traffic to a particular location. The new concept of small "get-away" payloads at low, economy rates as standby fill-in might improve the Shuttle utilization factor considerably.

D. CARGO ORBIT TRANSFER VEHICLES

Commencing by about 1985, a fully-effective space freight operation is conceivable. Transportation cost reduction by a factor of two or more can be realized on flights requiring upper stage propulsion. The primary cost of upper stage propulsion today is the cost of transport to LEO. By changing to a reusable vehicle at much higher exhaust velocity, cost to LEO is greatly reduced.

Net spacecraft mass to GEO from LEO is presently 25 percent or less. At a Shuttle transport gross cost of $700/kg, the net spacecraft transport cost to LEO becomes $2,800/kg. This is in addition to actual upper stage propulsion hardware cost to the payload of the order of $500-$1,000/kg. A new cargo orbit transfer vehicle with electric propulsion (either solar powered or nuclear powered) is able to deliver at least 60 percent net spacecraft. Propulsion hardware cost per net kilogram delivered is approximately the same as the chemical system. At three round trips or more, total hardware cost per trip is significantly lower than one-way chemical systems. On this basis, net spacecraft transport cost to LEO is reduced to $1,170/kg.

Because electric propulsion is low thrust, trip time may be four months or longer. Priority cargo (and personnel) for fast delivery will still need chemical propulsion at the higher cost. However, round-trip chemical systems of advanced design will offer more versatility, and perhaps equal or lower cost, than the present IUS and SSUS. Both the electric and chemical orbit transfer vehicle should be considered as urgently needed for the mid-1980s. This will assure a competitive transportation system well beyond the transition payload era.

There is an expected need for payloads larger than presently possible with a single Shuttle and IUS. With an electric vehicle already in orbit, GEO payloads of 17,000 kg to 23,000 kg can be accommodated within a single Shuttle launch. Even the chemical vehicle should allow GEO payloads of 8,000 kg to 10,000 kg. The larger spacecraft will also need larger power plants, which our studies indicate will be cost effective. The combination will produce major improvements in space applications at surprisingly low cost.

E. ON-ORBIT SERVICING AND RESUPPLY

In reviewing some of the studies of on-orbit servicing, it appears that value estimates are highly dependent on the input assumptions. When all superfluous assumptions are stripped away, there are just two basic elements that need to be compared:

(1) Long-life spacecraft.

(2) Re-serviceable spacecraft.

Space technology is now approaching a 7-year lifetime capability. However, if a mission lifetime of only one year is defined, it would be wasteful of resources to put 7-year equipment onboard. It is not too unusual to put 3-year equipment onboard a 1-year mission for greater confidence and for "extended mission" capability. Similarly, 7-year equipment may be put on 3-year missions.

At some point it is difficult to project just how long a spacecraft is to remain operational before it becomes obsolete. Present estimates are that a spacecraft will be obsolete in 6 years. This may be a little low as larger spacecraft and/or large and expensive networks are introduced into service. On the other hand, one has only to look at a very long past record of changes in TV and telephone systems, for instance, to question whether 7-10 years is not in the area of diminishing returns. As noted above, present technology is now moving into this 7-10 years lifetime capability. Servicing of such spacecraft would probably not be considered unless a failure occurred.

Another factor of complication to on-orbit servicing is that much of the traffic in Earth orbit is now going to orbits that are beyond the reach of Shuttle. This additional transportation is an added cost increment for servicing missions. Interim upper stages are in development now. Eventually electric cargo orbit transfer vehicles (OTV's) will be introduced.

Still another element for evaluation of spacecraft capability is the new technology in attitude control and stationkeeping. Until now, large inventories of consumables were needed. Recent advances in electric propulsion and magnetic torquing, however, will soon make 7-10 years of operation a straightforward job. Use of this hardware will produce extremely low system life cycle costs.

There is little argument that a 7-8 year spacecraft is more expensive than a 3-4 year spacecraft. The extent of the difference is not easily documented but is of the order of 50 percent. At present, and even with Shuttle, transportation cost is a significant fraction of the mission cost. For multiple-spacecraft missions, transportation cost may be 30-70 percent of the mission cost. Thus, on-orbit servicing cost would need to be somewhere below 15-25 percent of spacecraft cost. With such a low margin for cost improvement potential, further analysis was not done. Some interesting questions for future followup might, however, be contemplated:

(1) The large launch vehicles and advanced OTV's of the future are anticipating a transportation cost reduction by an order of magnitude. How does this affect the value of servicing?

(2) It is conceivable that spacecraft modularity could take into account future technology advances. Would there be increased benefit to on-orbit servicing if it provided new-technology modifications instead of just lifetime extension of old technology?

(3) Would on-orbit servicing be enhanced by putting high-technology robotics/teleoperator capability on an electric cargo OTV?

REFERENCES

A-1. Space Shuttle "Shuttlizing" Payloads, Engineering 805.13, UCLA Extension Short Course, October 17-21, 1977.

A-2. Execution Phase Project Plan for Multimission Modular Spacecraft (MMS), Revision 2, NASA/GFSC, July 1976.

A-3. McMurtrey, L. J., private communication, Boeing Aerospace Co., October 1977.

A-4. Aerospace Corporation advanced planning studies Report No. ATR76 (7377-01)-1, 30 November 1976.

A-5. Utility of Space Transportation System to Space Communication Community, Final Report, Hughes Aircraft Co., NASA-CR-144014, October 1975.

APPENDIX B

MISSION DESCRIPTIONS

The driver missions identified in Section V of the report are considered in more detail here. Since it was the expressed purpose of the mission model to provide a maximum reasonable exercise of power systems technology concepts, effort was made in each instance to describe a challenging mission. In this way we can expect that the worst-case requirements will be met, and future mission planning will yield no surprises in the space power area.

The first important concern is that spacecraft power level will be much larger than previously considered. There are a number of reasons for this increase. The first is that space science equipment might be large, low cost, and a fairly heavy power user. Power levels above a kilowatt might be anticipated for the more-demanding missions, with equipment mass measured in metric tons. So-called "off the shelf" science may not be achievable, but a reasonably-close approach to it may be. Synthetic aperture radar needs may be several kilowatts in addition to the above.

A second power requirement is for spacecraft refrigeration. The solar probe, Mercury and Venus vehicles need general equipment cooling, while RF masers and far infrared sensors require cryogenic refrigeration for 10 years or more in space. Using a Stirling heat engine/heat pump, power levels of 2-3 kWe may be needed.

A third power requirement is in communications. The solar probe with its high-temperature background noise, and the outer planets and solar escape have the largest need. Input power levels may be 5-10 kWe. Tradeoffs can be made with antenna size and on-board data processing.

Electric propulsion attitude control and stationkeeping for Earth orbit spacecraft may vary from a few watts to a few kilowatts, depending on mission. For large structures, not a part of this study, propulsion requirements could dominate the system design. For standard spacecraft, non-propulsive low-cost subsystems may also require power levels of a few watts to a few kilowatts. Adaptive control and rendezvous and docking requirements can be very significant.

The mission descriptions of this appendix attempt to identify the missions and the technology advancements required. Space power subsystems now usually provide a single, standard DC voltage. Requirements are usually 28 VDC ±4 VDC, although Ion Drive spacecraft may be at 200-400 VDC or even higher. Nuclear-electric power sources are expected to operate at approximately 50 VDC.

MISSION DESCRIPTION

MISSION TITLE: MERCURY ORBITER

LAUNCH DATES: 1984-1986

LIFETIME: 400 days transit, 200 days in orbit

MISSION DESCRIPTION:

- Shuttle/IUS launch with Ion Drive for heliocentric transfer
- One spacecraft in 12-h orbit with periapsis altitude 105 km (0.6 h sun occultation)
- One spacecraft in 1.5-h circular orbit at 105 km altitude (0.6 h sun occultation)

SCIENCE OBJECTIVES:

- Measure mass, size, temperatures, surface composition
- Imaging
- Atmosphere constituents
- Magnetic field, energy spectrum

POWER SOURCE:

- Solar photovoltaic array (S/C #1), 300 W
- Solar thermoelectric generator (S/C #2), 450 W

ENERGY STORAGE:

- Ni-Cd Battery (two 12-A-h each S/C)

TECHNOLOGY ADVANCEMENT REQUIRED:

- High temperature arrays and components
- Equipment cooling
- Electromagnetic Shielding (solar corona)

Mercury Orbiter (Continued)

REFERENCES:

- Mission Summaries for COMPLEX, Document 662-25, Jet Propulsion Laboratory Internal Document, Sept. 1976
- Report on Space Science 1975, National Academy of Sciences, 1976
- New Concepts for Mercury Orbiter Missions, JPL Report 760-190, Sept. 1977 (JPL internal document)

MISSION DESCRIPTION

MISSION TITLE: MERCURY LANDER/ROVER/PENETRATIONS

LAUNCH DATES: 1990-1994

LIFETIME: 400 days transit, 400 days on surface

MISSION DESCRIPTION:

- Shuttle/Ion Drive spiral and heliocentric transfer
- Orbiter at 500-1000 km altitude
- Semiautonomous lander/rover (~1500 kg)
- 3-4 penetrators deployed from orbiter

SCIENCE OBJECTIVES:

- Surface geology, chemistry and minerology evolutionary processes
- Multispectral imaging
- Seismometry
- Molecular analysis

POWER SOURCE:

- Solar photovoltaic array (orbiter), 50 kWe
- Radioisotope supply or laser beam (lander/rover), ~1 kWe

ENERGY STORAGE:

- Radioisotope and/or Ni-H_2 battery (1 kW-h)

TECHNOLOGY ADVANCEMENT REQUIRED:

- GaAs solar arrays
- Radioisotope thermionics or advanced RTG
- Laser power beaming (option)
- Electromagnetic shielding (solar corona)
- Equipment cooling

MISSION DESCRIPTION

MISSION TITLE: MERCURY SURFACE SAMPLE RETURN

LAUNCH DATES: 1994-1999

LIFETIME: 400 days transit, 100 days surface, 300 days return

MISSION DESCRIPTION:

- Shuttle/Ion Drive (round trip) spiral and heliocentric transfer, Earth storable land and return to orbit
- Lander/sample collector makes ~2 large hops before return to orbiter.

SCIENCE OBJECTIVES:

- Return surface samples from several sites
- Return small core samples if possible

POWER SOURCE:

- Solar photovoltaic array (orbiter), 100 kWe
- Radioisotope supply (lander/sample collector), 1 kWe

ENERGY STORAGE:

- 12 A-h Ni-Cd or Ni-H_2 battery for orbiter occultation periods

TECHNOLOGY ADVANCEMENT REQUIRED:

- GaAs solar arrays
- Radioisotope thermionics or advanced RTG
- Electromagnetic shielding (solar corona)
- Equipment cooling

REFERENCES:

- Outlook for Space, NASA SP-386, January 1976, Mission No. 1078

MISSION DESCRIPTION

MISSION TITLE: VENUS LANDER/ROVER/PENETRATOR

LAUNCH DATES: 1990-1991

LIFETIME: 200 days transit, 200 days on surface

MISSION DESCRIPTION:

- Shuttle/space storable propulsion
- Buoyant/aerodynamic lander with rover
- 3-4 penetrators deployed upon approach

SCIENCE OBJECTIVES:

- Atmospheric dynamics and composition
- Surface topography, composition, minerology
- Seismology
- Structure and evolution

POWER SOURCE:

- Radioisotope thermionics or advanced RTG, 2-5 kWe

ENERGY STORAGE:

- Fused salt batteries, 1-50 kW-h

TECHNOLOGY ADVANCEMENT REQUIRED:

- RTIG or advanced RTG
- Fused salt batteries
- Vehicle/equipment cooling and pressurization

REFERENCES:

- Colin, L., et al, Future Exploration of Venus, NASA TMX-62450, January 1976
- Lunar and Planetary Mission Handbook, Vol. IV, 760-167, JPL, May 1977 (JPL internal document)
- Outlook for Space, NASA SP-386, January 1976, Mission No. 1088

MISSION DESCRIPTION

MISSION TITLE: VENUS SURFACE SAMPLE RETURN

LAUNCH DATES: 1994-1999

LIFETIME: 250 days transit, 100 days surface, 200 days return

MISSION DESCRIPTION:

- Shuttle/Ion Drive (round trip) spiral and heliocentric transfer, Earth storable landing and return to orbit with aerodynamic/buoyant assist
- Lander/sample collector makes ~2 flights before return to orbiter. Return also includes several atmospheric samples

SCIENCE OBJECTIVES:

- Return several surface and atmospheric samples
- Return small core samples if possible

POWER SOURCE:

- Solar photovoltaic array (orbiter), 100 kWe
- RTIG or advanced RTG (lander/sample collector), 2-5 kWe

ENERGY STORAGE:

- 20-50 A-h Ni-H_2 battery for orbiter occultation periods
- Fused salt battery alternative to radioisotope

TECHNOLOGY ADVANCEMENT REQUIRED:

- GaAs solar arrays
- RTIG or advanced RTG
- Fused salt batteries
- Landed equipment cooling and pressurization

REFERENCES:

- Outlook for Space, NASA SP-386, January 1976, Mission No. 1079

MISSION DESCRIPTION

MISSION TITLE: AUTOMATED LUNAR BASE

LAUNCH DATES: 1991-1995

LIFETIME: 100 days transit, 10 years lunar surface operations

MISSION DESCRIPTION:

- Cargo OTV to lunar orbit
- Lander/rovers (placed in large crater) by space storable system
- Automated excavation for reactor with elevated radiator panels
- #1 Rover deploys reactor, unreels distribution cables, power receptacles and control station
- #2 Rover deploys materials processing station, processing station also has large deployable solar arrays for lunar day operations

SCIENCE OBJECTIVES:

- Collection and evaluation of lunar materials
- Lunar materials processing and controls
- Processed materials stockpiling and storage

POWER SOURCE:

- Nuclear reactor, 500 kWe
- Large solar array, 100-500 kWe

ENERGY STORAGE:

- Ni-Cd or $Ni-H_2$ batteries for rovers, 5 kW-h, or RTG's, 1-2 kWe

TECHNOLOGY ADVANCEMENT REQUIRED:

- GaAs solar arrays (COTV)
- Nuclear-electric power

Automated Lunar Base (Continued)

REFERENCES:

- Thematic Notes for Purple Pigeons, JPL, July 1976 (JPL internal document)
- Aviation Week & Space Technology, August 9, 1976
- Carrier, W., et al, The Nature of Lunar Soil, Amer. Soc. of Civil Eng., Soil Mech. & Foundations Division, Journal, Vol. 99, Oct. 1973, p. 813-832
- Outlook for Space, NASA SP-386, January 1976, Mission No. 1115

MISSION DESCRIPTION

MISSION: URANUS/NEPTUNE FLYBY/PROBE

LAUNCH DATES: 1985-1989

LIFETIME: 12 years

MISSION DESCRIPTION:

- Shuttle/IUS launch with Ion Drive for heliocentric transfer
- Probes at Uranus to evaluate rings and planetary atmosphere. Atmosphere probe at Neptune
- Solar escape for preliminary probe toward heliopause

SCIENCE OBJECTIVES:

- Comparison of composition, atmosphere structure, radiation balance, dimensions
- Scan of several satellites at each planet
- Investigation of local fields and particles

POWER SOURCE:

- Radioisotope supply, 0.5 kWe

ENERGY STORAGE:

- 8 A-h Ag-Zn batteries for probes

TECHNOLOGY ADVANCEMENT REQUIRED:

- Radioisotope thermionics or advanced RTG

REFERENCES:

- JPL Advanced Propulsion Concepts Study, Mission Definitions, Short Version, August 1972 (unpublished)
- Outlook for Space, NASA SP-386, January 1976, Mission No. 1084, 1108, 1109

MISSION DESCRIPTION

MISSION: PLUTO ORBITER/PROBE/SOLAR ESCAPE

LAUNCH DATES: 1998-2000+

LIFETIME: 20 years

MISSION DESCRIPTION:

- Shuttle/NEP to a solar escape velocity of approximately 80-100 km/s in upwind direction of the heliopause (~300 kg net science payload)
- Release ~2000 kg Pluto orbiter/probe at 5-6 AU, space storable retro after ~8 year flight

SCIENCE OBJECTIVES:

- Characteristics of heliopause and interstellar medium, total Solar System mass.
- Detailed characteristics of Pluto (and rings and satellites if any) by comprehensive in situ investigation (Follow-up of 1990 Jupiter/Pluto flyby)
- Determination of stellar and galactic distance scale

POWER SOURCE:

- Nuclear reactor (solar escape spacecraft), 400-500 kWe.
- Radioisotope thermionics or advanced RTG (Pluto Orbiter), 1 kWe

ENERGY STORAGE:

- Ag-Zn or Lithium battery for probe

TECHNOLOGY ADVANCEMENT REQUIRED:

- Nuclear-electric power
- Radioisotope thermionics or advanced RTG
- Long life (>20 years)
- Inductive energy storage

REFERENCES:

- Jaffe, L. D., et al, An Extraplanetary Mission, JPL Publication 77-70, October 1977

MISSION DESCRIPTION

MISSION: NUCLEAR WASTE DISPOSAL

LAUNCH DATES: 1994-

LIFETIME: 8 years

MISSION DESCRIPTION:

- Multiple Shuttle launches of NEP, propellant and waste material to logistics depot in low Earth orbit
- NEP spiral Earth escape and Solar System escape with 60-80 metric tons of nuclear waste material

OBJECTIVE:

- Low Cost Elimination of nuclear waste (~$1,000/kg total cost of transportation)

POWER SOURCE:

- Nuclear reactor, 400-500 kWe

ENERGY STORAGE:

- None

TECHNOLOGY ADVANCEMENT REQUIRED:

- Nuclear-electric power
- Radiation hardened components
- Inductive energy storage

REFERENCES:

- JPL/LASL Heat Pipe/Thermionic Reactor Technology Development Program, Progress Report, 730-6, June 1977 (JPL internal document)
- Perlich, W. F., Space Disposal of Nuclear Wastes, JSC Internal Note No. 75-ER-1 (JSC-10674), Nov. 1975

MISSION DESCRIPTION

MISSION: AUTOMATED PLANETARY STATION

LAUNCH DATES: 1993-2000^{+} (Jupiter, Saturn, Uranus, Neptune)

LIFETIME: 20 years

MISSION DESCRIPTION:

- Shuttle/NEP to planetary orbit
- Orbit individual satellites in sequence for mapping and laser scan of surface
- Release landers/rovers/sample pickup vehicles as required
- Release suborbital probes and atmosphere probes as required
- Rendezvous, if required, with another NEP vehicle for sample-return to Earth

SCIENCE OBJECTIVES:

- A large science laboratory with multiple ancillary vehicles for intensive study of each outer planet system (size, mass, atmospheres, ephemerides, chemistry, minerology, seismology, evolution, fields and particles, molecular analyses, etc.)
- Surface sample return to Earth of selective items

POWER SOURCE:

- Nuclear reactor, 400-500 kWe
- RTIG or advanced RTG for ancillary vehicles

ENERGY STORAGE:

- Lithium batteries as required for probes, penetrators, etc.

TECHNOLOGY ADVANCEMENT REQUIRED:

- Nuclear-electric power
- Radiation hardened components
- Laser power beaming
- RTIG or advanced RTG
- Inductive energy storage

Automated Planetary Station (Continued)

REFERENCES:

- Friedman, L., and C. V. Ivie, Automated Planetary Space Station at Jupiter, JPL Report 760-157, May, 1977 (JPL internal document)
- General Research Corporation, OAST Space Systems Technology Model, May 1977
- Outlook for Space, NASA SP-386, January 1976, Missions No. 1081-1084, 1091-1093, 1096-1097, 4009

MISSION DESCRIPTION

MISSION: SOLAR PROBE/ORBITER (0.01 AU)

LAUNCH DATES: 1984-1985

LIFETIME: 3.5 years transit, 1-2 years orbiting (elliptic)

MISSION DESCRIPTION:

- Shuttle/IUS launch with Ion Drive for heliocentric transfer to Venus
- Venus swingby and subsequent Venus swingbys to achieve ~0.01 AU perihelion

SCIENCE OBJECTIVES:

- Determine solar dynamic oblateness
- Examine and measure solar wind, energetic particles, neutrons, etc.
- Sense solar magnetic field, radio noise emissions
- Sample interplanetary dust
- Measure general relativistic terms
- Make solar observations

POWER SOURCE:

- Thermopile or thermionic power conversion, 50 kWe
- Solar photovoltaic array (Earth-Venus leg), 50 kWe

ENERGY STORAGE:

- Lithium battery for operation outside 0.3 AU, (10-30 kW-h)

TECHNOLOGY ADVANCES REQUIRED:

- High temperature components
- Thermionic or thermoelectric power conversion
- Heat shield/heat pipes
- Electromagnetic shielding
- Stirling refrigeration

Solar Probe/Orbiter (0.01 AU) (Continued)

REFERENCES:

- Friedman, L. D., A Solar Probe Mission, JPL Interim Report 760-133, October 1975 (JPL internal document)

MISSION DESCRIPTION

MISSION: ASTROPHYSICS SPACE LABORATORY

LAUNCH DATE: 1988-1993

LIFETIME: 7-10 years (serviceable)

MISSION DESCRIPTION:

- Geosynchronous orbit preferable
- Highly automated satellite
- May at times be manned as a space station module

SCIENCE OBJECTIVES:

- Multispectral telescopes - UV, visible, submillimeter, X-ray, cryogenic IR
- Gamma ray, cosmic ray and gravity wave detectors
- Programmable laboratory

POWER SOURCE:

- Solar photovoltaic arrays, 10-100 kWe

ENERGY STORAGE:

- Lithium batteries, ~20 kW-h

TECHNOLOGY ADVANCES REQUIRED:

- GaAs solar arrays (COTV)
- Stirling cryogenic refrigeration

REFERENCES:

- General Research Corporation, OAST Space Systems Technology Model, private communication, May 1977

MISSION DESCRIPTION

MISSION: LARGE SCALE MICROWAVE TELESCOPE

LAUNCH DATE: 1988-1993

LIFETIME: 10 years

MISSION DESCRIPTION:

- Low Earth orbit assembly
- COTV to geosynchronous orbit

SCIENCE OBJECTIVES:

- Intergalactic investigations
- Search for intelligent life

POWER SOURCE:

- Solar photovoltaic arrays, 80-100 kWe
- COTV's, 7-8 MWe (fleet of 10-15)

ENERGY STORAGE:

- Lithium batteries in GEO, 10-20 kW-h
- Lithium batteries or flywheels (COTV), 80 kW-h each

TECHNOLOGY ADVANCEMENT REQUIRED:

- GaAs solar arrays (COTV)
- Stirling cryogenic refrigeration

REFERENCES:

- General Research Corporation, OAST Space Systems Technology Model, private communication, May 1977

MISSION DESCRIPTION

MISSION: GEOLOGICAL MAPPING SYSTEM

LAUNCH DATE: 1986-1988

LIFETIME: 7-10 years

MISSION DESCRIPTION:

- A system of satellites as follow-on to ERTS, updating geological maps

OBJECTIVES:

- High-resolution, multispectral active and passive sensors and scanners with data management
- Precision pointing and mapping

POWER SOURCE:

- Thin-film GaAs solar arrays, 10-100 kWe

ENERGY STORAGE:

- Lithium batteries, 5-20 kW-h

TECHNOLOGY ADVANCEMENT REQUIRED:

- GaAs solar arrays
- Stirling cryogenic refrigeration

REFERENCES:

- General Research Corporation, OAST Space Systems Technology Model, private communication, May, 1977
- Nagler, R. G., "Surveying the Earth's Environment from Space: Spectral, Areal, Temporal Coverage Trends," AIAA Paper 77-1585.

MISSION DESCRIPTION

MISSION: CARGO ORBIT TRANSFER

LAUNCH DATES: 1985-

LIFETIME: 3-5 years

MISSION DESCRIPTION:

- Carry large cargo (5-150 metric tons) from low Earth orbit (LEO) to high Earth orbit at any required inclination, or to lunar orbit
- Return (with or without cargo) to LEO
- Maintain spacecraft attitude control during occultation periods in LEO
- Round trip flight time is between 10 and 400 days, 70% in the proton belts

OBJECTIVES:

- Low-cost ($100/kg) transport from LEO to high orbit or lunar supply

POWER SOURCE:

- Thin-film GaAs solar arrays, 500-1,000 kWe

ENERGY STORAGE:

- Lithium batteries, 50-100 kW-h
- Flywheel energy storage, 50-100 kW-h (alternative)

TECHNOLOGY ADVANCEMENT REQUIRED:

- GaAs solar arrays
- Radiation hardened components
- Electromagnetic shielding
- Flywheel energy storage
- Inductive energy storage (MPD thrusters)

Cargo Orbit Transfer (Continued)

REFERENCES:

- General Research Corporation, OAST Space Systems Technology Model, private communication, May 1977
- Stearns, J. W., Large-Payload Earth-Orbit Transportation with Electric Propulsion, JPL Report 33-793, Sept. 1976

MISSION DESCRIPTION

MISSION: SPACE PROCESSING/HIGH TEMPERATURE

LAUNCH DATES: 1984-1985

LIFETIME: 7-30 days (Spacelab)

MISSION DESCRIPTION:

- 15-20 cm diameter furnace on Spacelab operates at a thermal power of ~6 kWt and 2800°K or higher
- A 4.5 m diameter solar concentrator on an experiment pointing mount provides thermal input

SCIENCE OBJECTIVES:

- Glass and other refractory processing at 2800°K and higher can be enhanced by the absence of gravity (no contact with furnace walls)

POWER SOURCE:

- Solar concentrator, 15 kWt at 40-50 percent efficiency
- Thermionic power conversion bottoming available

ENERGY STORAGE:

- None

TECHNOLOGY ADVANCEMENT REQUIRED:

- Solar concentrator/solar furnace
- Thermionic bottoming cycle

REFERENCES:

- Space Processing Payload Equipment Study, NASA CR 150070, TRW Defense and Space Systems Group, July 1976
- Pichel, M., private communication, Pichel Industries, Inc., Temecula, CA, October 1977

APPENDIX C

TECHNOLOGY DESCRIPTIONS

The new Technology Readiness Program technology candidates were detailed here to provide inputs for a benefit/cost analysis in Appendix D, following. Technology program schedule and cost were discussed with those people most knowledgable of the technology. Costs reflect the risk associated with a flight-readiness date as shown in the schedule (item 9) and in Section V of the report.

C-2

TECHNOLOGY DESCRIPTION

TITLE Thin-film GaAs Solar Cells **NO.** 1-a-1

1. DESCRIPTION OF TECHNOLOGY

Development of gallium arsenide solar cells to replace silicon cells. Advantages:

a. Potential higher efficiency.
b. Potential higher radiation resistance.
c. Potential operation at higher temperatures.

2. TECHNOLOGY DRIVERS

The program described herein provides a state of the art level 7 by 1983. The COTV, toward which this program is oriented, is assumed to launch in 1985. Thus it appears that the needed technology will be obtained under high power applications.

3. CURRENT STATUS OF TECHNOLOGY: LEVEL 2

4. REQUIRED LEVEL 7 **DATE** 1985 **MISSIONS** Elect OTV

5. ESTIMATE OF UNPERTURBED ADVANCEMENT: LEVEL 4

6. ALTERNATIVE SOLUTIONS

1. Continue use of silicon cells.
2. Use alternative sources (RTG, etc.)

7. RELATED ACTIVITIES

Development of COTV.
Development of SEP.

8 RECOMMENDED NASA EFFORT

B/C Ratio: 3.6 (1 mission, COTV)
Benefits: $15,300K in 1979 dollars (10% rate), $27,200K undiscounted

ESTIMATED DEVELOPMENT COSTS: $4209K in 1979 dollars (10% rate), $5500 undiscounted

9. SCHEDULE (BY FISCAL YEAR)

	79	80	81	82	83	84	85			
Demonstrate high efficiency GaAs cells	—									
Demonstrate thin film GaAs substrates	—	—	—							
Develop thin film GaAs cells			—	—	—					
Flight qualify GaAs solar cells					—	—				
Elect OTV mission							—			

10. REFERENCES:

ATS-6 Solar Cell Experiment/Improvement Final Report, NAS5-22873, GSFC

1. Technology Description (Cont'd)

The gallium arsenide solar cell is of interest for three basic reasons:

(1) It is potentially capable of higher efficiency than silicon because the material response has a good match with the solar spectrum.

(2) It is potentially radiation resistant because of the high light absorption and low minority carrier diffusion length required. Additionally, radiation does not affect open circuit voltage of the Schottky barrier or AMOS (Anti-reflection coated Metal Oxide Semiconductor) cell.

(3) It can potentially operate at higher temperature than silicon, particularly with a junction-type AlGaAs cell.

Thin film concepts, achievable because of the high absorption of light in GaAs material, are now proposed for development. Such concepts may lead to further weight reductions in solar arrays but, more importantly, could result in cost savings. In large quantity production, material costs of GaAs must be reduced to a minimum. Material availability questions would also be eliminated by the thin film technology (5-10 micrometer substrate with CVD solar cell).

Higher efficiency and reduced radiation degradation are of significant value for large systems with long lifetime requirements. Low thrust earth orbit transfer requires long periods of travel through the proton belts. Solar flare activity is also a primary factor in array degradation. Cost effectiveness can be greatly improved by a low cost GaAs technology.

Recurring cost for the solar cells is expected to be approximately $5-$10 per watt, significantly lower than for silicon cells. Risk factors, however, must be applied. Expected availability date is 1983. In mass production, cost should reduce by at least another order of magnitude.

System cost per unit area for gallium arsenide arrays is expected to be close to that of the silicon array. Structure and labor costs would be very similar. Ultraviolet and meteoroid degradations are about the same, so coverglass is still needed with the GaAs cell. However, end of life power level for GaAs is expected to be higher because of higher cell efficiency and reduced proton and electron degradation. Mass per unit area may also be lower by about 10% to 15% because of the thinner cell material. The higher efficiency of GaAs would lead to an area reduction of 20% to 25%. Degradation, based on the latest ATS-6 solar cell equipment, is estimated as follows:

Silicon Cell Degradation (GEO)	First Year	Subsequent Years
UV & Meteoroid	5%	2%
Particle Radiation*	5%	5%
GaAs Cell Degradation (GEO)		
UV & Meteoroid	5%	2%
Particle Radiation*	2%	2%

*May increase significantly for high solar flare activity.

1. Technology Description (Cont'd)

For nearly-continuous operation in the proton belts of Earth, silicon solar cells are estimated to degrade approximately 60% and then stabilize (at about 40% of their initial power level). The thin-film GaAs cells still require additional radiation testing, but are estimated here to degrade approximately 15%, stabilizing at about 85% of their initial power level. Such assumptions are quite sensitive to proton fluence, and will require a significant test program to verify the above assumptions.

TECHNOLOGY DESCRIPTION

TITLE Radioisotope Thermionic Generator **NO.** 1-a-2

1. DESCRIPTION OF TECHNOLOGY

Develop RTIG technology to the point where it can be used for the Venus Rover/Lander missions, and missions to other planets and their satellites. (con't)

2. TECHNOLOGY DRIVERS

New technology in explosive forming and bonding of BeO and other high temperature ceramic materials is being investigated in a joint program at ERDA's Livermore laboratory and NASA's JPL. Applying this low cost approach to radioisotope (con't)
Probable first mission, VOIR, 1986 (2)

3. CURRENT STATUS OF TECHNOLOGY· LEVEL 2

4 REQUIRED LEVEL 7 **DATE** 1984 **MISSIONS** 1.5 kWe avg. power

VOIR (2), 1986 | Venus Rover/Orbiter (2), 1990
Venus Large Lander (2), 1988 | Venus Sample Return (2), 1994

5. ESTIMATE OF UNPERTURBED ADVANCEMENT. LEVEL 4

6. ALTERNATIVE SOLUTIONS

Advanced thermoelectric power conversion

7. RELATED ACTIVITIES

Thermionic power conversion at 1600K-1800K temperature is a primary effort under the nuclear space power technology program in (con't)

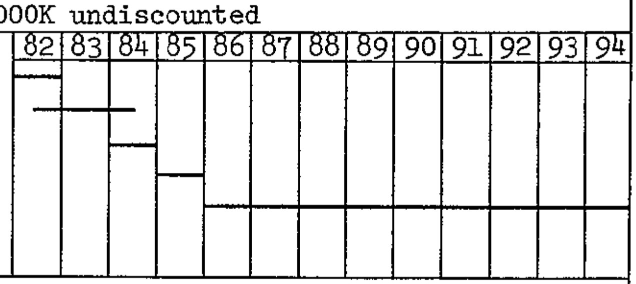
8. RECOMMENDED NASA EFFORT

Enabling for Venus long life lander/rover/sample return
Enhancing for other RTG missions

ESTIMATED DEVELOPMENT COSTS% $5482K in 1979 $ (10% rate), $8,000K undiscounted

9. SCHEDULE: (BY FISCAL YEAR)

System definition and design
System development
System test and evaluation
First mission (Venus Probe)
Succeeding missions



10 REFERENCES

1. GE Proposal N-30120, 15 April 1974, GE Space Div, Energy Systems Program

1. Technology Description (Cont'd)

RTIG technology development is required for a Venus Lander/Rover mission. However, once available, the technology would also be of significant interest for other applications, such as landers, rovers, sub-orbiters, etc. at all of the planets and their satellites. There are advantages of cost and weight reduction compared to the RTG/SIG because of (1) high efficiency and (2) high temperature heat rejection. Basic technology changes involve the need for operation at higher fuel temperature.

Thermionic power conversion is the only type that is compatible with a required heat rejection temperature of 800K or higher. Emitter temperature would need to be approximately twice this, or 1600K. Pressure on the surface of Venus would have minimal impact on system design. Temperatures are still within the range of stainless steel containment although a titanium honeycomb pressure vessel has also been studied.

The RTIG will provide power for the surface craft. The principal power requirement will be for thermal control through active, closed cycle refrigeration. Electrically energized compressor concepts are probably required. High temperature heat engine concepts appear costly and operate at low efficiency. RTIG power levels may be in the 2-5 kWe range for the Venus surface applications. A modular heat source/thermionic power conversion approach would cover this power level as well as the sub-kilowatt needs of other space missions.

The development of a suitable heat source will require a change to ceramic cladding and also the use of concentrator heat pipes. There will also be a need to assure thermal control for shuttle launch. Integration of thermionic power conversion and design for high pressure may also be significant. PuO_2 is still considered to be the most desirable fuel, although Cm_2O_3 has a higher power density. Curium-244 has a more difficult radiation problem than Plutonium-238.

The thermionic system is estimated to be about half the mass of the selenide thermoelectric system. Cost should be about 80% of the selenide system cost, partially because of somewhat reduced fuel requirements. Although the technology is enabling for Venus surface operation, it will also be a significant contribution to other missions where radioisotope power is needed.

2. Technology Drivers (Cont'd)

encapsulation could be very cost effective. Effort should be made to follow up on this possibility.

Advanced technology in molybdenum heat pipes with lithium working fluid is a current effort in nuclear reactor design at Los Alamos Scientific Laboratory. The effort, under NASA cognizance at JPL, also interfaces with the high temperature ceramic technology discussed above. This technology is directly applicable to the RTIG development need. Expansion of this effort to include concentrator heat pipe development should be scheduled, together with evaluation of thermal insulation requirements for the fuel. Cermet fuels should be used if possible. Peak fuel temperature is presently expected to be approximately 250K higher than thermionic operating temperature.

7. Related Activities (Cont'd)

NASA/OAST/RP. Projection is for 15% terminal efficiency in 1978, and 18-20% in FY'80. Comparable system efficiency is approximately 2% less, mainly because of busbar, cabling and switching losses. Thermionic converter technology appears to be directly applicable to the RTIG power system.

TECHNOLOGY DESCRIPTION

TITLE Systems With Heat Shields/Heat Pipes NO 1-a-3

1 DESCRIPTION OF TECHNOLOGY

Integrate hybrid solar thermoelectric generator/thermionic converters with thermal solar shields/solar collectors in order to provide both thermal control and electric generation capabilities to 0.01 AU.

High temperature power systems can take advantage of a heat shield/heat pipe radiator approach to spacecraft thermal control (see Description 1-b-2). Thermoelectric or thermionic power converters (cont'd)

2 TECHNOLOGY DRIVERS

0.01 AU Solar Probe, 1984
First use would be on the Solar Probe, 1984

3. CURRENT STATUS OF TECHNOLOGY: LEVEL 4

4 REQUIRED LEVEL 7 **DATE** 1981 **MISSIONS** Solar Probe

Other possible missions include the Mercury missions (1988 to 1996).

5 ESTIMATE OF UNPERTURBED ADVANCEMENT: LEVEL 4

6. ALTERNATIVE SOLUTIONS

Ablative heat shields; cost would be the same as the system proposed but spacecraft would burn up at about 0.02 AU.
Radioisotope supplies could not meet 20-50 kWe spacecraft demand.

7. RELATED ACTIVITIES

Development of solar thermoelectric generators, and thermionic power conversion technology. Also see Description 1-b-2.

8. RECOMMENDED NASA EFFORT

Enabling for Solar Probe mission.

ESTIMATED DEVELOPMENT COSTS: $2995K in 1979 $ (10% rate)
$3800K undiscounted

9. SCHEDULE. (BY FISCAL YEAR)

Activity	79	80	81	82	83	84				
Thermionic and Thermoelectric converter devel.	—	—								
Thermal control development	—	—								
Heat exchanger integration		—	—							
Flight hardware program			—	—	—					

10. REFERENCES.

Friedman, L.D., A Solar Probe Mission, JPL Interim Report 760-133, October 1975 (JPL internal document)

1. Technology Description (Cont'd)

operate from the waste heat to generate approximately 1 kWe at 0.3 AU and up to 50 kWe at 0.01 AU. A primary technology area is the "heat exchanger" interface with the heat pipes to provide electric power over a temperature range of 1000°K (0.3 AU) to 2100°K (0.01 AU).

Thermionic power conversion technology is being supported for nuclear space power, and will be available at the temperatures and efficiencies needed. High temperature, high efficiency thermoelectric power conversion concepts are not yet well defined.

TECHNOLOGY DESCRIPTION

TITLE Laser Power Beaming | NO 1-a-4

1. DESCRIPTION OF TECHNOLOGY

Large power on board offers possibilities for economical power beaming. At the outer planets, solar power is essentially non-existent. Nuclear power, initially radioisotopes and in the future the nuclear reactor, are required. But with a large reactor on an orbiting spacecraft, low cost power on the surface is possible with visible laser beaming. (Cont'd)

2 TECHNOLOGY DRIVERS

Outer planet missions.

3 CURRENT STATUS OF TECHNOLOGY: LEVEL 3

4. REQUIRED LEVEL 7 **DATE** 1985 **MISSIONS**

Intensive Study/Landers/Rovers/Sample Returns

5 ESTIMATE OF UNPERTURBED ADVANCEMENT: LEVEL 4

6. ALTERNATIVE SOLUTIONS

On board radioisotope power sources.

7. RELATED ACTIVITIES

8 RECOMMENDED NASA EFFORT

B/C Ratio (4 missions): 1.08

Benefits: \$6597K in 1979 \$ (10% rate), \$15,200K undiscounted

ESTIMATED DEVELOPMENT COSTS: \$6066K in 1979 \$ (10% rate), \$9000K undiscounted

9. SCHEDULE. (BY FISCAL YEAR)

	82	83	84	85	86	87	88	89	90	91
Receiver design and development	—	—	—							
Laser design and development	—	—	—							
Missions				—		—		—		—

10 REFERENCES:

See Reference 5-7 of report.

1. Technology Description (Cont'd)

Two different operations are considered: (1) photovoltaic receiver for conversion to electric power, and (2) illumination of the surface for photography. In addition, there are also good possibilities of utilizing the laser beam for guidance and navigation.

A copper chloride electrodynamic laser development is presently planned with 300 W beam power. This should be followed by a scale up to a beam power of 5-10 kW. Wavelength of operation is 5106 Å and 5782 Å (yellow and green). Operating efficiency may be as high as 2%, so that input power of 300-500 kWe is desired. This matches well with the power level of NEP spacecraft to the outer planets.

Gallium arsenide photovoltaics can be tailored to specific wavelengths of operation. By so doing, a 40% power conversion efficiency is achievable. An array area of approximately 1 m^2 would be adequate for the receiver. At this power density operating temperature would be 400-500K, so care would be needed to prevent thermal shock damage during startup of the operation. The power from the spacecraft would be beamed from a laser of less than a meter in diameter. A high pulse rate (approximately 20 kHz) may also be able to provide ac electric power at the surface vehicle for easy power processing.

A laser beam with several precisely defined visible wavelengths from multiple metal chlorides can be coordinated with vidicon cameras for significant photography. It is assumed that a surface spot size of 1000 m^2 can be illuminated adequately from an altitude of 1000 km or less. By appropriate pointing and orbit change, successive orbits would sweep out relatively large swaths of coverage for color photography analysis of the entire surface. It is also possible to incorporate a small laser on the lander for special illumination. These lower power level devices would have an efficiency of approximately 1%.

TECHNOLOGY DESCRIPTION

TITLE Solar Furnace NO 1-b-1

1. DESCRIPTION OF TECHNOLOGY

High temperature ceramic processing experiments in Spacelab will apparently require the development of a solar furnace. Waste heat from the furnace can also be used as input to thermionic power converters to provide electrical power, thus further reducing demands on the Shuttle. (Cont'd)

2 TECHNOLOGY DRIVERS

Space Industrialization

3 CURRENT STATUS OF TECHNOLOGY LEVEL 3

4 REQUIRED LEVEL 7 DATE 1985 MISSIONS Spacelab

5 ESTIMATE OF UNPERTURBED ADVANCEMENT LEVEL 4

6. ALTERNATIVE SOLUTIONS

Supply the electric power from an auxiliary electric power supply with large heat rejection radiators (not a viable solution at this time).

7. RELATED ACTIVITIES

Component technology is already being developed.

8 RECOMMENDED NASA EFFORT

Enabling for Spacelab

ESTIMATED DEVELOPMENT COSTS: $4444K in 1979 $ (10% rate), $5700K undiscounted

9 SCHEDULE (BY FISCAL YEAR)

	80	81	82							
Concentrator Design	—	—								
Furnace Development and Thermionic Fabrication	—	—								
Mounting and Pointing Design		—	—							
Furnace and Power Conversion Test			—							
Concentrator Fabrication and Test			—							

10. REFERENCES:

1. Technology Description (Cont'd)

The thermal input to the furnace may be several kWt. Providing this heating from an electrical power source by means of induction coils or laser beams is very inefficient. Since the Spacelab power supply and thermal radiator are limited to 14 kilowatts, there can be serious restrictions on the experiments. A solar furnace should eliminate the problem. Primary requirements are for high concentration ratio optics and high pointing accuracy.

Shuttle payload bay will limit the solar concentrator size to 4.5 m diameter. The concentrator will also require special packaging to avoid distortion and damage in the Shuttle environment. A Cassegranian surface is assumed, putting two reflective surfaces in series and increasing the mirror accuracy requirements.

Mirror technology was developed by JPL some years ago for solar thermionic space power experiments at 2000K. The mirror is a first surface silver which is electrochemically deposited on a nickel master. Reflectivity is approximately 0.92.

Solar pointing capability as anticipated by the Space Shuttle Experiment Pointing Mount Systems or the European Instrument Pointing System is expected to be within one arc second. This is sufficiently accurate to allow nearly the entire concentrator error tolerance to be assigned to the mirror surfaces.

The principal technology problem in the solar furnace will be cooling of the second mirror in the system. Approximately 2 kWt must be radiated to space, and this can require a sizable heat pipe radiator and introduce dimensional stability problems.

The solar furnace, at the base of the collector system, can be partially enclosed by refractory metal heat pipes that carry the waste heat to a set of thermionic power converters. Heat pipe temperature is 1650K, and the converter efficiency is 15% or higher. Converter heat rejection temperature can be approximately 800K.

Technology development cost for this furnace would be quite nominal. Component technology is already being developed, so that system development is the only effort not yet covered. Close mechanical tolerances with relatively large temperature shift is a primary point of concern. Design of the furnace to operate with a solar concentrator will also be a major task. Weight, however, is seemingly not the most crucial in a Spacelab experiment, thus allowing reasonable engineering design flexibility.

TECHNOLOGY DESCRIPTION

TITLE Heat Shield/Heat Pipes **NO.** 1-b-2

1. DESCRIPTION OF TECHNOLOGY

Integrate hybrid solar thermoelectric generator/thermionic converters with thermal solar shields/solar collectors in order to provide both thermal control and electric generation capabilities to 0.01 AU.

High temperature power systems can take advantage of a heat shield/heat pipe radiator approach to spacecraft thermal control. Spacecraft configuration in its simplest form is in the shape of a T; (Cont'd)

2. TECHNOLOGY DRIVERS

0.01 AU Solar Probe, 1984

3. CURRENT STATUS OF TECHNOLOGY: LEVEL 4

4. REQUIRED LEVEL 7 **DATE** 1981 **MISSIONS** Solar Probe

5 ESTIMATE OF UNPERTURBED ADVANCEMENT: LEVEL 4

6 ALTERNATIVE SOLUTIONS

Ablative heat shields; cost would be the same as the system proposed but spacecraft would burn up at about 0.02 AU.

7. RELATED ACTIVITIES

See Description 1-a-3

8. RECOMMENDED NASA EFFORT

Enabling for solar probe mission

ESTIMATED DEVELOPMENT COSTS. $9274K in 1979 $ (10% rate), $10,600K undiscounted

9 SCHEDULE: (BY FISCAL YEAR)

	79	80	81	82	83	84	85	86	87	88
Heat exchanger/radiator	—	—	—							
Thermal control	—	—	—	—						
Heat shield/collector			—	—						
Power conversion			—	—	—					
System design/test				—	—	—				

10 REFERENCES

1. Technology Description (Cont'd)

the radiators, preferably flat plate, are behind and normal to the plane of the heat shield. The temperature of the thermal control system is maintained as a heat input to thermoelectric and/or thermionic power converters.

Control is on the basis of radiator temperature. The system initially operates with thermoelectric power conversion at heat exchanger temperatures of up to 1400K. The heat exchanger controls the flow of heat to the thermoelectric power conversion unit, shunting the remainder to the primary radiator. As temperature continues to climb, the primary radiator switches from low temperature operation to high temperature operation, and fluid in the low-temperature radiator is eliminated. A second power conversion unit is proposed here, utilizing thermionic converters. However, additional thermal control may still allow the thermoelectric unit to operate. Temperature of the heat shield, heat exchanger, and primary radiator will continue to rise to 2100K-2200K as the spacecraft ultimately moves in to 0.01 AU.

A program is presently in progress for a solar thermoelectric generator for a Mercury orbiter mission. A reasonable extension of this technology to integrate it with a solar shield system would meet the requirements of the solar probe mission. Thermionic power conversion technology is being supported for nuclear space power, and will be available at the temperatures and efficiencies needed.

A reflective tungsten shield is proposed for thermal control inside 0.3 AU. Heat pipes will carry the absorbed heat through a tungsten heat exchanger to a tungsten heat pipe radiator. The spacecraft proper will be behind the heat shield and thermally isolated from the shield-radiator system. Thermal environment is thus completely stabilized, with solar plasma effects becoming the primary constraint on spacecraft operation.

TECHNOLOGY DESCRIPTION

TITLE Fiber Optics Cabling NO. 1-b-3

1 DESCRIPTION OF TECHNOLOGY

The use of fiber optics in the data transmission system between the spacecraft power subsystem and its support system can offer distinct advantages over a wire cable interface. These advantages include complete electrical isolation, freedom from ground loops, reduction in quantity of wire cable interfaces, no compensation required for impedance matching in long lengths of fiber cable and reduced susceptibility to external EMI sources. (Cont'd)

2 TECHNOLOGY DRIVERS

3 CURRENT STATUS OF TECHNOLOGY. LEVEL 4

4 REQUIRED LEVEL 7 **DATE** 1982 **MISSIONS**

Elect. OTV, Inner Planet Missions, Lunar Base, Pluto Orbiter, Nuclear Waste Disposal, Auto Station

5 ESTIMATE OF UNPERTURBED ADVANCEMENT. LEVEL 6

6 ALTERNATIVE SOLUTIONS

Continue the use of wire cable interfacing.

7. RELATED ACTIVITIES

8. RECOMMENDED NASA EFFORT

B/C Ratio: 0.95 based on 1 mission
3.96 based on five missions

Cost: 1 mission, $395K in 1979 dollars (10% rate), $450K undiscounted

ESTIMATED DEVELOPMENT COSTS: $395K in 1979 dollars (10% rate), $450K undiscounted

9. SCHEDULE (BY FISCAL YEAR)

	79	80	81	82	83	84	85	86	87	88
Preliminary design	—									
Design of interface units	—	—	—							
Design of fiber optic cable	—	—	—							
System testing			—	—						
Missions				*	*	*	*	*		

10 REFERENCES

See Reference 5-11 of report.

1. Technology Description (Cont'd)

The baseline concept for using fiber optics as a data transmission link was to improve the total efficiency of the data transmission system. The primary advantages considered are:

a) provide electrical isolation
b) minimize cross coupling
c) reduce interface wire cabling
d) provide increased design flexibility

A Remote Interface Unit (RIU) performs the function of converting the individual analog signals to a serial bit stream which modulates an LED for optical transmission through the fiber bundle to the Power SE. The RIU is physically located in close proximity to the power subsystem/spacecraft. The microprocessor within the RIU provides the software interface to control the format of the serial bit stream.

The data handling requirements within the Power SE are limited to decoding, displaying and recording the serial bit stream. Direct wire interfaces with the power subsystem are still required to provide external power, voltage stimuli, external loads and externally commanded relay functions.

8. Recommended NASA Effort (Cont'd)

Cost:	5 missions, \$395K in 1979 dollars (10% rate), \$450K undiscounted
Benefits:	1 mission, \$376K in 1979 dollars (10% rate), \$500K undiscounted
	5 missions, \$1566K in 1979 dollars (10% rate), \$2500K undiscounted

TECHNOLOGY DESCRIPTION

TITLE Stirling Refrigerator **NO.** 1-b-4

1. DESCRIPTION OF TECHNOLOGY

High performance infrared and microwave systems may require cryogenic cooling. Microwave maser temperature, for instance, is approximately 4.5K. In present systems, stored cryogenic fluids are exhausted in an open cycle cooling system. But if months or years of continuous operation are needed (and several proposed future NASA missions fall into this category) the pumped, closed cycle cryogenic refrigerator will also be needed. (Cont'd)

2. TECHNOLOGY DRIVERS

Probable first mission, Solar Probe, 1984 launch.

3. CURRENT STATUS OF TECHNOLOGY: LEVEL 4

4. REQUIRED LEVEL 7 **DATE** 1982 **MISSIONS** *

*Solar Probes, Earth Observers, Radio Astronomy, Mercury and Venus Missions, Lunar Bases, Nuclear Waste Disposal, Automatic Space Stations

5 ESTIMATE OF UNPERTURBED ADVANCEMENT: LEVEL 5

6. ALTERNATIVE SOLUTIONS

Open cycle cooling systems, which are extremely limited in time duration, particularly in missions involving large cooling requirements.

7. RELATED ACTIVITIES It appears that the Stirling engine and refrigerator might be combined in a single mechanical unit, further simplifying the design. Additional study should be done in this area.

8. RECOMMENDED NASA EFFORT

B/C Ratio: 0.39 based on single mission
1.20 based on four missions

Cost: 1 mission, $9902K in 1979 dollars (10% rate),
$12000K undiscounted

ESTIMATED DEVELOPMENT COSTS: $9902K in 1979 $ (discounted at 10%),
$12000K undiscounted

9 SCHEDULE: (BY FISCAL YEAR)

	79	80	81	82	83	84	85	86	87	88	89	90
Advance Development and Design	—											
Development		—	—	—								
Assembly and Test				—	—	—						
First Mission (Solar Probe)						—						
Succeeding Missions								—		—		—

10 REFERENCES·

1. Technology Description (Cont'd)

The refrigerator systems receiving the most attention presently are the Stirling cycle and the Vuilleumier (VM) refrigerator. For systems of kilowatt size and above, the VM refrigerator does not appear practical. However, the Stirling free piston type refrigerator can apparently provide a compact, lightweight system at high power with essentially the same advantages as the VM system and more. Stirling free piston operation offers a high thermal efficiency, can be completely hermetically sealed, and the few moving parts can be supported on gas bearings to preclude mechanical wear.

Considerable advantage can be taken of the large amount of technology effort already being expended in free piston Stirling, but that effort is basically for terrestrial power generation system applications. Space refrigeration, operating from either electrical or thermal supplies, requires a great deal of additional system design and development, culminating in a long life test demonstration. High temperature input above 800K, with heat rejection temperatures approaching 400K, will provide a reasonably efficient heat engine having a radiator area of less than 1 m^2/KW_t.

For a cooling temperature at 5-10K, we expect the Stirling to operate at 1-2 kW_t per watt of cooling power. Thus, large spacecraft system cryogenic requirements (2-5W) appear to be consistent with approximately a 5 kW_t Stirling engine/refrigerator unit. With redundancy the unit will have a mass of approximately 170 kg, including input and output heat exchangers. Heat pipe radiator area is estimated at 5 m^2, and if a cylindrical radiator is assumed with the Earth orbit environment, a mass of 30 kg may be achievable.

8. Recommended NASA Effort (Cont'd)

Cost:	4 mission, $9902K in 1979 dollars (10% rate), $12000K undiscounted
Benefits:	1 mission, $3875K in 1979 dollars (10% rate), $6240K undiscounted
	4 mission, $11906K in 1979 dollars (10% rate), $24960K undiscounted

APPENDIX D

BENEFIT/COST/RISK (B/C/R) ANALYSIS

As was discussed in Section VI of the main body of the report, Technology Readiness Program schedule, cost and risk assessments were made (Appendix C) to provide cost inputs to the benefit/cost analysis. Technology applications advantages and disadvantages were identified in Tables 6-1 and 6-2 for the conversion of benefits to dollars. Those benefits are listed in the first table of the B/C/R analysis, with value estimates assigned after considerable consultation with spacecraft and science experts. The benefits as a function of year are then called out in the second table, both discounted (10%) and actual dollars. The costs for the technology readiness program (non-recurring) are shown in the third table as a function of year (with a fourth table giving an estimate of cost category). A summary chart, the fifth table of the B/C/R analysis shows the benefit/cost ratio as a function of year and is the basis of Figures 6-a through 6-e in the main body of the report.

A. BENEFIT/COST/RISK ANALYSIS FOR GaAs SOLAR CELLS (No. 1-a-1)

Table D-A-1. Conversion of Benefits to Dollars (Technology: Thin-Film GaAs Solar Cells, No. 1-a-1)

	(Year)	($K)
A. Direct Benefits (1985 flight)		
(1) Equipment savings	84-85	12,500
(2) Launch savings	85	14,700
B. Utility of Data (1985 flight)		
(1) Cost breakthrough technology (materials)	85	1,000
Total Benefits per Flight		27,200

Table D-A-2. Dollar Benefit Flow (Technology: GaAs Solar Cells, No. 1-a-1)

Year	Benefit ($)	Discount Factor	Benefit in 1979 $
1984	13,600K	0.621	8,445K
1985	27,200K	0.564	15,341K
1986	27,200K	0.513	13,954K
1987	27,200K	0.467	12,702K
1988	27,200K	0.424	11,533K
1989	27,200K	0.386	10,500K
1990	27,200K	0.350	9,520K
1991	27,200K	0.319	8,677K
	204,000K		
Total benefit discounted at 10% in 1979 $			$90,472K

Table D-A-3. Dollar Cost Flow (Technology: GaAs Solar Cells, No. 1-a-1)

Year	Cost ($)	Discount Factor	Cost ($) in 1979 $
1979	600K	1.000	600K
1980	600K	0.909	545K
1981	900K	0.826	743K
1982	900K	0.751	676K
1983	1500K	0.683	1024K
1984	1000K	0.621	621K
	5500K		
Total cost discounted at 10% in 1979 $			$4209K

Table D-A-4. Cost of Technology (Technology: GaAs Solar Cells, No. 1-a-1)

Year	Engineering Cost		Manufacturing Cost		Material*	Facilities*
	$	Man Years	$	Man Years	Cost	Cost
1979	300K	5	300K	5	100K	50K
1980	300K	5	300K	5	100K	50K
1981	300K	5	600K	10	200K	100K
1982	300K	5	600K	10	200K	100K
1983	400K	7	1100K	20	300K	200K
1984	500K	9	500K	10	100K	100K

*Included under manufacturing cost.

Table D-A-5. Benefit/Cost Summary (Technology: GaAs Solar Cells No. 1-a-1)

Year	Σ (Benefit-Cost)		Σ (Benefit-Cost)	
	Actual $K	1979 $K	Actual $K	1979 $K
1979	-600	-600	0	0
1980	-1,200	-1,145	0	0
1981	-2,100	-1,888	0	0
1982	-3,000	-2,564	0	0
1983	-4,500	-3,588	0	0
1984	+8,100	+4,236	2.5	2.0
1985	+35,300	+19,577	7.4	5.7
1986	+62,500	+33,531	12.4	9.0
1987	+89,700	+46,233	17.3	12.0
1988	+116,900	+57,766	22.3	14.7
1989	+144,100	+68,266	27.2	17.2
1990	+171,300	+77,786	32.2	19.5
1991	+198,500	+86,463	37.1	21.5

B. BENEFIT/COST/RISK ANALYSIS FOR RADIOISOTOPE THERMIONIC GENERATOR (OR ADVANCED RTG), NO. 1-a-2)

Table D-B-1. Conversion of Benefits to Dollars (Technology: Radioisotope Thermionic Generator (or Advanced RTG), No. 1-a-2)

	Year	$K
A. Direct Benefits		
(1) Equipment savings (radioisotope)	85	1,000
(2) Launch savings	85	100
(3) Cost avoidance (heat source-one time)	85	2,000
B. Utility of Data		
(1) Cost breakthrough (materials, processes)	85	1,000
(5) Wider temperature range (enabling)	86	∞
Total Benefits per Flight	85 only	4,100
	Venus only	∞
	Other	2,100

Note: Table D-B-2 and D-B-5 not applicable to truly enabling technology.

Table D-B-3. Dollar Cost Flow (Technology: Radioisotope Thermionic Generator, No. 1-a-2)

Year	Cost ($)	Discount Factor	Cost ($) in 1979 $
1982	3000K	0.751	2253K
1983	2000K	0.683	1366K
1984	3000K	0.621	1863K
	8000K		5482K
Total cost discounted at 10% in 1979 $			$5482K

Table D-B-4. Cost of Technology (Technology: Radioisotope Thermionic Generator, No. 1-a-2)

Year	Engineering Manpower		Manufacturing Manpower		Material*	Facilities*
	Cost	Manpower	Cost	Manpower	Cost	Cost
1982	1,500	25	1,000	16	500	
1983	1,500	25	1,500	25	1,000	100
1984	1,000	16	1,500	25	1,000	100

*Included under manufacturing cost

C. BENEFIT/COST/RISK ANALYSIS FOR SYSTEMS WITH HEAT SHIELD/HEAT PIPES (No. 1-a-3)

Table D-C-1. Conversion of Benefits to Dollars (Technology: Systems with Heat Shield/Heat Pipes No. 1-a-3)

	Year	$K
A. Direct Benefits		
(3) Cost avoidance (minimum RTG @ 10 kWe)	84	40,000
B. Utility of Data		
(3) Increased lifetime	84	4,000
(5) Wider temperature range	84	∞
Total Benefits per Flight	0.01 AU only	∞

Note: Table D-C-2 and D-C-5 not applicable to truly enabling technology.

Table D-C-3. Dollar Cost Flow (Technology: Systems with Heat Shield/Heat Pipes, No. 1-a-3)

Year	Cost ($)	Discount Factor	Cost (1979 $)
1979	500K	1.000	500K
1980	600K	0.909	545K
1981	200K	0.826	165K
Flight Hardware			
1981	200K	0.826	165K
1982	1,000K	0.751	751K
1983	1,000K	0.683	683K
1984	300K	0.621	186K
	3,800K		2,995K
Total cost discounted at 10% in 1979 $			$2,995K

Table D-C-4. Cost of Technology (Technology: Systems with Heat Shield/Heat Pipes, No. 1-a-3)

Year	Engineering Cost		Manufacturing Cost		Material*	Facilities*
	Cost	Manpower	Cost	Manpower	Cost	Cost
1979	250K	4	250K	4	100K	50K
1980	300K	5	300K	5	100K	50K
1981	100K	2	100K	2		
Flight Hardware						
1981	100K	2	100K	2	50K	
1982	400K	7	600K	10	200K	50K
1983	400K	7	600K	10	200K	50K
1984	100K	2	200K	3	50K	

*Included under manufacturing cost.

D. BENEFIT/COST/RISK ANALYSIS FOR LASER POWER BEAMING (No. 1-a-4)

Table D-D-1. Conversion of Benefits to Dollars (Technology: Laser Power Beaming, No. 1-a-4)

	Year	$K
A. Direct Benefits (1985 flight)		
(1) Equipment savings	85	3,800

Table D-D-2. Dollar Benefit Flow (Technology: Laser Power Beaming, No. 1-a-4)

Year	Benefit ($)	Discount Factor	Benefit in 1979 $
1985	3,800K	0.564	2,143K
1987	3,800K	0.467	1,775K
1989	3,800K	0.386	1,467K
1991	3,800K	0.319	1,212K
1993	3,800K	0.263	1,000K
1995	3,800K	0.218	828K
1997	3,800K	0.180	684K
	26,600K		
Total benefit discounted at 10% in 1979 $			$9,109K

Table D-D-3. Dollar Cost Flow (Technology: Laser Power Beaming, No. 1-a-4)

Year	Cost ($)	Discount Factor	Cost ($) in 1979 $
1982	2,000K	0.751	1,502K
1983	3,500K	0.683	2,390K
1984	3,500K	0.621	2,174K
	9,000K		
Total cost discounted at 10% in 1979 $			$6,066K

Table D-D-4. Cost of Technology (Technology: Laser Power Beaming, No. 1-a-4)

Year	Engineering Cost		Manufacturing Cost		Material*	Facilities*
	Cost	Manpower	Cost	Manpower	Cost	Cost
1982	1,000K	17	1,000K	17	300	100
1983	1,500K	25	2,000K	33	700	200
1984	1,000K	17	2,500K	40	800	200

Table D-D-5. Benefit/Cost Summary (Technology: Laser Power Beaming, No. 1-a-4)

Year	Σ (Benefit/Cost)		Σ (Benefit/Cost)	
	Actual $K	1979 $K	Actual $K	1979 $K
1982	-2,000	-1,502	0	0
1983	-5,500	-3,892	0	0
1984	-9,000	-6,066	0	0
1985	-5,200	-3,923	0.42	0.35
1987	-1,400	-2,148	0.84	0.65
1989	+2,400	-681	1.27	0.89
1991	+6,200	+531	1.69	1.09
1993	+10,000	+1,531	2.11	1.25
1995	+13,800	+2,359	2.53	1.39
1997	+17,600	+3,043	2.96	1.50

E. BENEFIT/COST/RISK ANALYSIS FOR SOLAR FURNACE 9 (No. 1-b-1)

Table D-E-1. Conversion of Benefits to Dollars (Technology: Solar Furnace, No. 1-b-1)

	Year	$K
A. Direct Benefits		
(1) Equipment savings	85	5,000
(2) Launch savings	85	245
B. Utility of Data		
(5) Wider temperature range	85	∞
Total Benefits per Flight		∞

Note: Table D-E-2 and D-E-5 not applicable to truly enabling technology.

Table D-E-3. Dollar Cost Flow (Technology: Solar Furnace, No. 1-b-1)

Year	Cost ($)	Discount Factor	Cost ($) in 1979 $
1980	1,500K	0.909	1,364K
1981	2,000K	0.826	1,652K
1982	1,000K	0.751	751K
1985	1,200K	0.564	677K
	5,700K		
Total cost discounted at 10% in 1979 $			$4,444K

Table D-E-4. Cost of Technology (Technology: Solar Furnace, No. 1-b-1)

Year	Engineering Cost		Manufacturing Cost		Material*	Facilities*
	$	Man Years	$	Man Years	Cost	Cost
1980	1,000K	17	500K	10	100K	100K
1981	1,500K	25	500K	10	100K	100K
1982	500K	8	500K	10	100K	100K
1985			1,200K			

*Included under manufacturing cost.

F. BENEFIT/COST/RISK ANALYSIS FOR HEAT SHIELD/HEAT PIPES, (NO. 1-b-2)

Table D-F-1. Conversion of Benefits to Dollars (Technology: Heat Shields/Heat Pipe Radiators, No. 1-b-2)

	Year	$K
B. Utility of Data		
(3) Increased lifetime	84	10,000
(5) Wider temperature range	84	∞
Total Benefits per Flight	0.01 AU only	∞

Note: Table D-F-2 and D-F-5 not applicable to truly enabling technology.

Table D-F-3. Dollar Cost Flow (Technology: Heat Shields/Heat Pipe Radiators, No. 1-b-2)

Year	Cost ($)	Discount Factor	Cost ($) in 1979 $
1979	500K	1.000	500K
1980	2,400K	0.909	2,182K
1981	3,000K	0.826	2,478K
Flight Hardware			
1981	900K	0.826	743
1982	3,000K	0.751	2,253
1983	1,000K	0.683	683
1984	700K	0.621	435
	10,600K		
Total cost discounted at 10% in 1979 $			$9,274K

Table D-F-4. Cost of Technology (Technology: Heat Shields/Heat Pipe Radiators, No. 1-b-2)

Year	Engineering Cost		Manufacturing Cost		Material*	Facilities*
	$	Man Years	$	Man Years	Cost	Cost
1979	500K	8				
1980	1,000K	16	1,400K	21	300K	100K
1981	2,000K	33	1,900K	33	350K	100K
1982	1,500K	24	1,500K	25	350K	100K
1983	500K	8	500K	8	150K	50K
1984	300K	5	400K	6	100K	50K

*Included under manufacturing cost.

G. BENEFIT/COST/RISK ANALYSIS FOR FIBER OPTICS CABLING (No. 1-b-3)

Table D-G-1. Conversion of Benefits to Dollars (Technology: Fiber Optics Cabling, No. 1-b-3)

	Year	$K
A. Direct Benefits (1982 flight)		
(2) Launch savings (GSE)	82	100
B. Utility of Data		
(2) Data processing	82	150
(6) Risk reduction	82	250
Total Benefits per Flight		500

Table D-G-2. Dollar Benefit Flow (Technology: Fiber Optics Cabling, No. 1-b-3)

Year	Benefit ($)	Discount Factor	Benefit in 1979 $
1982	500K	0.751	376K
1983	500K	0.683	342K
1984	500K	0.621	310K
1985	500K	0.564	282K
1986	500K	0.513	256K
	2,500K		
Total benefit discounted at 10% in 1979 $			$1,566K

Table D-G-3. Dollar Cost Flow (Technology: Fiber Optics Cabling, No. 1-b-3

Year	Cost ($)	Discount Factor	Cost ($) in 1979 $
1979	100K	1.000	100K
1980	150K	0.909	136K
1981	125K	0.826	103K
1982	75K	0.751	56K
	450K		
Total cost discounted at 10% in 1979 $			$395K

Table D-G-4. Cost of Technology (Technology: Fiber Optics Cabling, No. 1-b-3)

Year	Engineering Cost		Manufacturing Cost		Material*	Facilities*
	$	Man Years	$	Man Years	Cost	Cost
1979	100K	2				
1980	100K	2	50K	2	20K	
1981	185K	1	40K	2	10K	
1982	75K	2				

*Included under manufacturing cost.

Table D-G-5. Benefit/Cost Summary (Technology: Fiber Optics Cabling, No. 1-b-3)

Year	Σ (Benefit/Cost)		Σ (Benefit/Cost)	
	Actual $K	1979 $K	Actual $K	1979 $K
1979	-100	-100	0	0
1980	-250	-236	0	0
1981	-375	-339	0	0
1982	+50	-19	1.11	0.95
1983	+550	+323	2.22	1.82
1984	+1,050	+633	3.33	2.60
1985	+1,550	+915	4.44	3.32
1986	+2,050	+1,171	5.55	3.96

H. BENEFIT/COST/RISK ANALYSIS FOR STIRLING REFRIGERATOR (No. 1-b-4)

Table D-H-1. Conversion of Benefits to Dollars (Technology: Stirling Refrigerator, No. 1-b-4)

	Year	$K
A. Direct Benefits (1984 flight)		
(1) Reduced coolant cost	84	40
(2) Launch savings	84	1,700
B. Utility of Data		
(2) Data processing	84	1,000
(3) Increased lifetime	84	1,000
(4) Increased payload	84	2,500
Total Benefits per Flight		6,240

Table D-H-2. Dollar Benefit Flow (Technology: Stirling Refrigerator, No. 1-b-4)

Year	Benefit ($)	Discount Factor	Benefit in 1979 $
1984	6,240K	0.621	3,875K
1986	6,240K	0.513	3,201K
1988	6,240K	0.424	2,646K
1990	6,240K	0.350	2,184K
Total	24,960K		
Total benefit discounted at 10% in 1979$			$11,906K

Table D-H-3. Dollar Cost Flow (Technology: Stirling Refrigerator, No. 1-b-4)

Year	Cost ($)	Discount Factor	Cost ($) in 1979 $
1979	1,000K	1.000	1,000K
1980	3,000K	0.909	2,727K
1981	4,000K	0.826	3,304K
1982	2,500K	0.751	1,878K
1983	1,000K	0.683	683K
1984	500K	0.621	310K
Total	12,000K		
Total cost discounted at 10% in 1979 $			$9,902K

Table D-H-4. Cost of Technology (Technology: Stirling Refrigerator, No. 1-b-4)

Year	Engineering Cost		Manufacturing Cost		Material*	Facilities*
	Cost	Manpower	Cost	Manpower	Cost	Cost
1979	1,000K	17				
1980	2,000K	33	1,000K	17	500K	
1981	1,000K	17	3,000K	33	900K	100K
1982	1,000K	17	1,500K	17	400K	100K
1983	400K	7	600K	10	200K	100K
1984	400K	7	100K	3		

*Included under manufacturing cost.

Table D-H-5. Benefit/Cost Summary (Technology: Stirling Refrigerator, No. 1-b-4)

Year	Σ (Benefit/Cost)		Σ (Benefit/Cost)	
	Actual $K	1979 $K	Actual $K	1979 $K
1979	-1,000	-1,000	0	0
1980	-4,000	-3,727	0	0
1981	-8,000	-7,031	0	0
1982	-10,500	-8,909	0	0
1983	-11,500	-9,592	0	0
1984	-5,760	-6,027	0.52	0.39
1986	+480	-2,826	1.04	0.71
1988	+6,720	-180	1.56	0.98
1990	+12,960	+2,004	2.08	1.20

APPENDIX E

RESEARCH HORIZONS

The definition of new technologies for NASA space power would be incomplete unless it includes those elements of research that will impact missions beyond the year 2000. The technology enablement studies have identified three basic items for which a modest level of research appears desirable. The potential value of these technologies makes it worthwhile to establish feasibility studies.

Lunar base activities will eventually require a nuclear reactor that utilizes thorium and converts it to U-233. It would be desirable to make this a completely automated operation, taking the thorium from the surface of the moon, breeding the U-233 in the reactor, and separating and concentrating the fission fuel. The primary goal is to be able to establish a self-sustaining lunar base power plant. A secondary goal may be to export U-233 to Earth as an energy fuel.

The manned Mars mission continues to be of interest, but is very difficult to achieve with presently considered technologies. Nuclear fusion power offers unusually good energy potential, provided that lightweight, controlled thermonuclear reactions can be achieved. As basic understanding moves ahead in nuclear fusion for terrestrial power, some small effort should also be applied to space utilization of this new field. There is even a possibility that plasma-core fission reaction devices may eventually prove feasible at both high power and high efficiency. If so, a great deal may yet be done with the fission reactor.

Studies already show that matter-antimatter reactions can produce energies sufficient for interstellar travel. There is a great deal that needs to be done in detailed studies, however, such as the interactions between antiprotons and hydrogen, the methods of production and storage of antiprotons, and the possibilities of the generation of electric power from positrons released from the antiprotons.

NASA—JPL—Coml L.A. Calif

jpl PUBLICATION 78-7