NASA Technical Memorandum 105231
AIAA-91-3518

IN-44
47076
p. 12

Trade Studies for Nuclear Space Power Systems

John M. Smith, David J. Bents,
and Harvey S. Bloomfield
Lewis Research Center
Cleveland, Ohio

(NASA-TM-105231) TRADE STUDIES FOR NUCLEAR SPACE POWER SYSTEMS (NASA) 12 p CSCL 10B N92-10221

Unclas
G3/44 0047076

Prepared for the
Conference on Advanced Space Exploration Initiative Technologies
cosponsored by AIAA, NASA, and OAI
Cleveland, Ohio, September 4-6, 1991

NASA

TRADE STUDIES FOR NUCLEAR SPACE POWER SYSTEMS

John M. Smith
David J. Bents
Harvey S. Bloomfield
NASA Lewis Research Center
Cleveland, Ohio

Abstract

As human visions of space applications expand and as we probe further and further out into the universe, our needs for power will also expand, and missions will evolve which are enabled by nuclear power. A broad spectrum of missions which are enhanced or enabled by nuclear power sources have been defined. These include earth orbital platforms, deep space platforms, planetary exploration and extraterrestrial resource exploration. The recently proposed Space Exploration Initiative (SEI) to the Moon and Mars has more clearly defined these missions and their power requirements. This paper presents results of recent studies of radioisotope and nuclear reactor energy sources combined with various energy conversion devices for earth orbital applications, SEI lunar/Mars rover and surface power, and planetary exploration.

Introduction

In response to President Bush's speech commemorating the 20th anniversary of the Appollo 11 Moon landing, NASA has embarked on a study of returning to the Moon to stay followed by a manned mission to Mars. NASA's initial response to this challenge was to complete a "90 Day Study" (ref. 1) which defined various mission scenarios (architectures) that emphasized different themes and long range goals. These were: science and exploration (emphasis on discovery and acquiring information), aggressive Mars mission (emphasis on getting to Mars with the lunar surface being used primarily as a training station), resource utilization (emphasis on lunar oxygen and helium 3 production) and a final emphasis on permanent lunar/Mars occupancy. To expand this national endeavor to include the best thoughts from within government, industry, academia and throughout the country a Synthesis Group was formed which has recently released their findings (ref. 2). While differing in detail and to some extent in emphasis from the "90 Day Study" there was broad agreement between the studies that space nuclear power was enabling for all the mission architectures that might be considered.

The need for nuclear power becomes evident when one considers the power requirements needed to support transportation, construction and mining vehicles; habitation systems and in-situ resource utilization systems. These power levels range from several kilowatts electric (kWe) to megawatts electric (MWe) and must eventually support the lunar base through the 14 earth-day night and Mars applications through a 12 hour night. The energy storage requirements for these long dark periods make solar energy prohibitively massive and expensive for these high power applications.

Several trade studies investigating the use of radioisotope and nuclear reactor energy sources combined with various energy conversion devices have been performed to address the power requirements of some lunar/Mars applications as well as power systems for some precursor robotic missions, earth orbital missions and future planetary exploration.

Radioisotope Power Systems For SEI Applications

Radioisotope power systems include thermoelectric generators energized by decay heat from a radioisotope heat source and dynamic heat engines energized by the same heat source. Radioisotope thermoelectric generators (RTG'S) have already found broad application for deep space missions as characterized by the Pioneer, Voyager, Galileo and Ulysses spacecraft. They also provided surface power for the Apollo Lunar Surface Experiments and the Mars Landers. They will find future use in deep space with the CRAF and Cassini missions and may well find application on precursor SEI missions which were left undefined in the "90 Day Study". These are low power (less than 1 kWe) missions.

The power requirements specified by the "90 Day Study" indicated that more robust power levels will be required for Lunar/Mars exploration. Power levels which range up to approximately 20 kWe will be more advantageously serviced by dynamic isotope power systems (DIPS) as shown in fig. 1 where a performance comparison between the presently used General Purpose Heat Source (GPHS) RTG, the anticipated performance of the next generation Mod RTG and a Brayton DIPS is made. Therefore, in the studies to follow the only radioisotope power system considered was the DIPS.

Lunar Rover and Surface Applications

On the basis of studies carried out during the "90 Day Study" and thereafter, a number of mission enabling or enhancing rover and service vehicles were identified. These were:

Lunar excursion vehicle payload unloader (LEVPU): Provides cargo off-loading and emplacement along with site preparation and construction. It is a large teleoperated crane with daytime operation only.

Mining excavator and regolith hauler: Vehicles used to mine and haul regolith for the in-situ resource utilization (ISRU) plant. Mining only occurs during the lunar day.

Pressurized Rover: Provides a "shirt sleeve" environment for transporting of personnel from the Lunar Excursion Vehicle (LEV) to the habitat, for 4 day exploration missions covering 100 km or more and as a temporary/emergency habitat. It requires both day and night operation.

Unpressurized Rover: Used for robotic missions of up to 1000 km to perform scientific experiments and for crew transport and site construction during the early phase of base development. Has both day and night operation.

LEV Servicer: Provides power to LEV and auxiliary/emergency power for the lunar base. It operates continuously.

The operational requirements and characteristics of these devices, as they can be presently defined, are listed in Table I. From these requirements and a characterization of the vehicle designs required to achieve the above operations, mission power profiles were obtained. These profiles fall into three general categories.

1. Cyclic operation with high peak power requirements, idle periods during the lunar day and little or no lunar night operation. The LEVPU, regolith hauler, mining excavator and certain pressurized and unpressurized rover short missions fit this category.

2. Cyclic operations where the active period is increased from the several hours of category 1 to one or more Earth-days, and may include operation during the lunar night. This category is representative of long duration pressurized or unpressurized rover missions.

3. Continuous operation (over one or more lunar day/night periods) with no cyclic or idle operational periods. The LEV servicer and certain robotic unpressurized rover missions characterize this category.

Representative power profiles for these three categories are shown in Figs. 2-4.

Systems capable of meeting these requirements can be developed now or within the timeframe for the SEI (early 21st century). They make use of the following technologies:

- Solar Photovoltaic (PV)
- Hydrogen/Oxygen Primary Fuel Cell (PFC)
- Hydrogen/Oxygen Regenerative Fuel Cell (RFC)
- Pressurized Gas Reactant Storage for PFC's and RFC's
- Cryogenic Reactant Storage
- High Energy Density Sodium Sulfur Rechargeable Battery
- Dynamic Isotope Power Systems (DIPS)--Brayton Power Conversion Unit

Two distinct strategies of providing vehicle power can be considered. The first method is the self contained power production system characterized by PV/RFC and DIPS. These systems will be required for long missions away from lunar base and for early missions when the base infrastructure is sparse. The other approach is to periodically refuel or recharge the system with fuel or power produced at the base by means of a solar and/or nuclear power system. In this case the on-board power system would be rechargeable batteries, reactant replenished PFC's, or RFC's.

The candidate power systems were investigated in refs. 3 and 4 for application to the previously discussed lunar missions. The results shown in figs. 2-4 characterized these applications based on mass, volume and area. The results are discussed below.

For the mission category representative of the regolith hauler, eight different power systems were compared. Three used on-board PV arrays to provide energy to RFC's or NaS batteries. A fourth system used NaS batteries to provide peaking power with baseload and recharge power provided by a DIPS. Four more systems utilized lunar base power to recharge NaS batteries, provide refuel for PFC's or to power a RFC electrolyzer. These results are shown in fig. 2. From the power profile it is seen that the peak power greatly exceeds the baseline power level but the energy requirement for the peaks is small due to the short time peaking power is required. Since the vehicle is inactive periodically during the day and not used at all during the lunar night, solar power is an acceptable energy source; however, lunar base refueled PFC's have a considerable mass advantage while lunar base recharged NaS batteries have a distinct volume and area advantage which could be a desirable feature for a highly maneuverable vehicle. Similar results were obtained for the LEVPU and mining excavator (ref. 4) and could pertain to certain pressurized and unpressurized rover short missions.

When the cycle period is increased from a few hours to several days, as in the case of the category 2 long range pressurized rover, the energy storage component becomes large enough to make batteries too heavy to consider. Therefore, fuel cells and DIPS are the only options. For this category RFC's with PV arrays were considered in which case the rover could leave the base for up to 14 Earth-days during the lunar day but normally would be dormant during the lunar night. Although the PV/RFC powered rover is recharged and could proceed 4 days into the lunar night it would do so only in emergency since it would not be recharged to begin the next daylight period if it did so. DIPS would have a tactical advantage in that it could also be used during the lunar night and does not require recharge periods. The four other systems using PFC's or RFC's refueled or recharged at lunar base were sized based on returning every 4 days. The results are shown in fig. 3 where it is seen that the unshielded DIPS has substantial mass advantage over all systems except the recharged PFC using cryo storage. Human-rated shielding for the DIPS can result in a severe mass penalty. The DIPS must accept either certain operational constraints on manned activity or a penalty for shielding mass. The extent of the penalty is very design oriented, depending strongly on separation distance and the use of the vehicle structure for shielding. This is discussed in further detail in ref. 3. The results shown in fig. 3 were based on a shadow shield design at a 2m separation distance and an allowable dose of 22 REM over a 90 day mission (22 REM from man-made sources, 50 REM overall).

When the power requirement is increased from several days to providing power continuously through the lunar night, the energy storage component becomes so large that it completely dominates the system. The only non-nuclear option which might be considered is a PV/RFC combined with a cryoplant and tankage (ref. 5). More a stationary power plant than a vehicle, this option has at least three times the mass and volume of the DIPS (fig. 4).

From these results, the dynamic isotope power system appears as an option which is competitive for the greatest number of missions and is the only competitive option for continuous power. Because its competitive attributes are more heavily influenced by application specific factors than the other systems, further examination is warranted. For example, shielding may be required for manned operation but the shielding is specific to the user vehicle configuration and operator schedule. Its impact on the power system cannot be fully assessed until the mission requirements and user installation are better defined.

Comparison of Brayton and Stirling DIPS

In the previous discussion the DIPS power conversion system was a Brayton unit. The continuing development of the high power (10's kWe) free piston Stirling engine (FPSE) as an alternative power conversion unit for the SP-100 space nuclear reactor (ref. 6) under the NASA Civil Space Technology Initiative High Capacity Power Program (CSTI/HCP) (ref. 7) and its application use at lower power levels (ref. 8) makes it an attractive alternative to the Brayton unit in the DIPS application. In ref. 9 a comparison of these two technologies for the DIPS application was performed. Both the Brayton and Stirling power systems used the same DOE General Purpose Heat Source (GPHS). The FPSE characterization was based on the CSTI/HCP Space Power Demonstrator Engine (SPDE) and small engine designs developed at MTI and the Lewis Research Center. Two heater head temperatures were considered: 1050K corresponding to the superalloy engine being constructed under the present phase of CSTI/HCP and 1300K corresponding to the refractory engine being designed for the next phase of CSTI/HCP. As shown in fig. 5 three methods of integrating the FPSE with the radioisotope heat source assembly (HSA) were considered. The first case uses a liquid metal pumped loop, while the second employs heat pipes embedded in a carbon/graphite block surrounded by GPHS modules to carry the heat to the FPSE heater head. This second configuration was studied in greater detail in ref. 10. In the third case the FPSE heater head is surrounded by the GPHS blocks which radiate thermal energy in a directly coupled configuration.

A schematic of the Brayton configuration is shown in fig. 6. The isotope heat source assembly (HSA) was modeled using an algorithium developed in ref. 11. The Brayton converter (turbomachinery, ducting and heat exchangers) was modeled using the Closed Cycle Engine Performance Code (CCEP) described in ref. 12. Two turbine inlet temperatures (TIT) were considered: 1144K corresponding to superalloy materials of construction and 1300K representative of a refractory metal alloy engine.

Fig. 7 shows the results of a minimum mass optimization analysis of the Brayton and Stirling power systems. The variation in mass of the Stirling power system as a function of output power for the three different FPSE heater head configurations is shown. As would be expected the directly coupled radiative case yields the lowest mass. However, its application is restricted to power levels below 1 kWe where the heater head volume to surface area is favorable. The next best design on a mass basis was the heat pipe coupled configuration. Comparison of the Stirling and Brayton results shows a factor of two decrease in mass for the directly coupled Stirling case as compared to the Brayton in the low power range (~200 We). With scaling to higher power levels (~20 kWe) the advantage is reduced to approximately 20%. The comparison also indicates that the Stirling units require radiator areas approximately half those of the Brayton. Similar results were obtained in ref. 13 where a Stirling power system using a heat pipe coupled heater head was compared to a Brayton system for use as a lunar mobile power source in the range from 2.5 kWe to 15 kWe. That study showed that the Stirling power module was 20% lower in mass and required 40% less radiator area than the Brayton system.

Mars Rover/Mars Aircraft

Although the lunar mission architectures were well defined in the "90 Day Study" and subsequent work, mission profiles and hence power requirements for Mars Missions are not well defined. To date the mission plan closely follows the lunar scenario using basically similar devices. Indeed, one of the main features of the ref. 2 architectures is to use the Moon environs and surface to test and gain operational experience with Mars systems and simulate Mars stay times. However, the martian conditions are very different from those of the Moon, e.g.; 24 hr. versus 28 day day/night cycles, increased gravity, CO2 atmosphere, dust storms, reduced solar intensity. Therefore, power requirements and system designs require more study.

One uniquely Martian device recently studied at NASA Lewis (ref. 14) is enabled and/or enhanced by the use of nuclear power. The device is a long endurance aircraft operating in the Martian atmosphere to perform such missions as magnetic and gravity field mapping, terrain mapping, atmospheric surveys and surveillance/reconnaissance missions. Since the aircraft is designed to fly continuously for up to a year, an inexhaustible source of energy is required. Two sources were compared: solar PV arrays and a radioisotope heat source. The design of the aircraft was based on studies of high altitude aircraft for Earth applications since the atmospheric density encountered at approximately 30.5 km above Earth is similar to that encountered near the surface of Mars. A number of Earth applications studies for high altitude aircraft were performed in the early 1980's by NASA Langley Research Center and Lockheed for flight times of up to 1 year. Their concept used solar power with regenerative fuel cells for energy storage.

For the solar powered aircraft "state-of-art" silicon solar cells at 14.2% efficiency and "far term" thin gallium arsenide solar cells at 25% efficiency were considered using a hydrogen/oxygen regenerative fuel cell for energy storage. The solar array panels are located over the solar exposed surface of the aircraft. The aircraft is propelled by a propeller attached through a gear box to an electric motor, as shown in fig. 8. For a system designed to operate at 0^{0} latitude during winter solstice, the aircraft could cover the region from 50^{0} S to 50^{0} N latitude with a flight path that follows the Martian seasons for a period of 1 Martian year. While ref. 14 considered cases allowing higher latitudes the aircraft rapidly increased in size and soon became prohibitively large. Therefore, a reasonable sized solar powered aircraft cannot reach to the Martian polar regions.

Two types of radioisotope heat sources were investigated for the Mars aircraft: Pu238 (material used in the GPHS RTG) and Cm244 which has more than a 7 fold increase in specific energy (535 vs 74 W/kg) over Pu238. Although Cm244 has been used in the terrestrial applications it has not been qualified for space missions and hence represents "far term" technology. To generate power from the heat source a closed cycle Brayton turboalternator was used. The aircraft propeller is driven through a gear box by a turbine and an alternator supplies electrical power for all of the other aircraft functions. A diagram of this system is shown in fig. 9.

The performance characteristics for the solar and DIPS powered Mars aircraft is presented in Table II. It is seen that for the "state-of-art" systems (14.2% silicon versus Pu 238 DIPS) and for the "far term" systems(25% GaAs versus Cm 244 DIPS) the size and mass characteristics of the DIPS powered aircraft are superior to those of the PV powered aircraft. However, both these aircraft due to their ability for controlled flight over large amounts of territory are able to perform mission scenarios beyond the capability of satellites, land rovers or balloons. For this capability the DIPS is again the much more desirable system because it can cover the polar regions which are inaccessible to the solar powered aircraft. Therefore, the DIPS is enabling technology for the aircraft surveillance of the Mars polar regions and significantly enhances missions over the rest of the planetary surface. An artist's conception of the solar powered Mars aircraft is shown in fig. 10.

Space Nuclear Reactor Systems

When power requirements exceed approximately 20 kWe radioisotope heat sources become far too massive and costly when compared to nuclear reactors. A broad spectrum of missions requiring power at these levels has been defined. These missions include earth orbital platforms; earth science and application experiments; earth orbit, lunar and Mars transport; planetary exploration and extraterrestrial resource exploration. The most widely investigated of these missions have been the earth orbiting platform, lunar/Mars base and planetary exploration. The application of nuclear power to these missions is discussed in this Section.

Earth Orbiting Platforms

When the need for earth orbital power exceeds approximately 50-100 kWe the use of large solar PV arrays becomes increasingly difficult. Nuclear power systems have the advantage of simplifying platform dynamics, eliminating the need for continual Sun orientation and, due to their compactness, reduce atmospheric drag in low earth orbit (LEO) with its attendant requirement for propulsive fuel makeup. The compactness of the nuclear system also will facilitate access to the platform by other vehicles, the assembly of large space structures such as antennas and increase the viewing area for on-board experiments and operations. Potential disadvantages are the limitations and constraints imposed by the reactor shielding required to protect platform instrumentation and/or humans.

The application of an SP-100 class nuclear power system to earth orbiting platforms has been studied (refs. 15-17). The major issues addressed in these studies was that of nuclear safety and radiation protection and assessment of the constraints of reactor shield designs.

In ref. 15, three different methods for coupling the nuclear power system to the platform were investigated. These methods were: attaching the reactor directly to the platform, attaching the reactor via a long flexible tether, or locating the reactor on a free-flying power platform. In addition, three options for power transmissions were investigated. These options were: electrical conduction, fuel transport, or electromagnetic beaming.

The recommended design for the platform mounted reactor was a shadow shielded reactor attached to a 70m boom with power transmitted by electrical conduction. The design for the tethered reactor occurred for a tether length of 30 km with power transmission by an electrolysis plant at the reactor. This plant produces and pumps gaseous hydrogen and oxygen through hoses to fuel cells on the main platform. The water produced by the fuel cells is then pumped back to the reactor. This concept proved to be considerably more massive than the boom mounted concept.

In the free flying reactor concept, hydrogen and oxygen are produced on the reactor platform by electrolysis and transported to the main platform by an orbital transfer vehicle which also returns the water produced by fuel cells aboard the main platform. While this concept allows the reactor to reside in a nuclear safe orbit it has the problem of the reactor platform and main platform being in non co-planar orbits for long periods of time due to the difference in drift rates at different altitudes. This results in long storage periods between resupply and/or large propellant consumption from the resulting Delta-V requirements.

In ref. 16 the tethered reactor concept was refined to incorporate electrical power transmission by means of a high voltage DC coaxial tube array, designed to operate in the meteoroid and plasma environment of LEO. Since the tethered reactor has already been shielded to protect its attached machinery, the tether must only be about 2 km in length to attenuate the reactor radiation from instrument safe levels to human rated levels. This was potentially the least massive of all the systems studied in refs. 15-17.

While refs. 15 and 16 investigated nuclear power concepts for earth orbiting platforms, ref. 17 studied the critical questions of installation, platform operation and disposal methodology. Human rated shielding configurations were generated for extravehicular activity (EVA), shuttle orbits approach, docking and departure, and EVA for end-of-life separation and disposal of shutdown nuclear reactor power system. A number of disposal destinations including nuclear safe Earth orbit, solar orbit, solar impact, solar escape, lunar impact and earth return were investigated. Nuclear safe Earth orbit was determined to be the most favorable method of disposal.

Lunar/Mars Applications

For Earth orbit applications at power levels up to about a hundred kWe the advantage of nuclear versus solar power is mainly logistical and hence enhancing rather than enabling. However, as one looks toward the exploration and commercialization of the Moon and Mars, nuclear power becomes the enabling technology on a mass basis for high capacity continuous power. This is due to the massive energy storage requirement for solar systems resulting from the long lunar (14 Earth-days) and Martian (12 hr) nights.

Although realized as necessary for lunar exploration, the exact scenario for introduction of nuclear reactor power into the architecture depends highly on the load profile. A widely considered scenario (ref. 18) is that the initial outpost power would be supplied by PV/RFC. As power requirements increase the outpost would add nuclear reactor power, using SP-100 thermoelectric conversion technology (Ref. 6). As power needs expand further to include such demands as In-Situ Resource Utilization (ISRU) plants an SP-100 reactor with highly efficient dynamic conversion would replace thermoelectrics to provide a significant increase in power level using the same reactor technology.

Of primary concern in the use of nuclear reactors at man-tended sites is human radiation protection. In ref. 19 a lunar base using a 2.5 MW thermal SP-100 nuclear reactor with FPSE power conversion to produce on electrical power output of 825 kWe was considered. As shown in fig. 11, several reactor radiation shielding options were investigated. The first option was to place the reactor in a cavity either provided by natural terrain, blasting or excavating. The radiation shield thus consists of indigenous lunar soil and a Boral bulkhead to prevent soil activation. The second option is a surface mounted reactor and doughnut shaped shield constructed of alternating layers of tungsten and lithium hydride which is transported from Earth. This option is prohibitive on a mass basis since the shield at 20 MT weighs as much as the entire rest of the power system. The third option consisted of mounding soil around the reactor. This requires nearly 20 times the amount of soil to be moved as in the hole excavation concept previously discussed. Since the soil thickness required is approximately 7m, it also requires long heat transport piping from the reactor to the Stirling power converters. As a result of this study the excavated cavity option was selected.

As a follow on to the above study, ref. 20 compares Stirling, Brayton and thermionics power systems for a lunar base application (ref. 18). With a common output power of 550 kWe at 1000 volts DC and a reactor-to-base distance of 250m the Stirling cycle configuration used was that being developed for the NASA CSTI/HCP program and included the 1050K state-of-the-art superalloy and 1300K advanced refractory metal engines. Two recuperated Brayton cycle concepts were considered: the first operates at a state-of-the-art turbine inlet temperature (TIT) of 1140K which corresponds to that developed and tested in a previous NASA program (ref. 21), and the second is an advanced design that operates at a TIT of 1300K utilizing refractory metals. In all of these cases a SP-100 derived (sized to provide appropriate input power) space nuclear reactor was the heat source.

Two thermionic concepts used for comparison were taken from the five cases analyzed in ref. 22. The first concept utilizes the technology being developed in the Thermionic Fuel Element (TFE) Verification Program (ref. 23). It has a reactor output voltage of $\pm$ 7.5 volt, a conservative interelectrode gap and a 1800K emitter temperature. The second case represents advanced technology with a reactor voltage output of $\pm$ 50 volts, a reduced interelectrode gap and an emitter temperature of 2000K.

Fig. 12 shows a mass comparison for the above cases. Of the "near term" technologies (Baseline TFE, 1140K Brayton and 1050K FPSE) the FPSE system has minimum mass. For the advanced technology cases (Adv. TFE, 1300K Brayton and FPSE) there is very little difference in overall mass. Radiator areas are shown in Fig. 13. It is seen that there is a significant reduction in radiator area for the thermionic cases due to their higher heat rejection temperatures. However, the advantage of a smaller radiator area may not be a system driver on the lunar surface and the effect of high temperature (900K) radiators on the proximity of other base elements, human presence and maintenance scenarios has not yet been studied.

Solar System Exploration

Another pontential need for space nuclear reactors is the exploration of the solar system beyond the Moon and Mars. For these missions the distance from the sun is so great that the reduction in solar intensity makes this source marginal and RTG's have been used in past NASA missions. However, with the planning of more ambitious missions and with the ability to maximize mission utilization with increased power the question arises as to the possible advantage of space nuclear reactors to enhance or enable these missions. In an attempt to understand this issue, ref. 24 studied the possible mission benefits of replacing the planned RTG power system on the Mariner Mark II Cassini spacecraft/mission with a small nuclear reactor.

In the first case analyzed a small 1 kWe reactor system was used to simply replace the RTG power system and provide twice the power as shown in fig. 14. In this case the additional mass of the 1 kWe reactor power system located on a 20m boom attached to the spacecraft resulted in a flight time penalty. The penalty ranged from 0.8 to 1.3 years (depending on the assumed radiation tolerance of the electronics) for the planned 6.8 year mission for the RTG powered spacecraft. Although no major advantage was seen in replacing the present RTG power source with a nuclear reactor for this mission, the elimination of the plutonium isotope and the addition of "power to burn" will make the spacecraft design and operation easier.

In a second case the reactor power was increased so that nuclear electric propulsion (NEP) could be used to replace the chemical propulsion system. In this case a relatively low power 25 to 30 kWe NEP system can deliver the Cassini spacecraft to Saturn with no flight time penalty. It also allows a direct trajectory which eliminates all Delta-V gravity assist maneuvers and therefore removes launch window constraints. Moreover, upon reaching Saturn the electric propulsion system can be shut down and the reactor power system can be used to dramatically enhance the science portion of the mission.

The attractiveness of small nuclear reactors to provide power for NEP and to enhance/enable mission science is being studied by JPL (ref. 25) and NASA Lewis for several other NASA planetary missions. These missions may include Neptune Orbiter, Pluto Orbiter, Jupiter Grand Tour, Jupiter Polar Orbiter, Multiple Mainbelt Asteroid Rendezvous, Comet Nucleus Sample Return and Uranus Orbiter. The power supply proposed for these 100 kWe class missions is the SP-100 thermoelectric system currently under development.. The advantages of using NEP for these missions are: shorter flight times (enabling in some cases); additional science with better performance, accessibility and maneuverability at mission site; and multiple rendezvous.

Concluding Remarks

Several recent studies have investigated the use of nuclear power for SEI missions and other space applications. For multi-hundred kWe SEI missions on the Moon and/or Mars nuclear power becomes an enabling technology for many applications due to the long 14 earth-day night on the Moon and the 12 hr night of Mars. These have prohibitively massive energy storage requirements if solar energy is used. For power requirements below tens of kWe radioisotope and solar energy sources can meet specific mission requirements. They can be used for robotic precursor missions, lunar/Mars rovers and small mobile/stationary power systems for baseload/emergency power for lunar/Mars surface applications. At the power level requirement suggested by the "90 Day Study" dynamic isotope power systems are found to be advantageous from a mass and cost basis. For these systems , free piston Stirling engine power converters show an advantage over Brayton cycles on a mass and radiator area basis.

As space power requirements increase beyond the tens of kWe range DIPS become too massive and costly and nuclear reactor power becomes enabling. It can be applied to earth orbiting platforms, lunar/Mars surface applications, planetary missions and nuclear electric propulsion. The developing SP-100 space nuclear reactor can accomplish all of these missions in the range of 10-100 kWe when combined with thermoelectric conversion units. Above this power level up to ~1MWe, SP-100 derived reactors with free piston Stirling converts appear to be advantageous. This fulfills the requirements identified by the "90 Day Study". However, NEP for lunar/Mars transport applications requires MWe's of power. For this application a scaled-up SP-100 derived reactor with Brayton or Rankine conversion is required. For unmanned planetary missions nuclear reactors replacing RTG's provide an advantage in allowing the use of NEP to shorten flight times, eliminate launch windows created by gravity assist maneuvers and provide additional power at mission sites to enhance and/or enable various science missions.

References

1. Cohen, A., editor: "Report of the 90 Day Study on Human Exploration of the Moon and Mars", NASA Johnson Space Center, November 1989.

2. Stafford, T. P., editor: "America at the Threshold", Synthesis Group Report, U. S. Gov. Printing Off., Washington, DC, May 1991.

3. Bents, D. J., et al, "SEI Power Source Alternatives for Rovers and Other Multi-kWe Distributed Surface Applications", NASA TM 104359, European Space Power Conference, Barcelona, Spain, September 1991.

4. Withrow, C. A., et al, "SEI Rover Solar-Electrochemical Power Systems Options", NASA TM 104402, May 1991.

5. Kohout, L., "Cryogenic Reactant Storage for Lunar Base Regenerative Fuel Cells", NASA TM-101980, International Conference on Space Power, International Astronautical Federation, Cleveland, Ohio, June 1989.

6. Mondt, J., "Overview of the SP-100 Program", AIAA/NASA/OAI Conference on Advanced SEI Technologies", AIAA Paper No. 91-3585, 1991.

7. Winter, J. M., "The NASA CSTI High Capacity Power Program", AIAA Paper No. 91-3629, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 1991.

8. Ross, B. A., Dudenhoefer, J. E., "Stirling Machine Operating Experience", 26th Intersociety Energy Conversion Engineering Conference, Boston, Mass., August, 1991.

9. Bents, D. J. et al, "Comparison of Dynamic Isotope Power Systems for Distributed Planetary Surface Applications", Proceedings of Eighth Symposium on Space Nuclear Power Systems, Vol II, pp. 586-597, Institute for Space Nuclear Power Studies, Albuquerque, NM, January 1991.

10. Schmitz, P. C., "Preliminary Design of a Mobile Lunar Power Supply", NASA TM-104471, June, 1991.

11. Brandewre, R., "Lockheed Missiles and Space Company BSTS Power System Definition Study, Rev. 1", RI/RD86-188, Rocketdyne Div., Rockwell International Corporation, Canoga Park, CA., 1986.

12. Klann, J. L., "Status of the 2 to 15 kWe Brayton Power System and Potential Gains from Component Improvements," Intersociety Energy Conversion Engineering Conference, 6th Proceedings, pg. 195-201, Society of Automotive Engineers, 1971.

13. Harty, R. B., "A Comparison of Radioisotope Brayton and Stirling Systems for Lunar Surface Mobile Power", Proceedings of Eighth Symposium on Space Nuclear Power Systems, Vol II, pp. 934-937, Institute for Space Nuclear Power Studies, Albuquerque, NM, January 1991

14. Colozza, A. J., "Preliminary Design of a Long-Endurance Mars Aircraft", AIAA Paper No. AIAA-90-2000, July, 1990.

15. Silverman, S. W., Willenberg, H. J. and Robertson, C., "Applicability of 100 kWe-Class of Space Reactor Power Systems to NASA Manned Space Station Missions", 0180-28461-1, Boeing Aerospace Co., NASA CR-174696, 1984.

16. Bents, D. J. , "Tethered Nuclear Power for the Space Station", NASA TM 87023, 1985.

17. Bloomfield, H. S. and Heller, J. A., "Feasibility Assessment of Installation, Operation and Disposal Options for Nuclear Reactor Power System Concepts for a NASA Growth Space Station," NASA TM-89923, 1987.

18. Petri, D. A., Cataldo, R. L. and Bozek, J. M., "Power System Requirements and Definition for Lunar and Mars Outposts", Proceeding of the 25th Intersociety Energy Conversion Engineering Conference, Volume I, pp. 18-27, 1990

19. Mason, L. S. and Bloomfield, H. S. and Hainley, D. C., "SP-100 Power System Concept Design for Lunar Base Applications", NASA TM 102090 Sixth Symposium on Space Nuclear Power Systems, Albuquerque, NM, January 1989.

20. Schmitz, P. C. and Mason L. S., "Space Reactor/Stirling Cycle Systems for High Power Lunar Application", NASA TM 103698, Eighth Symposium on Space Nuclear Power Systems, January 1991.

21. English, R. E., "Power Generation From Nuclear Reactors in Aerospace Applications", NASA TM 83342, November 1982.

22. Mason, L. S., Schmitz, P. C. and Gallop, D. R., "Lunar In-Core Thermionic Nuclear Reactor Power System Conceptual Design", Proceedings of the Eighth Symposium on Space Nuclear Power Systems", Vol II, pp. 651-656, Institute for Space Nuclear Power Studies, Albuquerque, NM, January 1991.

23. Bohl, R. J., et. al., "Thermionic Fuel Element Verification Program Overview", Eighth Symposium on Space Nuclear Power Systems, Vol II pp. 636-639, Institute for Space Nuclear Power Studies Albuquerque, NM, January 1991.

24. Bloomfield, H. S., "Small Space Reactor Power Systems for Unmanned Solar System Exploration Missions", NASA TM 100228, December 1987.

25. Private Communication, Kelly, J. H., Jet Propulsion Laboratory, Pasadena, CA, July 1991.

TABLE I.—MISSION ELEMENTS AND SPECIFIED REQUIREMENTS

Mission element	Mission								
	LEVPU	Mining excavator	Regolith hauler	Pressurized rover		Unpressurized rover			LEV servicer
				Short range	Long range	Scientific telerobotic	Man transport	Recharge and emergency power	
Crew size:									
Maximum	1	1	1	4	4	4	4	4	0
Minimum	0	0	0	2	2	0	0	0	0
Capability:									
Payload lifting and hauling capacity, kg	10 000	750	750	(a)	(a)	1200	1200	1200	(a)
Average velocity, m/s	1	2	2	2.8	2.8	2.8	2.8	2.8	(a)
Maximum slope, deg	6	6	6	20	20	20	20	20	(a)
LEV payload mass allocation, kg	15 000	1000	1000	4500	6000	600	600	600	(a)
Power requirement, kWe:									
Peak	10	40	15	(a)	(a)	3	0.7	5	10
Nominal	3	22	3	7	12	2	0.3	5	10
Standby	3	10	1.5	3	(a)	(a)	(a)	5	10
Operation parameters per cycle, hr at–									
Peak power	1	1	1	(a)	(a)	16	336	(a)	8560
Nominal power	11	8.6	8	10	96	24	336	960	8560
Standby	0	1.4	1.4	0	(a)	0	(a)	0	8560
Inactive	12	13.6	13.6	14	48	0	(a)	0	0

[a]No specification.

Table II.
Performance Characteristics for Solar and DIPS Powered Mars Aircraft

	Solar Cell Eff., %		Radioisotope	
	14.2	25	Pu 238	Cm 244
Wing Area, M^2	336.00	118.72	145.00	103.00
Wing Span, M	108.00	47.50	48.20	37.97
Airframe, kg	384.53	161.64	213.70	161.87
Propulsion, kg	26.84	16.87	65.90	46.82
Solar Cells, kg	159.60	56.41		
Fuel Cell, kg	151.18	94.07		
Isotope, kg			140.80	13.76
Payload, kg	100.00	100.00	100.00	100.00
Total, kg	**1189.44**	**438.20**	**521.95**	**307.97**

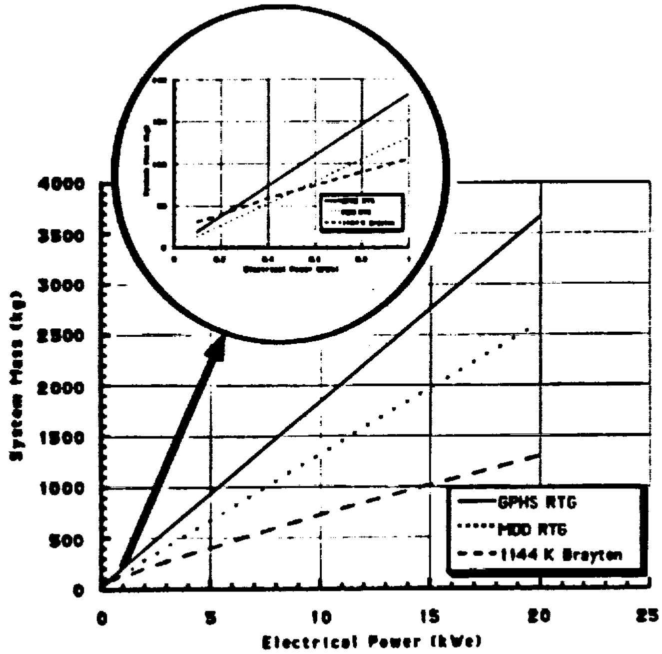

Figure 1. Comparison RTG and DIPS

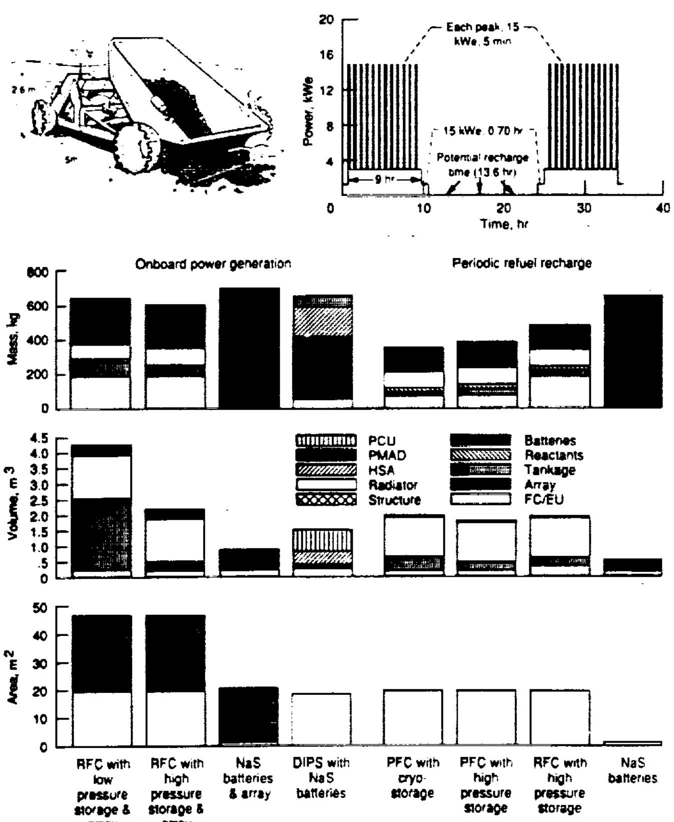

Figure 2. Regolith Hauler

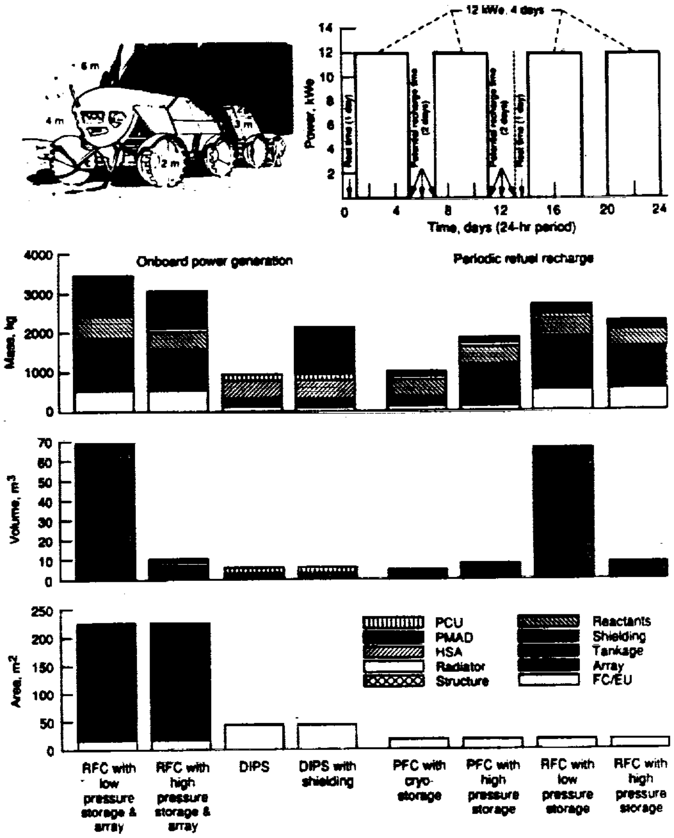

Figure 3. Pressurized Long Range Rover

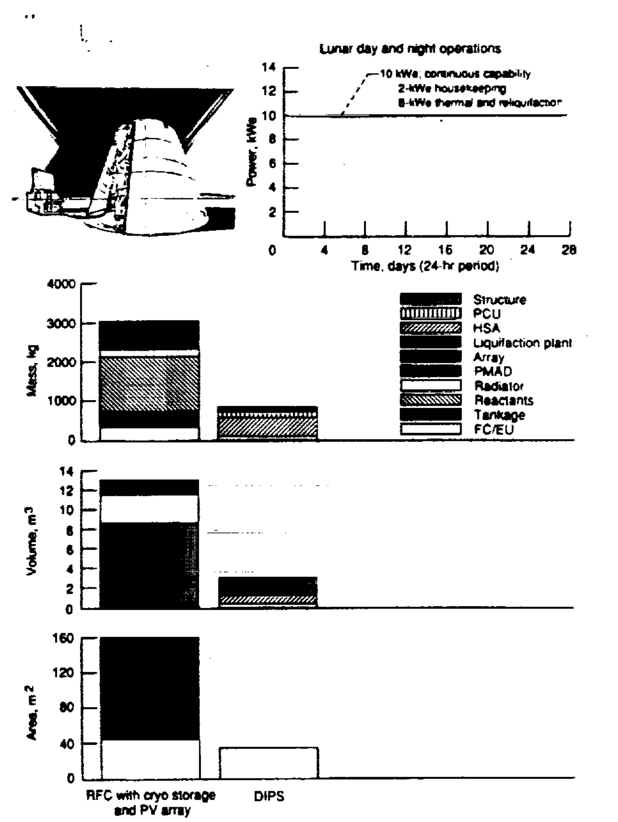

Figure 4. LEV Servicer

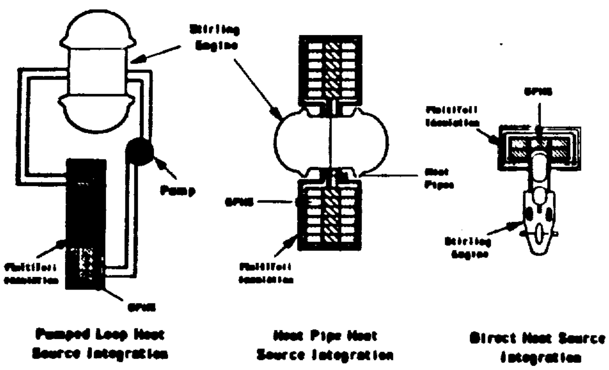

Figure 5. Free Piston Stirling DIPS

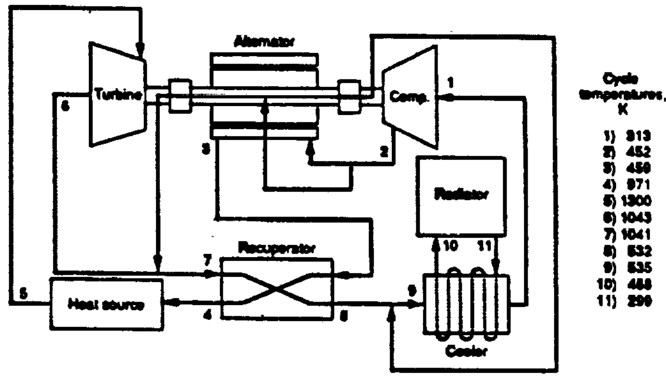

Figure 6. Brayton DIPS Schematic

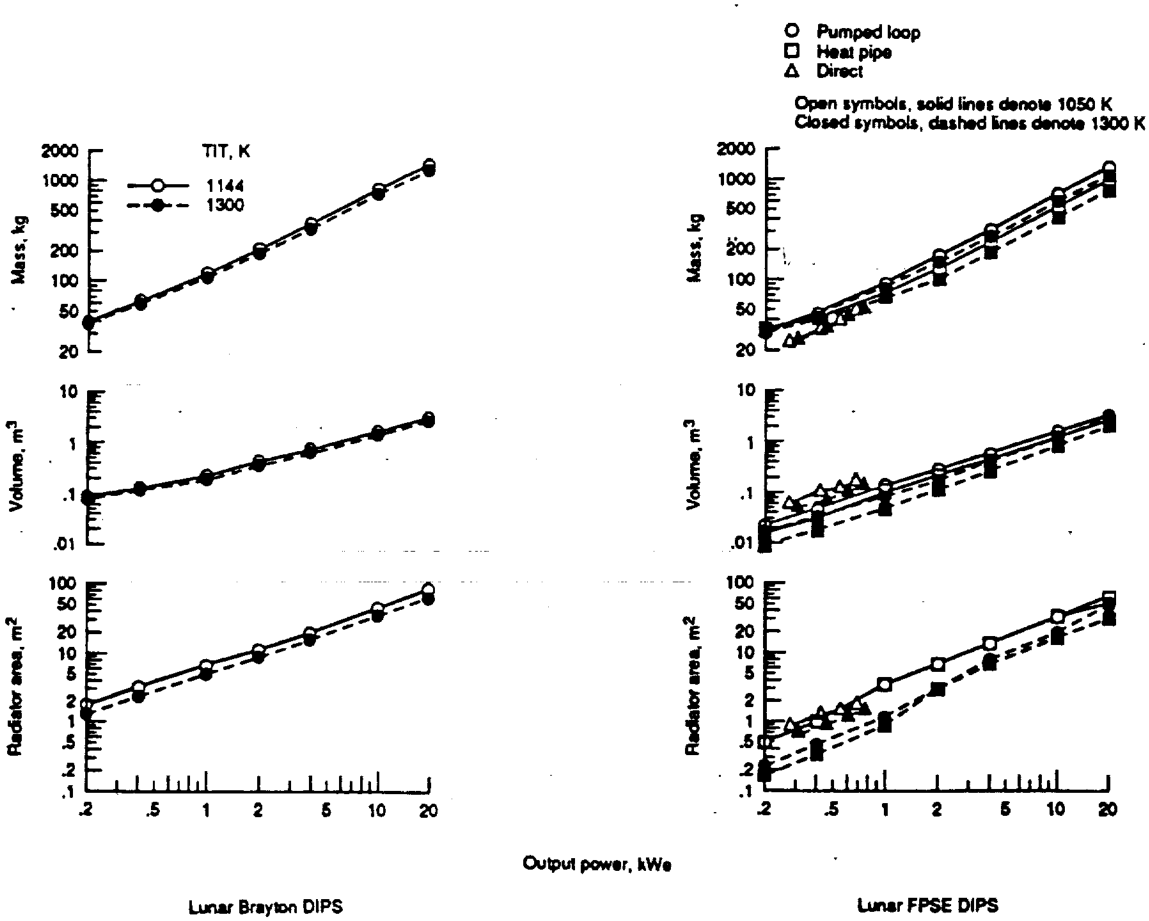

Figure 7.
Comparison of Brayton and FPSE
Performance Characteristics

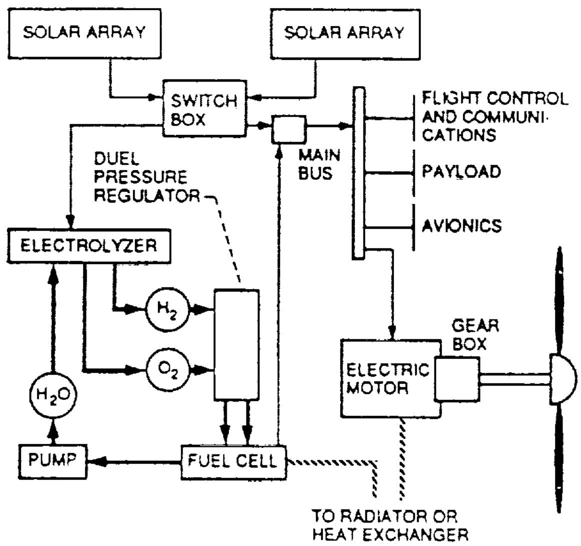

Figure 8. Solar Powered Aircraft Power and Propulsion System Diagram

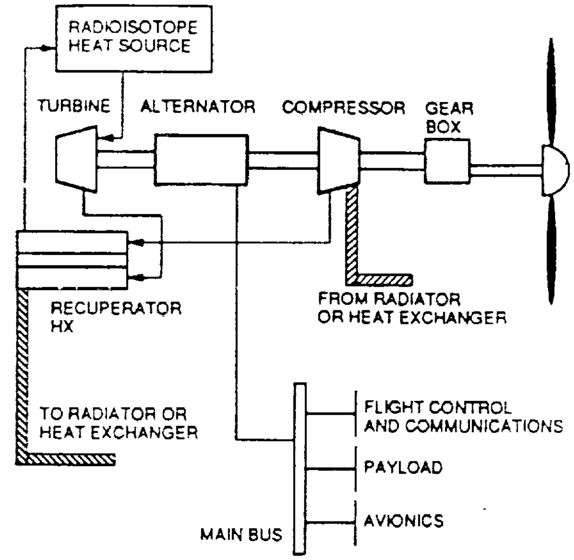

Figure 9. Radioisotope Powered Aircraft Power and Propulsion System Diagram

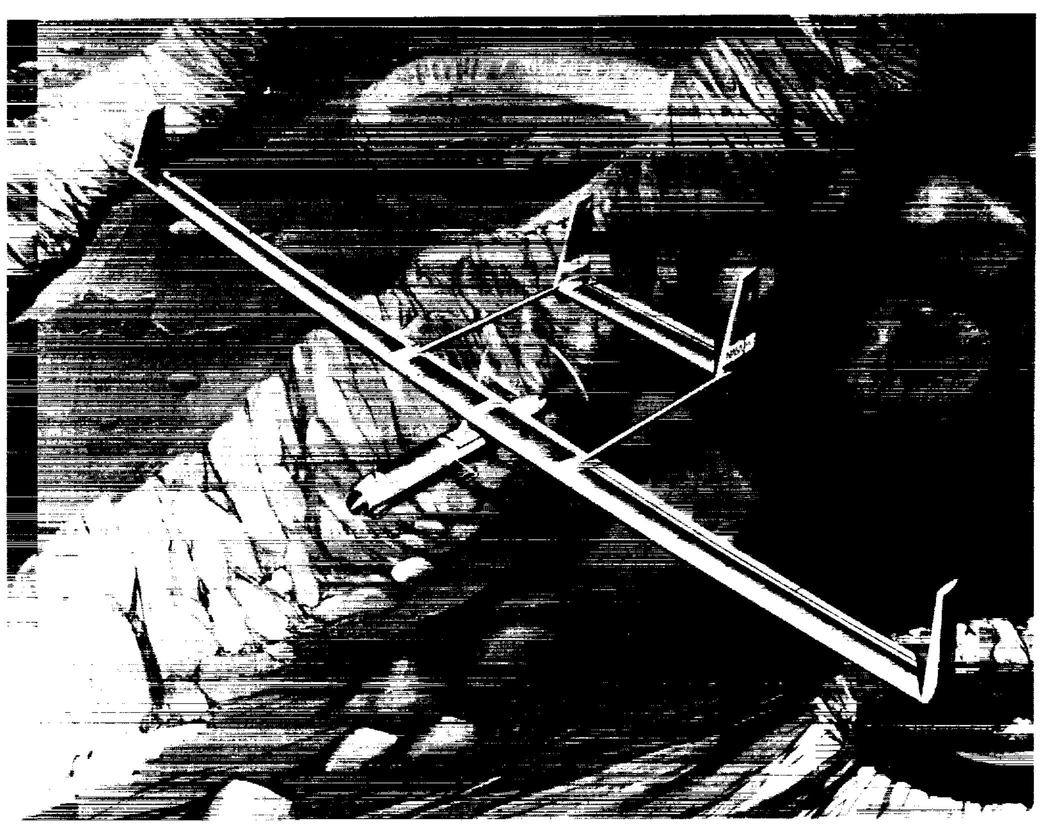

Figure 10. Artist's Conception of Solar Powered Mars Aircraft

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
		Technical Memorandum

4. TITLE AND SUBTITLE

Trade Studies for Nuclear Space Power Systems

5. FUNDING NUMBERS

WU-590-13

6. AUTHOR(S)

John M. Smith, David J. Bents, and Harvey S. Bloomfield

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135-3191

8. PERFORMING ORGANIZATION REPORT NUMBER

E-6554

9. SPONSORING/MONITORING AGENCY NAMES(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Washington, D.C. 20546-0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

NASA TM-105231
AIAA-91-3518

11. SUPPLEMENTARY NOTES

Prepared for the Conference on Advanced Space Exploration Initiative Technologies cosponsored by AIAA, NASA, and OAI, Cleveland, Ohio, September 4-6, 1991. Responsible person, John M. Smith, (216) 433-6130.

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Unclassified - Unlimited
Subject Category 44

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

As human visions of space applications expand and as we probe further and further out into the universe, our needs for power will also expand, and missions will evolve which are enabled by nuclear power. A broad spectrum of missions which are enhanced or enabled by nuclear power sources have been defined. These include earth orbital platforms, deep space platforms, planetary exploration and extraterrestrial resource exploration. The recently proposed Space Exploration Initiative (SEI) to the Moon and Mars has more clearly defined these missions and their power requirements. This paper presents results of recent studies of radioisotope and nuclear reactor energy sources combined with various energy conversion devices for earth orbital applications, SEI lunar/Mars rover and surface power, and planetary exploration.

14. SUBJECT TERMS

SP-100; Nuclear space power; Lunar power; Mars power

15. NUMBER OF PAGES

14

16. PRICE CODE

A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	