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POWER SYSTEMS FOR FUTURE MISSIONS

APPENDICES

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POWER SYSTEMS FOR FUTURE MISSIONS

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APPENDIX A

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INTRODUCTION

The following technology development plan discusses the reference power system design for the U.S. Department of Energy Dynamic Isotope Power System program and describes the development program needed to deploy this system as a primary source of power for lunar and Mars surface exploration missions. It specifically addresses the development tasks required to deploy a modular 2.5 kWe DIPS for the First Lunar Outpost mission (FLO), in 1999 and to design the DIPS module to be compatible with the Martian surface environment.

DIPS technology is suitable for use in both fixed and mobile power applications for the lunar or Martian surface as well as a reliable power source for space satellites. For fixed bases and mobile surface power system applications, the current DIPS program has focused on a standard power module design of 2.5 kWe. Surface power systems in the range of 1 kWe to 20 kWe can be developed from this standard module design.

A variety of potential DIPS remote or mobile applications have been identified by the National Aeronautics and Space Administration (NASA). These applications include remote power to science packages, surface rovers for both short and extended duration missions, and backup to central base power (Ref. A-1). For the scenarios and applications associated with the Space Exploration Initiative (SEI), a trade study was conducted which resulted in the selection of a standard power module design as the preferred approach (Ref. A-2). A 2.5 kWe power level was identified as the optimum module size. This trade study evaluated various cycle design options, turbine inlet temperature effects, technology readiness levels, development time, as well as overall power system costs, including delivery and support on the Moon and Mars. The 2.5 kWe power module approach had overall cost, schedule and technical advantages over application specific designs.

The use of multiple modular power units to supply power needs has many advantages. Modular units permit the development of a single-size module reducing development costs and it improving the power availability factor for most applications. These modular units are replaceable if a unit failure occurs. The fuel is simply removed from the failed unit and reused in a replacement unit providing a 100% fuel utilization factor with the use of non-fueled spares. For mobile units, sufficient capacity would be installed such that if one power module were to fail, the mobile unit could return to base.

The modular DIPS design requirements include a 15 year continuous lifetime. The fuel handling canister is designed for a 45 year life thereby allowing for the reuse of the fuel in future unfueled DIPS replacement modules. The fuel is transported only once to the destination and unfuel spare modules are sent as needed for the mission. After the initial 15 year lifetime, the output power level would decrease at the rate of 0.8% per year down to 2.2 kWe at 30 years if no improvements were included in the future unfueled DIPS modules.

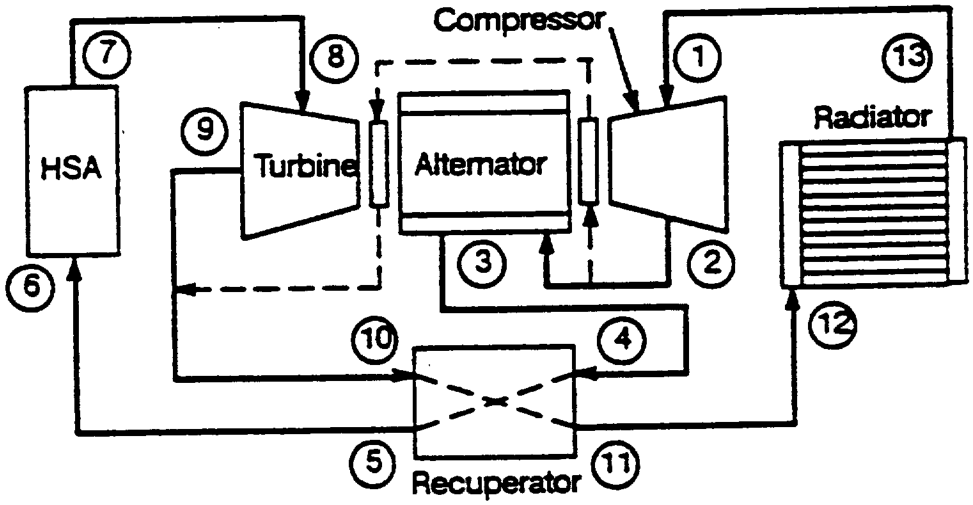
The DIPS system is required to operate in the rigors of the Martian environment as well as the lunar surface environment. This design challenge was accomplished by confining the use of refractory alloy materials to components located within the sealed heat source unit and utilizing protective coatings and vacuum jackets where refractory alloy materials might be exposed to the Martian environment even for short times during fuel canister transfer. The turbine inlet temperature for the working fluid was limited to 1133 °K (1579 °F) to insure that the gas containment boundary is totally constructed of nonrefractory alloy materials. Early in the trade studies, it was determined that higher temperature systems did not provide a significant improvement in system performance. This design change eliminated the need for high temperature dissimilar metal joints in the gas containment boundary and resulted in an all welded joint construction for the DIPS with a significant lowering of development costs and a faster development schedule. In addition, simplification of the system design was achieved by using a gas cooled permanent magnet alternator design which reduced the number of moving parts to one and simplified the method of voltage control. This design change resulted in a significant improvement in the overall system reliability, and provided additional improvements in unit cost, system efficiency and specific mass.

CONCEPT DESCRIPTION

The DIPS uses the decay of radioactive plutonium 238 as the source of heat and a closed Brayton cycle (CBC) power conversion cycle to convert this heat to electrical power. The CBC DIPS cycle diagram is illustrated in Figure A-1. The CBC uses an inert gas working fluid (helium-xenon mixture) which is heated by the radioactive heat source and then expanded through a turbine to convert heat energy to mechanical energy. From the turbine, the working fluid passes through a recuperator to recover heat and improve cycle efficiency. The waste heat from the cycle is then rejected through a gas tube and fin radiator assembly. From the radiator, the working fluid is compressed to the peak cycle pressure and then used to cool the alternator. The working fluid again passes through the recuperator for preheat before returning to the heat source.

The DIPS working fluid is contained within a loop that is hermetically sealed containing all full penetration inspectable welds. There are no valves or contacting mechanical parts that could cause wear and limit lifetime. The turboalternator permanent magnet rotor is suspended on noncontacting working fluid gas foil bearings, and there are no contacting seals or brushes.

Figure A-2 shows a conceptual layout of the system associated with a cart to provide mobility. The heat source units (HSUs) are located under the radiators and include fuel handling canisters that contain multiple General Purpose Heat Source (GPHS) modules. Figure A-3 shows an example of a typical HSU. The HSUs contain a reversible heat removal system (RHRS) that allows the radioisotope heat to be dissipated to space in the event the power conversion cycle is not operating.



No.	Stream	Temperature (K)	Pressure (kPa)	Flow (kg/s)
1	Compressor inlet	360.83	321.50	0.2007
2	Compressor discharge	469.23	538.55	0.2007
3	Alternator discharge	475.39	538.55	0.1987
4	HP recuperator inlet	475.39	536.69	0.1987
5	HP recuperator outlet	929.87	534.97	0.1987
6	HSA inlet	926.52	533.86	0.1987
7	HSA outlet	1136.67	531.59	0.1987
8	Turbine inlet	1133.31	530.49	0.1987
9	Turbine outlet	960.97	327.78	0.1987
10	LP recuperator inlet	953.79	327.16	0.2008
11	LP recuperator outlet	503.90	325.57	0.2008
12	Radiator inlet	503.90	324.88	0.2008
13	Radiator outlet	360.83	322.81	0.2008

Figure A-1. - CBC DIPS cycle diagram.

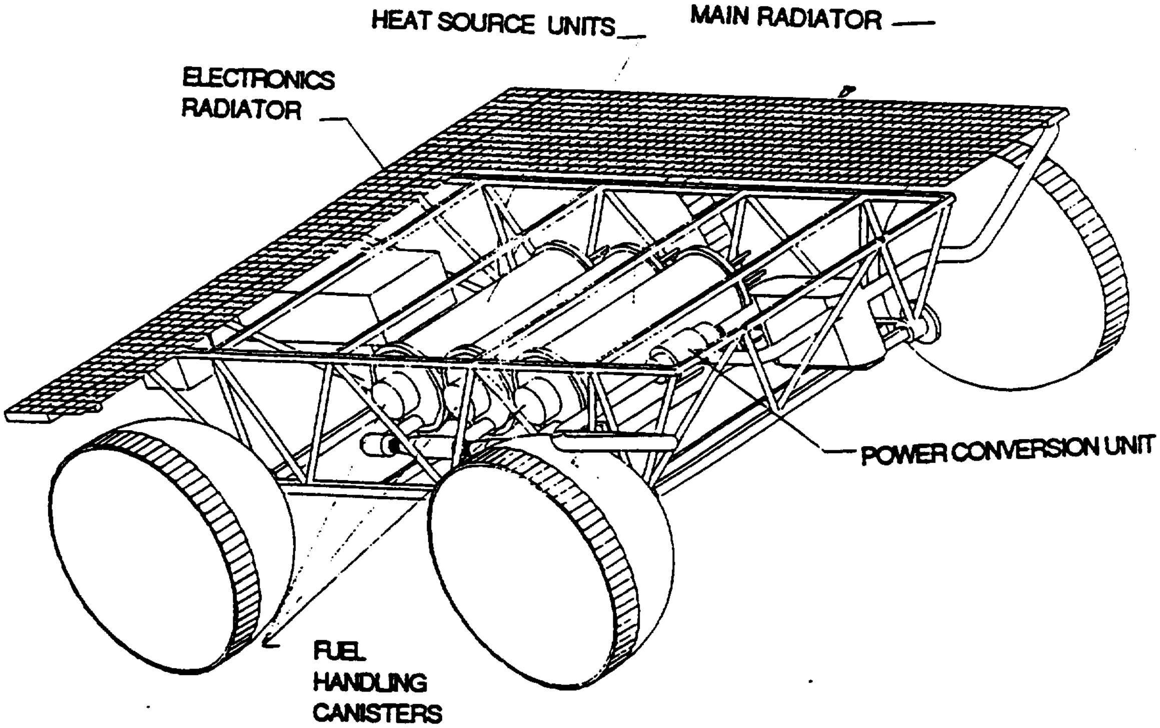


Figure A-2. Conceptual design of 2.5 kWe modular CBC DIPS power cart.

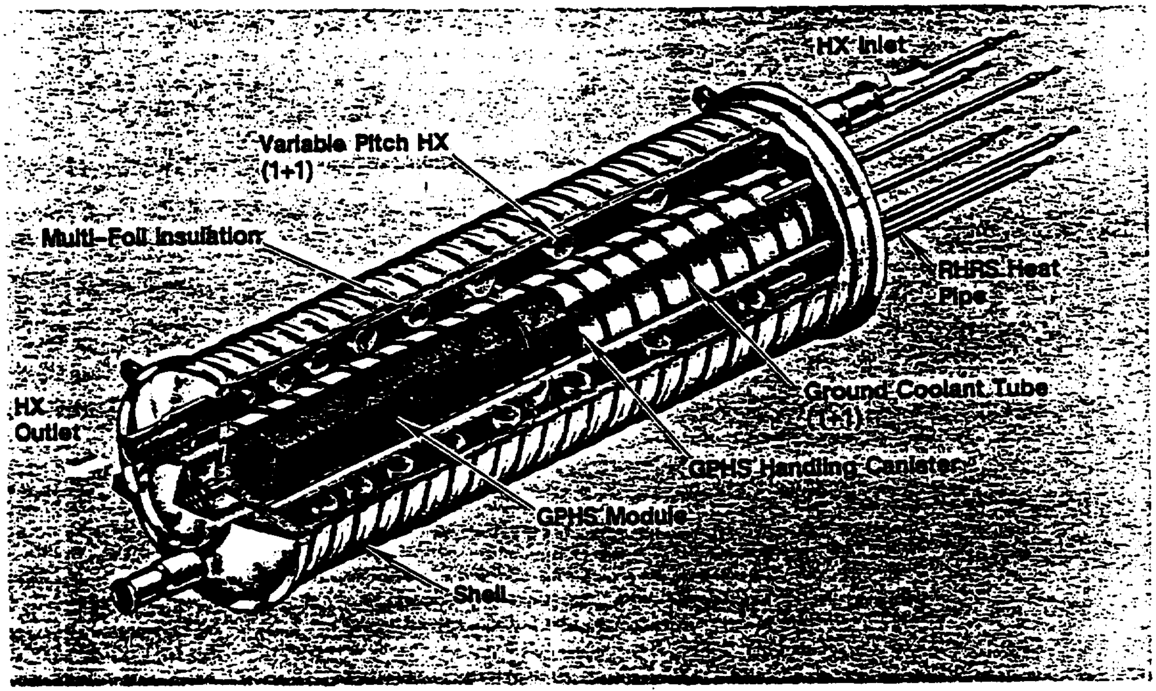


Figure A-3. 2.5 kWe DIPS HSU.

Typically, the DIPS heat source assembly consists of three heat source units (HSUs) plumbed together in parallel. This parallel piping connection provides a low pressure drop heat exchanger design. Each HSU fuel handling canister contains multiple GPHS modules and is removable from the HSU housing. The fuel handling canister is fabricated of Nb-1Zr which is coated with titanium diboride for emissivity control and oxidation protection from the Martian atmosphere. This coating must provide protection only while the canisters are being transferred from their transport and storage rack units to each HSU housing which would require less than one hour. The oxidation environment within the canister is minimized by providing a niobium wire wool getter to absorb the CO_2 diffusing through the helium release membrane and a niobium wire wrap at the slip joint end to "getter" any oxidizing atmosphere within the vacuum liner to canister annulus. A perfluoroelastomer O-ring seals the annulus to minimize CO_2 ingress. The canister materials and design were chosen to provide a viable concept able to meet the lifetime goal for either lunar or Martian missions. The getters would not be required for lunar surface operations.

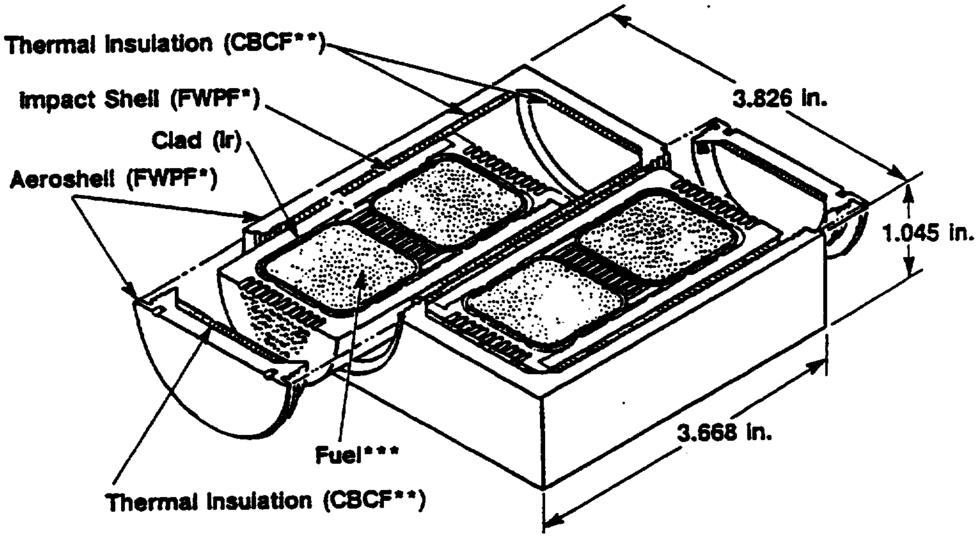
The HSU is composed of a multiple coil, helical pitch heat exchanger, multifoil insulation (MFI), MFI vacuum liner, reversible heat removal system (RHRS), and an outer shell. The heat exchanger will be fabricated from Inconel 617 with its outer surface coated with titanium diboride for emissivity control. Since the heat exchanger will be in a vacuum container, the emissive coating must only be able to accommodate the time-temperature intermetallic diffusion.

Backup cooling for the HSU is provided in the event the gas cooling loop is inactive. Variable conductance heat pipes provide a RHRS which can safely reject all the isotope heat at slightly above normal operating temperatures after the turbine gas flow stops.

The RHRS variable conductance heat pipes are provided to insure that the HSU component temperatures remain at acceptable levels before system start-up or during a temporary shutdown. They reject very little heat during normal operation of the DIPS. The heat pipes selected for this application use lithium as the working fluid. Neon is added as a noncondensible gas to limit heat loss at operating temperatures. The heat pipe containment boundary and wick will be made of Nb-1%Zr. As in the other internal HSU components, TiB_2 is used as an emissivity coating. For compatibility with the Martian environment, the condenser and gas reservoir sections are covered with a pre-oxidized Inconel 617 vacuum liner which protects the refractory metal from attack by the CO_2 rich Martian atmosphere. The vacuum enclosure is not necessary for use on the lunar surface and can be removed. In either application the entire heat pipe sees only vacuum conditions.

In the unlikely event that the gas loop and all the RHRS heat pipes are inoperative, the isotope fuel clad temperature is still maintained in a safe level due to the HSU meltable MFI insulation package. The MFI consists of 130 layers of foil which are designed to melt and provide a direct cooling path to space before damage to the fuel cladding occurs after loss of all other cooling. There are 80 layers of 0.0005 cm thick niobium surrounded by 50 layers of 0.00086 cm thick nickel, all separated by yttria particles. To assure integrity in the Martian environment, the insulation is contained within an evacuated chamber enveloped by the shell, liner, and end caps. Although not essential for lunar applications, the MFI vacuum liner provides for lunar/Mars compatibility with a mass penalty of less than 1.5 kg.

The GPHS module developed by the U.S. Department of Energy serves as the isotopic heat source in DIPS. The design of the GPHS module is shown in Figure A-4. There are 17 GPHS modules per HSU in the current 2.5 kWe modular DIPS design.



* Fine-Weave Pierced Fabric, a 90%-dense 3D carbon-carbon composite
** Carbon-Bonded Carbon Fibers, a 10%-dense high-temperature insulator
*** 62.5-W $^{238}PuO_2$ pellet (4 required)

Figure A-4. GPHS Module (250 W) Sectioned at Mid-Plane.

The DIPS Power Conversion Unit (PCU) is shown in Figure A-5. The PCU consists of a turboalternator compressor (TAC), recuperator, and interconnecting ducting. The high temperature interconnect ducting is made of Inconel and the low temperature ducting of aluminum or stainless steel. The TAC consists of a turbine, alternator and compressor mounted on a single common shaft supported by radial and axial foil gas bearings. A typical cross section of the TAC is shown in Figure A-6. The 1133 °K turbine inlet temperature is within demonstrated CBC technology for long-duration mission requirements using conventional super alloys. The top half of the figure shows a TAC cross-section for the reference turbine inlet temperature. The bottom half shows a advanced double wall scroll design for a 1300 °K turbine inlet temperature system. Significant design features of the TAC are called out on the figure. The turbine, alternator, and compressor are all located on a single solid shaft. The shaft, or rotating unit, is supported on hydrodynamic gas foil bearings so there are no rubbing parts to wear out. The TAC and its entire gas containment boundary are hermetically sealed with all joints fully inspectable per visual, dye penetrant, magnetic particle, and x-ray methods.

The TAC employs a two pole toothless (TPTL) permanent magnet generator (PMG). The TPTL PMG design consists of a permanent (Samarium-Cobalt) magnet encased in a sleeved rotor, as shown in the figure. The permanent magnet field eliminates the need for field coils and their associated losses. The high strength field results in the elimination of the stator teeth and their resultant weight and pole face loss penalties. The TAC design provides improved efficiency, lower cooling requirements, and lower unit weight when compared to the Rice alternator machine in this power range.

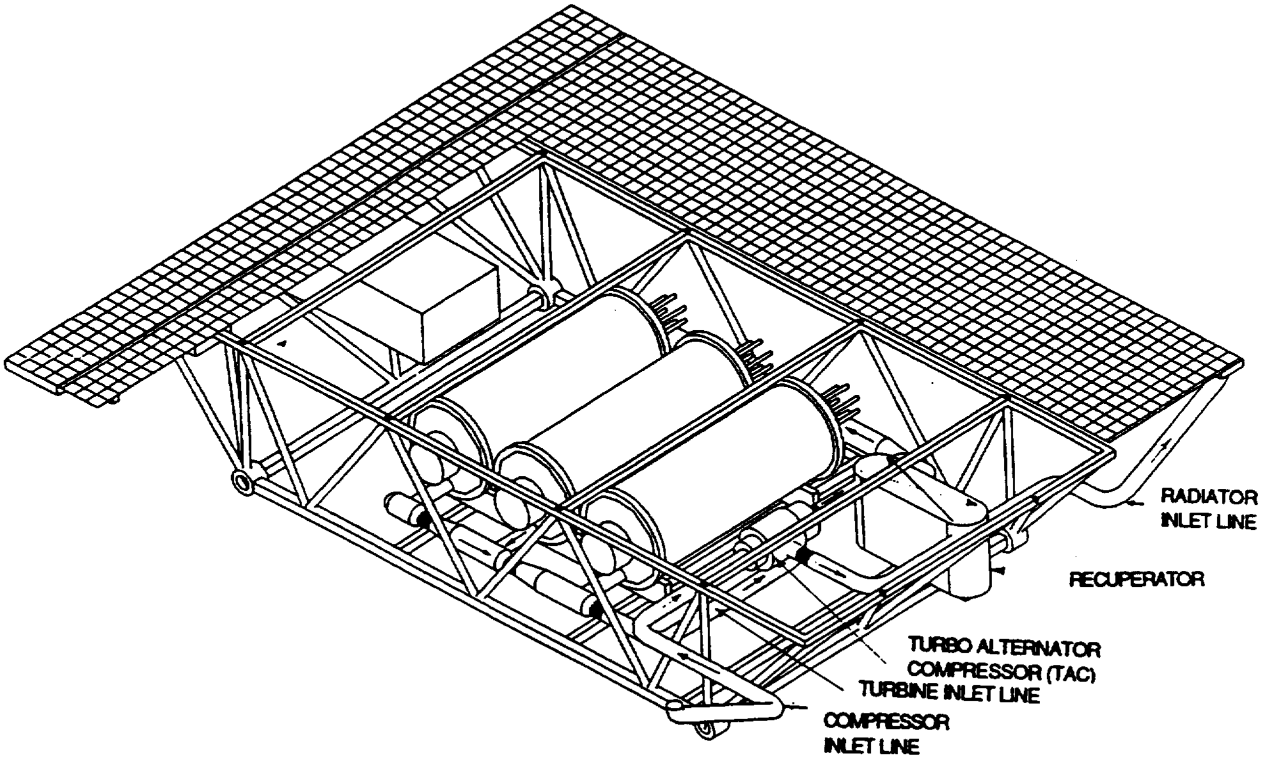


Figure A-5. Modular CBC DIPS PCU components.

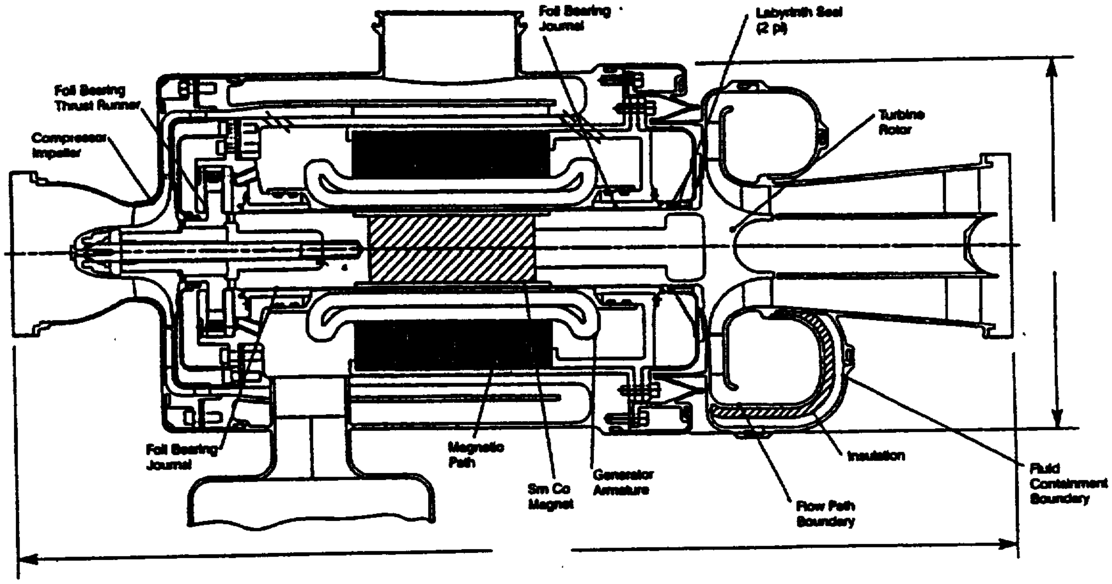


Figure A-6. CBC DIPS TPTL gas cooled TAC design.

A counterflow heat exchanger called a recuperator is used to increase PCU cycle efficiency. The recuperator consists of a compact, brazed, plate-fin heat exchanger core, and inlet and outlet fluid manifolds. The core matrix consists of a brazed assembly of rectangular offset fin counterflow sections These sections are composed of alternate layers of high pressure (HP) He-Xe side and low pressure He-Xe side fins separated by metal plates. Figure A-7 illustrates the favored recuperator structure that uses the plate-fin sandwiches stacked to form the heat exchanger core with integral manifolds. This concept of construction proved very successful in the BIPS program. When stacked and brazed the tube-sheet forms (1) the flow separation plate in the offset fin matrix, (2) the first flow boundary of the end sections and sidewalls, and (3) the inlet and discharge flow plena. This type of construction has the benefit of totally eliminating thick-to-thin stress risers and weld-over-braze assembly requirements.

This design has proven to be both rugged and predictable as demonstrated in highly accelerated cyclic life testing conducted in support of the BIPS program.

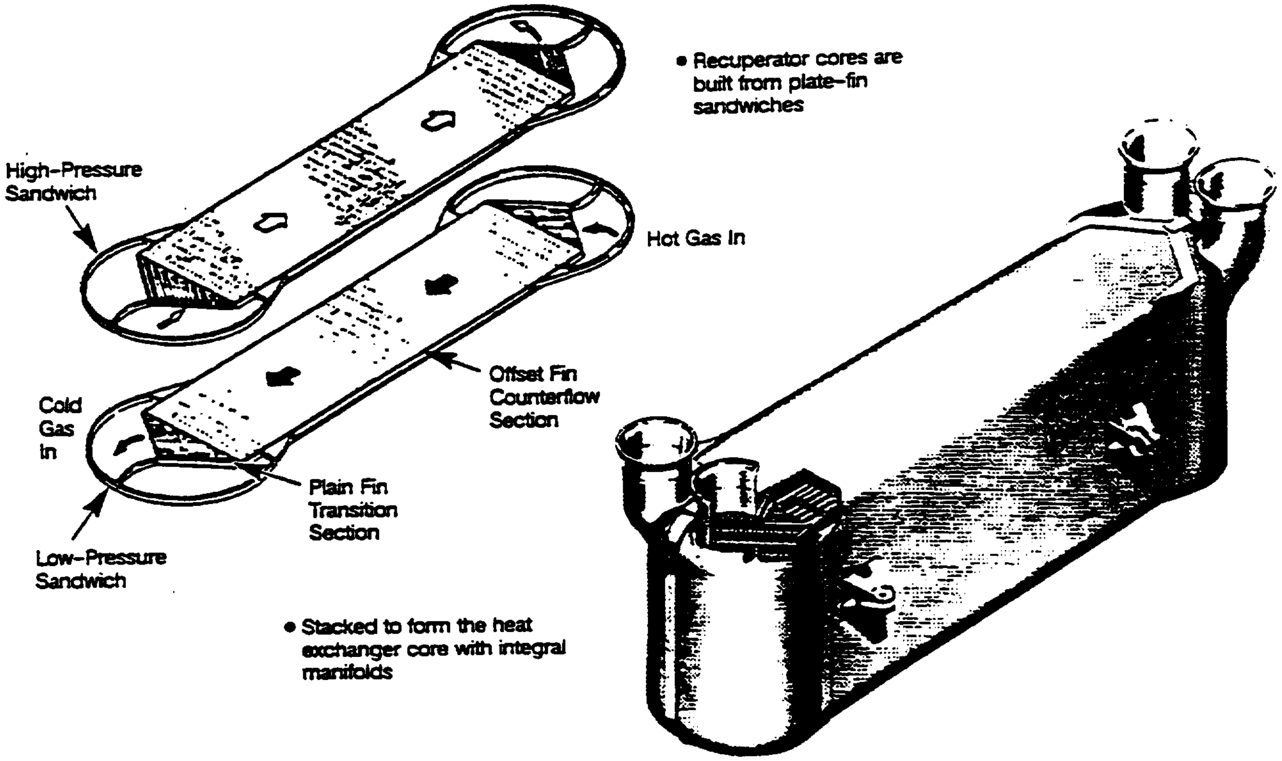


Figure A-7. CBC DIPS recuperator.

The main heat rejection radiator uses a conventional aluminum tube and fin assembly as shown in Figure A-8. The low pressure gas leaving the recuperator is cooled by passing through multiple parallel cooling tubes connected between inlet and outlet gas headers. The cooling tubes use highly efficient fins to develop radiating surface area. In combination with tube armor, these fins also serve as bumper armor to protect the gas tubes from meteoroid punctures. The gas tubes, headers, and fins are all constructed from aluminum alloys and all welds are fully inspectable. The finned radiating surface is covered with optical solar reflector (OSR) tiles to enhance the emissivity while limiting the absorption of solar energy (ε=0.8, α_s=0.08). These OSR tiles provide an effective sink temperature of 220K under lunar noon conditions for the DIPS radiators. Radiator surface coatings, such as Z93 (ε_s=0.9, α_s=0.3) were rejected for the DIPS application due to their higher solar energy absorption coefficient. The use of paint on the DIPS radiator surface would raise the effective sink temperature for the DIPS by 56 °K requiring a larger radiator surface area even with its improved thermal emissivity value of 0.9. Since the radiator surface must always be exposed to the environment, degradation of this coating with exposure time which appears as primarily an increase in solar energy absorptivity, would further raise the effective sink temperature and degrade the DIPS cycle efficiency. The effects of lunar and Martian dust particles on radiating surface properties will be established during DIPS radiator tests.

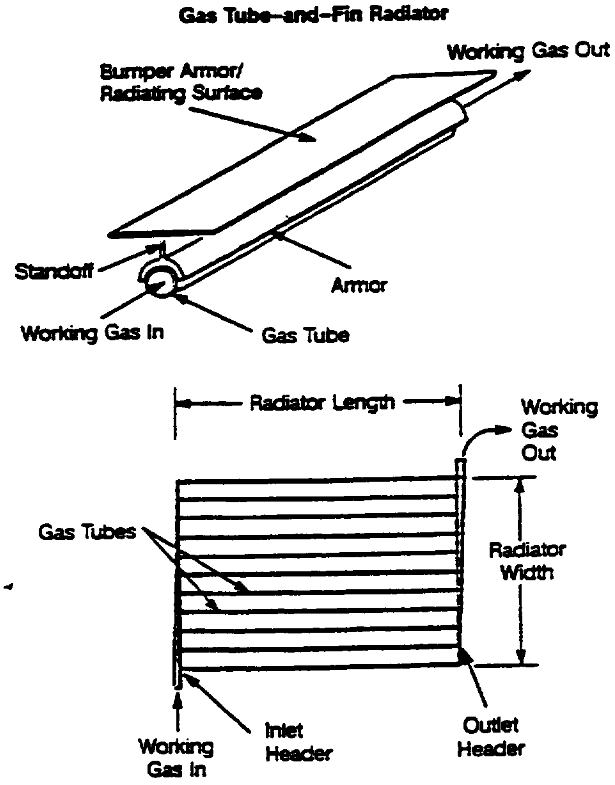


Figure A-8. Gas tube-and-fin radiator assembly.

In the current design, the DIPS power processing and control (PP&C) assembly consists of fully redundant power conditioning and control unit (PCCU) channels, a startup battery and battery control assembly, and the DIPS controller. The PCCU rectifies the ac power provided by the alternator and regulates the resulting 120 Vdc power output. This voltage was selected to match the SSF electrical power subsystem architecture since it is anticipated that any future lunar outpost would share some features from the SSF EPS as a means of providing cost savings on future NASA missions (if it is determined that for some applications, that ac power is preferable, the PCCU can be designed to deliver three phase ac power to the user bus). The PCCU also supplies DIPS power needs during alternator start up. The DIPS controller monitors and controls the PCCU operation and provides a communications bus for external data transfer and control of the system. The PP&C architecture is illustrated in Figure A-9.

Each PCCU channel includes a switching rectifier, inverter, speed regulator, filter, battery control assembly, battery, and switching assembly. During normal operation, power conditioning is accomplished in three steps utilizing the switching rectifier, speed regulator, and filter. The active PCCU channel and operating configuration are designated by using the switches comprising the switch assembly.

Energy storage requirements are met by the battery assembly. It performs two functions: (1) it furnishes instrumentation and control power prior to the DIPS alternator startup, and (2) it provides the power needed to motor the alternator up to its self-sustaining speed. The battery assembly contains a battery control assembly and a battery consisting of eighty 2 amp-hour cells. The battery control assembly contains a voltage converter and a battery monitoring unit. The converter uses a boost regulator to control the battery charging rate. The battery monitoring unit controls the battery temperature and monitors its operating status.

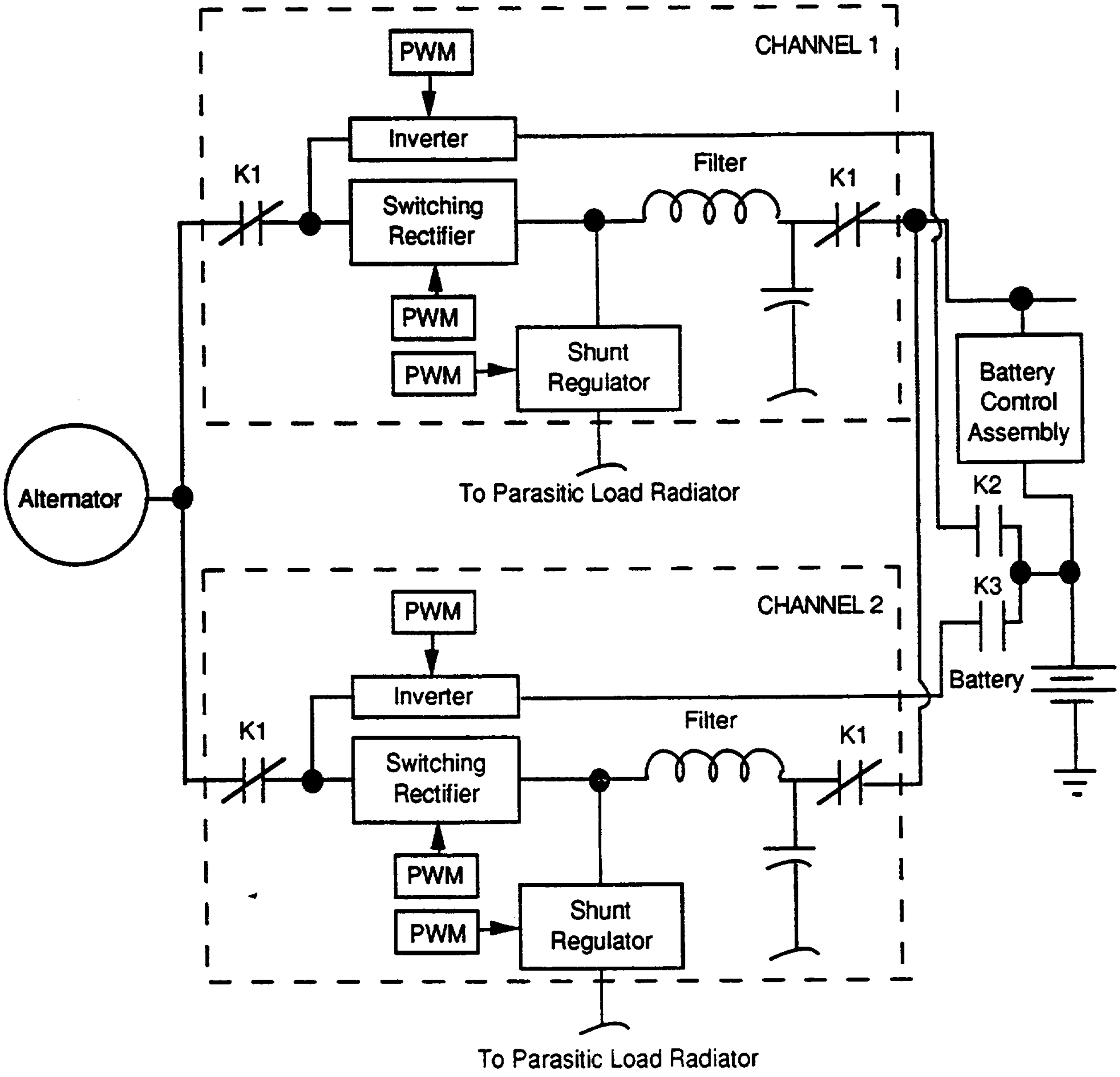


Figure A-9. DIPS PP&C architecture diagram.

KEY DEVELOPMENT ISSUES

The key issues for development of the baseline 1133 °K (1579 °F) DIPS concept and their system impacts are summarized in Table A-1.

TABLE A-1. DIPS TECHNOLOGY ISSUES, IMPACTS, AND DEVELOPMENT AREAS

Issue	Impact	Potential Development Areas
Isotope Cooling/ Nuclear Safety	•Active cooling during launch and flight •Passive emergency cooling	•High emissivity coatings •RHRS heat pipes •Meltable MFI package
Lunar/Mars Environment	•Refractory metal lifetime •HSA mass increase	•Coatings, getters, semi-permeable seals, dust protection, OSRs
Shock Loading	•Bearing lifetime •Reconfiguration of RHRS & electronics cooling radiator heat pipes	•Gas-foil bearings performance •Heat pipe design and verification testing
Alternator Temperature	•Alternator mass and radiator area •Cooling system complexity	•High temperature alternator insulation
Isotope Handling & Disposal	•Added mass for biological shielding •Added cost for non-recoverable isotope	•Fuel handling canister and tools •Launch and transport containers
Recuperator Heat Transfer Performance	•System efficiency, mass, and radiator area	•High performance laminar flow recuperator designs
Gas Leakage	•Life and reliability	•Full-penetration inspectable welded boundaries •Low-temperature dissimilar metal transition joints •Meteoroid protection

TECHNOLOGY ASSESSMENT

The DIPS Demonstration Program is focused on advanced technology development of dynamic space power systems to support the early needs of the SEI and other space exploration endeavors. By advancing the existing technology, the DIPS program provides a highly innovative and leveraged technology for the cost effective use of radioisotopes in space. The Rocketdyne Division of Rockwell International is the DIPS program leader with Allied Signal Aerospace Company's Garrett Fluid Systems Division, Teledyne Energy Systems, Thermoelectron Technologies Corporation, Physics International, and SCI Government Systems making up the other members of the DIPS team.

The DIPS is based on and utilizes the flight qualified, plutonium fueled GPHS modules currently being flown in the Galileo and Ulysses radioisotope thermoelectric generators (RTGs). The CBC power conversion technology has been demonstrated in prior programs such as DOE's Brayton Isotope Power System (BIPS) program and NASA's Brayton Rotating Unit (BRU) demonstration program, and is based on extensive experience with similar hardware in aircraft applications.

Current and nearterm component technology was chosen for DIPS to provide an opportunity for early deployment of this system with minimal development risk. As part of this technology evaluation, the technology bases were assessed for the following major DIPS assemblies:

- GPHS modules;
- HSU;
- TAC assembly;
- recuperator;
- radiator;
- ducting and bellows; and
- PP&C.

These evaluations are summarized in Table A-2 which shows that the 1133 °K (1579 °F) CBC DIPS has technology readiness levels ranging from 4 to 9, depending on the particular assembly. The technology base for each assembly is briefly discussed in the following sections.

GPHS Module State-of-the-Art

The technology base for the GPHS module is extensive and consists of the following items:

- materials, properties, and module performance characteristics have been well established in the government sponsored development program;

- manufacturing and quality control programs have been demonstrated with production of flight qualified modules for the Galileo and Ulysses programs;
- safety issues have been resolved in conjunction with the Galileo and Ulysses flights (both used RTGs with GPHSs as the heat source); and
- both the Galileo and Ulysses spacecraft have been launched and are operational.

TABLE A-2. 1133 °K (1579 °F) CBC DIPS TECHNOLOGY ASSESSMENT

Subsystem	Technology Readiness Level	Comments
GPHS modules	9	Successfully flown on Galileo and Ulysses missions
HSU; RHRS HSU; MFI HSU; Gas Containment	4	Laboratory demonstration Subscale tests complete Inconel 617 boundary; well established data base
TAC assembly	5	Similar to BIPS and BRU turboalternator-compressor designs but with high temperature alternator electrical insulation and an all welded, fully inspectable gas containment boundary
Recuperator	6	Successful closed-loop experience - BRU and BIPS system tests
Radiator	6	Aluminum tube and fin radiator design similar to STS radiator technology and SSF radiators
Interconnect ducting	5	Conventional high-temperature materials; low-temperature dissimilar materials joint; conventional materials bellows design with fully inspectable welds
PP&C	5	Space Station Freedom Electrical Power Subsystem component (SSF-EPS) technology

<u>HSU State-of-the-Art</u>

The fuel handling canister, the RHRS heat pipes, and internal HSU vacuum liners are fabricated from Nb-1%Zr. Materials properties for Nb-1%Zr are well known and have been qualified in the SP-100 program. Silicide or titanium diboride emissivity/oxidation protective coatings are proposed for the canister and liner surfaces. These coatings are known to resist oxygen attack at temperatures exceeding the design temperatures for extended periods of time (years). The coating protection is only required in the Martian environment during transfer of a fuel handling canister between HSUs and storage rack units (<1 hour per transfer). However, accelerated testing would be needed to establish the variables involved and confirm their application over the equivalent fifteen year lifetime. Long term stability of the coating in this application remains to be established.

Explosively bonded transition joints are used between the Nb-1%Zr and Inconel components for several joints within the HSU. Specifically, these joints are located in the HSU vacuum containment used for Mars applications and are not part of the HeXe pressure boundary. This is a well established fabrication technique. The joints are located in low temperature areas where intergranular effects are not expected to be an issue over the 15 year operational life. However, joint leak tightness and intergranular effects will need to be verified.

Lithium heat pipes, like those to be used for the RHRS, have a substantial data base as shown in Table A-3 (Ref. A-3), but the specific DIPS design is different than those comprising the data base. For this reason, a demonstration model was fabricated and tested at LANL, with successful results (Ref. A-4). At 1150 °K (1610 °F), the test heat pipe removed approximately 750 W of heat, which is in excess of the 710W target for the variable conductance heat pipe design. For the present DIPS RHRS design, the VCHP "on" temperature will be set at 1194 °K (1690 °F) by adjusting the neon gas inventory in the heat pipe.

The HSU uses multifoil insulation, not only to control heat losses from the HSU, but also to provide emergency cooling by melting and providing a direct heat path to space at a slightly higher than normal fuel cladding temperature. On the BIPS program, various combinations of multifoil insulation were subjected to subscale testing. The general conclusion that can be drawn from these tests is that the principle of progressive eutectic melting is valid. The tests also provide confidence that the meltdown characteristics of particular foil combinations can be accurately predicted.

TAC Assembly State-of-the-Art

The TAC design is similar to the 10 kWe BRU and 1.3 kWe mini-BRU units designed and tested by the Garrett Corporation in the 1970s.

The NASA BRU series of four units, with a flight configured recuperator, cooler, and ducting were tested for a total of 52,573 hours, most of which was at a turbine inlet temperature of 1144 °K (1599 °F). One of these units accumulated 41,000 hours, including a 13,600 hour continuous run. The BRU used pivoted pad bearings instead of foil bearings because foil bearing technology was not well developed at that time. Since then, the foil bearing has been used extensively in commercial aircraft auxiliary power units (APUs) for millions of hours and continuous improvements have been made to the bearing design. A summary of the APU foil gas bearing experience is given in Table A-4.

TABLE A-3. LITHIUM HEAT PIPE OPERATING EXPERIENCE

No.	Wall Material	Test Temperature (°K/°F)	Hours of Operation	Remarks
1	CVD-W	1000/1340	1000	
2	W-26Re	1000/1340	10000	
3	TZM	1500/2240	4600	Evaporator leak
4	TZM	1500/2240	10526	Weld leak
5	TZM	1500/2240	10400	Weld failure in end cap
6	TZM	1500/2240	9800	Weld failure in end cap
7	Nb-1Zr	1000/1340	132	
8	Nb-1Zr	1500/2240	9000	
9	Nb-1Zr deoxidized	1500/2240	1000	Grain growth, Zr loss, swelling
10	Nb-1Zr	1350/1970	2300	
11	Nb-1Zr	1100/1520	4300	
12	Nb-1Zr	1000/1340	3870	

TABLE A-4. APU FOIL GAS BEARING EXPERIENCE

	767/757	DC10	F-18	General Aviation	C5B
Systems in service	470	1,660	570	1,269	35
Total hours operating	6,735,800	59,073,400	406,180	3,200,500	42,000
Mean time between failures (MTBF), hours	57,100	61,700	45,135	35,170	N/A
Average number of start/stop cycles per unit	2,200	2,100	1,400	2,500	110
Annual bearing production rate	200	100	200	500	300

The mini-BRU was the turboalternator-compressor incorporated in the BIPS. After some initial problems with one of the foil bearings, design modifications were made and the unit was operated for 1000 hours in a relevant environment at a turbine inlet temperature of 1025 °K (1385 °F) with no further problems (i.e., TRL 6). The BRU and mini-BRU engines all used a Rice alternator. The DIPS TAC employs a two pole toothless (TPTL) permanent magnet generator (PMG). The use of Samarium-Cobalt permanent magnets for field excitation is a well established technology. Allied-Signal Samarium-Cobalt PMGs used in aircraft applications have in excess of 100,000 operating hours experience.

Recuperator State-of-the-Art

The applicable experience base for CBC recuperators includes the recuperators for the BRU series of power conversion units, and for BIPS. This experience base is summarized in Table A-5. As noted in the table, one of the BRU recuperators experienced a failure at 18,000 hours of operation. The nature of the failure was a leak between the core and the manifold ducting. Analysis of the cause of the failure attributed it to low-cycle fatigue due to deep thermal cycling. The problem was corrected by redesigning the internal support structure and shell, and using a welded and fully inspectable containment boundary for the recuperator shell. The BIPS recuperator used these improvements and was operated without incident. This recuperator is now being refurbished for use on the 2 kWe Solar Dynamic Space Power System Ground Test Demonstration Project being performed by the Allied Signal/Rocketdyne team for NASA LeRC. These tests will include 1000 hours of steady-state operation plus startup/shutdown, and system thermal cycling tests.

The DIPS recuperator design is based not only on the five successfully recuperated CBC systems, but is also founded on heat transfer equipment developed by Allied Signal and used in thousands of commercial, refinery, and aircraft applications.

Radiator State-of-the-Art

The tube and fin radiator concept is a well established technology. Fabrication of the radiator assembly is accomplished using standard materials and fabrication practices. Tube and fin radiators are used in many commercial applications. The space shuttle uses a redundant pumped loop tube and fin radiator which has been in use for more than 10 years. On the Space Station Freedom (SSF) program, the main thermal bus uses a pumped loop tube and fin radiator concept. For the DIPS application, the fluid within the tube is the He/Xe working gas which is different from the preceding applications where the pumped fluid is a liquid. The radiator tube and fin design concept is the same, however, with the thermal hydraulic characteristics being slightly different. Thermal hydraulic tests will be required to verify heat transfer coefficients.

Optical Solar Reflectors (OSRs) have been used for thermal management of solar arrays and spacecraft on a variety of satellites such as Explorer, Intelsat, Solar Maximum Mission and Satcom. The OSRs are attached to the radiator fins with a silicone bond. Prototypic tests of bonding OSRs to aluminum fins were recently carried out at Rockwell, employing several different types of silicone bonding agents to demonstrate bond integrity and develop optimum application methods. Test results indicated that the bond is stable and shows no degradation in thermal cycling tests. The tests used a lap configuration and in all tests for bond strength, the bond was stronger than the tiles themselves.

TABLE A-5. APPLICABLE CBC RECUPERATOR EXPERIENCE

Unit Name of Designation	Power Level (kWe)	Design Feature	Operating Parameters	Test/ Operational Hours	Key Test Results
BHXU (BRU)	10	Plate-fin plate-bar construction 347 SS	T_{in}=1230 °F P_{in}=27 psia W=7.5 lb/s	51,000	One failure at 18,000 h; unit repaired and test resumed
BHXUA (BRU)	10	Plate-fin double containment side plate Hastelloy-X	T_{in}=1230 °F P_{in}=27 psia W=7.5 lb/s He-Xe MW = 83	N/A	Not tested - 2 units delivered
BIPS	1.3	Plate-fin double containment side plate Hastelloy-X	T_{in}=950 °F P_{in}=66.7 psia W=0.25 lb/s He-Xe MW = 83	1,200	No failures

Ducting & Bellows State-of-the-Art

The DIPS system uses gas ducting components similar to those used and successfully demonstrated on the BRU and BIPS CBC systems. The ducting assembly consists of the metal ducting, bellows, and its multifoil insulation package. The BRU was tested in an environment simulating the pressures and temperatures of space. The BIPS was tested in space-vacuum conditions during a workhorse loop 1000 hour demonstration. In conjunction with the BIPS program, ORNL conducted a series of tests on BIPS bellows extending over several years at simulated space temperatures and pressures. The BIPS employed multifoil insulation which proved most effective for controlling thermal losses. For the DIPS ducting, Inconel 617 will be used for the hot leg ducting and aluminum or stainless steel will be used for the cold leg ducting. Dissimilar joint compatibility will be assessed for aluminum or stainless steel and Inconel 617.

PP&C State-of-the-Art

The DIPS PP&C system design is based on a reasonable electronics component evolution, from SSF component technologies and there are no significant technology issues associated with its development. However, PP&C mass will be critical item in the flight hardware design. It will be necessary to fabricate brassboard hardware for testing and evaluation purposes, but there is no need to initiate any advanced component development. Even though no space-based rotary alternator PP&C systems have been fabricated, most of the hardware elements have been or shortly will be operating in a relevant environment on other spacecraft or the SSF electrical power subsystem ground tests. The only new environmental factor is the radiation emitted by the DIPS HSUs. The DIPS radiation levels of 10^4 Rad (Si) are considered to be relatively low for electronic components. These radiation levels can be easily handled through proper component selection.

The electronics cooling radiator uses aluminum/ammonia heat pipes in an aluminum honeycomb/face sheet structure to reject the electronics waste heat (260 W thermal) to space. This technology has already been space qualified as indicated in Table A-6.

TABLE A-6. WATER AND AMMONIA HEAT PIPE EXPERIENCE

a)Ground and Shipboard Applications

Heat Pipe Material	Working Fluid	Application	No. Units Per System/ Module	Units Built	Years in Service	Estimated Hours x 10^6	Reported Failures
Cu/Cu	Water	Trident, SEM	108/3	7,000	14	458.65	0
Monel/Cu	Water	Mis. Spec.	1/1	300	10	8.74	0
SS/SS	Ammonia	Nassar Array	1000/1	1,200	16	467.39	0
Cu/Cu	Water	ALCM	12/3	12,000	8		0
SS-Cu/ SS-Cu	Methanol	MARM/ Missile	1/1	450	9		0
Cu/Cu	Water	Element G	1/1	397	2	2	0

b)Intended Applications - Shipboard, Avionics, and Space (Life Test Completed)

Heat Pipe Material	Wick Material	Working Fluid	No. Units	Operating Temperature (F)	Total Hours	No. of Failures
Cu	Cu	Water	2	60 to 75	77,376	0
Cu	Monel	Water	1	70	48,406	0
Monel	Monel	Water	2	100 to 102	69,264	0
SS	SS	Ammonia	3	40 to 60	102,110	0
Al	Al	Ammonia	1	50	59,832	0
Al	Grooves	Ammonia	1	50	29,664	0
SS	Grooves	Ammonia	1	50	34,632	0

c)Space Applications

Heat Pipe Material	Working Fluid	Applications	No. Units Per System/ Module	Units Built	Years in Service	Reported Failures
SS/SS	Ammonia	DSD, TWT baseplate	3/3	700	10.0	0
Al/Al	Ammonia	MSIII	19	22	0.5	0
Al/Al (Grooves)	Ammonia	Space telescope	3	3	0.5	0
Al/Al	Ammonia	Space sensor	1	2	0.2	0

MAJOR DEVELOPMENT TASKS

A 2.5 kWe CBC isotope power module is currently under development as part of the DIPS Demonstration Program which is sponsored by the U.S. Department of Energy (DOE) with support funding from NASA (Refs. A-1 to A-2). The development program defined in this roadmap is a continuation of the DIPS program. This development plan includes an initial component development and integrated ground system demonstration, a qualification program, and a flight program.

With the exception of the GPHS modules, the technology level of the DIPS components ranges from 4 to 6, and therefore development and demonstration testing is required. The development and testing tasks which are envisioned are briefly described in the following sections.

Task 1. Heat Source Unit Heat Exchanger (HSUHX) Development

Objectives: Develop a full scale flight HSUHX. Demonstrate adequate performance characteristics, means of protecting the unit from the effects of the Martian environment, long life at high temperatures, and structural integrity during a loss of coolant event.

Statement of Work: The work is divided into the following subtasks:

Subtask 1.1 Emissivity Coating Tests - Evaluate the emissivity and integrity of candidate coatings/surface treatments under the required thermal vacuum environments (including application techniques, surface preparations, and adherence) using subscale coupons. Select reference coatings (different coatings or surface treatments may be required due to the differing substrate materials) and apply to simulated canister, heat exchanger, and heat pipe wall surfaces. Conduct elevated temperature performance and thermal cycle tests to validate coupon emittance test results, and demonstrate adherence using prototypic configuration specimens.

Subtask 1.2 Multifoil Insulation Tests - Perform additional tests to verify the compatibility of candidate foil and oxide spacer materials under accelerated temperature conditions. Clad foils which require bonding of two foil materials, are expected to yield a better defined melting point and a more rapid melt down, but they require development. Conduct subscale tests on bonded foil specimens to determine if they are practical to fabricate, perform as predicted, retain adequate bond strength, survive thermal cycling, and do not diffuse excessively. Results of the clad foil tests will determine whether clad or conventional layered foils will be selected as the reference DIPS configuration.

Perform subscale meltdown tests similar to those performed on the multifoil system developed for the BIPS program. These assemblies will have the same combination of foil, oxide spacer, number of foils, foil thickness, layer spacing, ratio of length to diameter, and hot side material as candidate HSU designs. Test assemblies by equilibrating at the design temperature and then heat at a rate corresponding to that seen if all cooling were lost. Verify by experiment and analysis safe fuel cladding temperatures following the loss of cooling event

Conduct thermal performance testing on prototypic sections of multifoil insulation to verify that the calculated losses from high heat loss components are within acceptable limits. Measure the heat input versus heat source temperature relationship of a full scale HSU in an inert gas environment to define storage and handling parameters.

Conduct stability tests on prototypic sections of MFI to verify stability under long term (up to one year), elevated temperature conditions in simulated environments.

Subtask 1.3 Heat Exchanger Coil Test - Fabricate a hydraulic model of one complete helical coil heat exchanger. Measure the pressure drop across the heat exchanger and compare to calculated values. Based on these experimental results, incorporate any necessary refinements required into the design.

Subtask 1.4 Overall Heat Exchanger Tests - Test one complete HSUHX powered by an electric heater for performance using an inert gas loop with the HSU under internal vacuum. Establish prototypic thermal and flow conditions, and conduct a performance test including thermal mapping under varying flow conditions. Vibration test the assembly and repeat the performance test to identify any degradation. Establish the condition of the HSU and verify that the internals are undamaged. Run a final test in a vacuum chamber to verify that the meltable MFI performs as designed in the event of a loss of cooling. This test will render the HSUHX unsuitable for further testing.

Subtask 1.5 Electrical Heater Subassembly Tests - Test a complete set of electrical heater assemblies and their controls for performance, transient effects, operating environment, vibration, and shock effects. Perform endurance testing of the heaters to identify mean time between failure (MTBF) values for these major electrical components. Evaluate heater degradation, control functions, and monitoring functions during the endurance tests to establish instrument error bands to be used during the Integrated System Test Unit (ISTU) life test.

Task 2 - TAC Development

Objectives: Develop a full scale flight TAC. Demonstrate adequate steady state and transient performance characteristics, means of protecting the unit from the effects of the Martian environment, and long turbine life at high temperatures.

Statement of Work: The work is divided into the following subtasks:

Subtask 2.1 Compressor Tests - Operate a compressor stator/scroll assembly over its full range of mass flow to verify performance and map performance factors. In a separate test in a whirl pit, measure impeller growth at various speeds. Verify blade vibration mode shapes and frequencies by holographic testing.

Subtask 2.2 Turbine Tests - Operate a turbine/nozzle scroll assembly at a single design point speed and pressure ratio using air as the working gas to verify design performance. Evaluate test data and extend the results analytically to ensure a good match to the compressor at off design conditions. In a separate test, verify turbine wheel stress margins by measuring wheel growth at various speeds. Conduct performance testing in the modified compressor test rig and measure wheel growth in the whirl pit. Verify blade mode shapes and frequencies by holographic testing.

Subtask 2.3 Dynamic Simulator Tests - Determine turboalternator dynamic characteristics, critical speeds, and bearing losses using a rotor simulator with foil-gas bearings. Repeat these tests later using a functional alternator in an open-cycle TAC test. Determine alternator performance and dynamic characteristics. Conduct start-up and extended life tests.

Subtask 2.4 Alternator Tests - Conduct alternator stator and rotor tests with the rotor supported on conventional bearings. Measure the performance as an alternator and as a motor over a range of speeds, loads, and excitations. Conduct tests to determine generated power, electrical losses, thermal losses, wave shapes, starting torque and cooling required to maintain component temperatures.

Task 2.5 Alternator Insulation Tests - Develop a high temperature electrical insulation for the alternator stator. Perform voltage breakdown and lifetime tests of the electrical insulation to identify degradation mechanisms and lifetime limits as a function of component temperatures. Use these results to establish the cooling gas supply temperature limits in Subtask 2.4.

Task 2.6 Foil Bearing Tests - Design, fabricate and performance test the TAC foil bearing assemblies using a bearing rig test fixture to performance map each bearing configuration. Measurements should include torque versus speed, spring rate, load deflection, damping coefficient and starting torque versus load and acceleration. Perform shock and vibration testing for each bearing assembly.

Task 3. Recuperator Development

Objectives: Develop a prototype flight recuperator. Demonstrate adequate performance characteristics, means of protecting the unit from the effects of the Martian environment, and long life at high temperatures.

Statement of Work: The work is divided into the following subtasks:

Subtask 3.1 Recuperator Leakage Pressurize the recuperator core assembly with hot gas and thermally cycle it to verify its structural integrity. Measure leakage across the core and critical dimensions at temperature extremes.

Subtask 3.2 Heat Transfer Performance Conduct hot-gas heat transfer tests on a recuperator core section to verify performance. Measure pressure drops through the recuperator core section versus flow rate and inlet temperatures for He-Xe gas mixtures.

Task 4. Radiator Assembly Development

Objectives: Develop a prototypic gas tube and fin radiator assembly. Demonstrate adequate structural and thermal performance characteristics, and a means of protecting the unit from the effects of the Lunar and Martian environments, over its 15 year design lifetime.

Statement of Work: Performance test a module consisting of an aluminum gas tube and fin radiator assembly and associated headers. Verify performance of the radiator section under design point and extremes of temperature and radiating surface contamination conditions. Measure pressure drop, flow distribution, and heat transfer to determine radiator performance.

Conduct baseline performance tests in a vacuum chamber, with the assembly at temperature and radiating to a chamber cold wall. Perform random vibration tests of the assembly to determine its response and identify potential structural integrity problems. Repeat the performance testing to verify structural integrity.

Develop the OSR tile application process and establish the required bond strength. Demonstrate the OSR lifetime performance. Perform dust tests of the radiating surface effectiveness to simulate the lunar and Martian environmental effects on radiator performance.

Task 5. Ducting and Bellows Development

Objectives: Develop prototypic hardware for the interconnect gas ducting and bellows. Demonstrate adequate performance of the ducting.

Statement of Work: Design and fabricate prototypic interconnect ducting and bellows. As required, conduct static and dynamic structural verification testing on elbow ducting, branch connections, straight sections, bellows, all joints, and heat exchanger inlet/outlet components at room temperature and elevated temperatures. Perform deep thermal cycling tests on all joints and bellows to establish thermal cyclic life. Verify performance using pressure drop and flow distribution tests.

Task 6. PP&C Development

Objectives: Develop the necessary PP&C hardware and software to control the DIPS operation and process alternator output power. Demonstrate adequate steady state and transient voltage control at rated load conditions. Demonstrate lifetime, reliability and compatibility with the environment (including launch and operating environments). Develop and demonstrate the software capable of providing power system control under nominal, transient operations and during simulated failure modes.

Statement of Work: The work is divided into the following subtasks:

Subtask 6.1 Electrical Component Development - Build brassboard units to demonstrate and check functional performance of the individual component circuit designs. Incorporate design modifications and improvements, as necessary, into the brassboard units. Verify functional performance within the constraints of the actual component configuration. Fabricate prototype units and conduct a series of performance tests using simulated input and output loads. Conduct controller tests and validate the operating system software. Conduct the following other tests:

- start up, steady state and transient control simulation;
- failure simulation for detection and automatic switching to redundant control channel;
- effects of temperature extremes and thermal shock;
- effects of atmosphere;
- current limit protection demonstration;
- cold plate heat loads;

- Master/slave controller architecture demonstration (for integrated multiple unit power supplies); and
- EMI generation and susceptibility.

The prototype cable harnesses and parasitic load radiator will be checked out with the PP&C components during the ISTU calibration and set-up tests.

Subtask 6.2 Software Development - Develop the DIPS controller software. Check out the software in conjunction with tests of the controller, using simulated inputs and outputs. Integrate and checkout the software as part of the controller tests in Subtask 6.1.

Task 7. Integrated System Test Unit (ISTU) Development and Test

Objectives: Develop and test a full scale ISTU. Demonstrate adequate steady state and transient performance characteristics, long life at high temperatures, and suitable performance during failure modes. The ISTU shall be instrumented and calibrated during initial tests to validate the performance of the individual component designs.

Statement of Work: Assemble a complete power conversion unit consisting of two simulated HSU heat exchanges and one prototypic HSU, TAC, recuperator, interconnecting ducting, controls and instrumentation. Conduct tests in air using fibrous insulation on the ISTU loop with an electrical heat source, simulated user loads, and parasitic load resistors. Install a vacuum system to maintain the prototypic HSU containment vessel under internal vacuum. Demonstrate start-up, shutdown, and alternator design speed control under nominal operating conditions. Verify proper interface with the DIPS controller under conditions of varying power demand and simulated faulted conditions. Perform the following tests:

- time to start up from cold condition and motor KVA, invertor ramp rate;
- startup and battery recharge time;
- thermal balance;
- electrical power generating capability;
- steady state and dynamic stability;
- shutdown and start-up under simulated faulted conditions; and

- life test.

The work is divided into the following subtasks:

Task 7.1 ISTU Calibration and Set-up Tests - Pressure and leak test the ISTU. Fill the gas system with the He-Xe working fluid. Perform a complete electrical check-out of the unit including TAC shim and calibration tests and adjustments. Verify performance of thermal insulation.

Conduct a preliminary performance test sequence to demonstrate technical capabilities and measure critical unit performance parameters. Integrate acceptance level and design margin tests, up to qualification levels, into the initial test sequence to validate the individual component designs. Disassemble and inspect the individual components in the ISTU for wear and degradation effect and update component designs for the qualification and flight units as necessary.

Task 7.2 ISTU Life Test - Refurbish the ISTU components after completion of the performance, acceptance, and margin tests and place the unit on a multiyear life test. Refurbishment of the unit should include:

- replace or repair components to provide a prototypic hermetically-sealed gas containment boundary for the ISTU; and
- add special instrumentation required for the life test phase

Install prototypic instrumentation to provide a comprehensive diagnosis of the "health" of the ISTU and to monitor for degradation of major assemblies and individual components. Operate the ISTU at its nominal operating point, with expected ISTU variations in power output and environment.

Disassemble and inspect the ISTU at the end of the life test. Determine specific areas to be examined by an analysis of the health monitoring data and from the reliability analysis predictions. Typical characteristics to be determined by the examination should include, but are not limited to, bearing wear, welds and bellows wall metallurgical examinations, electrically controller setpoint drift, HSU MFI condition, and turbine blade erosion.

Task 8. Qualification Program Testing

Objectives: Design, fabricate, and test the flight system. Develop a low risk qualification program. Verify adequate performance and life for the entire system under flight qualification conditions.

Statement of Work: Perform a comprehensive performance and dynamic testing program of assemblies and the complete system to provide a formal demonstration that the DIPS will perform as designed after being subjected to simulated launch conditions.

Start with qualification of assemblies, as seen in Figure A-10. Fabricate qualified production items and assemble these parts into the QU. Qualify the QU by the rules for space vehicle qualification.

The work is divided into the following subtasks:

Subtask 8.1 Component Qualification Testing - Conduct performance testing at the component and assembly level to verify that each item performs as designed. Perform dynamic testing per MIL-STD-1540B to verify capability of the DIPS system to withstand launch loads, including acoustic, pyroshock and vibrational. The performance and dynamic qualification test sequence for components and assemblies is shown in the matrix of Figure A-11.

Subtask 8.2 Qualification Unit Testing. Fabricate, assemble, checkout, and test the QU. Use the same test facilities for component and assembly qualification testing as were used for the assembly level testing of the ISTU. The corresponding qualification test sequence for the QU is shown if Figure A-12.

Subtask 8.3 Qualification Life Testing (Optional) - Partially disassemble, examine, and refurbish the QU as required and modify for endurance testing as described for the ISTU. Life test the unit for 1.5 years (optional).

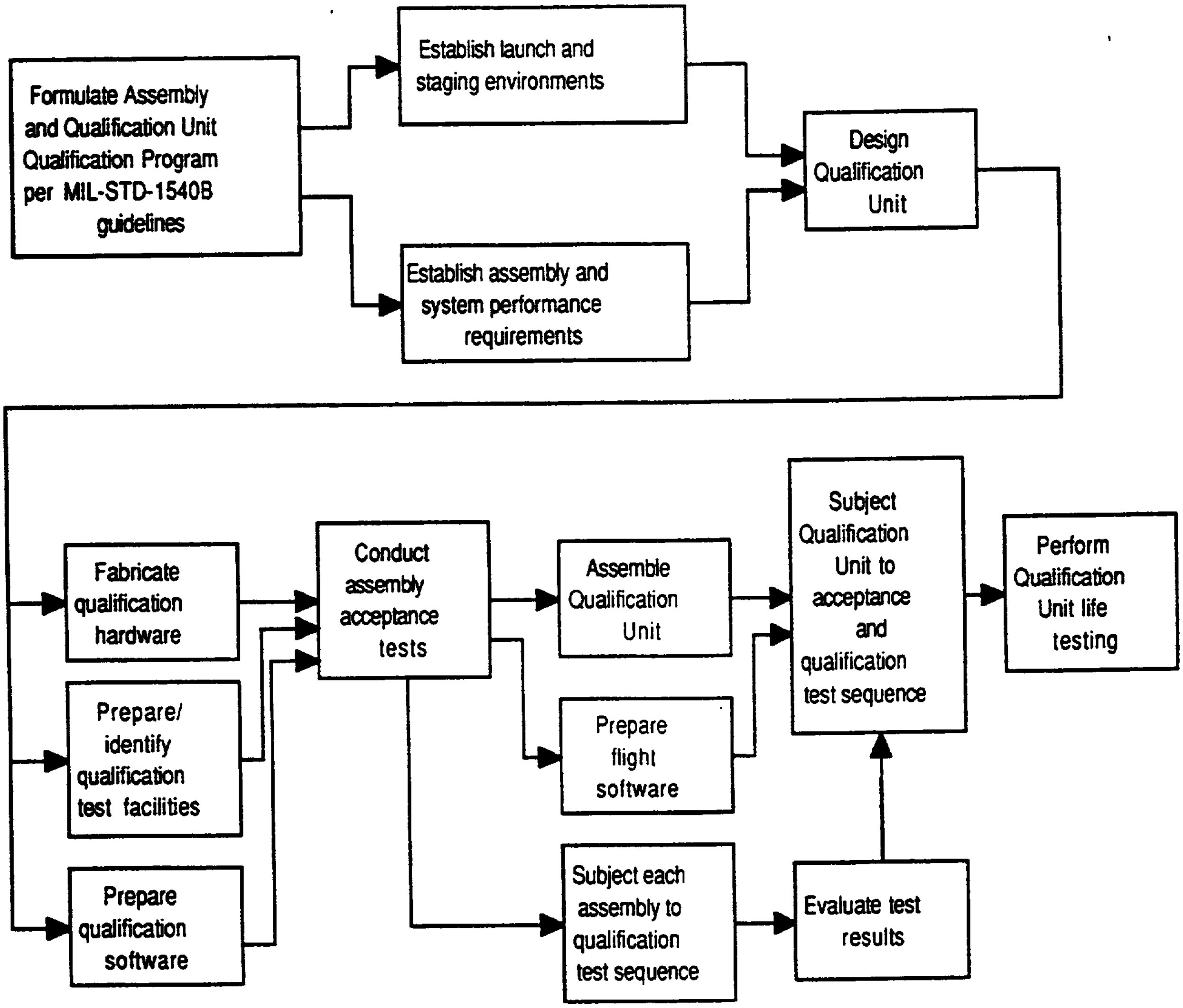


Figure A-10. DIPS qualification program.

Component or Subassembly	Functional (1)	Leak	Pyroshock	Functional (2)	Random Vibration	Functional (1)	Leak	Acceleration	Functional (1)	Thermal Cycling	Functional (2)	Thermal Vacuum	Functional (2)	Pressure	Leak	EMC	Life	Functional (1)
HSU	X	X			X	X	X		X			X	X	X	X		X	X
TAC and recuperator	X	X			X	X			X			X	X	X	X	X	X	X
Radiator and manifold	X	X			X	X	X		X			X	X	X	X		X	X
Electronics radiator	X	X			X	X	X		X			X	X	X	X		X	X
Parasitic load radiator	X				X	X			X			X	X			X	X	X
PP&C	X		X	X	X	X			X	X	X	X	X			X	X	X
Structure	X		X		X	X		X	X			X	X				X	X

Figure A-11. Assembly qualification test matrix.

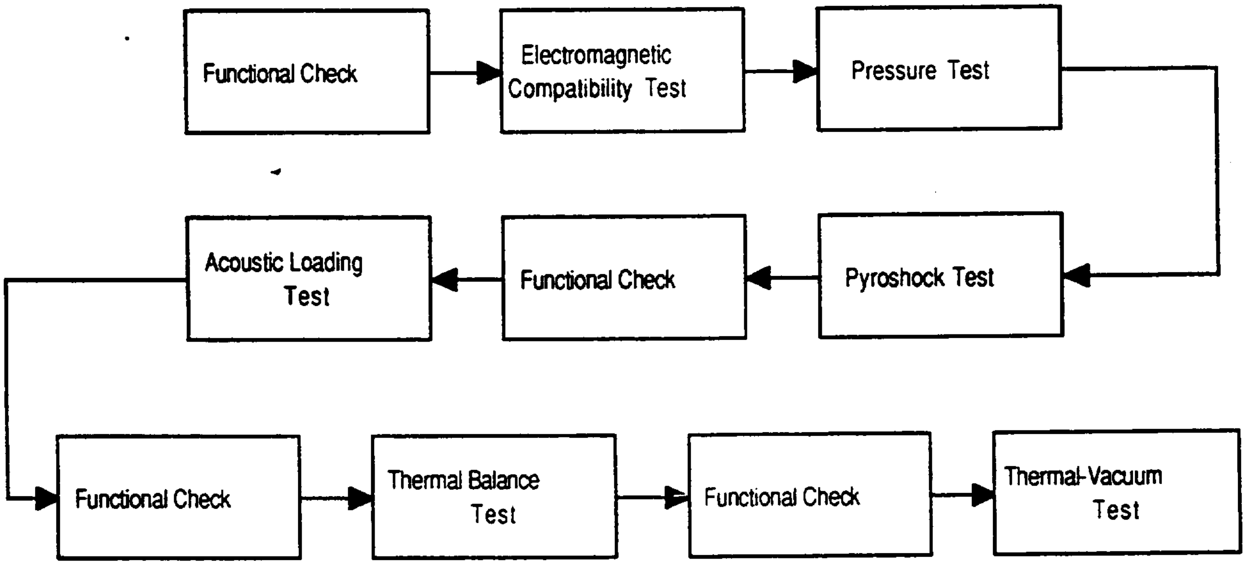


Figure A-12. QU test sequence (per MIL-STD-1540B).

Task 9. Flight Unit (FU) System Program

Objectives: Fabricate two flight systems, perform a flight safety program, and acceptance test the FUs to demonstrate required performance. Deliver the flight systems and provide integration support for the FU with the payload and the launch vehicle.

Statement of Work: Fabricate, acceptance test, and assemble parts to produce two DIPS flight systems. Subject both systems to acceptance testing per MIL-STD-1540B guidelines before shipment to the launch site. Use the same test facilities (vacuum chamber, vibration, acoustic) that were used for the qualification program for flight system acceptance testing. Perform safety studies and complete safety reports necessary to obtain launch approval. Provide launch support for the DIPS flight unit for integration with the payload and launch systems.

The work is divided into the following subtasks:

Subtask 9.1 Flight Component Fabrication. - Design, fabricate, inspect, and assemble the components and subassemblies required for the QU and FUs, including all spare parts and GSE as required.

Subtask 9.2 FU Assembly, Test, and Payload Integration - Assemble and inspect the two FUs. Acceptance test both FUs and ship to the launch site. Provide technical support for FU integration with the payload, launch vehicle, and launch support facilities.

Subtask 9.3 Flight Safety Program - Develop a flight safety program plan to support the safety studies and tests required to obtain launch approval. Prepare the safety analysis reports (SARs), and all supporting analyses and documents to assure launch approval.

Subtask 9.4 FU Launch Support - Provide launch support for the integrated FU/payload and launch vehicle systems. This includes FU monitoring during ascent, payload deployment, and FU startup on station.

DEVELOPMENT SCHEDULE

Figure A-13 presents the 2.5 kWe DIPS development schedule. The DIPS program has completed a 1 year conceptual design task. The preliminary design would be completed in the next two years. Concurrent component development of the heat source unit, power conversion unit (TAC, recuperator, and ducting), radiator assembly and PPCA is completed in 2.75 years. Detail design work is subsequently completed after 3.5 years.

Fabrication of components for ground testing for the ISTU starts with procurement of long lead materials and equipment in the second year of the program. This leads to assembly of the ISTU in the first half of the third year.

The ISTU will simulate the performance of a flight system but will have features such as additional instrumentation and readily accessible components to expedite gathering of engineering data and to permit modification of components. It will be performance tested in air under normal and off-normal design conditions. After this phase of testing, it will be partially disassembled, examined, refurbished as necessary, and put on life test, nominally for 1.5 yrs.

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components for three units, and a complete DIPS system qualification test. The redesign effort will be limited to minor modifications to the preliminary DIPS design.

The flight phase of the program includes assembly, and acceptance testing of two flight units and the associated safety analysis support for launch approval. Launch support activities would include the flight unit-payload and launch vehicle integration tasks as well as post launch support activities.

CLOSED BRAYTON CYCLE DYNAMIC ISOTOPE POWER SYSTEM TECHNOLOGY ROADMAP

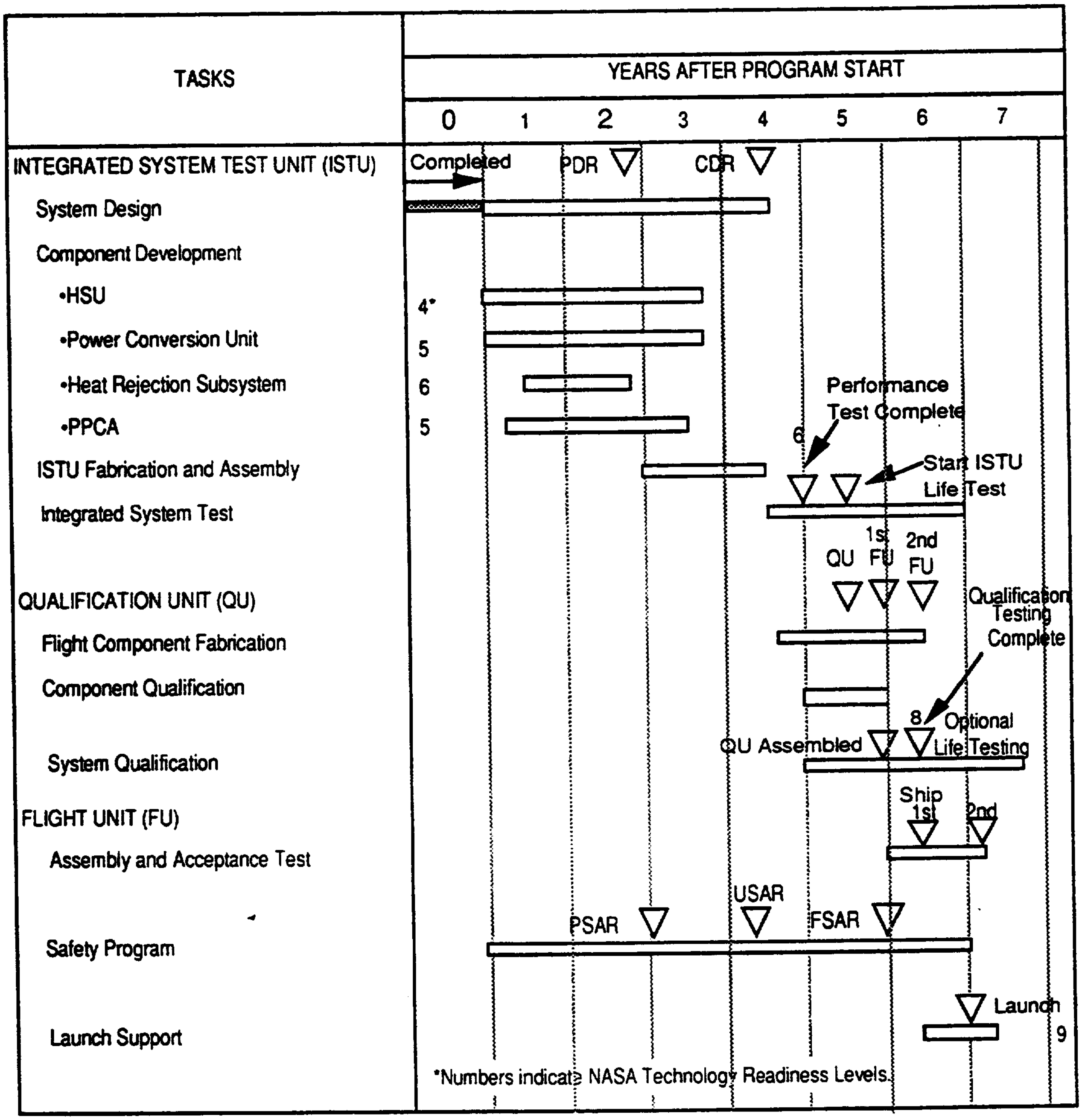


Figure A-13. 2.5 kWe DIPS development schedule.

REFERENCES

A-1 Johnson, R.A.; Hunt, M.E.; Pepping, R.E.; and Mason, L.S.: A Dynamic Isotope Power System Portable Generator for the Moon or Mars. 26th IECEC proceedings, Vol. 2, August 1991, p. 210.

A-2 Johnson, R.A.; Stadnik, A.G.; and Williams, R.: Compact, Long-Lived Dynamic Isotope Power Systems for the Exploration of Space.

A-3 Heat Pipe Design Handbook, June 1979.

A-4. Woldshun, K.: Rocketdyne Emergency Cooling Heat Pipe Design Summary. LANL draft report, 6 March 1987.

APPENDIX B

PEM PFC TECHNOLOGY ROADMAP

APPENDIX B - PEM RFC TECHNOLOGY ROADMAP

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REGENERATIVE FUEL CELL POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX B - PEM RFC TECHNOLOGY ROADMAP

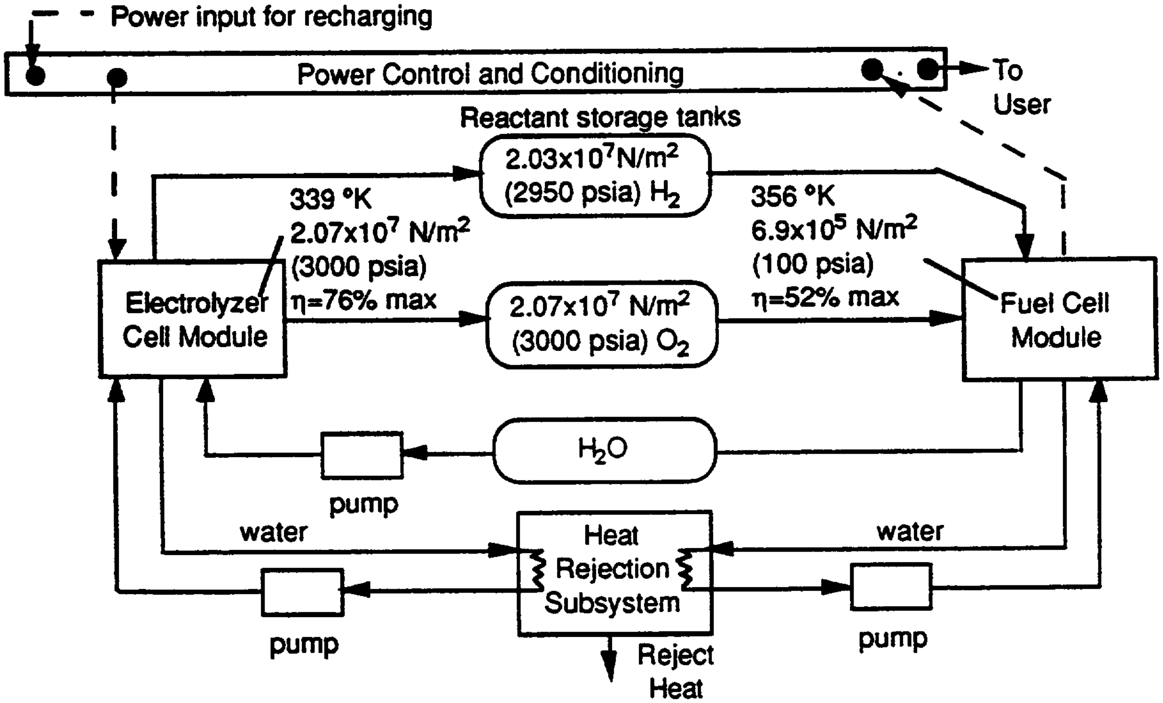
INTRODUCTION

This is a family of power systems based on common technology for use with portable or mobile power systems. Mobile RFC power systems are either incorporated into vehicles or are attached to vehicles as part of a separate power cart. These systems range in nominal power level from 3 to 22 kWe. These vehicles include the payload unloader (3 kWe normal [n]/10 kWe peak [p]), pressurized manned rover (7 kWe onboard, 12 kWe with power cart), regolith hauler (3 [n]/15 [p] kWe), and mining excavator (22 [n]/40 [p] kWe). Mobile Mars power systems will be similar to lunar systems except for the radiator size.

CONCEPT DESCRIPTION

The RFC system converts electrical energy into chemical energy and stores the energy for future use. An RFC is an energy storage device similar to a battery. The RFC system can be divided into six major subsystems for development purposes: (1)a fuel cell stack, which electrochemically converts hydrogen and oxygen into electricity; (2)an electrolyzer cell stack, which electrolyzes the fuel cell product water into gaseous hydrogen and oxygen reactants using externally provided power; (3)water management which removes moisture from the electrolysis cell product gases and humidifies fuel cell reactants to maintain proper cell membrane moisture content; (4)thermal management, which removes waste heat from the system, maintains the proper membrane temperature, prevents boiling or freezing in critical flow paths; (5)reactant storage (hydrogen, oxygen, and water); and (6)power processing and control (PP&C). The PP&C must be designed to allow for recharging from the base power system (either photovoltaic arrays or nuclear reactor) or from a portable Dynamic Isotope System (DIPS) cart.

A simplified schematic of a potential RFC system is shown in Figure B-1 (Ref. B-13). Figure B-1 does not show the details of the design such as electrical controls, fluid controls, trace heating, phase separation, gas humidification, gas drying, or redundant components. These items will be discussed in detail in the following sections.



Note: Necessary electrical and fluid controls, and redundant components not shown. Typical operating conditions and performance shown.

Figure B-1. PEM RFC power system schematic.

In the baseline RFC concept, high pressure oxygen and hydrogen gas were assumed for gaseous reactant storage tanks of relatively low volume. High pressure gas storage reduces the size and mass of the storage tanks. Cryogenic storage of oxygen and hydrogen may be desirable for large fixed systems, but will not be addressed in this roadmap. The complexity and power required for a liquefier is probably not warranted for a mobile system.

Two types of fuel and electrolysis cell technologies are available: alkaline and PEM. Proton Exchanger Membrane (PEM) fuel cells and electrolysis cells were selected for this study since these technologies were shown to be the preferred RFC approach for long life SEI applications in a recent LANL study done for NASA (Ref. B-1). The basic design and operation of the PEM RFC system are described in the following paragraphs.

Fuel Cell Description

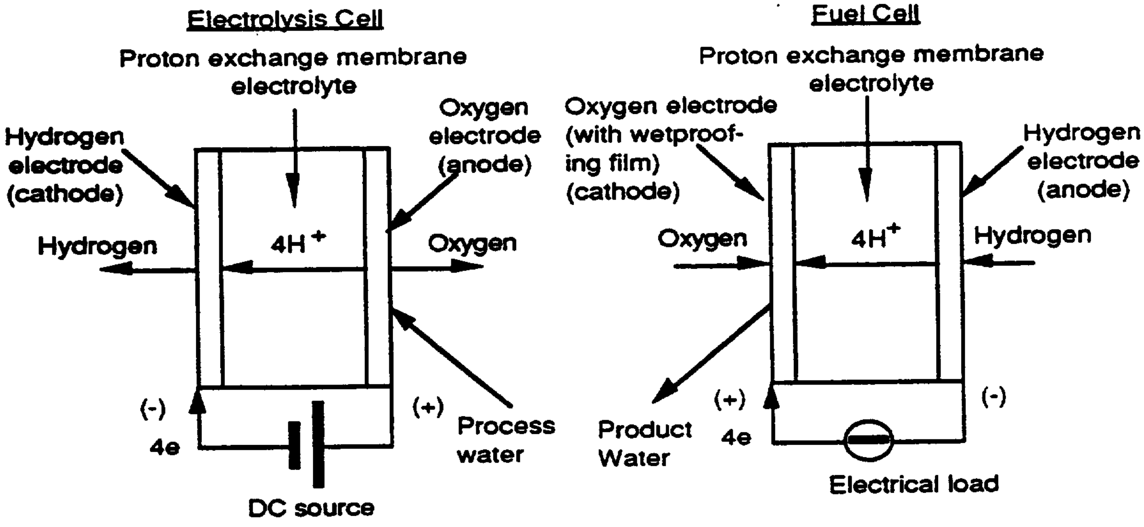
Fuel cells operate by separation of two electrocatalytic conversion reactions with an ionic conductor, as seen in Figure B-2 (Ref. B-2). Charge moves through electron conductors connecting the two electrocatalytic zones, where electron transfer results in chemical reactions. Ionic transport through the separator completes the process.

The PEM fuel cell incorporates an ion exchange membrane, typically a polyperfluorosulfonic (PFSA) acid sheet, as the ionic conductor. This component sustains transport of hydrated hydrogen ions, protons (H^+), associated with water. Protons are generated at the porous anode electrocatalytic layer and transport through the ionic conductor to the cathode electrocatalytic layer. At the cathode, protons react with oxygen to form water. Product water exhausts from the cathode compartment.

Hydrogen and oxygen gases are stored at 2.07×10^7 N/m^2 (3,000 psia) (Ref. B-3) for use in the fuel cell. The gases are regulated down to fuel cell operating conditions (4.14×10^5 to 6.9×10^5 N/m^2 or 60 to 100 psia). Oxygen is regulated to a few psi higher than hydrogen (for safety reasons) to insure that only oxygen is entrained in the product water. The reactant gases must be humidified prior to reacting in the stack. Humidification will be discussed in the water management section.

The hydrogen and oxygen gases are combined in the fuel cell to generate electricity and water. The product water is discharged into the cooling water loop. As the cooling water accumulator approaches the filled condition, the product water drain valve opens to allow water to flow to the storage subsystem.

The product water which leaves the fuel cell stack will be saturated with oxygen. This oxygen must be removed prior to entering the water storage tank. An approach for doing this has been demonstrated (external to the fuel cell) by Hamilton Standard, as seen in Figure B-3 (Ref. B-3). Water passes through an ion exchange membrane from the product water stream to humidify the dry hydrogen gas. Hydrogen diffuses from the hydrogen stream through the membrane to the water stream and combine with the oxygen to form water. Excess hydrogen is



CATHODE REACTION:

$4H^{+} + 4e^{-} \rightarrow 2H_2$

ANODE REACTION:

$2H_2O \rightarrow 4H^{+} + 4e^{-} + O_2$

CATHODE REACTION:

$4H^{+} + 4e^{-} + O_2 \rightarrow 2H_2O$

ANODE REACTION:

$2H_2 \rightarrow 4H^{+} + 4e^{-}$

Figure B-2. Acid PEM electrochemical cell reactions.

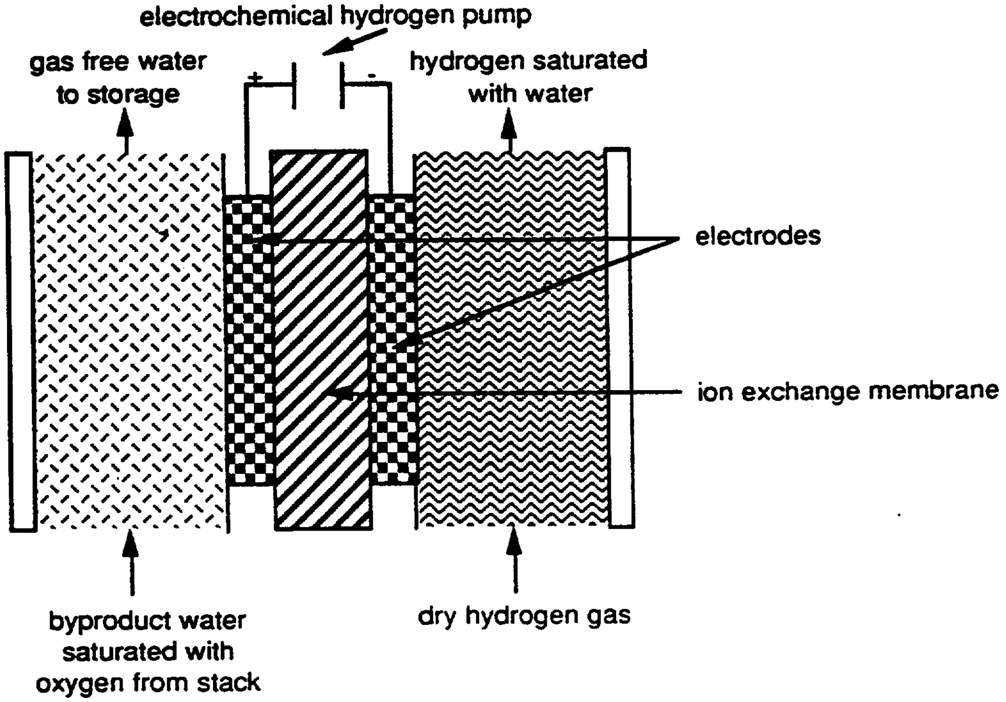


Figure B-3. Fuel cell product water deoxygenator.

returned to the hydrogen stream using an electrochemical hydrogen pump. Thus, only gas free water returns to the storage tank.

The fuel cell design options have to do with the type of membrane. Table B-1 (Ref. B-3) compares the fuel cell design options on a power density basis. The values in this table assume a system with 25 kWe net output continuously, 55% fuel cell thermal efficiency (based on 1.48 VDC) for 20,000 hours, and a design that is thermal vacuum compatible.

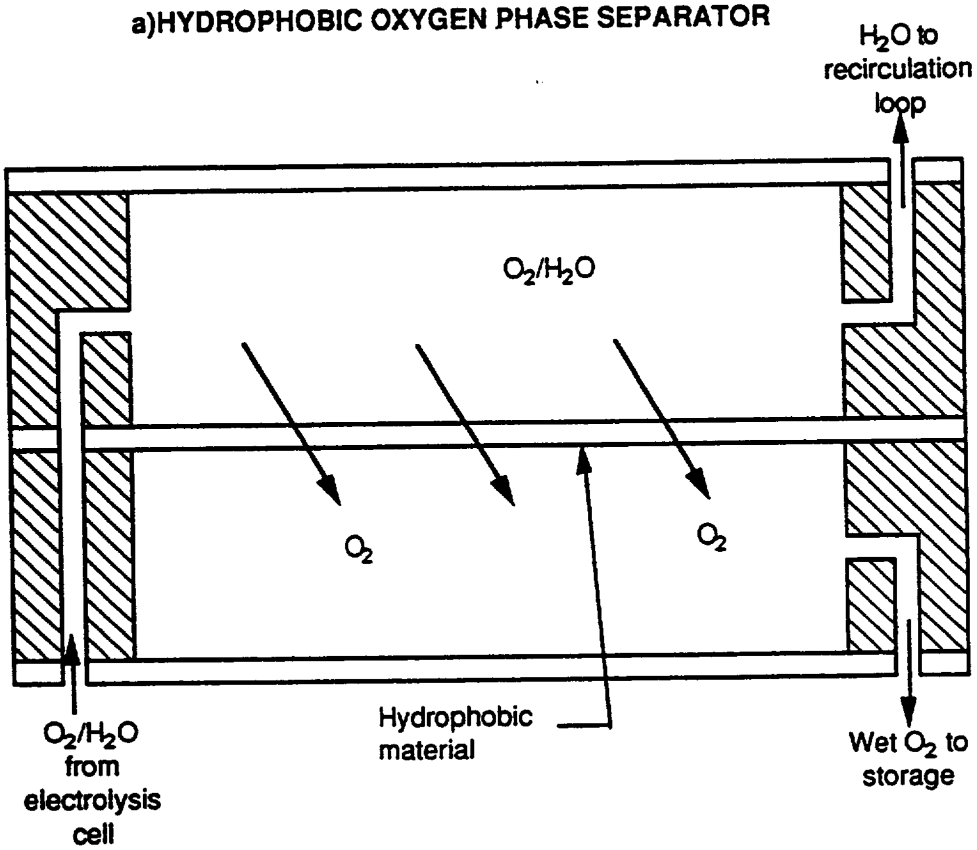
TABLE B-1. FUEL CELL DESIGN OPTION POWER DENSITY COMPARISON

Fuel Cell Subsystem Description	Nafion 120 Membrane (current)	Nafion 125/117 Membrane (advanced)	Dow Membrane (advanced)
	W/kg	W/kg	W/kg
"SOA Design" with Porous Hydrophillic Phase Separators (Space Station design)	103	184	307

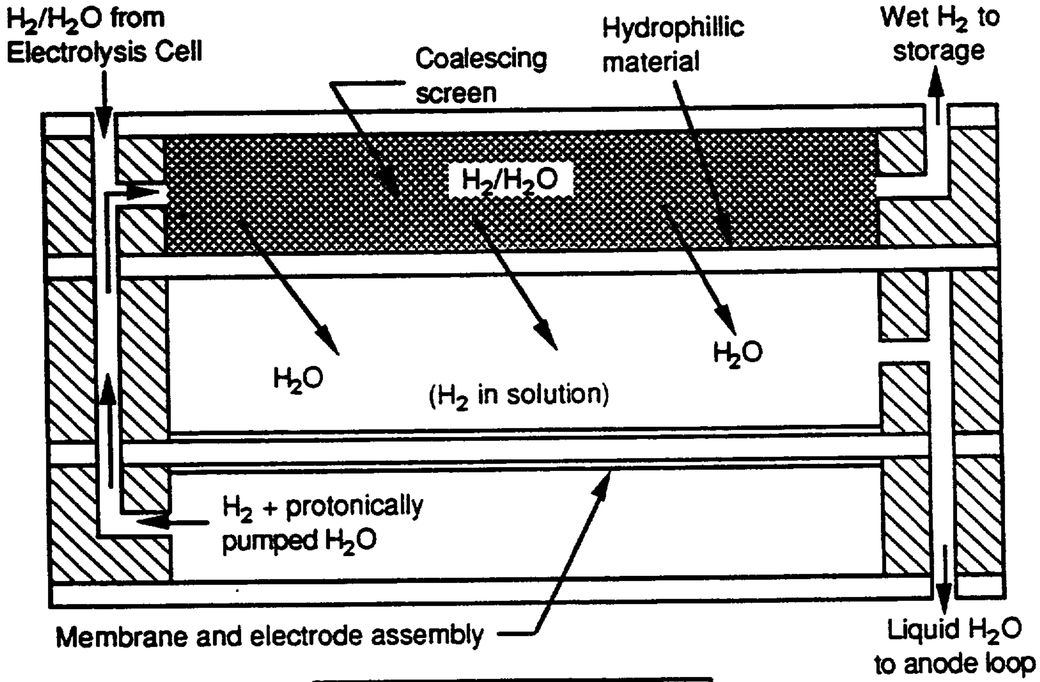
Electrolyzer Cell Stack Description

The PEM electrolyzer has the same type of ion exchange membrane as the PEM fuel cell to transfer H+ protons from the anode to cathode as was shown on the left of Figure B-2. Liquid water is pumped from the water storage tank by the water feed pump into the water recirculation loop, as was seen in Figure B-1. The water recirculation loop feeds the water into the cell stack on the anode side of each cell. Some water passes across the proton exchange membrane (PEM) forming a second water loop. Excess anode water loop flow is used to remove waste heat from the stack. Separators in the cell stack separate the hydrogen and oxygen gases from the liquid water streams, as seen in Figure B-4 (Ref. B-3). The separated gases (saturated with water vapor) are fed into regenerative dryers or are fed directly to the reactant storage tanks.

a)HYDROPHOBIC OXYGEN PHASE SEPARATOR



b)HYDROPHILIC/ELECTROCHEMICAL HYDROGEN PHASE SEPARATOR



Note: Dimensions not to scale.

Figure B-4. Oxygen and hydrogen phase separators in electrolyzer stack.

Various electrolyzer options were identified involving different cell spacings and different membranes. The estimated power densities for each design is shown in Table B-2 (Ref. B-3). Table B-2 values assume a system with 70% electrolyzer thermal efficiency for 20,000 hours, 2.07×10^7 N/m^2 (3,000 psia) gas generation pressure, thermal vacuum compatible design, and 13 kg/h of water electrolyzed. Only the Nafion 120 membrane has been life tested at 2.07×10^7 N/m^2 (3,000 psia).

TABLE B-2. ELECTROLYZER DESIGN OPTION POWER DENSITY COMPARISON

Electrolyzer Subsystem Description	Nafion 120 Membrane (current)	Nafion 125/117 Membrane (advanced)	Dow Membrane (advanced)
	W/kg	W/kg	W/kg
"SOA Design" with Static Separators	258	327	377
"Advanced Design" with Static Separators	347	392	414

The state-of-the-art (SOA) electrolyzer design utilizes the cell design which is used for U.S. and Royal Navy submarines. This cell design allows for 2.75 cells per centimeter (0.36 cm thick). The U.S. Navy utilizes a 2.07×10^7 N/m^2 (3,000 psi) stack while the Royal Navy uses a 1.03×10^6 N/m^2 (150 psi) stack design.

The "advanced" (Ref. B-3) electrolyzer design utilized a cell design of 12 cells per centimeter (0.083 cm thick). This cell was incorporated into a 120 cell stack for testing by the U.S. Navy as a low mass oxygen generator prototype. The advanced cell stack was designed for a maximum pressure of 2.76×10^6 N/m^2 (400 psi) without a housing. SOA cell stacks use a cell size of about 214 cm^2 (16.5 cm circular cell). This appears to be the optimum efficiency cell size for several 2.07×10^7 N/m^2 (3,000 psia) electrolyzer applications (Ref. B-4).

Thermal Management Subsystem Description

The thermal management subsystem provides temperature control, heat transport, and heat rejection functions. Pumped water and coolant loops provide the heat transport function, as seen in Figure B-5. A radiator is required to remove waste heat from the RFC system. There are inefficiencies in both the fuel cell stack and electrolysis cell stack which create waste heat. Membrane temperature must be controlled to prevent failures and meet life requirements. Water cools the stacks by collecting the waste heat and then transports the heat to one or more heat exchangers (one for electrolyzer and one for the fuel cell, or possibly a combined heat exchanger). These heat exchangers then transfer heat to the radiator coolant loop. Waste heat from the fuel cell may also be utilized to keep the electrolyzer from getting too cold.

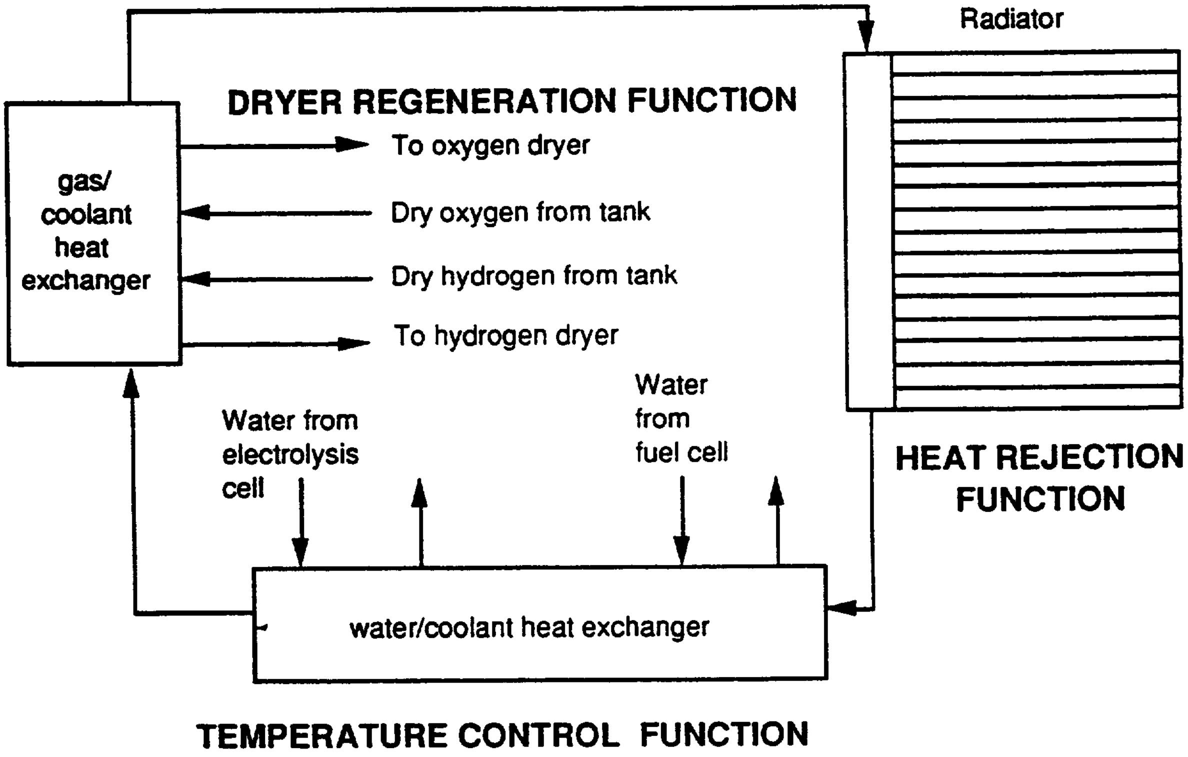


Figure B-5. Thermal management subsystem.

The heat rejection assembly provides a means for rejection of waste heat to the environment. Radiators for heat rejection are in some cases a major component of the power system mass. Radiators can also be quite large due to the low operating temperature.

Various options are available for the radiator design. Pumped loop radiators have been used successfully for the space shuttle and will be used for Space Station Freedom (SSF). This type of radiator is best applied to missions with limited duration or to systems which are serviceable. Pumped loop radiators are less massive than state-of-the-art heat pipe radiator designs. Heat pipes offer the advantage of improved reliability and a graceful failure mode. A recent Rocketdyne study has shown that advanced carbon-carbon (C-C) heat pipe radiators can be designed which are competitive in mass to pumped loop radiators. Thus, a heat pipe radiator was tentatively selected as the baseline design.

The baseline heat rejection assemblies for RFCs utilize lightweight, passive, reliable, heat pipe radiators that are sufficiently versatile to allow integration into a variety of configurations. The individual heat pipes operate independent from one another and thus the failure of a heat pipe will not result in failure of the complete radiator. The rectangular radiator heat pipe panel is attached to the coolant manifold. The cooling loop transfers heat to the heat pipes in the manifold heat exchanger. The heat pipe working fluid evaporates and travels to the top end of the heat pipes. The evaporated fluid is then condensed in the cooler section of the heat pipe. Both gravity and a small wick or grooves allow the liquid to return to the evaporator. A wick or groove is not absolutely necessary for vertically oriented radiators (due to gravity return) but is recommended to insure good control of the fluid transport. The heat pipes may be either carbon-carbon tubes with metal liners (Monel for water or aluminum for ammonia working fluids) or metal heat pipes.

Condensation and freezing of the water in critical locations must be prevented by maintaining fluid temperatures within limits. Thermal control of the lines and tanks may be accomplished by insulation, trace heating, insulation and trace heating, or convective heating/cooling using the radiator coolant. Composite tanks need to be kept above 219 °K to prevent tank failure. The composite tank liner will begin to separate from the overwrap at this temperature and buckling will occur. It may also be desirable to prevent the water vapor in the gases from freezing in the tanks. Fluid lines may also require thermal control to prevent water freezing and clogging of lines (especially in the pressure regulators).

Water Management Subsystem Description

Water management includes moisture control of the fuel cell membrane and the removal of moisture from electrolysis module product gases.

The moisture content of the fuel cell stack membranes must be carefully controlled to prevent dehydration and reduced life. The reactant gases must be humidified to maintain the proper membrane moisture content.

The traditional humidification approach for PEM fuel cells is shown in Figure B-6 (Ref. B-4). The humidifier automatically presaturates the incoming hydrogen and oxygen reactants to a dew point equal to the cell operating temperature. This latter approach was used in the Hamilton Standard SPE® fuel cell. The problem with this approach is that the product water going to the storage tank is saturated with oxygen. If the oxygen is not removed from the water, then the gas will accumulate in the water tank and have to be vented off (undesirable loss of reactant).

Another gas humidification approach, which is more appropriate for space systems, was shown in Figure B-4 (Ref. B-3). This approach also removes the oxygen gas from the product water and there is no gas buildup in the tank. This concept converts the oxygen to water by diffusion of hydrogen across the membrane. A hydrogen electrochemical pump keeps hydrogen from evolving in the water. Excess hydrogen is pumped back to the hydrogen side of the device. Although not shown if Figure 4, the oxygen reactant is also humidified in this same device.

If regenerative gas dryers are used in the system, then the gas from the tanks will be partially rehumidified during regeneration of the dryers as is seen in Figure B-7 (Ref. B-5; only the oxygen humidification is shown). Heat must be added to the cool dry gases in order to vaporize the water in the dryers. The purpose of this process is primarily to recover water from the gas dryers. Most of the gas humidification will be done using a humidifier such as was shown in Figure B-4.

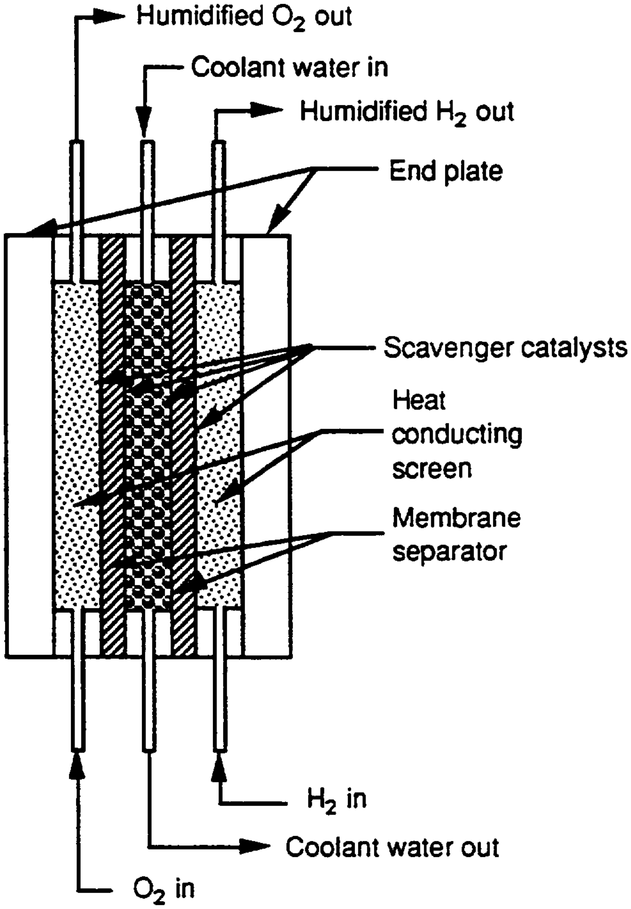


Figure B-6. SPE® fuel cell reactant prehumidification approach.

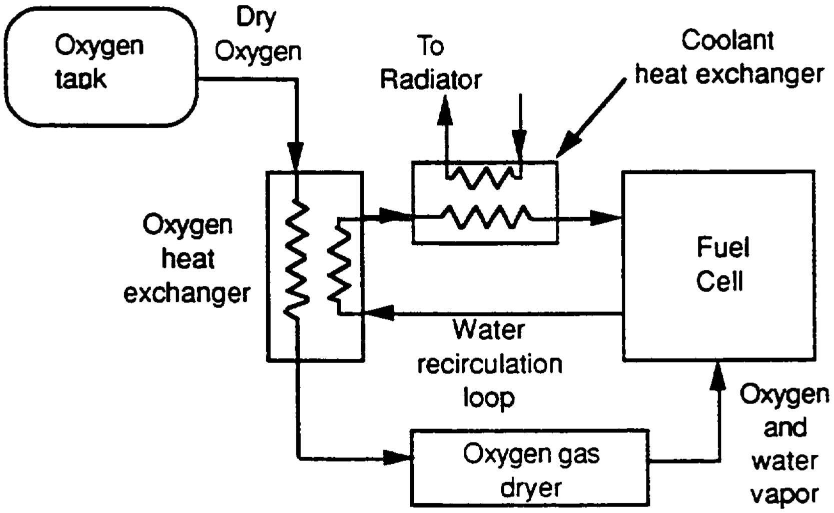


Figure B-7. Water recovery from the oxygen regenerative dryer.

Several approaches have been suggested for recovering moisture from the electrolyzer gases. The key concerns are preventing freezing of water in the lines and the mass loss from the system if the water is not fully recovered each operating cycle. The LANL study (Ref. B-1) proposes letting the water condense and freeze in the tank while keeping the feed lines heated. Some of the water is removed from the tank as a liquid by using a trap. The remainder of the water in the tank freezes. The residual water is recovered by heat input to the tank during the day as additional warm gas from the electrolyzer is introduced.

Another water recovery approach suggested by Hamilton Standard (Ref. B-5) is to dry the gases with regenerative desiccant dryers leaving only a trace of moisture in the gases. This would significantly lower the dew point for the remaining gases to a temperature which is below ambient or would not require much insulation or heat input to prevent condensation. A sacrificial dryer might also remove the remaining trace water. Moisture would be removed from the dryers during fuel cell operation by passing the dry gas from the tanks back through the dryers at the lower fuel cell pressure (need large driving force to recover the water). Potential desiccants include silica gel and molecular sieve (Ref. B-6). Silica gel is used at lower temperatures (below 363 °K) due to its high moisture recycling capacity compared with other industrial desiccants (about 0.35 kg water/kg gel at 303 °K and 80% humidity per Ref. B-7). Molecular sieve is used at higher temperatures (>393 °K) due to its high moisture recycling capacity and physical stability at higher temperatures. Potential configurations for desiccant dehumidifiers include a packed bed, Teflon fiber plates, a corrugated structure, and coated parallel-passages. A coated parallel-passage concept appears to have a great potential to provide an effective dehumidifier. This design consists of parallel-walled passages (laminar flow channels) with fine silica gel particles (80-250 mm) glued to the walls.

Storage Tank Description

For this study, it was assumed that oxygen and hydrogen would be stored at high pressure. The storage tanks will be made of composite materials. A metal liner is overwrapped with carbon or graphite fibers. The liner will probably be aluminum for the hydrogen tank and a corrosion resistant material such as Inconel, niobium, or tantalum for the oxygen and water tanks.

Reliability and life are the key areas of interest for tank design. Thus, materials must be selected which are stable against corrosion and hydrogen embrittlement for the system life. Current composite tank designs may exhibit high stress when driven through large temperature variations, so the liner materials must be carefully matched to the wrap material in terms of the coefficient of thermal expansion. Otherwise, tanks must be thermally controlled to limit temperature changes (may be difficult when going from non-operating to operating status). Tank linings must exhibit very limited corrosion even with pressurized oxygen storage and perhaps even with wet gas storage. Tanks must be rugged enough to survive the transportation phase of deployment. The use of multiple tanks may be required to meet system reliability requirements. Some elementary cladding may be required to provide an element of shielding from meteorites (the system housing may also be used for this purpose).

PP&C Subsystem Description

The PP&C subsystem has not as yet been defined for this concept. Two basic approaches can be taken. In the first approach, which is the same as for the DIPS, the system is designed to provide a constant power and voltage output. This approach requires a dc voltage regulator to process the fuel cell output. In the other approach, the power conditioning is done at the user loads. In this second approach, the fuel cell output can vary with time (i.e., voltage is unregulated). This approach allows the power processing to be optimized for each load. The power input to the electrolyzer module can also be regulated within the RFC power system or as part of the recharging power system.

TECHNOLOGY ISSUES

The key issues for development of a PEM RFC system and their impacts are summarized in Table B-3. Many of the remaining development issues for this concept are related to system integration. Limited work has been done in this area. Major issues of life and reliability can be resolved with simplified system concepts which utilize fewer components and fewer active components. Earlier PEM cells have already proven themselves in space applications (Gemini fuel cells and SSF RFC prototype) and naval applications (electrolysis cells). Thus, it appears that the remaining technical issues can be resolved and a space qualified system developed which has a reasonable life (20,000 hours or more with a 50% duty cycle). The technical issues and potential solutions will be discussed in more detail in the following paragraphs.

TABLE B-3. PEM RFC DEVELOPMENT ISSUES, IMPACTS, AND DEVELOPMENT AREAS

Issues	Impacts	Potential Development Areas
#1- Limited life components and reliability of many parts	•Increased frequency of replacement, maintenance, transportation cost •Mass and complexity of redundant components	•Development of passive system •Long life pumps, drives,valves, and controls
#2 - Material compatibility	•Reliability/life	•Materials for use with high pressure O_2 •Materials for wet gases •Materials immune to hydrogen embrittlement
#3 - Cell temperature and moisture control of fuel cell membrane	•Life	•Thermal control loops •Passive internal fuel cell gas humidifiers •Regenerative gas dryers
#4 - Oxygen in fuel cell water	•Mass/energy loss from the system due to venting of oxygen from water tank	•Internal deoxygenator in fuel cell
#5 - Water in electrolyzer gases	•Tank corrosion if wet gas stored (life and reliability) •Tank insulation mass •Complexity of gas dryer systems •Clogging of lines due to ice •Energy and mass loss due to unrecovered water	•Low mass desiccating regenerative dryers •Tank liner materials •Tank and/or line thermal control

TABLE B-3. PEM RFC DEVELOPMENT ISSUES, IMPACTS, AND DEVELOPMENT AREAS (CONTINUED)

Issues	Impacts	Potential Development Areas
#6 - Large, massive radiator due to low heat rejection temperature	•Increased transportation cost, complicated vehicle design (orientation for stability and mobility)	•Higher temperature cells (higher reject temperature) •Low mass carbon-carbon radiator •Heat pump
#7 - Long duration portable applications	•Transportation cost •Range limits or need to carry PV array	•Low mass storage tanks •Advanced membrane •Low mass PV arrays
#8 - Reactant storage system mass and volume	•Mass/number of trips (transportation cost)	•Low mass high pressure tanks
#9 - High pressure gas storage tank failure	•Increased system mass due to higher factor of safety	•Advanced composite tanks and stack pressure vessel
#10 - Efficiency of electrolysis cell reduced at higher pressure	•Transportation cost •Increased waste heat; larger radiator	•Low mass tanks, PV arrays, and radiators •Tank pressure following
#11 - High Water Purity Requirement	•Life	•Use materials that won't contaminate water •Deionizer

Issue #1 - Limited Life and Reliability

The life of an RFC system for SEI applications is presently unknown. The life of a system depends on the duty cycle for the application among other things. Fixed applications will generally have a 50% duty cycle (half of the time in the fuel cell mode and half of the time in the electrolysis mode). Degradation of the cell membranes is reduced when in a non-operating mode, especially if the temperature is reduced (Ref. B-5). Mobile equipment may have reduced duty cycles (i.e., short operating time and long recharging time). Only equipment common to both the fuel cell and electrolysis modes of operation such as the heat rejection system will be on continuously. A life goal of 20,000 hours for RFC systems (50% duty cycle) was assumed based on current technology and a desire to meet an IOC of 2001 for a lunar landing (Refs. B-1 and B-8). This life goal also corresponded to the life requirement for a lunar mission for earlier Pathfinder studies (Ref. B-8). Long life has been demonstrated (Ref. B-5) in the laboratory on a subscale level for PEM fuel cells (60,000 hours) and electrolysis cells (115,000 hours over 15 years). These have been primarily cell tests with state of the art membranes (Nafion). Advanced cell membranes made by Dow have not undergone long duration life tests. The major area of concern for life is the moving parts (pumps, valves, regulators, etc.) which have not demonstrated long life for space applications. In the LANL study (Ref. B-1), it is stated that this is a major concern based on the design of current space type hardware. However, pump life can be prolonged if the pumps are operated continuously rather than in a cyclic mode (this is probably required anyway for thermal control of the stacks while in a non-operating condition). Long life RFC systems will be obtained by development of advanced designs or with redundancy for less reliable components.

Passive component design is desirable in an RFC where possible to improve system reliability and life. Passive phase separation (i.e., separation of the water, oxygen, and hydrogen) is desirable and should be easy to accomplish based on terrestrial systems which use a gravity feed approach. Microgravity static phase separators developed for Space Station might also be used for SEI applications.

Issue #2 - Material Compatibility

The electrolysis gas products are saturated with water vapor. The presence of water in the oxygen tank may cause corrosion unless proper precautions are taken. The tank liner should be compatible with a corrosive environment or have a protective coating. Inconel 617, niobium, or tantalum will provide corrosion resistance. However, these materials have not been used as liners for composite tanks (at least not by Structural Composites Industries; Ref. B-9). Current composite tanks are made from aluminum or stainless steel. Plastic coatings could be used as long as the gas temperatures were less than about 370 °K. Aluminum could be utilized as the hydrogen tank liner to prevent hydrogen embrittlement.

Issue #3 - Temperature and Moisture Control

Thermal control and membrane moisture control are required elements for stable, high performance of PEM cells. Temperature control is important because the system produces liquid water. Freezing of the water in the lines or cell would result in system failure. Excessive temperatures, on the other hand, can cause separator (electrolyte) alterations. Severe membrane dehydration often can result in irreversible failure, due in part to shrinkage and rupture of the separator sheet. Consequently, the system design must include reliable temperature and relative humidity controls. Moisture must also be removed from electrolysis stack gases. Water and thermal control approaches were previously discussed.

Issue #4 - Oxygen in Fuel Cell Product Water

Oxygen gas will accumulate in the water tank if not removed from the product water stream. Eventually, the gas must be vented to prevent vapor lock. The gas which is vented must be replaced and this increases the system mass. A concept developed by Hamilton Standard (Fig. B-4) removes the oxygen from the water while humidifying the fuel cell reactant gases. However, this concept has not been incorporated into a fuel cell design.

Issue #5 - Water in Electrolyzer Gases

The phase separator will only remove the liquid water from the electrolysis cell product gases. A limited amount of water vapor will remain in a high pressure system due to the low operating temperature (water will condense until the partial pressure of the water gives a saturation temperature which matches the bulk gas temperature). The water vapor remaining in the gases can create problems if left in the system. Storage of wet gases in the tanks would require corrosion resistant materials. If not properly insulated or provided with waste heat, then the water may condense on the tank walls and freeze. Water could also freeze in the feed lines from the tanks and prevent gas flow. Several approaches are possible for dealing with these problems as were previously mentioned.

Major issues for desiccant gas dryers include repeatability (i.e., ability to regenerate the desiccant), efficiency (i.e., the percentage of the water which can be removed from the gas and the amount which can be recovered from the desiccant), parasitic power losses (i.e., pumping power required), performance degradation, heat input and temperature of gas stream required to regenerate the bed, and preventing freezing of the moisture during regeneration.

Work by Hamilton Standard with a molecular sieve dryer has shown the ability to remove over 99% of the water from a gas stream (Ref. B-5). Pressure drop through the bed can be minimized by proper bed design. A parallel flow path design developed by SERI is one option for a compact, low pressure drop dryer (Ref. B-6). Degradation in dryer performance is mainly due to contamination of the bed. Degradation can be controlled by eliminating contaminants in the system and by use of a closed system. Sufficient heat input must be available to the bed for regeneration to prevent freezing of the water. This is because the regeneration (desorption) is endothermic. Heat transfer enhancement through the use of fins or a high conductivity foam material matrix may be required. The bed must be designed such that regeneration can occur at the low temperatures (about 339 °K) which will be available due to heating of the dry gases by the fuel cell product water. In the past, silica desiccant dryers have been regenerated with air at 394 to 478 °K.

Issue #6 - Large Radiator Due to Low Heat Rejection Temperature

The RFC systems require large radiators due to the low reject temperature (333 °K) and high heat sink temperature (256 °K for vertical radiators and no ground cover). For this study, it was assumed that the radiator would be designed for nominal power levels. It was also assumed that peaking power waste heat loads can be handled by the thermal inertia of the system. This assumption needs to be verified analytically and empirically. Particularly, the effect of higher operating temperature and thermal cycling on cell degradation should be examined.

The large radiation area required for RFC systems results in massive radiators which are a significant portion of the total mass of portable RFC systems. The Martian winds have little effect on cooling of the radiators. It is possible that a forced convection cooling system using a fan might reduce the size of the radiators although at the cost of additional power to run the fan and reduced system reliability. This opens up other options in radiator placement which would be of benefit on mobile equipment on Mars. Development of low mass radiators is needed to minimize system mass. Development of higher temperature fuel and electrolysis cells would help to reduce radiator size by increasing the radiator heat rejection temperature. However, a higher operating temperature would reduce cell life (see Table B-4) and require the use of a porous metal support for the electrolyzer cell.

TABLE B-4. PROJECTED ELECTROLYSIS CELL MEMBRANE LIFE

Inlet Water Temperature (°K)	Exit Water Temperature (°K)	Projected Membrane Life (hrs)
322	339	262,800
356	378	87,600
422	444	8,760

Issue #7 - Long Duration Systems

Long duration applications such as the rover vehicles would require a large amount of reactants. Requirements for these vehicles would require heavy tanks (both due to the tank mass and reactant mass). Most of the power system mass for these applications is due to the reactant storage. It is not practical for mobile system to carry the large PV arrays required to run the electrolysis stack for Mars applications (due to low insolation rate). However, some cells may be placed on the vehicle exterior surfaces to provide part of the day power and reduce RFC power requirements. Reduced night time power requirements (i.e., for a manned or telerobotic controlled vehicle; assuming no travel during sleep period) for these applications would reduce the power system size and reactant storage required. The mass of the power system could be reduced by reducing tankage mass (i.e., using composite materials), improving cell efficiency (use of new membranes such as that made by Dow), and reducing parasitic power losses (use of passive thermal control, passive product water collection, etc.).

Issue #8 - Reactant Storage System Mass and Volume

The choice of the on-board reactant storage approach also impacts power system mass. Reactants can be stored either as low pressure gases, high pressure gases, or as low pressure cryogenic liquids. A tradeoff must be made between the mass/volume of gas storage tanks versus the mass/power of the liquefier for cryogenic liquid storage in order to select the best approach. High pressure storage offers smaller tanks but reduced safety over low pressure gas. Liquid cryogenic storage offers smaller, more costly tanks and more complex systems (i.e., thermal management). Cryogenic storage systems might have reduced reliability due to increased numbers of moving parts (refrigeration and liquefaction hardware) and complexity. Additional work would be needed to improve liquefier efficiency and reliability to make this approach viable (i.e., to carry along a liquefier as part of an RFC system).

Issue #9 - High Pressure Tank Safety

The safety issues involved with storing oxygen and hydrogen will be more severe a problem than current usage on the Shuttle. Fuel cell reactants on the Shuttle are stored as low pressure cryogens. However, high pressure (2.07×10^7 N/m^2 or 3,000 psia) PEM electrolyzers are used in submarine applications and the oxygen generated by these cells is stored at high pressure. The space application may require storage of reactants as high pressure gases. Design safety factors for burst pressure will be higher for man-rated systems. The higher pressure requirement will mean heavier tanks and pressure vessels for the electrolysis cells products. Composite gas storage tanks have the advantages of low mass and leakage before burst. Alternatively, reactants may be stored at low pressure as cryogenic liquids. Tank weight will be more of an issue for lunar systems than for Mars systems due to the longer storage time.

Issue #10 - Reduced Efficiency of Electrolysis Cells at Higher Pressure

There is a diffusional inefficiency in PEM electrolysis cells which increases as the operating pressure increases. High pressure operation of an electrolysis cell requires that part of the current input be used to compensate for this back diffusion. For the 2.07×10^7 N/m^2 (3,000 psia) NASA JSC unit, which operates at a nominal current from 180 to 350 amps, it takes 15 amps just for back diffusion compensation (up to 8% of the total input). This results in a larger charging power system to drive the electrolysis unit. Thus, one must tradeoff between the size and mass of a photovoltaic array (or other power system) and the size and mass of storage tanks required for lower pressure systems. For the short operating times required on Mars it is probably better to operate at a lower pressure than for a lunar system. However, this tradeoff needs to be quantified.

Issue #11 - High Water Purity Requirement

PEM separators are susceptible to ionic contamination in the electrolysis mode. The presence of metallic cations must be controlled. Thus, the use of high purity water is required for the PEM electrolysis cell to maintain performance (preserving high activity of the electrocatalysts and maintaining high ion exchange capability of the membranes). In addition, the impact of dust contamination in the system must be studied to determine potential system impacts. Potential development of new membranes which are more tolerant of impurities is a possibility. Alternatively, the system should be completely clean when assembled and be made of materials which will not contaminate the water. A closed system where reactants are never replaced would eliminate external contamination. These latter two approaches would eliminate the need for maintenance of a water purifier such as a deionizer and any particulate filters. However, a closed system would eliminate the option of replacing reactants from a centralized electrolyzer (i.e., primary fuel cell approach).

TECHNOLOGY ASSESSMENT

The technology bases were assessed for the following major RFC subsystems:

- fuel cell stack;
- electrolysis cell stack;
- thermal management;
- water management;
- reactant storage tanks; and
- PP&C.

The technology readiness of each assembly was evaluated using the NASA technology readiness levels. These evaluations are summarized in Table B-5 which shows that the RFC has technology readiness levels ranging from 3 to 5, depending on the particular subsystem. The technology base for each assembly is briefly discussed in the following sections.

TABLE B-5. RFC TECHNOLOGY ASSESSMENT

Subsystem	Technology Readiness Level	Comments
Fuel cell stack	3.5	Early design flown on Gemini; prototype developed for space station RFC based on earlier Hamilton Standard design; new International Fuel Cell design not flown (engineering qualified in 2.5 yrs for DARPA program)
Electrolysis cell stack	4	Large database for naval applications; prototype developed for space station applications
Active thermal management	3	Advanced C/C heat pipe radiator development currently underway; long life active components not developed for space applications; some sealed water pumps and drives for terrestrial applications have demonstrated long life
Water management	4	Passive phase separation designed for Space Station Freedom (SSF) prototype; silica gel dryers widely used for gas drying in terrestrial applications; limited experience with regenerative gas dryer systems
Reactant storage tanks	5	Small tanks successfully flown; need corrosion resistant liner development
PP&C	5	SSF

RFC System State-of-the-Art

Hamilton Standard designed, developed, tested, and delivered a 1.5 kW PEM breadboard RFC system to JSC in January 1983 (Ref. B-5). The RFC breadboard system was tested for 2,000 total ninety minute orbital cycles (1,630 by NASA). The fuel cell module of the RFC was later replaced by an advanced module and tested for about 500 hours. This breadboard was not tested in a relevant environment (i.e., vacuum or low pressure carbon dioxide, low gravity, day/night thermal cycles, etc.). In addition, the breadboard did not include key components such as composite tanks, radiators, and long life active thermal control components.

Although the fuel cell technology is fairly well developed in PEM systems, the system integration of the accessory components and cell stack is not as mature as that of the alkaline fuel cells. The technology readiness of the PEM RFC is estimated to be 3.5 for the current application.

Fuel Cell State-of-the-Art

A PEM fuel cell developed by Hamilton Standard (United Technologies) was used on the Gemini missions from 1962-66 (Ref. B-10). After the Gemini space flights, Hamilton Standard pursued further development of PEM fuel cell technology. The major breakthrough was the replacement of polystyrene sulfonic acid ion exchange membrane by perfluorinated sulfonic acid polymer, Nafion, produced by DuPont, as the electrolyte. PEM fuel cell technology has since advanced with the introduction of the Dow experimental membrane. The Dow membrane has greatly increased the current density of PEM systems over the current densities available from DuPont's former state-of-the-art membrane, Nafion 117. Dow has not yet started production of their membrane , but is supplying it to fuel cell manufacturers for testing and evaluation.

Cell lives of 60,000 hours (6.85 years) in the laboratory have been achieved by Hamilton Standard with PEM fuel cells (Refs. B-4 and B-5) because there are no corrosive electrolytes to cause contamination. PEM fuel cells can operate with high concentrations of gases like CO_2, whereas the KOH electrolyte of alkaline fuel cells would react with the CO_2 and cause precipitation.

A PEM fuel cell subsystem was developed by Hamilton Standard in the 1980s for a RFC demonstrator for the Space Station Freedom (Ref. B-2). The fuel cell had a 1 to 2 kWe rating. The RFC demonstrator underwent parametric testing at the factory prior to its delivery to NASA/JSC.

Treadwell Corporation has designed and built a PEM fuel cell stack and associated test stand (Ref. B-11). The stack was designed for an output power of 10 - 30 kW. Various stacks have been tested. This fuel cell system was designed for autonomous underwater vehicles.

Ballard Technologies Corporation in Canada has built small demonstrator stacks with the Dow membrane and was the first to achieve high power densities in a solid polymer electrolyte fuel cell (Ref. B-10). The Ballard design appears to be similar to the Hamilton Standard fuel cell design.

Siemens in Germany (under a license from Hamilton Standard) is also using the PEM technology to develop fuel cell systems for submarine power systems (Refs. B-10 and B-12).

LANL has two of the Dow PEM fuel cells on test, achieving 0.92 V at 2153 A/m^2 (Ref B-10). Dow has made a commitment to provide membranes to a product specification; whereas, they were previously in process development and membrane quality/consistency were not up to par for commercial use.

PEM fuel cells are well suited to passive water removal. The absence of a liquid electrolyte that has narrow concentration limits makes water management less of a problem. Ergenics Power Systems, Inc., is developing a flight-qualified 200 W fuel cell with passive water and heat removal for a Space Station extravehicular mobility unit (Ref. B-10).

International Fuel Cells (IFC) has tested a 16 cell, 5 kW stack using Nafion membranes and is now evaluating the Dow membranes. This is a new PEM design which is different from the fuel cell which flew on Gemini. IFC also worked on a "static" PEM fuel cell (Ref. B-13). This concept eliminated the power consuming pumps associated with the management of the product water. The design also incorporated heat pipes into the system to create a "static" waste heat management subsystem which eliminated the cooling subsystem parasitic power loss. This approach offered significantly improved reliability and higher system efficiency. IFC completed breadboard experiments and validated this system concept. IFC has since changed their design (Ref. B-14). The latest design has no heat pipes and requires a cooling loop with a pump. However, the water removal still utilizes a static approach. This concept is proprietary and few details were available from IFC. However, this concept uses graphite plates and either Dow or Nafion 117 membranes (Nafion is the baseline). IFC has tested both single cells and short stacks with its latest design. IFC has a DARPA contract to produce a 7.5 kWe fuel cell for unmanned underwater applications. This power plant will be available (engineering qualified) in 2.5 years. The life of this fuel cell is expected to be a few thousand hours. The DARPA fuel cell technology should be suitable for space since it is not affected by a zero gravity environment.

Electrolysis Cell Stack State-of-the-Art

Hamilton Standard has an extensive data base in high-pressure electrolysis. The 2.07×10^7 N/m^2 (3,000 psi) cell design is currently used in the oxygen generation plant (OGP) developed for the U.S. Navy and in the production units for the British Navy nuclear submarines. Hamilton Standard has over 20 years experience building PEM electrolyzers for the Navy (Ref. B-10). They have demonstrated 13 years of continuous usage of a PEM electrolyzer cell in the laboratory. U.K. Navy electrolyzer cells have accumulated a total of 69,000 system hours of usage as of January 1992 without any failures. One Navy electrolyzer cell stack has accumulated over 13,000 hours of usage at sea over a 5 year period (Ref. B-5).

During the 1980s, three demonstrators were developed by Hamilton Standard (Ref. B-2). These electrolyzers were fabricated and then tested by NASA. Each of these systems made use of the identical 213 cm^2 SPE® water electrolyzer design used for naval applications.

The first of these systems was a PEM RFC demonstrator for Space Station Freedom. Over 2,000 simulated orbital cycles were accumulated on this hardware. This program demonstrated a closed system fluid cycle balance, direct solar array/electrolyzer voltage/current control compatibility (no power conditioning required), and an energy storage efficiency of 48% with the electrolyzer at ambient temperature. Later in the program the PEM fuel cell was replaced by an alkaline fuel cell and the system was operated for 100 cycles with no problems. This test showed the compatibility between a PEM electrolyzer and an alkaline fuel cell (i.e., no KOH ions went to the PEM electrolyzer through the product water and no sulfonic acid groups passed through the PEM electrolyzer gases to the fuel cell; some people had thought that the acid and base in each unit would mix and neutralize each other). Recently, some of the electrolyzer cells were replaced by high performance cells using the Dow membrane. The electrolyzer module underwent additional testing and showed significant performance improvement, especially at higher current densities.

The second Hamilton Standard demonstrator was an oxygen generator assembly developed under contract to Boeing Aerospace and Electronics Company (Ref. B-2). The operating pressure, temperature and current density of this demonstrator are within the experience of naval applications. However this demonstrator differed from the Navy data base because of the need to operate in a microgravity environment and use processed hygiene water as the feedstock. Two membrane static phase separators are used to replace the pressure vessel phase separators used previously. The demonstrator was activated at NASA/MSFC in November 1990 and operated for 529 hours which exceeded the test objective of 450 hours. This program successfully demonstrated the operation of microgravity phase separators. There are continuing tests of this unit to improve the cell voltage performance.

The third Hamilton Standard demonstrator system was developed to show the feasibility of producing 2.07×10^7 N/m^2 (3,000 psi) hydrogen and oxygen on orbit for periodic rocket motor firing to maintain Space Station Freedom orbital altitude (Ref. B-2). Under NASA sponsorship, initial work was performed to convert the heavy 2.07×10^7 N/m^2 (3,000 psia) naval SPE® electrolyzer design into a space flight configuration (Ref. B-4). This required development of a lighter and smaller package. Changes were made to the supporting pressure vessel and fluid manifold. The use of two torispherical domes opposed on either side of a central fluid plate allowed for a wall thickness of as low as one quarter of an inch when using Inconel or other high strength materials. The fluid plate manifold is pressure balanced between the two pneumatic domes which eliminates the need for a thick plate to resist the gas pressure load, as used in the Navy hardware. This work produced a prototype cell stack for space applications that weighs less than 91 kg total (down from 454 kg for the 100-cell naval cell stack and pressure vessel). This unit was delivered in 1990. This demonstrator has been set up and operated intermittently at NASA/JSC during the last year. This cell was recently tested by JSC for 500 hours (Ref. B-2). This unit operates at 322 °K at a thermal efficiency greater than 70% (defined as the ratio of the power input minus the heat rejected to the power input).

Thermal Management Subsystem State-of-the-Art

NASA LeRC is currently carrying out an integrated multi-element project for the development of space heat rejection subsystems with special emphasis on low mass radiators in support of SEI power system technology (Ref. B-15). This effort involves both in-house and contracted efforts. Contracted efforts involving Rockwell International (RI) and Space Power Incorporated (SPI) are aimed at the development of advanced radiator concepts (ARC). NASA LeRC is also involved in a joint program with DOE to demonstrate a flexible fabric heat pipe radiator concept being developed by Pacific Northwest Laboratories (PNL). In-house work at NASA LeRC is designed to guide and support the overall program by system integration studies, heat pipe testing and analytical code development, radiator surface morphology alteration for emissivity enhancement, and composite materials research focused on the development of low mass, high conductivity fins. This program is concentrating on technologies capable of development before the end of the decade for both surface power and nuclear electric propulsion (NEP) applications.

Specific objectives of the ARC contracts are to achieve specific mass values <5 kg/m^2 with radiator surface emissivities of 0.85 or higher at typical radiator operating temperatures, and reliability values of at least 0.99 for the heat rejection subsystem over a ten year life. These figures represent a factor of two improvement over the currently considered heat rejection subsystem for SP-100, and even greater improvement factors for state-of-the-art heat rejection systems used in current spacecraft applications.

Presently, Rocketdyne division of Rockwell International is involved in the development of a carbon-carbon (C-C) or graphite heat pipe radiator panel. Rocketdyne has a NASA contract entitled "SP-100 Advanced Radiator Concepts" which started in 1987 and is part of the CSTI (Civilian Space Technology Initiative) program. This program will be completed in January of 1993. This program involves development of a C-C heat pipe suitable for use in an SP-100 radiator. However, this technology will be suitable for other radiator applications as well including RFC power systems (using other working fluids such as water and/or ammonia). The hope is that this program will develop generic technology which can be used for all future space radiators because of its reduced mass compared to metal heat pipe radiators. Carbon-carbon is a new structural material which requires different fabrication techniques than for metals or composites. A key objective of this program is to demonstrate the ability to fabricate C-C heat pipes with a thin metal liner, wicks, caps, and fill tube. The brazing technique required to attach liners is the same as would be required to attach manifolds to the heat pipes. Heat pipes filled with potassium will be designed, fabricated, and tested during this program. Thermal cycle testing will be done in a vacuum to demonstrate the integrity of the heat pipes.

Beginning in 1989, small samples of heat pipe panels were fabricated by Rocketdyne as part of the CSTI program. The next phase of the program will involve fabrication and testing of complete heat pipes about a meter long.

The working fluids and temperatures for the SP-100 system would be different than for lower temperature RFC systems. Thus, Rocketdyne has an on-going IR&D effort which involves development of a high pressure water C-C heat pipe for lower temperature applications such as

RFCs. This effort will eventually include fabrication and testing of a heat pipe. Rocketdyne is also looking at a higher thermal conductivity C-C heat pipe design.

Recent Rocketdyne studies have shown that a C-C heat pipe with water or ammonia is competitive on a mass basis with early metal heat pipe radiator technology developed in the late 1960s and early 1970s as well as pumped loop radiators. Rocketdyne studies have estimated the C-C radiator specific mass to be about 3 kg/m^2.

The objective of the joint NASA LeRC/Air Force program with PNL (Light Weight Advanced Ceramic Fiber [ACF] Heat Pipe Radiators) is to demonstrate the feasibility of a low mass ceramic fabric/metal liner heat pipe for a wide range of operating temperatures and working fluids (Ref. B-15). Specifically, the NASA LeRC objectives are to develop this concept for application to space radiators with operating temperatures below 500 °K using water as the working fluid. The specific mass goal for these heat pipes is 3 kg/m^2 or less at a surface emissivity of greater than 0.85.

Several heat pipes were built for the ACF program using titanium and copper foil material for containment of the water working fluid. A heat pipe with an eight mil Ti liner was demonstrated in early January 1991. An innovative "Uniskan Roller Extrusion" process has also been developed at PNL and used to draw 30 mil wall tubing to a 2 mil foil liner in one pass. The water heat pipes fabricated for LeRC have been subjected to a test program to evaluate performance and reliability at demanding operating conditions.

Future thrusts of the ACF program will be to perfect the heat pipe fabrication procedure using very thin (0.025 to 0.05 mm) foil liners which are internally texturized by exposure to high pressures (Ref. B-15). Plans will also be developed to perform hyper-velocity and ballistic velocity impact tests in order to determine if secondary fragments from a penetrated heat pipe will result in failures of neighboring heat pipes. Another major challenge will be to design a heat pipe with high conductivity, low mass fins as a first step toward the fabrication of low mass radiator panels.

The NASA LeRC in-house materials program includes radiator surface morphology alteration by arc texturing for emissivity enhancement purposes (Ref. B-15). Emissivity enhancement has been demonstrated for graphite-copper samples.

Another important part of the thermal management subsystem is the active controls for controlling fluid flowrates and pressures. This involves the development of long life pumps, valves, and regulators. Hamilton Standard has used 2.07x10^7 N/m^2 (3,000 psia) water pumps for its U.S. Navy electrolysis units (made by J.C. Carter Company per Ref. B-5). These pumps have lasted over 10,000 hours. Hamilton Standard also has used a small 0.91 m^3/h (4 GPM) pump for recirculating water to a fuel cell which lasted over 10,000 hours. Existing active convective transport hardware (pumps, centrifugal water separators, fans) have also been developed for space systems by GE and Hamilton Standard, but not for long life (Ref. B-1). Experts in the field believe that these components will have lifetimes of less than 2,000 hours in a space environment.

Water Management Subsystem State-of-the-Art

Water management involves both gas humidification to maintain proper cell membrane moisture content as well as gas dehumidification to prevent water condensation and freezing. Work has been done in both of these as areas, but not for space qualified hardware.

Gas humidification for production PEM fuel cells has been successfully done by Hamilton Standard using the approach shown in Figure B-6. An improved approach which removes oxygen from the product water was shown in Figure B-4. This approach was demonstrated in the laboratory in 1989, outside of a fuel cell (Ref. B-5). This approach has not yet been integrated into a PEM fuel cell design.

Water vapor can be removed from gases using condensation, absorption, adsorption, or a combination of these techniques. Gas drying is a well-known technical process. General Dynamics proposed (Ref. B-16) a system which utilizes condensation for removing 99.9% of the moisture from oxygen and hydrogen gases. The removal of the remaining moisture was assumed to be accomplished using absorption-adsorption techniques in existing ground liquefaction systems. General Dynamics adapted a system developed by Atlas Copco called MD sorption drying for propellant processing. This process recovers all of the water during the electrolysis portion of the cycle. However this is a very complex system with many moving parts (pumps, valves, and rotor in sorption dryer). In addition, this system increases the radiator size by reducing the heat rejection temperature.

Work was done with regenerative gas dryers for Hamilton Standard by the UTC Research Center in 1987 (Ref. B-5). These dryers use a molecular sieve to remove water. In tests, a humidified gas stream was run through a dryer bed for 36 hours without breakthrough. A 10% bed loading (amount of bed volume which was water) was achieved. An outlet dew point of less than 183 °K was obtained with a 100 to 1 gas volume ratio. Such a dryer removes about 99.9% of the water vapor from a gas. Based on this work, a 21 kg desiccant could be made regenerable and deliver 2,939 kg of dry gas (about twice that required for a 12.5 kW lunar base system). A much smaller unit would be required for a Mars system. Hamilton Standard has also looked at methods to reduce dew point temperature down to liquid hydrogen temperatures.

Reactant Storage Tank State-of-the-Art

Gas storage tanks have had considerable engineering advancement, both in earlier NASA programs and throughout the commercial sector. Gas tanks have become safe, reliable commodities, widely used in science and technology. Composite tanks with a metallic liner and a polymer wrap are generally considered for advanced space applications.

Composite tanks have had considerable recent development for a wide range of sizes (Ref. B-9). Structural Composite Industries (SCI) has built many tanks for both terrestrial and space applications. These cylindrical tanks have either aluminum or stainless steel liners. These liners are seamless and are made from plate stock without welding. SCI has developed tanks for Space Station, Brilliant Pebbles, HEDI, propellant tanks for launch vehicles, communications satellites, and Pegasus projects among others. Eleven of these tanks have been launched into space. Tank sizes have ranged from 754 cm^3 to 0.66 m^3 (Space Station). A recent effort involved a tank which is 4.06 m long and has a 0.53 m outer diameter. A current effort involves the development of a 1.42 m diameter tank. Tank pressures have gone as high as 1.034×10^8 N/m^2 (15,000 psia) for operation and 2.068×10^8 N/m^2 (30,000 psia) for burst. Tanks for SDI missions are designed for 5 year life. Additional liner development may be required for storage of wet oxygen.

PP&C Subsystem State-of-the-Art

The PP&C system design is based on a reasonable electronics component evolution, and there are no significant technology issues associated with its development. It will be necessary to fabricate breadboard hardware for testing and evaluation purposes, but there is no need to initiate any advanced component development. Most of the hardware elements have been or shortly will be operating in a relevant environment on other spacecraft or the SSF electrical power subsystem.

MAJOR DEVELOPMENT TASKS

The development program was divided into 9 major tasks. The first six tasks are component development tasks. The remaining tasks involve the system design, fabrication, integration, safety assurance, and testing.

Testing will be done on the component, subsystem (fuel cell stack, electrolysis cell stack, thermal management, gas dryers, storage tanks), and system level (qualification testing). The system level tests will show any possible negative interactions between subsystems.

The RFC development tasks are described in the following sections. Each task will include a section on objectives and a statement of work. The task descriptions are only approximate and depend on the RFC design chosen.

A 3 kWe nominal power (peaking power of 10 to 15 kWe) module with an energy storage capability of 32 kW h (net output) will be selected for the ground engineering system (GES) to take the RFC technology through Technology Maturity Level 6 (system validation model demonstrated in a simulated environment). Other power module sizes may be built and qualified for flight systems.

Task 1. PEM Fuel Cell Module Development

Objectives: Develop a full scale, long life (10,000 hours continuous) flight weight fuel cell module which can be integrated into a mobile RFC power system. Demonstrate materials compatibility, safety, and performance margins.

Statement of Work: This effort was divided into the following subtasks:

Subtask 1.1 Preliminary Fuel Cell Module Development. Demonstrate the feasibility of the selected fuel cell design for planetary surface applications. Investigate stack sealing materials, plates, membranes, humidifier, and diluent control. Test a prototype fuel cell stack to demonstrate materials compatibility, and safety.

Subtask 1.2 Final Fuel Cell Module Design. Design a full scale flight weight fuel cell modules (one or more different sizes).

Subtask 1.3 Fuel Cell Module Fabrication, Assembly, and Testing. Build and test (performance, mechanical, and thermal cycling) the fuel cell module. After initial breadboard validation testing is complete, perform additional module testing in a relevant environment.

Task 2. PEM Electrolysis Cell Module Development

Objectives: Develop a full scale, long life (10,000 hours continuous) flight weight electrolysis module which can be integrated into a mobile RFC power system. Demonstrate materials compatibility, safety, and performance margins.

Statement of Work: This effort was divided into the following subtasks:

Subtask 2.1 Preliminary Electrolysis Cell Module Development Design. Demonstrate feasibility of the selected concept for space applications. Investigate options for the cell membranes, pressure vessel, central fluid plate, and electrical connections. Demonstrate materials compatibility, performance, and safety using a prototype cell stack.

Subtask 2.2 Final Electrolysis Module Design - Design a full scale flight weight electrolysis cell module.

Subtask 2.3 Electrolysis Module Fabrication, Assembly, and Testing. Build and test (performance, mechanical, and thermal cycling) the full scale electrolysis cell module. After initial breadboard validation testing is complete, perform additional module testing in a relevant environment.

Task 3. Thermal Management Subsystem Development

Objectives: Develop and demonstrate a low mass, reliable heat pipe radiator. Develop and demonstrate long-life (20,000 hours continuous) active thermal control components (pumps, valves, regulators, etc.), as needed. Develop and demonstrate thermal control concepts for all components in the RFC system.

Statement of Work: This effort was divided into the following subtasks:

Subtask 3.1 Heat Pipe Demonstration. Fabricate and test representative length heat pipes to fully characterize heat pipe performance. Compare test results with predicted performance over the anticipated range of operating conditions including startup, shutdown, and restart.

Perform limited life testing of the heat pipes. Identify deterioration mechanisms, measure deterioration rates, and assess the adequacy of assembly and cleaning procedures. Withdraw heat pipe samples sequentially throughout the test period. Drain, section, and analyze the samples for corrosion, carbon diffusion, braze stability, and any other signs of deterioration.

In conjunction with the heat pipe performance and life testing, develop and demonstrate techniques for nondestructive examination (NDE) of the heat pipe elements and assembly. Assess the adequacy of liner bonding and weld joints. Determine if the correct fit-ups and interfaces were obtained in the assembled piece.

Subtask 3.2 Radiator Enhancement. Increase the applicability of the radiator concept by performing various enhancement tasks. Examine survivability options and extended length heat pipes.

Perform analytical assessment, testing, and enhancement design development to insure long life for both lunar and Mars missions. Consider natural threats such as micrometeoroids and dust erosion (Mars).

Fabricate a long heat pipe. Develop alternate techniques for fitting the liner into the tube. Investigate alternative tube fabrication, liner fabrication, and coating processes.

Subtask 3.3 Heat Pipe Integration and Testing. Develop and demonstrate techniques for thermal and mechanical bonding of the heat pipe to the radiator manifold. Subsequently, demonstrate heat pipe integration into a representative radiator section. Test the radiator section to provide an accurate overall heat transfer coefficient. Verify radiator dynamics and performance. Include a surrogate manifold section and a limited number of heat pipes in the demonstrator. Test the assembled unit in a cold wall, vacuum chamber, simulated space environment. Validate temperature drop predictions, verify manifold design, and assess component interactions.

Subtask 3.4 Radiator Module Design, Assembly, Fabrication, and Testing. Develop the detailed design of the radiator subsystem. Resolve specific design issues such as the effects of differential thermal expansion, deployment features, monitoring instrumentation and control equipment, local insulation, and trace heating of the manifold and piping.

Design, fabricate, and test a representative full scale heat pipe radiator and interface heat exchanger. Complete performance, mechanical (stress, shock, and vibration), and thermal cycling tests.

Subtask 3.5 Preliminary Active Component Development. Perform preliminary development of water circulation pumps and other active fluid components. Identify life limiting components and failure mechanisms of current designs. Develop design approaches with improved life capabilities. Design and build prototype components. Demonstrate materials compatibility, safety, and performance margins.

Subtask 3.6 Final Active Component Design. Design the full scale components and develop the active control subsystem concept.

Subtask 3.7 Active Control Subsystem Fabrication, Assembly, and Testing. The final components will be designed and integrated into a fluid subsystem test bed with simulated thermal loads and pressure drops for other RFC components. Testing shall include performance, shock, pressure, and safety tests. After initial breadboard validation testing is complete, perform additional testing in a relevant environment.

Task 4. Water Management Subsystem Development

Objectives: Develop reliable, long life (20,000 hours continuous) regenerative dryers for removing water vapor from the electrolyzer product gases. Develop an approach for recovering water from dryers. Develop a reactant gas humidification device for the fuel cell for maintaining a proper membrane moisture content.

Statement of Work: This effort was divided into the following subtasks:

Subtask 4.1 Gas Dryer Development. Develop regenerative gas dryers for use with wet hydrogen and oxygen gas streams. Design the dryers to remove the majority of the water vapor from the gas streams with reasonable size and mass hardware. Build prototype dryers. Test the dryers in the adsorption and desorption modes. Determine materials compatibility, regenerability, cyclic performance degradation, and efficiency.

Perform studies to determine the effect on system performance of any remaining moisture after leaving the regenerative dryers. Design an approach for recovery of the remaining moisture from the gas system. Develop additional components as necessary to handle this remaining moisture. Demonstrate analytically and empirically the performance of this additional hardware.

Subtask 4.2 Gas Humidifer Development. Complete the development of reactant humidifers which are integral to the fuel cell. This will include modification of existing humidifier concepts which currrently work external to the fuel cell. In addition, this task will be done in parallel with fuel cell development to allow for proper integration with the fuel cell stack design.

Task 5. Reactant Storage Tanks

Objectives: Develop reactant tanks for storage of wet hydrogen gas, wet oxygen gas, and water. Develop tank and tank feed line thermal management (insulation and/or heat recovery from fuel cell stack) approaches to prevent composite tank failure, and prevent ice blockage of gas lines and regulators.

Statement of Work: Design, fabricate, and test the reactant storage tanks. Develop corrosion resistant liners and fabrication techniques for the oxygen and water tanks. Consider the requirements for micrometeoride protection in the tankage design.

Develop insulated and/or heated tank concepts which prevent water condensation and freezing in the gas storage tanks and exit lines. Demonstrate proper transient performance of the tanks using analysis techniques. Build and test prototype tanks. Demonstrate materials compatibility and safety. Perform proof testing, pressure cycling (at different temperatures), thermal cycling (while under pressure), puncture resistance, and mechanical shock tests. Measure tank permeability to estimate leakage losses, especially for hydrogen.

Task 6. PP&C Subsystem Development

Objectives: Develop electronic devices for regulating output voltage and for controlling RFC operation. Demonstrate adequate steady state and transient performance, and immunity to the environment (including launch and operating). Demonstrate the software capability to handle power system nominal operation and failure modes.

Statement of Work: This effort was divided into the following subtasks:

Subtask 6.1 Electrical Components. Build breadboard units to demonstrate and check functional performance of the individual component circuit designs. Incorporate any necessary design modifications and improvements into brassboard units. Verify functional performance within the constraints of the actual component configuration. Fabricate prototype units and perform a series of tests using simulated input and output loads. Test the controller to validate the operating system software. Include the following testing:

- start up, steady state and transient control;
- failure simulation and detection and switching;
- effects of temperature extremes and thermal shock;
- effects of environment;
- shock and vibration;
- cold plate heat loads; and
- EMI generation and susceptibility.

Subtask 6.2 Software. Check out the controller software using simulated inputs and outputs.

Task 7. Ground Engineering System (GES) Testing

Objectives: Design a system concept which will meet both lunar and Mars mobile power applications. Show the concept feasibility. Verify adequate performance and life for an integrated system.

Statement of Work: This effort was divided into the following subtasks:

Task 7.1 GES Design. Identify and characterize specific power system applications. Determine power system requirements. Define optimum power system module size based on trade studies which minimize the overall life cycle cost of the power system for all applications (trade between development cost and mass).

Identify concepts which can meet early lunar base applications. Power system enhancements, if any, required for Mars applications will be identified. Perform tradeoff studies (performance, reliability, risk, safety, and life cycle cost) to select the optimum system design. Complete feasibility studies for the selected concept including off-design and transient analysis. Definitize the remaining hardware development tasks based on the system concept chosen.

Design a complete RFC system to at least meet the minimum life requirement of 20,000 hours (10,000 hours for the fuel cell and electrolysis cell stacks). Integrate the entire thermal and water control system including pumps, controls, lines, valves, heat exchangers, tanks, and radiator. Perform both design point and transient analyses to verify the design. Perform this task in parallel with the component development tasks to insure proper system characteristics

Task 7.2 GES Performance Tests. Manufacture the GES components and assemble. Pressure and leak check the assembly, fill the water storage tank, complete a electrical check-out, and check out all active devices to the extent possible.

Design the test program for the GES to verify all performance characteristics of the unit in conjunction with subjecting the GES to acceptance level tests. Conduct performance and thermal vacuum tests during a single thermal vacuum test sequence to demonstrate technical capabilities while minimizing program cost. Complete the following performance testing activities:

- checkout and refine subsystem assembly procedures;
- verify operation of ground support equipment (GSE) and interfaces with the GES;
- verify start up capability;
- verify steady state performance characteristics;
- establish effective operating range of the system;
- check out software and verify autonomy (include all operational modes and simulate failure modes);
- simulate normal switchover from one module to another;
- simulate failures to trigger module switchover; and

- establish system sensitivity to off-normal conditions such as: (1) partial loss of radiator cooling, (2) varying radiator heat removal profile, (3) partial loss of electronic cooling, and (4) temporary losses of heat sink with varying duration.

Task 7.3 GES Life Test. Refurbish the GES after completion of performance, acceptance, and margin tests. Refurbishment will include the following:

- replacing or repairing components as necessary to ensure that the GES is returned to its as-built condition;
- addition of special instrumentation required for the life test phase; and
- modification of the main radiator heat removal system or replacement of the main radiator with a dump heat exchanger.

Install instrumentation to provide a comprehensive diagnosis of the "health" of the GES and to monitor for degradation of major assemblies and individual components. Place the GES on a 20,000 hour life test (50% duty cycle for the fuel cell and electrolysis cell stacks). Operate the GES at its nominal operating point, with expected GES variations in power output and environment.

Disassemble the GES for diagnosis at the end of the life test. Determine areas to be examined by an analysis of the health monitoring data and from the reliability analysis predictions.

Task 8. Qualification Program Testing

Objectives: Fabricate the qualification units (QUs) Verify adequate performance and life for the QUs.

Statement of Work: Develop a comprehensive performance and dynamic testing program which will provide a formal demonstration that the RFC system will perform as designed after being subjected to simulated launch conditions. Select launch environmental conditions which envelope the probable intensities developed by various launch scenarios.

Begin the qualification effort (a typical approach is shown in Figure B-8) by qualifying the assemblies. Fabricate and assemble the qualified production items into the QU. Qualify the QU by the rules for space vehicle qualification.

This effort was divided into the following subtasks:

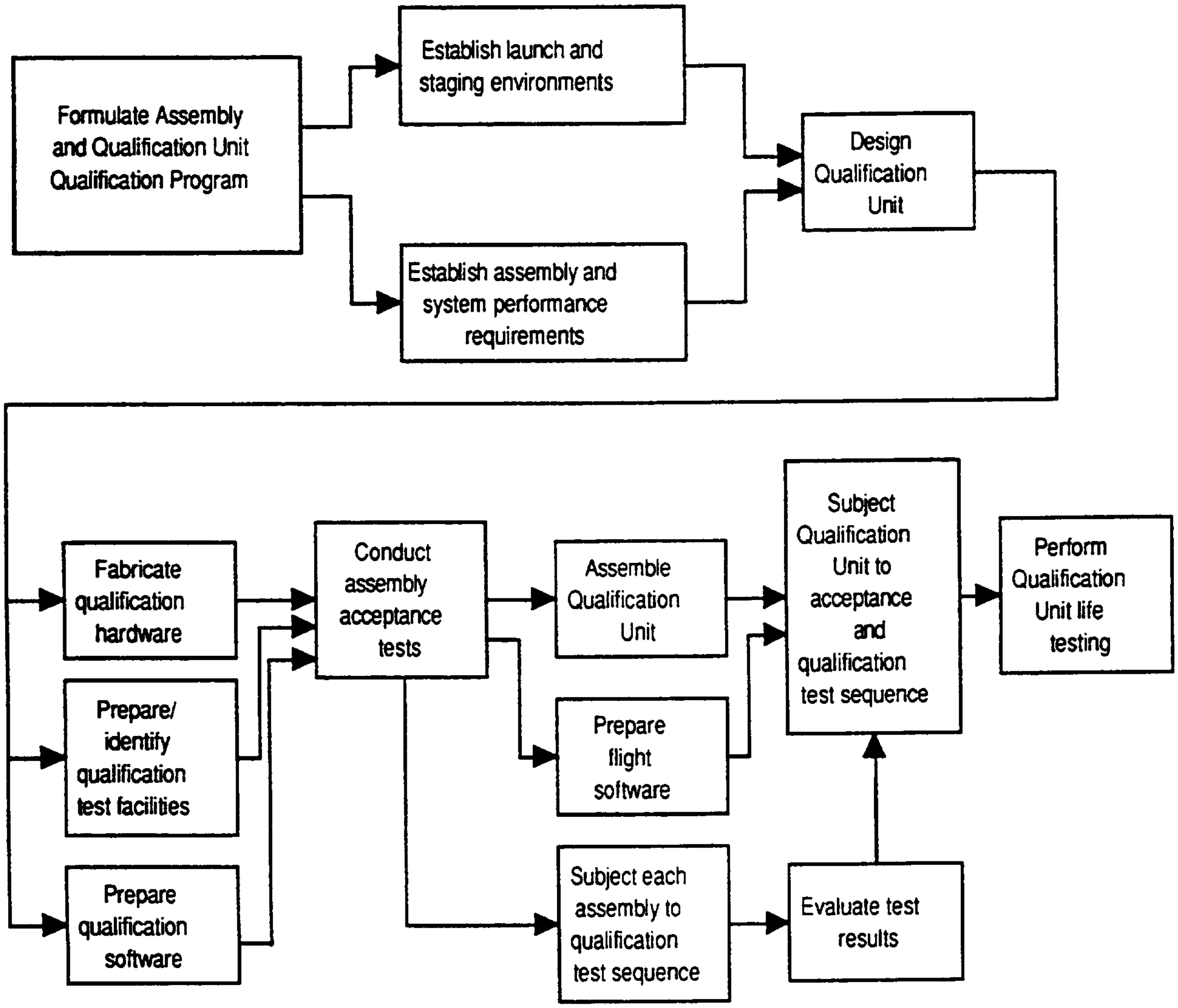


Figure B-8. RFC power system qualification program.

Subtask 8.1 Qualification Performance Testing. Conduct performance testing at each level to verify that each item performs as designed. Perform dynamic testing to verify the capability of the RFC system to withstand launch loads, including acoustic, pyroshock and vibrational. A possible performance and dynamic qualification test sequence for components and assemblies is shown in the matrix of Figure B-9. The corresponding qualification test sequence for the QU is shown in Figure B-10.

Subtask 8.2 Qualification Life Testing (Optional). After completion of the qualification performance testing, partially disassemble the QU and examine the unit. Refurbish the QU as required and modify for endurance testing in air, as described for the GES. Life test the QU for 8,800 hours or more.

Component or Subassembly	Functional (1)	Leak	Pyroshock	Functional (2)	Random Vibration	Functional (1)	Leak	Acceleration	Functional (1)	Thermal Cycling	Functional (2)	Thermal Vacuum	Functional (2)	Pressure	Leak	EMC	Life	Functional (1)
Fuel Cell Stack	X	X	X	X	X	X	X			X	X	X	X	X	X	X	X	X
Electrolysis Cell Stack	X	X	X	X	X	X	X			X	X	X	X	X	X	X	X	X
Reactant Storage	X	X	X	X	X	X	X			X	X	X	X	X	X		X	X
Thermal Management	X	X	X	X	X	X	X					X	X	X	X		X	X
Water Management	X	X	X	X	X	X	X			X	X	X	X	X	X		X	X
PMAD	X		X	X	X	X				X	X	X	X			X	X	X
Structure	X		X	X	X	X		X	X	X	X						X	X

Figure B-9. RFC power system assembly qualification test matrix.

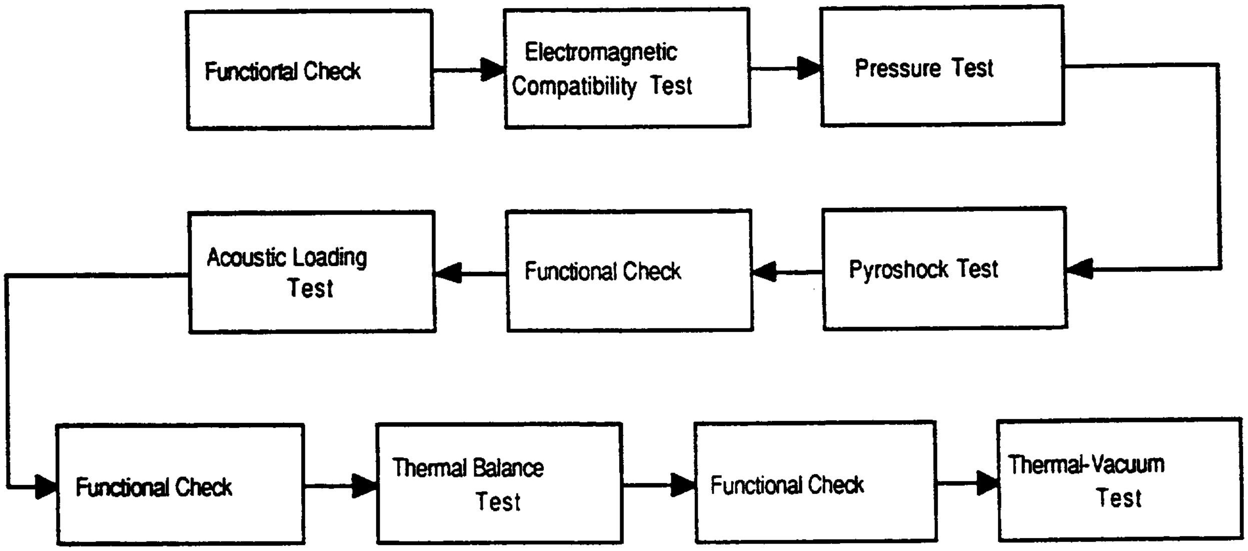


Figure B-10. RFC power system QU test sequence.

Task 9. Flight Unit (FU) System Program

Objectives: Fabricate the flight systems and acceptance test the FUs to demonstrate required performance. Deliver the flight systems and provide integration support for the FU with the payload and the launch vehicle.

Statement of Work: Fabricate, acceptance test, and assemble parts to produce the flight systems. Subject the systems to acceptance testing before shipment to the launch site. Use the same test facilities (vacuum chamber, vibration, acoustic) that were used for the qualification program for flight system acceptance testing. Finally, provide launch support activities to insure that the RFC FU meets its design and performance goals associated with integrated payload and launch systems.

The work is divided into the following subtasks:

Subtask 9.1 Flight Component Fabrication. Design, fabricate, inspect, and assemble the components and subassemblies required for the FUs, including all spare parts as required to support the flight system activities.

Subtask 9.2 FU Assembly, Test, and Payload Integration. Assemble and inspect the FUs. Acceptance test the FUs and ship to the launch site. Provide technical support for FU integration with the payload, launch vehicle, and launch support facilities.

Subtask 9.3 FU Launch Support. Provide FU launch support activities to insure that the FU meets its design and performance goals associated with the integrated payload and launch systems.

DEVELOPMENT SCHEDULE

The program starts with a conceptual system design task followed by preliminary system design. Development of the fuel cell stack, electrolysis cell stack, thermal management subsystem, water control subsystem, storage tanks, and PP&C proceed concurrently with the system design. The system design is subsequently completed by the middle of the fourth year.

Fabrication of components for the GES starts with procurement of long lead materials and equipment in the first part of the third year. This leads to assembly of the system by the end of the fourth year.

The GES will be similar to a flight system but will have features such as additional instrumentation and readily removable components to expedite gathering of engineering data and to permit modification of components. It will be tested in a vacuum chamber under normal and off-normal design conditions. After this phase of testing, it will be partially disassembled, examined, refurbished as necessary, and put on life test for 20,000 hours (50% duty cycle).

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components, and a complete system. The design effort will be minimal since it would involve only minor modifications to the GES design. An optional life testing of the QU may be done. This may be unnecessary after the component and GES life testing.

The flight phase of the program includes fabrication, assembly, and acceptance testing of the flight units and the associated safety analysis to obtain launch approval.

Component development was estimated to require 2-4 years of effort (Ref. B-14) without any life testing depending on the current technology readiness levels. Five years were required to develop and qualify the non-regenerative Shuttle alkaline fuel cells. It is felt that the required development effort for a PEM RFC will be longer due to the integration effort involved, the considerable amount of life testing required both on the component/subsystem level and system level, and the number of different applications (more than one module size may be required). The estimated DDT&E and production schedule for the PEM fuel cell and RFC is shown in Figure B-11. Development time was estimated to be 7 years.

REGENERATIVE FUEL CELL POWER SYSTEM TECHNOLOGY ROADMAP

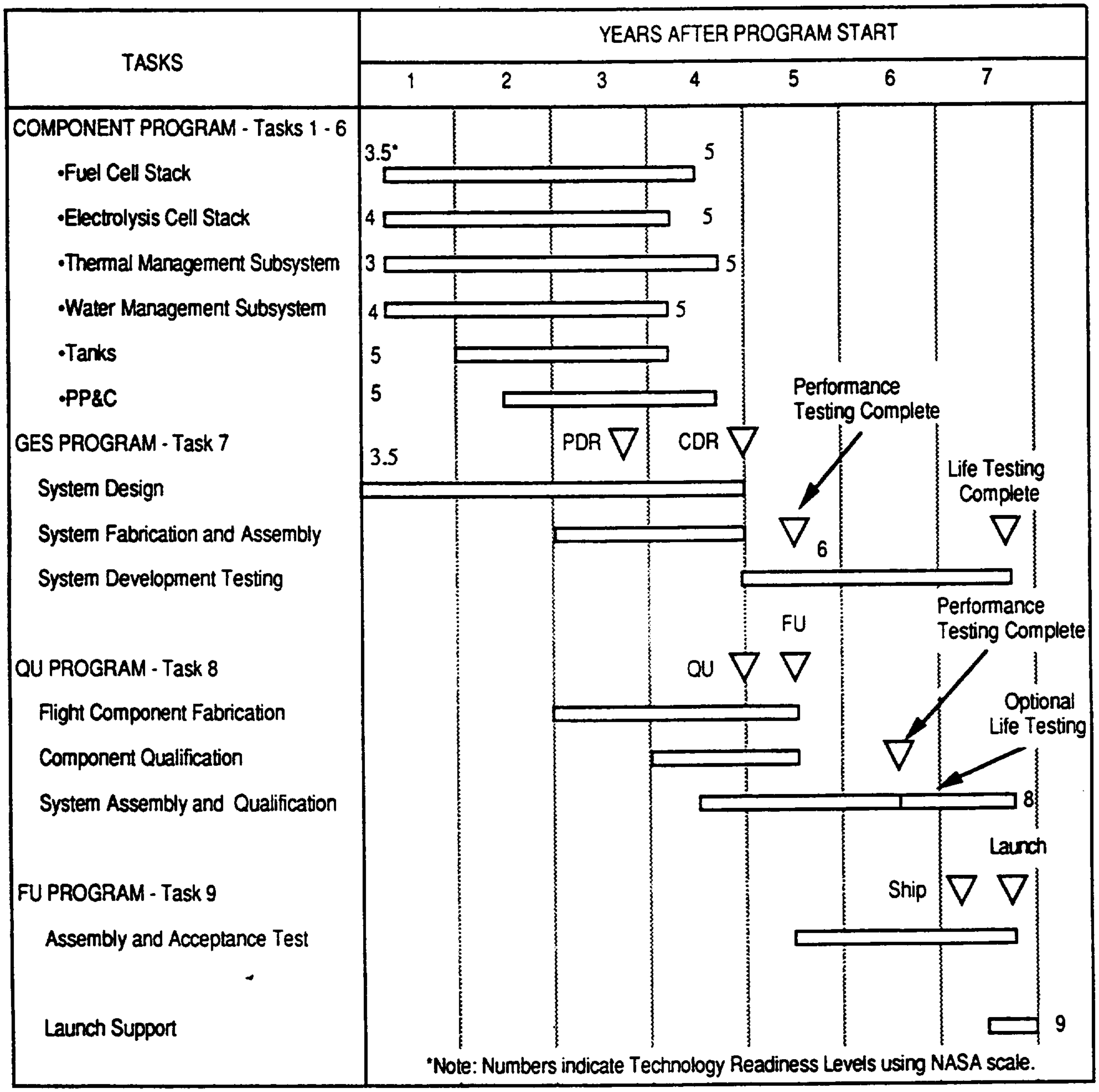


Figure B-11. PEM RFC power system development schedule.

REFERENCES

B-1. Technology Assessment and Tradeoff Study of Fuel Cell and Electrolyzer Technology for Project Pathfinder Energy Storage System. Los Alamos National Laboratory, February 1991.

B-2. McElroy, J.F.: SPE® Water Electrolyzers in Support of Mission From Planet Earth. Space Electrochemical Research and Technology Conference, NASA conference publication 3125, April 1991.

B-3. McElroy, J.F.: SPE® Regenerative Hydrogen/Oxygen Fuel Cells For Extraterrestrial Surface Applications. 24th IECEC Proceedings, August 1989, pp. 1631-1636.

B-4. McElroy, J.F.; et. al: SPE® Regenerative Hydrogen/Oxygen Fuel Cells For a Lunar Base Power System. 25th IECEC Proceedings, August 1990, Volume 3, pp. 143-148.

B-5. Personal Communications with Jim McElroy at Hamilton Standard. Rockwell International (Rocketdyne Division), Frank Littman, 8/13/91 - 3/25/92.

B-6. Pesaran, A.A.: Silica Gel Sorption Properties under Adiabatic Conditions. Solar Energy Research Institute report no. SERI/TR-252-2704, Febrary 1986.

B-7. Penny, T.R.; et. al: Capacity Degradation of Field-Tested Silica Gel Samples. Solar Energy Research Institute report no. SERI/TP-252-2741, June 1985.

B-8. Personal Communication with Paul Prokopius at NASA Lewis Research Center. Rockwell International (Rocketdyne Division), Frank Littman, August 21, 1991.

B-9. Personal Communications with Ian Devey of Structural Composites Industries. Rockwell International (Rocketdyne Division), Frank Littman, October 9, 1991.

B-10. Srinivasan, S.; et. al: Recent Advances in Solid Polymer Electroylyte Fuel Cell Technology with Low Platinum Loading Electrodes. 24th IECEC Proceedings, Vol. 3, August 1989, pp. 1623-1629.

B-11. Lawrance, R.J.; and Margiott, V.A.: Proton Exchange Membrance Fuel Cell Development. 24th IECEC Proceedings, Vol. 3, August 1989, pp. 1587-1591.

B-12. Hirschenhofer, J.H.: Commercial Fuel Cell Technology Status. 26th IECEC Proceedings, August 1991, pp. 521-525.

B-13. Meyer, A.P.: Status of Solid Polymer Fuel Cell Power Plant Development at International Fuel Cells Corporation. 24th IECEC Proceedings, August 1989, pp. 1619-1622.

B-14. Personal Communications with Jay Stedman at International Fuel Cells. Rockwell International (Rocketdyne Division), Frank Littman, September 13, 1991.

B-15. Juhasz, A.J.: An Overview of the LeRC CSTI Thermal Management Program. Paper no. AIAA-91-3528, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 4-6, 1991.

B-16. Bock, E.H.; and Fisher, J.G.: In-Space Propellant Processing Using Water Delivered as Shuttle Contingency Payload. AIAA Paper 78-941, July 1978.

APPENDIX C

NAS BATTERY TECHNOLOGY ROADMAP

APPENDIX C - NaS BATTERY TECHNOLOGY ROADMAP

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APPENDIX C - NaS BATTERY TECHNOLOGY ROADMAP

INTRODUCTION

This is a family of power systems based on common technology for use with portable or mobile power systems. Mobile battery power systems are either incorporated into vehicles or are attached to vehicles as part of a separate power cart. These systems range in nominal power level from 3 to 22 kWe. These vehicles include the payload unloader (3 kWe normal [n]/10 kWe peak [p]), pressurized manned rover (7 kWe onboard, 12 kWe with power cart), regolith hauler (3 [n]/15 [p] kWe), and mining excavator (22 [n]/40 [p] kWe). Mobile Mars power systems will be similar to lunar systems except for the radiator size.

CONCEPT DESCRIPTION

A typical power system schematic is shown in Figure C-1. The overall power system may be divided into the following subsystems for development purposes:

- Batteries;
- Thermal Management; and
- Power Processing and Control (PP&C).

The energy to the user is supplied by the batteries. The batteries are recharged on the next sun cycle by the base power system or by a Dynamic Isotope System (DIPS) power cart. The flow of energy from the array and to and from the batteries is controlled by the PP&C subsystem.

Since the batteries must operate at high temperature, thermal management is required to maintain the proper cell temperature and reject waste heat. In addition, the batteries need to be heated prior to startup. The electronic components in the PP&C also require cooling to remove waste heat.

The battery subsystem includes the cells and related structure to tie the cells together.

The thermal management subsystem includes battery insulation, battery isolation plates, battery radiator/interface heat exchanger, PP&C cold plates, and the PP&C radiator.

Power conditioning is included to process power for charging the batteries (down regulator) and processing output power (boost regulator). A shunt regular dissipates excess power from the array.

The system evaluated in this article uses sodium sulfur (NaS) batteries, DC-DC converters/regulators, and a heat pipe radiator. Each subsystem will be described in more detail in the following sections.

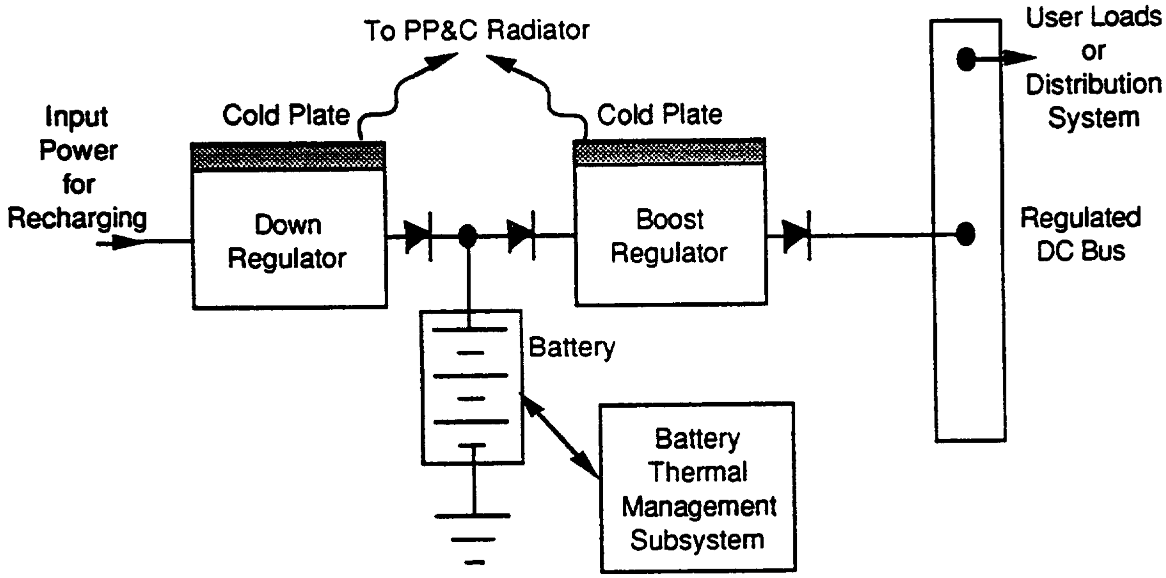

Figure C-1. Battery system schematic.

Battery Subsystem Description

Sodium sulfur batteries are high temperature (598 ^{O}K) secondary batteries which have been under development for a number of years. Due to the high theoretical specific energy, these batteries are being considered for electric vehicles, utility load leveling, satellite systems (Ref. C-1), and planetary surface power systems (Refs. C-2 and C-3).

A fully charged sodium sulfur cell consists of elemental sulfur and sodium separated by a beta alumina electrolyte as seen in Figure C-2. Beta alumina is a solid, ceramic electrolyte separator which is conductive only to sodium ions. The molten sodium serves as the negative electrode. During discharge, each sodium atom, entering into the discharge reaction, provides an electron to the external circuit and migrates through the electrolyte. The molten sulfur serves as the positive electrode. The sodium ions receive an electron, from the external circuit, on the sulfur side of the cell and combine with the sulfur to form sodium pentasulfide, Na_2S_5. After all the free sulfur is combined, a second conversion takes place in which the Na_2S_5 is converted to sodium trisulfide, Na_2S_3. The open circuit voltage of the cell during the first 59% of discharge, which corresponds to the sodium pentasulfide reaction, is nominally 2.08 volts per cell. During the sodium trisulfide part of the discharge, the corresponding voltage is 1.75 volts per cell. Thus, the sodium sulfur cell discharge voltage has a two level characteristic with the step change at about 59 percent. Experimental investigation has indicated that for best cycle life, the depth of discharge should be limited to 59 percent.

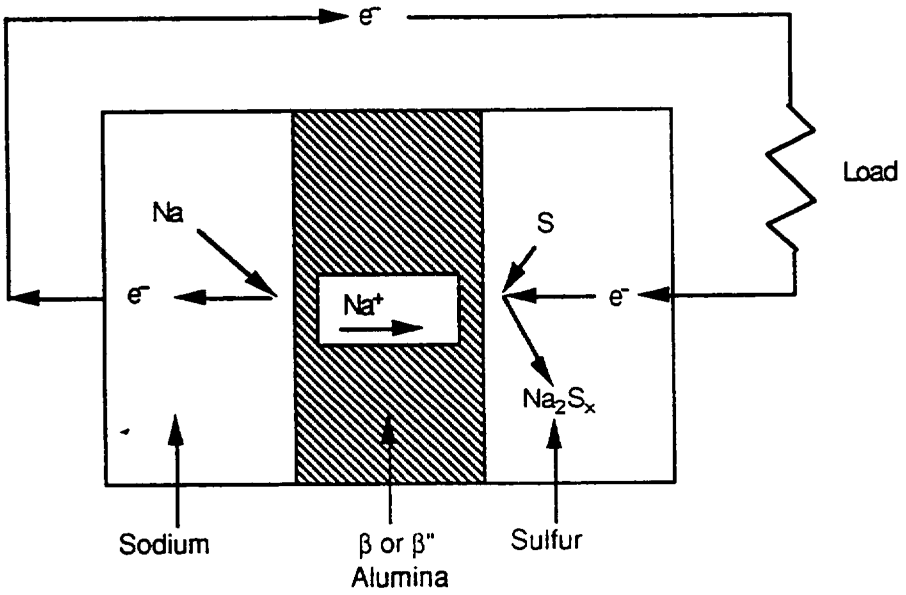

Figure C-2. NaS battery cell operation during discharge.

A tubular type NaS cell is illustrated in Figure C-3. This concept is most suitable for base load power systems (15 minutes to 12 hours of operation). These type of NaS cells have been

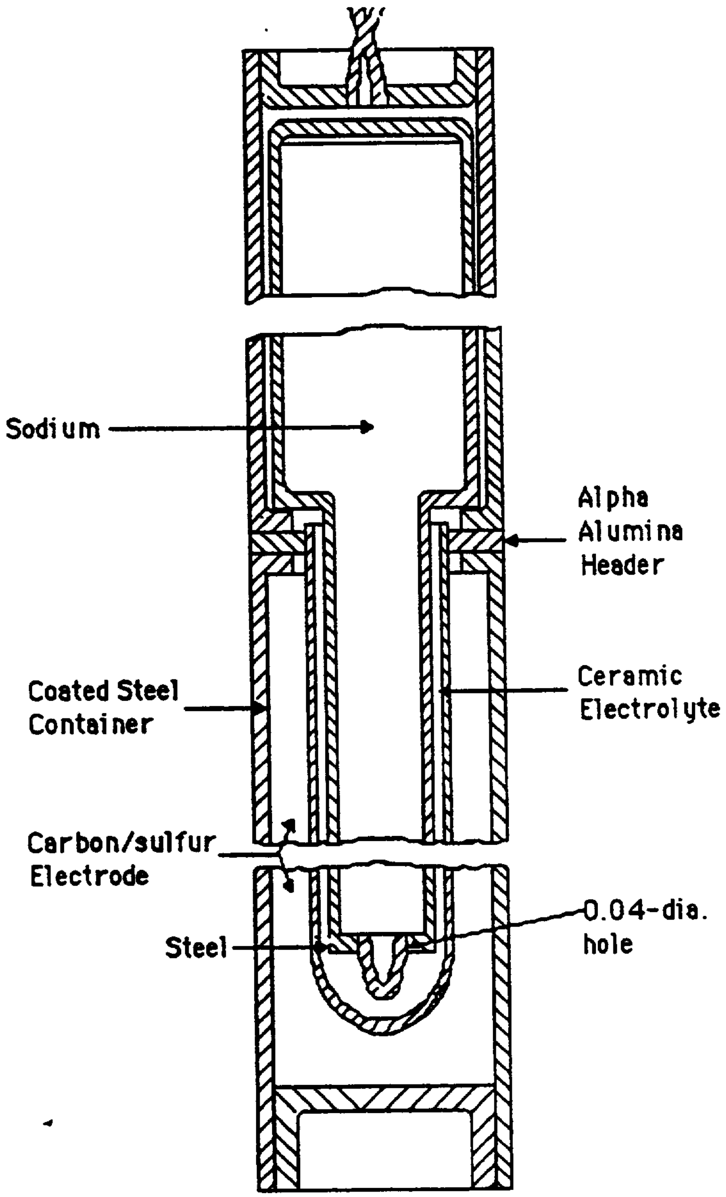

Figure C-3. Typical tubular NaS battery cell design.

undergoing evaluation testing by the Air Force for many years and have gone through several design iterations. As shown in the figure, the liquid sodium is contained in a closed steel tube with a metering orifice connecting to an annular chamber also containing sodium. The closed beta alumina electrolyte encloses the sodium and separates it from the liquid sulfur cathode. Liquid sulfur cathode material is held in a carbon felt matrix to provide electrical conductivity during charge and discharge. The steel tube is coated with a metal chosen for its resistance to the corrosive sodium sulfides.

Thermal Management Subsystem Description

The thermal management subsystem provides temperature control, heat transport, and heat rejection functions. Solid conduction paths and heat pipes provide the heat transport function. A radiator is required to remove waste heat from the battery and from the PP&C components.

An active thermal control system is required to maintain the battery at the proper operating temperature. Three different techniques are possible for controlling battery temperature (Ref. C-4): (1)louvers; (2)pumped loop radiator; and (3)heat pipe radiator.

The use of louvers allows the battery to radiate directly to space as needed to reduce its temperature. Louvers offer the greatest weight advantage (Ref. C-4). However, louvers require the greatest mechanical complexity and potentially the lowest reliability. Louvers also allow for a nonhomongeneous thermal distribution in the battery which may be detrimental.

Pumped loop (or tube-sheet) radiators offer the next best approach from a mass standpoint if no redundancy is built into them. Pumped loop radiators have been used successfully for the space shuttle and will be used for Space Station Freedom (SSF). This type of radiator is best applied to missions with limited duration or to systems which are serviceable. Potential working fluids include water, potassium, NaK, and Dowtherm A. NaK is the preferred working fluid for this application due to its low freezing point, low operating pressure, and stability in a thermal environment. The use of NaK as a reactor primary coolant indicate that it can be used with good reliability for prolonged periods of time. The development of highly reliable, long life, low flow rate, low mass pumping systems to accomodate the use of fluid loops in space would require considerable effort (Ref. C-4). Furthermore, the pump represents the source of a single point failure mode, thus requiring duplication within the system and thereby nullifying the slight mass advantage over heat pipes.

Thermal homogeneity could be accomplished with heat pipe systems for transport and heat rejection due to their isothermal heat transport characteristics. Heat pipes also offer the advantage of improved reliability and a graceful failure mode. A recent Rocketdyne study has shown that advanced carbon-carbon (C-C) heat pipe radiators can be designed which are competitive in mass to pumped loop radiators (estimated specific mass of 3 kg/m^2 for heat pipes with water or ammonia). Thus, a heat pipe radiator was tentatively selected as the baseline design.

The baseline heat rejection assemblies utilize lightweight, passive, reliable, heat pipe radiators that are sufficiently versatile to allow integration into a variety of configurations. The individual heat pipes operate independent from one another and thus the failure of a heat pipe will not result in failure of the complete radiator. The rectangular radiator heat pipe panel is attached to the battery assembly. Variable conductance heat pipes transfer heat to the radiator. The heat pipe working fluid evaporates and travels to the top end of the heat pipes. The evaporated fluid is then condensed in the cooler section of the heat pipe. A small wick or grooves allow the liquid to return to the evaporator (unless the heat pipe is vertical and gravity can be used for fluid return). A wick or groove is not absolutely necessary for vertically oriented radiators (due to gravity return) but is recommended to insure good control of the fluid transport. The heat pipes may be either carbon-carbon tubes with metal liners (titanium or nickel based alloy for biphenyl) or metal heat pipes. Carbon-carbon heat pipes are a factor of 2-3 times lower in mass than conventional all-metal heat pipes.

The problem with using a heat pipe for battery cooling is that no common heat pipe working fluid is suitable for operation at about 600 °K (Ref. C-5). Only three of the common heat pipe working fluids including water, cesium, and mercury show any promise for operation at this temperature. However, these fluids have several disadvantages. Both mercury and cesium are highly toxic and would have handling problems. Mercury has a poor contact angle and does not wet the wick surface, which causes heat pipe priming problems. Mercury also has a high density which results in a heavy heat pipe. Cesium has a very low vapor pressure which favors leakage into the pipe. Leakage is a severe problem since atmospheric water will react with the cesium to form hydrogen. Water develops a high pressure at the required operating pressure which requires a heavier heat pipe and may result in potential safety issues. A recent study (Ref. C-5) found that biphenyl is a suitable heat pipe working fluid for this application. Biphenyl is compatible with likely materials of construction and its liquid transport factor is suitable for use in a heat pipe. Also, its vapor pressure is in the desired range at the operating temperature of the battery. Thermal studies have shown that biphenyl should be stable for long-term operation at high temperature and non-condensable gas generation will not be a problem. Thus, biphenyl was selected as the working fluid for the battery heat pipes.

PP&C Subsystem Description

The PP&C subsystem is a collective term for the power system control electronics. This system includes items such as voltage regulators and battery charge controllers. Their function is to control the flow of energy though the power system to the payload.

Boost Regulator Description. The boost regulator increases the voltage from the batteries up to the nominal bus voltage. There are two basic types of boost regulators. The basic boost regulator handles all of the processed power and the efficiency of the circuit applies to all the power. This regulator places a "buck" inductor and switch in alternate positions such that the opening switch allows the inductor to force the current it had before the switch opened into the load impedence. The output voltage depends on the load impedence and current. The basic booster has a relatively low efficiency and may have loop stability problems.

The add-on booster only processes the boost voltage (usually only 10 volts). The boost power (voltage added times output current) is parallel to the load at the input. Most of the power is not processed and thus the overall efficiency is much higher than for the basic boost regulator. A failure of the add-on booster results in a lower DC bus voltage, but the system is still functional with degraded performance.

The boost regulator requires monitoring equipment to determine operating parameters and heat sinking to remove waste heat. EMI filters are required to prevent interference from the switching.

Down Regulator Description. The charging power system (especially in the case of a PV array) will be designed to provide a higher voltage than the nominal battery voltage. The down or "buck" regulator provides a constant output DC voltage for charging the battery subsystem from the array. The regulator output voltage is always lower than the input voltage. The voltage conversion is accomplished by switching the input on and off. The filter removes the AC component produced by switching, but allows the DC voltage throught.

The down regulator is reliable and very efficient (85-90% for nearterm hardware). The efficiency of the regulator depends on the internal operating frequency and the input-output voltage difference. As the frequency goes up the size goes down and so does the efficiency. This regulator does not isolate the input from the output loads. If the regulator is not properly used, then it can cause radio interference since it uses switching to interrupt the current. Proper EMI filters are required to eliminate this interference. Monitoring equipment is required to determine the proper operating conditions. Heat sinking is required to remove waste heat.

TECHNOLOGY ISSUES

The key issues for development of a NaS battery subsystem are summarized in Table C-1. The key issue with sodium sulfur batteries is achieving cycle life to permit operation as a satellite battery. While the specific energy, of the sodium sulfur system, is much higher than the current baseline nickel hydrogen batteries, the cycle life is much lower. Primary failure mechanisms are related to corrosion of the seals and cell components by the chemically active sulfides and the fragility of the beta alumina electrolyte. Design features have been developed to mitigate the effect of these failure mechanisms so that a failure is not catastrophic. Cracking of the electrolyte or loss of cell case integrity does however, result in loss of the cell however.

Continuing cell development is aimed at resolving these issues.

TABLE C-1. NaS BATTERY SUBSYSTEM ISSUES, IMPACTS, AND DEVELOPMENT AREAS

Issues	Impacts	Potential Development Areas
Cycle life	•Increased life cycle cost	•Physical and chemical stability of alpha alumina seal •Physical and chemical stability of electrolyte •Sealing technology for tubesheet to cell case
High operating temperature	•Heavy heat pipes, poor performance, toxic working fluids (cesium or mercury) or high pressure (water)	•Low mass C/C heat pipe radiator •Heat pipe working fluids (biphenyl)
Safety (explosion, fire, toxic fumes, formation of sulfuric acid)	•Increased manufacturing cost	•Battery casing design

TECHNOLOGY ASSESSMENT

The technology bases were assessed for the following major battery subsystems:

- Batteries;
- Thermal management; and
- PP&C.

The technology readiness of each assembly was evaluated using the NASA technology readiness levels. These evaluations are summarized in Table C-2 which shows that the power system has technology readiness levels ranging from 3 to 5, depending on the particular subsystem. The technology base for subsystems is discussed in the following sections.

TABLE C-2. NaS BATTERY TECHNOLOGY ASSESSMENT

Subsystem	Technology Readiness Level	Comments
NaS batteries	4	Breadboard cells tested by Air Force and prototype battery is under development
Battery thermal management	3	Radiator component development currently underway; biphenyl investigated as heat pipe fluid
PP&C	5	Similar components under development for Space Station Freedom (SSF)

System State-of-the-Art

The NaS battery system has a NASA Level of about 3.5. The concept of this type of a system has been formulated but the details of the system have not been worked out. Specifically, this includes the integration of the NaS battery storage into the system. Though NaS batteries are a basic secondary type battery they require special thermal control to work. The batteries require an operating temperature of 598 °K. The PP&C subsystem will have to work in conjunction with thermal control system to maintain this temperature. In addition, to overcome the fragility problems of NaS batteries, it has been proposed to launch the batteries in a frozen state (<573 °K). Once on the planetary surface, the batteries will have to be brought up to temperature through some type of external source. For a flight test experiment, the power source will be the Shuttle payload power bus. Another issue is that prior to the time the NaS batteries are brought on line the spacecraft will need power for basic operations such as ground telemetry and array deployment. Some other source of energy, such as a primary battery, will be necessary and will increase the complexity of the PP&C subsystem. In this system the operational characteristics of the NaS battery are the key issues. Due to the low readiness level of the NaS battery and the thermal management subsystem, the overall system readiness level is relatively low compared to current battery systems.

Battery Subsystem State-of-the-Art

Development of sodium sulfur batteries was initiated in the 1960s for electric vehicle applications (Ref. C-6). Since then, development has progressed to the point that prototypes and breadboards are now being fabricated or are in test. Advancement of the technology has been limited by the low level of funding and the concern over the safety and operating temperature of the battery. Several manufacturers are engaged in ceramics development on the electrolytes and, in the United States, Hughes and Eagle-Picher are actively engaged in the development of space type batteries (Ref. C-7).

Eagle-Picher, in conjunction with Hughes Aircraft, had an Air Force contract (administered by WPAFB) which ran from 9/86 to 1/91 to develop a NaS cell for Low Earth Orbit (LEO). The major goals of this program were the following:

- specific energy of 50 W-hr/lb;
- 30,000 charge-discharge cycles;
- 2C discharge rate;
- 5 year calendar life; and
- withstand LEO environment.

A Mars surface environment would be much more benign for a battery subsystem than a LEO application due to a reduced charging rate, a reduced number of cycles (only 1825 for 5 years), and a gravity environment (poor distribution of reactants in microgravity environment can cause current blockage due to formation of a non-conductive layer). In addition, the battery can be qualified using only ground testing.

For space applications, NASA Level 4 would apply. That is, breadboard cells have been in test for some time by the Air Force and a prototype of a satellite battery is in development under Air Force contract and is to be demonstrated in 1993 or 1994 (Ref. C-1). Further development is required to accelerate the fabrication and testing of cells and batteries in order to identify currently unknown problem areas. Specifically, testing will include space environmental testing, materials compatibility testing, and cycle testing. Under current funding levels, flight test of the battery is projected for the 1996 to 1997 timeframe. Availability for space applications is projected for the year 2000.

Thermal Management Subsystem State-of-the-Art

NASA LeRC is currently carrying out an integrated multi-element project for the development of space heat rejection subsystems with special emphasis on low mass radiators in support of SEI power system technology (Ref. C-8). This effort involves both in-house and contracted efforts. Contracted efforts involving Rockwell International (RI) and Space Power Incorporated (SPI) are aimed at the development of advanced radiator concepts (ARC). NASA

LeRC is also involved in a joint program with DOE to demonstrate a flexible fabric heat pipe radiator concept being developed by Pacific Northwest Laboratories (PNL). In-house work at NASA LeRC is designed to guide and support the overall program by system integration studies, heat pipe testing and analytical code development, radiator surface morphology alteration for emissivity enhancement, and composite materials research focused on the development of low mass, high conductivity fins. This program is concentrating on technologies capable of development before the end of the decade for both surface power and nuclear electric propulsion (NEP) applications.

Specific objectives of the ARC contracts are to achieve specific mass values <5 kg/m^2 with radiator surface emissivities of 0.85 or higher at typical radiator operating temperatures, and reliability values of at least 0.99 for the heat rejection subsystem over a ten year life. These figures represent a factor of two improvement over the currently considered heat rejection subsystem for SP-100, and even greater improvement factors for state-of-the-art heat rejection systems used in current spacecraft applications.

Presently, Rocketdyne division of Rockwell International is involved in the development of a carbon-carbon (C-C) or graphite heat pipe radiator panel. Rocketdyne has a NASA contract entitled "SP-100 Advanced Radiator Concepts" which started in 1987 and is part of the CSTI (Civilian Space Technology Initiative) program. This program will be completed in January of 1993. This program involves development of a C-C heat pipe suitable for use in an SP-100 radiator. However, this technology will be suitable for other radiator applications as well including battery power systems (using other working fluids). The hope is that this program will develop generic technology which can be used for all future space radiators because of its reduced mass compared to metal heat pipe radiators. Carbon-carbon is a new structural material which requires different fabrication techniques than for metals or composites. A key objective of this program is to demonstrate the ability to fabricate C-C heat pipes with a thin metal liner, wicks, caps, and fill tube. The brazing technique required to attach liners is the same as would be required to attach manifolds to the heat pipes. Heat pipes filled with potassium will be designed, fabricated, and tested during this program. Thermal cycle testing will be done in a vacuum to demonstrate the integrity of the heat pipes.

Beginning in 1989, small samples of heat pipe panels were fabricated by Rocketdyne as part of the CSTI program. The next phase of the program will involve fabrication and testing of complete heat pipes several feet long.

The working fluids and temperatures for the SP-100 system would be different than for lower temperature battery systems. Thus, Rocketdyne has an on-going IR&D effort which involves development of a C-C heat pipe for lower temperature applications. This effort will eventually include fabrication and testing of a heat pipe. Rocketdyne is also looking at a higher thermal conductivity C-C heat pipe design.

The objective of the joint NASA LeRC/Air Force program with PNL (Light Weight Advanced Ceramic Fiber [ACF] Heat Pipe Radiators) is to demonstrate the feasibility of a low mass ceramic fabric/metal liner heat pipe for a wide range of operating temperatures and working fluids (Ref. C-8). Specifically, the NASA LeRC objectives are to develop this concept for application to space radiators with operating temperatures below 500 °K using water as the working fluid. The specific mass goal for these heat pipes is 3 kg/m^2 or less at a surface emissivity of greater than 0.85.

Several heat pipes were built for the ACF program using titanium and copper foil material for containment of the water working fluid. A heat pipe with an 0.2 mm Ti liner was demonstrated in early January 1991. An innovative "Uniskan Roller Extrusion" process has also been developed at PNL and used to draw 0.76 mm wall tubing to a 0.05 mm foil liner in one pass. The water heat pipes fabricated for LeRC have been subjected to a test program to evaluate performance and reliability at demanding operating conditions.

Future thrusts of the ACF program will be to perfect the heat pipe fabrication procedure using very thin (0.025 to 0.05 mm) foil liners which are internally texturized by exposure to high pressures (Ref. C-8). Plans will also be developed to perform hyper-velocity and ballistic velocity impact tests in order to determine if secondary fragments from a penetrated heat pipe will result in failures of neighboring heat pipes. Another major challenge will be to design a heat pipe with high conductivity, low mass fins as a first step toward the fabrication of low mass radiator panels.

The NASA LeRC in-house materials program includes radiator surface morphology alteration by arc texturing for emissivity enhancement purposes (Ref. C-8). Emissivity enhancement has been demonstrated for graphite-copper samples.

PP&C State-of-the-Art

Assigning a NASA Readiness Level to PP&C is difficult. There are many systems that have flown which would make parts of the PP&C subsystem Level 9, but each PP&C system was custom design for each spacecraft design. Only until recently has the array and battery PP&C become standardized into packaged systems. In the Space Station Freedom program, standard PP&C components are undergoing qualification. Recent military programs have ended with qualified packaged PP&C component designs.

The PP&C system design is based on a reasonable electronics component evolution, and there are no significant technology issues associated with its development. It will be necessary to fabricate breadboard hardware for testing and evaluation purposes, but there is no need to initiate any advanced component development. Most of the hardware elements have been or shortly will be operating in a relevant environment on other spacecraft or the SSF electrical power subsystem.

DEVELOPMENT PLANS

The development program was divided into six major tasks. The first three tasks are component development tasks. The last three tasks includes the system design, fabrication, integration, and testing for the Ground Engineering System (GES), Qualification Unit (QU), and Flight Unit (FU).

Testing will be done on the component, subsystem, and system level (qualification testing). The system level tests will show any possible negative interactions between subsystems.

The battery development tasks are described in the following sections. Each task will include a section on objectives and the statement of work.

Task 1. Battery Subsystem Development

Objectives: Develop a full scale flight weight battery module. Demonstrate materials compatibility, safety and performance margins.

Statement of Work: This effort was divided into the following subtasks:

Subtask 1.1 Preliminary Battery Development. Demonstrate the feasibility of the selected battery design for planetary surface applications. Test a prototype battery to demonstrate materials compatibility, safety, and component life.

Subtask 1.2 Final Battery Module Design. Design a full scale flight weight battery module.

Subtask 1.3 Battery Module Fabrication, Assembly, and Testing. Build and test (performance, mechanical, and thermal cycling) the battery module. After initial breadboard validation testing is complete, perform additional module testing in a relevant environment.

Task 2. Thermal Management Subsystem Development

Objectives: Develop and demonstrate a low mass, reliable heat pipe radiator which will operate at about 600 °K. Develop and demonstrate components for thermally isolating the battery from the rest of the system and the environment. Develop a thermal management approach for the PP&C components.

Statement of Work: This effort was divided into the following subtasks:

Subtask 2.1 Heat Pipe Demonstration. Fabricate and test representative length heat pipes to fully characterize heat pipe performance. Compare test results with predicted performance over the anticipated range of operating conditions including startup, shutdown, and restart.

Perform limited life testing of the heat pipes. Identify deterioration mechanisms, measure deterioration rates, and assess the adequacy of assembly and cleaning procedures. Withdraw heat pipe samples sequentially throughout the test period. Drain, section, and analyze the samples for corrosion, carbon diffusion, braze stability, and any other signs of deterioration.

In conjunction with the heat pipe performance and life testing, develop and demonstrate techniques for nondestructive examination (NDE) of the heat pipe elements and assembly. Assess the adequacy of liner bonding and weld joints. Determine if the correct fit-ups and interfaces were obtained in the assembled piece.

Subtask 2.2 Radiator Enhancement. Increase the applicability of the radiator concept by performing various enhancement tasks. Examine survivability options and extended length heat pipes.

Perform analytical assessment, testing, and enhancement design development to insure long life for Mars missions. Consider natural threats such as micrometeoroids and dust erosion.

Fabricate a long heat pipe. Develop alternate techniques for fitting the liner into the tube. Investigate alternative tube fabrication, liner fabrication, and coating processes.

Subtask 2.3 Heat Pipe Integration and Testing. Develop and demonstrate techniques for attaching the heat pipes to the battery assembly. Subsequently, demonstrate heat pipe integration into a representative radiator section. Test the radiator section to provide an accurate overall heat transfer coefficient. Verify radiator dynamics and performance. Test the assembled unit in a cold wall, vacuum chamber, simulated space environment. Validate temperature drop predictions and assess component interactions.

Subtask 2.4 Thermal Management Subsystem Design, Assembly, Fabrication, and Testing. Develop the detailed design of the thermal management subsystem. Resolve specific design issues such as the effects of differential thermal expansion, deployment features, monitoring instrumentation and control equipment, local insulation, and trace heating of the battery for thawing.

Design, fabricate, and test a representative full scale thermal management subsystem. Complete performance, mechanical (stress, shock, and vibration), and thermal cycling.

Task 3. PP&C Subsystem Development

Objectives: Demonstrate adequate steady state and transient performance, and immunity to the environment (including launch and operating). Demonstrate the software capability to handle power system nominal operation and failure modes.

Statement of Work: This effort was divided into the following subtasks:

Subtask 3.1 Electrical Components. Build breadboard units to demonstrate and check functional performance of the individual component circuit designs. Incorporate any necessary design modifications and improvements into brassboard units. Verify functional performance within the constraints of the actual component configuration. Fabricate prototype units and perform a series of tests using simulated input and output loads. Test the controller to validate the operating system software. Include the following testing:

- start up, steady state and transient control;
- failure simulation and detection and switching;
- effects of temperature extremes and thermal shock;
- effects of environment;
- shock and vibration;
- cold plate heat loads; and
- EMI generation and susceptibility.

Subtask 3.2 Software - Check out the controller software using simulated inputs and outputs.

Task 4. Ground Engineering System (GES) Testing

Objectives: Design a system concept which will meet Mars power system applications. Show the concept feasibility. Verify adequate performance and life for an integrated system.

Statement of Work: This effort was divided into the following subtasks:

Task 4.1 GES Design. Identify and characterize specific power system applications. Determine power system requirements. Define optimum power system module size based on trade studies which minimize the overall life cycle cost of the power system for all applications (trade between development cost and mass).

Identify concepts and perform tradeoff studies (performance, reliability, risk, safety, and life cycle cost) to select the optimum system design. Complete feasibility studies for the selected concept. Definitize the remaining hardware development tasks based on the system concept chosen.

Design a complete power system to at least meet the minimum life requirement (10 years for arrays and 5 years for batteries). Perform both design point and transient analyses to verify the design. Perform this task in parallel with the component development tasks to insure proper system characteristics

Task 4.2 GES Performance Tests. Manufacture the GES components and assemble. Pressure and leak check the assembly, fill the batteries and heat pipes, complete a electrical check-out, and check out all active devices to the extent possible.

Design the test program for the GES to verify all performance characteristics of the unit in conjunction with subjecting the GES to acceptance level tests. Conduct performance and thermal vacuum tests during a single thermal vacuum test sequence to demonstrate technical capabilities while minimizing program cost. Complete the following performance testing activities:

- checkout and refine subsystem assembly procedures;
- verify operation of ground support equipment (GSE) and interfaces with the GES;
- verify start up capability;
- verify steady state performance characteristics;
- establish effective operating range of the system;

- check out software and verify autonomy (include all operational modes and simulate failure modes);
- simulate normal switchover from one module to another;
- simulate failures to trigger module switchover; and
- establish system sensitivity to off-normal conditions such as: (1) partial loss of radiator cooling, (2) varying radiator heat removal profile, (3) partial loss of electronic cooling, and (4) temporary losses of heat sink with varying duration.

Task 4.3 GES Life Test. Refurbish the GES after completion of performance, acceptance, and margin tests. Refurbishment will include the following:

- replacing or repairing components as necessary to ensure that the GES is returned to its as-built condition;
- addition of special instrumentation required for the life test phase; and
- modification of the main radiator heat removal system or replacement of the main radiator with a dump heat exchanger.

Install instrumentation to provide a comprehensive diagnosis of the "health" of the GES and to monitor for degradation of major assemblies and individual components. Place the GES on a multiyear life test in air. Operate the GES at its nominal operating point, with expected GES variations in power output and environment.

Disassemble the GES for diagnosis at the end of the life test. Determine areas to be examined by an analysis of the health monitoring data and from the reliability analysis predictions.

Task 5. Qualification Program Testing

Objectives: Fabricate the qualification units (QUs). Verify adequate performance and life for the QUs.

Statement of Work: Develop a comprehensive performance and dynamic testing program which will provide a formal demonstration that the power system will perform as designed after being subjected to simulated launch conditions. Select launch environmental conditions which envelope the probable intensities developed by various launch scenarios.

Begin the qualification effort (a typical approach is shown in Figure C-4) by qualifying the assemblies. Fabricate and assemble the qualified production items into the QU. Qualify the QU by the rules for space vehicle qualification.

This effort was divided into the following subtasks:

Subtask 6.1 Qualification Performance Testing. Conduct performance testing at each level to verify that each item performs as designed. Perform dynamic testing to verify the capability of the power system to withstand launch loads, including acoustic, pyroshock and vibrational. A possible performance and dynamic qualification test sequence for components and assemblies is shown in the matrix of Figure C-5. The corresponding qualification test sequence for the QU is shown in Figure C-6.

Subtask 6.2 Qualification Life Testing (Optional). After completion of the qualification performance testing, partially disassemble the QU and examine the unit. Refurbish the QU as required and modify for endurance testing in air, as described for the GES. Life test the QU.

Task 6 Flight Unit (FU) System Program

Objectives: Fabricate the flight systems, develop a flight safety program, and acceptance test the FUs to demonstrate required performance. Deliver the flight systems and provide integration support for the FU with the payload and the launch vehicle.

Statement of Work: Fabricate, acceptance test, and assemble parts to produce the flight systems. Subject the systems to acceptance testing before shipment to the launch site. Use the same test facilities (vacuum chamber, vibration, acoustic) that were used for the qualification program for flight system acceptance testing. Finally, provide launch support activities to insure that the FU meets its design and performance goals associated with integrated payload and launch systems.

The work is divided into the following subtasks:

Subtask 6.1 Flight Component Fabrication. Design, fabricate, inspect, and assemble the components and subassemblies required for the FUs, including all spare parts as required to support the flight system activities.

Subtask 6.2 FU Assembly, Test, and Payload Integration. Assemble and inspect the FUs. Acceptance test the FUs and ship to the launch site. Provide technical support for FU integration with the payload, launch vehicle, and launch support facilities.

Subtask 6.3 FU Launch Support. Provide FU launch support activities to insure that the FU meets its design and performance goals associated with the integrated payload and launch systems.

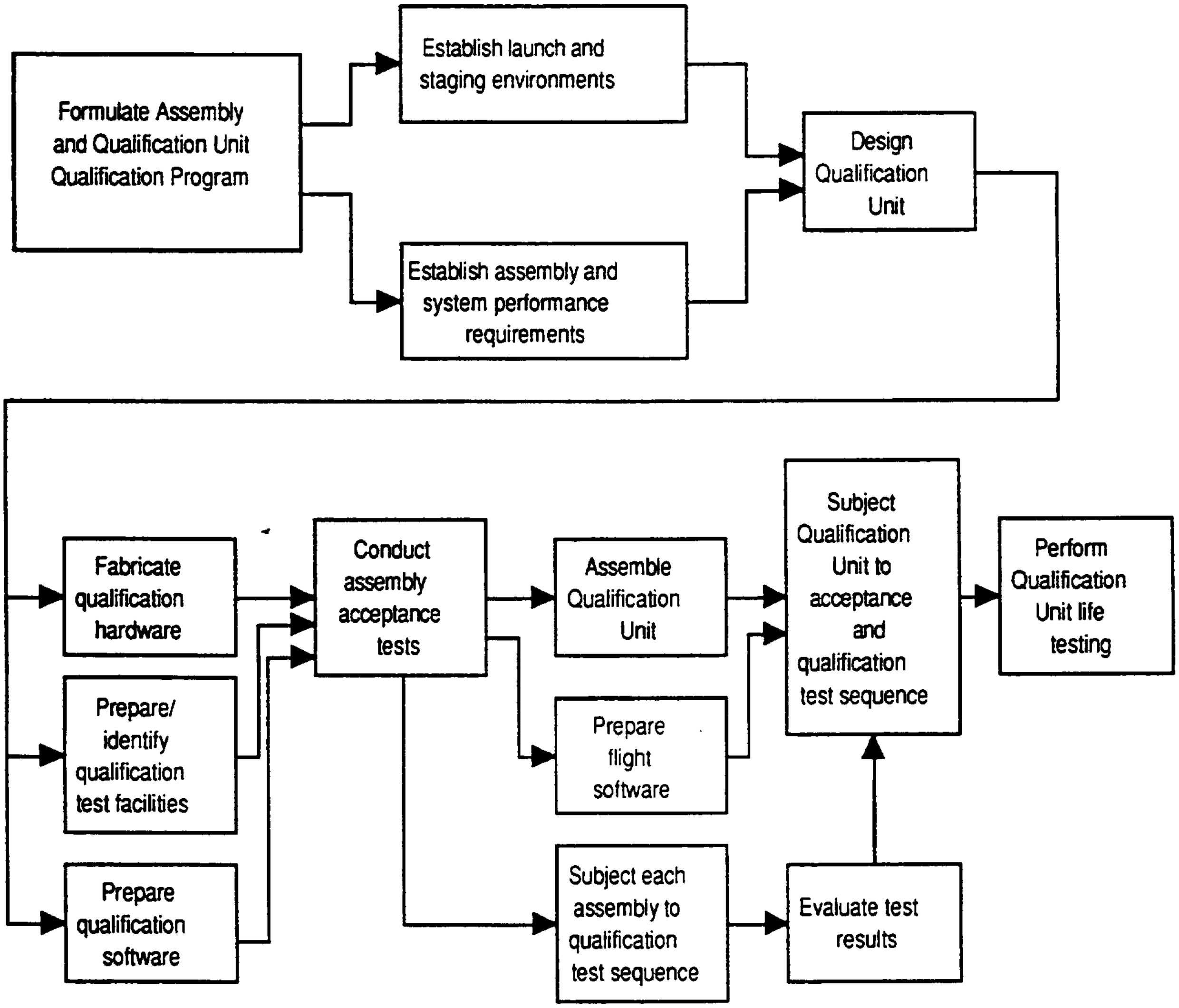

Figure C-4. NaS battery power system qualification program.

Component or Subassembly	Functional (1)	Leak	Pyroshock	Functional (2)	Random Vibration	Functional (1)	Leak	Acceleration	Functional (1)	Thermal Cycling	Functional (2)	Thermal Vacuum	Functional (2)	Pressure	Leak	EMC	Life	Functional (1)
Battery Module	X	X			X	X	X			X	X	X	X	X	X	X	X	X
Radiators	X	X			X	X	X			X	X	X	X	X	X		X	X
PP&C	X		X	X	X	X				X	X	X	X			X	X	X
Structure	X		X	X	X	X		X	X	X	X						X	X

Figure C-5. NaS battery assembly qualification test matrix.

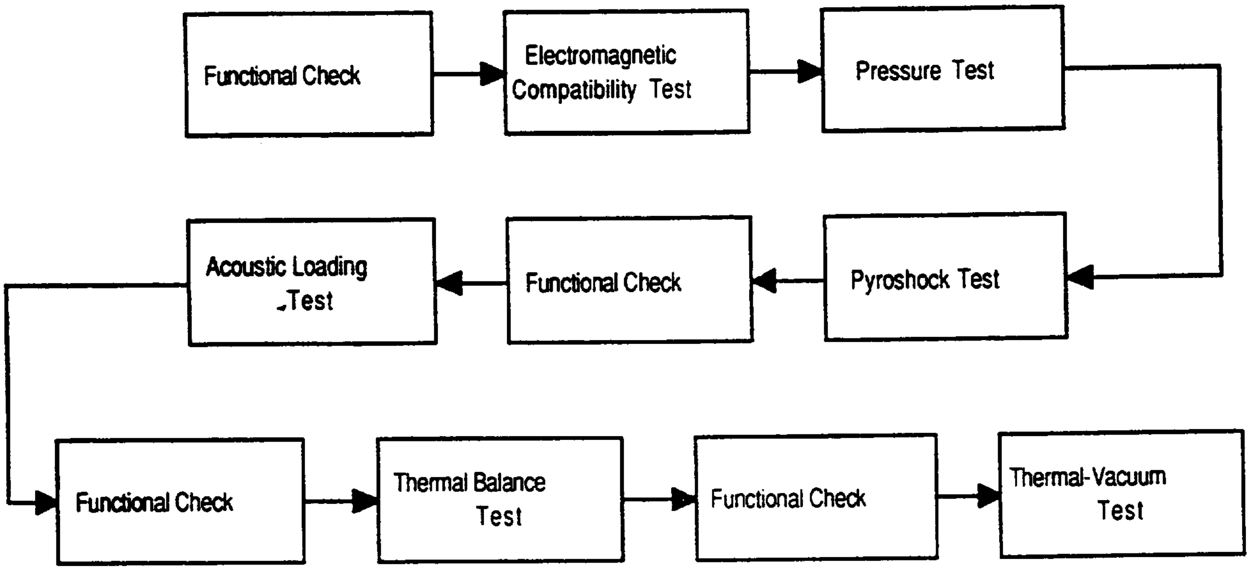

Figure C-6. QU test sequence.

DEVELOPMENT SCHEDULE

The program starts with a conceptual design task followed by preliminary design and concurrent component development or production of the battery, thermal management, and PP&C. The detailed design is subsequently completed by the middle of the third year.

Fabrication of components for ground testing for the Ground Engineering System (GES) starts with procurement of long lead materials and equipment in the first part of the fourth year. This leads to assembly of the system by the end of the fifth year.

The GES will be similar to a flight system but will have features such as additional instrumentation and readily removable components to expedite gathering of engineering data and to permit modification of components. It will be tested in a vacuum chamber under normal and off-normal design conditions. After this phase of testing, it will be partially disassembled, examined, refurbished as necessary, and put on life test, nominally for 1 year.

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components, and a complete system. The design effort will be minimal since it would involve only minor modifications to the GES design.

The flight phase of the program includes fabrication, assembly, and acceptance testing of the flight units and the associated safety analysis to obtain launch approval.

The estimated development and production schedule for the power system is shown in Figure C-7. A minimum development time is about 8 years.

SODIUM SULFUR BATTERY TECHNOLOGY ROADMAP

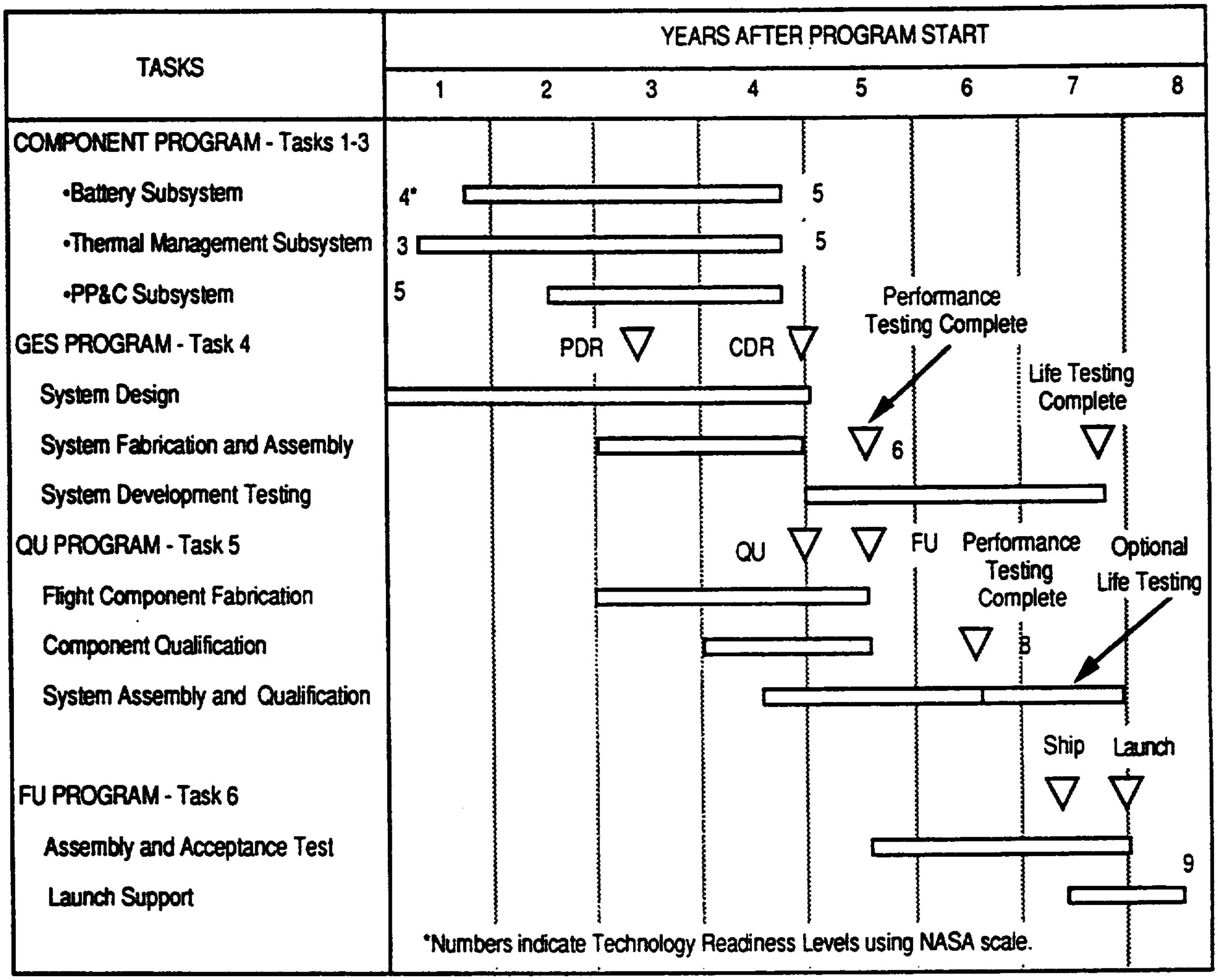

Figure C-7. NaS battery power system development schedule.

REFERENCES

C-1. High Energy Density Rechargeable Battery, Preliminary Design Review. Hughes Aircraft Company briefing, 27 April 1988.

C-2. Petri, D.A.; Cataldo, R.L.; and Bozek, J.M.: Power System Requirements and Definition for Lunar and Mars Outputs. 25th IECEC proceedings, Vol. 1, August 1990, p. 18.

C-3. Withrow, C.A.; et al: SEI Rover Solar-Electrochemical Power System Options. 26th IECEC, August 1991, p. 319.

C-4. Marcoux, L.; and Allen, D.M.: Conceptual Design of a Large Spacecraft Power System Utilizing a Sodium-Sulfur Battery. IECEC'83, Vol. 4, August 1983, p. 1470.

C-5. Grzyll, L.R.: Investigation of Heat Pipe Working Fluids for Thermal Control of the Sodium/Sulfur Battery. 26th IECEC proceedings, Vol. 3, p.1390.

C-6. Linden, D.: Handbook of Batteries and Fuel Cells. McGraw Hill Book Company, 1984.

C-7. Taenaka, R.: Personal Communications. Hughes Aircraft Co., 19 February 1992.

C-8. Juhasz, A.J.: An Overview of the LeRC CSTI Thermal Management Program. Paper no. AIAA-91-3528, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 4-6, 1991.

APPENDIX D

PV/RFC POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX D - PV/RFC POWER SYSTEM TECHNOLOGY ROADMAP

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PHOTOVOLTAIC/REGENERATIVE FUEL CELL POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX D

INTRODUCTION

This is a family of fixed power systems for both lunar and Mars applications. Mission applications require power levels from 0.9 kWe to 25 kWe (up to 100 kWe for lunar applications). Power system applications include communications (0.9 kWe), the excursion vehicle servicer (10 kWe), emergency power (12 kWe), and exploration site main power (25 kWe for habitat and associated external equipment). Solar powered systems such as these will require energy storage for operation at night.

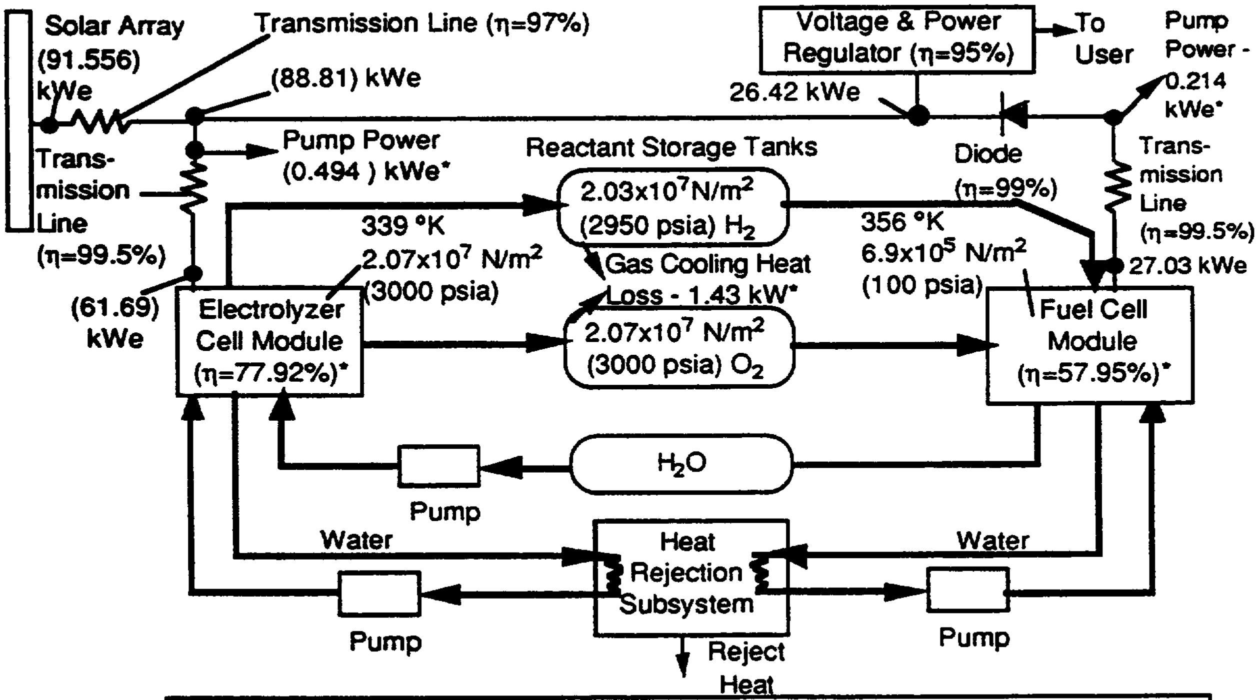
CONCEPT DESCRIPTION

A typical power system schematic is shown in Figure D-1. The overall power system may be divided into the following subsystems for development purposes:

- Photovoltaic (PV) Array;
- Regenerative Fuel Cell (RFC);
- Radiator; and
- Power Processing and Control (PP&C).

The solar array converts sun light directly into DC electricity. The energy from the array flows to the electrolysis cells to electrolyze water and to the user. When the system enters a period of darkness, the energy to the user is supplied by the fuel cells. The electrolysis process is repeated on the next sun cycle. The flow of energy from the array and to and from the RFC is controlled by the PP&C subsystem.

The PV subsystem includes the array panels, support structure, and wiring harness. For development purposes, specific subtasks will include cell development, array development, deployment development, and integration/ system testing. This power system concept utilizes high efficiency multijunction tandem photovoltaic cells to minimize array area. A large array area is required for Mars applications due to the low insolation rate (due to dust from local and global dust storms). Since the PV subsystem is the largest component of system mass for Mars systems, it is important to minimize array area and specific mass (kg/sq m).



Note: Necessary electrical and fluid controls, and redundant components not shown. Typical operating conditions and performance shown. Powers in () are for daytime operation. Heavy lines indicate fluid flow. Light lines indicate electrical flow.
*Values are variable depending on application (i.e., efficiencies vary).

Figure D-1. 25 kWe PV/RFC power system schematic.

The RFC subsystem includes fuel cells, electrolysis cells, water thermal control (humidifers, dehumidifiers, fluid controls), storage tanks (hydrogen, oxygen, water), thermal control (radiator, thermal control loops), and support structure.

Power conditioning is included to process power for running the electrolysis unit and to process the fuel cell output power. A shunt regular dissipates excess power from the array.

The current power system life goal is 15 years for SEI missions. A minimum life requirement of 10 years was assumed for the PVA subsystem. This appears to be a reasonable goal based on the demonstrated cell performance and low degradation rate. A 20,000 hr life (at 50% duty cycle) was assumed for the RFC subsystem. The RFC life requirement appears to be a realistic, though challenging, goal based on previously demonstrated cell performance and allowing for partial redundancy in the design of critical components (i.e., valves, pumps). Preliminary studies have also shown that 3 RFC units (exclusive of the radiator and tanks), each rated at 50% of the total power, can provide enough redundancy to meet a 5 year life with a 50% duty cycle (Ref. D-1).

Photovoltaic Array Subsystem Description

The baseline PV cell is one being developed by Boeing Defense and Space Group in Seattle (Refs D-2 through D-4). The multi-bandgap or tandem cell consists of a double-heterostructure GaAs or AlGaAs thin film top cell and a polycrystalline $CdZnS/CuInSe_2$ heterojunction thin film lower cell as seen in Figure D-2 (Ref. D-2). The leading technology for thin film photovoltaic cells is $CuInSe_2$ (CIS) (Ref. D-5). In the cascade structure shown, short wavelength (high energy) photons are absorbed in a high bandgap material on top of the solar cell. The high bandgap material is transparent to longer wavelength (low energy) photons which pass through and are absorbed by a second layer consisting of a photovoltaic material with low bandgap.

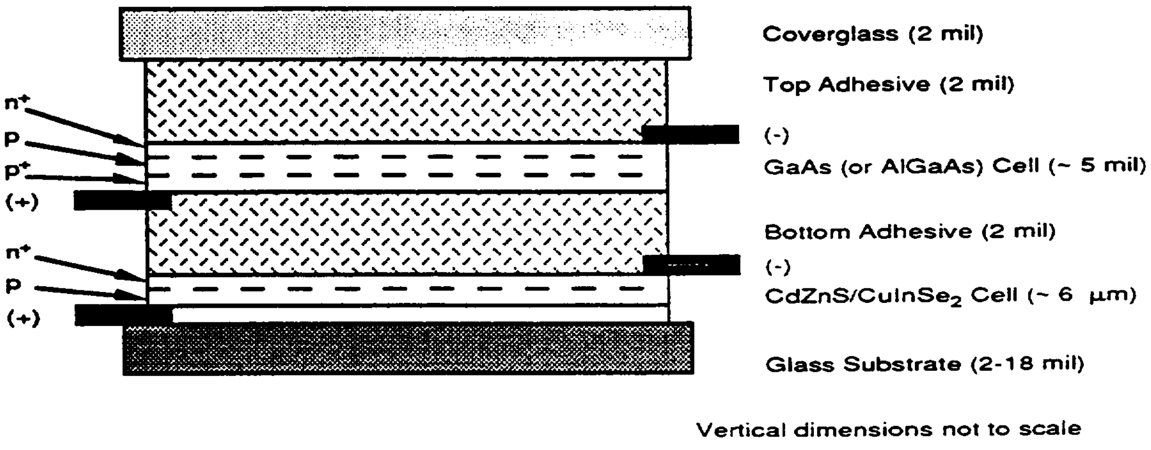


Figure D-2. Tandem cell schematic.

Cascades can be configured as a monolithic cell in which the top cell is integrally deposited on the bottom cell (or vice versa), or mechanically stacked, in which the two sets of cells are formed separately. A mechanically stacked tandem configuration was chosen by Boeing in order to provide wiring flexibility and to minimize fabrication constraints. Since both cells are deposited as thin films, a very high specific power is possible.

The current Boeing design is based on a 2 cm x 4 cm cell area (Ref. D-2). The cell has an improved two-terminal configuration with voltage-matched monolithic subcell units. Voltage-matching is achieved by stacking one GaAs CLEFT (Cleavage of Lateral Epitaxial Film for Transfer) cell on top of four $CuInSe_2$ subcells monolithically interconnected in series to form a single cell unit.

An improved cell design could be achieved by using AlGaAs rather than GaAs as the top cell. The bandgap of $CuInSe_2$ is better matched to the bandgap of AlGaAs than GaAs. A 26% Beginning-of-Life (BOL) cell efficiency at AM0 is projected without structural change for this advanced cell.

The upper thin film cell is fabricated by the CLEFT technique using MOCVD for cell structure growth. The CIS cell fabrication includes sequential depositions of the Mo back electrode, CIS absorber layer, and CdZnS window layer. This is followed by photolithographic patterning and etching to form a solar cell device. Deposition of grid metal and addition of an anti-reflection coating complete the cell fabrication process.

The individual PV panels are either small enough to be transported to Mars as designed or are hinged for easy deployment. The panels lie horizontal and do not track the sun. Trade studies done by Rocketdyne showed that there was no advantage to tracking arrays since much of the cell input is from diffuse light.

Planar arrays can be subdivided into two broad categories: rigid and flexible (Ref. D-6). Rigid arrays are PV arrays that are mechanically stiffened with a honeycomb structure, usually made of aluminum, sandwiched between two facesheets thatprovide back side shielding for the PV cells and provide enhanced mechanical support to the structure. Rigid arrays are used in high risk, unknown environments, where weight and cost are a secondary concern. Rigid arrays provide the most mechanical support and the most survivability. However, the rigid array is not suitable for planetary surface applications due to the mass penalty.

Flexible arrays are typically a Kapton substrate/superstrate which sandwich the PV cells and electrical circuitry, and are very attractive due to the significant mass savings over rigid panel arrays. Within the flexible arrays, there are three types: the roll-out type, the fold-out type, and the inflatable type. A flexible array is recommended for SEI surface power applications.

The Advanced Photovoltaic Solar Array (APSA) is an example of future flexible fold-out arrays (Ref. D-7). There are three elements to the APSA array: the flexible plastic and solar cell blanket (contributes about 50% of the mass), the deployment mast and mast housing (accounts for approximately 34% of the mass), and the blanket stowage compartment (16% of the mass) (Ref. D-6). The APSA wing consists of a flatfold, multiple panel, flexible blanket on which solar cell modules are installed and connected to printed circuit electrical harnesses that run along the outside longitudinal edges of the blanket assembly. Any type of solar cell can be utilized with the APSA approach. For launch, the accordion-folded blanket is stowed in a graphite/epoxy blanket during launch. The blanket is deployed (unfolded) by extending a motor-actuated, fiberglass, continuous longeron lattice mast that uncoils from an aluminum cylindrical canister structure. APSA is designed for zero g operation. Additional support structure would be required for planetary surface applications.

Inflatable arrays offer promise for missions which are driven by mass and/or the radiation environment. Inflatable arrays require the use of thin film PV solar cells. For a 100 W array (EOL) the mass breakdown is as follows: inflatable torus contributes 0.793 kg, the thin film solar array blanket 0.793 kg, and the support equipment 1.36 kg, resulting in a small satellite array specific power of about 34 W/kg (Ref. D-6).

An array deployment mechanism is required for automatic or robotic deployment. Array deployment alternatives include spring stored "one time deployment" and motor-driven deployment systems. The deployment mechanism is critical from a reliability standpoint and may also contribute significantly to the system mass. The deployment mechanism for Mars systems must also maintain array rigidity during wind storms and must be designed to prevent resonating of the structural due to the natural frequency.

Some form of structure may be used to keep the arrays off the surface or to prevent the arrays from being disturbed by Martian winds. This system would have increased mass over one which is simply rolled out onto the planetary surface.

RFC Description

The RFC converts electrical energy into chemical energy and stores the energy for future use. An RFC is an energy storage device similar to a secondary battery. The RFC can be divided into six major subsystems for development purposes: (1)a fuel cell stack, which electrochemically converts hydrogen and oxygen into electricity; (2)an electrolyzer cell stack, which electrolyzes the fuel cell product water into gaseous hydrogen and oxygen reactants using externally provided power; (3)water management which removes moisture from the electrolysis cell product gases and humidifies fuel cell reactants to maintain proper cell membrane moisture content; (4)thermal management, which removes waste heat from the system, maintains the proper membrane temperature, and prevents boiling or freezing in critical flow paths; and (5)reactant storage (hydrogen, oxygen, and water).

An RFC operates as a fuel cell during its discharge energy production phase and as an electrolyzer during its charge phase. An external power source must be used to provide power to the electrolyzer. The electrolyzer can produce high pressure oxygen and hydrogen gas for a gaseous reactant storage of relatively low volume.

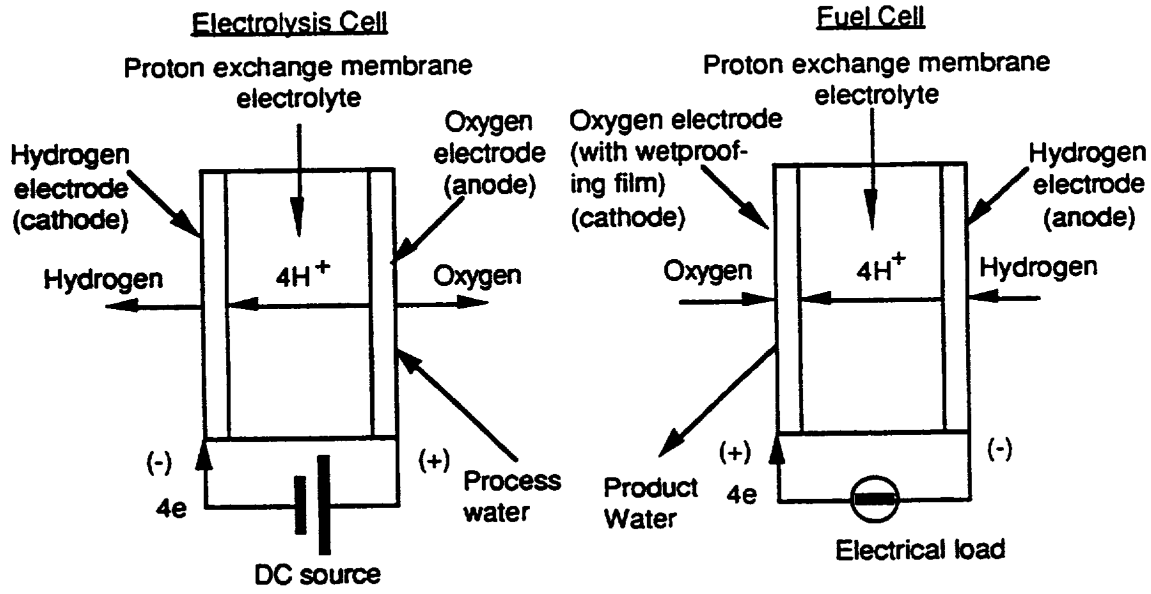
Two types of fuel and electrolysis cell technologies are available: alkaline and Proton Exchanger Membrane (PEM). PEM fuel cells and electrolysis cells were selected for this study since these technologies were shown to be the preferred RFC approach in a recent LANL study done for NASA (Ref. D-8). PEM cells were recommended since they are the only technology which can meet the efficiency, life, cycle life, and turndown required for lunar and Mars applications. In addition, PEM electrolysis cells are superior to alkaline cells in ruggedness and reliability. The basic design and operation of the PEM RFC system are described in the following paragraphs.

Fuel Cell Description. Fuel cells operate by separation of two electrocatalytic conversion reactions with an ionic conductor, as seen in Figure D-3 (Ref. D-1). Charge moves through electron conductors connecting the two electrocatalytic zones, where electron transfer results in chemical reactions. Ionic transport through the separator completes the process.

The PEM fuel cell incorporates an ion exchange membrane, typically a polyperfluorosulfonic (PFSA) acid sheet, as the ionic conductor. This component sustains transport of hydrated hydrogen ions, protons (H^+), associated with water. Protons are generated at the porous anode electrocatalytic layer and transport through the ionic conductor to the cathode electrocatalytic layer. At the cathode, protons react with oxygen to form water. Product water exhausts from the cathode compartment.

Hydrogen and oxygen gases are stored at 2.07×10^7 N/m^2 (3,000 psia) (Ref. D-9) for use in the fuel cell. The gases are regulated down to fuel cell operating conditions (4.14×10^5 to 6.9×10^5 N/m^2 or 60 to 100 psia). Oxygen is regulated to a few psi higher than hydrogen (for safety reasons) to insure that only oxygen is entrained in the product water. The reactant gases must be humidified prior to reacting in the stack. Humidification will be discussed in the water management section.

The hydrogen and oxygen gases are combined in the fuel cell to generate electricity and water. The product water is discharged into the cooling water loop. As the cooling water accumulator approaches the filled condition, the product water drain valve opens to allow water to flow to the storage subsystem.



CATHODE REACTION:

$4H^+ + 4e^- \rightarrow 2H_2$

ANODE REACTION:

$2H_2O \rightarrow 4H^+ + 4e^- + O_2$

CATHODE REACTION:

$4H^+ + 4e^- + O_2 \rightarrow 2H_2O$

ANODE REACTION:

$2H_2 \rightarrow 4H^+ + 4e^-$

Figure D-3. Acid PEM electrochemical cell reactions.

The product water which leaves the fuel cell stack will be saturated with oxygen. This oxygen must be removed prior to entering the water storage tank. An approach for doing this has been demonstrated (external to the fuel cell) by Hamilton Standard, as seen in Figure D-4 (Ref. D-9). Water passes through an ion exchange membrane from the product water stream to humidify the dry hydrogen gas. Hydrogen diffuses from the hydrogen stream through the membrane to the water stream and combine with the oxygen to form water. Excess hydrogen is returned to the hydrogen stream using an electrochemical hydrogen pump. Thus, only gas free water returns to the storage tank.

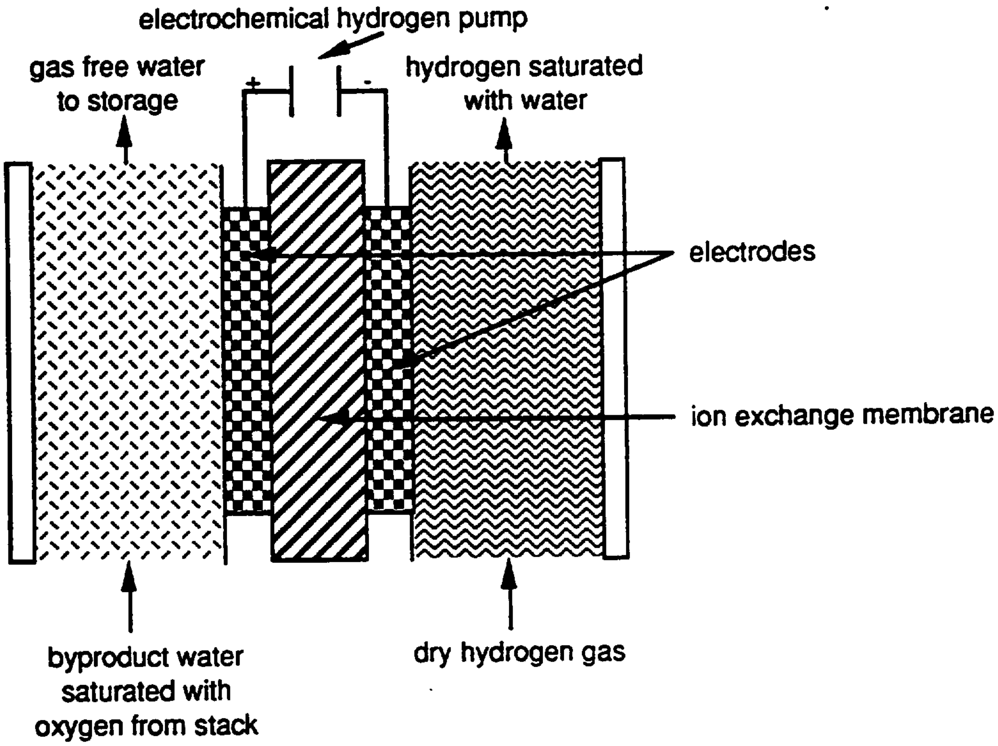


Figure D-4. Fuel cell product water deoxygenator.

The fuel cell design options have to do with the type of membrane. Table D-1 (Ref. D-9) compares the fuel cell design options on a power density basis. The values in this table assume a system with 25 kWe net output continuously, 55% fuel cell thermal efficiency (based on 1.48 VDC) for 20,000 hours, and a design that is thermal vacuum compatible.

TABLE D-1. FUEL CELL DESIGN OPTION POWER DENSITY COMPARISON

Fuel Cell Subsystem Description	Nafion 120 Membrane (current)	Nafion 125/117 Membrane (advanced)	Dow Membrane (advanced)
	W/kg	W/kg	W/kg
"SOA Design" with Porous Hydrophillic Phase Separators (Space Station design)	103	184	307

Electrolyzer Cell Stack Description. The acid PEM electrolyzer has the same type of ion exchange membrane as the PEM fuel cell to transfer H+ protons from the anode to cathode as was shown on the left of Figure D-3. Liquid water is pumped from the water storage tank by the water feed pump to the electrolyzer, as was seen in Figure D-1. The water feeds into the cell stack on the anode side of each cell. Some water passes across the proton exchange membrane (PEM) forming a second water loop. Excess anode water loop flow is used to remove waste heat from the stack. Separators in the cell stack separate the hydrogen and oxygen gases from the liquid water streams, as seen in Figure D-5 (Ref. D-9). The separated gases (saturated with water vapor) are fed into regenerative dryers or are fed directly to the reactant storage tanks.

Various electrolyzer options were identified involving different cell spacings and different membranes. The estimated power densities for each design is shown in Table D-2 (Ref. D-9). Table D-2 values assume a system with 70% electrolyzer thermal efficiency for 20,000 hours, $2.07\text{x}10^7$ N/m^2 (3,000 psia) gas generation pressure, thermal vacuum compatible design, and 13 kg per hour of water electrolyzed. Only the Nafion 120 membrane has been life tested at $2.07\text{x}10^7$ N/m^2 (3,000 psia).

TABLE D-2. ELECTROLYZER DESIGN OPTION POWER DENSITY COMPARISON

Electrolyzer Subsystem Description	Nafion 120 Membrane (current)	Nafion 125/117 Membrane (advanced)	Dow Membrane (advanced)
	W/kg	W/kg	W/kg
"SOA Design" with Static Separators	258	327	377
"Advanced Design" with Static Separators	347	392	414

The state-of-the-art (SOA) electrolyzer design utilizes the cell design which is used for U.S. and Royal Navy submarines. This cell design allows for 2.75 cells per cm (0.32 cm thick). The U.S. Navy utilizes a $2.07\text{x}10^7$ N/m^2 (3,000 psia) stack while the Royal Navy uses a $1.03\text{x}10^6$ N/m^2 (150 psi) stack design.

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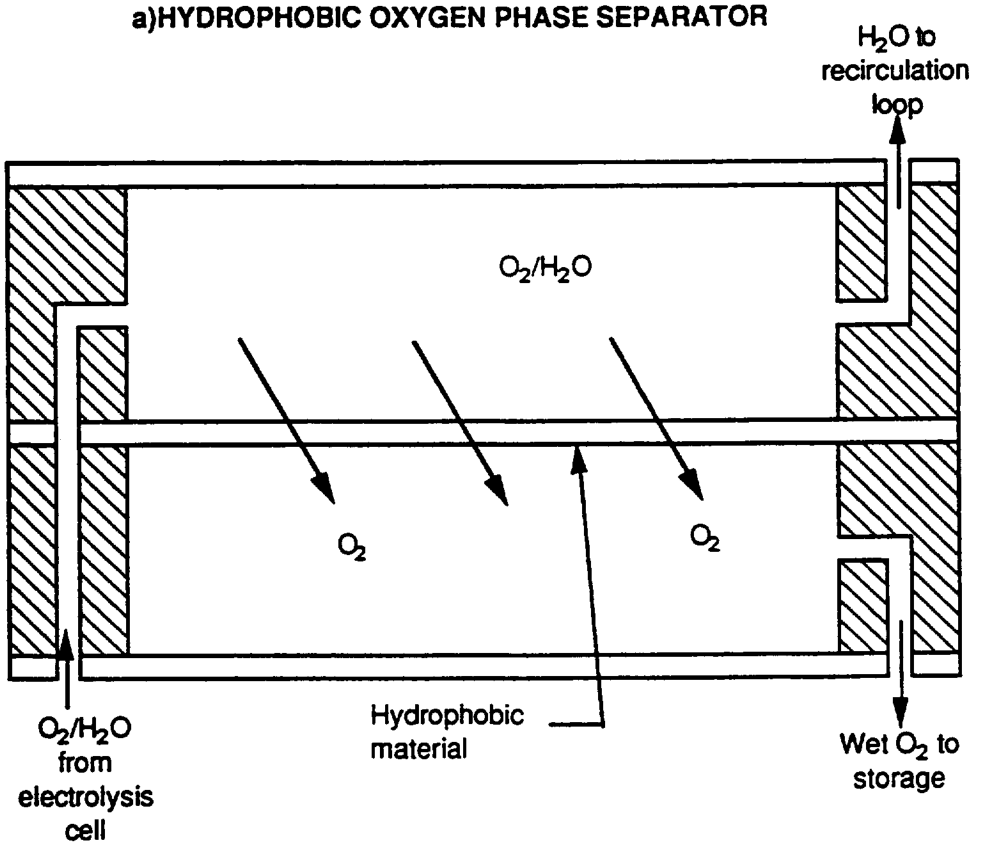
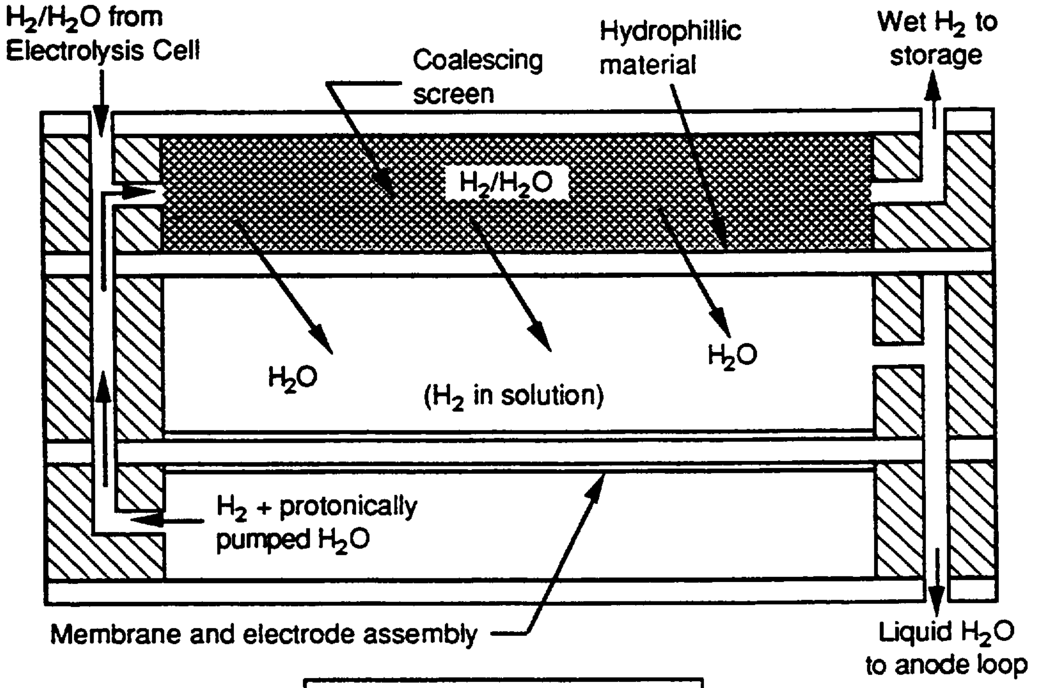
a)HYDROPHOBIC OXYGEN PHASE SEPARATOR



b)HYDROPHILIC/ELECTROCHEMICAL HYDROGEN PHASE SEPARATOR



Note: Dimensions not to scale.

Figure D-5. Oxygen and hydrogen phase separators in electrolyzer stack.

The "advanced" (Ref. D-9) electrolyzer design utilized a cell design of 12 cells per cm (0.083 cm thick). This cell was incorporated into a 120 cell stack for testing by the U.S. Navy as a low mass oxygen generator prototype. The advanced cell stack was designed for a maximum pressure of 2.76×10^6 N/m^2 (400 psi) without a housing. SOA cell stacks use a cell size of about 214 cm^2 (16.5 cm circular cell). This appears to be the optimum efficiency cell size for several 2.07×10^7 N/m^2 (3,000 psia) electrolyzer applications (Ref. D-10).

RFC Thermal Management Subsystem Description. The thermal management subsystem provides temperature control, heat transport, and heat rejection functions. Pumped water and coolant loops provide the heat transport function, as seen in Figure D-6. A radiator is required to remove waste heat from the RFC system. There are inefficiencies in both the fuel cell stack and electrolysis cell stack which create waste heat. Membrane temperature must be controlled to prevent failures and meet life requirements. Water cools the stacks by collecting the waste heat and then transports the heat to one or more heat exchangers (one for electrolyzer and one for the fuel cell, or possibly a combined heat exchanger). These heat exchangers then transfer heat to the radiator coolant loop. Waste heat from the fuel cell may also be utilized to keep the electrolyzer from getting too cold.

The heat rejection assembly provides a means for rejection of waste heat to the environment. Radiators for heat rejection are in some cases a major component of the power system mass. Radiators can also be quite large due to the low operating temperature.

Various options are available for the radiator design. Pumped loop radiators have been used successfully for the space shuttle and will be used for Space Station Freedom (SSF). This type of radiator is best applied to missions with limited duration or to systems which are serviceable. Pumped loop radiators are less massive than state-of-the-art heat pipe radiator designs. Heat pipes offer the advantage of improved reliability and a graceful failure mode. A recent Rocketdyne study has shown that advanced carbon-carbon (C-C) heat pipe radiators can be designed which are competitive in mass to pumped loop radiators. Thus, a heat pipe radiator was tentatively selected as the baseline design.

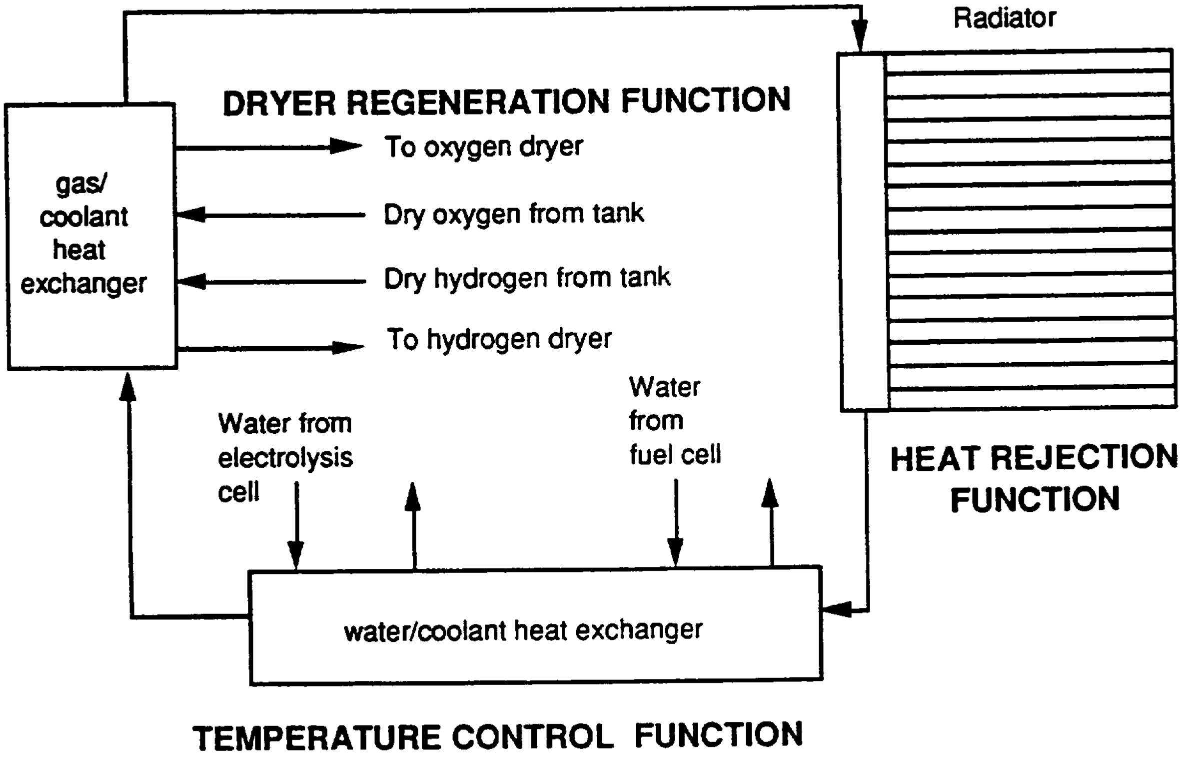


Figure D-6. Thermal management subsystem.

The baseline heat rejection assemblies for RFCs utilize lightweight, passive, reliable, heat pipe radiators that are sufficiently versatile to allow integration into a variety of configurations. The individual heat pipes operate independent from one another and thus the failure of a heat pipe will not result in failure of the complete radiator. The rectangular radiator heat pipe panel is attached to the coolant manifold. The cooling loop transfers heat to the heat pipes in the manifold heat exchanger. The heat pipe working fluid evaporates and travels to the top end of the heat pipes. The evaporated fluid is then condensed in the cooler section of the heat pipe. Both gravity and a small wick or grooves allow the liquid to return to the evaporator. A wick or groove is not absolutely necessary for vertically oriented radiators (due to gravity return) but is recommended to insure good control of the fluid transport. The heat pipes may be either carbon-carbon tubes with metal liners (Monel for water or aluminum for ammonia working fluids) or metal heat pipes.

Condensation and freezing of the water in critical locations must be prevented by maintaining fluid temperatures within limits. Thermal control of the lines and tanks may be accomplished by insulation, trace heating, insulation and trace heating, or convective heating/cooling using the radiator coolant. Composite tanks need to be kept above -65 °F to prevent tank failure. The composite tank liner will begin to separate from the overwrap at this temperature and buckling will occur. It may also be desirable to prevent the water vapor in the gases from freezing in the tanks. Fluid lines may also require thermal control to prevent water freezing and clogging of lines (especially in the pressure regulators).

RFC Water Management Subsystem Description. Water management includes moisture control of the fuel cell membrane and the removal of moisture from electrolysis module product gases.

The moisture content of the fuel cell stack membranes must be carefully controlled to prevent dehydration and reduced life. The reactant gases must be humidified to maintain the proper membrane moisture content.

The traditional humidification approach for PEM fuel cells is shown in Figure D-7 (Ref. D-10). The humidifier automatically presaturates the incoming hydrogen and oxygen reactants to a dew point equal to the cell operating temperature. This latter approach was used in the Hamilton Standard SPE fuel cell. The problem with this approach is that the product water going to the storage tank is saturated with oxygen. If the oxygen is not removed from the water, then the gas will accumulate in the water tank and have to be vented off (undesirable loss of reactant).

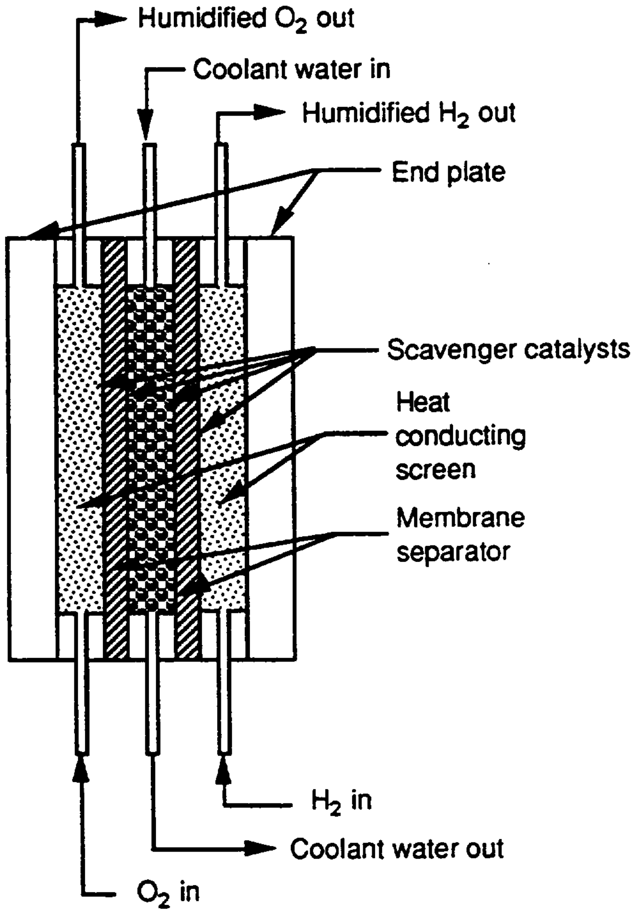


Figure D-7. SPE fuel cell reactant prehumidification approach.

Another gas humidification approach, which is more appropriate for space systems, was shown in Figure D-4 (Ref. D-9). This approach also removes the oxygen gas from the product water and there is no gas buildup in the tank. This concept converts the oxygen to water by diffusion of hydrogen across the membrane. A hydrogen electrochemical pump keeps hydrogen from evolving in the water. Excess hydrogen is pumped back to the hydrogen side of the device.

If regenerative gas dryers are used in the system, then the gas from the tanks will be partially rehumidified during regeneration of the dryers as is seen in Figure D-8 (Ref. D-11; only the oxygen humidification is shown). Heat must be added to the cool dry gases in order to vaporize the water in the dryers. The purpose of this process is primarily to recover water from the gas dryers. Most of the gas humidification will be done using a humidifier such as was shown in Figure D-4.

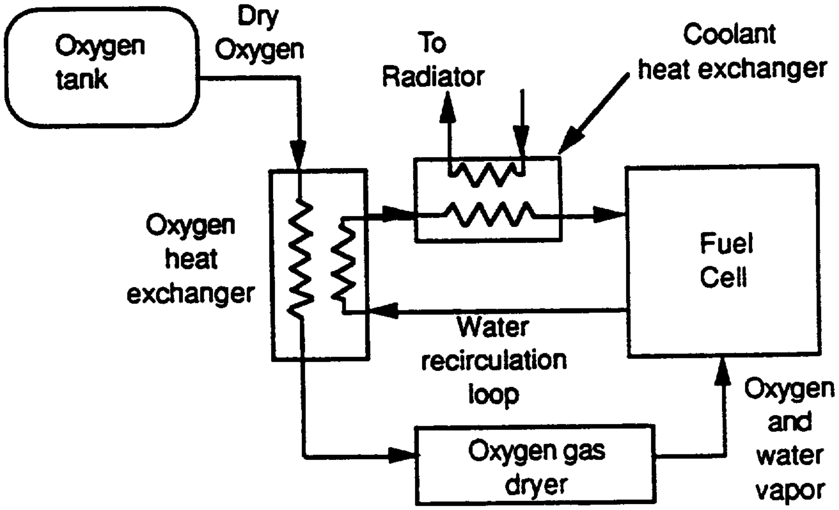


Figure D-8. Water recovery from the oxygen regenerative dryer.

Several approaches have been suggested for recovering moisture from the electrolyzer gases. The key concerns are preventing freezing of water in the lines and the mass loss from the system if the water is not fully recovered each operating cycle. The LANL study (Ref. D-8) proposes letting the water condense and freeze in the tank while keeping the feed lines heated. Some of the water is removed from the tank as a liquid by using a trap. The remainder of the water in the tank freezes. The residual water is recovered by heat input to the tank during the day as additional warm gas from the electrolyzer is introduced.

Another water recovery approach suggested by Hamilton Standard (Ref. D-11) is to dry the gases with regenerative desiccant dryers leaving only a trace of moisture in the gases. This would significantly lower the dew point for the remaining gases to a temperature which is below ambient or would not require much insulation or heat input to prevent condensation. A sacrificial dryer might also remove the remaining trace water. Moisture would be removed from the dryers during fuel cell operation by passing the dry gas from the tanks back through the dryers at the lower fuel cell pressure (need large driving force to recover the water). Potential desiccants include silica gel and molecular sieve (Ref. D-12). Silica gel is used at lower temperatures (below 363 °K) due to its high moisture recycling capacity compared with other industrial desiccants (about 0.35 kg water/kg gel at 303 °K and 80% humidity per Ref. D-13). Molecular sieve is used at higher temperatures (>393 °K) due to its high moisture recycling capacity and physical stability at higher temperatures. Potential configurations for desiccant dehumidifiers include a packed bed, Teflon fiber plates, a corrugated structure, and coated parallel-passages. A coated parallel-passage concept appears to have a great potential to provide an effective dehumidifier. This design consists of parallel-walled passages (laminar flow channels) with fine silica gel particles (80-250 mm) glued to the walls.

RFC Storage Tank Description. For this study, it was assumed that oxygen and hydrogen would be stored at high pressure. The storage tanks will be made of composite materials. A metal liner is overwrapped with carbon or graphite fibers. The liner will probably be aluminum for the hydrogen tank and a corrosion resistant material such as Inconel, niobium, or tantalum for the oxygen and water tanks.

Reliability and life are the key areas of interest for tank design. Thus, materials must be selected which are stable against corrosion and hydrogen embrittlement for the system life. Current composite tank designs may exhibit high stress when driven through large temperature variations, so the liner materials must be carefully matched to the wrap material in terms of the coefficient of thermal expansion. Otherwise, tanks must be thermally controlled to limit temperature changes (may be difficult when going from non-operating to operating status). Tank linings must exhibit very limited corrosion even with pressurized oxygen storage and perhaps even with wet gas storage. Tanks must be rugged enough to survive the transportation phase of deployment. The use of multiple tanks may be required to meet system reliability requirements. Some elementary cladding may be required to provide an element of shielding from meteorites (the system housing may also be used for this purpose).

PP&C Subsystem Description

The PP&C subsystem has not as yet been defined for this concept. Two basic approaches can be taken. In the first approach, which is the same as for the DIPS, the system is designed to provide a constant power and voltage output. This approach requires a dc voltage regulator to process the fuel cell output. In the other approach, the power conditioning is done at the user loads. In this second approach, the fuel cell output can vary with time (i.e., voltage is unregulated). This approach allows the power processing to be optimized for each load. The power input to the electrolyzer module can also be regulated within the RFC power system or as part of the recharging power system.

TECHNOLOGY ISSUES

Key issues for the PV subsystem are summarized in Table D-3. The key issues for development of a PEM RFC and their impacts are summarized in Table D-4.

TABLE D-3. PV SUBSYSTEM ISSUES, IMPACTS, AND DEVELOPMENT AREAS

Issues	Impacts	Potential Development Areas
Large array area due to low Martian insolation	•Increased life cycle cost (LCC) •Increased deployment time •Increased number of cells with reduced system efficiency and reliability	•Higher efficiency top cell •Robotic or automatic deployment system •Thin film arrays •Roll-out arrays
Small cell size	•Increased number of cells with reduced system efficiency and reliability	•Higher efficiency top cell •Larger size cells
Cell efficiency	•Increased array size and LCC	•Higher efficiency top cell (AlGaAs)
Cell cost	•Increased LCC	•Low cost production techniques
Deployment system and support structure weight	•Increased LCC	•Flexible roll-out array •Robotic deployment

Operating temperature fluctuation and extremes	•Reduced cell life due to thermal stress/increased LCC	•Design and test for appropriate environment •Test for thermal extremes
Martian wind	•Increased structure mass and LCC	•Lightweight structure and tie-downs
Dust accumulation	•Increased array area and LCC •Maintainence cost	•Robotic dust removal system

TABLE D-4. PEM RFC DEVELOPMENT ISSUES

Issues	Impacts	Potential Development Areas
Limited life components and system reliability	•Increased frequency of replacement, maintenance •Mass and complexity of redundant components	•Development of passive system •Long life pumps, drives,valves, and controls
Material compatibility	•Reliability/life	•Materials for use with high pressure O_2 •Materials for wet gases •Materials immune to H_2 embrittlement
Cell temperature and moisture control of fuel cell membrane	•Life	•Thermal control loops •Passive internal fuel cell gas humidifiers •Regenerative gas dryers
Oxygen in fuel cell water	•Mass/energy loss from the system due to venting of oxygen from water tank	•Internal deoxygenator in fuel cell
Water in electrolyzer gases	•Tank corrosion if wet gas stored (life and reliability) •Tank insulation mass •Complexity of gas dryer systems •Clogging of lines due to ice •Energy and mass loss	•Low mass desiccating regenerative dryers •Tank liner materials •Tank and/or line thermal control
Large, massive radiator due to low heat rejection temperature	•Increased transportation cost, complicated vehicle design	•Higher temperature cells (higher reject temperature) •Low mass carbon-carbon radiator •Heat pump
Reactant storage system mass and volume	•Transportation cost	•Cryogenic or supercritical storage
Efficiency of electrolysis cell reduced at higher pressure	•Transportation cost •Increased waste heat; larger radiator	•Low mass tanks, PV arrays, and radiators •Tank pressure following
High Water Purity Requirement	•Performance •Life	•Use materials that won't contaminate water •Deionizer

TECHNOLOGY ASSESSMENT

The technology bases were assessed for the following major PV/RFC subsystems:

- PV array;
- fuel cell stack;
- electrolysis cell stack;
- RFC thermal control subsystem;
- RFC water control subsystem;
- RFC reactant storage subsystem; and
- PP&C.

The technology readiness of each assembly was evaluated using the NASA technology readiness levels. These evaluations are summarized in Table D-5 which shows that the RFC has technology readiness levels ranging from 3 to 5, depending on the particular subsystem. The technology base for the PV and RFC subsystems is discussed in the following sections.

TABLE D-5. PV/RFC TECHNOLOGY ASSESSMENT

Subsystem	Technology Readiness Level	Comments
GaAs/CIS PV cells	5	Pilot development phase for cells; APSA program
Fuel cell stack	3.5	Early design flown on Gemini; prototype developed for space station RFC based on earlier Hamilton Standard design; new International Fuel Cell design not flown (engineering qualified in 2.5 years)
Electrolysis cell stack	4	Large database for naval applications; prototype developed for space station applications
RFC thermal management	3	Radiator component development currently underway; long life active thermal control components not developed for space applications; some sealed water pumps and drives for terrestrial applications have demonstrated long life
RFC water management	3	Silica gel dryers widely used for gas drying in terrestrial applications; limited experience with regenerative systems
RFC reactant storage tanks	5	Small tanks successfully flown; need corrosion resistant liner development
PP&C	5	Similar components under development for Space Shuttle Freedom

PV Array State-of-the-Art

The current cell technology level is between 5 and 6 on the NASA scale. The Boeing cell development is in the pilot line phase now (Ref. D-14). Production volume is expected by 1996. The GaAs top cell is currently made by Kopin company while the bottom CIS cell is made by Boeing. An advanced cell design (AlGaAs/CIS) could be available before 2008. Cell testing has progressed through the preliminary qualification stage. Testing has included thermal cycling, UV illumination, off-angle, vacuum stability, and humidity tests. Small coupon panels have been tested for mechanical shock, pyro shock, acoustics, vibration, and thermal cycling. Current cell size is 2 x 4 cm (1991). Boeing is also working on a thin plastic substrate to replace the current glass substrate.

Array development has been done for both rigid and flexible arrays. Various array technologies are summarized in Table D-6 (Ref. D-6). The roll-out type array approach was developed in the early 70's at Wright-Patterson Air Force Base during the Flexible Roll-Up Solar Array (FRUSA) program. The same basic tgechnology is used today for power generation on the Hubble Telescope. Some of the arrays in the flexible fold-out category are the SAFE array, Mil-Star, Space Station and the APSA array developed by TRW for JPL. The APSA array is designed for GEO orbit. The APSA program has been ongoing for several years and has significantly reduced the mass of arrays using a flexible array design. APSA has a goal of demonstrating a producible array system having a specific power greater than 130 W/kg (BOL) at 10 kW (BOL). A research and development array that should be seen as both "fold" and "roll" out is the inflatable array. The inflatable array approach is funded by DARPA and is being developed and build by L'Garde Inc . The DARPA program is called Inflatable Torus Solar Array Technology (ITSAT) and is scheduled to conduct a space flight experiment in the 4th quarter of 1993. The inflatable array is considered a high risk - high payoff approach.

TABLE D-6. DESCRIPTION OF VARIOUS ARRAY TECHNOLOGIES AND APPROXIMATE PERFORMANCE FIGURES OF MERIT (IN EARTH ORBIT)

Name	Description	PV CELS (BOL)	Base Power (EOL kWe)	Life (years)	Area Density (W/m^2)	Specific Power (W/kg)
Rigid Honeycomb (Typical)	Aluminum honeycomb	Si: BSF 8 mil	3.4	7	128.3	32
HUBBLE	Roll-out flexible blanket	Si: 2x4 cm^2 BSFR	4.4	2	117	19
Space Station	Flexible Kapton fold-out blanket	Si: wrap through contacts 8x8 cm^2	75	4	90	66
APSA	Flexible Kapton fold-out blanket	Si: 2x4 cm^2 BSF 13.8%	3.7	10	95	93
SUPER (Planar)	Flexible fold-out with Beryllium	GaAs/Ge 6x6 cm^2 18%	5.3	5	121.7	26.5
AHA	Ti honeycomb with 1.4 x concentrating shutters	GaAs/Ge 2 junction 24%	5	10	95.5	28.8
ESSA	Al honeycomb with protective shields	GaAs/Ge single junction 18%	5	10	137	23.2

The current application cell design could utilize either an inflatable array, an APSA type array (with additional support structure), or roll-out array.

RFC State-of-the-Art

Hamilton Standard designed, developed, tested, and delivered a 1.5 kW PEM breadboard RFC system to Johnson Space Center (JSC) in January 1983 (Ref. D-11). The RFC breadboard system was tested for 2,000 ninety minute orbital cycles (1,630 at JSC). The fuel cell module of the RFC was later replaced by an advanced module and tested for about 500 hours. This breadboard was not tested in a relevant environment (i.e., vacuum or low pressure carbon dioxide, low gravity, day/night thermal cycles, etc.). In addition, the breadboard did not include

key components such as composite tanks, radiators, and long life active thermal control components.

Although the fuel cell technology is fairly well developed in acid systems, the system integration of the accessory components and cell stack is not as mature as that of the alkaline fuel cells. The technology readiness of the PEM RFC power system is estimated to be 3.5 for the current application.

Fuel Cell State-of-the-Art

A PEM fuel cell developed by Hamilton Standard (United Technologies) was used on the Gemini missions from 1962-66 (Ref. D-15). After the Gemini space flights, Hamilton Standard pursued further development of PEM fuel cell technology. The major breakthrough was the replacement of polystyrene sulfonic acid ion exchange membrane by perfluorinated sulfonic acid polymer, Nafion, produced by DuPont, as the electrolyte. PEM fuel cell technology has since advanced with the introduction of the Dow experimental membrane. The Dow membrane has greatly increased the current density of PEM systems over the current densities available from DuPont's former state-of-the-art membrane, Nafion 117. Dow has not yet started production of their membrane , but is supplying it to fuel cell manufacturers for testing and evaluation.

Cell lives of 60,000 hours (6.85 years) in the laboratory have been achieved by Hamilton Standard with PEM fuel cells (Refs. 10 and 11) because there are no corrosive electrolytes to cause contamination. PEM fuel cells can operate with high concentrations of gases like CO_2, whereas the KOH electrolyte of alkaline fuel cells would react with the CO_2 and cause precipitation.

A PEM fuel cell subsystem was developed by Hamilton Standard in the 1980s for a RFC demonstrator for the Space Station Freedom (Ref. D-1). The fuel cell had a 1 to 2 kWe rating. The RFC demonstrator underwent parametric testing at the factory prior to its delivery to NASA/JSC.

Treadwell Corporation has designed and built a PEM fuel cell stack and associated test stand (Ref. D-16). The stack was designed for an output power of 10 - 30 kW. Various stacks have been tested. This fuel cell system was designed for autonomous underwater vehicles.

Ballard Technologies Corporation in Canada has built small demonstrator stacks with the Dow membrane and was the first to achieve high power densities in a solid polymer electrolyte fuel cell (Ref. D-15). The Ballard design appears to be similar to the Hamilton Standard fuel cell design.

Siemens in Germany (under a license from Hamilton Standard) is also using the PEM technology to develop fuel cell systems for submarine power systems (Ref. D-15).

LANL has two of the Dow PEM fuel cells on test, achieving 0.92 V at 2,153 A/m^2 (Ref. D-15). Dow has made a commitment to provide membranes to a product specification; whereas, they were previously in process development and membrane quality/consistency were not up to par for commercial use.

Acid (PEM) fuel cells are well suited to passive water removal. The absence of a liquid electrolyte that has narrow concentration limits makes water management less of a problem. Ergenics Power Systems, Inc., is developing a flight-qualified 200 W fuel cell with passive water and heat removal for a Space Station extravehicular mobility unit (Ref. D-15).

International Fuel Cells (IFC) has tested a 16 cell, 5 kW stack using Nafion membranes and is now evaluating the Dow membranes. This is a new PEM design which is different from the fuel cell which flew on Gemini. IFC also worked on a "static" PEM fuel cell (Ref. D-17). This concept eliminated the power consuming pumps associated with the management of the product water. The design also incorporated heat pipes into the system to create a "static" waste heat management subsystem which eliminated the cooling subsystem parasitic power loss. This approach offered significantly improved reliability and higher system efficiency. IFC completed breadboard experiments and validated this system concept. IFC has since changed their design (Ref. D-17). The latest design has no heat pipes and requires a cooling loop with a pump. However, the water removal still utilizes a static approach. This concept is proprietary and

few details were available from IFC. However, this concept uses graphite plates and either Dow or Nafion 117 membranes (Nafion is the baseline). IFC has tested both single cells and short stacks with its latest design. IFC has a DARPA contract to produce a 7.5 kWe fuel cell for unmanned underwater applications. This power plant will be available (engineering qualified) in 2.5 years. The life of this fuel cell is expected to be a few thousand hours. The DARPA fuel cell technology should be suitable for space since it is not affected by a zero gravity environment.

Electrolysis Cell Stack State-of-the-Art

Hamilton Standard has an extensive data base in high-pressure electrolysis. The 2.07×10^7 N/m^2 (3,000 psi) cell design is currently used in the oxygen generation plant (OGP) developed for the U.S. Navy and in the production units for the British Navy nuclear submarines. Hamilton Standard has over 20 years experience building PEM electrolyzers for the Navy (Ref. D-15). They have demonstrated 13 years of continuous usage of a PEM electrolyzer cell in the laboratory. U.K. Navy electrolyzer cells have accumulated a total of 69,000 system hours of usage as of 1/92 without any failures. One Navy electrolyzer cell stack has accumulated over 13,000 hours of usage at sea over a 5 year period (Ref. D-11).

During the 1980s, three demonstrators were developed by Hamilton Standard (Ref. D-1). These electrolyzers were fabricated and then tested by NASA. Each of these systems made use of the identical 213 cm^2 SPE water electrolyzer design used for naval applications.

The first of these systems was a PEM RFC demonstrator for Space Station Freedom. Over 2,000 simulated orbital cycles were accumulated on this hardware. This program demonstrated a closed system fluid cycle balance, direct solar array/electrolyzer voltage/current control compatibility (no power conditioning required), and an energy storage efficiency of 48% with the electrolyzer at ambient temperature. Later in the program the PEM fuel cell was replaced by an alkaline fuel cell and the system was operated for 100 cycles with no problems. This test showed the compatibility between a PEM electrolyzer and an alkaline fuel cell (i.e., no KOH ions went to the PEM electrolyzer through the product water and no sulfonic acid groups passed through the PEM electrolyzer gases to the fuel cell; some people had thought that the acid and base in each unit would mix and neutralize each other). Recently, some of the electrolyzer cells were replaced by high performance cells using the Dow membrane. The electrolyzer module underwent additional testing and showed significant performance improvement, especially at higher current densities.

The second Hamilton Standard demonstrator was an oxygen generator assembly developed under contract to Boeing Aerospace and Electronics Company (Ref. D-1). The operating pressure, temperature and current density of this demonstrator are within the experience of naval applications. However this demonstrator differed from the Navy data base because of the need to operate in a microgravity environment and use processed hygiene water as the feedstock. Two membrane static phase separators are used to replace the pressure vessel phase separators used previously. The demonstrator was activated at NASA/MSFC in November 1990 and operated for 529 hours which exceeded the test objective of 450 hours. This program successfully demonstrated the operation of microgravity phase separators. There are continuing tests of this unit to improve the cell voltage performance.

The third Hamilton Standard demonstrator system was developed to show the feasibility of producing 2.07×10^7 N/m^2 (3,000 psi) hydrogen and oxygen on orbit for periodic rocket motor firing to maintain Space Station Freedom orbital altitude (Ref. D-1). Under NASA sponsorship, initial work was performed to convert the heavy 2.07×10^7 N/m^2 (3,000 psi) naval SPE electrolyzer design into a space flight configuration (Ref. D-10). This required development of a lighter and smaller package. Changes were made to the supporting pressure vessel and fluid manifold. The use of two torispherical domes opposed on either side of a central fluid plate allowed for a wall thickness of as low as 0.64 cm when using Inconel or other high strength materials. The fluid plate manifold is pressure balanced between the two pneumatic domes which eliminates the need for a thick plate to resist the gas pressure load, as used in the Navy hardware. This work produced a prototype cell stack for space applications that weighs less than 91 kg total (down from 454 kg for the 100-cell naval cell stack and pressure vessel). This unit was delivered in 1990. This demonstrator has been set up and operated intermittently at NASA/JSC during the last year. This cell was recently tested by JSC for 500 hours (Ref. D-1). This unit operates at 322 °K at a thermal efficiency greater than 70% (defined as the ratio of the power input minus the heat rejected to the power input).

RFC Thermal Management Subsystem State-of-the-Art

NASA LeRC is currently carrying out an integrated multi-element project for the development of space heat rejection subsystems with special emphasis on low mass radiators in support of SEI power system technology (Ref. D-19). This effort involves both in-house and contracted efforts. Contracted efforts involving Rockwell International (RI) and Space Power Incorporated (SPI) are aimed at the development of advanced radiator concepts (ARC). NASA LeRC is also involved in a joint program with DOE to demonstrate a flexible fabric heat pipe radiator concept being developed by Pacific Northwest Laboratories (PNL). In-house work at NASA LeRC is designed to guide and support the overall program by system integration studies, heat pipe testing and analytical code development, radiator surface morphology alteration for emissivity enhancement, and composite materials research focused on the development of low mass, high conductivity fins. This program is concentrating on technologies capable of development before the end of the decade for both surface power and nuclear electric propulsion (NEP) applications.

Specific objectives of the ARC contracts are to achieve specific mass values <5 kg/m^2 with radiator surface emissivities of 0.85 or higher at typical radiator operating temperatures, and reliability values of at least 0.99 for the heat rejection subsystem over a ten year life. These figures represent a factor of two improvement over the currently considered heat rejection subsystem for SP-100, and even greater improvement factors for state-of-the-art heat rejection systems used in current spacecraft applications.

Presently, Rocketdyne division of Rockwell International is involved in the development of a carbon-carbon (C-C) or graphite heat pipe radiator panel. Rocketdyne has a NASA contract entitled "SP-100 Advanced Radiator Concepts" which started in 1987 and is part of the CSTI (Civilian Space Technology Initiative) program. This program will be completed in January of 1993. This program involves development of a C-C heat pipe suitable for use in an SP-100 radiator. However, this technology will be suitable for other radiator applications as well including RFC power systems (using other working fluids such as water and/or ammonia). The hope is that this program will develop generic technology which can be used for all future space radiators because of its reduced mass compared to metal heat pipe radiators. Carbon-carbon is a new structural material which requires different fabrication techniques than for metals or composites. A key objective of this program is to demonstrate the ability to fabricate C-C heat pipes with a thin metal liner, wicks, caps, and fill tube. The brazing technique required to attach liners is the same as would be required to attach manifolds to the heat pipes. Heat pipes filled with potassium will be designed, fabricated, and tested during this program. Thermal cycle testing will be done in a vacuum to demonstrate the integrity of the heat pipes.

Beginning in 1989, small samples of heat pipe panels were fabricated by Rocketdyne as part of the CSTI program. The next phase of the program will involve fabrication and testing of complete heat pipes about a meter long.

The working fluids and temperatures for the SP-100 system would be different than for lower temperature RFC systems. Thus, Rocketdyne has an on-going IR&D effort which involves development of a high pressure water C-C heat pipe for lower temperature applications such as RFCs. This effort will eventually include fabrication and testing of a heat pipe. Rocketdyne is also looking at a higher thermal conductivity C-C heat pipe design.

Recent Rocketdyne studies have shown that a C-C heat pipe with water or ammonia is competitive on a mass basis with early metal heat pipe radiator technology developed in the late 1960s and early 1970s as well as pumped loop radiators. Rocketdyne studies have estimated the C-C radiator specific mass to be about 3 kg/m^2.

The objective of the joint NASA LeRC/Air Force program with PNL (Light Weight Advanced Ceramic Fiber [ACF] Heat Pipe Radiators) is to demonstrate the feasibility of a low mass ceramic fabric/metal liner heat pipe for a wide range of operating temperatures and working fluids (Ref. D-19). Specifically, the NASA LeRC objectives are to develop this concept for application to space radiators with operating temperatures below 500 °K using water as the working fluid. The specific mass goal for these heat pipes is 3 kg/m^2 or less at a surface emissivity of greater than 0.85.

Several heat pipes were built for the ACF program using titanium and copper foil material for containment of the water working fluid. A heat pipe with a 0.2 mm Ti liner was demonstrated in early January 1991. An innovative "Uniskan Roller Extrusion" process has also been developed at PNL and used to draw 0.76 mm wall tubing to a 0.2 mm foil liner in one pass. The water heat pipes fabricated for LeRC have been subjected to a test program to evaluate performance and reliability at demanding operating conditions.

Future thrusts of the ACF program will be to perfect the heat pipe fabrication procedure using very thin (0.025 to 0.05 mm) foil liners which are internally texturized by exposure to high pressures (Ref. D-19). Plans will also be developed to perform hyper-velocity and ballistic velocity impact tests in order to determine if secondary fragments from a penetrated heat pipe will result in failures of neighboring heat pipes. Another major challenge will be to design a heat pipe with high conductivity, low mass fins as a first step toward the fabrication of low mass radiator panels.

The NASA LeRC in-house materials program includes radiator surface morphology alteration by arc texturing for emissivity enhancement purposes (Ref. D-19). Emissivity enhancement has been demonstrated for graphite-copper samples.

Another important part of the thermal management subsystem is the active controls for controlling fluid flowrates and pressures. This involves the development of long life pumps, valves, and regulators. Hamilton Standard has used 2.07×10^7 N/m^2 (3,000 psi) water pumps for its U.S. Navy electrolysis units (made by J.C. Carter Company per Ref. D-11). These pumps have lasted over 10,000 hours. Hamilton Standard also has used a small 0.91 m^3/h (4 GPM) pump for recirculating water to a fuel cell which lasted over 10,000 hours. Existing active convective transport hardware (pumps, centrifugal water separators, fans) have also been developed for space systems by GE and Hamilton Standard, but not for long life (Ref. D-8). Experts in the field believe that these components will have lifetimes of less than 2,000 hours in a space environment.

RFC Water Management Subsystem State-of-the-Art

Water management involves both gas humidification to maintain proper cell membrane moisture content as well as gas dehumidification to prevent water condensation and freezing. Work has been done in both of these as areas, but not for space qualified hardware.

Gas humidification for production PEM fuel cells has been successfully done by Hamilton Standard using the approach shown in Figure D-6. An improved approach which removes oxygen from the product water was shown in Figure D-4. This approach was demonstrated in the laboratory in 1989, outside of a fuel cell (Ref. D-11). This approach has not yet been integrated into a PEM fuel cell design.

Water vapor can be removed from gases using condensation, absorption, adsorption, or a combination of these techniques. Gas drying is a well-known technical process. General Dynamics proposed (Ref. D-20) a system which utilizes condensation for removing 99.9% of the moisture from oxygen and hydrogen gases. The removal of the remaining moisture was assumed to be accomplished using absorption-adsorption techniques in existing ground liquefaction systems. General Dynamics adapted a system developed by Atlas Copco called MD sorption drying for propellant processing. This process recovers all of the water during the electrolysis portion of the cycle. However this is a very complex system with many moving parts (pumps, valves, and rotor in sorption dryer). In addition, this system increases the radiator size by reducing the heat rejection temperature.

Work was done with regenerative gas dryers for Hamilton Standard by the UTC Research Center in 1987 (Ref. D-11). These dryers use a molecular sieve to remove water. In tests, a humidified gas stream was run through a dryer bed for 36 hours without breakthrough. A 10% bed loading (amount of bed volume which was water) was achieved. An outlet dew point of less than minus 183 °K was obtained with a 100 to 1 gas volume ratio. Such a dryer removes about 99.9% of the water vapor from a gas. Based on this work, a 21 kg desiccant could be made regenerable and deliver 2,939 kg of dry gas (about twice that required for a 12.5 kW lunar base system). A much smaller unit would be required for a Mars system. Hamilton Standard has also looked at methods to reduce dew point temperature down to liquid hydrogen temperatures.

RFC Reactant Storage Tank State-of-the-Art

Gas storage tanks have had considerable engineering advancement, both in earlier NASA programs and throughout the commercial sector. Gas tanks have become safe, reliable commodities, widely used in science and technology. Composite tanks with a metallic liner and a polymer wrap are generally considered for advanced space applications.

Composite tanks have had considerable recent development for a wide range of sizes (Ref. D-21). Structural Composite Industries (SCI) has built many tanks for both terrestrial and space applications. These cylindrical tanks have either aluminum or stainless steel liners. These liners are seamless and are made from plate stock without welding. SCI has developed tanks for Space Station, Brilliant Pebbles, HEDI, propellant tanks for launch vehicles, communications satellites, and Pegasus projects among others. Eleven of these tanks have been launched into space. Tank sizes have ranged from 754 cm^3 to 0.66 m^3 (Space Station). A recent effort involved a tank which is 4.06 m long and has a 0.53 m outer diameter. A current effort involves the development of a 1.42 m diameter tank. Tank pressures have gone as high as 1.034×10^8 N/m^2 (15,000 psia) for operation and $2.068 X 10^8$ N/m^2 (30,000 psia) for burst.

Tanks for SDI missions are designed for 5 year life. Additional liner development may be required for storage of wet oxygen.

PP&C Subsystem State-of-the-Art

The PP&C system design is based on a reasonable electronics component evolution, and there are no significant technology issues associated with its development. It will be necessary to fabricate breadboard hardware for testing and evaluation purposes, but there is no need to initiate any advanced component development. Most of the hardware elements have been or shortly will be operating in a relevant environment on other spacecraft or the SSF electrical power subsystem.

DEVELOPMENT PLANS

The development program was divided into eleven major tasks. The first eight tasks are component development tasks. The last three tasks includes the system design, fabrication, integration, and testing for the Ground Engineering System (GES), Qualification Unit (QU), and Flight System (FS).

Testing will be done on the component, subsystem, and system level (qualification testing). The system level tests will show any possible negative interactions between subsystems. Additional testing will be done on the lunar surface.

The PV/RFC development tasks are described in the following sections. Each task will include a section on objectives and the statement of work. The task descriptions are only approximate and depend on the PV/RFC design chosen.

Task 1. PV Array Development

Objectives: Complete development of high efficiency tandem cell, a low mass flexible array, and an automatic or robotic deployment approach.

Statement of Work: The following subtasks are identified:

Task 1-1 Subscale PV Array Development - Complete development of GaAs/CIS or advanced (AlGaAs/CIS) tandem cell. Develop larger cell size. Test subscale PV panels to verify performance under appropriate environmental conditions (thermal cycling, vibration tests, strength and stiffness, radiation, etc.). Develop array structure and deployment concept.

Task 1-2 Full Scale PV Array Development - Develop and test full scale PV arrays. Verify adequate performance.

Task 2. PEM Fuel Cell Module Development

Objectives: Develop a full scale flight weight fuel cell module which can be integrated into a mobile RFC power system. Demonstrate materials compatibility, safety, and performance.

Statement of Work: This effort was divided into the following subtasks:

Subtask 2.1 Preliminary Fuel Cell Module Development - Demonstrate the feasibility of the selected fuel cell design for planetary surface applications. Investigate stack sealing materials, plates, membranes, humidifier, and diluent control. Test a prototype fuel cell stack to demonstrate performance, materials compatibility, and safety.

Subtask 2.2 Final Fuel Cell Module Design - Design a full scale flight weight fuel cell modules (one or more different sizes).

Subtask 2.3 Fuel Cell Module Fabrication, Assembly, and Testing - Build and test (performance, mechanical, and thermal cycling) the fuel cell module. After initial breadboard validation testing is complete, perform additional module testing in a relevant environment.

Task 3. PEM Electrolysis Cell Module Development

Objectives: Develop a full scale flight weight electrolysis module which can be integrated into a mobile RFC power system. Demonstrate materials compatibility, safety, and performance.

Statement of Work: This effort was divided into the following subtasks:

Subtask 3.1 Preliminary Electrolysis Cell Module Development Design - Demonstrate feasibility of the selected concept for space applications. Investigate options for the cell membranes, pressure vessel, central fluid plate, and electrical connections. Demonstrate materials compatibility, performance, and safety using a prototype cell stack.

Subtask 3.2 Final Electrolysis Module Design - Design a full scale flight weight electrolysis cell module.

Subtask 3.3 Electrolysis Module Fabrication, Assembly, and Testing - Build and test (performance, mechanical, and thermal cycling) the full scale electrolysis cell module. After initial breadboard validation testing is complete, perform additional module testing in a relevant environment.

Task 4. Thermal Management Subsystem Development

Objectives: Develop and demonstrate a low mass, reliable heat pipe radiator. Develop and demonstrate long-life active thermal control components (pumps, valves, regulators, etc.), as needed. Develop and demonstrate thermal control concepts for all components in the RFC system.

Statement of Work: This effort was divided into the following subtasks:

Subtask 4.1 Heat Pipe Demonstration - Fabricate and test representative length heat pipes to fully characterize heat pipe performance. Compare test results with predicted performance over the anticipated range of operating conditions including startup, shutdown, and restart.

Perform limited life testing of the heat pipes. Identify deterioration mechanisms, measure deterioration rates, and assess the adequacy of assembly and cleaning procedures. Withdraw heat pipe samples sequentially throughout the test period. Drain, section, and analyze the samples for corrosion, carbon diffusion, braze stability, and any other signs of deterioration.

In conjunction with the heat pipe performance and life testing, develop and demonstrate techniques for nondestructive examination (NDE) of the heat pipe elements and assembly.

Assess the adequacy of liner bonding and weld joints. Determine if the correct fit-ups and interfaces were obtained in the assembled piece.

Subtask 4.2 Radiator Enhancement - Increase the applicability of the radiator concept by performing various enhancement tasks. Examine survivability options and extended length heat pipes.

Perform analytical assessment, testing, and enhancement design development to insure long life for both lunar and Mars missions. Consider natural threats such as micrometeoroids and dust erosion (Mars).

Fabricate a long heat pipe. Develop alternate techniques for fitting the liner into the tube. Investigate alternative tube fabrication, liner fabrication, and coating processes.

Subtask 4.3 Heat Pipe Integration and Testing - Develop and demonstrate techniques for thermal and mechanical bonding of the heat pipe to the radiator manifold. Subsequently, demonstrate heat pipe integration into a representative radiator section. Test the radiator section to provide an accurate overall heat transfer coefficient. Verify radiator dynamics and performance. Include a surrogate manifold section and a limited number of heat pipes in the demonstrator. Test the assembled unit in a cold wall, vacuum chamber, simulated space environment. Validate temperature drop predictions, verify manifold design, and assess component interactions.

Subtask 4.4 Radiator Module Design, Assembly, Fabrication, and Testing - Develop the detailed design of the radiator subsystem. Resolve specific design issues such as the effects of differential thermal expansion, deployment features, monitoring instrumentation and control equipment, local insulation, and trace heating of the manifold and piping.

Design, fabricate, and test a representative full scale heat pipe radiator and interface heat exchanger. Complete performance, mechanical (stress, shock, and vibration), and thermal cycling tests.

Subtask 4.5 Preliminary Active Component Development - Perform preliminary development of water circulation pumps and other active fluid components. Identify life limiting components and failure mechanisms of current designs. Develop design approaches with improved life capabilities. Design and build prototype components. Demonstrate materials compatibility, safety, and performance margins.

Subtask 4.6 Final Active Component Design - Design the full scale components and develop the active control subsystem concept.

Subtask 4.7 Active Control Subsystem Fabrication, Assembly, and Testing - The final components will be designed and integrated into a fluid subsystem test bed with simulated thermal loads and pressure drops for other RFC components. Testing shall include performance, shock, pressure, and safety tests. After initial breadboard validation testing is complete, perform additional testing in a relevant environment.

Task 5. Water Management Subsystem Development

Objectives: Develop regenerative dryers for removing water vapor from the electrolyzer product gases. Develop an approach for recovering water from dryers. Develop a reactant gas humidification device for the fuel cell for maintaining a proper membrane moisture content.

Statement of Work: This effort was divided into the following subtasks:

Subtask 5.1 Gas Dryer Development - Develop regenerative gas dryers for use with wet hydrogen and oxygen gas streams. Design the dryers to remove the majority of the water vapor from the gas streams with reasonable size and mass hardware. Build prototype dryers. Test the dryers in the adsorption and desorption modes. Determine materials compatibility, regenerability, cyclic performance degradation, and efficiency.

Perform studies to determine the effect on system performance of any remaining moisture after leaving the regenerative dryers. Design an approach for recovery of the remaining moisture from the gas system. Develop additional components as necessary to handle this remaining moisture. Demonstrate analytically and empirically the performance of this additional hardware.

Subtask 5.2 Gas Humidifer Development - Complete the development of reactant humidifers which are integral to the fuel cell. This will include modification of existing humidifier concepts which currrently work external to the fuel cell. In addition, this task will be done in parallel with fuel cell development to allow for proper integration with the fuel cell stack design.

Task 6. Reactant Storage Tanks

Objectives: Develop reactant tanks for storage of wet hydrogen gas, wet oxygen gas, and water. Develop tank and tank feed line thermal management (insulation and/or heat recovery from fuel cell stack) approaches to prevent composite tank failure, and prevent ice blockage of gas lines and regulators.

Statement of Work: Design, fabricate, and test the reactant storage tanks. Develop corrosion resistant liners and fabrication techniques for the oxygen and water tanks. Consider the requirements for micrometeoride protection in the tankage design.

Develop insulated and/or heated tank concepts which prevent water condensation and freezing in the gas storage tanks and exit lines. Demonstrate proper transient performance of the tanks using analysis techniques. Build and test prototype tanks. Demonstrate materials compatibility and safety. Perform proof testing, thermal cycling (while under pressure), puncture resistance, and mechanical shock tests. Measure tank permeability to estimate leakage losses, especially for hydrogen.

Task 7. PP&C Subsystem Development

Objectives: Develop electronic components for controlling system operation, maintaining constant output voltage, and for eliminating excess solar energy. Demonstrate adequate steady state and transient performance, and immunity to the environment (including launch and operating). Demonstrate the software capability to handle power system nominal operation and failure modes.

Statement of Work: This effort was divided into the following subtasks:

Subtask 7.1 Electrical Components - Build breadboard units to demonstrate and check functional performance of the individual component circuit designs. Incorporate any necessary design modifications and improvements into brassboard units. Verify functional performance within the constraints of the actual component configuration. Fabricate prototype units and perform a series of tests using simulated input and output loads. Test the controller to validate the operating system software. Include the following testing:

- start up, steady state and transient control;
- failure simulation and detection and switching;
- effects of temperature extremes and thermal shock;
- effects of environment;
- shock and vibration;
- cold plate heat loads; and
- EMI generation and susceptibility.

Subtask 7.2 Software - Check out the controller software using simulated inputs and outputs.

Task 8. Ground Engineering System (GES) Testing

Objectives: Design a system concept which will meet both lunar and Mars mobile power applications. Show the concept feasibility. Verify adequate performance and life for an integrated system.

Statement of Work: This effort was divided into the following subtasks:

Task 8.1 GES Design - Identify and characterize specific power system applications. Determine power system requirements. Define optimum power system module size based on trade studies which minimize the overall life cycle cost of the power system for all applications (trade between development cost and mass).

Identify concepts which can meet early lunar base applications. Power system enhancements, if any, required for Mars applications will be identified. Perform tradeoff studies (performance, reliability, risk, safety, and life cycle cost) to select the optimum system design. Complete feasibility studies for the selected concept including off-design and transient analysis. Definitize the remaining hardware development tasks based on the system concept chosen.

Design a complete power system to at least meet the minimum life requirement (10 years for the arrays and 20,000 hrs for the RFC with a 50% duty cycle for electrolysis and fuel cell stacks). Integrate the entire RFC thermal and water control system including pumps, controls, lines, valves, heat exchangers, tanks, and radiator. Perform both design point and transient analyses to verify the design. Perform this task in parallel with the component development tasks to insure proper system characteristics

Task 8.2 GES Performance Tests - Manufacture the GES components and assemble. Pressure and leak check the assembly, fill the water storage tank, complete a electrical check-out, and check out all active devices to the extent possible.

Design the test program for the GES to verify all performance characteristics of the unit in conjunction with subjecting the GES to acceptance level tests. Conduct performance and thermal vacuum tests during a single thermal vacuum test sequence to demonstrate technical capabilities while minimizing program cost. Complete the following performance testing activities:

- checkout and refine subsystem assembly procedures;
- verify operation of ground support equipment (GSE) and interfaces with the GES;
- verify start up capability;
- verify steady state performance characteristics;
- establish effective operating range of the system;

- check out software and verify autonomy (include all operational modes and simulate failure modes);
- simulate normal switchover from one module to another;
- simulate failures to trigger module switchover; and
- establish system sensitivity to off-normal conditions such as: (1) partial loss of radiator cooling, (2) varying radiator heat removal profile, (3) partial loss of electronic cooling, and (4) temporary losses of heat sink with varying duration.

Task 8.3 GES Life Test - Refurbish the GES after completion of performance, acceptance, and margin tests. Refurbishment will include the following:

- replacing or repairing components as necessary to ensure that the GES is returned to its as-built condition;
- addition of special instrumentation required for the life test phase; and
- modification of the main radiator heat removal system or replacement of the main radiator with a dump heat exchanger.

Install instrumentation to provide a comprehensive diagnosis of the "health" of the GES and to monitor for degradation of major assemblies and individual components. Place the GES on a multiyear life test. Operate the GES at its nominal operating point, with expected GES variations in power output and environment.

Disassemble the GES for diagnosis at the end of the life test. Determine areas to be examined by an analysis of the health monitoring data and from the reliability analysis predictions.

Task 9. Qualification Program Testing

Objectives: Fabricate the qualification units (QUs) Verify adequate performance and life for the QUs.

Statement of Work: Develop a comprehensive performance and dynamic testing program which will provide a formal demonstration that the power system will perform as designed after being subjected to simulated launch conditions. Select launch environmental conditions which envelope the probable intensities developed by various launch scenarios.

Begin the qualification effort (a typical approach is shown in Figure D-9) by qualifying the assemblies. Fabricate and assemble the qualified production items into the QU. Qualify the QU by the rules for space vehicle qualification.

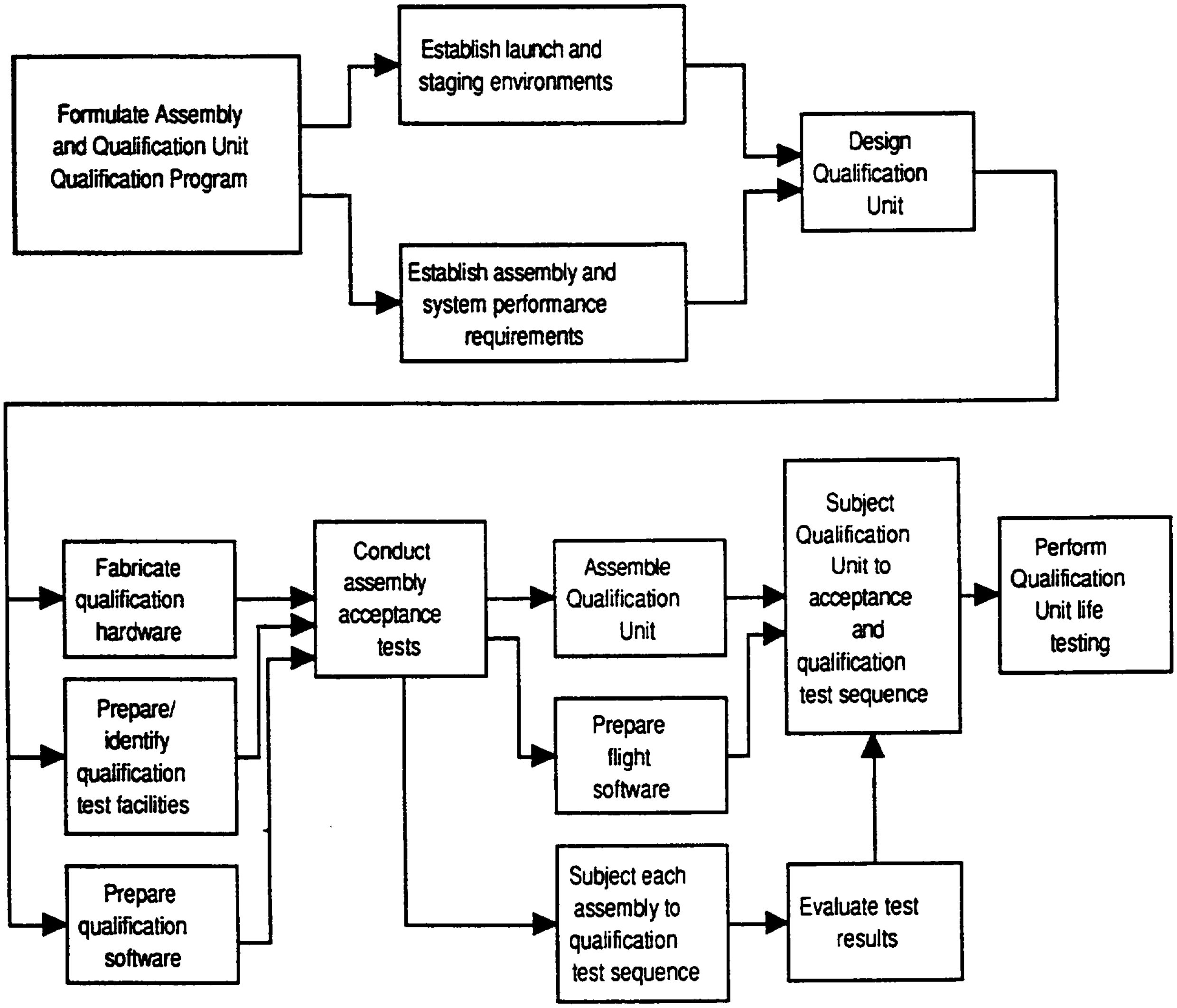


Figure D-9. PV/RFC power system qualification program.

This effort was divided into the following subtasks:

Subtask 9.1 Qualification Performance Testing - Conduct performance testing at each level to verify that each item performs as designed. Perform dynamic testing to verify the capability of the RFC system to withstand launch loads, including acoustic, pyroshock and vibrational. A possible performance and dynamic qualification test sequence for components and assemblies is shown in the matrix of Figure D-10. The corresponding qualification test sequence for the QU is shown in Figure D-11.

Component or Subassembly	Functional (1)	Leak	Pyroshock	Functional (2)	Random Vibration	Functional (1)	Leak	Acceleration	Functional (1)	Thermal Cycling	Functional (2)	Thermal Vacuum	Functional (2)	Pressure	Leak	EMC	Life	Functional (1)
Fuel Cell Stack	X	X			X	X	X			X	X	X	X	X	X	X	X	X
Electrolysis Cell Stack	X	X			X	X	X			X	X	X	X	X	X	X	X	X
Reactant Storage	X	X	X	X	X	X	X			X	X	X	X	X	X		X	X
Thermal Control	X	X			X	X	X					X	X	X	X		X	X
Radiator and Manifold	X	X			X	X	X			X	X	X	X	X	X		X	X
PMAD	X		X	X	X	X				X	X	X	X			X	X	X
Structure	X		X	X	X	X		X	X	X	X						X	X
PV Array	X		X	X	X	X				X	X	X	X			X	X	X

Figure D-10. PV/RFC power system assembly qualification test matrix.

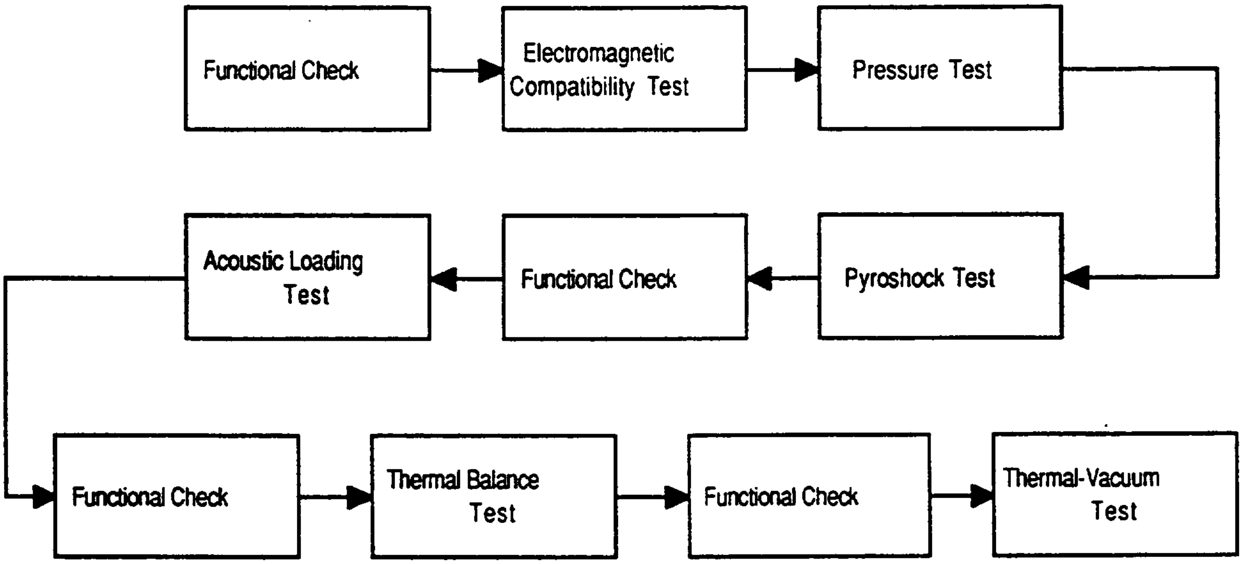


Figure D-11. PV/RFC power system QU test sequence.

Subtask 9.2 Qualification Life Testing (Optional) - After completion of the qualification performance testing, partially disassemble the QU and examine the unit. Refurbish the QU as required and modify for endurance testing in air, as described for the GES. Life test the QU for 20,000 hours or more.

Task 10. Flight Unit (FU) System Program

Objectives: Fabricate the flight systems and acceptance test the FUs to demonstrate required performance. Deliver the flight systems and provide integration support for the FU with the payload and the launch vehicle.

Statement of Work: Fabricate, acceptance test, and assemble parts to produce the flight systems. Subject the systems to acceptance testing before shipment to the launch site. Use the same test facilities (vacuum chamber, vibration, acoustic) that were used for the qualification program for flight system acceptance testing. Perform safety studies and complete safety reports necessary to obtain launch approval. Finally, provide launch support activities to insure that the FU meets its design and performance goals associated with integrated payload and launch systems.

The work is divided into the following subtasks:

Subtask 9.1 Flight Component Fabrication. - Design, fabricate, inspect, and assemble the components and subassemblies required for the FUs, including all spare parts as required to support the flight system activities.

Subtask 9.2 FU Assembly, Test, and Payload Integration - Assemble and inspect the FUs. Acceptance test the FUs and ship to the launch site. Provide technical support for FU integration with the payload, launch vehicle, and launch support facilities.

Subtask 9.3 FU Launch Support - Provide FU launch support activities to insure that the FU meets its design and performance goals associated with the integrated payload and launch systems.

DEVELOPMENT SCHEDULE

The program starts with a conceptual design task followed by preliminary design and concurrent component development of the PV array, RFC (fuel cell stack, electrolysis cell stack, active controls, gas dryers, and tanks), radiator, and PMAD. The detailed design is subsequently completed by the middle of the fourth year.

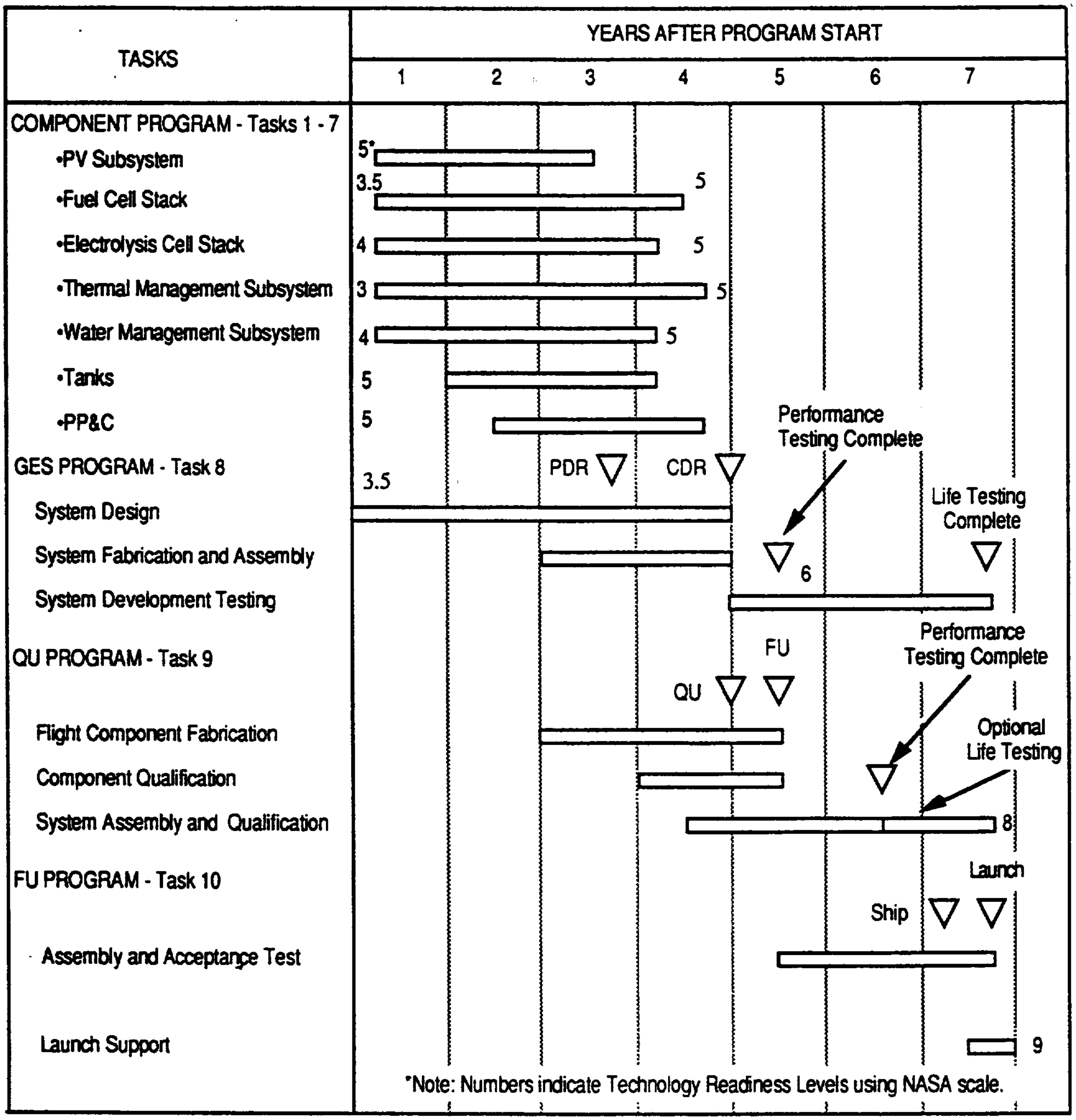
Fabrication of components for ground testing for the Ground Engineering System (GES) starts with procurement of long lead materials and equipment in the first part of the third year. This leads to assembly of the system by the end of the fourth year.

The GES will be similar to a flight system but will have features such as additional instrumentation and readily removable components to expedite gathering of engineering data and to permit modification of components. It will be tested in a vacuum chamber under normal and off-normal design conditions. After this phase of testing, it will be partially disassembled, examined, refurbished as necessary, and put on life test for 20,000 hours (50% duty cycle for the fuel cell and electrolysis cell stacks).

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components, and a complete system. The design effort will be minimal since it would involve only minor modifications to the GES design.

The flight phase of the program includes fabrication, assembly, and acceptance testing of the flight units and the associated safety analysis to obtain launch approval.

The estimated development and production schedule for the PV/RFC power system is shown in Figure D-12. The development time to achieve a flight proven system is estimated to be 7 years. If the program is initiated in FY'93, then power systems could be available for early lunar missions.



PHOTOVOLTAIC/REGENERATIVE FUEL CELL POWER SYSTEM TECHNOLOGY ROADMAP

Figure D-12. PV/RFC power system development schedule.

REFERENCES

D-1. McElroy, J.F.: SPE Water Electrolyzers in Support of Mission From Planet Earth. Space Electrochemical Research and Technology Conference, NASA conference publication 3125, April 1991.

D-2. Kim, N.P.; and Devaney, N.E.: Low Cost $CuInSe_2$ Based Solar Cells for Space Application. 26th IECEC Proceedings, Volume 2, August 1991, pp. 314-320.

D-3. Kim, N.P.; et. al: High Specific Power (AlGaAs)GaAs/$CuInSe_2$ Tandem Junction Solar Cells for Space Application. 24th IECEC Proceedings, Volume 2, August 1989, pp. 779-783.

D-4. Stanbery, B.J.; et. al: Thin Film GaAs/$CuInSe_2$ Solar Cells for Space Power Application. 19th IECEC Proceedings, August 1987, pp. 280-284.

D-5. Landis, G.A.: Review of Thin Film Solar Cell Technology and Applications for Ultra-Light Spacecraft Solar Arrays. Proceedings of the 10th Space Photovoltaic Research and Technology (SPRAT) Conference, 1989, pp. 180-197.

D-6. Cooley, Lt. W.T.: Air Force Photovoltaic Array Alternatives. 26th IECEC Proceedings, Volume 2, August 1991, pp. 275-280.

D-7. Stella, P.M.; and Kurland, R.M.: Development of the Advanced Photovoltaic Solar Array. 26th IECEC Proceedings, Volume 2, August 1991, pp. 269-274.

D-8. Technology Assessment and Tradeoff Study of Fuel Cell and Electrolyzer Technology for Project Pathfinder Energy Storage System. Preliminary report to NASA Lewis Research Center, Los Alamos National Laboratory, February 1991.

D-9. McElroy, J.F.: SPE Regenerative Hydrogen/Oxygen Fuel Cells For Extraterrestrial Surface Applications. 24th IECEC Proceedings, August 1989, pp. 1631-1636.

D-10. McElroy, J.F.; et. al: SPE® Regenerative Hydrogen/Oxygen Fuel Cells For a Lunar Base Power System. 25th IECEC Proceedings, August 1990, Volume 3, pp. 143-148.

D-11. Personal Communications with Jim McElroy at Hamilton Standard. Rockwell International (Rocketdyne Division), Frank Littman, 8/13/91, 9/12/91, 9/23/91, 1/31/92, and 3/25/92.

D-12. Pesaran, A.A.: Silica Gel Sorption Properties under Adiabatic Conditions. Solar Energy Research Institute report no. SERI/TR-252-2704, Febrary 1986.

D-13. Penny, T.R.; et. al: Capacity Degradation of Field-Tested Silica Gel Samples. Solar Energy Research Institute report no. SERI/TP-252-2741, June 1985.

D-14. Kim, N.P.: Personal Communications. Boeing Defense and Space Group, Seattle, WA, 10 September 1991 and 3 October 1991.

D-15. Srinivasan, S.; et. al: Recent Advances in Solid Polymer Electroylyte Fuel Cell Technology with Low Platinum Loading Electrodes. 24th IECEC Proceedings, Vol. 3, August 1989, pp. 1623-1629.

REFERENCES

D-16. Lawrance, R.J.; and Margiott, V.A.: Proton Exchange Membrance Fuel Cell Development. 24th IECEC Proceedings, Vol. 3, August 1989, pp. 1587-1591.

D-17. Meyer, A.P.: Status of Solid Polymer Fuel Cell Power Plant Development at International Fuel Cells Corporation. 24th IECEC Proceedings, August 1989, pp. 1619-1622.

D-18. Personal Communications with Jay Stedman at International Fuel Cells. Rockwell International (Rocketdyne Division), Frank Littman, September 13, 1991.

D-19. Juhasz, A.J.: An Overview of the LeRC CSTI Thermal Management Program. Paper no. AIAA-91-3528, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 4-6, 1991.

D-20. Bock, E.H.; and Fisher, J.G.: In-Space Propellant Processing Using Water Delivered as Shuttle Contingency Payload. AIAA Paper 78-941, July 1978.

D-21. Personal Communications with Ian Devey of Structural Composites Industries. Rockwell International (Rocketdyne Division), Frank Littman, October 9, 1991.

APPENDIX E

PV/NAS BATTERY TECHNOLOGY ROADMAP

APPENDIX E - PV/NAS BATTERY TECHNOLOGY ROADMAP

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APPENDIX E

INTRODUCTION

This is a family of fixed power systems for Mars applications. Mission applications require power levels from 0.9 kWe to 75 kWe. Power system applications include communications (0.9 kWe), emergency power (5 kWe), the excursion vehicle servicer (10 kWe), and exploration site main power (25 kWe and 75 kWe modules for habitat and associated external equipment). Solar powered systems such as these will require energy storage for operation at night.

CONCEPT DESCRIPTION

A typical power system schematic is shown in Figure E-1. The overall power system may be divided into the following subsystems for development purposes:

- Photovoltaic (PV) Array;
- Batteries;
- Thermal Management; and
- Power Processing and Control (PP&C).

The solar array converts sun light directly into DC electricity. The energy from the array flows to the batteries, for later use, and to the user. When the system enters a period of darkness, the energy to the user is supplied by the batteries. The batteries are recharged on the next sun cycle by the solar array. The flow of energy from the array and to and from the batteries is controlled by the PP&C subsystem.

Since the batteries must operate at high temperature, thermal management is required to maintain the proper cell temperature and reject waste heat. In addition, the batteries need to be heated prior to startup. The electronic components in the PP&C also require cooling to remove waste heat.

This power system concept utilizes high efficiency photovoltaic cells to minimize array area. A large array area is required for Mars applications due to the low insolation rate (due to dust from local and global dust storms). Since the PV subsystem is the largest component of system mass for Mars systems, it is important to minimize array area and specific mass (kg/sq m).

The PV subsystem includes the array panels, support structure, and wiring harness. For development purposes, specific subtasks will include cell development, array development, deployment mechanism development, and integration/system testing.

The battery subsystem includes the cells and related structure to tie the cells together.

The thermal management subsystem includes battery insulation, battery isolation plates, battery radiator/interface heat exchanger, PP&C cold plates, and the PP&C radiator.

Power conditioning is included to process power for charging the batteries (down regulator) and processing output power (boost regulator). A shunt regular dissipates excess power from the array.

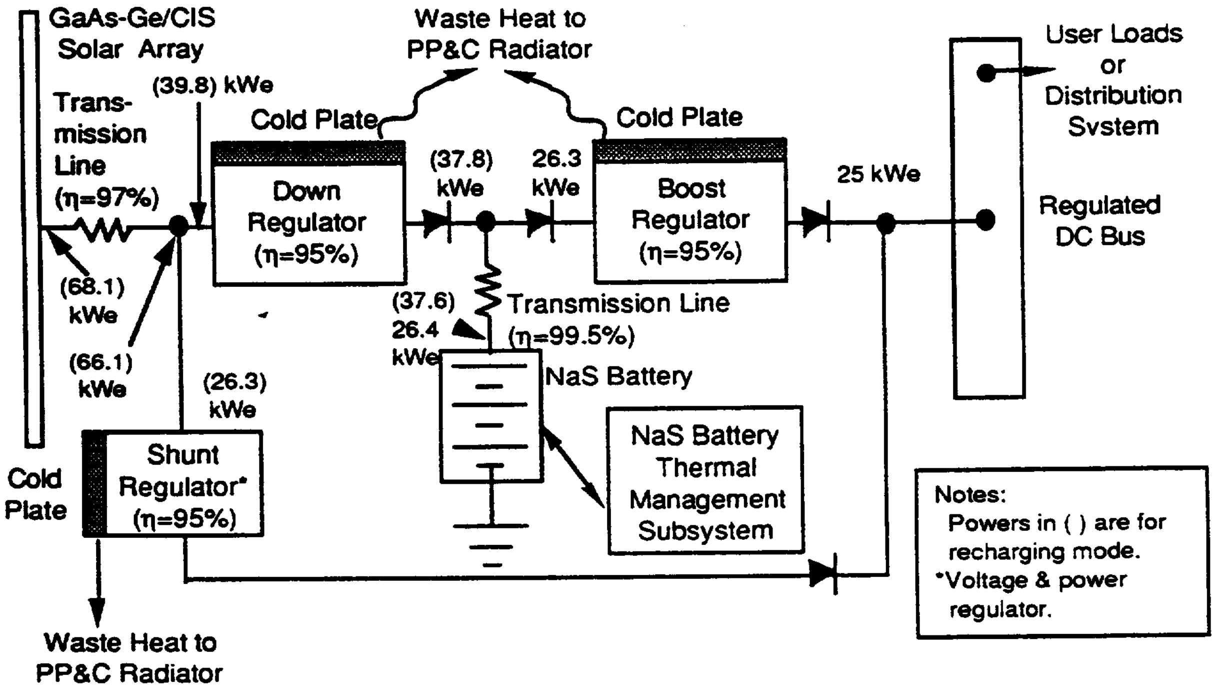


Figure E-1. PV/NaS battery system schematic.

The PV panels are assembled into modules on site. The modules are then connected to the batteries. The battery subsystem housing is a separate unit which includes the PP&C subsystem. The battery/PP&C unit is connected to the arrays after array deployment.

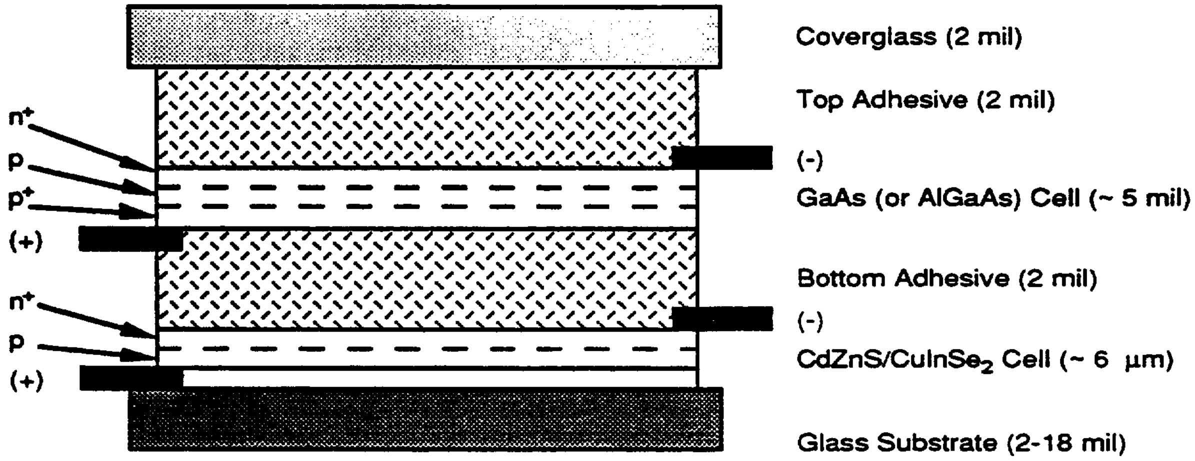
The system evaluated in this article uses advanced multijunction tandem solar cells, sodium sulfur (NaS) batteries, DC-DC converters/regulators, and a heat pipe radiator. Each subsystem will be described in more detail in the following sections.

Solar Subsystem Description

A solar array is made up of panels of photovoltaic cells mounted on a substrate structure. A photovoltaic cell is a semiconductor device that turns light in to direct current electricity (DC).

The baseline PV cell is one being developed by Boeing Defense and Space Group in Seattle (Refs. E-1 through E-3). The multi-bandgap or tandem cell consists of a double-heterostructure GaAs or AlGaAs thin film top cell and a polycrystalline $CdZnS/CuInSe_2$ heterojunction thin film lower cell as seen in Figure E-2 (Ref. E-1). The leading technology for thin film photovoltaic cells is $CuInSe_2$ (CIS) (Ref. E-4). In the cascade structure shown, short wavelength (high energy) photons are absorbed in a high bandgap material on top of the solar cell. The high bandgap material is transparent to longer wavelength (low energy) photons which pass through and are absorbed by a second layer consisting of a photovoltaic material with low bandgap.

Cascades can be configured as a monolithic cell in which the top cell is integrally deposited on the bottom cell (or vice versa), or mechanically stacked, in which the two sets of cells are formed separately. A mechanically stacked tandem configuration was chosen by Boeing in order to provide wiring flexibility and to minimize fabrication constraints. Since both cells are deposited as thin films, a very high specific power is possible.



Vertical dimensions not to scale

Figure E-2. Tandem PV cell schematic.

The current Boeing design is based on a 2 cm x 4 cm cell area (Ref. E-1). The cell has an improved two-terminal configuration with voltage-matched monolithic subcell units. Voltage-matching is achieved by stacking one GaAs CLEFT (Cleavage of Lateral Epitaxial Film for Transfer) cell on top of four $CuInSe_2$ subcells monolithically interconnected in series to form a single cell unit.

An improved cell design could be achieved by using AlGaAs rather than GaAs as the top cell. The bandgap of $CuInSe_2$ is better matched to the bandgap of AlGaAs than GaAs. A 26% Beginning-of-Life (BOL) cell efficiency at AM0 is projected without structural change for this advanced cell.

The upper thin film cell is fabricated by the CLEFT technique using MOCVD for cell structure growth. The CIS cell fabrication includes sequential depositions of the Mo back electrode, CIS absorber layer, and CdZnS window layer. This is followed by photolithographic patterning and etching to form a solar cell device. Deposition of grid metal and addition of an anti-reflection coating complete the cell fabrication process.

The individual PV panels are either small enough to be transported to Mars as designed or are hinged for easy deployment. The panels lie horizontal and do not track the sun. Trade studies done by Rocketdyne showed that there was no advantage to tracking arrays since much of the cell input is from diffuse light.

Planar arrays can be subdivided into two broad categories: rigid and flexible (Ref. E-5). Rigid arrays are PV arrays that are mechanically stiffened with a honeycomb structure, usually made of aluminum, sandwiched between two facesheets that provide back side shielding for the PV cells and provide enhanced mechanical support to the structure. Rigid arrays are used in high risk, unknown environments, where weight and cost are a secondary concern. Rigid arrays provide the most mechanical support and the most survivability. However, the rigid array is not suitable for planetary surface applications due to the mass penalty.

Flexible arrays are typically a Kapton substrate/superstrate which sandwich the PV cells and electrical circuitry, and are very attractive due to the significant mass savings over rigid panel arrays. Within the flexible arrays, there are three types: the roll-out type, the fold-out type, and the inflatable type. A flexible array is recommended for SEI surface power applications.

The Advanced Photovoltaic Solar Array (APSA) is an example of future flexible fold-out arrays (Ref. E-6). There are three elements to the APSA array: the flexible plastic and solar cell blanket (50% of the mass), the deployment mast and mast housing (34% of the mass), and the blanket stowage compartment (16% of the mass) (Ref. E-5). The APSA wing consists of a flat fold, multiple panel, flexible blanket on which solar cell modules are installed and connected to printed circuit electrical harnesses that run along the outside longitudinal edges of the blanket assembly. Any type of solar cell can be utilized with the APSA approach. For launch, the accordion-folded blanket is stowed in a graphite/epoxy blanket during launch. The blanket is deployed (unfolded) by extending a motor-actuated, fiberglass, continuous longeron lattice mast that uncoils from an aluminum cylindrical canister structure. APSA is designed for zero g operation. Additional support structure would be required for planetary surface applications.

Inflatable arrays offer promise for missions which are driven by mass and/or the radiation environment. Inflatable arrays require the use of thin film PV solar cells. For a 100 W array (EOL) the mass breakdown is as follows: inflatable torus contributes 0.793 kg, the thin film solar array blanket 0.793 kg, and the support equipment 1.36 kg, resulting in a small satellite array specific power of about 34 W/kg (Ref. E-5).

An array deployment mechanism is required for automatic or robotic deployment. Array deployment alternatives include spring stored "one time deployment" and motor-driven deployment systems. The deployment mechanism is critical from a reliability standpoint and may also contribute significantly to the system mass. The deployment mechanism for Mars systems must also maintain array rigidity during wind storms and must be designed to prevent resonating of the structural due to the natural frequency.

Some form of structure may be used to keep the arrays off the surface or to prevent the arrays from being disturbed by Martian winds. This system would have increased mass over one which is simply rolled out onto the planetary surface.

Battery Subsystem Description

Sodium sulfur batteries are high temperature (598 ^{O}K) secondary batteries which have been under development for a number of years. Due to the high theoretical specific energy, these batteries are being considered for electric vehicles, utility load leveling, satellite systems (Ref. E-7), and planetary surface power systems (Refs. E-8 and E-9).

A fully charged sodium sulfur cell consists of elemental sulfur and sodium separated by a beta alumina electrolyte as seen in Figure E-3. Beta alumina is a solid, ceramic electrolyte separator which is conductive only to sodium ions. The molten sodium serves as the negative electrode. During discharge, each sodium atom, entering into the discharge reaction, provides an electron to the external circuit and migrates through the electrolyte. The molten sulfur serves as the positive electrode. The sodium ions receive an electron, from the external circuit, on the sulfur side of the cell and combine with the sulfur to form sodium pentasulfide, Na_2S_5.

After all the free sulfur is combined, a second conversion takes place in which the Na_2S_5 is converted to sodium trisulfide, Na_2S_3. The open circuit voltage of the cell during the first 59% of discharge, which corresponds to the sodium pentasulfide reaction, is nominally 2.08 volts per cell. During the sodium trisulfide part of the discharge, the corresponding voltage is 1.75 volts per cell. Thus, the sodium sulfur cell discharge voltage has a two level characteristic with the step change at about 59 percent. Experimental investigation has indicated that for best cycle life, the depth of discharge should be limited to 59 percent.

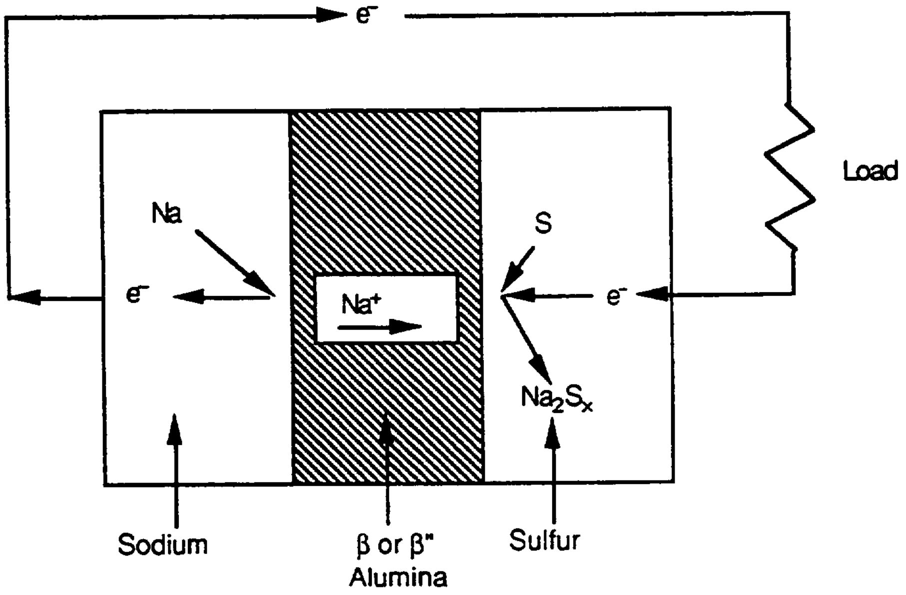


Figure E-3. NaS battery cell operation during discharge.

A tubular type NaS cell is illustrated in Figure E-4. This concept is most suitable for base load power systems (15 minutes to 12 hours of operation). These type of NaS cells have been undergoing evaluation testing by the Air Force for many years and have gone through several design iterations. As shown in the figure, the liquid sodium is contained in a closed steel tube with a metering orifice connecting to an annular chamber also containing sodium. The closed beta alumina electrolyte encloses the sodium and separates it from the liquid sulfur cathode. Liquid sulfur cathode material is held in a carbon felt matrix to provide electrical conductivity during charge and discharge. The steel tube is coated with a metal chosen for its resistance to the corrosive sodium sulfides.

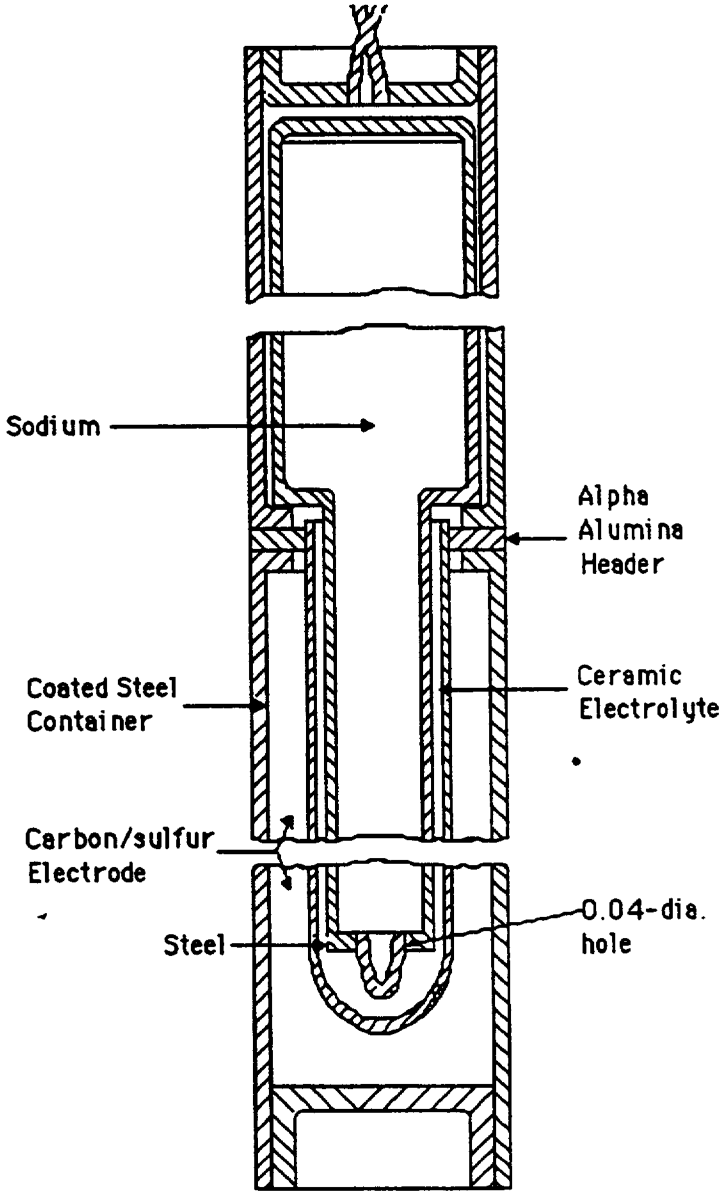


Figure E-4. Typical tubular NaS battery cell design.

Thermal Management Subsystem Description

The thermal management subsystem provides temperature control, heat transport, and heat rejection functions. Solid conduction paths and heat pipes provide the heat transport function. A radiator is required to remove waste heat from the battery and from the PP&C components.

An active thermal control system is required to maintain the battery at the proper operating temperature. Three different techniques are possible for controlling battery temperature (Ref. E-10): (1)louvers; (2)pumped loop radiator; and (3)heat pipe radiator.

The use of louvers allows the battery to radiate directly to space as needed to reduce its temperature. Louvers offer the greatest weight advantage (Ref. E-10). However, louvers require the greatest mechanical complexity and potentially the lowest reliability. Louvers also allow for a nonhomongeneous thermal distribution in the battery which may be detrimental.

Pumped loop (or tube-sheet) radiators offer the next best approach from a mass standpoint if no redundancy is built into them. Pumped loop radiators have been used successfully for the space shuttle and will be used for Space Station Freedom (SSF). This type of radiator is best applied to missions with limited duration or to systems which are serviceable. Potential working fluids include water, potassium, NaK, and Dowtherm A. NaK is the preferred working fluid for this application due to its low freezing point, low operating pressure, and stability in a thermal environment. The use of NaK as a reactor primary coolant indicate that it can be used with good reliability for prolonged periods of time. The development of highly reliable, long life, low flow rate, low mass pumping systems to accomodate the use of fluid loops in space would require considerable effort (Ref. E-10). Furthermore, the pump represents the source of a single point failure mode, thus requiring duplication within the system and thereby nullifying the slight mass advantage over heat pipes.

Thermal homogeneity could be accomplished with heat pipe systems for transport and heat rejection due to their isothermal heat transport characteristics. Heat pipes also offer the advantage of improved reliability and a graceful failure mode. A recent Rocketdyne study has shown that advanced carbon-carbon (C-C) heat pipe radiators can be designed which are competitive in mass to pumped loop radiators (estimated specific mass of 3 kg/m^2 for heat pipes with water or ammonia). Thus, a heat pipe radiator was tentatively selected as the baseline design.

The baseline heat rejection assemblies utilize lightweight, passive, reliable, heat pipe radiators that are sufficiently versatile to allow integration into a variety of configurations. The individual heat pipes operate independent from one another and thus the failure of a heat pipe will not result in failure of the complete radiator. The rectangular radiator heat pipe panel is attached to the battery assembly. Variable conductance heat pipes transfer heat to the radiator. The heat pipe working fluid evaporates and travels to the top end of the heat pipes. The evaporated fluid is then condensed in the cooler section of the heat pipe. A small wick or grooves allow the liquid to return to the evaporator (unless the heat pipe is vertical and gravity can be used for fluid return). A wick or groove is not absolutely necessary for vertically oriented radiators (due to gravity return) but is recommended to insure good control of the fluid transport. The heat pipes may be either carbon-carbon tubes with metal liners (titanium or nickel based alloy for biphenyl) or metal heat pipes. Carbon-carbon heat pipes are a factor of 2-3 times lower in mass than conventional all-metal heat pipes.

The problem with using a heat pipe for battery cooling is that no common heat pipe working fluid is suitable for operation at about 600 °K (Ref. E-11). Only three of the common heat pipe working fluids including water, cesium, and mercury show any promise for operation at this temperature. However, these fluids have several disadvantages. Both mercury and cesium are highly toxic and would have handling problems. Mercury has a poor contact angle and does not wet the wick surface, which causes heat pipe priming problems. Mercury also has a high density which results in a heavy heat pipe. Cesium has a very low vapor pressure which favors leakage into the pipe. Leakage is a severe problem since atmospheric water will react with the cesium to form hydrogen. Water develops a high pressure at the required operating pressure which requires a heavier heat pipe and may result in potential safety issues. A recent study (Ref. E-11) found that biphenyl is a suitable heat pipe working fluid for this application. Biphenyl is compatible with likely materials of construction and its liquid transport factor is suitable for use in a heat pipe. Also, its vapor pressure is in the desired range at the operating temperature of the battery. Thermal studies have shown that biphenyl should be stable for long-term operation at high temperature and non-condensable gas generation will not be a problem. Thus, biphenyl was selected as the working fluid for the battery heat pipes.

PP&C Subsystem Description

The PP&C subsystem is a collective term for the power system control electronics. This system includes items such as shunt regulators, voltage regulators, and battery charge controllers. Their function is to control the flow of energy though the power system to the payload.

Shunt Regulator Description. At the beginning of life of a solar power system the solar array produces excess power. To prevent the rest of the system from being overpowered, a shunt regulator is installed. This regulator shunts off excess current from the array. This energy is dissipated in the form of heat.

Boost Regulator Description. The boost regulator increases the voltage from the batteries up to the nominal bus voltage. There are two basic types of boost regulators. The basic boost regulator handles all of the processed power and the efficiency of the circuit applies to all the power. This regulator places a "buck" inductor and switch in alternate positions such that the opening switch allows the inductor to force the current it had before the switch opened into the load impedence. The output voltage depends on the load impedence and current. The basic booster has a relatively low efficiency and may have loop stability problems.

The add-on booster only processes the boost voltage (usually only 10 volts). The boost power (voltage added times output current) is parallel to the load at the input. Most of the power is not processed and thus the overall efficiency is much higher than for the basic boost regulator. A failure of the add-on booster results in a lower DC bus voltage, but the system is still functional with degraded performance.

The boost regulator requires monitoring equipment to determine operating parameters and heat sinking to remove waste heat. EMI filters are required to prevent interference from the switching.

Down Regulator Description. The PV array is designed to always provide a higher voltage than the nominal battery voltage. The down or "buck" regulator provides a constant output DC voltage for charging the battery subsystem from the array. The regulator output voltage is always lower than the input voltage. The voltage conversion is accomplished by switching the input on and off. The filter removes the AC component produced by switching, but allows the DC voltage throught.

The down regulator is reliable and very efficient (85-90% for nearterm hardware). The efficiency of the regulator depends on the internal operating frequency and the input-output voltage difference. As the frequency goes up the size goes down and so does the efficiency. This regulator does not isolate the input from the output loads. If the regulator is not properly used, then it can cause radio interference since it uses switching to interrupt the current. Proper EMI filters are required to eliminate this interference. Monitoring equipment is required to determine the proper operating conditions. Heat sinking is required to remove waste heat.

TECHNOLOGY ISSUES

Key issues for the PV subsystem are summarized in Table E-1. The key issues for development of a NaS battery subsystem are summarized in Table E-2.

The key issue with sodium sulfur batteries is achieving cycle life to permit operation as a satellite battery. While the specific energy, of the sodium sulfur system, is much higher than the current baseline nickel hydrogen batteries, the cycle life is much lower. Primary failure mechanisms are related to corrosion of the seals and cell components by the chemically active sulfides and the fragility of the beta alumina electrolyte. Design features have been developed to mitigate the effect of these failure mechanisms so that a failure is not catastrophic. Cracking of the electrolyte or loss of cell case integrity does however, result in loss of the cell however. Continuing cell development is aimed at resolving these issues.

TABLE E-1. PV SUBSYSTEM ISSUES, IMPACTS, AND DEVELOPMENT AREAS

Issues	Impacts	Potential Development Areas
Large array area due to low Martian insolation	•Increased life cycle cost (LCC) •Increased deployment time •Increased number of cells with reduced system efficiency and reliability	•Higher efficiency top cell (AlGaAS) •Robotic or automatic deployment system •Thin film arrays •Roll-out arrays
Small cell size	•Increased number of cells with reduced system efficiency and reliability	•Higher efficiency top cell (AlGaAS) •Larger size cells
Cell efficiency	•Increased array size and LCC	•Higher efficiency top cell (AlGaAS)
Cell cost	•Increased LCC	•Low cost production techniques
Deployment system and support structure weight	•Increased LCC	•Flexible roll-out array •Robotic deployment
Operating temperature fluctuation and extremes	•Reduced cell life due to thermal stress/increased LCC	•Design and test for appropriate environment •Test for thermal extremes
Martian wind	•Increased structure mass and LCC	•Lightweight structure and tie-downs
Dust accumulation	•Increased array area and LCC •Maintainence cost	•Robotic dust removal system

TABLE E-2. NaS BATTERY SUBSYSTEM ISSUES, IMPACTS, AND DEVELOPMENT AREAS

Issues	Impacts	Potential Development Areas
Cycle life	•Increased life cycle cost	•Physical and chemical stability of alpha alumina seal •Physical and chemical stability of electrolyte •Sealing technology for tubesheet to cell case
High operating temperature	•Heavy heat pipes, poor performance, toxic working fluids (cesium or mercury) or high pressure (water)	•Low mass C/C heat pipe radiator •Heat pipe working fluids (biphenyl)
Safety (explosion, fire, toxic fumes, formation of sulfuric acid)	•Increased manufacturing cost	•Battery casing design

TECHNOLOGY ASSESSMENT

The technology bases were assessed for the following major PV/battery subsystems:

- PV array;
- Batteries;
- Thermal management; and
- PP&C.

The technology readiness of each assembly was evaluated using the NASA technology readiness levels. These evaluations are summarized in Table E-3 which shows that the power system has technology readiness levels ranging from 3 to 5, depending on the particular subsystem. The technology base for subsystems is discussed in the following sections.

TABLE E-3. GaAs-CIS PV/NaS BATTERY TECHNOLOGY ASSESSMENT

Subsystem	Technology Readiness Level	Comments
GaAs/CIS PV cells	5	Pilot development phase for cells; APSA program
NaS batteries	4	Breadboard cells tested by Air Force and prototype battery is under development
Battery therma management	3	Radiator component development currently underway; biphenyl investigated as heat pipe fluid
PP&C	5	Similar components under development for Space Station Freedom (SSF)

System State-of-the-Art

The GaAs/CIS PV array and NaS battery system has a NASA Level of about 4. The concept of this type of a system has been formulated but the details of the system have not been worked out. Specifically, this includes the integration of the NaS battery storage into the system. Though NaS batteries are a basic secondary type battery they require special thermal control to work. The batteries require an operating temperature of 598 ^{O}K. The PP&C subsystem will have to work in conjunction with thermal control system to maintain this temperature. In addition, to overcome the fragility problems of NaS batteries, it has been proposed to launch the batteries in a frozen state (<573 ^{O}K). Once on the planetary surface, the batteries will have to be brought up to temperature through some type of external source. For a flight test experiment, the power source will be the Shuttle payload power bus. Another issue is that prior to the time the NaS batteries are brought on line the spacecraft will need power for basic operations such as ground telemetry and array deployment. Some other source of energy, such as a primary battery, will be necessary and will increase the complexity of the PP&C subsystem. In this system the operational characteristics of the NaS battery are the key issues. Due to the low readiness level of the NaS battery and the thermal management subsystem, the overall system readiness level is relatively low compared to current PV/battery systems.

PV Array State-of-the-Art

The current cell technology level is between 5 and 6 on the NASA scale. The Boeing cell development is in the pilot line phase now (Ref. E-12). Production volume is expected by 1996. The GaAs top cell is currently made by Kopin company while the bottom CIS cell is made by Boeing. An advanced cell design (AlGaAs/CIS) could be available well before 2008. Cell testing has progressed through the preliminary qualification stage. Testing has included thermal cycling, UV illumination, off-angle, vacuum stability, and humidity tests. Small coupon panels have been tested for mechanical shock, pyro shock, acoustics, vibration, and

C-3.

thermal cycling. Current cell size is 2 x 4 cm (1991). Boeing is also working on a thin plastic substrate to replace the current glass substrate.

Array development has been done for both rigid and flexible arrays. Various array technologies are summarized in Table E-4 (Ref. E-5). For large systems, the support structure development maturity is at a NASA Level 7. The only large system now being planned for launch is the Space Station Freedom array. The array support structure was flight tested under the SAFE program. Under the current schedules these designs will reach Level 9 by the year 2000.

TABLE E-4. DESCRIPTION OF VARIOUS ARRAY TECHNOLOGIES AND APPROXIMATE PERFORMANCE FIGURES OF MERIT (IN EARTH ORBIT)

Name	Description	PV CELLS (BOL)	Base Power (EOL kWe)	Life (years)	Area Density (W/m^2)	Specific Power (W/kg)
Rigid Honeycomb (Typical)	Aluminum honeycomb	Si: BSF 8 mil	3.4	7	128.3	32
HUBBLE	Roll-out flexible blanket	Si: 2x4 cm^2 BSFR	4.4	2	117	19
Space Station	Flexible Kapton fold-out blanket	Si: wrap through contacts 8x8 cm^2	75	4	90	66
APSA	Flexible Kapton fold-out blanket	Si: 2x4 cm^2 BSF 13.8%	3.7	10	95	93
SUPER (Planar)	Flexible fold-out with Beryllium	GaAs/Ge 6x6 cm^2,18%	5.3	5	121.7	26.5
AHA	Ti honeycomb with 1.4 x concentrating shutters	GaAs/Ge -2 junction,24%	5	10	95.5	28.8
ESSA	Al honeycomb with protective shields	GaAs/Ge single junction,18%	5	10	137	23.2

The roll-out type array approach was developed in the early 70's at Wright-Patterson Air Force Base during the Flexible Roll-Up Solar Array (FRUSA) program. The same basic tgechnology is used today for power generation on the Hubble Telescope. Some of the arrays in the flexible fold-out category are the SAFE array, Mil-Star, Space Station and the APSA array developed by TRW for JPL. The APSA array is designed for GEO orbit. The APSA program has been ongoing for several years and has significantly reduced the mass of arrays using a flexible array design. APSA has a goal of demonstrating a producible array system having a specific power greater than 130 W/kg (BOL) at 10 kW (BOL). A research and development array that should be seen as both "fold" and "roll" out is the inflatable array. The inflatable array approach is funded by DARPA and is being developed and build by L'Garde Inc. The DARPA program is called Inflatable Torus Solar Array Technology (ITSAT) and is scheduled to conduct a space flight experiment in the 4th quarter of 1993. The inflatable array is considered a high risk - high payoff approach.

The current application cell design could utilize either an inflatable array, an APSA type array (with additional support structure), or roll-out array.

Battery Subsystem State-of-the-Art

Development of sodium sulfur batteries was initiated in the 1960s for electric vehicle applications (Ref. E-13). Since then, development has progressed to the point that prototypes and breadboards are now being fabricated or are in test. Advancement of the technology has been limited by the low level of funding and the concern over the safety and operating temperature of the battery. Several manufacturers are engaged in ceramics development on the electrolytes and, in the United States, Hughes and Eagle-Picher are actively engaged in the development of space type batteries (Ref. E-14).

Eagle-Picher, in conjunction with Hughes Aircraft, had an Air Force contract (administered by WPAFB) which ran from 9/86 to 1/91 to develop a NaS cell for Low Earth Orbit (LEO). The major goals of this program were the following:

- specific energy of 397 kJ/kg (50 W-hr/lb);
- 30,000 charge-discharge cycles;
- 2C discharge rate;
- 5 year calendar life; and

- withstand LEO environment.

A Mars surface environment would be much more benign for a battery subsystem than a LEO application due to a reduced charging rate, a reduced number of cycles (only 1825 for 5 years), and a gravity environment (poor distribution of reactants in microgravity environment can cause current blockage due to formation of a non-conductive layer). In addition, the battery can be qualified using only ground testing.

For space applications, NASA Level 4 would apply. That is, breadboard cells have been in test for some time by the Air Force and a prototype of a satellite battery is in development under Air Force contract and is to be demonstrated in 1993 or 1994 (Ref. E-7). Further development is required to accelerate the fabrication and testing of cells and batteries in order to identify currently unknown problem areas. Specifically, testing will include space environmental testing, materials compatibility testing, and cycle testing. Under current funding levels, flight test of the battery is projected for the 1996 to 1997 timeframe. Availability for space applications is projected for the year 2000.

Thermal Management Subsystem State-of-the-Art

NASA LeRC is currently carrying out an integrated multi-element project for the development of space heat rejection subsystems with special emphasis on low mass radiators in support of SEI power system technology (Ref. E-15). This effort involves both in-house and contracted efforts. Contracted efforts involving Rockwell International (RI) and Space Power Incorporated (SPI) are aimed at the development of advanced radiator concepts (ARC). NASA LeRC is also involved in a joint program with DOE to demonstrate a flexible fabric heat pipe radiator concept being developed by Pacific Northwest Laboratories (PNL). In-house work at NASA LeRC is designed to guide and support the overall program by system integration studies, heat pipe testing and analytical code development, radiator surface morphology alteration for emissivity enhancement, and composite materials research focused on the development of low mass, high conductivity fins. This program is concentrating on technologies capable of development before the end of the decade for both surface power and nuclear electric propulsion (NEP) applications.

Specific objectives of the ARC contracts are to achieve specific mass values <5 kg/m^2 with radiator surface emissivities of 0.85 or higher at typical radiator operating temperatures, and reliability values of at least 0.99 for the heat rejection subsystem over a ten year life. These figures represent a factor of two improvement over the currently considered heat rejection subsystem for SP-100, and even greater improvement factors for state-of-the-art heat rejection systems used in current spacecraft applications.

Presently, Rocketdyne division of Rockwell International is involved in the development of a carbon-carbon (C-C) or graphite heat pipe radiator panel. Rocketdyne has a NASA contract entitled "SP-100 Advanced Radiator Concepts" which started in 1987 and is part of the CSTI (Civilian Space Technology Initiative) program. This program will be completed in January of 1993. This program involves development of a C-C heat pipe suitable for use in an SP-100 radiator. However, this technology will be suitable for other radiator applications as well including battery power systems (using other working fluids). The hope is that this program will develop generic technology which can be used for all future space radiators because of its reduced mass compared to metal heat pipe radiators. Carbon-carbon is a new structural material which requires different fabrication techniques than for metals or composites. A key objective of this program is to demonstrate the ability to fabricate C-C heat pipes with a thin metal liner, wicks, caps, and fill tube. The brazing technique required to attach liners is the same as would be required to attach manifolds to the heat pipes. Heat pipes filled with potassium will be designed, fabricated, and tested during this program. Thermal cycle testing will be done in a vacuum to demonstrate the integrity of the heat pipes.

Beginning in 1989, small samples of heat pipe panels were fabricated by Rocketdyne as part of the CSTI program. The next phase of the program will involve fabrication and testing of complete heat pipes several feet long.

The working fluids and temperatures for the SP-100 system would be different than for lower temperature battery systems. Thus, Rocketdyne has an on-going IR&D effort which involves development of a C-C heat pipe for lower temperature applications. This effort will eventually include fabrication and testing of a heat pipe. Rocketdyne is also looking at a higher thermal conductivity C-C heat pipe design.

The objective of the joint NASA LeRC/Air Force program with PNL (Light Weight Advanced Ceramic Fiber [ACF] Heat Pipe Radiators) is to demonstrate the feasibility of a low mass ceramic fabric/metal liner heat pipe for a wide range of operating temperatures and working fluids (Ref. E-15). Specifically, the NASA LeRC objectives are to develop this concept for application to space radiators with operating temperatures below 500 °K using water as the working fluid. The specific mass goal for these heat pipes is 3 kg/m^2 or less at a surface emissivity of greater than 0.85.

Several heat pipes were built for the ACF program using titanium and copper foil material for containment of the water working fluid. A heat pipe with an 0.2 mm Ti liner was demonstrated in early January 1991. An innovative "Uniskan Roller Extrusion" process has also been developed at PNL and used to draw 0.76 mm wall tubing to a 0.05 mm foil liner in one pass. The water heat pipes fabricated for LeRC have been subjected to a test program to evaluate performance and reliability at demanding operating conditions.

Future thrusts of the ACF program will be to perfect the heat pipe fabrication procedure using very thin (0.025 to 0.05 mm) foil liners which are internally texturized by exposure to high pressures (Ref. E-15). Plans will also be developed to perform hyper-velocity and ballistic velocity impact tests in order to determine if secondary fragments from a penetrated heat pipe will result in failures of neighboring heat pipes. Another major challenge will be to design a heat pipe with high conductivity, low mass fins as a first step toward the fabrication of low mass radiator panels.

The NASA LeRC in-house materials program includes radiator surface morphology alteration by arc texturing for emissivity enhancement purposes (Ref. E-15). Emissivity enhancement has been demonstrated for graphite-copper samples.

PP&C State-of-the-Art

Assigning a NASA Readiness Level to PP&C is difficult. There are many systems that have flown which would make parts of the PP&C subsystem Level 9, but each PP&C system was custom design for each spacecraft design. Only until recently has the array and battery PP&C become standardized into packaged systems. In the Space Station Freedom program, standard PP&C components are undergoing qualification. Recent military programs have ended with qualified packaged PP&C component designs.

The PP&C system design is based on a reasonable electronics component evolution, and there are no significant technology issues associated with its development. It will be necessary to fabricate breadboard hardware for testing and evaluation purposes, but there is no need to initiate any advanced component development. Most of the hardware elements have been or shortly will be operating in a relevant environment on other spacecraft or the SSF electrical power subsystem.

DEVELOPMENT PLANS

The development program was divided into seven major tasks. The first four tasks are component development tasks. The last three tasks includes the system design, fabrication, integration, and testing for the Ground Engineering System (GES), Qualification Unit (QU), and Flight Unit (FU).

Testing will be done on the component, subsystem, and system level (qualification testing). The system level tests will show any possible negative interactions between subsystems.

The PV/battery development tasks are described in the following sections. Each task will include a section on objectives and the statement of work.

Task 1. PV Array Development

Objectives: Complete development of high efficiency tandem cell, a low mass flexible array, and an automatic or robotic deployment approach.

Statement of Work: The following subtasks are identified:

Task 1.1 Subscale PV Array Development. Complete development of GaAs/CIS or advanced (AlGaAs/CIS) tandem cell. Develop a larger cell size. Test subscale PV panels to verify performance under appropriate environmental conditions (thermal cycling, vibration tests, strength and stiffness, radiation, etc.). Develop the array structure and deployment concept.

Task 1.2 Full Scale PV Array Development. Develop and test full scale PV arrays. Verify adequate performance.

Task 2. Battery Subsystem Development

Objectives: Develop a full scale flight weight battery module which can be integrated into a PV/battery power system. Demonstrate materials compatibility, safety, and performance margins.

Statement of Work: This effort was divided into the following subtasks:

Subtask 2.1 Preliminary Battery Development. Demonstrate the feasibility of the selected battery design for planetary surface applications. Test a prototype battery to demonstrate materials compatibility, safety, and component life.

Subtask 2.2 Final Battery Module Design. Design a full scale flight weight battery module.

Subtask 2.3 Battery Module Fabrication, Assembly, and Testing. Build and test (performance, mechanical, and thermal cycling) the battery module. After initial breadboard validation testing is complete, perform additional module testing in a relevant environment.

Task 3. Thermal Management Subsystem Development

Objectives: Develop and demonstrate a low mass, reliable heat pipe radiator which will operate at about 600 °K. Develop and demonstrate components for thermally isolating the battery from the rest of the system and the environment. Develop a thermal management approach for the PP&C components.

Statement of Work: This effort was divided into the following subtasks:

Subtask 3.1 Heat Pipe Demonstration. Fabricate and test representative length heat pipes to fully characterize heat pipe performance. Compare test results with predicted performance over the anticipated range of operating conditions including startup, shutdown, and restart.

Perform limited life testing of the heat pipes. Identify deterioration mechanisms, measure deterioration rates, and assess the adequacy of assembly and cleaning procedures. Withdraw heat pipe samples sequentially throughout the test period. Drain, section, and analyze the samples for corrosion, carbon diffusion, braze stability, and any other signs of deterioration.

In conjunction with the heat pipe performance and life testing, develop and demonstrate techniques for nondestructive examination (NDE) of the heat pipe elements and assembly. Assess the adequacy of liner bonding and weld joints. Determine if the correct fit-ups and interfaces were obtained in the assembled piece.

Subtask 3.2 Radiator Enhancement. Increase the applicability of the radiator concept by performing various enhancement tasks. Examine survivability options and extended length heat pipes.

Perform analytical assessment, testing, and enhancement design development to insure long life for Mars missions. Consider natural threats such as micrometeoroids and dust erosion.

Fabricate a long heat pipe. Develop alternate techniques for fitting the liner into the tube. Investigate alternative tube fabrication, liner fabrication, and coating processes.

Subtask 3.3 Heat Pipe Integration and Testing. Develop and demonstrate techniques for attaching the heat pipes to the battery assembly. Subsequently, demonstrate heat pipe integration into a representative radiator section. Test the radiator section to provide an accurate overall heat transfer coefficient. Verify radiator dynamics and performance. Test the assembled unit in a cold wall, vacuum chamber, simulated space environment. Validate temperature drop predictions and assess component interactions.

Subtask 3.4 Thermal Management Subsystem Design, Assembly, Fabrication, and Testing. Develop the detailed design of the thermal management subsystem. Resolve specific design issues such as the effects of differential thermal expansion, deployment features, monitoring instrumentation and control equipment, local insulation, and trace heating of the battery for thawing.

Design, fabricate, and test a representative full scale thermal management subsystem. Complete performance, mechanical (stress, shock, and vibration), thermal cycling tests.

Task 4. PP&C Subsystem Development

Objectives: Demonstrate adequate steady state and transient performance, life, and immunity to the environment (including launch and operating). Demonstrate the software capability to handle power system nominal operation and failure modes.

Statement of Work: This effort was divided into the following subtasks:

Subtask 4.1 Electrical Components. Build breadboard units to demonstrate and check functional performance of the individual component circuit designs. Incorporate any necessary design modifications and improvements into brassboard units. Verify functional performance within the constraints of the actual component configuration. Fabricate prototype units and perform a series of tests using simulated input and output loads. Test the controller to validate the operating system software. Include the following testing:

- start up, steady state and transient control;
- failure simulation and detection and switching;

- effects of temperature extremes and thermal shock;
- effects of environment;
- shock and vibration;
- cold plate heat loads; and
- EMI generation and susceptibility.

Subtask 4.2 Software - Check out the controller software using simulated inputs and outputs.

Task 5. Ground Engineering System (GES) Testing

Objectives: Design a system concept which will meet Mars power system applications. Show the concept feasibility. Verify adequate performance and life for an integrated system.

Statement of Work: This effort was divided into the following subtasks:

Task 5.1 GES Design. Identify and characterize specific power system applications. Determine power system requirements. Define optimum power system module size based on trade studies which minimize the overall life cycle cost of the power system for all applications (trade between development cost and mass).

Identify concepts and perform tradeoff studies (performance, reliability, risk, safety, and life cycle cost) to select the optimum system design. Complete feasibility studies for the selected concept. Definitize the remaining hardware development tasks based on the system concept chosen.

Design a complete power system to at least meet the minimum life requirement (10 years for arrays and 5 years for batteries). Perform both design point and transient analyses to verify the design. Perform this task in parallel with the component development tasks to insure proper system characteristics

Task 5.2 GES Performance Tests. Manufacture the GES components and assemble. Pressure and leak check the assembly, fill the batteries and heat pipes, complete a electrical check-out, and check out all active devices to the extent possible.

Design the test program for the GES to verify all performance characteristics of the unit in conjunction with subjecting the GES to acceptance level tests. Conduct performance and thermal vacuum tests during a single thermal vacuum test sequence to demonstrate technical capabilities while minimizing program cost. Complete the following performance testing activities:

- checkout and refine subsystem assembly procedures;
- verify operation of ground support equipment (GSE) and interfaces with the GES;
- verify start up capability;
- verify steady state performance characteristics;
- establish effective operating range of the system;
- check out software and verify autonomy (include all operational modes and simulate failure modes);
- simulate normal switchover from one module to another;
- simulate failures to trigger module switchover; and
- establish system sensitivity to off-normal conditions such as: (1) partial loss of radiator cooling, (2) varying radiator heat removal profile, (3) partial loss of electronic cooling, and (4) temporary losses of heat sink with varying duration.

Task 5.3 GES Life Test. Refurbish the GES after completion of performance, acceptance, and margin tests. Refurbishment will include the following:

- replacing or repairing components as necessary to ensure that the GES is returned to its as-built condition;
- addition of special instrumentation required for the life test phase; and
- modification of the main radiator heat removal system or replacement of the main radiator with a dump heat exchanger.

Install instrumentation to provide a comprehensive diagnosis of the "health" of the GES and to monitor for degradation of major assemblies and individual components. Place the GES on a multiyear life test in air. Operate the GES at its nominal operating point, with expected GES variations in power output and environment.

Disassemble the GES for diagnosis at the end of the life test. Determine areas to be examined by an analysis of the health monitoring data and from the reliability analysis predictions.

Task 6. Qualification Program Testing

Objectives: Fabricate the qualification units (QUs). Verify adequate performance and life for the QUs.

Statement of Work: Develop a comprehensive performance and dynamic testing program which will provide a formal demonstration that the power system will perform as designed after being subjected to simulated launch conditions. Select launch environmental conditions which envelope the probable intensities developed by various launch scenarios.

Begin the qualification effort (a typical approach is shown in Figure E-5) by qualifying the assemblies. Fabricate and assemble the qualified production items into the QU. Qualify the QU by the rules for space vehicle qualification.

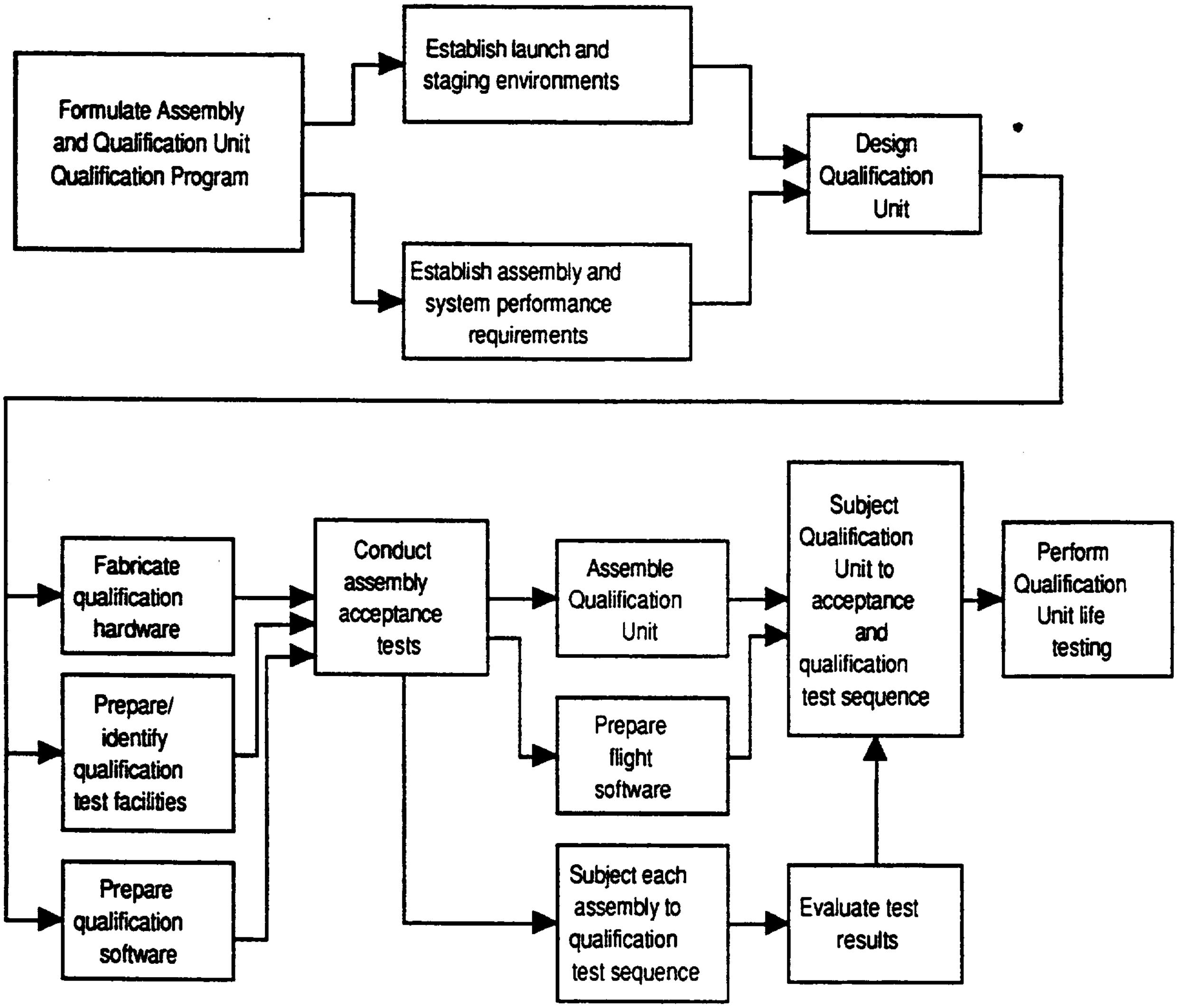


Figure E-5. PV/battery power system qualification program.

This effort was divided into the following subtasks:

Subtask 6.1 Qualification Performance Testing. Conduct performance testing at each level to verify that each item performs as designed. Perform dynamic testing to verify the capability of the power system to withstand launch loads, including acoustic, pyroshock and vibrational. A possible performance and dynamic qualification test sequence for components and assemblies is shown in the matrix of Figure E-6. The corresponding qualification test sequence for the QU is shown in Figure E-7.

Subtask 6.2 Qualification Life Testing (Optional). After completion of the qualification performance testing, partially disassemble the QU and examine the unit. Refurbish the QU as required and modify for endurance testing in air, as described for the GES. Life test the QU.

Component or Subassembly	Functional (1)	Leak	Pyroshock	Functional (2)	Random Vibration	Functional (1)	Leak	Acceleration	Functional (1)	Thermal Cycling	Functional (2)	Thermal Vacuum	Functional (2)	Pressure	Leak	EMC	Life	Functional (1)
Battery Module	X	X			X	X	X			X	X	X	X	X	X	X	X	X
Radiators	X	X			X	X	X			X	X	X	X	X	X		X	X
PP&C	X		X	X	X	X				X	X	X	X			X	X	X
Structure	X		X	X	X	X		X	X	X	X						X	X
PV Array	X		X	X	X	X				X	X	X	X			X	X	X

Figure E-6. PV/battery power system assembly qualification test matrix.

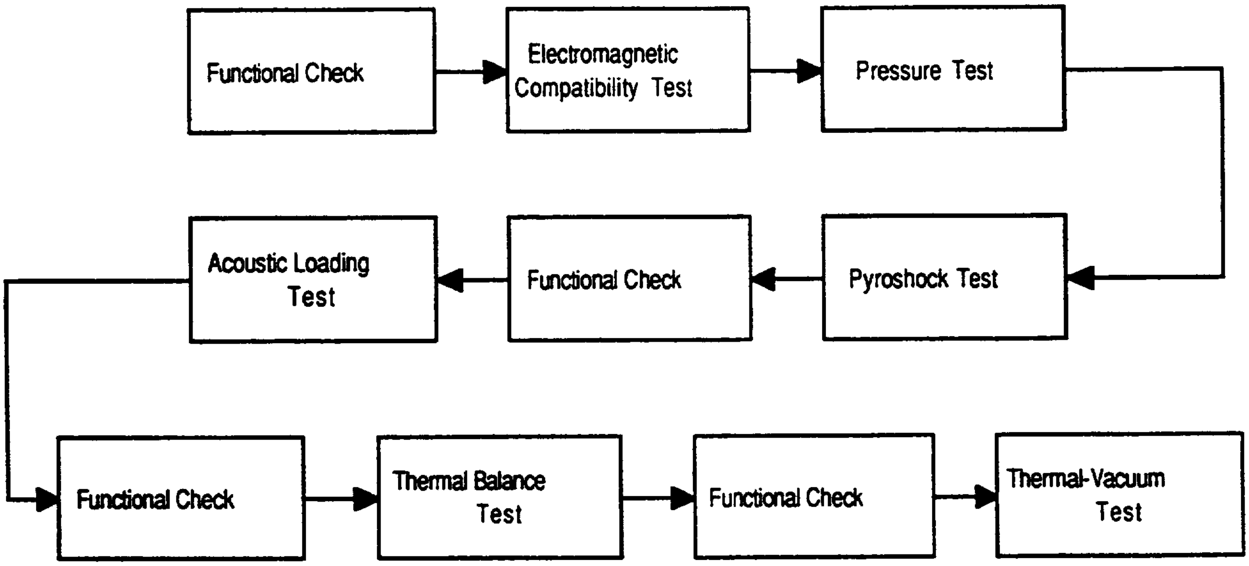


Figure E-7. PV/battery power system QU test sequence.

Task 7 Flight Unit (FU) System Program

Objectives: Fabricate the flight systems and acceptance test the FUs to demonstrate required performance. Deliver the flight systems and provide integration support for the FU with the payload and the launch vehicle.

Statement of Work: Fabricate, acceptance test, and assemble parts to produce the flight systems. Subject the systems to acceptance testing before shipment to the launch site. Use the same test facilities (vacuum chamber, vibration, acoustic) that were used for the qualification program for flight system acceptance testing. Finally, provide launch support activities to insure that the FU meets its design and performance goals associated with integrated payload and launch systems.

The work is divided into the following subtasks:

Subtask 7.1 Flight Component Fabrication. Design, fabricate, inspect, and assemble the components and subassemblies required for the FUs, including all spare parts as required to support the flight system activities.

Subtask 7.2 FU Assembly, Test, and Payload Integration. Assemble and inspect the FUs. Acceptance test the FUs and ship to the launch site. Provide technical support for FU integration with the payload, launch vehicle, and launch support facilities.

Subtask 7.3 FU Launch Support. Provide FU launch support activities to insure that the FU meets its design and performance goals associated with the integrated payload and launch systems.

DEVELOPMENT SCHEDULE

The program starts with a conceptual design task followed by preliminary design and concurrent component development or production of the PV array, battery, thermal management, and PP&C. The detailed design is subsequently completed by the middle of the third year.

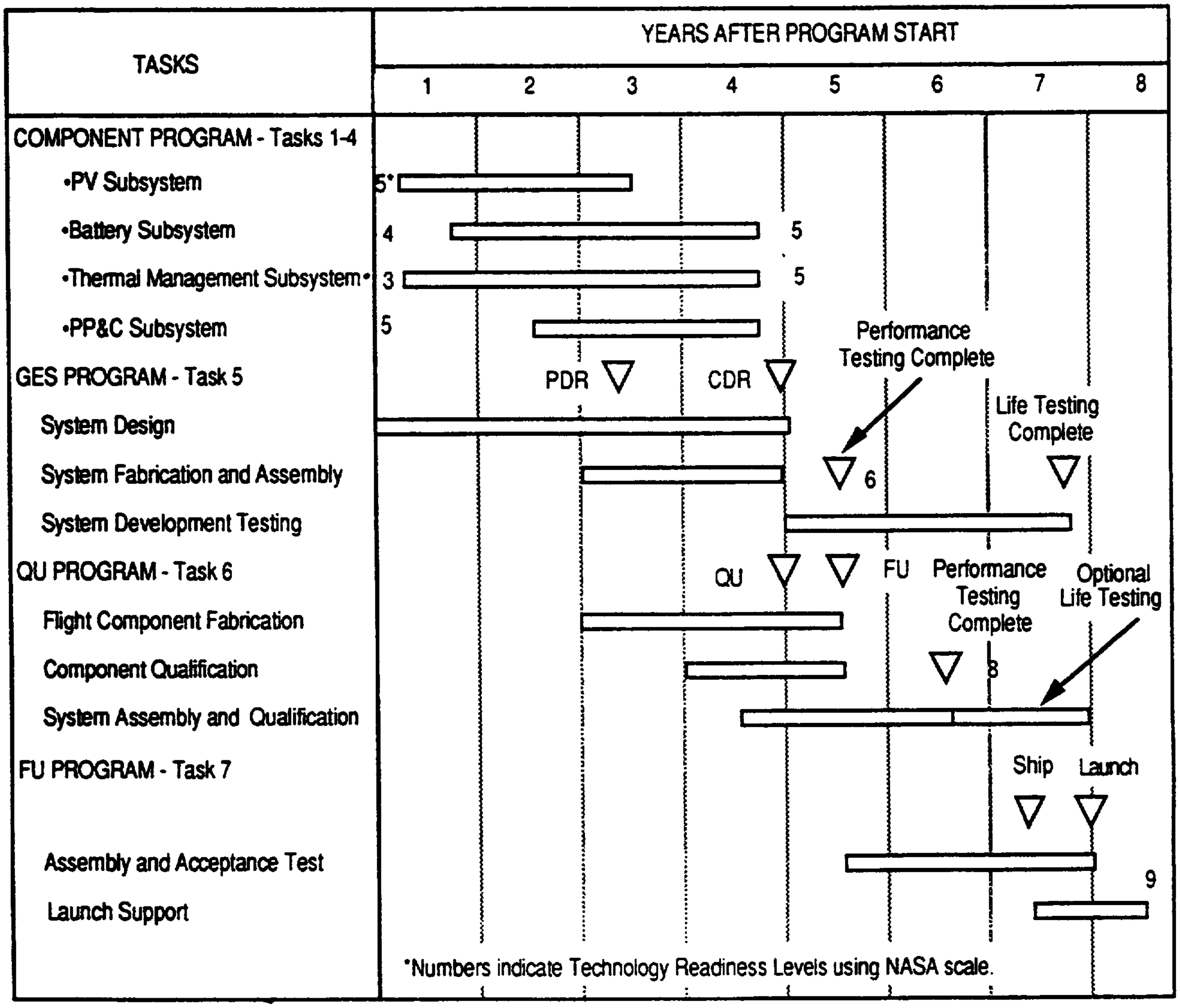
Fabrication of components for ground testing for the Ground Engineering System (GES) starts with procurement of long lead materials and equipment in the first part of the fourth year. This leads to assembly of the system by the end of the fifth year.

The GES will be similar to a flight system but will have features such as additional instrumentation and readily removable components to expedite gathering of engineering data and to permit modification of components. It will be tested in a vacuum chamber under normal and off-normal design conditions. After this phase of testing, it will be partially disassembled, examined, refurbished as necessary, and put on life test, nominally for 1 year.

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components, and a complete system. The design effort will be minimal since it would involve only minor modifications to the GES design.

The flight phase of the program includes fabrication, assembly, and acceptance testing of the flight units.

The estimated development and production schedule for the power system is shown in Figure E-8. A minimum development time is about 8 years.



*Numbers indicate Technology Readiness Levels using NASA scale.

Figure E-8. GaAs-CIS PV array/NaS battery power system development schedule.

REFERENCES

1. Kim, N.P.; and Devaney, N.E.: Low Cost $CuInSe_2$ Based Solar Cells for Space Application. 26th IECEC Proceedings, Volume 2, August 1991, pp. 314-320.

2. Kim, N.P.; et. al: High Specific Power (AlGaAs)GaAs/$CuInSe_2$ Tandem Junction Solar Cells for Space Application. 24th IECEC Proceedings, Volume 2, August 1989, pp. 779-783.

3. Stanbery, B.J.; et. al: Thin Film GaAs/$CuInSe_2$ Solar Cells for Space Power Application. 19th IECEC Proceedings, August 1987, pp. 280-284.

4. Landis, G.A.: Review of Thin Film Solar Cell Technology and Applications for Ultra-Light Spacecraft Solar Arrays. Proceedings of the 10th Space Photovoltaic Research and Technology (SPRAT) Conference, 1989, pp. 180-197.

5. Cooley, Lt. W.T.: Air Force Photovoltaic Array Alternatives. 26th IECEC Proceedings, Volume 2, August 1991, pp. 275-280.

6. Stella, P.M.; and Kurland, R.M.: Development of the Advanced Photovoltaic Solar Array. 26th IECEC Proceedings, Volume 2, August 1991, pp. 269-274.

7. High Energy Density Rechargeable Battery, Preliminary Design Review. Hughes Aircraft Company briefing, 27 April 1988.

8. Petri, D.A.; Cataldo, R.L.; and Bozek, J.M.: Power System Requirements and Definition for Lunar and Mars Outputs. 25th IECEC proceedings, Vol. 1, August 1990, p. 18.

9. Withrow, C.A.; et al: SEI Rover Solar-Electrochemical Power System Options. 26th IECEC, August 1991, p. 319.

10. Marcoux, L.; and Allen, D.M.: Conceptual Design of a Large Spacecraft Power System Utilizing a Sodium-Sulfur Battery. IECEC'83, Vol. 4, August 1983, p. 1470.

11. Grzyll, L.R.: Investigation of Heat Pipe Working Fluids for Thermal Control of the Sodium/Sulfur-Battery. 26th IECEC proceedings, Vol. 3, p.1390.

12. Kim, N.P.: Personal Communications. Boeing Defense and Space Group, Seattle, WA, 10 September 1991 and 3 October 1991.

13. Linden, D.: Handbook of Batteries and Fuel Cells. McGraw Hill Book Company, 1984.

14. Taenaka, R.: Personal Communications. Hughes Aircraft Co., 19 February 1992.

15. Juhasz, A.J.: An Overview of the LeRC CSTI Thermal Management Program. Paper no. AIAA-91-3528, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 4-6, 1991.

APPENDIX F

THERMIONIC REACTOR POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX F - THERMIONIC REACTOR POWER SYSTEM TECHNOLOGY ROADMAP

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APPENDIX F

INTRODUCTION

Space Nuclear Reactor Power Systems (SNRPS) are primarily suitable for application to satellites with high power requirements and fixed base high power planetary surface applications and electric propulsion applications. Currently envisioned Mars surface power systems include power levels of 25 and 75 kWe.

Thermionic reactor systems use a passive power conversion approach to generate electricity. Typical thermionic power conversion systems include STAR-C (out-of-core TI, conduction cooled UC_2/graphite core), Driver Fuel In-Core Thermionic Fuel Element (TFE), and the In-Core TFE Heat Pipe Cooled Reactor (Refs. F-1 through F-10).

The thermionic fuel element (TFE) based power systems have the potential to be designed for use at sufficiently low temperatures such that the reactor structure and containment can be fabricated entirely from stainless steel. Thus, a thermionic system would not need special protection from the Martian environment as would a higher temperature system using refractory metals.

CONCEPT DESCRIPTION

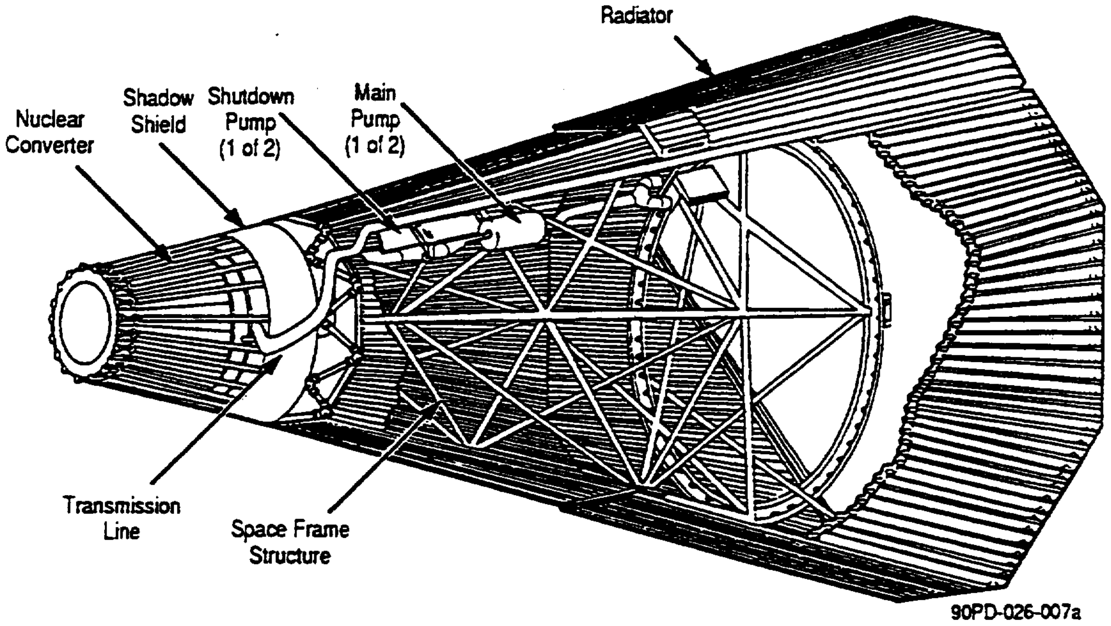
Based on its superior mass characteristics, scalability aspects, and the possibility that it can be operated at a sufficiently low temperature to permit the use of an all stainless steel external structure the Driver Fuel In-core TFE system was selected as the most likely candidate for a thermionic planetary surface based power system of less than 100 kWe output.

The Driver Fuel In-core TFE concept couples in-core TFEs with UO_2 driver fuel pins (where required) for criticality purposes. A pumped liquid metal heat transport loop removes waste heat from the reactor core. The waste heat is rejected to space by a heat pipe radiator. Rotating radial reflector drums are used for both control and primary reactor shutdown. In-core safety rods provide the backup shutdown function. The driver fuel is fully enriched.

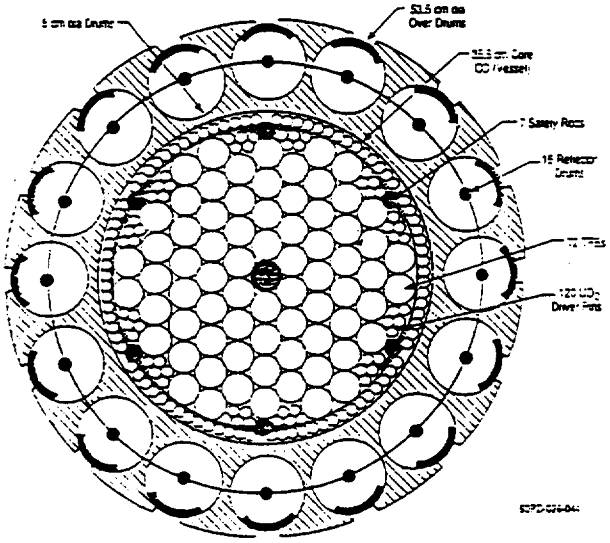
Figure F-1 illustrates the key features of the fast driver fuel in-core TFE system concept, which is based on existing or presently emerging technology. The system is easily scalable over the range of 10 to 100 kWe. The system contains an in-core thermionic reactor coupled to a fixed radiator by a single pumped, liquid metal cooling loop. NaK at a maximum temperature of 970 °K is circulated through the core by one of two redundant EM pumps similar in design to those developed for SNAP 8. A TEM pump similar to the one used in SNAP 10A provides passive decay heat removal. A sodium heat pipe radiator was designed to reject waste heat to space. A redundant power processing and control (PP&C) system, based on a 5 year extrapolation of Space Station Freedom technology, completes the major subsystems in the concept.

The TFE design, also shown in Figure F-1, is based on the UO_2 fueled F-series thermionic converter. The fast neutron spectrum of this fuel provides scalability to higher power levels where no drivers are needed.

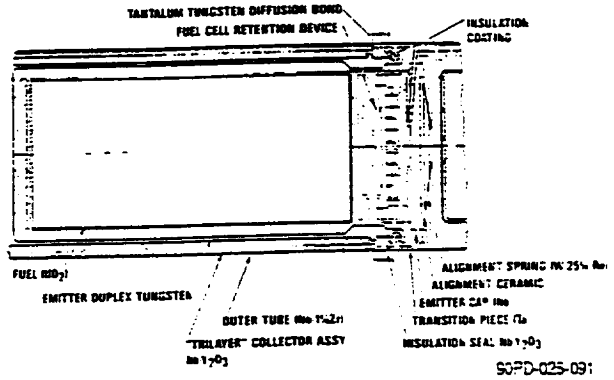
The ability to use the same basic reactor concept over a full range of power outputs reduces the amount of development required and the amount of qualification testing required. In the case of the fast driver concept scaling is accomplished by using the same TFE and driver pin design and adjusting their quantities within the reactor vessel to meet the required power output.



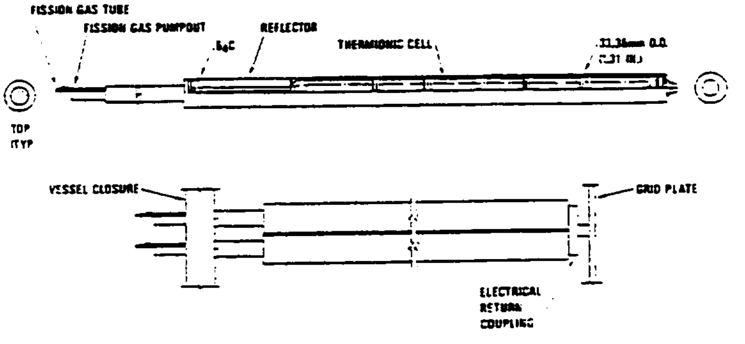
40 kWe Baseline Power System



Reactor Core Layout



Thermionic Cell Assembly



Thermionic Fuel Element Assembly

Figure F-1. Driver Fuel In-Core TFE system concept.

KEY ISSUES

The key issues are summarized in Table F-1.

TABLE F-1. SUMMARY OF KEY ISSUES, IMPACTS AND DEVELOPMENT AREAS

Issues	Impacts	Potential Development Areas
High development cost due to safety assurance program	•Increased life cycle cost	•Flight demonstration program
Safety of nuclear systems during operation	•Increase in system mass for shielding - may be especially significant if in-situ materials are not used	•Use of in-situ materials for shielding
Safety of nuclear systems during launch	•Public pressure may prevent system from being launched and mission from not being completed	•PV/RFC power system •SD power system
TFE lifetime and performance	•Lower emitter temperatures if lifetime cannot be met and higher system mass	•In-reactor TFE and cell tests (TFE Verification Program) •High strength emitter materials
C-C heat pipe Fabricability	•Higher system mass if must use all-metal heat pipes	•C-C metal lined heat pipe development
Radiation hardenability of electronic components	•Heavier shielding mass required if components cannot be hardened	•Under development for AF programs

TECHNOLOGY ASSESSMENT

The technology bases (Refs. F-1 through F-7) were assessed for the following major Fuel Driver In-Core TFE assemblies:

- thermionic power reactor;
- thermionic fuel elements;
- heat rejection subsystem; and
- PP&C subsystem.

The technology readiness of each assembly was evaluated using the NASA technology readiness levels. These evaluations are summarized in Table F-2 which shows that Fuel Driver In-Core TFE assemblies have technology readiness levels ranging from 3 to 5. The technology base for each assembly is briefly discussed in the following sections.

TABLE F-2. IN-CORE TI TECHNOLOGY ASSESSMENT

Assembly	Technology Readiness Level	Comments
Reactor	3	No reactor assemblies built; new military programs due soon; EM pumps under development for SP-100; stainless steel reactor built for SNAP 10 program
TFE	4	Extensive data base from earlier testing; long term tests nearly complete
Heat Rejection Subsystem	4 / 3 *	Currently under development
Electrical	4	Radiation hardened components

*Metal heat pipe/carbon-carbon heat pipe.

Thermionic Power Reactor State-of-the-Art

The driver fuel TFE reactor has been designed for safety from the ground up and requires no technology changes over the complete power range from 10 to 100 kWe. The reactor uses the F series thermionic converter, which has a significant data base from earlier thermionic programs and permits the reactor to scale from 10 kWe to well beyond the 100 kWe level. The UO_2 fuel in the TFE and driver pins has a broad technology base from the Liquid Metal Fuel Breeder Reactor (LMFBR) program and previous TFE test programs (Refs F-6 and F-7).

It is possible to design this reactor to use stainless steel for its containment shell and piping, thus obviating the need to expose any refractory materials to a planetary atmosphere such as on Mars. SP-100 control rod and sliding reflector segment control assemblies are undergoing laboratory testing. These would be used in the design to minimize development costs.

Several basic EM pump design concepts have evolved through the years. Only two designs are inherently low in weight, the ALIP and the thermoelectric electromagnetic (TEM) pump. A TEM pump is being developed for the SP-100 program to-pump molten lithium at higher temperature than this application.

A very substantial liquid metal handling data base exists at the Energy Technology Engineering Center (ETEC), operated by Rocketdyne for the Department of Energy, which was responsible for liquid metal testing of many of the component destined for use in the Clinch River Breeder Reactor. This included operation of numerous EM pumps.

An extensive data base exists for the TFE elements of the driver fuel TFE reactor, however, a full-up assembly of such elements in a critical assembly has not been built. This results in a NASA technology readiness rating of 3.0 for this subsystem, which is the same as that of the SP-100 power reactor.

Thermionic Power Conversion Unit (PCU) State-of-the-Art

The thermionic power conversion device is a static converter, requiring no moving parts. Device efficiency is about double that of thermoelectrics. The TFE is the key technology item in the fast driver fuel TFE design concept. The important technology areas include thermionic converter performance, cesium vapor management, emitter lifetime and electrical insulator stability in the 1000 °K temperature regime and fast flux reactor environment.

The F-series converter is particularly suited to this application because it offers the scalability not available with other thermionic conversion devices. Due to this scalability, critical technology and experience developed at the lower power applications will be valid for the higher power versions.

A TFE verification program, currently in progress, builds directly on an extensive data base developed in the 1960s and early 1970s in an AEC/NASA program. In this program, TFEs were developed and tested and the processes necessary to fabricate long lived components were developed. Thermionic converters were operated for more than 40,000 hrs out-of-pile and a TFE lifetime of 12,000 hrs was demonstrated in-pile.

Fueled emitters were put under test in January 1985 in the TRIGA reactor at General Atomics (GA) during the SP-100 program, to study emitter distortion. As of June 1989, about 35,000 hours of real time testing had been attained. In addition, insulators and intercalated cesium graphite reservoirs are under test and the preliminary results have shown good stability and performance.

The entire TFE verification program, which will include in-pile testing of complete 6 cell TFEs is scheduled for completion in CY 1994 and is currently on track. These factors lead to a NASA technology readiness rating of 4.0.

Heat Rejection Subsystem (HRS) State-of-the-Art

Presently, Rocketdyne division of Rockwell International is involved in the development of a carbon-carbon (C-C) or graphite heat pipe radiator panel. Rocketdyne has a NASA contract entitled "SP-100 Advanced Radiator Concepts" which started in 1987 and is part of the CSTI (Civilian Space Technology Initiative) program. This program will be completed in January of 1993. This program involves development of a C-C heat pipe suitable for use in an SP-100 radiator. However, this technology will be suitable for other radiator applications as well including TFE power systems. The hope is that this program will develop generic technology which can be used for all future space radiators because of its reduced mass compared to metal heat pipe radiators. Carbon-carbon is a new structural material which requires different fabrication techniques than for metals or composites. A key objective of this program is to demonstrate the ability to fabricate C-C heat pipes with a thin metal liner, wicks, caps, and fill tube. The brazing technique required to attach liners is the same as would be required to attach manifolds to the heat pipes. Heat pipes filled with potassium will be designed, fabricated, and tested during this program. Thermal cycle testing will be done in a vacuum to demonstrate the integrity of the heat pipes.

Beginning in 1989, small samples of heat pipe panels were fabricated by Rocketdyne as part of the CSTI program. The next phase of the program will involve fabrication and testing of complete heat pipes. A planned 3.5 year follow-on to this program could involve the development of a full scale C-C heat pipe radiator panel complete with manifolds and interface heat exchanger. This effort would be generic in scope and thus not include flight qualification. It is not clear whether this follow-on would be part of the CSTI program or part of the SP-100 program.

The current technology level of the radiator subsystem is estimated to be about 4 for the metal heat pipe radiators and about 3 for carbon-carbon heat pipe radiators.

Power Processing and Controls State-of-the-Art

The electronic component technology required for thermionic power processing equipment is being developed for future nuclear space power system applications under the SP-100 program. Currently available analytical and design technologies will be used along with radiation testing. Radiation hardened high reliability Class S parts coupled with component redundancy, where necessary, will ensure a reliable and low risk power processing and control system. Rockwell is currently developing the Space Station Power Management and Distribution System; some of that technology is directly transferable to large thermionic power systems. The use of multiple buses, components for the switching of loads, and switching of subassemblies within the power assembly are technologies common to the two programs.

The required technology for the control and data acquisition functions currently exists, and the technology for autonomous control is under active development for Space Station Freedom (SSF), ESSA and other space power programs. The need for radiation hardened components with these capabilities remains to be demonstrated which results in a technology rating of 4 for this subsystem.

MAJOR DEVELOPMENT TASKS

The technology level of the driver fuel in-core reactor power system components range from 3 to 5, and therefore development testing is required. The testing currently envisioned is briefly described in the following sections. Four component development tasks (reactor, power conversion subsystem, HRS, and PP&C) and three system development tasks (Ground Engineering System, Qualification Unit, and Flight System) are described.

Task 1. Thermionic Power Reactor Development

Objectives: Complete development and testing of the reactor components so that the final reactor design can proceed with a high degree of confidence and the ground engineering and flight test programs can be completed on schedule.

Statement of Work: Develop a fast driver thermionic reactor power subsystem conceptual design. Develop and test components that support the reactor subsystem preliminary design. These components would include such items as control rod drivers, reflectors, drum drives, core assembly, bearings, heat exchangers, EM pumps and expansion compensators.

Task 2. Thermionic PCS Development

Objectives: Complete development of the thermionic fuel element design which will meet the desired performance and life requirements.

Statement of Work: The components for these devices are currently in an advanced state of development. The assembly and test of prototype TFEs in a reactor environment. This would require that the devices currently under test at GA be modified to the exact specification required for the fast driver fuel TFE device and be tested in a nuclear reactor similar to the TRIGA reactor.

Task 3. HRS Development

Objectives: Development and demonstration of a small scale radiator panel. Development of full scale heat pipe radiator and interface heat exchanger.

Statement of Work: The following subtasks are identified:

Subtask 3.1 Heat Pipe Demonstration - Various development efforts are required to bring heat pipe technology to a space qualifiable level. Specific efforts include performance testing, life testing, and nondestructive evaluation.

For performance testing, representative length heat pipes will be fabricated and tested to fully characterize the heat pipe performance. Test results will be compared with predicted performance over the anticipated range of operating conditions including startup, shutdown, and restart.

Limited life testing will be performed. These tests will identify deterioration mechanisms, measure deterioration rates, and assess the adequacy of assembly and cleaning procedures. Samples will be withdrawn sequentially throughout the test period, drained, sectioned, and analyzed for corrosion, carbon diffusion, braze stability, and any other signs of deterioration.

In conjunction with the performance and life testing tasks, techniques for nondestructive examination (NDE) of the heat pipe elements and assembly will be developed and demonstrated. This effort will assess the adequacy of liner bonding and weld joints. It will also be determined if the correct fit-ups and interfaces were obtained in the assembled piece.

Subtask 3.2 Radiator Enhancement - To increase the applicability of the radiator concept, various enhancement tasks will be performed. These tasks include survivability options and extended length heat pipes.

Analytical assessment, testing, and enhancement design will be done to insure long life for both lunar and Mars missions. Natural threats to be considered include micrometeoroids and dust erosion (Mars).

Significant radiator performance improvements can be achieved with the use of longer heat pipes in some applications. A long heat pipe will be fabricated. Alternate techniques will be developed for fitting the liner into the tube will be developed. Alternative tube fabrication, liner fabrication, and coating processes will also be investigated.

Subtask 3.3 Heat Pipe Integration and Testing - The first step in this task is to develop and demonstrate techniques for thermal and mechanical bonding of the heat pipe to the radiator manifold. Subsequently, heat pipe integration into a representative radiator section will be demonstrated. Testing of the radiator section will be done to provide an accurate overall heat transfer coefficient. System dynamics and performance will also be verified. The demonstrator will include a surrogate manifold section and a limited number of heat pipes. The assembled unit will be tested in a cold wall, vacuum chamber, simulated space environment. The purposes of this series of tests will be to validate temperature drop predictions, verify manifold design, and assess component interactions.

Subtask 3.4 HRS Design, Assembly, Fabrication, and Testing - This task will provide a focus for the other radiator development tasks. This task will involve the detailed design of a radiator subsystem. Specific design issues will be resolved including the effects of differential thermal expansion, deployment features, monitoring instrumentation and control equipment, local insulation, and trace heating of the manifold and piping

A representative full scale heat pipe radiator and interface heat exchanger will be designed, fabricated, and tested. Testing will include performance, mechanical (stress, shock, and vibration), thermal cycling, and life tests.

Task 4. PP&C Development

Objectives: Complete electronic component development. Verify adequate performance and life.

Statement of Work: Breadboard units will be built to demonstrate and check functional performance of the individual component circuit designs. Design modifications and improvements will be incorporated into brassboard units, which will be used to verify functional performance within the constraints of the actual component configuration. Prototype units will be fabricated and a series of performance tests run using simulated input and output loads. Controller tests will include validation of operating system software. Tests will include:

- Start up, steady state and transient control;
- failure simulation, failure detection, and switching to a backup circuit;
- effects of temperature extremes and thermal shock;
- shock and vibration;

- cold plate heat loads; and
- EMI generation and susceptibility.

Prototype cable harnesses and a prototype parasitic load radiator will be checked out with the electrical components during the system test.

Software will be checked out in conjunction with tests of the controller, using simulated inputs and outputs. Integration and checkout of the software will be performed as part of the system test.

Task 5. Ground Engineering System (GES) Testing

Objectives: Complete the ground engineering system design. Verify all performance characteristics of the unit in conjunction with subjecting the GES to MIL-STD-1540 acceptance level tests. Verify adequate system life. Take the system technology to level 6 (system validation model demonstrated in a simulated environment).

Statement of Work: As a cost reduction measure it has been proposed that GES testing be dispensed with and that space reactor power programs proceed to the flight qualification phase after the component development needed to support final design has been completed. Acceptance of this approach by the technical community is not certain as of this writing, therefore, a discussion of GES testing is included as part of the overall development of a space reactor power system.

After assembly, the GES will be pressure and leak checked, a complete electrical checkout will be performed, and all active devices will be checked out.

Performance and thermal vacuum tests will be conducted during a single thermal vacuum test sequence to demonstrate technical capabilities while minimizing program cost. The performance evaluation will consist of the following major activities:

- checkout and refine subsystem assembly procedures;
- check out procedures for loading NaK into the system;
- verify operation of ground support equipment (GSE) and interfaces between GSE and the GES;

- check out ground cooling systems;
- verify start up capability;
- verify steady state performance characteristics;
- establish effective operating range of the system;
- check out software and verify autonomy (include all operational modes and simulate failure modes);
- establish system sensitivity to off-normal conditions such as: (1)partial loss of radiator cooling, (2)varying radiator heat removal profile, (3)partial loss of electronic radiator cooling, and (4)temporary losses of heat sink with varying duration; and
- determine magnitude of effects on spacecraft for: (1)jitter and vibration, (2)EMI/EMC, (3)thermal radiation to spacecraft, and (4)maneuvering loads.

The MIL-STD-1540 test sequence will integrate acceptance level and design margin tests as appropriate up to qualification levels, to provide a technically sound, minimum cost approach.

When the performance, acceptance, and margin tests are complete, the GES will placed on a multiyear life test. Instrumentation will be used to monitor for degradation of major assemblies and individual components. The GES will be operated at its nominal operating point, with expected GES variations in power output and environment.

The GES will be disassembled for examination at the end of the life test. Specific areas to be examined will be determined by an analysis of the health monitoring data and from the reliability analysis predictions.

Task 6. Qualification Program

Objectives: A comprehensive performance and dynamic testing program for assemblies and the complete system will provide a formal demonstration that the Driver Fuel In-Core TFE system will perform as designed after being subjected to simulated launch conditions. A low-risk approach, incorporating qualification of individual assemblies, followed by qualification of a complete system, termed the qualification unit (QU), characterizes this program. This phase will take the system technology to level 8 (flight qualified).

Statement of Work: The required qualification tests for the assemblies of the in-core TI system are those specified in Section 6.4, "Component Qualification Tests", of MIL-STD-1540B. For the QU, the required qualification tests are in Section 6.2, "Space Vehicle Qualification Tests", of MIL-STD-1540B. Qualification will be to launch environmental conditions selected to envelope the probable intensities developed by various launch scenarios.

The overall approach, starts with qualification of assemblies. Qualified production items are then fabricated and assembled into the QU which is then qualified by the rules for space vehicle qualification. This approach minimizes the risk that some assembly of the QU will fail during the qualification test sequence, thereby nullifying the qualification, and requiring backtracking to recover from the failure.

Performance testing at each level will be conducted to verify that each item performs as designed. The qualification item will be similar to the corresponding engineering item and, therefore, performance testing will be less time consuming.

Dynamic testing will be performed per MIL-STD-1540B, to verify capability of the in-core TI system to withstand launch loads, including acoustic, pyroshock and vibrational. The performance and dynamic qualification test sequence for components and assemblies is shown in Figure F-2. Fabrication, assembly, checkout, and testing of the QU will be similar to the corresponding operations for the GES and therefore, will benefit from the experience with the GES. Test facilities for component and assembly qualification testing will be the same as those for the assembly level testing of the QU test program.

When qualification testing is successfully completed, the QU will be scheduled for life testing for nominally 1.5 years (optional).

Task 7. Flight System Program

Objectives: Produce, acceptance test, and deliver one or more flight systems. Take the system technology to level 9 (flight proven).

Statement of Work: In the flight system program, parts will be fabricated, acceptance tested, and assembled. Systems will be subjected to acceptance testing per MIL-STD-1540B guidelines before shipment to the launch site. The same test facilities (vacuum chamber, vibration, acoustic) that were used for the qualification program would be used for flight system acceptance testing. The flight phase of the program also will include the safety studies and reports necessary to obtain launch approval.

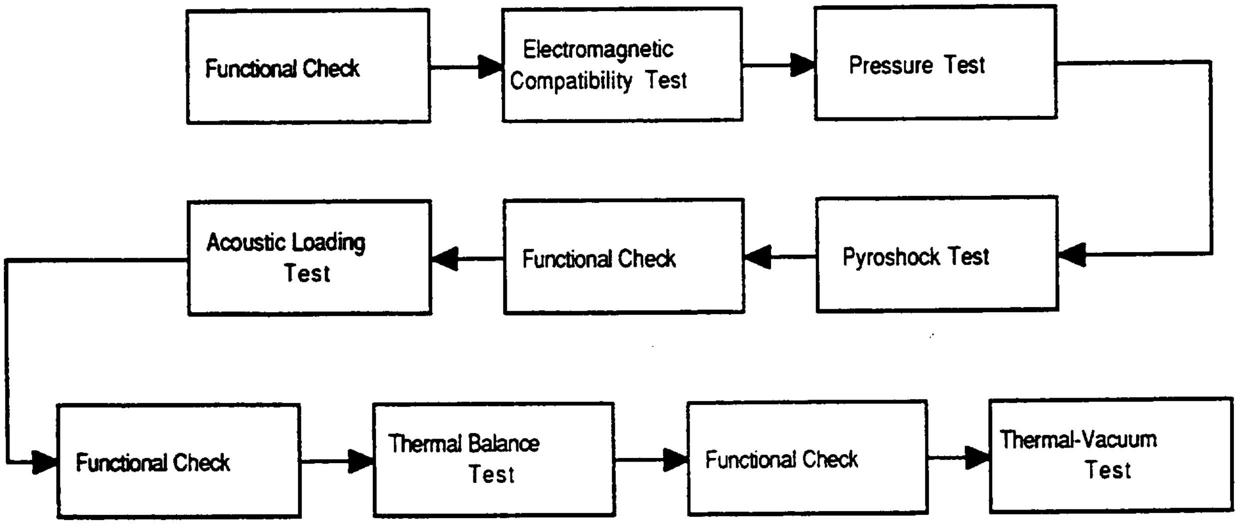


Figure F-2. Thermionic power system QU test sequence (per MIL-STD-1540B).

DEVELOPMENT SCHEDULE

Figure F-3 presents a development schedule for the Fuel Driver In-Core TFE reactor. The total development time to flight is 8.5 years. Development time is expected to be fairly short for this concept since this is a relatively simple system (working fluids always liquid; control done at PP&C; and small radiator) and much of the component technology is state-of-the-art (i.e., stainless steel materials). Significant development work has been done on the TFE elements and the Air Force has recently expressed an interest in developing a relatively high power reactor unit by way of an upcoming RFP.

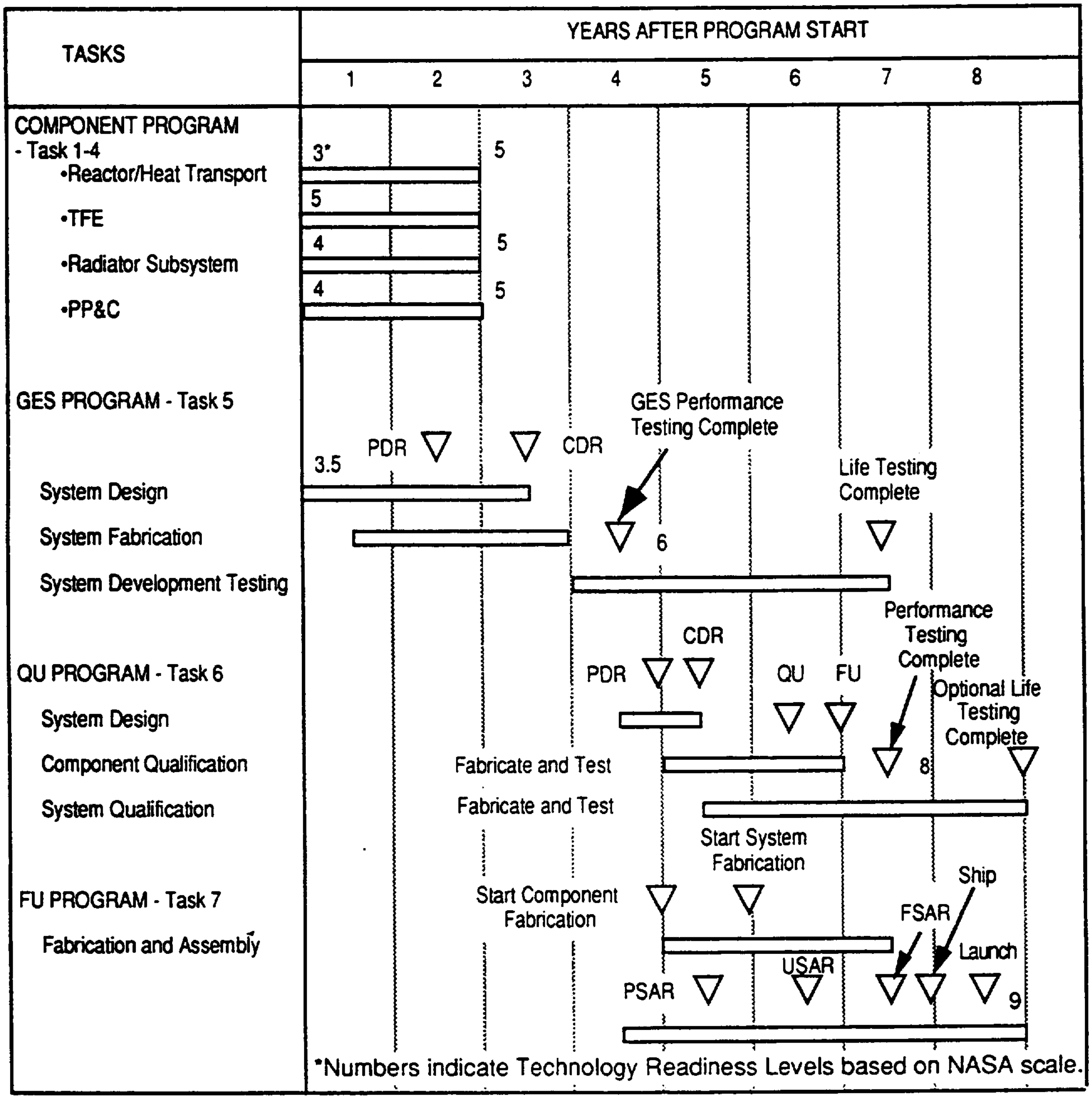
The program starts with a conceptual design task followed by preliminary design and concurrent component development of the reactor and heat transport components, thermionic fuel elements, and electronic assemblies. The detailed design is subsequently completed by the middle of the third year.

It is expected that the program critical path initially will be through design, development, and fabrication of the thermionic fuel elements. The first complement of TFEs must be ready by the middle of the fourth year for assembly in the GES reactor. From there, the critical path is through the nuclear subsystem, assembly of the qualification system, and finally the testing of the qualification system.

The GES will be similar to a flight system but will have features such as additional instrumentation and readily removable components to expedite gathering of engineering data. It will be tested in a vacuum chamber under normal and off-normal design conditions. After this phase of testing, it will be put on life test for 3 yrs.

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components, and a complete system. The design effort will be minimal since it would involve only minor modifications to the GES design.

The flight phase of the program includes fabrication, assembly, and acceptance testing of the flight units and the associated safety analysis (including Preliminary Safety Analysis Review [PSAR], Updated Safety Analysis Review [USAR], and Final Safety Analysis Review {FSAR]) to obtain launch approval.



*Numbers indicate Technology Readiness Levels based on NASA scale.

Figure F-3. Fuel driver in-core TFE power system development schedule.

REFERENCES

F-1. Van Hagan, T.H.; and Mills, J.C.: Fast Driver: A Thermionic Power System for Low Power Applications. Proceedings of the 26th IECEC, Volume 1, August 1991, p.431.

F-2. Mills, J.C.; and Dahlberg, R.C.: Thermionic Systems for DOD Missions. Proceedings of the Eighth Symposium on Space Nuclear Power Systems, Part Three, January 1991, p. 1088.

F-3. TFE Fast Driver Reactor System: A Concept for Low Power Applications. Rockwell International (Rocketdyne Division) briefing to USAF Phillips Laboratory, 31 July 1991.

F-4. Lamp, T.; and Donovan, B.: The Advanced Thermionics Initiative Program. Proceedings of the 26th IECEC, Volume 3, August 1991, p.132.

F-5. Jalichandra, P.; et. al: Thermionic Critical Technology Investigation. Proceedings of the 26th IECEC, Volume 3, August 1991, p.148.

F-6. Bohl, R.J; et. al: Thermionic Fuel Element Verification Program - Overview. Proceedings of the Eighth Symposium on Space Nuclear Power Systems, Part Two, January 1991, p. 636.

F-7. Bohl, R.J.; et. al: Thermionic Fuel Element Verification Program - Overview. Proceedings of the 26th IECEC, Volume 3, August 1991, p.121.

APPENDIX G

SP-100 POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX G - SP-100 POWER SYSTEM TECHNOLOGY ROADMAP

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INTRODUCTION

The SP-100 program is currently developing a high temperature power reactor coupled to a thermoelectric generating system (Refs. G-1 and G-2). The nominal system power level has been selected as 100 kWe. This system uses SiGe GaP for the thermoelectric power conversion.

Use of the SP-100 reactor on the Martian surface would require that the reactor and any of the power conversion equipment fabricated from refractory alloys be isolated from the Martian environment. Also, recent tests indicated that the lunar regolith is not compatible with refractory alloys; consequently, it would also have to be isolated from the lunar environment.

CONCEPT DESCRIPTION

SP-100 is a joint DOD/DOE/NASA program to develop, qualify and flight demonstrate a space power reactor system. The basic configuration of the system currently being developed by the SP-100 program is shown in Figure G-1. The reactor provides thermal energy to a lithium coolant that is pumped by 12 thermoelectromagnetic (TEM) pump assemblies to an equal number of TE converter assemblies. The TE converter assemblies, located at the rear of the conical structure, convert thermal energy to electrical energy. Waste heat from each Thermoelectric Converter Assembly (TCA) is rejected to a secondary lithium loop which transports the waste heat to heat pipe space radiator panels. The radiator panels are deployable by use of flexible bellows in the secondary lithium lines. Generated power is conditioned for the user in the power processing module, which establishes the primary mechanical and electrical interfaces with the mission payload.

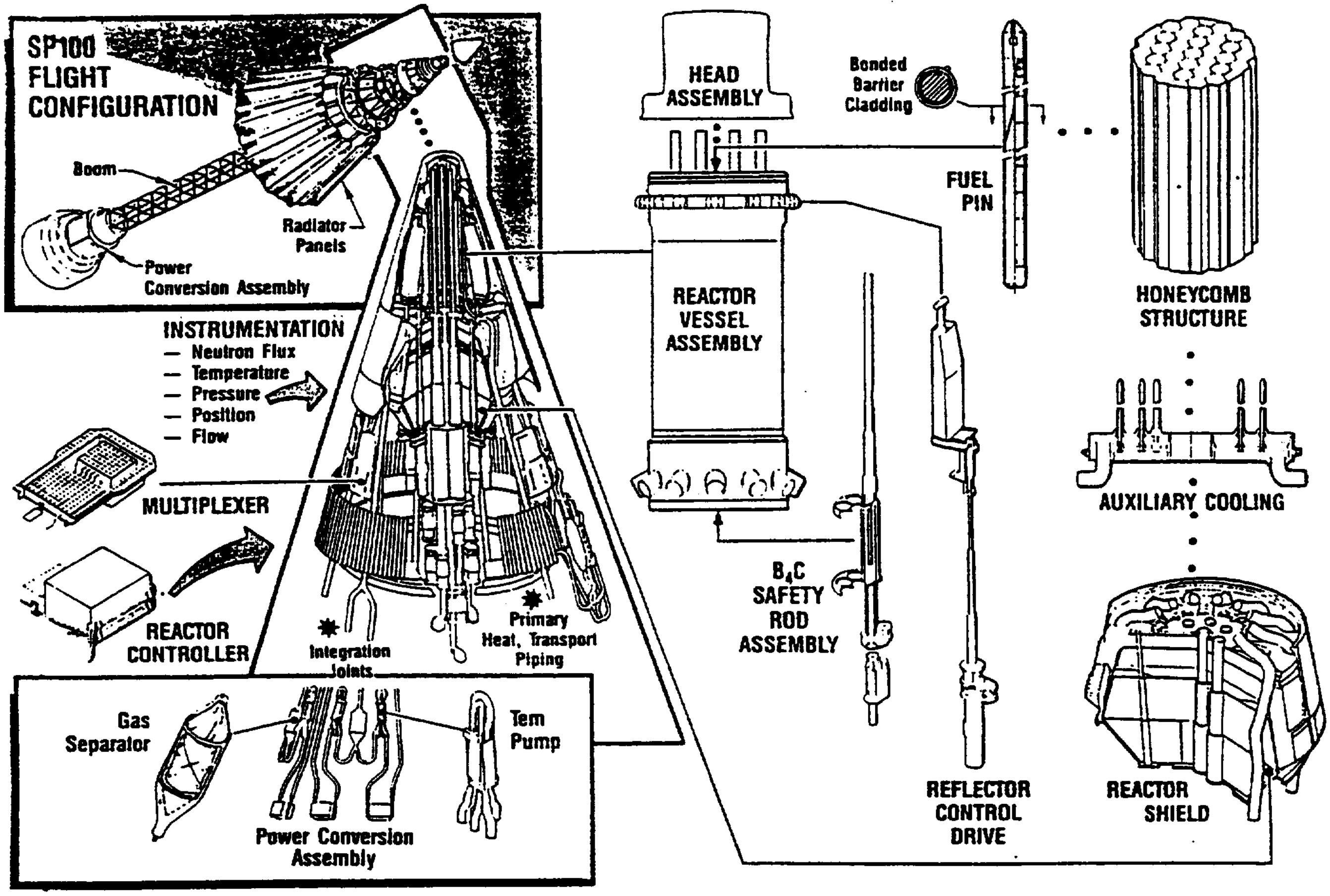


Figure G-1. SP-100 TE generic flight system configuration.

Figure G-1 illustrates the SP-100 Generic Flight System (GFS) configuration. The configuration will require considerable modification for surface applications to provide containment of the refractory alloy components, provide additional shielding consistent with the emplacement geometry, and reconfigure the radiator geometry for packaging.

The reactor designed for the SP-100 system is a fast spectrum design with sealed uranium nitride (UN) fuel pins contained in a single vessel with liquid lithium circulated as the coolant. The reactor is approximately 0.55 meters in diameter by 0.75 meters high. Twelve sliding block reflector control segments provide reactivity control through neutron leakage. PWC-11 refractory metal is used for the reactor fuel pin cladding and for the reactor structure. Three large safety rods are inserted into the reactor core during launch and ascent and are extracted only after a nuclear safe orbit is achieved. The reactor is nominally rated at

2.4 MWt and delivers its thermal energy to liquid lithium at 1350 °K. SP-100 design goals, requirements, and design features for the generic flight system are shown in Table G-1.

TABLE G-1. SP-100 GENERIC FLIGHT SYSTEM DESIGN GOALS, REQUIREMENTS AND FEATURES

Parameter	Requirement	Design Feature (s)
Design lifetime	7 years	•Fuel inventory •Fission gas accommodation
Reliability	0.95	•TE conversion flight proven •Established reactor data base
Main bus power	100 kWe	•Modular design provides scalability
Main bus voltage	200 Vdc	•Option range (28 to 400) readily provided
Load following	Rapid, continuous	•Full shunt
Shielded diameter at user interface	15.5 m (50 ft)	•Larger areas provided at minimum penalty
Radiation at user interface	1.0×10^{13} n/cm^2 5×10^5 Rad	•Reactor shield assembly
Thermal flux at user interface	0.07 W/cm^2	•Meets specified requirement (0.14 W/cm^2) •Easily moderated by boom length
Solar orientation	No restrictions	•Full sun design for radiator
Natural radiation and meteoroids/debris	Mass allowance in baseline for worst case envelope	•Meteoroid armor radiator shields

The flight system radiation shield is mounted directly behind the reactor and consists of both gamma and neutron shield segments. The gamma shield consists of depleted uranium. The neutron shield is made up of a series of axial lithium hydride sections with intervening thermal conductors that carry the gamma and neutron generated heat to the shield's radial surface where it is radiated to space. Shielding for planetary applications may be configured to maximize the use of indigenous materials. Alternatively, where deployment time and effort must be minimized, a 4Π shield configuration could be included with the reactor system design.

Thermal transport is accomplished by liquid lithium that is pumped by 12 thermoelectric driven electromagnetic pumps. The TEs for the pumps are powered by the temperature differential between the working fluids in the primary and secondary loops. This approach ensures pumping of the working fluid as long as the reactor is at an operating temperature higher than the heat rejection loop. It also facilitates cooldown of the reactor with removal of decay heat when electrical power to the payload is no longer required. A pair of supply and return lines feed the liquid lithium to the 6 thermopile heat exchangers located in each Power Conversion Assembly (PCA). The supply and return lines are connected to common headers located within the reactor subsystem. The TEM pumps are located between the PCA and the common return header to the reactor vessel as shown in Figure G-2. The secondary lithium lines provide cooling to the power conversion assemblies and the TEM pumps while transporting the waste heat to a liquid manifold located on the centerline of each of the 12 radiator panels. The lithium is in a frozen condition until system startup. Thawout of the lithium for lunar surface application is accomplished using electrical trace heaters.

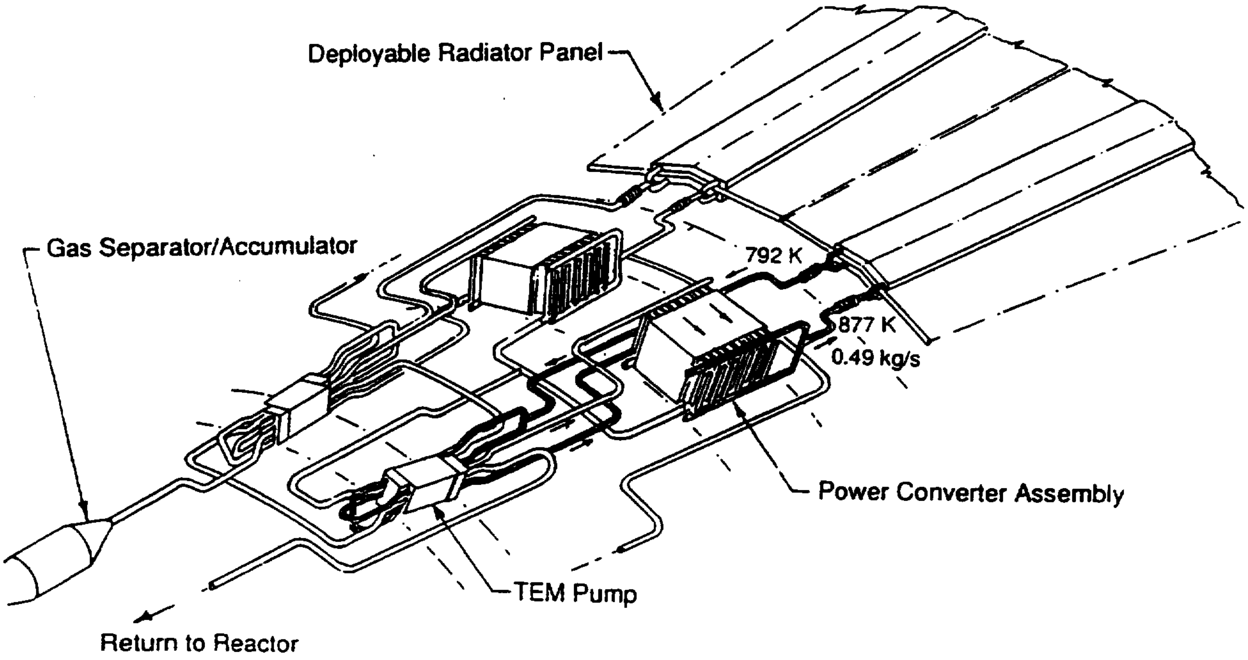


Figure G-2. Radiator interface with heat transport piping.

Figure G-3 illustrates the detail design of the SP-100 TE cell. Each cell consists of 24 thermocouples electrically arranged with four rows or strings of six couples each. The "N" and "P" TE legs are connected at the hot and cold ends by electrical straps to form the series electrical arrangement. The electrical straps extend laterally to intertie the four strings electrically in parallel. This results in an electrical intertie at every half couple. Terminals at each end of the cell provide the series connections to adjacent cells. Low voltage electrical insulators provide insulation for the 1.0 volt potential within the cell outboard of the electrical straps. The compliant pads are outboard of these insulators and are used for differential thermal expansion. Finally, outboard of the pads, 100 volt insulators are used to insulate the cells from the heat source and the radiator subsystems.

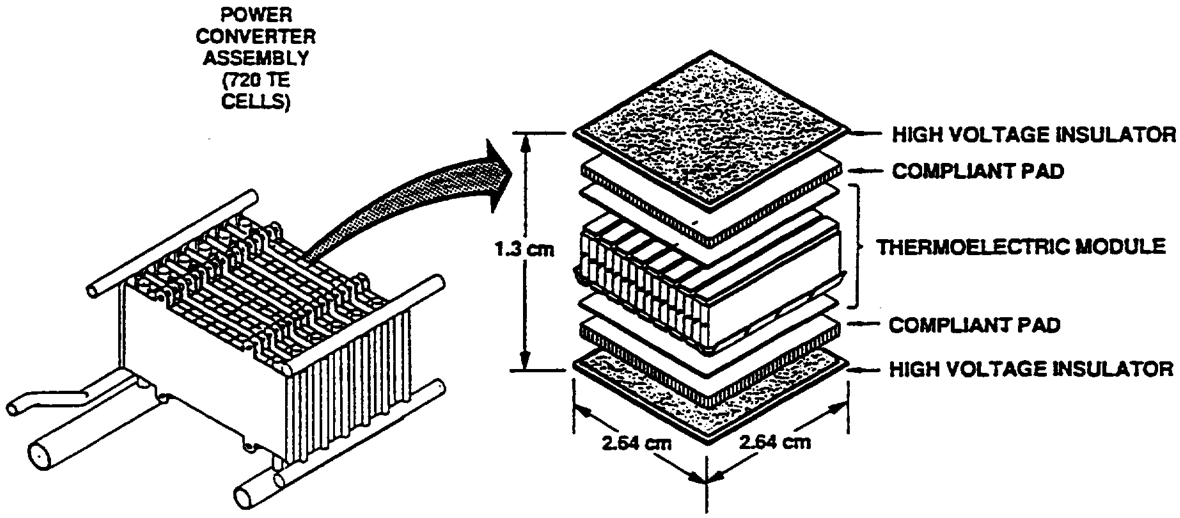


Figure G-3. Thermoelectric cell design.

The radiator panels use heat pipes to transport heat from the liquid lithium secondary manifolds along the length of the panel. The current design approach uses a carbon-carbon heat pipe radiator panel with potassium as the working fluid and a thin metal niobium liner to prevent contact between the potassium and the carbon-carbon structure.

The power conditioning electronics and shunt dissipators are located on a remote power processing module that remains attached to the payload after the boom is deployed. The shunt regulator provides +/-2% voltage tolerance on a nominal 200 Vdc output and is capable of load following without restriction as long as loads do not exceed the power generation level capability. The remote location provides several distinct advantages. It allows state of the art electronics to be used with standard materials of construction. If located near the reactor, the electronics would be required to operate in relatively intense nuclear and thermal radiation fields.

KEY ISSUES

The key issues for the SP-100 GFS are summarized in Table G-2.

TABLE G-2. SUMMARY OF KEY ISSUES AND IMPACTS

Issues	Impacts	Potential Development Areas
High development cost and risk	•Increased life cycle cost and schedule •Heavier system specific mass	•Accelerated (high-risk) development program
Safety of nuclear systems during operation	•Increase in system mass for shielding - may be especially significant if in-situ materials are not used	•Use of in-situ materials for shielding
Low PCA efficiency compared to other PCA options	•Increased launch mass and costs	•Dynamic SP-100 or in-core TI reactor system
Limited TE system power level	•May require multiple systems - increases installation effort and time •New development program to develop higher power systems would be costly	•Parallel development of dynamic SP-100 power system designs

TECHNOLOGY ASSESSMENT

The technology bases were assessed for the following major SP-100 assemblies:

- SP-100 space power reactor and primary loop;
- thermoelectric power conversion;
- heat rejection subsystem; and
- power processing and control.

The technology readiness of each assembly was evaluated using the NASA technology readiness levels (1-9). These evaluations are summarized in Table G-3 which shows that SP-100 assemblies have technology readiness levels ranging from 3 to 4. The technology base for each assembly is briefly discussed in the following sections and summarized in Figure G-4.

TABLE G-3. SP-100 TECHNOLOGY ASSESSMENT

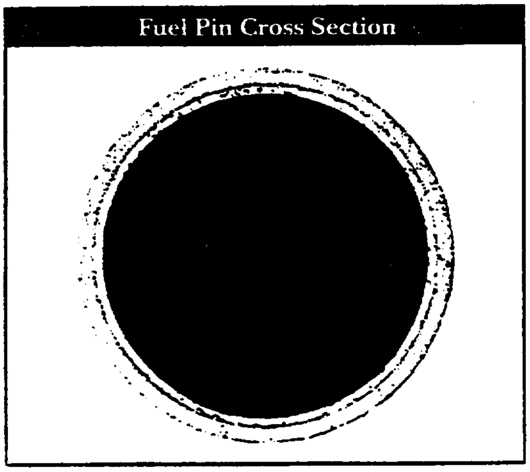
Assembly	Technology Readiness Level	Comments
SP-100 Reactor and Primary Loop	3	Currently funded development program
Thermoelectric Power Conversion	3	Good progress in development program
Heat Rejection Subsystem	4 / 3*	Currently under development
Power Processing and Control	4	Radiation hardened electronics

*All-metal heat pipe/carbon-carbon heat pipe.

SP-100 Space Reactor and Primary Loop State-of-the-Art

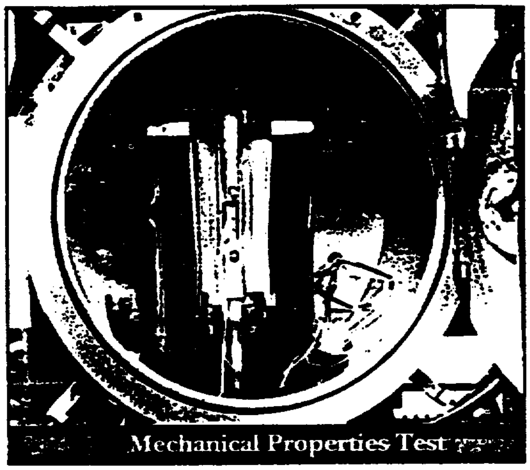
The SP-100 power reactor is actively being developed by the General Electric Corporation under a joint DOD/DOE/NASA contract. Considerable progress has been made in the development with the expenditures of more than $420M to date.

A fuel pin irradiation testing program has demonstrated the capability of the generic flight system fuel pin to meet performance requirements that call for full power operation to about 6.0 atom percent burnup for seven years. Irradiation tests have included UN fuel pins with Nb-1Zr cladding and tungsten or rhenium liners, rhenium liners bonded to Nb-1Zr cladding, and rhenium cladding. The accelerated tests (fluences and burnup levels are achieved



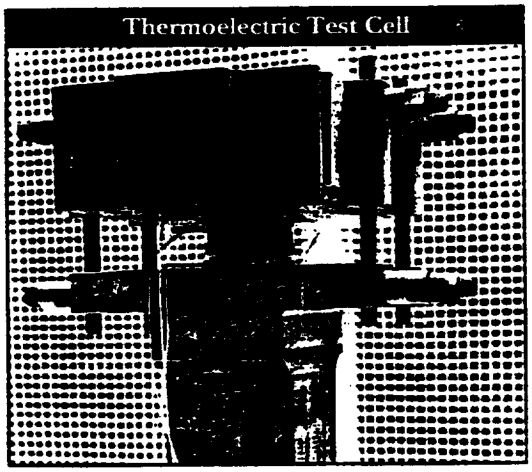
Fuel Pins

76 UN fueled pins with Nb-1Zr/rhenium cladding have been in-reactor tested to full burn-up with no failures. The manufacturing processes are qualified and technology readiness has been achieved.



Materials

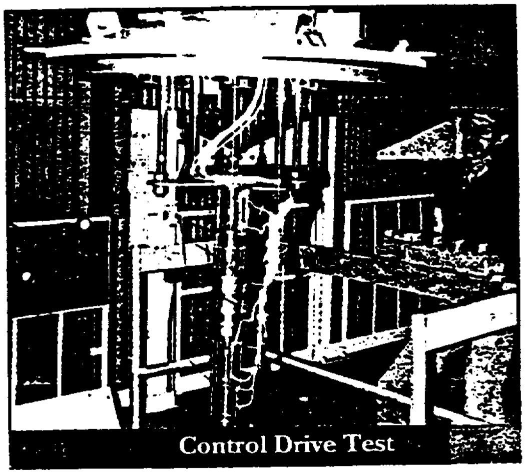
Nb-1Zr and PWC-11 refractory metal mechanical properties tests and irradiation experiments provide high confidence in meeting all requirements for full ten year life.



Thermoelectrics

2nd generation TE cell ingradient tests produced 8.7 watts. Full power, with SiGe cells, is expected in next generation tests. High-voltage insulator and compliant pad tests, conducted under prototypic conditions, provide high confidence in meeting all requirements.

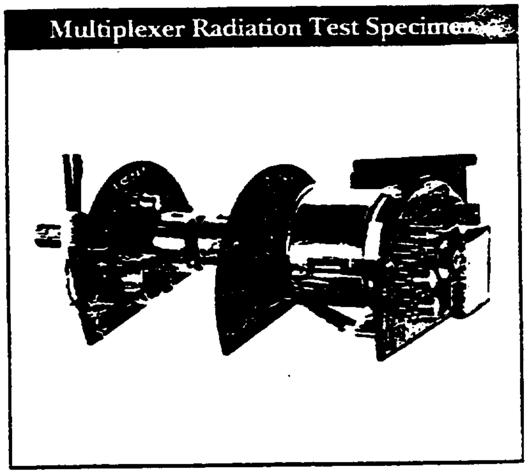
Figure G-4. SP-100 hardware development.



Control Drive Test

Control Systems

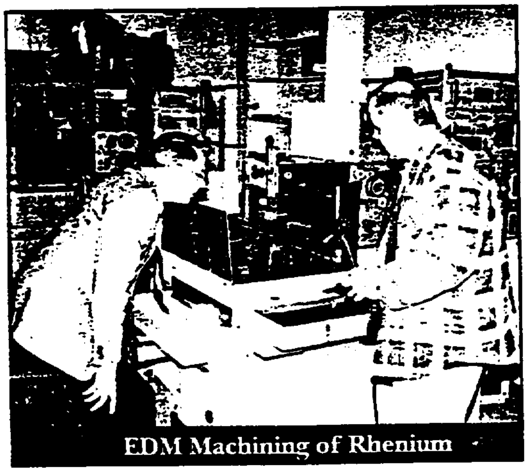
Prototypic first generation control components (clutches, motors, brakes, bearings and coatings) have been developed and tested in high temperature high vacuum environments. Feasibility has been demonstrated.



Multiplexer Radiation Test Specimen

Radiation Hard Electronics

Feasibility has been established through testing of electronic parts required for signal multiplexing in a high gamma and neutron environment. End-to-end signal verification of special sensors, such as the Johnson Noise Temperature Sensor, has been achieved.

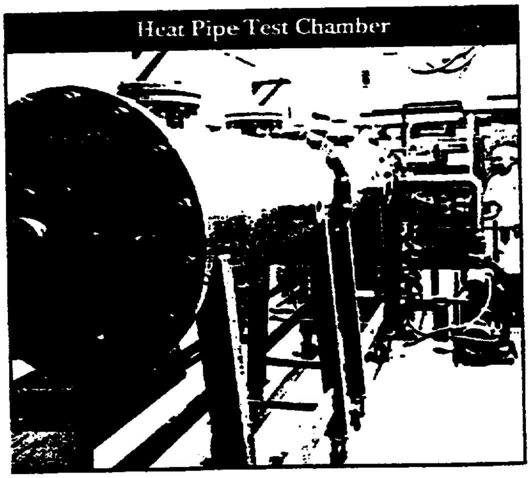


EDM Machining of Rhenium

Refractory Fabrication

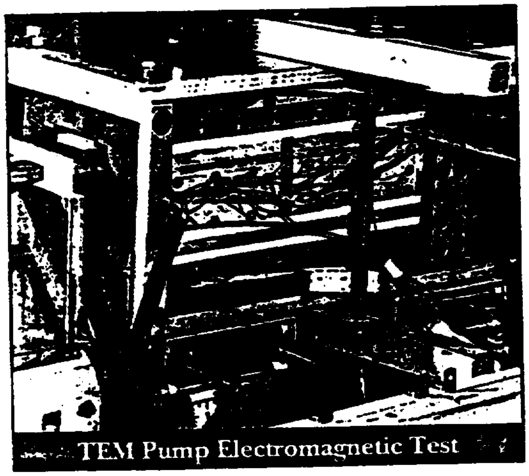
Fabrication procedures and specifications are qualified including welding, EDM machining, bonding and cold forging. Compatibility of Nb-1Zr/-PWC-11 welding and bonding has been demonstrated by over 2000 hours of testing at 1350 °K.

Figure G-4. SP-100 hardware development (continued).



Heat Pipes

Nb-1Zr, titanium and stainless steel foil wick heat pipes with potassium working fluid, have been tested for over 7500 hours at 885 °K. Repeated restart capability has been demonstrated.



Component Testing

Control drives, TEM pump and electronic components have been under feasibility and prototypic environmental testing. A component test loop is being constructed for flight prototypic testing in lithium at 1400 °K. Testing of TEM pumps, TE converter assemblies and radiator panel segments will be conducted to demonstrate technology readiness.

Figure G-4 SP-100 hardware development (completed).

in a fraction of the actual service durations) will provide the information required for formulating the analytical models and validating the fuel pin design. Extended life testing is not planned.

Nuclear assembly fabrication development was investigated and a process to extrude nozzles in the inlet/outlet nozzle course portion of the reactor vessel. A demonstration nozzle has been formed to the correct dimensions by the extrusion process at room temperature.

Nb-1Zr weld specimens have been exposed to lithium to evaluate the effect of welding atmosphere purity and post weld heat treatment on the sensitivity to lithium attack at grain boundaries. Results of the tests indicate that the welds containing bulk oxygen contents greater than 176 ppm experienced lithium attack at the Nb-1Zr grain boundaries. No attack was observed for bulk oxygen levels less than 140 ppm. The results of this work indicate that if low oxygen levels are maintained during welding, the potential for lithium attack is essentially eliminated.

Nondestructive examination methods continued to be developed and improved with most efforts being concentrated on support of the barrier fuel program. Production techniques for nondestructive testing of bonded fuel tubes were developed. A microfocus rod anode X-ray system was used in conjunction with ultrasonic testing to evaluate the detection of defects in various weld configurations.

A reactor flow test was completed including test article fabrication, test planning, and the completion of a comprehensive series of tests. Results of the test program provide important fuel assembly hydraulic design correlations as well as a validated orifice sizing procedure. TEM pump electromagnetic tests are being performed to verify magnetic circuit performance predictions. A test loop is being constructed for primary loop component prototypic testing in lithium at 1400 °K. The technology readiness level of the SP-100 reactor assembly and primary lithium loop components is judged to be at level 3. Sufficient background work has been done to enable the fabrication and testing of a breadboard model to be started.

Thermoelectric Power Conversion Device State-of-the-Art

A first generation TE cell (PD-1) has been fabricated and tested. This cell incorporated the key features of the prototypic design (high voltage insulator, compliant pad, TE module, and module glass) but was limited in power output by several features which resulted from the state of the technology at the time. Most notable of these features were: (1) the incorporation of graphite layers between the compliant pads and the TE module for the purpose of compensating for a coefficient of thermal expansion (CTE) mismatch between those parts; (2) and the limitation of the maximum hot side temperature due to the use of copper braze which was used since the high temperature braze had not yet been fully developed. These features limited the expected output power to approximately 5 watts. This power level compares to an expected 11 watts for a prototypic cell using SiGe T/E legs and 13 watts for one using SiGe/GaP TE legs. The test did demonstrate the ability to predict the cell performance as a function of cell configuration and test conditions. It was the first demonstration that the SP-100 conductively coupled, multicouple cell concept would work under in-service conditions using the enabling features of the high voltage insulator and the compliant pad.

Tests have recently been completed on a second generation cell. Designated PD-2, the design eliminates the graphite layers and low temperature braze features of the PD-1 design to more closely approach the prototypical design. The PD-2A cell yielded 8.7 watts. Figure G-5 summarizes the thermoelectric multicell development effort under the SP-100 program to date.

In addition to the development of TE cells, significant accomplishments have been made in the development of single crystal alumina high voltage insulators, tungsten/graphite compliant pads, and electrodes. These factors contribute to the TE cell technology level rating of 3.

Heat Rejection Subsystem (HRS) State-of-the-Art

Heat rejection loop components operate at less than 900 °K and thus, there are no major materials problems with this subsystem. Radiator manifold liquid-metal thaw experiments have indicated that the GE manifold design using bleed holes to promote progressive melting of the frozen metal coolant within the manifold may be feasible.

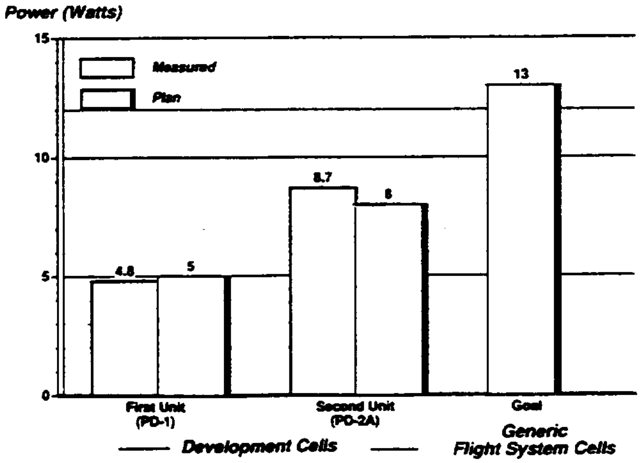


Figure G-5. SP-100 conductively coupled multicouple evolution.

Development of the GE heat pipe radiator/manifold assembly design, which uses a titanium lined beryllium structure with potassium as the heat pipe working fluid, has been delayed in order to permit more resources to be devoted to reactor development and in anticipation of the successful development of a carbon-carbon configuration. A program to develop advanced radiator concepts for SP-100 power systems was initiated by NASA-LeRC in 1987. To meet the SP-100 requirements, while minimized mass, a carbon-carbon heat pipe radiator concept was selected.

The "SP-100 Advanced Radiator Concepts" program, which started in 1987 as part of the CSTI (Civilian Space Technology Initiative) program, will be completed in January of 1993. This program involves development of a C-C heat pipe suitable for use in an SP-100 radiator assembly. However, this technology will be suitable for other radiator applications as well including other nuclear and non-nuclear power systems. This program will develop generic technology which can be used for all future space radiators because of its reduced mass compared to metal heat pipe radiators. Carbon-carbon is a new structural material which requires different fabrication techniques than for metals or composites. A key objective of this program is to demonstrate the ability to fabricate C-C heat pipes with a thin metal liner, wicks, caps, and fill tube. The brazing technique required to attach liners is the same as would be required to attach manifolds to the heat pipes. Heat pipes filled with potassium will be designed, fabricated, and tested during this program. Thermal cycle testing will be done in a vacuum to demonstrate the integrity of the heat pipes. No mechanical testing for stress or shock are planned.

The current technology level of the radiator subsystem is estimated to be about 4 for the all-metal heat pipe radiators and about 3 for carbon-carbon heat pipe radiators.

Power Processing and Controls (PP&C) State-of-the-Art

The electronic component technology required for the SP-100 power processing equipment is being developed for future power system applications under the SP-100 program. Currently available analytical and design technologies will be used along with radiation hardness testing. Radiation hardened high reliability Class S parts will ensure that a reliable and low risk power processing and control system is provided. Rockwell is currently developing the Space Station Freedom (SSF) Electrical Power Subsystem; some of this technology is directly transferable to SP-100 lunar and Mars surface power systems concepts. The use of multiple buses, components for the switching of loads, and switching of subassemblies within the power assembly are technologies common to the two programs.

Several active efforts are underway to develop expert system techniques for real-time control applications. These include a design for the evaluation of Space Shuttle Main Engine (SSME) performance, sensors and data processing for fault detection, and the electrical load power allocations in the SSF electrical power subsystem. The algorithms for power scheduling and balancing on the space station are in development. Expert system software for the SSF electrical power subsystem is being developed using the ADA environment. This work includes implementing and evaluating an ADA based inference engine, which will enable an expert system to run on standard microprocessors for SSF applications.

The required technology for the control and data acquisition functions currently exists, and the technology for autonomous control is under active development for Space Station Freedom (SSF), ESSA and other space power programs. The need for radiation hardened components with these capabilities remains to be demonstrated, which results in a technology rating of 4.0 for the PP&C subsystem.

MAJOR DEVELOPMENT TASKS

The technology level of SP-100 components range from 3 to 4, and therefore development testing is required. The testing currently envisioned is briefly described in the following sections.

Task 1. SP-100 Space Power Reactor Development

Objectives: Complete development testing of the reactor components so that the final reactor design can proceed with a high degree of confidence and allow the ground engineering and flight test programs can be completed on schedule.

Statement of Work: Complete additional test and evaluation efforts to verify the utility of several mass reduction design features which include the safety rods, the aft axial reflector, the sliding radial reflector control, the hemispherical reactor vessel head, the in-vessel thaw accumulator, and materials changes in the shield vessel. Other development areas include the verification of the reactor design; completion of the fuel pin irradiation test program, fabrication methods development, NDE development, multiplexer development, temperature sensor development, pressure sensor development, materials of construction development, verification of control drive assembly design, and the development of tribological coatings. Develop the thermoelectric electromagnetic (TEM) pump power supply and demonstrate the feasibility of the pump design in a 1400 °K lithium environment.

Task 2. Thermoelectric Power Conversion Device Development

Objectives: Complete development of TE power conversion devices with improved performance.

Statement of Work: Additional development efforts in the areas of cell technology development and life verification will continue. Continued development is required in the high voltage insulator, compliant pad, and electrode development areas. Accelerated life testing of molybdenum barrier equipped insulators is being conducted. The development of a tungsten-graphite bilayer electrode for use in the third generation cell is continuing. Significant development work is currently being concentrated on the thermal conductivity reduction in fine particulate p-type SiGe material. Increases in performance of the thermoelectric conversion material of 30 to 40 % have been predicted if the thermal conductivity can be reduced by adding several volume percent of 50 angstrom particles to the materials matrix.

Additional development effort is needed in the areas of SiGe/UT87 bonds. Also effort is required to find a graphite material that has better strength than UT87. Materials with a finer grain size or possibly the use of carbon-carbon composites would be appropriate. Compliant pad materials development also requires additional development effort. Near term efforts will be directed toward the development of more flexible materials which do not diffusion bond and lose their flexibility.

Task 3. HRS Development

Objectives: Develop and test a small scale radiator panel. Develop and test a full scale heat pipe radiator/interface heat exchanger assembly.

Statement of Work: The following subtasks are identified:

Subtask 3.1 Heat Pipe Demonstration - Various development efforts are required to bring heat pipe technology to a space qualifiable level. This effort is assumed to be a follow-on to the Rocketdyne "SP-100 Advanced Radiator Concepts" contract. Specific efforts include performance testing, life testing, and nondestructive evaluation.

For performance testing, representative length heat pipes will be fabricated and tested to fully characterize heat pipe performance. Test results will be compared with predicted performance over the anticipated range of operating conditions including startup, shutdown, and restart.

Limited life testing will be performed. These tests will identify deterioration mechanisms, measure deterioration rates, and assess the adequacy of assembly and cleaning procedures. Samples will be withdrawn sequentially throughout the test period, drained, sectioned, and analyzed for corrosion, carbon diffusion, braze stability, and any other signs of deterioration.

In conjunction with the performance and life testing tasks, techniques for nondestructive examination (NDE) of the heat pipe elements and assembly will be developed and demonstrated. This effort will assess the adequacy of liner bonding and weld joints. It will also be determined if the correct fit-ups and interfaces were obtained in the assembled piece.

Subtask 3.2 Radiator Enhancement - To increase the applicability of the radiator concept, various enhancement tasks will be performed. These tasks include survivability options and extended length heat pipes.

Analytical assessment, testing, and enhancement design will be done to insure long life for both lunar and Mars missions. Natural threats to be considered include micrometeoroids and dust erosion (Mars).

Significant radiator performance improvements can be achieved with the use of longer heat pipes in some applications. A long heat pipe will be fabricated. Alternate techniques will be developed for fitting the liner into the tube will be developed. Alternative tube fabrication, liner fabrication, and coating processes will also be investigated.

Subtask 3.3 Heat Pipe Integration and Testing - The first step in this task is to develop and demonstrate techniques for thermal and mechanical bonding of the heat pipe to the radiator manifold. Subsequently, heat pipe integration into a representative radiator section will be demonstrated. Testing of the radiator section will be done to provide an accurate overall heat transfer coefficient. System dynamics and performance will also be verified. The demonstrator will include a surrogate manifold section and a limited number of heat pipes. The assembled unit will be tested in a cold wall, vacuum chamber, simulated space environment. The purposes of this series of tests will be to validate temperature drop predictions, verify manifold design, and assess component interactions.

Subtask 3.4 HRS Design, Assembly, Fabrication, and Testing - This task will provide a focus for the other radiator development tasks. This task will involve the detailed design of a radiator system. Specific design issues will be resolved including the effects of differential thermal expansion, deployment features, monitoring instrumentation and control equipment, local insulation, and trace heating of the manifold and piping.

A representative full scale heat pipe radiator and interface heat exchanger will be designed, fabricated, and tested. Testing will include performance, mechanical (stress, shock, and vibration), thermal cycling, and life tests.

Task 4. Power Processing and Controls Development

Objectives: Complete development of radiation hardened components. Verify adequate performance and life.

Statement of Work: Breadboard units will be built to demonstrate and check functional performance of the individual component circuit designs. Design modifications and improvements will be incorporated into brassboard units, which will be used to verify functional performance within the constraints of the actual component configuration. Prototype units will be fabricated and a series of performance tests run using simulated input and output loads. Controller tests will include validation of operating system software. Tests will include:

- Start up, steady state and transient control;
- failure simulation and detection and switching;
- effects of temperature extremes and thermal shock;
- shock and vibration;
- cold plate heat loads; and
- EMI generation and susceptibility.

Prototype cable harnesses and a prototype parasitic load radiator will be checked out with the electrical components during the system test.

Software will be checked out in conjunction with tests of the controller, using simulated inputs and outputs. Integration and checkout of the software will be performed as part of the system test.

Task 5. Ground Engineering System (GES) Testing

Objectives: Complete the system design. Verify all performance characteristics of the unit in conjunction with subjecting the GES to MIL-STD-1540 acceptance level tests. Take the system technology readiness to a level of 6 (system validation model demonstrated in a simulated environment).

Statement of Work: As a cost reduction measure it has been proposed that GES testing be dispensed with and that space reactor power programs proceed to the flight qualification phase after the component development needed to support final design has been completed. Since acceptance of this approach by the technical community is not certain as of this writing, a discussion of GES testing is included as part of the overall development of a space reactor power system.

After assembly, the GES will be pressure and leak checked, as appropriate, a complete electrical check-out will be performed, and all active devices will be checked out to the extent possible.

Performance and thermal vacuum tests will be conducted during a single thermal vacuum test sequence to demonstrate technical capabilities while minimizing program cost. The performance evaluation will consist of the following major activities:

- Check out and refine subsystem assembly procedures;
- check out procedures for loading coolant into the reactor coolant loop;
- verify operation of ground support equipment (GSE) and interfaces between GSE and the GES;
- check out ground cooling systems as required;
- verify start up capability;
- verify steady state performance characteristics;
- establish effective operating range of the system;
- check out software and verify autonomy (include all operational modes and simulate failure modes);
- establish system sensitivity to off-normal conditions such as: (1) partial loss of radiator cooling, (2) varying radiator heat removal profile, (3) partial loss of electronic cooling, and (4) temporary losses of heat sink with varying duration; and

The MIL-STD-1540 test sequence will integrate acceptance level and design margin tests as appropriate up to qualification levels, to provide a technically sound, minimum cost approach.

When the performance, acceptance, and margin tests are complete, the GES will be refurbished and will be disassembled for Post Irradiation Examination (PIE).

Task 6. Qualification Program

Objectives: A comprehensive performance and dynamic testing program for assemblies and the complete system will provide a formal demonstration that the SP-100 system will perform as designed after being subjected to simulated launch conditions. Minimize risk by first qualifying individual assemblies and then qualifying a complete system, termed the qualification unit (QU). Take the system technology readiness from a level of 6 to a level of 8 (flight-qualified system).

Statement of Work: The required qualification tests for the assemblies of the SP-100 system are those specified in Section 6.4, "Component Qualification Tests", of MIL-STD-1540B. For the QU, the required qualification tests are in Section 6.2, "Space Vehicle Qualification Tests", of MIL-STD-1540B. Qualification will be to launch environmental conditions selected to envelope the probable intensities developed by various launch scenarios.

The overall approach, starts with qualification of assemblies. Qualified production items are then fabricated and assembled into the QU which is then qualified by the rules for space vehicle qualification. This approach minimizes the risk that some assembly of the QU will fail during the qualification test sequence, thereby nullifying the qualification, and requiring backtracking to recover from the failure.

Performance testing at each level will be conducted to verify that each item performs as designed. The qualification item will be similar to the corresponding engineering item and, therefore, performance testing will be less time consuming.

Dynamic testing will be performed per MIL-STD-1540B, to verify capability of the SP-100 system to withstand launch loads, including acoustic, pyroshock and vibrational.

The performance and dynamic qualification test sequence for components and assemblies is shown in Figure G-6. Fabrication, assembly, checkout, and testing of the QU will be similar to the corresponding operations for the GES and therefore, will benefit from the experience with the GES. Test facilities for component and assembly qualification testing will be the same as those for the assembly level testing of the QU test program. Various component and breadboard subsystems may be demonstrated in space for additional validation.

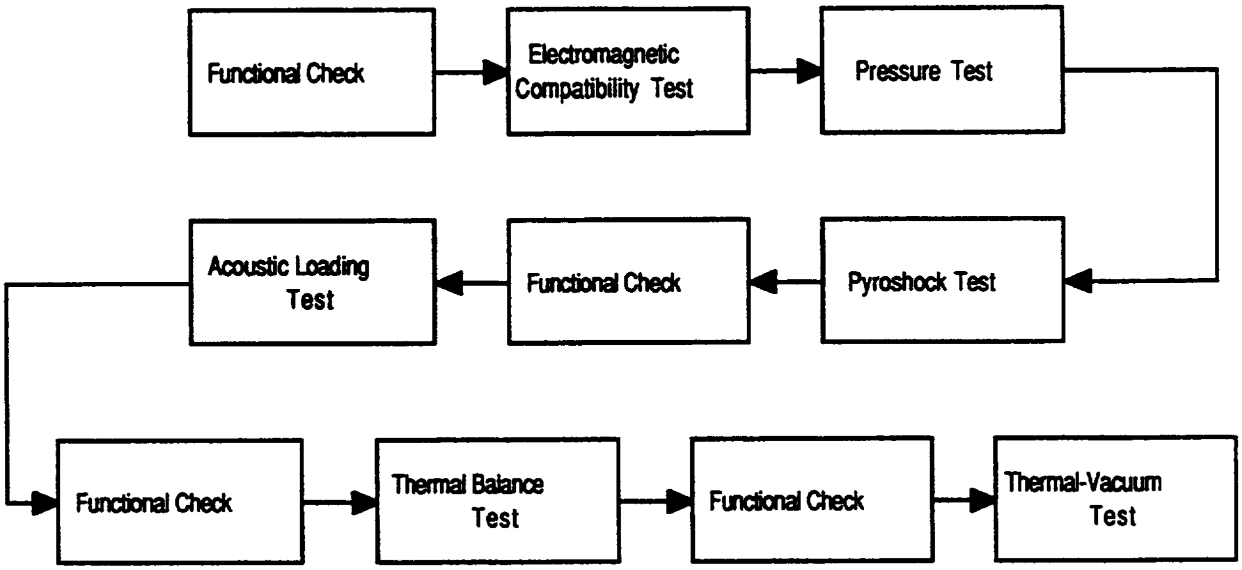


Figure G-6. QU test sequence (per MIL-STD-1540B).

Task 7. Flight System Program

Objectives: Produce, acceptance test, deliver, and launch one or more flight systems. Show successful performance for actual application in space. Take the system from a technology readiness level of 8 to a level of 9 (flight-proven system).

Statement of Work: In the flight system programs, parts will be fabricated, acceptance tested, and assembled. Systems will be subjected to acceptance testing per MIL-STD-1540B guidelines before shipment to the launch site. Many of the same test facilities (thermo-vacuum chamber, vibration, acoustic) that were used for the qualification program would be used for flight system acceptance testing. The flight phase of the program also will include the safety studies and reports necessary to obtain launch approval.

DEVELOPMENT SCHEDULE

The SP-100 program is currently funded but has been experiencing year by year funding reductions that are resulting in significant schedule slippage. Resolution of the funding question will become more apparent with the identification of a specific mission. No mission has been identified to date. A significant amount of development work has been accomplished and the program will probably be ready to start preliminary design with another 3-4 years of component development effort.

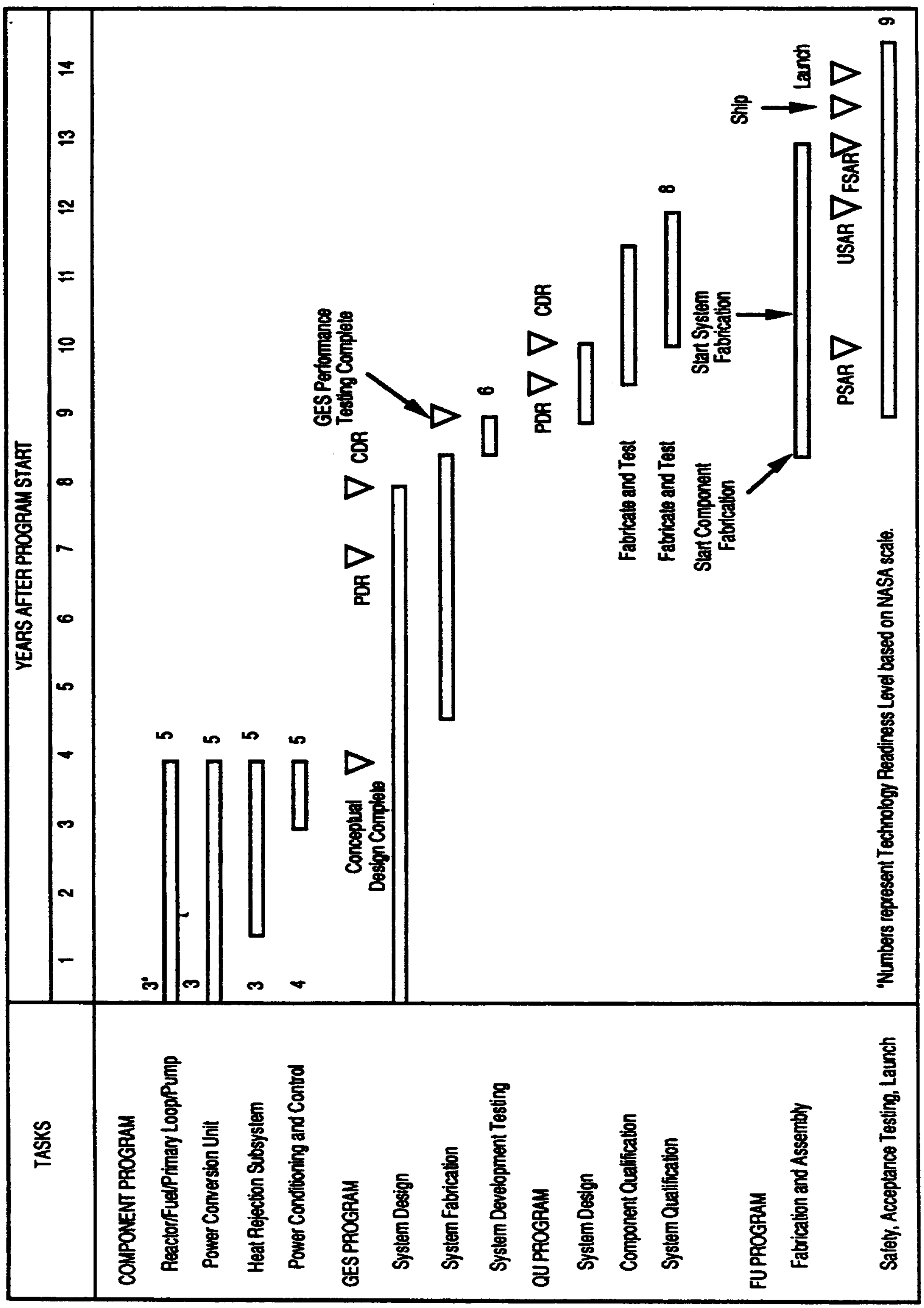
Figure G-7 presents a development schedule for the SP-100 TE generic flight system (additional effort would be required to modify the design for compatibility with a Martian environment). The program starts with a conceptual system design task followed by preliminary system design. Concurrently with the system design, the reactor, power conversion unit, heat rejection subsystem, and power processing and control components will be developed. The preliminary system design is complete by the middle of the seventh year. The detailed system design is subsequently completed by the middle of the eighth year.

Fabrication of components for ground testing for the GES starts with the development and testing of component hardware. The procurement of long lead materials and equipment in the first part of the fourth year leads to assembly of the system in the last half of the eighth year.

The GES will be similar to a flight system but will have features such as additional instrumentation to expedite gathering of engineering data and to permit PIE of components. It will be tested in a vacuum chamber under normal and off-normal design conditions. After this phase of testing, it will be partially disassembled and examined.

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components, and a complete system. The design effort will be minimal since it would involve only minor modifications to the GES design.

The flight phase of the program includes fabrication, assembly, and acceptance testing of the flight units and the associated safety analysis to obtain launch approval.



*Numbers represent Technology Readiness Level based on NASA scale.

Figure G-7. SP-100 TE generic flight system development schedule.

REFERENCES

G-1. Mondt, J.F.: SP-100 Space Subsystems Development Process. AIAA paper 91-3584, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 1991.

G-2. Armijo, J.S.; et. al: SP-100 Progress. AIAA paper 91-3588, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 1991.

G-3. Lunar Electric Power Systems Utilizing the SP-100 Reactor Coupled to Dynamic Conversion Systems. NASA Lewis Research Center Contract NAS3 25808, Rockwell International (Rocketdyne Division), NASA Contractor Report, October 1991.

G-4. Harty, R.B.; Durand, R.E.; and Masin, L.S.: Lunar Electric Power Systems Utilizing the Sp-100 Reactor Coupled to Dynamic Conversion Systems. AIAA paper 91-3520, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 1991.

G-5. Upton, H.; et. al: The Application of SP-100 Technology in a Lunar Surface Power System. AIAA paper 91-3584, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 1991.

APPENDIX H

DYNAMIC SP-100 POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX H - DYNAMIC SP-100 POWER SYSTEM TECHNOLOGY ROADMAP

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INTRODUCTION

Space Nuclear Reactor Power Systems (SNRPS) are primarily suitable for application to satellites with high power requirements and fixed base high power planetary surface applications. Currently envisioned systems include power levels of 25 and 75 kWe for use on the surface of Mars and 100 and 550 kWe units for use on the Moon.

The SNRPS units can be integrated with dynamic or static power conversion systems to provide electric power. The SP-100 program is currently developing a high temperature power reactor coupled to a thermoelectric (TE) generating system. The nominal system power level has been selected as 100 kWe. This system makes extensive use of refractory materials and could used on the lunar surface.

It was assumed that an advanced SNRPS using a dynamic power conversion system would be developed following development of the SP-100 TE power system (some time after the completion of testing of the Ground Engineering System). The electrical power output of the basic reactor can be significantly enhanced by the use of dynamic power conversion technologies. Dynamic power systems concepts include Closed Brayton Cycles (CBC), Stirling Cycles (SC), and Potassium Rankine Cycles (PRC) integrated in various ways with the nuclear power source. Recent studies have shown that electrical power outputs of over 550 kWe can be obtained by the use of CBC, SC or PRC power conversion equipment with the SP-100 reactor.

Use of the SP-100 reactor on the Martian surface would require that the reactor and any of the power conversion equipment fabricated from refractory alloys be isolated from the Martian environment.

DYNAMIC SP-100 POWER SYSTEM TECHNOLOGY ROADMAP

CONCEPT DESCRIPTION

The SP-100 reactor is coupled to a dynamic power conversion system and the resultant power output attainable is increased to about 550 kWe. This increase is due to the greater power conversion efficiency of the dynamic systems.

For each power system case, a common set of ground rules and assumptions has been used for lunar application. These include the following:

- An intermediate heat exchanger is incorporated in all designs to isolate the primary reactor coolant (lithium) from a secondary loop that contained the cycle working fluid (will facilitate maintenance including removal and replacement of the power conversion units);
- auxiliary power is available for thaw and startup;
- a vacuum/guard vessel is provided as part of the system;
- a passive cooling system transports waste heat from all sources to the planetary or lunar surface, where the heat is rejected to space by a radiator;
- a free convection loop is provided to remove decay heat if required;
- an expansion tank with a free surface is provided to accommodate both lithium expansion and helium gas generation in the reactor;
- all power conversion equipment, reactor control actuators and heat rejection equipment is located at grade level to provide for maintainability; and
- a voltage output of 1000 Vac or dc.

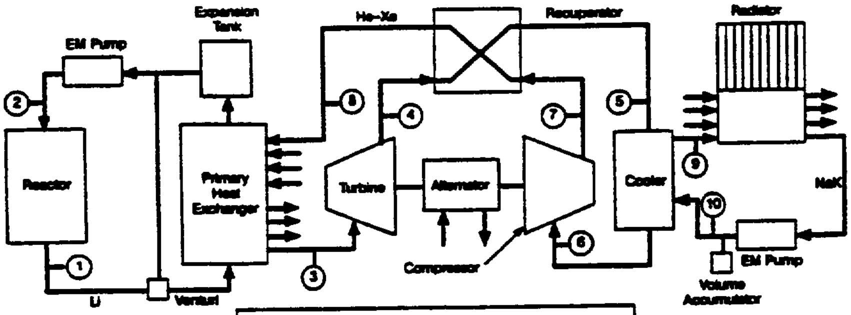
The reactor designed for the SP-100 system is a fast spectrum design with sealed uranium nitride (UN) fuel pins contained in a single vessel with liquid lithium circulated as the coolant. The reactor is approximately 0.55 meters in diameter by 0.75 meters high. Twelve sliding blocks in the radial reflector assembly open to allow neutron leakage from the core, which provides reactivity control. PWC-11 refractory metal is used for the reactor fuel pin cladding and for the reactor structure. Three safety rods are inserted into the reactor core during launch and ascent and are extracted only after a nuclear safe orbit is achieved. The reactor is nominally rated at 2.4 MWt and delivers its thermal energy to liquid lithium at 1350°K.

Shielding for planetary applications may be configured to maximize the use of indigenous materials. Alternatively, where deployment time and effort must be minimized, a 4Π shield would be integral with the reactor system.

The flow schematic and nominal operating state points for the Brayton system are shown in Figure H-1. A single lithium primary loop transports heat from the reactor to the primary heat exchanger. The Brayton cycle uses a helium-xenon gas mixture for the working fluid. Heat is rejected from the cycle through a NaK heat rejection loop. Each Brayton loop is cross coupled to each of the four radiator panels so that if a power conversion system failure occurs there is no loss of radiator area. The NaK loop is also used to cool the alternator. The Brayton turbine uses a single stage radial inflow design and the compressor uses a single-stage radial outflow design. The turbine wheel, alternator and compressor are all mounted on a single shaft. Both the journal and thrust bearings use compliant pad gas lubricated foil bearings. Gross cycle efficiency for the Brayton unit was estimated at 25.7%.

The Free-Piston Stirling Engine (FPSE) is a thermally driven mechanical oscillator operating on a Stirling engine cycle which derives power from the heat flow between a heat source and a sink. The desired displacer motion in the FPSE is produced by gas forces rather than kinematic linkages. This engine operates at the highest overall device efficiency of all known heat engines. In addition, the FPSE is uniquely suited to driving direct coupled reciprocating loads, such as linear alternators, in a hermetically sealed configuration and without the need for high pressure shaft seals or contaminating lubricants.

The flow schematic and nominal operating state points for the Stirling system are shown in Figure H-2. A primary and secondary lithium loop are used to transport heat to the Stirling engine heater. Helium is used for the working fluid for the Stirling cycle. Heat is rejected from the Stirling cooler through a NaK heat rejection loop, which is also used to cool the alternator. As with the Brayton system, the heat rejection loops are cross coupled to each radiator panel. The Stirling engine used in this application is an opposed piston design which is selected for its minimal vibration characteristics. The gross cycle efficiency for the Stirling is 33.0%.



State Point	Fluid	Temperature (K)	Pressure (kPa)
1	Li	1355	—
2	Li	1276	—
3	He-Xe	1300	1,290
4	He-Xe	1041	690
5	He-Xe	640	680
6	He-Xe	405	675
7	He-Xe	556	1,340
8	He-Xe	957	1,330
9	NaK	629	—
10	NaK	395	—

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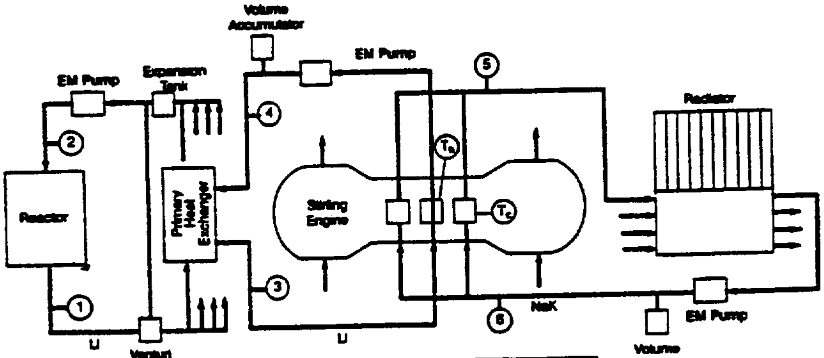
Figure H-1. 550 kWe Brayton cycle power system schematic.

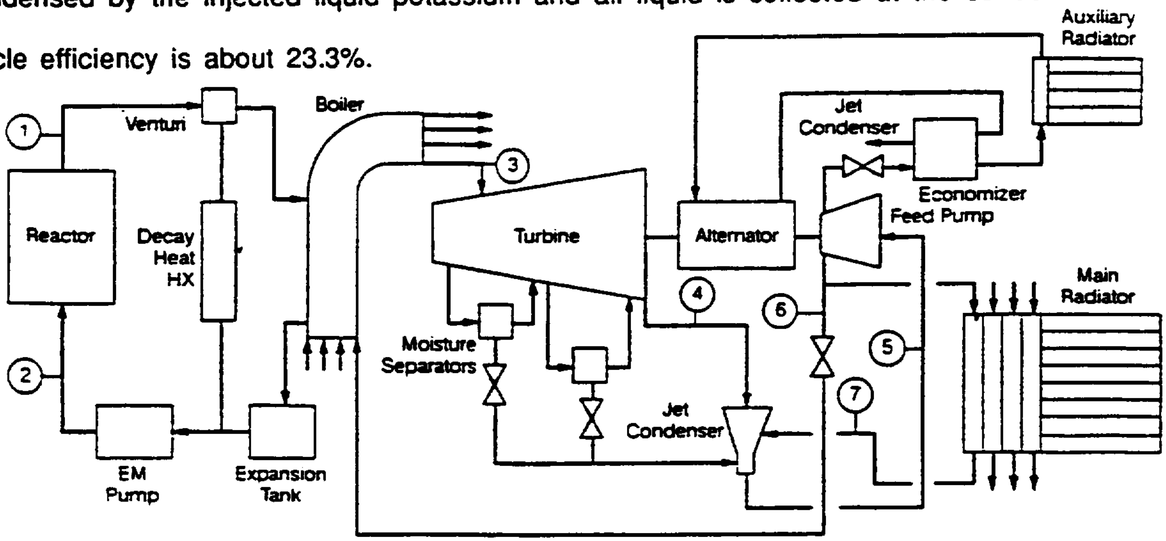


State Point	Fluid	Temperature (K)
1	Li	1355
2	Li	1276
3	Li	1304
4	Li	1225
5	NaK	625
6	NaK	552
T_h	—	1285
T_c	—	602

5953-7

Figure H-2. 550 kWe Stirling cycle power system schematic.

The flow schematic and nominal operating state points for the potassium Rankine system are shown in Figure H-3. The power system contains a lithium cooled primary loop and a potassium power conversion loop. The primary heat transport loop includes the reactor, potassium boiler, EM-pump, expansion tank, flow control venturi, decay heat removal system,and interconnecting piping. All primary loop components are electrically trace heated to provide for controlled lithium thaw during start-up. The boiler supplies dry saturated potassium vapor to the turbines. The power conversion unit consists of a turbine, alternator and boiler feed pump mounted on a common shaft. Moisture separators are used in the turbine to maintain vapor quality above 89% to minimize turbine erosion. A salient pole alternator is used to minimize mass. The heat rejection subsystem is made up of the main cycle radiator as well as an auxiliary radiator for alternator and bearing coolant. The jet condenser diverts a major portion of the potassium feed from the feed pump outlet to the main radiator manifolds then back to the jet condenser where it is injected. The low pressure turbine exhaust is condensed by the injected liquid potassium and all liquid is collected at the condenser throat. Cycle efficiency is about 23.3%.



State Point	Fluid	Temperature (K)	Pressure (kPa)
1	Li	1355	-
2	Li	1276	-
3	K	1275	629
4	K	850	13
5	K	844	227
6	K	845	649
7	K	744	611

Figure H-3. Potassium Rankine cycle power system schematic.

The overall deployed arrangements for the 550 kWe lunar surface Stirling, Brayton, and potassium Rankine systems are shown in Figure H-4. The radiators are arranged in a vertical cruciform configuration with the cooling headers located at the bottom of the radiator panels. Because of the small radiator required for the potassium Rankine system, a cruciform configuration was not required. Vertical reflux condenser tubes are connected to the header and reject the waste heat to space. The guard vessel radiators are located between the main radiator panels.

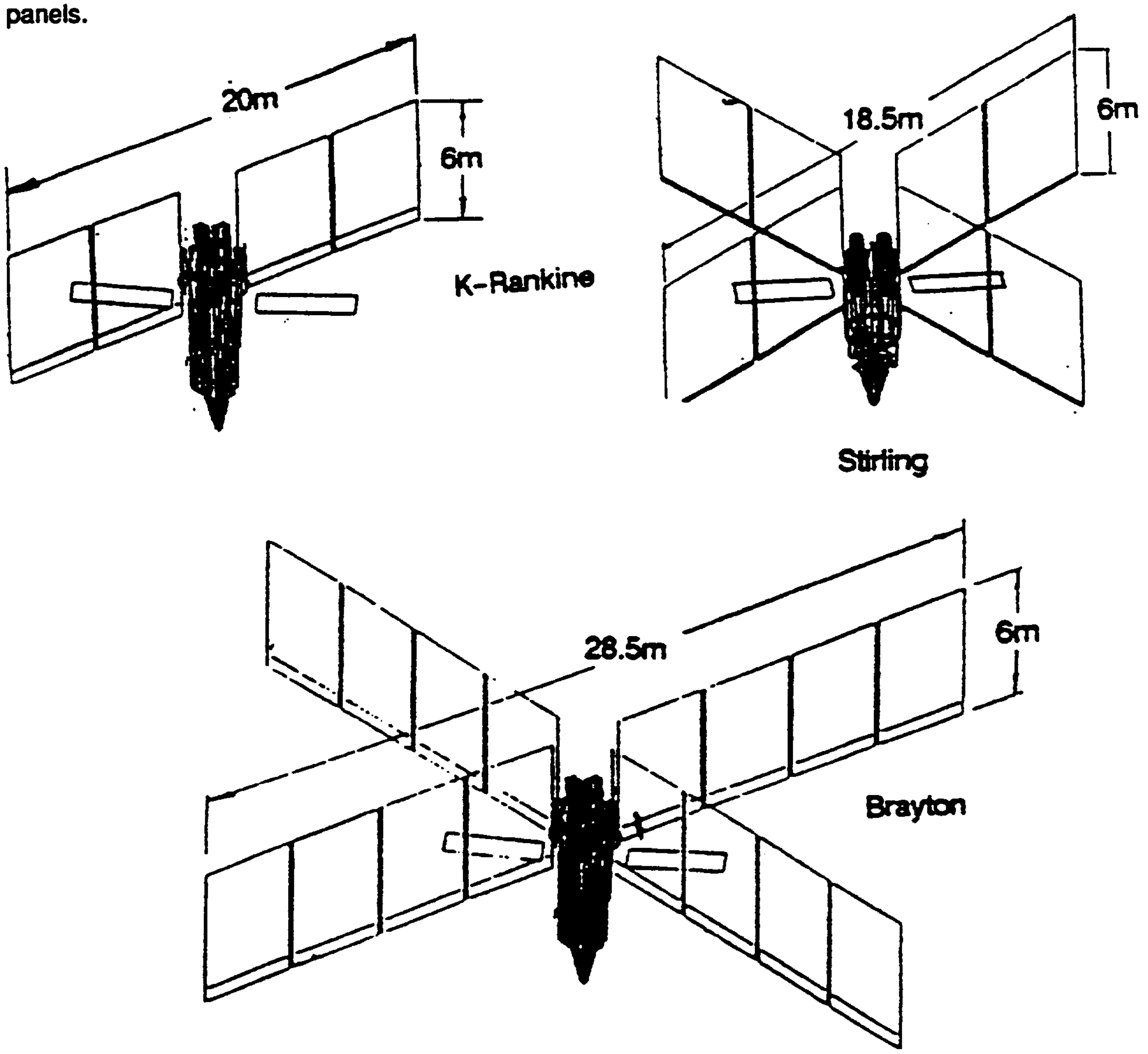


Figure H-4. 550 kWe lunar surface power system configurations.

In each case the entire primary loop is contained within a stainless steel guard vessel. The vessel is closely fitted to the reactor in order to prevent uncovering of the reactor should a breach in the lithium primary loop occur. The vessel is borated in the vicinity of the reactor to minimize the effect on reactor control from back scattered neutrons. The guard vessel is passively cooled by reflux condenser pipes bonded to the outer surface. Heat generated in the vessel is dissipated in a separate radiator located above grade.

In the case of a buried reactor, a regolith shield is located between the primary loop equipment gallery and the power conversion equipment. The shield tank is filled with the local regolith during installation and functions as a shutdown shield. A typical buried power system installation is shown in Figure H-5. The power system is placed in an excavated pit and then back filled. The regolith shield tank is filled with regolith and vibration compacted. The radiator panels are extended and electrical connections are made before the system is ready for startup.

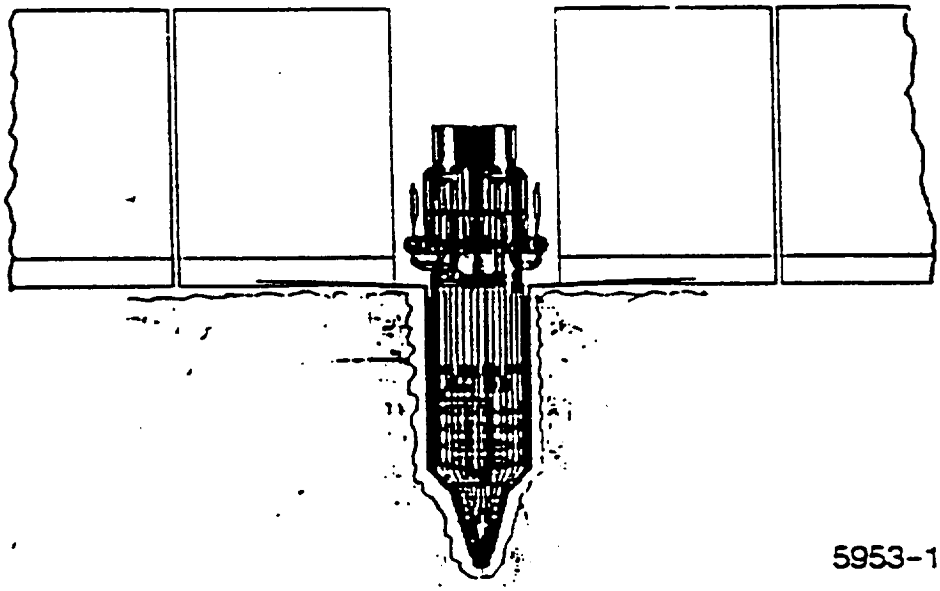


Figure H-5. - Buried 550 kWe lunar surface power system installation.

Thermal transport (from the reactor to the Power Conversion Unit [PCU] and from the PCU to the radiator) is accomplished by liquid metal coolant that is pumped by electromagnetic (EM) pumps. A pair of supply and return lines feed the liquid to thermopile heat exchanger located in each PCU. The supply and return lines are connected to common supply and return headers located within the reactor subsystem. The EM pump is located between the common return header and the reactor vessel. The liquid metal is in a frozen condition until system startup. Thawout of the frozen lithium lines is accomplished using electrical trace heaters.

Reject heat from the power conversion systems are transported to the radiator panels by a NaK pumped loop. The radiator panels use carbon-carbon heat pipes with potassium as the working fluid and a thin metal niobium liner to prevent contact between the potassium and the carbon-carbon structure.

The power conditioning electronics and shunt dissipators are located on a remote power processing module. The shunt regulator provides voltage regulation on the nominal buss output and is capable of load following without restriction as long as loads do not exceed the power generation capability.

KEY ISSUES

The key systems issues are summarized in Table H-1. Due to prior development of the SP-100 TE system, the development costs and risks should be relatively low for dynamic SP-100 systems.

TABLE H-1. KEY ISSUES, IMPACTS, AND DEVELOPMENT AREAS

Issues	Impacts	Potential Development Areas
Safety of nuclear systems during operation	•Increase in system mass for shielding - may be especially significant if in-situ materials are not used	•Use of in-situ materials for shielding or 4Π shield designs
Lunar/Mars Environment	•Refractory metal lifetime •Increased system mass	•Coatings, getters, liners, dust protection

TECHNOLOGY ASSESSMENT

The technology bases were assessed for the following major Dynamic SP-100 assemblies:

- SP-100 space power reactor;
- power conversion devices;
- radiator; and
- power processing and control system (PP&C).

The current technology readiness of each assembly and the estimated technology readiness at program start (year 2001 or beyond) were evaluated using the NASA technology readiness levels. It was assumed that the SP-100 TE power system and several key components (1300 °K SC PCU, carbon-carbon radiator, PP&C) would be developed to at least level 6 prior to program start. These evaluations are summarized in Table H-2. The technology base for each assembly is briefly discussed in the following sections.

TABLE H-2. DYNAMIC SP-100 TECHNOLOGY ASSESSMENT

Assembly	Current Technology Readiness Levels	Program Start Technology Readiness Levels	Comments
SP-100 Reactor	3	6	Currently funded development program
PCU (1300 °K)	4 / 3 / 3*	4 / 6 / 3*	CBC-good previous experience, DIPS; SC-Rapid progress on current NASA programs; PRC-no currently active development; 1300 °K materials proven
Radiator	4 / 3**	6 (C-C)	Currently under development
PP&C	4	6	SP-100 program; Space Station technology

*Brayton cycle/Stirling cycle/potassium Rankine cycle.
**Metal heat pipe/carbon-carbon heat pipe.

SP-100 Space Reactor State-of-the-Art

The SP-100 power reactor is actively being developed by the General Electric corporation under a joint DOD/DOE/NASA program. Considerable progress has been made in the development of the fuel pins and reactor components.

A fuel pin irradiation testing program has demonstrated the capability of the generic flight system fuel pin to meet performance requirements that call for full power operation to about 6.0 atom percent burnup for seven years. Irradiation tests have included UN fuel pins with Nb-1Zr cladding the W or Re liners, Re liners bonded to Nb-1Zr cladding, and Re cladding. The accelerated tests (fluences and burnup levels are achieved in a fraction of the actual service durations) will provide the information required for formulating the analytical models and validating the fuel pin design.

Nuclear assembly fabrication development was investigated and a process to extrude nozzles in the inlet/outlet nozzle course portion of the reactor vessel. A demonstration nozzle has been formed to the correct dimensions by the extrusion process at room temperature.

Nb-1Zr weld specimens have been exposed to lithium to evaluate the effect of welding atmosphere purity and post weld heat treatment on the sensitivity to lithium attack at grain boundaries. Results of the tests indicate that the welds containing bulk oxygen contents greater than 176 ppm experienced lithium attack at the Nb-1Zr grain boundaries. No attack was observed for bulk oxygen levels less than 140 ppm. The results of this work indicate that if low oxygen levels are maintained during welding, the potential for lithium attack is essentially eliminated.

Nondestructive examination methods continued to be developed and improved with most efforts being concentrated on support of the barrier fuel program. Production techniques for nondestructive testing of bonded fuel tubes were developed. A microfocus rod anode X-ray system was used in conjunction with ultrasonic testing to evaluate the detection of defects in various weld configurations.

A reactor flow test was completed including test article fabrication, test planning, and the completion of a comprehensive series of tests. Results of the test program provide important fuel assembly hydraulic design correlations as well as a validated orifice sizing procedure. The technology readiness level of the SP-100 reactor assembly is judged to be at level 3. Sufficient background work has been done to enable the fabrication and testing of a breadboard model to be started.

Potassium Rankine PCU State-of-the-Art.

In the 1960s, a large amount of work was done on Rankine PCSs for space power that used potassium for the working fluid. By 1970, the level of work had been greatly reduced. In terms of high temperature test hours, the largest contributors to boiling potassium research and development were AirResearch, General Electric, ORNL and Pratt & Whitney.

Table H-3 lists these organizations and summarizes the hours of operation on various systems and components for both potassium and cesium. Subtotals for the boiling corrosion subsystems, component test boiling subsystems, simulated Rankine cycle power plant subsystems, and the total operating time for all boiling subsystems are presented. The dimensions of all the components in the simulated subsystems were proportioned to provide operating characteristics and responses similar to those expected in the operation of a Rankine cycle space power plant. This is the major distinguishing feature between the simulated and the component test boiling subsystems. The total operating time for each component operated in these subsystems is also provided. Note that the total amount of operating experience at each installation is smaller than the sum of the operating concurrently for all of the components because two or more components were operated in each subsystem.

The major attraction of the potassium Rankine PCS is that it is compatible with higher source temperatures. This potential provides the higher Carnot efficiency and higher overall efficiency since the device efficiency can be expected to be about 40% of Carnot.

Several potassium Rankine system conceptual designs exist but none have progressed beyond the drawing board stage. The technology rating of potassium Rankine power conversion is judged to be approximately 3.0.

TABLE H-3.OPERATING TIMES WITH POTASSIUM AND CESIUM POWER CONVERSION SUBSYSTEMS AND COMPONENTS AT TEMPERATURES ABOVE 810°K

	Potassium Systems									Cesium Systems			
	AGN[a]	AS[b]	Allison[c]	GE[d]	JPL[e]	NASA[f]	ORNL[g]	P&WA[h]	RD[i]	UNC[j]	BNL[k]	AGN[a]	WA[m]
Corrosion Test Systems													
Boiling systems													
Thermal convection				10,500			43,600	12,000					420
Forced convection													
One loop		1,300					14,200				1,100	2,000	
Two loops				5,000			5,000						
Subtotal		1,300		15,500			62,800	12,000			1,100	2,000	420
All liquid systems													
Thermal convection										100,000			
Forced convection			53,000							3,000			
Component Test Systems (Boiling)													
One loop	100	5,000		11,900			2,800	3,600	200			2,500	
Two loops		900		7,700		400		1,300					
Three loops						600							
Subtotal	100	5,900		19,600		1,000	2,800	4,900	200			2,500	
Simulated Power Plant Systems													
One loop							10,200						
Two loops					1,000								
Subtotal					1,000		10,200						
Total for all boiling systems	100	7,200		35,100	1,000	1,000	75,800	16,900	200		1,100	4,500	420
Component - Power Plant													
Boilers													
<35 kW		1,300			1,000		71,400	16,900					
>35 kW	100	5,900		15,500		1,000	4,400		200		1,100	4,500	420
Subtotal	100	7,200		19,600	1,000	1,000	75,800	16,900	200		1,100	4,500	420
Turbines				35,100									
K lubricating bearings, <10 kW		3,000[n]					5,000[n]		100				
K lubricating bearings, >10 kW		50[n]											
Oil lubricating bearings, >10 kW													
Boiler feed pumps				5,100[n]									
Electromagnetic	100	7,200			3,600	1,000	26,900	4,900	200	3,000	1,100	4,500	
Turbine-driven centrifugal				24,600			5,000[n]						
Radiator						600							
Condenser - liquid metal loop						600	5,000						
Condenser - air cooled		5,900		2,500			43,600	4,900	200				
Condenser - radiator (combined)	100	1,300		16,100	1,000	400	27,200	12,000			1,100	4,500	420
Subtotal	100	7,200		16,500	1,000	1,600	75,800	16,900	200		1,100	4,500	420
Electrical generator				35,100									
Operation in potassium vapor													
External load device		50											
Pumps in all-liquid systems		5,900	53,000		3,600	1,600	6,100	1,300	1,600	3,000			
Potassium seals		3,050[n]		18,600									
Seal test				5,100[n]									
Bearing test rig				300									
Instrumented		3,000[n]							1,600				
Single bearing endurance test		300					4,500						

[a] Aerojet-General Nucleonics; San Ramon, CA.

[b] Allied Signal; Phoenix, AZ and Los Angeles, CA.

[c] Allison Division of General Motors; Indianapolis, IN.

[d] General Electric Company; Cincinnati, OH.

[e] Jet Propulsion Laboratory; Pasadena, CA.

[f] National Aeronautics and Space Administration; Cleveland, OH.

[g] Oak Ridge National Laboratory; Oak Ridge, TN.

[h] Pratt and Whitney Aircraft (CANEL); Middletown, CT.

[i] Rocketdyne; Canoga Park, CA.

[j] United Nuclear; White Plains, NY.

[k] Brookhaven National Laboratory; Upton, Long Island, NY.

[m] Westinghouse Astronuclear; Large, PA.

[n] The time shown for boiler feed pump, potassium seal and test bearing was accumulated during operation of the turbine indicated in that column.

Brayton Cycle PCU State-of-the-Art.

The closed cycle gas turbine Brayton PCU is adaptable to a wide range of applications and power levels. Practical competitive systems have been designed and/or built for space, terrestrial, marine, and underwater environments with power levels ranging from 500 watts to more than 500 Megawatts. Input thermal energy can be from solar, nuclear, and fossil or chemical combustion with waste heat rejected to the environment by a wide variety of methods. The major reason for interest in the closed Brayton cycle is its high demonstrated efficiency, simplicity and mature technology. Commercial combined cycle machines have been constructed with overall cycle efficiencies of 50%. Low temperature space power units (e.g., DIPS) have cycle efficiencies in the 20 to 25% range. Selection of the system operating points is usually done on the basis of minimizing system mass while observing sensible design limits for the different components.

During the late 1960s and 1970s, NASA conducted several near-prototypical development programs using Brayton rotating machinery and accessories. These included the Mini-BRU, the CRU and the BRU. Four "B" engines were built using BRUs with 1144 °K TIT capability were developed under NASA contract by Garrett Corporation (now Allied Signal) and sent to NASA LeRC for performance and endurance testing. The purpose of these tests was to demonstrate the technology for space application of the closed-cycle Brayton engine. One of these engines accumulated 38,000 hrs of successful operation, while achieving cycle efficiencies of 30 to 33%. The two units tested met or exceeded performance objectives with no detectable performance degradation during endurance testing.

Although low temperature CBC PCUs are well developed (TRL of 5), high temperature CBC PCUs are yet to be built. Existing refractory-metal alloys are available that will be suitable for high temperature operation (though they must be protected from the Martian environment). PWC-11 (Nb-1Zr-0.1C) is now being exploited under the SP-100 program (Ref. H-1). JPL and Los Alamos National Laboratory will operate PWC-11 as reactor-fuel clad at 1400 °K. The U.S. has a great deal of experience with Nb-base refractory-metal alloys. This includes use of Nb-1Zr in the Brayton solar-heat receiver to C-103 (Nb-10Hf-1Ti-0.72r) in both the Heat Source Assembly and the turbine for the Mini-Brayton powerplant. Thus, the 1300 °K CBC PCU was given a technology rating of 4 (Refs. H-1 and H-2).

Stirling Cycle PCU State-of-the-Art.

Major interest for space based power systems has been centered on the free piston Stirling engine (FPSE). The FPSE concept was first invented in the U.S. in 1963. NASA has been sponsoring an extensive test program of several free piston Stirling engines with linear alternators built by Mechanical Technologies Incorporated (MTI). The NASA Stirling Space Power Converter Program originated in 1984 as part of the SP-100 program. A summary of the test times accumulated on these engines is given in Table H-4. One engine, the EM-2 has accumulated over 5000 hrs in tests at MTI.

The currently funded NASA FPSE program will result in a space capable 1050 °K FPSE by 1996 (Ref. H-3). The ultimate goal of the space FPSE program is to develop the technologies for a refractory metal Stirling power converter with a hot end temperature of 1300 °K and a cold end temperature of 650 °K. The NASA approach is to take the 1050 °K superalloy technology and to evolve into 1300 °K technology by direct substitution of refractory materials. The Materials Division at NASA LeRC has substantial experience in the application of refractory materials and has been developing materials for use in the hot components of the SP-100 reactor. The Materials Division has compiled a list of refractory material candidates as seen in Table H-5. The LeRC ratings for each alloy as to joinability, fabricability, availability, and data were rated by NASA on a scale of 0 to 10 with 10 being the best. A NASA contract with MTI called for the refractory design to be carried through the conceptual design phase during FY'91. Funding for full scale development of the 1300 °K machine will be included under the Exploration Technology Program, scheduled for initiation in 1993 (Ref. H-3).

The success of this technology depends upon supporting research and technology efforts including heat pipes, bearings, refractory metal joining technologies, high efficiency alternators, life and reliability testing, and predictive methodologies. Based on the FPSE experience base and the NASA plans for the high temperature FPSE, the 1300 °K FPSE is given a technology readiness rating of 3.0 (the 1050 °K machine has a rating of 4.0 per Ref. H-4).

TABLE H-4. FREE PISTON STIRLING ENGINE ACCUMULATED TEST TIMES

Engine	Test Hours
RE-1000	280 (NASA)
EM-2	5385 (MTI)
SPDE	253 (MTI)
SPRE-I	349 (NASA), 74 (MTI)
SPRE-II	333 (MTI)

TABLE H-5. REFRACTORY METAL CANDIDATES FOR 1300 °K STIRLING

Base Material	Alloy Name	Composition (wt%)	Melting Point (°K)	Density (kg/m^3)	Join-ability	Fabric-ability	Alloy Avail-ability	Data Avail-ability
W	W-25Re-HfC	24-26% Re 1% HfC	1380	19,300	5	4	4	3
Ta	ASTAR-811C	8% W 1% Re 1% HfC	3270	16,600	8	8	10	5
Mo	TZM	0.08% Zr 0.5% Ti	2880	10,200	2	8	10	4
	TZC	1.25% Ti 0.1% Zr 0.15% C	2880	10,200	2	6	10	4
Mo/Re	Mo-47.5Re	47.5% Re bal. Mo	2780	15,500	8	6	8	3
Nb	FS-85	11% W 28% Ta 1% Zr	2740	8,600	8	8	5	4
	B-88	27% W 2% HfC	2740	8,600	7	7	4	2
	C-103	10% Hf 1% Ti 0.7% Zr	2740	8,600	10	10	10	7
	PWC-11	1% Zr, 0.1% C	2740	8,600	10	10	10	7
	Nb-1Zr	1% Zr	2740	8,600	10	10	10	8

Radiator State-of-the-Art

Presently, Rocketdyne division of Rockwell International is involved in the development of a carbon-carbon (C-C) or graphite heat pipe radiator panel. Rocketdyne has a NASA contract entitled "SP-100 Advanced Radiator Concepts" which started in 1987 and is part of the CSTI (Civilian Space Technology Initiative) program. This program will be completed in January of 1993. This program involves development of a C-C heat pipe suitable for use in an SP-100 radiator. However, this technology will be suitable for other radiator applications as well including dynamic power systems. The hope is that this program will develop generic technology which can be used for all future space radiators because of its reduced mass compared to metal heat pipe radiators. Carbon-carbon is a new structural material which requires different fabrication techniques than for metals or composites. A key objective of this program is to demonstrate the ability to fabricate C-C heat pipes with a thin metal liner, wicks, caps, and fill tube. The brazing technique required to attach liners is the same as would be required to attach manifolds to the heat pipes. Heat pipes filled with potassium will be designed, fabricated, and tested during this program. Thermal cycle testing will be done in a vacuum to demonstrate the integrity of the heat pipes.

The current technology level of the radiator subsystem is estimated to be about 4 for the metal heat pipe radiators and about 3 for carbon-carbon heat pipe radiators.

PP&C State-of-the-Art

The electronic component technology required for the PP&C equipment is being developed for future power system applications under the SP-100 and Space Station Freedom Electrical Power Subsystem (SSF EPS) programs. Currently available analytical and design technologies will be used along with radiation hardness testing. Radiation hardened high reliability Class S parts will ensure that a reliable and low risk power processing and control system is provided. Rockwell is currently developing the SSF EPS, and this technology is directly transferable to Dynamic SP-100 lunar and Mars surface power systems concepts. The use of multiple buses, components for the switching of loads, and switching of subassemblies within the power assembly are technologies common to the two programs.

Several active efforts are underway to develop expert system techniques for real-time control applications. These include a design for the evaluation of Space Shuttle Main Engine (SSME) performance, sensors and data processing for fault detection, and the electrical load power allocations in the SSF electrical power subsystem. The algorithms for power scheduling and balancing on the space station are in development. Expert system software for the SSF electrical power subsystem is being developed using the ADA environment. This work includes implementing and evaluating an ADA based inference engine, which will enable an expert system to run on standard microprocessors for SSF applications.

The required technology for the control and data acquisition functions currently exists, and the technology for autonomous control is under active development for Space Station Freedom (SSF), ESSA and other space power programs. The need for radiation hardened components with these capabilities remains to be demonstrated, which results in a technology rating of 4.0 for the PP&C subsystem.

MAJOR DEVELOPMENT TASKS

The technology level of Dynamic SP-100 components is expected to range from 3 to 6 at the beginning of full scale development. Most of the remaining work will be in system integration, qualification, and acceptance testing. The development and testing currently envisioned is briefly described in the following sections.

Task 1. Intermediate Heat Exchanger (Integration of Reactor and PCU)

Objectives: Develop an interface between the heat source and PCU to allow heat transport.

Statement of Work: Develop and test an intermediate heat exchanger for transferring heat from the reactor cooling loop to the PCU.

Task 2. Power Conversion Unit Development

Objectives: Complete development of power conversion devices which meet the desired performance and life requirements.

Statement of Work: The following task descriptions apply, depending on the PCU selected:

Potassium Rankine PCU. A significant amount of the enabling technology for potassium Rankine power conversion exists, but particular components have not been designed in any detail for many years. Test loops, materials specifications and performance parameters for virtually all potassium Rankine power system components will require development.

Brayton Cycle PCU. Develop a 1300 °K refractory alloy heat engine based on the CBC PCU for the Dynamic Isotope Power System.

Task 3. Radiator Integration With PCU

Objectives: Provide a means for removing heat from the PCU and transporting the heat to the main radiator.

Task Description: Design and test a heat exchanger and heat transport loop from the PCU to the radiator panels.

Task 4. Ground Engineering System (GES) Testing

Objectives: Complete the system design. Verify all performance characteristics of the unit in conjunction with subjecting the GES to MIL-STD-1540 acceptance level tests. Verify adequate system life.

Statement of Work: As a cost reduction measure it has been proposed that GES testing be dispensed with and that space reactor power programs proceed to the flight qualification phase after the component development needed to support final design has been completed. Acceptance of this approach by the technical community is not certain as of this writing, therefore, a discussion of GES testing is included as part of the overall development of a space reactor power system.

After assembly, the GES will be pressure and leak checked, as appropriate, a complete electrical check-out will be performed, and all active devices will be checked out to the extent possible.

Performance and thermal vacuum tests will be conducted during a single thermal vacuum test sequence to demonstrate technical capabilities while minimizing program cost. The performance evaluation will consist of the following major activities:

- Check out and refine subsystem assembly procedures;
- check out procedures for loading and filling the reactor coolant subsystem;
- verify operation of ground support equipment (GSE) and interfaces between GSE and the GES;
- check out ground cooling systems as required;
- verify start up capability;
- verify steady state performance characteristics;
- establish effective operating range of the system;
- check out software and verify autonomy (include all operational modes and simulate failure modes);
- simulate normal switchover from one PCU to another;
- simulate failures to trigger PCU switchover; and
- establish system sensitivity to off-normal conditions such as: (1) partial loss of radiator cooling, (2) varying radiator heat removal profile, (3) partial loss of electronic cooling, and (4) temporary losses of heat sink with varying duration.

The MIL-STD-1540 test sequence will integrate acceptance level and design margin tests as appropriate up to qualification levels, to provide a technically sound, minimum cost approach.

When the performance, acceptance, and margin tests are complete, the GES will be refurbished and placed on a multiyear life test. Instrumentation will be installed to provide a comprehensive diagnosis of the "health" of the GES and to monitor for degradation of major assemblies and individual components. The GES will be operated at its nominal operating point, with expected GES variations in power output and environment.

The GES will be disassembled for diagnosis at the end of the life test. Specific areas to be examined will be determined by an analysis of the health monitoring data and from the reliability analysis predictions.

Task 5. Qualification Program

Objectives: A comprehensive performance and dynamic testing program for assemblies and the complete system will provide a formal demonstration that the Dynamic SP-100 system will perform as designed after being subjected to simulated launch conditions. A low-risk approach, incorporating qualification of individual assemblies, followed by qualification of a complete system, termed the qualification unit (QU), characterizes this program.

Statement of Work: The required qualification tests for the assemblies of the Dynamic SP-100 system are those specified in Section 6.4, "Component Qualification Tests", of MIL-STD-1540B. For the QU, the required qualification tests are in Section 6.2, "Space Vehicle Qualification Tests", of MIL-STD-1540B. Qualification will be to launch environmental conditions selected to envelope the probable intensities developed by various launch scenarios.

The overall approach, starts with qualification of assemblies. Qualified production items are then fabricated and assembled into the QU which is then qualified by the rules for space vehicle qualification. -This approach minimizes the risk that some assembly of the QU will fail during the qualification test sequence, thereby nullifying the qualification, and requiring backtracking to recover from the failure.

Performance testing at each level will be conducted to verify that each item performs as designed. The qualification item will be similar to the corresponding engineering item and, therefore, performance testing will be less time consuming.

Dynamic testing will be performed per MIL-STD-1540B, to verify capability of the Dynamic SP-100 system to withstand launch loads, including acoustic, pyroshock and vibrational.

The performance and dynamic qualification test sequence for components and assemblies is shown in Figure H-6. Fabrication, assembly, checkout, and testing of the QU will be similar to the corresponding operations for the GES and therefore, will benefit from the experience with the GES. Test facilities for component and assembly qualification testing will be the same as those for the assembly level testing of the EU test program.

When qualification testing is successfully completed, the QU will be partially disassembled, examined, refurbished as required and modified for endurance testing in air, as described for the GES. The unit will be scheduled for life testing for nominally 1-1/2 years as a basis for estimating program costs. The period can be extended if required.

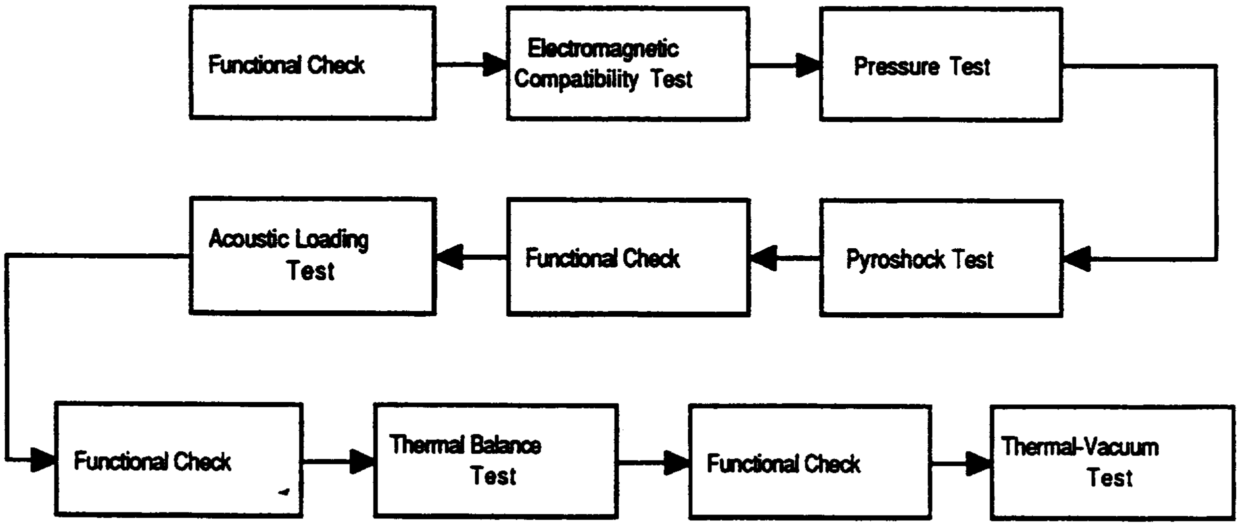


Figure H-6. QU test sequence (per MIL-STD-1540B).

Task 6. Flight System Program

Objectives: Produce, acceptance test, and deliver one or more flight systems.

Statement of Work: In the flight system programs, parts will be fabricated, acceptance tested, and assembled. Systems will be subjected to acceptance testing per MIL-STD-1540B guidelines before shipment to the launch site. The same test facilities (vacuum chamber, vibration, acoustic) that were used for the qualification program would be used for flight system acceptance testing. The flight phase of the program also will include the safety studies and reports necessary to obtain launch approval.

DEVELOPMENT SCHEDULES

The SP-100 program is currently funded but has been experiencing year by year funding reductions, which are resulting in significant schedule slippage. Resolution of the funding question will become more apparent with the identification of a specific mission. A significant amount of development work has been accomplished and the program is considered to be nearly ready to start preliminary design.

Figures H-7 through H-9 present development schedules for the Dynamic SP-100 system with a SC PCU, CBC PCU, and PRC PCU, respectively. This roadmap assumes prior development of the SP-100 TE system, a 1300 °K SC PCU, a carbon-carbon heat pipe radiator, and the PP&C. The different PCU systems will take different development times and will affect the overall system development schedule . The SC Dynamic SP-100 is expected to take the least time (9.5 years) due to prior component development. The potassium Rankine PCU is expected to require three more years (13.5 years) to develop than the CBC PCU (10.5 years). The development times do not include the time to modify the system for use in the Martian environment.

The GES will be similar to a flight system but will have features such as additional instrumentation and readily removable components to expedite gathering of engineering data and to permit modification of components. It will be tested in a vacuum chamber under normal and off-normal design conditions. After this phase of testing, it will be partially disassembled, examined, refurbished as necessary, and put on life test, nominally for three years.

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components, and a complete system. The design effort will be minimal since it would involve only minor modifications to the GES design.

The flight phase of the program includes fabrication, assembly, and acceptance testing of the flight units and the associated safety analysis to obtain launch approval.

DYNAMIC SP-100 POWER SYSTEM TECHNOLOGY ROADMAP

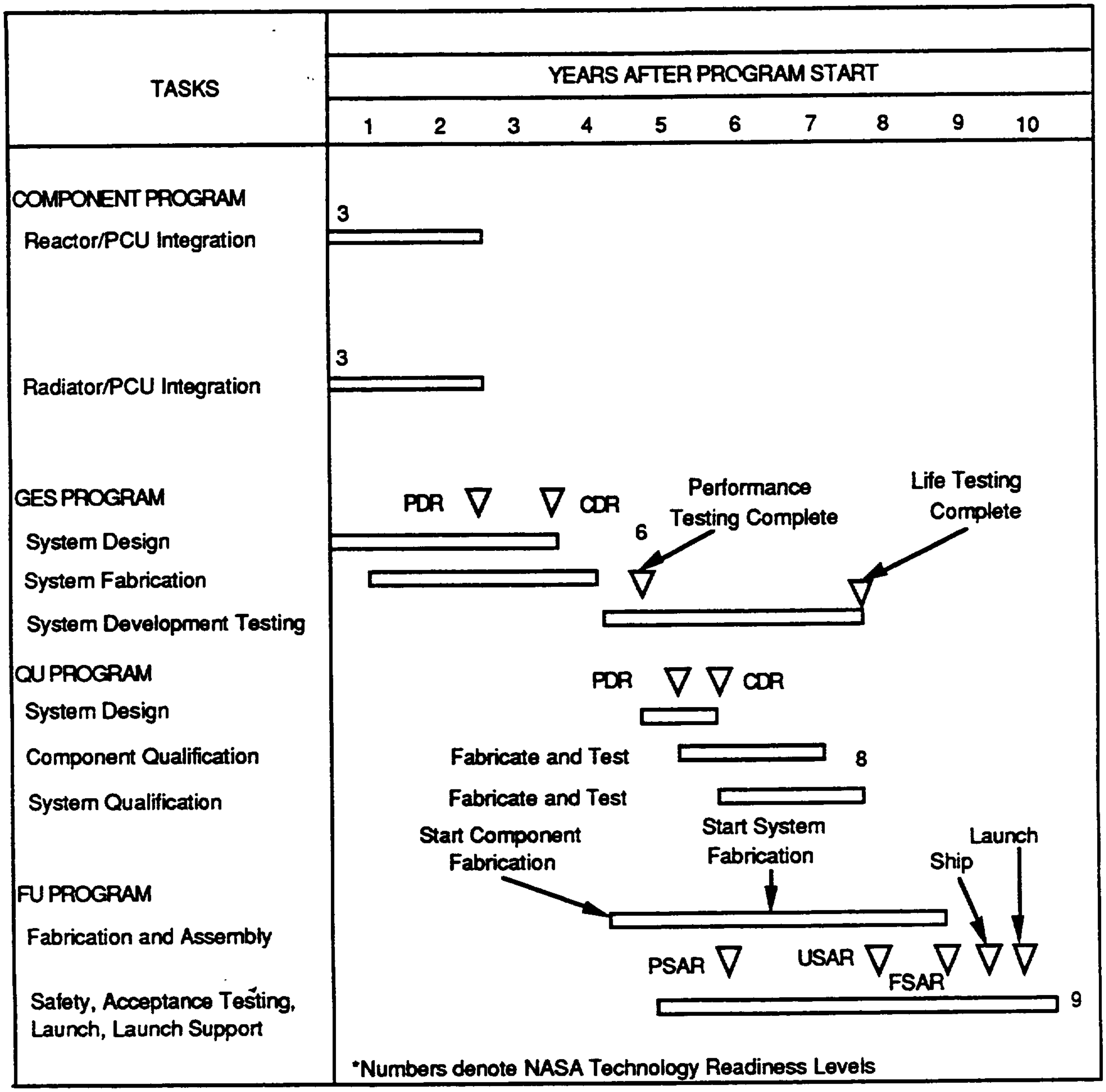


Figure H-7. SC dynamic SP-100 development schedule.

DYNAMIC SP-100 POWER SYSTEM TECHNOLOGY ROADMAP

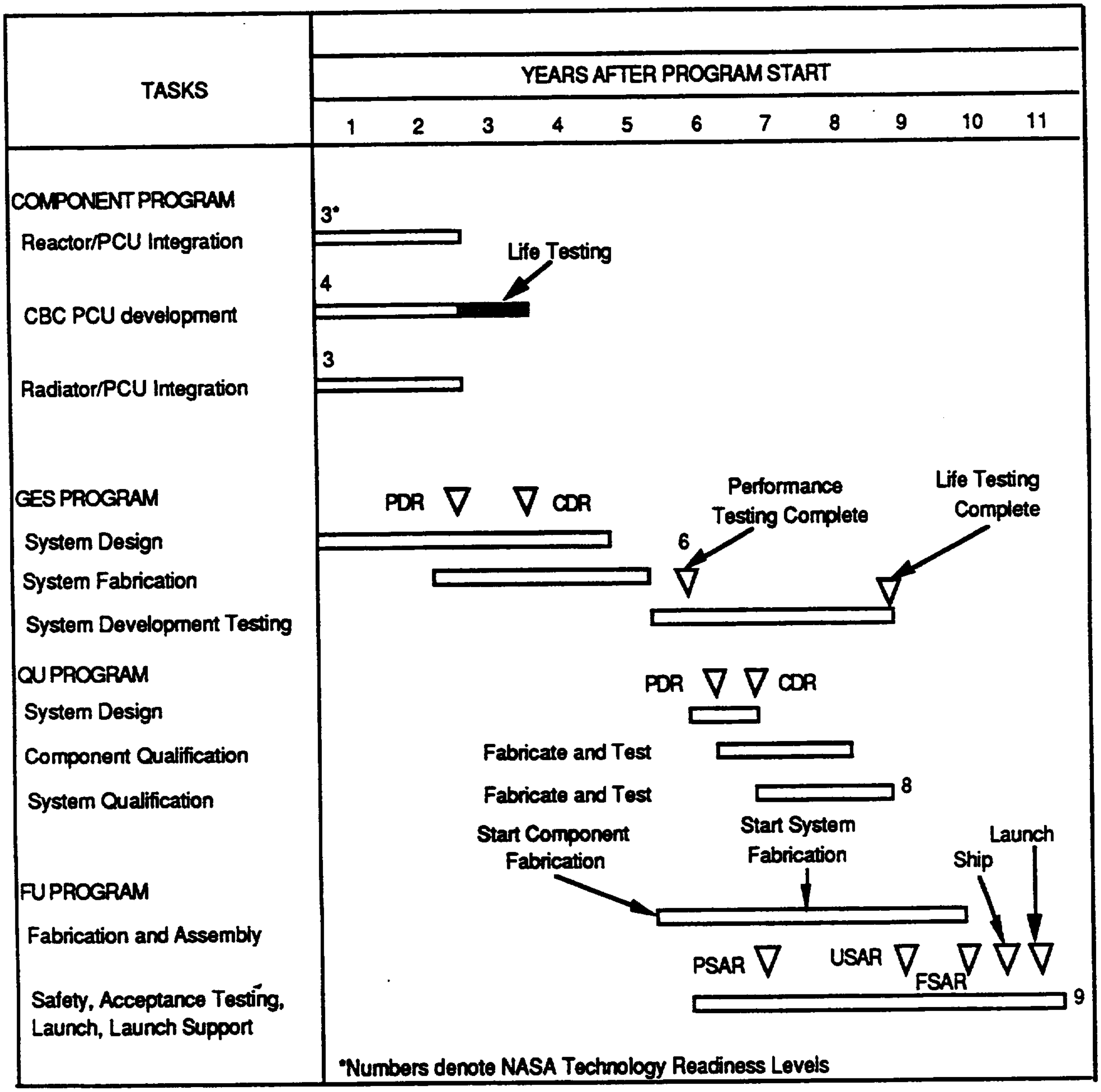


Figure H-8. CBC dynamic SP-100 development schedule.

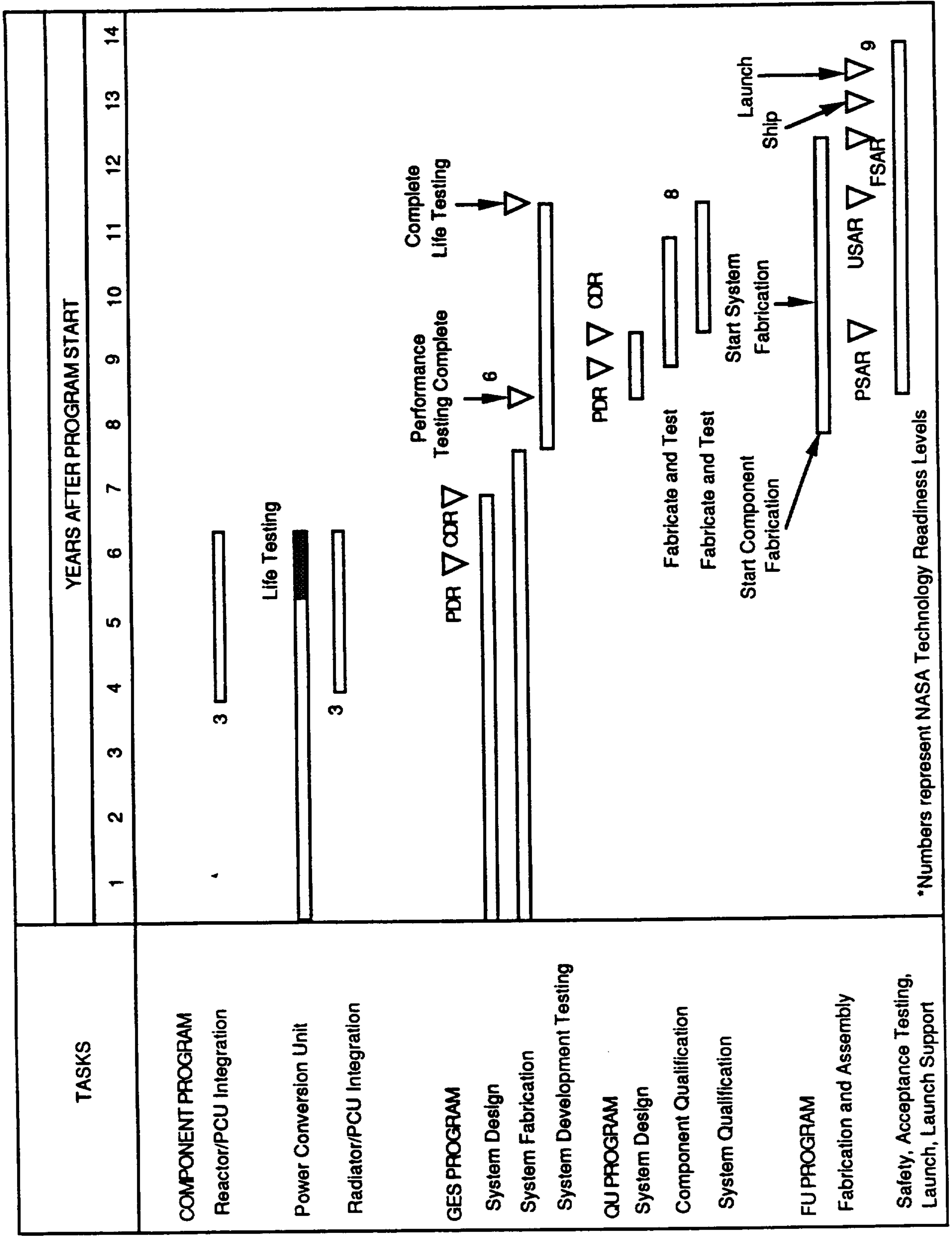


Figure H-9. PRC dynamic SP-100 development schedule.

C-4.

REFERENCES

H-1. English, R.: Rockwell Roadmap for Isotope-Brayton (DIPS). Letter to Martin Valgora (NASA LeRC), 4/21/92.

H-2. English, R.; Mason, L.: Technology Maturation Levels. Personal communications, 3/92.

H-3. Dudenhoefer, J.E.; and Winter, J.M.: Status of NASA's Stirling Space Program, 26th IECEC Proceedings, Volume 2, 8/91, pp. 38-43.

H-4. Dudenhoefer, J.E.: Stirling Space Engine Program Status. Personal communication, 3/4/92.

APPENDIX I

NEAR-TERM SOLAR DYNAMIC POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX I - NEAR-TERM SOLAR DYNAMIC POWER SYSTEM TECHNOLOGY ROADMAP

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APPENDIX I - NEARTERM SOLAR DYNAMIC POWER SYSTEM TECHNOLOGY ROADMAP

CONCEPT DESCRIPTION

A concept was selected for a nearterm solar dynamic (SD) system to provide power to manned space platforms in low Earth orbit (LEO) and other space missions after the year 2002. This concept was based on the work done for the Space Station Freedom (SSF) SD option (Ref. I-1). An SD space power system includes four basic subsystems: (1)the heat source; (2)the power conversion unit (PCU); (3)the heat rejection subsystem (HRS); and (4)the power processing and control (PP&C) subsystem. Figure I-1 shows a schematic of the Space Station SD system module (Ref. I-2).

The heat source includes a concentrator and receiver. The solar flux is captured and reflected into the receiver cavity by the concentrator. The receiver is both a heat exchanger for transferring concentrated solar heat to the heat engine working cycle and a device for storing excess thermal energy during the sunlit portion of the orbit. The energy storage device releases energy to the cycle working fluid during eclipse.

This power system includes a closed Brayton cycle (CBC) system for the conversion of the heat generated in the receiver to electrical power. A helium and xenon gas mixture is used as the CBC working fluid. The PCU consists of a turboalternator-compressor (TAC), a recuperator/cooler, and interconnecting ducting. The working gas is heated in the receiver tubes and flows to the turbine end of the TAC where it expands and performs work. Typically, the gas pressure decreases by a factor of 2 due to the expansion in the turbine. From the turbine, the gas passes through a recuperator where energy is transferred to cycle gas returning to the heat source. Recuperation improves efficiency and reduces the size of the radiator and concentrator.

From the recuperator, the gas flows to the heat rejection subsystem. A gas cooler transfers the cycle waste heat to a pumped liquid which circulates between the cooler and the radiator. The cycle waste heat is radiated to space from the radiator. From the heat rejection system, the cooled cycle gas (typically around 300 °K) returns to the TAC. Bleed gas is used for cooling the alternator and turbine bearings.

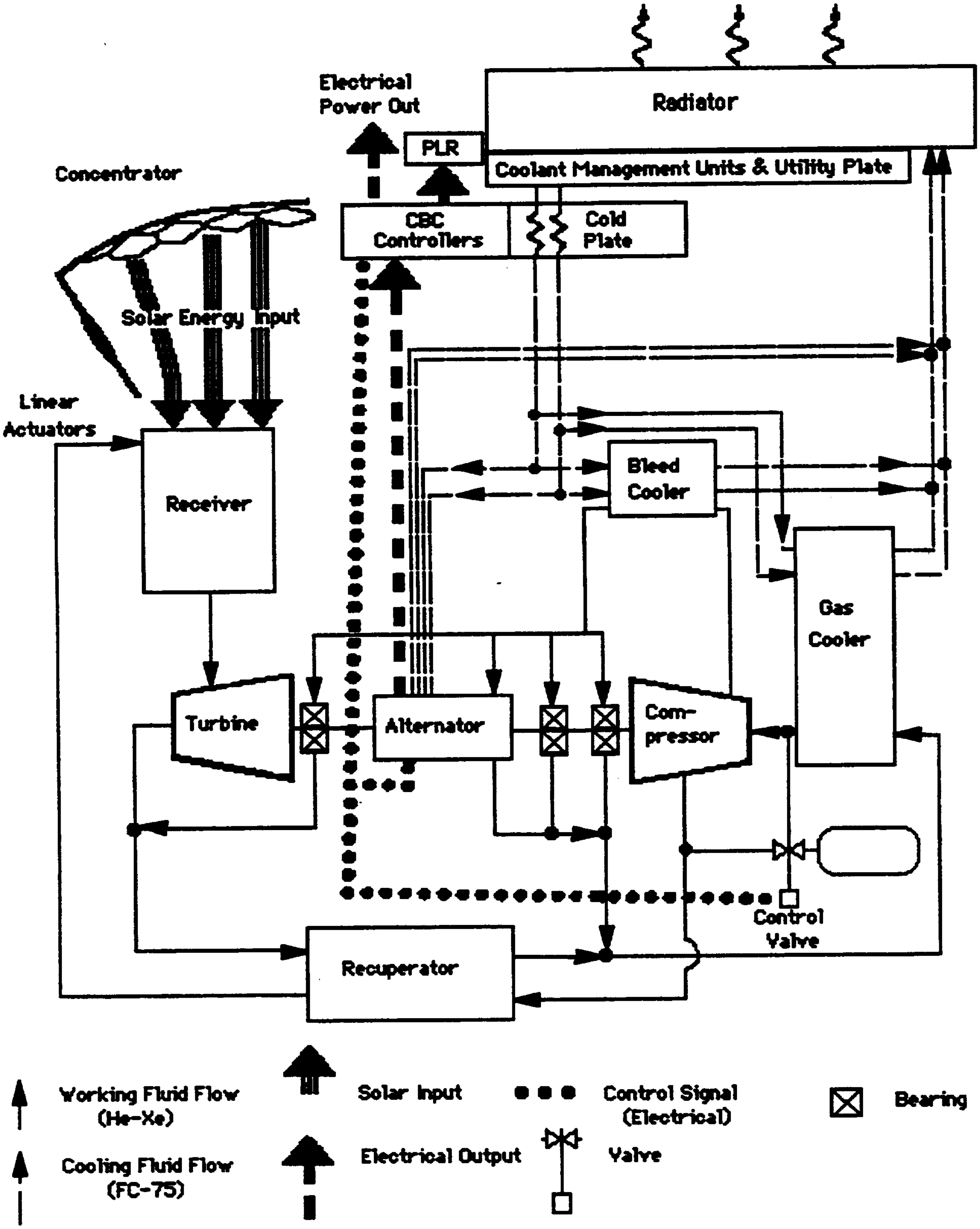


Figure I-1. CBC-SD power system schematic.

The power system is designed for LEO with a nominal 500 km altitude at 28.5 degrees inclination. The power system platform is assumed to be serviceable. Excess energy management (EEM) or coping with solar availability in excess of the design minimum value is not a major concern for this particular orbit.

It is expected that 25 kWe modules will be developed. The power system concept is based on current technologies for the concentrator, receiver, radiator, and power conversion technologies.

Solar Concentrator

An offset parabolic concentrator is used to capture the solar rays and focus them on the inner surface of the receiver. A truss hex concentrator is the baseline for the Space Station (Ref. I-1), although it was recognized that lower mass concepts are under development which may prove to be more appropriate for more weight-sensitive missions such as geosynchronous earth orbits (GEO). The Space Station CBC truss hex configuration is shown in Figure I-2 (Ref. I-1). The truss hex concept utilizes a heavier beam construction than other concepts. The concentrator surface is segmented for reasons of launch packaging and deployment.

Solar Receiver

As seen in Figure I-3 (Ref. I-1), the receiver design consists of a cylindrical cavity and a series of tubes running the length of the cavity. The CBC working fluid from the recuperator flows through an external duct to a toroidal manifold at the aperture end of the receiver. The manifold distributes the fluid to each tube. The flow is collected in the outlet manifold and is then sent to the turbine. The receiver is shown schematically in Figure I-4 (Ref. I-3).

A $LiF\text{-}CaF_2$ eutectic salt is a phase change material (PCM) used for thermal energy storage (TES). The PCM is contained in a series of sealed metallic containment canisters. The canisters are stacked and brazed to the working fluid tube as seen in Figure I-5 (Refs. I-2 and I-3). The canisters are not bonded to each other, but are separated by ceramic fiber spaces.

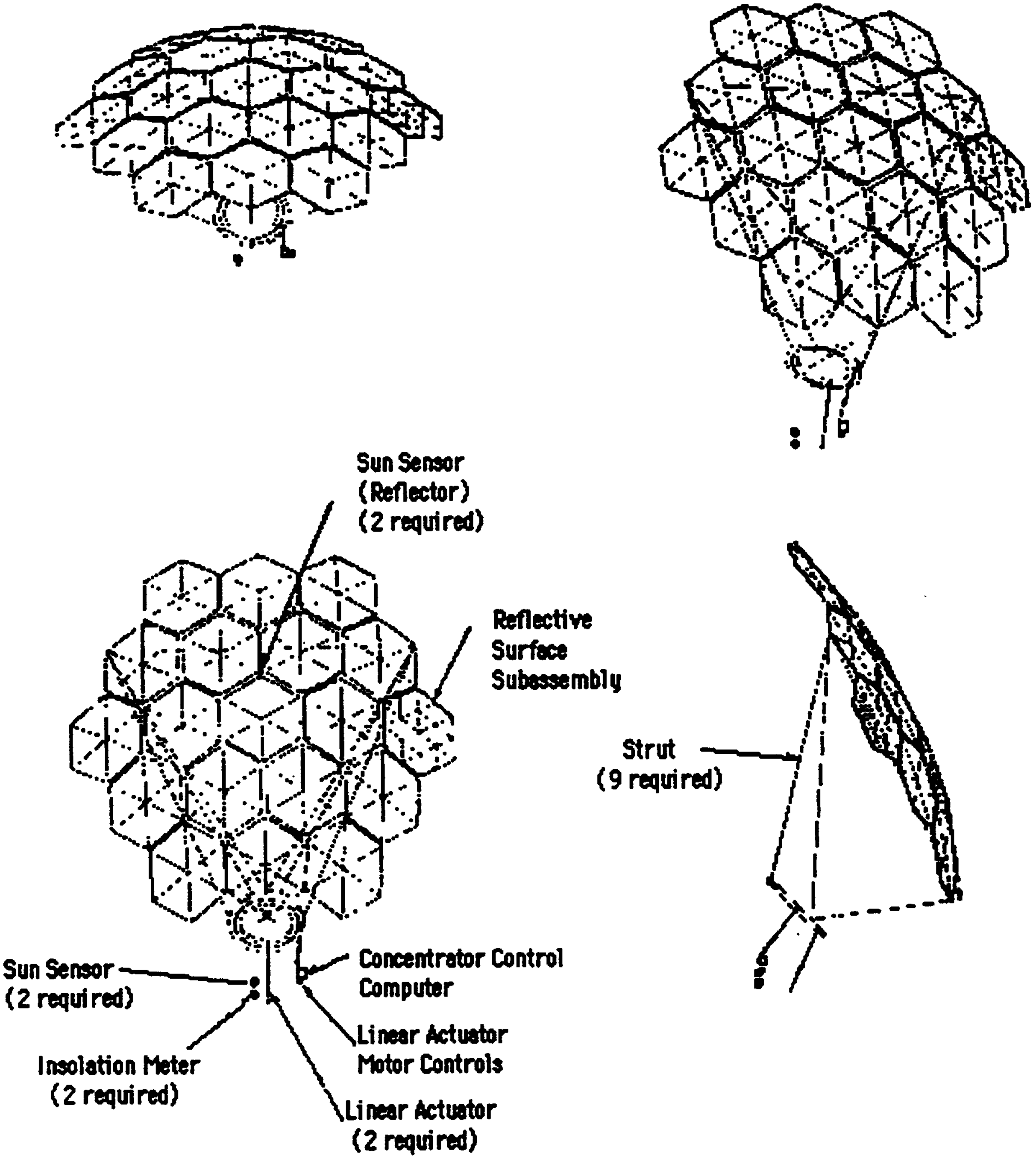


Figure I-2. Solar concentrator for SSF.

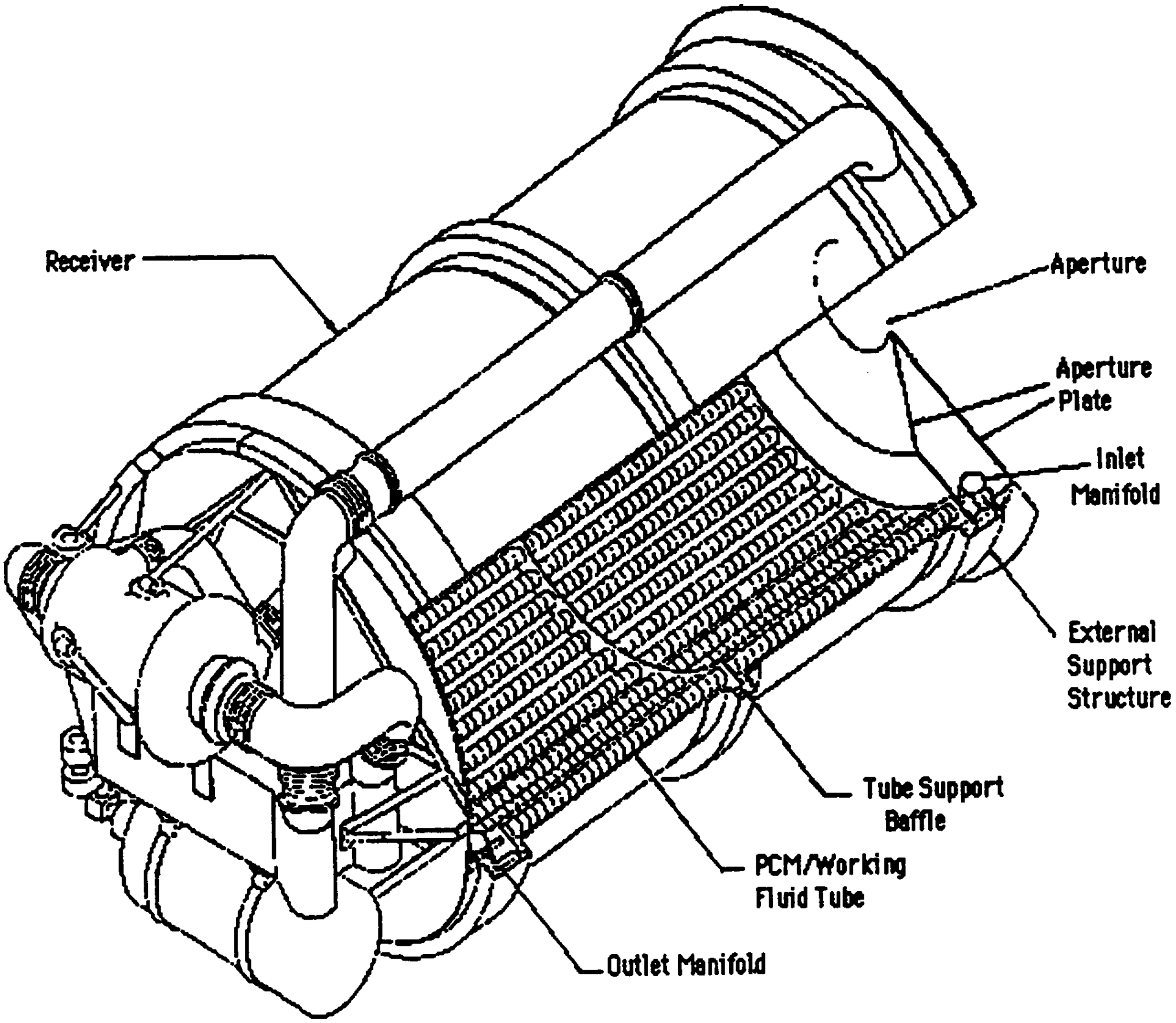


Figure I-3. Space Station Freedom CBC integrated solar receiver/PCU.

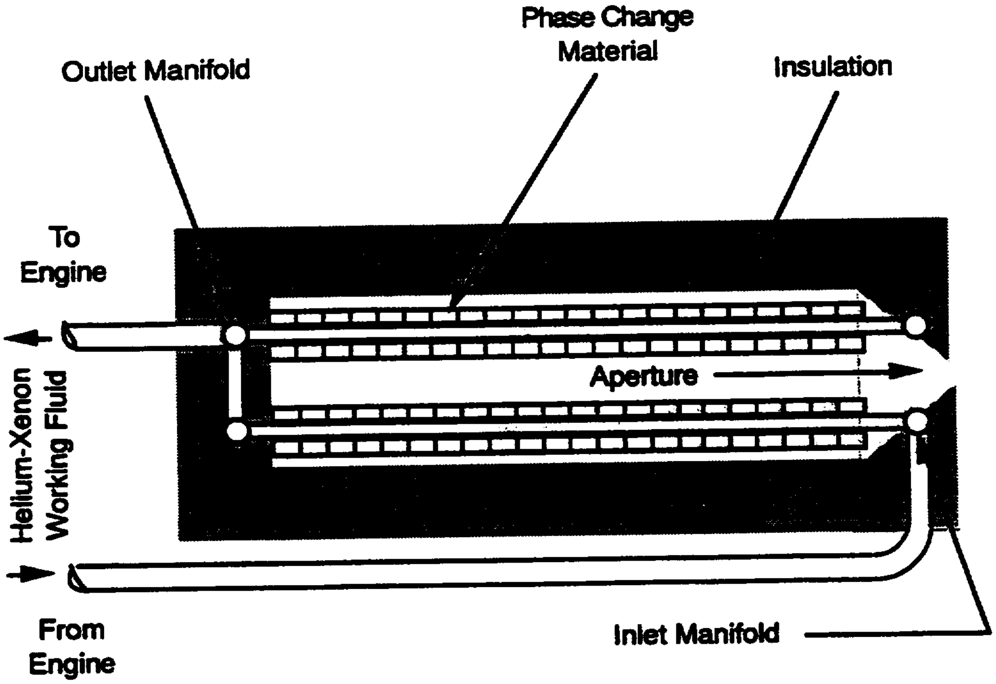


Figure I-4. Solar receiver schematic.

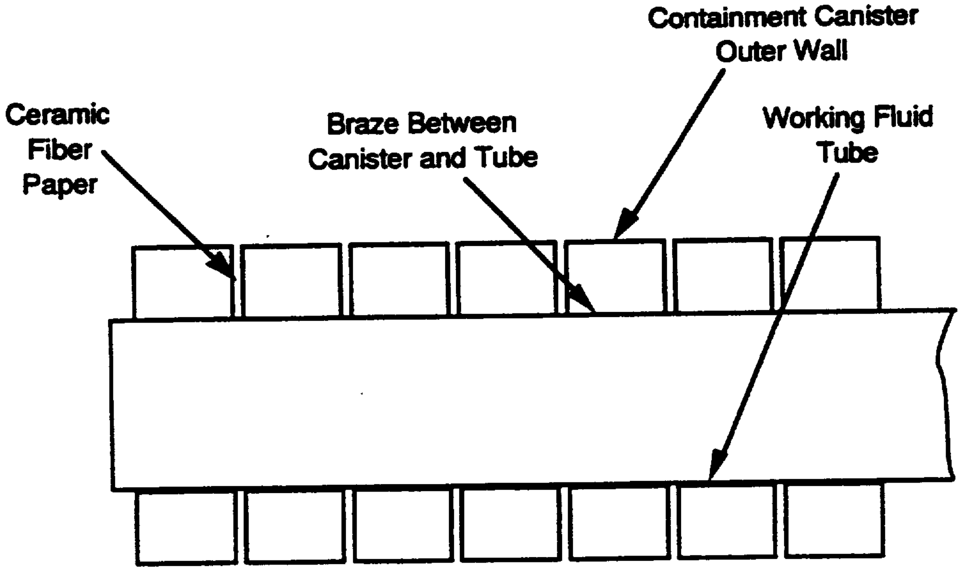


Figure I-5. Receiver tube configuration.

The use of small canisters for the PCM reduces the problem of void formation (due to the lower density of the liquid as compared to the solid) during freezing by minimizing stress buildup. In addition, the failure of a canister affects only that canister and the heat receiver continues to operate with only a small loss of capacity.

The cavity walls are made up of a thin layer of high-temperature insulation. Due to gaps between the tubes, some of the radiation entering the receiver will impinge directly on these walls. The walls will reradiate the incoming flux to the back side of the tubes. This reduces flux variations around the tubes and thereby reduces thermal stress levels. The cavity walls also act as a mandrel to wrap sheets of very low conductivity multifoil insulation which reduces heat losses from the receiver.

The insulated cavity is enclosed in an aluminum support structure. The tubes are supported by baffles which are, in turn, connected to reinforced regions of the support structure. The tubes fit loosely in the baffle holes and are free to expand. The backwall of the cavity moves as the tubes expand. Tube expansion is accommodated by two external bellows.

The receiver consists of 82 tubes, 2.5 m in length, with 96 containment cannisters per tube. The material for the working fluid tubes and containment canisters is the commonly used cobalt-base superalloy Haynes 188. The receiver outside dimensions are approximately 1.9 m in diameter and 3 m long. The receiver module weights about 1,794 kg. The receiver mass breakdown is shown in Table I-1 (Ref. 3).

TAC Description

The Brayton power conversion unit (PCU) design is based on the BRU-F engine which was tested at NASA LeRC (Refs. I-1 and I-4). The turbomachinery (turbine and compressor) and the alternator are on a common shaft forming a single rotating unit. A cross-section of a typical TAC is shown in Figure I-6.

TABLE I-1. RECEIVER MASS BREAKDOWN

Component	Mass, kg
Phase change material	340
Working fluid tubes	120
Containment canister	707
Inlet manifold	15
Outlet manifold	20
Ducts (including bellows and insulation)	24
Formed insulation mandrel	82
Backwall support plate	41
Multifoil insulation (nickel and aluminum)	117
Aluminum outer shell	144
Baffles and supports	61
Aperture plate	85
Aperture shield	39
Total	1,795

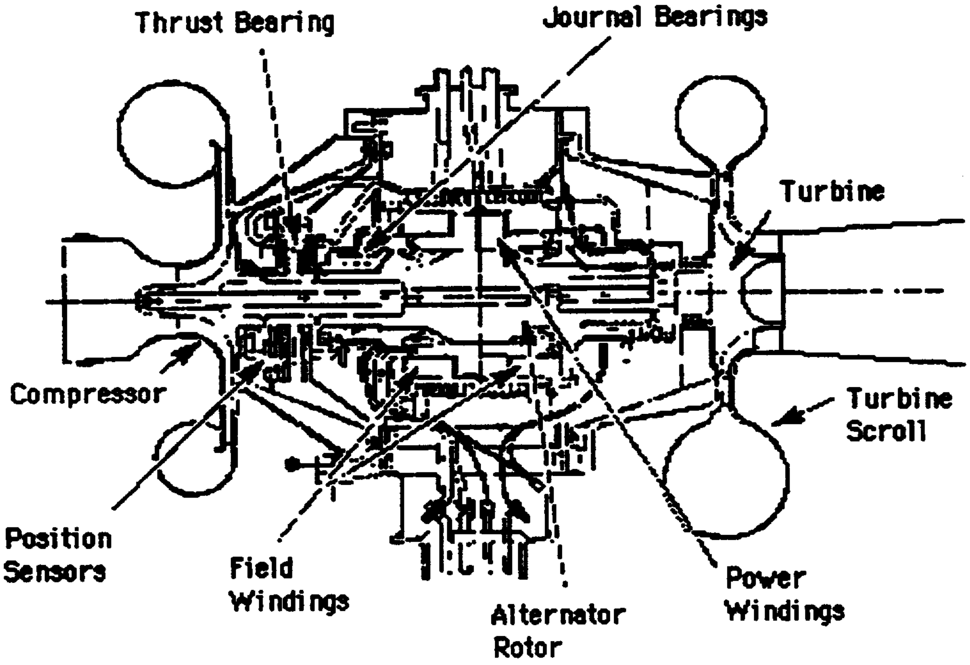


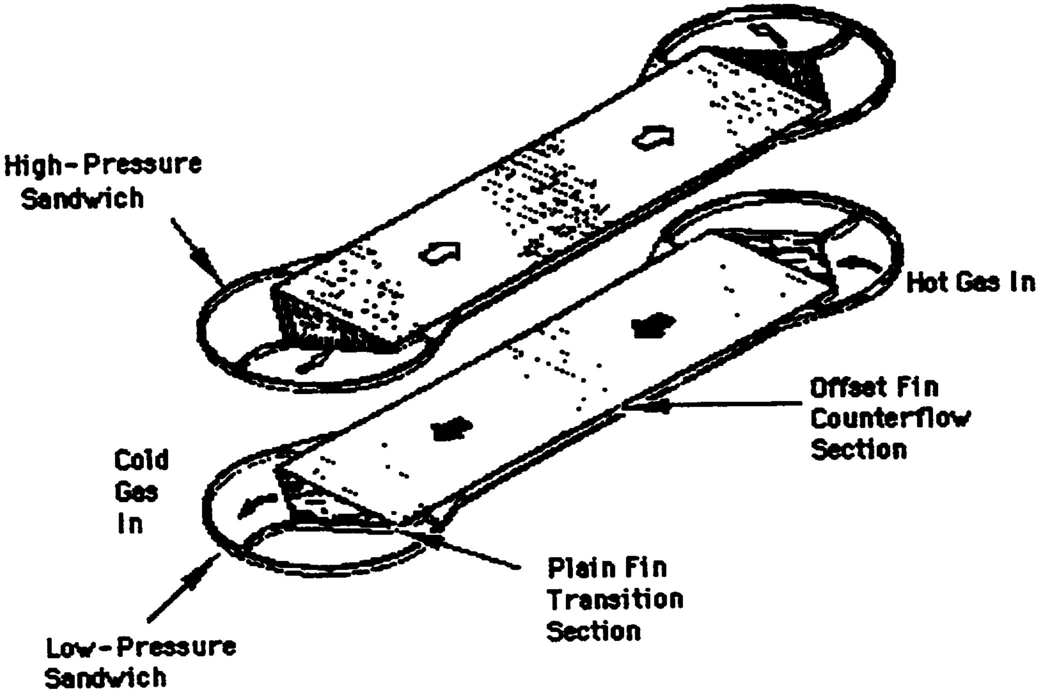
Figure I-6. Typical TAC assembly.

Both the turbine and the compressor are single stage radial machines and operate at a design speed of 32,000 rpm (Ref. I-4). The alternator is a solid rotor, modified Lundell type machine which converts shaft power to three phase ac electric power at 120 V line-to-neutral with a nominal frequency of 1,067 Hz. All of the electrical windings of the alternator are on the nonrotating part of the TAC, thereby providing for effective cooling paths and minimizing mechanical stresses on the windings and their insulation. Cooling of the rotor and bearings is achieved by a small bleed gas cooler which cools gas bled from the compressor outlet and feeds the cooled gas into the rotor cavity. The alternator is cooled by a combination of the cooled gas and liquid coolant from the heat rejection system. Ducting and structural ties connect the TAC with the balance of the PCU and receiver.

Recuperator/Cooler Description

The recuperator is a high-effectiveness, high-temperature heat exchanger with a low-pressure loss characteristic (Ref. I-1). The basic core configuration is shown in Figure I-7. The counterflow portion of the core is rectangular in shape. The design is of the plate-fin type, employing very compact offset fins to enhance the heat transfer. Triangular end sections and manifolds provide access to the appropriate passages of the counterflow core. These were designed to provide good flow distribution into the core with minimal pressure drop and weight penalties. The recuperator design effectiveness was 0.98 (high pressure) and 0.96 (low pressure).

A gas cooler is integrated with the recuperator in order to transfer the remaining waste heat in the working fluid to a pumped liquid which is circulated through the radiator. This cooler would be a dual fluid, cross-counterflow plate-fin design. The cooler gas-side design effectiveness would be in the range of 0.95 to 0.97 with a liquid-side effectiveness of approximately 0.67. This would result in a compact package easily welded to the low-pressure discharge side of the recuperator. A space-configured design would incorporate a redundant liquid loop.



- **Stacked to form the heat exchanger core with integral manifolds**
- **Recuperator cores are built from plate-fin sandwiches**

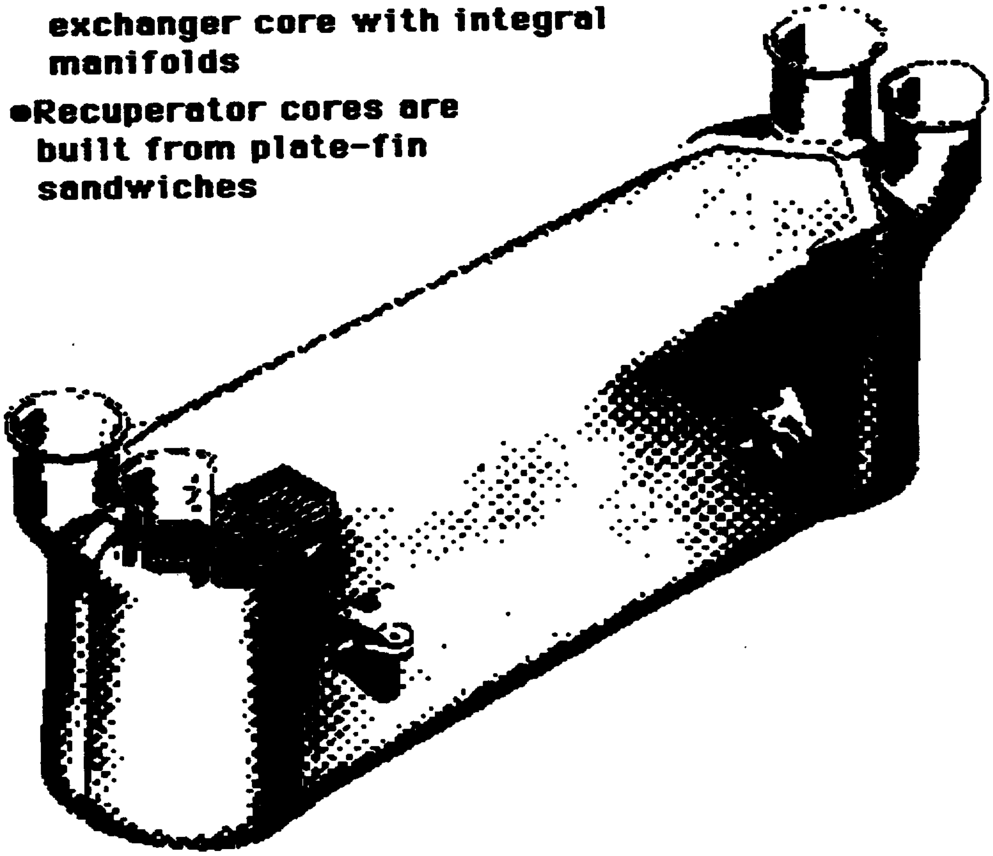


Figure I-7. Recuperator.

Heat Rejection Subsystem Description

The heat rejection assembly consists of a pumped liquid loop which collects the waste heat from the PCU and electrical components and transfers it to a radiator, and the radiator itself. Figure I-8 shows the dominant part of this assembly (Ref. I-4).

Since SSF is serviceable, a pumped loop radiator was selected to minimize radiator mass. The radiator is designed to reject about 100 kW of thermal power to space. The radiator is a deployable structure consisting of eight 8.7 m by 2.5 m panels connected together by flexible liquid-carrying hoses. Several radiator panels are installed on a scissored deployment mechanism. Each panel on the Space Station radiator assembly is made up of alternating extruded heat exchanger tubes and honeycomb structure, bonded between aluminum face sheets. The tubes consist of high conductivity extrusions, with the liquid passage in the center of the extrusions. The aluminum tubes transport coolant liquid from the inlet manifold on one side of the panels to the outlet on the opposite side. The maximum expected temperature at the inlet is 445 °K. The heat in the pumped fluid is conducted through the tube walls to the panels and then is radiated to space. The design features double-sided, high-emissivity radiator surfaces. The pumped-liquid radiator loop is integrated with the gas loop using the gas cooler previously described. Dual coolant paths provide reliability and full operational capability after one failure.

The deployment mechanism is a cable-actuated scissors which extends the panels from a stowed layered position to the extended position as seen in Figure I-8. The scissors mechanism also provides the structural support to maintain the radiator panels in position.

The radiator together with its deployment mechanism is expected to have a mass of 1350 kg (3000 lb). The total heat rejection subsystem complete with pumps, lines, and cold plate (for electronics thermal control) is expected to have mass of about 1550 kg.

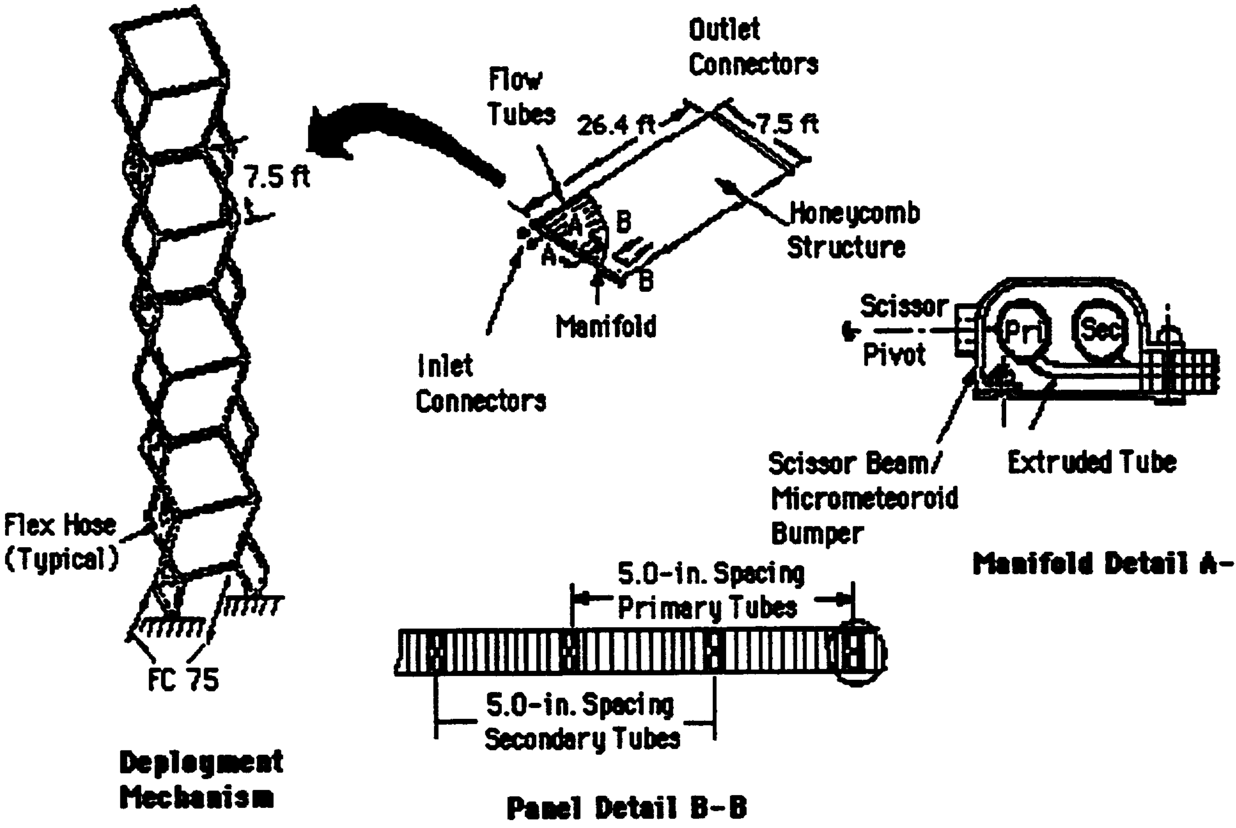


Figure I-8. Heat rejection subsystem.

Electrical Equipment Assembly (EEA) Description

This electronic/electrical assembly contains a variety of hardware which is mounted on a cold plate. For thermal control of this hardware, cooling liquid from the heat rejection subsystem circulates through this cold plate. The EEA contains all of the controls for the SD power module. This includes pointing controllers which assure that the concentrator is pointed to within 0.1 degree of the sun line. These controllers drive both the beta and fine pointing gimbal actuators and are interfaced with alpha gimbal controls in the central portion of the manned base. Control of the PCU includes control for voltage and speed, startup and shutdown sequencing, and mass flowrate of the gas (power level). Control of the mass flowrate assures maintenance of proper temperatures in the receiver. Pump motors for the heat rejection assembly and their drive electronics are also in the EEA. Finally, the EEA includes the frequency changer which converts the nominal 1067 Hz, three phase power from the alternator to 20 kHz single phase power for primary distribution. The mass of the EEA is expected to be about 260 kg.

Parasitic Load Radiator (PLR) Description

The PLR, under control of the EEA, is used to dissipate excess electric power from the Brayton system and thereby control the speed of the TAC as the user load varies. The CBC is designed to provide a specific power output at constant solar input (orbit-to-orbit variations not included). Variation in user demand creates the necessity to dissipate the excess electrical power.

TECHNOLOGY ISSUES

Various issues arose during the SSF Solar Dynamic Program. These issues include the following:

- flux tailoring (adjustment of the flux profile within the receiver for uniform distribution) and the effect on receiver life;
- concentrator pointing accuracy;
- concentrator fabrication and assembly;
- TES canister manufacturing techniques (forming, joining, filling and sealing);
- void formation in the TES during freezing;
- control methodology (i.e., gas inventory or speed control and determination of the receiver state-of-thermal-charge);
- electromagnetic interference (as emitted from the alternator and from the PLR driver circuit); and
- flight qualification using only ground testing (to reduce program costs and risk).

The above issues are primarily fabrication and integration issues since much of the component technology has been previously demonstrated. No integrated SD space power system has been built and component interactions need to be assessed before full scale development can proceed.

The SSF TES design minimizes void formation and thermal fatigue through the use of many small cannisters (reduces the likelihood of failure and the impact of a failure). The cannister metal walls are also sufficiently thick so that most of the heat transfer to the coolant tubes is through the metal rather than the salt. This insures that operation during zero g conditions will be similar to operation on the ground. However, this approach greatly increases the mass of the TES subsystem since about 40% of the mass is for the metal containment.

These and other features of the SSF SD option design were included to allow development and qualification testing to be based solely on ground testing while insuring proper operation in space (Ref. I-5). This approach was taken to reduce development costs.

TECHNOLOGY ASSESSMENT

The technology bases were assessed for the following major SD subsystems:

- concentrator;
- receiver/TES;
- HRS (radiator and deployment mechanism);
- CBC PCU; and
- EEA

The technology readiness of each assembly was evaluated using the NASA technology readiness levels. These evaluations are summarized in Table I-2 which shows that the SD has technology readiness levels ranging from 4.5 to 6, depending on the particular subsystem. The technology base for each assembly is briefly discussed in the following sections.

TABLE I-2. CBC-SD TECHNOLOGY ASSESSMENT

Subsystem	Technology Readiness Level	Comments
Concentrator	5	Harris full scale concentrator
Receiver/TES	5	Rockwell Phase B proposal effort prototype tested; Allied Signal work
EEA	5 - 6	Space Station Freedom, SSME technology
HRS	6	Apollo, Skylab, Shuttle radiators; Skylab Apollo Telescope Mount solar array deployment mechanism
CBC	4.5	BIPS and BRU-F

Concentrator State-of-the-Art

Much progress has been made with concentrators (Ref. I-6). Several experimental concentrators of various sizes - 1.5, 3.0, 9.7, and 12.2 meters diameter were fabricated and thoroughly ground tested, but were not flight-qualified. The 1.5 meter concentrator was a one piece unit, whereas the others were an assembly of segmented hinged reflector panels. Autodeployability was demonstrated with the segmented concentrators. Each also withstood the shake tests that simulated launch loads. The surface contour of the smallest concentrator was accurate to less than a 1-sigma value of 1.0 milliradian slope error, and weighed 2.7 kg/m^2. The surface contours of the larger segmented mirrors were not as accurate. These concentrators did fly as antennas, one on the Pioneer spacecraft.

Key accomplishments have been realized in concentrator reflective and protective coatings, optical characterization, and structural rigidity. The Harris Corporation, NASA Lewis, and 3M have demonstrated the reflective capability of the individual facets and the resistance of protective coatings to atomic oxygen. In addition, a full scale concentrator (19 panels) has been fabricated by the Harris Corporation. Successful assembly and repeatability tests were completed. The concentrator was delivered to NASA Lewis for a series of optical tests.

Receiver/TES State-of-the-Art

A strong technology base involving thermal storage materials compatibility, mechanical strength, thermal energy storage performance, and receiver thermal performance has been developed (Ref. I-4). Over 5000 hours of exposure of the thermal storage containment materials to the LiF-CaF2 salt show negligible corrosion. Extensive testing at NASA Lewis and at contractors has shown that the thermal performance is completely verifiable by ground testing and analysis (i.e., no flight test is required). Also, tests conducted by Allied-Signal with a single tube segment of the receiver has verified the operation of the baseline thermal energy storage configuration.

Rocketdyne has completed a system demonstration test in which separate receiver and

TES units were mounted on a Vanguard solar concentrator. This system successfully demonstrated essentially uniform receiver thermal/storage performance. The Rocketdyne receiver design was a simple tube coil. One inch diameter LiOH PCM cannisters were used for the TES. The LiOH PCM cannisters were tested through many freeze/thaw cycles, although the volume change upon LiOH phase change is quite small.

Boeing Aerospace & Electronics (BA&E) designed and built a full-size solar dynamic heat receiver for NASA as part of the advanced development effort to support solar dynamic power module design for the Space Station Freedom (Refs. I-7 and I-8). A follow-on program was also awarded to BA&E to test this heat receiver by operating it over the range of expected or required interfaces for space operation.

The Boeing heat receiver was designed to meet the requirements specified for the solar dynamic power modules on the Space Station Freedom. The 25 kW of electrical power supplied to the user requires a nominal 102 kW of thermal power delivered to the closed-Brayton cycle (CBC) heat engine throughout a 94 minute orbit, including when the spacecraft is eclipsed for up to 36 minutes from the sun. The receiver employs an integral thermal energy storage system that utilizes the latent heat available through phase change of a eutectic salt mixture of lithium fluoride and calcium difluoride. The salt mixture has a melt temperature of about 1420 °F. The salt is contained within a nickel felt matrix used to enhance heat transfer and to control the locations of voids that form during solidification.

Boeing developed test hardware, facilities, and procedures to conduct ground testing of a full size, solar dynamic heat receiver in a partially simulated, low Earth-orbit environment. The purpose of the test program was to quantify the receiver thermodynamic performance, its operating temperatures, and thermal response to changes in environmental and power module interface boundary conditions. The heat receiver was tested in a vacuum chamber using liquid nitrogen cold shrouds and an aperture cold plate. The receiver was tested inside a vacuum chamber to preclude convection effects and installed in a horizontal orientation to minimize the influence of gravity on the salt void distribution in the felt metal material. Special test equipment were designed to provide the required ranges in interface boundary conditions that typify those expected or required for operation as part of the solar dynamic power module on the Space Station Freedom. The support hardware included an 250 W infrared quartz lamp heater with 30 independently controllable zones and a closed-Brayton cycle engine simulator to circulate and condition the helium-xenon gas mixture. The CBC engine simulator circulates the gas through the receiver, removes heat, and conditions the gas to achieve the inlet temperatures, pressures, and flowrates required to simulate various power cycle operating modes.

The Boeing test program successfully demonstrated that a full-size solar dynamic heat receiver can be operated on Earth and in vacuum to quantify performance. The heat receiver met almost all of its design requirements during the simulation of Space Station Freedom operational modes. Thermodynamic performance compared with predictions although receiver losses through the cavity insulation were higher than expected. Cavity temperatures remained below design limitations. Comparison of the test mode data showed that receiver performance, maximum operating temperatures, and temperature gradients do not vary significantly between the power module operating modes simulated. The cavity radiation exchange appears to effectively smooth maldistributed incident flux although further analyses is required to estimate what incident flux profiles were actually produced from the off-design quartz lamp power distributions.

Closed Brayton Cycle State-of-the-Art

Dynamic power conversion systems based upon the CBC engine have long been considered for space-based applications (Ref. I-2). Design and development activities, dating back to 1962 have resulted in several CBC demonstration engines which have accumulated tens of thousands of operating hours utilizing laboratory heat sources. The CBC technology draws upon an unparalleled data base of open Brayton cycle engines utilized for aircraft propulsion, auxiliary power systems, and industrial power applications. The simplicity and flexibility of

CBC power conversion has resulted in the selection of the technology for several space-based applications, most notably the growth power requirement of the Space Station Freedom, with solar energy as the heat source.

A 10.5 kW Brayton power conversion system was developed by Allied-Signal under the management of NASA Lewis in the 1960's and 1970's as part of the BRU-F program. Several sets of hardware were produced and tested. These included Brayton Rotating Units (BRUs), heat exchangers, controls, accumulators, etc. These BRUs have physical size, turbine inlet temperature (1600 °F) and shaft speed (36,000 rpm) roughly comparable to the design of the SSF solar dynamic power module. The same type of working fluid, a mixture of helium and xenon gases, was also used. Two BRUs were successfully operated without any failures, maintenance or signs of wear or excessive creep.

Turboalternator - Compressor State-of-the-Art .The TAC consists of a turbine, alternator and compressor mounted on a single common shaft supported by radial and axial foil gas bearings. The design is similar to the 15 kWe Brayton rotating unit (BRU) and 1.3 kWe mini-BRU units designed and tested by the Garrett Corporation in the 1970s.

The NASA BRU series of four units, with a flight configured recuperator, cooler, and ducting were tested for a total of 52,573 hours, most of which was at a turbine inlet temperature of 1144 °K. One of these units accumulated 41,000 hours, including a 13,600 hour continuous run. The BRU used pivoted pad bearings instead of foil bearings because foil bearing technology was not well developed at that time. Since then, the foil bearing has been used successfully in aircraft auxiliary power units (APUs) for millions of hours and improvements have been made in the design. A summary of foil gas bearing experience is given in Table I-3.

The mini-BRU was the turboalternator-compressor incorporated in the BIPS. After some initial problems with one of the foil bearings, design modifications were made and the unit was operated for 1000 hours at a turbine inlet temperature of 1025 °K with no further problems.

Aircraft Auxiliary Power Units (APUs) have endured over 200 million unit-hours with over 200 million start/stop cycles with turbine and compressor designs that are similar to the SSF TAC. Because of the experience in related aircraft components, and conservative design, extreme confidence exists in the capability of producing a space qualified PCU.

TABLE I-3. SUMMARY OF APU FOIL GAS BEARING EXPERIENCE

	767/757	DC10	F-18	General Aviation	C5B
Systems in service	470	1,660	570	1,269	35
Total hours operating	6,735,800	59,073,400	406,180	3,200,500	42,000
Mean time between failures, hours	57,100	61,700	45,135	35,170	N/A
Average number of start/stop cycles per unit	2,200	2,100	1,400	2,500	110
Annual bearing production rate	200	100	200	500	300

Recuperator State-of-the-Art. The applicable experience base for CBC recuperators includes the recuperators for the BRU series of power conversion units, and for BIPS. This experience base is summarized in Table I-4. As noted in the table, one of the BRU recuperators experienced a failure at 18,000 hours of operation. The nature of the failure was a leak between the core and the manifold ducting. Analysis of the cause of the failure attributed it to fatigue due to thermal cycling. The problem was corrected by changing the structural material and the braze alloy. The BIPS recuperator used these improvements and was operated without incident.

TABLE I-4 APPLICABLE EXPERIENCE DATA BASE - CBC RECUPERATOR

Unit Name of Designation	Power Level (kWe)	Design Feature	Operating Parameters	Test/ Operational Hours	Key Test Results
BHXU (BRU)	10	Plate-fin plate-bar construction 347 SS	T_{in}=939 °K P_{in}=1.86x10^5 N/m^2 W=3.4 kg/s	51,000	One failure at 18,000 h; unit repaired and test resumed
BHXUA (BRU)	10	Plate-fin double containment side plate Hastelloy-X	T_{in}=939 °K P_{in}=1.86x10^5 N/m^2 W=3.4 kg/s He-Xe MW=83	N/A	Not tested - 2 units delivered
BIPS	1.3	Plate-fin double containment side plate Hastelloy-X	T_{in}=783 °K P_{in}=4.6x10^5 N/m^2 W=0.11 kg/s He-Xe MW=83	1,200	No failures

Interconnect Ducting State-of-the-Art. The high temperature interconnect ducting is made of Inconel and the low temperature ducting of aluminum. Both materials are well characterized.

Radiator and Manifold State-of-the-Art

Single phase, pumped liquid radiators have been previously flown aboard Apollo and Skylab and are now in use on the Shuttle (Ref. I-4). The SD radiator, which uses adhesively bonded honeycomb construction techniques, takes advantage of the current state-of-the-art fabrication methods which are demonstrated by the Shuttle's large single phase pumped loop radiators mounted on the cargo bay doors. In addition, the SD radiator uses an integrated automatic deployment mechanism and support structure which has also been used in orbit. The Skylab Apollo Telescope Mount (ATM) solar arrays were successfully deployed using the same basic concept that a SD radiator will use (i.e., a scissors arm with cable actuator). The SD power module will incorporate all the successful radiator technologies demonstrated previously. Building on this technology provides a low risk and low cost approach to producing a flight qualified radiator.

EEA State-of-the-Art

No special electrical or electronic hardware will be required for the SD power system. Power distribution equipment will already be installed as part of SSF when the SD system is deployed.

The technology required for the power processing equipment is being developed for Space Station Freedom. Rockwell is currently developing the Space Station Power Management and Distribution System. The use of multiple buses, components for the switching of loads, and switching of subassemblies within the power assembly are technologies which are being developed.

The required technology for the control and data acquisition functions currently exists, and the technology for autonomous control is under active development for Space Station Freedom (SSF), ESSA and other space programs.

MAJOR DEVELOPMENT TASKS

The SD CBC power system described utilizes existing designs and technology. The viability of this system has been established through detailed analyses, conceptual/preliminary designs, and component/technology development programs. However, demonstration of a complete power system has yet to be accomplished.

The SD development includes an initial NASA program for a 2 kWe demonstrator which is currently in the initial phase (Ref. I-9). Ground testing an integrated solar dynamic power system is the next step toward demonstrating the availability of solar dynamic technology for application to Space Station Freedom and other space platforms of future missions. A solar dynamic power system that maximizes synergism with the current SSF Solar Dynamic Power Module design configuration will be designed and built using existing hardware from prior NASA programs and designs from the SSF program (Ref. I-1). This approach maximizes program efficiency by minimizing component design, fabrication, and development effort while ensuring applicability to the SSF.

To this demonstration program was added an additional full scale 25 kWe power system development task (including any necessary component design modifications), qualification program, and flight program. The full scale power system development program was based on information in the Rocketdyne Phase C/D proposal (Ref. I-10). It is likely that the full scale program can be reduced in scope from that originally proposed by Rocketdyne due to work which will be done for the 2 kWe demonstrator. As a result of the SSF SD option conservative design approach no flight testing will be required to verify proper performance and life of the system. In addition, due to the early 2 kWe ground demonstrator program, their is no need for an additional full scale ground demonstrator system during development (although some integrated assembly testing is anticipated). System testing will be done during qualification and flight acceptance testing. Each of these tasks will be described in the following sections.

Advanced SD component development is expected to continue at a low level in parallel with the development of the nearterm SD system. Development of an advanced SD power module with improved performance will be described in a separate document.

Task 1. 2 kWe Ground Test Demonstrator (GTD) Development

Objectives: Demonstrate that a dynamic system does not have any limitations that preclude its successful operation in space. Resolve system integration issues, and provide a tool to perform future investigations and demonstrations (including evaluations of advanced technology components) to resolve issues arising from currently undefined, future space endeavors.

Statement of Work: The following subtasks were identified:

Task 1.1 GTD System Integration. Develop a comprehensive understanding of the characteristics and performance of each existing component. Use an existing CBC design point code to establish system performance, mass, and size and to establish the performance criteria for the refurbished and new components. Based on the capabilities of the existing components and designs, establish a system design and prepare requirements documents to provide direction for the modification of existing designs and the generation of the new required components.

Define the interfaces between the various components and subsystems, and the interface activity of each supplier. Towards the end of the component development phase, integrate the components into a complete system for delivery to NASA for system tests. Once a "design point" is established, incorporate the design into an off-design code in order to predict the performance throughout an orbit, or in response to changes in the operating environment or conditions. Support the development of component and system specifications as well as the design activity of any new component.

Task 1.2 GTD Product Assurance. Utilize plans and practices currently in place to assure that safety and reliability features are built into the GTD designs. Define the specific quality assurance practices which will be used in the program.

Task 1.3 GTD Component Design, Fabrication and Testing. Design, fabricate, and test the major new components identified in the system analysis layout. Base the design and fabrication of these components to the maximum extent practical on the existing SSF Solar Dynamics Power Module (SDPM) designs and make use of existing fabrication processes. Design the components for a minimum of 4 years operational life. Perform functional, thermal, stress and fatigue analyses, as required, to demonstrate that the hardware design satisfies the GTD program requirements. Major new components will include the solar concentrator, solar receiver, radiator, gas cooler, power conditioning and control unit (PCCU), parasitic load radiator, load simulator, and integration and test support hardware.

Conduct component tests in order to verify the performance and operation of that component. Perform these tests in accordance with previously developed test instructions. Check the performance against the requirement and, if any discrepancy is noted, determine the impact of the deviation on overall system performance using the system performance code noted previously. Confirm or correct the performance algorithms of the components to maintain the currency of design tools.

Task 1.4 GTD Existing Components Assessment, Refurbishment, and Testing. Refurbish two TACs including disassembly, cleaning, inspection, and repair or replacement, as appropriate. Conduct functional tests, as required, to assure that the rebuilt TAC units meet the new system design requirements.

Develop analytical models of existing components for functional, thermal, and stress evaluation using current analytical techniques and tools.

Assess the recuperator performance against the system requirements using analytical models. Pressure test the recuperators to the maximum required test pressure to verify the physical condition. Clean, repair, and functionally test the units to assure compliance with requirements.

Task 1.5 GTD System Test. After completion of the component tests, assemble and test the system in a local ambient test facility for final check-out prior to delivery for vacuum testing. In some instances, component performance in an ambient test cell will be significantly different from the expected performance in a vacuum. In this situation, exercise the performance codes to determine the performance under the imposed conditions. If agreement is achieved between the test and the predicted performance, then vacuum testing of the unit will proceed. Utilize the design point and off-design performance codes throughout the vacuum testing to confirm the performance or investigate any abnormalities discovered.

Task 2. Full Scale Concentrator Development

Objectives: Develop and demonstrate full scale panels for a 25 kWe power system. Demonstrate the capability of the full scale production facility to produce panels which meet design objectives for optical and thermal performance to the extent that such may be proven with ground testing.

Statement of Work: Design a full scale concentrator based on previous SSF work and any necessary changes as a result of the GTD program. Perform design trades such as facet fabrication process and materials selection, evaluation of the benefit of a rate gyro-derived error feedback for fine pointing and tracking, passive thermal control system selection for the reflector, structure and fine pointing mechanism, and final fine pointing algorithm selection in order to optimize the concentrator design. Consider alternate materials and fabrication techniques to reduce mass. Perform reliability/redundancy trades to minimize life cycle costs.

Demonstrate that the full scale panels can achieve and maintain satisfactory optical characteristics when deployed and under normal thermal operating conditions. Optically align the facets during ground assembly. Insure the ability of the hardware to maintain this alignment, once it has been disassembled, packaged for launch, launched, and then reassembled on orbit by verifying this design requirement during tests on prototype hardware. Demonstrate the ability to align the prototype hardware optically before, during, and after qualification tests of the launch configuration.

Task 3. Full Scale Receiver/TES Development

Objectives: Develop and demonstrate a full scale solar receiver with integral thermal storage for a 25 kWe power system. Demonstrate adequate performance and long cycle life.

Statement of Work: Design a full scale receiver/TES subsystem. Construct the receiver/TES subsystem. Perform component testing to verify adequate performance and life.

Determine the thermal-hydraulic and structural performance of the receiver heat source heat exchanger by analysis and testing. Include the effect of environments such as thermal stress, vibration, launch acceleration, acoustic, pressurization, and shock loads in these studies. Select design features to control working fluid flow rate distribution and accommodate variance in the flow rate between heat exchanger tubes, and define methods of implementation. Include accommodation of differential thermal expansion in heat exchange tubes and in ducting bellows, and accommodation of pressure-drop limitations while preserving good thermal performance in the receiver design. Design the receiver structure to support the receiver equipment through the range of environments that include ground transport, launch, and operation. Evaluate alternate materials and fabrication techniques of the thermal storage subassembly to minimize life cycle costs. Examine the design to insure the accommodation of the high-load environment anticipated during launch followed by large thermal expansions during startup and operation. Include the effects of walkoff tolerance, the ability to accommodate multiple concentrator slew operations, and cyclic sunrise/sunset atmospheric refraction effects in the aperture shield design. Study thermal leak paths through structural supports, and reduce total surface losses to the be within limits set through performance requirements.

Initiate an intensive materials characterization effort to establish detailed design information for the eutectic salt, canister material, aperture shield material, and insulation materials. Include tests of the properties, off-gassing, thermal shock tolerance, particulate generation, compatibility, and accelerated aging, as appropriate, in addition to thermodynamic and transport properties.

Perfect the manufacturing techniques for automated canister production, canister-to-tube bonding, multifoil insulation layup, and retention and assembly of the segmented aperture shield.

Complete performance testing starting with a partial receiver tube thermal cycling and progressing sequentially to a full length tube, partial receiver core, full receiver core, and integrated receiver testing. Perform thermal/hydraulic, pressure/leak, cyclic stress, dimensional stability and fitup, acoustic, acceleration, and shock tests to determine the performance of the receiver elements both singly and in combination under the range of environments.

Perform development testing and associated analysis in order to supply sufficient data on performance, safety, producibility, reliability, life, and operability to support a decision to commit resources to final detailed design and fabrication of flight-quality receivers for qualification. Perform long-term canister cycling, receiver thermal energy charge and discharge characterization, working fluid pressure boundary mechanical integrity, and aperture shield walkoff tolerance studies.

Task 4 - Full Scale CBC PCU Development

Objectives: Develop a full scale flight CBC PCU for a 25 kWe power system. Demonstrate adequate performance and life.

Statement of Work: The CBC PCU development includes development of equipment in four basic areas: power conversion unit, parasitic load radiator, engine controllers, and wire harness. The PCU development includes development of the TAC, recuperator/cooler, and interconnect ducting.

Design the TAC. Perform analysis and verification of rotor dynamics and vibration, aerodynamic and thermal-hydraulic performance, structural performance and integrity of the working fluid pressure boundary. Investigate containment integrity over the full range of conditions that the device will see during its ground handling, launch, and orbital operations. Study the effect of environments such as thermal stress, vibration, launch acceleration, acoustic, pressurization, and shock loads. Select and define methods of implementation for design features to provide thermal management and working fluid bleed flow distribution.

Design and build heat exchangers for the PCU including a recuperator and gas cooler. Minimize thermal stress and enhance reliability in the design of the heat exchangers. Pressurize the recuperator core assembly with hot gas and thermally cycle the unit to verify its structural integrity. Measure leakage across the core and critical dimensions at temperature extremes. Measure the pressure drop through the recuperator core. Give the complete recuperator a hydraulic proof pressure test and a pneumatic proof pressure test. Perform a hot-gas performance test on the assembly. Conduct a resonant frequency scan test (ring-test) on a complete assembly.

Design ducting, gas accumulator tankage and bellows based on aircraft turbine engine ducting experience. Design gas control valves based on nuclear safety-qualified valve designs currently in production.

Design the PCU electric loop control including the alternator magnetics, power electronics, instrumentation, logic components, software, parasitic load radiator, cabling, and packaging.

Design and construct an engineering model of the PCU for performance against the full requirements set. Initiate a materials characterization effort to establish detailed design information for candidate PLR resistance element and insulator materials. Perform tests to determine electrical properties, off-gassing, thermal shock tolerance, particulate generation, compatibility, and accelerated aging, as appropriate. Perfect manufacturing techniques for bellow joints, PLR assembly, gas control valves, special reliability features associated with the heat exchanger tasks, and PCU components integration.

Complete performance testing of the turboalternator rotor, bearings, compressor, and turbine wheels. Perform a fully integrated test of the PCU. In parallel, begin the heat exchanger test program with core fabrication and test, followed by integration and test with manifolds and wrap structure. Perform a characterization and accelerated aging test of the bellows and solenoid control valves. Perform electric controls testing with individual functional elements breadboarded in early performance tests followed by progressive integration to brassboard and engineering fidelity units.

Complete development testing and associated analysis in order to supply sufficient data on performance, safety, producibility, reliability, life, and operability to support a decision to commit resources to final detailed design and fabrication of flight-quality PCUs for qualification.

Task 5. Integrated Receiver/Power Conversion Unit (IRPCU) Development

Objectives: Develop a full scale flight IRPCU. Demonstrate adequate steady state and transient performance characteristics, long life at high temperatures, and suitable performance during failure modes.

Statement of Work: Assemble and test a complete power conversion unit consisting of a receiver and PCU. Tests will be conducted under vacuum using a closed loop with simulated heat source, user loads and parasitic load resistors. Demonstrate start-up, shutdown and design speed conditions under normal operating conditions. Verify interface with the SD controller under conditions of varying power demand and simulated faulted conditions. Tests will include:

- time to start up from cold condition and motor KVA, invertor ramp rate;
- thermal balance;
- electrical power generating capability;
- steady state and dynamic stability;
- shutdown and start-up under simulated faulted conditions; and
- short term endurance test.

Task 6. Heat Rejection Subsystem Development

Objective: Develop and demonstrate a full scale pumped loop radiator and interface heat exchanger.

Statement of Work: Complete design studies which define the heat rejection assembly. Design studies will include thermal and hydraulic design, structural and dynamic design, materials selection, and configuration design and description. Optimize the heat rejection assembly design based on studies of weight, efficiency, and reliability considerations as they affect life cycle costs.

Fabricate test articles for coatings, adhesive and impact testing, and a development radiator panel, a development deployment test article with simulated radiators, and development flex hoses. Performance test the heat rejection assembly in a vacuum. Verify performance of the radiator under design point and extremes of temperature conditions.

Task 7. Full Scale EEA Development

Objectives: Demonstrate adequate steady state and transient performance, life, and immunity to the environment (including launch and operating). Demonstrate the software capability to handle power system nominal operation and failure modes.

Statement of Work: Define the EEA requirements. Develop algorithms for pointing and tracking and other controller functions. Integrate and test the electrical/electronic equipment at the subsystem level. The electrical equipment assembly will include two redundant SD controllers and a frequency changer, as well as the SD module cable set. The heat rejection assembly provides the utility plate cooling for these SD electronics.

Breadboard units will be built to demonstrate and check functional performance of the individual component circuit designs. Design modifications and improvements will be incorporated into brassboard units, which will be used to verify functional performance within the constraints of the actual component configuration. Prototype units will be fabricated and a series of performance tests run using simulated input and output loads. Controller tests will include validation of operating system software. Tests will include:

- start up, steady state and transient control;
- failure simulation and detection and switching;
- effects of temperature extremes and thermal shock;
- shock and vibration;
- cold plate heat loads; and
- EMI generation and susceptibility.

Prototype cable harnesses and a prototype parasitic load radiator will be checked out with the electrical components during the integrated subsystem tests.

Software will be checked out in conjunction with tests of the controller, using simulated inputs and outputs. Integration and checkout of the software will be performed as part of the system test.

Task 8. Full Scale Power System Design

Objectives: Define the requirements for a full scale power system. Define a full scale 25 kWe module design.

Statement of Work: Based on the results of the ground demonstration unit tests, identify a power system concept which can meet flight system requirements. Define subsystem sizes. Perform feasibility studies for the selected concept including off-design and transient analysis.

Design and integrate the full scale SD system. This task shall be done in parallel with the component development tasks to insure proper system characteristics. Design the system to meet the launch configuration.

Task 9. Full Scale Power System Qualification Program Testing

Objectives: Assemble qualification units and verify adequate performance for the entire system.

Statement of Work: Complete a comprehensive performance and dynamic testing program for assemblies and the complete system to provide a formal demonstration that the SD system will perform as designed after being subjected to simulated launch conditions. Qualification will be to launch environmental conditions selected to envelope the probable intensities developed by various launch scenarios.

The overall approach, shown in Figure I-9, will start with qualification of assemblies. Qualified production items will then be fabricated and assembled into the QU which is then qualified by the rules for space vehicle qualification. This approach will minimize the risk that some assembly of the QU will fail during the qualification test sequence, thereby nullifying the qualification, and requiring backtracking to recover from the failure.

Conduct performance testing at each level to verify that each item performs as designed. Perform dynamic testing to verify capability of the SD system to withstand launch loads, including acoustic, pyroshock and vibrational.

The performance and dynamic qualification test sequence for components and assemblies is shown in the matrix of Figure I-10. The corresponding qualification test sequence for the QU is shown in Figure I-11.

Task 10. Flight System Program

Objectives: Build the flight systems and perform acceptance testing to demonstrate required performance. Deliver the flight systems.

Statement of Work: Fabricate the flight unit parts, acceptance test the components, and assemble the components to produce flight systems. Subject the systems to acceptance testing before shipment to the launch site. Complete reports necessary to obtain launch approval.

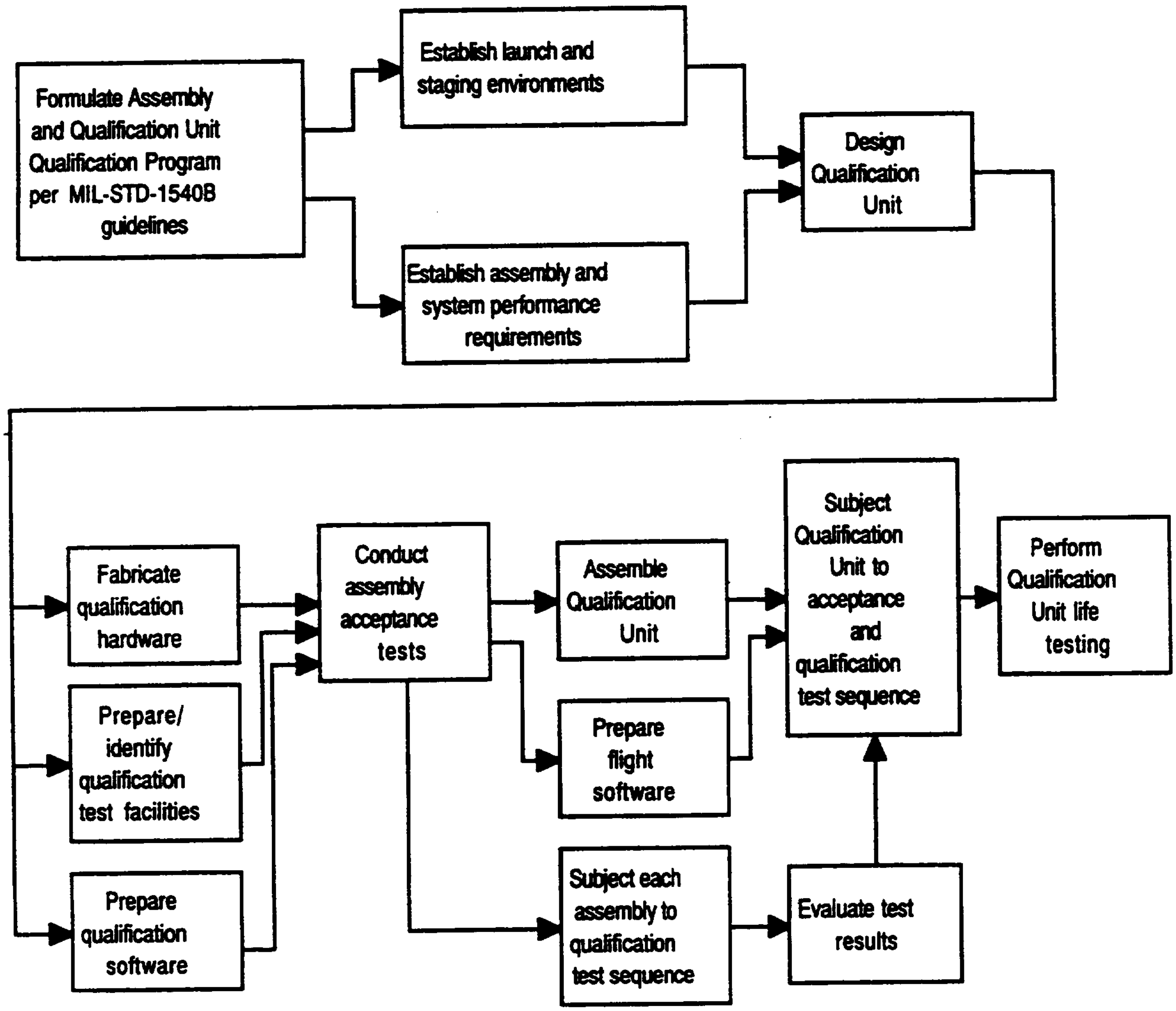


Figure I-9. SD qualification program.

Component or Subassembly	Functional (1)	Leak	Pyroshock	Functional (2)	Random Vibration	Functional (1)	Leak	Acceleration	Functional (1)	Thermal Cycling	Functional (2)	Thermal Vacuum	Functional (2)	Pressure	Leak	EMC	Life	Functional (1)
Concentrator	X		X	X	X	X		X	X	X	X	X	X				X	X
Solar Receiver/TES	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X
PCU	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
IRPCU	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
HRS	X	X	X	X	X	X	X			X	X	X	X	X	X		X	X
EEA	X		X	X	X	X				X	X	X	X			X	X	X
Structure	X		X	X	X	X		X	X	X	X						X	X

Figure I-10. Assembly qualification test matrix.

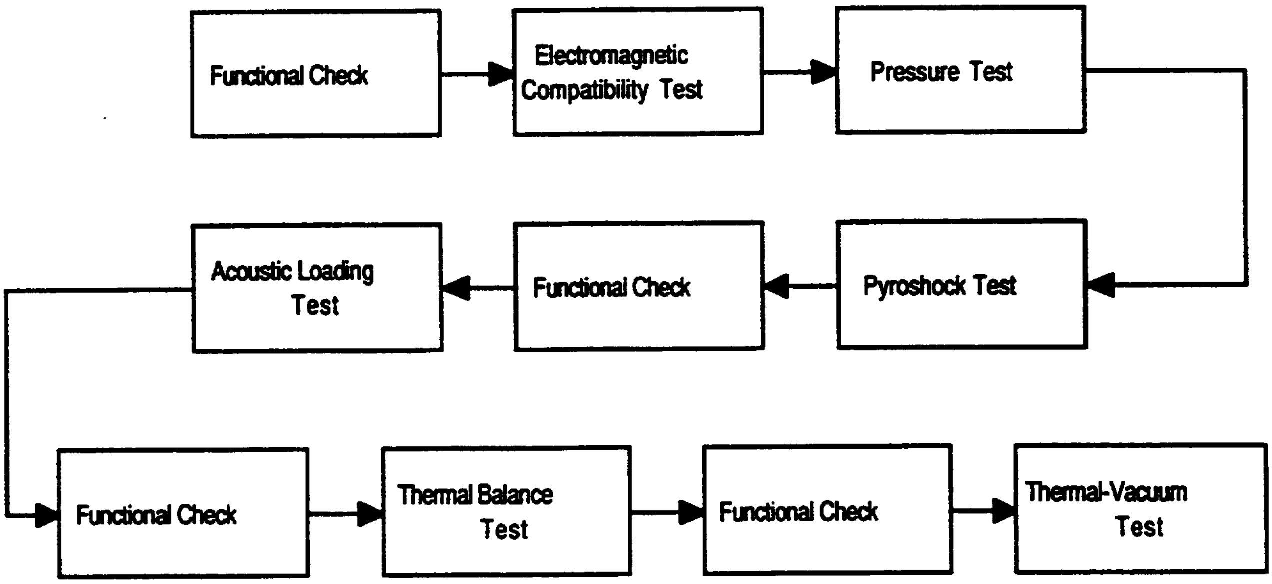


Figure I-11. QU test sequence.

DEVELOPMENT SCHEDULE

Figure I-12 shows the development schedule for this system (note: PSAR=Preliminary Safety Analysis Review, USAR=Updated Safety Analysis Review, FSAR=Final Safety Analysis Review, PDR=preliminary Design Review, and CDR=Critical Design Review). The total development time to reach a technical maturity level of 9 is estimated to be 9.5 years (assuming no gaps in the program). The development effort begins with a 2 kWe SD system ground demonstrator program which is scheduled to last 4 years.

A full scale development program was assumed to start one year before the end of the GTD program or the beginning of 1995. This is probably the earliest this new start could occur. Realistically, it is unlikely that NASA will begin another large power system development program such as this while SSF has yet to be deployed. SSF is scheduled to be fully operational by the year 2000 with three 25 kWe PV modules. The first growth phase for SSF would be probably to add another PV module. It would be sometime after this that an SD system might be deployed for additional SSF power growth. Thus, there is likely to be additional slippage in the beginning of the full scale SD development program. In the meantime, it is expected that advanced SD component development will continue at a low level. The GTD facility will be used to test these advanced components as part of an integrated power system. These advanced components could be developed to a point where they could be integrated into the SSF SD baseline design. This development option is discussed in more detail in the Advanced SD Power System Technology Roadmap (Appendix I).

The full scale development program for SSF starts with a full scale system design task and concurrent component development of the concentrator, receiver/TES, PCU, HRS, and PP&C. The detailed design is subsequently completed during the sixth year (2.5 years after start of full scale development).

The SSF power system qualification phase will last about 2 years and includes fabrication, assembly, and qualification testing of individual components, and a complete system. It is expected that fabrication of qualification units can begin toward the end of the full scale system development.

The flight phase of the SSF program includes fabrication, assembly, and acceptance testing of the flight units and the associated safety analysis to obtain launch approval. The safety program is carried on in parallel with the qualification program.

NEARTERM SOLAR DYNAMIC POWER SYSTEM TECHNOLOGY ROADMAP

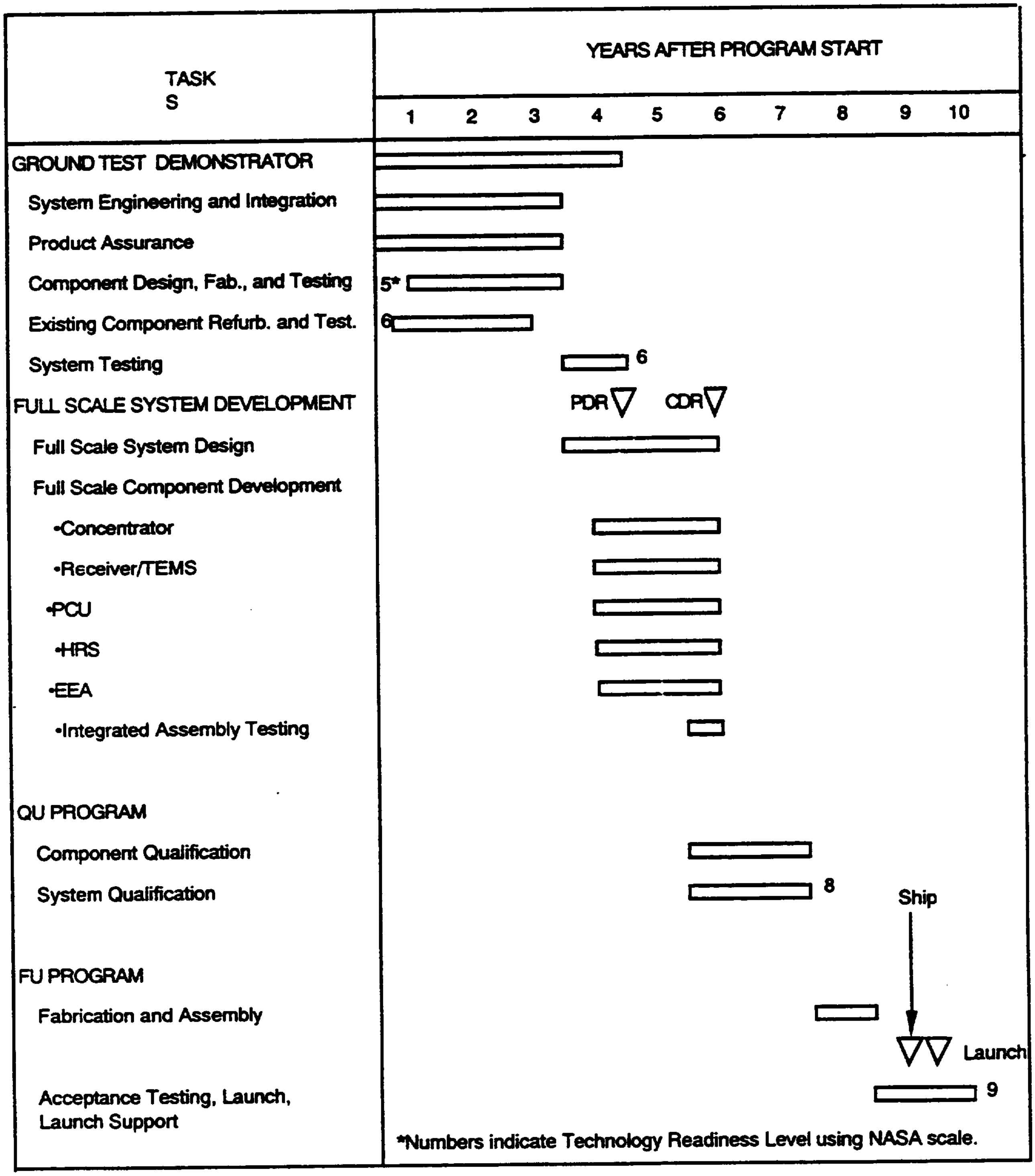


Figure I-12. Nearterm SD power system development schedule.

REFERENCES

I-1. Space Station WP-04 Power System, Contract Number NAS3-24666, Preliminary Analysis and Design Document DR-02. RI/RD85-320-2, Rockwell International (Rocketdyne Division), 19 December 1986.

I-2. Harper, W. B.; et. al: Solar Dynamic Closed Brayton Cycle Power Generating System Ground Demonstrator. 26th IECEC Proceedings, Vol. 6, August 1991, p. 10.

I-3. Strumpf, H.J.; et. al: Design Analysis and Life Prediction for the Brayton Engine Solar Receiver for the Space Station Freedom Dynamic Option. 26th IECEC proceedings, Vol. 1, August 1991, p. 241.

I-4. Secunde, R.R.; Labus, T.L.; and Lovely, R.G.: Solar Dynamic Power Module Design. 24th IECEC Proceedings, August 1989, p. 299.

I-5. Friefeld, J.: Personal Communications. Rockwell International (Rocketdyne Division), 13 February 1992.

I-6. Naujokas, G.J.; and Savino, J. M.: A Program for Advancing the Technology of Space Concentrators. 24th IECEC proceedings, Vol. 2, August 1989, p. 861.

I-7. L.M. Sedgwick; K.J. Kaufmann; K.L. McLallin; T.W. Kerslake: Ground Test Program for a Full-Size Solar Dynamic Heat Receiver. 26th IECEC Proceedings, Vol. 1, August 1991, p. 268.

I-8. L.M. Sedgwick, K.J. Kaufmann, K.L. McLallin, T.W. Kerslake: Full-Size Solar Dynamic Heat Receiver Thermal-Vacuum Tests. 26th IECEC Proceedings, Vol. 1, August 1991, p. 262.

I-9. 2 kWe Solar Dynamic Power System Ground Test. NASA-Lewis Research Center solicitation number RFP3-438039, 12 December 1991.

I-10. Space Station Freedom Phase 3/4 Proposal. Rockwell International (Rocketdyne Division) proposal RI/RD87-200P-2, 1987.

APPENDIX J

ADVANCED SOLAR DYNAMIC POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX J - ADVANCED SOLAR DYNAMIC POWER SYSTEM TECHNOLOGY ROADMAP

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APPENDIX J - ADVANCED SOLAR DYNAMIC POWER SYSTEM TECHNOLOGY ROADMAP

INTRODUCTION

This is a technology development plan for advanced solar dynamic (SD) power systems. The term "advanced" is used to denote technology which is beyond the baseline system previously selected for Space Station Freedom (SSF) SD option. The SSF baseline system represents current technology and is intended for nearterm applications (i.e., within 10 years). Due to the cancellation of the SD option for SSF and the subsequent funding cut for advanced SD research, it is not known when or if an SD power system will be deployed in space or for what applications. Potential SD power system applications include Low Earth Orbit (LEO) and lunar applications (manned and unmanned). An SD growth module or modules for SSF is still in the NASA plans although this program is currently unfunded and not approved by Congress. However, there are significant advantages for SD over PV systems in terms of reduced size and extended life. Thus, it is expected that SD power systems will eventually be utilized in space applications.

This roadmap presents two options for development of advanced SD power systems. The first development option is to fold development of advanced components (i.e., the concentrator, receiver, and heat pipe radiator) into the SSF design. This could result in some delay in development of the SSF SD option (compared to the optimistic schedule estimated for the nearterm SD system) to allow for validation of the advanced component designs. However, it is likely that sufficient time for advanced component development will occur prior to the actual start of full scale development of an SD power system.

In the second option, the nearterm baseline SD system is developed first for SSF or similar applications. The advanced SD Thermal Management and Stirling cycle (SC) power converter component development would continue in parallel. Development of an advanced SD system using a Stirling power converter would be completed several years after deployment of the nearterm system. This advanced SC SD power system could be utilized for manned or unmanned applications over a range in power levels (depends on the module size developed).

This concept would be especially suitable for unmanned or unserviceable spacecraft although there would be significant advantages for manned applications (reduced mass and size).

CONCEPT DESCRIPTION

The advanced SD power system concept has not yet been selected. This concept could either be an advanced closed Brayton cycle (CBC) power system or an SC power system. In addition, the power level for this system has not been selected. Current goals for the power converters are about 25 kWe. These system concepts and key components are described in the following sections.

Advanced CBC SD Power System Description

The CBC system would be similar in operation to that described for the nearterm SD system. However, the truss hex concentrator (massive, fixed facet, constructible) would be replaced by a self-deployable, low mass concentrator. In addition, the massive Allied Signal receiver design would be replaced by the low mass heat pipe cavity design.

Advanced SC SD Power System Description

This concept would be similar to the previously described SD system except that the power conversion would be done by a Stirling engine with an integral linear alternator instead of a CBC converter. For an integral receiver/TES subsystem (which is the current NASA baseline), the receiver design would also have to be modified to provide heat to the Stirling engine heater head. Heat pipes (or wicked heat transfer surfaces) would replace the gas tubes of the CBC receiver. This concept would require further development of the SC converter which is not as well developed as the CBC converter. This concept offers improved system efficiency and reduced system mass over the advanced CBC system.

Advanced Concentrator Description

Many concentrator concepts have been examined for space applications. These include both reflective and refractive approaches (Ref. J-1). Work has been done to define low mass, low packaged volume, self-deployable concentrator concepts, and to develop materials and fabrication techniques for reflector surfaces. No concept has been selected as the baseline, primarily due to a lack of funding. However, the Sunflower approach, which will be discussed later, is being pursued by NASA as the best nearterm solution. Goals for the concentrator are summarized in Table J-1 (Refs. J-1 and J-2). Some of these concepts are described in the following sections.

TABLE J-1. CONCENTRATOR GOALS

Parameter	Goal
concentration ratio	minimum - 2000 goal - 5000
efficiency	minimum - 90% goal - 95%
maximum end-of-life degradation	10%
specific mass	maximum - 1.5 kg/m^2 goal - 1.0 kg/m^2
operational life	10 years in LEO
deployment	automatic without EVA
size range	1 to 100 kWe @ a system efficiency of 30%
survivability	terrestrial, launch, and LEO environment
slope accuracy	0.5 to 1.5 mrad
misorientation	0
pointing error	0

Acurex Cartridge Deployed Rigid Panel Concentrator Description. This concentrator is made up of many rigid pie-shaped, parabolically contoured reflective panels as seen in Figure J-1 (Ref. J-3). The primary structural material selected for the concentrator is PEEK/carbon fiber composite. The assembled concentrator takes a circular shape with a void in the center. In the packaged transportation configuration, the panels are stacked vertically, similar to a deck of playing cards, in a compact dispenser cartridge. Deployment of the concentrator is via passive potential energy stored in spring motors that are sequentially unlatched to move the respective positions and latching them there. Upon deployment of the concentrator, the dispenser remains part of the assembly as the interface between the concentrator, the receiver/engine support structure, and the space station or satellite.

Harris Splined Radial Panel Concentrator Description. This concept consists of many thin, slender, flexible reflector strips on an umbrella shaped support system of ribs and cords as seen in Figure J-2 (Ref. J-1). The semi-rigid slat surfaces are bowed into a spline parabolic curve by a catenary cord and tie system attached to the deployable truss structure. The slats are bonded one to another and are not easily replaceable. This concept is self-deployable.

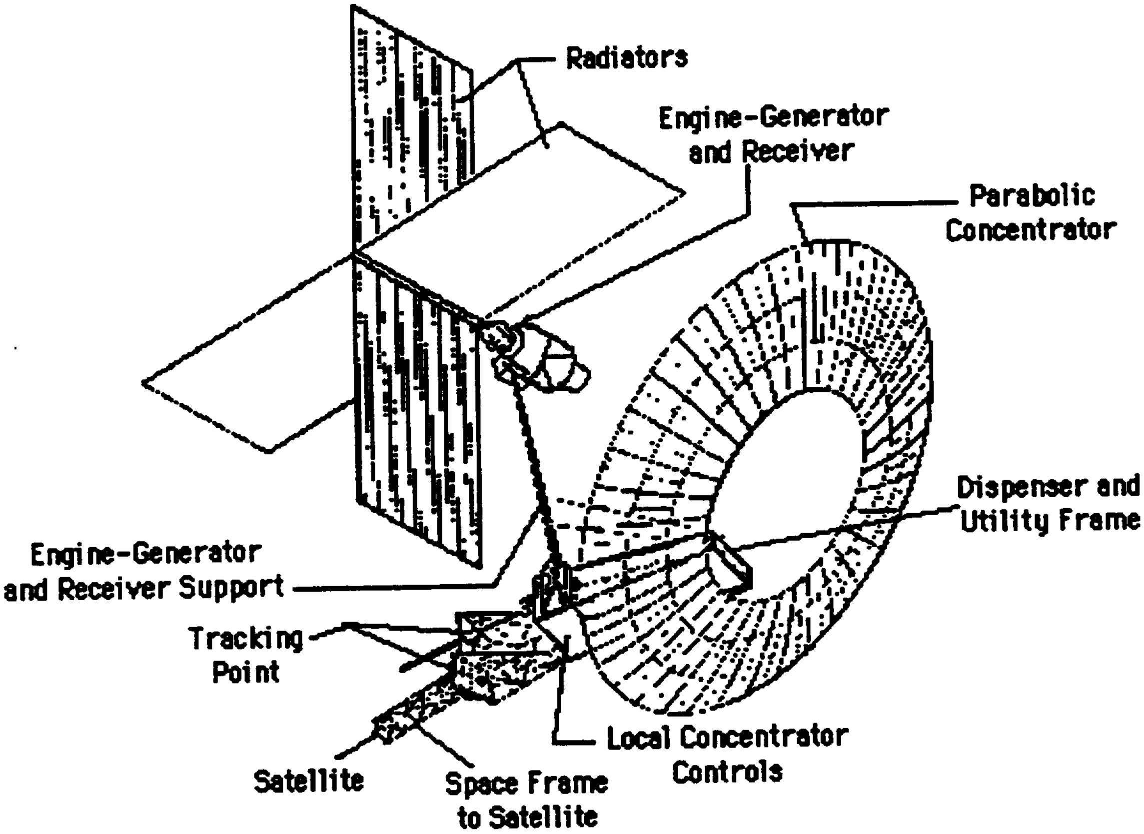


Figure J-1. Acurex self-deployable concentrator.

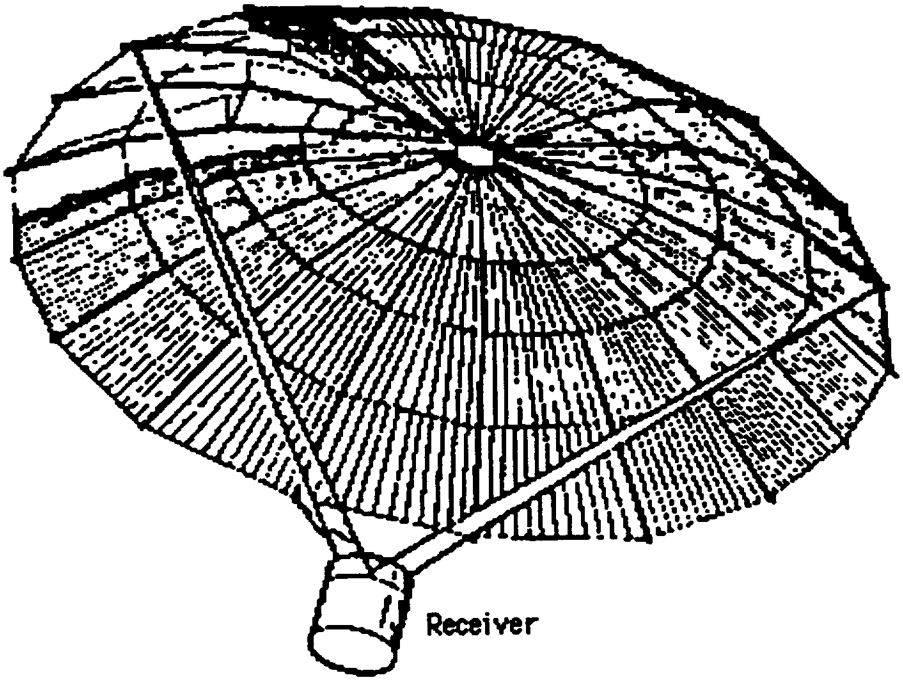


Figure J-2. Harris splined radial composite panel concentrator.

Fresnel Lens Concentrator Description. A domed Fresnel concept is shown in Figure J-3. This concentrator is composed of many flat transparent thin panels embossed with refractive prisms (Ref. J-1). The Fresnel dome is shaped by a catenary cord and tie system attached to a deployable truss structure. Lens panels are bonded to one another and are not easily replaceable. There are also a large number of panels. The lens is shadowed by the truss structure. The concept is self-deployable. The mass and stowed volume for this concept are similar to the splined radial panel.

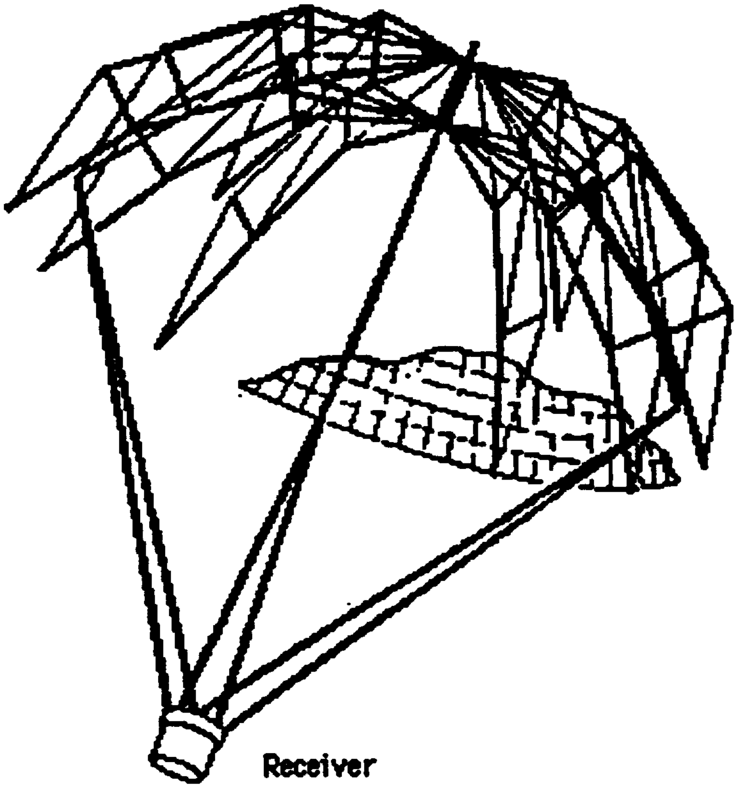


Figure J-3. Domed Fresnel refractive concentrator.

The main obstacle to development of the refractive type concentrator is the lack of a lightweight Fresnel lens material that will resist degradation by atomic oxygen (AO) and ultraviolet radiation (UV) in low earth orbit (LEO).

Sunflower Concentrator Concept Description. Figure J-6 (Ref. J-5) shows a picture of a 2 meter concentrator using the Sunflower concentrator approach. This appears to be the preferred NASA approach to space solar concentrators (Refs. J-1 and J-5). This concept has achieved a specific mass of 1.5 kg/m^2, which is only 38% of the Space Station Freedom concentrator specific mass. This concentrator has also achieved a surface accuracy of 1.0 milliradians. This concept is self-deployable and uses developed technology.

The Sunflower concentrator is parabolic with a hole in the center. The concentrator is divided up into several petal shaped panels. This concept is limited to a size of about 13 m diameter when deployed (10 kWe) based on shuttle bay packaging limitations (Ref. J-5). A variation of this concept which looks like a "vegetable steamer" can be used for larger sizes (2 kWe to about 35 kWe per Ref. J-5). This advanced concept uses petal shaped panels attached to a parabolic, rigid circular center section as seen in Figure J-5 (Ref. J-5).

The baseline reflector is an all metallic approach (Ref. J-5). The structure or substrate consists of an aluminum honeycomb with aluminum face sheets (0.3 mm thick). The face sheets are bonded onto the honeycomb. Alternatively, the face sheets may be soldered onto the honeycomb. The goal is to have a smooth surface with no imperfections. A leveling coating is then applied to the front side of the substrate to eliminate any remaining imperfections in the surface. Both polymers and microsheet glass are being considered for leveling layers. A polyimid leveling coat would be about 0.025 mm thick. A microsheet glass sheet would be about 0.076-0.102 mm thick and would be bonded to the aluminum facesheet. Over the leveling coat would be put a thin layer (800 - 1000 Angstroms) of vapor deposited aluminum. Aluminum is not affected by the LEO environment.

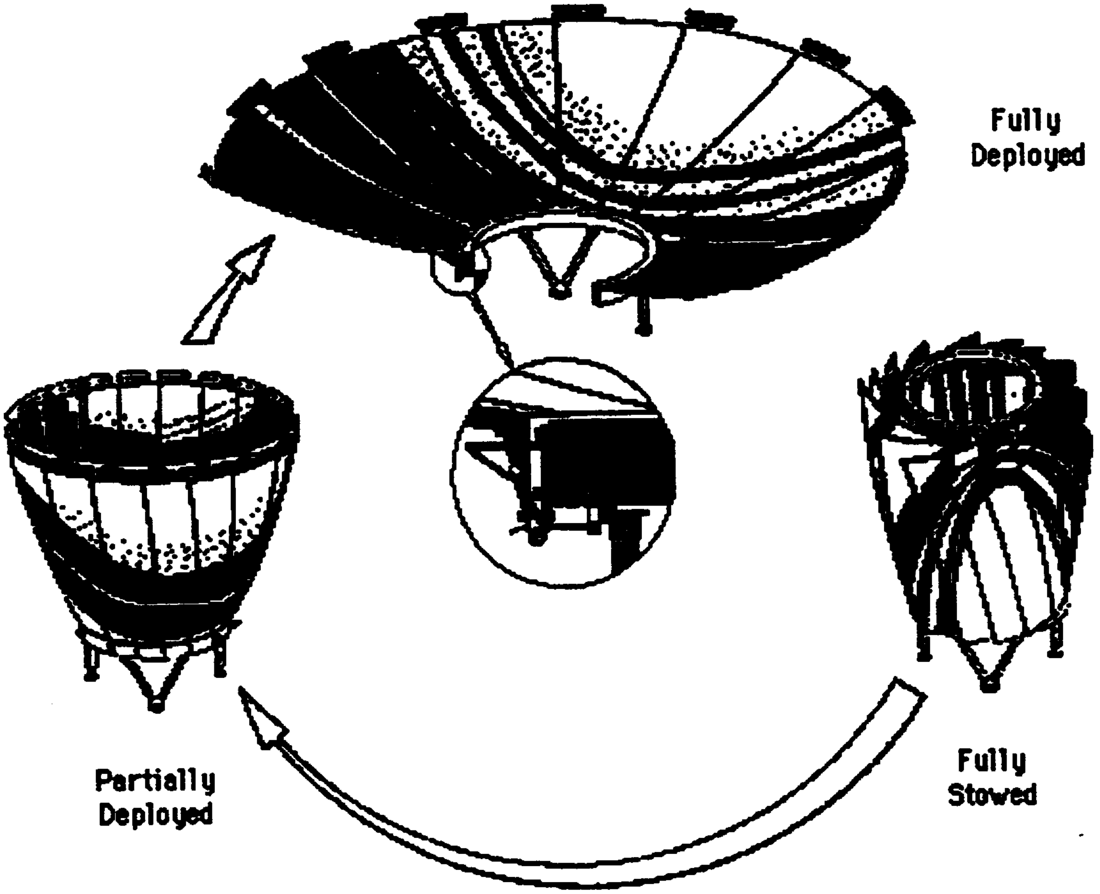


Figure J-4. NASA/Cleveland State University self-deployable 2 meter prototype concentrator.

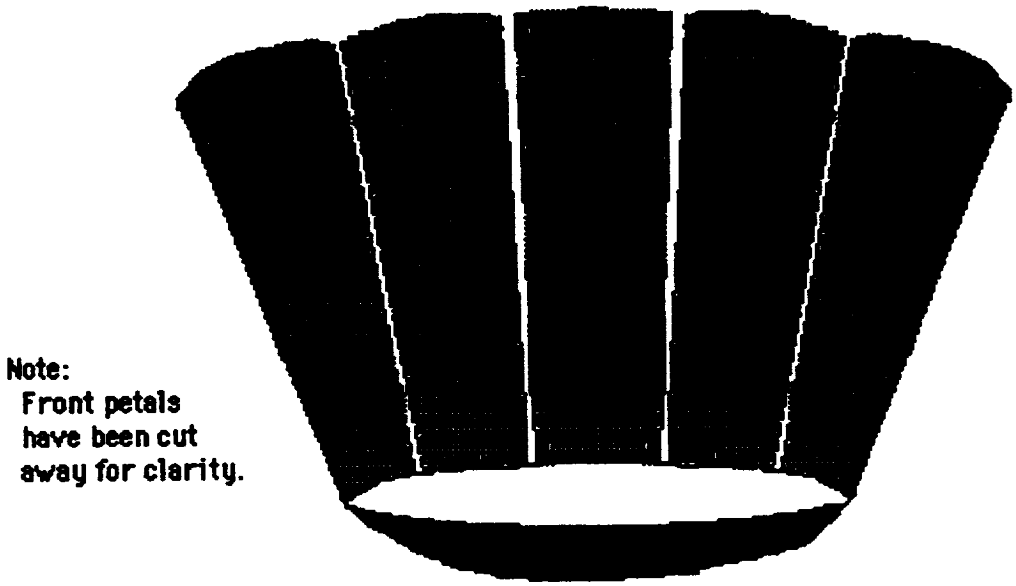


Figure J-5. Vegetable steamer configuration concentrator.

Advanced Receiver/Thermal Energy Storage (TES) Description

Solar heat receiver concepts with integral TES are dependent on the design of the heat engine. This results in different receiver concepts for CBC and SC engines. Goals for an advanced receiver are summarized in Table J-2 (Ref. J-6). The following sections describe concepts evaluated during a NASA LeRC funded program.

TABLE J-2. ADVANCED RECEIVER GOALS

Parameter	Goal
interfaces	•optical compatibility with selected concentrator •appropriate engine interface
operating temperature range	•978 to 1,228 °K
operational life	•7 to 10 years in LEO
scalability	•goal - 25 kWe and above
robustness	•tolerance to asymmetric solar flux from concentrator •tolerance to micrometeoroids •tolerance to external insolation due to pointing and tracking errors/malfunctions
thermal energy storage void management	•elimination/minimization of thermal ratcheting

Sundstrand CBC Receiver/TES Description. This concept uses a single annular heat pipe which contains the heat engine gas tubes and phase change LiF canisters, as seen in Figure J-6 (Ref. J-6). A wedge-shaped TES canister design is used to minimize thermal ratcheting effects (see Fig. J-7, which is from Ref. J-4). The annular heat pipe has the ability to spread thermal loads both axially and circumferentially. Thus, concentrator types with asymmetric flux patterns, e.g. offset paraboloid, are compatible with this receiver.

The heat pipe inner diameter is formed by the cavity side wall and the outer wall coincides with the TES canister outer wall. The annulus is subdivided into eight compartments necessary for ground testing as defined by the maximum pumping height of the proposed heat pipe wick structure and sodium properties. Each compartment is hermetically sealed from the others. The heat pipe provides a high thermal conductance path between the solar energy

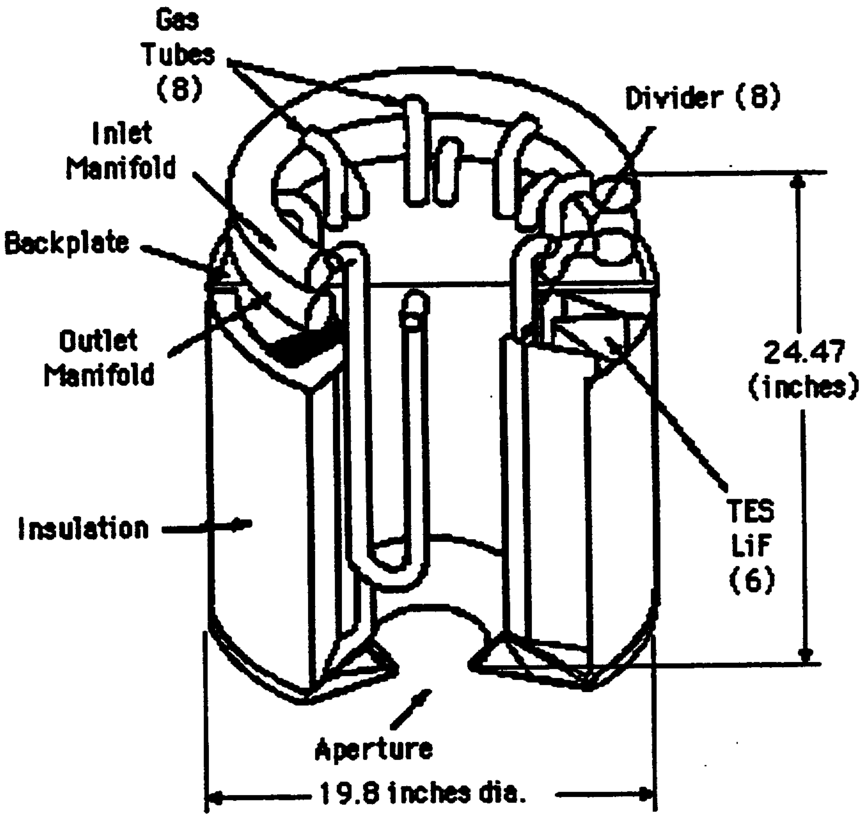


Figure J-6. Advanced CBC receiver/TES concept.

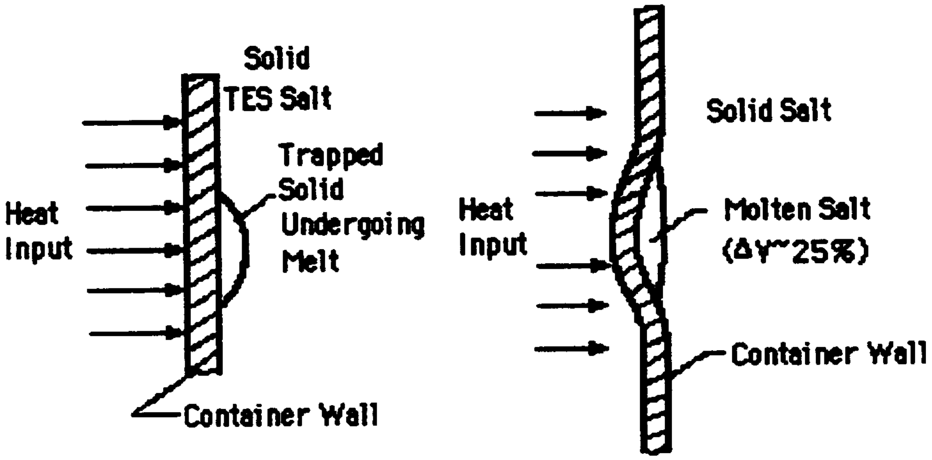
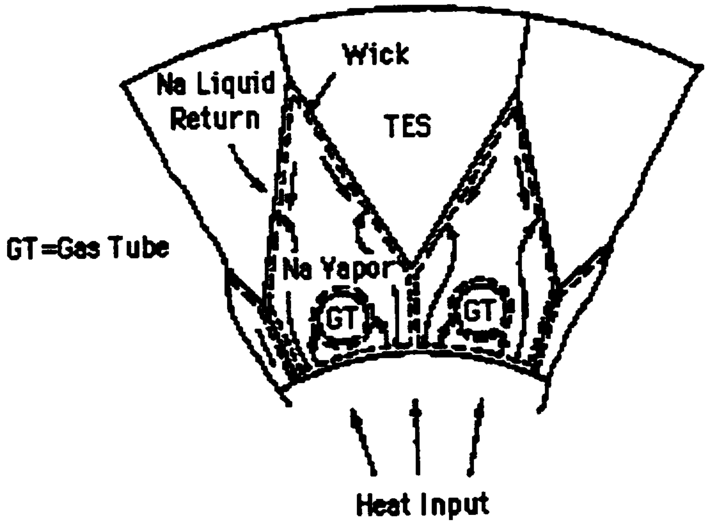


Figure J-7. Canister distortion caused by thermal ratcheting.

absorbing surface, the TES, and the gas tubes. During eclipse, the heat pipe transfers energy from the TES to the gas tubes. The heat pipe is required to smooth the 25:1 axial incident flux variation on the receiver side wall. The heat pipe insures near isothermal heat addition to the TES canisters and the gas tubes. Additionally, all heat pipe surfaces will remain isothermal, thus reducing thermal stresses. Heat pipe operation modes for insolation and eclipse portions of an orbit are shown in Figure J-8 (Ref. J-7).



a) Daylight operation.

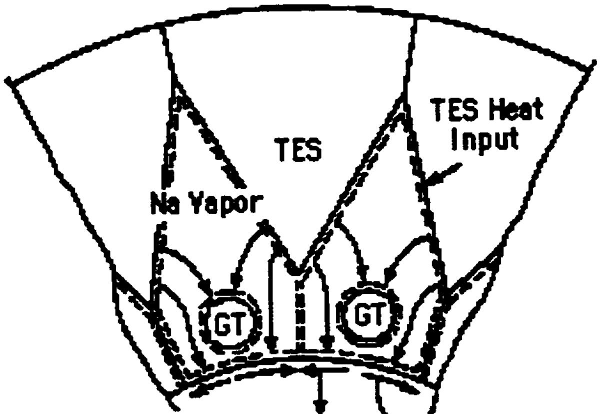


Figure J-8. Advanced CBC receiver/TES concept operation.

Liquid salt completely fills the TES cross section at the end of insolation. As energy is extracted during eclipse, a solid/liquid front marches radially outward from the heat removal surfaces. A void will form on the insulated surface (under zero g). During the subsequent insolation period, a liquid front would uniformly cover the entire heat surface. LiF has a large volume change upon fusion (about 30%). The volume increase associated with melting will pressurize the enclosing structure. Because the PCM expands upon melting, the liquid would be required to communicate with the void to prevent thermal ratcheting problems. In order to prevent mechanical TES canister failure, the canisters are designed to insure the void is nearly contiguous to the heat addition surface. The wedge-shaped TES canister provides minimum mass and good void control.

Sanders Associates SC Receiver/TES Description. A technology option for the receiver/TES is the cavity heat pipe receiver, as seen in Figure J-9 (Ref. J-8). The cavity heat pipe design relies on the heat pipe principle for energy transfer to the engine and TES. The receiver/TES would be similar to the Sanders Cavity Heat Pipe Receiver. The Sanders Associates receiver design includes TES within the receiver proper (integral TES). This concept has a low specific mass of 27 kg/kWe, which is 60% lower than the SSF CBC receiver value. This concept also has only 5% of the volume of the baseline SSF receiver.

Cavity heat pipe (CHP) describes a concept for integrating conventional heat pipe technology with a solar receiver cavity. In the Sanders' CHP concept, a single sodium vapor-filled cavity couples the receiver absorber, the engine heater head, and the PCM module interface, as seen in Figure J-10. During the solar period sodium evaporates off the back side of a domed solar absorber and condenses simultaneously in the engine heater head and internal PCM container modules. A continuous wick/artery system connects the heater head condenser, the PCM modules and the absorber/evaporator zone. During the orbital eclipse, the sustained thermal release from the PCM continues to supply vapor to the heater head. Liquid returning from the condenser sections flows through a low fluid resistance sealed volume to the evaporating surfaces. It is vented to the vapor cavity only through the various wick pores of the dome, PCM modules, and heater head. This feature allows for the several different wick structures to be reliably connected during the assembly process.

The surface areas and geometry of the dome, the PCM modules, and the heater head are the critical factors in heat heat exchanger design. The CHP, unlike conventional tubular heat pipes, allows these three heat exchangers to be designed independently to best accommodate the specific stress and thermal performance requirements. The three heat exchangers must operate at 1039 °K ± 20 °K (due to fatigue limits of wall materials).

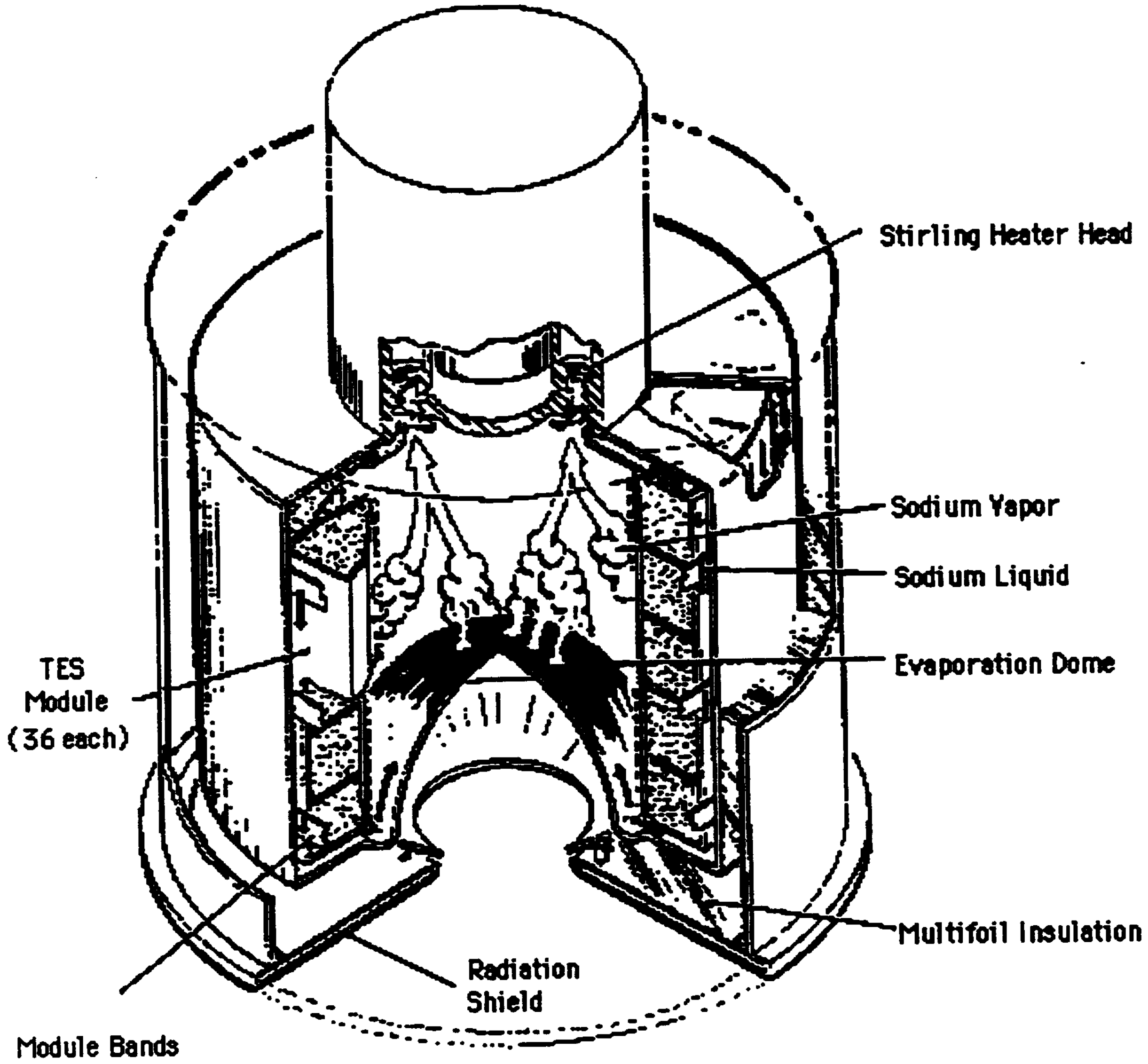


Figure J-9. Sanders advanced Stirling receiver/TES concept.

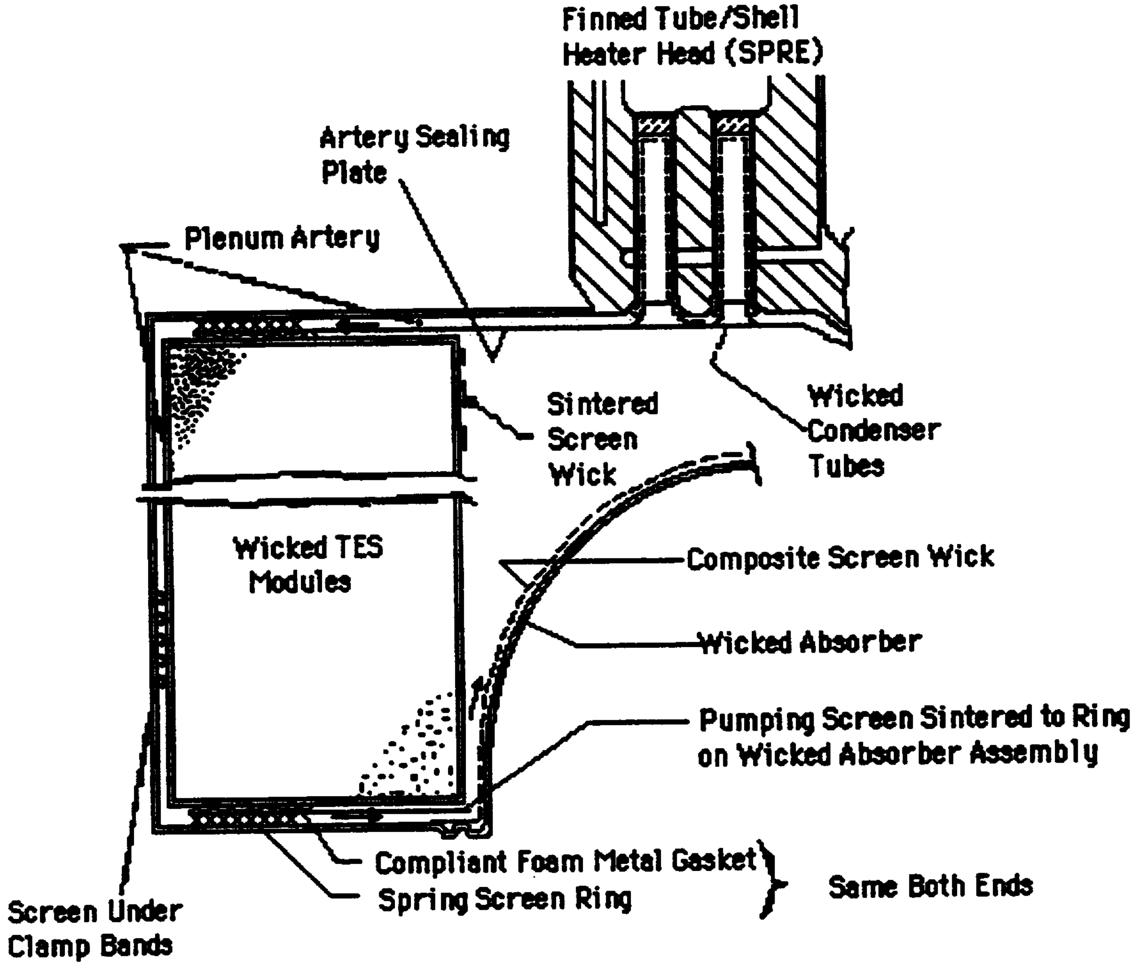


Figure J-10. Advanced Stirling receiver/TES concept cross-section.

Closed Brayton Cycle Power Conversion Unit Description

This PCU would be the same as previously described in Appendix I.

Stirling Power Conversion Unit Description

This PCU uses a regenerative closed cycle piston engine with cyclic recirculation of the working gas. There are typically two pistons per cylinder. The displacer shuttles working gas from the expansion (hot) space to the compression (cold) space through the heater, regenerator, cooler, and then back again. The power piston, integrated with a linear alternator, compresses the gas when cold and allows gas expansion when hot. The Stirling PCU has a dynamic balance unit or opposed single cylinder engine to minimize dynamic disturbances.

The major advantage of the free-piston Stirling engine over the kinematic Stirling engine is that it has only a few moving parts (a displacer piston and power piston; Ref. J-9). It can use noncontacting gas bearings and can be hermetically sealed, thereby increasing the potential for high reliability and very long life. Free-piston engines have no mechanical mechanism coupling the reciprocating elements to each other. Instead, free-piston engines elements are coupled by the forces exerted by the working fluid. The engine will resonate at a frequency determined by the combined dynamics of the piston and the displacer.

This heat engine has a high thermal efficiency (the ideal efficiency is equal to the Carnot efficiency). The efficiency remains almost constant over a wide range of operating conditions. High operating pressures are required ($>1.5\times10^7$ N/m^2 for power outputs greater than 40 kWe per Ref. J-10). Low molecular working fluids such as helium must be used. This heat engine is currently limited to less than 100 kWe per module. Power output is almost directly proportional to speed and can be regulated by the applied alternator voltage. Power output can also be controlled by adjusting the working fluid pressure level. The linear alternator has a low output power factor (high inductance) and is hermetically sealed. Power conversion is required to convert from the alternator output (50-200 Hz) to the desired frequency and to correct the power factor.

Figure J-11 shows the current NASA space Stirling engine design (Ref. J-21). The single-cylinder free piston SSE (with dynamic balancer) is being designed for a 1050 °K hot temperature and an engine temperature ratio of 2.0.

The key advantage to using a Stirling engine over a Brayton engine is an increase of cycle efficiency and resultant decrease in system mass (smaller concentrator, receiver, and radiator) of about 20% (Ref. J-11) without increasing the peak cycle temperature.

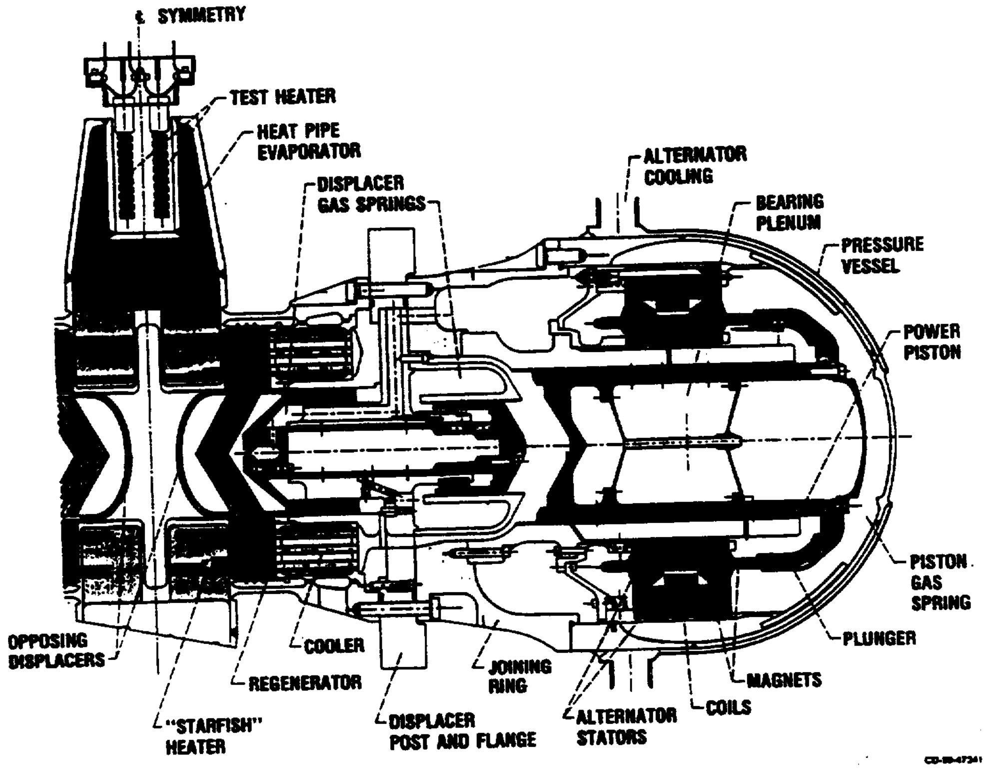


Figure J-11. NASA space Stirling engine.

Heat Rejection Subsystem Description

The heat rejection assembly selected for SSF consists of a pumped liquid loop which collects the waste heat from the PCU and electrical components and transfers it to a radiator, and the radiator itself (Ref. J-12). This concept was selected since it has a somewhat lower mass than a heat pipe radiator. However, a pumped loop radiator has lower reliability (due to the potential for puncture) than a heat pipe radiator and would be unsuitable for applications which do not allow servicing. It is expected that advanced heat pipe radiators will be lower in mass than current heat pipe radiators. For added flexibility in mission selection, a heat pipe radiator was tentatively selected for advanced SD power systems.

A conventional heat pipe radiator approach was studied for the SD option of SSF (Ref. J-11. This heat-rejection radiator concept consists of individual heat pipes with fins to reject heat to space. For CBC cycle heat rejection, an intermediate transport loop fluid (FC-75 coolant) was selected to take engine waste heat to a heat exchanger. The transport fluid cools from 423 °K to 296 °K in this design. The baseline heat rejection system design uses, methanol-titanium dual-slot heat pipes in the radiator panels and wiffletree clamping mechanisms to attach the radiator panels to the heat exchanger. The heat exchanger is a single pass unit with a standard plate-fin design. Figure J-12 (Ref. J-11) shows a cross-section of the CBC baseline heat pipe radiator panel design.

The use of methanol limits heat pipe operation to a maximum temperature of about 450 °K. Mercury heat pipes would be more suitable for Stirling engine space power systems.temperatures, which are being designed to reject heat at temperatures between 500 °K and 600 °K. The operating temperature range for mercury heat pipes is between 513 °K and 823 °K. A mercury heat pipe would require steel for the heat pipe material and beryllium fins (Ref. J-13).

A deployable heat pipe radiator design option was also studied for the SSF SD option. This radiator uses a scissors type deployment mechanism as seen in Figure J-13 (Ref. J-11).

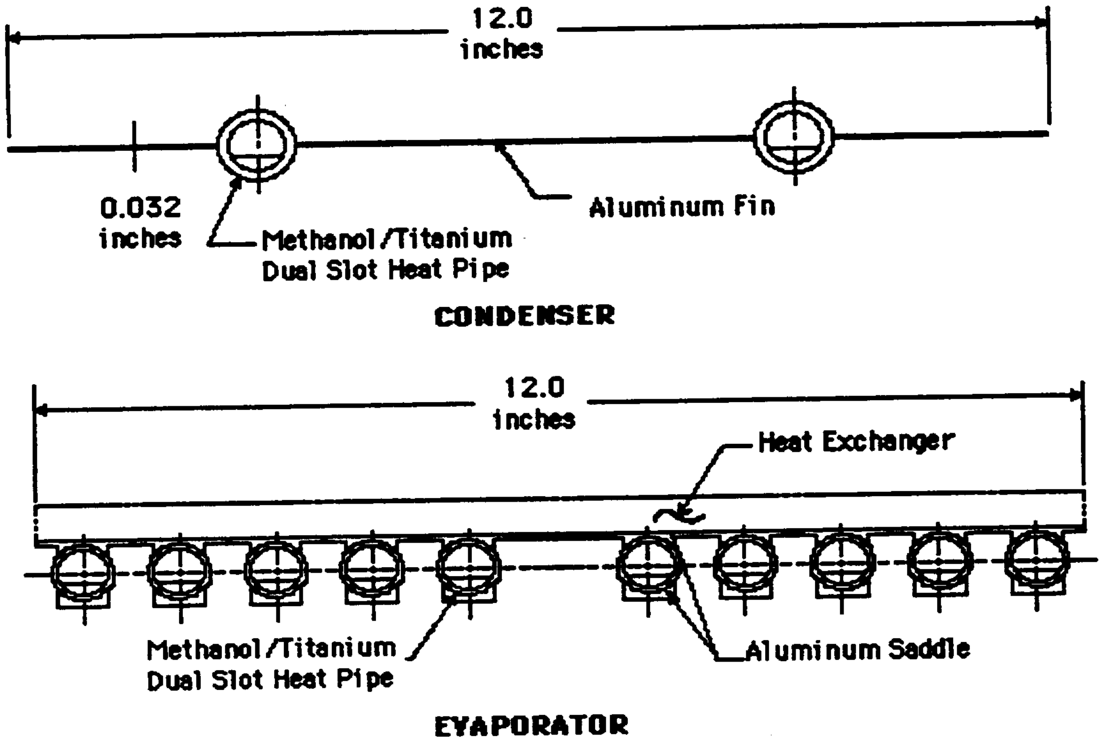
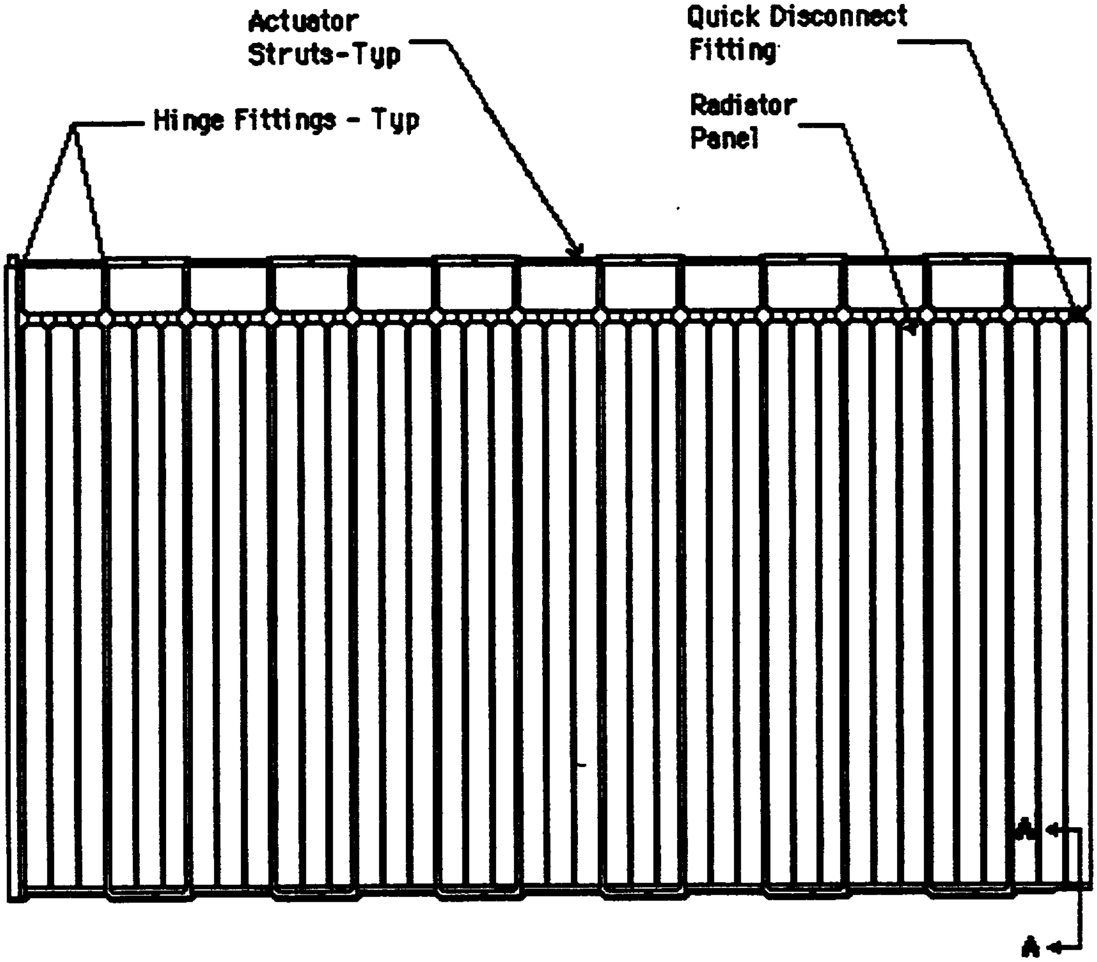
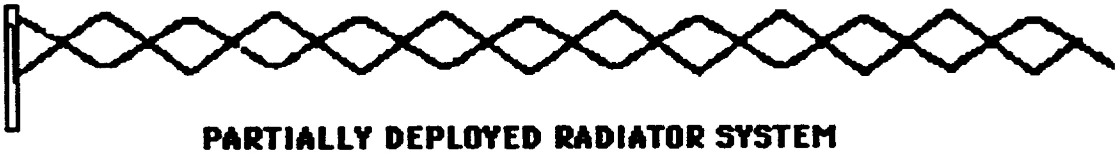


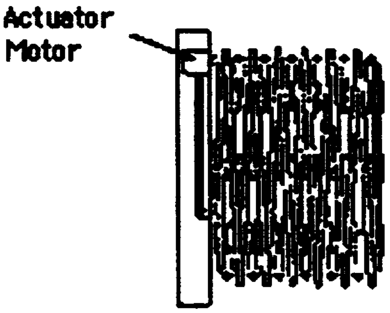
Figure J-12. CBC methanol/titanium heat pipe radiator panel cross-sections.



DEPLOYED RADIATOR SYSTEM



PARTIALLY DEPLOYED RADIATOR SYSTEM



RADIATOR SYSTEM IN STOWED POSITION

Figure J-13. Deployable heat pipe radiator.

An advanced radiator concept called the "Folding Panel Radiator" is being developed by Space Power Incorporated (SPI). This concept is illustrated in Figure J-14. This concept is based on a pumped binary lithium/sodium potassium (Li/NaK) loop and was designed to avoid the use of mercury, which is highly toxic. It combines the high heat capacity and low pumping power of Li (melting point 452 °K) with the liquid pumping capability of NaK, down to its freezing temperature of 225 °K. This concept utilizes high conductivity fins for the heat transport.

The Folding Panel Radiator works as follows. During startup (Li frozen), liquid NaK would be pumped through the inner cores of radiator tube passages in hydraulic contact with the frozen layers of Li coating the inner passage surfaces. As the NaK is heated during power system startup, it will eventually melt the solid Li shells by direct contact forced convection heat transfer, progressively mixing with the NaK to form the all liquid Li/NaK coolant. Conversely, on shut-down of the power system, the molten lithium with its higher freezing point will selectively "cold trap" or freeze on the inner passage surfaces as their temperatures drop below 452 °K, while the NaK continues to be pumped in its liquid state through the inner cores of the radiator passages.

Other potential advanced radiator concepts include the use of an integrally woven carbon-carbon (C-C) heat pipe/fin with a metal liner or a ceramic fiber fabric/metal liner heat pipe. The metal liner provides protection from the working fluid. Water can be used as a working fluid below 500 °K. Titanium or copper foil would be material options when using water.

Electrical Equipment Assembly (EEA) Description

This electronic/electrical assembly contains a variety of hardware which is mounted on a cold plate. For thermal control of this hardware, cooling liquid from the heat rejection subsystem circulates through this cold plate. The EEA contains all of the controls for the SD power module. This includes pointing controllers which assure that the concentrator is pointed to within 0.1 degree of the sun line. Control of the PCU includes control for voltage and speed, startup and shutdown sequencing, and mass flowrate of the gas (power level). Control of the mass flowrate assures maintenance of proper temperatures in the receiver. Pump motors for the heat rejection assembly and their drive electronics are also in the EEA. Finally, the EEA includes power conditioning which converts the nominal 50-200 Hz power from the alternator to whatever the user needs.

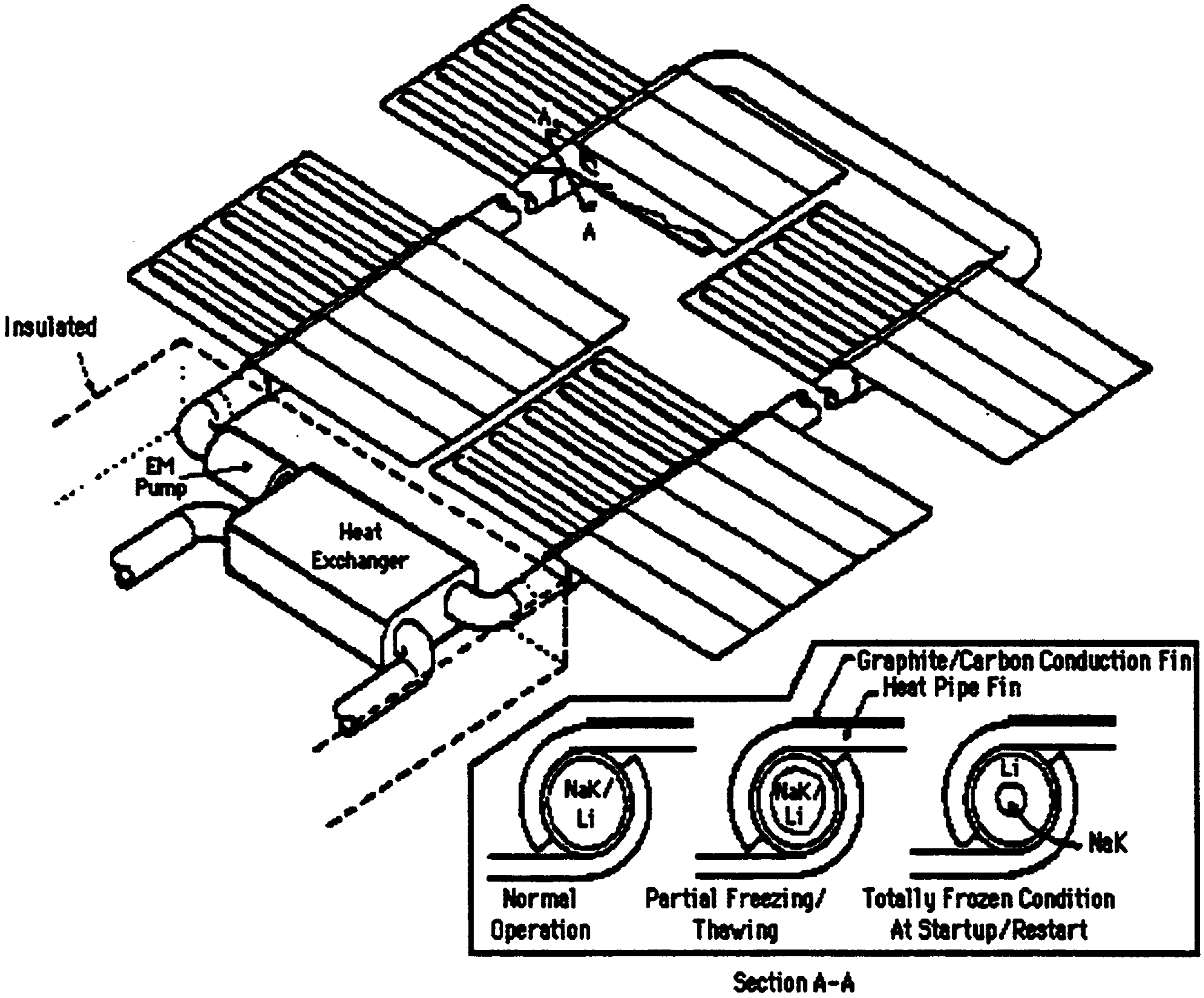


Figure J-14. Pumped Li/NaK Binary Loop Radiator Concept Under Development at SPI.

Parasitic Load Radiator (PLR) Description

The PLR, under control of the EEA, is used to dissipate excess electric power from the PCU and thereby control the speed of the PCU as the user load varies. The PCU provides a specific power output at each case of solar input (orbits vary). Variation in user demand below the PCU capability creates the necessity to dissipate any excess electrical power.

TECHNOLOGY ISSUES

This power system concept is an evolution of the nearterm SSF SD power system. Technology advancement is foreseen for several of the major subsystems: the concentrator, the receiver/TES, the Stirling engine PCU, and heat rejection subsystem. The following sections discuss the remaining technology issues for each of these subsystems.

Concentrator Technology Issues

Concentrators are a critical element of SD systems due to their size and mass. Future solar concentrators must be low in mass, be packageable into a small volume for the launch phase, and be automatically deployable on orbit without astronaut assistance. Innovative designs have been developed to a conceptual or prototype level which meet these requirements. In addition to these system level issues, it is important that highly specular, efficient reflectors be developed which can survive the LEO environment and the orbital day/night cycles for more than 10 years. The solution to this later problem lies in the selection of suitable LEO environmental resistance materials and the reflector substrate design. Significant work has been made with reflector materials and fabrication techniques.

Low mass reflective concentrator designs have been developed as a result of recent NASA funded programs (Ref. J-1). These concepts, as seen in the concept description section, utilize innovative geometries and materials to achieve low mass. Materials such as aluminum and honeycomb structures have reduced mass. Minimization of structure support has also reduced concentrator mass to meet NASA goals. Self-deployment has been demonstrated for the SAIC membrane concentrator and the NASA Sunflower approach. These new designs also appear scalable to 10s of kilowatts electric. Remaining issues appear to be development of full scale concentrators and flight testing. Flight testing could resolve any remaining issues regarding deployment of large concentrators and life issues. In addition, integration issues with the receiver and PCU need to be resolved through testing..

Highly specular reflection is achieved by depositing a highly reflective material on a very smooth substrate surface. The ability to produce a very smooth surface is a critical problem whose solution has not been straight forward. However, recent NASA programs, both inhouse and contractor, have mostly resolved this issue. The use of a leveling layer (a thin coating of a polyimid or microsheet glass) provides a very smooth surface for the reflective layer. Issues still open include the handling and attachment of thin sheets of microsheet glass to the substrate (microsheet glass would give better performance than a polyimid leveling layer). Another issue is the attachment of honeycomb facesheets. Shrinkage of the adhesive may cause an imprint of the honeycomb or markoff in the surface (slight dip in surface at the attachment points). Soldering may resolve this attachment issue. An all metallic substrate was chosen over a composite structure since the composite fiber contour tended to print through the surface (Ref. J-5). The issue of longterm compatibility with the LEO environment has been resolved by selection of a SiO_x protective layer and aluminum reflecting layer. Future work needs to be done to scale up the size of reflectors and demonstrate the appropriate manufacturing techniques.

Receiver/TES Technical Issues

The second challenge will be the receiver/TES. The chosen PCM, LiF, has a solid to liquid volumetric ratio of about 3:4. LiF also has a higher melting point than the $LiF\text{-}CaF_2$ PCM selected for SSF which complicates the materials problems. The challenge is to achieve an overall TES design concept which does not subject the canisters to undue stress and thermal ratcheting damage at the onset of heat addition to the frozen PCM (occurring at sunrise of each orbit). A major factor in seeking a solution to the challenge of TES design was to achieve a low mass design.

Distortion of the TES canister is caused when a wall of the canister is exposed to uneven heating. As the TES material adjacent to the wall melts, it expands and forces the canister wall material to be stressed beyond its yield point. Over time, the wall may be gradually stretched to the point of failure. The SSF design avoided this problem by encasing the TES material in small canisters which is a conservative but higher mass approach. The advanced CBC receiver avoids the problems of hot spots and thermal ratcheting through the use of a heat pipe cavity. in daylight operation, the solar receptor is the heat pipe evaporator and the combination of the TES wedge-shape canisters and heat engine gas tubes serve as the heat pipe condenser. Since the heat pipe is inherently an isothermal device, the heat pipes redistribute the incoming flux evenly to the TES canisters and the heat engine gas tubes. This results in uniform melting of the TES material and no hot spots are generated.

Thermal conductivity of the PCM had been another issue in receiver programs. The SSF design utilized thick metal walls to conduct heat rather than depending on the low conductivity PCM. Thermal conductivity enhancement using a nickel foam insert was found to be ineffective and probably detrimental since the PCM appears to be transparent to thermal radiation. The use of a wedge-shaped TES and heat pipes to smooth the heat flux results in the proper PCM void management necessary to minimize temperature gradients in the TES.

TES containment materials compatibility, which had been an issue in earlier NASA programs, has been resolved (Ref. J-16) due to extensive testing at NASA LeRC (9,000 hours with LiF and 15,000 hours with $LiF\text{-}CaF_2$). Materials such as Haynes 188 and Hastelloy B have proved to be satisfactory containment materials. Although this issue has been resolved, it is critical that the maximum temperature of the metal be kept below 1050 °K to maintain the creep strength of these materials. Otherwise, more costly refractory materials such as Niobium would be required.

Additional TES design issues include operation in zero-g and one-g testing. Flight testing of the receiver/TES unit is desirable to resolve performance issues, such as void management, which are difficult to resolve with one-g testing (Ref. J-15). Another issue is the tight ± 20°K temperature tolerance for the receiver.

Stirling PCU Technical Issues

The Stirling PCU should not present any unsurmountable technology challenges, since the technology will be similar to the 25 kWe Stirling engine currently in design and development by NASA LeRC for space application (Ref. J-11). The success of the Stirling engine development depends on research efforts involving heat pipes (unconventional design for heater head), bearings, superalloy joining technologies (verify technology for Utemid 720), high efficiency alternators, life and reliability testing, and predictive methodologies. Key technical issues include heater head life, alternator life, and engine efficiency. The use of an opposed engine configuration for the PCU eliminates the need for a dynamic balancer. The low cold end engine temperatures will reduce the challenges in the alternator design.

TECHNOLOGY ASSESSMENT

The technology bases were assessed for the following major SD subsystems:

- concentrator;
- receiver/TES;
- heat rejection subsystem (HRS);
- Stirling PCU; and
- electrical equipment assembly (EEA).

The technology readiness of each assembly was evaluated using the NASA technology readiness levels. These evaluations are summarized in Table J-3 which shows that the advanced SD power system has subsystem technology readiness levels ranging from 3 to 6, depending on the particular subsystem. The technology base for each assembly is briefly discussed in the following sections.

TABLE J-3. ADVANCED SD TECHNOLOGY ASSESSMENT

Subsystem	Technology Readiness Level	Comments
Concentrator	3	NASA Advanced SD program
Receiver/TES	3	NASA Advanced SD program (Sundstrand, Sanders work)
EEA	5 - 6	Space Station Freedom, SSME technology
HRS	3 / 4 / 8*	DIPS, SP-100, Shuttle, SSF programs
Brayton PCU	4.5	BRU-F,BIPS
Stirling PCU	4	SPDE,SPRE,CTPC; Stirling space cryocoolers

*3 for carbon-carbon heat pipes/4 for all metal heat pipes/8 for pumped loop radiator.

Concentrator State-of-the-Art

The Advanced Solar Dynamics Power Systems Project was started by NASA LeRC in 1985 for the purpose of advancing the technology of SD power systems for space applications beyond 2000 (Ref. J-1). Since then, technology development activities have been initiated for the major components and subsystems such as the concentrator, heat receiver and engine, and the radiator. Current SD activities at NASA are being funded at a low level due to the cancellation of the SD growth option for SSF.

The Advanced Solar Concentrator (ASC) Program began in 1986 and is a continuing effort involving both contractual and NASA inhouse efforts (Ref. J-2). This program has been successful in developing both low mass, self-deployable concepts (up through the preliminary design stage in some cases with some prototypes having been developed) and manufacturing techniques for high quality reflectors. It is possible that this technology will be utilized for SSF (if the SD option is revived) as well as other earth orbital applications.

The general objective of the ASC Program (Ref. J-1) is to develop the enabling technology for parabolic concentrators that will power advanced SD power systems in LEO. The specific objectives are to:

- Identify at least one on-axis low mass concentrator concept that will be packageable into a small volume for launch and automatically deployable on orbit.
- Identify the materials and construction techniques that will enable the concentrators to operate in LEO environment for more than 10 years.
- Identify techniques and materials for producing reflecting surfaces with high reflectance and specularity.
- Build and ground test a reduced scale (2-meter diameter) concentrator of the most promising concept, to demonstrate some of the technology.

The first element of the ASC Program was Concentrator Concepts Identification and Selection. Four contractual efforts were awarded by NASA LeRC and completed. In addition, there is an on-going cooperative effort between Cleveland State University and NASA LeRC to continue development of the Sunflower concept.

The second element of the ASC Program is Supporting Research and Technology which involves materials identification and development of manufacturing techniques for reflective layers, leveling layers, and protective coatings. This is an inhouse NASA LeRC effort (Ref. J-2).

Concentrator concept contracts were awarded to the following companies (these concepts were described in more detail in the concept description section):

- Entech Inc., Dallas, Texas - This contract was awarded in 1985 and completed in 1988. This concept involved an assembly of thin refractive panels on a dish-shaped support frame (Fresnel lens).
- Acurex Inc., Mountain View, CA - This was a 24 month effort starting in 1988 (Refs. J-3 and J-14). This contract involved a concentrator made up of rigid pie shaped, parabolically contoured, reflector panels. They completed a preliminary design study and fabricated some panels.
- Harris Corp., Melbourne, FL - This was a 12 month effort starting in 1988. Harris proposed a splined radial composite panel.

- Science Applications International Corp., San Diego, CA - This was a 12 month effort starting in 1988. SAIC began development of a stainless steel membrane concentrator similar to terrestrial concentrators which they have built. SAIC built a 2 meter prototype and tested a deployment approach.

Recent space solar concentrator efforts sponsored by NASA LeRC have involved development of all metal concentrators. Solar Kinetics, Inc. in Dallas recently completed a NASA LeRC funded program to develop an all metal concentrator, including appropriate fabrication techniques (Ref. J-20). Solar Kinetics bonded aluminum face sheets to both sides of a 0.625 cm aluminum honeycomb. The honeycomb sandwich provides a very stiff and low mass concentrator. Leveling of these surfaces was accomplished by the application of a thin monomer coating over the front face sheet. Then a very thin aluminum reflective coating was applied. Finally, a layer of Al_2O_3 protective coating was added.

NASA LeRC is also involved in a cooperative effort with Cleveland State University (Advanced Manufacturing Center) to build a 2 meter diameter prototype concentrator based on the Sunflower concept developed at TRW in the 1960's (Refs. J-1 and J-6). The preliminary design of the prototype is completed. The radial panels are doubly curved to a parabolic shape.

NASA has been developing fabrication processes for the 2 meter prototype (Ref. J-5) and this work is nearly complete. To date, NASA has built and coated petals for the prototype. The petals are made using stretch forming of aluminum sheets to create the petals (Solar Kinetics uses a similar hydroforming approach). A good reflective surface was achieved. The Sunflower design has achieved a specific mass of 1.5 kg/m^2 and a surface accuracy of 1.0 millirads. NASA has examined two different approaches for bonding the aluminum face sheets to the aluminum honeycomb. Adhesive bonding works well with minimal markoff (imprint of honeycomb on face sheet due to shrinkage after curing). NASA has also experimented on a small scale with soldering the face sheets to the core and have achieved decent results. NASA has also been developing techniques for applying leveling coatings. This has included the identification of a higher temperature polymer (polyimid) coating. NASA has also experimented with microsheet glass which provides the best surface. A coating of 0.08-0.1 mm has been bonded successfully to aluminum. NASA will be doing additional work with microsheet glass concepts (i.e., forming and attaching thin, large sheets). Also, NASA has fabricated test coupons of both microsheet glass and Eccocoat EP-3 coating for an upcoming shuttle flight to evaluate materials compatibility with atomic oxygen (Ref. J-2).

Advanced CBC Receiver/TES State-of-the-Art

The major objective of the receiver portion of the Advanced SD Technology Program is to eliminate the cyclic distortion of the TES canister and at the same time reduce the mass of the receiver to one half that of the SSF design while retaining reliability and long life. Both advanced CBC engine and SC receiver concepts have been examined to meet this objective.

The advanced CBC receiver, designed by Sundstrand Corporation, was previously described. This concept avoids the problems of hot spots and thermal ratcheting through the use of a heat pipe cavity which redistributes the solar flux evenly to the TES canisters and the heat engine gas tubes. The Sundstrand receiver design has a specific mass which is well within NASA's goal for a low mass receiver concept. The ultimate goal of the Sundstrand portion of the ASD Program is to develop a 7 kWe CBC heat receiver.

Phase 1 of the NASA Advanced Solar Dynamic (ASD) Brayton Heat Receiver Program was completed by Sundstrand in mid-1991 (Ref. J-15). This effort was a 2 year program. This program included two tasks: (1)design of critical technology experiments; and (2)performance of critical technology experiments (Ref. J-6).

Critical technology experiments were performed by Sundstrand on TES modules (Ref. J-7). TES canisters were designed to address the void control, thermal conductivity enhancement (TCE), and metal oxide issues. A wedge-shaped design was chosen because it has the lowest mass, an acceptable internal temperature gradient, and good void control. The modules, wedge-shaped canisters containing LiF, were designed to minimize the mechanical stresses that occur during the phase change of the LiF. Nickel foam inserts were placed in two of the test canisters to provide TCE and to distribute the void volume throughout the canister. A procedure was developed for reducing the nickel oxides on the nickel foam to enhance the wicking ability of the foam. The canisters were filled with LiF and closure-welded at NASA LeRC. The canisters, one with a nickel foam insert, the other without an insert were then thermally cycled (simulations of worst-case LEO environment with a temperature variation from 978 to 1240 °K) in various orientations in a fluidized bed furnace at Sundstrand. Computer-aided tomography was successfully used to nondestructively determine void locations in the canisters. Finally, canister dimensional stability was measured after thermal cycling.

The objectives of the TES canister experiments were the following:

- determine the ability of the baseline TES configuration to provide void control;
- determine the wicking ability of a metal foam TCE;
- find the qualitative effects of a metal foam TCE on cycle ΔT; and
- determine the orientation sensitivity of the configuration in a terrestrial environment in which a receiver would have to operate for qualification purposes.

Sundstrand found very similar results for one-g testing of both canisters. NASA LeRC has found that for LiF most of the heat transfer through the TES is by radiation. Lithium fluoride is transparent to thermal radiation. Thus, the use of an insert may increase thermal conduction through the salt, but will decrease thermal radiation. Thus, the use of an insert is now considered undesirable by NASA, since no suitable transparent materials exist.

During the Sundstrand testing, the TES canisters maintained mechanical integrity, having dimensional changes within acceptable limits, in spite of the rigorous temperature profiles. No indication of thermal ratcheting was found. Also, temperature profiles and void locations were predictable and repeatable. Computed tomography was verified as an accurate nondestructive inspection technique.

Sundstrand was awarded a new 2 year contract at the end of 1991 (Refs. J-6 and J-13) to continue work with its CBC receiver concept. Sundstrand will design, fabricate, and test one segment of their heat pipe cavity receiver. This experiment will simulate a section of the cavity including the cylindrical solar flux receptor, a TES canister, a heat engine gas tube (for removing heat), wicking, and the heat pipe working fluid. Operation and performance of the cavity will be observed with carefully located instrumentation over the equivalent of LEO cyclic operation.

NASA has a program with the Navy (ORL) to design a hybrid SD/PV power system for a communications satellite (Ref. J-15). This program has been going on for about 1 year so far and has looked at the interface requirements between the satellite and the power system. This program will result in a conceptual design for an advanced satellite power system.

Stirling Engine Heat Receiver State-of-the-Art

No work is currently being done on heat receiver for space Stirling engine power systems. Some receiver work is being done for a 25 kWe terrestrial SD power system for the DOE. This work may or may not be applicable to space power systems.

Past work was done by Sanders Associates on a cavity heat pipe Stirling receiver as part of the ASD Program (Refs. J-8 and J-13). This concept was described earlier. A conceptual design was developed by Sanders. Sanders performed some comparison weight studies of their concept with the SSF CBC baseline receiver as well as some stress/fatigue analyses. Sanders build a heat pipe cavity with no TES. The hardware was delivered to NASA LeRC, but was never tested.

NASA LeRC currently is sponsoring a contract for work on an advanced Stirling solar receiver (Ref. J-22).

Stirling PCU State-of-the-Art

There is no known experience of Stirling dynamic power conversion systems having ever been flown in space. However, a database does exist for numerous Stirling cyrocoolers which have flow aboard space missions. Stirling cyrocoolers operate using the same basic Stirling cycle as does a power converter, and the basic components are essentially the same (Ref. J-16). A power converter operates in reverse of a cryocooler. In addition, power conversion units operate at much higher temperatures than cryocoolers. All of the space cryocoolers used kinematic Stirling machines. Cryocoolers flown to date have operated from 6 days to 6 years All experienced some performance degradation. New missions (from May of 1991 through 1998) will use a new 5 year life Stirling cryocooler. The fact that Stirling technology in the form of cryocoolers has successfully flown in space, implies that Stirling technology in the form of power converters should also be suitable for a space environment.

NASA LeRC is sponsoring two contracts to develop Stirling engines for terrestrial applications. One contract is with Stirling Technology Company (STC) and Allied Signal. The other contract is with Cummins Engine, Sunpower, and CFIC. These programs will develop an engine with a peak temperature between 950 and 975 °K.

Development of a Stirling engine for space applications is being conducted in the NASA LeRC Free-Piston Stirling Space Power Converter Technology Program (Ref. J-16). This work is being conducted under NASA's Civil Space Technology Initiative. This program originated in 1983 as part of the SP-100 program.

Only a few organizations in the United States are currently developing free-piston Stirling technology (Ref. J-9). These include Sunpower Inc. of Athens, Ohio, Mechanical Technology, Inc. (MTI) of Latham, New York, STC of Richland, Washington, DOE Oak Ridge National Laboratory and NASA LeRC. Free-piston Stirling engines have been designed and built in power levels ranging from fractional kW up to 25 kWe. By 1982, a few free-piston engines had been built in the 3-4 kW size range. The free-piston/linear alternator Stirling program began at NASA in 1985 as technology demonstration. The NASA space power demonstrator engine

(SPDE) was designed and built in less than 18 months by MTI, and had delivered more than 22 kW of output power by 1986. Currently, power output of the MTI design is in good agreement with the original predictions at a temperature ratio of 2.

In 1986, the 650 °K Space Power Demonstration Engine (SPDE) developed 25 kWe of engine P-V power. The SPDE was a dual-opposed configuration consisting of two 12.5 kWe converters. After this successful demonstration, the engine was cut in half. One half is undergoing testing at NASA-LeRC and the other half has completed testing at the contractors site in Latham, New York. These power converters are now called Space Power Research Engines (SPRE) and serve as test beds for evaluating key technology areas and components. A power output of 11.2 kWe at an overall efficiency of 195 (electrical power out/heat in) has been achieved with these engines. This is approaching the SPRE design goal of 12.5 kWe and an efficiency greater than 20%. Much of the recent engine work was focused on producing experimental data for validating the HFAST engine performance code being developed by MTI as part of a NASA contract. Two areas of study have been : 1)the sensitivity of the engine performance to the displacer seal clearance, and 2)the effects of varying piston centering port area. Testing at the end of 1991 was to verify potential performance improvements suggested by HFAST studies conducted by MTI.

Development of advanced Stirling power conversion is proceeding due to the high efficiency and long-life potential of free-piston, linear alternator concept first invented in the U.S. in 1963. The Stirling technology development contract with MTI specifies a superalloy power converter which must be capable of operating for 60,000 hours at a hot end temperature of 1050 °K and a cold end temperature of 525 °K (temperature ratio of 2). This converter could be considered for operation with the SP-100 reactor operating at 1100 °K or for solar dynamic power conversion.

NASA LeRC is now funding development of the Component Test Power Converter (CTPC). The CTPC is an intermediate step providing an incremental evaluation of critical technologies to be incorporated into the 25 kWe Space Stirling Power Converter. Those critical technologies have been identified as: bearings, materials, coatings, linear alternators, mechanical and structural issues, and heat pipes (Ref. J-16). The CTPC "cold" end provides a test vehicle to identify problem areas and develop individual "cold" end components and sub-assemblies. Specifically, the CTPC "cold end" will provide an early test and evaluation of the mechanical design at 525 °K, bearing operation at 525 °K, and structural and mechanical design of the linear alternator. The impact of temperature on close clearance seal and bearing surfaces (the potential for binding increases as temperature increases) was evaluated prior to delivery of the CTPC hot end in the Fall of 1991. The complete engine is currently on test at NASA LeRC. Limited life testing of this engine may be run starting in December of 1992 (Ref. J-22).

The prior generation SPRE linear alternator demonstrated greater than 90% efficiency in an operating power converter at design conditions of 1.5×10^7 N/m^2, a temperature ratio of 2, and 10 mm piston amplitude. This alternator demonstrated a basic understanding of linear machine technology and validated the design codes used in its development.

The current generation CTPC linear alternator is quite similar in design to the SPRE with the exception that it must operate in an environment which is 200 °K hotter than the SPRE. A test facility was developed at NASA LeRC to characterize samples of magnets from a variety of vendors at power converter operating temperatures. Results indicate that sufficient design margin exists for Sm-Co magnets operating at this temperature. Early testing of the CTPC alternator has yielded an 85% efficiency. It was found that this unit was damaged during fabrication. A second alternator is being fabricated using a deep oxide technique to guarantee good electrical insulation and to demonstrate the expected efficiency. This technique should be acceptable for the final space design because the approach eliminates organic insulating materials which can outgas and contaminate the helium working fluid.

Life and reliability are critical issues for a space power system. Some free piston Stirling engines have achieved long life by incorporating non-contact bearings. Development of Stirling cycle cryocoolers have shown that a 5 year life is attainable. In 1983, under a NASA contract, MTI began endurance testing of the EM-2 free-piston Stirling engine. The EM-2 was a nominal 2 kWe machine incorporating a combustion heater, hydrostatic gas journal bearings, and saturated plunger type linear alternator. The power converter was operated at low-power and full-power conditions over 262 starts/stops. At the end of 5385 hours, only minor scratches were discovered due to the numerous dry starts/stops and no debris was generated. The heater temperature for this power converter was 1033 °K.

The heater head of Stirling power conversion systems is the major design challenge for long life. Heater head creep is predicted to be the life-limiting mechanism for Stirling engines. Analyses have already shown that start-stop cycles and the phenomena of thermochemical ratcheting must be considered when designing Stirling heater heads. The proposed Starfish heater head design of the Space Stirling Power Converter simplifies the manufacture and extends the life capability of the Stirling engine hot end.

The current design for the CTPC uses Inconel 718 for the hot end. The next generation machine will probably utilize Udimet 720 for long life at 1050 °K (Ref. J-22). It is believed that a 7 year life can be achieved with this new material (7 year life could also be achieved with Inconel 718 operating at 950 °K). However, this material is not well characterized at the current time. Thus, NASA LeRC has a material characterization running in parallel with their Stirling engine program. The long life Stirling engine is schedule to be on test by December of 1993. The long life CTPC will be designed to develop 25 kWe per cylinder. An endurance demonstration (1 year or more) of this machine is scheduled for May 1994. Power system tests in a relevant environment are scheduled for January of 1996, if funding is available. The system test will include a simulated heat source, vacuum environment, radiator, and PP&C.

Radiator and Manifold State-of-the-Art

Past heat pipe radiator concepts have involved the use of all metal heat pipes and metal fins. However, these radiators are heavier than pumped loop radiators. The experience base for ammonia and water heat pipes is substantial as shown in Table J-4.

A multi-element project is currently being carried out at NASA LeRC for the development of advanced, low mass space heat rejection subsystems (Ref. J-17). The Lewis Thermal Management Program is part of the Civilian Space Transportation Initiative (CSTI) and supports SEI power system technology, especially the SP-100 program. Key elements of this program are summarized in Figure J-15. Goals of the program are a radiator specific mass of 5 kg/m^2, 10 year life in LEO, and a reliability of 0.99 or higher. The specific mass goal is a factor of 2 less than the current SP-100 baseline radiator design specific mass. The project is pursuing technologies which can be developed before the year 2000.

The Advanced Radiator Concepts (ARC) contractual development effort is aimed at the development of improved space heat rejection systems, with special emphasis on space radiator hardware, for several power system options including thermoelectric (TE) and Free-Piston Stirling. Although the principal heat sources for these systems is nuclear, the technology being developed for the heat rejection subsystem is also applicable to LEO dynamic power systems with solar energy input.

TABLE J-4. WATER AND AMMONIA HEAT PIPE TECHNOLOGY BASE

a)Ground and Shipboard Applications

Heat Pipe Material	Working Fluid	Application	No. Units Per System/ Module	Units Built	Years in Service	Estimated Hours x 10^6	Reported Failures
Cu/Cu	Water	Trident, SEM	108/3	7,000	14	458.65	0
Monel/Cu	Water	Mis. Spec.	1/1	300	10	8.74	0
SS/SS	Ammonia	Nassar Array	1000/1	1,200	16	467.39	0
Cu/Cu	Water	ALCM	12/3	12,000	8		0
SS-Cu/ SS-Cu	Methanol	MARM/ Missile	1/1	450	9		0
Cu/Cu	Water	Element G	1/1	397	2	2	0

b)Intended Applications - Shipboard, Avionics, and Space (Life Test Completed)

Heat Pipe Material	Wick Material	Working Fluid	No. Units	Operating Temperature (°K)	Total Hours	No. of Failures
Cu	Cu	Water	2	289-297	77,376	0
Cu	Monel	Water	1	294	48,406	0
Monel	Monel	Water	2	311-312	69,264	0
SS	SS	Ammonia	3	278-289	102,110	0
Al	Al	Ammonia	1	283	59,832	0
Al	Grooves	Ammonia	1	283	29,664	0
SS	Grooves	Ammonia	1	283	34,632	0

c)Space Applications

Heat Pipe Material	Working Fluid	Applications	No. Units Per System/ Module	Units Built	Years in Service	Reported Failures
SS/SS	Ammonia	DSD, TWT baseplate	3/3	700	10.0	0
Al/Al	Ammonia	MS111	19	22	0.5	0
Al/Al (Grooves)	Ammonia	Space telescope	3	3	0.5	0
Al/Al	Ammonia	Space sensor	1	2	0.2	0

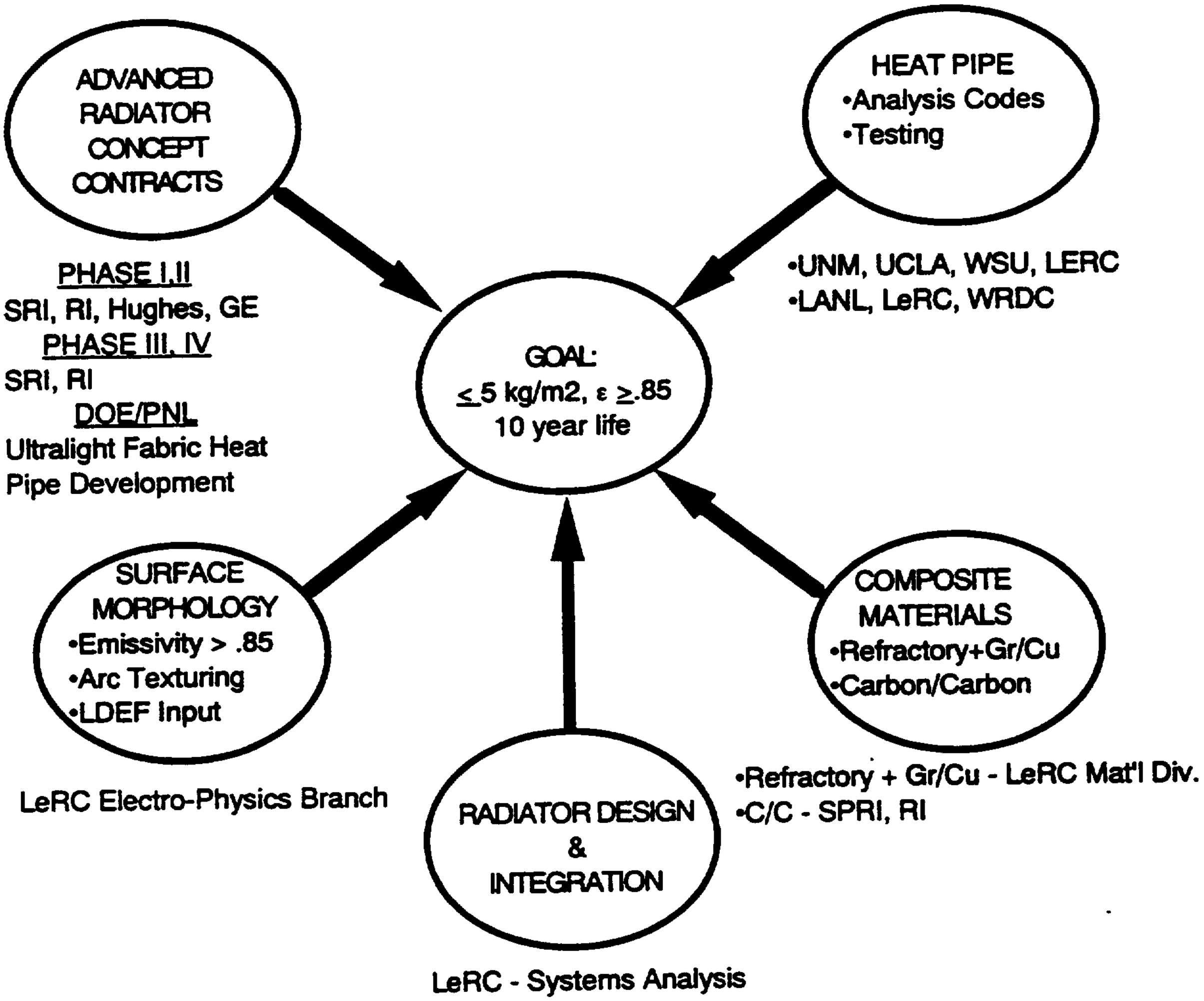


Figure J-15. NASA LeRC CSTI Thermal Management Program elements.

Phases I, II, and III of the ARC contracts have been completed by both contractors SPI and RI. Both contractors were selected to proceed into component development, fabrication, and demonstration to be accomplished under Phase IV over a two year period ending in January of 93. Both a high temperature heat rejection option (800 - 830 °K) applicable to TE power conversion systems and a low temperature option (500 - 600 °K) applicable to Stirling power conversion systems will be developed by SPI, while RI will concentrate only on the high temperature option.

SPI is developing a concept called the "Folding Panel Radiator" for the 500 - 600 °K heat rejection temperature range. This concept was described previously in the concept description section. SPI has demonstrated the feasibility of a heat rejection system based on a binary Li/NaK pumped loop during transient operating conditions representative of both the cooldown (Li freezing) and the warmup (Li melting). The thawing process during the warmup condition had to be controlled very closely to avoid plugging of the flow loop due to molten Li re-freezing downstream at certain operating conditions. Current efforts focus on widening the operating envelop by a variety of techniques. One of these techniques involves the use of fine mesh screens which act as semi-permeable membranes to NaK under certain operating conditions. SPI also completed a two dimensional computer analysis of the cooldown and warmup process.

Progress was also made by SPI in identifying potential subcontractors, named Applied Sciences Inc. (ASI) and Science Applications International Company (SAIC), with demonstrated capabilities in the development and fabrication of high thermal conductivity composite materials for space radiator fin applications. Use of composite materials has the potential of reducing radiator specific mass by over 60% for radiators that are radiative heat transfer surface limited. Technology requirements for joining the high conductivity fins to the heat pipes by advanced brazing or welding techniques have also been identified.

The RI ARC program involves development of an integrally woven graphite carbon tubes with an internal metallic barrier that is compatible with the intended potassium working fluid. The metallic barrier provides safe containment of the working fluid. Concurrently with this task a high temperature braze or other joining process is to be developed to insure good mechanical and thermal contact between the thin metallic liner and the C-C internal tube surface. Bonding of the entire liner surface to the tube needs to be achieved to prevent collapse of the liner at conditions where the external atmospheric pressure exceeds the internal pressure of the working fluid.

Although the baseline RI heat pipe concept is being developed for SP-100, this technology will be suitable for other radiator applications as well including other nuclear and non-nuclear power systems (using different working fluids and liners). This program will develop generic technology which can be used for all future space radiators because of its reduced mass compared to metal heat pipe radiators. Carbon-carbon is a new structural material which requires different fabrication techniques than for metals or composites. A key objective of this program is to demonstrate the ability to fabricate C-C heat pipes with a thin metal liner, wicks, caps, and fill tube. The brazing technique required to attach liners is the same as would be required to attach manifolds to the heat pipes. Heat pipes filled with potassium will be designed, fabricated, and tested during this program. Thermal cycle testing will be done in a vacuum to demonstrate the integrity of the heat pipes.

Beginning in 1989, small samples of heat pipe panels were fabricated by RI as part of the ARC program. During Phase III carbon-carbon (C-C) heat pipe tube sections with integrally woven fins were fabricated.

The objective of the Light Weight Advanced Ceramic Fiber (ACF) Heat Pipe Radiator Program is to demonstrate the feasibility of ceramic fabric/metal liner heat pipes for a range of operating temperatures and working fluids. This is a joint NASA LeRC and Air Force program with Pacific Northwest Laboratory. Specifically, this concept applies to Stirling engine space radiators with operating temperatures below 500 °K using water as the working fluid. The specific mass goals for these heat pipes are < 3 kg/m^2 at a surface emissivity of > 0.85.

An innovative "Uniskan Roller Extrusion" process has been developed at PNL and used to draw 0.8 mm wall tubing to a 0.05 mm foil liner in one pass. Several heat pipes were built by PNL using titanium and copper foil material for containment. A heat pipe with an 0.2 mm Ti liner was demonstrated at operating temperatures up to 475 °K in January 1991. The heat pipes fabricated for LeRC have been subjected to a test program to evaluate performance and reliability at demanding operating conditions, including operating pressures up to 2.5×10^6 N/m^2.

Additional in-house work at NASA LeRC has been done to enhance surface emissivity by the use of arc texturing. Emittance enhancement by atomic oxygen bombardment has also been studied.

Heat pipe performance modelling, both under steady state and transient operating conditions is being conducted in-house and under university grants respectively, with UCLA, UNM and WSU. Working versions of the codes have been developed and validation of predicted performance is underway at LeRC, LANL, and WRDC. Efforts have also been initiated to compile an experimental heat pipe performance database.

Electrical System State-of-the-Art

No special electrical or electronic hardware will be required for the SD power system. Power distribution equipment will already be installed as part of SSF when the SD system is deployed.

The technology required for the power processing equipment is being developed for Space Station Freedom. Rockwell is currently developing the Space Station Power Management and Distribution System. The use of multiple buses, components for the switching of loads, and switching of subassemblies within the power assembly are technologies which are being developed.

The required technology for the control and data acquisition functions currently exists, and the technology for autonomous control is under active development for Space Station Freedom (SSF), ESSA and other space programs.

ON-GOING COMPONENT TECHNOLOGY DEVELOPMENT PROGRAM PLANS

The following current programs will continue and will contribute to development of an advanced SD power system. Figure J-16 shows a tentative schedule for these continuing programs and potential follow-ons.

ADVANCED SOLAR DYNAMIC POWER SYSTEM TECHNOLOGY ROADMAP

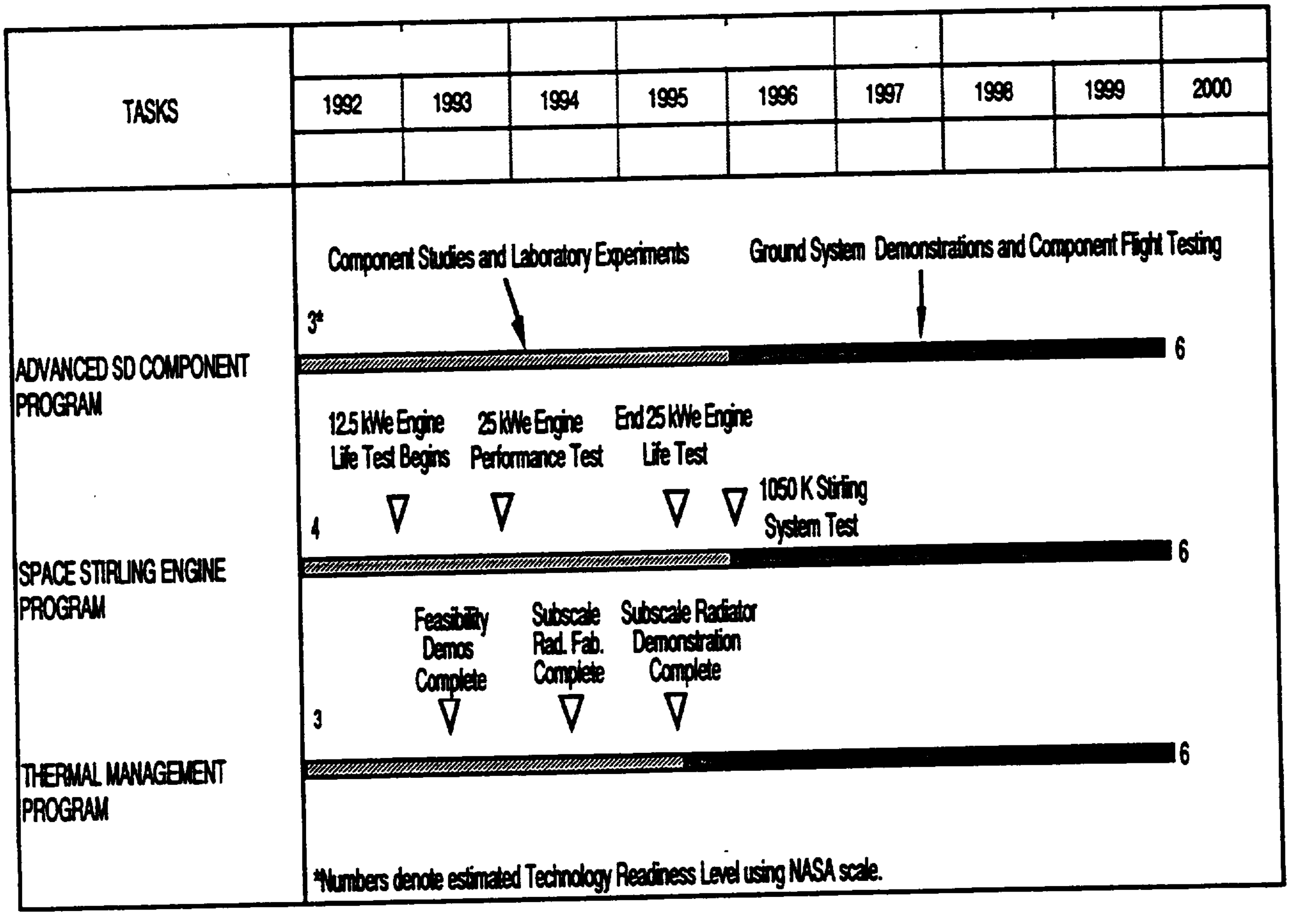


Figure J-16. On-going/planned SD component and subsystem technology programs.

NASA LeRC Advanced SD Component Program

NASA plans to demonstrate, as early as possible, the technology of a packageable and auto-deployable concentrator with a high concentration ratio. The approach is to design, build, and ground test a 2 meter prototype. The intent is to utilize as much of the technology of the Sunflower concept as is possible.

NASA will establish the engineering feasibility and practicality of the concentrator concept. NASA will demonstrate concentrator fabrication techniques and optical surface quality both isothermally and under simulated thermal conditions. NASA will build and test the subscale concentrator in the GTD facility.

NASA will show the feasibility of the selected receiver concept using a ground test of one segment of the receiver. NASA hopes to perform flight testing of integrated heat receiver/TES units to verify zero-g performance (Ref. J-15). An initial flight experiment using a TES canister is planned for the shuttle in 1995. The purpose of this experiment would be to verify the results (using several freeze/melt cycles) of the NORVEX computer code developed by Oak Ridge National Laboratory for NASA. This code predicts how voids are formed and how heat transfer takes place. This code has been validated for one-g TES systems.

Most of the technical issues have been resolved for advanced CBC receivers (Ref. J-15). NASA needs to prove this to the world by building and flying a receiver. Eventually, NASA would like to build and test an entire CBC receiver for a 2 kWe system. This receiver could be tested in NASA's planned SD Ground Test Demonstrator facility.

NASA LeRC Stirling Program

The current Stirling technology development contract with MTI will continue and the baseline 25 kWe engine design and component testing is expected to be completed in 1995. An important goal of the current program is to study heater head design problems both experimentally and analytically, and then to incorporate these results into the final design. This will also include a life assessment of the final heater heat design.

NASA LeRC CSTI Thermal Management Program

Phase IV of the RI ARC program will involve fabrication and testing of complete heat pipes. This phase of the program may also utilize higher conductivity, but lower conductivity graphite fibers. Phase IV is expected to be completed in 1993.

A planned 3.5 year follow-on to the RI ARC program could involve the development of a full scale C-C heat pipe radiator panel complete with manifolds and interface heat exchanger. This effort would be generic in scope and thus not include flight qualification. It is not clear whether this follow-on would be part of the CSTI program or part of the SP-100 program.

FULL SCALE DEVELOPMENT PROGRAM OPTION #1 - ADVANCED CBC SD SYSTEM

A technology development plan or roadmap has been generated for an advanced SD power system concept which would replace the SD baseline concept previously defined for SSF (Ref. J-18). This roadmap assumes continued component development will occur as outlined previously. The development plan is similar to that proposed for the SSF SD option (Ref. J-19), but is far less detailed. This roadmap is divided into several parts corresponding to the major subsystems of an advanced SD power system plus system integration:

- Task J-1, Concentrator Development;
- Task J-2, Solar Receiver/TES Development;
- Task J-3, CBC Power Conversion Unit Development;
- Task J-4, EEA Development;
- Task J-5, HRS Development;
- Task J-6, Integrated Assembly Testing;
- Task J-7, System Design;
- Task J-8, Qualification; and
- Task J-9, Flight.

The technology task descriptions include the objectives and the statement of work (SOW).

There are a total of 9 major tasks identified. The tasks are divided into subtasks which will generally be performed in serial order and would ordinarily be grouped together for program execution.

The 9 full scale system development tasks are described in the following sections.

Task J-1. Concentrator Development

Objectives: Develop and demonstrate full scale panels. Demonstrate the capability of the full scale production facility to produce panels which meet design objectives for optical and thermal performance to the extent that such may be proven with ground testing.

Statement of Work: Design, build, and test a full scale concentrator based on previous advanced

SD concentrator development work. Perform design trades in order to optimize the concentrator design. Consider alternate materials and fabrication techniques to reduce mass. Perform reliability/redundancy trades to minimize life cycle costs.

Demonstrate panel deployment. Demonstrate that the full scale panels can achieve and maintain satisfactory optical characteristics when deployed and under normal thermal operating conditions.

Task J-2. CBC Receiver/TES Development

Objectives: Develop and demonstrate a full scale solar receiver with integral thermal storage suitable for use with a CBC PCU. Demonstrate adequate performance and long cycle life.

Statement of Work: Design a full scale receiver/TES subsystem. Construct the receiver/TES subsystem. Perform component testing to verify adequate performance and life.

Determine the thermal-hydraulic and structural performance of the receiver heat source heat exchanger by analysis and testing. Include the effect of environments such as thermal stress, vibration, launch acceleration, acoustic, pressurization, and shock loads in these studies. Select design features to control working fluid flow rate distribution and accommodate variance in the flow rate between heat exchanger tubes, and define methods of implementation. Include accommodation of differential thermal expansion in heat exchange tubes and in ducting bellows, and accommodation of pressure-drop limitations while preserving good thermal performance in the receiver design. Design the receiver structure to support the receiver equipment through the range of environments that include ground transport, launch, and operation. Evaluate alternate materials and fabrication techniques of the thermal storage subassembly to minimize life cycle costs. Examine the design to insure the accommodation of the high-load environment anticipated during launch followed by large thermal expansions during startup and operation. Include the effects of walkoff tolerance, the ability to accommodate multiple concentrator slew operations, and cyclic sunrise/sunset atmospheric refraction effects in the aperture shield design. Study thermal leak paths through structural supports, and reduce total surface losses to be within limits set through performance requirements.

Perform thermal/hydraulic, pressure/leak, cyclic stress, dimensional stability and fitup, acoustic, acceleration, and shock tests to determine the performance of the receiver elements both singly and in combination under the range of environments.

Perform development testing and associated analysis in order to supply sufficient data on performance, safety, producibility, reliability, life, and operability to support a decision to commit resources to final detailed design and fabrication of flight-quality receivers for qualification. Perform long-term canister cycling, receiver thermal energy charge and discharge characterization, working fluid pressure boundary mechanical integrity, and aperture shield walkoff tolerance studies.

Task J-3. CBC PCU Development

Objectives: Develop a full scale flight CBC PCU. Demonstrate adequate performance and life.

Statement of Work: The CBC PCU development includes development of equipment in four basic areas: power conversion unit, parasitic load radiator, engine controllers, and wire harness. The PCU development includes development of the TAC, recuperator/cooler, and interconnect ducting.

Design the TAC. Perform analysis and verification of rotor dynamics and vibration, aerodynamic and thermal-hydraulic performance, structural performance and integrity of the working fluid pressure boundary. Investigate containment integrity over the full range of conditions that the device will see during its ground handling, launch, and orbital operations. Study the effect of environments such as thermal stress, vibration, launch acceleration, acoustic, pressurization, and shock loads. Select and define methods of implementation for design features to provide thermal management and working fluid bleed flow distribution.

Design and build heat exchangers for the PCU including a recuperator and gas cooler. Minimize thermal stress and enhance reliability in the design of the heat exchangers.

Pressurize the recuperator core assembly with hot gas and thermally cycle the unit to verify its structural integrity. Measure leakage across the core and critical dimensions at temperature extremes. Measure the pressure drop through the recuperator core. Give the complete recuperator a hydraulic proof pressure test and a pneumatic proof pressure test. Perform a hot-gas performance test on the assembly. Conduct a resonant frequency scan test (ring-test) on a complete assembly.

Design ducting, gas accumulator tankage and bellows based on aircraft turbine engine ducting experience. Design gas control valves based on nuclear safety-qualified valve designs currently in production.

Design the PCU electric loop control including the alternator magnetics, power electronics, instrumentation, logic components, software, parasitic load radiator, cabling, and packaging.

Design and construct an engineering model of the PCU for performance against the full requirements set. Initiate a materials characterization effort to establish detailed design information for candidate PLR resistance element and insulator materials. Perform tests to determine electrical properties, off-gassing, thermal shock tolerance, particulate generation, compatibility, and accelerated aging, as appropriate. Perfect manufacturing techniques for bellow joints, PLR assembly, gas control valves, special reliability features associated with the heat exchanger tasks, and PCU components integration.

Complete performance testing of the turboalternator rotor, bearings, compressor, and turbine wheels. Perform a fully integrated test of the PCU. In parallel, begin the heat exchanger test program with core fabrication and test, followed by integration and test with manifolds and wrap structure. Perform a characterization and accelerated aging test of the bellows and solenoid control valves. Perform electric controls testing with individual functional elements breadboarded in early performance tests followed by progressive integration to brassboard and engineering fidelity units.

Complete development testing and associated analysis in order to supply sufficient data on performance, safety, producibility, reliability, life, and operability to support a decision to commit resources to final detailed design and fabrication of flight-quality PCUs for qualification.

Task J-4. HRS Development

Objective: Develop a full scale heat pipe radiator and interface heat exchanger.

Statement of Work: Complete design studies which define the heat rejection assembly. Design studies will include thermal and hydraulic design, structural and dynamic design, materials selection, and configuration design and description. Optimize the heat rejection assembly design based on studies of weight, efficiency, and reliability considerations as they affect life cycle costs.

Fabricate test articles for coatings, adhesive and impact testing, and a development radiator panel, a development deployment test article with simulated radiators, and development flex hoses. Perform component level testing to verify the design.

Task J-5. EEA Development

Objectives: Demonstrate adequate steady state and transient performance, life, and immunity to the environment (including launch and operating). Demonstrate the software capability to handle power system nominal operation and failure modes.

Statement of Work: Define the EEA requirements. Develop algorithms for pointing and tracking and other controller functions. Integrate and test the electrical/electronic equipment at the subsystem level. The electrical equipment assembly will include two redundant SD controllers and a power conditioner, as well as the SD module cable set. The heat rejection assembly provides the utility plate cooling for these SD electronics.

Breadboard units will be built to demonstrate and check functional performance of the individual component circuit designs. Design modifications and improvements will be incorporated into brassboard units, which will be used to verify functional performance within the constraints of the actual component configuration. Prototype units will be fabricated and a series of performance tests run using simulated input and output loads. Controller tests will include validation of operating system software. Tests will include:

- start up, steady state and transient control;
- failure simulation and detection and switching;
- effects of temperature extremes and thermal shock;
- shock and vibration;
- cold plate heat loads; and
- EMI generation and susceptibility.

Prototype cable harnesses and a prototype parasitic load radiator will be checked out with the electrical components during the integrated subsystem tests.

Software will be checked out in conjunction with tests of the controller, using simulated inputs and outputs. Integration and checkout of the software will be performed as part of the system test.

Task J-6. Integrated Assembly Testing

Objectives: Verify proper operation of various integrated assemblies including the receiver/TES and PCU, the concentrator and receiver, and the heat rejection subsystem. Demonstrate adequate steady state and transient performance characteristics, long life at high temperatures, and suitable performance during failure modes.

Statement of Work: The following subtasks are defined:

Task J-6.1 Integrated Concentrator/Receiver Assembly Testing. Fabricate a prototype concentrator and heat receiver (without PCU). Assemble an integrated concentrator and receiver. Test the assembly using a PCU simulator. Demonstrate the ability of the hardware to provide proper alignment between the concentrator and the receiver after deployment based on prototype hardware.

Task J-6.2 Integrated Receiver/PCU Assembly Testing. Assemble and test a complete power conversion unit consisting of a receiver and PCU. Tests will be conducted under vacuum using a closed loop with simulated heat source, user loads and parasitic load resistors. Demonstrate start-up, shutdown and design speed conditions under normal operating conditions. Verify interface with the SD controller under conditions of varying power demand and simulated faulted conditions. Tests will include:

- time to start up from cold condition and motor KVA, invertor ramp rate;
- thermal balance;
- electrical power generating capability;
- steady state and dynamic stability;
- shutdown and start-up under simulated faulted conditions; and
- short term endurance test.

Task J-6.3 Heat Rejection Subsystem Assembly Testing. Fabricate a prototype radiator and heat exchanger. Assemble the components into a subsystem. Performance test the heat rejection assembly in a vacuum. Verify performance of the radiator under design point and extremes of temperature conditions.

Task J-7. Full Scale CBC SD Power System Design

Objectives: Define the requirements for a full scale power system. Define a full scale module design.

Statement of Work: Based on the results of the ground demonstration unit tests, identify a power system concept which can meet flight system requirements. Define subsystem sizes. Perform feasibility studies for the selected concept including off-design and transient analysis.

Design and integrate the full scale SD system. This task shall be done in parallel with the component development tasks to insure proper system characteristics. Design the system to meet the launch configuration.

Task J-8. Full Scale CBC SD Power System Qualification Program Testing

Objectives: Assemble qualification units and verify adequate performance for the entire system.

Statement of Work: Complete a comprehensive performance and dynamic testing program for assemblies and the complete system to provide a formal demonstration that the SD system will perform as designed after being subjected to simulated launch conditions. Qualification will be to launch environmental conditions selected to envelope the probable intensities developed by various launch scenarios.

The overall approach, shown in Figure J-17, will start with qualification of assemblies. Qualified production items will then be fabricated and assembled into the QU which is then qualified by the rules for space vehicle qualification. This approach will minimize the risk that some assembly of the QU will fail during the qualification test sequence, thereby nullifying the qualification, and requiring backtracking to recover from the failure.

Conduct performance testing at each level to verify that each item performs as designed. Perform dynamic testing to verify capability of the SD system to withstand launch loads, including acoustic, pyroshock and vibrational.

The performance and dynamic qualification test sequence for components and assemblies is shown in the matrix of Figure J-18. The corresponding qualification test sequence for the QU is shown in Figure J-19.

Task J-9. CBC SD Power System Flight System Program

Objectives: Build the flight systems and perform acceptance testing to demonstrate required performance. Deliver the flight systems.

Statement of Work: Fabricate the flight unit parts, acceptance test the components, and assemble the components to produce flight systems. Subject the systems to acceptance testing before shipment to the launch site. Complete reports necessary to obtain launch approval.

C-5.

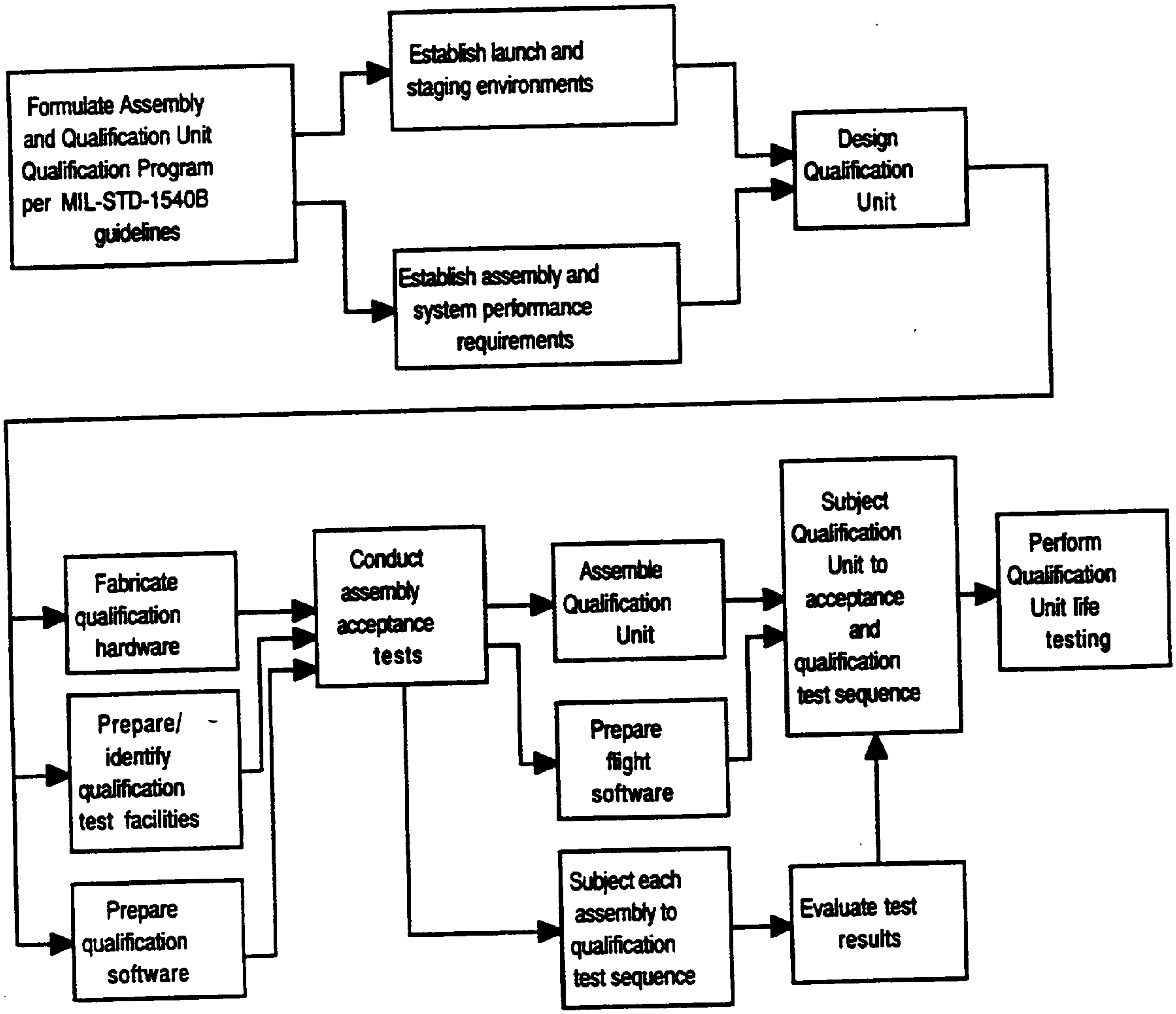


Figure J-17. Advanced SD qualification program.

Component or Subassembly	Functional (1)	Leak	Pyroshock	Functional (2)	Random Vibration	Functional (1)	Leak	Acceleration	Functional (1)	Thermal Cycling	Functional (2)	Thermal Vacuum	Functional (2)	Pressure	Leak	EMC	Life	Functional (1)
Concentrator	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Solar Receiver	X	X	X	X	X	X	X			X	X	X	X	X	X		X	X
TES	X	X	X	X	X	X	X			X	X	X	X	X	X		X	X
EM Pump	X	X			X	X	X					X	X	X	X	X	X	X
Radiator and Manifold	X	X			X	X	X			X	X	X	X	X	X		X	X
PMAD	X		X	X	X	X				X	X	X	X			X	X	X
Structure	X		X	X	X	X		X	X	X	X						X	X
PCU	X	X	X	X	X	X	X			X	X	X	X	X	X	X	X	X

Figure J-18. Assembly qualification test matrix.

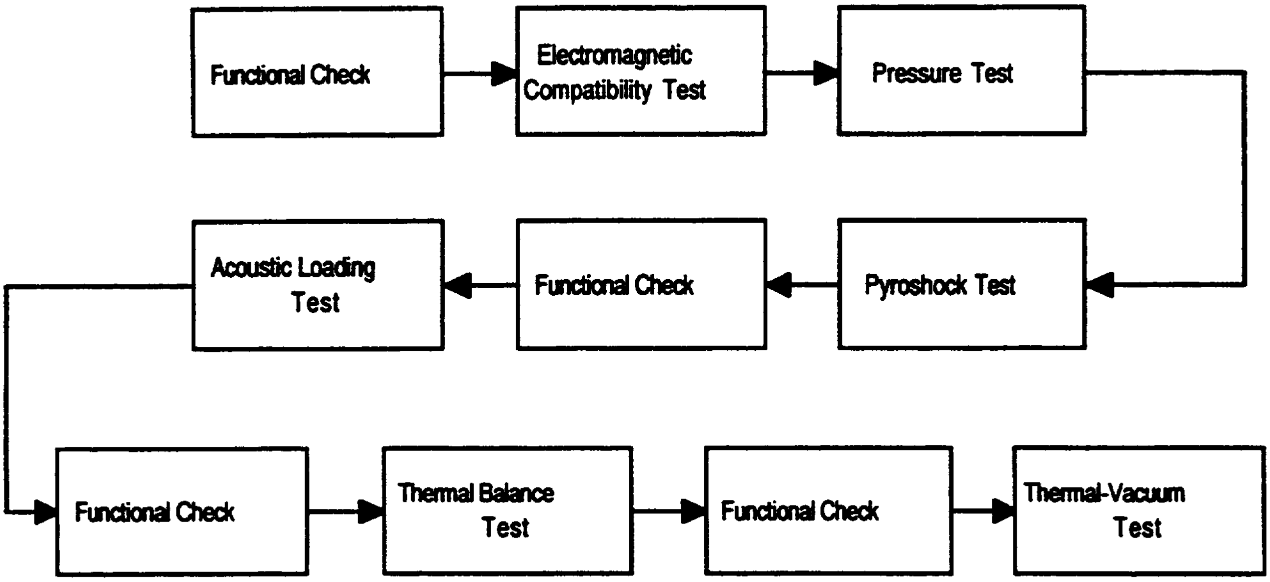


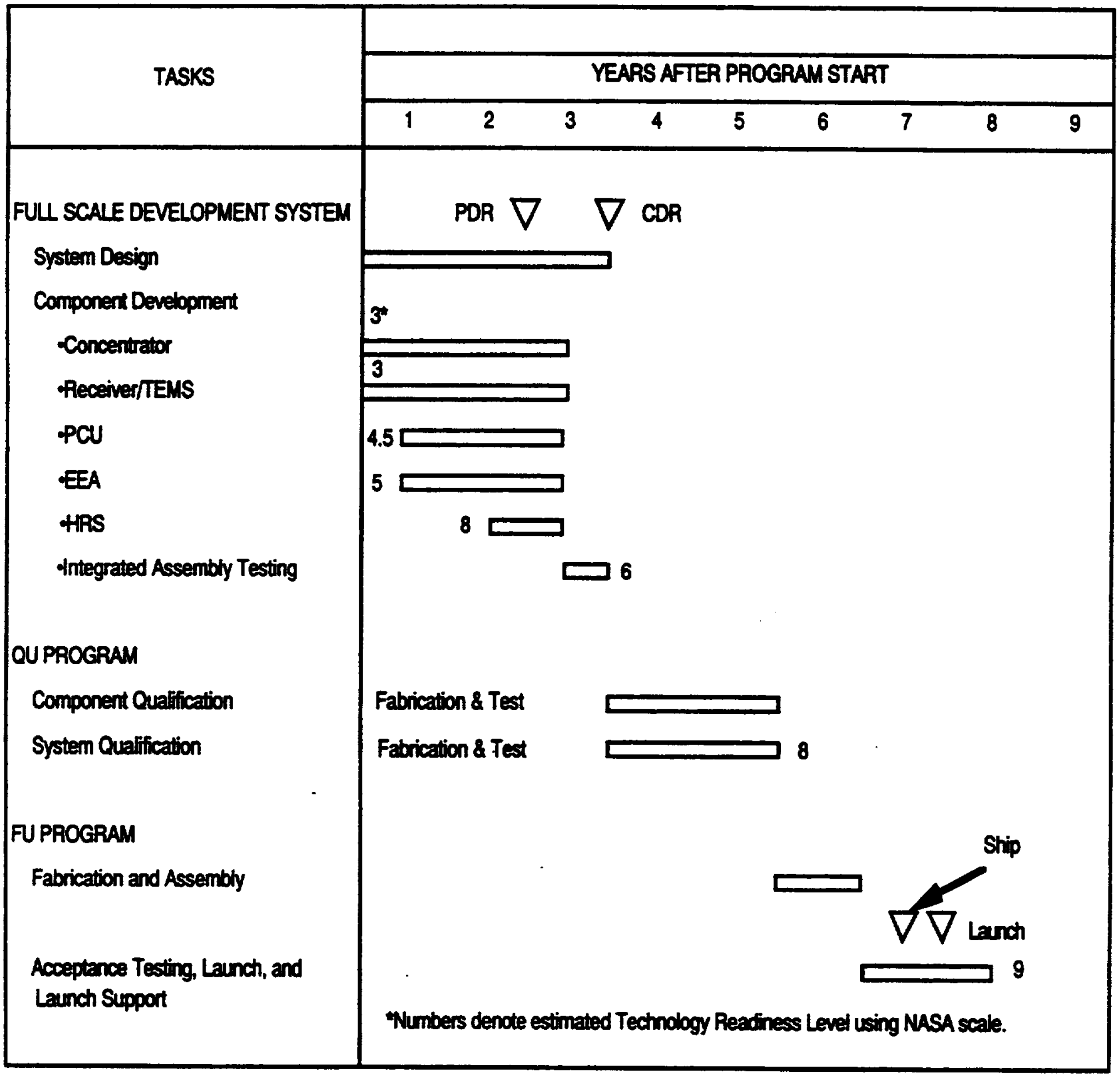
Figure J-19. QU Test Sequence.

OPTION #1 DEVELOPMENT SCHEDULE

Figure J-20 shows a schedule for the advanced CBC SD power system development program. The program starts with a conceptual system design task followed by a preliminary system design. The detailed system design is subsequently completed by the end of the 3rd year. The system design shall be suitable for qualification and flight units. Concurrently, component development of the concentrator, receiver/TES, Stirling PCU, EEA, and HRS will proceed. Additional component development would be required if the technology level is not at Level 6 as indicated in the schedule. The integrated assembly testing begins during the 3rd year of the program. An additional full scale ground engineering system demonstration task was not included in the schedule due to the subscale system testing which was assumed to be done using the 2 kWe Ground Test Demonstrator (GTD) facility.

The qualification phase includes fabrication, assembly, and qualification testing of individual components, and a complete system.

The flight phase of the program includes fabrication, assembly, and acceptance testing of the flight units.



*Numbers denote estimated Technology Readiness Level using NASA scale.

Figure J-20. Advanced CBC SD power system development schedule.

FULL SCALE DEVELOPMENT PROGRAM OPTION #2 - ADVANCED SC SD SYSTEM

A technology development plan or roadmap has been generated for an advanced SC SD power system concept which would follow development of the SSF SD baseline concept previously defined (Ref. J-18). This roadmap assumes continued component development and system testing will occur as outlined previously. This roadmap is divided into several parts:

- Task IJ-1, System Design;
- Task IJ-2, Component Design;
- Task IJ-3, Integrated Assembly Testing;
- Task IJ-4, Qualification; and
- Task IJ-5, Flight.

There are a total of 5 major tasks identified. Description of the Option 2 tasks would be similar to those described for Option 1. Figure J-21 shows a schedule for full scale development of an advanced SC SD power system.

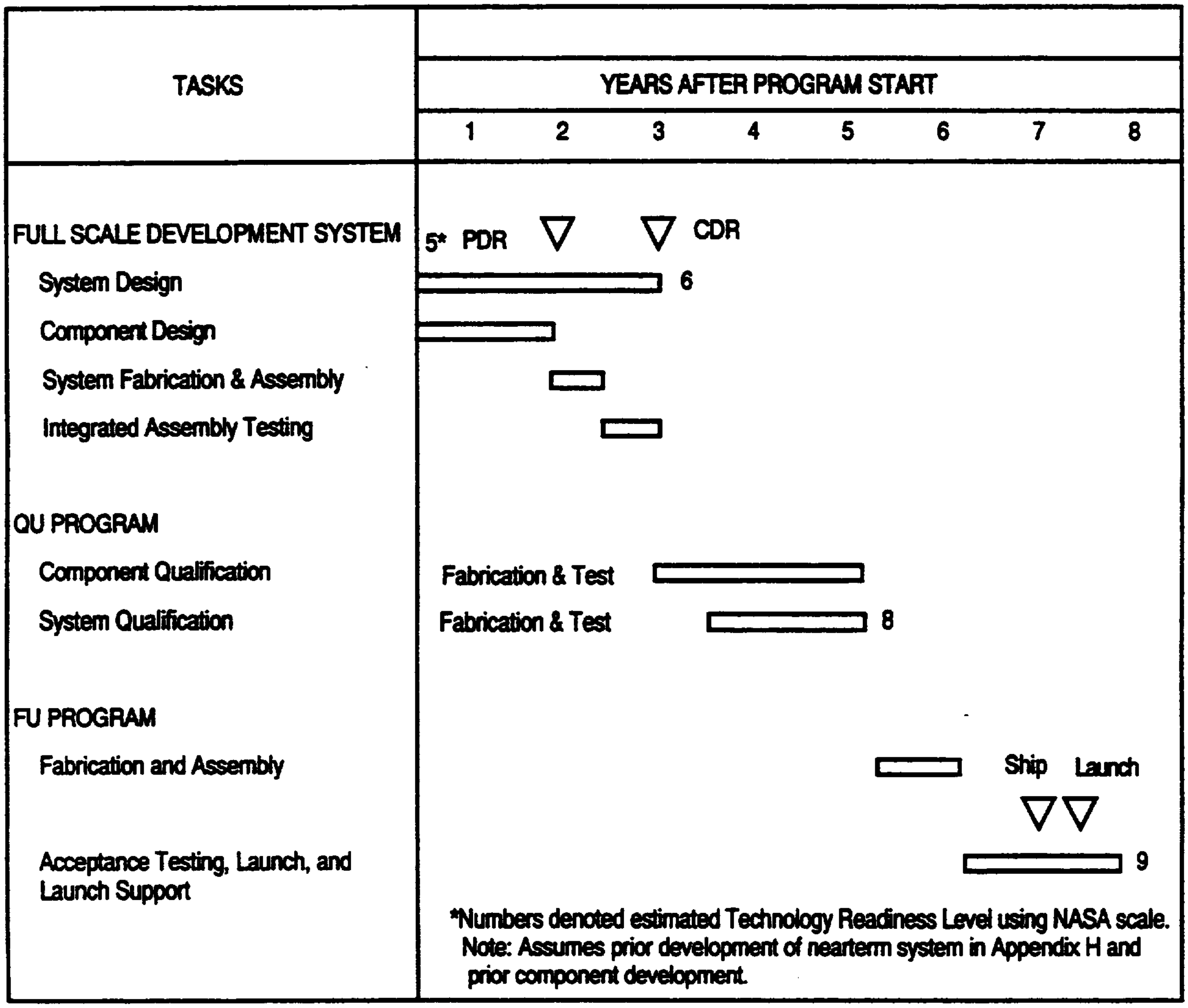


Figure J-21. Advanced SC SD power system development schedule.

REFERENCES

J-1. Naujokas, G.J.; and Savino, J. M.: A Program for Advancing the Technology of Space Concentrators. 24th IECEC proceedings, Vol. 2, August 1989, p. 861.

J-2. Richter, S.W.: Technology Development of Fabrication Techniques for Advanced Solar Dynamic Concentrators. 26th IECEC proceedings, Vol. 1, August 1991, p. 301.

J-3. Dehne, H.J.;: Conceptual Design of a Self-Deployable, High Performance Parabolic Concentrator for Advanced Solar-Dynamic Power Systems. NASA Contractor Report 187061, May 1991.

J-4. Calogeras, J.E.; Dustin, M.O.; and Secunde, R.R.: Solar Dynamic Power for Earth Orbital and Lunar Applications. 26th IECEC proceedings, Vol. 1, August 1991, p. 274.

J-5. Mroz, T.: Personal Communication. NASA LeRC, 20 February 1992.

J-6. Heidenreich, G.R.; Downing, R.S.; and Lacey, D.E.: Brayton Advanced Heat Receiver Development Program. 24th IECEC proceedings, Vol. 1, August 1989, p. 937.

J-7. Schneider, M.G.; et. al: Critical Technology Experiment Results for Lightweight Space Heat Receiver. 26th IECEC proceedings, Vol. 1, August 1991, p. 284.

J-8. Kesseli, J.B.; and Lacy, D.E.: The Cavity Heat Pipe Stirling Receiver for Space Solar Dynamics. 24th IECEC proceedings, Vol. 1, August 1989, p. 931.

J-9. Shaltens, R.K.: Solar Powered Stirling Cycle Electrical Generator. Technology 2000, November 1990.

J-10. Reader, G.T.; and Hooper, C.: Stirling Engines. E. & I.N. Spon, NY, 1983.

J-11. Wallin, W.E.; and Friefeld, J.M.: Solar Dynamic Power System Definition Study, Final Report. NASA CR 180877 or Rocketdyne report RI/RD87-250, March 1988.

J-12. Secunde, R.R.; Labus, T.L.; and Lovely, R.G.: Solar Dynamic Power Module Design. 24th IECEC Proceedings, August 1989, p. 299.

J-13. Lecocq, P.; and Alleau, T.: Heat Pipe Cooled Radiator for a Brayton Cycle Space Power System. 3rd Symposium on Space Nuclear Power Transactions, January 1987, p. TM-2.1.

J-14. 2 kWe Solar Dynamic Power System Ground Test. NASA-Lewis Research Center solicitation number RFP3-438039, 12 December 1991.

J-15. Dustin, M.: Personal Communication. Sverdrup Technologies, 19 February 1992.

J-16. Dudenhoefer, J.E.; and Winter, J.M.: Status of NASA's Stirling Space Power Converter Program. 26th IECEC, Vol. 2, August 1991, p. 38.

J-17. Juhasz, A.J.: An Overview of the Lewis Research Center CSTI Thermal Management Program. AIAA-91-3528, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 1991, Cleveland, Ohio.

J-18. Space Station WP-04 Power System, Contract Number NAS3-24666, Preliminary Analysis and Design Document DR-02. RI/RD85-320-2, Rockwell International (Rocketdyne Division), 19 December 1986.

J-19. Space Station Freedom Phase C/D Proposal. Rockwell International (Rocketdyne Division) proposal RI/RD87-200P-2, 1987.

J-20. Schertz, P.S.; and Lammert, L.: Development of an Improved Mirror Facet for Space Applications. NASA CR-189109, October 1991.

J-21 Dudenhoeffer, J. E.: Letter and enclosure to F. Littman (Rockwell International, Rocketdyne Division), 19 March 1992.

J-22 Dudenhoeffer, J.E.: Personal Communication, 4 March 1992.

APPENDIX K

ADVANCED STIRLING CYCLE DYNAMIC ISOTOPE POWER SYSTEM TECHNOLOGY ROADMAP

APPENDIX K - ADVANCED STIRLING CYCLE DYNAMIC ISOTOPE POWER SYSTEM TECHNOLOGY ROADMAP

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APPENDIX K - ADVANCED STIRLING CYCLE DIPS TECHNOLOGY ROADMAP

INTRODUCTION

The following technology development plan discusses an advanced high temperature dynamic isotope system concept which utilizes a Stirling cycle Power Conversion Unit (SC PCU) rather than a closed Brayton cycle PCU as in the reference power system design for the U.S. Department of Energy Dynamic Isotope Power System (DIPS) program. It is assumed that this advanced concept would be developed some time after development of the nearterm CBC DIPS concept.

DIPS technology is suitable for use in both fixed and mobile power applications for the lunar or Martian surface as well as a reliable power source for space satellites. For fixed bases and mobile surface power system applications, the current DIPS program has focused on a standard power module design of 2.5 kWe. Power systems in the range of 1 kWe to 20 kWe can be developed from multiple modules.

A variety of potential DIPS remote or mobile applications have been identified by the National Aeronautics and Space Administration (NASA). These applications include remote power to science packages, surface rovers for both short and extended duration missions, and backup to central base power (Ref. K-1).

The use of multiple modular power units to supply power needs has many advantages. Modular units permit the development of a single-size module reducing development costs and it improving the power availability factor for most applications. These modular units are replaceable if a unit failure occurs. The fuel is simply removed from the failed unit and reused in a replacement unit providing a 100% fuel utilization factor with the use of non-fueled spares. For mobile units, sufficient capacity would be installed such that if one power module were to fail, the mobile unit could return to base.

The advanced SC DIPS design life is assumed to be 7 years of continuous operation. The fuel handling canister is designed for a 45-year life thereby allowing for the reuse of the fuel in future unfueled DIPS replacement modules.

CONCEPT DESCRIPTION

The DIPS uses the decay of radioactive plutonium 238 as the source of heat and a SC PCU to convert this heat to electrical power. Figure K-1 is a layout of a 2.5 kWe power system.

The Free-Piston Stirling Engine (FPSE) is a thermally driven mechanical oscillator operating on a Stirling engine cycle which derives power from the heat flow between a source and a sink. The desired displacer motion in the FPSE is produced, unlike in most commonly known Stirling engines, by gas forces rather than kinematic linkages. This engine operates at the highest overall device efficiency of all known heat engines. In addition, the FPSE is uniquely suited to driving direct coupled reciprocating loads, such as linear alternators, in a hermetically sealed configuration and without the need for high pressure shaft seals or contaminating lubricants.

Helium is used for the working fluid for the Stirling cycle. Heat is rejected from the Stirling cooler through a NaK-78 liquid metal heat rejection loop, which is also used to cool the alternator. The Stirling engine used in this application is an opposed piston design which is selected for its superior vibration damping characteristics.

The HSUs contain a reversible heat removal system (RHRS) that allows the radioisotope heat to be dissipated to space in the event the power conversion cycle is not operating. The HSU design for the nearterm CBC DIPS concept was modified for this advanced system. The gas heat exchanger required for the CBC PCU was eliminated. Six heat pipes located within each HSU are coupled to the Stirling engine heater head as shown in Figure K-2.

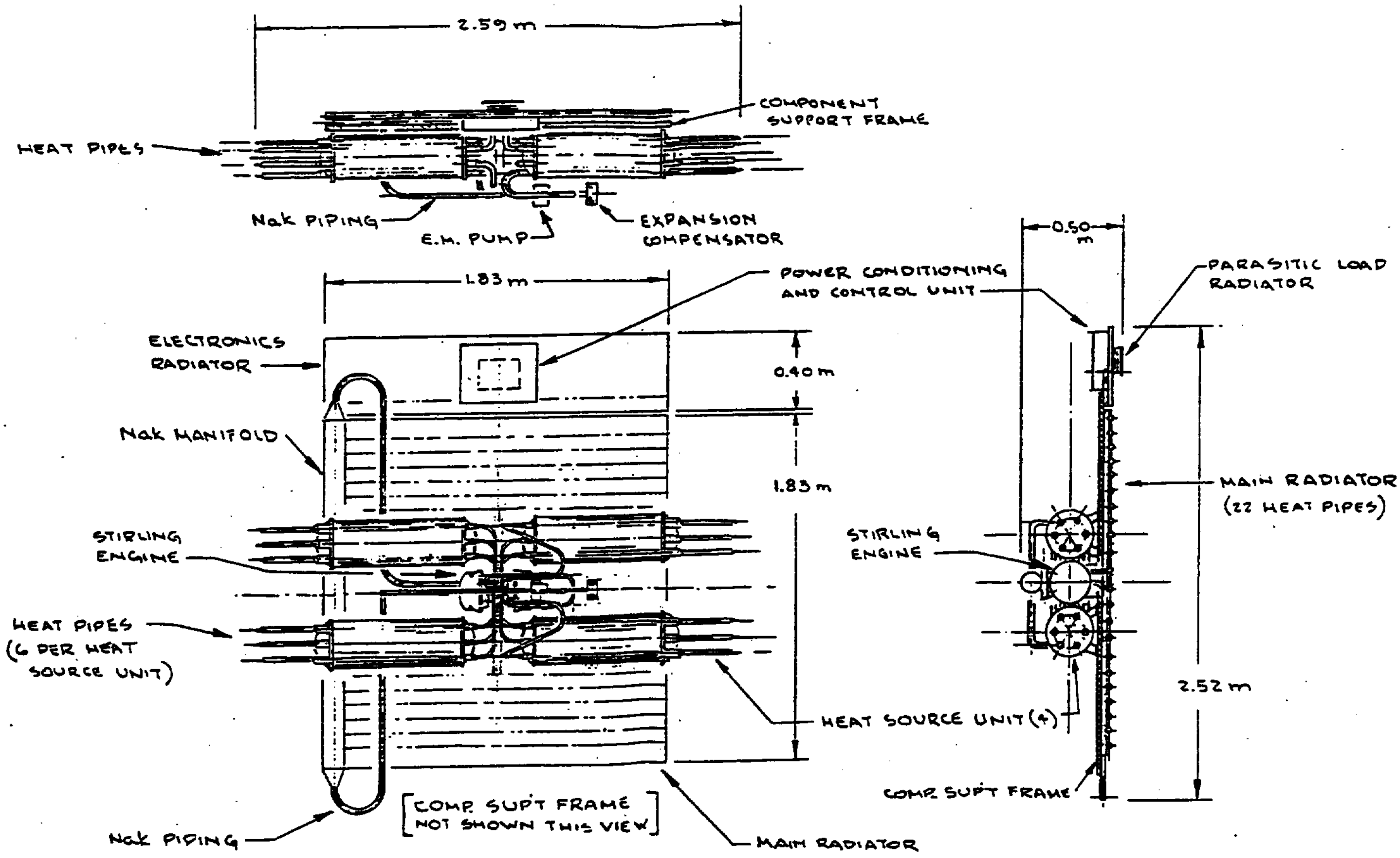

Figure K-1. 2.5 kWe SC DIPS module layout.

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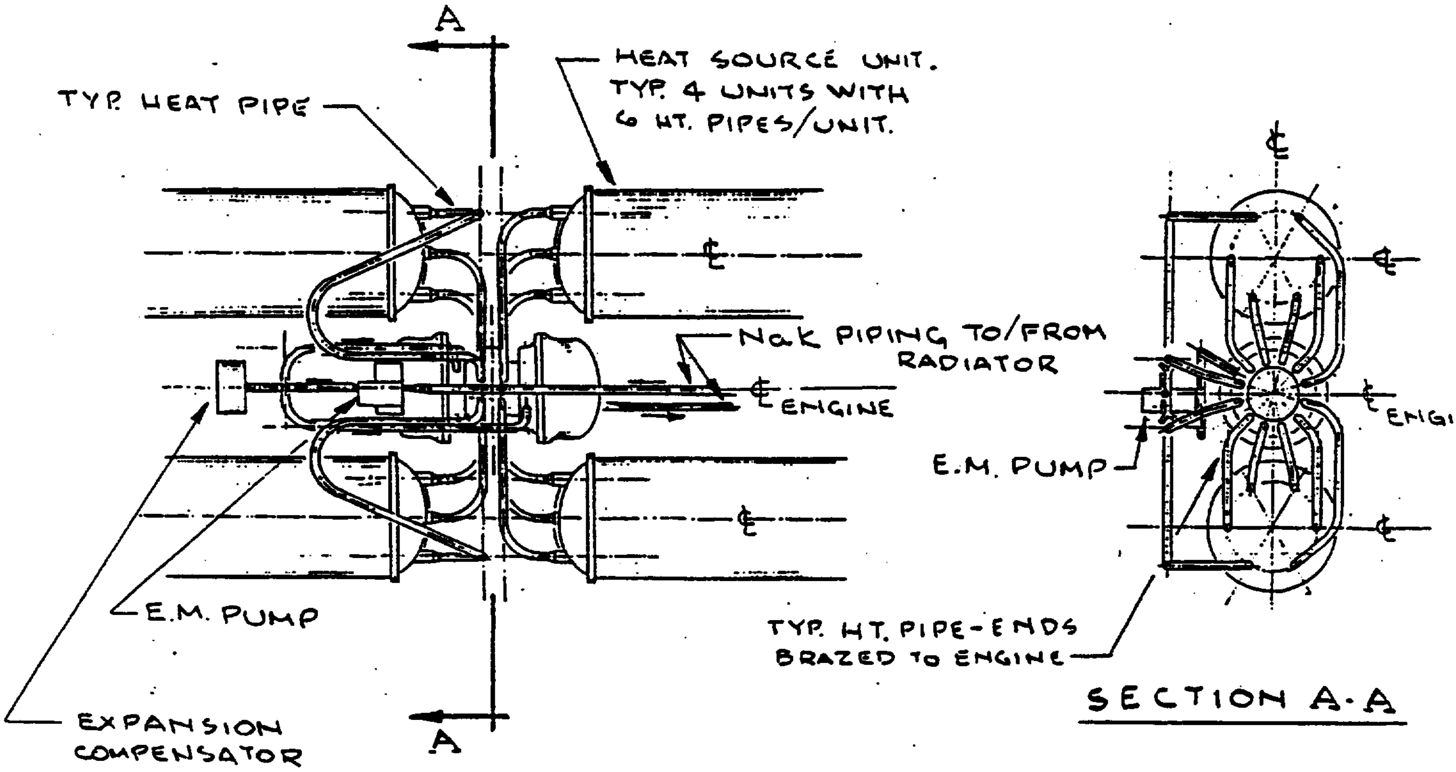


Figure K-2. Stirling engine/HSU integration.

Typically, the DIPS heat source assembly consists of three Heat Source Units (HSUs). Each HSU fuel handling canister contains multiple GPHS modules and is removable from the HSU housing. The fuel handling canister is fabricated of Nb-1Zr which is coated with titanium diboride for emissivity control and oxidation protection from the Martian atmosphere. This coating must provide protection only while the canisters are being transferred from their transport and storage rack units to each HSU housing which would require less than one hour. The oxidation environment within the HSU is minimized by providing a niobium wire wool getter to absorb the CO_2 diffusing through the helium release membrane and a niobium wire wrap at the slip joint end to "getter" any oxidizing atmosphere within the vacuum liner to canister annulus. A perfluoroelastomer O-ring seals the annulus to minimize CO_2 ingress. The canister materials and design were chosen to provide a viable concept able to meet the lifetime goal for either lunar or Martian missions. The getters would not be required for lunar surface operations.

Each HSU contains six Reversible Heat Removal System (RHRS) heat pipes. These heat pipes turn on when the Stirling engine is not operating to dissipate the radioisotope heat.

The heat pipes use lithium as the working fluid. Neon is added as a noncondensible gas to limit heat loss at operating temperatures. The heat pipe containment boundary and wick will be made of Nb-1%Zr. As in the other internal HSU components, TiB_2 is used as an emissivity coating. For compatibility with the Martian environment, the condenser and gas reservoir sections are covered with a pre-oxidized Inconel 617 vacuum liner which protects the refractory metal from attack by the CO_2 rich Martian atmosphere. The vacuum enclosure is not necessary for use on the lunar surface and can be removed. In either application the entire heat pipe sees only vacuum conditions.

In the unlikely event that all the RHRS heat pipes are inoperative, the isotope fuel clad temperature is still maintained in a safe level due to the HSU meltable MFI insulation package. The MFI consists of 130 layers of foil which are designed to melt and provide a direct cooling path to space before damage to the fuel cladding occurs after loss of all other cooling. There are

80 layers of 0.0005 cm thick niobium surrounded by 50 layers of 0.00086 cm thick nickel, all separated by yttria particles. To assure compatibility with the Martian atmosphere, all components with exposed refractory alloys must be contained in a vacuum vessel with an active vacuum pumping system. In addition, recent tests by General Electric indicate that lunar dust has a catastrophic effect on refractory alloys. Consequently, a vessel similar to that used for a Martian environment would be required for the lunar surface. An active vacuum pumping system, however, would not be required for lunar operation. This vacuum enclosure is not shown in Figures K-1 or K-2.

The GPHS module developed by the U.S. Department of Energy serves as the isotopic heat source in DIPS. The design of the GPHS module is shown in Figure K-3.

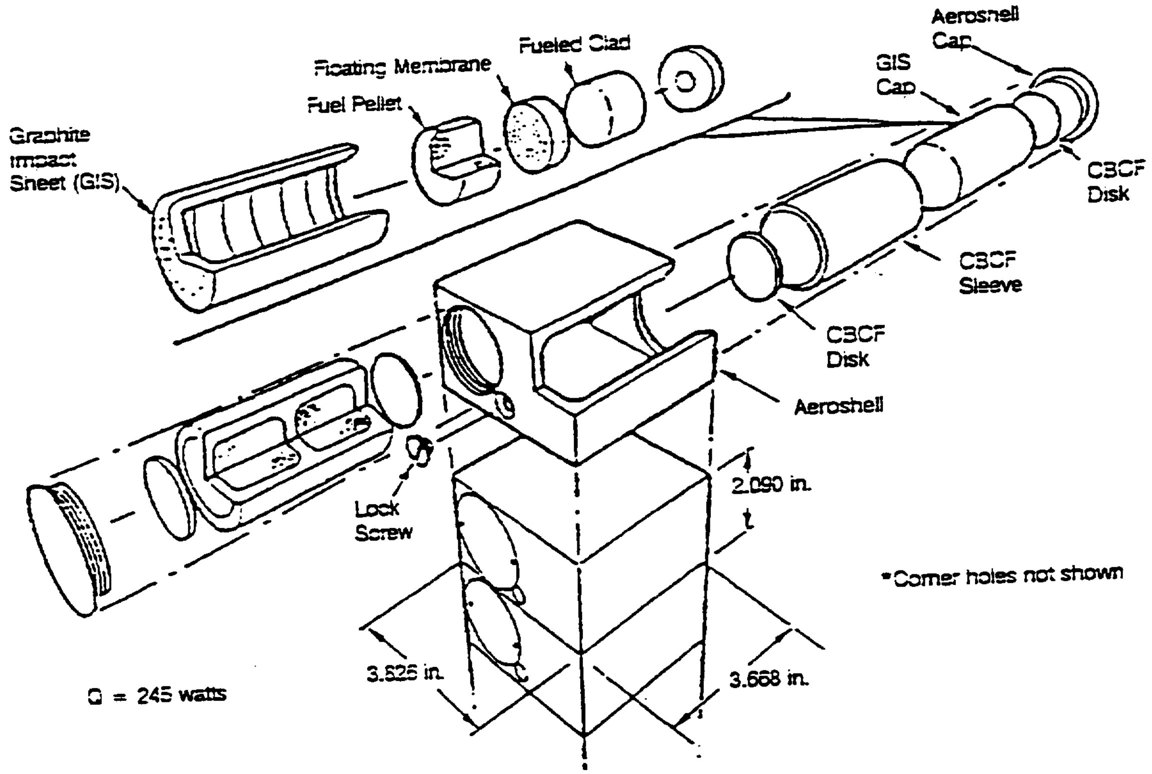


Figure K-3. GPHS Module (250 W) Sectioned at Mid-Plane.

The Power Conversion Unit (PCU) uses a regenerative closed cycle piston engine with cyclic recirculation of the working gas. There are typically two pistons per cylinder. The displacer shuttles working gas from the expansion (hot) space to the compression (cold) space through the heater, regenerator, cooler, and then back again. The power piston, integrated with a linear alternator, compresses the gas when cold and allows gas expansion when hot. The Stirling PCU has a dynamic balance unit or opposed single cylinder engine to minimize dynamic disturbances.

The major advantage of the free-piston Stirling engine over the kinematic Stirling engine is that it has only a few moving parts (a displacer piston and power piston). It can use noncontacting gas bearings and can be hermetically sealed, thereby increasing the potential for high reliability and very long life. Free-piston engines have no mechanical mechanism coupling the reciprocating elements to each other. Instead, free-piston engines elements are coupled by the forces exerted by the working fluid. The engine will resonate at a frequency determined by the combined dynamics of the piston and the displacer.

This heat engine has a high thermal efficiency (the ideal efficiency is equal to the Carnot efficiency). The efficiency remains almost constant over a wide range of operating conditions. High operating pressures are required ($>1.5\times10^7$ N/m^2 for power outputs greater than 40 kWe). Low molecular working fluids such as helium must be used. This heat engine is currently limited to less than 100 kWe per module. Power output is almost directly proportional to speed and can be regulated by the applied alternator voltage. Power output can also be controlled by adjusting the working fluid pressure level. The linear alternator has a low output power factor (high inductance) and is hermetically sealed. Power conversion is required to convert from the alternator output (50-200 Hz) to the desired frequency and to correct the power factor.

Figure K-4 shows the current NASA space Stirling engine design (Ref. K-6). The single-cylinder free piston SSE (with dynamic balancer) is being designed for a 1050 °K hot temperature and an engine temperature ratio of 2.0.

The key advantage to using a Stirling engine over a Brayton engine is an increase of cycle efficiency and resultant decrease in system mass.

Waste heat from the Stirling engine is transported to the radiator by a pumped NaK heat rejection loop. The radiator panel uses heat pipes to transport heat from the NaK heat rejection loop and radiate heat to space. The design uses a carbon-carbon heat pipe radiator panel with potassium as the working fluid and a thin metal niobium liner to prevent contact between the potassium and the carbon-carbon structure.

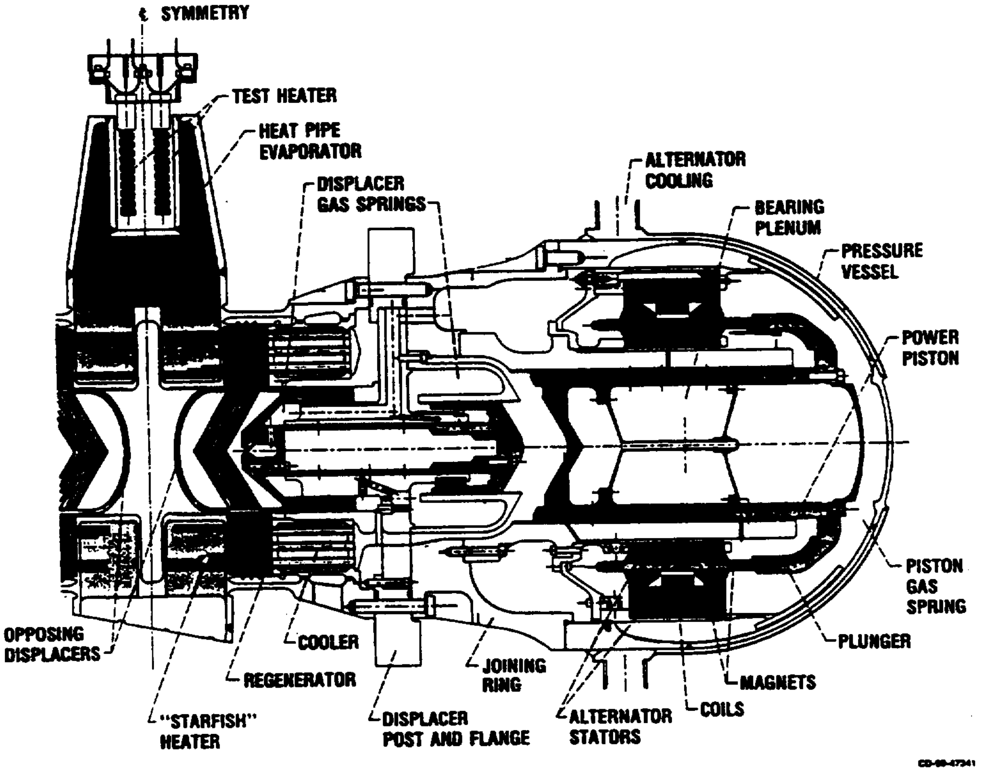


Figure K-4. NASA space Stirling engine.

A power conditioning and control system (PC&C) converts the 120 Vac alternator output to 120 Vdc. An electronics cooling radiator operating at 330 °K rejects the PC&C waste heat. A

Parasitic Load Radiator (PLR) is mounted on this electronics radiator and provides 0-2.5 kWe power output control to the user load.

KEY DEVELOPMENT ISSUES

The key issues for development of the baseline 1300 °K SC DIPS concept and their system impacts are summarized in Table K-1.

TABLE K-1. 1300 °K SC DIPS TECHNOLOGY ISSUES, IMPACTS, AND DEVELOPMENT AREAS

Issue	Impact	Potential Development Areas
Isotope Cooling/ Nuclear Safety	•Active cooling during launch and flight •Passive emergency cooling	•High emissivity coatings •RHRS heat pipes •Meltable MFI package
Lunar/Mars Environment	•Long term vacuum operation	•Vacuum system life
Shock Loading	•Reconfiguration of RHRS & electronics cooling radiator heat pipes	•Heat pipe design and verification testing
Isotope Handling & Disposal	•Added mass for biological shielding •Added cost for non-recoverable isotope	•Fuel handling canister and tools •Launch and transport containers
Stirling engine heater head life	•Impacts number of redundant and replace units needed	•Life testing •Option to run at reduced temperature for longer life

TECHNOLOGY ASSESSMENT

The SC DIPS concept is based on and utilizes the flight qualified, plutonium fueled GPHS modules currently being flown in the Galileo and Ulysses radioisotope thermoelectric generators (RTGs). The SC power conversion technology is being developed as part of the NASA Space Stirling Engine and terrestrial Stirling engine programs.

Advanced SC PCU component technology was chosen for this concept to provide improved performance for future space missions (late midterm to farterm). As part of this technology evaluation, the technology bases were assessed for the following major DIPS assemblies:

- GPHS modules;
- HSU;
- SC PCU;
- radiator; and
- PP&C.

These evaluations are summarized in Table K-2 which shows that the 1300 °K SC DIPS has technology readiness levels ranging from 3 to 9 (current status), depending on the particular assembly. Since full scale development of this system is assumed to start after development of the baseline CBC DIPS concept, the technology readiness levels were also estimated for the program start time (assumed to be the year 2000 or beyond). The technology base for each assembly is briefly discussed in the following sections.

GPHS Module State-of-the-Art

The technology base for the GPHS module is extensive and consists of the following items:

- materials, properties, and module performance characteristics have been well established in the government sponsored development program;
- manufacturing and quality control programs have been demonstrated with production of flight qualified modules for the Galileo and Ulysses programs;
- safety issues have been resolved in conjunction with the Galileo and Ulysses flights (both used RTGs with GPHSs as the heat source); and
- both the Galileo and Ulysses spacecraft have been launched and are operational.

HSU State-of-the-Art

The fuel handling canister, the RHRS heat pipes, and internal HSU vacuum liners are fabricated from Nb-1%Zr. Materials properties for Nb-1%Zr are well known and have been qualified in the SP-100 program. Silicide or titanium diboride emissivity/oxidation protective coatings are proposed for the canister and liner surfaces. These coatings are known to resist oxygen attack at temperatures exceeding the design temperatures for extended periods of time (years). The coating protection is only required in the Martian environment during transfer of a fuel handling canister between HSUs and storage rack units (<1 hour per transfer). However, accelerated testing would be needed to establish the variables involved and confirm their application over the equivalent fifteen year lifetime. Long term stability of the coating in this application remains to be established.

TABLE K-2. 1300 °K SC DIPS TECHNOLOGY ASSESSMENT

Subsystem	Current Technology Readiness Levels	Program Start Technology Readiness Levels	Comments
GPHS modules	9	9	Successfully flown on Galileo and Ulysses missions
HSU; RHRS HSU; MFI HSU; Gas Containment	4	9	Laboratory demonstration. Subscale tests complete. Need refractory alloy HSU.
1300 °K SC	3	6 *	Rapid progress is being made on current NASA 1050 °K SC program; 1300 °K program no longer being pursued.
Radiator	3	6	Currently under development for SP-100 as part of NASA LeRC CSTI Thermal Management Program
PP&C	5	6	Space Station Freedom Electrical Power Subsystem component (SSF-EPS) technology

*Assumes 1300°K program is started by 1996 and completed by the year 2000.

Explosively bonded transition joints are used between the Nb-1%Zr and Inconel components for several joints within the HSU. Specifically, these joints are located in the HSU vacuum containment used for Mars applications and are not part of the HeXe pressure boundary. This is a well established fabrication technique. The joints are located in low temperature areas where intergranular effects are not expected to be an issue over the 15 year operational life. However, joint leak tightness and intergranular effects will need to be verified.

Lithium heat pipes, like those to be used for the RHRS, have a substantial data base as shown in Table K-3 (Ref. K-2), but the specific DIPS design is different than those comprising the data base. For this reason, a demonstration model was fabricated and tested at LANL, with successful results (Ref. K-3). At 1150 °K, the test heat pipe removed approximately 750 W of heat, which is in excess of the 710W target for the variable conductance heat pipe design. For the present DIPS RHRS design, the VCHP "on" temperature will be set at 1194 °K by adjusting the neon gas inventory in the heat pipe.

The HSU uses multifoil insulation, not only to control heat losses from the HSU, but also to provide emergency cooling by melting and providing a direct heat path to space at a slightly higher than normal fuel cladding temperature. On the BIPS program, various combinations of multifoil insulation were subjected to subscale testing. The general conclusion that can be drawn from these tests is that the principle of progressive eutectic melting is valid. The tests also provide confidence that the meltdown characteristics of particular foil combinations can be accurately predicted.

TABLE K-3. LITHIUM HEAT PIPE OPERATING EXPERIENCE

No.	Wall Material	Test Temperature (°K/°F)	Hours of Operation	Remarks
1	CVD-W	1000/1340	1000	
2	W-26Re	1000/1340	10000	
3	TZM	1500/2240	4600	Evaporator leak
4	TZM	1500/2240	10526	Weld leak
5	TZM	1500/2240	10400	Weld failure in end cap
6	TZM	1500/2240	9800	Weld failure in end cap
7	Nb-1Zr	1000/1340	132	
8	Nb-1Zr	1500/2240	9000	
9	Nb-1Zr deoxidized	1500/2240	1000	Grain growth, Zr loss, swelling
10	Nb-1Zr	1350/1970	2300	
11	Nb-1Zr	1100/1520	4300	
12	Nb-1Zr	1000/1340	3870	

Stirling Cycle PCU State-of-the-Art.

Major interest for space based power systems has been centered on the free piston Stirling engine (FPSE). The FPSE concept was first invented in the U.S. in 1963. NASA has been sponsoring an extensive test program of several free piston Stirling engines with linear alternators built by Mechanical Technologies Incorporated (MTI). The NASA Stirling Space Power Converter Program originated in 1983 as part of the SP-100 program. A summary of the test times accumulated on these engines is given in Table K-4. One engine, the EM-2 has accumulated over 5000 hrs in tests at MTI.

The currently funded NASA FPSE program will result in a space capable 1050 °K FPSE by 1996 (Ref. K-4). This technology could be extended to 1300°K using refractory alloys. The Materials Division at NASA LeRC has substantial experience in the application of refractory materials and has been developing materials for use in the hot components of the SP-100 reactor. The Materials Division has compiled a list of refractory material candidates as seen in Table K-5. The LeRC ratings for each alloy as to joinability, fabricability, availability, and data were rated by NASA on a scale of 0 to 10 with 10 being the best. A NASA contract with MTI called for the refractory design to be carried through the conceptual design phase during FY'91. Funding for full scale development of the 1300 °K machine will be included under the Exploration Technology Program, scheduled for initiation in 1993 (Ref. K-4).

The success of this technology depends upon supporting research and technology efforts including heat pipes, bearings, refractory metal joining technologies, high efficiency alternators, life and reliability testing, and predictive methodologies. Based on the FPSE experience base and the NASA plans for the high temperature FPSE, the 1300 °K FPSE is given a technology readiness rating of 3.0 (the 1050 °K machine has a rating of 4.0 per Ref. K-5).

TABLE K-4. FREE PISTON STIRLING ENGINE ACCUMULATED TEST TIMES

Engine	Test Hours
RE-1000	280 (NASA)
EM-2	5385 (MTI)
SPDE	253 (MTI)
SPRE-I	349 (NASA), 74 (MTI)
SPRE-II	333 (MTI)

TABLE K-5. REFRACTORY METAL CANDIDATES FOR 1300 °K STIRLING

Base Material	Alloy Name	Composition (wt%)	Melting Point (°K)	Density (kg/m^3)	Join-ability	Fabric-ability	Alloy Avail-ability	Data Avail-ability
W	W-25Re-HfC	24-26% Re 1% HfC	1380	19,300	5	4	4	3
Ta	ASTAR-811C	8% W 1% Re 1% HfC	3270	16,600	8	8	10	5
Mo	TZM	0.08% Zr 0.5% Ti	2880	10,200	2	8	10	4
	TZC	1.25% Ti 0.1% Zr 0.15% C	2880	10,200	2	6	10	4
Mo/Re	Mo-47.5Re	47.5% Re bal. Mo	2780	15,500	8	6	8	3
Nb	FS-85	11% W 28% Ta 1% Zr	2740	8,600	8	8	5	4
	B-88	27% W 2% HfC	2740	8,600	7	7	4	2
	C-103	10% Hf 1% Ti 0.7% Zr	2740	8,600	10	10	10	7
	PWC-11	1% Zr, 0.1% C	2740	8,600	10	10	10	7
	Nb-1Zr	1% Zr	2740	8,600	10	10	10	8

Radiator State-of-the-Art

Past heat pipe radiator concepts have involved the use of all metal heat pipes and metal fins. However, these radiators are heavier than pumped loop radiators. The experience base for ammonia and water heat pipes is substantial as shown in Table K-6.

A multi-element project is currently being carried out at NASA LeRC for the development of advanced, low mass space heat rejection subsystems (Ref. K-7). The Lewis Thermal Management Program is part of the Civilian Space Transportation Initiative (CSTI) and supports SEI power system technology, especially the SP-100 program. Key elements of this program are summarized in Figure K-5. Goals of the program are a radiator specific mass of 5 kg/m^2, 10 year life in LEO, and a reliability of 0.99 or higher. The specific mass goal is a factor of 2 less than the current SP-100 baseline radiator design specific mass. The project is pursuing technologies which can be developed before the year 2000.

TABLE K-6. WATER AND AMMONIA HEAT PIPE TECHNOLOGY BASE

a)Ground and Shipboard Applications

Heat Pipe Material	Working Fluid	Application	No. Units Per System/ Module	Units Built	Years in Service	Estimated Hours x 10^6	Reported Failures
Cu/Cu	Water	Trident, SEM	108/3	7,000	14	458.65	0
Monel/Cu	Water	Mis. Spec.	1/1	300	10	8.74	0
SS/SS	Ammonia	Nassar Array	1000/1	1,200	16	467.39	0
Cu/Cu	Water	ALCM	12/3	12,000	8		0
SS-Cu/ SS-Cu	Methanol	MARM/ Missile	1/1	450	9		0
Cu/Cu	Water	Element G	1/1	397	2	2	0

b)Intended Applications - Shipboard, Avionics, and Space (Life Test Completed)

Heat Pipe Material	Wick Material	Working Fluid	No. Units	Operating Temperature (°K)	Total Hours	No. of Failures
Cu	Cu	Water	2	289-297	77,376	0
Cu	Monel	Water	1	294	48,406	0
Monel	Monel	Water	2	311-312	69,264	0
SS	SS	Ammonia	3	278-289	102,110	0
Al	Al	Ammonia	1	283	59,832	O
Al	Grooves	Ammonia	1	283	29,664	0
SS	Grooves	Ammonia	1	283	34,632	0

c)Space Applications

Heat Pipe Material	Working Fluid	Applications	No. Units Per System/ Module	Units Built	Years in Service	Reported Failures
SS/SS	Ammonia	DSD, TWT baseplate	3/3	700	10.0	0
Al/Al	Ammonia	MS111	19	22	0.5	0
Al/Al (Grooves)	Ammonia	Space telescope	3	3	0.5	0
Al/Al	Ammonia	Space sensor	1	2	0.2	0

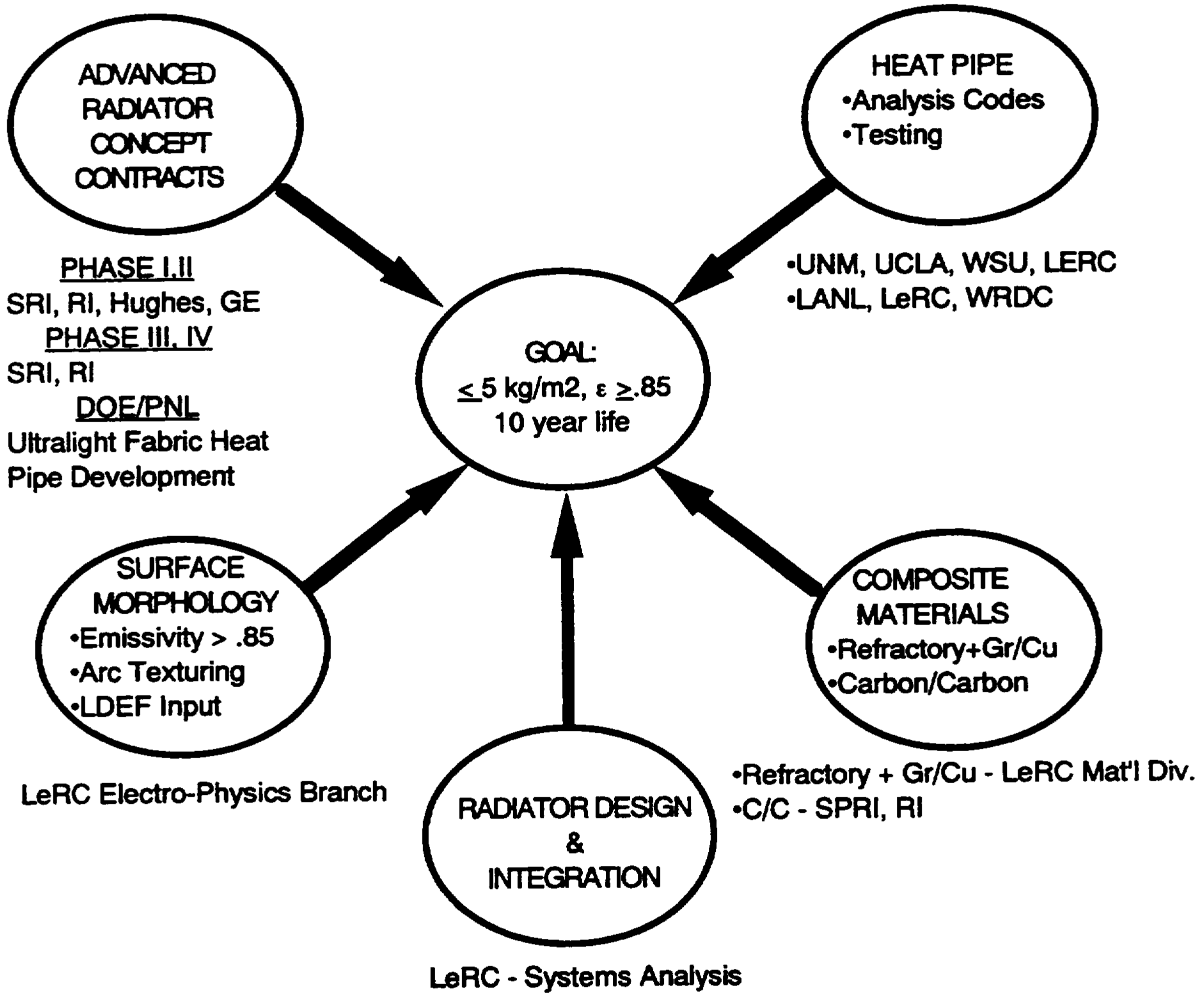


Figure K-5. NASA LeRC CSTI Thermal Management Program elements.

The Advanced Radiator Concepts (ARC) contractual development effort is aimed at the development of improved space heat rejection systems, with special emphasis on space radiator hardware, for several power system options including thermoelectric (TE) and Free-Piston Stirling. Although the principal heat sources for these systems is nuclear, the technology being developed for the heat rejection subsystem is also applicable to LEO dynamic power systems with solar energy input.

Phases I, II, and III of the ARC contracts have been completed by both contractors SPI and RI. Both contractors were selected to proceed into component development, fabrication, and demonstration to be accomplished under Phase IV over a two-year period ending in January

1993. Both a high temperature heat rejection option (800 - 830 °K) applicable to TE power conversion systems and a low temperature option (500 - 600 °K) applicable to Stirling power conversion systems will be developed by SPI, while RI will concentrate only on the high temperature option.

PP&C State-of-the-Art

The DIPS PP&C system design is based on a reasonable electronics component evolution, from SSF component technologies and there are no significant technology issues associated with its development. However, PP&C mass will be critical item in the flight hardware design. It will be necessary to fabricate brassboard hardware for testing and evaluation purposes, but there is no need to initiate any advanced component development. Even though no space-based Stirling engine linear alternator PP&C systems have been fabricated, most of the hardware elements have been or shortly will be operating in a relevant environment on other spacecraft or the SSF electrical power subsystem ground tests. The only new environmental factor is the radiation emitted by the DIPS HSUs. The DIPS radiation levels of 10^4 Rad (Si) are considered to be relatively low for electronic components. These radiation levels can be easily handled through proper component selection.

The electronics cooling radiator uses aluminum/ammonia heat pipes in an aluminum honeycomb/face sheet structure to reject the electronics waste heat (260 W thermal) to space. This technology has already been space qualified was indicated in Table K-6.

MAJOR DEVELOPMENT TASKS

The development program defined in this roadmap is a potential long term followon to the CBC DIPS program. This development plan includes limited component development (due to prior development), an integrated ground system demonstration, a qualification program, and a flight program.

The development and testing tasks which are envisioned are briefly described in the following sections.

Task 1. Heat Source Unit Modification

Objectives: Modify the CBC DIPS HSU to allow interfacing with the SC PCU.

Statement of Work: Modify the design of the HSU. Remove the heat exchanger coil required by the CBC PCU in the nearterm DIPS concept. Extend the RHRS heat pipes to allow heat transport to the SC PCU.

Task 2. 1300 °K SC PCU Integration with HSU

Objectives: Integrate the Stirling heater head to allow interfacing with the HSU (prior development of Stirling engine assumed).

Statement of Work: If necessary, modify the heater head design to allow interfacing with the heat pipes from the HSU.

Task 3. Radiator Interface with PCU

Objectives: Insure proper heat transport to reject heat from the PCU (prior development of radiator assumed).

Statement of Work: Design an interface between the cold end of the SC PCU and the heat pipe radiator.

Task 4. Integrated System Test Unit (ISTU) Development and Test

Objectives: Develop and test a full scale ISTU. Demonstrate adequate steady state and transient performance characteristics, long life at high temperatures, and suitable performance during failure modes. The ISTU shall be instrumented and calibrated during initial tests to validate the performance of the individual component designs.

Statement of Work: Assemble a complete power conversion unit, HSU, herat rejection system, controls and instrumentation. Conduct tests in air using fibrous insulation on the ISTU loop with an electrical heat source, simulated user loads, and parasitic load resistors. Install a vacuum system to maintain the prototypic HSU containment vessel under internal vacuum. Demonstrate start-up and shutdown under nominal operating conditions. Verify proper interface with the DIPS controller under conditions of varying power demand and simulated faulted conditions. Perform the following tests:

- time to start up from cold condition and motor KVA, invertor ramp rate;
- thermal balance;
- electrical power generating capability;
- steady state and dynamic stability;
- shutdown and start-up under simulated faulted conditions; and
- life test.

The work is divided into the following subtasks:

Task 4.1 ISTU Calibration and Set-up Tests - Pressure and leak test the ISTU. Fill the gas system with the He-Xe working fluid. Perform a complete electrical check-out of the unit and calibration tests and adjustments. Verify performance of thermal insulation.

Conduct a preliminary performance test sequence to demonstrate technical capabilities and measure critical unit performance parameters. Integrate acceptance level and design margin tests, up to qualification levels, into the initial test sequence to validate the individual component designs. Disassemble and inspect the individual components in the ISTU for wear and degradation effect and update component designs for the qualification and flight units as necessary.

Task 4.2 ISTU Life Test - Refurbish the ISTU components after completion of the performance, acceptance, and margin tests and place the unit on a multiyear life test. Refurbishment of the unit should include:

- replace or repair components to provide a prototypic hermetically-sealed gas containment boundary for the ISTU; and
- add special instrumentation required for the life test phase

Install prototypic instrumentation to provide a comprehensive diagnosis of the "health" of the ISTU and to monitor for degradation of major assemblies and individual components. Operate the ISTU at its nominal operating point, with expected ISTU variations in power output and environment.

Disassemble and inspect the ISTU at the end of the life test. Determine specific areas to be examined by an analysis of the health monitoring data and from the reliability analysis predictions.

Task 5. Qualification Program Testing

Objectives: Design, fabricate, and test the flight system. Develop a low risk qualification program. Verify adequate performance and life for the entire system under flight qualification conditions.

Statement of Work: Perform a comprehensive performance and dynamic testing program of assemblies and the complete system to provide a formal demonstration that the DIPS will perform as designed after being subjected to simulated launch conditions.

Start with qualification of assemblies, as seen in Figure K-6. Fabricate qualified production items and assemble these parts into the QU. Qualify the QU by the rules for space vehicle qualification.

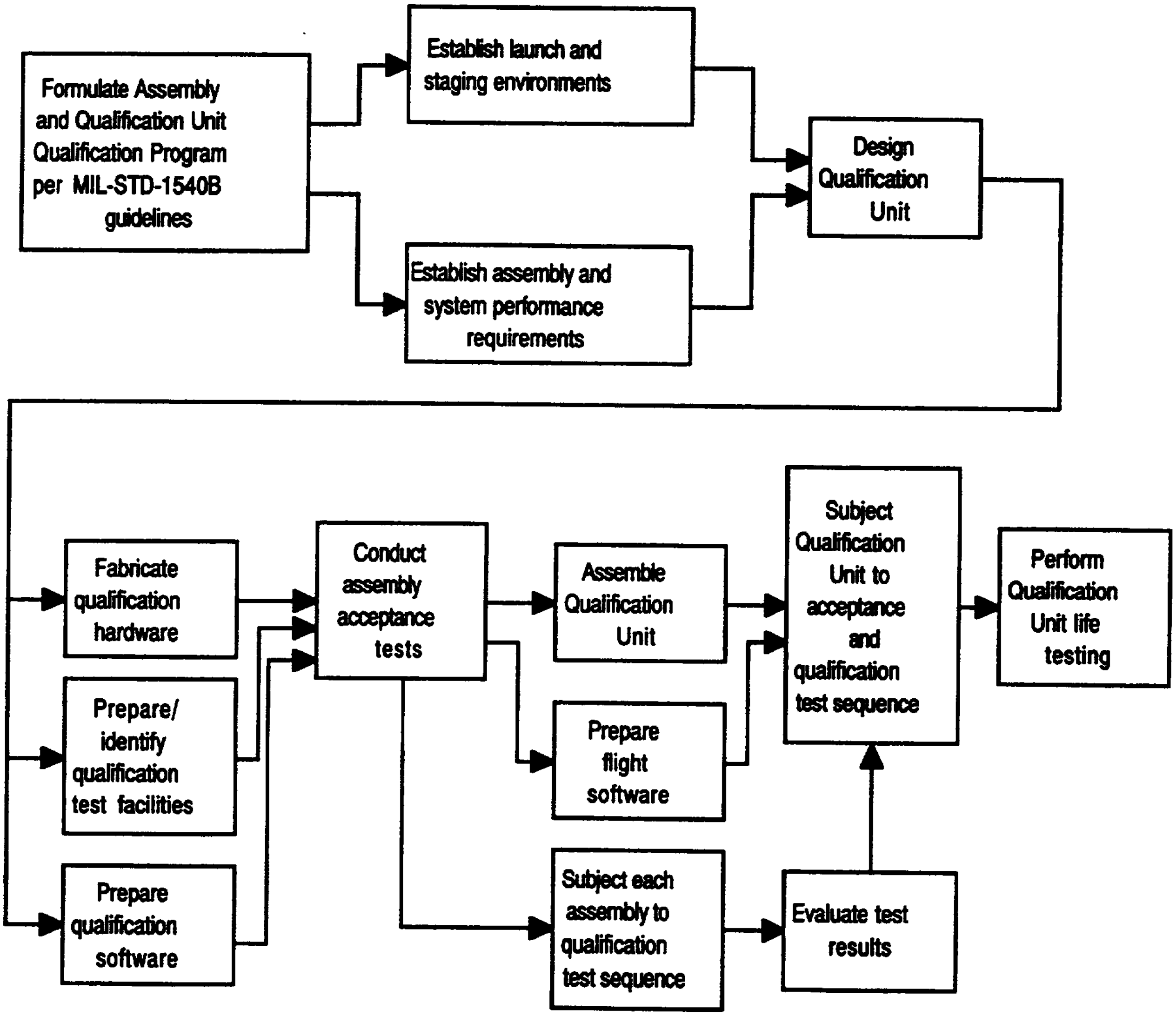


Figure K-6. SC DIPS qualification program.

The work is divided into the following subtasks:

Subtask 5.1 Component Qualification Testing - Conduct performance testing at the component and assembly level to verify that each item performs as designed. Perform dynamic testing per MIL-STD-1540B to verify capability of the DIPS system to withstand launch loads, including acoustic, pyroshock and vibrational. The performance and dynamic qualification test sequence for components and assemblies is shown in the matrix of Figure K-7.

Subtask 5.2 Qualification Unit Testing. Fabricate, assemble, checkout, and test the QU. Use the same test facilities for component and assembly qualification testing as were used for the assembly level testing of the ISTU. The corresponding qualification test sequence for the QU is shown if Figure K-8.

Component or Subassembly	Functional (1)	Leak	Pyroshock	Functional (2)	Random Vibration	Functional (1)	Leak	Acceleration	Functional (1)	Thermal Cycling	Functional (2)	Thermal Vacuum	Functional (2)	Pressure	Leak	EMC	Life	Functional (1)
HSU	X	X			X	X	X		X			X	X	X	X		X	X
SC PCU	X	X			X	X			X			X	X	X	X	X	X	X
Radiator and manifold	X	X			X	X	X		X			X	X	X	X		X	X
Electronics radiator	X	X			X	X	X		X			X	X	X	X		X	X
Parasitic load radiator	X				X	X			X			X	X			X	X	X
PP&C	X		X	X	X	X			X	X	X	X	X			X	X	X
Structure	X		X		X	X		X	X			X	X				X	X

Figure K-7. SC DIPS Assembly qualification test matrix.

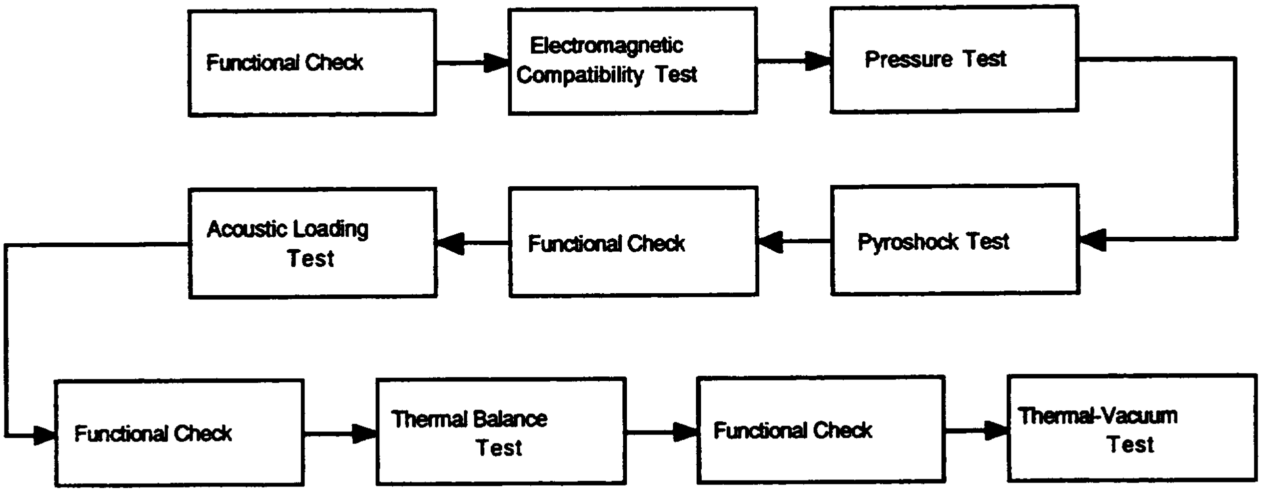


Figure K-8. QU test sequence (per MIL-STD-1540B).

Subtask 5.3 Qualification Life Testing (Optional) - Partially disassemble, examine, and refurbish the QU as required and modify for endurance testing as described for the ISTU. Life test the unit for 1.5 years (optional).

Task 6. Flight Unit (FU) System Program

Objectives: Fabricate two flight systems, perform a flight safety program, and acceptance test the FUs to demonstrate required performance. Deliver the flight systems and provide integration support for the FU with the payload and the launch vehicle.

Statement of Work: Fabricate, acceptance test, and assemble parts to produce two DIPS flight systems. Subject both systems to acceptance testing per MIL-STD-1540B guidelines before shipment to the launch site. Use the same test facilities (vacuum chamber, vibration, acoustic) that were used for the qualification program for flight system acceptance testing. Perform safety studies and complete safety reports necessary to obtain launch approval. Provide launch support for the DIPS flight unit for integration with the payload and launch systems.

The work is divided into the following subtasks:

Subtask 6.1 Flight Component Fabrication. - Design, fabricate, inspect, and assemble the components and subassemblies required for the QU and FUs, including all spare parts and GSE as required.

Subtask 6.2 FU Assembly, Test, and Payload Integration - Assemble and inspect the two FUs. Acceptance test both FUs and ship to the launch site. Provide technical support for FU integration with the payload, launch vehicle, and launch support facilities.

Subtask 6.3 Flight Safety Program - Develop a flight safety program plan to support the safety studies and tests required to obtain launch approval. Prepare the safety analysis reports (SARs), and all supporting analyses and documents to assure launch approval.

Subtask 6.4 FU Launch Support - Provide launch support for the integrated FU/payload and launch vehicle systems. This includes FU monitoring during ascent, payload deployment, and FU startup on station.

DEVELOPMENT SCHEDULE

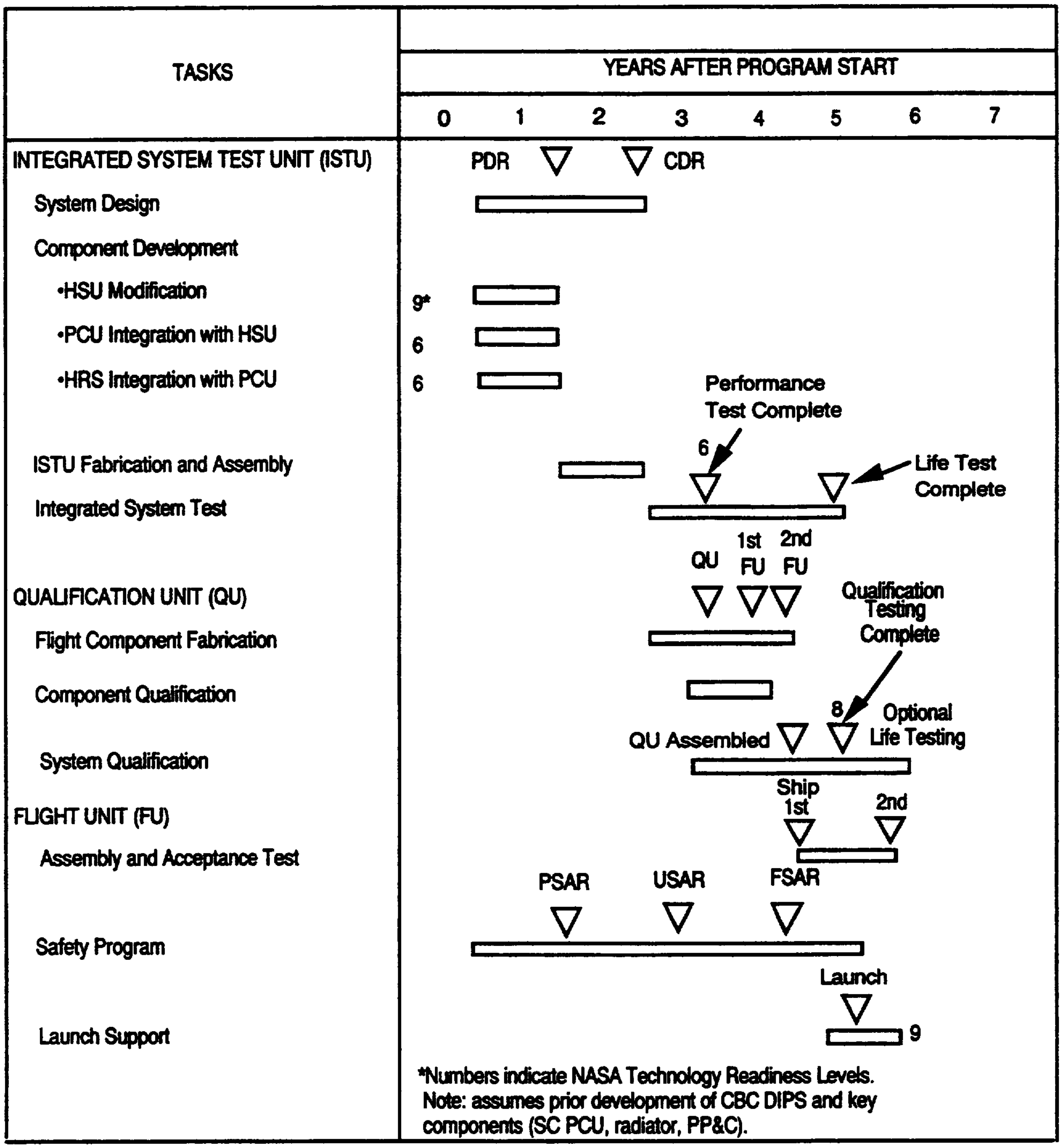
Development of this system is assumed to start after the CBC DIPS system has been developed. In addition, as mentioned previously, development of the C/C heat pipe radiator and 1300°K SC PCU was assumed to occur previous to this program. Development of the 1300°K SC engine during the SC DIPS program would add about 4-6 years to the system development time. Figure K-9 presents the high temperature SC DIPS development schedule. The time from program start to launch is estimated to be 5 years. The preliminary system design would be completed in one years. Concurrent component development of the heat source unit, power conversion unit, and radiator assembly is completed in 1 year. Detail design work is subsequently completed after 2 years.

Fabrication of components for ground testing for the ISTU starts with procurement of long lead materials and equipment in the first year of the program. This leads to assembly of the ISTU in the first half of the second year.

The ISTU will simulate the performance of a flight system but will have features such as additional instrumentation and readily accessible components to expedite gathering of engineering data and to permit modification of components. It will be performance tested in air under normal and off-normal design conditions. After this phase of testing, it will be partially disassembled, examined, refurbished as necessary, and put on life test, nominally for 1.5 yrs.

The qualification phase includes design, fabrication, assembly, and qualification testing of individual components for three units, and a complete DIPS system qualification test. The redesign effort will be limited to minor modifications to the preliminary DIPS design.

The flight phase of the program includes assembly, and acceptance testing of two flight units and the associated safety analysis support for launch approval. Launch support activities would include the flight unit-payload and launch vehicle integration tasks as well as post launch support activities.



*Numbers indicate NASA Technology Readiness Levels.
Note: assumes prior development of CBC DIPS and key components (SC PCU, radiator, PP&C).

Figure K-9. 1300 °K SC DIPS development schedule.

REFERENCES

K-1 Johnson, R.A.; Hunt, M.E.; Pepping, R.E.; and Mason, L.S.: A Dynamic Isotope Power System Portable Generator for the Moon or Mars. 26th IECEC proceedings, Vol. 2, August 1991, p. 210.

K-2 Heat Pipe Design Handbook, June 1979.

K-3. Woldshun, K.: Rocketdyne Emergency Cooling Heat Pipe Design Summary. LANL draft report, 6 March 1987.

K-4 English, R.; and Mason, L.: Technology Maturation Levels. Personal communications, 3/92.

K-5. Dudenhoefer, J.E.; and Winter, J.M.: Status of NASA's Stirling Space Program, 26th IECEC Proceedings, Volume 2, 8/91, pp. 38-43.

K-6. Dudenhoefer, J.E.: Stirling Space Engine Program Status. Personal communication, 3/4/92.

K-7. Juhasz, A.J.: An Overview of the Lewis Research Center CSTI Thermal Management Program. AIAA-91-3528, AIAA/NASA/OAI Conference on Advanced SEI Technologies, September 1991, Cleveland, Ohio.

APPENDIX L

ESPPRS USER'S GUIDE

ESPPRS USER'S GUIDE

This user's guide describes the function and use of the **ESPPRS** (Evolutionary Space Power and Propulsion Requirements System) software. The **ESPPRS** program was developed primarily to complete the conversion of timeline/profile software task of the Space Station Evolutionary Power Technology Requirements Study (Contract NAS3-24902). The conversion was performed using the TREES-pls language, the FOREST-pls scheduling library, and the TWM (TREES WINDOW MANAGER) library developed by Information Sciences, Inc. The transfer of technical data generated in Task 1A of the SSEP study to the **ESPPRS** program was performed as well. Several pre- and post-processing modules were also developed to assist with input of scenario/activity information and report generation.

Operating System Information

A few commands will need to be issued in the Apollo operating system (Apollo OS) while using **ESPPRS** and its pre- and post-processing modules. These commands include "trees", and "cpf". To execute one of the pre-or post-processing modules the following command should be entered at the **$** prompt of the Apollo OS:

```
"trees module_name" <CR>
```

Frequently the name of the input data file (e.g. "espprs.dat") required by one of the program modules will be different than the name of an **ESPPRS** output data file (e.g. "rock.dat_cg"). In this case a "cpf" command must be executed to copy the contents of one file to another (e.g. "rock.dat_cg" into "espprs.dat"). The -r option is used with "cpf" when the "espprs.dat" file already exists and the "rock.dat_cg" file is to replace it. An example of copying files with "cpf" is to type the following at the **$** prompt:

```
"cpf -r rock.dat_cg espprs.dat" <CR>
```

Pre-Processing Modules

Three pre-processing modules were written to enable and assist the input of activity and requirements information for a scenario. These modules include **Convert**, **Mk_Input**, and **Merge**. A brief description of each module and its use will follow.

Convert

This module was used to convert ascii data files containing SSEP study data from the PC (RBASE V and Microsoft Project software) into the file format required by the **ESPPRS** computer code. Therefore **Convert** was only used to transfer data from the PC to the Apollo. The utility called **Mk_Input** (described in the next section) should be used to establish new scenarios (data files) for the **ESPPRS** code. The input files used by **Convert** are ascii include "activity.dat" and "timeline.dat". The "activity.dat" input file contains power requirements and platform assignment data from the PC (RBASE) software. The "timeline.dat" input file contains scheduling information (window start/end dates and durations) and predecessor relationships from the PC (Microsoft Project) software. Sample listings of the input files used by **Convert** ("activity.dat" and "timeline.dat") are provided in Appendix A.

Flags are used in each file as end-of-file markers denoting the last line of data to be read. For the "activity.dat" file this flag is "AAAAAA" placed in the first field of the last line of data to be read. For the "timeline.dat" file the flag is "-99" placed in the first field of the last line of data to be read. To execute **Convert**, type the command:

```
"trees convert" <CR>
```

Upon execution of **Convert**, a file named "rocketdyne.dat" is generated. This file contains activity and requirements information in the proper (tree structure) format required by ESPPRS.

Mk_Input

This module is a simpler, more compact version of **Convert** that needs only one input file, "mk_input.dat", which can be keyed in using the Apollo editor. The structure of "mk_input.dat" consists of a table that contains 14 fields of data which characterize each activity in a scenario. In the first field an integer (3-digit maximum) is used to number the activities line by line. The second field contains a 6-character name for each activity. The third and fourth fields contain window start and finish dates for the activity. The string "ASAP" may be placed in either the start or finish field to indicate the activity may be scheduled anytime during the duration of the scenario. Otherwise the window start or finish date for the activity may be specified using a 10-character string in the format "mm/dd/yyyy". For this format "mm" is the month, "dd" is the day, and "yyyy" is the year of the date to be specified. Temporal relations for each activity are identified in the fifth field by a integer (3-digit maximum) which points to the line number (first field) of its predecessor. Several predecessors may be identified in this manner by simply copying the information for the activity onto multiple lines (one for each predecessor). These multiple lines will be identical except for the fifth field in which a different pointer for each predecessor will be labeled. An integer equal to "-99" in the fifth field is a flag which indicates the activity has no predecessors.

Field six contains the platform assignment, a 3-character string, for each activity. The next five fields (7 - 11) contain resource requirements data for each activity. This resource data includes duty cycle (field 7), peak power level (field 8), personnel (field 9), initial tonnage (field 10), and resupply tonnage (field 11). The units for the resource fields are kilowatts (kWe) for peak power, man-hours for personnel, and kilograms (kg) for tonnages. Field 12 contains the duration in days of each activity. Field 13 contains a description (40-character maximum) for each activity. A flag is used as the end-of-file marker for the "mk_input.dat" file and is "-99" placed in the first field of the last data line to be read.

After the "mk_input.dat" file has been keyed in, **Mk_Input** may be executed in order to transform the information into the trees format required by the ESPPRS program. A sample listing of the file "mk_input.dat" is provided in Appendix B. To execute **Mk_Input**, type the command:

```
"trees mk_input" <CR>
```

Upon execution **Mk_Input** a file called "rock.dat" is created that is identical in structure to the "rocketdyne.dat" file created by **Convert**. Thus the file "rock.dat" will serve as the input/output file containing all the necessary scenario information for the **ESPPRS** computer code.

Merge

This program is used to merge data contained in two separate files into a single file. In other words, **Merge** will combine the schedule and resource information of two different scenarios into a single scenario. Both files to be merged must be identical in format to the input/output files created by **Convert** or **Mk_Input**. The default names for the input files are "rock1.dat" and "rock2.dat". These files may be produced with the following procedure:

1. Key in the data file ("mk_input.dat") for the first scenario.
2. Execute **Mk_Input** which creates an output file called "rock.dat"
3. Use the Apollo OS to copy the file "rock.dat" to "rock1.dat" (i.e. execute the command "cpf rock.dat rock1.dat" <CR>)
4. Repeat the previous three steps for the second scenario.

Once the "rock1.dat" and "rock2.dat" input files have been created, the **Merge** program may be executed. To do this type the following command:

```
"trees merge" <CR>
```

Upon execution of **Merge**, "rock1.dat" and "rock2.dat" will be combined into a single output file called "big.dat". This file will be in the appropriate format usable by **ESPPRS**.

ESPPRS COMPUTER CODE

The **ESPPRS** computer code is the result of the effort to convert the SSEP mission database codes to the TREES/FOREST software on the Apollo workstation. The primary function of the **ESPPRS** software is to perform projection and resource summation for an evolutionary space scenario. In fulfilling this function, the **ESPPRS** code provides a highly flexible tool that may be used to generate resource requirements for a multitude of future scenarios.

Six data files must be resident in the same directory as the executable module of the **ESPPRS** code ("espprs.run"). These files are "rock.dat", "mfg.dat", "config.dat", "hardware.dat", "templates.dat", and "directory.dat". As discussed previously, the "rock.dat" file contains scheduling and resource requirement information used to describe a scenario. The "rock.dat" file is generated by the execution of the **Mk_Input** utility as described previously. The "mfg.dat" file contains additional resource information for manufacturing activities. These activities have variable resource requirements which depend on the scheduled start date of the activity and are computed during run-time execution of the **ESPPRS** code. The "config.dat", "hardware.dat", and "templates.dat" files contain default information used for the execution of **ESPPRS**.

The "directory.dat" file is used to specify the names of primary input/output files for the **ESPPRS** computer code. A timeline name is associated with each file specified in "directory.dat". This timeline name will appear in the Timeline **Directory** menu of **ESPPRS**. As discussed previously in the **Merge** section, the utility **Mk_Input** may be used to create multiple input data files for ESPPRS. Thus multiple I/O files may be specified in "directory.dat". Each data file produced using **Mk_Input** will contain three data trees. At the top of each tree is a default label. The default label currently used by **Mk_Input** is "ROCK". In order to specify a data file generated by **Mk_Input** in the file "directory.dat", the timeline name in the "directory.dat" file must be the same as the label of each tree in its associated data file. The following procedure is recommended for preparing the "directory.dat" and **Mk_Input** generated data files for use with **ESPPRS**:

1. Key in the data file ("mk_input.dat") using the Apollo editor.
2. Execute **Mk_Input** which creates an output file called "rock.dat".
3. Use the Apollo OS to copy "rock.dat" to a new file name (e.g. execute the command "cpf rock.dat mars.dat" <CR>).
4. Make the following changes to "directory.dat" using the Apollo editor.
    a. Insert a new timeline name (e.g. MARS), indented 3 spaces, on the line below the string "$DIRECTORY".
    b. Insert the string "FILE - file name", indented 6 spaces, on the line below the new timeline name.
5. Change all the tree labels in file name to the timeline name specified in step 4b.

The **ESPPRS** program may be executed for smaller data sets (i.e. "rock.dat" files less than 1 megabyte in size) by entering the command:

```
"run-espprs" <CR>
```

For large data sets (i.e. "rock.dat" files greater than 1 megabyte) the command entered is:

```
"run-bigespprs" <CR>
```

Upon execution of the **ESPPRS** program, the entire screen will be replaced by a gray background with the **Main** menu in the lower left-hand portion of the screen. There are 11 boxes on the **Main** menu which represent different options of the **ESPPRS** computer code. In order to select one of these options the mouse is used to point at the desired option while the middle button depressed to execute that option. This is also referred to as clicking, choosing, or picking a menu option. There are 11 options on the Main menu: **Load**, **Select**, **View**, **Schedule**, **Unschedule**, **Shift**, **Move**, **Nodes**, **Edit**, **Print**, **Save**, and **Exit**. Each of the Main menu options is described in detail in the following sections.

**** Note ****

The middle button is used to click on a menu option for the standard Apollo mouse. For other (non-standard) input devices, try different buttons to determine which is appropriate for clicking on menu options in **ESPPRS**.

Various error states may occur on during execution of **ESPPRS** that will hang (i.e. freeze execution) the run-time environment of **ESPPRS**. Error states may occur if the user tries to open a view window for a platform that already has a view window open or if the user opens several view windows at once (e.g. more than 10 view windows). It is a good idea to have no more than five view windows open at once to prevent the screen from becoming too cluttered. If the run-time environment hangs, "CTRL-Q" may be pressed to immediately stop execution of **ESPPRS** and return to the Apollo OS **$** prompt.

Load

This option allows the user to load one or more data sets (e.g. "rock.dat") during the run-time execution of **ESPPRS**.

1. Choose **Load** from the **Main** menu.
2. Select a timeline (i.e. scenario data set) from the **Timeline Directory** menu.

The available timelines listed in the **Timeline Directory** menu were specified in the "directory.dat" file prior to execution. Once a timeline is loaded the name of that timeline is displayed in the current timeline box located in the lower right-hand corner of the screen. Several timelines may be loaded in succession by performing repeated picks of the **Load** option.

Select

This option allows the user to change the current timeline selection when multiple timelines have been loaded.

1. Choose **Select** from the **Main** menu.
2. Select a timeline from the **Select Timeline** menu.

The **Select Timeline** menu lists the loaded timelines that may be chosen as the current timeline. Choosing one of the timelines from the **Select Timeline** menu will identify that timeline as the current timeline. This timeline is labeled in the current timeline box located in the lower right-hand corner of the screen.

View

This option can be used to view the timeline profiles, resource profiles, or activity problems (i.e. scheduling conflicts) for a platform. Choosing **View** from the **Main** menu will display the **View** menu in the middle of the screen. This menu has three options which are **Platform**, **Resource**, and **Activity_Problems**. This section provides procedures for

1. Viewing a timeline profile.
2. Using the **Digital Adjust**.
3. Display activity characteristics.
4. Viewing a resource profile.
5. Displaying aggregate sums.
6. Viewing activity problems.

Viewing a timeline profile:

You can view the timeline profile of a selected platform. The timeline profile is a chronological representation of the scheduled start date, end date and duration of each activity on a selected platform. The following procedure is used to view a timeline profile:

1. Choose **View** from the **Main** menu.
2. Choose **Platform** from the **View** menu.
3. Select the platform to be viewed from the **Platform** menu.

The timeline profile will be displayed for the selected platform. The profile image will consist of a list of all activities on the platform, a horizontal timeline axis, and a corresponding timeline symbol to the right of each activity. Two types of symbols are used to represent the timeline of an each activity. The first symbol is a horizontal green bar that indicates the start and end dates and duration of a standard activity. The second symbol is an inverted blue triangle representing the date of a milestone activity.

If there are more activities on a platform than can be displayed in the view window, the scroll bar to the right of the timeline profile may be used to view the additional activities. Using the mouse to click on the box at the bottom portion of the vertical scroll column will move the view window down and display the timelines of the additional activities on the viewed platform. Clicking on the box at the top portion of the scroll bar column will move the view window back up. Clicking on the scroll bar with the mouse and holding the button down will enable the user to slide the bar to any position along the scroll bar column.

Using the **Digital Adjust**:

You can change the portion of the timeline axis being viewed.

1. Use mouse to click on the **Digital Adjust** box at the bottom of the view window.
2. The start and end times on the timeline axis will then be displayed in format (dd month year hh mm) where:
    - dd is a 2-digit integer representing days
    - month is a 3-character abbreviation for the month
    - year is a 4-digit integer for the year
    - hh is a 2-digit integer representing hours
    - mm is a 2-digit integer representing minutes
3. To modify the start and end times do the following:
    - click above a time unit to increase its value
    - click below a time unit to decrease its value
4. Click on the abort box to ignore all changes and return to the view window.
5. Click on the execute box to implement changes and re-display the view window.

Changes made using the **Digital Adjust** will be altered slightly to maintain regularly spaced intervals along the timeline axis. This results in the start and end times on the modified timeline axis being close but not equal to those selected using the **Digital Adjust**.

Viewing resource profiles:

You can view the resource profile (e.g. power, personnel, or transportation mass) of a selected platform. The resource profile is a plot of the total of a selected resource for all activities on a selected platform. To view a resource profile perform the following steps:

1. Choose **View** from the **Main** menu.
2. Click on **Resource** from the **View** menu.
3. Select a platform from the **Platform** menu (e.g. P01).
4. Select a resource from the **Resource** menu (e.g. Tonnage).

The resource profile for the selected platform and resource will then be displayed in a view window. The resource profile will depict the aggregate of the selected resource at each point along the timeline axis. A scale for the resource sum is on the left side of the profile (y-axis) and the units are labeled with the profile's title. Clicking at any point along the timeline axis will display a resource quantity window. This window will contain a list of each activity and its corresponding quantity of the selected resource at that point. The aggregate of the quantities of the activities at that point will also be displayed. The portion of the timeline axis being viewed can be modified as described above in the **Digital Adjust** section.

Viewing activity problems:

The activities which have scheduling conflicts can be viewed. This will display a list of activities with scheduling problems and a short description of the type of conflict encountered during the scheduling process. The procedure used to view the activity problems (i.e. scheduling conflicts) is detailed below.

1. Choose **View** from the **Main** menu.
2. Choose **Activity_Problems** from the **View** menu.
3. Select which activities to view problems for from the **Problem Activities** menu and click on **Done** when finished making selections.

Schedule

The activities defined in a scenario are scheduled based on their window start and end dates and temporal relations (i.e. predecessors). The window start and end dates represent the boundary (i.e. earliest and latest possible times) during which the activity must be scheduled. The predecessor relationships describe the general order of the activities (i.e. which activities must precede other activities). To schedule the activities in the current scenario choose **Schedule** from the **Main** menu.

Unschedule

Some or all of the activities in a scenario may be unscheduled. This is useful because the characteristics of an activity can only be edited when the activity is unscheduled. To unschedule an activity perform the following steps.

1. Choose **Unschedule** from the **Main** menu.
2. Select one of the three options on the **Unschedule** menu detailed below.
    a. **All_Platforms** - unschedules all activities on all platforms.
    b. **Platform(s)** - user selects one or more platforms from the **Platform** menu to unschedule.
        - select **Done** when finished making selections.
    c. **Activity** - user selects the platform from the **Platform** menu which contains the activities to be unscheduled.
        - user selects one or more activities from the **Activity** menu to unschedule.
        - select **Done** when finished making selections.

Shift

You may change the platform assignments of activities or create new platforms using the **Shift** option. Therefore the **Shift** option is critical in implementing the platform assignments determined from a branching analysis of the scheduled activities. To shift an activity from its current platform to another platform (pre-defined or new) do the following:

1. Choose **Shift** from the **Main** menu.
2. Select the platform which contains the activity to be shifted from the **From Platform** menu.
3. Select the activity to be shifted from the **Activity** menu.
4. Select the platform to which the activity is to be shifted from the To **Platform** menu.
    - Click **New** from the **To Platform** menu if the activity is being shifted to a new (i.e. undefined) platform and key in the name of the new platform.

Move

You can alter the scheduled assignment interval of an activity. This is useful for making small changes to the scheduled start and end dates without editing and rescheduling the activities. An activity assignment interval may be moved such that the new assignment interval causes a conflict. This will most likely be caused by violating a pre-defined temporal relation. If such conflicts occur, each activity having such a conflict will be flagged on the screen display. The conflicting activities will be flagged by changing the color of their timeline symbols on the screen. Instead of the default green for standard activities or blue for milestone activities, the timeline symbol displayed on the screen will be red if that activity has a conflict. The procedure for moving an activity's assignment interval is:

1. Choose **Move** from the **Main** menu.
2. Select the platform which contains the activity to be moved from the **Platform** menu.
3. Select the activity to be moved from the **Activity** menu.
4. Use the **Digital Adjust** to modify the start date of the activity.

The **Digital Adjust** feature used with the **Move** option is slightly different from the one used in the view window. Here the target date labeled inside the **Digital Adjust** box represents the start date the activity to be moved. The target date is modified by clicking on one of the time units at the top of the **Digital Adjust** box and then clicking the "+" box to increase the target date in steps of that unit or the "-" box to decrease the target date in steps of that unit. The **Execute** box is clicked to implement a modified target date, the **Reset** box is clicked to reset the target date to its starting value, and the **Abort** box is clicked to exit the **Digital Adjust** without implementing changes to the target date.

Nodes

You can perform a nodal (e.g. LEO, GEO, Mars) analysis on the total resources in a scenario. To do this an external file called "nodes.dat" must be in the current working directory. The "nodes.dat" file contains information concerning the nodal location of each platform. Note the node assignment must be in quotes

(" "). Clicking on **Nodes** from the **Main** menu will perform a nodal analysis on the current timeline and write the analysis results to two output files. Both files will be identical with one having the same name as the current timeline with a ".prn" extension and the other being named "nodes.prn". Either file can be transferred to a MS-DOS diskette and imported to a Lotus-123 spreadsheet labeled "nodes.wkl". This spreadsheet can be used to produce graphs on the nodal analysis data and is provided with the **ESPPRS** software. Sample listings of the "nodes.dat" and "nodes.prn" data files are provided in Appendix C.

Edit

You can edit the characteristics of an unscheduled activity. This can be used to add, change or delete an activity and its associated characteristics (i.e. resources and temporal relations). To do this perform the following steps:

1. Choose **Edit** from the **Main** menu.
2. Select the platform which contains the activity to be edited from the **Platform** menu.

The activity characteristics which can be edited will then be displayed in an edit window. The activities that are on the platform will be listed under the **Activities** heading on the left side of the edit window. The platform name is labeled at the top of the edit window next to the **Operations Sequence** heading. Click on the name of an activity under the **Activities** heading to display and edit its characteristics. The activity characteristics are displayed under three headings: **Activity Editor**, **Resources**, and **Temporal Relations**. The characteristics displayed under the **Activity Editor** include the name, description, window start and end, duration, and phase. The characteristics displayed under the **Resources** heading include resource name, start and end time, and quantity. The characteristics displayed under the **Temporal Relations** heading include S/E, >=<, E/S, op sequence, activity name, and constant. Any of these characteristics may be changed by simply clicking on the characteristic and keying in a new value.

The first three fields under the **Temporal Relations** heading (S/E, >=<, and E/S) are used to specify the type of temporal relationship between two activities. The first field refers to the start (S) or end (E) of the activity being edited. The second field basically determines whether the temporal relation is a successor or a predecessor. A predecessor relation requires that the activity being edited must be scheduled greater than or equal (>=) to its related activity. Thus a ">=" is placed in the second field. A successor relation requires that the activity being edited be scheduled less than or equal (<=) to its related activity. Thus a "<=" is placed in the second field. The third field refers to the end (E) or start (S) of the activity that is being linked (via a temporal relation) to the activity being edited. The op sequence field (fourth) displays the platform that the related activity is assigned to. The fifth field contains the name of the related activity. The sixth field is a constant that represents a scheduling time delay for the temporal relationship.

An activity, resource, or temporal relation may be added, copied, or deleted using the editor. To add an activity, resource, or temporal relation, click on the **Add** box located in the upper right corner of the edit window. Then click on the appropriate box (**Act**, **Res**, **Rel**, or **Abort**) from the **Add** window to add an activity, resource, or temporal relation. To copy or delete an activity, click on the **Copy** or **Delete** box from the upper right corner of the edit window and then select the activity (from the list under the **Activities** heading) to be copied or deleted. To copy or delete a resource, click on the resource name under the **Resources** heading and then click the **Copy** or **Delete** box from the small resource window. To copy or delete a temporal relation, click on the item in the S/E field under the **Temporal Relations** heading.

To implement editing changes, additions, and deletions made to activities on the platform being edited and exit the editor, click on the **Execute** box in the upper right corner of the edit window. To return all activities to their original settings before editing began, click on the **Reset** box in the upper right corner of the edit window. The **Abort** box in the upper right corner can be used to exit the editor without implementing any modifications to activities on the platform.

Print

This option can be used to print timeline profile or resource profile reports for a platform. Choosing **Print** from the **Main** menu will display the **Timeframe per Page** menu in the middle of the screen. This menu has two options which include **Decade** or **Complete Timeline**. Choosing the **Decade** option allows the user to generate a series timeline or resource reports of a platform for the entire scenario at decade intervals. The following procedure describes the use of the **Decade** option:

1. Choose **Print** from the **Main** menu.
2. Select **Decade** from the **Timeframe per Page** menu.
3. Select **Platform Activities** or **Platform Resources** from the **Plot Type** menu.
    - **Platform Activities** denotes a timeline profile report.
    - **Platform Resources** denotes a resource profile report.
4. Select a platform from the **Platform** menu (e.g. P02)
5. Use the **Digital Adjust** to select the start date for the reports.
6. If **Platform Resources** was chosen in step 3 then choose the type of resource for the report (e.g. Electric_Power, Personnel, or Tonnage).

Choosing the **Complete Timeline** option will generate timeline or resource profile reports for a specified range of dates. The following procedure describes the use of the **Complete Timeline** option:

1. Choose **Print** from the **Main** menu.
2. Select **Complete Timeline** from the **Timeframe per Page** menu.
3. Select **Platform Activities** or **Platform Resources** from the **Plot Type** menu.
    - **Platform Activities** denotes a timeline profile report.
    - **Platform Resources** denotes a resource profile report.
4. Select a platform from the **Platform** menu (e.g. P02)
5. Use the **Digital Adjust** to specify the start to end date range for the profile report.
6. If **Platform Resources** was chosen in step 3 then choose the type of resource for the report (e.g. Electric_Power, Personnel, Tonnage).

The **Print** option is configured to print the profile reports directly to an Apple LaserWriter or similar printer physically connected to an Apollo serial port. Each report that is printed during the run-time execution of **ESPPRS** is saved to a file in the current working directory (i.e. the directory from which **ESPPRS** was executed). The names of saved print files are print0001, print0002, print0003, etc. labeled in the order in which they were printed from **ESPPRS**. These files may be printed from the Apollo OS by executing the command:

```
"prf -trans print0001" <CR>
```

Sample profile reports generated from **ESPPRS** for both timeline and resource profiles are presented in Appendix D.

Save

You can save the current timeline to a external data file by choosing **Save** from the **Main** menu. The save file name is the input file name listed in the "directory.dat" file.

Exit

You can exit the **ESPPRS** program by choosing **Exit** from the **Main** menu. The program will then ask whether or not you really want to exit and you may click either the **Yes** or **No** box in response. If the first response is **Yes** the program will ask if you want to save each timeline that was loaded during the **ESPPRS** run-time session. You can answer each of these queries by clicking a **Yes** or **No** response.

Post-Processing Modules

Three post-processing modules have been written to provide additional report capabilities for **ESPPRS**. These modules include "report1", "report2", and "report3" and a brief description of each will follow.

Report1

This module uses "espprs.dat" as an input file and produces a tabular report in an output file called "report1.dat". The report is a listing of activities from an **ESPPRS** scenario including the activity name, platform assignment, resources, scheduled start and end dates, and duration. The listing is sorted first by platform assignment and then by scheduled start date. A sample listing of the "report1.dat" file generated by **Report1** is presented in Appendix E.

Report2

This module uses "espprs.dat" as an input file and produces a tabular report in an output file called "report2.dat". The report is a listing of activities from an **ESPPRS** scenario including the activity name, platform assignment, resources, duration, and activity description. This listing is sorted by activity name. A sample listing of the "report2.dat" file generated by **Report2** is presented in Appendix E.

Report3

This module uses "espprs.dat" as an input file and produces a tabular report for each activity resource (i.e. electric power, personnel, initial tonnage, repeat tonnage, and tonnage). These reports contain resource totals for each platform for an **ESPPRS** scenario. The "plats.dat" file is a second input file which contains a list of platform names to be processed by **Report3**. The output files generated by **Report3** are called "power.dat", "pers.dat", "iton.dat", "rton.dat", and "mass.dat" and contain platform totals for each of their respective resources. Sample listings of "plats.dat" and the associated resource output files are provided in Appendix F.

Glossary

Activity
A space-based operation which provides a useful function or purpose towards fulfilling the established objectives and goals of a scenario.

Node
A physical space-based location used to quantify the transportation requirements of a scenario.

Platform
A grouping of activities onto a physical space platforms by process, location, and environmental compatibility.

Resource
A need/requirement for an activity such as power, personnel, transportation mass, or equipment.

Resource Profile
A graphical representation of the resources needed by the set of activities on a platform over time.

Scenario
Assumptions and guidelines for an logical expansion into space. Defines a consistent framework of activities to meet a set of pre-determined goals.

Temporal Relation
A time and/or sequence constraint between two activities.

Timeline
A traceable, evolutionary projection of future space activities and their associated resources.

Timeline Profile
A graphical representation of timelines for a set of activities.

Appendix A

- add.out
- add.run
- add.t
- atran
- clip.dat
- clip.out
- clip.run
- clip.t
- cltran

Trees Software

- inerr.dat
- createlv.t
- forlib.dat
- forest.lib
- rterr.dat
- sortlv
- trees
- trees.small
- trees.big
- twm.lib
- tree.dat
- trest
- treeslib.lib

crystal_growth.prn
nodes.prn

Convert Utility

activity.dat
timeline.dat
convert.out
convert.run
convert.t
ctran
rocketdyne.dat

Mk_Input Utility

mk_input.dat
mk_input.out
mk_input.run
mk_input.t
mktran
rock.dat

Merge Utility

merge.out
merge.run
merge.t
mtran
rock1.dat
rock2.dat
big.dat

Report Utilities

report1.dat
report1.out
report1.run
report1.t
report2.dat
report2.out
report2.run
report2.t
report3.dat
report3.out
report3.run
report3.t
rtran1
rtran2
rtran3
mass.dat
power.dat
pers.dat
iton.dat
rton.dat
plats.dat
espprs.dat

Miscellaneous Utilities

add.dat

ESPPRS Directory Hierarchy

ESPPRS Source Code

- digital.t
- espprs.t
- espprslib.t
- nodes_espprs.t
- print.t
- timeline.t
- utilities.t
- y-scale.t

ESPPRS Libraries

- avyx.lib
- digital.lib
- espprs.lib
- nodes.lib
- print.lib
- timeline.lib
- utilities.lib
- y-scale.lib

ESPPRS Executable Files

- espprs.run
- run-espprs
- run-bigespprs
- translate
- ntran

ESPPRS Setup Files

- config.dat
- hardware.dat
- templates.dat

ESPPRS I/O Files

- espprs.out
- directory.dat
- mfg.dat
- nodes.dat
- rock.dat_cd
- rock.dat_cg
- rock.dat_cp
- rock.dat_ff
- rock.dat_gl
- rock.dat_ls
- rock.dat_lunar
- rock.dat_mars
- rock.dat_mm
- rock.dat_mp
- rock.dat_ph
- rock.dat_rm
- rock.dat_ss
- rock.dat_sv
- rock.dat.xp
- print0001

year	P01	P02	P09	P11	P12	P99
1992	0.0	0.0	560.0	0.0	0.0	0.0
1993	0.0	0.0	2880.0	0.0	0.0	0.0
1994	0.0	0.0	2880.0	0.0	0.0	0.0
1995	60.0	0.0	2880.0	0.0	0.0	0.0
1996	1440.0	0.0	0.0	0.0	0.0	0.0
1997	1440.0	0.0	0.0	0.0	0.0	0.0
1998	1440.0	0.0	0.0	0.0	0.0	0.0
1999	0.0	0.0	0.0	399.3	0.0	0.0
2000	0.0	17060.0	0.0	447.2	0.0	0.0
2001	0.0	30.2	0.0	500.9	0.0	0.0
2002	0.0	30.2	0.0	561.0	0.0	0.0
2003	0.0	30.2	0.0	628.3	0.0	0.0
2004	0.0	30.2	0.0	703.7	0.0	0.0
2005	0.0	34150.2	0.0	788.2	0.0	0.0
2006	0.0	1470.2	0.0	882.8	0.0	0.0
2007	0.0	1470.2	0.0	988.7	0.0	0.0
2008	0.0	36925.2	0.0	1107.4	0.0	0.0
2009	0.0	1460.0	0.0	1240.2	1240.2	0.0
2010	0.0	1460.0	0.0	1389.1	1389.1	0.0
2011	0.0	1460.0	0.0	1555.8	1555.8	0.0
2012	0.0	0.0	0.0	1742.4	4284.4	0.0
2013	0.0	0.0	0.0	1951.5	4532.1	0.0
2014	0.0	0.0	0.0	2185.7	4805.5	0.0
2015	0.0	0.0	0.0	2448.0	5107.6	0.0
2016	0.0	0.0	0.0	2741.8	5441.8	0.0
2017	0.0	18395.0	0.0	3070.8	5811.8	0.0
2018	0.0	40.0	0.0	3439.3	6222.0	0.0
2019	0.0	40.0	0.0	3852.0	6677.0	0.0
2020	0.0	37370.0	0.0	4314.2	7182.2	0.0
2021	0.0	380.0	0.0	4831.9	7743.5	0.0
2022	0.0	380.0	0.0	5411.8	8367.6	0.0
2023	0.0	380.0	0.0	6061.2	9061.9	0.0
2024	0.0	60.0	0.0	6788.5	9902.8	0.0
2025	0.0	19310.0	0.0	7603.1	10771.8	0.0
2026	0.0	480.1	0.0	8515.5	11740.3	0.0
2027	0.0	480.1	0.0	9537.4	12820.2	0.0
2028	0.0	480.1	0.0	10681.9	14024.5	0.0
2029	0.0	0.0	0.0	11963.7	27332.1	0.0
2030	0.0	0.0	0.0	13399.3	30267.6	0.0
2031	0.0	0.0	0.0	15007.3	33550.3	0.0
2032	0.0	0.0	0.0	16808.1	37221.5	1.2
2033	0.0	0.0	0.0	18825.1	41327.8	1.2
2034	0.0	0.0	0.0	21084.1	45921.5	1.2
2035	0.0	0.0	0.0	23614.2	51060.8	1.2
2036	0.0	0.0	0.0	26447.9	56811.2	1.2
2037	0.0	0.0	0.0	29621.7	63246.0	1.3
2038	0.0	0.0	0.0	33176.3	70447.1	1.3
2039	0.0	0.0	0.0	37157.4	78506.4	1.3
2040	0.0	0.0	0.0	41616.3	87526.8	1.3
2041	0.0	0.0	0.0	46610.3	97623.6	1.3
2042	0.0	0.0	0.0	52203.5	108925.9	1.4
2043	0.0	0.0	0.0	58467.9	121578.1	1.4

year	P01	P02	P09	P11	P12	P99
1993	0.0	0.0	2880.0	0.0	0.0	0.0
1994	0.0	0.0	2880.0	0.0	0.0	0.0
1995	0.0	0.0	2880.0	0.0	0.0	0.0
1996	1440.0	0.0	0.0	0.0	0.0	0.0
1997	1440.0	0.0	0.0	0.0	0.0	0.0
1998	1440.0	0.0	0.0	0.0	0.0	0.0
2001	0.0	30.2	0.0	0.0	0.0	0.0
2002	0.0	30.2	0.0	0.0	0.0	0.0
2003	0.0	30.2	0.0	0.0	0.0	0.0
2004	0.0	30.2	0.0	0.0	0.0	0.0
2005	0.0	30.2	0.0	0.0	0.0	0.0
2006	0.0	1470.2	0.0	0.0	0.0	0.0
2007	0.0	1470.2	0.0	0.0	0.0	0.0
2008	0.0	1470.2	0.0	0.0	0.0	0.0
2009	0.0	1460.0	0.0	0.0	0.0	0.0
2010	0.0	1460.0	0.0	0.0	0.0	0.0
2011	0.0	1460.0	0.0	0.0	0.0	0.0
2018	0.0	40.0	0.0	0.0	0.0	0.0
2019	0.0	40.0	0.0	0.0	0.0	0.0
2020	0.0	40.0	0.0	0.0	0.0	0.0
2021	0.0	380.0	0.0	0.0	0.0	0.0
2022	0.0	380.0	0.0	0.0	0.0	0.0
2023	0.0	380.0	0.0	0.0	0.0	0.0
2024	0.0	60.0	0.0	0.0	0.0	0.0
2025	0.0	60.0	0.0	0.0	0.0	0.0
2026	0.0	480.1	0.0	0.0	0.0	0.0
2027	0.0	480.1	0.0	0.0	0.0	0.0
2028	0.0	480.1	0.0	0.0	0.0	0.0
2029	0.0	1.0	0.0	0.0	0.0	0.0
2030	0.0	1.0	0.0	0.0	0.0	0.0
2031	0.0	1.0	0.0	0.0	0.0	0.0

year	P01	P02	P09	P11	P12	P99
1992	0.0	0.0	560.0	0.0	0.0	0.0
1995	60.0	0.0	0.0	0.0	0.0	0.0
2000	0.0	17060.0	0.0	0.0	0.0	0.0
2005	0.0	34120.0	0.0	0.0	0.0	0.0
2008	0.0	35455.0	0.0	0.0	0.0	0.0
2017	0.0	18395.0	0.0	0.0	0.0	0.0
2020	0.0	37330.0	0.0	0.0	0.0	0.0
2025	0.0	19250.0	0.0	0.0	0.0	0.0
2028	0.0	315.0	0.0	0.0	0.0	0.0

```
2047        0.0      5000.0         0.0      8180.4    17441.8        163.1
2048        0.0      5000.0         0.0      9162.0    19493.0        165.6
2049        0.0      5000.0         0.0     10261.5    21789.7        168.1
2050        0.0      5000.0         0.0     11492.8    24361.4        170.6
```

```
year      P01       P02       P09       P11       P12       P99


1995    530.0       0.0       0.0       0.0       0.0       0.0
1996    530.0       0.0       0.0       0.0       0.0       0.0
1997    530.0       0.0       0.0       0.0       0.0       0.0
1998    530.0       0.0       0.0       0.0       0.0       0.0
1999      0.0       0.0       0.0      35.5       0.0       0.0
2000      0.0    3936.0       0.0      39.8       0.0       0.0
2001      0.0    3936.0       0.0      44.5       0.0       0.0
2002      0.0    3936.0       0.0      49.9       0.0       0.0
2003      0.0    3936.0       0.0      55.9       0.0       0.0
2004      0.0    3936.0       0.0      62.6       0.0       0.0
2005      0.0    8333.0       0.0      70.1       0.0       0.0
2006      0.0    4397.0       0.0      78.5       0.0       0.0
2007      0.0    4397.0       0.0      87.9       0.0       0.0
2008      0.0    8863.0       0.0      98.5       0.0       0.0
2009      0.0    4466.0       0.0     110.3      82.7       0.0
2010      0.0    4466.0       0.0     123.5      92.6       0.0
2011      0.0    4466.0       0.0     138.3     103.7       0.0
2012      0.0       0.0       0.0     154.9     351.3       0.0
2013      0.0       0.0       0.0     173.5     368.8       0.0
2014      0.0       0.0       0.0     194.3     388.0       0.0
2015      0.0       0.0       0.0     217.7     409.2       0.0
2016      0.0       0.0       0.0     243.8     432.5       0.0
2017      0.0    3936.0       0.0     273.0     458.3       0.0
2018      0.0    3936.0       0.0     305.8     486.7       0.0
2019      0.0    3936.0       0.0     342.5     518.1       0.0
2020      0.0    8402.0       0.0     383.6     552.9       0.0
2021      0.0    4466.0       0.0     429.6     591.4       0.0
2022      0.0    4466.0       0.0     481.2     634.2       0.0
2023      0.0    4466.0       0.0     538.9     681.6       0.0
2024      0.0    3936.0       0.0     603.6     753.0       0.0
2025      0.0    8402.0       0.0     676.0     813.9       0.0
2026      0.0    4466.0       0.0     757.2     881.5       0.0
2027      0.0    4466.0       0.0     848.0     956.9       0.0
2028      0.0    5311.0       0.0     949.8    1040.8       0.0
2029      0.0     530.0       0.0    1063.8    2520.1       0.0
2030      0.0     531.0       0.0    1191.4    2790.7       0.0
2031      0.0     531.0       0.0    1334.4    3093.3       0.0
2032      0.0       0.0       0.0    1494.5    3431.6     130.5
2033      0.0       0.0       0.0    1673.9    3810.1     132.4
2034      0.0       0.0       0.0    1874.7    4233.5     134.4
2035      0.0       0.0       0.0    2099.7    4707.2     136.5
2036      0.0       0.0       0.0    2351.7    5237.2     138.5
2037      0.0       0.0       0.0    2633.9    5830.3     140.6
2038      0.0       0.0       0.0    2949.9    6494.0     142.7
2039      0.0    5000.0       0.0    3303.9    7236.8     144.8
2040      0.0    5000.0       0.0    3700.4    8068.1     147.0
2041      0.0    5000.0       0.0    4144.4    8998.7     149.2
2042      0.0    5000.0       0.0    4641.8   10040.4     151.4
2043      0.0    5000.0       0.0    5198.8   11206.5     153.7
2044      0.0    5000.0       0.0    5822.6   12512.0     156.0
2045      0.0    5000.0       0.0    6521.3   13973.5     158.4
2046      0.0    5000.0       0.0    7303.9   15609.7     160.7
```

2044	0.0	0.0	0.0	682.1	44963.9	17.5
2045	0.0	0.0	0.0	764.0	50305.6	17.7
2046	0.0	0.0	0.0	855.7	56287.4	18.0
2047	0.0	0.0	0.0	958.3	62986.4	18.3
2048	0.0	0.0	0.0	1073.3	70488.2	18.6
2049	0.0	0.0	0.0	1202.1	78889.5	18.8
2050	0.0	0.0	0.0	1346.4	88298.1	19.1

year	P01	P02	P09	P11	P12	P99
1992	0.0	0.0	10.0	0.0	0.0	0.0
1993	0.0	0.0	10.0	0.0	0.0	0.0
1994	0.0	0.0	10.0	0.0	0.0	0.0
1995	10.0	0.0	10.0	0.0	0.0	0.0
1996	10.0	0.0	0.0	0.0	0.0	0.0
1997	10.0	0.0	0.0	0.0	0.0	0.0
1998	10.0	0.0	0.0	0.0	0.0	0.0
1999	0.0	0.0	0.0	4.2	0.0	0.0
2000	0.0	36.3	0.0	4.7	0.0	0.0
2001	0.0	36.3	0.0	5.2	0.0	0.0
2002	0.0	36.3	0.0	5.8	0.0	0.0
2003	0.0	36.3	0.0	6.5	0.0	0.0
2004	0.0	36.3	0.0	7.3	0.0	0.0
2005	0.0	217.6	0.0	8.2	0.0	0.0
2006	0.0	181.3	0.0	9.2	0.0	0.0
2007	0.0	181.3	0.0	10.3	0.0	0.0
2008	0.0	376.2	0.0	11.5	0.0	0.0
2009	0.0	194.9	0.0	12.9	155.0	0.0
2010	0.0	194.9	0.0	14.5	173.6	0.0
2011	0.0	194.9	0.0	16.2	194.5	0.0
2012	0.0	0.0	0.0	18.2	535.5	0.0
2013	0.0	0.0	0.0	20.3	566.5	0.0
2014	0.0	0.0	0.0	22.8	600.7	0.0
2015	0.0	0.0	0.0	25.5	638.5	0.0
2016	0.0	0.0	0.0	28.6	680.2	0.0
2017	0.0	99.7	0.0	32.0	726.5	0.0
2018	0.0	99.7	0.0	35.8	777.8	0.0
2019	0.0	99.7	0.0	40.1	834.6	0.0
2020	0.0	455.3	0.0	44.9	897.8	0.0
2021	0.0	355.6	0.0	50.3	967.9	0.0
2022	0.0	355.6	0.0	56.4	1045.9	0.0
2023	0.0	355.6	0.0	63.1	1132.7	0.0
2024	0.0	156.1	0.0	70.7	1264.6	0.0
2025	0.0	475.1	0.0	79.2	1376.4	0.0
2026	0.0	319.0	0.0	88.7	1501.1	0.0
2027	0.0	319.0	0.0	99.3	1640.1	0.0
2028	0.0	330.9	0.0	111.3	1795.2	0.0
2029	0.0	11.9	0.0	124.6	8531.3	0.0
2030	0.0	11.9	0.0	139.6	9512.0	0.0
2031	0.0	11.9	0.0	156.3	10609.8	0.0
2032	0.0	0.0	0.0	175.1	11838.6	14.6
2033	0.0	0.0	0.0	196.1	13214.2	14.8
2034	0.0	0.0	0.0	219.6	14754.2	15.1
2035	0.0	0.0	0.0	246.0	16478.3	15.3
2036	0.0	0.0	0.0	275.5	18408.6	15.5
2037	0.0	0.0	0.0	308.6	20569.8	15.7
2038	0.0	0.0	0.0	345.6	22989.6	16.0
2039	0.0	0.0	0.0	387.1	25699.0	16.2
2040	0.0	0.0	0.0	433.5	28732.9	16.5
2041	0.0	0.0	0.0	485.5	32130.0	16.7
2042	0.0	0.0	0.0	543.8	35934.0	17.0
2043	0.0	0.0	0.0	609.0	40193.8	17.2

```
@
   P01
   P02
   P09
   P11
   P12
   P99
END
```

Appendix B

```
 1 CG0001  07/03/1989 ASAP       -99 P99 0.00    0.0    0     0.0    0.0 1096 CG0001:STS Crystal Growth Experiments
 2 CG0002  ASAP       ASAP         1 P09 1.00   10.0    0   560.0 2880.0 1089 CG0002:FF Crystal Growth Experiments
 3 CG0003  ASAP       ASAP         2 P99 0.00    0.0    0     0.0    0.0    0 CG0003:Decision Growth Method
 4 CG0004  06/27/1995 ASAP         2 P99 0.00    0.0    0     0.0    0.0    0 CG0004:Space Station IOC
 5 CG0005  ASAP       ASAP         3 P01 1.00   10.0  530    60.0 1440.0 1091 CG0005:Pilot Plant Dev & Production
 5 CG0005  ASAP       ASAP         4 P01 1.00   10.0  530    60.0 1440.0 1091 CG0005:Pilot Plant Dev & Production
 6 CG0006  ASAP       ASAP         5 P11 0.00    0.0    0     0.0    0.0  727 CG0006:Commercial Plant Development
 7 CG0007  ASAP       ASAP         6 P99 0.00    0.0    0     0.0    0.0    0 CG0007:Commercial Microchip Production
 8 CG0008  ASAP       ASAP         6 P99 0.00    0.0    0     0.0    0.0    0 CG0008:M/S*FL1* Link from SPACESCI
 9 CG0009  ASAP       ASAP         6 P02 0.20  181.3 3936 17060.0   30.2 1819 CG0009:Exps. Lattice Const and Bandgap
 9 CG0009  ASAP       ASAP         8 P02 0.20  181.3 3936 17060.0   30.2 1819 CG0009:Exps. Lattice Const and Bandgap
10 CG0010  ASAP       ASAP         9 P99 0.00    0.0    0     0.0    0.0    0 CG0010:M/S*CG2* Link to GLASSES,MPOI
11 CG0011  ASAP       ASAP         9 P99 0.00    0.0    0     0.0    0.0    0 CG0011:Dec. Photochip Production
12 CG0012  ASAP       ASAP         9 P02 0.80  181.3  461 17060.0 1440.0 1091 CG0012:Pilot Plant Dev & Production
12 CG0012  ASAP       ASAP        11 P02 0.80  181.3  461 17060.0 1440.0 1091 CG0012:Pilot Plant Dev & Production
13 CG0013  ASAP       ASAP        12 P12 0.00    0.0    0     0.0    0.0  727 CG0013:Dev Commercial Plant
14 CG0014  ASAP       ASAP        13 P99 0.00    0.0    0     0.0    0.0    0 CG0014:Commercial Photochip
15 CG0015  ASAP       ASAP         9 P02 0.20  181.3 3936 17060.0   30.2 1091 CG0015:Exp.SLS Lattice Matching
16 CG0016  ASAP       ASAP        15 P02 0.80  181.3  530 17060.0 1440.0 1091 CG0016:Develop Pilot Plant
17 CG0017  ASAP       ASAP        16 P12 0.00    0.0    0     0.0    0.0  727 CG0017:Commercial Plant Development
18 CG0018  ASAP       ASAP        17 P99 0.00    0.0    0     0.0    0.0    0 CG0018:Commercial Tandem Cells
19 CG0019  ASAP       ASAP        15 P99 0.00    0.0    0     0.0    0.0    0 CG0019:M/S*CD3* Link from COATINGS
20 CG0020  ASAP       ASAP        15 P02 0.20  249.3 3936 18395.0   20.0 1091 CG0020:Exp. MBE for Dopant AD.
20 CG0020  ASAP       ASAP        19 P02 0.20  249.3 3936 18395.0   20.0 1091 CG0020:Exp. MBE for Dopant AD.
21 CG0021  ASAP       ASAP        20 P99 0.00    0.0    0     0.0    0.0    0 CG0021:M/S*CG1* Link to GLASSES,METALLIC
22 CG0022  06/05/2017 ASAP        20 P99 0.00    0.0    0     0.0    0.0    0 CG0022:M/S*MMD3* Link from METALLIC
23 CG0023  ASAP       ASAP        20 P02 0.40  249.3 3936 18395.0   40.0 1089 CG0023:Opt.Filter Proof-of-Concept
23 CG0023  ASAP       ASAP        22 P02 0.40  249.3 3936 18395.0   40.0 1089 CG0023:Opt.Filter Proof-of-Concept
24 CG0024  ASAP       ASAP        23 P02 0.80  249.3  530 18395.0  320.0 1089 CG0024:Pilot Plant Dev & Production
25 CG0025  ASAP       ASAP        24 P12 0.00    0.0    0     0.0    0.0  725 CG0025:Commercial Plant Development
26 CG0026  ASAP       ASAP        25 P12 0.00    0.0    0     0.0    0.0    0 CG0026:Commercial Optical Filters
27 CG0027  ASAP       ASAP        23 P02 0.40  390.3 3936 18935.0   60.0 1817 CG0027:Expmts-Optical Storage Informatn
28 CG0028  ASAP       ASAP        27 P02 0.80  395.0  530 18935.0  480.0 1089 CG0028:Optical Stor.Pilot Plt Dev/Prodn
29 CG0029  ASAP       ASAP        28 P12 0.00    0.0    0     0.0    0.0  725 CG0029:Comm'l Optical Stor.Plant Devlpmt
30 CG0030  ASAP       ASAP        29 P12 0.00    0.0    0     0.0    0.0    0 CG0030:Begin Comm'l Prodn Opt.StrgDevice
31 CG0031  ASAP       ASAP        27 P02 0.20   14.9 3936   315.0    0.1 1089 CG0031:Expmts,Pseudo-Cryst,Prec.OptMsrmt
32 CG0032  ASAP       ASAP        31 P02 0.80   14.9  530   315.0    1.0 1089 CG0032:Pseudo-Crystl Pilot Plt Dev/Prodn
33 CG0033  ASAP       ASAP        32 P99 0.00    0.0    0     0.0    0.0  725 CG0033:Comm'l Pseudo-Crystl Plant Prodn
34 CG0034  ASAP       ASAP        33 P12 0.00    0.0    0     0.0    0.0    0 CG0034:Begin Prodn PseudoCrystl MsmrtDev
```

Appendix C

```
NODES
   P01 - 'NODE_01'
   P02 - 'NODE_01'
   P03 - 'NODE_01'
   P04 - 'NODE_01'
   P06 - 'NODE_01'
   P08 - 'NODE_01'
   P09 - 'NODE_01'
   P10 - 'NODE_01'
   P11 - 'NODE_01'
   P12 - 'NODE_01'
   P15 - 'NODE_01'
   P18 - 'NODE_01'
   P19 - 'NODE_01'
   P20 - 'NODE_01'
   P21 - 'NODE_01'
   P23 - 'NODE_01'
   P24 - 'NODE_01'
   P26 - 'NODE_01'
   P29 - 'NODE_01'
   P31 - 'NODE_01'
   P32 - 'NODE_01'
   P22 - 'NODE_02'
   P05 - 'NODE_03'
   P28 - 'NODE_04'
   P07 - 'NODE_05'
   P14 - 'NODE_05'
   P16 - 'NODE_06'
   P30 - 'NODE_06'
   P13 - 'NODE_07'
   P17 - 'NODE_07'
   P25 - 'NODE_07'
   P33 - 'NODE_08'
   P34 - 'NODE_08'
   P35 - 'NODE_08'
   P36 - 'NODE_08'
   P37 - 'NODE_08'
   P38 - 'NODE_08'
   P39 - 'NODE_08'
   P41 - 'NODE_08'
   P42 - 'NODE_08'
   P44 - 'NODE_08'
   P45 - 'NODE_08'
   P46 - 'NODE_08'
   P47 - 'NODE_08'
   P48 - 'NODE_09'
   P40 - 'NODE_10'
   P27 - 'NODE_11'
   P43 - 'NODE_12'
   P50 - 'NODE_13'
   P51 - 'NODE_13'
   P52 - 'NODE_13'
   P53 - 'NODE_13'
   P54 - 'NODE_13'
   P55 - 'NODE_13'
```

```
"INITIAL_TONNAGE"

"year"  "  NODE_01"

1989          0.0
1990          0.0
1991          0.0
1992        560.0
1993          0.0
1994          0.0
1995         60.0
1996          0.0
1997          0.0
1998          0.0
1999          0.0
2000      17060.0
2001          0.0
2002          0.0
2003          0.0
2004          0.0
2005      34120.0
2006          0.0
2007          0.0
2008      35455.0
2009          0.0
2010          0.0
2011          0.0
2012          0.0
2013          0.0
2014          0.0
2015          0.0
2016          0.0
2017      18395.0
2018          0.0
2019          0.0
2020      37330.0
2021          0.0
2022          0.0
2023          0.0
2024          0.0
2025      19250.0
2026          0.0
2027          0.0
2028        315.0
2029          0.0
2030          0.0
2031          0.0
2032          0.0
2033          0.0
2034          0.0
2035          0.0
2036          0.0
2037          0.0
2038          0.0
```

```
    P56 - 'NODE_13'
    P60 - 'NODE_14'
    P64 - 'NODE_14'
    P67 - 'NODE_14'
    P68 - 'NODE_14'
    P69 - 'NODE_14'
    P70 - 'NODE_14'
    P58 - 'NODE_15'
    P66 - 'NODE_15'
    P62 - 'NODE_16'
    P61 - 'NODE_17'
    P57 - 'NODE_18'
    P99 - 'NODE_99'
    X01 - 'NODE_XX'
    X02 - 'NODE_XX'
    X03 - 'NODE_XX'
    X04 - 'NODE_XX'
    X05 - 'NODE_XX'
    X06 - 'NODE_XX'
END
```

PRECEDING PAGE BLANK NOT FILMED

```
2039          0.0
2040          0.0
2041          0.0
2042          0.0
2043          0.0
2044          0.0
2045          0.0
2046          0.0
2047          0.0
2048          0.0
2049          0.0
2050          0.0

"REPEAT_TONNAGE"

"year"  "  NODE_01"

1989          0.0
1990          0.0
1991          0.0
1992          0.0
1993       2880.0
1994       2880.0
1995       2880.0
1996       1440.0
1997       1440.0
1998       1440.0
1999          0.0
2000          0.0
2001         30.2
2002         30.2
2003         30.2
2004         30.2
2005         30.2
2006       1470.2
2007       1470.2
2008       1470.2
2009       1460.0
2010       1460.0
2011       1460.0
2012          0.0
2013          0.0
2014          0.0
2015          0.0
2016          0.0
2017          0.0
2018         40.0
2019         40.0
2020         40.0
2021        380.0
2022        380.0
2023        380.0
2024         60.0
2025         60.0
2026        480.1
```

```
2027        480.1
2028        480.1
2029          1.0
2030          1.0
2031          1.0
2032          0.0
2033          0.0
2034          0.0
2035          0.0
2036          0.0
2037          0.0
2038          0.0
2039          0.0
2040          0.0
2041          0.0
2042          0.0
2043          0.0
2044          0.0
2045          0.0
2046          0.0
2047          0.0
2048          0.0
2049          0.0
2050          0.0

"ELECTRIC_POWER"

"year"  "  NODE_01"

1989          0.0
1990          0.0
1991          0.0
1992         10.0
1993         10.0
1994         10.0
1995         20.0
1996         10.0
1997         10.0
1998         10.0
1999          0.0
2000         36.3
2001         36.3
2002         36.3
2003         36.3
2004         36.3
2005        217.6
2006        181.3
2007        181.3
2008        376.2
2009        194.9
2010        194.9
2011        194.9
2012          0.0
2013          0.0
2014          0.0
```

```
2015          0.0
2016          0.0
2017         99.7
2018         99.7
2019         99.7
2020        455.3
2021        355.6
2022        355.6
2023        355.6
2024        156.1
2025        475.1
2026        319.0
2027        319.0
2028        330.9
2029         11.9
2030         11.9
2031         11.9
2032          0.0
2033          0.0
2034          0.0
2035          0.0
2036          0.0
2037          0.0
2038          0.0
2039          0.0
2040          0.0
2041          0.0
2042          0.0
2043          0.0
2044          0.0
2045          0.0
2046          0.0
2047          0.0
2048          0.0
2049          0.0
2050          0.0

"PERSONNEL"

"year"  "  NODE_01"

1989          0.0
1990          0.0
1991          0.0
1992          0.0
1993          0.0
1994          0.0
1995        530.0
1996        530.0
1997        530.0
1998        530.0
1999          0.0
2000       3936.0
2001       3936.0
2002       3936.0
```

```
2003       3936.0
2004       3936.0
2005       8333.0
2006       4397.0
2007       4397.0
2008       8863.0
2009       4466.0
2010       4466.0
2011       4466.0
2012          0.0
2013          0.0
2014          0.0
2015          0.0
2016          0.0
2017       3936.0
2018       3936.0
2019       3936.0
2020       8402.0
2021       4466.0
2022       4466.0
2023       4466.0
2024       3936.0
2025       8402.0
2026       4466.0
2027       4466.0
2028       4996.0
2029        530.0
2030        530.0
2031        530.0
2032          0.0
2033          0.0
2034          0.0
2035          0.0
2036          0.0
2037          0.0
2038          0.0
2039          0.0
2040          0.0
2041          0.0
2042          0.0
2043          0.0
2044          0.0
2045          0.0
2046          0.0
2047          0.0
2048          0.0
2049          0.0
2050          0.0

"TONNAGE"

"year"  "  NODE_01"

1989          0.0
1990          0.0
```

```
1991          0.0
1992        560.0
1993       2880.0
1994       2880.0
1995       2940.0
1996       1440.0
1997       1440.0
1998       1440.0
1999          0.0
2000      17060.0
2001         30.2
2002         30.2
2003         30.2
2004         30.2
2005      34150.2
2006       1470.2
2007       1470.2
2008      36925.2
2009       1460.0
2010       1460.0
2011       1460.0
2012          0.0
2013          0.0
2014          0.0
2015          0.0
2016          0.0
2017      18395.0
2018         40.0
2019         40.0
2020      37370.0
2021        380.0
2022        380.0
2023        380.0
2024         60.0
2025      19310.0
2026        480.1
2027        480.1
2028        795.1
2029          1.0
2030          1.0
2031          1.0
2032          0.0
2033          0.0
2034          0.0
2035          0.0
2036          0.0
2037          0.0
2038          0.0
2039          0.0
2040          0.0
2041          0.0
2042          0.0
2043          0.0
2044          0.0
2045          0.0
```

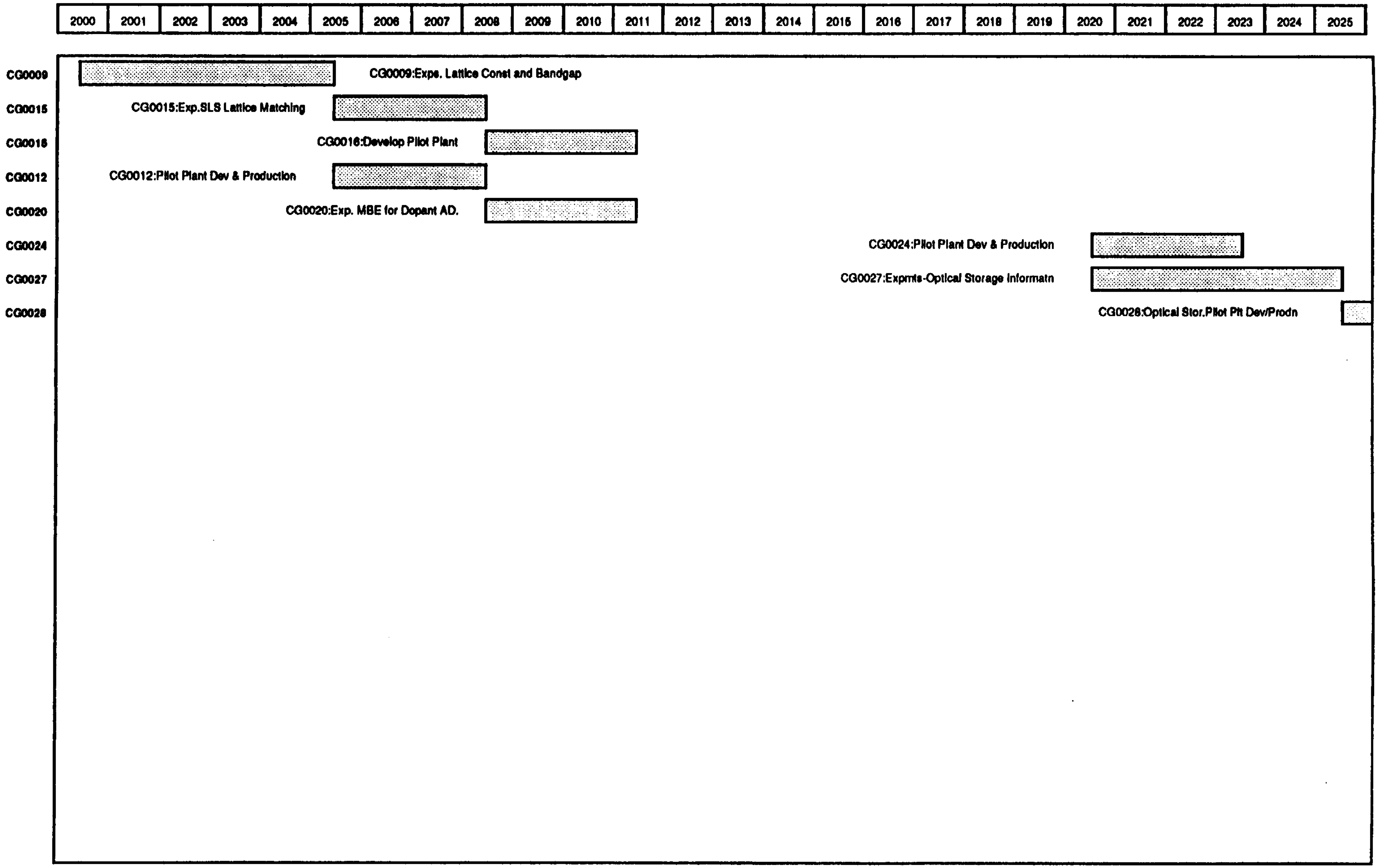
```
2046          0.0
2047          0.0
2048          0.0
2049          0.0
2050          0.0
```

Appendix D

Scenario Activities

Scenario: P02
Report printed 05/10/1990 15:55

Start: Jan 01 2000
End: Dec 31 2025

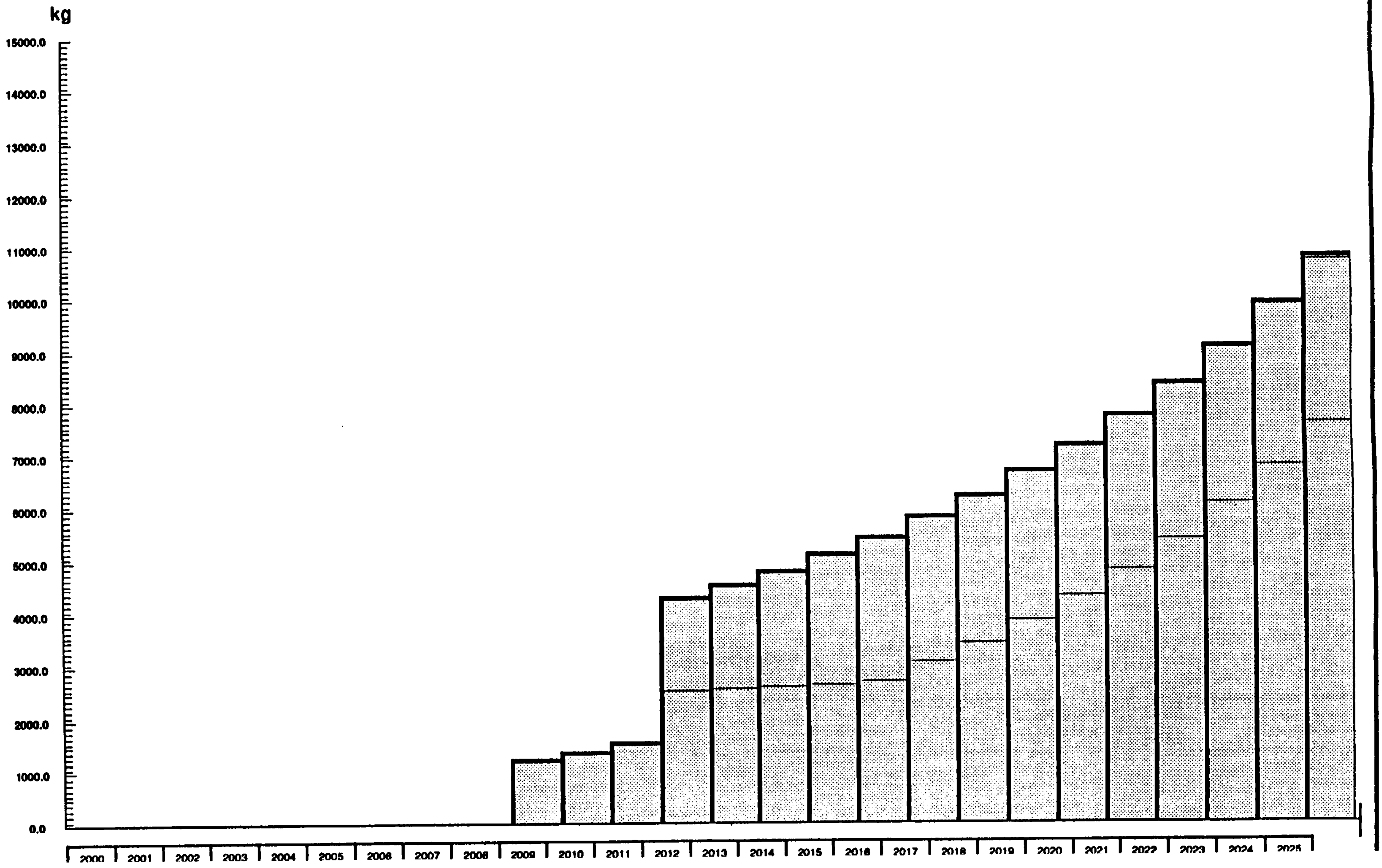


Platform Resources

Scenario: P12
Report printed 05/10/1990 15:59

Start: Jan 01 2000
End: Dec 31 2025

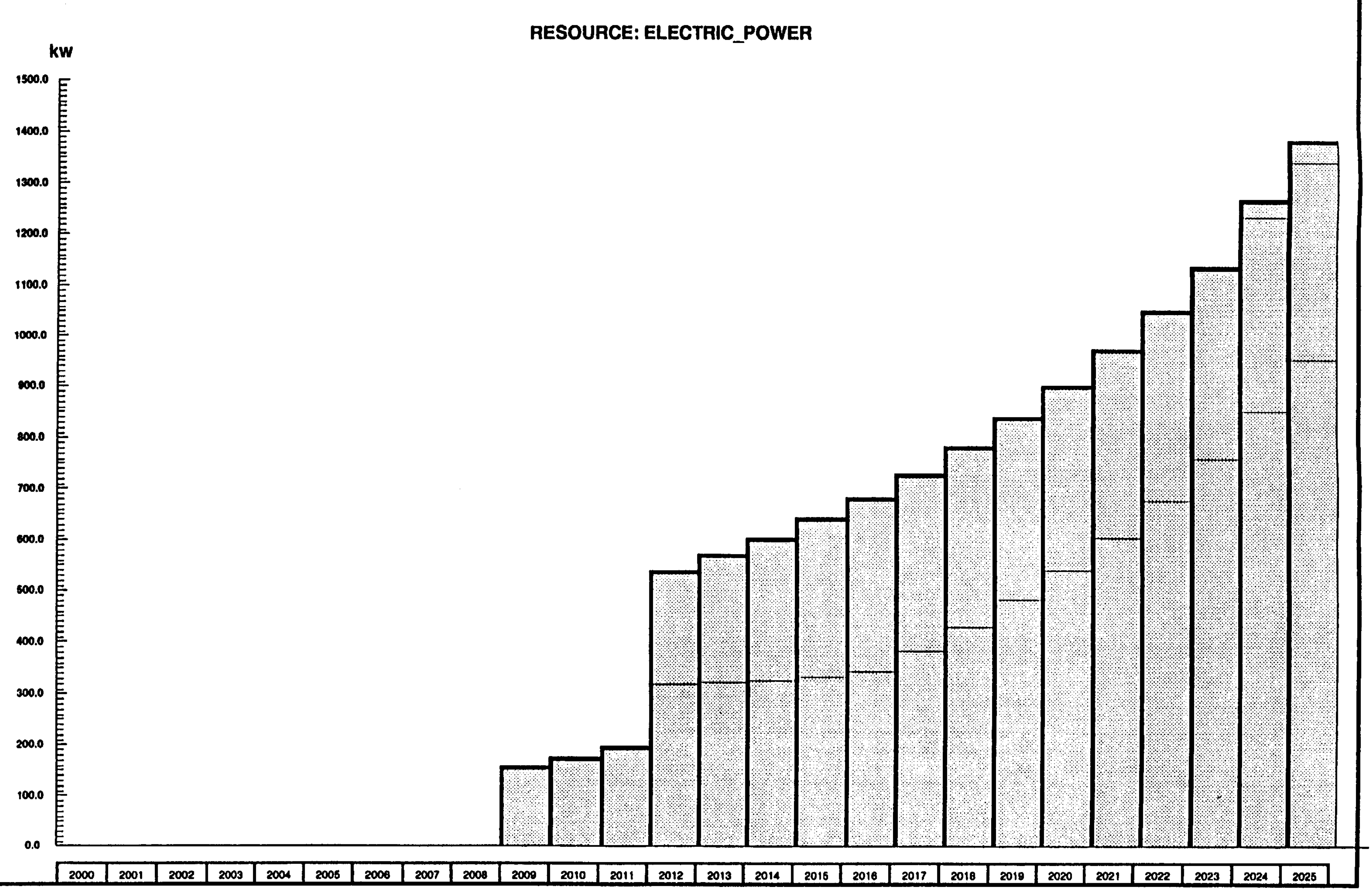
RESOURCE: TONNAGE



Platform Resources

Scenario: P12
Report printed 05/10/1990 15:58

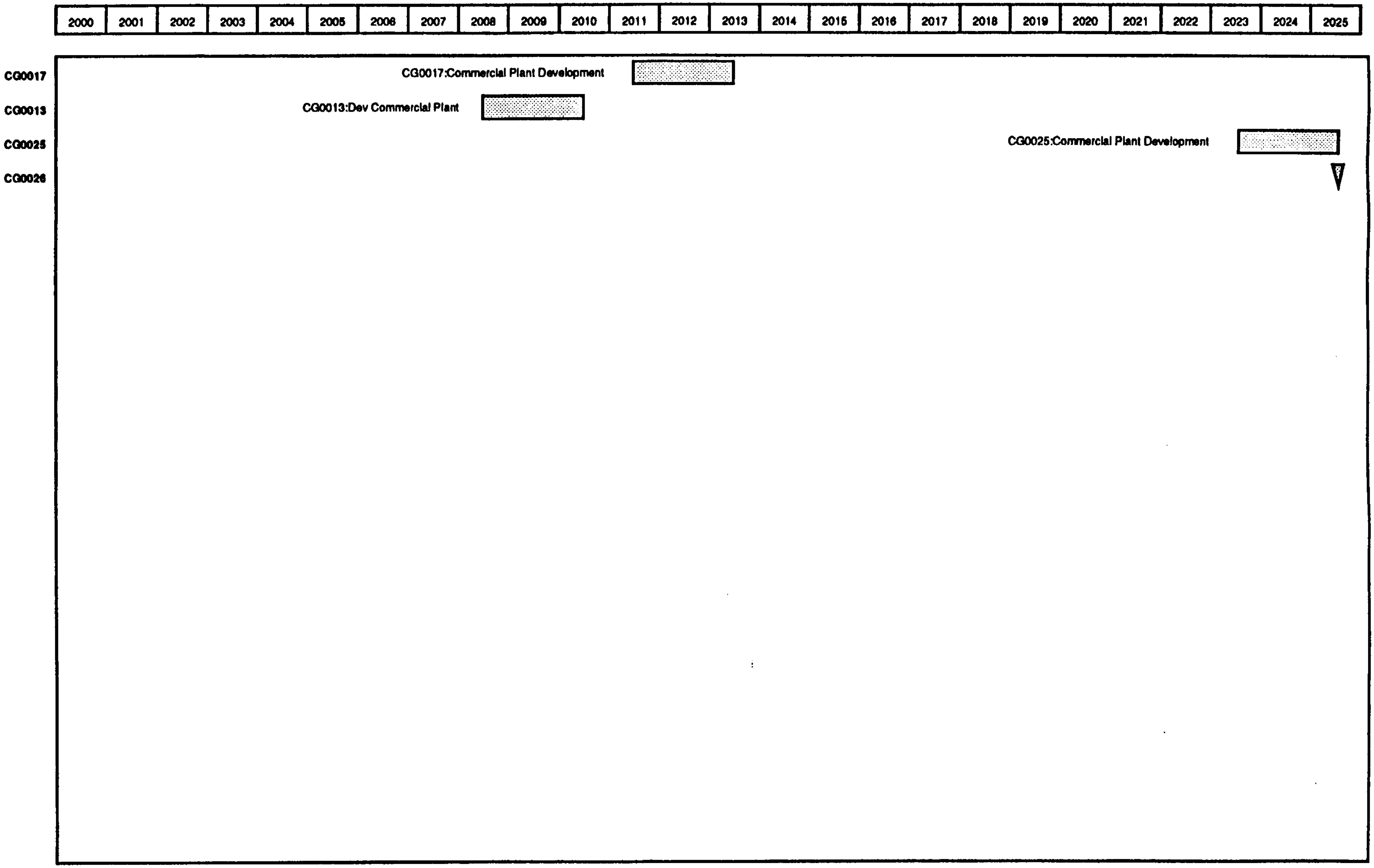
Start: Jan 01 2000
End: Dec 31 2025



Scenario Activities

Scenario: P12
Report printed 05/10/1990 15:57

Start: Jan 01 2000
End: Dec 31 2025

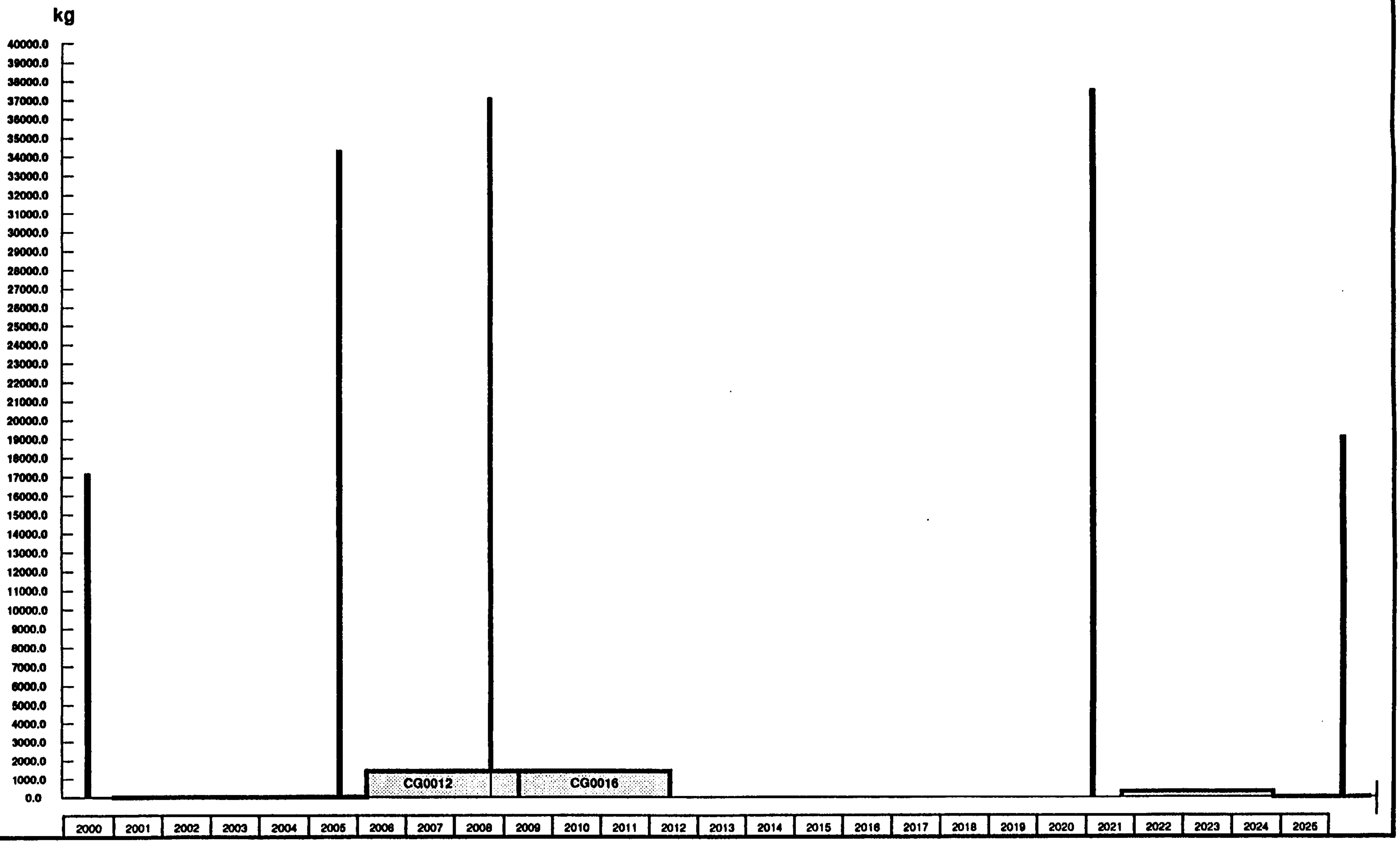


Platform Resources

Scenario: P02
Report printed 05/10/1990 15:58

Start: Jan 01 2000
End: Dec 31 2025

RESOURCE: TONNAGE

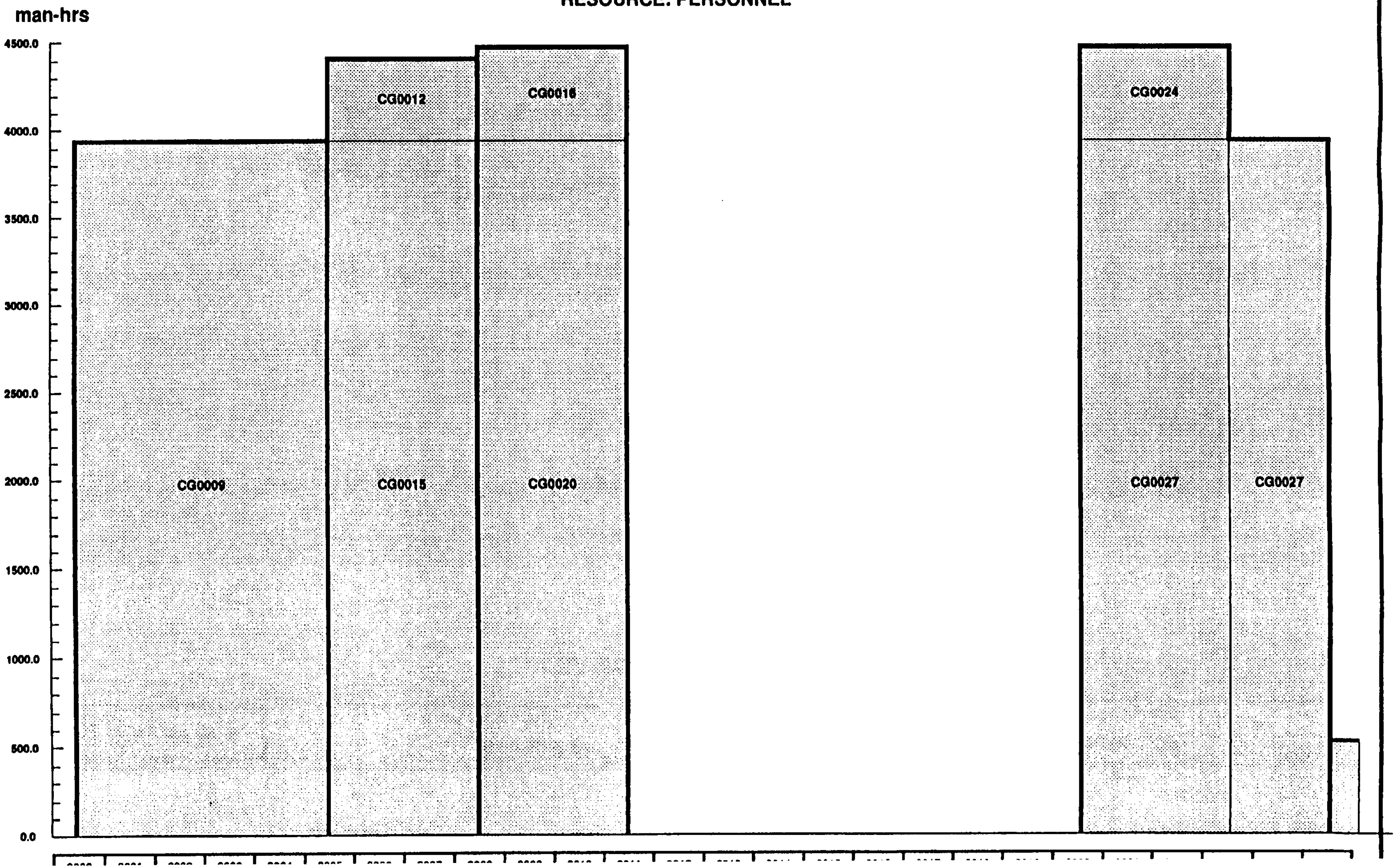


Platform Resources

Scenario: P02
Report printed 05/10/1990 15:58

Start: Jan 01 2000
End: Dec 31 2025

RESOURCE: PERSONNEL

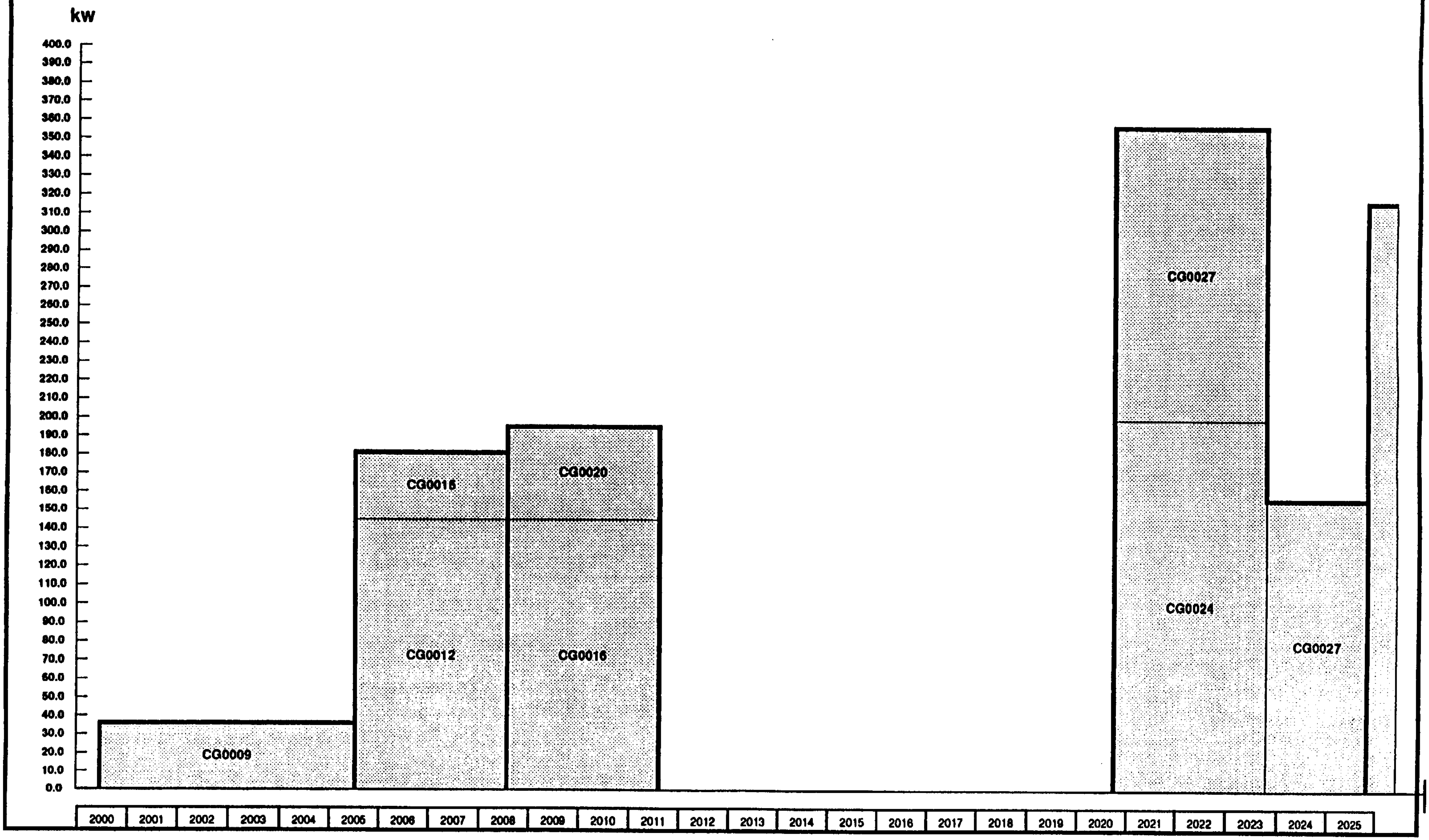


Platform Resources

Scenario: P02
Report printed 05/10/1990 15:58

Start: Jan 01 2000
End: Dec 31 2025

RESOURCE: ELECTRIC_POWER



Appendix E

```
platf  activity   duty     power     imass      rmass      pers   start date   end date   duration
                            kWe        kg         kg     man-hrs                              days

 P01    CG0005    1.00     10.00      60.00    1440.00    530.00  06/27/1995  06/22/1998    1091
 P02    CG0009    0.20    181.30   17060.00      30.20   3936.00  06/20/2000  06/13/2005    1819
 P02    CG0015    0.20    181.30   17060.00      30.20   3936.00  06/14/2005  06/09/2008    1091
 P02    CG0012    0.80    181.30   17060.00    1440.00    461.00  06/14/2005  06/09/2008    1091
 P02    CG0020    0.20    249.30   18395.00      20.00   3936.00  06/10/2008  06/06/2011    1091
 P02    CG0016    0.80    181.30   17060.00    1440.00    530.00  06/10/2008  06/06/2011    1091
 P02    CG0023    0.40    249.30   18395.00      40.00   3936.00  06/05/2017  05/29/2020    1089
 P02    CG0027    0.40    390.30   18935.00      60.00   3936.00  06/01/2020  05/23/2025    1817
 P02    CG0024    0.80    249.30   18395.00     320.00    530.00  06/01/2020  05/26/2023    1089
 P02    CG0028    0.80    395.00   18935.00     480.00    530.00  05/26/2025  05/19/2028    1089
 P02    CG0031    0.20     14.90     315.00       0.10   3936.00  05/26/2025  05/19/2028    1089
 P02    CG0032    0.80     14.90     315.00       1.00    530.00  05/22/2028  05/16/2031    1089
 P09    CG0002    1.00     10.00     560.00    2880.00      0.00  07/03/1992  06/27/1995    1089
 P11    CG0006    0.00      0.00       0.00       0.00      0.00  06/23/1998  06/19/2000     727
 P12    CG0013    0.00      0.00       0.00       0.00      0.00  06/10/2008  06/07/2010     727
 P12    CG0017    0.00      0.00       0.00       0.00      0.00  06/07/2011  06/03/2013     727
 P12    CG0025    0.00      0.00       0.00       0.00      0.00  05/29/2023  05/23/2025     725
 P12    CG0026    0.00      0.00       0.00       0.00      0.00  05/23/2025  05/23/2025       0
 P12    CG0029    0.00      0.00       0.00       0.00      0.00  05/22/2028  05/17/2030     725
 P12    CG0030    0.00      0.00       0.00       0.00      0.00  05/17/2030  05/17/2030       0
 P12    CG0034    0.00      0.00       0.00       0.00      0.00  05/13/2033  05/13/2033       0
 P99    CG0001    0.00      0.00       0.00       0.00      0.00  07/03/1989  07/03/1992    1096
 P99    CG0004    0.00      0.00       0.00       0.00      0.00  06/27/1995  06/27/1995       0
 P99    CG0003    0.00      0.00       0.00       0.00      0.00  06/27/1995  06/27/1995       0
 P99    CG0007    0.00      0.00       0.00       0.00      0.00  06/19/2000  06/19/2000       0
 P99    CG0008    0.00      0.00       0.00       0.00      0.00  06/19/2000  06/19/2000       0
 P99    CG0011    0.00      0.00       0.00       0.00      0.00  06/13/2005  06/13/2005       0
 P99    CG0010    0.00      0.00       0.00       0.00      0.00  06/13/2005  06/13/2005       0
 P99    CG0019    0.00      0.00       0.00       0.00      0.00  06/09/2008  06/09/2008       0
 P99    CG0014    0.00      0.00       0.00       0.00      0.00  06/07/2010  06/07/2010       0
 P99    CG0021    0.00      0.00       0.00       0.00      0.00  06/06/2011  06/06/2011       0
 P99    CG0018    0.00      0.00       0.00       0.00      0.00  06/03/2013  06/03/2013       0
 P99    CG0022    0.00      0.00       0.00       0.00      0.00  06/05/2017  06/05/2017       0
 P99    CG0033    0.00      0.00       0.00       0.00      0.00  05/19/2031  05/13/2033     725
```

```
activity  platf   duty      power      imass       rmass      pers   duration    description
                              kWe        kg          kg     man-hrs   days

 CG0001    P99    0.00       0.00       0.00        0.00       0.00   1096   CG0001:STS Crystal Growth Experiments
 CG0002    P09    1.00      10.00     560.00     2880.00       0.00   1089   CG0002:FF Crystal Growth Experiments
 CG0003    P99    0.00       0.00       0.00        0.00       0.00      0   CG0003:Decision Growth Method
 CG0004    P99    0.00       0.00       0.00        0.00       0.00      0   CG0004:Space Station IOC
 CG0005    P01    1.00      10.00      60.00     1440.00     530.00   1091   CG0005:Pilot Plant Dev & Production
 CG0006    P11    0.00       0.00       0.00        0.00       0.00    727   CG0006:Commercial Plant Development
 CG0007    P99    0.00       0.00       0.00        0.00       0.00      0   CG0007:Commercial Microchip Production
 CG0008    P99    0.00       0.00       0.00        0.00       0.00      0   CG0008:M/S*FL1* Link from SPACESCI
 CG0009    P02    0.20     181.30   17060.00       30.20    3936.00   1819   CG0009:Exps. Lattice Const and Bandgap
 CG0010    P99    0.00       0.00       0.00        0.00       0.00      0   CG0010:M/S*CG2* Link to GLASSES,MPOI
 CG0011    P99    0.00       0.00       0.00        0.00       0.00      0   CG0011:Dec. Photochip Production
 CG0012    P02    0.80     181.30   17060.00     1440.00     461.00   1091   CG0012:Pilot Plant Dev & Production
 CG0013    P12    0.00       0.00       0.00        0.00       0.00    727   CG0013:Dev Commercial Plant
 CG0014    P99    0.00       0.00       0.00        0.00       0.00      0   CG0014:Commercial Photochip
 CG0015    P02    0.20     181.30   17060.00       30.20    3936.00   1091   CG0015:Exp.SLS Lattice Matching
 CG0016    P02    0.80     181.30   17060.00     1440.00     530.00   1091   CG0016:Develop Pilot Plant
 CG0017    P12    0.00       0.00       0.00        0.00       0.00    727   CG0017:Commercial Plant Development
 CG0018    P99    0.00       0.00       0.00        0.00       0.00      0   CG0018:Commercial Tandem Cells
 CG0019    P99    0.00       0.00       0.00        0.00       0.00      0   CG0019:M/S*CD3* Link from COATINGS
 CG0020    P02    0.20     249.30   18395.00       20.00    3936.00   1091   CG0020:Exp. MBE for Dopant AD.
 CG0021    P99    0.00       0.00       0.00        0.00       0.00      0   CG0021:M/S*CG1* Link to GLASSES,METALLIC
 CG0022    P99    0.00       0.00       0.00        0.00       0.00      0   CG0022:M/S*MMD3* Link from METALLIC
 CG0023    P02    0.40     249.30   18395.00       40.00    3936.00   1089   CG0023:Opt.Filter Proof-of-Concept
 CG0024    P02    0.80     249.30   18395.00      320.00     530.00   1089   CG0024:Pilot Plant Dev & Production
 CG0025    P12    0.00       0.00       0.00        0.00       0.00    725   CG0025:Commercial Plant Development
 CG0026    P12    0.00       0.00       0.00        0.00       0.00      0   CG0026:Commercial Optical Filters
 CG0027    P02    0.40     390.30   18935.00       60.00    3936.00   1817   CG0027:Expmts-Optical Storage Informatn
 CG0028    P02    0.80     395.00   18935.00      480.00     530.00   1089   CG0028:Optical Stor.Pilot Plt Dev/Prodn
 CG0029    P12    0.00       0.00       0.00        0.00       0.00    725   CG0029:Comm'l Optical Stor.Plant Devlpmt
 CG0030    P12    0.00       0.00       0.00        0.00       0.00      0   CG0030:Begin Comm'l Prodn Opt.StrgDevice
 CG0031    P02    0.20      14.90     315.00        0.10    3936.00   1089   CG0031:Expmts,Pseudo-Cryst,Prec.OptMsrmt
 CG0032    P02    0.80      14.90     315.00        1.00     530.00   1089   CG0032:Pseudo-Crystl Pilot Plt Dev/Prodn
 CG0033    P99    0.00       0.00       0.00        0.00       0.00    725   CG0033:Comm'l Pseudo-Crystl Plant Prodn
 CG0034    P12    0.00       0.00       0.00        0.00       0.00      0   CG0034:Begin Prodn PseudoCrystl MsmrtDev
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Appendix F

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CG0001  NA  P99   0.00     0.0     0      0.0     0.0  07/03/1989  07/03/1992  1096  CG0001:STS Crystal Growth Experiments
CG0002  RD  P09   1.00    10.0     0    560.0  2880.0  06/29/1992  06/23/1995  1089  CG0002:FF Crystal Growth Experiments
CG0003  XX  P99   0.00     0.0     0      0.0     0.0  06/23/1995  06/23/1995     0  CG0003:Decision Growth Method
CG0004  XX  P99   0.00     0.0     0      0.0     0.0  06/27/1995  06/27/1995     0  CG0004:Space Station IOC
CG0005  RD  P01   1.00    10.0   530     60.0  1440.0  06/27/1995  06/22/1998  1091  CG0005:Pilot Plant Dev & Production
CG0006  MF  P11   0.00     0.0     0      0.0     0.0  06/23/1998  06/19/2000   727  CG0006:Commercial Plant Development
CG0007  XX  P99   0.00     0.0     0      0.0     0.0  06/19/2000  06/19/2000     0  CG0007:Commercial Microchip Production
CG0008  XX  P99   0.00     0.0     0      0.0     0.0  06/19/2000  06/19/2000     0  CG0008:M/S*FL1* Link from SPACESCI
CG0009  MS  P02   0.20   181.3  3936  17060.0    30.2  06/20/2000  06/13/2005  1819  CG0009:Exps. Lattice Const and Bandgap
CG0010  XX  P99   0.00     0.0     0      0.0     0.0  06/13/2005  06/13/2005     0  CG0010:M/S*CG2* Link to GLASSES,MPOI
CG0011  XX  P99   0.00     0.0     0      0.0     0.0  06/13/2005  06/13/2005     0  CG0011:Dec. Photochip Production
CG0012  RD  P02   0.80   181.3   461  17060.0  1440.0  06/14/2005  06/09/2008  1091  CG0012:Pilot Plant Dev & Production
CG0013  MF  P12   0.00     0.0     0      0.0     0.0  06/10/2008  06/07/2010   727  CG0013:Dev Commercial Plant
CG0014  XX  P99   0.00     0.0     0      0.0     0.0  06/07/2010  06/07/2010     0  CG0014:Commercial Photochip
CG0015  MS  P02   0.20   181.3  3936  17060.0    30.2  06/14/2005  06/09/2008  1091  CG0015:Exp.SLS Lattice Matching
CG0016  RD  P02   0.80   181.3   530  17060.0  1440.0  06/10/2008  06/06/2011  1091  CG0016:Develop Pilot Plant
CG0017  MF  P12   0.00     0.0     0      0.0     0.0  06/07/2011  06/03/2013   727  CG0017:Commercial Plant Development
CG0018  XX  P99   0.00     0.0     0      0.0     0.0  06/03/2013  06/03/2013     0  CG0018:Commercial Tandem Cells
CG0019  XX  P99   0.00     0.0     0      0.0     0.0  06/09/2008  06/09/2008     0  CG0019:M/S*CD3* Link from COATINGS
CG0020  MS  P02   0.20   249.3  3936  18395.0    20.0  06/10/2008  06/06/2011  1091  CG0020:Exp. MBE for Dopant AD.
CG0021  XX  P99   0.00     0.0     0      0.0     0.0  06/06/2011  06/06/2011     0  CG0021:M/S*CG1* Link to GLASSES,METALLIC
CG0022  XX  P99   0.00     0.0     0      0.0     0.0  06/05/2017  06/05/2017     0  CG0022:M/S*MMD3* Link from METALLIC
CG0023  MS  P02   0.40   249.3  3936  18395.0    40.0  06/05/2017  05/29/2020  1089  CG0023:Opt.Filter Proof-of-Concept
CG0024  RD  P02   0.80   249.3   530  18395.0   320.0  06/01/2020  05/26/2023  1089  CG0024:Pilot Plant Dev & Production
CG0025  MF  P12   0.00     0.0     0      0.0     0.0  05/29/2023  05/23/2025   725  CG0025:Commercial Plant Development
CG0026  XX  P12   0.00     0.0     0      0.0     0.0  05/23/2025  05/23/2025     0  CG0026:Commercial Optical Filters
CG0027  MS  P02   0.40   390.3  3936  18935.0    60.0  06/01/2020  05/23/2025  1817  CG0027:Expmts-Optical Storage Informatn
CG0028  RD  P02   0.80   395.0   530  18935.0   480.0  05/26/2025  05/19/2028  1089  CG0028:Optical Stor.Pilot Plt Dev/Prodn
CG0029  MF  P12   0.00     0.0     0      0.0     0.0  05/22/2028  05/17/2030   725  CG0029:Comm'l Optical Stor.Plant Devlpmt
CG0030  XX  P12   0.00     0.0     0      0.0     0.0  05/17/2030  05/17/2030     0  CG0030:Begin Comm'l Prodn Opt.StrgDevice
CG0031  MS  P02   0.20    14.9  3936    315.0     0.1  05/26/2025  05/19/2028  1089  CG0031:Expmts,Pseudo-Cryst,Prec.OptMsrmt
CG0032  RD  P02   0.80    14.9   530    315.0     1.0  05/22/2028  05/16/2031  1089  CG0032:Pseudo-Crystl Pilot Plt Dev/Prodn
CG0033  MF  P99   0.00     0.0     0      0.0     0.0  05/19/2031  05/13/2033   725  CG0033:Comm'l Pseudo-Crystl Plant Prodn
CG0034  XX  P12   0.00     0.0     0      0.0     0.0  05/13/2033  05/13/2033     0  CG0034:Begin Prodn PseudoCrystl MsmrtDev
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  1 CG0001  0 4  36.0 100.4 07/03/1989 ASAP     02/27/1987 02/23/1990 07/03/1995 06/26/1998 -99
  2 CG0002  0 4  36.0  72.2 ASAP       ASAP     06/29/1992 06/23/1995 06/29/1998 06/22/2001   1
  3 CG0003  0 4   0.0  72.2 ASAP       ASAP     06/23/1995 06/23/1995 06/25/2001 06/25/2001   2
  4 CG0004  0 4   0.0  72.2 06/27/1995 ASAP     06/23/1995 06/23/1995 06/25/2001 06/25/2001   2
  5 CG0005  0 4  36.0  72.1 ASAP       ASAP     06/27/1995 06/22/1998 06/25/2001 06/18/2004   3
  5 CG0005  0 4  36.0  72.1 ASAP       ASAP     06/27/1995 06/22/1998 06/25/2001 06/18/2004   4
  6 CG0006  0 4  24.0  72.1 ASAP       ASAP     06/23/1998 06/19/2000 06/21/2004 06/16/2006   5
  7 CG0007  0 4   0.0 396.1 ASAP       ASAP     06/19/2000 06/19/2000 05/13/2033 05/13/2033   6
  8 CG0008  0 4   0.0  72.1 ASAP       ASAP     06/19/2000 06/19/2000 06/19/2006 06/19/2006   6
  9 CG0009  0 4  60.0  72.1 ASAP       ASAP     06/20/2000 06/13/2005 06/19/2006 06/10/2011   6
  9 CG0009  0 4  60.0  72.1 ASAP       ASAP     06/20/2000 06/13/2005 06/19/2006 06/10/2011   8
 10 CG0010  0 4   0.0 336.1 ASAP       ASAP     06/13/2005 06/13/2005 05/13/2033 05/13/2033   9
 11 CG0011  0 4   0.0 276.1 ASAP       ASAP     06/13/2005 06/13/2005 05/22/2028 05/22/2028   9
 12 CG0012  0 4  36.0 276.1 ASAP       ASAP     06/14/2005 06/09/2008 05/22/2028 05/16/2031   9
 12 CG0012  0 4  36.0 276.1 ASAP       ASAP     06/14/2005 06/09/2008 05/22/2028 05/16/2031  11
 13 CG0013  0 4  24.0 276.1 ASAP       ASAP     06/10/2008 06/07/2010 05/19/2031 05/13/2033  12
 14 CG0014  0 4   0.0 276.1 ASAP       ASAP     06/07/2010 06/07/2010 05/13/2033 05/13/2033  13
 15 CG0015  0 4  36.0  72.1 ASAP       ASAP     06/14/2005 06/09/2008 06/13/2011 06/06/2014   9
 16 CG0016  0 4  36.0 240.1 ASAP       ASAP     06/10/2008 06/06/2011 05/22/2028 05/16/2031  15
 17 CG0017  0 4  24.0 240.1 ASAP       ASAP     06/07/2011 06/03/2013 05/19/2031 05/13/2033  16
 18 CG0018  0 4   0.0 240.1 ASAP       ASAP     06/03/2013 06/03/2013 05/13/2033 05/13/2033  17
 19 CG0019  0 4   0.0  72.1 ASAP       ASAP     06/09/2008 06/09/2008 06/09/2014 06/09/2014  15
 20 CG0020  0 4  36.0  72.1 ASAP       ASAP     06/10/2008 06/06/2011 06/09/2014 06/02/2017  15
 20 CG0020  0 4  36.0  72.1 ASAP       ASAP     06/10/2008 06/06/2011 06/09/2014 06/02/2017  19
 21 CG0021  0 4   0.0 264.1 ASAP       ASAP     06/06/2011 06/06/2011 05/13/2033 05/13/2033  20
 22 CG0022  0 4   0.0  72.1 06/05/2017 ASAP     06/06/2011 06/06/2011 06/05/2017 06/05/2017  20
 23 CG0023  1 4  36.0   0.0 ASAP       ASAP     06/05/2017 05/29/2020 06/05/2017 05/29/2020  20
 23 CG0023  1 4  36.0   0.0 ASAP       ASAP     06/05/2017 05/29/2020 06/05/2017 05/29/2020  22
 24 CG0024  0 4  36.0  96.0 ASAP       ASAP     06/01/2020 05/26/2023 05/22/2028 05/16/2031  23
 25 CG0025  0 4  24.0  96.0 ASAP       ASAP     05/29/2023 05/23/2025 05/19/2031 05/13/2033  24
 26 CG0026  0 4   0.0  96.0 ASAP       ASAP     05/23/2025 05/23/2025 05/13/2033 05/13/2033  25
 27 CG0027  1 4  60.0   0.0 ASAP       ASAP     06/01/2020 05/23/2025 06/01/2020 05/23/2025  23
 28 CG0028  0 4  36.0  36.0 ASAP       ASAP     05/26/2025 05/19/2028 05/22/2028 05/16/2031  27
 29 CG0029  0 4  24.0  36.0 ASAP       ASAP     05/22/2028 05/17/2030 05/19/2031 05/13/2033  28
 30 CG0030  0 4   0.0  36.0 ASAP       ASAP     05/17/2030 05/17/2030 05/13/2033 05/13/2033  29
 31 CG0031  1 4  36.0   0.0 ASAP       ASAP     05/26/2025 05/19/2028 05/26/2025 05/19/2028  27
 32 CG0032  1 4  36.0   0.0 ASAP       ASAP     05/22/2028 05/16/2031 05/22/2028 05/16/2031  31
 33 CG0033  1 4  24.0   0.0 ASAP       ASAP     05/19/2031 05/13/2033 05/19/2031 05/13/2033  32
 34 CG0034  0 4   0.0   0.0 ASAP       ASAP     05/13/2033 05/13/2033 05/13/2033 05/13/2033  33
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