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FEASIBILITY OF ISOTOPIC POWER FOR MANNED LUNAR MISSIONS

VOLUME 1--SUMMARY

May 1964

MND-3296-1

SPECIAL REREVIEW FINAL DETERMINATION	Reviewers	Class.	Date
	DBW	U	12/13/80
	KAW	U	05/13/82
Class: <u>U</u>			

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iii

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FOREWORD

The results of a study on the application of radioisotopic power to the Apollo mission are provided in this series of reports. The study was performed for the U. S. Atomic Energy Commission by the Nuclear Division of the Martin Company under Contract AT(30-1)-3296. The work was sponsored by the Division of Reactor Development, Isotopic Power Branch in keeping with its responsibility for the development of isotopic power systems for space missions of established or possible interest to the National Aeronautics and Space Administration (NASA) and the Department of Defense (DOD).

The complete results of the study are presented in eight volumes, as follows:

Volume 1--Summary

Volume 2--Mission Requirements, Fuel Availability, and
Nuclear Safety and Radiation

Volume 3--Thermoelectric System

Volume 4--Thermionic System

Volume 5--Brayton Cycle System

Volume 6--Stirling Cycle System

Volume 7--Dowtherm-A Rankine System

Volume 8--Mercury Rankine System

The assistance of the following companies who are recognized leaders in their respective fields and who generously provided state-of-the-art information is gratefully acknowledged; Brayton Cycle--AiResearch Manufacturing Company of the Garrett Corporation; Dowtherm-A Rankine--Sundstrand Corporation; Mercury Rankine--Thompson Ramo-Wooldridge Corporation; and Stirling--Allison Division of the General Motors Corporation.

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CONTENTS

	Page
Legal Notice	ii
Foreword	iii
Contents	v
Summary	vii
I. Introduction	1
II. Advanced Mission Requirements	3
III. Summary of Apollo Mission Requirements	5
IV. Fuel Availability	7
V. Safety	9
A. Nuclear Safety Philosophy	9
B. Prelaunch Phase Nuclear Safety	9
C. Launch Phase Nuclear Safety	10
D. Re-entry Phase Nuclear Safety	10
VI. Conversion System Selection and Comparison	13
A. Selection Criteria	13
B. State of the Art and Development Problems	17
C. System Selection	36
VII. Development Program for Thermoelectric System	41
A. Phase I--Design and Component Development.	41
B. Phase II--Prototype Manufacture and Component Qualification	41
C. Phase III--Unmanned Flight Operational Systems Manufacture and Flight Rating Tests.	44

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CONTENTS (continued)

	Page
D. Phase IV--Manned Flight Operational Systems Manufacture and Flight Rating Tests	44
VIII. Costs	45
References	49

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SUMMARY

CONCLUSIONS

The results of this study have shown conclusively that the application of radioisotopic power to the Apollo mission is entirely feasible and that a radioisotope power system (RIPS) would weigh less than the current fuel cell system; that nuclear safety can be achieved and radiation exposure to the crew maintained well within acceptable limits; that the RIPS can be developed within the time limits imposed for a very reasonable cost; and, that the power system reliability specification not only can be met but can be exceeded.

It is further concluded that among the energy conversion devices considered, the thermoelectric system is the most nearly representative of current state of the art and therefore, development risks are minimal. Furthermore, this static energy converter has the reliability necessary for manned missions beyond earth orbit. The growth potential of the thermoelectric system for application at higher power levels is limited by its relatively low efficiency and consequent burden on isotopic fuel supplies.

To reduce the amount of fuel required for higher powered isotopic systems--three to five kilowatts(e)--higher efficiency systems are required. Potentially, higher efficiency can be achieved with a number of dynamic systems. Each of these potential systems possesses significant problem areas which must first be resolved before individual system feasibility can be established. The choice of a single, high efficiency system for further development depends on the technical assessment of the probability that the basic problems associated with that system will be solved. Further, should these basic problems be solved, the resultant system must meet the intended mission requirements. Among the most important of these mission requirements is that of system reliability extending over mission durations of a few days to years.

Unfortunately, there is no simple formula by which the technical assessments mentioned can be made. We can evaluate the history of the technology behind each system and attempt, using our best technical judgment, to determine the probability that the problems will be solved and that mission requirements will be met.

Inherent problems in dynamic systems include bearing, heat transfer and dynamic stability problems. For some systems, these problems are associated with the high temperatures at which they must operate to achieve attractive performance. Other systems involve boiling, condensing and/or lubrication problems in a zero g environment.

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Among the dynamic systems considered, two stand out as offering highly desirable features. These are the Brayton cycle system and the Dowtherm-A Rankine system. Each of these systems requires the early solution of certain development problems before a logical technical judgment can be made between them. It is our considered judgment that it would be presumptuous at this time to select between these two systems based on the facts at hand.

Based upon a careful evaluation of the facts which are summarized in the volumes associated with this report, the Martin Company makes the following recommendations.

- (1) Initiate a full-scale development program to provide the complete thermoelectric RIPS for Apollo as soon as possible.
- (2) Initiate a development program on the Brayton system to determine the feasibility of achieving reliable gas bearings at system operating temperatures, satisfactory turbine inlet temperatures and adequate component efficiencies.
- (3) Initiate a development program on the Dowtherm-A Rankine system specifically directed toward component development fabrication and test and a zero g demonstration of boiler/condenser performance.
- (4) Initiate a fuel capsule development and safety program oriented to cover the range of fuel temperatures required by the direct conversion and dynamic systems of Items (1), (2) and (3).

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I. INTRODUCTION

The growth in scope of the national space program is dependent upon many factors, among which is the requirement for dependable power during the mission lifetime. This need is self-evident and particularly demanding for systems which include men. The recent and successful use of isotopic power in space has led to significantly greater scientific and public acceptance of the use of this technology. Further, it has provided a sound basis for review of all missions to which isotopic power can be meaningfully applied.

Two unique characteristics of this technology are: the ability to predict the energy of the thermal source, as a function of time, with a reliability of 1.0; and the virtual independence of the power supply to system orientation. These factors immediately suggest its potential application to manned missions.

Recognizing this potential, the U. S. Atomic Energy Commission requested the Martin Company, under Contract AT(30-1)-3296, to examine the applicability of RIPS to the Apollo mission.

This mission is paramount to our national space effort and is representative of the integration problems associated with the complex manned missions. These factors, coupled with the recent Department of Defense decision to undertake the MOL (Manned Orbiting Laboratory) to which this technology could well be applied, makes this examination technically compelling and very timely.

This report includes a discussion of the Apollo system and the major integration problems associated with incorporating a radioisotope power supply. Fuel, fuel availability and fuel form selection are discussed and the optimum fuel for this mission established. The safety criteria and constraints imposed on system design are analyzed. A detailed appraisal of conversion systems including Stirling, Brayton, Rankine (Mercury and Dowtherm-A) thermoelectric and thermionic is performed and the unique development problems associated with each system are analyzed. Selection criteria are established and applied to the several systems in a comparative evaluation. An affirmative assessment of the feasibility of the application of isotopic power to the Apollo mission is reached and a recommended development program for the heat source and conversion systems is presented.

This entire program has been conducted on an extremely critical and objective basis. Major advancements required over current technology to bring a given system to flight operational status have been considered as grounds for elimination.

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II. ADVANCED MISSION REQUIREMENTS

One of the conversion system selection criteria was the application of a given system to other missions--system versatility. The most pressing of these is the Gemini mission which requires approximately 500 watts for a 14-day life. The 500-watt thermoelectric module (presented in Volume 3) would be directly applicable to this spacecraft within the stringent schedule requirements. A detailed integration study should be performed to examine the RIPS-Gemini interfaces. However, such a study was beyond the scope of this program.

Aside from the requirements of the established Gemini and Apollo missions, the need to initiate the early development of long-lived, reliable isotope power systems in the 0.5- to 5.0-kilowatt range is very clear. In this sector of the power spectrum, no other nuclear system can compete in weight or volume, and no nonnuclear system offers the life and reliability inherent in a power system energized with an isotope heat source.

The man-in-space program beyond Apollo, though lacking the development incentive of a funded national objective, is rapidly being defined through extensive study programs. Of immediate interest for the 1965 to 1970 time period are the several concepts now envisioned for the establishment of a small earth orbiting space station. Among the concepts receiving most attention are the recently announced Gemini X or MOL that will carry two men for a period up to six weeks; the extended Apollo mission accommodating three men up to a year; the Manned Orbiting Research Laboratory which would house four men for a year or more; and the U. S. Air Force Orbital Space Station for which definition study contracts have recently been awarded. While it is premature to attempt to predict the ultimate concept or concepts that will be launched, the need for a reliable power supply within the range under consideration here is a factor common to all. Studies* have recognized that about one kilowatt per man, more or less, depending upon the degree of closure of the ecological cycle, is required for minimum life support in space and that the total average power required for these missions is probably less than five kilowatts. Peak power as high as 10 kilowatts for short duty cycles may be needed to service communications and scientific experiments which are still in the formative stage. A rechargeable battery system could store energy and supply these peaks. Space station designers have considered isotope power systems for this application but to date have not specifically requested them, primarily due to lack of high powered system demonstration and to the erroneous belief that suitable fuels are unavailable. Volume 2 of this report discusses fuel availability in detail.

*Manned Space Flight Symposium on Manned Interplanetary Mission, NASA, Huntsville, 1963.

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Increased space station capability is foreseen in the large rotating earth orbiting station accommodating as many as 40 men for periods up to five years. Obviously, a five-kilowatt system could not provide the base power for such a station. However, the need for a highly reliable emergency power source points directly to the desirability of having an off the shelf and proven isotope power supply in the 1970s. The concept of a ferry vehicle transporting as many as six men to and from the station and remaining on station for as long as six months as a lifeboat suggests still another need within the spectrum of power advanced for the isotope systems.

EMPIRE (Early Manned Planetary Interplanetary Roundtrip Expedition) and other similar studies have shown that the manned Mars mission might be accomplished with minimum average power as low as 10 kilowatts. The duration of the mission could vary from about 360 to 880 days, depending upon the launch conditions and the projected stay time at Mars. The Mars Excursion Module (MEM), not unlike the Lunar Excursion Module (LEM), might descend upon the Martian surface for a period of about 45 days. The two- to three-kilowatt isotope system would appear to be the ideal power supply for MEM.

The initial concept of a lunar base with two men for 90 days and subsequent modular growth also suggests the applicability of the isotope system. Expansion of the colony would eventually outstrip the isotope capability; still, the need for auxiliary power would continue.

The scope of unmanned mission programs under study is replete with requirements for power in the range under consideration here. Active communication satellites, weather satellites, orbiting observatories, deep space probes and potential military applications all seek long-lived, reliable power. Of particular interest among these are the Voyager interplanetary spacecraft requiring power on the order of 2.5 kilowatts; the Orbiting Astronomical Observatory requiring power up to one kilowatt; the Advanced Solar Probe requiring up to 750 watts; and unspecified communications satellites needing in the neighborhood of three kilowatts for direct communications with airborne stations.

Although it is beyond the scope of this study to analyze RIPS application to specific space missions other than Apollo, the trend in requirements is clear and points to a significant void in the power supply spectrum. The decision to develop isotope systems to fill this void in time to enable designers to take full advantage of them is definitely warranted. The availability of lightweight, compact, highly reliable isotope systems will promote their application and substantially enhance the national space capability.

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III. SUMMARY OF APOLLO MISSION REQUIREMENTS

A detailed discussion of the Apollo mission requirements is presented in Volume 2 of this report. The most significant requirements imposing constraints on the power supply are summarized here:

1. Power supply life requirements (days)

Mission duration	14
Launch pad hold	30
Total	44

2. Launch azimuth from Cape Kennedy

Window from 72 to 108 degrees

3. Parking orbit--115-miles circular

4. Power required (watts)

Average power over mission	1200
Power required for crew safety	900
Peak powers (short duration)	Up to 4000
Voltage	27 to 31 volts dc at bus

5. Temperature

+30° - +140° F

6. Vibrations

Constant at 9 g from 400 to 2000 cps (see detailed specification)

7. Shock

30 g for 11-millisecond rise

8. Reliability

Mission success (assuming 14 days at 1200 watts)	0.9986
Crew safety (assuming 6 days at 900 watts)	0.99997

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9. Maximum allowable radiator surface*

300 square feet

10. Crew radiation dose

Total 54 r per mission

Assumes 24 r from solar and Van
Allen belts; 30 r from nuclear sup-
ply acceptable

11. Equivalent sink temperature

Average 300° R

High noon conditions up to
550° R for approximately 12 minutes
per lunar orbit

12. Schedule

Unmanned flights starting July 1966

Manned flights starting July 1967;
manned lunar flights starting January
1968.

*Criterion determined by NASA for this program.

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IV. FUEL AVAILABILITY

The thermal energy required from the isotopic fuel is dependent on the specific power conversion system selected. Mission requirements at time of launch range from 8 to 28 kilowatts(t). A review of most recent data of the AEC current and planned production of isotopes indicated that polonium-210 is the best isotopic fuel that could be available in sufficient quantities to meet a 1967 to 1968 launch date.

A 30-day period from time of fuel encapsulation (heat source fabrication) to launch and a 14-day mission requires a fuel loading of 72 and 154 grams polonium for the Stirling and thermoelectric systems, respectively. The thermionic, Rankine and Brayton cycles are between these extremes.

Polonium is currently produced at Mound Laboratory, an installation operated for the USAEC. Present production capabilities are 50 grams annually. Additional processing is planned and will increase the capability to 300 to 500 gm/yr during Fiscal 1966 and 1967.

A capacity of approximately 1000 gm/yr would provide a fuel loading for any of the systems studied in a two-month or less production period. Minimum production times are of paramount importance with short half life isotopes and multiple launches.

According to information obtained from the AEC Division of Isotopes Development regarding fuel requirements defined in this study, an annual production capability in excess of 1000 grams could be made available two years after receipt of official request to the AEC. The two-year lead time is based on installing the additional required processing facilities at Savannah River with an investment of \$1.5 million. Resulting estimated production cost on a per curie basis is \$1.13.

Should demands for multiple missions exceed 1000 gm/yr, the AEC could install a capacity approaching 10,000 grams (equivalent to 1400 kilowatts(t) at Hanford. A lead time of three to four years and a facility expenditure of \$4.5 million would be required. The AEC estimate of Hanford fuel costs in Fiscal 1968 and 1969 is \$0.45/curie.

The estimated cost of a fuel loading, based on an annual production of 1000 grams, ranges from \$159 to \$907 thousand, depending on the power conversion cycle, time of launch and location of the processing facility.

In addition to its availability potential, polonium-210 is a most promising fuel for short duration, manned space missions and is able to offer the following desirable features and advantages:

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- (1) High power density and minimum fuel weight.
- (2) Low radiation values resulting in minimum shielding requirements.
- (3) High melting point fuel form (GdPo) with proven container material compatibility.
- (4) Minimum safety problems as related to manned space missions.
- (5) Current R&D programs, sponsored by the AEC, are adding to the state of the art.
- (6) Established production technology.

Volume 2 includes a detailed discussion on criteria for fuel selection, current and planned production of isotopes, fuel properties, power cycle fuel requirements and evaluation and the technology of polonium as related to its properties, production and processing.

Critical examination of the data presented in Volume 2 clearly indicates that Po-210 is the only fuel that can be made in sufficient quantities to satisfy the Apollo mission thermal requirements, schedule, and radiation levels. Strontium-90, Cs-137 and Ce-144 require prohibitive shielding weights. Promethium-147, Pu-238, Cm-242 and Cm-244 will not be available in sufficient quantity to meet the schedule requirements. Polonium-210, on the other hand, meets all of the fuel requirements of the mission.

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V. SAFETY

A nuclear safety study of the Apollo manned lunar spacecraft, utilizing a nuclear powered electrical system, is summarized in this chapter.

The objectives of the study were to develop a nuclear safety philosophy, establish basic nuclear safety criteria and determine design requirements imposed by the general philosophy and specific criteria established for this mission. Nuclear safety considerations were based upon potential radiobiological hazards that could occur throughout the Apollo mission and postmission life of the radioactive fuel.

A. NUCLEAR SAFETY PHILOSOPHY

The total quantity of polonium-210 required by the various energy conversion systems varies from 74 grams (334,000 curies) to 154 grams (696,000 curies). Fuel inventory is based upon the criterion of satisfying the power requirements after 44 days of encapsulation, allowing a maximum hold time of 30 days prior to launch.

The determination of a particular re-entry mode for the radioisotope fuel, based on a study of the nominal liftoff to earth orbital insertion and the postmission re-entry trajectories, indicates that it is feasible from an aerospace nuclear safety standpoint to utilize either an intact or burn-up philosophy. For this preliminary study, an intact philosophy was considered conservative, owing to the maximized system weight attendant to this choice. Regardless of the final re-entry mode selected, the radioisotopic fuel must be contained throughout all ground operations.

B. PRELAUNCH PHASE NUCLEAR SAFETY

The prelaunch safety considerations encompass all tasks or activities from fuel encapsulation to the initiation of launch countdown. Other important stages in the prelaunch operation are the transportation and handling of the fuel to the launch area, and the installation of the fuel into the energy conversion system in the Apollo spacecraft. The stringent safety requirements associated with these tasks are not in themselves unique to the Apollo mission: they are similar to the problems which are posed in the movement or utilization of any large radioactive source.

The basic criterion of the prelaunch phase is that no person shall be exposed to a radiobiological hazard.

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Decontaminated fuel capsules must be shipped either separately or in the fuel block in a shipping cask which meets Federal regulations.

The maximum credible accident that could occur during the prelaunch operation would be an accidental dropping of the fuel source from the top of the gantry to the concrete pad. The impact force upon the fuel containment structure would, however, be less severe than terminal velocity impact resulting from re-entry.

C. LAUNCH PHASE NUCLEAR SAFETY

The launch phase of the Apollo spacecraft on its lunar mission is defined as the time interval from the initiation of countdown until the spacecraft is injected into an earth-moon trajectory (thrust termination of the S-IVB stage engine). The primary safety considerations are: (1) adequate radiobiological shielding of spacecraft crew members under normal launch conditions; and (2) the occurrence of aborts and their resulting hazards to the spacecraft crew, launch site personnel and the general populace.

An abort occurring during the ascent portion of the trajectory could result in exposing the nuclear power system to aerodynamic heating and, subsequently, to earth impact at or less than terminal velocity. The design criteria for the system requires that the radioisotopic fuel be adequately protected from aerothermodynamic forces such that intact re-entry can be assured. This mode of re-entry significantly reduces possible radiobiological hazards by eliminating the grey area or partial burnup zone. This is a major factor in terms of safety since the radio-nuclide cannot be released to the atmosphere for a concentrated area dispersion. In addition, intact re-entry enhances containment of the fuel as a result of earth impact by protecting the structural integrity of the containment structure prior to impact.

An aerothermodynamic evaluation of the radioisotope fuel containment structures used in nuclear power systems was performed. The objective of the study was to design the fuel containment structure for each energy conversion system under consideration to conform to the established design criteria of intact re-entry.

D. RE-ENTRY PHASE NUCLEAR SAFETY

The re-entry phase is defined as the period from the time the service module is jettisoned from the command module during the earth re-entry trajectory until the astronauts are recovered and the radioisotopic power source is disposed of safely.

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The basic conclusion that can be drawn with respect to the re-entry disposal of the radioisotope at this stage in the development is that safe conditions are equal to or less severe than early mission conditions and that disposal of the radioisotope can be accomplished; but the particular disposal mode cannot be defined until more specific transearth trajectories and re-entry characteristics of the service module have been established.

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VI. CONVERSION SYSTEM COMPARISON AND SELECTION

The selection of the recommended systems is based on a detailed comparison of each system with the criteria established for the Apollo mission, systems relative cost and development time, and their ability to fulfill power requirements of anticipated space missions in the 1967 to 1970 time period. This chapter presents this comparison with a detailed discussion of the relative merits of and problems involved with each system.

A. SELECTION CRITERIA

The six candidate systems considered in this study were subjected to a rigorous comparison against the following criteria (not necessarily in order of importance). Table 1 presents the data accumulated from the reference designs in Volumes 3 through 8 of this report and compares them against the system criteria.

1. Weight

The RIPS total system weight shall be equal to or less than that of the fuel cell system. The net weight of the latter was estimated to be 840 pounds, taking into account potable water credit.

2. Efficiency

Assuming four fuel loadings per year and a fuel production of 1000 grams of Po-210 per year, the minimum net system efficiency for a 1500-watt(e) output at end of life is 6.3%. Therefore, system efficiency shall be not less than 6.3%.

3. Reliability

The Apollo mission power system specification for reliability is stated as 0.9986 for mission completion (1200 watts for 14 days) and 0.99997 (900 watts for 6 days) for crew safety. The selected power supply must, as a minimum, meet these specifications.

4. Development Time

The permissible development time through man-rated system fabrication and acceptance testing is approximately 36 months. The development program is divided into four phases: Phase I--Design and Component Development; Phase II--Prototype Manufacture and Component Qualification; Phase III--Unmanned Flight Operational Systems, Manufacture and Flight Rating Tests; and Phase IV--Manned Flight Operational Systems, Manufacture and Flight Rating Tests.

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TABLE 1

Application of Fundamental Criteria

Criterion	Brayton	Dowtherm-A Rankine	Mercury Rankine	Stirling	T/E	T/I
Weight (lb)--840-lb maximum						
Unshielded--32 rem	634--619	411	341	537--544	415--432	251
Shielded--3 rem	700--685	487	414	606--613	539--556	421
Efficiency (%)--6.3% minimum	15.8--18.4	14.4	12.1	17.7--19.3	8.55--6.69	10.8
Reliability	-----Mechanical system--test data required-----				Electronic system-- superior reliability	Electronic system-- insufficient test data to establish reliability
Crew safety--0.99997					Crew safety 0.999999	
Mission success--0.9986					Mission success 0.999486	
Safety						
Fuel block surface temper- ature (°F)	≥ 1950*	1650	> 1400*	≥ 1765*	≥ 1600	2600
Fuel center temperature (°F)	≥ 2060	1750	> 1500	≥ 1870	≥ 1680	2800
Confidence level that con- tainment can be met	0.70	0.90	0.70	0.90	0.95	0.40
System integration						
Power quality	-----ac dc-----				dc-direct	dc-direct
Control	-----Acceptable-----				Superior	Superior
Radiator area (ft ²)	90--120	60	22	90--120	195--255	
Radiator coolant average temperature (°F)	250--150	290	570	150	225	
Mechanical	-----Acceptable-----				Superior	Superior
Operational	-----Startup problems-----				Superior	Superior
Mission confidence level**	0.70	0.70	0.50	0.40	1.00	0.40
Development time (months inclusive) delivery flight system						
Phase I	0 to 13	0 to 12	0 to 17	0 to 18	0 to 16	0 to 20
Phase II	9 to 21	10 to 18	9 to 25	17 to 27	10 to 23	17 to 32
Phase III	16 to 27	18 to 26	16 to 31	27 to 36	17 to 28	20 to 36
Phase IV	21 to 32	26 to 36	23 to 36	35 to 45	21 to 31	31 to 41
Total (36 mo)	32	36	36	45	31	41

*See Table 2--Remarks

**Relative probability of meeting Apollo mission requirements (relative to T/E = 1.00).

TABLE 1 (continued)

<u>Criterion</u>	<u>Brayton</u>	<u>Dowtherm-A Rankine</u>	<u>Mercury Rankine</u>	<u>Stirling</u>	<u>T/E</u>	<u>T/I</u>
Estimated cost--development						
Fuel block and GSE safety and management	5.47 M	5.47 M	5.47 M	5.47 M	5.47 M	Not estimated
Phase I	2.77	1.00	3.41	3.06	1.84	
Phase II	1.73	2.06	1.65	2.06	1.82	
Phase III	3.99	3.71	3.65	3.00	1.59	
Phase IV	1.75	3.53	0.88	2.76	1.08	
Total	15.71	15.77	15.06	16.34	11.81	Not estimated

Estimated cost--production unit

Fuel at \$0.45/curie	189 K	216 K	241 K	159 K	361 K	300 K
Fuel block	50	60	60	50	60	60
Conversion system	340	160	175	150	145	140
Total	579	436	476	359	566	500

<u>Application Potential</u>	<u>Power</u>	<u>Mission Duration</u>	<u>Brayton</u>	<u>Dowtherm-A Rankine</u>	<u>Mercury Rankine</u>	<u>Stirling</u>	<u>T/E</u>	<u>T/I</u>
Gemini	500	14 d	No (2)	No (2)	No (2)	No (2)	Yes (5)	No (3)
Advanced MetSat	500	5 yr	No (2)	No (2)	No (2)	No (2)	Yes (6)	Yes (6)
Advanced Solar Probe	750	3 yr	No (2)	No (2)	No (2)	No (2)	Yes (6)	Yes (6)
LEM	750	14 d	No (2)	No (2)	No (2)	No (2)	Yes (5)	No (3)
Advanced ComSat	1000	5 yr	No (2)	No (2)	No (2)	No (2)	Yes (6)	Yes (6)
OAQ	1000	1 yr	No (2)	No (2)	No (2)	No (2)	Yes (6)	Yes (6)
Extended Mission Apollo	1500	100 d	Yes	Yes	Yes	Yes	Yes	No (3)
MEM	3000	2 yr	Maybe (1)	Maybe (1)	Maybe (1)	Maybe (1)	Maybe (6)	Unknown
Lunar base	3000	90 d	Maybe (1)	Maybe (1)	Maybe (1)	Maybe (1)	Maybe (4)	Unknown
MORL-NASA	4000	1 yr	Yes	Yes	Yes	Yes	Maybe (4)	No (3)
MOL-USAF	< 5000	30 to 90 d	Yes	Yes	Yes	Yes	Maybe (4)	No (3)

- (1) Mission duration and crew safety reliability constraints very demanding. Depends on demonstrated reliability.
- (2) Not competitive with direct conversion system.
- (3) Not state of the art in 1960's; unable to meet total fuel containment safety criteria.
- (4) Qualified yes. Extends fuel production capabilities.
- (5) Provided system integration can be achieved.
- (6) Alternate fuel required.

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5. Safety and Crew Radiation Exposure

Total fuel containment through 10 half lives and under all credible conditions including re-entry under maximum aerothermodynamic heating and terminal impact is specified.* Impact containment of the fuel is extremely dependent on the operating temperature of the fuel core. Since the strength of oxidation resistant alloys falls off rapidly above 1800° F, the fuel core surface temperature should not exceed this value. The maximum accumulated radiation dose to the crew shall not exceed the stated NASA requirement of 30 rem due to the radioisotope source.

6. Estimated Cost--Development

The development cost of the systems have been estimated by phase. Fuel block development is broken out as a separate item to delineate the conversion systems development cost.

7. Estimated Cost--Production Unit

These figures reflect the cost per system after development in quantities of 10.

8. System Integration

Major areas of system integration are qualitatively evaluated. Of primary interest is the radiator area required for heat rejection. NASA-Houston has stated that up to 300 square feet of radiator can be made available with a one-year lead time. Logic dictates, however, that the smaller the radiator area required the easier the integration.

All of the systems are designed to produce 28 volts dc to the bus and hence can directly replace the fuel cell power supply. The dynamic systems all will have some ripple in their output. The percent of ripple will ultimately depend on the filter used. No specification for ripple is listed in the power supply specification since the fuel cell is a direct current device.

The response time for supplying peak power is essentially instantaneous since each of the power supplies studied trickle charges a battery system, in addition to supplying main bus power. There is no rise time problem for any of the systems investigated.

9. Application Potential

Since the objective of this program is to determine the feasibility of RIPS for Apollo, primary emphasis in selection of a conversion system is placed on applicability to Apollo. Justification for the development

*See Volume 2; burnup also may be permissible.

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of a radioisotope power supply, however, cannot only be based on this mission, but rather its ability to fulfill Apollo and advanced systems requirements. Typical advanced mission requirements are evaluated with respect to each power supply. The missions evaluated include: Gemini, advanced MetSat, advanced Solar Probe, LEM, advanced ComSat, OAO, Extended Mission Apollo, Mars Excursion Module, Lunar Base, MORL (NASA) and MOL (USAF) A power supply.

B. STATE OF THE ART AND DEVELOPMENT PROBLEMS

The confidence level as shown in Table 1 represents an estimate of the probability that the system can be developed consistent with the Apollo requirements and schedule. Although stated numerically, these values represent a qualitative judgment that a given system can meet its reference design value as determined by consideration of the development problems involved versus the state of the art. To arrive at these values, Table 2 was prepared. It compares development problems versus state of art for each major component of each conversion system. The asterisked parameters are those in which development problems exist and are discussed here for further clarification.

1. Thermoelectric System

The reference design uses PbSnTe-BiTe segmented thermoelectric elements, a concept currently under development in the Martin laboratories. Experimental couples have achieved 9.5% thermoelectric efficiency at the temperature conditions stated. Since these couples have not as yet been subjected to flight qualification tests, the probability of success is somewhat less than unity. The alternate design using plain PbTe is based on current performance of flight-qualified systems and can meet the Apollo system criteria. Thus, no major development problem exists.

The radiator area of 195 square feet must be increased to 255 square feet if the PbTe couple is used. This is within the criteria specification given by NASA-Houston.

2. Thermionic System

The major problems indicated in Table 2 for the development of a thermionic system are as follows:

- (1) Maximum fuel temperature of 1600° C is very close to the melting point of GdPo. Since a liquid fuel is extremely difficult to contain at impact, the possibility of having to develop a new fuel form may become a major development problem. Approximately five years of research has gone into the development of GdPo.

TABLE 2
Comparative Analysis of Development Problems and State of the Art

<u>Conversion System</u>	<u>Major Subsystem</u>	<u>Significant Parameters</u>	<u>Design Objective</u>	<u>Achieved as of 2/15/64</u>	<u>Approach to Achieve Design Objective</u>	<u>Remarks</u>
Thermoelectric	Heat source	Centerline temperature	1680° F	1800° F SNAP 1A, SNAP 11	Multiple capsule	Temperature can be reduced by increasing fuel block surface area. Preliminary calculations indicate maximum temperature = 1480° F. Surface temperature = 1400° F. Weight increases to 60 lb/500-watt fuel block.
		Be and graphite ablator	Capable of fabrication	SNAP 11 similar ablator	Use of current fabrication technology	
		Cladding	Protection of Be	SNAP 11 similar cladding	Use of current fabrication technology	
		Surface temperature (max)	1600° F			
		Weight	50 lb	As-designed		
		Impact resistance	Fuel containment at 450 fps	SNAP 11 impact at 1100 fps	Use of superalloys	
	T/E converter assembly	T/E couple efficiency	9.4%	9.5%	Segmented PbSnTe-BiTe	Based on segmented PbSnTe-BiTe elements, use of straight PbTe gives $\eta_{\text{gen}} = 6.7\%$, wt = no change, fuel inventory = 28,000 watts, system in use in SNAP 3, 7, 9A and 11. System entirely feasible. State-of-the-art improvement to segmented couple considered nominal. Segmented couples fabricated and laboratory tested, also being developed for SNAP 21.
		$T_h = 900^\circ \text{F}$, $T_c = 250^\circ \text{F}$				
		Power output	28 v dc		Voltage regulator	
		Heat losses for 500-watt module	461 watts	Loss based on current technology	No problem	
		Generator efficiency	0.865	6% (PbTe)	Segmented PbSnTe-BiTe	
		Weight/500-watt module	49 lb	Based on design	No problem	
		Reliability (failure rate per couple)	$\lambda = 10^{-6}/\text{hr}$	$\lambda = 10^{-7}/\text{hr}$	No problem	
						See Volume 3, Reliability.

TABLE 2 (continued)

<u>Conversion System</u>	<u>Major Subsystem</u>	<u>Significant Parameters</u>	<u>Design Objective</u>	<u>Achieved as of 2/15/64</u>	<u>Approach to Achieve Design Objective</u>	<u>Remarks</u>
Thermoelectric	Heat rejection system	Radiator reliability (14 d)	0.9999	Vehicle skin sufficient	No problem	Calculations based on NASA data.
		Radiator emissivity	0.90	0.92		
		Coolant pumps reliability (14 d)	0.9999	0.9999 by redundant pumps	No problem	Single pump = 0.991 based on Titan and Minuteman data.
		Power required/pump	1.5 watts	Insignificant effect on system		
		Coolant flow	730 lb/hr			
		Pump efficiency	0.60			
		Area for 1500-watt(e) system*	195 ft ²	PbTe = 255 ft ²	Within specification	
		Coolant temperature (max)				
		Inlet	290° F			
		Outlet	265° F			
		Pressure drop	0 5 psi			
	Emergency heat dump	Reliability	0.99999999	Achieved by ordinance and temperature sensor, could be accomplished manually by astronauts.		Requirement does not affect mission success or crew safety, primarily general isotope safety criteria.
Overall		System efficiency (end of life)	0.855			PbTe gives 6.69%.
		Weight (unshielded)	415 lb			PbTe gives 432 lb.
		Reliability				
		Crew safety	0.999999			
		Mission success	0.999486			

*Development problems--see text.

TABLE 2 (continued)

Conversion System	Major Subsystem	Significant Parameters	Design Objective	Achieved as of 2/15/64	Approach to Achieve Design Objective	Remarks
Thermionic	Heat source	Fuel power density	150 watts/cc	250 (for GdPo)		
		Fuel temperature (max)*	2800° F	Melting point GdPo ~ 2900° F	May require higher melting point fuel form	Major problem.
		Fuel capsule corrosion*	No interaction	Liners resist corrosion at 2000° F	Development of inert liner	Major problem.
		Capsule resistance to Cs corrosion	No interaction	TaWHf alloy OK for 100 hr at 4500° F	No problem	
		Impact resistance*	Fuel containment at 450 fps		Multiple fuel capsule construction	Major problem.
	T/I conversion	Cs seal	Containment of Cs in diode	Niobium to alumina seals		
		Electrical lead-through*	Capable of 200 amp, seal must withstand impact	Leadthrough seals successful at smaller diameters		Impact integrity serious problem.
		Reliability*	$\lambda = 10^{-7}$ failure/hr	Minimum data available	Testing of significant number of diodes	Difficult.
	Heat rejection	Emissivity	0.70	0.70	Current technology	
	Power conditioner	Stepup converter	Convert 7.7 v dc (at 260 amp) 28 v dc	Similar systems built at lower current		
		Efficiency	0.85	0.85 at lower current		
Brayton	Heat source	Maximum fuel temperature*	$\geq 2060^\circ \text{F}$	1800° F SNAP 1A	Reduce design temperature	Core temperature reduced to 1790° F by reducing turbine inlet temperature to 1140° F and consequently reduced η to 12%. Reduction assures fuel containment.
		Be and graphite ablator	Operation below Be melting point	Be melting point 2250° F		
		Cladding	Protection of Be	OK		
		Surface temperature*	$\geq 1950^\circ \text{F}$	Not demonstrated		
		Weight (fuel block)	59 lb	--	Design indicates can be met	
		Impact resistance*	Containment at 450 fps	Not demonstrated	Reduce design temperature	

TABLE 2 (continued)

<u>Conversion System</u>	<u>Major Subsystem</u>	<u>Significant Parameters</u>	<u>Design Objective</u>	<u>Achieved as of 2/15/64</u>	<u>Approach to Achieve Design Objective</u>	<u>Remarks</u>
Brayton	Heat source heat exchanger	Weight	39 lb		Design indicates can be met	
		High temperature cladding on fins	Prevent copper oxidation		State of the art	
	Energy conversion-CRU	Alternator (P-M type) efficiency	0.91			
		Turbine				
		Inlet temperature	1540° F	Achieved in prototype system		Reduce to 1140° F to solve fuel block temperature problem. High efficiency still required.
		Inlet pressure	7.49 psia			
		Pressure ratio	1.56			
		Rotor diameter	5.12 in.			
		Efficiency	0.85	Not measured as component		
		Compressor				
		Inlet temperature*	85° F	Achieved in prototype system	Function of radiator effectiveness	System efficiency dependent on compressor efficiency. May present development problem in matching compressor to turbine at lower temperature operation.
		Inlet pressure	4.51 psia			
		Pressure ratio	1.77			
		Rotor diameter	4.15 in.			
		Efficiency*	0.775	Not measured		
		Bearings				
		Reliability*-- 14 day		High with roller bearings		Roller bearings OK for 14-day mission.
		100 day		Gas bearings-- unknown		Gas bearings for longer life; development problem.
		Overall shaft speed	45,000 rpm	Achieved		
	Recuperator	Effectiveness	0.90	Parameters achieved prototype system		Recuperator parameters change for lower temperature operation.
	Valves and ducts	Recuperator bypass valve	Capable of regulating system	Not demonstrated	Electromechanical butterfly valve	Accuracy of sensor and positioner may require development.
		Shutdown valve	Reliable operation	Not demonstrated	Electromechanical guillotine valve	Simple on-off system; no problem anticipated.
		Duct heat losses	260 watts	Prototype system demonstration	No problem	

TABLE 2 (continued)

<u>Conversion System</u>	<u>Major Subsystem</u>	<u>Significant Parameters</u>	<u>Design Objective</u>	<u>Achieved as of 2/15/64</u>	<u>Approach to Achieve Design Objective</u>	<u>Remarks</u>
Brayton	Heat rejection	Radiator heat exchanger				
		Effectiveness radiator	0.90	Demonstrated		
		Emissivity	0.90	0.92		
		Reliability	0.9999	Vehicle skin sufficient		Lunar noon operation will create problems at low heat rejection temperature.
		Area*	120 ft ²	Design value		
		Evaporator	Provide heat sink for pre-launch and launch	Not demonstrated	See text	
	Power conditioning and controls	Coolant pumps	35 watts (e)/pump	State of the art on Apollo pumps		
		Power loss	2%	No problem anticipated		
		Power output characteristics	28 v dc	No problem anticipated		
		Ripple	2%	No problem anticipated		
	Overall	Reliability	25 x 10 ⁻⁶ /hr	Demonstrated		Minuteman data.
		System efficiency (end of life)	0.18			Reduces to 12% at 1140° F turbine inlet.
		Gross shaft power output	2.01 kw			
		Net power output	1.5 kw			
		Weight (unshielded)	619 lb			
Hg Rankine	Heat source* (conduction design)	Fuel temperature (max)	1500° F		Requires design revision	Conduction boiler not designed for capsule integrity. (Radiation boiler is greater problem.)
		Be cladding			Requires design revision	Design does not include Be cladding, must be revised.
		Core surface temperature	1400° F			
		Weight	65 lb		Design revision will up weight	

TABLE 2 (continued)

<u>Conversion System</u>	<u>Major Subsystem</u>	<u>Significant Parameters</u>	<u>Design Objective</u>	<u>Achieved as of 2/15/64</u>	<u>Approach to Achieve Design Objective</u>	<u>Remarks</u>
Hg Rankine		Corrosion of tubes*	Corrosion must be minimized	TRW data 125 mg/in. ² (2500 hr at 1100° F)		Mass transfer may be problem in superheat region. SNAP 8 considering use of columbium to prevent corrosion.
		Impact resistance	Containment at 450 fps		Requires design revision	
		Vapor quality and flow*	100% vapor, no oscillation	Spiral tubes prevent plug boiling	Vapor quality through superheat	Vapor quality requires further investigation.
	CRU					
		Turbine				
		First stage	3.8 in., 6% admission	Similar to existing units		
		Second stage	4.26 in., 10% admission	Similar to existing units		
		Vapor quality second stage inlet *	100%	Not demonstrated	Design assumes sufficient superheat	Poor quality inlet vapor may cause serious erosion problems. May require reheat or separator.
		Efficiency	0.55	Demonstrated at TRW		
		Alternator				
		Output at design condition	1200 cps ± 1%	Demonstrated Sunflower, SNAP 2		
		Efficiency	88%	Demonstrated Sunflower, SNAP 2		
		Bearings (hydrodynamic)				
		Spiral groove axial thrust	210 watts total consumption	Higher power systems have demonstrated corrosion at temperature		
		Three sector journal, radial loads	No corrosion problems	Not demonstrated		Liquid metal corrosion may be problem
		Reliability*	MTTF 40,000 hr	Life tests up to 6500 hr		More testing required.

TABLE 2 (continued)

<u>Conversion System</u>	<u>Major Subsystem</u>	<u>Significant Parameters</u>	<u>Design Objective</u>	<u>Achieved as of 2/15/64</u>	<u>Approach to Achieve Design Objective</u>	<u>Remarks</u>
Hg Rankine	Heat rejection	Radiator system				
		Tapered tube condenser sub-cooler*	Stability under zero g	Not demonstrated, negative g OK	Need flight tests	Zero g demonstration required.
		Join Haynes 25 tube to Al fin*	Braze	Not demonstrated		Braze technique appears unfeasible.
		Aluminum fin	Operation up to 575° F	Not demonstrated		Aluminum integrity very poor at temperature
	Power conditioning	Weight	7.6 lb	Not demonstrated	All steel construction	Significant weight increase for all steel construction.
		Rectifier regulator				
		Output	28 v dc, 4% ripple	No anticipated problem		Ripple may be too high, weight increase to rectify small (< 10 lb).
		Weight	15 lb			
	Overall	η	0.91			
		System efficiency (end of life)	12.1%			Probably will be reduced with new boiler design.
		Net power output	1.5 kw			
		Weight (unshielded)				
Dowtherm-A Rankine	Heat source and heat exchanger	Radiation boiler	341 lb			Both low due to boiler and radiator design.
		Conduction boiler	403 lb			
		Fuel temperature (max)	1793° F	Current technology	No problem	
		Fuel block surface temperature	1650° F	Current technology	No problem	
		Weight (source and boiler)	70 lb	As designed	No problem	
		Polymer formation in fluid*	< 1.0%	Figure 2 indicates < 0.06%	No problem	
		Impact resistance	Fuel containment at 450 fps	Current technology	No problem	
	Boiler stability	No plug or bubbly boiling under zero g		Not demonstrated	Spiral turbulator in tube	

TABLE 2 (continued)

<u>Conversion System</u>	<u>Major Subsystem</u>	<u>Significant Parameters</u>	<u>Design Objective</u>	<u>Achieved as of 2/15/64</u>	<u>Approach to Achieve Design Objective</u>	<u>Remarks</u>
Dowtherm-A Rankine	CRU	Turbine				
		Inlet temperature	650° F	Demonstrated at Sundstrand	No problem anticipated	
		Efficiency	0.65	Calculated, now under test	No problem anticipated	
		Vapor quality	100%	Achieved without superheat	No problem anticipated	T-S diagram indicates why possible.
		Pump (pitot)	0.25	0.50 demonstrated	Little effect on system performance	
		Alternator (main and control)	0.85	State of the art	No problem anticipated	
		Bearings (hydro-dynamic) (tilting pad)				
		Reliability	MTTF = 10 ⁶ hr	Technology well established	Additional tests	Accuracy of MTTF to be demonstrated.
		Weight/CRU	50 lb	Demonstrated		
	Regenerator	Effectiveness	0.90	Demonstrated		
	Heat rejection system	Condenser fabrication	Al tapered tubes brazed or cemented to skin	OK as designed	OK temperature (max) in tubes 323° F	
		Liquid--vapor interface stability*	Stable in zero g	Not demonstrated	Flight test needed	Zero g demonstration required.
		Auxiliary radiator	Dump excess heat	As designed	No problem	
		Sensor and valve*	Accurate setting of heat dump	Not demonstrated	Development program	
		Radiator area (total)	60 ft ²	As designed	No problem	
	Control system	Speed control	Dissipative alternator, parasitic load	Demonstrated	No problem	
		Dump inlet pressure	Simple relief valve	State of the art	No problem	

TABLE 2 (continued)

Conversion System	Major Subsystem	Significant Parameters	Design Objective	Achieved as of 2/15/64	Approach to Achieve Design Objective	Remarks
Stirling	Overall	Turbine inlet temperature sensor and control valve	Bypass valve, electromechanically actuated	Not demonstrated	Development program	
		System efficiency (end of life)	0.144			
		Weight (unshielded)	411 lb			
	Heat source	Fuel temperature (max)	$\geq 1870^{\circ}\text{F}$	1800° F SNAP 1A	Reduce fuel temperature to $< 1800^{\circ}\text{F}$	Could reduce temperature by increasing area for heat transfer as in T/E system.
		Capsule surface temperature	$\geq 1765^{\circ}\text{F}$	1600° F SNAP 1A	Reduce fuel temperature to $< 1800^{\circ}\text{F}$	
		Weight fuel block	50 lb	As designed		Weight increases with above modification.
		Weight René heat exchanger and tubing	16 lb	As designed		
		Impact resistance	Containment at 450 fps	SNAP 11 impact at 1100 fps	Use of superalloys	
	Engine	Roller bearing lubrication (crank pin and crank-shaft)	0 g operation	Not demonstrated	Mist system	Design approach appears reasonable; zero g demonstration required.
		Displacer and power piston shaft seals	Acceptable deterioration	Allison tests	Spring loaded white metal rubbing contact	Displacer piston shaft seals failed three times in 950 hr.
		Power piston seal	Acceptable deterioration	Not demonstrated	Clearance type white metal	Tested and discarded Rulon R-5 ring. Insufficient test data.
		Displacer piston seal	Acceptable deterioration	Unknown	Clearance type white metal	Operates in high temperature zone (1300°F).
		NaK pump heater temperature	1250°F	OK	Canned rotor or EM	
	Heat rejection system	Emissivity	0.90	0.92	No problem	
		Area	120 ft^2	As designed	No problem	
	Control	Parasitic load	As designed	No problem	No problem	
	Overall	System efficiency (end of life)	0.193			Probably will be reduced with reduced fuel temperature.
		Reliability (14 d)	0.9968	Not established		Major problem.
		Weight (unshielded)	544 lb			Probably will increase with reduced fuel temperature.

- (2) Existence of corrosive attack of the fuel form on the tantalum liner may present a significant problem at 1600° C. Mound Laboratory has found no such attack at 1000° C but feels that corrosion will take place as the melting point of the fuel form is approached. A development program is required to investigate this problem.
- (3) Operating temperature of the thermionic capsule at approximately 2700° F indicates that a major containment problem may result when this core impacts at its terminal velocity. Actually two problems are involved, the impact resistance of the fuel core itself which must be a refractory metal at these temperatures and, second, the oxidation resistance of the fuel core after impact has taken place. Primary containment of the fuel is a refractory tantalum-tungsten-hafnium alloy. Oxidation corrosion is dependent upon the structural integrity of the outer shell of the thermionic system. Any leak in this outer shell after impact would cause the fuel containment structure to oxidize and eventually release the fuel.
- (4) Electrical leadthrough--Since the basic characteristic of a thermionic system is one of low voltage and high current, the system operates at a current of 250 amperes. The conduit which must carry this current starts at the emitter surface and leads through the containment structure. It must be insulated from this structure by either a metal-glass-metal seal or a metal-ceramic-metal seal. This point represents one of the weakest areas in fuel containment since the integrity of this seal must be maintained during and after the impact phenomena.
- (5) Reliability--A diode mean time to failure of approximately 10⁶ hours is required to meet the system reliability requirements of the Apollo mission. Very little data are available on the reliability of thermionic diodes in this power range.

3. Brayton System

There appears to be no question that the fundamental Brayton conversion system can operate in a zero g space environment. Several problems do exist, however, in the reduction of this concept to hardware. These problems are discussed.

a. Heat source

Since the Brayton system operates at a relatively high ΔT , the fuel core which drives the system must operate at relatively high temperatures. The reference design system has a maximum fuel temperature

Fig. 1. Optimized CRU Efficiency Versus Turbine Inlet Temperature--Bravton Cycle

of approximately 2062° F and a fuel core surface temperature of about 1950° F (assuming a turbine inlet temperature of 1540° F). Fuel encapsulation from a safety standpoint has not been achieved at these operating temperatures. Two approaches may be used in providing a satisfactory design.

(1) Multiple fuel core construction

The development of a fuel core that can operate at these temperatures may be approached in the following manner. The gadolinium-polonide is encapsulated in a tantalum liner. This in turn is encapsulated in a high strength TZM refractory alloy. Oxidation resistance of the refractories is accomplished by a third capsule utilizing Haynes 25 or dispersion-hardened Haynes 25. Since the impact resistance of such a capsule has not been demonstrated, it is felt that this area represents a significant development problem.

(2) Lower operating temperatures

Discussion of this problem with the Garrett Corporation prompted them to run an analysis of lower turbine inlet temperatures under optimized conditions. The results of this analysis are presented in Fig. 1. Assuming optimized turbine and compressor efficiencies, this indicates that a CRU efficiency of 14% and a systems performance efficiency of approximately 12% can be achieved by reducing the turbine inlet temperature to 1140° F resulting in a fuel core temperature of approximately 1787° F. It greatly reduces the problem of developing a satisfactory containment structure. It is recommended that this second approach be followed in developing a system to meet the nuclear safety requirements.

b. Combined rotating unit (CRU)

If the course of reducing the heat source heat exchanger temperature is taken, the matching of turbine efficiency and compressor efficiency becomes a more difficult task. This manifests itself principally in the compressor efficiency since the total system efficiency is extremely dependent upon this factor. This parameter bears further investigation.

The bearings proposed in the Brayton cycle system are of two types. AiResearch feels that a high reliability bearing system can be achieved for a 14-day mission with oil-lubricated roller bearings. The reliability, however, falls off rapidly as a function of time and, for missions of 100 days or longer duration, a gas bearing must be developed. The concept of gas hydrodynamic bearings has been studied for a long period of time. AiResearch has done significant research in this area and feels confident that such a bearing can be developed. To our knowledge, however, no one has demonstrated gas bearings which can operate at the thrust and power levels of this system.

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c. Heat rejection system

The design radiator area of 120 square feet is required to reduce the compressor inlet temperature to 85° F. This low heat rejection temperature will present a problem especially during the lunar noon operation of the Apollo mission. It may be necessary to provide an auxiliary loop in the system, possibly the same as the evaporator loop, which is capable of accepting the excess heat during this lunar noon operation. Such a system has yet to be designed.

In summary, the three developmental areas remaining in the Brayton system are the temperature of the heat source, the efficiency of the compressor, the development of a gas bearing system and satisfactory design and analysis to show that systems operation can be achieved under lunar mission conditions.

4. Mercury Rankine System

The mercury Rankine energy conversion system has been studied in detail for space missions since 1956. A number of systems have proposed this conversion technique including the SNAP 1, 2 and 8, SPUR, SPUD and Sunflower. The funding for these programs has been heavy but significant problems still remain in developing the mercury Rankine conversion cycle. These problems are discussed in detail in the following paragraphs.

a. Heat source

There are two basic problems associated with liquid mercury within the boiler or heat exchanger of this system. The first of these is the corrosion of liquid mercury with the Haynes 25 tubing used in its containment. Thompson-Ramo Wooldridge has measured a corrosion rate of 125 mg/in.² of tubing surface area for a period of 2500 hours at 1100° F. On the basis of these data, they state that there is no mass transfer or corrosion problem. Since the superheater of this system operates at temperatures considerably in excess of this, namely 1350° F, we question this conclusion. It is significant to point out that Aerojet General, operating at a similar turbine inlet temperature, is considering use of a columbium tubing system to prevent corrosion and mass transfer. Further, the turbine inlet temperature was dropped to approximately 1250° F on that system.

The second problem associated with the heat source has to do with the quality of the mercury vapor and the flow from the boiler to the turbine inlet. Early in the SNAP 1 program, plug boiling was observed in the isotope boiler system. This caused serious oscillations in the exit gas of the boiler and caused erratic turbine operation. This prob-

CONFIDENTIAL

MND:3296-1

lem was eased to a great extent by placing a spiral turbulator within the tubing causing the vapor and mercury droplets to be thrown against the boiler tubes. In a mercury vapor system, it is necessary to significantly superheat the vapor before any expansion takes place. This is caused by the negative slope of the temperature entropy diagram at points of high entropy. Any expansion decreases the quality of the vapor. Thus, superheating is required or serious turbine erosion can result. This problem requires further analysis.

The fuel capsules required for the boiler of this system were not designed for intact re-entry or abort integrity. To meet the fuel safety requirements this boiler must be redesigned to incorporate the integrity and ablator materials and the operating temperatures reduced to keep the fuel core below 1800° F. These changes will further reduce the overall efficiency of this system.

b. CRU

A two-stage turbine is used for this system; the first allowing 6% vapor admission, the second 10%. This two-stage turbine was required to optimize system performance at this power level. Reference to the temperature entropy diagram (Volume 8) indicates that significant superheating would be required for the vapor quality to remain 100% as the vapor enters the second-stage turbine. Stated in another way, as vapor expansion takes place in the first-stage turbine, the tendency is for the vapor to go through the saturation curve and be less than 100% quality. This vapor must then drive the second-stage turbine without superheating. It is felt that this is an optimistic assumption and certainly requires further analysis.

Hydrodynamic liquid film bearings are proposed for the mercury Rankine cycle. A spiral groove axial thrust bearing is proposed while a three-sector channel bearing takes the radial loads. Thompson has run life tests up to 6500 hours on such bearings without failure.

c. Heat rejection system

One of the major problems associated with a Rankine cycle in a zero g environment is that of adequate and stable condensation of the vapor phase back to the liquid phase. Systems utilizing this conversion technique have generally proposed a tapered tube condenser. This concept essentially assumes that liquid droplets form on the condenser walls, converge into sheets of liquid and, as the tube diameter decreases, eventually form a stable interface which does not vary in position within the tube. A number of tests have been run on this concept, notably by Thompson, Aerojet General, and Sundstrand, in various conditions of condenser attitude ranging from one g in the vertical position to the horizontal position to a slightly negative g field in a tilted position. In

Fig. 2. Rate of Polymer Formation Versus Fluid Temperature--Dowtherm-A Rankine Cycle

addition, very short duration zero g condensation-tests have been performed in KC-135 aircraft and in drop tests. These tests have all been positive and indicate that such a condenser can be built and will operate. Some question still remains, however, as to film stability in the zero g environment.

Thompson proposes to use Haynes 25 tubing throughout the conversion system. They further propose to braze the Haynes 25 condenser tubing to the aluminum skin. From a metallurgical standpoint, it is felt that such a braze represents a significant step beyond current technology. Since the inlet temperature is 575° F to the condenser, there is further doubt that an aluminum skin section would have sufficient strength at this temperature to support the radiator section. A Haynes or stainless steel skin section for the mercury Rankine radiator would add significant weight to the system.

To summarize the problems associated with the mercury Rankine system, the Martin Company feels that containment of liquid mercury at the specified temperatures still represents a problem in this cycle. Mass transfer of material from the hot area to cooler portions of the system, such as the turbine blades, the condenser or pump, could present a serious problem to efficient system operation. The vapor quality also remains a significant problem. It is further felt that insufficient analysis has been performed to adequately design the boiler and condenser radiator system. There is inadequate evidence to prove that tapered tube condensation can be accomplished in a zero g environment.

5. Dowtherm-A Rankine Cycle

The Sundstrand Corporation has proposed the use of a Dowtherm-A Rankine cycle to fulfill the power requirements for the Apollo mission. This system has certain significant advantages which make it quite attractive for use with an isotope heat source. The working fluid temperature at the turbine inlet is 650° F. This allows a fuel temperature of approximately 1795° F to be achieved, thus assuring safe operation. The specific problems associated with this system are discussed here.

a. Polymer formation in working fluid

Throughout the history of the organic reactor program when biphenyl was considered as a moderator-working fluid, pyrolytic and radiolytic polymer formation has been considered a problem. Sundstrand feels that a system can be operated with as much as 5% high molecular weight polymer present in the working fluid. To be on the safe side, however, they have stated that a design objective of less than 1% would be conservative. Figure 2 shows the available information on polymer formation in biphenyl and Dowtherm-A as a function of temperature. The curve shown is above nearly all of the experimental points and has been

Fig. 3. 1000-Hour Endurance Test Performance History--Stirling Cycle

used as the reference design curve for this study. At a soak temperature of 650° F, less than 0.06% of polymer is formed over a 44-day mission according to the curve. Since radiation levels within the fuel core heat exchanger are extremely low in comparison to a reactor, no radiolytic decomposition is anticipated. From a design standpoint, the use of Dowtherm-A in this system appears to be conservative.

b. Boiling condensing problem

The problem of boiling and condensing in a zero g environment is present in this system as in all Rankine systems. Boiling stability can be achieved through the use of the spiral turbulator within the boiler tube according to experts in the field of Rankine conversion systems. Zero g condensation appears to be a more significant problem and has been discussed to some extent previously. Since this has not been demonstrated, it is felt that an orbital flight test program is needed to assess the extent of this problem.

c. CRU

The low operating temperature of the Dowtherm-A Rankine cycle, coupled with the fact that the CRU operates at a relatively low speed of 25,000 rpm, minimizes the development problems associated with this system. The design values for the turbine, pump, alternator and bearings look reasonable. Some development is required in the control system, especially in the electromechanical bypass valve used to flatten the turbine inlet temperature as the isotope decays. No major problems can be foreseen in this development, however.

6. Stirling Cycle

An applied research program to determine the potential capabilities of the Stirling engine for supplying satellite and space vehicle power was initiated by the Allison Division of General Motors Corporation in the fall of 1959. This effort culminated in demonstrating the engine under a 1000-hour endurance test, conducted in 1962. Figure 3 presents a graphic log of the performance history over the 1000-hour endurance test. The major problems associated with this system are discussed in the following paragraphs.

a. Seals

The Stirling engine is a reciprocating device with the inherent problems associated with sliding seals. During the 1000-hour performance test, the following seal failures were noted:

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<u>Time (hr)</u>	<u>Failure</u>
427.5	Rulon R-5 rings on power piston
745	Rulon R-5 rings on power piston
788	Displacer piston shaft seal
838	Displacer piston shaft seal
947.5	Displacer piston shaft seal

After 745 hours, the Rulon rings were discarded, a white metal clearance seal was adopted, and the test completed without further power piston seal difficulty. Allison reports that the displacer piston shaft seal failures showed evidence of a pure fatigue failure. After 947.5 hours, a new seal was made of bronze with a 0.002-inch lining of white metal and the test resumed.

b. Lubrication

Roller bearing lubrication (crankpin and shaft) is achieved by a closely baffled oil mist lubrication system. The design approach appears satisfactory; however, operation in a zero g environment has not been demonstrated.

While the predictable high efficiency of the Stirling cycle is very attractive, the lack of demonstrated reliability at the present time, coupled with the inherent sliding seal development problems, mitigates against selection of this system over other dynamic systems.

C. SYSTEM SELECTION

A comparative evaluation of all systems leads to the conclusion that the thermoelectric system is best suited to the Apollo mission. In addition, the proposed 500-watt module is directly applicable to a number of advanced space missions.

Table 3 summarizes the major advantages and disadvantages of the six conversion systems studied. Examination of this table in conjunction with the more detailed examination made earlier, and further detailed in the individual volumes discussing each of these systems, leads to the conclusion that for the lunar mission the thermoelectric system is the only one which can satisfy the reliability and development schedule required for the Apollo mission. Using today's state-of-the-art technology, a 6.7% efficiency can be achieved with very high reliability.

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No major development problems are involved. With a reasonable extension of today's technology, namely that of developing the higher efficiency segmented thermoelectric couples, this efficiency can be increased to 8.55%.

TABLE 3
Summary of Conversion System Analysis

<u>System</u>	<u>Major Advantages</u>	<u>Major Disadvantages or Problems</u>
Thermoelectric	PbTe state-of-the-art technology High reliability No major development problems Modular concept--versatile Low weight	Segmented couples under development Low efficiency Large radiator area Large fuel inventory
Thermionic	Compact Very low radiator area Light weight	Questionable use of GdPo fuel form Requires development of fuel liner Very difficult containment problem (high temperature electrical lead-through) Insufficient data on reliability
Brayton	High efficiency All gas cycle--no boiling or condensing	High fuel core temperature (safety) Gas bearing development Low heat rejection temperature Reliability unsure--insufficient data Heavy

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TABLE 3 (continued)

<u>System</u>	<u>Major Advantages</u>	<u>Major Disadvantages or Problems</u>
Mercury Rankine	Many components already developed Small radiator area	Mass transfer Mercury corrosion-erosion T-S cause vapor quality problem Zero g boiling and condensing Fuel core safety design Radiator integration problem (temperature) Reliability unsure-insufficient data
Dowtherm-A Rankine	Low temperature turbine inlet No corrosion-erosion problems, inert fluid Light and efficient No safety problem Small radiator area	Zero g boiling-condensing Reliability unsure-insufficient data Possible polymer problem if gamma emitter used
Stirling	Very high efficiency Low fuel inventory	Zero g lubrication Reliability unsure-insufficient data Seal problems Long life applicability doubtful High fuel core temperature (safety)

The reliability requirements for the Apollo mission are extremely high due primarily to the fact that once committed to the translunar injection trajectory, the spacecraft may require up to 147 hours to return to the earth. Loss of power for this time period could well be fatal to the astronauts. An orbital space station or similar device is not limited

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by this same requirement since under emergency conditions, earth re-entry could generally be accomplished within one hour. Many of the NASA and DOD advanced requirements will, therefore, have lower reliability constraints. This fact, coupled with the desire to develop a power system of higher efficiency and thereby minimize fuel inventory, prompted us to examine in more detail the high efficiency engine systems studied under this contract. Further, once demonstrated and developed, the reliability of these high efficiency systems may well prove satisfactory for extra orbital missions. A critical analysis of the systems proposed led us to the conclusion that two of the engine systems appeared to have highly desirable features. These two systems are the Brayton cycle system and the Dowtherm-A Rankine system.

The Brayton cycle system couples relatively high efficiency with a gas cycle wherein no boiling or condensing in a zero g environment must be performed. The system is capable of development without testing in space. The major problems in the Brayton system center around the high ΔT required for its operation. This involves a high temperature fuel core and the consequent questionable impact integrity of that core. Further, a development program must be performed to evolve a satisfactory gas bearing that can operate through launch loads and later in a zero g environment. The low heat rejection temperature may also be a limitation, especially on spacecraft wherein the radiator area is limited. The specific power of this system is somewhat low.

The Dowtherm-A Rankine cycle finds its primary advantage in the low temperature required for its operation. Operating at a ΔT from 650° to 250° F eliminates any safety problems specifically associated with the conversion system. Dowtherm-A is a noncorrosive inert fluid and consequently turbine erosion and/or corrosion is not a problem. The system is light and can perform at high efficiency and, in addition, requires a relative small radiator area enabling easy integration into most space vehicles. The disadvantage of this concept primarily centers around the fact that it is a two-phase system operating in a zero gravity environment. This entails a boiling and condensing portion of the cycle and the dynamics of the system associated thereto have not been demonstrated in a zero g environment.

It is not possible at this time to make a technical judgment in selecting between the two cycles. Solutions to the major problem areas associated with each have not been demonstrated. On this basis, we would recommend that the U.S. Atomic Energy Commission embark on a development program for both cycles with primary emphasis devoted to those problem areas indicated. In the case of the Brayton cycle, this emphasis should be placed on reducing the turbine inlet temperature consistent with fuel core containment requirements and examining in detail the effect that this reduction has on systems per-

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formance. In addition, effort should be expended in the development of a reliable gas bearing with tests to determine its applicability under launch load and zero g conditions.

The Dowtherm-A Rankine cycle needs continued development on its major components but major emphasis should be placed on the design of a zero g boiling-condensing experiment. Such an experiment should be performed in orbit and perhaps could use a radioisotope heat source for the energy required. This experiment should be as simple as possible, containing only a heat source, a boiler, a condenser and a pump. The details of this program are outside the scope of this study, but it is felt that a minimum time would be required to conceive a statement of work for the Phase I studies recommended.

It is believed that at the end of the development program for these engine systems, a selection can be made on a much sounder technical basis than is possible today.

It is recommended that a fuel core be developed that is capable of withstanding terminal velocity impact at the highest anticipated operating temperature of the three systems involved. The exact determination of this temperature cannot be made at this time since the minimum turbine inlet temperature for the Brayton cycle has not been established. It is anticipated, however, that a fuel core surface temperature of 1700° to 1800° F would be required. Thus, the fuel core development program should be directed at this number. The fuel core development must include a detailed test program which analyzes each phase of missile operation for the Saturn vehicle. Early in this program, a determination must be made on the safety criteria for this type of fuel capsule. As stated in the ensuing volumes, the systems investigated have been designed for random impact re-entry. The question of intact re-entry or burnup re-entry must be resolved. Volume 2 of this series indicates that from a preliminary analysis either mode of re-entry would be satisfactory for the Apollo trajectory.

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VII. DEVELOPMENT PROGRAM FOR THERMOELECTRIC SYSTEM

This program provides for the development, unmanned flight demonstration and delivery of two completely integrated radioisotope fueled thermoelectric power systems for manned flight operational systems. Figure 4 is a program plan which graphically displays the proposed development test and delivery plan with its associated time span. It can be seen that the program is divided into four basic phases of work which overlap in calendar time.

Each of the four phases of the development program terminates with a logical development milestone and objective. These four phases of work and objectives are discussed.

A. PHASE I--DESIGN AND COMPONENT DEVELOPMENT

The objective of this phase is to produce, at the end of 16 months, development tested components for fuel capsules, fuel blocks, segmented thermoelectric couples, generator modules, ground handling components and a radiation system. The phase allows sufficient time for the iterative process of design and testing and redesign and re-testing to eliminate component deficiencies while arriving at the optimum system design. The various design parameters and components will each be investigated over a sufficiently broad range so as to permit identification of either the optimum design point or best tradeoff between parameters and/or components. The establishment of these design points will provide the basis for the Phase II Prototype Design.

B. PHASE II--PROTOTYPE MANUFACTURE AND COMPONENT QUALIFICATION

The Phase II objectives are to qualify the major elements of the system as components and/or subsystems, and to operate a complete system made up of prototype components with an electrical heat source. Operation and testing of these prototype components and systems will provide the necessary test data and information required to verify predicted system performance and establish system and component control, safety compatibility, structural integrity and reliability.

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Fig. 4. Program Schedule

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Fig. 4. (continued)

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C. PHASE III--UNMANNED FLIGHT OPERATIONAL
SYSTEMS MANUFACTURE AND FLIGHT RATING TESTS

All work and data evaluations and designs of the previous phases will be incorporated into a final unmanned flight operational system design. This will include any redesign or drawing changes that result from the prototype tests or safety tests program.

Two systems, with the necessary ground support equipment, will be manufactured and acceptance tested for installation and flight testing in the Apollo vehicle. These systems will be delivered for unmanned vehicle launches.

D. PHASE IV--MANNED FLIGHT OPERATIONAL
SYSTEMS MANUFACTURE AND FLIGHT RATING TESTS

During Phase IV, support will be provided for the installation and launch of the unmanned system. Concurrently, additional qualification and reliability testing will be completed to produce a fully qualified manned flight operational system design. This design will incorporate the results of the unmanned flight tests.

Two systems will be manufactured and acceptance tested during Phase IV. These systems will be qualified and delivered for manned vehicle launches.

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VIII. COSTS

The following material itemizes the costs for the thermoelectric system according to phases.

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Radioisotope Fueled Thermoelectric Power System

<u>Phase</u>	<u>Definition</u>	<u>Isotopic Fuel Block Cost</u>	<u>Power Conversion System Cost</u>	<u>Total</u>
I	Design and Component Development	\$1,757,000	\$1,843,000	\$ 3,600,000
II	Prototype Manufacture and Component Qualification	1,953,000	1,821,000	3,774,000
III	Unmanned Flight Operational Systems Manufacture and Flight Rating Tests (two systems)	1,150,000	1,594,000	2,744,000
IV	Manned Flight Operational Systems Manufacture and Flight Rating Tests (two systems)	605,000	1,085,000	1,690,000
	Total Development Costs (less fee)	<u>\$5,465,000</u>	<u>\$6,343,000</u>	<u>\$11,808,000</u>

Estimated per Vehicle Set of Hardware

Isotopic Fuel Processing Cost at \$0.45/curie	\$ 361,000
Isotopic Fuel Block Cost	60,000
Power Conversion System Cost	<u>145,000</u>
Total Cost*	\$ 566,000

*Based on quantities of 10.

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