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FEASIBILITY OF ISOTOPIC POWER FOR MANNED LUNAR MISSIONS

VOLUME 2--MISSION, FUEL, AND NUCLEAR SAFETY AND RADIATION

May 1964

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MARTIN **MARTIN**
MARIETTA

NUCLEAR DIVISION Baltimore 3, Maryland

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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,

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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I. APOLLO MISSION REQUIREMENTS

A. MISSION DESCRIPTION

The Apollo mission provides for manned exploration of the lunar surface. The guidelines developed for this lunar landing reconnaissance are:

- (1) A maximum 14-day lunar landing mission.
- (2) A three-man astronaut crew.
- (3) Launch from Cape Kennedy, Florida, on an azimuth which may vary from 72 to 108 degrees.
- (4) Earth parking orbit nominally circular at 115 statute miles.
- (5) One to three earth parking orbits before translunar injection is permitted.
- (6) Utilization of lunar orbit rendezvous mode.
- (7) Nominally circular equatorial lunar orbit between 69 and 115 statute miles altitude.
- (8) Minimum of one lunar orbit to observe the landing site, before initiating lunar excursion module (LEM) transfer orbit.
- (9) The LEM landing transfer orbit and command and service module (CSM) will have the same period.
- (10) Lunar landing nominally occurring on the lunar equator between 30° E and 40° W longitude.
- (11) LEM will do all maneuvering required for rendezvous and docking with CSM. CSM also has maneuver capability.
- (12) A re-entry module capable of withstanding both land and water impact.

The Apollo system at various stages during the mission is illustrated in Fig. 1. The characteristics and performance data for all spacecraft vehicles in time sequence for the complete mission are presented in Table 1. The S-IC, S-II and S-IVB (first burn) stages of the Saturn S-5 vehicle are used to launch Apollo from Cape Kennedy, Florida. It would insert the S-IVB, LEM and service and command modules into a circular parking orbit about the earth as shown in Fig. 2.

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Fig. 1. Diagrammatic Representation of Mission Configurations

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TABLE 1
Apollo Spacecraft Launch Vehicles --Characteristics and Performance
Data in Time Sequence

Events	Stage	Description	Weights			Burnout					Thrust Level (lb)	Time From Liftoff (hr, min, sec)		
			Time of Stage Ignition (lb)	Burnout/ Stage (lb)	Burn Time (sec)	Velocity (fps)	Altitude (ft)	Latitude (deg)	Longitude (deg)					
1	S-IC	Liftoff	6,000,008	1,753,010	150	8,954	200,640	28.4N	79.7W	Maximum 8,634,000 Sustainer 6,907,700	From:	0	0	0
											To:	0	2	30.0
2	S-II	2nd boost from liftoff after staging	1,363,110	449,320	387.4	22,279	598,224	27.5N	65.1W	1,000,000	From:	0	2	31.5
											To:	0	8	59.0
3	S-IVB	1st burnout--obtain earth orbit	359,274	281,356	165.1	25,574	607,200	25.6N	54.2W	200,000	From:	0	8	59.9
											To:	0	11	45.1
4	S-IVB	2nd burn--obtain translunar insertion	281,356	123,244	335.1	35,725				200,000	From:	3	25	39.7
											To:	3	31	14.9
5	Service module	1st burn--insert into lunar orbit	86,500	64,283	324.1	5,298*				21,900	From:	73	54	45.6
											To:	74	0	19.7
6	LEM	1st burn--insert into elliptic orbit around moon	21,968	21,097	395.3	5,353*				1,100	From:	78	40	28.5
											To:	78	47	3.8
7	LEM	2nd burn--descent from 50,000 to 10,000 ft	21,097	11,784	326.3	100*				8,800	From:	79	14	9.6
											To:	79	21	35.9
8	LEM	3rd burn--translation and descent from 10,000 ft	11,784	8,700										
9	LEM	4th burn--liftoff from lunar surface	8,700	4,894	134	5,593*				8,800**	From:	98	28	45.3
											To:	98	30	59.3
10	LEM	5th burn--guidance maneuver	4,894	4,812	9.0	5,759*					From:	99	14	27.8
											To:	99	14	36.8
11	LEM	6th burn--completion of rendezvous	4,812	4,751	300						From:	99	34	59.3
											To:	99	39	59.3
12	Service module	2nd burn--obtain transearth insertion	42,715	31,942	156.7	8,243*				21,900	From:	108	54	0.7
											To:	108	56	37.5
13	Command module	Re-entry	9,500								From:	185	17	27.0
											To:	Depending upon landing site		

*Velocities are referred to rotating lunar surface. Unmarked velocities are inertial. Weights in Column 3 are total configuration weights.

**Time values vary for initiation of lunar takeoff from 98:30:45 for 7-day mission to 258 hr for 14-day mission.

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Fig. 2. Launch Ground Track and Radar Acquisition

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This parking orbit will have a duration of from one to three orbits. The S-IVB is then restarted for its second burn and injects the service module, LEM and command module into the translunar trajectory. During this translunar trajectory, the service and command modules will be oriented such that the command module is attached to the LEM. As the moon is approached, the service module fires to insert itself, the command module and LEM into a circular lunar orbit between 60 and 115 statute miles.

The LEM will then go into a transfer orbit from which it will descend to the lunar surface, carrying two of the astronauts. After completion of the on-site lunar exploration, LEM will ascend to rendezvous with the CSM. The astronauts will transfer to the command module, after which LEM will be jettisoned. The CSM will then be injected into a transearth trajectory by the service module engine. After the service module engine has performed the midcourse correction to achieve the proper earth re-entry trajectory, the service module will be jettisoned perpendicular to the command module trajectory. The command module will then re-enter for either a land or water impact.

Significant events in the Apollo program schedule are shown. This schedule is subject to change as the program evolves. The goal of providing a qualified electrical power system for the initial manned lunar flight within three years is based on this schedule.

Calendar Year														
1966					1967					1968				
Saturn IB	201	202	203	204	205	206	207	208	209					
	○	○	○	○	○	○	○	○	○					
				U	M	M	M	M	M					
Saturn V				501	502	503	504	505	506	507	508	509	510	512
				○	○	○	○	○	○	○	○	○	○	●
				U	U	U	U	U	U	M	M	M	M	M

U = unmanned

M = manned

201	Vehicle test	206	Manned S/C and LEM
202	Vehicle test	207	Manned S/C and LEM
203	Vehicle test	208	Manned S/C and LEM
204	Orbital test S/C and LEM	209	Manned S/C and LEM
205	Manned S/C and LEM	501	Orbital test spacecraft

*Not required for RIPS.

Fig. 3. Average Power Requirements for Apollo 14-Day Lunar Landing Mission

502	Launch environment test	506	Emergency destruct system qualification test
503	CM re-entry test	507	Manned circumlunar
504	CM re-entry test	512	Manned lunar landing
505	CM re-entry test		

B. ELECTRICAL POWER REQUIREMENTS AND DESIGN CONSIDERATIONS

The normal average electrical power requirement for the Apollo system is approximately 1200 watts.* A power level of approximately 900 watts is required to maintain the crew under emergency conditions. Hydrogen-oxygen fuel cells currently provide this power to the bus in the form of 28-volt direct current. The power profile for the 14-day mission is shown in Fig. 3. Included are 513 electrical watts for heaters for the fuel cells not on line. Electrical power for the entry, landing and postlanding phases is provided by storage batteries installed in the command module. These same batteries appear to be adequate to meet the peak load demands encountered for a very short time during the mission itself. However, because the possibility exists that these batteries will not be available for these power peaks, additional batteries are included in determining weights of the several radioisotope systems.

Although great differences exist between the basic nature of the fuel cell and radioisotope power systems, certain portions of the existing power system specifications* can be assumed to be applicable to a radioisotope system. These are discussed in the following.

1. Performance

a. Voltage output

The voltage output of the fuel cell power plant will be 27 to 31 volts during all phases of normal operations.

b. Transient voltage

Within the limits of normal operations, the fuel cell power plant shall not require more than 0.7 second to attain 90% of voltage output when subjected to an instantaneous change in the resistive load of 570 watts.

*Apollo Spacecraft Fuel Cell Electrical Power Plant, NAA Spec MC464-0015 Rev. B, 24 June 1963 (Confidential).

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c. Orientation

The fuel cell power plant must be capable of operating in zero gravity and sustaining accelerations up to 7 g in one axis. The subcontractor shall specify orientation required to minimize fuel cell power plant failure probability.

2. Environmental Conditions

a. Temperature

Operating. The fuel cell power plant shall be capable of operating in a temperature ranging from +30° to +130° F for 14 days.

Nonoperating. The fuel cell power plant shall be capable of withstanding without damage, a temperature of -20° F for 14 days, and +140° F for 14 days including solar radiation.

b. Vibration

Operating. The fuel cell power plant shall be capable of operating as specified herein while being subjected to the vibration levels listed:

Launch Phase

Peak Vibration Acceleration

Sinusoidal

5 to 100 cps	*Linear increase from 0.3 to 7 g
100 to 2000 cps	Constant at 7 g

Random

5 to 50 cps	*Linear increase from 0.005 to 0.086 g^2/cps
50 to 150 cps	Constant at 0.086 g^2/cps
150 to 2000 cps	*Linear decrease from 0.086 to 0.005 g^2/cps

*Denotes linear scale when plotted logarithmically

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Inflight Phase

Peak Vibration Acceleration

Sinusoidal

5 to 400 cps	*Linear increase from 0.23 to 9 g
400 to 2000 cps	Constant at 9 g

Random

5 to 100 cps	*Linear increase from 0.0086 to 0.07 g^2/cps
100 to 550 cps	Constant at 0.07 g^2/cps
550 to 2000 cps	*Linear decrease from 0.07 to 0.05 g^2/cps

Shock. The fuel cell power plant shall be capable of withstanding, without damage, a shock of 30 g for 11 ± 1 millisecond, while in a non-operating condition.

Solar radiation. The fuel cell power plant shall be capable of withstanding, without damage, 360 Btu/ft²/hr of solar radiation for 14 days.

3. Reliability

Apportionment for mission success (1200 watts(e) average for 336 hours)	0.998600
Apportionment for crew safety (900 watts(e) average for 147 hours)	0.999970

4. Radiation Specification

Specification of the dose permissible from a nuclear unit on board the Apollo spacecraft involves consideration of the currently recommended allowable radiation levels for space missions and the expected radiation levels from space radiation. The recommended allowable radiation levels which have been established for space missions by NASA (Ref. 1) are presented in Table 2. Interpretation of this data (neglecting detailed consideration of different criteria for different parts of the body) is that

*Denotes linear scale when plotted logarithmically.

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Fig. 4. Total Radiation Dosage Versus Time from Liftoff for the Total Flight Plan

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- (1) The column titled Average Yearly Dose will apply to radiation encountered in the natural and artificial radiation belts, galactic cosmic radiation and radiation from a nuclear source on board the spacecraft.
- (2) The column titled Maximum Permissible Single Acute Emergency Exposure will apply to solar flare radiation.

The data presented in Fig. 32 of Ref. 2 (reproduced as Fig. 4 of this section), as defined, pertains to the total radiation dosage received from the radiation belts from the time of liftoff for the total flight plan. The level at the end of the mission is 11.3 rad. For a conservative estimate, however, this level is taken as 24 rad to include a possible increase in the contribution from the radiation belts as well as an expected 3 to 4 rem from galactic cosmic radiation over the 14-day maximum mission duration. A maximum permissible dose from a nuclear system on board is computed, thusly,

$$54 \text{ rad} - 24 \text{ rad} = 30 \text{ rad assuming rem/rad} = 1.0$$

This computation is based on interpreting the data in Table 2 that pertains to the blood-forming organs as the whole body dose. The practical minimum dose from a nuclear system was arbitrarily chosen as 3 rem integrated over the 14-day period.

5. Additional Requirements

In addition to the stated specifications, certain rather arbitrary assumptions had to be made to guide the radioisotope power systems design study activity. They are:

- (1) Waste thermal energy must be dissipated from the surface of the service module. This places an upper limit on the total area available for the radiator at about 300 square feet.* Further, this area must be assumed to be scattered in a structurally convenient manner.
- (2) Average equivalent sink temperatures are as shown in Fig. 5.
- (3) The specified weight of the fuel cells and flexible lines is 241.2 pounds. Fuel consumption integrated over the power spectrum is 621 pounds for the 14-day mission. Nitrogen weight, tankage weight and venting requirements are not di-

*This value was determined by NASA-Houston, who stated that such a radiator area could be made available on the service module within a one-year lead time.

Fig. 5. Estimated Sink Temperatures for Apollo Mission

rectly specified, * but they are assumed to be 230 pounds (conservative estimate based on 1/3 lb tankage/lb of fuel). Crediting this system with 250 pounds potable water gives a net fuel cell system weight of approximately 840 pounds. Any isotopic power supply should be equal to or less than this weight.

TABLE 2
Recommended Allowable Radiation Levels for
Space Missions (Ref. 2)

<u>Critical Organ</u>	<u>Maximum Permissible Integrated Dose (rem)</u>	<u>RBE (rem/rad)</u>	<u>Average Yearly Dose (rad)</u>	<u>Maximum Permissible Single Acute Emergency Exposure (rad)</u>
Skin of whole body	1630	1.4	233	500 ¹
Blood-forming organs	271	1.0	54	200
Feet, ankles and hands	3910	1.4	559	700 ²
Eyes	271	2 ³	27	100

¹Based on skin erythema level

²Based on skin erythema level but these appendages believed to be less radiosensitive

³Slightly higher RBE assumed since eyes are believed more radio-sensitive.

*Apollo Spacecraft Fuel Cell Electrical Power Plant, NAA Spec MC464-0015.

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II. SAFETY

A nuclear safety study of the Apollo manned lunar spacecraft, utilizing a nuclear powered electrical system, is presented 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. This required a detailed investigation into the mission, mission requirements, nuclear characteristics and quantity of radioisotopic fuel, fuel accountability and spacecraft environmental radiation resulting from the nuclear power source. Because of the large quantity of radioactive fuel needed to provide the necessary spacecraft electrical power and the nature of the mission, nuclear safety considerations are of paramount importance.

A. NUCLEAR SAFETY PHILOSOPHY

In establishing an aerospace nuclear safety philosophy, the primary concern is with the radioisotope fuel. The basic radioisotope fuel considerations are inherently influenced by the mission and involve the degree of containment and accountability of the fuel, and its associated radiobiological effects. The spacecraft launch trajectory, the large quantity of radioisotope fuel required for the mission, the presence of astronauts and the safety of the general populace serve to emphasize the need for a sound and realistic nuclear safety philosophy. The only plausible philosophy acceptable is one in which neither the astronauts nor the general populace are subjected to undue radiobiological hazards resulting from the planned Apollo mission and from any credible accident during any phase of the mission or postmission life of the radioisotope.

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.

Major considerations in developing the nuclear safety philosophy were the earth launch trajectory and relatively low earth parking orbit. A study of a nominal Apollo spacecraft trajectory liftoff to earth-orbital insertion was performed to determine: (1) probable impact regions for intact re-entry and (2) partial burnup regions. The study was based upon malfunctions of the Saturn V propulsion system, considering both the second

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Fig. 6. Probable Impact Region for Partial Burnup During Earth Boost Phase of Mission

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stage, S-II, and third stage, S-IVB. Analysis of these stages provides: (1) the earliest possible abort trajectory and partial burnup; (2) the critical point of the stage interaction; and (3) the maximum downrange impact point resulting from a S-IVB malfunction occurring slightly before orbital insertion. The results of the study are illustrated in Fig. 6. The probable impact (intact re-entry) and partial burnup regions occurred between approximately 1350 and 3775 nautical miles from Cape Kennedy, Florida, or approximately 2965 to 540 nautical miles, respectively, from the African Coast.

In case the generator and fuel capsule are completely burned up on re-entry, the activity will be injected into the upper atmosphere. The injection will take place above the stratosphere, but it is conservative to neglect the residence time in the mesosphere and consider the injection as taking place directly into the stratosphere. It is the consensus that the average residence time for such particles in the stratosphere is two years, and for the troposphere, 21 days (Ref. 3).

It can be shown (Ref. 4) that for an injection of 696,000 curies of Po-210 into the stratosphere, the maximum ground deposition occurs at 160 days, at which time the total deposition is 1.586×10^5 curies. The maximum tropospheric concentration occurs at 49 days when the troposphere inventory is 5.924×10^3 curies.

It has been found that for injection of radioactive material into the upper atmosphere the fallout is distributed fairly uniformly in the hemisphere of injection between the 20th and 70th parallels of latitude (Ref. 5). The area of this zone is 5.4×10^{14} square meters. If it is assumed that at the time of maximum ground activity 80% of it is deposited in this zone, the average ground concentration is 2.65×10^{-10} curies/m² (Ref. 6).

Some studies of the occurrence of polonium-210 in the biosphere have been performed. In soil samples collected in the Pittsburgh region, 5 and 7 $\mu\mu\text{c/gm}$ were found. These samples were collected in the top five centimeters of soil. If it is assumed that the activity of this top five centimeters of soil can be compared to ground deposition, it is found, for soil with an average density of 2.5 gm/cc that the 7 $\mu\mu\text{c/gm}$ is equivalent to 8.75×10^{-7} curie/m². This concentration is more than 3000 times the deposition resulting from burnup of an isotopic power system using 696,000 curies of Po-210.

The troposphere extends to a height of 10,700 meters (standard atmosphere). The volume of the zone between 20 and 70 degrees latitude is 5.78×10^{18} m³. If it is assumed that 80% of the tropospheric

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inventory is uniformly distributed through this volume there is, at the time of maximum concentration, 9×10^{-16} curie/m³ (or $\mu\text{c/cc}$).

This concentration may be compared with the maximum permissible concentration of polonium-210. For the soluble form, the MPC_a, for a 168-hour week, is 2×10^{-10} $\mu\text{c/cc}$. The spleen and the kidney are the critical organs. For the insoluble form, the lung is the critical organ, and the MPC_a is 7×10^{-11} $\mu\text{c/cc}$. These values are seen to exceed, by several orders of magnitude, the predicted tropospheric concentration.

The formulation of the aerospace nuclear safety philosophy was governed by:

- (1) Potential radiobiological hazards associated with environmental radioactive contamination resulting from a release of 696,000 curies of Po-210.
- (2) Establishment of component re-entry burnup and launch pad intact integrity.
- (3) Partial burnup regions.
- (4) Distribution rates and dispersion patterns of injected radioactivity into the atmosphere.
- (5) Relatively defined impact points for intact re-entry.
- (6) General safety requirements with respect to mission phases--prelaunch, launch, lunar exploration and postlunar.

The determination of a particular re-entry mode of 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 burnup 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 absolutely contained throughout all ground operations.

B. NUCLEAR SAFETY DESIGN CRITERIA

The safety design criteria for a nuclear powered system for this mission are, in part, dictated by the magnitude of the potential hazards

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concomitant with the use of a large quantity of radioisotopic fuel. Adherence to safety philosophy requires that the fuel containment structure (FCS) must be capable of satisfying the following specific design criteria.

- (1) The FCS and/or shipping cask shall provide for absolute fuel containment under all earth transport and storage environments.
- (2) The structure must retain its mechanical, thermal and chemical integrity under launch pad abort environments.
- (3) It shall be capable of intact re-entry resulting from any trans-lunar injection malfunction.
- (4) The FCS shall retain its containment integrity after earth impact at the terminal velocity resulting from re-entry. (For purposes of this study only; a burnup design may be equally attractive.)
- (5) The structure must remain intact when exposed to thermal conditions resulting from underground burial.
- (6) The FCS shall retain its structural integrity when under maximum helium gas internal pressure resulting from the alpha decay of the fuel.
- (7) Structural integrity of the FCS must be retained under the corrosive action and hydrostatic pressure of the sea at depths up to 1000 fathoms (6000 ft) for a period of 1400 days, approximately 10 half-lives of the polonium-210 fuel.
- (8) The FCS shall retain its integrity if the coolant is lost.
- (9) Sufficient radiobiological shielding shall be provided to achieve:
 - (a) Acceptable radiation levels established by the Interstate Commerce Commission regulations.
 - (b) Mission requirements and to protect the crew from any excessive radiational environment.

Safety considerations with respect to the various phases of the mission and the safety design criteria are discussed in the following sections of this report.

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C. 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.

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

- (1) The shipping cask shall be designed to provide for the removal of thermal energy resulting from radioactive decay. This condition must be met during prolonged storage, and at an ambient temperature of 120° F without the surface temperature exceeding 180° F, including the case where loss of coolant, if used, occurs. Hence, a design which utilizes conduction and natural convection precludes any potential problem associated with a circulating liquid coolant.
- (2) The structural characteristics of the shipping cask must be sufficient to withstand any credible transportation accident and still retain its shielding and fuel containment integrity. This would prevent any possible radiobiological hazard to the general public should a transportation accident occur.
- (3) Sufficient radiobiological shielding must be provided during transportation to comply with Interstate Commerce Commission regulations which permit maximum dose rates of 200 mr/hr at surface and 10 mr/hr at one meter from the surface of the shipping cask.

In addition to these specific requirements, launch site operation imposes several unique requirements. For example, the handling and installation of the fuel source must be minimized to prevent or reduce possible radiobiological hazards. Site operations can be reduced by designing the shipping cask so the heat source can be directly installed in the conversion system in the service module. Since higher surface temperatures can be tolerated at the site, the source might be placed in a special gantry shield (lighter in weight than the shipping cask) for transport to the service module of the Apollo spacecraft atop the Saturn

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V launch vehicle. Remote handling equipment, designed specifically for transferring the nuclear power source to the service module, will be used atop the gantry. Since certain operations in the transfer may expose the fuel source, portable shielding will be used to assure a low radiation environment to minimize personnel exposure. Procedural changes such as limited access can also be used to minimize exposure.

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.

D. 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). In determining the feasibility of radioisotope power, the added safety considerations existing as a result of the linking together of man and a nuclear powered system must be thoroughly evaluated as well as the safety considerations of an unmanned launch. 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.

The nuclear powered conversion system is located in the service module of the spacecraft as far as possible from the command module to minimize shield weight requirements.

It is only possible to set limits for the overpressures that may develop in case of a launch pad abort with detonation of the fuel-oxidizer mixture. The mode of mixing upon rupture of the tanks is random and may give a result utilizing anything from about 20% of the theoretically available TNT equivalent down to zero. Usually only a small fraction of the available energy is manifested in overpressure. A. D. Little Company has carried out a number of scaled tests and Space Technology Laboratories (STL) (Ref. 7) has proposed empirical equations based on these data for the overpressures developed in LOX-RP-1 detonations as a function of distance. The average found is given by

$$p = \frac{593}{Z^3} - \frac{28.8}{Z^2} + \frac{20.7}{Z} \quad (1)$$

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Fig. 7. Overpressure from Detonation of S-I Stage

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and the value for the 98% confidence level (98% of the explosive yields are exceeded by)

$$p = \frac{9862}{Z^3} - \frac{187.9}{Z^2} + \frac{52.8}{Z} \quad (2)$$

where

$$Z = R/W^{1/4}$$

and

$$R = \text{distance (ft)}$$

$$W = \text{total weight of propellant mixture (lb)}$$

The overpressures for a detonation of the S-I stage of the Saturn have been calculated for distances from 30 to 150 feet and are shown in Table 3 and the graph of Fig. 7. It is to be expected that the origin of the detonation shock wave would be at the junction of the fuel and oxidizer tanks. The generator location is about 120 feet from this. It is seen that the overpressures developed are not in a range that endangers the integrity of the fuel capsules which will withstand 15,400 psi. If the detonation should be that of the S-II or S-IV stage, the overpressures developed are relatively less but the generator location is nearer the probable epicenter. Even in this case the overpressures are a great deal less than the failure point of the capsule. The overpressures for detonation of the S-II and S-IV stages have been calculated using Eqs 1 and 2 and are given in Table 3. These values are not strictly accurate since these stages use LH_2 instead of RP-1.

TABLE 3
Overpressure for Propellant Detonation (psi)

Saturn Stages						
Distance (ft)	S-I		S-II		S-IV	
	Average	98% Con- fidence	Average	98% Con- fidence	Average	98% Con- fidence
30	543.0	9120	294	4840	128	2090
60	72.0	1140	40.7	607	16	265
90	24.3	396	14.4	183	4.7	82
120	12.0	147	7.6	80	2.0	37
150	7.4	77	4.9	43	1.0	21

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It is estimated that the maximum diameter of the fireball for detonation of the total Saturn vehicle is 1300 feet and that its duration would be 17.5 seconds. Tests carried out by the Martin Company (Ref. 8) show that the fireball temperature may reach 4000° to 5000° F with a maximum heat flux of about 20 Btu/ft²-sec. The temperature of the subsequent fire is about 2000° F. This fire will persist for 20 to 30 minutes. The melting point of the Haynes 25 encapsulating material is exceeded for only a few seconds, and the capsules within the generator structure of a thermoelectric generator will not receive a sufficient heat flux and a temperature to violate the containment.

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.

1. Re-entry Aerothermodynamics Study

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. The energy conversion systems investigated were:

- (1) Thermoelectric
- (2) Brayton
- (3) Dowtherm-A Rankine
- (4) Stirling
- (5) Thermionic.

However, owing to similarity in the design of the fuel structures from aerothermodynamic considerations (shape, size and weight), the thermoelectric system fuel structure was considered typical of the first four systems. Three different thermionic system configurations--one-module, two-module and four-module designs--were evaluated (500 watts/module).

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The spacecraft flight trajectory from liftoff to earth re-entry after lunar exploration was evaluated to determine the most stringent aerothermodynamic forces that could be seen by the fuel containment structure for either normal or abort conditions. The most severe thermal environment was found to occur in the last portion of the transearth trajectory, in which the nuclear system re-enters the earth's atmosphere at superorbital velocity. The following initial superorbital re-entry condition at the outer edge of the earth's atmosphere was used to determine the total heat available for the fuel containment structure (Refs. 9 and 10):

Inertial velocity, $V_o = 36,000$ fps

Inertial flight path angle, $\gamma_o = -6$ degrees

Altitude, $h_o = 400,000$ feet

The total heat available from this trajectory was corrected for the proper ballistic coefficient and characteristic radius. Additional factors were applied for shape and orientation of the generator, i. e., a tumbling fuel block with a flat face. All bodies were considered to be re-entering in a tumbling mode.

The heat required to expose the fuel capsules was determined by the following assumptions and calculations:

- (1) Only the ablator was considered as a heat sink.
- (2) Total heat required is equal to the sum of the following quantities:
 - (a) Amount of heat necessary to raise the ablator to its melting or sublimation temperature.
 - (b) Amount of heat absorbed by the ablator's change of phase, such as heat of fusion or sublimation.
 - (c) Radiation losses at some average temperature for a period of time and at the ablator melting temperature for another period of time.
- (3) No heat transferred internally.

The minimum ablator thickness was determined when the total heat required to expose the capsules was equal to the total heat available from a tumbling re-entry trajectory.

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The results of the aerodynamic study indicate that the following system requirements conform to the safety design criteria of intact re-entry.

- (1) The thermoelectric system (similar to Brayton Dowtherm-A Rankine and Stirling systems) can be protected from super-orbital re-entry conditions by placing approximately 1.5 to 2 inches of beryllium over the flat faces and approximately one inch of pyrolytic graphite over the four smaller faces. The pyrolytic graphite on the edges of the fuel block reduces the edge heat losses, thereby increasing generator efficiency. In addition, it contributes to prevention of capsule re-entry burnup.

The resultant fuel blocks have the following weights and sizes:

	<u>Thermo- Electric</u>	<u>Stirling</u>	<u>Brayton</u>	<u>Dowtherm-A Rankine</u>
Capsule diameter (in.)	0.75	0.75	0.75	0.75
Capsule length (in.)	5.00	5.00	6.00	6.00
Beryllium block size (in.)	10 x 10.5 x 3.75	11 x 10.5 x 3.75	20 x 6.5 x 3.75	11 x 12.5 x 3.75
Fuel block size* (in.)	12 x 12.5 x 3.75	13 x 12.5 x 3.75	22 x 8.5 x 3.75	13 x 14.5 x 3.75
Fuel block weight (lb)	45.0	49.7	58.5	59.8

*Pyrolytic graphite, one inch thick is keyed into the four small faces of the beryllium block.

- (2) The thermionic system can be protected from superorbital re-entry by:
 - (a) One-module design--encapsulating the module in approximately 1.2 inches of beryllium over all surfaces.
 - (b) Two-module design--encapsulating the modules in approximately 1.3 inches of beryllium over all surfaces.
 - (c) Four-module design--encapsulating the modules in approximately 1.6 inches of beryllium over all surfaces.

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Since aerodynamic heating theory at superorbital velocities and the ablation characteristics of the ablator are not clearly defined, the results of the study are not to be considered as absolute, but as an indication of the feasibility of intact re-entry design.

2. Land Impact

An abort during the ascent phase of the trajectory would result in a high velocity impact of the fuel containment structure onto earth. The probability of land impact is much less than that of water impact, with the exception of aborts very close to the launch pad. However, this does not reduce the stringent requirement of the fuel structure surviving land impact. For evaluating structure integrity, the most rigid condition of impact was assumed to exist, that of impact at terminal velocity on a hard media, such as granite. Analysis shows that the fuel structure, if re-entering with the flat surface projected, will have a terminal velocity of approximately 160 fps: an end-on re-entry gives a velocity of approximately 450 fps. Terminal velocity calculations for the single fuel capsule give: flat side--160 fps; tumbling--210 fps; and end-on condition--430 fps. In the evaluation, it was assumed that the capsule impacted at its maximum velocity of 450 fps. Since the capsules contain an end weld, it was assumed that the weld was the most critical and could withstand only 50% of the parent material tensile strength at capsule operating temperatures. The impact pressure was determined to be 11,900 psi. Considering only 50% of the Haynes 25 0.060-inch wall thickness, the capsule could withstand a pressure of 14,800 psi at operating temperature. In addition to the Haynes 25, other components that offer structural support are the inner capsule liners of 0.090 inch of TZM (titanium, zirconium, molybdenum) and 0.010-inch tantalum. Also, the cooling of the capsule in the lower atmosphere will decrease the surface temperature to less than the 1370° F assumed in the calculations. These extremely conservative calculations indicate that the integrity of the capsule is retained under conditions of land impact. Verification of capsule integrity must be substantiated by an impact test program.

Some impact tests have been performed under the SNAP 9A program where fuel capsules have fallen at terminal velocity and impacted on the desert. The capsules, upon impact on the surface of the desert, failed to bury themselves or lose containment integrity. There is, however, a high probability that if the total fuel block becomes buried below a depth of 12 inches, the increase in temperature will melt the capsules and the fuel will flow into the surrounding soil. Once the GdPo fuel is sufficiently dispersed it will solidify and basically remain at the point of solidification since the material is quite inert. The dose rate at one meter from this source (assuming six inches below ground) would be approximately 10 rem and would become less as the GdPo fuel decays.

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Impact considerations for the thermionic system are more difficult to analyze as the outer structure of Haynes 25 is designed to absorb the impact energy. However, its fuel containment integrity must be established through actual testing of the system.

3. Water Impact

There is a higher probability of water impact if an abort occurs, since the ascent trajectory will be primarily over water during the launch phase.

The system might impact in shallow offshore water. In this case, the mechanical forces imposed on the containment structure would be less severe than in the case of land impact. The excess heat of the system would be removed by the water, thus preventing any meltdown. Impact in deep downrange water is the most probable.

The problems to be considered are external hydrostatic pressure due to the depths and seawater corrosion. The fuel capsules will be intact within the block since the block is designed to preclude burnup during re-entry.

The capsule temperature will decrease after immersion in seawater due to the increased heat transfer coefficient and will rapidly approach the seawater temperature. Neglecting the internal gas pressure due to helium buildup, the external pressure calculated to cause collapsing of the Haynes 25 alloy outer cladding was 15,400 psi. To this value could be added the pressure strength of the TZM and tantalum inner liners. The abyssal depth of the major portion of the ocean reaches to 2150 fathoms for an external hydrostatic pressure value of about 5600 psi. The individual fuel capsules, without benefit of the TZM inner liner, can withstand the pressure exerted by any possible ocean depth without collapsing and exposing the radioactive fuel.

Corrosion tests by the Martin Company on container material in seawater gave a corrosion rate of less than 0.1 mil per day for Haynes 25. The refractories, such as tantalum and TZM, exhibit excellent corrosion resistance, usually less than a mil per year in almost any media. The Haynes 25 should maintain integrity for at least two years, greater than five half lives of decay for the polonium and, thus, decrease the activity per fuel capsule to less than 480 curies. Many additional years of protection will be provided by the 90-mil TZM and 10-mil tantalum liners. After five years the polonium will have decayed four orders of magnitude, with a resulting activity considered to be insignificant. After the integrity of the capsule has been violated, the release of polonium to the seawater, should any remain, will be limited since the GdPo is a solid compound with a finite rate of solution and because the capsule leak rate will be controlled by the cross sectional area of the hole or crevice in the

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corroded capsule wall. When a material is introduced into the ocean, its dispersion is a function of depth. The upper mixed region may extend from a few meters to approximately 200 meters in depth and is affected by winds, tides and the currents which promote both vertical and horizontal mixing. The thermocline or pynocline layer exists immediately below the mixed region and extends to a depth of 400 to 500 meters, depending on season and location. The isotope, if an abort occurs, will probably impact in the deep sea region known as the abyssal plain at a depth of greater than 2000 fathoms. Dilution occurs by horizontal diffusion in the deep sea area (vertical diffusion is considered to be only two to five meters per year), thus the fuel will not reach the surface for many years.

Current AEC practice for deep sea burial is to compound the isotope into a reasonably insoluble form, place it in a corrosion-resistant container and immerse the package at a depth of at least 1000 fathoms. The Apollo fuel satisfies all these requirements for the most probable abort occurrence.

The good corrosion resistance of the Haynes 25 and TZM capsule material, combined with the deep seawater, precludes the radioisotope being ingested directly or indirectly by the consumption of contaminated sea life.

To illustrate the conservatism of deep sea disposal, it was assumed that 30% of the total activity at the end of two years (4300 curies) was released instantaneously. Calculations, based upon maximum permissible concentration (MPC) and intake through ingestion of 1500 grams per week of marine life, showed that limits would be exceeded for only eight days in a radius of up to 7000 meters of the release point. A release at the end of five years would be negligible.

E. LUNAR EXPLORATION PHASE

The lunar exploration phase is defined as the period from spacecraft injection into an earth-moon trajectory and the jettisoning of the service module after the proper earth re-entry trajectory of the command module has been established. Lunar exploration consists of three distinct tasks: earth-moon flight; lunar orbit; and moon-earth flight. The major nuclear safety considerations associated with the use of a nuclear electric power system on the Apollo manned lunar exploration during these tasks are:

- (1) Elimination of all direct radiobiological hazards to the astronauts.
- (2) Minimization of the total dose received by the astronauts.

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Fig. 8. Decay of Polonium-210--696,000 Curies

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F. POSTLUNAR PHASE NUCLEAR SAFETY

The postlunar 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. The feasibility of integrating a nuclear power system into the manned exploration of lunar space necessitates that the postmission safety problem areas be investigated. The radiobiological hazards associated with the use of isotopes during the mission can only be postulated as a result of an improbable series of events or failures. Postmission safety consideration does not necessarily depend upon random events and, therefore, must be precisely planned and implemented accordingly. Hence, the presence of astronauts in this respect are an asset rather than a liability.

Several safe disposal methods for the postlunar phase are applicable and can be integrated in the design of the power system as more information concerning the postmission re-entry characteristics of the service module are known. Therefore, general postlunar safety considerations of several disposal methods are discussed.

1. Service Module--Earth Orbit Abandonment

Earth orbital abandonment of the service module containing the nuclear power system is probably the most desirable, because this method presents little or no hazard from a radiological viewpoint. The degree of effectiveness that can be achieved by this technique, however, depends upon the orbital lifetime of the system which is a function of its orbital altitude and its ballistic coefficient ($W/C_d A$). If both of these parameters are favorable, the radioisotope will decay to an insignificant level, as illustrated in Fig. 8. For example, a five-year orbital lifetime would reduce the radioactivity from one megacurie to approximately 100 curies--a factor of 10^4 . At end of its orbital lifetime the service module will undergo normal orbital decay and either burn up or re-enter intact according to its design. Since re-entry would be random, the radioisotope containment structure could impact on land or water, although probability favors the latter.

The safety of this method is totally dependent upon the orbital lifetime of the service module. Figure 9 shows generalized curves of orbital lifetime as a function of altitude and ballistic coefficient. The curves were obtained from satellite lifetime data (Ref. 11).

2. Service Module Re-entry at End of Mission

If the service module re-enters the earth's atmosphere immediately after jettison from the command module, the service module containing

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Fig. 9. Orbital Lifetime Versus Orbital Altitude

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the FCS will follow a similar trajectory to the command module, with the possible exception of skip-out. The containment structure would, therefore, impact the earth's surface in a semicontrolled manner, probably a water impact. If the impact event occurred in the ocean, there would be very little, if any, possible radiological hazard to the general populace resulting from this re-entry mode. Because the fuel containment structure is designed such that aerodynamic forces do not degrade the integrity of the structure, the corrosion resistance capability of the structure is unaffected. Since the radioisotope is encapsulated in a highly corrosion-resistant material, the radioisotope will be contained for a period sufficiently long to reduce the activity to acceptable release values. These conditions are discussed in the Launch Safety Phase in this report.

If impact should occur on land, the radioisotope would remain in its encapsulation which is designed to survive terminal velocity impact (see Launch Phase). There does exist some question as to the accountability of the fuel for a land impact, however, as more information concerning the specific characteristics of trajectory of the jettisoned service module is known, the accountability of the radioisotope fuel containment structure can be better defined.

The basic conclusion that can be drawn with respect to the postlunar mission 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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III. SPACECRAFT NUCLEAR RADIATION ANALYSIS

A. REQUIREMENTS

Spacecraft radiation environments resulting from the integration of a radioisotopic powered system were analyzed with respect to:

- (1) The total integrated radiation dose received by the astronauts.
- (2) The radiobiological shielding requirements to meet radiation design criteria for manned space exploration.

The radiation design criteria were based on the average yearly dose of 54 rem as established by NASA for the Apollo lunar mission (Ref. 1) and the total integrated radiation dose of 22 rem from space radiation for a 14-day lunar mission (Ref. 2). From these guidelines, a nominal radiation criterion was established--the permissible total integrated dose from the radioisotope power supply (RIPS) to the astronauts shall be less than 30 rem. The allowable dose rate to the astronauts corresponding to the 30 rem design dose over a 14-day mission is 89.3 millirem/hr from the nuclear system.

B. INTEGRATION FACTORS

Integration features which were considered in the evaluation of radiation dose and shielding requirements for the conceptual designs of the nuclear systems are:

- (1) The location and orientation of the nuclear system in the spacecraft.
- (2) The effectiveness of the command module ablator and its structure as shielding.
- (3) The crew exposure time.

The nuclear systems are located at a maximum distance from the command module to take full advantage of the $1/R^2$ attenuation afforded by geometry. Orientation of the nuclear system heat source in the spacecraft was selected primarily to yield the lowest radiation levels to the crew area in the command module. The spacecraft and associated equipment provide some protection to the crew from the nuclear source. The maximum shielding afforded by the structure in both the command and service modules was not evaluated; however, a minimum shielding effectiveness was determined based on the ablator and its associated structure in the command module re-entry surface. Consideration of this material amounted to a dose rate reduction factor of 0.8.

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An integration feature which was not considered in the radiation analysis is the optimization accrued by positioning individual heat sources to take advantage of attenuation afforded by other heat sources and/or system components. In this analysis, the dose rate contribution in the command module from each heat source was considered unaffected by any of the other units in the system.

C. ANALYSIS

Beginning of life thermal power requirements were used in the analysis, thus providing an additional conservatism. The end of life thermal fluxes are used in the text as a means of identifying the various systems.

The types of radiation associated with Po-210 are gamma photons, alpha particles and neutrons produced by alpha particle reactions with the low Z material impurities in the fuel. The alpha radiation presents no radiation problem since it is completely absorbed within the nuclear fuel block.

1. Summary

Each nuclear system design studied meets the maximum radiation design dose of 30 rem without the use of radiation shielding. Table 4 presents the neutron and gamma dose rate contributions as well as the total dose rate and dose to the astronauts in the command module from the unshielded nuclear systems. Table 5 contains the shielding requirements necessary for each nuclear system to meet the minimum practical radiation design dose of 3 rem. The results presented in Table 4 show that the dose to the astronauts over a 14-day mission from the unshielded nuclear systems range from approximately 14 to 16 rem from the dynamic systems to about 20 to 27 rem from the static systems. All values are below the 30-rem constraint established as the maximum permissible dose from a nuclear power system. On this basis, the nuclear power system can be integrated into the spacecraft without shielding and presents no radiobiological hazard to the astronauts.

2. Design Description

The configurations of the various nuclear systems located in the aft end of the service module of the spacecraft are shown in Figs. 10 and 11. Also shown in these figures are the orientations of the various nuclear systems and the location of shielding necessary to satisfy the minimum design dose of 3 rem.

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TABLE 4
Unshielded Dose Rates* at Center of Crew Area due to Nuclear Heat Source
in Command Module
(14-day mission)

<u>System</u>	Unshielded Dose Rate (mrem/hr)			<u>Unshielded Dose (rem)</u>
	<u>Gammas</u>	<u>Neutrons</u>	<u>Total</u>	
Dynamic				
Brayton cycle	40	0.8	40.8	13.7
Rankine cycle (mercury working fluid)	40	0.8	40.8	13.7
Stirling cycle	48	0.75	48.75	16.4
Rankine cycle (Dowtherm-A working fluid)	60	0.93	60.93	20.5
Thermionic				
1 unit	60/unit	1.2/unit	61.2/unit	20.6 all units
2 units	35/unit	0.6/unit	35.6/unit	23.9 all units
4 units	20/unit	0.3/unit	20.3/unit	27.3 all units
Thermoelectric				
3 units	17/unit	0.3/unit	17.3/unit	17.4 all units

*Shielding effectiveness of spacecraft included using a minimum dose rate reduction factor of 0.8.

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TABLE 5
 Shielding Requirements for Various Nuclear Systems
 (radiation design dose = 3.0 rem over 14-day mission)

<u>System</u>	<u>Beginning of Life Thermal Power (watt)</u>	<u>Depleted Uranium Maximum Thickness (in.)</u>	<u>Total Shield Weight (lb)</u>
Dynamic			
Brayton cycle	10,200	0.45	66
Rankine cycle (mercury working fluid)	15,300	0.45	73
Stirling cycle	9,690	0.50	69
Rankine cycle (Dowtherm-A working fluid)	13,000	0.56	76
Thermionic			
1 unit at 20,000 watts(t)	20,000	0.57	102
2 units at 10,000 watts(t)	20,000	0.60	176
4 units at 5000 watts(t)	20,000	0.60	170
Thermoelectric			
3 units at 7500 watts(t)	22,500	0.52	124

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The three configurations of thermionic systems considered in the radiation study are:

- (1) A single unit producing the total requirement of 20,000 watts of thermal power at the end of life.
- (2) Two units each producing one-half of the total thermal power requirement.
- (3) Four units each producing one-quarter of the total thermal power requirement.

The location of the single unit heat source in the spacecraft is shown in Fig. 10. All units are cylindrical in shape. The positioning of the two- and four-unit designs in the spacecraft are:

Two units--each unit is located at the aft end of the spacecraft, at two positions typified by the single unit.

Four units--two units are placed adjacent end-to-end and are located at the aft end of the spacecraft at two positions typified by the single unit.

Figure 12 presents the unshielded dose rates from each of the nuclear systems. This figure shows the variation of dose rates with increasing distance from the nuclear source. The resultant unshielded radiation levels in the command module are tabulated in Table 4 for each of the nuclear systems considered. Table 5 presents the shielding requirements necessary to reduce the unshielded radiation dosages to the minimum design dose level of 3 rem. The thickness of depleted uranium was obtained from Fig. 13. Figures 10 and 11 show the real dimensions of the shields for each of the systems.

D. ANALYTICAL METHOD

In the evaluation of the radiation dose rates from the nuclear systems, the various heat source configurations were approximated with cylindrical geometry. The fuel, its encapsulation and containment were homogenized and treated as a uniform density source region. The fuel block and all associated materials were included by adapting the system materials and geometry to the material and geometric requirements of the calculation techniques.

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Fig. 10. Schematics of Shield Configurations for Thermionic and Thermoelectric Heat Source Designs

Fig. 11. Schematics of Shield Configurations for Stirling and Brayton Cycle Heat Source Designs

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Fig. 12. Radiation Dose Rate from Various Nuclear Systems as Function of Distance Along Centerline of Spacecraft

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Fig. 13. Attenuation Factor for the 0.8-Mev Gamma from the Po-210 Fuel as a Function of Depleted Uranium Thickness

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Radiation source strengths of the homogenized fuel and dose rates from the various nuclear systems were determined, based on the thermal power requirements and the following nuclear characteristics of the Po-210 radioactive fuel.

<u>Dimension</u>	<u>Quantity</u>
Watts/curie	0.031456
Photons/disintegration	1.2×10^{-5}
Disintegrations/sec-curie	3.7×10^{10}
Mev/photon	0.8
Neutrons/sec-curie	100.0

The basic data necessary for the calculation of radiation dose rates were taken from Refs. 10 and 12 through 15. The type of data required included gamma linear attenuation coefficients, gamma dose buildup factors and neutron cross sections.

The calculations of dose rate were performed using the IBM-7094 program SPEND (Shield Penetration and Energy Deposition) which utilizes cylindrical geometry and analytical shielding approximations (point kernel techniques); and numerically integrates over the volume of the source to the receiver point to obtain the radiation dose rate. In this program, the nuclear data on each material in the configuration is used with the geometry routine to compute the total number of relaxation lengths between each source element and receiver point. Then the material attenuation and buildup factors (if applicable) are evaluated and geometric attenuation is included according to the inverse square law. The energy flux is computed at the receiver point and the appropriate flux-to-dose rate conversion factor is applied.

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IV. FUEL AVAILABILITY

A. GENERAL

Eight radionuclides, presently being produced, can be considered for evaluation as isotopic power heat sources. The heat source provides the required thermal energy output necessary to operate a conversion device in the production of electrical energy. These isotopes can be grouped into the categories of alpha emitters and reclaimed fission products.

The alpha emitters are the primary isotopes that have been receiving major attention for use on manned space missions. This is due mainly to the minimum shielding requirements for both launch and orbit operations, a less serious re-entry and safety problem and fuel availability and weight.

The fission products, although considerably cheaper in most cases since they are a reactor waste product, are at somewhat of a disadvantage for a manned mission due to the beta- and, in some instances, gamma-emitting qualities that require considerable biological shielding. However, in some instances such as unmanned missions and terrestrial applications where the requirements are not totally dependent upon shielding weight, specific activity and power density, the fission product isotopes are quite competitive with the alpha emitters.

1. Fuel Selection Criteria

The basic criteria for the selection of a fuel for space application, independent of cost per thermal watt, are discussed as follows:

a. Isotope half life

A short half life isotope limits the useful mission life. For example, isotopes such as curium-242 and polonium-210 are limited to six months. Plutonium-238, with a half life of nearly 90 years, and strontium-90, at 28 years, have a useful life of greater than 10 years and are ideal for deep space probes or remote land applications. A reasonably long half life in relation to length of mission precludes the use of power flattening schemes and optimizes the initial fuel loading.

b. Specific power

The specific power, thermal watts per gram of fuel, is of prime importance when light weight and high power density is of concern. Specific powers vary from 0.07 watt/gm for cesium-137 to 140 watt/gm in the case of polonium-210 in the metal form. Low power densities are

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more adaptable to thermoelectric systems than to thermionics due to the lower temperatures that can be attained as a result of the heat losses per unit volume of fuel.

The alpha emitters have a higher specific activity (watts per curie), than the fission products as is the case with the weight density.

c. Radiation levels

It follows that the minimum weight of the overall device is realized if little or no shielding is required. The fission product isotopes require considerable shielding for a manned mission and must also be considered in the ground handling aspects of an unmanned mission. The hard gamma radiation resulting from the fission products must be evaluated in terms of the electronic equipment and radiation-hardened circuitry of the vehicle. The alpha-emitting isotopes require minor shielding relative to the fission isotopes although a minor amount of neutrons may be present from spontaneous fissions or impurities introduced in the fuel during the production process.

d. High melting point

A high melting point permits the fuel to remain in the solid form at all times. This gives a uniform heat distribution per unit area and, essentially, a constant thermal conductivity for heat transfer considerations at operational conditions. A solid fuel permits the inclusion of a constant and known void volume for helium pressure buildup in the case of the alpha emitters. Solid fuel is easier to handle and fabricate without having the attendant problems of maintaining a low temperature or preventing volatilization.

e. Containment and fuel form

In many cases it is desirable to have the fuel compounded to increase the melting point, to adjust the specific power and reduce the operating pressure. Strontium has been prepared in both the titanate and oxide form. Polonium has been compounded with the rare earths and most of the other isotopes are available as the oxides. Plutonium and polonium have also been used as metals.

In general, fuels at high temperature in the solid form are more compatible with containment materials than are the molten fuels, thus minimizing any corrosion problems.

f. Availability

Availability, in sufficient quantities within reasonable production periods and consistent with the time intervals between a series of mission launches, is of extreme importance. It is not desirable to require

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the installation of expensive new facilities at an accelerated rate for a specific mission nor to install a capacity for a given production rate that will not be utilized for subsequent missions. For short half life isotopes, the production period should be as short as possible to minimize the cost on a per curie basis at time of launch. As a general rule, the time to produce a given fuel loading should not exceed one half life of the isotope.

2. USAEC Current and Planned Production

Most radionuclides suitable for processing into isotopic heat sources are produced as a byproduct in uranium-fueled reactors. The exception is polonium-210 which is produced by irradiation of bismuth. The potential demand for isotopes has caused the USAEC to plan for increased production.

One major problem normally confronts the AEC when large quantities of an isotope are required which are not available without expanding capacity. The user is somewhat reluctant to commit missions based on quantities of fuel that exceed production and in turn, the USAEC cannot justify additional investments to expand production without having an official request.

Table 6 presents the most recent information available on projected annual fuel production and cost. This information has been obtained from personal communications with Division of Isotopes Development personnel in December 1963 and January 1964.

3. Fuel Properties

In the criteria for selection, it was stated that specific activity, specific power, power density and melting point were major considerations. State of the art in terms of fuel chemical form and the half life of the isotope are of importance. Table 7 lists the major properties of the eight isotopes being evaluated.

4. Fuel Selection

The mission duration is a period of 14 days. Allowing 30 days from time of fuel encapsulation through launch gives a fuel life requirement of 44 days, thus allowing the use of a short half life isotope without major problems involving power flattening.

From 1500 to 2000 watts electrical, depending on the power conversion cycle, is required throughout the mission. This electrical output, a function of the overall cycle efficiency, then requires a thermal output of the isotope in the range of 10 to 20 kilowatts. As-

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TABLE 6
Availability of Isotopic Power Fuels
Present and Projected^(1, 2)

Isotope	Decay Product	Half Life	Present	Annual Production (kw(t)/yr)		Projected Cost with Expanded Capacity (\$/curie or gram)
				Planned	Could Be ⁽³⁾	
Sr-90	Beta	28 yr	19.5	33	66 ⁽⁴⁾	0.13/c
Cs-137	Beta-gamma	27 yr	4.8	16.8	48 ⁽⁴⁾	0.10/c
Ce-144	Beta-gamma	285 d	As needed	27.7	800 ⁽⁴⁾	0.007/c
Pm-147	Beta	2.52 yr	0.1	0.2	12 ⁽⁴⁾	0.033/c
Po-210	Alpha	138 d	7	40 to 70 ⁽⁵⁾	140 to 1440 ⁽⁶⁾	0.45 to 1.13/c ⁽⁷⁾ 2000 to 5000/gm
Cm-242	Alpha-neutron	162 d	12	--	120 ⁽⁸⁾	5.43/c 12,700/gm
Cm-244	Alpha-neutron	18.4 yr	0.5	35 ⁽⁹⁾	-- ⁽¹⁰⁾	2.62 to 3.50/c 1500 to 2000/gm

NOTES:

- (1) (2) Private communications with DID personnel, 12/16/63 and 1/8/64.
- (3) AEC requires official request for increased production to justify expanded capacity.
- (4) Requires that AEC construct isotope plant at Hanford (HIP). If authorized, HIP could start operations in 1/68.
- (5) Will be available FY 66. Processing performed at Mound Laboratory.
- (6) 140-kilowatts could be available from Savannah River with a two-year lead time (early FY 67). Larger quantities, if demands justify, could be available by FY 68 to 69 at HIP.
- (7) Price in orbit allowing 60% decay from production. \$1.5 to 4.5 million for plant modification, depending on location, is included in price.
- (8) Could be available during 68.
- (9) First one-kilowatt production in 67.
- (10) Many kilowatts could be available. Requires six-year lead time and large capital investment.

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TABLE 7
Properties of Isotopic Fuels

<u>Isotope</u>	<u>Fuel Form</u>	<u>Melting Point (°C)</u>	<u>Power Density (watt/cc)</u>	<u>Specific Power (watt/gm)</u>	<u>Weight Density (gm/cc)</u>	<u>Specific Activity (watt/kcurie)</u>	<u>Half Life</u>
Sr-90	SrO ₂	2430	0.93	0.35	2.65	6.5	28 yr
Cs-137	Glass	--	0.22	0.07	3.1	4.8	27 yr
Ce-144	CeO ₂	2680	13.8	2.3	6.0	7.9	285 d
Pm-147	Pm ₂ O ₃	2300	1.5	0.27	5.55	0.37	2.52 yr
Po-210	GdPo	1630	247 ⁽²⁾	82.4	3.0	31.7	138 d
Pu-238	PuO ₂	2400	4.38	0.384	11.4 ⁽¹⁾	34.5	89.6 yr
Cm-242	CmO ₂	2250	1050 ⁽³⁾	95	11.0	36.0	162 d
Cm-244	CmO ₂	2250	24	2.42	9.9	35.0	18.4 yr

NOTES:

- (1) PuO₂ crystalline density 11.4 gm/cc; high temperature sintered density 10 gm/cc.
- (2) Gadolinium polonide powder. Power density = 247 watts/cc. Can be further diluted to adjust power to 100 to 150 watts/cc.
- (3) Further dilution required.

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suming that three missions per year could develop, it is desirable that the fuel be produced over a period not greater than four months production time. This is also true from an economic standpoint. This would require an annual fuel production of 30 to 60 kilowatts.

For this specific application, polonium-210 is selected as it more nearly fulfills the requirements of the manned mission than any other isotope. It has the following advantages:

- (1) High power density--the GdPo power density of 247 watts/cc can be adjusted by dilution to the optimum value for heat transfer considerations.
- (2) Low radiation values--polonium is an alpha emitter with low radiation, thus minimizing shielding requirements.
- (3) Fuel form and compatibility--it is easily compounded with gadolinium and has been proven compatible with containment materials.
- (4) High melting point--gadolinium polonide has a melting point of approximately 2900° F and this assures that the fuel will remain solid.
- (5) Low fuel weight--the total fuel loading of any system will be less than one pound.
- (6) Minimum safety problems--fuel is inert, solid, low radiation and can be contained.
- (7) Established production technology--Mound Laboratory, operated for the USAEC, has long-term production experience in the wet process.
- (8) Current development programs--Mound Laboratory continues to improve the state of the art through USAEC-sponsored development programs on production processes, fuel form compounds, encapsulation and physical properties determinations.
- (9) Plans for increased production--installed capacity will be increased ten-fold by fiscal 1966 thus reducing costs.
- (10) Availability--polonium is the only isotope that can be available in sufficient quantities with a two-year lead time.

A detailed discussion on the technology of polonium may be found in the Appendix. Included are nuclear and atomic properties, production techniques, fuel form characteristics, fuel form containment and a production schedule of each operation.

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B. FUEL REQUIREMENTS

1. Power Cycle Requirements

Six power conversion systems are being considered for use with polonium-210 as the isotopic heat source. Each has a different overall system efficiency, operating temperature, heat transfer media and electrical characteristics to the power conditioner. The fuel requirements, in terms of thermal watts and the quantity of fuel required at time of encapsulation are given in Table 8.

2. Fuel Cycle Evaluation

Starting July 1, 1965 the polonium-210 production capacity will be at an annual rate of 300 to 500 grams (40 to 70 kilowatts(t)). However, production rates might not reach this point. The production of polonium from bismuth by irradiation will be performed at Hanford and the processing operations, fuel form preparation and encapsulation will be conducted at the Mound Laboratory.

As shown in Table 9, the production of polonium could be in the range of 140 to 1400 kw/yr. This can be accomplished at either Savannah River or Hanford. The increased capacity, over that planned at Mound Laboratory and available in fiscal 1966, requires an official request to the USAEC.

In accordance with recent information obtained from the AEC Division of Isotopes Development (Ref. 16) on polonium production, Savannah River could produce polonium in annual quantities in excess of 1000 grams (140 kilowatts) with a two-year lead time after the official request is made. This means that the process could be in production by early FY 1967. Plant modifications and equipments to produce this quantity was estimated by USAEC as \$1.5 millions.

The Savannah River production would be geared to getting a stream on line with minimum time and costs involved and is of the single order, no carry-on type. Estimated cost at the time of launch is \$5000 per gram of polonium-210, equivalent to \$1.13 per curie.

Should user requirements develop over long periods for several missions, as may well be the case, and thus provide a repetitive business, the production and processing capacities would be installed at Hanford. The Hanford facility would require a greater capital investment--\$4.5 millions compared to \$1.5 millions at Savannah River, but would have a greater capacity and produce at a lower cost per curie. The USAEC estimate (Ref. 17) of cost at Hanford is \$2000/gm, equivalent to about \$0.45/curie. This compares to present estimated cost

TABLE 8
Polonium-210 Fuel Requirements per Mission

<u>Power Cycle</u>	<u>Thermo- electric</u>	<u>Thermionic</u>	<u>Stirling</u>	<u>Brayton</u>	<u>Rankine Mercury</u>	<u>Dowtherm-A Rankine</u>
Electrical power to bus ⁽¹⁾ (watts(e))	1500	1700	1500	1500	1500	1500
Overall system efficiency (%)	8.5	10.8	17.7 ⁽²⁾	15.75 ⁽²⁾	12.1	14.4
Power at time of encapsu- lation ⁽³⁾ (watt(t))	22,000	19,600	10,600	12,500	15,500	13,900
Fuel required:						
Kilocuries	696	618	334	394	488	438
Po-210 metal (gm)	154	137	74	88	109	97
GdPo fuel volume (cm ³)	147	131	71	84	104	93
GdPo fuel weight (gm) ⁽⁴⁾	441	393	213	252	312	279

NOTES:

- (1) Four fueled thermionic devices, each 500 watts(e) gross. Stirling, Brayton and Rankine cycles contain redundant 1500-watt systems with only one system fueled.
- (2) Increasing radiator area from 90 to 120 square feet increases overall efficiency of Stirling to 19.3% and Brayton to 18.4%.
- (3) Assumes 30-day period between fuel encapsulation and launch.
- (4) Based on gadolinum-polonide power density of 150 watt-cm³ and density of 3 gm/cm³.

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(Ref. 18) of \$6 per curie at Mound Laboratory. However, Hanford would require from three to four years lead time after the request for production. Table 9 summarizes the present, planned and possible production of polonium.

TABLE 9
Potential Polonium Production

<u>Fiscal Year</u>	<u>Annual Production</u> (gm) (kw(t))		<u>Status or Action</u> <u>Required</u>
1965	50	7	Available at Mound Laboratory.
1966	300 to 500	40 to 70	Will be available at Mound Laboratory.
1967	1000 to 10,000	140 to 1400	500 gm from Mound. In excess of 1000 gm from Savannah River possible with expenditure of \$1.5 million (presently on hand as equipment funds). Two-year lead time required for Savannah River.
1968 to 1969	1000 to 10,000	140 to 1400	Quantity possible from Hanford with lead time of 3 to 4 yr and expenditure of \$4.5 million for facilities.

To determine the quantity of fuel that could be made available at any given time, an equation can be derived in terms of annual production rates as follows:

$$Q_t = P \left(\frac{T}{0.693} \right) \left[1 - \exp^{-0.693 t/T} \right]$$

where

Q_t = amount at time (t) (grams)

T = half life of isotope (138.4 days)

t = time (days production)

P = production rate (gm/day)

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Table 10 presents the quantity accumulated as a function of days production starting at time zero with none on hand. The quantity is based on annual production rates of 300, 500 and 1000 grams (the lower and upper range of Mound Laboratory and that possible with a two-year lead time at Savannah River). Table 10 lists relative cost per curie, defined as the total production over a given period divided by quantity accumulated at the end of the production period. This shows that cost increases sharply due to the short half life; for example the cost per curie accumulated at 180 days is 22.3% greater than the 90-day cost.

TABLE 10
Quantity of Polonium Available as Function of
Time and Production Rate

Production (days)	Quantity Accumulated			Relative Cost per Curie (\$)
	300 gm/yr	500 gm/yr	1000 gm/yr	
0	0	0	0	1.00
30	23.0	38.4	77	1.07
60	42.7	71.0	142	1.17
90	59.7	99.5	199	1.25
120	74.0	123.2	246	1.35
150	87.7	146.0	292	1.44
180	97.6	162.5	325	1.53
210	107.0	178.0	356	1.63
240	115.0	191.5	383	1.74
270	122.0	203.0	406	1.84
300	128.0	213.0	426	1.95
330	133.0	221.5	443	2.07
360	137.2	228.5	457	2.19

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Days production are plotted against grams accumulated at three production rates, in Fig. 14. The figure also shows the quantity required for each of the power conversion cycles. For example, at a production rate of 300 gm/yr, the time to accumulate the quantity of fuel for the Stirling cycle requires four months. The 154 grams required for the thermoelectric system can be obtained in approximately two months at the 1000-gram annual production rate.

Table 11 presents the estimated fuel cost per load for each system based on an annual production rate of 1000 grams. Costs are given on total curies produced and at estimated Savannah River and Hanford costs (\$1.13 and \$0.45/curie, respectively).

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Fig. 14. Quantity of Polonium-210 Available as Function of Time and Production Rate

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TABLE 11
Estimated Fuel Cost per Load
(assumes 1000 gm/yr production)

	<u>Thermo- electric</u>	<u>Thermionic</u>	<u>Mercury Rankine</u>	<u>Dowtherm-A Rankine</u>	<u>Brayton</u>	<u>Stirling</u>
Production time required (mo)	2.18	1.90	1.45	1.30	1.14	0.96
Total production (gm)	181.5	158.2	120.8	108.2	95.1	80.0
Fuel load (gm)	154	137	109	97	88	74
Total produced (10^3) (curies)	802	700	535	479	420	353
Total per load (10^3) (curies)	696	618	492	438	397	334
Savannah River cost ($\$10^3$)	907	791	604	541	475	399
Hanford cost ($\$10^3$)	361	315	241	216	189	159

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V. RELIABILITY

The Apollo mission electrical power system specification for reliability is stated as 0.9986 for successful mission completion and 0.99997 for crew safety.

Although reliability analyses of the six electrical power systems is beyond the scope of this contractual effort, attention has been given to reliability in a qualitative manner except in the case of the thermoelectric system. A preliminary analysis is included for this system.

Reliability estimates and projections are fraught with a number of dangers. In electronic systems, reliability estimates have proven to be quite accurate in the light of actual experience with equipment in the laboratory and in the field. The reasons for the relatively good agreement between analytical estimates and actual experience results from:

- (1) Statistical process controls leading to predictable statistical patterns and behaviors.
- (2) Random selection from very large populations.
- (3) Large numbers per system tend to offset drift tolerance-type failures.

In dealing with mechanical systems, the problem of reliability estimating is changed to some extent. First, most mechanical systems have far fewer total number of components in comparison with electronic systems. Second, a mechanical system has only one or two of each kind of component as compared to 10 to 1000 of one kind of component in a typical electronic system. As a result of these smaller numbers, there is much less opportunity for the reliability estimate to settle out statistically as it does in an electronic system.

Of the six power systems evaluated, four are definitely of the mechanical system type. These are the Brayton, Stirling and two Rankine systems. The approach taken for each of these systems has been to adopt two separate systems employing a common heat source. Each system is capable of delivering 1500 watts(e) net. This approach to providing redundancy is considered the most practical one for meeting reliability specifications. In addition to having the inherent difficulties of most mechanical systems, demonstrating predicted reliability with reasonable confidence presents a difficult and expensive task due to environmental considerations. It should be mentioned, however, that of the four dynamic systems, the Brayton system offers the best chance for demonstrating reliability because the space environment has little effect on the feasibility of its operation. Thus, ground tests are more meaningful.

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17. Personal meeting with personnel of Department of Isotopes Development, USAEC, December 16, 1963.
18. Personal communication with Dr. G. Richard Grove, Mound Laboratory, November 12, 1963.

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APPENDIX

POLONIUM FUEL TECHNOLOGY

Polonium-210 is a toxic radioisotope and certain safety measures are necessary to be designed into the heat source, its shipping container and all phases of its handling to preclude a release of the containment to the environs of both earth and space. The following discussion covers the atomic and nuclear properties of polonium, its production and the gadolinium-polonide fuel form properties and containment.

A. ATOMIC AND NUCLEAR PROPERTIES

Polonium-210 is an alpha emitting isotope with a half life of 138.4 days (1.19232×10^7 seconds). It has an atomic number of 84 and an atomic mass of 210.049. Its decay is according to the following scheme:

B. POLONIUM PRODUCTION

Polonium-210 is produced by the irradiation of specially purified bismuth, according to the following reaction:

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The purity of the bismuth feed is of prime importance in order to reduce the quantity of radioisotope impurities present. The feed specification⁽¹⁾ for the bismuth process is:

Iron	Less than 0.1%
Silver	Less than 0.1 ppm
Sulphur	Less than 0.1%
Nitrate (insoluble)	Less than 0.1%
Bismuth	More than 99.5%

Cast bismuth slugs are canned in aluminum. The slugs are 1-1/2 inch in diameter by six inches long and contain three pounds bismuth. Irradiation at present is normally performed at Hanford. Some irradiation has been done in Idaho. The optimum period for production of polonium-210 by neutron irradiation of bismuth is approximately 146 days, the time required to reach a 50% saturation value⁽²⁾ in the Hanford operation. Higher neutron flux, for example the ETR, requires a shorter irradiation period. Prolonged irradiation is not practical since the rate of growth decreases as it approaches the saturation value. However, polonium, unlike isotopes such as plutonium, can remain in the reactor for longer periods if its removal is inconvenient at a specified time without deleterious effects on the production. Normal irradiation time can be scheduled as three to four months⁽³⁾. The yield of polonium is quite low per unit weight of bismuth (approximately 6 ppm). At present, the yield is normally 12 curies/lb of bismuth but 20 curies have been readily attained. By optimum irradiation times and improvements in the purification process, it is possible that 50 curies/lb will be available by the mission date⁽³⁾.

At present, polonium is being produced by the wet process at Mound Laboratory, this utilizes a nitric acid process to dissolve the aluminum-clad bismuth slugs. Some development work has been done on the dry

(1) Lange, R. C., "Radioactive Impurities in the Polonium Process at Mound Laboratory," MLM-1158, July 12, 1963.

(2) Mound Laboratory Progress Report, MLM-1145, November 1962.

(3) Meeting at Mound Laboratory, Dr. G. Richard Grove, et al., November 12, 1963.

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process which takes advantage of the difference in volatility of the polonium and bismuth. In this latter process, the bismuth can be reused.

The wet process at Mound Laboratory will be described⁽⁴⁾. The bismuth slugs are dissolved in nitric acid, followed by a denitration process. The solution, containing certain radioactive impurities, is passed through a bismuth powder scrub bed for the first concentration. The Po-210, Se-75 and isotopes of tellurium are deposited on the bed. The Fe-59, Co-60, Sb-124, 95% Ag-110M and 99% of the bismuth pass through the bed.

After an additional scrubbing, the second bismuth powder bed is dissolved and the polonium precipitated with stannous chloride. Additional dissolution and the addition of ammonia precipitates the polonium, tellurium and 1% of the selenium-75. This product is again dissolved and the solution acidity adjusted for the electro-deposition step in the process. The final product is electrolytically deposited on fine mesh platinum gauzes. The composition of the final polonium product is as follows:

	(%)
Polonium-210	~ 95
Tellurium	1 to 3
Selenium	~ 0.1
Oxygen	4 to 2

The 22 kilowatts(t) at encapsulation, in the case of the thermoelectric system, is equivalent to 696,000 curies (about 154 grams of polonium) prior to fuel form preparation. This quantity would be reduced to 74 grams for the Stirling cycle. Allowing for decay during the process, requires the production of about 802,000 curies. Assuming that 20 curies of polonium are produced per pound of bismuth, 10 to 30 tons of bismuth, depending on the cycle chosen, must be irradiated and processed per system launch. However, bismuth raw metal is readily available commercially and sufficient reactor space at Hanford can be scheduled to irradiate this quantity⁽³⁾.

⁽⁴⁾Mound Laboratory Progress Report, MLM-1144, October 1962.

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The polonium metal is silvery gray⁽⁵⁾, much like lead in appearance. It has a melting point of 485° F and a boiling point of 1765° F. Its density is 9.4 gm/cm³ and its coefficient of thermal expansion is 23×10^{-6} cm/cm/°C.

C. FUEL FORM CHARACTERISTICS

Recently a program was undertaken by Mound Laboratory to prepare suitable polonium fuel forms for use in space vehicles⁽⁶⁾. Rare earth metals (gadolinium, cerium, lutetium, neodymium, yttrium, etc.) were reacted with polonium in the mole ratio of two parts rare earth to three parts polonium at 1800° F and the reaction products determined. A continuation of this effort indicated that gadolinium-polonide in the one-to-one mole ratio was the best fuel form from a stability standpoint⁽³⁾. The melting point of this compound is 2900° F, well above the operating temperature of the systems with the possible exception of the 2600° F thermionic. This will assure that the fuel will be solid for heat transfer and containment considerations.

Polonium has one of the highest power densities and specific powers of the presently available isotopes. Dilution of the fuel with gadolinium permits lower power densities that can be safely handled from a thermal standpoint. The dilution of the metal with rare earth does not affect the radiation value per curie and does not significantly alter the density of the dilutant. Table A-1 presents a comparison of the fuel form characteristics.

For the power conversion system, it is required to further dilute the fuel to a power density of approximately 150 watts/cm³ to obtain the desired heat output per unit surface area of the fuel capsules and to provide sufficient void volume at nominal pressures for helium buildup.

(5) Polonium, H. V. Moyer, Editor, TID-5221, July 1956.

(6) Progress Report for August, MLM-1140, 1962.

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TABLE A-1
Polonium-210 Fuel Form Comparison⁽³⁾

<u>Form</u>	<u>Specific Power (watt/gm)</u>	<u>Power Density (watt/cm³)</u>	<u>Mr/hr/C (at 1 M)</u>	<u>Number (n/sec/c)</u>
Metal	144.0	1350	3.65×10^{-3}	70
GdPo (crystalline)*	82.4	634	3.65×10^{-3}	212**
GdPo (powder)*	82.4	247		

*GdPo crystalline density estimated at 7.7 gm/cm³

GdPo powder density estimated at 3.0 gm/cm³

Melting point ~2900° F

Specific activity 31.7 watt/K curie

**Measured from 100-curie source

The neutron value in Table A-1 of 212 n/sec/curie for the rare earth compound is higher than will be the case for any fuel used on this program. Recent technological developments in the fuel form preparation utilized tantalum as the reaction vessel⁽³⁾, thus eliminating oxygen contamination from the previously used quartz. Mound personnel are confident that the neutron level for the GdPo form will approach the value of 70 n/sec/curie reported for polonium metal.

D. FUEL FORM CONTAINMENT

The fuel must be contained at all times to preclude biological intake and contamination of the environs. It is encased in a refractory metal liner, compatible with the fuel and in turn, the liner is encapsulated in a high strength alloy. The alloy capsule must be chemically compatible with the fuel block and possess the ability to withstand any imposed conditions from the external environs. Both the liner and capsule must have high mechanical strength at high temperature to provide resistance to impact loads and to withstand chemical corrosion by seawater immersion should a launch abort occur.

The Mound Laboratory has and is presently conducting high temperature compatibility tests of GdPo with tantalum. Recent test results

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show that tantalum was not affected by the fuel for 1500 hours at 1800° F⁽³⁾. Currently, a R&D program of general capsule evaluation is under way to investigate the suitability of other refractory materials for containment.

The actual amount of polonium in terms of weight is quite small, ranging from 74 to 154 grams. After compounding with gadolinium and diluting to 150 watts/cm³, the total fuel weight will approximate 213 to 441 grams.

Alpha particles produced by the decay of the fuel pick up free electrons and produce helium gas. Provisions must be made to contain this gas within the fuel capsule without exceeding specified pressure limits.

Helium production amounts to 0.0259 cm³/curie at standard temperature and pressure when completely decayed, i.e., infinite time. A recent test on a gadolinium-polonide source at Mound Laboratory confirmed this theoretical value by experimentally measuring a pressure corresponding to 97% of the calculated helium buildup pressure. The mass of the helium generated and the temperature of the fuel vary with time as a function of the decay constant of polonium-210. The mass increases and the temperature decreases in time.

E. FUEL SEQUENCE AND SCHEDULE

1. Fuel Process Flow Diagram

A block diagram of the entire fuel sequence from the casting of bismuth slugs through launch and possibly the recovery and reprocessing is shown in Fig. A-1.

The bismuth metal is cast into slugs and canned. From 10 to 30 tons of bismuth is shipped to the reactor site and loaded into the reactor for irradiation. After irradiation, the slugs are cooled for several days and then shipped to the processing facility for decanning and processing of the polonium. The decanned bismuth is dissolved, passed to the concentration operation, thence to the process operation and eventually to the electrolysis operation (based on the wet process). The gadolinium-polonide is prepared and diluted to the required power density, compacted, loaded into fuel capsules and welded. The individual capsules are smear tested, assayed and loaded into the fuel block assembly.

The assembled fuel block is then loaded into a shielded shipping cask and sent to the launch site for system checkout and loading aboard the launch vehicle. After completion of prelaunch operations and checkout tests, the fuel is launched to spend 14 days in orbit. Should the spent fuel be recovered, it will be returned to the processing facility for dissolution and reprocessing of the residual polonium.

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Fig. A-1. Fuel Process Flow Diagram

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2. Estimated Time Span for One Fuel Load

[illegible]

This schedule represents the estimated span time for one fuel loading. Items 3 through 6 would be repeated for each subsequent fuel load. Assuming that the first fuel load is needed for an April 1, 1967 launch date, the initial bismuth irradiation must be initiated no later than July 1, 1966. The canning of bismuth slugs is expected to require three months so it must be initiated no later than April 1966. Bismuth procurement orders must be placed by January 1966 to enable meeting an April 1967 launch for a total procurement-to-launch-span-time of 15 months.

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