Technical Memorandum No. 33-180

Heat-Sterilizable Power Source Study for Advanced Mariner Missions

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A limited study was performed to evaluate heat-sterilizable power sources for the *Mariner* Mars landing capsule for missions in 1969 and 1971. It was found that a radioisotope thermoelectric or thermionic generator (RTG) in conjunction with a small sterilizable battery—or a dynamic system such as a hydrazine-turboalternator system—would meet the mission requirements. For a power level of 700 w and a demand time of 60 min, these systems may be built at a weight of about 20-25 lb.

Presently, sterilizable batteries with specific energies of 15 whr/lb are too heavy for the mission. Uncertainties in the development of sterilizable batteries with specific energies of 50 whr/lb are such that additional systems should also be considered.

A more detailed study is planned for both the RTG and hydrazine-turboalternator systems, and will include the problems of integrating these systems into the landing capsule. Author

I. INTRODUCTION

The missions assumed for this study are the advanced *Mariner* Mars missions in 1969 and 1971. The missions are planetary flybys with a bus carrying a landing capsule. The flight time before landing capsule separation is about 200 days. At capsule separation it is assumed that there is a requirement for 700 w of raw power for direct communication to Earth for about 10 min. Since the landing capsule enters the Mars' atmosphere 10 to 15 hr after separation from the bus, it is assumed that another power demand of 700 w for 10 min is required for communication. After the capsule has landed, collected data would be transmitted to Earth at 10-to 15-hr intervals or as long as power is available (or at least up to 60 hr after landing). This, therefore, requires at least six transmissions back to Earth or a total energy of at least 700 whr.

The mission power requirement is based on the following rough estimate:

Transmitter power output (*RF* power)	100 w
Transmitter power input (regulated power)	450 w
Regulation efficiency	~65%
Required raw power	700 w
Voltage (transmitter)	1800-2200 v (dc)
Voltage (power supply)	25-50 v (dc)
Duration	10-250 min

In order to avoid a biological contamination of Mars, it is current policy that all equipment on the landing capsule be sterilized at 300°F for about 36 hr. Another severe requirement is the resistance to high shock loads upon landing (possibly as high as 2000 to 5000 g). This requirement, however, may be relaxed if high speed drogues and low speed parachutes are used for terminal descent. A summary of system weights vs number of power cycles for various power systems is shown in Fig. 1.

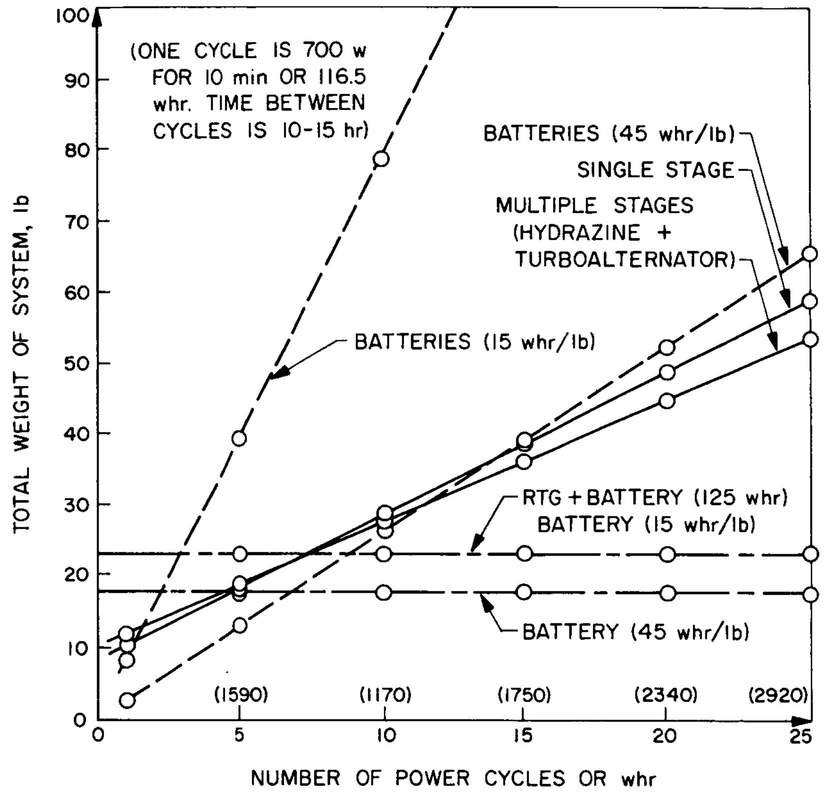


Fig. 1. Weight vs power cycles for various landing capsule power systems

II. POWER SOURCES FOR LANDING CAPSULE

The following power sources were evaluated for use with a landing capsule for a Mars mission:

1. Batteries
2. Solar panel arrays (photovoltaic cells)
3. Fuel cells
4. Radioisotope thermoelectric or thermionic generators
5. Solar thermoelectric or thermionic generators
6. Dynamic power systems

Some of the most important requirements for the landing capsule power-source are:

1. It must be sterilizable and light weight
2. It must withstand a high shock load at landing
3. It must operate reliably after storage for 200 days (flight time)
4. It must operate in the environment of Mars
5. It must operate in shadow and sunlight
6. It must not interfere with the science experiments and the other subsystems onboard the landing capsule

The above requirements rule out the use of solar-powered systems. The length of storage time and the requirement of sterilizing also eliminate fuel cells and probably batteries as sole power sources. Since the power system must be stored for long periods of time, chemical systems requiring storage tanks (such as fuel cells) are not desirable. With the best of available insulation and cryogenic storage, the weight of such systems is very great. No RTG unit with a power output of 700 w has yet been built. If such a unit were built for a 1969 mission, its weight would probably be at least 140 lb. However, in combination with a rechargeable battery, an RTG of 10-to 15-w power output may meet the requirement of low weight if the period between power demands is about 10 to 15 hr. Such a system would have a long service life and be limited only by the number of charge-discharge cycles of the battery. Of the dynamic systems, the hydrazine-turboalternator power system appears to be best suited for the mission. The hydrazine fuel can be stored for a 300-day period and the unit can be started and stopped by remote control or by a timer.

A. Batteries

Batteries have been used extensively on the early satellites. For short-time low power, they represent a ready solution to the problem of providing electrical energy in space. For space applications, only sealed batteries such as the nickel-cadmium (Ni-Cd), the silver oxide cadmium (Ag-Cd), and the silver oxide-zinc (Ag-Zn) batteries are used. The outstanding characteristic of the Ni-Cd and Ag-Cd batteries is their capability of withstanding large numbers of charge-discharge cycles, while the Ag-Zn battery has the ability to store a large amount of energy in a relatively small volume and weight. The Ag-Zn battery is limited to about 50 charge-discharge cycles at 50°C and 65% discharge (Ref. 1). Lab experiments have shown several hundred cycles at lower temperatures.

The main problem with batteries is the mission requirement for sterilization. Preliminary data indicate that a heat-sterilizable Ag-Zn battery may have a specific energy of only 10-15 whr/lb. This has already been achieved with sterilized batteries in an informed condition. Recent information from General Electric, however, indicates that *discharged* Ni-Cd batteries have been heat-sterilized without any apparent degradation in specific energy. Another problem for batteries is the mission duration of about 200 days. To avoid self-discharge during storage, the battery may be stored in a dry condition and activated by a mechanism when power is required (subsequent charging would be necessary). However, the addition of an activating mechanism increases the weight significantly. JPL is sponsoring work which is expected to yield approximately 50 whr/lb.

Information was received during a recent meeting at the NASA-Lewis Research Center about a "thermal battery" which is under development. In this concept, the electrolyte is separated from the electrodes by a thermal barrier with a built-in heating element. When power is demanded, the heating element melts the barrier and the chemical reaction can take place. Such a battery can be stored for a long period and may be heat-sterilizable without high degradation. The barrier can be made of a material that melts above the sterilization temperature.

B. Solar-Powered Systems

Solar-powered systems may be in the forms of solar photovoltaic cells, solar thermoelectric or themionic generators, and solar dynamic systems. A solar-powered system deployable after capsule landing is very cumbersome, and its expected reliability and performance are unpredictable due to the uncertainties about the Martian environment (such as solar constant, dust storms and terrain). As solar mirrors or solar cells must be located outside the capsule, they will bum up or be damaged by entry heating. Also, the requirement of the power system to withstand high shock loads rules out the use of solar-powered systems for the landing capsule.

C. Fuel Cells

Fuel cell technology has advanced rapidly in recent years. This is mainly due to the selection by NASA of the hydrogen-oxygen fuel cells as power supplies for the Apollo and Gemini spacecraft. Fuel cell operation is dependent on the chemical kinetics of the conversion of molecular hydrogen and oxygen into ions. This ionic conversion takes place within the cell at the interface between the gas and the electrolyte. The amount of current available from the cell is a direct function of the total surface area available for the ionization reactions to proceed. The efficiency of a fuel cell may be as high as 65%. Current fuel cells with large electrodes weigh about 48 lb/kw-hr and occupy 1.1 ft^3/kw-hr. From a weight and volume standpoint, it appears that the crossover point between fuel cells and batteries is about 10 hr or less; for dynamic systems, the crossover point may be about 50 hr (Ref. 2).

The Bacon type fuel cell must operate at about 475°F to support a usable current density. However, this requires that the cell is maintained at about 600 psi in order to keep the electrolyte from boiling. The Apollo fuel cell is a modified version of the Bacon cell and is designed to operate at about 60 psi and 500°F. It takes about an hour to bring the fuel cell up to operating temperature.

The requirements of heat-sterilization and fuel storability for 200 days make the use of fuel cells unattractive for the Mars mission.

D. Radioisotope Thermoelectric or Thermionic Generators (RTG)

Auxiliary space power systems using radioisotopes as power sources have become an operational reality with the launchings of the transmit satellites which have on board SNAP radioisotope power units. Radioisotopes provide self-contained sources of energy in relatively high power densities. Their natural decay rates determine, in major part, the lifetime designed into the power unit. These power systems lend themselves especially to remotely located electrical devices such as unmanned satellites or missions to other planets (Ref. 3).

The principles of radioisotopic power are simple. Heat is generated when radiation is absorbed in the fuel and in the surrounding containment material. The heat is then partially converted into electricity, using a suitable energy conversion device, with the remainder of the heat being dissipated to the external environment. The radioisotopic power unit, therefore, consists of three main components: a radioisotopic fuel, an energy conversion device, and a heat sink.

1. Radioisotopic Fuel

For space applications, it is advantageous to use isotopes and fuel forms with low external radiation to minimize shield weight. Alpha emitters such as Pu-238, Cm-244, Cm-242, and Po-210 are therefore best suited to space missions. In selecting a fuel for a specific mission, a balance must be made between system weight, half-life, power density, fuel availability, cost and radiation tolerance. Some of the more useful fuels are listed in Table 1. For the 200-day Mars *Mariner* mission, Pu-238 or Cm-242 are the power sources considered. Cm-242 has a very high specific power. However, because of its short half-life, excess thermal energy at the start of the mission must be provided in order to meet end of mission power demand.

Table 1. Characteristics of isotopic heat sources*

	Sr-90	Cs-137	Pm-147	Pu-238	Cm-244	Cu-242	Po-210	Ce-144	Co-60
Type of decay	Beta	Beta-Gamma.	Beta	Alpha	Alpha	Alpha	Alpha	Beta-Gamma.	Beta-Gamma.
Half-life, yr	28	30	2.7	89	18	0.45	0.38	0.78	5.3
Specific power of isotope, thermal w/g	0.90	0.42	0.33	0.56	2.8	120	141	25.6	17.4
Estimated isotopic purity, %	50	35	95	80	98	90	95	18	10
Typical fuel form	SrO	Glass	Pm_2O_3	PuO_2	Cm_2O_3	Cm_2O_3	Metal	CeO_2	Metal
Active Isotope in compound,%	42	16	82	71	89	82	95	15	10
Specific power of compound, thermal w/g	0.38	0.067	0.27	0.39	2.49	98	134	3.8	1.7
Density of compound, g/cc	3.7	3.2	6.6	8.9	10.6	11.75	9.3	6.4	8.9
Power density of compound, thermal w/cc	1.40	0.21	1.8	3.5	26.4	1150	1210	24.5	15.5
Shielding requirement	Heavy	Heavy	Minor	Minor	Moderate	Minor	Minor	Heavy	Heavy
Emission requiring shielding	Bremst'ng	Gamma			Neutron			Gamma	Gamma

* Rohrmann, C. A., "Radioisotopic Heat Sources," HW-76323 Rev 1. October 15, 1963.

2. Energy Conversion Devices

The heat generated by a radioisotope can be converted into electrical energy by thermoelectric, thermionic, or turboelectric Rankine cycle. Thermoelectric and thermionic conversion are similar in principle. In thermoelectricity, an electric current is produced when a suitable material is subjected to a temperature gradient. In thermionics, a current flows between two electrodes when they are held at different temperatures. No radioisotope-turboelectric system has yet been built; however, many studies have been performed on systems from 1 kw(c) to 40 kw(e). These studies have indicated that the turboelectric system is best applied to short-lived, relatively high-power applications. At the present time, attainable conversion efficiencies for thermoelectric and thermionic systems are of the order of 5 to 10%. It is reasonable to assume, however, that within the next few years the conversion efficiency can be at least doubled.

The electrical characteristics of RTG must be matched with the electrical requirements of the spacecraft. The power output, of course, is one of the design features of the generator itself, but the voltage out of the power supply generally has to be altered with a converter so that it matches the system payload requirements of the spacecraft. The converter can be a completely static solid state system using a solid state switchings circuit, a transformer, and a rectifier (if required). Converter efficiencies of 85% have been achieved at voltages between 3 and 30 v. A high voltage converter (3 to 35v) may weight about 3 lb and a low-voltage converter (0.5 to 3v) with charge regulator may weigh about 4 lb.

3. Thermionic RTG System

SNAP-13 is a thermionic demonstration device of about 12.5 w power output under development by the Martin Company for the AEC. It represents the first marriage of a radioisotope heat source to a thermionic conversion unit. Cm-242 is used as a heat source coupled with a cesium filled thermionic diode. The diode is designed to operate in the ionization mode and has been tested for more than 4000 hr using electrical heaters to simulate isotope thermal conditions. The weight of the unit, fueled for 120 days of operation, is 4.5 lb without a re-entry heat shield.

SNAP-13 appears well suited as a power source for the Mars *Mariner* landing capsule. To meet the mission requirements, extra fuel and a re-entry heat shield must be added (bringing its weight to 8-10 lb). In order to supply the short 700-w power peaks, a combination of a SNAP-13 type RTG and a 125-whr battery is proposed (Fig. 2). The RTG would recharge the battery through a charge regulator and converter. A second converter between the battery and capsule load would provide the constant small load and the 700-w peak loads.

Due to the low number of charge-discharge cycles required, a sterilizable Ag-Zn battery may be used. Preliminary weight and volume estimates are as follows:

Components	Weight, lb	Volume, in.^{3}
SNAP-13 (including extra fuel and heat shield)	8	225
Charge regulator and converter (low voltage)	4	60
Battery (125 whr)	3-8	214
Converter (high voltage)*	4-8	36
System Total	19-28	535

The cost of SNAP-13 will depend upon the extent of AEC funding for development and fuel costs. The Martin Company estimates a cost of $50,000 to $70,000 per unit after an AEC safety and development program. This program could be completed for a 1969 Mars *Mariner* flight.

4. Thermoelectric RTG System

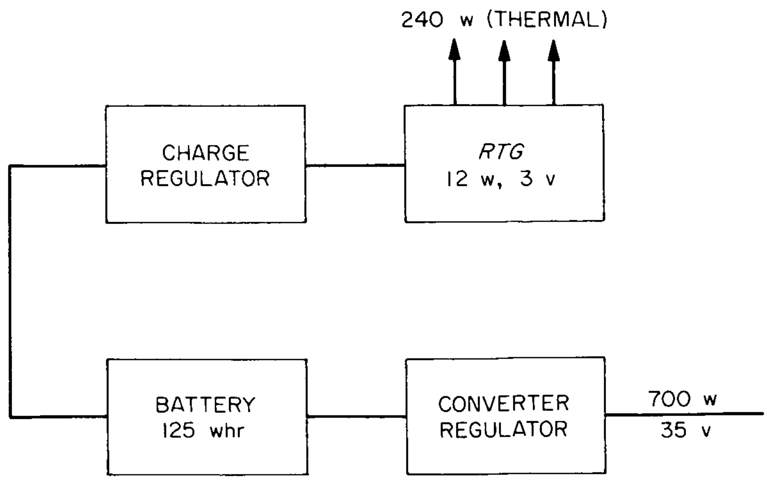
A light-weight thermoelectric RTG device of about 10 w power output is under development at General Atomic for the AEC. In this device, the heat from a Pu-238 source is transferred to the *T/E* panels by radiation. The unit is expected to produce about 500 w/lb of *T/E* elements for a hot junction temperature of only 400-450°C and a cold junction or radiator temperature of about 100°C. The design goal is about 5 w/lb of generator weight excluding heat shield and power flattening device. No radiator fins are required, as the area of the *T/E* cold side is sufficient to dissipate the heat.

The *T/E* panels consist of *n*- and *p*-type thermoelectric elements sandwiched between aluminum sheets which serve as energy collector and radiator surfaces. For structural strength, the radiator sheet is bonded to an aluminum honeycomb through which penetrations are made for the *T/E* elements. Pb-Sn-Te material is used for the *T/E* elements which are very small (1 x 1 x 2 mm). By using a metallurgical bond with a very low contact resistance, the power output is maintained independent of element size, and the size and weight of the *T/E* panels can be drastically reduced.

When this device is fully developed (1965-1966), its performance may compare favorably with that of the SNAP-13 thermionic generator.

*Converter may be eliminated if 30-v battery is used.

Fig. 2. RTG - Battery System



5. Waste Heat and Thermal Control

The thermal heat generated by the RTG device must be disposed of by thermal radiation to space. It is therefore advantageous to locate the RTG device close to the outer surface of the capsule so that heat can be radiated without affecting other capsule components. It is, however, expected that the landing capsule will be located in the shade during most of flight time and may need a heat source for thermal control during that period. All or part of the RTG waste heat may therefore be used to keep the capsule heated, and the generated electric power can be used by the bus during the 200-day flight.

6. Isotope Availability and Cost

Isotopes which are currently being produced as power sources for space radioisotope thermoelectric generators are Pu-238 for SNAP-9A and Cm-242 for SNAP-11 and 13. Sr-90 is used as a power source in the SNAP-7 series terrestrial generators.

The AEC has experience in producing Pu-238 and plans to make 30 to 35 kg available prior to 1968, which at a 5% over-all RTG efficiency, would provide about 900 eW of power. Thereafter, estimated annual availability is equivalent to about 550 eW by the end of 1968, increasing to about 800 eW by the end of 1972. For planning purposes, these availability figures roughly represent a lower limit to the amount of Pu-238 that could be produced by the AEC at a cost of about $450 per gram ($16,000 per eW). The upper limit on availability at this cost is not likely to exceed twice the amounts stated above. Alternatives for substantially increasing Pu-238 availability are concomitant with unit costs of several thousand dollars per gram. Hence, expanding AEC production of Pu-238 with its concomitantly higher unit cost may be undesirable, unless the user agencies specifically request additional amounts of this isotope in spite of its cost.

Radioisotopes which have possible use as heat sources and which would be available during the next several years, in addition to Pu-238, are Sr-90, Cs-137, Pm-147, and Cm-242. The anticipated annual availabilities and unit costs during the 1964-66 period are:

Isotope	Annual capacity	Unit cost
Strontium-90	3 to 5 Mc	\$1.00-1.50/c
Cesium-137	1 to 3.5 Mc	\$0.80/c
Promethium-147	0.3 Mc	\$0.75/c
Curium-242	150 g	\$12,700/g

Again, assuming a 5% over-all RTG efficiency in converting isotopic heat output to usable electric power, the thermal output from the above isotopes at the time of separation and purification might provide the following amounts of electric power:

Isotope	Annual capacity	Unit cost
Strontium-90	1,000 to 1,600 eW/yr	\$3,000-4,500/eW.
Cesium-137	250 to 850 eW/yr	\$3,500/eW
Promethium-147	6 eW/yr	\$40,000/eW
Curium-242	900 eW/yr	\$2,000/eW

7. Design of Radioisotope T/E Generator

The following assumptions were made in the design of an RTG device as a power source for the landing capsule:

Assumptions

Battery capacity	125 whr
Charging time	13 hr
Efficiency of charge regulator and converter	80%
Efficiency of RTG device	5%
Radioisotope fuel	Pu-238

For Pu-238

$$\text{Electric power out of RTG} = \frac{125}{13 \times 0.8} = 12 \text{ w}$$

$$\text{Total heat in (end of mission)} = \frac{12}{0.05} = 240 \text{ w}$$

Thermal Analysis

Heat used by T/E elements (90%)	216 w_{th}
Heat loss through insulation (7%)	17 w_{th}
Heat loss through structure (3%)	7 w_{th}
Useful power density (Pu C)	6.9 w/cc
Fuel volume: $\dfrac{240\ w_{th}}{6.9\ \text{w/cc}}$	34.8 cc
Fuel weight: 34.8 cc × 11.46 g/cc =	400 g
34.5 w/Kilocurie (Pu^{238})	
Source Strength: $\dfrac{240\ \text{w}}{34.5\ \text{w/1000 curie}}$ =	7000 curie
Disintegrations/sec-g: 6.4×10^{11} Alpha/sec-g	
Alpha/sec: $6.4 \times 10^{11} \times 400$ =	2.56×10^{14}
Neutron/sec: $\dfrac{400\ \text{g} \times 1.32 \times 10^{16}\ \text{fiss}}{238_A \times 4.9 \times 10^{10}\ \text{sec}}\ \dfrac{\times 2.5\ n/\text{fiss}}{T/2}$ =	1.15×10^{6}
Gamma energy	0.8 Mev
YIELD, %	7×10^{-5}
Disintegrations/sec/g	6.45×10^{11}
Gamma/sec: $6.45 \times 10^{11} \times 400 \times 7 \times 10^{-7}$ =	1.81×10^{8}

Size of Fuel Element

Design for helium containment ($V_{air} = V_{fuel}$)

Assume length = 6 in. = 15 cm

$\pi/4 \times (D_0^2 - D_1^2) \times 15 = 34.8,\ D_1^2 = D_0^2 - D_1^2$

$D_0 = 2.45$ cm, $D_1 = 1.74$ cm

Radiation Dose from Fuel

The neutron flux reduction is given by

$$n/\text{cm}^2 - \text{sec} = n/\text{sec}\ \frac{e^{-\Sigma X}}{4\pi r^2}$$

where Σ = neutron removal cross section, cm^{-1}

X = shield thickness, cm

r = distance from fuel, cm

The gamma flux reduction is given by

$$\gamma/\text{cm}^2 - \text{sec} = \gamma/\text{sec}\ \frac{B\, e^{-\mu X}}{4\pi r^2}$$

where

B = buildup factor

μ = linear absorption coefficient, cm^{-1}

X = shield thickness, cm

r = distance from shield, cm

The fuel is assumed to be surrounded by a heat shield (3/4-in. Beryllium) with the following physical constants (Ref. 4):

$\rho = 1.84\ \text{g/cm}^3$

$\Sigma = 0.131\ \text{cm}^{-1}$

$\mu = 0.115\ \text{cm}^{-1}$

$B = 1$

The neutron flux at $1m$ from fuel is

$$n/\text{cm}^2 - \text{sec} = 1.15 \times 10^6 \times \frac{e^{-0.131 \times 1.9}}{4\pi \times 100^2} = 7.15$$

For 1-Mev neutron energy,

$$1 \text{ mrem/hr} = 9\ n/\text{cm}^2\text{-sec}$$

Therefore,

$$\text{dose rate} = 0.8 \text{ mrem/hr}$$

The gamma flux at $1m$ from fuel is

$$\gamma/\text{cm}^2\text{-sec} = 1.81 \times 10^8 \ \frac{e^{-0.115 \times 1.9}}{4\pi \times 100^2} = 1.14 \times 10^3$$

For 0.8-Mev gamma energy,

$$1 \text{ mr/hr} = 6 \times 10^2\ \gamma/\text{cm}^2\text{-sec}$$

Therefore,

$$\text{dose rate} = 1.9 \text{ mr/hr}$$

8. Summary of Thermoelectric RTG System

Using the light weight T/E panels developed by General Atomic, a preliminary design of a thermoelectric RTG system gave the following results:

Power output, total	12 w
Number of panels	5
Voltage per panel	0.84 v
Size of panel	7 in. × 7 in.
Number of T/E couples	200
T/E material	Pb - Sn - Te
Hot junction temperature	400°C
Cold junction temperature	150°C
Total efficiency	4.7%

Total weight (excluding heat shield)	3 lb
Weight of heat shield (estimated)	2 lb
Physical size	7-in. w × 7-in. h × 7-in. l
Volume	341 in.^{3}

Preliminary weight and volume of the total thermoelectric RTG device in combination with a 125 whr battery are as follows:

Components	Weight, lb	Volume, in.^{3}
T/E RTG device (including heat shield)	5	341
Charge regulator and converter (low voltage)	4	60
Battery (125 whr)	3-8	214
Converter (high voltage)*	4-8	36
System Total	16-25	651

E. Hydrazine – Turboalternator Power System

A hydrazine-turboalternator power system with a raw power output of 700 w for a duration of one hour can be developed at a total weight of about 15 to 20 lb. The power system shown schematically in Fig. 3 consists of an alternator directly driven by a single-stage, impulse turbine. The fuel for the turbine is prepackaged liquid hydrazine. The fuel is pressurized and expelled from a bladder in the storage tank by nitrogen which is regulated to maintain a constant fuel pressure. The fuel is catalytically decomposed in a gas generator and enters the turbine at 1650°F and 150 psi. To control the power system, the output voltage which is proportional to speed is sensed to provide a signal for a solenoid fuel control value.

Systems similar in principle to the one described above have been built by several companies. Turbomachinery is now operating at speeds up to 250,000 rpm and turbine tip speeds at 2000 ft/sec. The maximum efficiency of a small single-stage turbine is about 35%. By designing for several stages, an efficiency of about 50% may be obtained. The result is a lower fuel consumption and therefore lower fuel weight; however, the turbine weight will increase. It is therefore necessary to optimize the turbine with regard to efficiency and weight.

* Converter may be eliminated if 30-v battery is used.

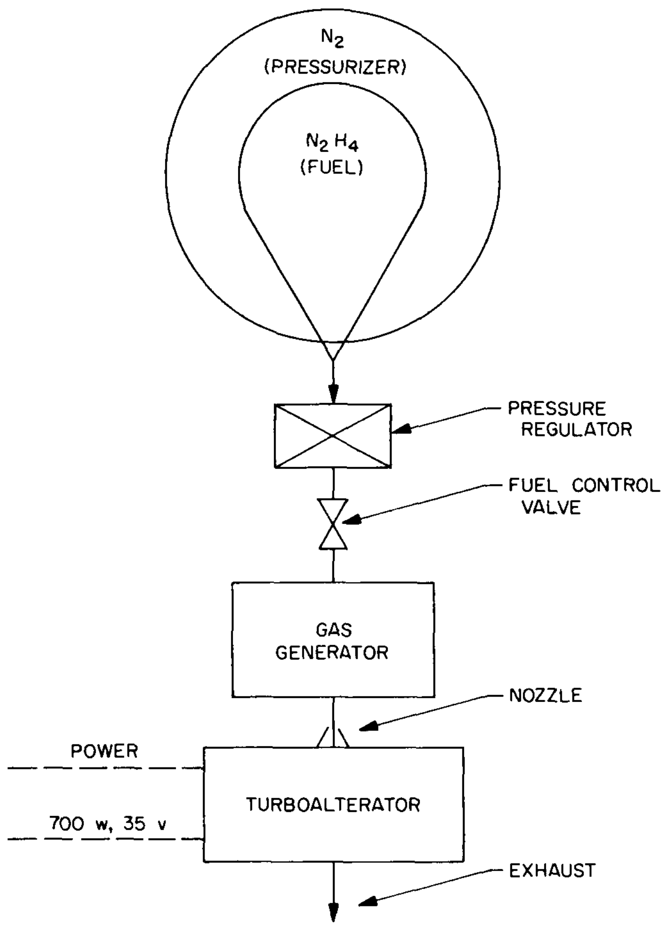


Fig. 3. Hydrazine-turboalternator system

1. Hydrazine Propellant

Hydrazine has both a relatively high available energy and a low decomposition temperature. The available energy of 682 Btu/lb from a partial decomposition process makes it an attractive propellant for small, high-speed turbines (Ref. 5). Since it is a monopropellant with excellent long-term storage properties, the liquid storage and gas generator designs are greatly simplified.

Hydrazine is a clear, oily, water-white liquid with an odor of ammonia. It has a density very close to that of water and a low vapor pressure of only 2.5 psia at 150°F. This minimizes toxicity and fire hazards. Hydrazine freezes at about 35°F; however, it contracts upon freezing so that there is no damage to the containing vessel. In addition, hydrazine is stable to shocks. It is recommended that stainless steel (304 and 347), pure aluminum, and certain aluminum alloys be used in conjunction with hydrazine.

The fuel consumption of a hydrazine-turboalternator system can be found from the following equation:

$$M_{fuel} = 0.09 + 0.00085\, t + 0.0001\, e, \text{ lb}$$

where

t = running time, sec

e = electrical energy, wmin

2. Hydrazine Gas Generator

Hydrazine is admitted under pressure to a reaction chamber where it is decomposed either catalytically or by exposure to a heat source which raises it to the decomposition temperature. With properly prepared catalysts, decomposition can be initiated at temperatures on the order of 40°F. As the hydrazine decomposes, energy is liberated which acts to heat the chamber rapidly to a temperature above 1500°F. The catalyst can either be a true cold catalyst (shell catalyst) or a solid oxidant type of material.

The exact temperature of the gases discharged from the gas generator can be controlled by the design of the unit because hydrazine decomposition proceeds in two stages. The first stage is a decomposition of hydrazine into ammonia and nitrogen. The second stage is a breakdown of the ammonia into nitrogen and hydrogen. The initial reaction liberates heat which results in a theoretical temperature of 2500°F for the ammonia and nitrogen decomposition products at 100% decomposition. The relative quantity of decomposed ammonia can be controlled by the design of the catalyst chamber and the discharge gas temperatures may be between 1200°F and 2000°F.

3. Turbine Design

A preliminary design of a single-stage, impulse-type turbine gave the following results:

Turbine power	820 w
Tip diameter	3.50 in.
Material	Titanium
Blade velocity	2000 ft/sec
Turbine velocity	130,000 rpm
Inlet pressure	150 psia
Inlet temperature	1650°F

Adiabatic efficiency	34%
Hydrazine flow rate	0.002 lb/sec
Weight of turbine (estimated)	2 lb

4. Alternator

The alternator proposed for this power unit is a unique design recently developed by the Garrett Corporation. The design principles of this alternator are as follows: There are two flat cylindrical rotors, each of which contains eight permanent magnets. The two rotors are fixed in relation to each other so that opposite poles face each other across the control air gap. The stator windings are located in this air gap. Contrary to normal practice, the stator windings are not wound upon a laminated iron core, since the magnets are sufficiently powerful to provide an adequate magnetic flux across the air gap. The absence of the iron core eliminates the usual iron losses and results in a low-weight machine.

The above design has been used in an 8-pole, three-phase, 8-kcps, 2-kw machine which rotates at 120,000 rpm.

5. Estimated Weight and Volume

The estimated weight of the 700-w, 700-whr hydrazine-turboalternator power unit is about 20 lb, and the unit occupies a volume of about 226 in.^{3} A breakdown of the estimated weights of the principal components are as follows:

Turboalternator and housing	6.50 lb
Fuel (hydrazine)	7.34
Nitrogen	0.20
Tankage (fuel and nitrogen)	4.50
Valving	0.30
Gas generator and nozzle assembly	0.23
Electronics	0.60
Insulation	0.03
Tubing, connectors and miscellaneous	0.30
Total	20.00 lb

6. 1-kw Turboalternator Power-System

The following is a rough estimate of a 1-kw turboalternator power system. The estimate was made by R. R. Breshears and A. D. Harper.

Estimate of Propellant Requirements

a. Assumptions

1. N_2H_4 gas generator driven turbine
2. Over-all efficiency = 40% [electrical output (raw)] /available mechanical energy
3. Impulse turbine
4. Effective vacuum I_{sp} of gas generator products = 200 lb sec/lb

b. Calculated Propellant Consumption Rate Estimate

c = Exhaust velocity = $(I_{sp})\,(g) = 200 \times 32.2 \simeq 6000$ ft/sec

Assume equivalent flow rate for 1 lb-thrust rocket motor ($F = 1.0$ lb)

Optimum blade velocity = $c/2$ = 3000 ft/sec

Force on blade (180° turn) = 1 lb

Power (100% eff) = 1 lb × 3000 ft/sec = 6000 lb-ft/sec $\simeq$ 4 kw

Output power (40% eff) = (0.40) 4.0 = 1.6 kw

$$\text{Propellant flow rate} = \frac{F}{I_{sp}} = \frac{1.0}{200} = 0.005 \text{ lb/sec}$$

$$\text{Specific propellant consumption} = \frac{0.005 \text{ (lb/sec)}}{1.6 \text{ kw}} \times 3600 \text{ sec/hr} = 10 \;\frac{\text{lb}}{\text{kw-hr}}$$

c. Alternate Propellant Consumption Rate Estimate from the Literature

Reference: "Design of Space Power Plants," by D. B. Mackay, Prentice-Hall, page 282, 1963.

Specific propellant consumption for N_2H_4 from Reference = 5 lb/bhp-hr (turbine output)

Assume alternator efficiency = 80%

$$\text{Over-all specific propellant consumption} = \frac{5\ (\text{lb/bhp-hr})}{0.746\ (\text{kw/bhp})\ 0.80} = 8.5\ \frac{\text{lb}}{\text{kw-hr}}$$

8. Estimates of System Masses

Item	Mass	
	1.0-hr duration	**6.0-hr duration**
Propellant @ 8.5 lb/kw-hr	8.5	51
Propellant supply system	8.5	20
Gas generator	0.1	0.1
Turboalternator*	10.0	10.0
	27.1 lb	81.1 lb
Energy-to-weight ratio	37 whr/lb	74 whr/lb

*Aeronutronics estimates the turboalternator mass at about 5.0 lb.

The estimates of the chemically driven turboalternator given here are felt to be quite conservative. The specific propellant consumption rate is for an over-all efficiency of about 47%.

Because of the trade-off between turboalternator mass and propellant mass, a design with this efficiency is probably about optimum for guided missile power systems which are required to operate for only a few minutes. When operating times on the order of hours are required, the optimum design will probably be attained by increasing the efficiency of the turboalternator with additional turbine stages (with some increase in mass). This would substantially reduce the propellant mass required. If the efficiency were increased to 75%, the power-to-weight ratios would be increased to about 45 whr/lb at 1 kw for 1 hr and 100 whr/lb at 1 kw for 6 hr. All of these performance figures appear to be very attractive in comparison with power-to-weight ratios of 10-15 whr/lb for heat-sterilizable batteries estimated as state-of-the-art by 1967.

Based upon these data, it appears that a turboalternator should definitely be considered as a power source for advanced *Mariner* capsules.

III. CONCLUSIONS

It appears that the RTG plus battery system or the hydrazine turboalternator system will be capable of meeting *Mariner* mission requirements. The final selection between these two types of power sources and others will depend mainly on whether the communication system will operate continuously or be cycled and on the required lifetime of the system after landing. Communications and experiments after landing will determine the final power level requirement of the system. The required lifetime of the system will be determined by the duration of the experiments.

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