AN EVALUATION OF SOME SPECIAL TECHNIQUES FOR

NUCLEAR WASTE DISPOSAL IN SPACE

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ABSTRACT

This note presents a preliminary examination of several special ways for space disposal of nuclear waste material which utilize the radioactive heat in the waste to assist in the propulsion for deep space trajectories. These include use of the wastes (or an extract of the ^{90}Sr or ^{137}Cs compounds contained in the waste) in a thermoelectric generator (RTG) which operates an electric propulsion device and a radioisotope — thermal thruster which uses hydrogen or ammonia as the propellant. These propulsive devices are compared to the space tug and the space tug/solar electric propulsion (SEP) combination for disposal of waste on a solar system escape trajectory. Such comparisons indicate that the waste-RTG approach has considerable potential (disposing of perhaps four times as much waste) provided the combined specific mass of the waste container — RTG system does not exceed approximately 150 kg/kw$_e$.

Although this study stresses the solar system escape destination, several exploratory numerical calculations have been made for high Earth orbit and Earth escape destinations. These show that some care must be exercised in selecting an Earth escape path in order to avoid future near encounters with the Earth or Venus. In general, it is believed that useful calculations are possible using numerical integration which could help in an orbit or trajectory selection process.

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Introduction

If the projected future United States power demands are to be partly satisfied by stationary nuclear power plants, then there will be an associated increase in the amount of nuclear waste material which results from the reprocessing of the spent reactor fuel elements to recover the unused fuel. While such reprocessing is an inherent part of the economical operation of such nuclear power stations, it can lead to surprising amounts of radioactive residue. Reference 1, for example, contains some projections which lead to an annual output of over 5×10^5 kg of waste by the year 2000. This, of course, depends on the electrical power demands continuing to increase as they have in the past and also that no new power producing methods (such as nuclear fusion reactors) emerge to meet the increasing demand.

As a result of such nuclear waste possibilities, the Atomic Energy Commission has asked NASA to study the feasibility of disposing of the waste products in space. Reference 1 and its associated documents constitute a direct response to the AEC request. This memorandum has been stimulated by that effort but is not an official part of the response to the AEC.

The purpose of this paper, then, is to consider several concepts by which the energy still contained in the nuclear waste material could be utilized to augment or complete the space disposal process of such waste. As indicated in figure 1 (taken from ref. 1), it can be seen that the energy output per unit mass (shown on the ordinate in figure 1) is about 300 w/kg if the waste is obtained at one year from the time of reprocessing.

This is a specific energy value equivalent to that of ^{238}Pu. However the nature of the waste is such that (see ref. 2) it is much less dense (in its solid form) and much more difficult to shield. The radiation is due to the high percentage of short lifetime radioactive elements in the waste. As indicated in figure 1, the energy output eventually becomes an exponential function of time. This represents a transition to activity from a few dominant, longer half life elements (i.e. ^{139}Cs and ^{90}Sr). This is shown more clearly in figure 2 (taken from ref. 3).

Thus, it would appear that there is at least an interesting amount of energy in the waste if it can be obtained early enough (i.e. the specific energy output is similar to that of isotopes usually used in space applications). Another possibility is the separation of special high heat output elements out of the waste. This will be considered as an alternate but more expensive way of utilizing the heat in the waste.

Two propulsive techniques will be considered which use the waste heat. One is the direct conversion of the heat into electricity by a thermoelectric generator (RTG) and the other is heating of some working fluid such as liquid hydrogen or ammonia and subsequently expelling the fluid at high velocity to produce thrust. This is similar to the "poodle" thruster concept described in reference 4, and will be referred to here as an isotope thermal device. Another somewhat related concept—solar electric propulsion (SEP) plus waste (RTG)—will also be considered but not evaluated in quantitative terms. These various concepts are illustrated in figure 3.

Waste Material Form

Considering the energy output only, it would follow that early acquisition and containment of the solid waste would be desirable. This would utilize the high energy output of the short half-life elements before they decay to less active states. However, it is recognized that the existing processing facilities may restrict the acquisition time to one or two years after reprocessing of the fuel elements. Thus, one form of

the waste products which will be considered here will be the solid state one and two years after reprocessing.

Of the various solid forms which are currently being considered (see ref. 2) their density varies between 1.33 and 2.8 gm/cm^3. In reference 1, the spray melt solidification process was selected as a desirable form for packaging and heat conductivity purposes. The properties of this type of solid waste are listed in table 1.

TABLE 1. SOME PROPERTIES OF SPRAY MELT SOLID WASTE MATERIAL
(from refs. 1 and 2)

Heat output (one year)	300 watts/kg
Heat output (two years)	150 watts/kg
Density	3.0 gm/cm^3
Thermal conductivity	1.8 $watts/cm^3/°K$
Maximum (center line) temperature	1170°K

These properties will be used here as those of typical solidified but otherwise unprocessed waste material.

The second waste form which will be considered here is the separation of ^{137}Cs and/or ^{90}Sr from the main waste stream. This could be done at about five years or more after initial reprocessing operation. At the present time, facilities for separating these elements out of the waste stream do exist (see ref. 5); thus the cost of increasing output or purity may not be excessive. However, the additional cost of performing the separation must still be included.

The purpose of separating out these isotopes is that they have a high specific energy output combined with a rather long half life. Some properties of these isotopes in their common (usually oxide) forms are given in table 2.

TABLE 2.   SOME PROPERTIES OF ^{137}Cs and ^{90}Sr ISOTOPE COMPOUNDS
(from ref. 6)

Isotope	Shield Density* cm of Uranium	Density gm/cc	watts/gm	Half-life years
^{90}Sr (SrO)	2.3	4.7	0.453	27.7
^{137}Cs (CsF)	5.5	3.586	0.134	30

*The shield densities shown in this table are computed for a spherical 1 kw_t source and are for 10 rem/hr at 1 meter from the center of the source.

^{90}Sr, for example, has an energy output similar to ^{238}Pu which is often used as a heat source for flight RTG power supplies, but has a much more severe shielding problem and other safety disadvantages. However, both isotopes are relatively good heat sources if separated from the rest of the solid waste. Also, they could be bothersome to store on Earth because of their relatively long half lives.

Another waste form which also results from special processing is the separation of the actinide compounds from the waste material. These isotopes have very long half lives which implies extensive ground storage time if they are left in the waste. However, the heat output is very low and as such, does not constitute an interesting heat source for propulsive or power generating purposes. Rather it probably represents the most compact form the waste can take without utilizing some form of nuclear transmutation. However, the actinides are only the most troublesome part of the waste material and facilities for the storage or use of the rest must be provided.

Because the actinides represent only a fraction of the total waste, they are probably best considered along with more conventional space disposal techniques such as the shuttle/tug or shuttle/centaur. Use of such launch vehicle systems is being examined by the NASA, Lewis Research Center (see ref. 1) and will not be considered here.

Disposal Destinations

A number of destinations for the proposed waste containers are currently being studied. They include high Earth orbit, Earth escape, 0.99 AU and 1.1 AU circular orbits about the sun, solar system escape and solar impact (see figure 4). A comparison of the propulsive energy requirement of these destinations can be found in reference 1.

The nearby destinations such as high Earth orbit or Earth escape are of interest because the propulsive energy expenditure required to achieve such orbits is low and would probably be most attractive as a destination for chemical rocket systems such as the space tug or the centaur. However, they may also be of interest for the waste heated fluid concept as such devices may have low specific impulses, depending on the working fluid used (e.g., ammonia). However, while such destinations may be easy to reach, they create another problem regarding whether or not the waste is actually disposed of in such a case. This is particularly true of the Earth escape case where there is some chance, however small, that the waste container may someday return. Several exploratory numerical integrations were carried out in order to illustrate some of the problems that can arise in certain cases. Specifically, the cases so far studied are Earth escape and high Earth orbit.

All numerical integrations have been performed on a CDC 7600 computer using a version of the LeRC N-Body program (see ref. 7). The CDC 7600 has a 60 bit word length which allows single precision 14 digit arithmetic. Thus, very accurate numerical integrations are possible without the usual need for double precision arithmetic or accumulation.

Considering first the high Earth orbit case, it was first determined that the important perturbations were those due to the moon, sun and the Earth's oblateness. Inclusion of Jupiter and several other planets had little noticeable effect after several years of integration. The predominant changes in the orbit's elements were precession of the

line of nodes and the argument of pericenter. The orbit chosen for study
was circular at 50,000 nautical miles altitude and inclined at 28.5° to
the equatorial plane. These calculations for the Earth orbit case were
very time consuming, requiring about 25 seconds of computer time per year
of orbit time.

The Earth escape case was less expensive, using less than 50 seconds
of computer time for 500 years of interplanetary flight. The orbit under
investigation was an Earth escape trajectory with a perigee altitude of
100 nautical miles and an eccentricity of 1.1.

The main results for the Earth escape case were that some care must
be taken to keep the orbit as much inside the Earth's orbit as possible.
That is, Earth departure conditions should be such that the trajectory
enters heliocentric space at aphelion. Otherwise it was found the
trajectory would re-enter the Earth's sphere of influence several times
within a 100 year span. On one other occasion a trajectory passed
through the sphere of Venus at 273 years during a 514 year integration
even when special care is taken to inject at aphelion. This indicates
that care must also be exercised in selecting the trajectory perihelion.
In the Earth escape cases, it was found necessary to include all the
planets out to Jupiter. More planets could not be included because of
present limitations of the program.

In all cases the planets and the moon were included with fixed orbit
elements chosen from some particular epoch. This is a serious omission
only in the case of the moon, which precesses around the Earth at the
rate of about 18°/year. However it has become clear that useful calcula-
tions can be made which can very likely be of value in orbit selection
and simulation.

Unlike the low energy cases, there are at least two other destina-
tions which probably constitute true disposal. They are solar impact
and solar system escape. Of these two, solar system escape is perhaps

the most preferable because of the generally lower energy requirements. One objection to solar system escape is that it may become someone else's problem in due time. However, the time to reach the nearest star is enormous and could leave the package no more harmful than a meteorite. (A detailed treatment of the probable hazards associated with these destinations can be found in ref. 10.)

Thus, it would appear that the most preferable destination is solar escape with solar impact a second choice. Of the other destinations, high Earth orbit or solar orbit are perhaps the least likely to return to Earth. However, the preliminary numerical calculations which have been completed for the Earth orbit case have indicated that the orbit will precess (not unlike the moon) due to solar, lunar and Earth oblateness perturbations. Thus, it may be difficult to track the waste containers for the hundreds or perhaps thousands of years which may be required by safety considerations.

Consequently, this section will consider only solar escape and impact as likely destinations for the propulsion systems under consideration herein. The other destinations will be given ample consideration (in ref. 1) and need not be considered here in any further detail.

It has been shown in reference 1, direct solar impact requires a velocity relative to the Earth of 30 km/sec. This stops the package relative to the sun and it falls straight down on a radial line to impact. Very few propulsion systems presently under consideration (with the possible exception of the laser ignited fusion devise described in reference 8) could accomplish such a mission. For example, the waste-RTG and solar electric propulsion systems can simulate such a mission only by a slow spiral into the sun. Unfortunately, the effective velocity change for such a maneuver is the difference between the circular orbit speeds at the different radii. Thus, to reach the surface of the sun (a radius of 0.698×10^6 km) would require:

$$\Delta V = V_{c,\odot} - V_{c,\oplus} \approx 436. - 30 = 406 \text{ km/sec} \tag{1}$$

where V_c is the orbital speed at the indicated distance, which is probably beyond the capabilities of any ion thruster system.

A more optimal approach, even for chemical rocket systems, would be to first proceed outward to some high aphelion and then nullify the velocity at aphelion and drop into the sun. This, as well as the direct method, is illustrated in figure 4. The limit in this process is to first essentially escape the solar system (i.e., very high aphelion) and then apply a very small correction and return to solar impact. Unfortunately, the time involved in such a maneuver is excessive and some compromise must be made between the time and ΔV involved in the maneuver. Figure 5 illustrates the interchange between time of flight and ΔV.

Thus, it is clear that low energy (i.e. low ΔV) solar impact missions and solar escape are closely related and considering one is equivalent to considering the limiting case of the other. For this reason, only solar escape will be considered in the following sections.

Radioisotope Waste RTG System

As noted previously, it is best to use the waste early. Suppose, for example, that we obtain the waste at one year; then, from figure 1:

$$P/m = 0.3 \text{ kw/kg}$$

$$\alpha_{min} = \frac{1}{P/m \; \eta_c} = \frac{1}{.3 \times .05} = 67 \text{ kg/kw} \tag{2}$$

where

α_{min} = minimum specific mass of the power supply

η_c = efficiency of thermoelectric converter

As indicated, this assumes that the RTG converter efficiency is 5 percent, which is typical of present day technology.

However, since the value of P/m falls off so rapidly in figure 1, some average value should be chosen. To do this, it will be assumed that the electro-static thruster system can operate no longer than 20,000 hr. (Again, this is typical of current estimates from test and flight data. A general description of electrostatic thruster developments and operations can be found in reference 11.) Thus, the value of P/m taken from figure 1 should be between one and three years. This gives an average α_{min} of about 134 kg/kw.

Assuming the thruster efficiency to be of the form:

$$\eta_{th} = \frac{B}{1 + (\frac{D}{C})^2} = \frac{0.842}{1 + (\frac{16}{C})^2} \tag{3}$$

where

C = ion exhaust velocity, km/sec
B = propellant utilization efficiency
D = ionization loss factor, km/sec,

and that the propulsion time (t_p) is limited to 20,000 hr, a value of C can be found which gives the highest initial acceleration. This is given by the relation:

$$C_{opt} = \sqrt{\frac{2000\ B\ t_p}{\alpha_{min}} + (D \times 1000)^2}$$

$$= 34,200\ m/sec$$

The payload ratio, for optimum C, can be shown to be:

$$\mu_L = 1 - \frac{a_0\ \alpha_{min}\ C_{opt}}{B} \tag{4}$$

where a_0 is the initial thrust/mass ratio.

Since no payload is to be carried in addition to the waste-RTG package $\mu_L = 0$ and,

$$a_0 = \frac{B}{\alpha_{min} C_{opt}} = 1.84 \times 10^{-4} \text{ m/sec}^2 \tag{5}$$

Therefore, the propellant fraction, μ_p is

$$\mu_p = \frac{a_0 t_p}{C_{opt}} = 0.388 \tag{6}$$

Thus the ΔV capability of the system is

$$\Delta V = -C_{opt} \ln (1 - \mu_p) = 16,800 \text{ m/sec} \tag{7}$$

Using the same criterion noted before, (see equation 1) it follows that the package will spiral out into the asteroid belt before it runs out of propellant. (This includes Earth escape which requires an additional ΔV of about 8 km/sec.)

The above example illustrates that some additional velocity may be required at Earth departure in order to escape the sun's gravity field. To investigate this possibility, some computer calculations were made to determine the value of a_0 required to reach solar escape starting from various values of velocity relative to Earth (supplied by some chemical rocket stage such as the Centaur). These are shown in figure 6. This has been done with a limit on t_p of 20,000 hr. and a fixed value of C = 30,000 m/sec (this is the lowest practical value based on current thruster technology work. Lower values of C develop difficulties in accelerator grid spacing required).

Given the data shown in figure 6, it is then possible to determine what values of a_0 and $V_{\infty,1}$ are required to escape the solar system for

any chosen value of α_{min}. These are determined from the following equation:

$$\mu_L = 0 = 1 - \mu_p - \frac{a_o \, C}{2 \, \eta_{th}} \, \alpha_{min} \qquad (8)$$

where μ_p and η_{th} can be determined from previous relations (equations (3) and (6)).

Assuming the use of the shuttle/centaur, we have the following relation between $V_{\infty,1}$ and m_o (at Earth escape):

TABLE 3. SHUTTLE/CENTAUR EARTH DEPARTURE MASS CAPABILITY

$V_{\infty,1}$ (km/sec)	m_o (kg)
0	10,400
3	8,500
6	4,900
9	2,050

For this launch system and using an α_{min} of 134 as before, it is found that $V_{\infty,1} \approx 3.0$ gives $\mu_L = 0$ and leads to an ejected system mass (final mass) of 4,850 kg (10,700 lb). This is considerably better than the direct solar escape payload of 1,230 kg. given in reference 1 for the shuttle/tug (expended). Continuing this process for other values of α other than α_{min} leads to the results shown in figure 7 where the mass sent to solar escape is shown as a function of the propulsion system specific mass. As indicated in this figure, all cases above $\alpha \approx 60$ kg/kWe will require some assistance (from a Centaur, tug or some other chemical rocket stage) during Earth escape. These results indicate that the best case of $\alpha = \alpha_{min}$ is interesting and that more work should probably be done to better define α.

$\mathcal{B}$

Solar-Electric Propulsion

Another set of calculations has been made for the case of a 20 kw solar electric propulsion (SEP) system as the ejection stage. These were performed for the same shuttle/centaur departure mode but used an α of 30 kg/kw (typical for SEP stages) and a typical solar cell profile of power as a function of distance from the sun. In this case, it was found that $V_{\infty,1}$ = 6.0 was required (with I_{sp} = 3000 sec. as before) which gave an injected mass (excluding the SEP system) of 1320 kg at solar escape. As indicated in figure 7, this is essentially the same as the shuttle/tug system. Thus it would appear that the SEP approach would not have any great advantage over a simple expended tug. Also, the combined cost of both the SEP stage and the Centaur would probably exceed or equal that of a single tug.

Waste Thermal Thruster

Another device examined was one in which the heat in the waste is transferred into some fluid which is ejected to produce thrust. A preliminary set of calculations for such a device was made assuming that liquid hydrogen could be heated to a maximum temperature of $\approx 2000°F$. (This is near the maximum centerline temperature of most solid waste forms given previously.) Assuming complete expansion into a vacuum, an exhaust velocity of about 7 km/sec ($I_{sp} \approx 700$ sec.) is theoretically possible. However, the amount of waste material (or any isotope) needed to produce a sizable thrust is critical. For example, to achieve a thrust/mass of .10 requires:

$$\frac{P}{m_0} \approx \frac{F}{m_0} \frac{C}{2} = \frac{.10 \times 7000}{2} = 350 \frac{watts}{kg}$$

From figure 1, it is clear that this is about as much heat output as can be expected from any radioisotope heat source.

In order to escape the solar system with an initial acceleration of .10 m/sec^2, more ΔV than the 8.5 km/sec required with very high values of a_0

must be supplied. This is due to the energy expended in lifting the unused propellant through a gravity field. Such "gravity loss" factors can be found in such documents as reference 9. Specifically, it is found from reference 9 that ΔV must be increased by 1.75 to overcome the "gravity losses" associated with an initial acceleration of 0.10 m/sec^2.

At this point it is appropriate to try and size a stage which will escape the solar system. Assuming that the liquid H_2 can be contained in tank with a mass of about 10 percent of the contained propellant, it can be shown that the mass ratio for the maneuver (excluding tanks) is:

$$\frac{m_f}{m_o} = (1 + \sigma)e^{-\Delta V/C} - \sigma = .0314 \tag{9}$$

where σ = tankage factor = 0.10.

If the transfer began in low Earth orbit with m_o = 29600 kg, then the propulsion system could be no more massive than

$$m_{eng} = m_o \left(\frac{m_f}{m_o}\right) = 0.0314 \times 29600 = 927 \text{ kg}$$

Therefore, the thrust, F, can be no more than:

$$F = \frac{2P}{C} = \frac{2 \times 927 \times 300}{7000} = 79 \text{ N}$$

Thus,

$$a_o = \frac{79}{29600} = 2.68 \times 10^{-3} \text{ m/sec}^2$$

At this low a value of a_o, it would require a ΔV of about 8 km/sec just to escape the Earth (see equation 1). This indicates that the system can't escape from low Earth orbit without some high thrust assistance.

For the case of high thrust assist, new data similar to figure 6 have been generated. Using the data and table 3, it has been found that $V_{\infty,1} = 6$ gives the highest mass of ejected waste isotope. This mass is shown in figure 8 where the values for shuttle/tug and SEP (20 kWe) are also shown for comparison.

In figure 8 it can be seen that α has no effect on the injected mass over the range shown. This results from the higher values of a_0 (to the right in figure 8) that result when the Centaur is used for Earth escape. Thus, α has very little effect on ΔV (through changes in a_0) until very high values of α are assumed.

It is therefore concluded from figure 8 that there is insufficient heat energy in the waste (or some of its components such as ^{90}Sr) to be of much interest as a thermal thrust producing device. However, all of this has assumed only one shuttle launch. If more launches are used, the system may be made to compete with a single shuttle/tug. However, considering the optimistic assumptions about the propulsion system size, it would not seem likely that the device could have much economic advantage.

Combined Systems

It may also be possible that some combination of the systems so far discussed could be a better choice. This is perhaps most true of an SEP waste-RTG system. As the power from the solar cells drop off, the RTG power would remain to give a more uniform power distribution throughout the flight. However, the same solar cell cost argument stated before still applies here. Thus, this would appear to be another system worthy of further investigation along with the pure waste-RTG devices.

Other Heat Sources

As noted earlier, the possibility of using some extracted compound of ^{137}Cs or ^{90}Sr should also be considered. From tables 1 and 2 it is

clear that ^{90}Sr would very likely be a much better heat source than the waste or ^{137}Cs. The 450 watts/kg output plus the long half life would combine to give a value of α_{min} = 44 kg/kw. This clearly gives superior performance (shown in figure 7) to the shuttle/tug combination. However, the cost of separating the ^{90}Sr compounds from the waste must be included. Also, it must be recognized that the compounds may be mixed and of a type giving lower energy output than the SrO value given in the table. Most important, however, are containment, shielding, and other safety considerations which will probably increase α much above α_{min}. This is a detailed design problem which is beyond such a preliminary survey. Our purpose here is to try to narrow the alternatives for more intensive future study.

Summary

This investigation, although very preliminary in nature, has indicated that there could be some useful ways in which the heat in the nuclear waste can be used to augment space disposal of the waste. In particular, it appears that an RTG system operating on waste or ^{90}Sr compounds separated from the original waste could be used together with an electric propulsion system to reach solar system escape. In each case the results are very attractive in the extreme case of no-containment weight estimates. This does not mean the scheme will be ultimately useful, but does indicate that further consideration could be worthwhile.

Other approaches, such as SEP only and a nuclear-thermal thruster using the waste heat and liquid hydrogen, do not appear attractive, even with the aid of optimistic assumptions.

Although this study stresses the solar systems escape destination, several exploratory numerical calculations have been made for the high Earth orbit and Earth escape destinations. These show that some care must be exercised in selecting an Earth escape path in order to avoid future near encounters with the Earth and Venus. In general, it is believed that useful calculations are possible using numerical integration which could help in an orbit selection process.

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SYMBOL TABLE

a   acceleration, m/sec^2

B   constant in thruster efficiency expression

C   exhaust velocity, km/sec

D   constant in thruster efficiency expression, km/sec

F   thrust, N

I_{sp} specific impulse, kg-sec/kg

m   mass kg

P   power, watts

t   time, sec

V   velocity, km/sec

α propulsion system specific mass, kw/kg

ΔV velocity increment, km/sec

η efficiency

μ mass ratio

σ tank mass/propellant mass

Subscripts

c   converter

eng   engine

f   final

L   payload

min   minimum

opt   optimum

p   propellant or propulsion

th   thruster

o   initial (t = o)

1   Earth departure

∞   indicating conditions on the asymptote of a hyperbolic orbit

☉   Sun

⊕   Earth
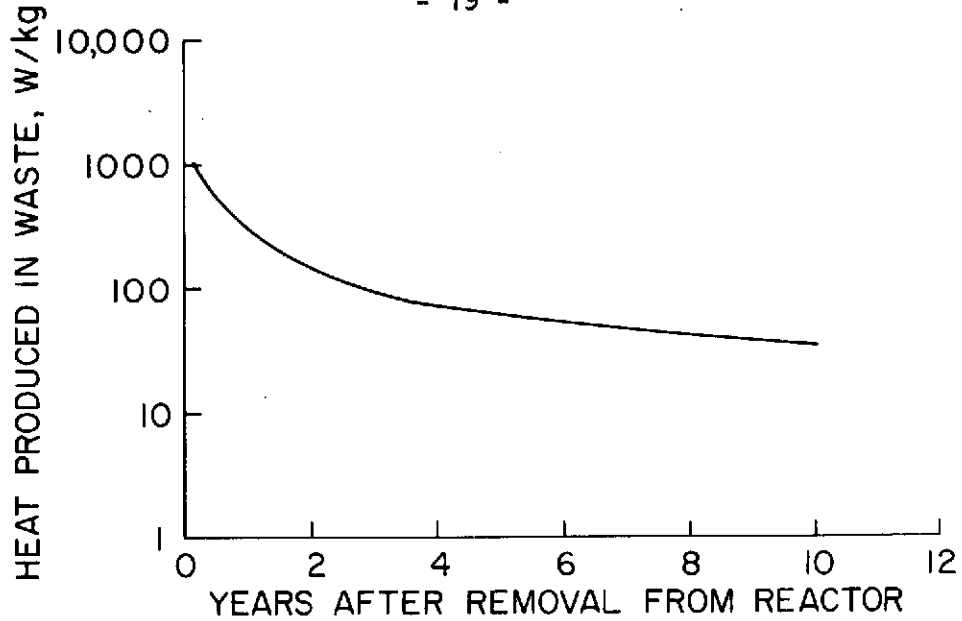
Figure 1. - Fission product thermal power as a function of time after reprocessing of spent fuel elements. (From NASA TMX-68147)
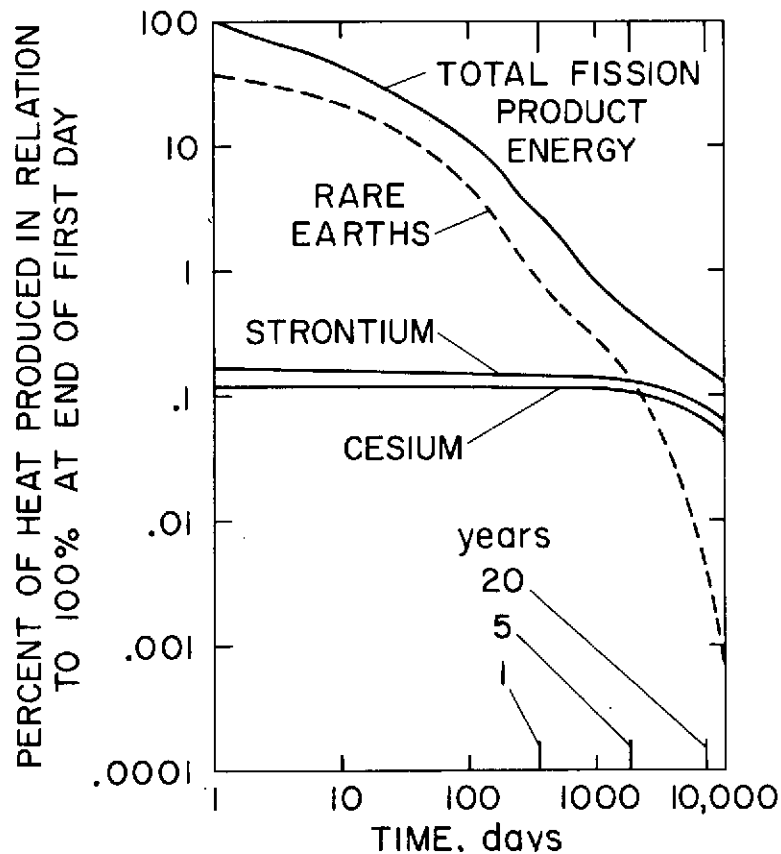

Figure 2. - Heat production in high-level wastes from spent fuel processing. (After five years, Strontium and Cesium account for most of the heat production. Note that removal of these isotopes before several years of aging would have little effect on heat production of the remaining mixture.)
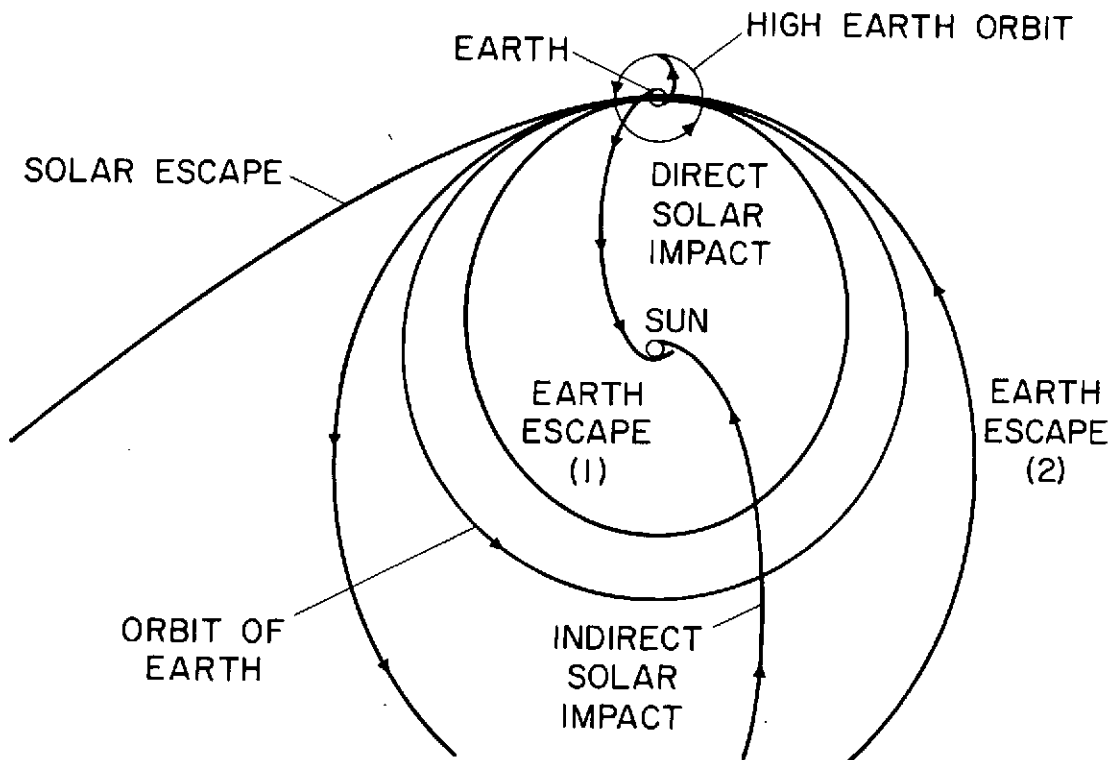
ISOTOPE- THERMAL

PROPELLANT
(LH$_2$ OR
AMMONIA)

ISOTOPE
WASTE
HEAT

H$_2$ ⟶

C≅7 km/sec

ISOTOPE-ELECTRIC

POWER
CONDITIONING

NEUTRALIZER
ELECTRONS

ISOTOPE
WASTE HEAT
SOURCE

PROPELLANT
Hg

ION BEAM

Hg$^+$

C≅30 km/sec

RTG CONVERTER

ELECTROSTATIC
THRUSTERS
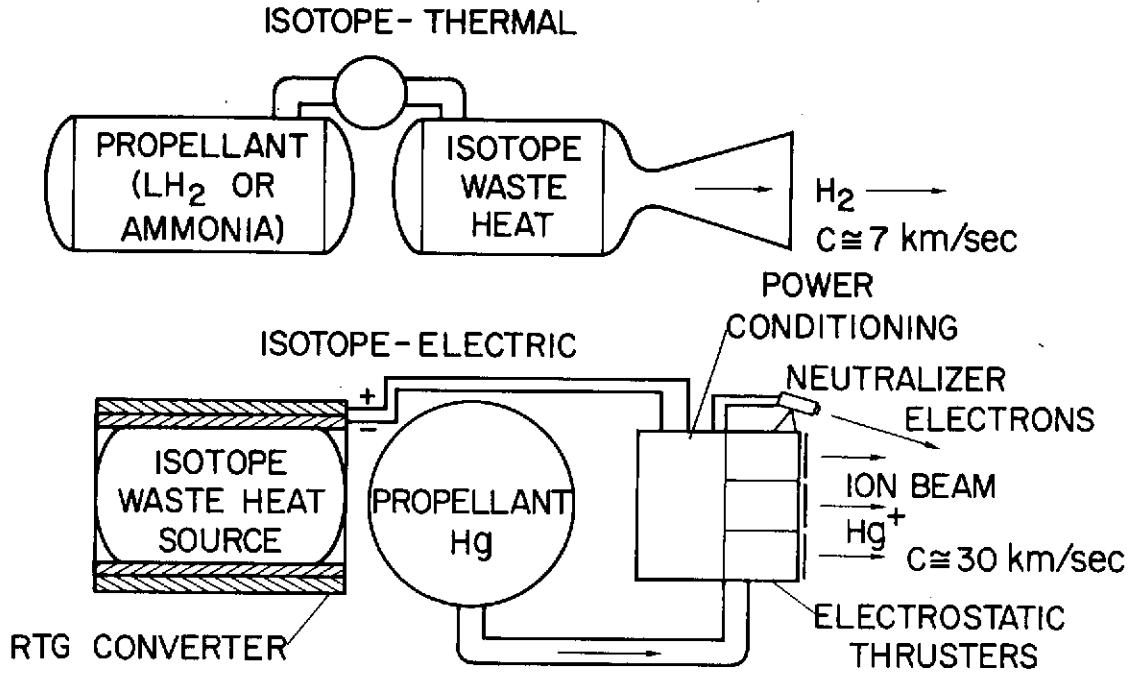
Figure 3. - Isotope waste heat propulsion concepts.

HIGH EARTH ORBIT

EARTH

SOLAR ESCAPE

DIRECT
SOLAR
IMPACT

SUN

EARTH
ESCAPE
(I)

EARTH
ESCAPE
(2)

ORBIT OF
EARTH

INDIRECT
SOLAR
IMPACT

Figure 4. - Nuclear waste disposal space trajectories.
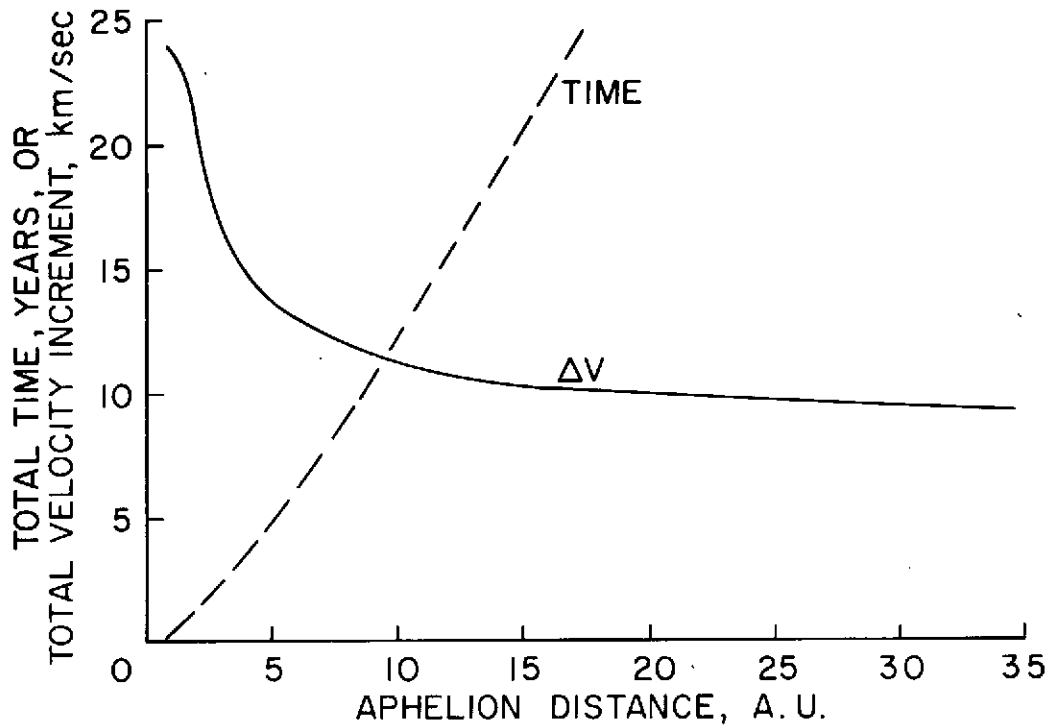
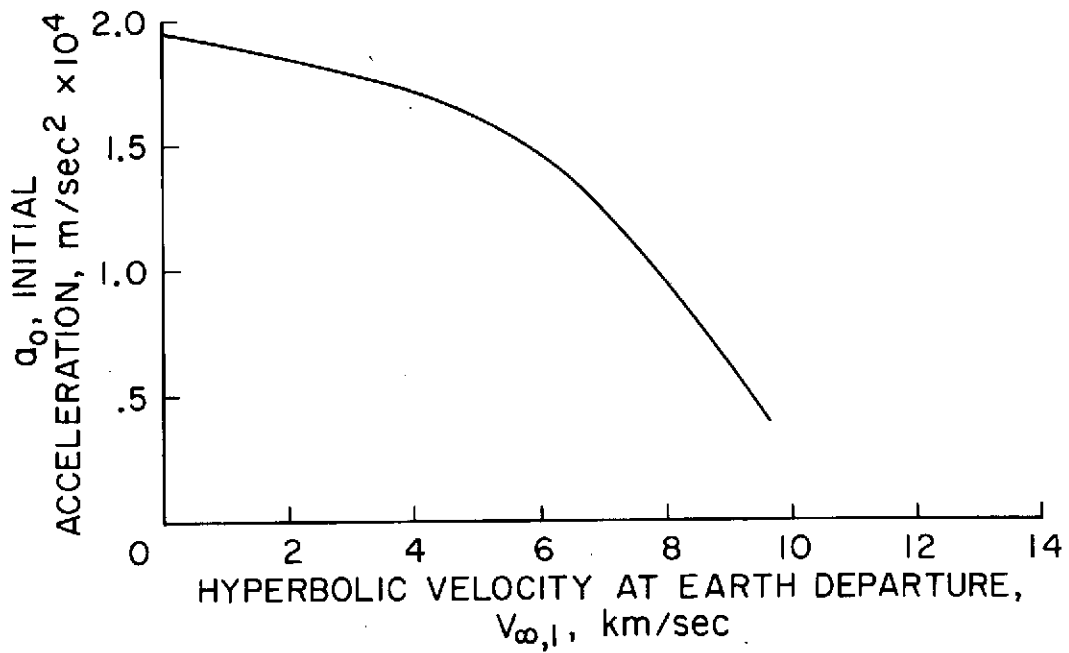Figure 5. - Propulsive velocity increment and flight time for solar impact missions; H_p = 556 km.

Figure 6. - Initial acceleration required for solar system escape mission; t_p = 20,000 hr; I_{sp} = 3000.

$$\eta_c = 0.05$$
$$\eta_{th} = \frac{0.842}{1+(16/C)^2}$$
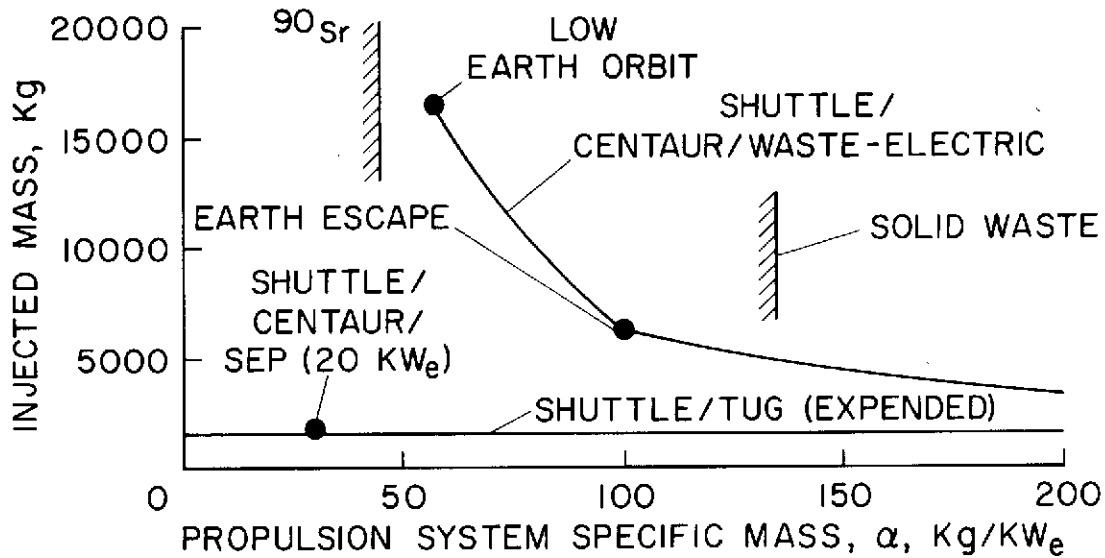$$\alpha \text{ (SEP)} = 30 \text{ Kg/KW}_e$$

Figure 7. - Mass delivered to solar system escape by various propulsion system concepts; shuttle payload = 65,000 lb. at 100 n.mi.; t_p = 20,000 hr.
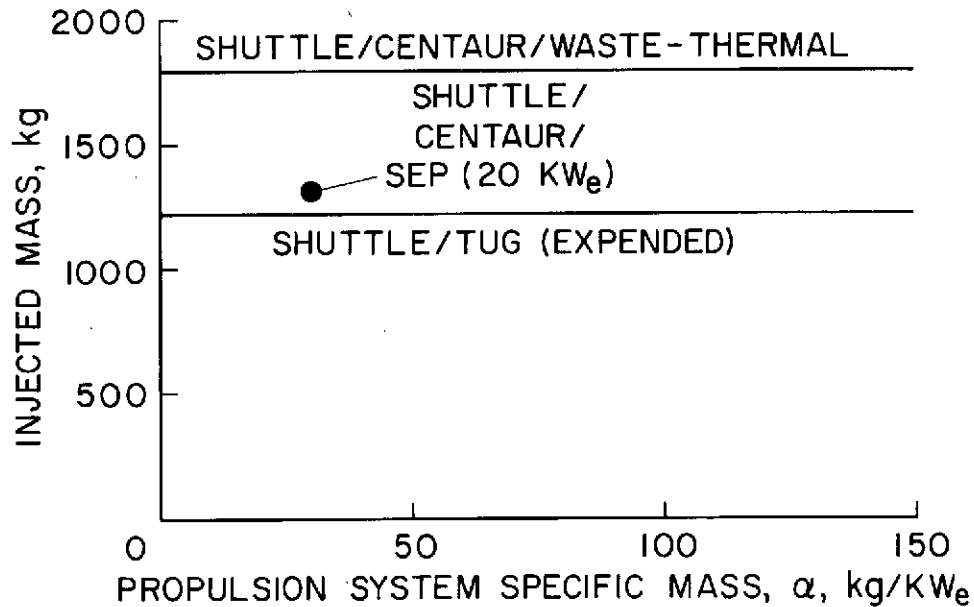

Figure 8. - Mass delivered to solar system escape; comparison of some alternative systems; I_{sp} = 700 (isotope-thermal); α(SEP) = 30 kg/kw.