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PERFORMANCE POTENTIAL OF GAS-CORE AND FUSION ROCKETS: A MISSION APPLICATIONS SURVEY

by Laurence H. Fishbach,
and Edward A. Willis, Jr.
Lewis Research Center
Cleveland, Ohio

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Laurence H. Fishbach, Aerospace Engineer
Edward A. Willis, Jr., Head, Flight Systems Section
NASA-Lewis Research Center
Cleveland, Ohio

Abstract

This paper reports an evaluation of the performance potential of five nuclear rocket engines for four mission classes. These engines are: the regeneratively cooled gas-core nuclear rocket; the light bulb gas-core nuclear rocket; the space-radiator cooled gas-core nuclear rocket; the fusion rocket; and an advanced solid-core nuclear rocket which is included for comparison. The missions considered are: Earth-to-orbit launch; near-Earth space missions; close-interplanetary missions; and distant interplanetary missions. For each of these missions, the capabilities of each rocket engine type are compared in terms of payload ratio for the Earth launch mission or by the initial vehicle mass in Earth orbit for space missions (a measure of initial cost). Other factors which might determine the engine choice are discussed. It is shown that a 60 day manned round trip to Mars is conceivable.

I. Introduction

The relatively low specific impulse I_{sp} of chemical rockets severely restricts mission capability. For missions more difficult than lunar exploration and one-way planetary probes, high propulsive velocity requirements or ΔV's result in extremely large initial vehicle masses at Earth. These high masses in turn imply the need for large and presumably expensive vehicles and launch facilities, and for very complex operations such as orbital assembly, multi-staging, etc. When a more ambitious program of space exploration is considered, there is an evident need for higher specific impulses together with moderate engine weights.

Solid-core nuclear rockets (SCNR) would develop relatively high thrusts at about twice the specific impulse of the best chemical rockets (800 + sec as opposed to 400 + sec). Even this increase in Isp may be inadequate for very energetic, very high payload missions, partly because the SCNR is considerably heavier in relation to its thrust than is the chemical rocket. At the opposite end of the advanced-engine spectrum, electric rockets can develop extremely high specific impulse. On the other hand, their very low thrust implies long propulsion times and in many cases, undesirably long mission times.

In this study we have investigated the performance of several nuclear rockets for a variety of potentially interesting missions. Included in the study are:

1. An advanced solid-core nuclear rocket (SCNR), 930 sec I_{sp}, thrust to engine mass ratio (F/M) of 100 Newtons/kg.

2. Regeneratively cooled gas-core nuclear rockets (REGEN.GCNR), 1000 to 3000 sec I_{sp}, F/M of 14 to 25 N/kg.

3. Light bulb gas-core nuclear rockets (LBGCNR), 1700 to 2650 sec I_{sp}, F/M of 10 to 20 N/kg.

4. Space radiator cooled gas-core nuclear rockets (SRGCNR), 2600 to 6500 sec I_{sp}, F/M of 1 to 3 N/kg.

5. Fusion rockets (FUSION) up to 200,000 sec I_{sp}, power to engine mass ratio (P/M, or 1/α) of 1 kW/kg.

The purpose of this paper is to determine which engine offers the best performance potential for each of four mission classes:

1. Earth-to-orbit launch
2. Near-Earth space missions
3. Close interplanetary missions
4. Distant interplanetary missions.

For Earth-to-orbit launch vehicles, payload ratio is inversely related to total initial mass (vehicle initial mass times the number of launches) required to put up a given payload. Initial mass is presumably a measure of vehicle initial cost. (Costs per unit of payload, however, depend significantly upon whether the vehicle in question is reusable or not.)

For the other three missions, performance is measured in terms of initial mass in Earth orbit (IMEO) required to perform the given mission with the thrust level optimized. In addition, for reusable vehicles, propellant loadings are presented since propellant and payload would have to be replenished for missions subsequent to the first.

II. Symbols

F	thrust, N
g	standard value of gravity, 9.80665 m/sec^2
H/U	hydrogen-to uranium-flow rate ratio
I_{sp}	specific impulse, sec
M_e	mass of engine, kg
M_{is}	interstage structure mass, kg
$M_{jettison}$	mass jettisoned, such as Mars lander, kg
M_p	propellant mass, kg

M_{pay}	payload mass, kg
M_{pstr}	propellant-structure mass, kg
M_{ts}	thrust-structure mass, kg
M_0	mass at beginning of maneuver, kg
α	specific mass of low-thrust propulsion system, kg/kW
V_∞	hyperbolic excess velocity, km/sec
ΔV	propulsive velocity increment, km/sec

Subscripts:

i	i^{th} maneuver
i_{max}	last maneuver

III. Characteristics and Masses of Prospective Nuclear Engines

Before taking up a detailed discussion of missions, trajectories, mass fractions, etc., it is appropriate to briefly review the performance and operating characteristics used for the advanced engine concepts considered in this study.

Solid-Core Nuclear Rocket (SCNR)

This is the only one of the engines to be discussed that could be brought to operational status within a decade. It operates on a familiar principle: the heating of hydrogen to produce thrust. The heating is done by a reactor constructed of solid fuel elements and structure. To keep these parts from melting, the reactor temperature, and hence the hydrogen temperature, is limited to values less than 3300 K. Correspondingly, the maximum specific impulse attainable by this concept is only a little greater than 900 seconds. The NERVA program has already demonstrated that a value of at least 800 seconds can be attained in practice.

Unfortunately, the masses estimated for such engines have increased significantly during the development process. This has resulted, in part, simply from an increased awareness of the need to shield certain vehicle and system elements against nuclear radiation. Nevertheless, from early estimates as high as 1/160 (kg mass per Newton thrust), SCNR specific masses have now increased to values approximating 1/30 or 1/25. For the purposes of this study an improvement in technology has been assumed: A specific impulse of 930 sec for the SCNR was chosen and the engine mass was assumed to be represented by

$$M_e = .01 \times F + 2250 \qquad (1)$$

which may be achievable for very large thrust levels.

The SCNR by its nature retains sizable inventory of radioactive fission fragments and, therefore, creates a severe post-firing radiation hazard. Even if other problems are neglected, it is difficult to see how this one characteristic of the SCNR could be reconciled with the manned, recoverable, rapid-turnaround style of operation envisioned in modern recoverable launch vehicle concepts. Perhaps for these reasons recent SCNR application studies such as reference 1 consider only high orbit, lunar or interplanetary missions with orbital startup.

Regeneratively Cooled Gas-Core Nuclear Rockets (REGEN.GCNR)

One way to avoid the above mentioned limitations is to have the fuel in the form of a fissioning plasma ball which is at extremely high temperatures and heats hydrogen primarily by radiation. The plasma is approximately separated from the hydrogen by hydrodynamic means. A small amount of "seed" material, e.g., tungsten powder, is added to improve the hydrogen's radiative absorption properties.

Basic research has resulted in apparent progress toward a demonstration of feasibility. Laboratory tests applicable to this concept (reference 2) indicate that the necessary types of flow pattern, uranium plasma confinement and separation, etc. can be achieved under conditions suggestive of a GCNR reactor. The favorable engine weight and impulse characteristics predicted by that study (see reference 3) yield good performance for interplanetary missions (reference 4) and also appear promising for Earth launch missions. An advantage in this application is the fact that the fission fragment inventory is not retained in the reactor after shutdown. Thus, it would probably not present a major radiation hazard after shutdown. Unfortunately, it does discharge a small amount of partly-fissioned uranium (e.g., 1 percent by mass of the hydrogen flow rate) at all times during engine operation, and this is evidently an undesirable feature from the atmospheric-pollution viewpoint if an Earth to orbit launch vehicle with low-altitude startup is envisioned.

This engine might also be undesirable for near-Earth and lunar missions since it would tend to contaminate the area with radioactive wastes due to the large numbers of such missions that would be contemplated.

The mass of the regeneratively cooled GCNR is based on equations presented in reference 4 (which in turn updated those of references 2 and 3). These equations relate engine mass to specific impulse, engine diameter, hydrogen to uranium flow rate ratio, critical mass, and thrust. For the REGEN.GCNR's studied herein, the diameter is 3.66 m, the critical mass is 48 kg, the flow rate ratio is 100 and the specific impulse is 2500 secs. The engine mass (reference 4) becomes

$$M_e = 0.1522F^{0.6169} + 8.113 \times 10^{-5}F^{1.2554} + 4189\left[(5.184 + 6.534 \times 10^{-4}F^{0.3831})^3 - 139.31\right] + 54365 \qquad (2)$$

Light Bulb Gas-Core Nuclear Rockets (LBGCNR)

The light bulb gas-core nuclear rocket (reference 5) is essentially the same as the

REGEN.GCNR. The major differences are that the light bulb employs a transparent mechanical barrier to separate the uranium plasma from the hydrogen and has a small radiator to remove excess heat. It, therefore, has all the benefits of the REGEN.GCNR, but does not discharge radioactive debris. Thus, the LBGCNR appears to be an attractive candidate for near-Earth missions.

The mass and specific impulse of the LBGCNR are given in Table I as a function of thrust. These values are from computations by United Aircraft Research Laboratories based on a radiator mass of 135 kg/MW and a chamber pressure of 1000 atmospheres.

TABLE I: LBGCNR DATA

F-n	M_e-kg	I_{sp}-sec
133 370	14 050	1780
222 370	15 650	1905
311 360	17 200	1990
400 360	19 050	2050
667 100	23 125	2180
1 334 200	34 475	2355
1 778 900	44 000	2425
2 668 400	83 000	2530
3 113 100	122 450	2570
3 557 900	204 075	2605
4 002 800	385 500	2635

Space Radiator Cooled Gas-Core Nuclear Rockets (SRGCNR)

The space radiator cooled gas-core rocket is identical to the REGEN.GCNR with the addition of a heat exchanger loop. These two rockets are shown schematically in Figure 1. The addition of the radiator allows for the cooling of the reactor walls and moderator without using up the regenerative cooling capacity of the liquid hydrogen. The latter is used selectively to cool critical areas such as the nozzle throat. These measures allow the core to operate at much higher temperatures than those of the REGEN.GCNR and hence, at higher specific impulse.

The SRGCNR engine mass is also based on tabular lookup and interpolation of mass as a function of thrust, radiator specific mass and chamber pressure. For this study the radiator specific mass was also chosen to be 135 kg/MW and the chamber pressure 1000 atmospheres. Table II presents the tabular data used for these conditions (reference 6).

TABLE II: SRGCNR DATA

F-n	Me-kg	I_{sp}-sec
22 240	36 280	3400
44 470	45 590	4150
88 950	68 210	4850
133 420	82 830	5200
177 900	101 440	5500
222 370	120 060	5700
266 850	136 680	5800
311 320	157 300	5900
355 800	175 920	6000
400 270	194 530	6000
444 750	213 150	6000

Fusion Rockets (Fusion)

In the fusion rocket (reference 7) a fusion fuel such as a mixture of deuterium and helium-3 is injected into a reaction chamber. A few percent of the injected fuel undergoes fusion reactions. The energy released heats the unreacted fuel to extremely high temperatures where it ionizes to form a plasma. Magnetic fields hold the plasma fuel away from the reaction chamber walls and divert some of it into a mixing chamber. Hydrogen propellant is injected into the mixing chamber where it is ionized and heated. The thermal energy of the propellant is converted into directed motion by a magnetic nozzle to produce thrust. Mixing with a propellant is required because the escaping fusion-reaction products by themselves would have a specific impulse in the range of 200 000 sec, far above the optimal value for most planetary missions. By adjusting the amount of hydrogen added, the specific impulse can be varied as desired throughout the mission.

The specific powerplant mass, α, has been taken to be 1 kg/kW as estimated in reference 7. This reference assumes a radiator specific mass of about 15 kg/MW as compared to the 135 kg/ MW used herein for the LBGCNR and SRGCNR.

IV. Propulsive Velocity Increments

The required propellant loading for each stage may be easily computed from the chemical rocket equation, i.e.

$$(M_p)_i = (M_0)_i \ (1 - \exp(-\Delta Vi/I_{sp}g) \qquad (3)$$

The propulsive velocity increment or ΔV ultimately is obtained from trajectory simulations. For Earth-to-orbit launch missions, the launch-trajectory code of reference 8 was used. This comprises realistic Earth and atmosphere models, representative vehicle aerodynamic coefficients, and an accurate numerical-integration package. Calculus of variations steering logic is used above the sensible atmosphere to maximize the payload delivered.

Ideal impulsive ΔV's for outer-planet missions were taken from reference 9, a standard trajectory data compilation. Similar data for Mars missions were obtained from unpublished Lewis data which, like reference 9, was generated by means of the trajectory program described in reference 10. Precomputed gravity loss data (c.f. reference 11) was tabulated and used by means of a tabular lookup routine to account for ΔV penalties for finite thrust. This permits the optimization of thrust level without re-integrating the actual trajectories.

For the fusion rocket, an approximate low-thrust trajectory computer code based on reference 12 was used. In order to facilitate comparison with the higher thrust systems, all planet physical constants and circular/coplanar orbit elements were chosen to match those of reference 9.

V. Vehicle Mass

Vehicle mass is calculated by summing the masses required for each maneuver. As previously mentioned, propellant mass is calculated from equation (3). The initial mass for the next maneuver is then

$$(M_0)_{i+1} = (M_0)_i - (M_p)_i - (M_{pstr})_i - M_{jettison} - (M_{is})_i \qquad (4)$$

The final maneuver requires that the initial mass equal

$$(M_0)_{i_{max}} = (M_p)_{i_{max}} + M_{pstr} + M_{ts} + M_{pay} + M_e \qquad (5)$$

$$M_{ts} = 0.002 \times F \qquad (6)$$

$$(M_{pstr})_i = 0.20 \times (M_p)_i \qquad (7)$$

$$(M_{is})_i = 0.01 \times (M_0)_i \qquad (8)$$

For Mars and major planet missions,

$$M_{jettison} = 150\ 000 \text{ kg} \qquad (9)$$

For the missions studied herein there are usually two payloads, one left at the destination which is called $M_{jettison}$ and one returned to Earth consisting of a command module, crew, etc. called M_{pay}.

VI. Earth-to-Orbit Launch

Launch costs have been an item of major concern since the earliest days of the space program. Cost reduction principles yield only incremental improvements in cost effectiveness as long as the launch vehicle is discarded after each use. In order to reduce launch costs to really attractive levels; e.g., less than $400 per kilogram in orbit, a fully reusable system, "the space shuttle" concept, is under intensive study at this time. As presently conceived, it consists of two chemical rocket-propelled airplane-like stages.

Unfortunately, the low I_{sp} of chemical engines combined with the considerable mass of reentry structure result in an uncomfortably small payload ratio, e.g., about 1 percent of initial mass. An alternative approach which could be considered for a later operational date would involve the use of advanced nuclear engines. By taking advantage of the presumed high specific impulse and other favorable characteristics of (e.g.) the LBGCNR, it is possible in principle to achieve excellent payload ratios and to do it with a single stage vehicle. This latter concept, illustrated in figure 2, has apparent advantages in terms of operational simplicity and a presumably lower initial cost.

A parametric study of this possibility is presented in figure 3 where burnout mass (excluding the nuclear engine) is plotted against the nuclear engine's specific impulse and specific mass (mass per unit thrust) for a typical Earth surface to low circular parking orbit launch mission. In all cases, the nuclear engine thrust level is optimized, and is generally only slightly greater than the vehicle's initial gross weight. Note that the burnout mass parameter includes structure, other dead mass, and payload. (Typical structure plus dead mass fractions for chemical shuttle vehicles are indicated by the horizontal band on the figure.) The payload is then the difference between the structural mass fraction and the burnout fraction shown on the figure. For example, a hypothetical engine of 1800 sec I_{sp} and specific mass of 0.02 kg/Newton would yield a 0.30 burnout mass fraction. Assuming a 0.25 dead mass fraction, a comfortable 5 percent payload fraction remains.

Unfortunately, even the advanced SCNR (930 sec I_{sp}) would require a specific mass of 0.005 (less than half the presently estimated value, c.f. equation (1)) before it could deliver any payload at all. To deliver a 5 percent payload ratio, it would have to be essentially massless. The outlook for the GCNR type of engine is considerably better. Based on the most optimistic combination of present estimates of I_{sp} and engine specific mass, payload ratios of 15 percent may be obtainable. Hence, it is concluded provisionally (pending more refined structure and engine mass analyses) that the LBGCNR is an attractive candidate for this mission.

VII. Near-Earth Missions

Lunar Ferry

The Lunar Ferry mission and its associated vehicle are illustrated schematically in figure 4. In this mission, the reusable nuclear rocket stage, which is initially in a parking orbit about Earth, follows a minimum energy transfer trajectory to deliver various amounts of payload into a lunar orbit. The vehicle then returns with a 50 000 kg payload (crew, command module, etc.) on a minimum energy transfer to Earth and into the original parking orbit. There it would be refueled, pick up another lunar payload, and depart for lunar orbit. The capability to deliver large payloads to the moon will be a necessity if and when permanent lunar bases or colonies are formed.

The performance of four of the nuclear rocket concepts was measured for this mission. The fusion rocket was eliminated because its low thrust levels result in excessive mission times. The results are shown in figure 5. The SCNR requires the highest IMEO for this mission. The LBGCNR and REGEN.GCNR each offer a potential 20 percent reduction in IMEO over that of the SCNR. The reader may recall that these two engines are very similar in performance; the major difference between them is the method of separating the hydrogen from the uranium plasma. Their specific impulses are about the same, as are their masses.

The SRGCNR requires about 15 percent less IMEO than the LBGCNR and REGEN.GCNR, and about 35 percent less than the SCNR.

Since this vehicle will be reused, it may be more meaningful to compare the performance of the four rockets on the basis of propellant loading. This is what would have to be replenished after each mission. The differences are dramatic. The SRGCNR requires only 25 percent of the propellant of the SCNR and 60 percent of that of the LBGCNR and REGEN.GCNR. The propellant requirements are strongly determined by the I_{sp} of the engine and this more than compensates for the increase in engine mass of the SRGCNR due to the space radiator.

The REGEN.GCNR and SRGCNR radiative emissions might be of concern for near-Earth space missions. Although the high jet velocity of the particles should allow them to escape Earth's gravity, the physics of the situation are not well defined. Therefore, for a large number of missions such as contemplated for a space tug or lunar ferry, these emissions might be of concern. The LBGCNR, which retains the fission products, might then emerge as the most likely candidate for this mission. A sensitivity study of IMEO for 500 000 kg of payload to specific impulse level of the LBGCNR is shown in figure 7. This study assumed that the thrust and mass values in Table I remained constant while the specific impulse increased or decreased by 50 percent from the base values shown. The thrust level for the mission was then reoptimized. The value of specific impulse for the base case was 2240 sec. As can be seen from the figure, going to 3000 sec results in an IMEO reduction of about 7 percent while dropping to 1000 sec results in an IMEO increase of 40 percent.

Slingshot

The second near-Earth mission studied is the "slingshot", which is essentially an advanced version of the "space tug". This mission and its associated vehicle are illustrated schematically in figure 8. The vehicle is initially in an Earth parking orbit, then boosts out of orbit to a given hyperbolic excess velocity, V_∞, and separates, with a 500 000 kg payload continuing along the initial path. The nuclear rocket then retrofires, returning to low Earth orbit where an additional impulse places the vehicle with a 50 000 kg payload back into a circular parking orbit. This mission is analogous to the reusable launch vehicle; the "slingshot" is ready to boost another payload as soon as it has been refueled.

The same four engines are studied for this mission as for the lunar ferry. The results are shown in figure 9. The SCNR again requires the highest IMEO. The LBGCNR and REGEN.GCNR performance curves were indistinguishable and have been plotted together. The SRGCNR again shows the best performance. At a V_∞ of 5.5 km/sec (about the requirement for a Earth departure maneuver for a 300 day Mars round trip) the SRGCNR requires 93 percent as much IMEO as the LBGCNR and 70 percent as much as the SCNR. The savings of 7 and 30 percent, respectively, increase with increasing V_∞ since propellant loading increases and the higher I_{sp} of the SRGCNR proves more beneficial. As in the lunar ferry mission, the differences in terms of propellant loading requirements is considerably larger with the gas cores having a 3 or 4 to 1 advantage over the solid core rocket at high energies.

VIII. Manned Interplanetary Missions

A typical recoverable manned interplanetary vehicle is shown in figure 10. Manned interplanetary trips are of two major types: near planets (exemplified by Mars); and far planets (Jupiter and beyond).

Mars Mission

Fast Missions. For fast missions to Mars, the vehicle is assumed to start in a 600 km circular Earth parking orbit, proceed to Mars, enter a 0.9 eccentricity parking orbit with periapsis at 1.1 planet radii and then return to Earth with a 50 000 kg payload (command module, crew, etc.) plus a reentry vehicle and no mass is jettisoned at Mars. The reentry vehicle employs atmospheric braking with no limit on entry velocity (actual velocities turn out to be between 2 and 3 times the circular velocity in Earth orbit).

This mission is called a "courier" and in some respects resembles the circumlunar flight of Apollo 8. As shown in figure 11, three engines are candidates for this type of mission, the three gas core nuclear rockets. This mission requires between 60 and 200 days round trip travel time. The SRGCNR can do the trip in 60 days for an IMEO of 2 million kg. The "Skylab" program will hopefully have demonstrated by the time an SRGCNR can be built, that man can stay in space for periods of this duration.

By increasing travel time, IMEO can be reduced dramatically. At 80 days, for example, an IMEO of only 1 million kg is required. For the rest of this discussion on Mars trips it is assumed that the 80 day fast trip would be of prime interest since it has only half the IMEO of the 60 day trip yet does not assume any appreciable extension of space time over the Skylab program.

The emission of radioactive waste of the SRGCNR is not thought to be significant for a mission of this nature since there would not be as many repetitions. The high I_{sp} of the SRGCNR at the same time as having high thrust capability (compared to electric or even fusion rockets) make it a clear choice for this mission. The shortest travel time that either of the other GCNR's can do the mission in is about 100 days and even then require 6 times as much IMEO as the SRGCNR.

Conventional Mars Missions. Most studies of Mars round trip missions have considered trip times of 1 year and longer (e.g., reference 13). These missions have much lower ΔV requirements than the fast ones; thus, they can be more ambitious in terms of large payloads, reusable vehicles, etc., and still result in reasonable IMEO's.

These "Science/Exploration" missions also start in low Earth orbit and proceed to a 0.9 eccentricity parking orbit at Mars. The transfer times and angles are much longer, however, than those for the courier missions. The vehicle remains in Martian orbit for 40 days. A payload of 150 000 kg is left which might have been used to go down to the surface, build an orbiting observatory, etc. The vehicle then returns to Earth with a payload of 100 000 kg and reenters the initial parking orbit using propulsive braking.

All five of the nuclear rocket types in this paper were studied for this mission. As previously mentioned, even the 930 sec I_{sp} SCNR may not be sufficient for high energy, high payload missions. This was the case for this mission. In order to bring the IMEO for SCNR trips into the 1 to 2 million kg range, it was necessary to reduce the Earth return payload to 50 000 kg and reenter atmospherically (not enter into the initial parking orbit). Thus, this vehicle is not recovered.

The results of the study are also shown in figure 11. Even with an easier mission profile the SCNR still requires the highest IMEO. The LBGCNR and REGEN.GCNR again yield almost identical results. A 500 day mission requires about 1 million kg IMEO. The trip time can be reduced to 1 year for an increase in IMEO to about 1.6 million kg.

The SRGCNR appears to offer significant savings in terms of IMEO or trip time and, in addition, a flattening of the curve. At 500 days the SRGCNR requires only 60 percent as much IMEO as the LBGCNR. On the other hand, for a 1 million kg IMEO, the SRGCNR can do the mission in 280 days or a reduction of 45 percent in mission time.

The fusion rocket appears promising at long trip times, excelling the other four engines. A rocket of this nature propels for a significant portion of the trip and, thus, can follow more optimum flight paths.

With decreasing trip time, the long propulsion times of the fusion rocket force it to progressively higher thrust and lower specific impulse. This increases propellant mass and powerplant fraction resulting in a crossover between the SRGCNR and fusion rocket curves at about 1 year mission time.

Based on the results illustrated herein, it would appear that SRGCNR would be the best candidate for these Science/Exploration missions to Mars. Trip times are relatively short and potential mass savings of the fusion rocket do not appear to be decisively large. The story could be changed, however, if the fusion rocket had a specific powerplant mass of 0.5 kg/kW instead of 1.0. This would shift the fusion rocket curve down and to the left, resulting in higher mass savings and reduced mission times. It should be pointed out that the equivalent specific powerplant mass or α of the SRGCNR is about .01 to .1 kg/kW, but it suffers somewhat in not being able to achieve the high I_{sp} of the fusion rocket.

The uranium requirements for the courier and Science/Exploration missions are shown in figure 12 for hydrogen to uranium flow rate ratios of 100 and 200 for the SRGCNR and for a uranium mass to thrust ratio of .008 kg/N for the SCNR. The LBGCNR fuel requirements are very small since no fuel is lost and is determined by burnup rate and reuses. As an example, a 400 000 Newton thrust engine contains about 45 kg of uranium. The shape of these curves is the same as those for IMEO and the relative magnitudes are about the same also. The propellant requirements can be calculated by multiplying the uranium requirements by the H/U ratio. This propellant requirements is important since the Science/Exploration vehicles are reusable. The 80 day SRGCNR Courier requires about $\$35 \times 10^6$ worth of U^{235} (3500 kg x $\$10^4$/kg) when H/U = 200. Although a considerable sum, this in percentage terms, does not represent a major cost increment when compared (e.g.) to a direct cost of about $\$220 \times 10^6$ for launch operation alone (10^6 kg IMEO x about \$220/kg in orbit).

The optimum thrust levels for LBGCNR's and SRGCNR's are shown in figure 13. The LBGCNR thrust levels are about 5 times those of the SRGCNR. Since the I_{sp} level is about 1/2 that of the SRGCNR, the LBGCNR reactor power is 2.5 times that of the SRGCNR. It appears that one SRGCNR reactor could be used to perform a large portion of the missions whereas the optimum thrust level changes more rapidly with mission time for the LBGCNR.

This point is demonstrated more clearly in figure 14 where it can be seen that for the 80 day Courier and 400 day Science/Exploration missions, IMEO is fairly insensitive to the thrust level of the SRGCNR over quite a long range. A 150 000 Newton engine would perform both missions with essentially the same IMEO as the optimum thrust engine (180 000/80 days, 130 000/400 days). Thus it is conceivable that a standard-design SRGCNR might be usable for a large variety of space missions.

Since the IMEO appears to be relatively insensitive to the thrust level, and since there is only a 2 to 1 variation in optimum F for the LBGCNR (fig. 13), perhaps a fixed-size LBGCNR would also have multi-mission capability.

The SRGCNR discussed so far operated at a chamber pressure of 1000 atm. If the chamber pressure is increased to 2000 atm, the achievable specific impulse increases by approximately 1000 sec at a thrust level of 50 000 Newtons to 500 sec at 500 000 Newtons. If the chamber pressure is decreased to 500 atm the I_{sp} decreases by 1000 sec at 50 000 Newtons to 600 sec at 500 000 Newtons. The engine mass does not change significantly at 50 000 Newtons and is $\pm$ 10 000 kg at 500 000 Newtons. The effect of achievable specific impulse is shown in figure 15. High I_{sp} is for a chamber pressure of 2000 atm, Nominal for 1000 atm. (table II) and Low for 500 atm.

The effect of I_{sp} is small for the 400 day Science/Exploration mission and strong for the much higher energy 80 day Courier mission. For the 500 atm engine 80 day mission a 40 percent increase in IMEO is shown while the 2000 atm engine decreases the IMEO by 20 percent.

<u>Major Planet Missions</u>

The last class of missions studied are trips to the major planets Jupiter, Saturn, and Uranus.

A peculiarity of major planet round trips, as discussed in reference 9, is that feasible trips for relatively high thrust rockets such as those discussed herein (exclusive of the fusion rocket) occur only at discrete intervals of 12 to 13 months. Thus, for these rockets, missions do not exist at times intermediate to the data points shown in figure 16 which have been connected with straight lines to identify the rocket type and indicate trends.

Consider first the Jupiter missions shown in figure 16(a). For these missions there are again two modes: Courier (no payload to planet, 50 000 kg back to Earth and atmospheric reentry) and Science/Exploration (150 000 kg to planet, 100 000 kg to Earth, 200 day stay time in 0.9 eccentricity parking orbit with periapsis at 1.1 planet radii, and recovery into low Earth orbit).

The Courier mode requires 1.67 years and an IMEO of 350 000 kg for the SRGCNR and 840 000 kg for the LBGCNR. The fusion rocket which is shown by the long-short-short dashed curve gives continuous performance but cannot perform a 1.67 year trip to Jupiter.

The next opportunity for high thrust rockets occurs at 2.8 years. The IMEO increases over that at 1.67 years as a result of switching from Courier to Science/Exploration type trips. At this trip time, fusion and LBGCNR IMEO's are at about 1.4 million kg while the SRGCNR requires 750 000 kg or about 54 percent as much.

Going to longer trip time, the IMEO for the fusion rockets drops rapidly and for trips beyond 4.3 years outperforms the other two rockets. As missions become difficult and trip times longer, the practically unlimited I_{sp} capability of fusion rockets makes them increasingly attractive.

Thus in figure 16(b) it can be seen that fusion rockets always outperform LBGCNR's for Saturn missions and offers slight improvement over SRGCNR's beyond 4.8 years.

For missions to Uranus (fig. 16(c)) the fusion rocket outperforms both GCNR's for Science/Exploration missions. Even so, the performance gains of the FUSION rocket are not decisively large for the missions considered here. By extrapolating the trends shown in figures 16(a), (b), and (c), the fusion rocket appears to be the best candidate for very high ΔV, very long time missions such as Neptune and Pluto round trips and Solar System escape. These missions were not considered here.

IX. Concluding Remarks

For the five nuclear rockets studied herein the space-radiator-cooled gas-core rocket appears to always require the least IMEO for the missions studied if excessive trip times are ruled out. Other ecological restructions may make the light-bulb gas-core nuclear rocket the choice for the near-Earth missions since it does not emit radioactive wastes.

A significant class of new fast trips to Mars has been proposed where for mission times as low as 80 days, manned round trips can be accomplished for only 1 million kg IMEO. In addition, should trip times be of prime importance, 60 day duration round trips to Mars appear feasible for a 2 million kg IMEO.

All of the gas-core and fusion concepts examined promise appreciably better performance than the solid-core engine. However, selection of a preferred concept must await continued work to establish the feasibility of the concepts and better definition of engine characteristics, followed by consideration of development and operational costs.

X. References

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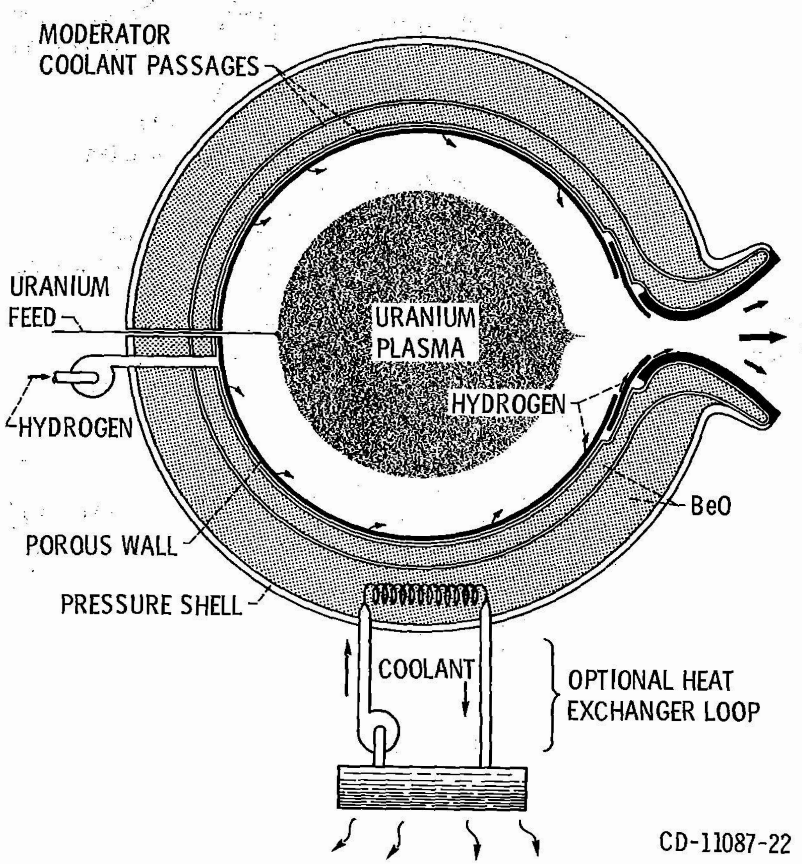

Figure 1. - Open cycle gas-core nuclear rocket engine - schematic diagram.

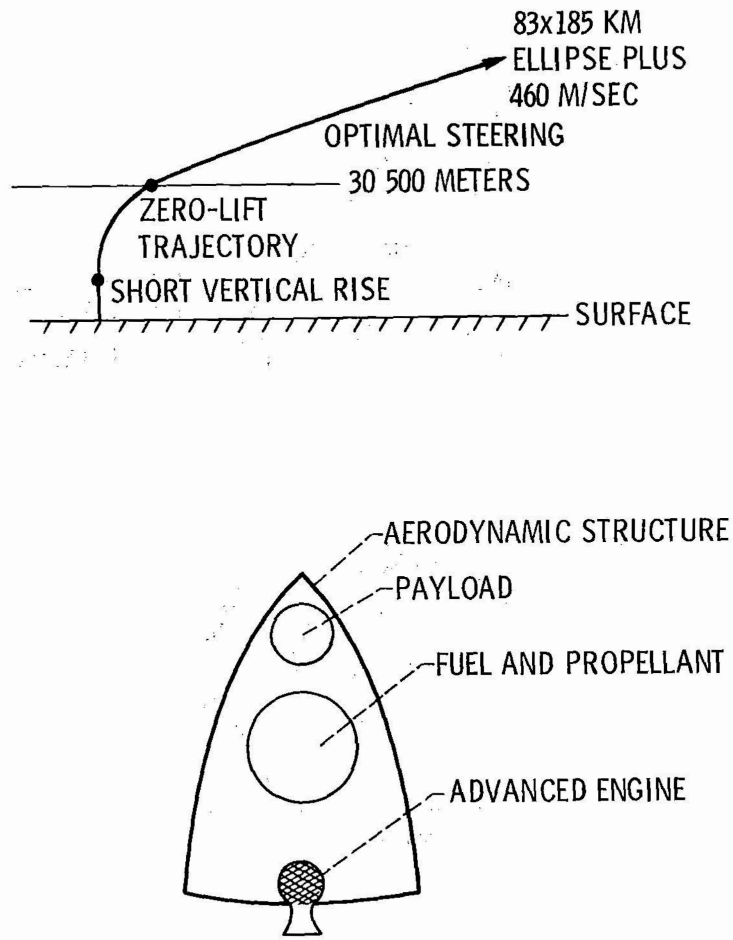

Figure 2. - Earth launch vehicle and trajectory.

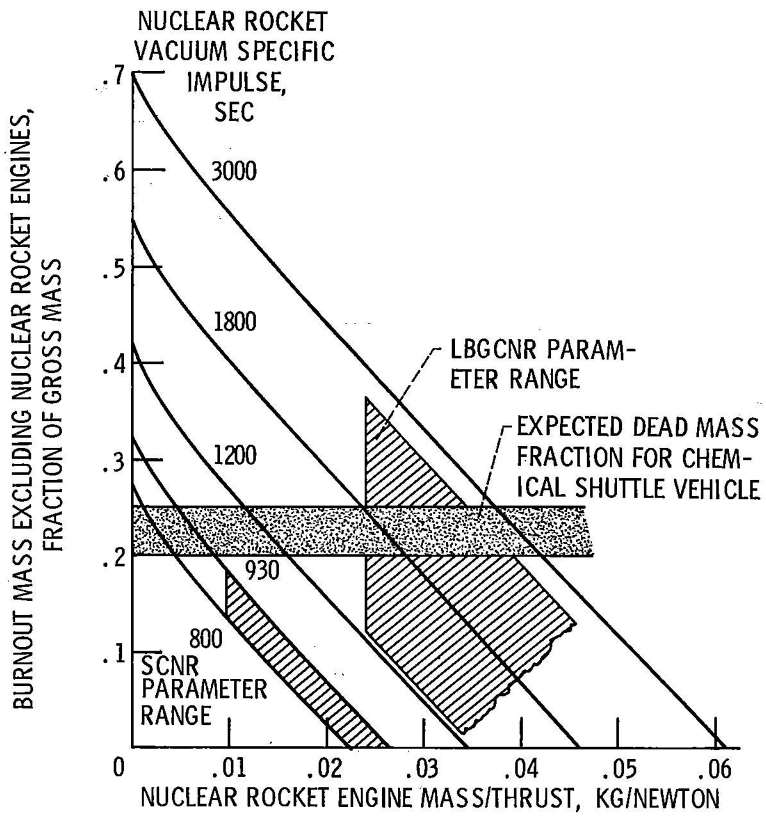

Figure 3. - Performance of single stage nuclear rocket launch vehicle. Final orbit: 83x185 km, 460 m/sec ΔV reserve.

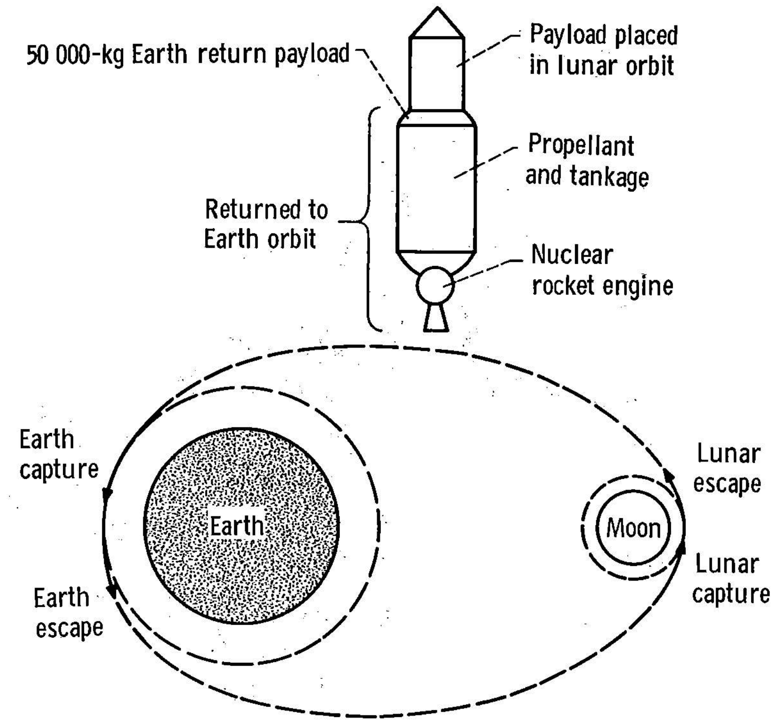

Figure 4. - Lunar ferry mission.

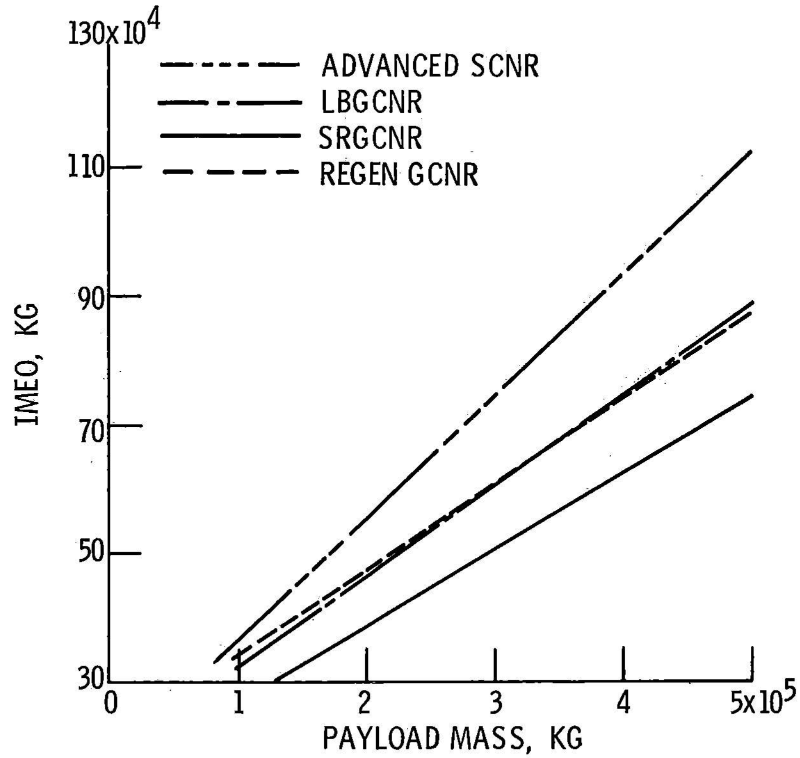

Figure 5. - Effect of payload mass on lunar ferry mission.

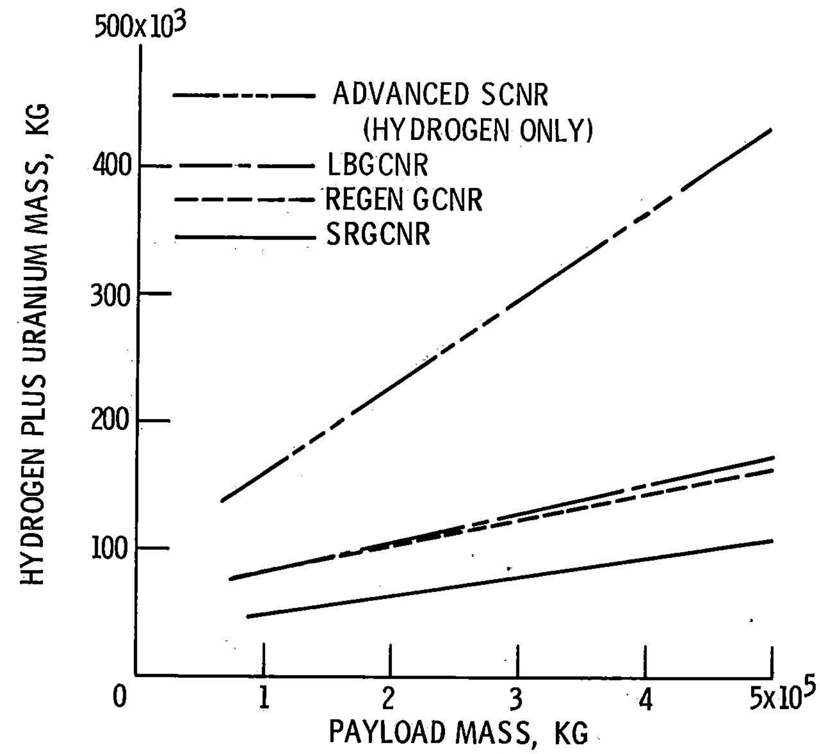

Figure 6. - Propellant requirements for lunar ferry mission.

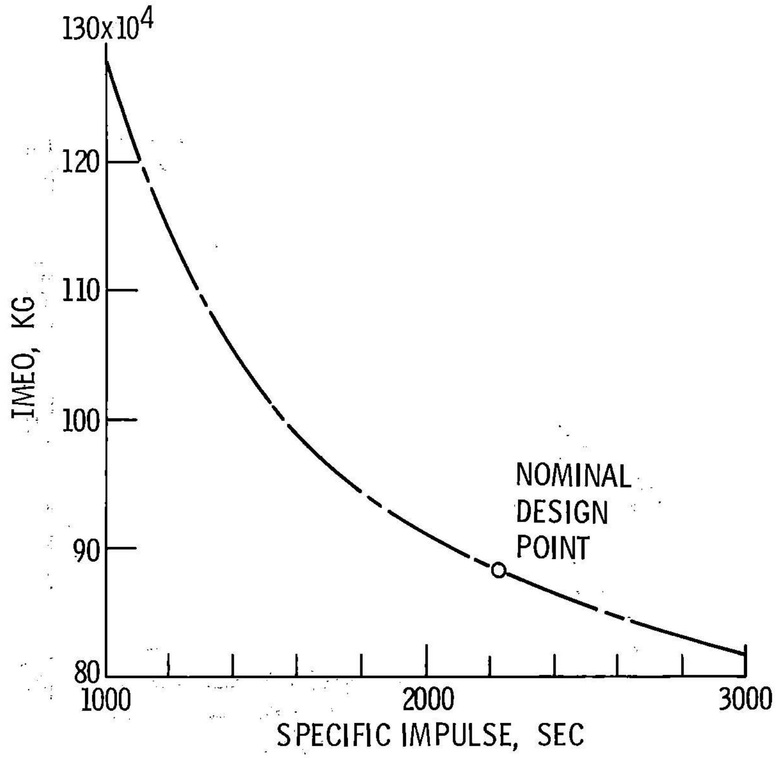

Figure 7. - Effect of specific impulse on lunar ferry mission. Light bulb gas core nuclear engine, 500 000 kg payload.

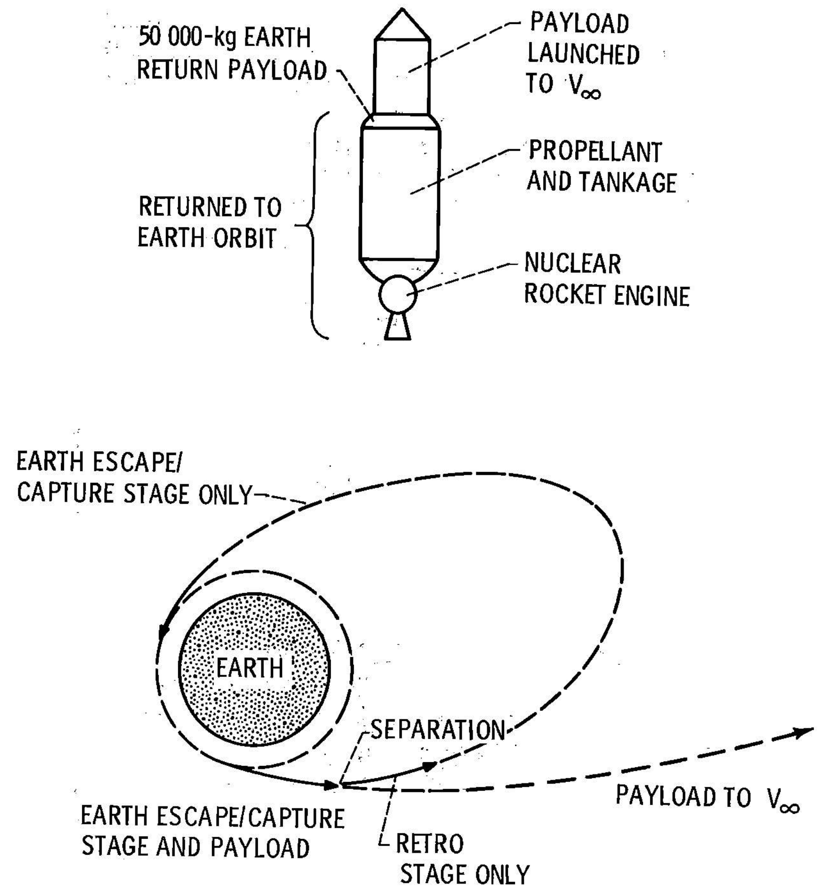

Figure 8. - Slingshot mission.

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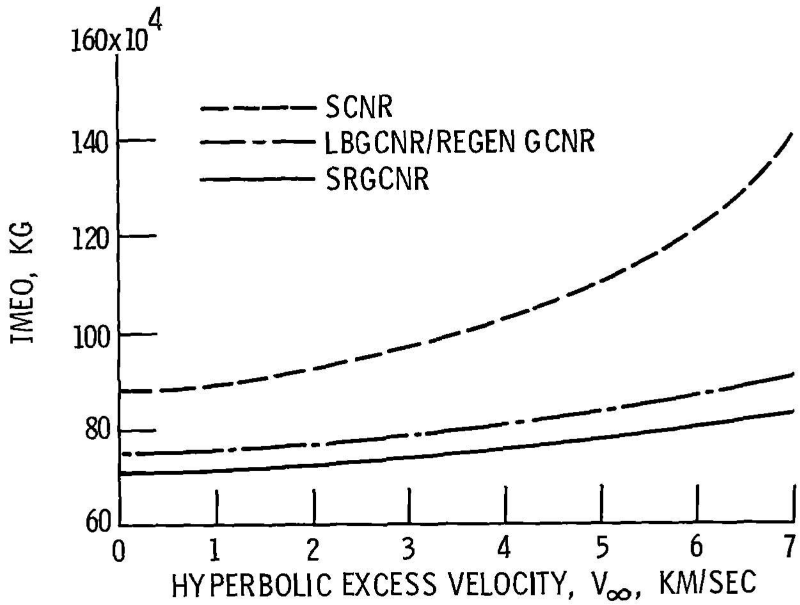

Figure 9. - Slingshot mission to various hyperbolic excess velocities.

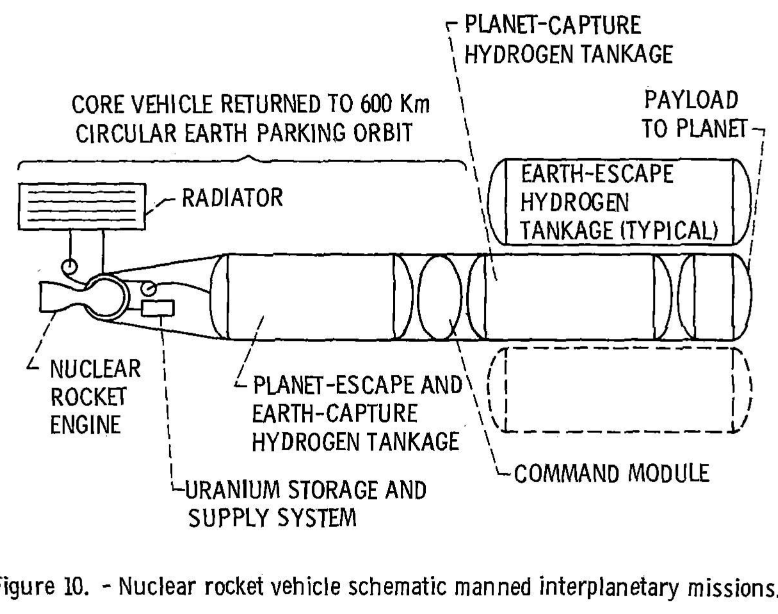

Figure 10. - Nuclear rocket vehicle schematic manned interplanetary missions.

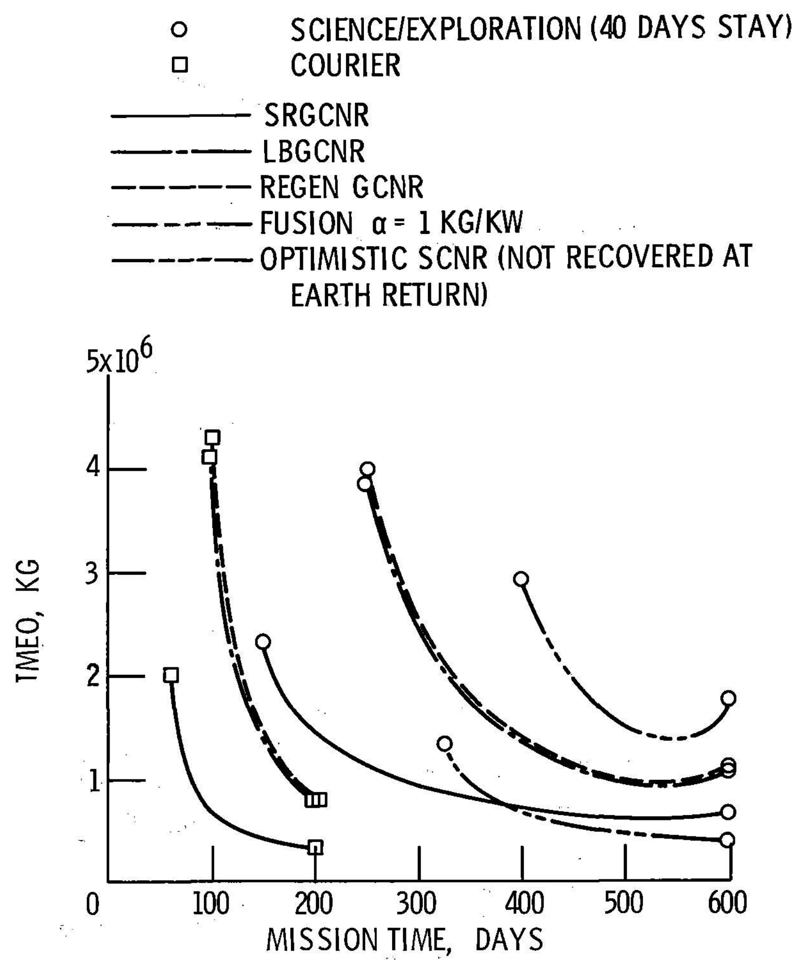

Figure 11. - Effect of Mars round trip mission time, for various nuclear rocket engines.

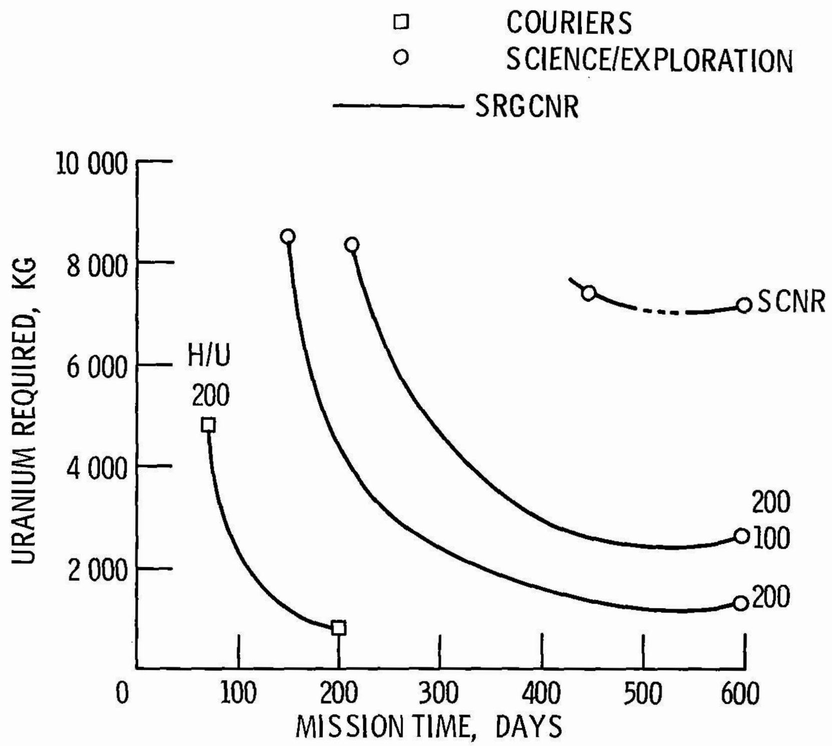

Figure 12. - Total uranium propellant requirements for SCNR and for SRGCNR rocket engines as a function of propellant to fuel flow rate ratios. Mars round trips.

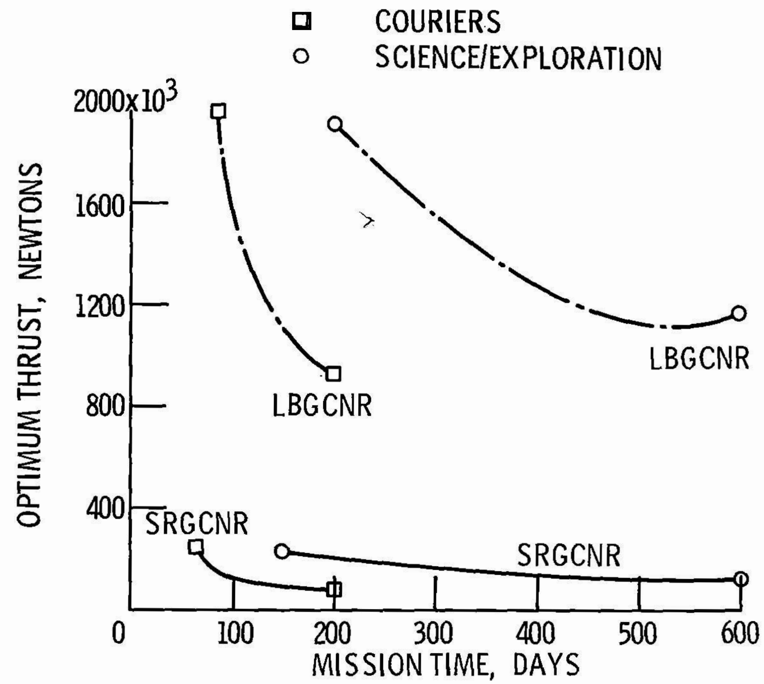

Figure 13. - Optimum thrust levels for various nuclear rocket engines. Mars round trips.

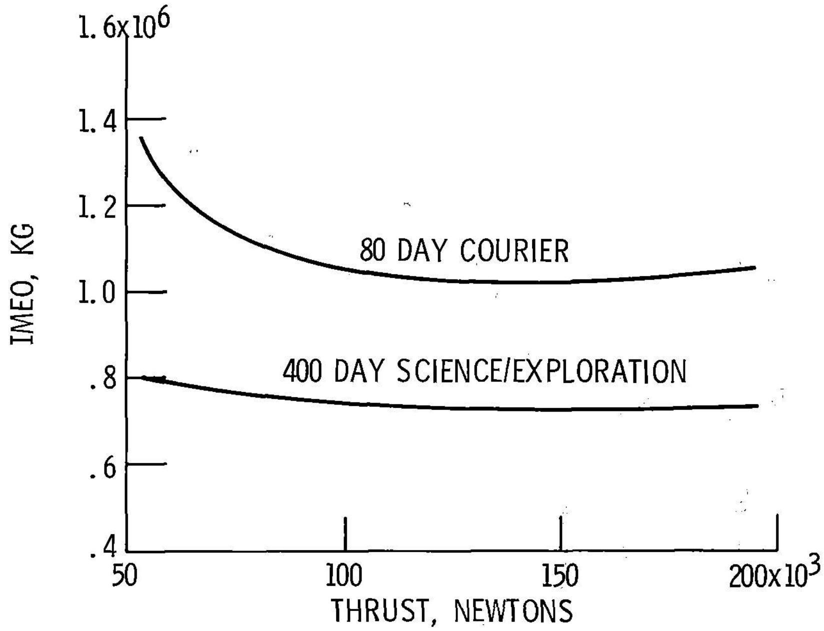

Figure 14. - Effect of thrust level on IMEO for Mars round-trip missions. Space radiator cooled GCNR.

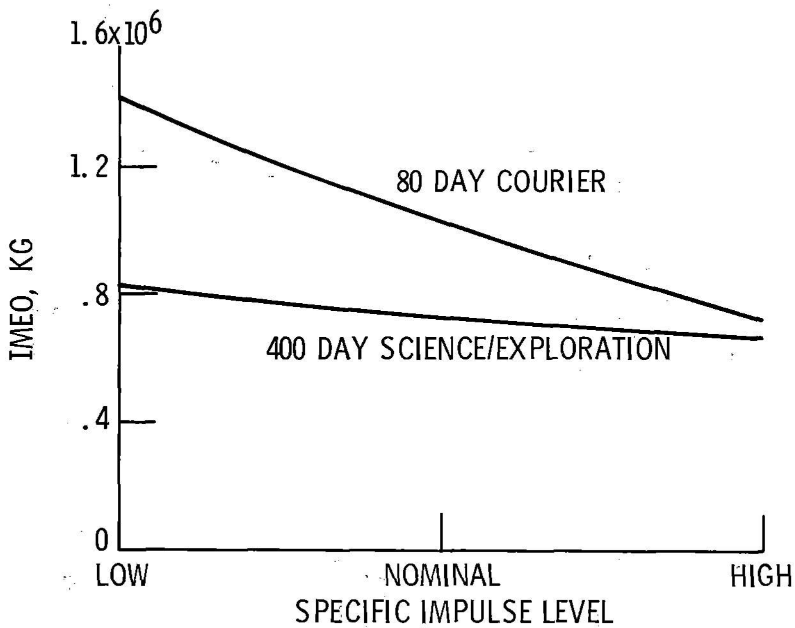

Figure 15. - Effect of attainable specific impulse of radiator-cooled GCNR - thrust levels reoptimized.

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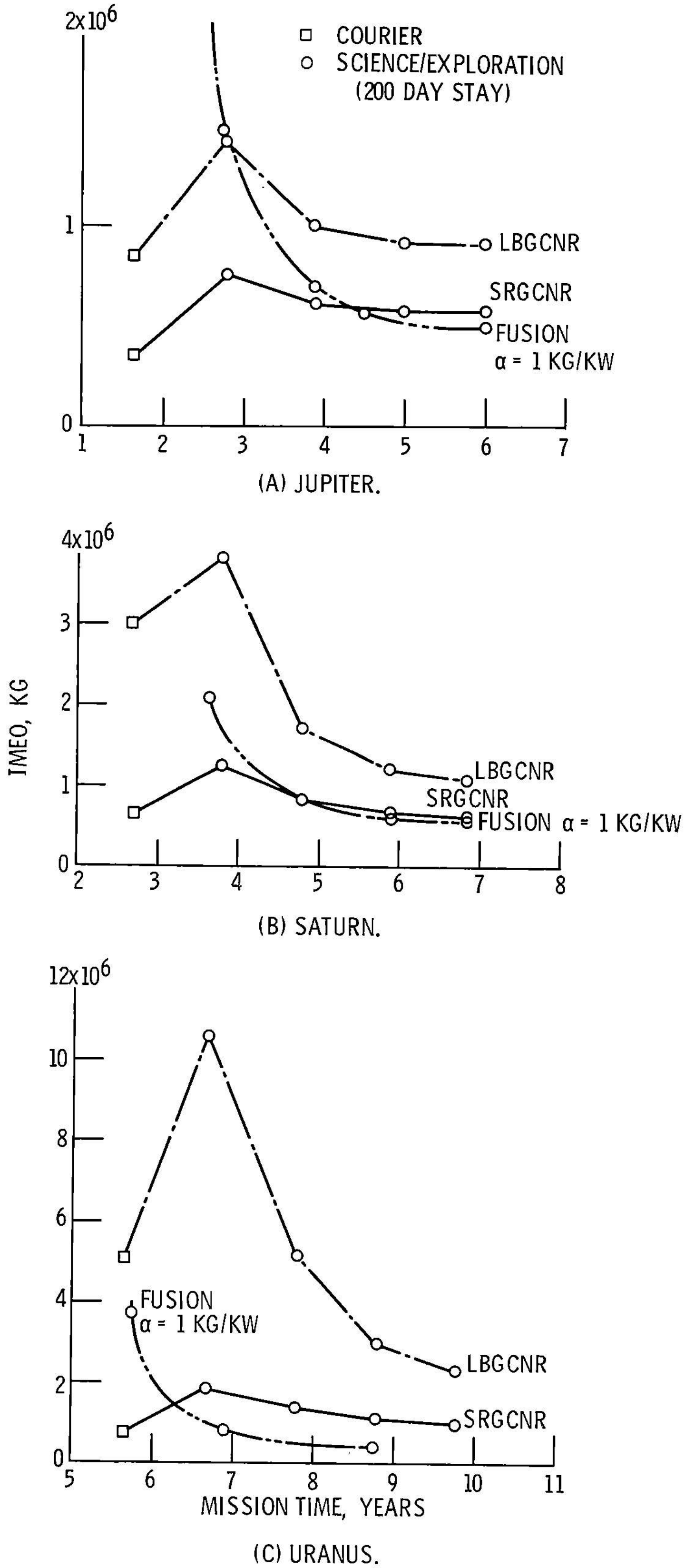

Figure 16. - Effect of major planet round trip mission time.