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by Robert G. Ragsdale and Edward A. Willis, Jr.
Lewis Research Center
Cleveland, Ohio

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GAS-CORE ROCKET REACTORS - A NEW LOOK

Robert G. Ragsdale and Edward A. Willis, Jr.
Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio

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Abstract

The current status of research on the feasibility of an open-cycle gas-core nuclear rocket engine is presented. In the technology research program, reactor critical experiments are being conducted, cold- and hot-flow tests are being performed, and both experiments and analyses are underway to determine compositions and gas opacities at reactor conditions. The use of a space radiator may permit attainment of a specific impulse ranging from 2500 to 6500 seconds, for engine thrust ranging from 20,000 to 400,000 newtons, and engine pressure ranging from 0.5×10^8 to 2×10^8 newtons per square meter. Mission application studies show that this class of rocket engines are capable of 80-day-roundtrip missions to Mars, with the engine and command module reinserted into the starting orbit at Earth.

I. Introduction

Virtually all existing or proposed rocket propulsion engines can be categorized as either high-thrust systems or high-specific-impulse systems. What is really needed for fast interplanetary travel is both characteristics,(1) namely a high specific impulse and a high engine thrust-weight ratio. The characteristics of an open-cycle gas-core nuclear rocket engine are examined in this paper to see how closely it might meet these requirements. The current status of the research program is reviewed, engine thrust-weight-specific-impulse characteristics are estimated using the most recent results from the research studies, and mission application study results are presented to show the rocket performance capabilities of this class of engines.

Over the past 10 or 15 years, the goal of gas-core research has been to assess the feasibility of an engine with a specific impulse in the range 1500 to 3000 seconds, and a thrust-weight ratio in the range 1 to 10. Both open-cycle(2) (uranium plasma is in direct contact with rocket propellant) and closed-cycle(3) (has solid, cooled transparent barrier between uranium plasma and rocket propellant) concepts are under investigation. The upper limit of 3000 seconds on specific impulse resulted from the use of only regenerative cooling to remove heat deposited in various parts of the engine structure. Higher specific impulses would require a space radiator to reject part or all of the waste heat. This is not really a new idea; it was first mentioned by Meghreblian over ten years ago.(4)

The results of a new look at this old idea are presented in this paper. Engine thrust-weight-specific-impulse characteristics are estimated using the most recent information from current research studies that are being conducted on fluid flow, gaseous radiant heat transfer, and reactor physics phenomena related to gas-core reactors. Radiator weight estimates are based on a recent study of radiators for Rankine-cycle space power systems.(5) The new upper limit on specific impulse of a radiator-cooled gas-core engine is based on an analysis of the radiant heat flux reaching the inside surface of the reactor cavity wall.(6) Results of a previous gas-core engine study are used to obtain the weights of the moderator-reflector and the engine pressure shell.(7)

In the following sections of this paper, the principle of operation of a gas-core engine will be briefly discussed. Then the highlights of current research will be presented to show the basis for estimates of engine performance characteristics that are given in the next section. Finally, results of mission application studies using these projected engine characteristics will be presented. The overall objective of the paper is to show what a radiator-cooled, open-cycle gas-core engine is, what is being done to determine its feasibility, what engine characteristics are currently projected, and what utility such an engine might have in space.

II. Principle of Operation

Like the solid-core nuclear rocket engine, the job of a gas-core engine is to heat hydrogen and then expand it through a nozzle to convert the thermal energy into thrust. In order to obtain a higher specific impulse than the 825 seconds of the solid core, a gas core has to produce hotter hydrogen. For a specific impulse of 825 seconds, the hydrogen temperature at the nozzle inlet is approximately 2500 K. A temperature of 22,000 K is required for a specific impulse of 5000 seconds. The temperature levels required for specific impulses in the range 3000 to 5000 seconds cannot be obtained by simply running solid-core-type fuel elements at a higher temperature because the uranium must be hotter than the hydrogen, and at these elevated temperatures the fuel elements would melt and vaporize.

The gas-core concept is to use an incandescent, radiating ball of fissioning uranium plasma as the "fuel element." The nuclear heat released within the uranium plasma leaves its surface in the form of thermal radiation, or photons. This thermal energy is picked up by a surrounding stream of hydrogen propellant, which is then expanded through a nozzle to produce thrust.

Figure 1 illustrates schematically how this basic notion might be translated into a rocket engine. It is not unreasonable to picture this kind of engine as a nuclear "sun" with the central fireball and surrounding gas flow contained within a chamber surrounded by structural materials. The analogy is not exact, of course, because the heat generation is due to fission rather than fusion. However, in both cases, the amount of energy that can be generated in, and released from, the fireball is essentially unlimited. There is, however, a limitation on how much energy can be absorbed by the hydrogen and turned into thrust without over-heating the cavity wall or the exhaust nozzle. It is the amount of energy that reaches various solid, temperature-limited regions of the engine that ultimately limits the power generation and therefore the specific impulse of this kind of device.

The proposed reactor shown in Fig. 1 is basically spherical. It is composed of an outer pressure vessel, a beryllium oxide moderator, and finally a porous or slotted cavity liner. Unfortunately, some of the heat that is produced by the fissioning uranium finds its way into these regions. This occurs in two ways. A small amount of the reactor power, less than 0.5 percent, is transferred to the porous wall by thermal radiation from the hot gases in the reactor cavity. Approximately 7 percent of the reactor power is deposited in the three solid regions of the reactor due to attenuation of high-energy gamma and neutron radiation, the hydrogen gas in the core having no capacity to intercept and absorb these two forms of radiation. This gamma and neutron heat is removed either by a coolant in an external space radiator loop, or regeneratively by the hydrogen propellant before it enters the central reactor cavity. The beryllium oxide region is operated at a temperature of about 1400 K, which is below the upper materials limit of approximately 1700 K and yet above the radiator temperature of 1100 K.

The hydrogen is pumped to a pressure of 5.07×10^7 to 20.34×10^7 newtons per square meter by means of a turbopump operated by hydrogen bled from an intermediate station in the propellant circuit. The hydrogen then is ducted into a plenum between the porous or slotted wall and the moderator. Appropriate seed particles which are about the size of smoke particles are introduced into the hydrogen as it enters this plenum region. The seeded hydrogen then flows through the porous or slotted wall. By properly designing the shape of the porous wall and by proper injection and distribution of the hydrogen flow through this wall, a relatively stagnant nonrecirculating central region forms within the cavity. The cavity is about 2.44 meters in diameter. The central fuel region occupies about 40 percent of the cavity volume. However, this region may also contain some (up to 50 at.%) hydrogen that would diffuse in from the outer edge of the fuel. Thus the "effective" volume of pure uranium would probably range from 20 to 30 percent of the cavity volume.

Uranium metal is injected into this region. It vaporizes and rises to temperatures sufficient to thermally radiate the energy that is generated by the fissioning uranium. A possible fuel injection technique consists of pushing a very thin rod of solid uranium metal through a shielded pipe (perhaps made of cadmium oxide) that penetrates the moderator. As it enters the cavity, the uranium vaporizes and rises in temperature to about 55,000 K. Reactor startup could be achieved by first establishing the hydrogen flow. Next uranium particles would be blown into the dead cavity region to achieve nuclear criticality. The power would then be increased to a level sufficient to vaporize the incoming uranium rod.

The seeded hydrogen is heated as it flows radially inward from the porous cavity wall by absorbing the thermal radiation from the central fissioning uranium fireball. The cavity walls receive only 0.5 percent or less of the thermal radiation issuing from the fireball. This wall protection is accomplished by introducing 5 to 10 percent by weight of a solid particle seeding material, such as graphite, tungsten, or natural uranium, into the hydrogen before it enters the cavity. This same technique is used in the nozzle region to reduce the hydrogen radiation heat load and the hydrogen temperature near the nozzle wall to tolerable levels. Figure 1 shows that some cold seeded hydrogen can be introduced through the nozzle walls at the downstream end of the engine if it is required. Seed concentrations of about 20 percent may be required here, somewhat more than was required in the cavity. The addition of the nozzle cooling gas would tend to reduce the specific impulse.

III. Research Program Status

The research work being conducted in the gas-core program falls rather naturally into the three categories of reactor physics, fluid flow, and heat transfer. The discussion in this section is also divided into these categories. Not all of the work will be discussed here. More detailed and comprehensive program reviews can be found elsewhere.(2,8,9,10)

Reactor Physics

Through an AEC/NASA-Lewis interagency agreement, extensive critical experiments have been carried out by Idaho-Nuclear Corporation.(11,12) These experiments have been done for both cylindrical and spherical geometries, and both solid (foil) and gaseous (uranium hexafluoride) nuclear fuel forms have been studied. These experimental measurements have provided a good basis for developing reliable calculational techniques for obtaining the critical mass requirements of cavity reactors.(13)

After the reactor computer codes have shown good agreement with the experiments, these same codes are then used to predict how much uranium would be required in a rocket reactor. Results from some recent calculations by the author of Ref. 13 are shown in Fig. 2. These calculations were performed in the same way as described in Ref. 13, but are for a different moderator-reflector material (beryllium oxide) and are for somewhat smaller numerical ranges of cavity diameter and moderator thickness. All of the values on Fig. 2 are for a fixed fuel region radius that is two-thirds of the cavity radius, and for uranium-235 as the nuclear fuel. Effects of hot hydrogen upscattering on the neutron spectrum were not included in these calculations for the reactor operating conditions of this study. This upscattering can be an adverse effect, but to include it would require a detailed radial distribution of temperature and composition through the engine cavity. Studies are now underway to incorporate this and other effects into gas-core nuclear analysis.

Figure 2(a) shows that critical mass ranges from 10 to 35 kilograms for the cavity diameters and moderator thicknesses considered. Larger cavity diameters require larger critical mass, but not larger critical density, as will be discussed in the following paragraph. For a constant cavity diameter the critical mass requirement can be reduced (which would allow engine operation at a lower pressure) by adding on more moderator-reflector. Lower pressure would result in a lower pressure shell weight. But the additional moderator thickness makes the engine heavier, which is an adverse effect that could more than offset the reduction in pressure shell weight.

For a fixed moderator thickness, an increase in cavity diameter decreases the critical density, but increases the moderator weight. This is shown in Fig. 2(b). It turns out that there is an "optimum" moderator thickness that gives a minimum critical density for a given moderator weight. This is shown in Fig 2(c). The reason for the optimum moderator thickness is as follows. When the moderator is very thin, neutron leakage becomes excessive, and extremely large diameters are required to avoid very high critical densities. When the moderator is very thick, it's weight becomes large even though small cavity diameters are attainable without very high critical densities.

For the particular fuel and moderator considered here, Fig. 2(c) shows that the optimum moderator thickness is 0.46 meters. Figure 2(c) further shows that beyond 40,000 kilograms of moderator weight, going to larger (and heavier) cavities causes only a slight reduction in critical density. All of the engine weight estimates presented later in this paper are based on a constant moderator-reflector thickness of 0.46 meters.

Fluid Mechanics

Isothermal flow experiments are being carried out under NASA-Lewis support and direction at United Aircraft Research Laboratories[14] and at Idaho-Nuclear Corporation.[15]

The United Aircraft work is aimed at obtaining a fairly detailed understanding of the factors that govern the amount of fuel that would reside in an engine and the factors that govern the fuel loss rate. These experiments are carried out at room temperature and pressure. Air is used to simulate the hydrogen in an engine, and both air and freon are used to simulate the fuel. Recent experiments using air/air in a cylindrical cavity have produced quite favorable results that indicate that the effective fuel volume might be 20 or 30 percent of the cavity volume for a uranium flow rate that is less than one percent of the hydrogen flow rate. Experiments are now underway on a spherical, porous-wall geometry.

The Idaho-Nuclear Corporation flow studies are being conducted in two-dimensional cavity mock-ups that incorporate as many of the engine features that have an influence on the flow as possible. A photograph of the experimental apparatus is shown in Fig. 3(a). Clear air is used to simulate the hydrogen propellant of an engine, and smoky air or smoky argon are used to simulate the gaseous uranium fuel. The objective of the experiment is to determine what cavity wall shape and propellant injection technique gives the largest fuel volume inside the cavity for the lowest fuel flow rate. The apparatus is constructed to allow rapid changes in the cavity geometry so that new ideas can be quickly evaluated and the best features retained in subsequent tests.

Figure 3(b) shows a time-exposure photograph of the flow in an air/air test. This picture shows the time-average, or "smeared out" appearance of the flow. The propellant flow rate for this run was 125 times greater than the flow rate of the fuel gas. In other tests, either the propellant flow or the fuel flow is turned off then back on, with the other gas flow held constant. These tests provide information about flow patterns in the reactor cavity during engine startup and shutdown. One series of such experiments showed that the propellant flow creates a pressure distribution throughout the flow field that counterbalances the tendency of the heavier fuel gas to "fall" through the propellant. This same kind of body force effect would exist in an engine in space due to the acceleration of the vehicle during firing. The engine acceleration force would be considerably less (0.01 to 0.05 "g's") than in the experiments, but the absolute dimensions, velocities, and density ratios would also be different. Both analysis and additional experiments are underway to determine the scaling law that describes this bouyancy effect.

Hot plasma flow experiments are being conducted by TAFA Division of Humphreys Corporation, under NASA-Lewis contract support and direction.[16] Induction heating is used to simulate the heat generation that would occur by nuclear fission in an engine. Induction heating is also used by United Aircraft Research Laboratories in their closed-cycle studies.[17]

The objective of the TAFA work is to determine if heat-generating plasma flows show the same general flow characteristics as cold flows, and to develop a technique for non-nuclear simulation of gas-core flow conditions. Figure 4(a) shows a schematic of the induction heated test apparatus. In the flow visualization portion of this work, transparent walls are used so that the interaction of the central plasma with the surrounding cold flow can be directly observed, and still and motion pictures obtained.

The induction heating simulation research has been quite productive, and has provided a number of positive inputs to the program. As a result of experiments conducted in parallel on a number of induction heater configurations, it is now possible to build a heater that incorporates many important engine features such as a curved porous wall, seeded propellant flow, solid-rod fuel injection, a seeded-transpiration cooled nozzle, and operation at a pressure of 5×10^5 newtons per square meter. In one megawatt (electrical) heater tests, a volumetric heat generation rate of 900 megawatts per cubic meter was achieved, as compared to representative engine values which might range from 400 to 4000 megawatts per cubic meter.

To come closer to the large size of an engine in these electrical simulation experiments, part of the work is aimed at building larger diameter induction heaters. Because of the physical laws of induction plasma heating, larger dimensions require lower electrical frequency to maintain efficient coupling between the work coil and the plasma. Thus tests have been carried out at successively lower frequencies and larger heater diameters. Successful operation was recently accomplished, at power levels of approximately one megawatt, at a frequency of 9600 hertz with a 20-centimeter diameter heater, and at a frequency of 960 hertz with a 30-centimeter diameter heater. Photographs of these two heaters are shown in Figs. 4(b) and (c). Work is now underway to build an induction-heated gas-core simulator that will be designed to operate at a power level up to 15 megawatts. Studies are also underway to determine facility availability and modification requirements to provide the electrical power for such tests.

Heat Transfer

The heat-transfer problem in a gas-core reactor comes down to two requirements. First, the reactor power leaving the edge of the fuel region must be absorbed in the hydrogen propellant. Second, this internal energy must be kept in the hydrogen until it is converted into thrust by a nozzle, without an excessive heat load to either the cavity wall or the nozzle. Information on the opacity and composition of uranium, hydrogen, and seed materials is required in order to determine the heat transfer situation in a gas-core engine. Both experimental and theoretical studies are underway to provide this information.[10,18]

Under NASA-Lewis support and direction, experiments are being conducted at Georgia Institute of Technology to determine what seed material to add to the hydrogen to make it absorb at temperatures where it would ordinarily be transparent. These experiments involve measuring the attenuation of a light beam as it passes through a mixture of hot hydrogen and seed material.

Figure 5(a) shows a schematic of the experimental apparatus, and Fig. 5(b) shows some recent data for tungsten-seeded hot hydrogen. Data have been obtained at pressures up to 10^7 newtons per square meter and temperatures up to 2500 K. The data show that the addition of a few percent by weight of tungsten in hydrogen will produce a mixture absorption cross section ranging from 2000 to 100,000 square centimeters per gram. The data also show that the solid-particle-seeded hydrogen becomes more absorptive at elevated pressure. This is a favorable, though unanticipated, characteristic since the engine is a high pressure device.

As will be discussed in the following section, these measured solid-seed absorption cross sections are high enough to provide cavity wall thermal protection for specific impulses in the 4000 to 7000 second range. As the solid-seed-hydrogen mixture flows radially inward, its temperature rises due to the absorption of thermal radiation. At some distance from the wall (typically about 1 centimeter), the temperature is high enough that the seed particles vaporize, or melt and then vaporize depending on the particular seed material. From that point on, the absorption of radiant energy is due to the opacity of the mixture of seed vapor and the hydrogen. Eventually, some temperature is reached (~15,000 K) where the hydrogen by itself is quite absorptive.

In this "opacity window" region where seed vapor opacity is required to absorb the radiant energy, we have depended to date on theory to provide estimates of the vaporized seed opacity. Work is now underway at Georgia Tech to obtain experimental data in this interesting and important temperature range by using an induction heater.

IV. Projected Engine Characteristics

The latest results from the reactor physics, fluid mechanics, and heat-transfer studies have been used to estimate the characteristics of a gas-core engine. These characteristics are expressed in terms of engine weight, thrust, and specific impulse. Such results are then coupled with mission application studies to determine which of the available combinations of thrust-weight and impulse are most desirable for various classes of missions.

The important inputs to this calculation from the research program are as follows. The critical density of the uranium as a function of the cavity dimensions and moderator material is taken from Fig. 2. Based on fluid mechanics work such as illustrated in Figs. 3 and 4, it is assumed that the uranium plasma occupies 30 percent of the reactor cavity volume, and that the uranium loss rate from the engine is one percent or less of the hydrogen flow rate. The optical properties of the solid-seeded hydrogen are taken from Fig. 5. The gaseous opacities are taken from theoretical predictions. A radiant heat-transfer analysis[6] is used to determine the maximum specific impulse achievable consistent with the limitation of a cavity wall temperature less than 1000 K. The engine pressure is a dependent variable of the calculation; it is simply the value required to have a critical mass of uranium in the reactor cavity.

The engine weight is taken to be the sum of the three major components- the moderator, pressure shell, and the radiator. The weight of the spherical pressure shell is based on a strength-to-density value[7] of 1.7×10^5 N-meters/kg. The radiator weight is calculated using a unit weight of 140 kilograms per megawatt of radiated power, and on a heat deposition rate that is seven percent of the reactor power. This is the same radiator weight per unit area (19 kilograms per square meter of planform area) as the vapor fin radiator of Ref. 5, but operated at 1100 K instead of the 945 K used in Ref. 5, which affords a factor-of-two reduction in required area and therefore in weight.

The characteristics of an open-cycle gas-core rocket engine are shown in Fig. 6 for engine thrust levels from 20,000 to over 400,000 newtons. The corresponding values of specific impulse and engine weight are given in Figs. 6(a) and (b), respectively. Engine pressure is included as a parameter over a range from 0.5×10^8 to 2×10^8 newtons per square meter. The three specific impulse curves are also labeled "low," "nominal," and "high" a notation to be discussed in the mission applications section to follow.

For the entire range of thrust and pressure, specific impulse varies from 2500 to 6500 seconds, and engine weight varies from 40,000 to 210,000 kilograms. Higher engine pressure allows operation at a higher specific impulse because the higher pressure makes the gases more opague, but it also makes the engine heavier. A higher thrust engine can deliver more specific impulse because there is more hydrogen flow for wall cooling, but it also weighs more. A choice of a "best" engine, or a best combination of these engine characteristics can only be made in the context of a specific mission requirement.

There is certainly more information required for an engine feasibility evaluation than the relatively simple results displayed in Fig. 6. Two obvious examples are the limitation placed on specific impulse by nozzle cooling requirements, and the control requirements placed by the engine dynamic characteristics. It is also clear that large-scale, hot-flow experiments are needed to bridge the gap between the laboratory tests done to date and the size and power levels envisioned for an actual engine. The idea of incorporating a space radiator into a gas-core engine system has not yet received any great depth of study, so there is surely more to learn in this direction. The addition of a radiator has more than doubled the specific impulse potential of a gas-core engine, without adding enough weight to offset the gain. If radiator weight could be reduced below the values estimated in this study, further engine performance improvements could be realized. Some work is being initiated in each of these foregoing areas, and in addition the basic studies described earlier are continuing.

V. Mission Performance Potential

As previously mentioned, several advanced propulsion concepts have been seriously considered in recent years.(1) These range from high-thrust, low-specific-impulse systems such as the solid-core nuclear rocket (SCNR) to high-specific-impulse, low thrust systems such as a nuclear-electric rocket or fusion device. Among these, the gas-core nuclear rocket (GCNR) concept described above occupies an intermediate area (in regards to both thrust and specific impulse) which has not been included in prior comparative performance studies. It is therefore of considerable interest to examine the performance of this system in typical mission application.

Vehicle System and Mission Profiles

Upon reviewing Fig. 6, it will be seen that the radiator-cooled GCNR becomes progressively more efficient in terms of increasing specific impulse and decreasing specific weight, as the thrust level is raised. Therefore, the most favorable mission applications for it will be those involving large payloads and high propulsive effort or ΔV requirements. A group of "fast" manned roundtrip missions to Mars and Jovian planets was somewhat arbitrarily selected to suit this guideline.

For each mission, the space vehicle, as illustrated in Fig. 7, consists basically of a single GCNR installation, a command module, payload items, jettisonable liquid hydrogen tankage, and interconnecting structure. The GCNR provides the four burns required-Earth-orbit escape, planet-orbit capture and escape, and Earth-orbit capture at the mission's end. Thus, the "core" vehicle comprising the GCNR engine, its uranium storage and supply system, the command module, and part of the hydrogen tankage is recovered in Earth orbit, presumably to be re-used. The basic input values and data sources used for numerical calculations are indicated in Table 1. The output of the calculation is the initial mass in earth orbit (IMEO) of the entire vehicle, including the hydrogen propellant. The IMEO is calculated as a function of trip time for various planetary missions.

Fast Mars Roundtrips

The Mars roundtrip mission performance of the radiator-GCNR system is compared to that of several alternative engine concepts in Fig. 8. Initial mass in Earth orbit (IMEO), a rough measure of initial cost, is plotted against the roundtrip mission time.

The upper curve on the figure is for the SCNR, and, paralleling previous studies such as Ref. 21, shows a very pronounced minimum at about 500 days trip time and 1.5×10^6 kg. With this type of engine recoverable mission trip times below 400 days are unobtainable.

The next curve, for the regenerative-GCNR, shows a factor-of-two improvement in IMEO and also a significant broadening of the minimum. At the 1.5×10^6 kg IMEO level associated with the optimum SCNR trip, the regenerative-GCNR can support a mission time of about 250 days-which is also a factor-of-two saving. This, however, is the practical limit of its performance.

Another factor-of-two performance increment is offered by the present concept of a radiator-cooled, open cycle GCNR as illustrated by the solid curve. At optimum mission time, the IMEO of 400,000 kg is only twice the total payload. At the 700,000 kg level which was minimum for the regenerative engine, the radiator-cooled GCNR now yields a 250 day mission time. Further, the IMEO curve has now flattened still more, so that trips as short as 150 days are feasible to consider.

At this point, it should be noted that the Mars staytime of 40 days has become an appreciable fraction of the mission time; the Earth/Mars transit times are so small that extremely high ΔV's (and high propellant fraction) result. This difficulty is mitigated in the "Courier" mode by using all of the available mission time for Earth/Mars transits. The basic IMEO level is further decreased by reducing the Mars payload to a negligible level. These measures, added to the radiator-GCNR's basic characteristics, result in startling performance at short mission times. This is shown by the asterisked curve at the bottom left of Fig. 8. Trips as short as 75-80 days can reasonably be considered under the present ground rules.

To complete the comparison, the final curve at the lower right gives the performance of a hypothetical, very advanced low thrust system, e.g. a fusion rocket, which may have a specific powerplant mass, or "α," as low as 1 kg/kw. Its specific impulse can be extremely high and, following customary low-thrust practice, is optimized for every mission shown. In the courier mode, this system yields extremely low IMEO's at mission times greater than 250 days. Its performance is disappointing, however, in the area of very rapid trips. This is because, despite its low α, it is still a low thrust, power-limited system. Even when thrusting continuously, it must resort to comparatively low I_{sp} in order to meet the acceleration, velocity, distance and time requirements of this class of mission. Therefore it cannot take advantage of its extremely high I_{sp} potential.

The radiator-GCNR, on the other hand, is I_{sp}-limited (recall fig. 6). Per unit of thrust or jet power, however, this is a very light weight propulsion system in comparison with any known electric or fusion rocket concept. Thus, it is possible to use enough thrust to hold burn time to, say, two or three days total out of an 80 day mission, and the system's full I_{sp} potential can be utilized. As a point of interest, one may compute an "equivalent α" for the GCNR by dividing the engine masses shown in Fig. 6 by their associated values of jet kinetic power in kilowatts. The result, for the data covered in Fig. 6 is that α ranges from 0.01 to 0.1 kg/kw-which is roughly two orders of magnitude better than the best ever estimated for an advanced low thrust system.

In summary, Fig. 8 demonstrates an unique mission capability of the present radiator-GCNR concept: Mars manned round trips in less than three months. No other known alternative concept can duplicate this performance. Preliminary calculations, not illustrated in Fig. 8, indicate that essentially the same result is found for missions to Mercury, Venus, and representative asteroids.

Uranium and Hydrogen Requirements

The IMEO's shown on Fig. 8 are a rough measure of the space vehicle systems initial cost. This excludes development but does include the first mission use. But as pointed out previously, the mission mode being considered here involves a re-usable core vehicle which is parked in a low Earth orbit at the mission's end. To re-use this vehicle, it is necessary to replace the propellants and the jettisoned payload; these items basically determine the cost of the second and subsequent uses of a given vehicle.

The uranium and hydrogen requirements for the previously illustrated radiator-GCNR Mars mission are shown in Fig. 9. The three solid curves show the uranium requirements for the Science/Exploration mission mode at several hydrogen/uranium flow ratios. The asterisked curve shows the courier mission's requirements at a flow ratio of 200/1. In each case, the indicated uranium requirement is multiplied by the flow ratio to obtain the hydrogen requirement.

For example, the 80 day Mars courier mission mentioned previously would require about 3350 kg of uranium and 3550x200 or 670,000 kg of hydrogen. These amounts to about 3/4 of the 900,000 kg IMEO. At longer trip times, the uranium and hydrogen requirements are much smaller, and are also smaller fraction (e.g. 25 percent) of the IMEO.

It is of interest to note, per the dotted curve, that the GCNR's uranium requirement for one mission may be considerably less than the amount invested in the case of a SCNR. When considering re-use mission, however, it must be noted that the SCNR can probably be re-used several times without adding uranium, whereas the SCNR's uranium must be replaced for every mission. Allowing for this, it appears that flow ratios in the 200/1 to 400/1 range are needed if the GCNR is to be competitive with the SCNR on the basis of uranium costs per average mission. This is not an insignificant point, since 1000 kg of uranium (per fig. 9 for 150-day Mars courier) times roughly \$10,000/kg equals \$10,000,000 worth of uranium per mission. Still, the uranium cost is not dominant, when compared to a launch cost that might range from \$44,000,000 to \$440,000,000 for the 200,000 kg of hydrogen propellant required for each round-trip, based on a unit launch cost ranging from \$220 per kilogram (\$100 per pound) to \$2200 per kilogram (\$1000 per pound) of mass in orbit.

Selecting an Initial-Design Goal

The preceding results are based on using optimum thrust levels for each mission time and engine combination. The optimum values for the present radiator-GCNR concept (as defined by the fig. 6 data) are shown as a function of round trip time in Fig. 10. It appears that a thrust in the range of 70,000 to 90,000 newtons (roughly, 15,000 to 20,000 pounds) would be most suitable if low-IMEO missions are desired. For the very fast mission emphasized in the preceding sections, however, a rating of 112,000 to 224,000 newtons (25,000 to 50,000 pounds) seems to be more appropriate.

The effect of using fixed, non-optimum thrust levels is displayed in Fig. 11, where IMEO is plotted against thrust for two representative missions. For the 80-day mission, it appears that any thrust rating between 100,000 and 250,000 n.mil. yield essentially optimum performance. The 400-day mission, similarly is little penalized for using a thrust as large as 225,000 n. A rating of about 150,000 N seems to be a good compromise between these two missions if both are of interest. On the other hand, it has been found that radiator-GCNR vehicles with higher-than-optimum thrusts are relatively insensitive to I_{sp} penalties. This is displayed in Fig. 12 where IMEO is plotted against I_{sp} and thrust level for the 80-day courier mission. Notice that the I_{sp} scale is not numbered, but merely consists of three points labelled "low," "nominal" and "high." These adjectives refer to similarly-labelled portions of the Fig. 6 engine data. As previously explained, the "low" and "high" figures are believed to represent comparable perturbations for each engine size about its nominal specific impulse. The 112,000 N. engine, which nominally operates at a lower I_{sp} level than the larger engine, is clearly the more sensitive to an I_{sp} decrement. Based on these considerations, we have provisionally chosen the 224,000 newton (50,000 pounds) thrust engine as being the most appropriate for further more detailed study and design analysis.

Multimission Performance

Up to this point, we have emphasized the manned Mars mission in our discussion. This was partly because of the radiator-GCNR's really startling performance potential for this mission-e.g. a round trip in 80 days. Also, the Mars mission, having been extensively studied and discussed in the latter 1960's, should be a familiar starting point for most readers. But a radiator-GCNR engine, once developed, could be used to great advantage in many other applications as well. For instance, we have already mentioned that unpublished LeRC calculations indicate the radiator-GCNR is also very effective for missions to all of the nearby planets and asteroids, not must Mars.

A second, and significantly different class of possible missions for the radiator-GCNR, consists of round trips to the major planets-Jupiter, Saturn, Uranus and Neptune. These planets are of considerable scientific interest in their own right-e.g., because of their distance from the Sun, and because they are physically and chemically so different from the Earth. Moreover, these planets posess extensive lunar systems which, collectively, account for most of the potential manned planetary landing sites in the solar system.

To evaluate the radiator-GCNR's multi-mission performance potential for this class of objectives, we chose the 224,000 newton engine suggested in the preceding section. This was used as the sole powerplant for the range of major planet mission results displayed in Fig. 13. As in the earlier plots, IMEO is presented as a function of mission time. The Science/Exploration mission made with 200 days stay time is used for all but the fastest trips shown for each planet; those use the courier mode.

A peculiarity of major planet round trips discussed in Ref. 19 is that feasible trips occur only at discrete intervals (of 12 to 13 months). Thus, only the "data points" shown in Fig. 13 represent real missions, the connecting lines serve merely to group the planets and to indicate trends.

Consider first the Jupiter mission results, shown in Fig. 13. Using the courier mode, a 600 day or 1.67 year round trip can be made for an IMEO of about 1.3×10^6 kg. At the next feasible trip time, 1000 days or 2.75 years, the Science/Exploration mode requires under 10^6 kg which is comparable to the Mars mission results. For Saturn missions, we find approximately the same IMEO's but at 400-days larger trip time. The results for Uranus and Neptune are discouraging, however, and we suspect that an alternative engine with higher I_{sp} (e.g., a nuclear-electric or fusion rocket) would be more suitable.

In brief, this data indicates that courier and Science/Exploration missions to Jupiter and Saturn can be made for IMEO's which are comparable to the corresponding Mars-mission results. The associated mission times, however, are considerably larger-in rough proportion to the greatly increased Earth-to-planet travel distances. Because of this, the Jupiter and Saturn missions are not such an appealing or spectacularly favorable application as were the fast near-planet trips. Nevertheless, these missions may eventually prove to be of very great scientific interest, and when added to the missions discussed earlier, they indicate that the radiator-GCNR engine could serve as the basis of a truly versatile, recoverable multimission space exploration system.

VI. Conclusions

Research studies related to an open-cycle gas-core nuclear rocket engine fall into the general categories of reactor physics, fluid mechanics, heat transfer, engine systems, and mission applications. The current research work includes zero power reactor critical experiments; cold-flow and hot-flow fluid mechanics experiments; gas opacity and composition experiments and analyses; projections of engine thrust-weight-specific-impulse characteristics; and mission application studies. The work to date indicates the following conclusions.

1. The use of a space radiator to reject waste heat increases the specific impulse potential to 7000 seconds, from the previous limit of 3000 seconds available if only regenerative cooling is used. The 7000 second limit is placed by the ability to thermally protect the cavity wall.

2. There will be less than one percent by mass uranium in the hydrogen propellant exhausted from the engine.

3. For engine thrust ranging from 20,000 to 400,000 newtons and engine pressure ranging from 0.5×10^8 to 2×10^8 newtons per square meter, the maximum specific impulse was found to range from 2500 to 6500 seconds. The corresponding range of engine weight, including the radiator, is presently estimated to be from 40,000 to 210,000 kilograms.

4. Mission application studies show that an engine reflecting this technology would yield outstandingly good performance for a wide range of interplanetary destinations and round-trip times. It is especially suitable for very fast round trips to nearby planets-e.g., the 80-day Mars courier-and in such applications it outperforms all conceptually-known competition by a large margin.

VII. References

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Command module*	50,000 kg (all mission)
Payload to planet*	150,000 kg (Science/Exploration mode) 0 kg (courier mode)
Expendables*	50 kg/day
Hydrogen tankage*	20 percent of hydrogen mass
Interstage structure*	2 percent of transmitted load
Thrust structure*	5 percent of thrust
GCNR installation	per figure 6 (includes uranium storage and supply system)
Parking orbits	600 km circular at Earth; high ellipse at planet
Propulsive effort	ideal ΔV; from ref. 19; gravity-loss corrections Cg from ref. 20
Propellant fraction	$1 - \exp^{(-\Delta V_i \times C_g / I_{sp} g)}$ (per maneuver)

*Arbitrary values, estimated from prior manned-Mars mission studies such as Ref. 21.

TABLE 1 INPUR DATA FOR INITIAL-MASS-IN-EARTH-ORBIT COMPUTATION

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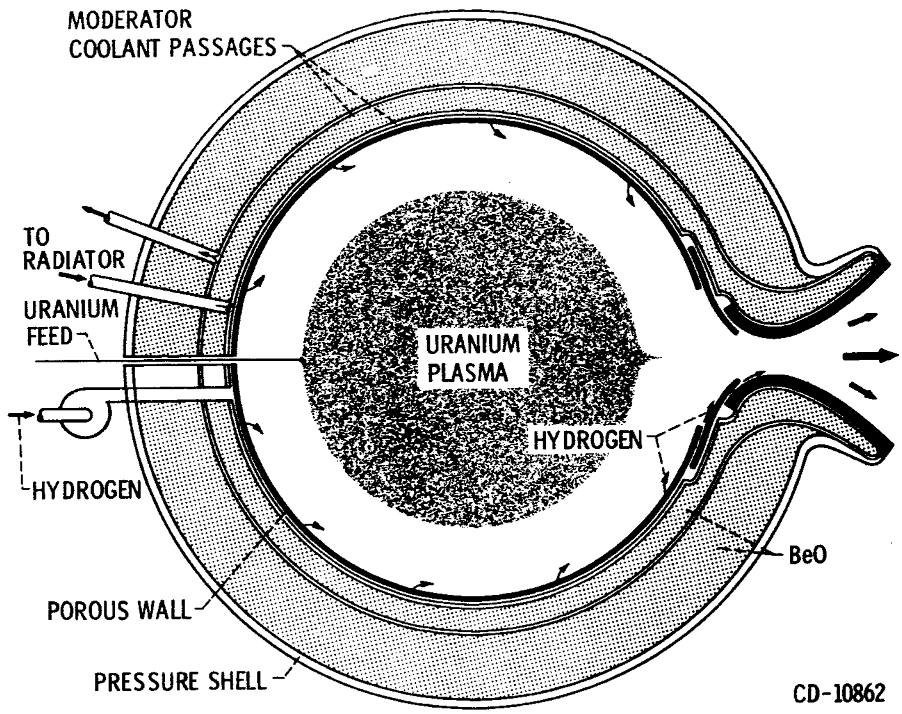
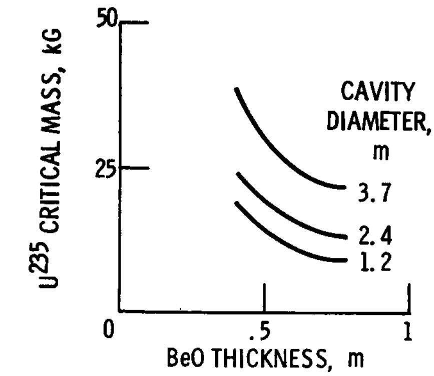
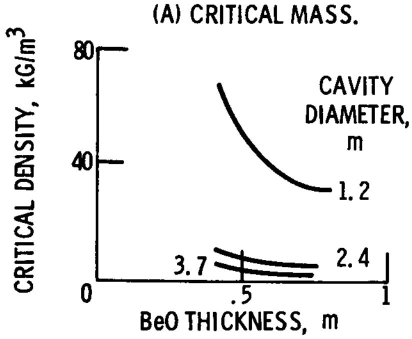


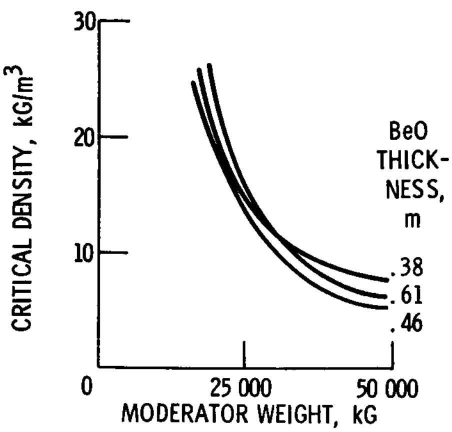
Figure 1. - Porous wall gas core engine.



(A) CRITICAL MASS.



(B) CRITICAL DENSITY.



(c) Moderator weight.

Figure 2. - Criticality requirements of a beryllium-oxide-moderated, spherical gas-core reactor; for a fuel-to-cavity radius ratio of 0.67, and a 0.63 centimeter graphite cavity liner containing 5 percent niobium.

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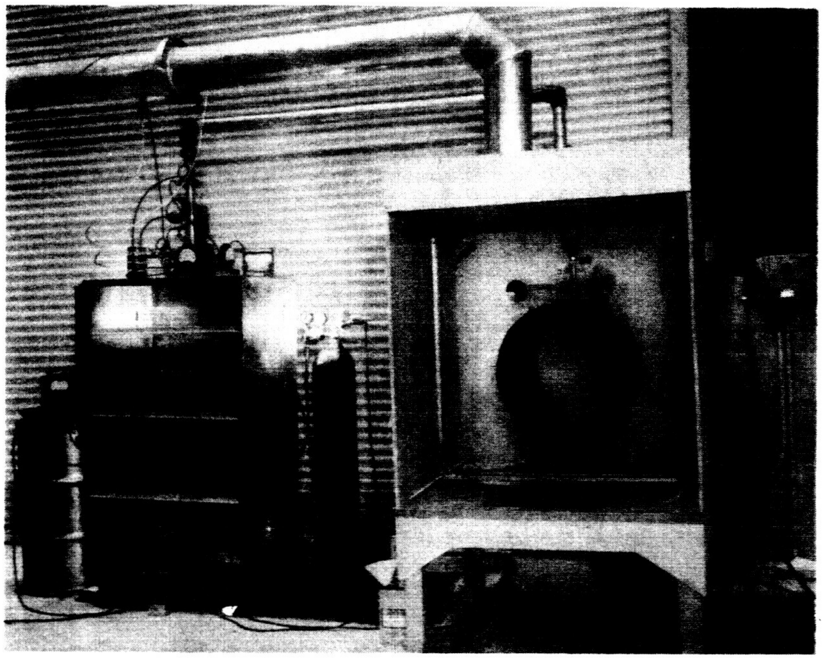
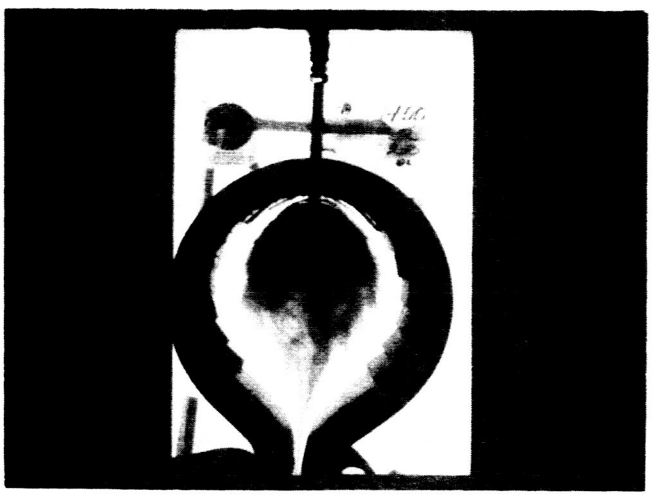
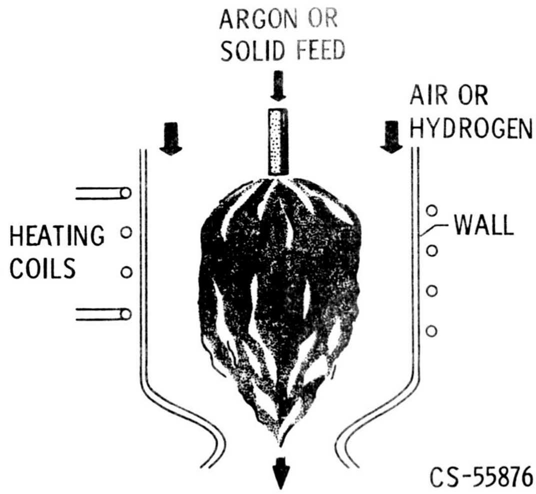


(a) Photograph of apparatus.

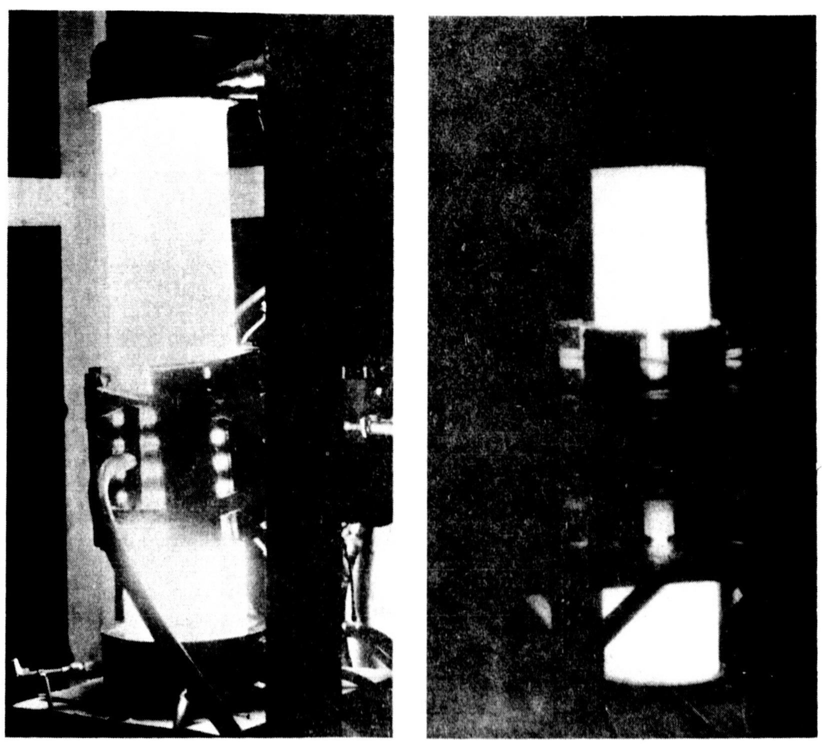


(b) Air-air smoke test at mass flow ratio of 125/1.

Figure 3. - Curved porous wall flow experiment.



(a) Schematic.



(b) 9600-Hertz, 20-centimeter diameter test section.

(c) 960-Hertz, 30-centimeter diameter test section.

Figure 4. - Induction - heated flow experiments.

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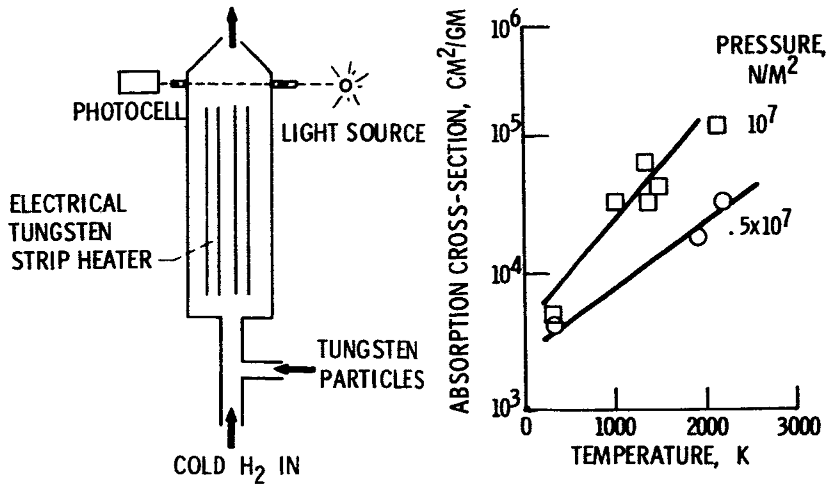
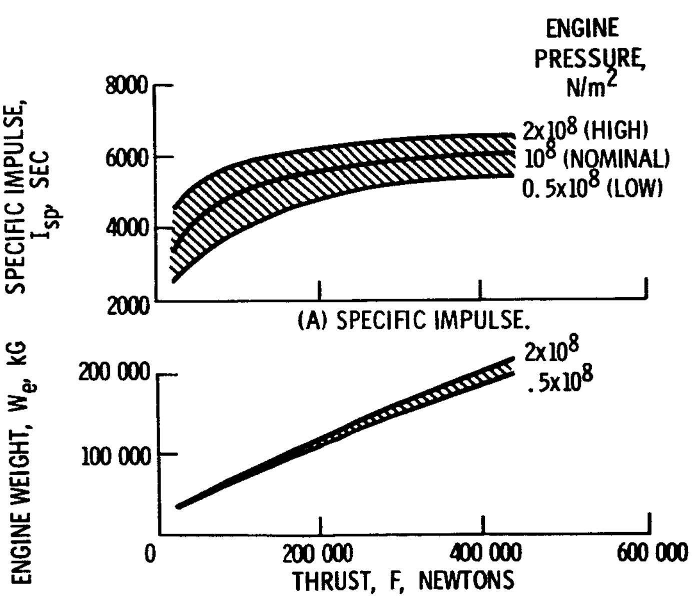


(A) SCHEMATIC OF APPARATUS.

(B) MEASURED ABSORPTION CROSS-SECTIONS FOR WAVELENGTH RANGE FROM 0.25×10^{-6} TO 0.4×10^{-6} METERS.

Figure 5. - Absorption properties of hot-hydrogen seeded with solid tungsten particles.



(B) ENGINE WEIGHT.

Figure 6. - Gas-core engine weight and specific impulse.

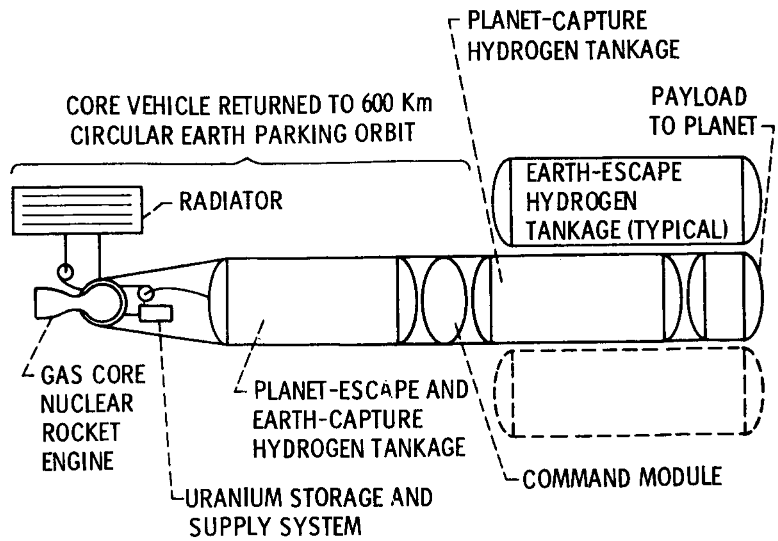


Figure 7. - Radiator cooled gas core nuclear rocket vehicle schematic manned interplanetary missions.

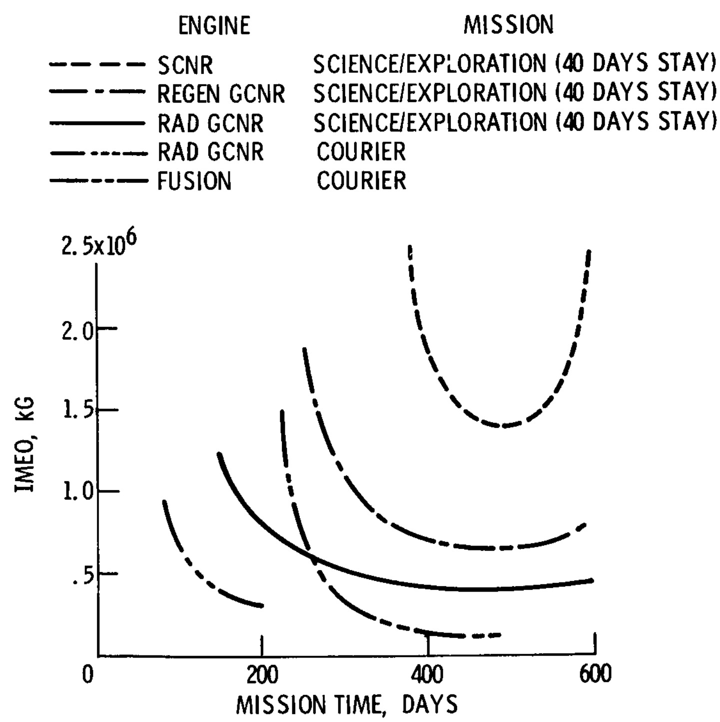


Figure 8. - Effect of Mars round trip mission time, for various nuclear rocket engines.

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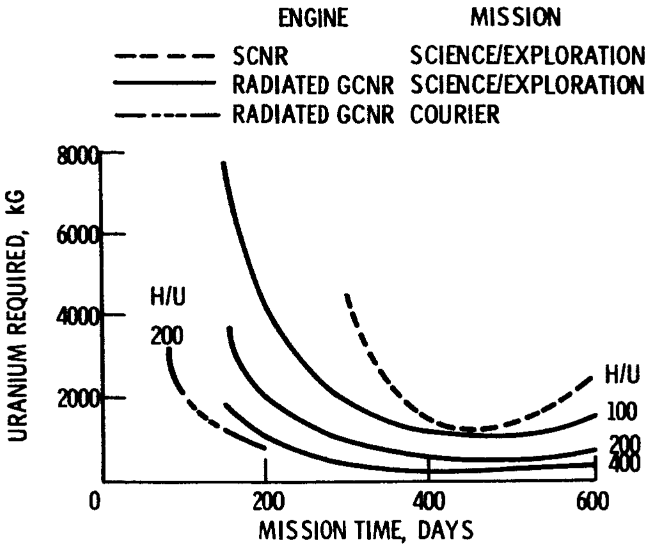

Figure 9. - Total uranium propellant requirements for radiator-cooled GCNR and SCNR Mars round trips.

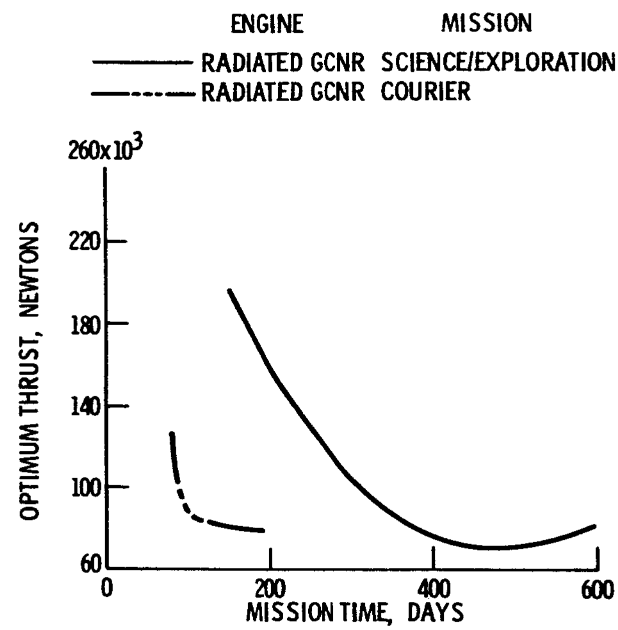

Figure 10. - Optimum thrust levels for radiator cooled gas core nuclear rockets (Mars round trips).

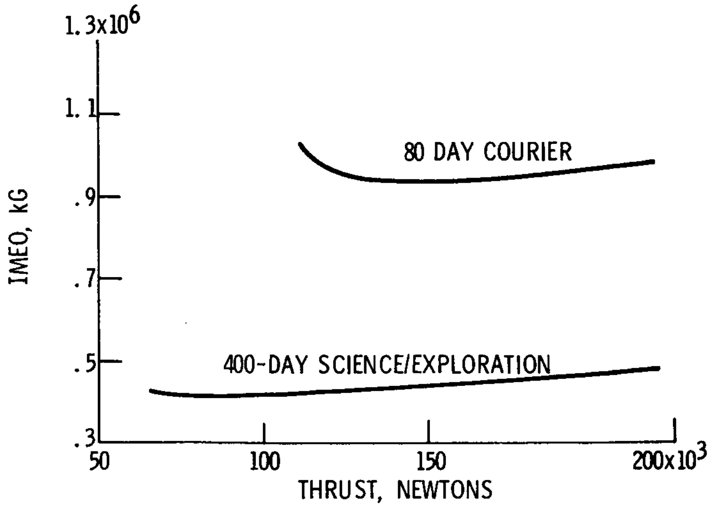

Figure 11. - Effect of thrust level on IMEO for Mars round-trip missions. (Radiator-cooled GCNR per figure 6, nominal ISP).

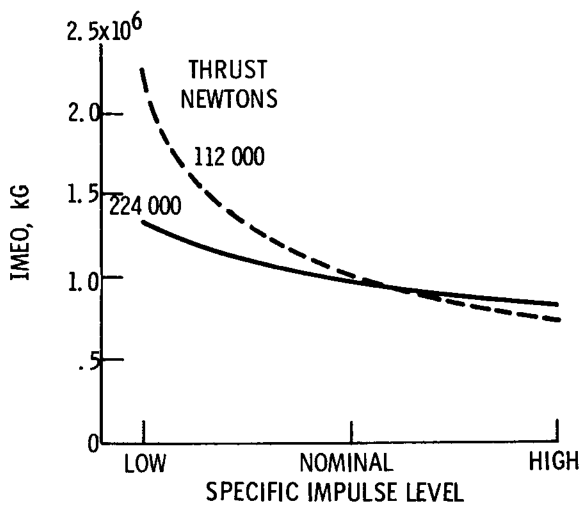

Figure 12. - Effect of attainable specific impulse of radiator-cooled GCNR (80 day Mars courier mission).

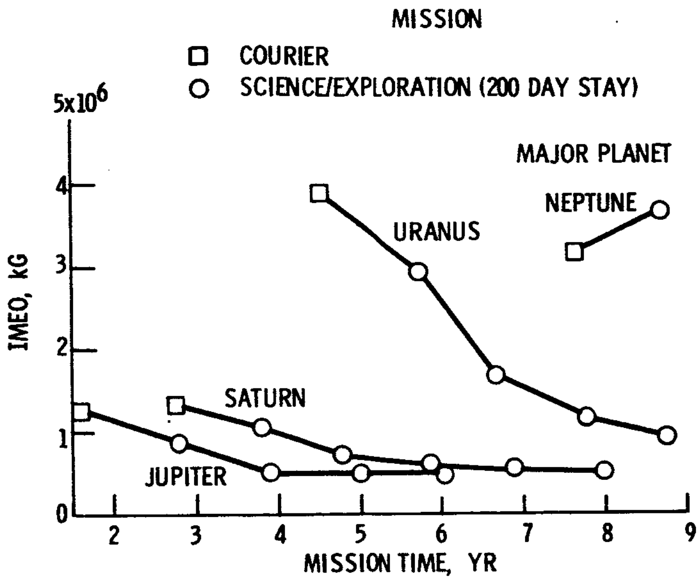

Figure 13. - Major planets round trip missions 224 000 Newton 5650 seconds radiated GCNR.

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