Dual Electric-Nuclear Engine

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The dual electric-nuclear rocket system combines a nuclear rocket for high thrust phases of manned interplanetary space missions with a cluster of electric engines powered by the same reactor for high-specific-impulse, low-thrust phases. For a seven-man fast Mars mission, the dual electric-nuclear engine can reduce the space vehicle gross weight in initial earth orbit to between 40 and 60% of that for an all-nuclear engine. The nuclear reactor is designed to operate in both open-cycle and closed-cycle systems. During open-cycle high-thrust phases, hydrogen propellant is heated as it flows through the reactor core passages and then exhausts through a nozzle to give I_{sp} of about 833 sec. These open-cycle, high-thrust phases are used for orbital escape or orbital capture. In the closed-cycle phases, the reactor, operating at reduced power, heats a circulating fluid that drives a turbogenerator. The generator, in turn, powers a cluster of MHD accelerators, providing low thrust at an I_{sp} of 4900 sec. These low-thrust phases are used for shortening trip times in what otherwise would be the coast periods of the nuclear rocket mission.

Introduction

THE use of nuclear rockets for more difficult space missions (high payloads and/or fast trips) appears promising. In earlier work,[1, 2] it was shown that nuclear rockets become competitive with chemical rockets as the reactor weight becomes a small fraction of the rocket gross weight. For very difficult trips, such as a seven-man Mars exploratory mission, even a nuclear rocket cannot reduce the space vehicle weight in initial 300-naut-mile earth orbit to below the order of several million pounds for relatively fast missions.[3] But a reduction in space vehicle weight to about 0.47 of the nuclear rocket value can be effected with a dual electric-nuclear engine system. For shorter trip times than the 347-day trip considered here, this hybrid system becomes still more useful in reducing the gross weight.

In a manned round-trip mission using an all-nuclear space vehicle, reactors would be used at each phase of thrust application into and out of Earth and Mars orbits. (Independent planetary landing vehicles are assumed in this work.) Figure 1 indicates the overall trajectory profile. In the present work, reactors were staged in the all-nuclear vehicle in order to minimize gross weight. This staging is expensive in terms of reactor cost and might not be carried out in practice, but it was assumed here to give conservative estimates of dual electric-nuclear vehicle advantage.

The concept of the dual electric-nuclear engine develops from the fact that, for fast interplanetary trips, considerably more than escape ΔV is required on entering or departing from planetary orbits. If the high-thrust nuclear engine were used for attaining only escape velocity or perhaps somewhat more than escape velocity, and if high-specific-impulse, low-thrust electric engines were used for attaining the balance of the required vehicle velocity during what otherwise would be coast periods, then reductions in gross vehicle weight could be realized. Recent work by Edelbaum[4] indicates that, although such advantages do not accrue for some space missions, the improvement is marked for fast missions with ΔV's in the range of those for a fast round trip to Mars (see Fig. 8 of Ref. 4, at lower values of τ).

If a gas or a liquid metal closed cycle were to be designed into the existing open-cycle hydrogen-cooled reactor, it would be possible to produce large quantities of electrical energy for powering the electric engines. Such a design appears feasible within existing technology. The reactor thermal power made available for electrical power generation would be about 100 times greater than the maximum now being considered for auxiliary power reactors, ~214 mw (thermal) compared to 2 mw. The 214 mw would be less than 1% of the reactor's full power value of ~26,000 mw, and the reactor would operate at about 3500°F rather than the full power temperature of 4500°F, although the electric engine operating time would be much longer. The redesigned reactor would not weigh significantly more (about an 8% increase in weight) but might involve problems due to the higher critical mass required with consequent higher uranium loading, and the faster neutron spectrum. In addition, greater control problems may be encountered. In such a dual engine, the nuclear phases would have an I_{sp} of 833 sec and an acceleration of ~0.4g; the electric phases would have an I_{sp} of ~4900 sec and an acceleration of ~0.0002g. There would be a ($gtb \sin\theta$) loss because of the sun during the electric propulsion phases, but this would not be excessive. This loss, for operation in the Earth's gravitational field (or Mars) would be excessive, leading to high ΔV's if one attempted to use electric propulsion to raise the vehicle from the 300-naut-mile orbit. Also, the earth escape time would be large. For these reasons, the nuclear high-thrust phases are necessary.

If two separate reactors were used for the high-thrust application and the power-generating requirement, the penalty would be about a 350,000-lb increase in vehicle gross weight because of the added reactor weight, as well as increased reactor fuel and construction costs. Furthermore, although not included in the present work, it is quite feasible to use the closed loop in the open-cycle reactor for removal of a portion of the reactor afterheat after each high-power phase. Normally, for the all-nuclear engine (if reactors are not staged), a considerable amount of hydrogen propellant must be exhausted inefficiently, at low temperatures, during afterheat removal. Use of a closed loop in the open-cycle reactor would lead to a very significant additional reduction in gross weight in this case. (Or, alternately stated, the all-nuclear vehicle weighs more because of the afterheat hydrogen required with nonstaging.)

Table 1 shows a comparison of a typical dual electric-nuclear vehicle with a comparable all-nuclear vehicle for a seven-man Mars fast round trip. The vehicle requirements such as shields, ecological system, and landing vehicles are derived from Ref. 3; the mission trajectory and characteristics for the all-nuclear vehicle are derived from Refs. 7 and 8. Electric engine component weights are derived from Refs. 6 and 11.

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The relatively large gross weights of the vehicles of Table 1 are due to conservative estimates of solar flare shielding and structural weight and to the fact that earth atmospheric drag for re-entry was not considered for the return to 300-naut-mile orbit. (Re-entry drag is assumed, however, for the Earth landing vehicle.) Nevertheless, a parametric study, as indicated in Fig. 2, gave similar reductions for smaller vehicles. Furthermore, round trips faster than the present example will require heavier vehicles. It is held as a basic premise in this work that manned interplanetary missions will require the shortest possible transfer time.

Table 2 indicates some of the design parameters for the typical dual electric-nuclear engine. The cycle thermodynamic efficiency was taken as 30% based on estimated improvement for large space powerplants over the next 15 to 20 years. For a similar mission and time period, Stuhlinger and King quoted an efficiency of 35%.[9] The dual electric-nuclear system improvement is sensitive to cycle efficiency. For example, if the highest attainable cycle efficiency were reduced to 20%, the reduction in gross weight would change from 0.47 to ~0.70 of the all-nuclear vehicle gross weight. For this particular vehicle and mission, the optimum I_{sp} of the electric engine for maximum payload was 4900 sec. A crossed-field MHD engine appears applicable for this I_{sp} with a projected engine efficiency (approximately 15 to 20 years from now) of about 70%. Ion engines also might be applicable, depending on attainable efficiencies at this relatively low I_{sp}. Arc jet engines, at present, would not appear to be usable here because of excessive heat losses and high temperatures of materials at the high I_{sp}'s required.[10] It should be added that there is an optimum electric engine power that minimizes electric-engine-weight/power-in-jet, and it appears likely, therefore, that a cluster or clusters of electric engines will be required for the relatively high thrust and power in the electric propulsion phases of these missions.

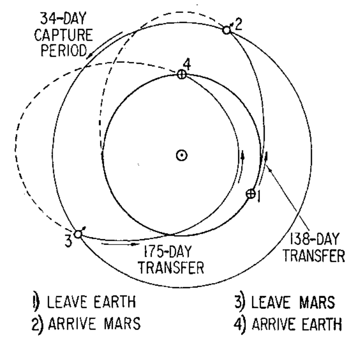


Fig. 1 Fast round-trip Mars mission profile

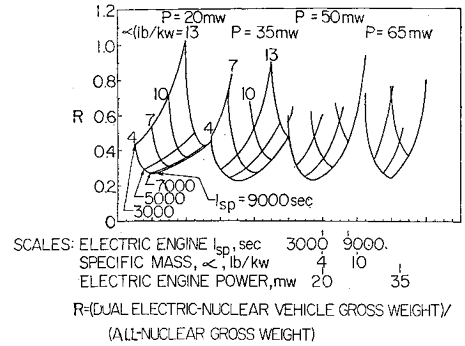


Fig. 2 Gross weight ratio *R* for a series of dual electric-nuclear space vehicles with 3 × 10^6-lb gross weight

Table 1 Typical dual electric-nuclear vehicle and all-nuclear vehicle comparison

Parameters	Dual electric-nuclear	All-nuclear
Total space vehicle weight, lb	~3,600,000	~7,660,000
Total propellant, lb	2,648,000	6,500,000
Tankage, lb	265,000	650,000
Nuclear engine, lb		
1st stage	46,000	120,000[a]
2nd stage		57,000
3rd stage		17,000
4th stage		10,000
Vehicle structures, lb	20,000	40,000[a]
Electric engine and power supply, lb	355,000	...
Total shield, lb[b]	122,000	122,000
Total ecological storage (at 12 lb/man-day), lb	34,000	34,000
Crew quarters	30,000	30,000
Mars landing-and-return vehicle, lb	60,000	60,000
Earth landing vehicle	20,000	20,000
Specific impulse, sec	4900 (electric) 833 (nuclear)	850

[a] A lower all-nuclear vehicle gross weight was achieved by reactor staging. The higher reactor weights are due to the larger reactors required for sufficient propellant flows and power to maintain 0.4 g acceleration during orbital escape and capture. The greater vehicle structure weight is required in the heavier vehicle.
[b] For a 125-rem total dose including one giant major solar flare.

Table 2 Dual electric-nuclear vehicle parameters for typical design

Electric engine specific impulse, sec	4900
Nuclear phase specific impulse, sec	833
Electric engine power, mw	
In the jet	48
At the generator	69
At the reactor (mw, thermal)	214
Electric engine efficiency, %	70
Specific mass, lb/kw	~8.3
Total weight of engine plus power supply, lb[a]	~401,000

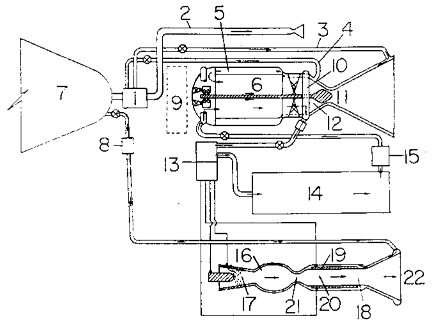
[a] This includes the weight of the reactor (46,000 lb), closed loop in the core (~5000 lb), and all other closed cycle components.

Engine Design Considerations

Brayton Cycle

There are several techniques for obtaining a closed cycle using hydrogen gas as the working fluid in a Brayton cycle. One involves plugging the nozzle and flowing the hydrogen through the reactor, as in the full power phase. This is shown schematically in Fig. 3. (The electric engine in Fig. 3 is not to scale and is intended to represent a cluster of engines.) A major problem, perhaps insurmountable, is the leakage of H_2 which would occur around a plug. The loss of as little as 0.1%, or even smaller amounts, of the hydrogen per circuit in the closed loop would degrade the performance of the engine seriously. One possible improvement of the scheme shown in Fig. 3 might be noted: the turbogenerator in the closed cycle might be combined with the high-flow, open-cycle liquid hydrogen turbopump with a net reduction in weight. Spline couplings would be required between the pump, turbine, and generator (actuated during "standstill") and a dynamic seal between the turbine and generator.

To avoid the leakage problem, an alternate approach uses closed-flow loops in the reactor core, as suggested in Fig. 4 (where it is applied to a Rankine cycle). Another design improvement might be obtainable here by the use of the closed-flow channels in the reactor during the high-thrust open-cycle phase for heating bleed hydrogen to drive the liquid hydrogen turbopump. In this way, the separate bleed line could be eliminated and closed-loop cooling maintained during the high-thrust phases. Again, the high-flow liquid



1 LIQUID H_2 TURBOPUMP
2 TURBOPUMP EXHAUST
3 PROPELLANT
4 DRIVE GAS (BLEED)
5 REFLECTOR
6 REACTOR
7 CRYOGENIC H_2 TANKAGE
8 H_2 PUMP
9 SHIELD
10 HEADER
11 PLUG
12 NUCLEAR ROCKET
13 TURBO-GENERATOR
14 RADIATOR
15 H_2 GAS COMPRESSOR
16 MIXING CHAMBER
17 ARC
18 PLASMA ACCELERATOR
19 ELECTRODES
20 MHD DUCT
21 NOZZLE
22 AFTER-NOZZLE

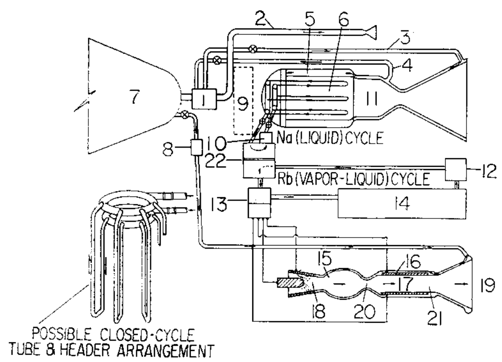
Fig. 3 Dual electric-nuclear system with a Brayton cycle using a plugged nozzle

hydrogen turbopump conceivably could be coupled with the generator for closed-cycle power generation.

Rankine Cycle

The Brayton cycle, even with use of the closed-loop approach, has a serious disadvantage. At presently allowable turbine inlet temperatures, the radiator size required for transfer of the waste heat into space is formidable. At these turbine inlet temperatures, a Rankine liquid-vapor cycle permits much smaller radiators than those of a Brayton cycle.[11] In time, if material problems for turbines can be overcome, allowable inlet temperatures might increase to a level where alkali metal-vapor usage becomes less feasible. At such high temperatures, the gas cycle will become more attractive. This possibility and the relative simplicity of the gas cycle suggest detailed comparison studies.

The Rankine cycle also presents some problems. One problem will be boiling and condensing under conditions of fluid weightlessness or near weightlessness with low vehicle accelerations (0.0002 g). Another problem will involve the startup and shutdown of the closed-cycle system with respect to filling the radiator passages with liquid metal.



1 LIQUID H_2 TURBOPUMP
2 TURBOPUMP EXHAUST
3 PROPELLANT
4 DRIVE GAS (BLEED)
5 REFLECTOR
6 REACTOR
7 CRYOGENIC H_2 TANKAGE
8 H_2 PUMP
9 SHIELD
10 Na PUMP
11 NUCLEAR ROCKET
12 Rb PUMP
13 TURBO-GENERATOR
14 CONDENSER RADIATOR
15 MIXING CHAMBER
16 ELECTRODES
17 MHD DUCT
18 ARC
19 AFTER-NOZZLE
20 NOZZLE
21 PLASMA ACCELERATOR
22 Na-Rb HEAT EXCHANGER

Fig. 4 Dual electric-nuclear system with a Rankine cycle

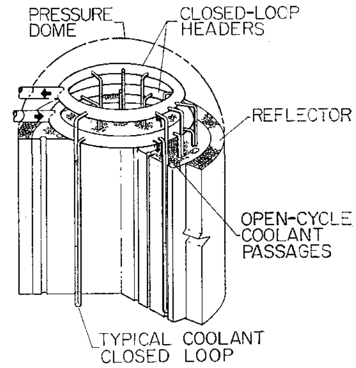


Fig. 5 Schematic of a closed loop in the reactor core

Nevertheless, because of the much lower radiator weights (on the order of 0.1 that of a Brayton cycle) for feasible turbines in the near future, a Rankine cycle was chosen for the present study.

Figure 4 shows a schematic of the system for a Rankine cycle. A separate sodium reactor coolant loop is used in this illustration. Rubidium in a second loop is heated and vaporized in a sodium to rubidium heat exchanger. Other liquid metal combinations, or use of a single loop, also are possible.

An alternate scheme to the dynamic cycles would be the use of thermionic conversion in the reactor core.[5] This technique also appears promising. However, in the cited reference, the authors stated that the thermionic core is not applicable to Earth surface escape because of thermal stress considerations. Although surface escape was not considered for the Rankine cycle system discussed here, thermal stress problems do not appear to restrict such usage, and, consequently, the Rankine cycle system may prove to be more versatile.

Core Design

In the reactor core, heat transfer from the fuel element to the coolant tubes during the electric propulsion phases may be achieved by radiation, by conduction, or by some combination of these. The tubes may be of molybdenum or, preferably, of W^{184} if the cost can be reduced by new separation methods, such as high-speed centrifuges.

One possible core fabrication technique would use graphite fuel elements loaded with uranium carbide in conjunction with separate molybdenum closed-loop hairpin tubes (Fig. 4). In this case, radiation would be the primary heat transfer mechanism, and the core would be so designed that the tubes "see" as much fuel element surface as possible. Furthermore, the tubes would be coated with a thin layer of high-emissivity material such as Rokide C. Rokide C, primarily Cr_2O_3, has been applied successfully to Mo with temperatures over 2000°F to give an emissivity over 0.90.

During the high-thrust phase, when fuel element temperatures are about 4500°F, the empty Mo tubes will be held at about 2000°F by flowing excess hydrogen through the open-cycle flow passages surrounding them. Because of this, the overall performance of the rocket will be degraded by about 2% (I_{sp} of 833 sec instead of 850 sec). The tubes will be filled with inert gas at ~1000 psia to pressure-balance the external hydrogen that typically will be at an average core pressure of 1000 psia.

The hairpin tubes of Fig. 4 lead to a rather complicated core coolant passage geometry at the nozzle end because of the tube bends and because, in the high-thrust phase, the hot hydrogen propellant at over 4000°F must not be allowed to flow directly over these tube bends. A rather complicated core design at the exhaust end would be required. Furthermore, the alignment of coolant passages and the insertion of the hairpin tubes add to the fabrication problem.

To avoid these problems, the scheme of Fig. 5 is suggested. The hairpin tube downcomer and return legs are touching, leading to effectively larger tubes of flattened cross section. (About 80% of the surface area of these double tubes is effective in heat transfer.) The coolant tubes now can be inserted in "bayonet heater" fashion from the header end.

With this latter scheme and a peak fuel element temperature during the electric propulsion phases of about 3500°F, about 800 $\frac{1}{2}$-in. "double tubes" are required. The molybdenum tube wall volume is about 1.4% of core volume; the total tube volume is about 7% of core volume; and the total weight, filled with Na coolant, is about 8% of total reactor weight.

Condenser-Radiator

The heaviest component is the condenser-radiator. Optimization of this is essential. In addition to the usual thermodynamic, flow, and dimensional parameters and the choice of fluid and radiator materials, the meteoroid puncture problem enters optimization evaluations.

With all other parameters fixed, as radiator area increases, necessary meteoroid protection in terms of pounds per kilowatt increases at a faster rate, since the larger area presents a greater target. Consequently, radiator weight increases sharply at higher powers, such as those required for the missions discussed here. To lessen this increase, the radiator must be divided into a number of parallel sections capable of isolation from the main system, and additional "emergency standby" sections must be added. There is an optimum number of parallel sections for minimum weight where, at one extreme, shielding weight is high, and at the other extreme, valve and separation plumbing weights are high. There is also an optimum standby area increase beyond which the added weight of radiator exceeds the savings in shield weight. This double optimization must be carried out to obtain minimum radiator weight for any particular electric power level.

If the "belt radiator" concept discussed in Ref. 12 were to be developed and found feasible, a reduction in radiator weight to about 25% of the more conventional tube and header radiator might be possible. Such reductions would make the electric-nuclear rocket appear even more favorable.

Some problem areas should be mentioned. It will be necessary to drain the radiator when not in use to prevent freezing of rubidium or sodium and consequent plugging. (Plugging of a small number of tubes can be tolerated.) Liquid metal valves and helium purge valves will require electrical heaters. It might be noted that the radiators in their extended position will undergo about 0.4 g acceleration during the nuclear phases, requiring some additional structural support (included in the 20,000 lb of Table 1.)

Missions

The mission trajectories are indicated in Fig. 1.[7] For the fast transfers considered, the trajectory ellipses intersect the Mars and Earth orbits. In the all-nuclear trips, full staging of both tankage and reactor was used for each thrust phase. There are four stages in all. In the electric-nuclear trip, tankage was jettisoned after each low-thrust plus high-thrust phase, again resulting in four stages, but only with respect to tankage. Because of the smaller reactor weight in the electric-nuclear vehicle and the added complexity of a closed-cycle core loop, it was considered neither necessary nor advisable to attempt reactor staging in this instance. The jettisoning of human waste material, landing vehicles, and other equipment was performed in a uniform manner for both the electric-nuclear and all-nuclear systems. It was assumed that 85% of the electric propulsion system (i.e., radiators and other components external to the reactor) could be discarded before the final nuclear thrust into return-Earth-orbit. Cost studies might prove that this system and/or the reactor system should be recovered.

The required ΔV's for the mission are presented in Table 3 for both the nuclear and electric-nuclear vehicles. In the case of the dual electric-nuclear trips, the high thrust nuclear phase ΔV's were assumed equal to the orbital escape velocities, except in the Earth-departure phase. For the latter, the minimum gross weight occurred when the entire ΔV was gained using high thrust, i.e., no electric propulsion was used during Earth departure. This was true only for Earth departure. In the other trip phases, it was found disadvantageous to use high thrust for more than planetary escape ΔV. This seemingly anomalous behavior on Earth departure can be explained in the light of the following discussion.

During most of each electric engine thrust phase (on Earth departure, for example), the vehicle velocity relative to the planet will be close to zero. The vehicle will behave as if it had, after 1 to 5 days, escaped effectively to infinity. If it is assumed (as in the present study) that all the energy of the electric phase is added when the vehicle velocity relative to the planet is zero, the required electric phase ΔV's must be large in order to give the same final total vehicle energy. The dual system then will result in a typical space vehicle gross weight equal to 0.47 of the all-nuclear vehicle gross weight. If it is assumed that the energy of the electric phase is added entirely when the vehicle velocity is at the planetary escape velocity, the electric phase ΔV will be lower, and the space vehicle gross weight for the typical electric-nuclear vehicle will be about 0.41 of the all-nuclear vehicle weight. The actual value lies somewhere between these two extremes and will be determined more accurately in future work.

Because V_{escape} on Earth departure is only slightly less than V_∞ for the mission considered here, it is more favorable to add ($V_\infty - V_{escape}$) by continuing the high-thrust phase rather than by adding the required ΔV with electric propulsion at a relative vehicle velocity of zero. In other words, it pays to follow a high thrust with a low thrust when the remaining (ballistic) ΔV is of the order of, or greater than, the high thrust ΔV but not when it is a small fraction of the high thrust ΔV.[4]

Table 3 Mission parameters for a 347-day Mars round trip

Parameters	Leave Earth orbit		Arrive Mars		Leave Mars		Arrive Earth		Totals
ΔV nuclear, fps (I_{sp} = 850 sec)	a) 13,390		b) 20,260		c) 24,400		d) 21,270		79,320
Plasma-nuclear: ΔV nuclear, fps (I_{sp} = 833 sec)	a) 13,390			d) 2370	e) 2370			h) 10,400	
ΔV electric, fps[a] (I_{sp} = 5000 sec)		b) 0	c) 32,100			f) 33,800	g) 33,500		127,930
"Burning time," days	0		45		31		24		100
Transit time and hold time, days	138			34		175			347

[a] These values of ΔV are valid only for this particular vehicle and mission.

The electric propulsion phase calculational procedure followed the technique of Irving [Eq. (10-10) of Ref. 13] for constant-thrust acceleration. But in addition, the effect of the sun's gravity was included in a first-order manner.

In all cases, the nuclear thrust gave a 0.4 g acceleration, resulting in negligible $gt_b \sin\theta$ losses with respect to Earth or Mars. At lower accelerations, these losses would become significant.

Parametric Analysis

A parametric analysis was performed with the following variables: electric engine specific impulse, electric propulsion system specific mass, electric engine power, and vehicle gross weight. The vehicles were compared on the basis of a merit ratio R (dual electric-nuclear vehicle gross weight/all-nuclear vehicle gross weight), where both vehicles for a particular case deliver equal payloads in the seven-man Mars trip of Fig. 1. Figure 2 shows a carpet plot of possible dual electric-nuclear vehicles with gross weights equal to 3×10^6 lb. Similar plots, not shown, were prepared for 1×10^6-, 2×10^6-, and 4×10^6-lb gross weights. It is worth noting that, with a fixed dual electric-nuclear gross weight, the lower the value of R, the higher the payload ratio of the vehicle. Using working plots such as Fig. 2, with interpolations, optimum values of electric engine power giving minimum R values could be found for each combination of dual vehicle gross weight, α, and specific impulse.

For purposes of this study, the operating time of the electric engine is significant, since relatively short operating times are desired in order to approximate better the ballistic fast transfer ellipses of Fig. 1. Although more detailed mission analyses are planned to find the effect of operating time on total transfer time, for this study it was assumed that total operating time should be restricted to about 30% of the total transfer time. The typical vehicle of Table 1 then was found by appropriate interpolations for optimum electric engine power and for the forementioned restriction on operating time.

Conclusions

The dual electric-nuclear space vehicle appears useful in reducing the required gross weight of space vehicles performing fast interplanetary missions, in particular, a seven-man Mars round trip. Dual vehicle gross weight reductions to 0.4 to 0.6 of the all-nuclear vehicle gross weight appear feasible. A reduction to 0.47 of the all-nuclear vehicle weight was found for a typical vehicle in the seven-man Mars round trip. Primarily, the concept involves a combining of present, or soon to be developed, technologies rather than requiring new fundamental breakthroughs. Therefore, the system appears applicable to a time period about 15 to 20 years from now, when manned interplanetary missions probably will assume the importance of present manned lunar missions.

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