A COMPARISON OF PROPULSION SYSTEMS FOR POTENTIAL SPACE MISSION APPLICATIONS[1]

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ABSTRACT

A derivative of the NERVA nuclear rocket engine was compared with a chemical propulsion system and a nuclear electric propulsion system to assess the relative capabilities of the different propulsion system options for three potential space missions. The missions considered were (1) orbital transfer from low earth orbit (LEO) to geosynchronous earth orbit (GEO), (2) LEO to a lunar base, and (3) LEO to Mars. The results of this comparison indicate that the direct-thrust NERVA-derivative nuclear rocket engine has the best performance characteristics for the missions considered. The combined high thrust and high specific impulse achievable with a direct-thrust nuclear stage permits short operating times (transfer times) comparable to chemical propulsion systems, but with considerably less required propellant. While nuclear-electric propulsion systems are more fuel efficient than either direct-nuclear or chemical propulsion, they are not stand-alone systems, since their relatively low thrust levels require the use of high-thrust ferry or lander stages in high gravity applications such as surface-to-orbit propulsion. The extremely long transfer times and inefficient trajectories associated with electric propulsion systems were also found to be a significant drawback.

INTRODUCTION

Nuclear rocket propulsion development, based on solid-fueled reactors, was begun in 1955 as the ROVER Program (1). The program, directed at a manned mission to Mars, was expanded in 1961 by the addition of the NERVA program to develop a prototype of a flight-rated engine. The NERVA program successfully designed, developed, built, and tested a number of developmental engines, culminating in the NRX-A6, which operated for 3600 s at its rated power of 1100 MW(t), and the XE-prime engine system, which demonstrated 28 startup and shutdown cycles along with 1100 MW(t) full power operation. At the time of program termination in 1972, the design of the prototype flight-rated engine was well underway, having successfully passed the equivalent of an Air Force Preliminary Design Review (2).

In 1985, the Air Force Rocket Propulsion Laboratory (AFRPL), now the Air Force Astronautics Laboratory, was given the lead to develop an advanced nuclear propulsion system for future Air Force needs. In support of this developmental effort, the Idaho National Engineering Laboratory (INEL) established a team of experts from the aerospace and nuclear industries and initiated studies to assist the AFRPL in establishing requirements for the nuclear rocket program. In addition, mission analyses were performed and the performance characteristics of a nuclear rocket engine were compared with the performance characteristics of chemical and nuclear electric propulsion (NEP) systems (3). This paper describes the direct-thrust nuclear propulsion system concept used in the INEL study and summarizes the results of the comparison of the performance characteristics of the three propulsion system options considered.

NUCLEAR PROPULSION SYSTEM CONCEPT

The propulsion system selected for the INEL study was a derivative of the NERVA nuclear rocket engine. This concept was selected because, as described earlier, the NERVA technology is fully developed and represents the lowest risk and earliest nuclear engine deployment option available to the Air Force.

A schematic of the NERVA-derivative nuclear propulsion system, referred to as the Advanced Nuclear Rocket Engine (ANRE), is shown in Figure 1. The ANRE utilizes a full-flow, partial topping cycle powered by a solid core, graphite-moderated hydrogen-cooled epithermal reactor. Liquid hydrogen coolant/propellant, from a storage tank is delivered to the ANRE reactor via a turbopump. The hydrogen flow is initially split, with most of the flow used to cool the ANRE exhaust nozzle, and the remaining flow used to cool the core support elements (described later). The hydrogen from the nozzle then flows through an annulus between the reflector and reactor vessel, and joins the flow from the core support elements. Part of this flow is used

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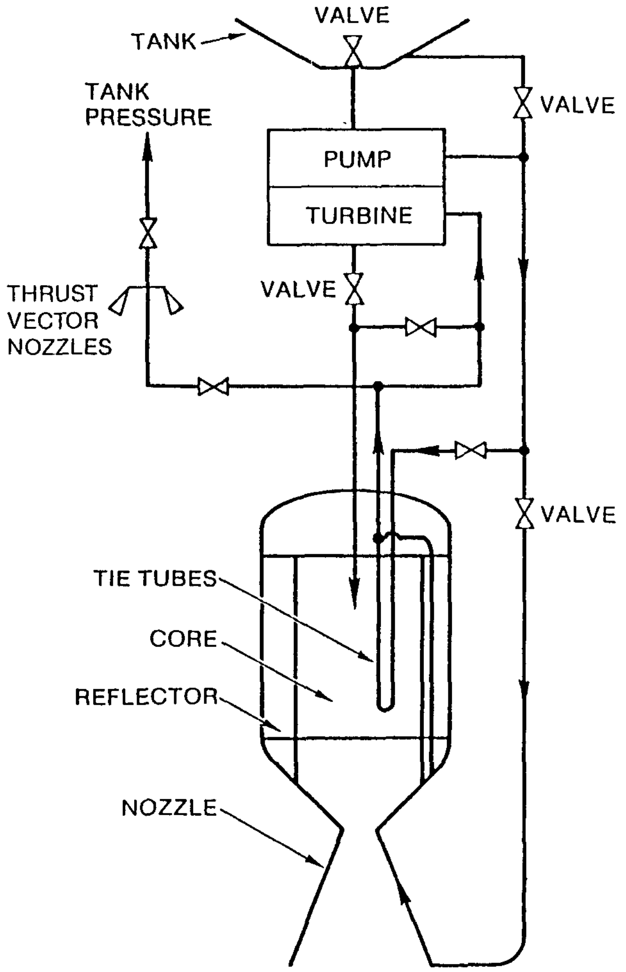


FIG. 1 NUCLEAR ENGINE SCHEMATIC

to drive the turbopump, and then the combined hydrogen flow passes through the reactor core, where it is heated to temperatures between 2700-3000 K. The hydrogen is then expanded through the ANRE exhaust nozzle to provide the engine thrust.

Many of the basic design features of the NERVA reactor, depicted in Figure 2, are included in the ANRE design. These include the NERVA fuel module design, the basic NERVA fuel element and tie element, the fuel module support and the radial support structure. The ANRE core is a cylindrical assembly of extruded graphite fuel modules held together by a highly-damped, spring-loaded, radial/lateral support system. The reactor core is surrounded by a beryllium reflector, in which are located rotating drums containing neutron absorbers for reactor control and shutdown.

A schematic cutaway of an individual fuel module is shown in Figure 3. The reactor thermal energy is provided through the fission of ^{235}U contained in fuel beads in the graphite elements. Multiple coolant channels, coated with ZrC, form flow passages through the elements. The exterior surfaces of the hexagonal fuel elements are also coated with ZrC, which protects the graphite from reaction with the hydrogen working fluid, and acts as an additional barrier to fission product or fuel diffusion release. Each fuel module consists of a central hexagonal tie element surrounded by six hexagonal fuel elements. The tie elements are structural members designed to provide axial support for the reactor core.

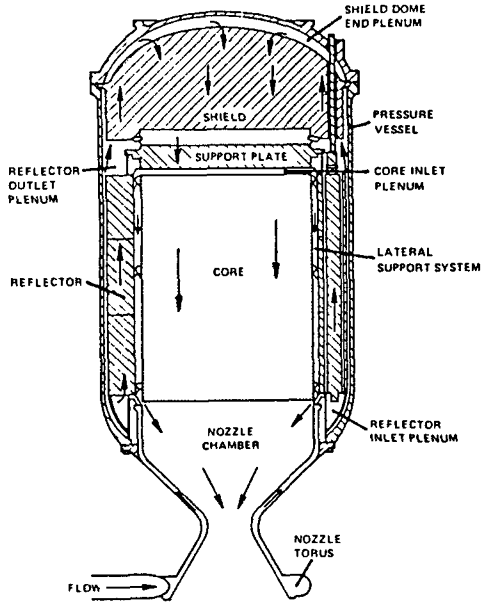


FIG. 2 SCHEMATIC OF NERVA NUCLEAR ROCKET REACTOR From Westinghouse Astronuclear Laboratory, "NRX-A6 Test Predictions, "WANL-TME-1613, November 1967

Figure 4 shows the ANRE baseline design along with the overall stage dimensions and weights. The basic performance characteristics of the ANRE concept are summarized in Table 1. The ANRE is designed to provide about 67 kN (15,000 lb) thrust. This thrust level was selected because it produced acceptable engine performance for the smallest achievable critical core design. The small ANRE baseline design is also believed to be in the range of engine sizes that would be most attractive for many of the currently planned Earth orbital missions (discussed later).

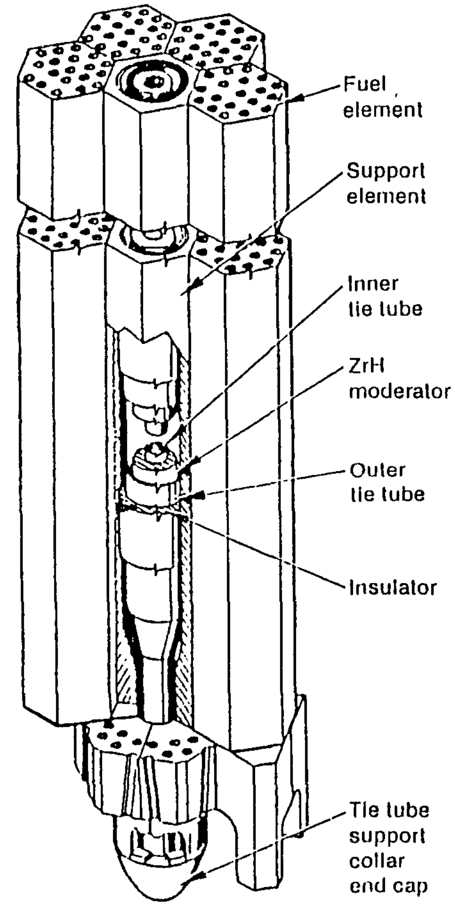
Major improvements in the ANRE engine concept over NERVA are expected to be (1) lower specific engine mass associated with improvements in structural materials, and (2) higher specific impulse associated with improvements in nuclear fuel technology. The higher specific impulse of the ANRE design will be discussed in more detail later because it represents the major performance advantage of nuclear propulsion systems over chemical propulsion systems.

DISCUSSION

To evaluate the performance characteristics of the ANRE concept, the ANRE operational characteristics were compared with a chemical and a nuclear-electric propulsion system for three different space missions. The three different space missions were (1) orbital transfer from low earth orbit (LEO) to geosynchronous earth orbit (GEO), (2) LEO to a lunar base, and (3) LEO to Mars.

LEO to GEO

The Earth orbital transfer missions of greatest interest to the Air Force can be divided into unmanned and manned LEO to GEO transfers. The most frequent unmanned missions are the multiple-manifest delivery missions made up of small geostationary satellites and GEO logistics missions. These packages are expected to weigh about 5,500 kg, and they impose no



- Six fuel elements around a central tie element
- Tie elements support the fuel elements through the support plate and provide the flow path for the gas coolant
- Fuel elements contain the hydrogen coolant channels
- Extruded fuel elements contain fuel beads in the graphite matrix
- The fuel elements are coated with ZrC to protect them from hydrogen corrosion
- Uc-ZrC fuel beads with ZrC coating use fully enriched U-235

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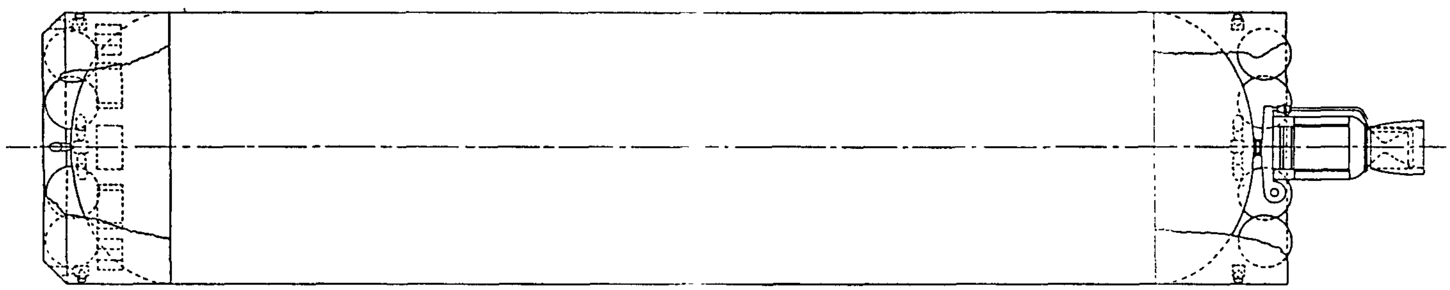
FIG. 3 ANRE FUEL MODULE



Dimensions (m):

Length	14.79
Diameter	4.57

Weight Statement (kg)

Gross Flight Weight	22,172
Cooling Propellant	909
Dry Weight	4,266
Total Propellant	11,543
Boiloff Propellant	84
Tank Weight	866

FIG. 4 ANRE BASELINE STAGE DESIGN

Table 1 ANRE Performance Characteristics

Power	320 MWt max
Chamber temperature	3000 K max
Chamber pressure	6.0 MPa
Thrust	0.9 - 67 kN
Specific impulse	970 s max
Mass	1818 kg
Operating cycles	80 (120 to 3600 s)
Operating life	36,000 s (full-power)

requirement for performance below optimum thrust levels (no g-load limitations). A few large GEO satellites that can be divided into packages no larger than 5,500 kg are anticipated. These larger satellites generally are g-sensitive, and cannot tolerate a thrust acceleration in excess of 0.1 g. These missions are all delivery-only missions, and do not themselves pose any retrieval requirements. Retrieval of transfer vehicles and mission equipment, however, may be beneficial from an overall mission cost standpoint.

A modest number of manned GEO sortie missions are anticipated in the time period through 2010. These missions are anticipated to require delivery of 5,500 kg to GEO and return of a 4,500 kg manned capsule. These missions do not impose unique transfer stage thrust and size requirements, but will impose unique radiation constraints on nuclear propulsion systems.

Lunar Missions

Lunar missions were considered in the propulsion system concept comparisons because one of the long-term objectives being considered for the nation is the development and support of a permanent base on the surface of the moon. Very large mission payload capabilities and frequent logistics support missions will be required for such an enterprise.

Mars Missions

The third category of missions considered, manned sorties to Mars are being considered in the same context as the development of a permanent lunar base. These potential missions are currently envisioned as single events, and it is not currently possible to speculate beyond that. However, the required vehicle size, if a chemical propulsion system were used, could be in excess of 0.5×10^6 kg (one million pounds).

Concept Comparisons

A summary of the general operational characteristics for the three engine technologies (chemical, nuclear, electric) for each of the above missions is provided in Table 2.

As indicated earlier, the primary advantage of the nuclear engine over a chemical engine is its high specific impulse (I_{sp}) defined as:

$$I_{sp} = \frac{\text{thrust}}{\text{mass flow rate}} \tag{1}$$

For equivalent thrust levels the higher I_{sp} of the nuclear engine requires a smaller propellant mass flow rate and less total propellant for a given mission. The high I_{sp} of the ANRE is a result of the use of hydrogen (the lowest molecular weight gas), and the relatively high exhaust temperatures that can be achieved.

Figure 5 shows the calculated specific impulse for the ANRE engine as a function of exhaust temperature. The specific impulse was calculated using the NETAP code (4) which was calibrated against nozzle

Table 2 Operational Characteristics of Three Engine Technologies

CHEMICAL

Low Specific Impulse (∿480 s)
High Thrust to Weight (∿0.5 kN/kg)
Bipropellant
Lunar Based Oxygen Generation Alternative
Impulsive Transfers
Flight Times
- LEO-GEO <1 day
- LEO-Lunar Base ∿3 days
- LEO-Mars Base ∿200 days

DIRECT THRUST NUCLEAR

Medium Specific Impulse (∿950 s)
Medium Thrust to Weight (∿0.03 kN/kg)
Monopropellant
Earth Based Propellant (LH2)
Impulsive Transfers
Flight Times
- LEO-GEO <1 day
- LEO-Lunar Base ∿3 days
- LEO-Mars Base ≤200 days

NUCLEAR ELECTRIC (ADVANCED ELECTROMAGNETIC OR ELECTROSTATIC)

High Specific Impulse (∿4000 s)
Low Thrust to Weight (∿2×10^{-6} kN/kg)
Monopropellant
Earth Based Propellant (Ar)
Spiral Transfers
Flight Times
- LEO-GEO >50 days
- LEO-Lunar Base >300 days
- LEO-Mars Base ∿2 years

and engine tests during the NERVA program. Also indicated on the figure are the temperature ranges for the different NERVA tests. An important consideration in the calibration was the hydrogen properties which were corrected to parahydrogen and for dissociation. The specific impulse at different exhaust pressures was calculated for a nozzle area ratio of 300 and included all losses associated with the engine cycle while at full power. From Figure 5, for an exhaust temperature of 3000 K, the predicted specific impulse is about 970 s. This I_{sp} is about twice that achievable with a chemical propulsion system (∿480 s).

The high I_{sp} advantage of the nuclear engine in reducing the required propellant mass is offset somewhat by the higher required engine mass-to-thrust ratio (∿30 kg/kN) compared to that of a chemical propulsion system (∿2 kg/kN). However, for most missions, the combined engine and propellant mass (discussed later) is considerably less for the direct-thrust nuclear engine than for the chemical engine. In addition, the risks associated with the possibility of explosive combination of propellant and oxidizer during launch and storage of propellants for a chemical engine are eliminated by the use of a monopropellant in the direct-thrust nuclear engine. This allows the ANRE to be designed so that the pressure vessel can retain the core under all coolant and nuclear accident conditions.

As indicated in Table 2, the chemical and direct-thrust nuclear propulsion systems both have sufficient thrust capability to utilize low energy impulsive

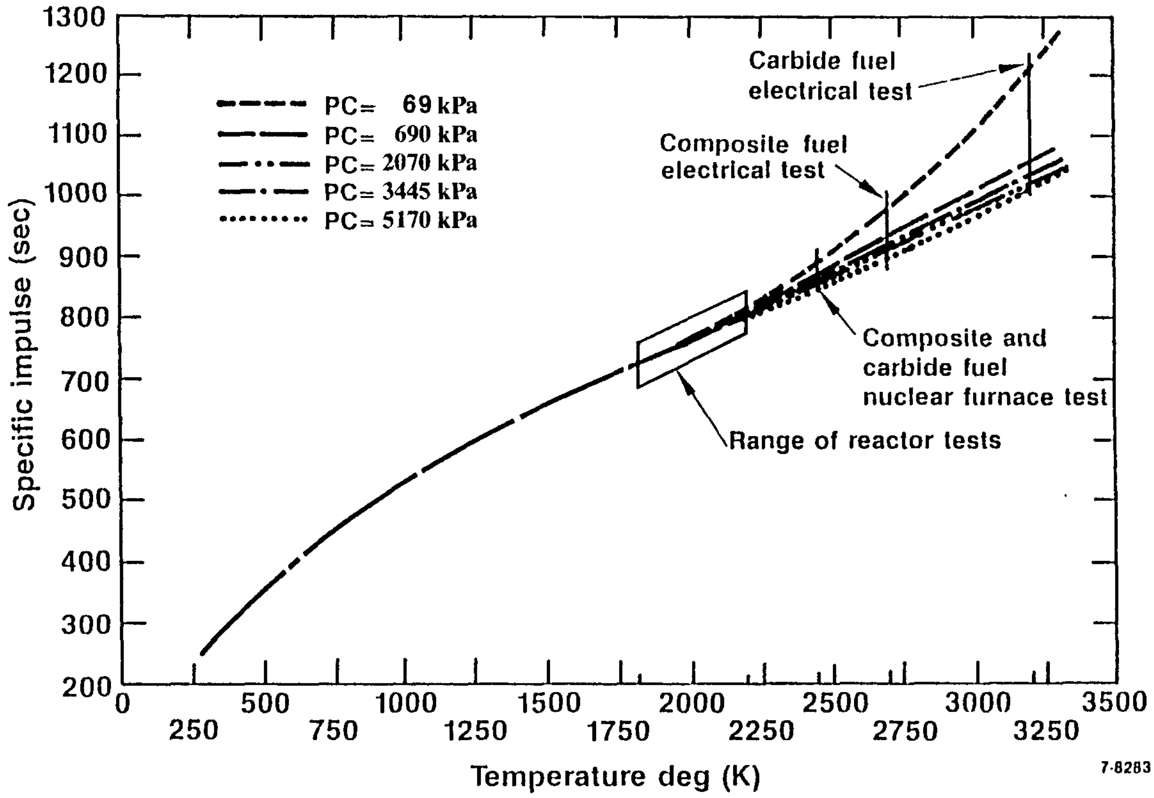


FIG. 5 ISP VS. EXHAUSE NOZZLE TEMPERATURE (area ratio = 300)

transfers. Because of the relatively high impulsive thrust levels achievable with the chemical and direct-thrust nuclear systems, both engines are capable of completing the three missions considered in this study in relatively short time frames with minimal fuel consumption.

Table 2 shows the nuclear-electric propulsion system is characterized by considerably higher specific impulses than are achievable with chemical or direct-thrust nuclear engines. Specific impulses for NEP systems typically range from 1500 s to 10,000 s. Despite the potential saving in propellant, however, NEP systems have very low thrust capability and very high mass-to-thrust ratios. This precludes their use in high gravity applications such as surface-to-orbit propulsion on any body within the solar system except very small asteroids.

The low accelerations available with electric propulsion also preclude the use of low-energy impulsive transfers possible with direct-thrust nuclear or chemical systems. As a result, electric-propelled vehicles typically follow spiral trajectories that require significantly more total propulsive energy. The necessary energy is characterized by the mission ΔV.[2] For example, a low-energy impulsive (Hohmann) transfer from low Earth to geosynchronous orbit requires a ΔV of 4160 m/s. For a low-thrust spiral trajectory, the ΔV is 5850 m/s. This difference partially offsets the potential savings available with higher specific impulse electric propulsion systems. In this example, the ΔV has increased by 41%, thus the electric propulsion specific impulse must be 41% higher than the comparable impulsive system, or no net propellant mass reduction will result. If the comparable impulsive system is a nuclear rocket with an I_{sp} of 970 s, a minimum specific impulse of 1370 s would be required for electric propulsion. This is easily met by more advanced electromagnetic or electrostatic thrusters, but could severely challenge nearer-term electrothermal systems.

The lower acceleration levels and higher energy requirements typical of electric propulsion systems and trajectories lead directly to system operating times (i.e., transfer times) that are much longer than those typical of chemical or direct-thrust, nuclear impulsive systems. For example, as indicated in Table 2, a LEO-GEO transfer requires in excess of 50 days even with very high power systems. Longer trip times represent a significant limitation of electric propulsion system applications. In particular, any manned mission in cislunar space would not be done using electric propulsion because of the long-duration life support requirements implicit in multiweek flights. In addition, electric propelled transfer to GEO from LEO includes significant times spent in the Earth's Van Allen belts, thus any payload sensitive to radiation would require more shielding than if an impulsive transfer were used.

The very long LEO to lunar and LEO to Mars flight times indicated in Table 2 for NEP also impose some technical issues that go beyond the undesirably long mission times. The very low propellant flow rates used result in storage requirements measured in months rather than hours. Long-duration storage of cryogens like argon is not yet entirely proven and would need to be, prior to the use of electric propulsion orbit transfer vehicles. The power requirements for electric propulsion also necessitate continuous source operation for long times. If the source is nuclear, the resultant potential radiation exposure could be

[2] ΔV (m/s) is the velocity increment imparted to the vehicle as a result of engine thrust over a period of time.

significant without additional shielding. Furthermore, the same long exposure concerns arise with propellant plume backflow contamination. While the propellant flow rates in electric propulsion systems are extremely small, the long exposure time can, if not carefully designed for, lead to undesirable build-up of propellant contaminants on spacecraft surfaces. A related concern with the propellant plume is its ionization level and resulting interactions with microwave radar and communications systems.

Since transfer time and propellant consumption are the major differentiating factors between the three propulsion system options, additional calculations were performed to quantify the propellant requirements for each of the three missions considered. The results of these calculations are shown in Table 3

Table 3 Mission/Stage Performance Summary

Mission/Stages	One-Way Transfer Time (Days)	Propellant Required (kg)
LEO-GEO		
Direct Thrust Nuclear	1	11,543
Chemical	1	24,091
Nuclear Electric	58	1,505
LUNAR BASE SUPPORT		
Direct Thrust Nuclear	3	42,123
Chemical	3	110,818
Nuclear Electric	>300	40,909
MARS BASE SUPPORT		
Direct Thrust Nuclear	200	59,545
Chemical	200	253,636
Nuclear Electric	>700	115,909

which summarizes the one-way flight transfer times and required propellant for the different options. For the LEO-GEO transfer the nuclear electric system has a clear advantage in propellant consumption and a clear disadvantage in the time required to perform the orbit transfer. The direct-thrust nuclear engine consumes about half the propellant of a chemical engine and accomplishes the mission in about the same time. For this application the chemical engine is the least desirable option and the selection between direct-thrust nuclear and nuclear electric depends on the value of time. There are some missions which require a fast transit time and if only one system is to be developed, direct-thrust nuclear would appear to be the best choice.

The propellant requirements for the three options for lunar base support indicate that both the direct-thrust nuclear and nuclear electric vehicles would have a large operational advantage over a chemical stage because of the latter's large propellant requirements. The long transfer time of the NEP provides no incentive to select it and thus the direct thrust nuclear vehicle is again the preferred choice.

For the Mars base support, the direct-thrust nuclear engine is again indicated as the preferred choice. The large propellant requirements of the chemical propulsion system and the long transfer time of the nuclear electric propulsion vehicle, make these options less desirable than the direct-thrust nuclear engine.

CONCLUSIONS

The results of this study demonstrate that the ANRE is an extremely versatile and flexible propulsion system capable of performing a variety of space propulsion missions. The principal advantage of direct nuclear propulsion over chemical propulsion is the reduced propellant requirements due to the higher specific impulse.

While nuclear electric propulsion systems can achieve higher specific impulses than direct thrust nuclear rocket engines, the relatively low thrust obtainable from NEP systems leads to long-duration transit times. Therefore, when mission time is critical, or where a single propulsion system is required to perform several missions a year, NEP is generally not desirable.

In considering the direct thrust nuclear options, the ANRE concept appears to be the most desirable because of the low technology risk involved. The NERVA technology on which the ANRE is based is fully developed. Therefore, utilization of this technology represents the lowest risk and earliest deployment strategy to develop an advanced nuclear propulsion system. The major performance improvements in ANRE over earlier NERVA engines are being accomplished by utilizing technology advancements in the materials and fuels areas to lower overall propulsion system weight and reduce propellant requirements.

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