Evaluation of High-Performance Space Nuclear Electric Generators for Electric Propulsion Application

Gordon Woodcock

Gray Research, Inc.

675 Discovery Drive, Suite 302, Huntsville AL 35806

256-880-9708; grw@mindspring.com

Abstract. Electric propulsion applications are enhanced by high power-to-mass ratios for their electric power sources. At multi-megawatt levels, we can expect thrust production systems to be less than 5 kg/kWe. Application of nuclear electric propulsion to human Mars missions becomes an attractive alternative to nuclear thermal propulsion if the propulsion system is less than about 10 kg/kWe. Recent references have projected megawatt-plus nuclear electric sources at specific mass values from less than 1 kg/kWe to about 5 kg/kWe. Various assumptions are made regarding power generation cycle (turbogenerator; MHD) and reactor heat source design. The present paper compares heat source and power generation options on the basis of a parametric model that emphasizes heat transfer design and realizable hardware concepts. Pressure drop (important!) is included in the power cycle analysis, and MHD and turbogenerator cycles are compared. Results indicate that power source specific mass less than 5 kg/kWe is attainable, even if peak temperatures achievable are limited to 1500K. Projections of specific mass less than 1 kg/kWe are unrealistic, even at the highest peak temperatures considered.

INTRODUCTION AND PURPOSE

Electric propulsion applications are enhanced by high power-to-mass ratios for their electric power sources. Present-day ion thrusters are about 2 kg/kWe, power processors at the 10's kWe level about 4 kg/kWe. Solar electric systems at the 25 kWe level are predicted to be about 25 kg/kWe, and nuclear systems at 100 kWe are predicted to have similar values. At multi-megawatt levels, we can expect thrust production systems to be less, perhaps considerably less, than 5 kg/kWe. Application of nuclear electric propulsion to human Mars missions becomes, performance-wise, an attractive alternative to nuclear thermal propulsion if the system (source plus thrust production) is less than about 10 kg/kWe.

Recent references have projected megawatt-plus nuclear electric sources at specific mass values from less than 1 kg/kWe to about 5 kg/kWe. Various assumptions are made regarding power generation cycle (turbogenerator; MHD) and reactor heat source design.

NUCLEAR ELECTRIC PROPULSION MISSION CONSIDERATIONS

Basic Principles - Electric propulsion systems are power-limited, in contrast to chemical propulsion systems, which are energy-limited. By power-limited we mean that system design is dominated by consideration of the fixed mass of hardware needed to generate the necessary power. Energy-limited systems design is dominated by the mass of propellant needed to produce the mission energy.

Ideal velocity increments (delta Vs) for in-space transportation missions range from a few to over 20 km/sec, with most interest for application of nuclear electric propulsion falling in the range 10 km/sec to 20 km/sec. These values are large compared to the maximum practically attainable jet velocity for chemical propulsion systems, about 4.7 km/s. Achieving jet velocities for chemical propulsion as near as possible to the maximum is therefore very important, and even then, high propellant fractions and often staging are necessary. Mission designs often make use of gravity assists to enhance performance; for example, the Cassini mission to Saturn used four such assists. The large propellant mass required to achieve high propellant fraction increases the launch mass required, and places great premium on minimizing spacecraft mass. Both effects are costly.

Electric propulsion can achieve any desired jet velocity, up to the speed of light ($3x10^8$ m/s). However, the mass required to produce the jet is a limiting factor, and this leads to an optimum Isp for any mission, depending on mission parameters and the performance of the electric propulsion system. Consider what is required to accelerate a 1-t. spacecraft by 20 km/s with a speed-of-light jet. The momentum transferred is 20 million kg-m/s = 20 million N-s. The momentum of light is E/c where c is the speed of light. The energy required is $(20x10^6)(3x10^8) = 6x10^{15}$ Joules = 1670 GWh, the output of a 1000-megawatt electric powerplant for about 2_ months. We must convert 0.06 kg of mass to radiation energy. With nuclear fission, considering typical powerplant efficiency, about 200 kg of uranium must be fissioned to generate this much energy.

If, however, we use a jet velocity 40 km/s (roughly optimum) the mass ratio is 1.65 and, neglecting electric propulsion mass, the propellant required is 650 kg. The energy required to accelerate the propellant is $5x10^{11}$ Joules, over 2_ months, 80 kW. This is a typical power output for a near-term space nuclear powerplant. On the other hand, if chemical propulsion were used to deliver the 20 km/sec the propellant required would be, again neglecting the mass of the propulsion system, about 70,000 kg. It is clear from this example that we need "enough" jet velocity but more jet velocity is not always better.

<u>Options</u> - Nuclear power is one of the main options for electric propulsion, the other being solar power. Beamed power, e.g. from a laser or microwave power beaming station on Earth, has also been investigated, and isotope power has been proposed. Nuclear power has the obvious advantage that its power availability does not depend on distance from the Sun. Some missions need power and/or propulsion far from the Sun, and nuclear power is the clear choice (for power levels of watts to hundreds of watts this may mean isotope nuclear power). At high power levels (multi-hundred kilowatts and up) it appears to offer mass advantages over solar power. On the other hand, at power levels below 100 kWe, solar power has the mass advantage. Solar electric systems also have a lifetime advantage for most applications but either system offers lifetimes on the order of years.

<u>Mission Application</u> - High power nuclear electric propulsion has been most notably considered for the human Mars mission application. The reason one would select nuclear electric propulsion for this mission is its flexibility to perform either conjunction-like or opposition-like profiles, and be reused for more than one mission opportunity. For an opposition-like mission the propulsion system will need to deliver about 25 to 35 km/s in about 200 days. (Thrusting half the trip time is typical.) Taking the median, 30 km/s in 200 days is 0.00174 m/s^2. Simple algebra shows f/m=a; p = fu/2; p/m = au/2. Assuming Isp 4000, u = 40,000 (approximately) and p/m is 35 watts per kg. Taking into account typical propulsion efficiency 60%, the electric power needs to be about 60 watts/kg. If 1/3 of the vehicle start mass is propulsion system, the propulsion system needs to generate 180 watts/kg which is 5.5 kg/kWe. Also note that if the vehicle start mass is 150 t. the power level is 60 watts/kg x 150,000 kg = 9 megawatts. Thus while electric propulsion performance analysis can become quite complex, a rudimentary calculation illustrates the approximate propulsion performance targets.

Ranges of Achievable Mass/Power Performance

We may note that for a wider range of applications, the useful range of mass/power performance is also wider. The following calculation is normalized to a unit mass (1 kg) spacecraft, which is presumed to be 75% powerplant and propulsion and 25% customer payload. Propellant is added to the 1 kg.

(1) Propellant Mass = 0.65*burnout mass = 0.65 kg; (2) Burnout mass = 75% powerplant & propulsion; (3) Jet velocity, Vj = 40 km/s; (4) Jet power = mV2/2 = $8x10^8$ watts for 1 kg/sec mass flow; (5) For typical efficiency, electric power ~ $13x10^5$ kWe for 1 kg/sec; (6) Powerplant & propulsion = 0.75 kg/kWe = 0.075 kW; (7) Flow, kg/sec = 0.075/$13x10^5$ = $5.7x10^{-8}$; (8) Duration = 0.65kg/$5.7x10^{-8}$ kg/s = 132 days.

For most missions, the velocity needs to be delivered in less than 2 years as a maximum. Multiply 10 kg/kWe by 730/132 to obtain 55 kg/kWe as a rough maximum acceptable mass/ power ratio.

Many studies and papers have been published on mass/power performance for nuclear electric propulsion systems. Reasonable agreement seems to exist for near-term technology, 100 kWe-class systems. Near term technology typically implies uranium oxide/stainless steel heat-pipe-cooled reactor technology, Brayton cycle energy conversion, and rotating electromagnetic generation of electricity. At lower power levels, Stirling cycle energy conversion may offer better mass/power performance. Several energy generation cycles have been proposed and analysed, as summarized in Table 1 in Section 4.

Mid-term technology is usually considered to employ refractory metal reactor fuel elements, probably still with uranium oxide, and heat pipe cooling. Turbines may require refractory materials, but the heat exchangers, except for the heat pipe unit, could be made of conventional materials.

Advanced technology implies direct reactor cooling by the cycle gas flow, graphite or carbide reactor fuel elements, and advanced materials for turbines and the recuperator heat exchanger. Note that a substantial technology legacy exists from the "high-temperature gas-cooled reactor (HTGR)" commercial power reactor programs in the UK and Canada.

Specific Observations Regarding Performance Estimates

Turbine temperatures: For helium gas-cooled reactors and turbines, it should be possible to use high-temperature materials which are not usable in chemically reactive gas flows. Carbon-carbon or carbon-SiC blades should be serviceable in a helium environment and could operate at temperatures above those considered practical for jet engine turbines, which operate in a hot oxidizing environment.

Reactor temperatures: Some authors seem to have extrapolated from nuclear rocket reactor experience, which has demonstrated 1-hour life and hoped for 10-hour life, to 10,000 hour life at the same reactor operating temperature. This is a major extrapolation. As far as I know, there is no test experience with graphite-based core materials at such lifetimes. The life limit in the nuclear rocket environment is hydrogen corrosion, which does not apply to an inert-gas-cooled reactor. However, fission products and fission product gas release, radiation damage, as well as other degradations, are applicable to long-life reactors and were not considered in the nuclear rocket case because life was limited due to hydrogen corrosion. If the helium flow is seeded by cesium (for an MHD generator), reactions between cesium and the hot reactor core must be evaluated and may affect temperature limits. Cesium has one stable isotope, which has a neutron cross section low enough to not be concerned about poisoning the reaction, but high enough to be concerned about depleting the seed concentration.

My view is that temperature limits 1500K - 2000K are more realistic, based on operating experience with graphite, helium-cooled high-temperature gas-cooled reactors for commercial power generation. Maximum short-term fuel temperature (hot channel max) was cited at about 1600K, with normal fuel operating temperature about 1150K. Fuel was rated at 3 full-power years, with burnup approaching 100,000 MWD/t. (Another source gave 50,000 MWD/t.) These reactors used highly enriched U235, with thorium 232 as a "phoenix fuel" rather than U238.

Reactor: For this application, the reactor design must include burnup as well as heat transfer limits. Rocket reactors have very low burnup and it is not an issue. Rocket reactors are also high pressure drop designs; closed-cycle Brayton systems must be very low pressure drop, as described below.

Superconducting Magnets: The referenced paper describes superconducting magnets for producing the magnetic field for the MHD generator. These are presumably located near the reactor. The reactor will leak a megawatt or so of radiation ... neutrons and gamma rays. Some (a kilowatt?) will be deposited in the magnets. Removing heat from a superconducting magnet at liquid helium temperatures is difficult. There is a tradeoff among distance from the reactor, shielding and cryostat mass, to minimize total mass. We are confident this mass penalty is greater than zero.

Turbo-compressors: Specific mass projections, based on aircraft engine experience, appear to be applicable. Note that a helium compressor may be considerably more massive. Air has 7 times the molecular weight of helium, and hence 7 times the density and 40% the speed of sound. A helium compressor is likely to need at least twice the number of stages for a given pressure ratio compared to an air compressor. Some analysts have proposed helium-xenon mixtures to solve the molecular weight issue; the mix apparently has most of the conductivity and heat capacity per unit volume of helium but is much easier to pump.

In an MHD design, an electric motor must be used to drive the compressor, and appears to have been neglected in some references. Its specific mass will be many times that of the compressor. I referred back to one of the solar power satellite thermal cycle studies of several years ago. It described a 32-megawatt electrical generator at 0.14 kg/kWe, not including its thermal control system. This estimate was made by General Electric, a builder of high-power aerospace electric generators.

Of course, if one uses a conventional turbine, the compressor may be driven by a shaft but the power output must come from a generator which will be as heavy per unit power as the motor. Note that for a typical closed Brayton cycle the compressor power is about twice the output power, so the advantage still goes to the conventional turbine.

Regenerator (also called recuperator): The regenerator mass per unit heat transfer area is estimated as 1 kg/m^2. This may be appropriate for a lightweight, moderate-temperature industrial design. Note that if the recuperator is a tube-in-shell design, the mass of a tube is pDLtρ (thin wall approximation) where terms are D diameter, L length, t thickness, and ρ material density. The heat transfer area is pDL, and the ratio m/A is just tρ, which is intuitive. For the temperatures of operation, up to over 1400K (over 2100F) the material must be a turbine-type nickel-based alloy. For these, ρ is about 8000 kg/m^3. For m/A to be 1 (just for the tubes), wall thickness must be 0.125 mm = 0.005".

Radiator: The radiator mass per unit area is a significant contributor to overall mass. 1 kg/m^2 is equivalent to a sheet of aluminum 1/2800 m = 0.36 mm thick. This is 0.014". If the material were a copper alloy as probably necessary at the planned radiator temperatures 500 - 700K (440 - 800F), the thickness would be 1/8000 = 0.125 mm = 0.005". Small fin radiators on spacecraft may indeed be so thin, but this radiator is another animal entirely and will be several times as massive. One cannot afford the mass penalty, pressure drop, or leak risk of piping the helium all over the large radiator area (for the cycle I analyzed, 10 MWe. the radiator area is about half a football field). Therefore, the design needs to be a compact(!) heat pipe heat exchanger which transfers waste heat from the helium flow to a large number of heat pipes which then distribute the heat over the radiator area. It will be > 1 kg/m^2.

MHD vs turbine: As cycle peak temperatures are reduced in the interest of realism, and radiator masses become more realistic, the higher efficiency of a turbine versus an MHD generator, combined with the reduced size of output generator versus compressor drive motor, may tip the balance in favor of a conventional turbine, if turbine materials and designs can be developed for helium use at selected cycle temperatures. The tradeoff should be based on point designs for comparative systems at realistic temperatures and component mass characteristics.

Sensitivities

Mission/performance sensitivities and representative estimates are presented in Figure 1. Estimates from other sources, especially at high power levels, varied widely, with some estimates well below 1 kg/kWe. Some of these estimates were linked to MHD generators (rather than turbine-generators). Others considered gas-phase (plasma) reactors along with MHD.

Note that specific power is sensitive to technology level and power output, and that NEP does not scale to low power well. Consequently, it may not make sense to produce a reactor at less than 100 kWe capability.

Since the efficacy of nuclear electric propulsion for human Mars missions seems to depend on achieving low values of mass/power, the present investigation was focused on high-power advanced technology reactors.

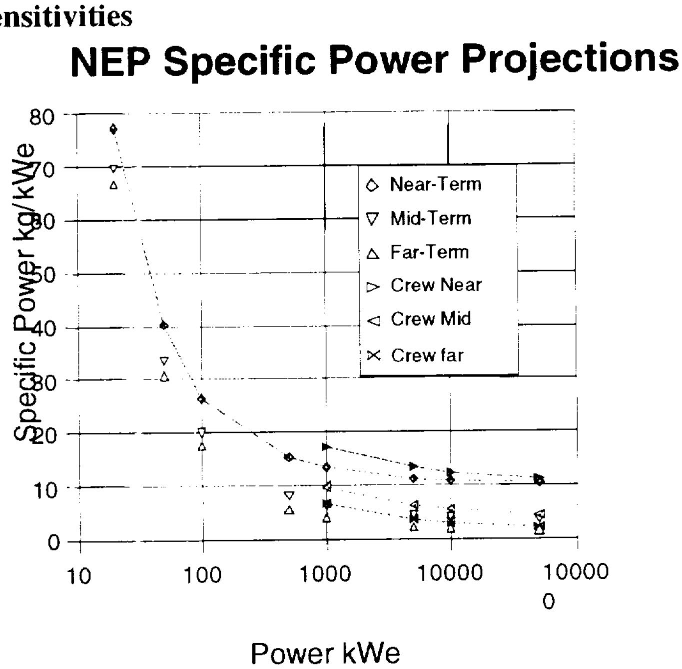


FIGURE 1. General NEP Mass/Power Sensitivity

Selection of Systems for Analysis

Table 1. Summary of Potential Cycles for NEP Power Conversion

Potential Cycles	Considerations
Thermoelectric	Cycle efficiency very low and max temperature restricted; thus mass/power relatively high.
Thermionic	Promise of good efficiency has never materialized; plagued by materials problems.
Brayton	Tends to large radiator areas but cycle is high efficiency.
Turbine	"Traditional" design; turbine temperatures may be limiting.
MHD	Potential for high cycle temperatures if reactor materials and life are capable.
MHD gas-core	Removes reactor (but not other) temperature limits; very speculative and difficult to develop.
Rankine	Higher average radiator temperature for same cycle bottom temperature; working fluids usually corrosive.
Steam	Classical terrestrial thermal power cycle; radiator temperatures too low for space.
Liquid Metal	SNAP-8 tried mercury (nasty material); modern designs use potassium; materials problems rampant.
Stirling	Because it involves a lot of heat exchange, tends to be preferred only for low-power (10's kW) systems.

Based on the considerations in the table, Brayton turbine and MHD cycles were selected. A specific objective was to estimate the advantages for MHD generation.

Cycle Analysis

The specific cycle analyzed was taken from the referenced paper. It is diagrammed in Figure 2. Helium is compressed by a compressor, shaft-driven in the case of a turbine expander and motor-driven in the case of an MHD expander. Two intercooler stages reduce the average heat rejection temperature. This improves cycle efficiency for a given cycle temperature ratio but increases the radiator area per unit heat rejection. There is an obvious trade here; the trade was not performed.

Helium leaves the compressor and enters a recuperator which preheats it by transferring heat from the helium leaving the turbine or MHD expander. This also improves cycle efficiency by increasing the average cycle temperature ratio for a given max/min temperature ratio. The recuperator enables practical cycle efficiencies above 25%, not otherwise achievable.

Leaving the recuperator, the helium enters the reactor where it is heated to the cycle maximum temperature. It then enters the expander (MHD or turbine). Leaving the expander, the helium enters the recuperator where it is further cooled by transferring heat to the compressor discharge flow. Leaving the recuperator the helium enters the radiator heat exchanger and is cooled to the cycle minimum temperature.

State points are presented in the Figure. Red text shows a representative MHD expander case, with maximum temperature 2000K, and black data are for a turbine expander with maximum temperature 1500K. These values represent my estimates of maximum practical cycle temperatures for these cases. Cycle minimum temperature was not optimized but is not far off optimum. Temperatures are K and mass flows kg/s.

<u>Pressure Drop Effect on Cycle Efficiency:</u> We used the same cycle diagram as the referenced paper. The pressure ratio across the expander can be expressed as a product of all the pressure ratios from each cycle state point to the next. Since multiplication is commutative, the pressure drop ratios may all be grouped together and combined into a single pressure drop factor G (G will be less than or greater than 1 depending on how the pressure ratios are expressed.) Then, one can substitute $G\, r_c^N$ for r_c^N, where r_c is the pressure ratio of one compressor stage and N is the number of stages, assumed all having the same pressure ratio.

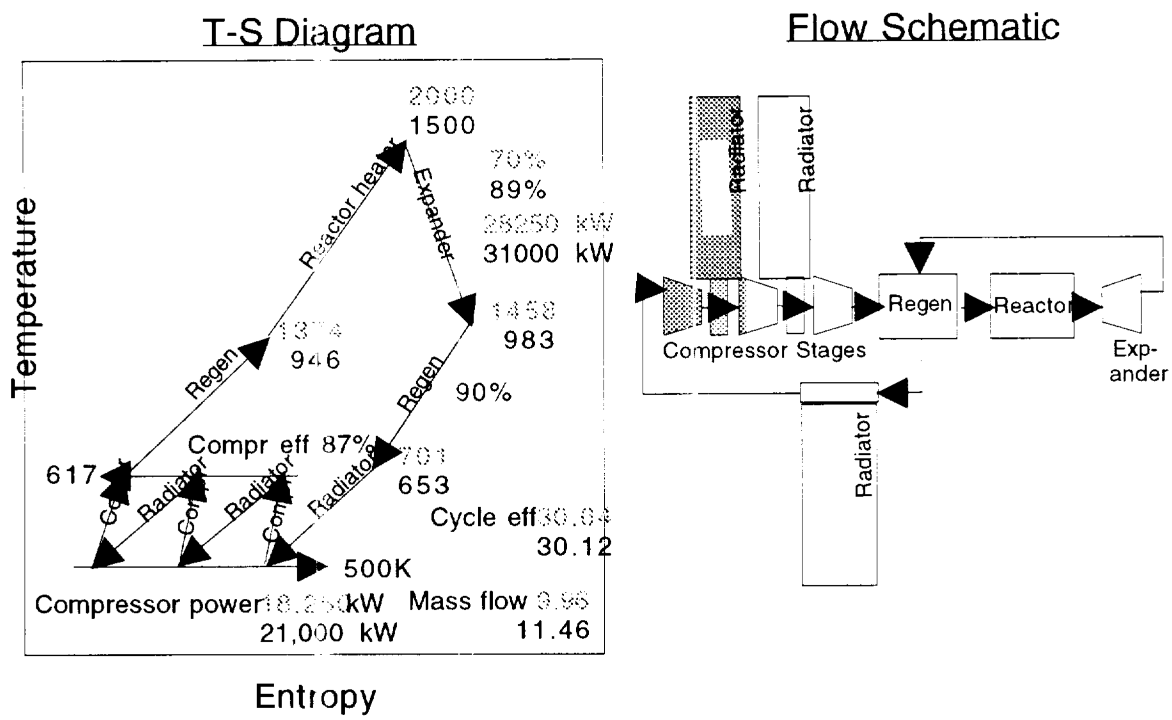


FIGURE 2. Brayton Cycle Diagram

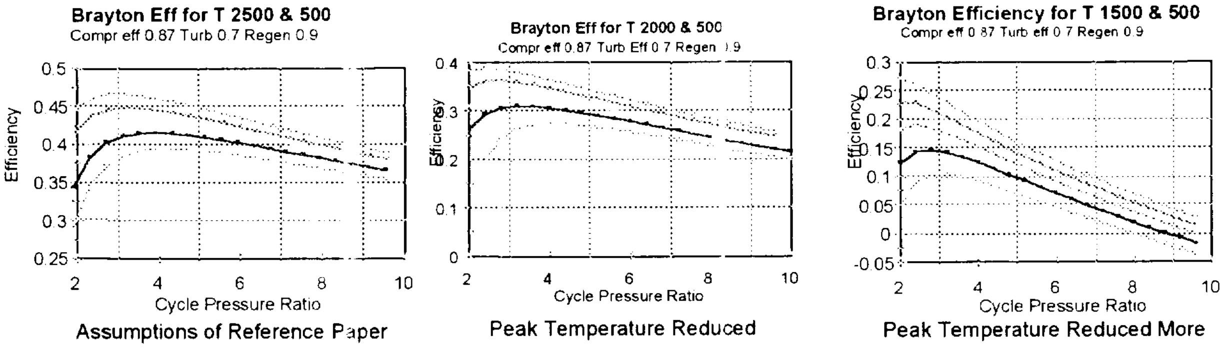
Without pressure drop the highest cycle efficiency occurs at a low compressor pressure ratio (and high mass flow, if one were to calculate it). With pressure drop, the highest cycle efficiency is less and occurs at reasonable, but still rather low, pressure ratios, as shown in Figure 3. Full optimization of a Brayton cycle requires, in addition to this, trading pressure drop versus duct and heat exchanger size and mass, compressor pressure ratio versus mass, recuperator effectiveness versus size and mass, and so on. However, Figure 3 permits selection of reasonable, if not fully optimized, state points.

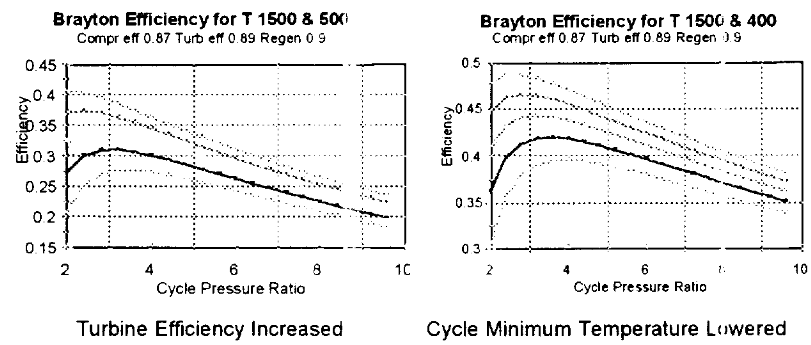
For purpose of analysis of achievable power-to-mass ratio, I selected the top center chart with pressure ratio 4 and pressure drop ratio 0.85, and cycle efficiency 30%. This reflects my skepticism of operating the reactor with a helium outlet temperature of 2500K for a long period of time. The pressure ratio is near optimum; I saw no reason to stay with the reference pressure ratio 8.

I also analyzed a representative turbomachine (as opposed to MHD) conversion cycle, with cycle maximum temperature 1500K and minimum temperature 500K, also with pressure drop ratio 0.85. This case, coincidentally, also has cycle efficiency 30%.

Full optimization of the cycle requires optimizing on pressure ratio, low temperature limit (assuming high temperature is fixed at maximum hardware capability), pressure drop versus mass of each major component, and radiator design.

I used a small C code to generate the cycle efficiency curves and a spread sheet to analyze mass/power ratio. Cycle state points were picked off from the C code and manually transferred to the spread sheet.





Effects of Cycle Parameters

Curves of pressure drop are from 1.0 (no pressure drop; top curve) to 0.9 (bottom curve) in 0.025 steps. Note that increase of turbine efficiency from typical MHD to typical turbomachine has a large effect on cycle efficiency. The baseline for the analysis of this paper was the lower left chart, with cycle pressure ratio 4 and pressure drop 0.925. However, the somewhat lower minimum temperature of the immediate left chart would raise the cycle efficiency about 10 "points".

FIGURE 3. Example Sensitivities of Cycle Efficiency to Pressure Ratio, Pressure Drop and Temperatures

Reactor Performance: The reactor design was assumed cylindrical, similar to a NERVA reactor. Two considerations were used to size the reactor: fuel burnup and heat transfer. For simplicity I assumed the reactor core was U235C2 and graphite. A practical design might add thorium-232, as needed to get the right criticality and to provide some breeding to counteract burnup. No neutronics analyses were done. The reactor is certainly large enough. The main reasons for a neutronics analysis are to size the reflector, assess controllability based on reflector drums, and determine reasonable burnup and benefits of thorium addition.

Fuel load was based on 80,000 MWD/ton for 2 to 5 year full power life, about 9% burnup, and the physical size of the reactor was based on a 20% void fraction for helium passages, an assigned pressure drop of 3 psi (about a fifth of the allowable for the entire circuit), and the necessary heat transfer area. The graphite mass was determined by balance of volume after fuel load. Viscosity was determined by a kinetic theory relationship:

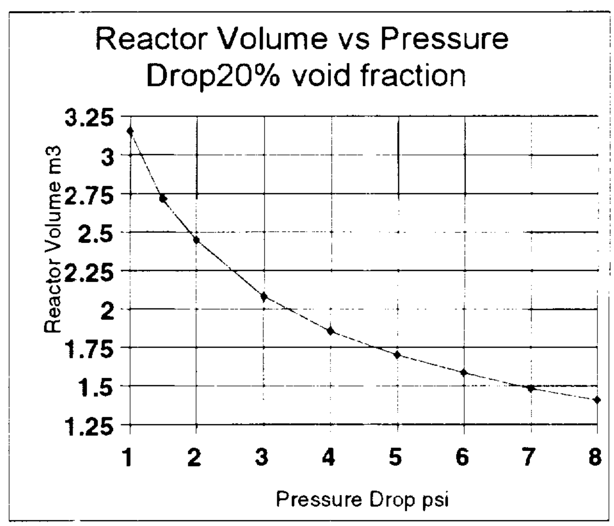
$m = 2.6693 \times 10^{-5} (MT)^{0.5}/(d^2\Omega)$ where the result is in cgs units. For mks units, divide by 10, which was done on the spread sheet.

Averages were used, where a real heat transfer analysis would consider several points in the helium passages to assess heat transfer versus helium temperature and other flow conditions. The Reynolds' number in the passages (3000) is lower than I would like, but is probably OK. Friction coefficient was an assumed value. A 20 cm (8") reflector was assumed, with reflector controls assumed included in the reflector mass. The reactor size result is somewhat too small for mass flow (ρAV), so further design iteration would be required for a real design. However, this seems to be in the ballpark. Main reactor parameters are given in Table 2 and Figure 4.

Turbomachine: Used a specific mass of 0.025 kg/kW shaft power. Various sources suggest this is about right. However, none of these sources described helium turbomachines; it is quite possible that because of the low molecular weight, helium machines will need so many more stages they will be significantly heavier. For the MHD expander, I used a specific mass of 0.05 kg/kWe. There is little data on which to base this estimate. It has only a small effect on overall power-to mass ratio unless the specific mass is much greater.

Table 2. Main Reactor Parameters

Parameter	MHD	Turbine	Parameter	MHD	Turbine
Electric Power Output (MW)	10	(same)	Graphite Mass (kg)	2560	2608
Thermal Power (MW)	32.6	33.2	Reflector Thickness (m)	0.2	0.2
Cycle Max Temperature (K)	2000	(same)	Reflector Mass (kg)	2148	2171
Cycle Min Temperature (K)	500	(same)	Vessel Mass (kg)	540	545
Cycle Max Pressure (Mpa)	10	(same)	Total Mass (kg)	5580	5658
Cycle Pressure Ratio	4	(same)	Alpha, reactor only (kg/kWe)	0.56	0.57
Pressure Drop Ratio	0.85	(same)	Heat Transfer Passage L/D	400	(same)
Reactor Void Fraction (%)	20	(same)	Passage Size (mm)	5	(same)
Delign Life (yr)	2	(same)	Delta P (Pascals)	20,700	(same)
Total Output (MW Days Thermal)	23.480	24.255	Reynolds' Number	2900	(same)
Total Uranium Burn (kg)	27	27.5	H, kcal/m^2-K	0.19	(same)
Assumed Burnup (MWD/t)	80,000	(same)	Reactor Volume (cu m)	2.04	2.07
Fuel Load (kg U235)	298	303	Reactor Length (m)	2	(same)
Burnup (%)	9.1	(same)	Reactor Diameter (m)	1.14	1.15
UC2 Load (kg)	328	334			



Regenerator/recuperator: A tube-in-shell design was assumed, and heat transfer area required was factored from the reactor heat transfer analysis, considering delta Ts and total heat transfer required. I used a somewhat greater mass/area than in the reference paper, because the latter results in very thin wall tubes. Also, I added a calculated allowance for shell mass. Since this shell will run quite hot, I used a low stress value for the shell, and assumed it would have the density of a turbine alloy.

Radiator: Radiator area was calculated based on total heat rejection and assumed average temperature. The average temperature will trend close to or below the cycle minimum temperature because of temperature drops between the helium minimum temperature and the actual heat rejection temperature. The radiator was assumed to be a finned heat pipe design, with flat fins between the pipes externally and circular fins inside the helium-to-heat-pipe heat exchanger manifold. Sodium or potassium appear to be suitable heat pipe fluids for the temperature range considered. At a somewhat lower cycle minimum temperature, water could work. Thermal power per heat pipe, and length of the pipes, is probably pushing the state of the art. Capillary-pumped loops might be better.

I used a numerical integration to roughly iterate on fin thickness. Fins too thin, too much delta T and radiator weight goes up. Fins too thick, fins weigh too much. There is an optimum, and getting the complete optimization is a fair amount of work; for example, it also involves varying the heat pipe size and spacing. My optimization was rough, but I think the radiator mass is representative.

Main recuperator and radiator parameters are shown in Table 3 and Figure 5.

The radiator is actually in 3 parts. One section rejects heat in cooling the helium from regenerator outlet to compressor inlet, and the other two sections reject heat from the compressor intercooler segments of the cycle. The radiator total area is so large as to dwarf the rest of the system, although at 3743 sq m (about 3/4 of a football field) this area would only generate a little over 1 megawatt as a high-performance solar array.

Table 3. Main Recuperator and Radiator Parameters

Parameter	MHD	Turbine	Parameter	MHD	Turbine
Recuperator Heat Transfer (kcal/s)	9427	4726	Emissivity	0.9	(same)
Recuperator Heat Transfer (MWth)	39.46	19.782	Sides	2	(same)
Required Heat Transfer Area (m^2)	586	675	Heat/Unit Area (Stefan-Boltz) kW/m^2	6.05	(same)
Tube Diameter (mm)	6	(same)	Area Required (m^2)	3743	3837
Vol/Area (m^3/m^2)	0.0045	(same)	Radiator delta T (recup out–compr in K)	201	153
Recuperator Volume (m^3)	2.64	3.04	Heat Radiated (MWth)	10.46	9.2
Total Tube Flow Area (m^2)	0.31	0.356	Radiator Delta T Intercoolers (K)	117	(same)
Number of Tubes	10,940	12,593	Heat Rejected Each (MWth)	6.08	7.0
Recuperator Cross-Section (m^2)	0.93	1.07	Estimated Radiator HTX area (m^2)	1132	1160
Recuperator Diameter (m)	1.09	1.17	Heat Pipe Diam (cm) & Length (m)	5; 20	(same)
Recuperator Length (m)	2.84	(same)	Heat Pipe Spacing (cm)	15	(same)
Tube Mass per Unit Area (kg/m^2)	1.5	(same)	Area per Pipe (m^2)	3	(same)
Tube Wall Thickness (mm)	0.2	(same)	Number of Pipes	1248	1279
Shell Stress (Mpa)	34.5	(same)	Thermal Power per Pipe (kWth)	18.14	(same)
Shell Wall (mm)	4	4.2	Pipe Wall (mm)	0.2	(same)
Tube Mass (kg)	880	1012	Mass per Pipe (kg)	5.03	(same)
Shell Mass (kg)	365	365	Fin Thickness (mm)	0.2	(same)
Baffles & Misc. Mass (kg)	73	85	Fin Area (m^2)	2495	2558
Total Recuperator Mass (kg)	1318	1521	Fin Mass (kg)	3992	4092
Heat Rejected (MWth)	22.63	23.2	Radiator Mass not incl manifold (kg)	10,263	10,520
Radiator HTX Delta T (K)	25	(same)	Heat Transfer Area per Pipe (m^2)	0.91	(same)
Fin Delta T (K)	50	(same)	Manifold Wall (mm)	1	(same)
Average Temp (K)	475	(same)	Manifold Mass (kg)	5763	5908

RESULTS

The specific mass summary for the MHD system is as follows:

MHD Element	Raw Alpha	Turbogenerator Element	Raw Alpha
Reactor	0.558	Reactor	0.566
Generator	0.141	Compressor and Turbine	0.130
Recuperator	0.132	Recuperator	0.152
Compressor & Drive	0.314	Generator	0.15
Radiator	1.026	Radiator	1.052
Radiator Manifold	0.576	RadiatorManifold	0.591
Total	2.752	Total	2.64
Integration (25%)	0.688	Integration (25%)	0.66
Total Estimate (kg/kWe)	3.441	Total Estimate (kg/kWe)	3.301

The turbogenerator system differs from the reference system as follows:
(1) Cycle max temperature 1500K instead of 2000K; (2) Expander is turbine rather than MHD device, efficiency 0.89 instead of 0.70; (3) No motor required to drive compressor (it's shaft-driven); and (4) Shaft-driven rotating generator required to produce electrical power

Although much greater than the estimates of the reference paper, these are still very lightweight systems compared to most estimates of space nuclear-electric systems. The reasons for the high performance are high power (10 megawatts) and high cycle temperature.
Comparing the two systems, the reactor, recuperator and radiator are almost identical. Cycle efficiencies are almost the same, the reduced maximum temperature for the turbogenerator system is compensated by the greater turbine efficiency compared to the MHD machine. The rotating system has slightly less mass than the MHD machine, compressor and drive. This is mainly because the generator has about half the power rating of the compressor drive.

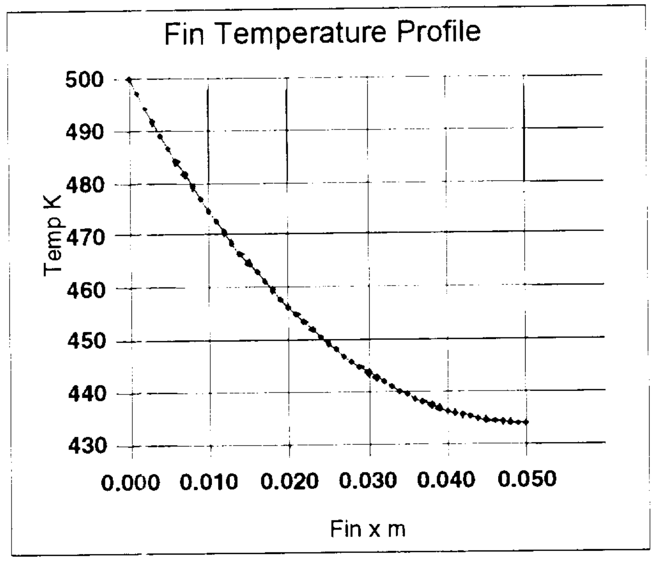


Figure 5. Fin Temperature Profile

CONCLUSIONS

(1) Nuclear electric propulsion can reach performance levels applicable to most mission categories, including (a) Inner solar system complex profile; (b) Outer solar system simple and complex profiles; (c) Beyond solar system, and (d) HEDS Mars and asteroids. A simple profile is one that merely reaches an objective, such as a flyby, while a complex profile inserts into orbit, or lands, or retrieves a sample.
(2) The technology is well-understood in principle. (a) Numerous reactor and power conversion technology progams have developed basic data; and (b) mature analytical capabilities exist.
(3) Mass/power ratios less than 5 kg/kWe are probably achievable. While conceptual analyses such as presented here are generally optimistic, I calculated 3.5 kg/kWe and this indicates 5 is probably achievable.
(4) A direct-cooled closed cycle helium or helium-xenon cycle and reactor are needed to achieve low mass/power ratios.
(5) A turbine-based system appears to provide performance about equal to MHD system with significantly lower maximum temperatures (e.g. 1500K vs 2000K) and more mature technology
(6) Projections of mass/power 1 kg/kWe or less do not appear realistic for any foreseeable technology.

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REFERENCES

Litchford, Bitteker & Jones (2001) "Prospects for Nuclear Electric Propulsion Using Closed-Cycle Magneto-hydrodynamic Energy Conversion", AIAA 2001-0961