Nuclear Electric Propulsion and Human Space Exploration

William C Strobl[1]
Retired General Dynamics Vice President and
Program Director of Advanced Launch Systems
Carlsbad, CA 92009

Richard Hora [2] and James Mildice [3]
Earth Space Applications, Inc.
San Diego, CA 92074

Abstract

Today, many countries are evaluating future human missions to the moon, near planets, and asteroids. However, in the early 1960's the United States funded studies to define vehicle systems to send humans to and from the planet Mars. Early development was in progress of a nuclear thermal rocket engine (NERVA) for the potential vehicle propulsion system. The mission studies identified that we would begin with the NERVA rocket and transition to nuclear electric type propulsion, when it became available. The human mission studies were cancelled in 1965.

Since then, we have defined new technologies and learned much more about the Moon, Mars and asteroids. Finding large deposits of water on the Moon and Mars, and finding that Mars' atmosphere is dense enough to allow aero-braking are major discoveries that will aid in human space flight. Developing an infrastructure for traveling from Earth's Moon to Mars could use some assets of the Moon, reduce the energy required to go to Mars, reduce departure and capture thrust requirements, reduce the travel time and reduce the size of the Earth to Earth-orbit boosters. We need to define, evaluate and select a propulsion system that could take advantage of these desired capabilities.

Scientists have identified and begun development testing of a nuclear electric system and others have defined a type of laser propulsion. Thus all of these propulsion systems should be evaluated for technical viability and ability to satisfy a budget-driven schedule. If it is shown that these systems can be brought to fruition within two or three years of each other then the most economical, safest, lowest astronaut stress system, that delivers comfortably the above projected need should be selected.

The senior author of this paper has had experience designing conceptual human Mars propulsion systems in 1962-1965, 1986 –1988 and 2008-2010 using both types of nuclear propulsion as well as chemical propulsion. At the AIAA Space 2010 conference we presented a paper comparing chemical, nuclear thermal and nuclear electric propulsion systems. One of the attendees questioned the selection of the nuclear electric system because of the very large radiator it required. The question was answered verbally, that answer is expanded herein.

This paper will illustrate the use of system element selection and sizing slopes to reduce the radiator size without significantly increasing the system weight.

I. Introduction

President Obama has directed NASA to rethink the design and development of a Heavy-Lift Launch vehicle to satisfy requirements for human space flight to the moon, Mars, and Asteroids. A comprehensive Architecture study should be conducted of all potential propulsion systems to determine the proper requirements for the propulsion systems to transport humans from Earth to the moon, Mars, and Asteroids. Some scientific personnel want to select the nuclear thermal rocket technology of the middle 1960's for the Mars and Asteroid missions and ignore new and evolving technology. Most likely we won't fly humans to Mars for at least 20 to 25 years, so why rush into selecting a system now, especially since Dr Chang-Diaz has manufactured and ground tested a magneto-plasma rocket (Vasimr VX-200) that is planned for flight test in the next few years.

We presented a technical paper (Propulsion Technology and Human Space Exploration [1]) at the AIAA 2010 Space conference comparing chemical, nuclear thermal and nuclear electric propulsion systems that indicated nuclear electric was a viable contender. Table 1 lists some of the nuclear electric propulsion system capabilities. We believe that the nuclear electric propulsion system is more Astronaut friendly and will survive much longer than a nuclear thermal system.

---

[1] Retired, 2906 Via Pepita, Carlsbad, CA., AIAA Senior Member
[2] Vice President – Business Development, PO Box 224, Poway, CA 92074, AIAA Member
[3] CEO, PO Box 224, Poway, CA 92074, AIAA Senior Member

Table 1. Nuclear Electric Propulsion Capabilities

1. It completes a round trip to Mars (via Earth's moon) in 230 days. It can get to Mars in 75 days.
2. It thrusts continually there and back.
3. It can provide variable artificial gravity for the entire mission
4. The system is reusable not expendable. It can go again and again.
5. It stays in space, so it's not scarred with reentry protection
6. It parks at and flies from lunar orbit, so there is no danger of radiation or unwanted materials falling to Earth. It could obtain (some or all) of its propellant, structural materials, water, and oxygen from the moon saving the cost of provisioning from Earth
7. It can dispose of itself by flying to the Sun
8. It is designed for a 5-year life without refueling the nuclear fuel.
9. It can significantly expand robotic space exploration (more science to farther targets in less time).

The major negative of the nuclear electric propulsion is the size of the radiator to expend the waste heat. The radiator size can be reduced significantly with the combination of some new technology, element design selection and selection of a solution slightly off optimum weight. We will disclose our approach for a major reduction of the size of the radiator.

Discussion

Vehicle System concepts with nuclear electric propulsion for human transportation to Mars and Asteroids proposed in the 2000 to 2003 time period reflected the "in hand" technology and design approach of that time. Many concepts applied a Brayton/Regenerator Cycle with a liquid metal cooled nuclear reactor, a single sided heavily armored radiator, a turbine inlet temperature of 1300 K and a radiator heat sink temperature of 250 K. At that time there were predictions that in 10 to 15 years technology advances would increase the turbine inlet temperature to 1500 K, and reduce the size and weight of other Brayton Cycle elements. Also predicted was that the technology trend would continue to evolve and in 20 to 25 years the turbine inlet temperature would increase to 2000 K.

Ten years have passed and commercial aircraft are already approaching a turbine inlet temperature of 1500K for cruise and 1800 K for short 5 minute bursts (like take-off). NASA Glen Research Center has had a contractor design, manufacture and test a prototype heat pipe radiator segment that is much lighter than that which was employed in past designs. One meteorite puncture of the armored radiator design would be a loss of mission whereas a heat pipe design could accommodate many punctures with only a slight reduction of capability. Some commercial nuclear industry companies in the United Sates are considering a gas cooled, graphite moderated Pebble based reactor, similar to those used commercially in England and South Africa. It is considered safer and lighter than the liquid metal cooled reactor. Also the magneto-plasma Vasimr VX 200 thruster is planned for flight in the next few years.

Since we will probably not fly to Mars and Asteroids for 20 to 25 years these technologies will most likely be considered "in hand" technologies for that timeframe. Technology development of a turbine inlet temperature capability of 2000 K for cruise would make the commercial aircraft industry very, very happy.

The first step we illustrate the weight benefits acquired with these technologies and design approaches to achieve the **minimum** weight design for the electric power system for the overall vehicle system. The second step will describe a methodology to significantly reduce the size of the radiator with a **total vehicle system** weight increase of only about 10% or less.

II. Nuclear Electric Propulsion System Weight Minimized

The example that we selected for analysis in this paper is a 20 MWe nuclear electric propulsion system because it can satisfy all the conditions stated in Table 1 with safety and performance margins. The vehicle system would consist of three elements: Life Support System for 4 Astronauts (including a solar flare shelter and artificial gravity capability), Propulsion System (magneto-plasma thrusters, propellant tanks and lines, structure, and cables), and a 20 MWe Brayton/Regenerator Cycle electric power generating system.

A standard Brayton/Regenerated Cycle is illustrated in Figure1. The figure also illustrates values of temperature, pressure, and electric and thermal power as required for each element. Figure 1 also portrays the system in an excel spreadsheet as an analysis tool. The data is analyzed on two sheets the first of which is shown. On this sheet we can change any parameter to determine sensitivities of each on the effect on other elements. Sheet two (not shown) calculates the size of the radiator based on the NASA Glen Research Center radiator sizing equations. Excel Solver iterates on the mass flow and turbine pressure ratio to achieve a system thermal balance. Currently a manual process calculates the weight. As a result minimum weight can be determined as well as sensitivities to that weight by changing any parameter. Comparing results to a NASA Glen Research Center detailed software program validated the excel spreadsheet model.

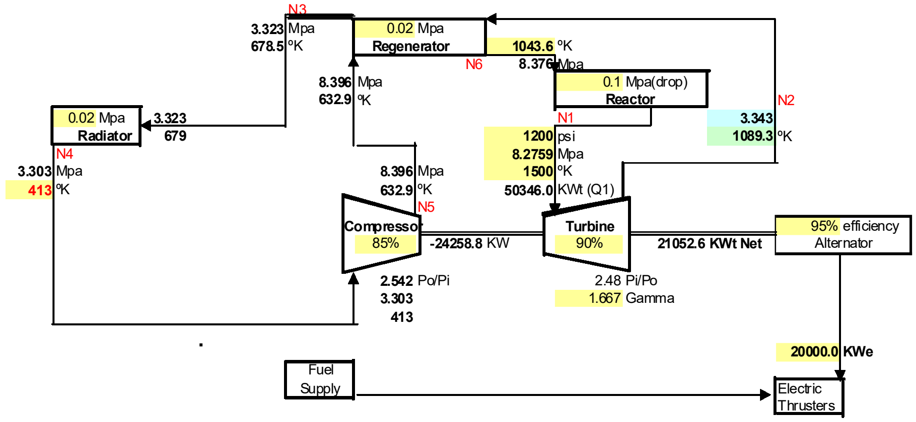


Figure 1. Standard Brayton/ Regenerator Cycle

The Brayton/Regenerator Cycle evaluated contains the following elements:

1. A Gas Cooled, graphite moderated, Pebble Bed Reactor - it weighs about 50% of a liquid metal cooled reactor.
2. A two sided radiating Carbon-Carbon Heat Pipe Radiator Design – NASA Glenn Research Center prototype radiator section design that was manufactured and tested. – 40% lighter than an armored standard radiator.
3. A new ESAI concept AC Alternator (Variable Speed Universal Machine) and Power Management And Distribution (PMAD) – reduces weight of Alternator and PMAD by 40%.
4. 1500 Kelvin Turbine inlet temperature – reduced flow rate which in turn reduced the size/weight of the turbine, compressor, radiator, regenerator and connecting lines.
5. Gaseous Helium is the working fluid

These nuclear electric power-generating elements yield an optimum system dry weight of about 2.71 kg/kWe. Figure 2 illustrates a comparison of the example electric power generating system to that which was envisioned in 2000 to 2003 timeframe. The mass values of the elements of the systems illustrated were collected from various Internet sources, technical papers, and published NASA Technical Notes, then averaged and put into our excel spreadsheet software.

The 2000 to 2003 timeframe system assumed a liquid metal cooled reactor, a 1300 K turbine inlet temperature, an armored single sided radiating surface radiator, a 250 K radiator heat sink temperature, and DC power and related PMAD, which yielded an approximate mass of 100,300 kg and a radiating area of about 15,600 square meters. Applying the same 1300 K turbine inlet but changing to AC electric power, a gas cooled Pebble based reactor,

Table 2. Electric Power Generating System Mass and Size Comparison

	20 MWe	Modifications	20MWe	20MWe	20MWe
EstimatedTimeframe	2000-2003			2012	2025
Turbine Inlet Temperature (K)	1300		1300	1500	2000
Radiator Exit Temperature (K)	400		400	413	420
Radiator Sink Temperature (K)	250	1. Lower Heatsink Temperature	20	20	20
Weight Items (kg)					
Reactor & Shield	18000	2. Liquid Metal Cooled to Gas Cooled,	9000	8000	7300
Power Generating	23900	Graphite Moderated, Pebble Bed	21900	16000	8900
Alternator & PMAD System	12400	3. DC to AC system	10700	10700	10700
Main Radiator	37000	4. Armored Design to Heat	22500	14500	8100
Structure	9000	Pipe System	6500	5000	4500
Power Generating Total	100300		70600	54200	39500
System Size (kg/kwe)	5.02		3.53	2.71	1.98
Radiating Area (Sq. Meters)	15,538		14,222	9200	5116
Area/2	7769		7111	4600	2558

a two sided radiating surface heat pipe radiator design, and a lower heat sink temperature reduces the weight by almost 30,000 kg and reduces the radiator size by about 50% . The 20 K radiator heat sink temperature assumes that the vehicle fly's with the edge of the radiator to the sun. This approach needs further evaluation verses slow rolling the vehicle for optimum thermal control of the propellant and life support system as well as the method selected for artificial gravity. However increasing the turbine inlet temperature to either 1500 K or 2000 K will drastically reduce the weight and the size of the radiator, reactor, regenerator, structure, fluid lines, and the helium mass flow rate. A two-sided radiating heat pipe radiator size is reduced by 35% with a turbine inlet temperature of 1500 K and 65% with an inlet temperature of 2000 K.

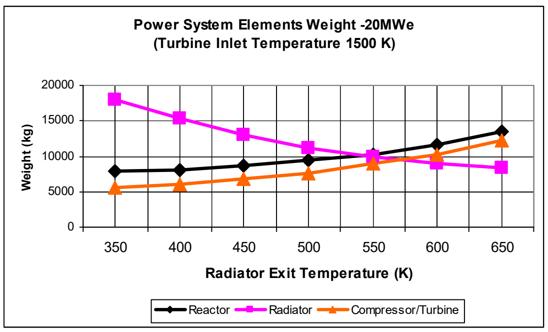
III. Nuclear Electric Propulsion with Reduced Radiator Size and Weight

The 20 MWe system with a 1500 K turbine inlet temperature was evaluated with our electric power generating model for weight sensitivity to variations in the radiator exit temperature. As the radiator exit temperature decreases the size of the radiator and increases the helium working fluid flow, the size and weight of the reactor, turbine, compressor, and regenerator. Figure 2 Plots weight verses radiator exit temperature of every system element. The lowest total weight occurs at a radiator exit temperature of about 400 K with a radiator size of 4600 square meters. At about a radiator exit temperature of 575 K the total weight is 62,700 kg with a radiator size of about 2960 square meters. Continuing to a higher radiator exit temperature increases total weight more rapidly, mostly due to the steeper slope of the regenerator curve.

20 MWe Power Generating System Weights

1500 K Turbine Inlet Temperature

Lowest Weight		Selected Weight
54,200	Weight (kg)	62,700
4,600	Radiating Area/2 (m^2)	2,960



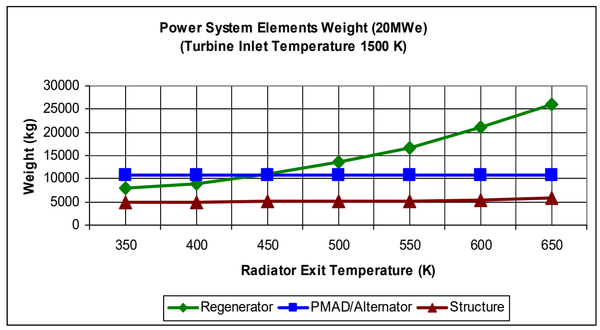
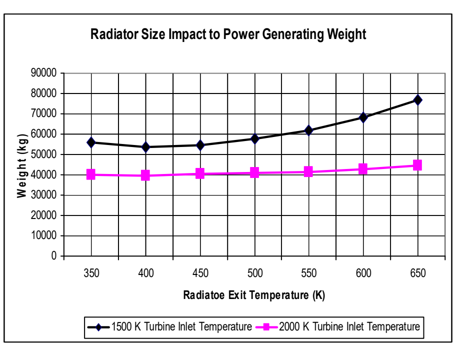


Figure 2. Plot of Weight verses Radiator Exit Temperature for a 20 MWe System

The reduction in radiator size is much greater for the electric power system at a turbine inlet temperature of 2000 K as illustrated in Figure 3. Assuming a Mars mission departing and arriving Earth lunar altitude and 6,000 kilometer altitude at Mars with a total timeline of 230 days requires a vehicle total dry weight as shown in the figure. At a radiator exit temperature of 700 K, the total dry vehicle weight increase to reduce the radiator to 1100 square meters for the 2000 K turbine inlet temperature is about 8.7% greater than the minimum weight case. At 575 K radiator exit temperature, the increase in weight for the 1500 K turbine inlet temperature case is about 7.4 %. After a radiator exit of 575 K temperature, the latter vehicle dry weight grows more rapidly than the reduction in radiator size. The 2000 K turbine inlet temperature case continues to decrease radiator size more rapidly than vehicle dry weight growth up to 750 K radiator exit temperature.

20 MWe System for a 100–30–100 = 230 Days Mars Mission

1500 K ←		Turbine Inlet Temperature	→ **2000 K**	
Lowest Weight	Selected		Lowest Weight	Selected
54,200	62,700	Power Generating	39,500	48,800
44,350	45,500	Balance Propulsion System	43,500	44,300
32,585	32,585	Crew (4) and Life Support	32,585	32,585
131,135	**140,785**	**Total Vehicle Dry Weight (kg)**	**115,585**	**125,685**
4,600	**2,960**	**Radiating Area/2 (m^2)**	**2,560**	**1,100**



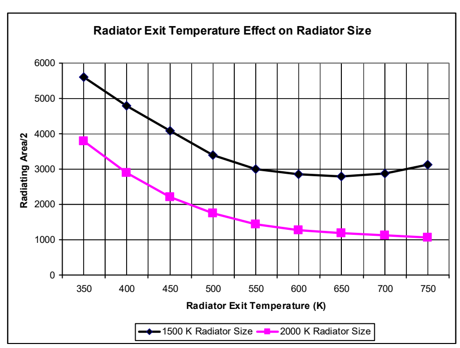


Figure 3. Reduced Radiator Size Effect on Mars Mission Vehicle Total Weight

The effect of the radiator size on the gross weight of the vehicle (including propellants) is illustrated in Figure 4. The weight increase to the vehicle gross weight (for the 1500 K turbine inlet temperature case) is only about 5% to reduce the radiator size from 4600 to 2960 square meters. The weight increase to the vehicle gross weight (for the 2000 K turbine inlet temperature case) is less than 4% to reduce the radiator size from 2560 to 1100 square meters.

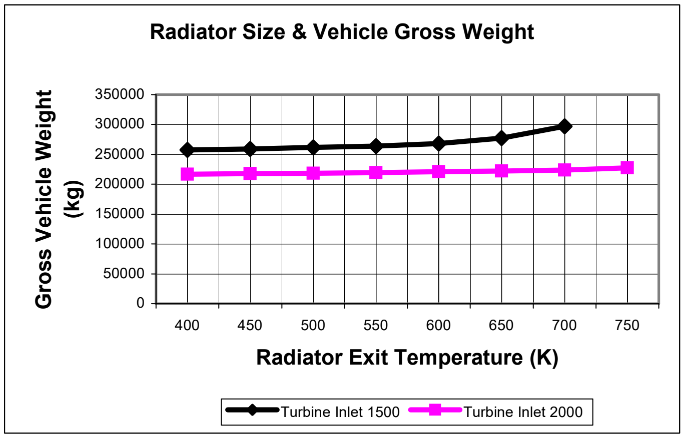


Figure 4 Radiator Size Effect on Vehicle Gross Weight

Another characteristic of the total vehicle system is the nozzle efficiency of the magneto-plasma thruster. Since the thrust and specific impulse are derived from the electric power level, the higher efficiency will allow a greater specific impulse at each thrust level. Figure 4 illustrates the weight effect of nozzle efficiency on the fully fueled vehicle system for our example Mars mission departing from Lunar altitude and capturing Mars at 6000 kilometer orbit. We assumed a nozzle efficiency of 60% to achieve the capabilities described in the Introduction and achieve the 100-30-100=230 Days timeline. The same vehicle with a little more fuel can also fly to Mars in 75 days

Example Thruster	Efficiency %
Smart 1 (Hall Thruster)	45.6
NSTAR Ion Engine	64.7
NEXT Ion Engine	89.7
Vasimr VF- 200	61.3

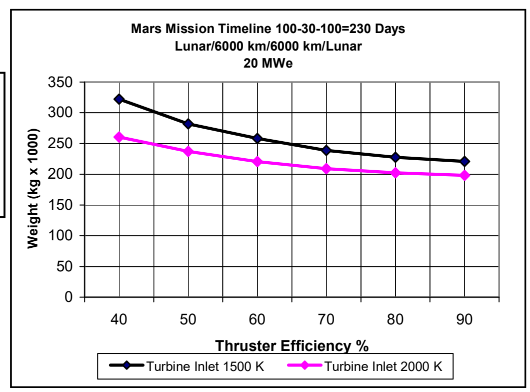


Figure 5. Rocket Nozzle Efficiency Effect on Vehicle System Total Weight

and return in 125 days.

The ion engine nozzle efficiency trend indicates a possible similar growth for the magneto-plasma system. These efficiencies were derived from an Ad Astra Rocket Company web page. The web page also reported that a November 2010 test of the VX-200 engine produced an efficiency of 72%. As the nozzle efficiency increases one could either shorten the mission travel time or select a smaller electric power system. For example, with a nozzle efficiency of 90%, a 15 MWe system can perform as well as the 20 MWe system at a lower weight and smaller radiator. The 15 MWe system, with a 2000 K inlet temperature, totally fueled weight would be about 170,000 kg (verses 198,000 kg for 20 MWe system) and a radiator size of 845 square meters (verses 1100 square meters). However, the 20 MWe system may be a better choice knowing1.) That a 25% loss in power will still be enough for completing the mission within the travel time allowance and 2.) Travel time to Mars could be reduced even below 75 days.

IV. Observations

The basic technology required to define a nuclear electric propulsion system is at a stage that could be developed in time to provide the capability for human flights to Mars and Asteroids in 20 to 25 years. Radiator size would not be a problem. The information presented herein defines 3 methods to reduce the radiator area to a size that would be easily manageable to assemble in low Earth orbit. Each will reduce the radiator size for an electric power baseline system; the amount of reduction depends upon turbine inlet temperature.

1.0 For the 1500 K turbine case, the design of a nuclear electric vehicle with about an 5% off optimum increase in dry weight will allow a 37% reduction (over an optimum system design) in the size of a two sided radiating heat pipe radiator.

2.0 Additional increases of the turbine inlet temperature will continue to decrease the power system weight and radiator size. An increase to 2000 K will reduce the vehicle-fueled weight by 18% and reduce the radiator size an additional 44%.

3.0 The Magneto-plasma thruster efficiency projection from 60% to 90% seems possible. This approach will allow usage of a smaller megawatt system to perform the same basic Mars mission as described herein for less total weight and an additional radiator size reduction.

4.0 A combination of these methods could reduce the radiator size up to 81%

V. Recommendations

1.) Include nuclear electric propulsion as a candidate for human missions to Mars and asteroids and automated missions to other planets and beyond.

2.) Conduct a comprehensive Architecture study that includes human and automated Earth orbit, Lunar and Mars and Asteroid requirements and candidate propulsion systems.

3.) Our country should begin a program to determine the requirements to bring the following to a system development Preliminary Design Review (PDR) status.
   a. Adapting a commercial gas cooled, graphite moderated, Pebble based reactor for space flight.
   b. Pursue the next step for the NASA Glen Research Center heat pipe radiator section to design and manufacture reasonable size radiator.
   c. Pursue ideas to improve turbine inlet temperatures.
   d. Follow and support the development of the magneto-plasma rocket

VI. Acknowledgements

William C Strobl, author, would like to thank Dr. Albert Juhasz, NASA Glenn Research Center for the support of element characteristics of a Brayton Cycle electric generating system. Also, I would like to thank both of my co-authors, Richard Hora and Jim Mildice, for recognizing and assisting me to emphasize the importance of the issues described herein and how to bring them up to the surface.

References

1. Strobl, W. C., Hora, R., Mildice, J. “Propulsion Technology and Human Space Exploration”, AIAA Space 2010, Anaheim Convention Center, Anaheim, California.
2. Chang Diaz, F. R., Glover, T. W., Cassady, L. D., Ilin, A.V.,“Vasimr Human Missions to Mars”, Space, Propulsion & Energy Sciences International Forum, March 15 – 17, 2011, University of Maryland, College Park, MD.
3. Ad Astra Rocket Company Web Page
4 Ehricke, K. A. et al, “A Study of Early Manned Interplanetary Missions – *Final Summary Report,”(AOK63-0001)* Contract NAS 8-5026, General Dynamics/Astronautics, San Diego, CA., 1963
5. Ehricke, K. A. et al, “A Study of Manned Interplanetary Missions – *Condensed Summary Report,”(AOK64-0006)* Contract NAS 8-5026 (follow-on), General Dynamics/Astronautics, San Diego, CA., 1964
6. Ehricke, K. A. et al, “A Study of Planetary Transportation System Model – *Final Report,”(AOK65-0001-3)* Contract NAS 8-5026, General Dynamics/Astronautics, San Diego, CA., 1963
7. Ehricke, K. A. et al, “A Study of Planetary Transportation System Model – *Final Report,”(AOK65-0001-3)* Contract NAS 8-5026, General Dynamics/Astronautics, San Diego, CA., 1963
8. Barter, N. J., editor, “*Space Data – Fifth Edition – Second Printing*” Northrop Grumman Space Technology, March 2003
9. Cordell, B. M. “The Space Transportation Initiative: Plans and Prospects”, *SAWE Paper NO. 2028*, (1991).
10. Chang Diaz, F. R. et al, “*The Physics for Lightweight and Engineering of the VASIMR Engine*”, AIAA 2000-3756, 36th AIAA/ASME/SAE/ASEE, Joint Propulsion Conference, Huntsville, Alabama
11. Juhasz, A. J., EL-Genk, M. S., Harper W. B.,“ *Closed Brayton Cycle Power System with High Temperature Pellet Bed Reactor Heat Source for NEP Applications*”, NASA Technical Memorandum 105933,1993.

12. Juhasz, A. J., "*Design Considerations for Lightweight Space Radiators Based on Fabrication and Test Experience With a Carbon-Carbon Composite Prototype Heat Pipe",* NASA /TP-1998-207427/rev1, 2002.

13. Juhasz, A. J., "*Heat Transfer Analysis of a Closed Brayton Cycle Space Radiator",* NASA/TM-2007-21503, 2007

14. Juhasz, A. J., *"Program Computes Outer-Space Heat-Sink Temperatures",* NASA Tech Briefs LEW-16852

15. Mason, L. S., *"A Comparison of Brayton and Sterling Space Nuclear Systems for Power Levels from 1 Kilowatt to 10 Megawatts",* NASA TM –2001-210593, January 2001

16. Juhasz, A. J., El-Genk M. S., Morley N.J., *"Pellet Bed Reactor Concept for Nuclear Electric Propulsion",* American Institute of Physics, 1993.

17. Mildice, J. W., *"Variable Speed Universal Machine"*, Exploration Systems Enterprise Request for Information, Topic: Power and Propulsion, May 2004.

18. Strobl, W. C., Nottke N., "*High Leverage Technologies Impacting Heavylift Vehicle Requirements for Manned Mars Missions"*, AIAA published paper, 1989, General Dynamics, Space Systems Division, San Diego, CA.

19. Strobl, W. C., "*The New Launch System (NLS)-the Low Cost Approach to Satisfy Space Users Needs"*, AIAA published paper, 1989, General Dynamics, Space Systems Division, San Diego, C A.