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"Positron Propelled and Powered Space Transport Vehicle for Planetary Missions"

4001 Office Court Dr. Suite 303
Santa Fe, NM 87507

Gerald A. Smith
Principal Investigator
(505) 438-2654
(413) 254-1142 (fax)
gerry@pr-llc.com

Distribution List:

Dr. Robert Cassanova
Director, NIAC
bcass@niac.usra.edu

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ABSTRACT

The energy density of antimatter (positrons) is the largest known to man. It is ten orders of magnitude greater than chemical and three orders of magnitude greater than nuclear fission or fusion energy. With separate support from the AFRL, Positronics Research LLC (PRLLC) is developing traps for long-term and high-density storage of positrons. Annihilation of a positron and an electron results in the creation of two 511 keV gamma rays. Unlike nuclear fission, nuclear fusion, or antiproton systems, no residual radioactivity is created, and disposition of positron fuel in the case of an accident can be done predictably and safely. Propulsion systems making use of positrons can improve engine performance and make them an attractive substitute for chemical systems toward manned exploration of Mars and other planetary systems. A system using direct gamma ray products of annihilation for thrust is not investigated, as there are no current means of deflecting gamma ray into preferential flow. PRLLC has investigated three hybrid positron propulsion systems. The first is a solid-core concept, in which positrons heat a hydrogen working fluid through an attenuating solid such as tungsten. The second is a gas-core concept, where gamma rays directly heat propellant through a one- or two-fluid process. Results from the gas-core concept prompted investigation of a third system. This is a concept where solid propellant ablates off a surface bombarded by pulses of photons emanating from positron annihilation. Systems and performances of the three systems are discussed in detail in this report.

1. MARS MANNED PLANETARY MISSION STUDIES

PRLLC has investigated engine parameters for manned Mars missions as a baseline for positron engine development. One of the boldest challenges for chemical propulsion is a manned mission to Mars. Onboard propellant requires an overall interplanetary system mass that prohibits use of any type of existing launch vehicle, including the Saturn V. A new launch system is massive and may be cost-prohibitive. The need to protect astronauts' health from radiation hazards in space inhibits use of low-impulse interplanetary trajectories to reduce propellant mass. Missions must be established which can transport astronauts to Mars within 180 days.

The demand of a positron-based engine to get from LEO to Mars is based in principle on two parameters: the mass of spacecraft after burnout, and the delta-V provided by orbital mechanics. Efforts to minimize burnout mass for a positron-based rocket interplanetary spacecraft prompted examination of previously designed systems. The NASA Mars Exploration Study Team studied such systems during the years 1997 to 1998.[1,2,3]

Some of the conclusions reached by NASA and largely embraced for this current study include:

- To make the Mars mission economically feasible, multiple payloads should be launched to Mars instead of a single "all-in-one" vehicle containing all items for a rendezvous with Mars and return. This reduces payload masses for each launch within tolerance of existing chemical propulsion systems.
- A solid core nuclear thermal rocket (NTR) was studied. Specifically, the study adopted existing NERVA rockets with I_{sp} = 900 second, and a core temperature near 2800ºC. The 1993 study examined 15 klbf and 20 klbf rockets.[3]
- Each launch had a payload consisting of said NTR with its own Mars system payload.
- Un-piloted cargo was sent on a low-energy ("C3") Hohmann-type transfer to Mars, which is generally the slowest means of reaching Mars given appropriate launch information.

- It was proposed that the Mars Excursion Vehicle (MEV) be sent on a “fast-transit” to Mars. The team had evaluated factors such as human bone and tissue degeneration in weightlessness and cosmic radiation, and estimated that a 180 to 230 day mission was adequate so as not to require artificial gravity on the spacecraft.
- The Earth Return Vehicle (ERV) sits in Mars orbit at 250-km periapsis and waits until humans have docked from Mars using the Zubrin-devised LOX/methane propulsion system. The ERV uses a chemical propulsion system to return home, presumably because the ERV may pose environmental concerns if it enters Earth orbit with an active fission-based propulsion plant.
- Minimization of ΔV to Mars is performed by launching s/c during estimated planetary conjunctions (which occur every 778 days) and by an aero-braking procedure at Mars.
- The aero-brake procedure occurs using the chemical propulsion system of the cargo vessel or the lander. The payload is jettisoned from the NTR system (called the Trans-Mars Insertion system (TMI)) sometime during the transit to Mars.
- To reduce probability of impact with Earth, an additional ΔV is given to the TMI stage after the payload has separated.

The conclusions from the study suggest that a positron-based engine must meet or exceed performance levels of predicted NERVA nuclear-thermal engines. However, positrons do not share the environmental issues of nuclear reactors. Each positron annihilates with an electron to create two 511-keV gamma rays. Each gamma ray is below nuclear activation threshold; there is no residual radioactivity associated with positrons. Moreover, once all the necessary positrons are expended for heating propellant, there is no source of radiation remaining in the system.

The benefits of using positrons for a Mars mission include:

1. The “disposable ΔV” used to propel TMI stages into low-probability Earth or Mars intercepts can be eliminated, thereby reducing total propellant mass.
2. The reduction in shielding and engine mass should translate into a lower Initial Mass Low Earth Orbit (IMLEO) for launch vehicles or can allow for faster transit times for piloted missions.
3. The ERV can contain the positron engine instead of LOX/CH_4 if the storage life for positrons can exceed four years, the time in which the ERV waits in orbit for astronauts to dock. This means a significant mass savings or an equivalent reduction in Mars-to-Earth return time for astronauts.
4. The improvement in I_{sp} can translate to either reduced launch payload mass for cargo missions or reduced transit times for piloted missions to Mars (similar to Item #2).
5. Alternatively, more chemical propellant can be stored on the lander to improve aero-braking or landing strategies that often impose hazards on human health or safety.

The launch dates have been augmented from the Mars Ref. Mission study of c.2015 to a more realistic timeframe c.2030. Again, assuming minimum delta-V for Mars opposition-class missions, a few interplanetary scenarios are illustrated in Figure 1. The predicted delta-V necessary for an insertion trajectory into Mars for the manned mission of 2031 (Figure 1b) is about $\Delta V = 3.7$ km/sec. Each trajectory assumed a maximum transit time of 180 days, which can increase for unmanned payload on a lower-energy trajectory.

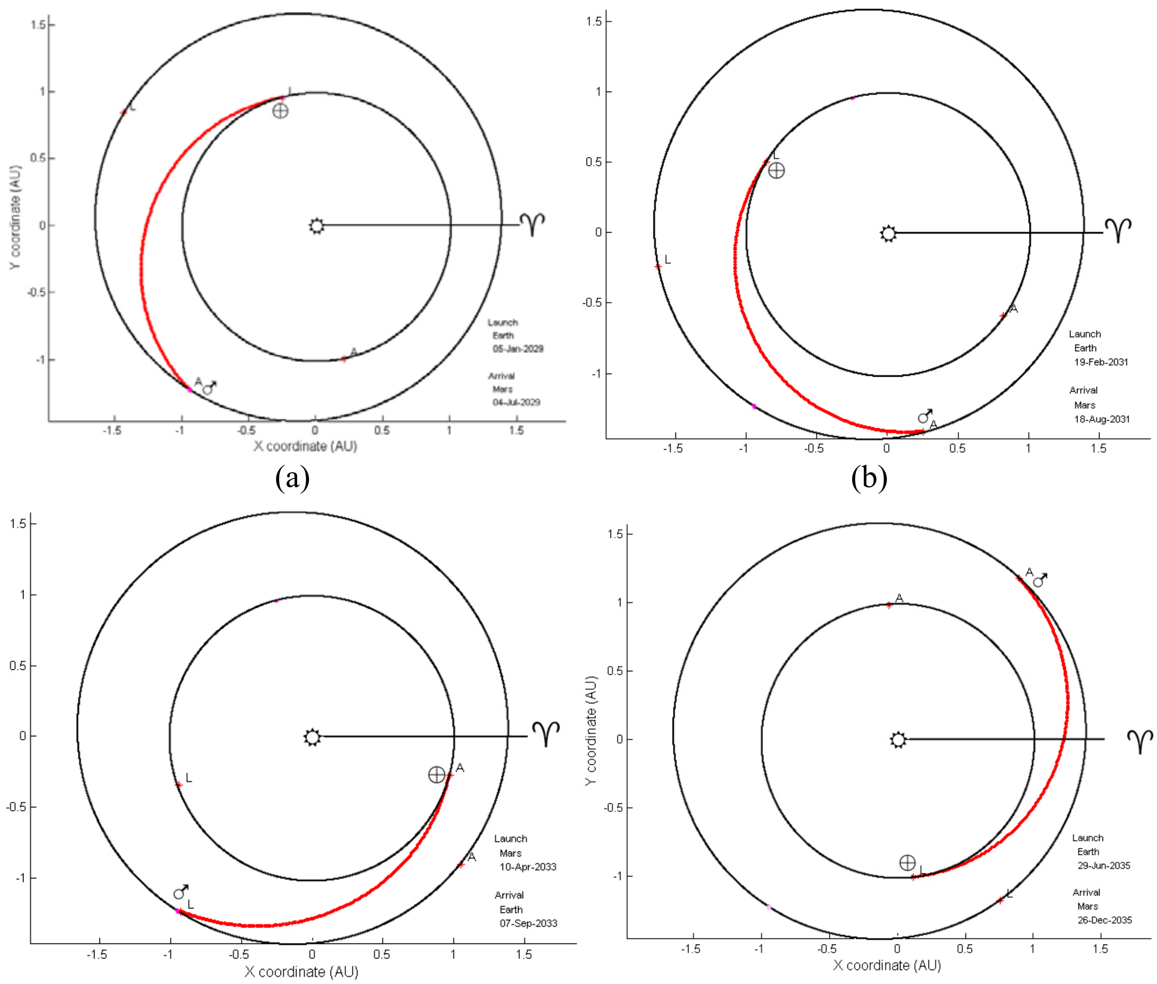
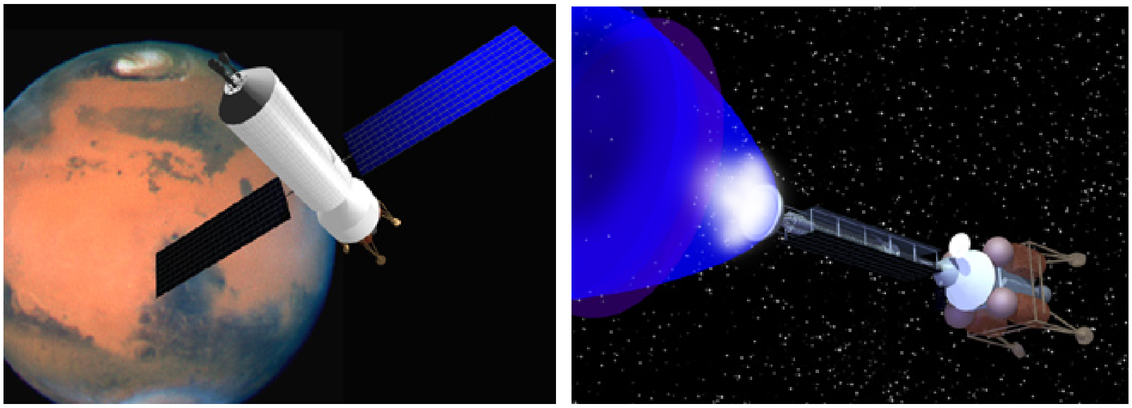


Figure 1. Possible trajectories for Mars missions using the positron rocket. X-coordinates defined in direction of Aries: (a) 2029 pre-lander cargo mission trajectory, of which additional mass savings can incur if on low-energy trajectories; (b) 2031 manned lander to Mars mission; (c) 2033 manned return to Earth (shorter time); (d) 2035 additional lander to Mars, if necessary.

The Mars reference mission suggested payload masses near 60,000 kg for year 2015 missions. This can be reduced assuming technological advances to around 45,000 kg for year 2030 missions. Future sections make use of this range of payload masses for further evaluation of system performances. A complete interplanetary spacecraft mass of 90,000 kg (including the propellant) is predicted.

In summary, the mission scenario for a positron-based spacecraft is similar to existing studies, but using launch vehicles more tolerable in cost. Every 778-day period, two ~45,000 kg payloads are launched from Earth using a Saturn V or equivalent chemical rocket. One payload is the unmanned system or manned crew lander sent to Mars. The other contains the positron propulsion system and the propellant tank. They are assembled as one complete unit in Low Earth Orbit (LEO). One or two unmanned systems are launched in advance of the crewed system in order to ensure that the Martian habitat is well established. The crew arrives at Mars in late

2033, performs research over approximately one-year's duration, then returns home via a smaller positron-based spacecraft using a shorter trajectory. Illustrations of positron-based spaceships are shown in Figure 2.



(a) (b)

Figure 2. Spacecrafts using positron-powered engines: (a) A solid-core system enters Mars orbit; (b) An ablation system burns for landing missions to the outer planets.

PRLLC has also developed architecture for far-term exploration involving positrons in virtually every phase of the mission. These far-term opportunities should be sought after the positron engine has been benchmarked for the first Mars mission and when positrons are in more abundant supply:

- Prior to humans leaving for Mars on initial flights, cargo ships will precede them to Mars on low-energy trajectories to take the components of a Mars Space Station (MSS) and necessary supplies, including a Mars Surface Lander (MSL). The MSS will be similar to an Earth Space Station (ESS). The cargo ships will utilize positron-rocket engines.
- Manned positron-powered SSTO Reusable Space Vehicles (RSV) that are launched from Earth fly to rendezvous in LEO with the ESS. The RSV is a horizontal takeoff, horizontal landing (HTHL) winged-body, manned vehicle where the first stages of flight use air-breathing engines with positrons used for heating the air. It can switch to the same type of rocket engine ultimately used to get to Mars in the final ascent phase.
- Once prepared for interplanetary flights at the ESS, including refueling, the RSV will fly to Mars on a fast, high-energy trajectory, carrying a crew of five to six astronauts, powered by positron-based rocket engines. These spacecraft may additionally use Stirling or Brayton cycle-based positron plants. The RSV then rendezvous with the MSS, and the astronauts descend to the Mars surface on the MSL. The MSL may use a high thrust variant of the positron rocket engine.

2. POSITRON PROPULSION SYSTEMS

A photonic rocket using gamma rays from positron annihilation was devised by the German engineer Eugen Sänger in 1953[4] The gamma rays were reflected off a parabolic mirror in order to impart momentum to the aircraft. The concept remains unrealistic to this date chiefly because a means of deflecting gamma rays by reflection through large angles has not been invented. Other means must be devised to utilize the gamma ray energy. Three concepts that PRLLC has investigated for this report are hybrid systems where the two 511-keV gamma rays generated from positron annihilation heat a working fluid to produce thrust. The first, the solid-core concept, is an indirect means of heating the propellant, but is based largely on proven technology. The remaining two, the gas-core and Sänger ablation concepts, provide a direct means of heating propellant but will require more extensive computational and experimental efforts to validate.

2.1. THE SOLID-CORE CONCEPT

The solid-core concept behaves similarly to the NERVA nuclear-thermal concept.[5] A hot-bleed system is required to make the engine self-sustaining. The cryogenic hydrogen propellant in a hot-bleed system is supplied from a storage tank through a high-pressure pump and routed to cool the regenerative nozzle, the casing of the heat exchanger, and the central positron target tubes. This results in pre-heating the propellant. The pre-heated propellant enters the inlet plenum to the positron heater-attenuator, where it is heated to high temperatures. A small fraction of the hot exit propellant is bled off to drive the turbine that drives the high-pressure feed pump; the majority of the hot propellant is exhausted through a De Laval nozzle to generate thrust. The high-temperature bleed can either be mixed with cold hydrogen to reduce its temperature or directly fed to the turbine. If it is directly fed to the turbine, the turbine must be made of materials that can withstand high temperatures; however, this can result in a reduced turbine mass. The bleed flow is exhausted from a turbine exit nozzle to space after driving the turbine. Figure 3 depicts a positron-powered rocket with a hot-bleed cycle.

As with the NERVA system, the positron solid-core concept is thermally limited by materials in the heating chamber. The crucial difference is that the fission system requires a reactor and complex machinery, whereas the positron system simply relies on positrons or positronium (Ps, bound state of a positron and electron) atoms injected upstream from a storage unit shown in Figure 3. This has two advantages: a reduction in the engine mass for a given thrust (excluding positron trap mass), and greater choice in materials to be used in the heating chamber to reach the highest performance possible.

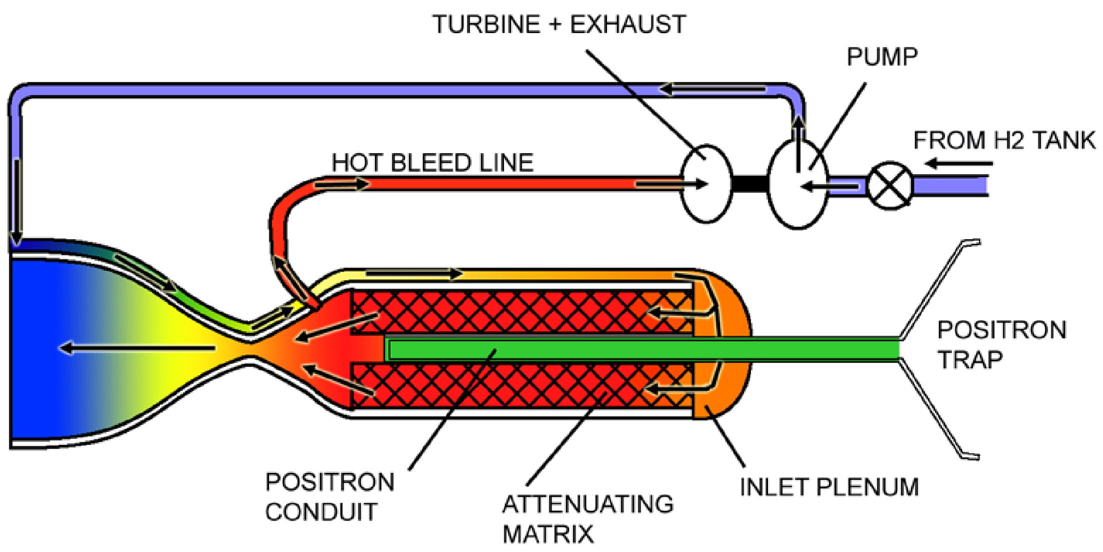


Figure 3. The solid-core positron rocket engine, with the hot-bleed configuration.

A thermal-fluids analysis was conducted to predict the performance of the positron solid-core system and compare it to NERVA/Rover systems. The results suggest that a specific impulse of 920 seconds is attainable with chamber temperatures at 3000 K, with further improvement is expected once materials with higher melting temperatures have been identified to permit an optimal flat-power profile throughout the attenuation region. The corresponding thrust and power emulate the fission systems. Mars burn times, on the order of 30 minutes, suggest that a spacecraft employing three 72 kN solid-core engines would require a total of 6-9 mg of positrons per mission, assuming 100% energy conversion efficiency.

Research was further conducted on positron utilization in a closed-loop power system. A Brayton cycle system was investigated with output power of 100 kW, consistent with Mars Ref. Mission specifications.[1,2] Results showed efficiencies of 25-30%, and positron consumption of about 7 μg/hr. This power system may therefore have practical implications in far-term commitments of faster transits to Mars, where positron consumption in the plant does not dominate over consumption in the rocket.

Specific advantages and disadvantages of the two systems are summarized in Table 1.

Table 1. Comparison of Space Propulsion and Power Systems – Solid Core.

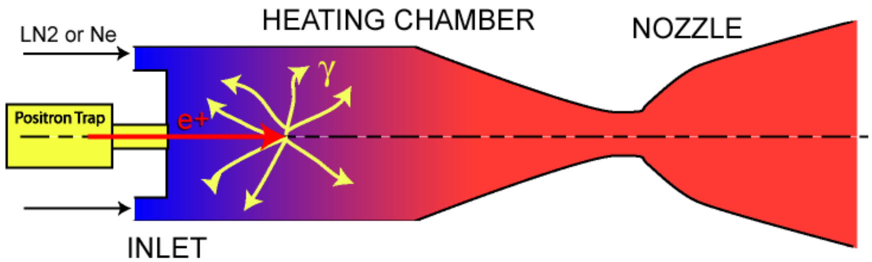
	Fission-Based	Positron Powered
Technology	• Demonstrated, NERVA / Rover • Never flight tested	• Conceptual • Must demonstrate positron storage and controlled injection • Projected near-term technology demonstration for positron storage
Performance	• $I_{sp} \approx$ 950 sec • Thrust ≈ 72 to 1123 kN • Power ≈ 367 to 5320 MW (matched to thrust) • Lifetime ≈ 2 hours total operation	• $I_{sp} \approx$ similar to fission-based systems • Thrust – variable & similar to other systems • Power –matched to thrust • Lifetime – set by material considerations, should exceed fission-based systems
Operation	• Design is dictated by neutronic criticality, fuel burn up and fission product poisoning considerations • High neutron & gamma radiation source during operation • Requires active, accurate & massive control • Requires shutdown cooling (waste of propellant) to decay heat • Radiation source after shutdown	• Flexibility in design since there are no criticality, burn up or poison accumulation issues • Simple design based only upon heat transfer and gamma attenuation issues • Does not require shutdown cooling • Simple on-off control, power controlled by rate of positron utilization • Not a radiation source

	due to fission products & activation	
Materials	• Material choices dictated by neutronic criticality considerations • Propellant – Direct-heated H_2 • Working fluids for power systems – inert gas • Uranium fuel dispersed in graphite media – chosen for neutronic considerations • Requires complex fuel form for corrosion considerations	• Flexible material choices dictated only by temperature and hydrogen corrosion considerations. • Propellant – Direct-heated H_2 • Working fluids for power systems – inert gas
Payload Integration	• Requires massive shield from reactor • Requires separation from reactor • Complex design considerations due to neutron scattering	• No shield required • Propulsion and power sources can be integrated into vehicle
Post Operation	• Not able to return to earth or inhabited surface or station • Not reusable or refuelable	• Able to return to Earth or inhabited surface or station. • Reusable and refuelable

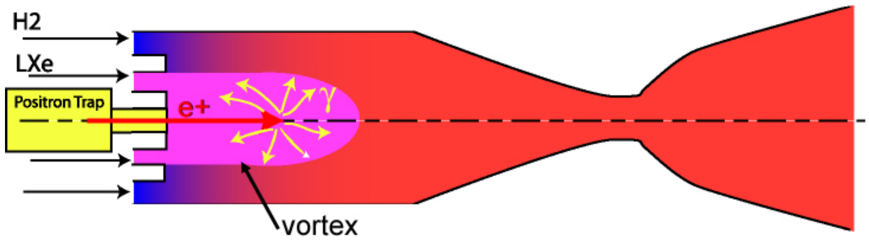
2.2. THE GAS-CORE CONCEPT

Research in the positron gas-core concept follows on the heals of open-cycle nuclear gas-core concepts[6,7]. The gas-core concept differs primarily from the solid-core concept in that gamma rays from each positron annihilation directly heat a fluid under pressure. The primary constraint of the solid-core approach is melting temperatures of the solid matrix required to attenuate gamma rays before heating the hydrogen propellant. By direct heating, attenuation media are removed, the chamber temperature can increase significantly, and the outstanding materials issue simply pertains to wall temperatures.

Nuclear gas-core systems may never reach flying status because they release fission fragments through the exhaust. Positrons again offer the advantages of having no residual radioactivity, as well as the need to only shield un-attenuated 511-keV gamma rays during burn periods. The 511-keV gamma rays are largely transparent to the propellant unless higher densities and/or higher molecular weights are enforced.



(a)



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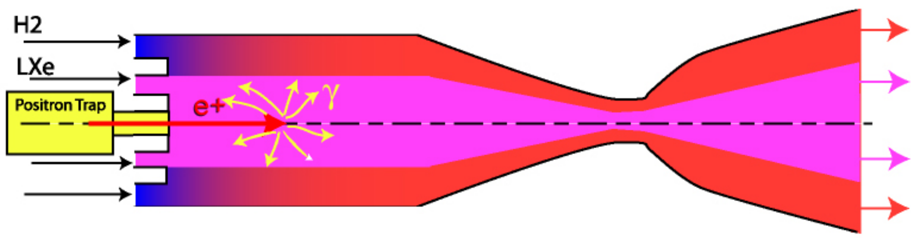

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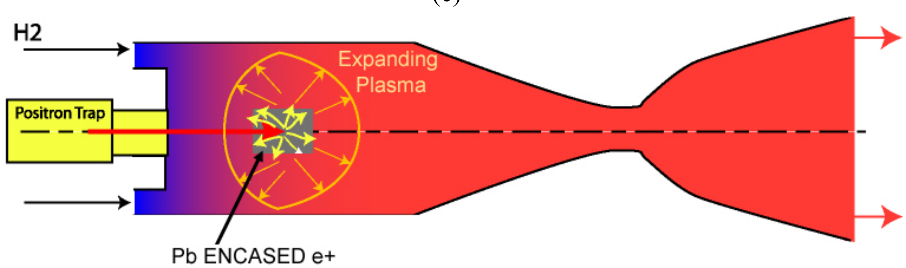

(d)

Figure 4. Concepts studied for the positron gas-core concept. (a) A single-fluid system containing propellant with higher molecular weight such as LN_2; (b) A two-fluid system using injected vortices of high molecular weight xenon to attenuate gamma rays; (c) two-fluid flow through system using H_2 injected at higher mass flow rates than the attenuating xenon; (d) one-fluid cartridge system with solid attenuating material (lead) moving relative to the positrons.

Several versions of the gas-core concept are shown in Figure 4. In Figures 4(a-c), positrons are injected along axis though a miniature high-pressure orifice into a heating chamber. In Figure 4(d) positrons are emitted as cartridges consisting of a positron core surrounded by high molecular weight lead. In sync with the cartridges are pulses of hydrogen gas or liquid. In all cases, the primary fluid must be brought in at high density, at pressures above 100 atm. This means a turbo-pump must be located upstream, and its power source must be obtained from additional machinery such as the aforementioned Brayton cycle system or by mixing a cold fluid from some bled hydrogen into a turbine.

Summing up, the single-fluid concept must rely purely on the sole propellant to cause attenuation. Results from a CFD code and 1-D system code revealed that high-density regions of the fluid moved away from the photon source under powers in excess of 300 MW. The propellant does not absorb the gamma rays, and this, in turn, causes performance to greatly diminish with efficiencies below 10%. The amount of heat required for continuous operation also suggests that a vortex configuration (Figure 4(b)) will break down and cause the engine to behave as in the two-fluid flow through model.

However, both the two-fluid flow through model and the cartridge concept shows promise if the mass flow rate of the hydrogen propellant exceeds that of the xenon or lead by about a factor of five. By operating in a pulsed mode one should be able to mitigate the means of delivering the positrons into the chamber core. The cartridge concept may be less complicated than the flow through concept. Results show that if complete absorption of photons in the lead occurs (approximately 2 cm of lead), the performance of the system reaches that of previously examined systems.[6] Thrusts near 130 kN and I_{sp} = 3200 sec are predicted for a single-engine system. The efficiency is about 85%. Burn times are again on the order of 30 minutes for ΔV = 3.7 km/sec. About 25 mg of positrons are consumed for a ~50,000 kg burnout mass, but the total system mass is reduced to only 56000 kg. The burnout mass can therefore increase to 63,000 kg with a positron mass of 30 mg, or extra propellant can be carried onboard for a greater delta-V. A reduced input power may provide greater positron utilization, as discussed in the ablation section.

The limit of the gas-core concept occurs near the ionization thresholds for hydrogen in the rocket chamber. It is undesirable to consider magnetic nozzles in such concepts with high-density flow. This roughly occurs in the above case, but even a reduction of chamber temperature by a factor of two may be sufficient. Therefore, a limit to the gas-core concept may be $I_{sp} \sim 2500$ sec.

2.3. THE SÄNGER ABLATION CONCEPT

A spacecraft employing this concept is shown in Figure 2(b), and a schematic is shown in Figure 5. Positrons are emitted in "pellets" from several storage banks located behind the engine. The positrons are programmed (e.g. by destabilization of supporting fields) to detonate behind a stiffened pressure plate. The shape of this pressure plate will be optimized in further studies, but for the purposes of this discussion it can be in the form of a parabolic plate. The means of making the Sänger photonic rocket more realistic is to replace/add such solid propellant to this plate. This is an ablative substance. Instead of reflecting gamma rays, the gamma rays deposit sufficient energy local to the surface of the ablation material to jettison high-energy particles with large I_{sp}. A solid generally maintains a high-density profile near the annihilate target. Of course, over series of ablations the solid material may slowly recede from the target. This effect can be mitigated if the positrons annihilate slightly upstream over time. Alternatively, the ablation material can creep slowly toward a fixed target.

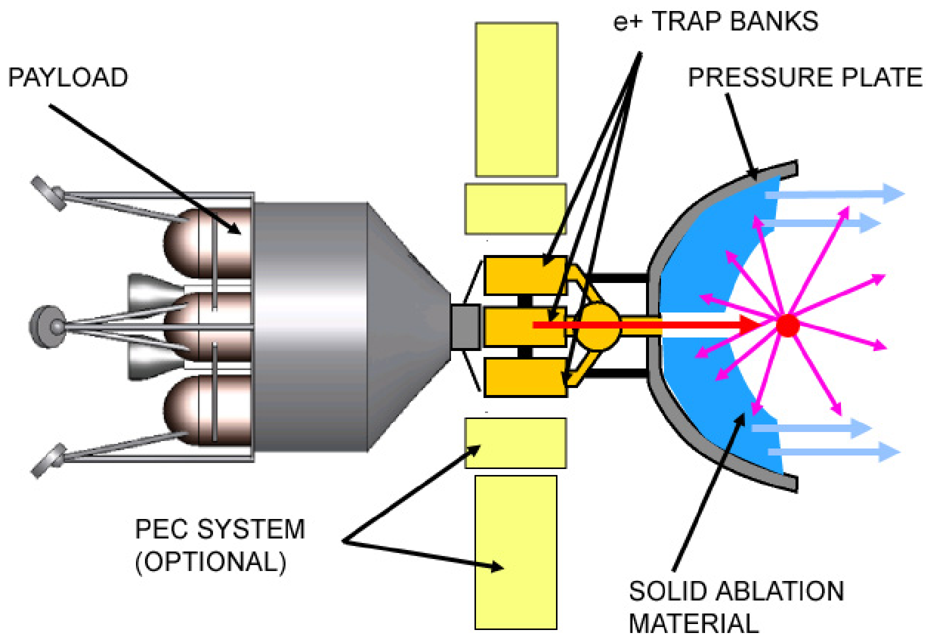

Figure 5. Modified Sänger concept where a photon reflector is replaced by an ablation target. Material is heated with photons by positron annihilation to generate thrust and exit velocity. This engine cannot generate onboard power, and a positron energy conversion (PEC) system is optionally shown.

Unlike the solid-core or gas-core concepts, there are no means to draw off some of the ablated material to provide power. A Brayton-cycle positron energy conversion system can provide power to the pellet mass driver and other components (including the payload). The predicted I_{sp} for this system may result in fast transit times to Mars, warranting a positron power plant. Alternatively, solar collectors or closed-loop nuclear reactors could be employed, depending on the mission scenario.

The gamma rays must be shifted to shorter wavelengths through a high-Z intermediary material before intercepting an ablation material. This same lead can serve as the shell of a Ps or positron pellet. The pellet vaporizes into high-energy plasma, which then propagates to the ablation material. PRLLC staff previously at Penn State University examined wavelength shifting (WLS) using silicon carbide (SiC) ablation material for the Antiproton Catalyzed Microfission/fusion (ACMF) concept.[8,9] There, photon distributions with a mean near 37 keV were shifted to 1 keV energies with 85% efficiency, using a 200 gram lead mass.

An analytical model was used to investigate the performance of the concept (see Appendix I). The performance depends primarily on the mean energy of the shifted photons and the energy per pellet. An energy redistribution of mean 8 keV resulted in higher I_{sp}, with a range of about 1200 sec < I_{sp} < 3000 sec to optimize positron mass expenditures. The total quantity of positrons consumed in this range was 15-40 mg. This is because the ablation concept has a theoretical limit of 50% efficiency since no less than half of the photons are lost to space. However, there is little concern of ionization and wall temperatures limiting this concept.

One of the limiting parameters for the ablation concept is the positron density. One can reach I_{sp} of 5000 sec with a positron mass of 70 mg. Larger pellets to hold additional positrons can be created, although the performances of such have not been fully explored.

2.4. SYSTEM COMPARISON

A side-by-side comparison of three positron propulsion concepts is presented in Table 2, and is meant to summarize results from the preceding sections. The positron mass calculated was for a one-way transit to Mars using a delta-V of 3.7 km/sec. The lower limits of the gas-core concept have not yet been determined; at present, it is practical to issue a lower bound of I_{sp} = 1000 sec in favor of the more proven solid-core concept. However, the efficiencies of all three concepts must be further evaluated to refine their range of operations.

Table 2. Comparisons of the three positron propulsion concepts for Mars space missions.

	Solid-Core	Gas-Core	Sänger Ablation
Isp	650 – 920 sec (< 3000ºK)	1000 - 2500 sec	1200 – 5000 sec, initially predicted
Thrust	72 kN, small class	130 kN (1000 atm)	40 kN to 145 kN (1 Hz pellet rate)
Limits	• Attenuating material melting temperatures. • Wall temperatures .	• Hydrogen ionization temperature. • Nozzle temperatures.	• Positron density per pellet.
e+ mass	• 6-9 mg (100% attenuation eff)	• < 30 mg (85% efficiency)	• 15 – 40 mg (50% efficiency)
Special Notes	• Continuous operation for burn period. • Multiple engines allowed. • Hot-bleed line to draw power possible.	• Pulsed system preferred. • Multiple engines may be allowed pending further study. • Hot-bleed to draw power possible. • Could be throttled for greater thrust using xenon.	• Pulsed system. • Efficiency limited to 50%. • Multiple engines have not been studied. • Cannot provide onboard power.
Research Notes	• Chamber efficiencies have not been determined.	• Lower pressures still acceptable. • Efficiencies at smaller geometries to be determined • Additional research on photon energies from lead needed.	• Additional research on photon energies from lead needed. • Efficiencies from radiation transport to be determined.

3. POSITRON PRODUCTION AND STORAGE

Our NIAC contract does not does include in its tasks the requirement that means of production and long-term storage of positrons be investigated. However, we note that PRLLC has been engaged in such research since 2001, and this work will continue into the foreseeable future with continuing support from the Air Force Research Laboratory (AFRL), Munitions Directorate, Elgin AFB, FL. The relevant contracts are F08630-02-C-0017, F08630-02-C-0018, F08630-03-C-0032 and FA8651-04-C-0140. The POC is Mr. Kenneth Edwards, (850) 883-2707. The AFRL maintains strict control on release of information from the project. We are permitted to state here that the research involves stabilization of Ps atoms in the presence of porous materials and crossed magnetic and electric fields at high vacuum. Work to date at PRLLC is promising, and suggests long-term, high-density storage of Ps atoms may be feasible. We are similarly carrying out research on production of large (mg) quantities of positrons using advanced concepts with intense electron beams.

4. SUMMARY

PRLLC has identified three propulsion systems capable of sending manned payload to Mars within six months. These positron-based engines offer environmental and performance advantages over nuclear-thermal engines, the only class of propulsion currently capable of providing such transportation. The solid-core concept uses technology established from the nuclear-thermal program, and is limited by chamber temperatures. The gas-core and Sänger ablation concepts can reach significantly higher specific impulse, in which the propulsion mass savings should be used in faster transits to Mars or other celestial objects. These concepts rely on redistributing the mean energy of the photons emitted by a positron-electron annihilation to attain high efficiency. The mass of positrons required for Mars transits, on the order of 10 mg, may be realized within 30 years. PRLLC continues to make advances in long-term, high-density confinement of antimatter to bring such systems to existence. Methods for making mg's of positrons are currently under investigation.

5. REFERENCES

1. Hoffman, S.J. and D.I. Kaplan, ed. "Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team," NASA Special Publication 6107, JSC, 1997.
2. B.G. Drake, ed. "Reference Mission Version 3.0: Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team," http://ares.jsc.nasa.gov/ HumanExplore/ Exploration/EXLibrary/docs/MarsRef/addendum/index.htm, June 1998.
3. Borowski, S.K., et al. "Nuclear Thermal Rocket/Vehicle Design Options for Future NASA Missions to the Moon and Mars," AIAA-93-4170 (NASA Tech Memorandum 107071), 1993.
4. E. Sanger, "Zur Theorie der Photonenrakteten", Ing. Arch. 21, 213, 1953.
5. D. R. Koenig, "Experience Gained from the Space Nuclear Rocket Program (Rover)," LA-10062-H, May 1986, Los Alamos National Laboratory, Los Alamos, NM.
6. L.E. Thode, M.C. Cline, and S.D. Howe, "Vortex Formation and Stability in a Scaled Gas-Core Nuclear Rocket Configuration," J. Propulsion and Power, **14**, 4 (1998).
7. A Computational Model for an Open-Cycle Gas Core Nuclear Rocket, D. I. Poston and T. Kammash, Nuclear Science and Engineering **122**, 32 (1996).
8. G.Gaidos, R.A. Lewis, G.A. Smith, et al, "Antiproton-Catalyzed Microfission/fusion Propulsion Systems for Exploration of the Outer Solar System and Beyond," AIAA-98-3589, Presented at the 34th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, 1998.
9. W. Lance Werthman, "Antiproton-Catalyzed Microfission/fusion Space Propulsion," MS Thesis, Dept. of Aerospace Engineering, Penn State University, 1995.

2. Appendices

APPENDIX I

APPENDIX I. SÄNGER ABLATION CONCEPT

I.1. Background

For the gas-core concept, simulations have shown that upstream propagation of high-density fluid transitions significantly reduce the attenuation efficiency of 511-keV gamma rays resulting from positron annihilation. High power inputs tend to exacerbate this effect. The overall performance for the gas-core concept is considered poor unless the gamma rays are pre-attenuated in a secondary fluid or solid that must be expelled with the primary propellant. This can be in the form of liquid xenon or solid lead. Photons must be re-emitted at very low energies in lieu of expected low density H_2 around the source. We have also determined that there is some limitation on the gas-core concept due in part to ionization thresholds for hydrogen. Fluid densities are too large to warrant use of magnetic nozzles, which invoke additional mass and complexity. Wall temperatures have not yet been addressed.

The amount of power delivered suggests behavior similar to a detonation engine. A detonation engine is generally a pulsed system in which an abundance of power is delivered in a local spot that causes intense heating to the propellant closest to the spot and can cause outward pressure expansion. The pressure can impart a force on the spacecraft as the primary thrust mechanism. In this respect, it is important to compare feasible positron concepts with nuclear detonation engines of the past. During the late 1950's and early 60's, many spacecraft concepts took advantage of fission-based nuclear propulsion. One project that lasted from approximately 1958 to 1964 was Orion.[1], the basis of which was ejection and explosion of 'nuclear bombs' in a pulsed detonation (PDE) scheme. Jettisoned at the same time was a disc containing plastic or other similar material with carbon and hydrogen. Carbon and hydrogen have low molecular weight, and the propellant can quickly ionize and escape prior to arrival of additional fission fragments. The plasma formed from the explosion applied a force to a "pusher-plate" mounted on the back of and which accelerated the spacecraft. Specific impulses were expected to be around 3000 seconds.

Orion suffered from many conceptual problems, causing its abandonment in the 60's:

1) Each 'bomb' detonated was fission-based; therefore, a 'dirty' system.
2) Because of mass criticality, the spacecraft had to be designed to handle excess energy per detonation. In other words, the spaceship had to be designed around the engine.

Replacing fissionable material with positrons could provide enormous benefits to the design, which are the following:

1) Gamma rays emitted from positron annihilation are below nuclear activation threshold, leaving no residual radiation.
2) Because positron ejections can be tailored, the engine can be designed around the spacecraft, not the other way around.
3) Certain massive components such as shock absorbers may be eliminated depending on mission scenario. A single ablation core may be possible.
4) The efficiency may increase depending on the arrangement of the ablation material.

In 1953 the German engineer Eugen Sänger developed a simple photonic rocket as a means of propelling aircraft,[2] which remains untested to this date because there no means to deflect gamma rays into unidirectional flow. Positronics Research LLC (PRLLC) has introduced an ablation engine that is a combination of the detonation and photonic rockets, dubbed the Sänger Ablation Concept.

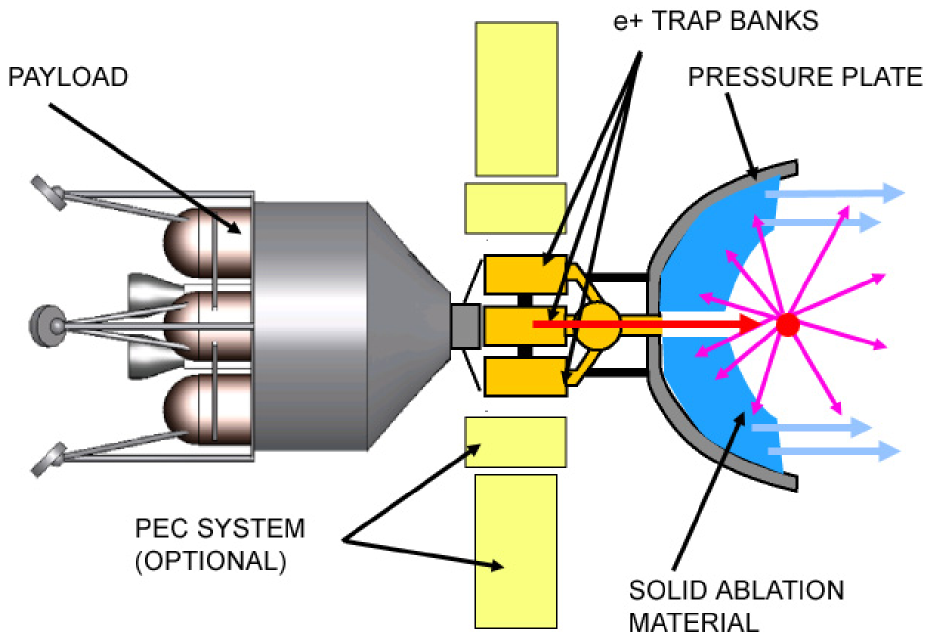

Figure I.1. Modified Sänger concept where a photon reflector is replaced by an ablation target. Material is heated by photons from positron annihilation to generate thrust and exit velocity. This engine cannot generate onboard power, and a positron energy conversion (PEC) system is optionally shown.

I.2. Overall Design

A spacecraft employing this concept is shown in Figure I.1. Positrons are emitted in "pellets" from several storage banks located behind the engine. The positrons are programmed (e.g. by destabilization) to detonate behind a stiffened pressure plate. The shape of this pressure plate will be optimized in further studies, but for the purposes of this discussion it can be in the form of a parabolic mirror. The means of making the Sänger photonic rocket more realistic is to replace/add such solid propellant to this plate. This is an ablative substance. Instead of reflecting gamma rays, the gamma rays deposit sufficient energy local to the surface of the ablation material to jettison high-energy particles with large I_{sp}. A solid generally maintains a high-density profile near the annihilate target. Of course, over series of ablations the solid material may slowly recede from the target. This effect can be mitigated if positrons annihilate slightly upstream over time. Alternatively, the ablation material can creep slowly toward a fixed target.

Unlike the solid-core or gas-core concepts, there are no means to draw off some of the ablated material to provide power. Therefore, Figure I.1 additionally shows a Brayton-cycle positron energy conversion system in order to provide power to the pellet mass driver and other components (including the payload). The predicted I_{sp} for this system may result in fast transit times to Mars, warranting a positron power plant. Alternatively, solar collectors or closed-loop nuclear reactors could be employed, depending on the mission scenario.

From inspection, it appears that the maximum efficiency for this concept is 50%, since half of the emitted photons are eventually lost; however, note that these photons will not convert to heat since there are no components downstream. The design is attractive because the pressure plate can be made to attenuate any unabsorbed photons, which implies little or no risk to crew or payload. Pulsing also ameliorates thermal conditions on the pressure plate.

I.3. Pellet Design

In addition to upstream motion of the high-density interface, the other problem derived for the gas-core concept regards the long attenuation lengths for 511 keV gamma rays. Attenuation lengths for 511 keV gamma rays for even low-Z *solids* are the order of tens of centimeters.[3] This is not desirable, as it implies that the local energy deposition at the surface may be too weak to strip atoms. It is important to optimize the kinetic energy per unit volume in the ablation material so that I_{sp} is > 1000 sec, but thrust values are still reasonable, well above electric propulsion concepts.

The gamma rays can be shifted to lower wavelengths through a high-Z intermediary material before intercepting an ablation material. This same lead can serve as the shell of a positronium (Ps, bound electron-positron) atom or positron pellet. The pellet vaporizes into high-energy plasma, which then propagates to the ablation material. PRLLC ataff previously examined wavelength shifting (WLS) at Penn State University using silicon carbide (SiC) ablation material for the Antiproton Catalyzed Microfission/fusion (ACMF) concept.[4,5]. Photon distributions with a mean near 37 keV were shifted to 1 keV energies with 85% efficiency, using a 200 gram lead mass.

This is in reasonable correlation with the lead absorption efficiency. The 3-D Monte Carlo particle tracking code GEANT[6] was used to examine the total absorption efficiency in a spherical geometry of varying radius. Results are shown in Figure I.2. The 1/e attenuation length is about 0.54 cm, so three to four attenuation lengths sets the mass between 200 and 500 grams, yielding efficiencies from 90 to 95%.

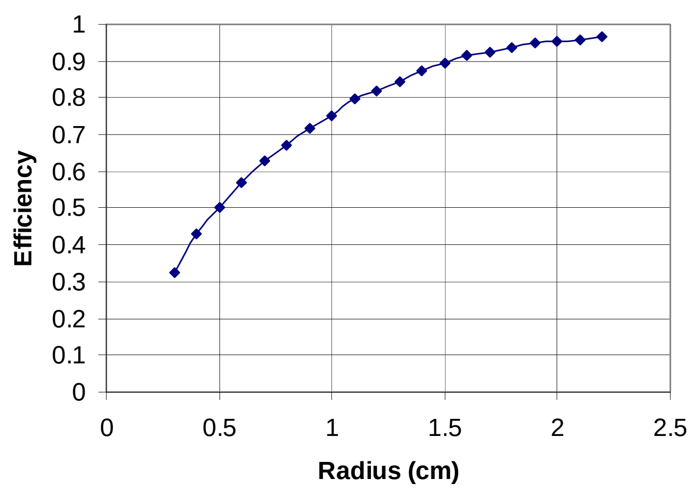

Figure I.2. 511-keV photo-absorption efficiency versus spherical lead radius using GEANT. A value of 1.6 cm correlates to approximately 200 grams.

A geometry illustrating the type of pellet considered for the ablation concept is shown in Figure I.3. It assumes that the positron core is small (perhaps 1 cm diameter) in order to minimize the mass of lead.

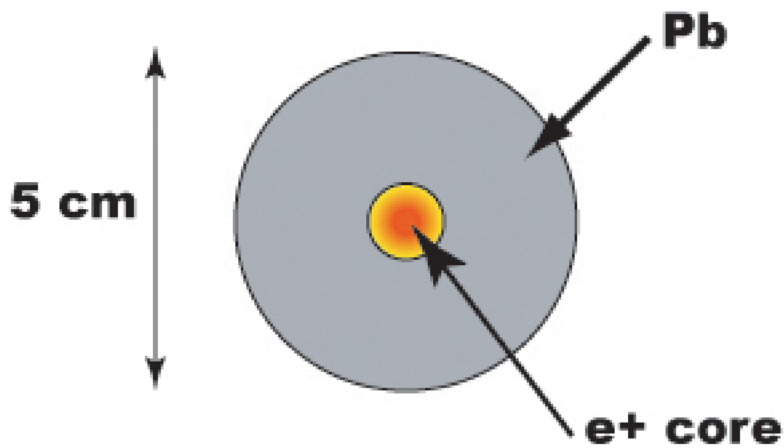

Figure C.3. Illustration of a 500 gram lead pellet for ablation concept.

I.4. Performance

Generally, the first determination is if the positron/lead system can be decoupled into two components: absorption of 511-keV gamma rays in the lead, and hydrodynamics of the lead plasma. This can only be true if the time of absorption is significantly less than the transport process, or $t_{abs} << t_{trans}$. The decoupled system is significantly easier to solve through two discrete codes: a 1-D spherical radiation attenuation and transport code, and an MHD solver. GEANT shows that 511 keV photons are absorbed by the lead in under 50 psec. Penn State studies have suggested evolution of the lead plasma in about 100 nsec.[5] A decoupled system can be therefore be invoked if positrons or Ps atoms achieve annihilation in under 1 nsec.

We assume that about 1 TJ of onboard energy is necessary for a one-way transit to Mars of ~180 days. The energy is distributed over about t_b = 1000 seconds, somewhat consistent with Mars Ref. Mission suggestions for solid-core engines, which was around 30 minutes.[7] An approximation to the amount of ablation energy required per pellet is:

$$E_P = E_{TOT} \cdot \frac{s}{t_b}, \tag{I.1}$$

where s is the cartridge pulse rate. The choice of the pulse rate is largely arbitrary without invoking an MHD computation, but a minimum time between pulses can be evaluated by dividing the exit velocity into a canonical length of the engine's control volume. If this length were to be 10 meters divided by 1000 sec * 9.81 m/sec^2, then $1/s \cong 1$ msec. Conversely, the lower end of s must be around 1 sec^{-1}, since lower rep rates may require significant shock absorbers, and it is important to treat the problem as quasi-steady-state. Here, to facilitate certain computations, we let s = 1 sec^{-1}. Substituting these values into Eq. (I.1), an average energy per pulse on the ablation target becomes **E_P = 1 GJ**.

This is the amount of energy that must be transferred to the ablation material. If one needs to address the potential energy in each pellet E_c, i.e. the stored energy of the positrons, then $E_C = E_P/\eta$, where η is some inefficiency associated with stored energy not making it to the target. The theoretical limit to this efficiency is 50%, because half of all photons are eventually lost. Other efficiencies such as the 90% absorption efficiency and the 85% photon re-emission fraction have been considered, but not applied to the following analysis.

The ablation equations are derived by Ripin et al[8] for laser systems, and are reproduced below. The momentum equation for a rocket of mass M, traveling in a relative coordinate frame as it is in the middle of an ablation period, is:

$$\frac{d}{dt}(Mv_i) = -\frac{dM}{dt}(u - v_i), \tag{I.2}$$

where u is the exit velocity of the ablative material, and v_i is the incremental change in the spacecraft's forward speed. The alternative form is:

$$M\frac{dv_i}{dt} = -u\frac{dM}{dt}. \tag{I.3}$$

One adds a subscript to denote that M becomes a function of ablation interval or M_i, in the event that the specific impulse is low and therefore the amount of onboard ablation propellant is nontrivial. The resulting integral is the well-known rocket equation, or

$$v_i / u = \ln(M_i / (M_i - \Delta M)). \tag{I.4}$$

We can assume the incremental mass change ΔM is small per pulse i, or:

$$v_i / u \approx \ln\left(1 + \frac{\Delta M}{M_i}\right) = \Delta M / M_i. \tag{I.5}$$

Therefore, the total trip ΔV is calculated from

$$\Delta V = \sum_i^N v_i \approx u\sum_i^N \frac{\Delta M}{M_i}, \tag{I.6}$$

where N is the number of pulses. The delta-V is Nv_i, where M_i is approximately constant.

For first-order approximations, the mass of the ablated material ΔM (when mass of lead is << mass of ablation material) found on a 1-D cylindrical slab of radius L is:

$$\Delta M \approx \frac{3\pi L^2}{\mu'} = \langle \dot{m} \rangle s, \tag{I.7}$$

so the average mass flow rate is defined in terms of the rep rate of the system, s. Note that the attenuation coefficient for photons, μ', is finally introduced here. We assume that about 3 attenuation depths approximate the amount of mass that is ablated per event. It is related to the energy of from each ablation E_P as:

$$E_P = \frac{1}{2}u^2 \Delta M. \tag{I.8}$$

Last, the thrust is approximately:

$$T = \langle \dot{m} \rangle (u - v_i). \tag{I.9}$$

Equation (I.5) suggests that the change in spacecraft momentum v_i will play a negligible role in the thrust equation.

Assume that the mass of the lead is 500 g. Information from the gas-core concept suggests that the minimum mass flow ratio of the primary propellant to the lead be 5:1. This implies ΔM = 2.5 kg. Choosing L = 100 cm, Eq. (I.7) results in μ' = 38 cm^2/g. Assume that the ablation material is SiC. At photon energies of 5 keV, the attenuation coefficients for C and Si are 19 and 245 cm^2/g, respectively. At 10 keV, the coefficients are 2.37 and 33.9 cm^2/g. This suggests a mean photon energy near 8 or 9 keV for SiC.

If this wavelength shift is possible, then upon electing E_p = 1 GJ, u = 28300 m/sec (I_{sp} = 2900 sec), and $v_i \sim$ 1.18 m/sec. To obtain a Mars spacecraft of engine and payload mass of 60000 kg delta-V of 3700 m/sec, N = 3100 pellets, and the total propellant mass is 9300 kg (this is near the threshold of assuming M_i = constant = 60000 kg). The thrust for a pulse rate of 1 Hz is about 85 kN over 3100 sec, which may be a practical lower limit for the engine, and the total positron mass may be a minimum of 34 mg.

Reduction of the energy per pellet required computation in order to resolve the change in mass of the spacecraft M_i through Eq. (I.6). A short C++ code was written, and two of the more valuable parameters are shown in Figure I.4. It assumed a payload of 60000 kg and an engine mass of presently 3000 kg. It illustrates that the total positron mass decreases as the energy ablated per pellet decreases. However, the number of shots *N* increases in excess of 10000 below 100 MJ; moreover, the propellant and lead mass (included with the code as 500 g per shot) reach levels in excess of 40000 kg. Below E_P = 150 GJ, the I_{sp} is < 1100 sec. Certain trade studies would have to be performed to compare cost of positrons vs. system mass, but it is clear at lower energies a small reduction in total positrons is insignificant compared to buildup of system mass.

The mass density does not factor into these equations. The only introduced material property is μ'. Figure I.5 shows the effect of reducing μ' to 5 cm^2/g. The energy per ablation E_P must increase to about 2 GJ for reasonable system mass reduction (E_c = 4 GJ), or an I_{sp} of about 1500 sec. The corresponding thrust is 270 kN, and the burn time is reduced down to 1000 seconds. Note the positron mass is still reasonable at 20 mg; however, it is clear from I_{sp} comparison in Figure I.6 that better performance can be gained with lower mean energy hitting the ablation material.

An optimal range of E_c (at 50% ablation efficiency) is $0.4 < E_c < 2$ GJ, above which the e+ mass increases to values which may be undesirable unless the I_{sp} (> 3000 sec) is absolutely necessary to reduce transit times or decrease system mass. Thrusts values center around 50 kN. The burn time is around 5000 sec. Both thrust and burn time can be improved at higher pulse rates. The optimal pulse rate factors heavily on radiation and mass transport and will require use of an MHD code.

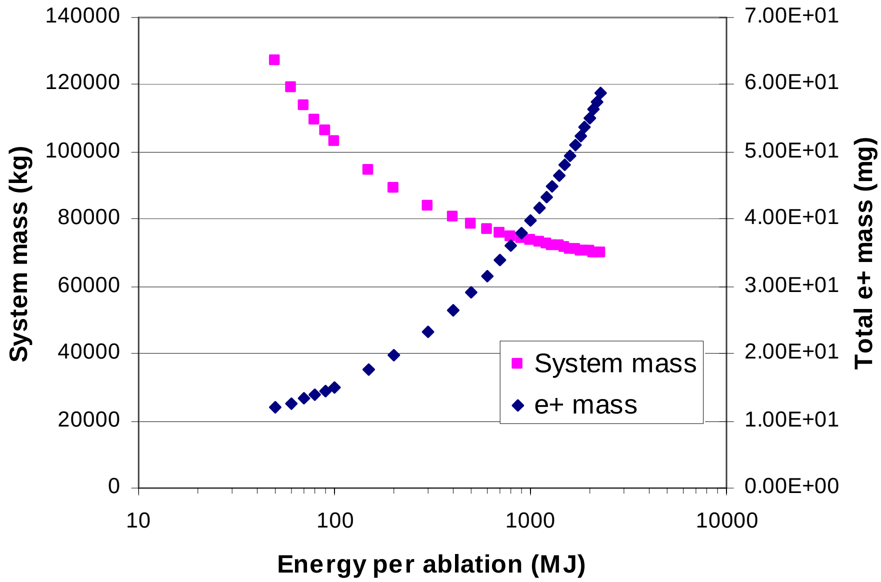

Figure I.4. Comparison of total positron mass and overall initial spacecraft mass as a function of energy ablation per pellet, E_P. The burnout mass of the spacecraft is 63000 kg. The material has a attenuation coefficient of 38 cm^{-1} (SiC at 8 keV mean photon energy, Cu at 20 keV).

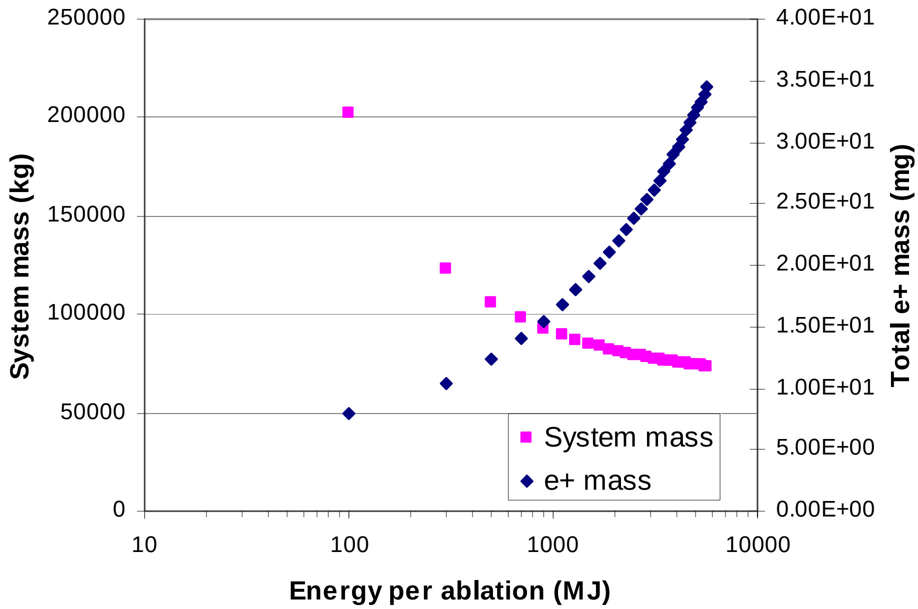

Figure I.5. Comparison of total positron mass and overall initial spacecraft mass as a function of energy ablation per pellet, E_P. The burnout mass of the spacecraft is 63000 kg. The material has a attenuation coefficient of 5 cm^2/g (SiC at 20 keV mean photon energy, Cu at 50 keV).

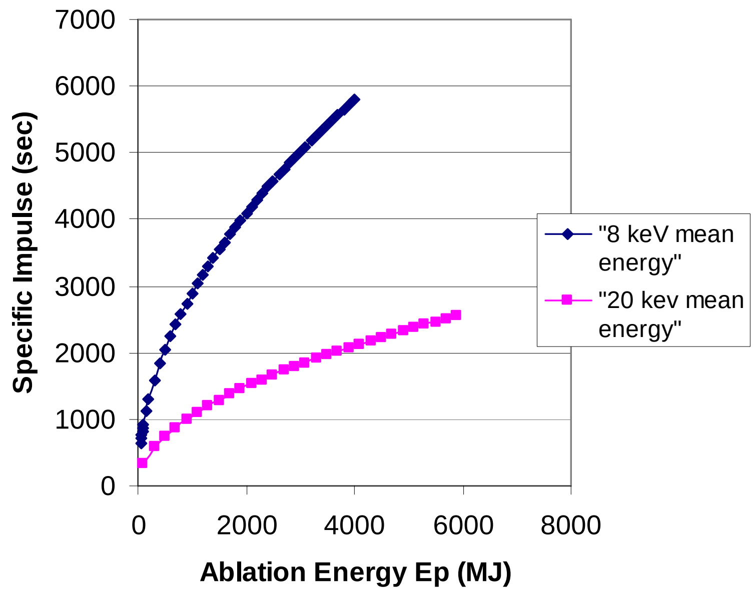

Figure I.6. A comparison of engine I_{sp} for SiC with 8 keV and 20 keV input photon mean energies. This correlates to about 20 keVand 50 keV mean energies for Cu.

The means of achieving a strong μ' is based on 1) the material's effective molecular weight, and 2) the mean photon energy emitted from the lead plasma. A calculation performed in GEANT reveals that the K-edge energy absorption is at approximately 90 keV (Figure I.7). This provides an upper bound at which the lead plasma will re-emit photons. The means of selecting the actual energy is largely based on the thickness of the lead. The studies performed at Penn State suggest that these mean shifted energies may be permissible at the lead thicknesses derived.

Some earlier solid-core system analyses suggest ~5 mg consumed during transit, which was at 100% efficiency. The results are in good agreement with such predictions given the intrinsic 50% efficiency of the ablation concept. Future studies should include a comparison of the inefficiencies and limitations (wall temperatures) of the gas-core cartridge concept versus the solid-angle efficiency of the ablation concept, which should not have significant thermal problems.

I.5. Future Research

There are several parameters that must be refined to produce exact specification. The first is the ablation efficiency η. This is function of losses due to radiation transport and geometry of the system. The ablation material and thickness of lead must also be carefully determined through plasma studies with lead. MHD studies are clearly required to examine mean blow-off rates *u*, along with optimization of pulse rates.

Finally, it must be restated that one or two research topics in the gas-core concept have not been dismissed. Using a lead-based cartridge immersed in a hydrogen gas is still feasible design. There are many parallels between such a design and the ablation concept, particularly towards the optimization of the lead pellet shell.

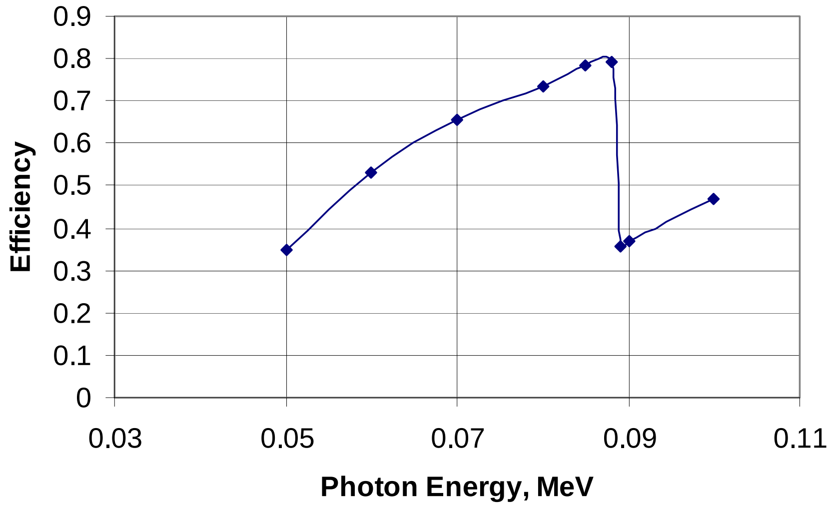

Figure I.7. Transmission efficiency of photons in a 0.13-cm lead sphere by varying input energy of photons. K-shell absorption energy near 89 keV corroborates other findings.[3]

I.6. References

1. M. Flora, "Project Orion: Its Life, Death, and Possible Rebirth," http://www.islandone.org/Propulsion/ProjectOrion.html. Last accessed: 20 Feb 2006.
2. E. Sanger, "Zur Theorie der Phonenenraketen", Ing. Arch. 21, 213, 1953; "Photon Drive," http://spectech.bravepages.com/DSP_Article_PHOTON%20DRIVE.htm. Last accessed: 20 Feb 2006.
3. CRC Handbook of Chemistry and Physics. 83rd edition, 2002.
4. G.Gaidos, R.A. Lewis, G.A. Smith, et al, "Antiproton-Catalyzed Microfission/fusion Propulsion Systems for Exploration of the Outer Solar System and Beyond," AIAA-98-3589, Presented at the 34th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, 1998.
5. W. Lance Wertham, "Antiproton-Catalyzed Microfission/fusion Space Propulsion," MS Thesis, Dept. of Aerospace Engineering, PSU, 1995.
6. "GEANT 4 Home Page," wwwasd.web.cern.ch/wwwasd/**geant4**/**geant4**.html. Last accessed: 1 February 2005.
7. Borowski, S.K., et al. "Nuclear Thermal Rocket/Vehicle Design Options for Future NASA Missions to the Moon and Mars," AIAA-93-4170 (NASA Tech Memorandum 107071), 1993.
8. B.H. Ripin, R. Decoste, et al, "Laser-plasma Interaction and Ablative Acceleration of Thin Foils at 10^{12}-10^{15} W/cm^2," Phys. Fluids 23(5), 1980.