

Manned Spacecraft Propulsion through Direct Conversion of Nuclear Energy

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Abstract—Fission-based power sources have been used in space as early as the Voyager missions. However, the use of nuclear power for manned space exploration has yet to be accomplished. Reactor-based (thermal or electric) propulsion efforts have been stymied by enormous launch costs and extremely long mission times, which endanger the health of the astronauts, and increase the risk of failure due to the need for complex space rendezvous and assembly. All of this is avoided with a compact, high efficiency, single-launch spacecraft such as that projected for the Fission-Powered Pulsed Plasma Propulsion concept (F4P) proposed here. In the F4P scheme, the immense energy of the nuclear reaction is to be harnessed in a more direct and powerful way to propel the spacecraft with the optimal range of jet power (30-100 MW), and at the most efficacious exhaust velocity (30-50 km/sec). It is accomplished in the following manner: (1) Large, thin hoops of Lithium (Li) metal are driven at the proper angle and speed for convergence onto fissile target injected from the F4P thruster throat. (2) The converging Li shells form a thick blanket compressing the fissile material thereby producing a large increase in reactivity and fission energy release. (3) Virtually all of the energy released (plasma thermal, radiant, shell-ohmic, and fission-fragment) is absorbed by the Lithium shell which is subsequently vaporized and ionized. This Li plasma plume expands out against the thruster's divergent magnetic field which acts as a barrier and directs the plasma flow out of the thruster resulting in minimal interaction with the spacecraft. In this manner, both high exhaust velocity and thrust are achieved with only minor interaction with the spacecraft. To understand the issues involved with this approach, as well as design an experimental test, a 1D analytical model was developed and 3D ANSYS Explicit Dynamics® of the liner implosion were carried out. The inductive coupling and drive efficiency of the compression for a single liner was evaluated with the one-dimensional model which included the key circuit, magnetic field and liner parameters. The model was then used to characterize the liner motion as a function of liner mass, resistivity, stored energy, coil voltage and initial magnetic field. The stability and structural behavior of multiple liners was studied with the 3D ANSYS code. Techniques for controlling liner bucking as well as liner rotation for Rayleigh-Taylor stability were examined with these codes. Experimental testing at the relevant scale have validated the metal hoop compression process. The major unknown to be resolved is the functional dependence of the fission energy yield on the several key parameters of the compression process. This must be fully understood and tested to prove the viability of the F4P concept. The current effort is focused on (1) a detailed physics model of the heating and compression of the fissile fuel from a solid to an ionized gas. (2) Calculations to determine the fission amplification and energy yield throughout the compression cycle as a function of fissile mass and density.

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1. INTRODUCTION

Even with spectacular breakthroughs in electronics, materials science, and computer technology in the past 50 years, human exploration of space has not progressed beyond what was basically accomplished in the 1970s despite the substantial financial commitment and passionate interest from virtually every modern nation on the planet. The future of manned space exploration and development of space depends critically on the creation of a dramatically more efficient propulsion system for in-space transportation. Fission-Powered Pulsed Plasma Propulsion (F4P) represents a possible solution to this impasse. Current fission-based nuclear power schemes rely on MW class reactors operating in space. Nuclear thermal employs a light-gas cooled reactor core which limits the specific impulse, $I_{sp} \sim 900$ sec, which is well shy of the 3-5ksec desired for optimal transport. With propellant storage and reactor mass being significant issues, only minor reductions in trip time and in-space mass and assembly are predicted. Nuclear electric can achieve the much higher I_{sp} desired as Electric Plasma (EP) thrusters can be employed. However, the low conversion efficiency of the reactor as well as middling efficiency of high-power EP thrusters creates a massive heat rejection problem. Nuclear fusion is always 30 years away and is likely to stay that way.

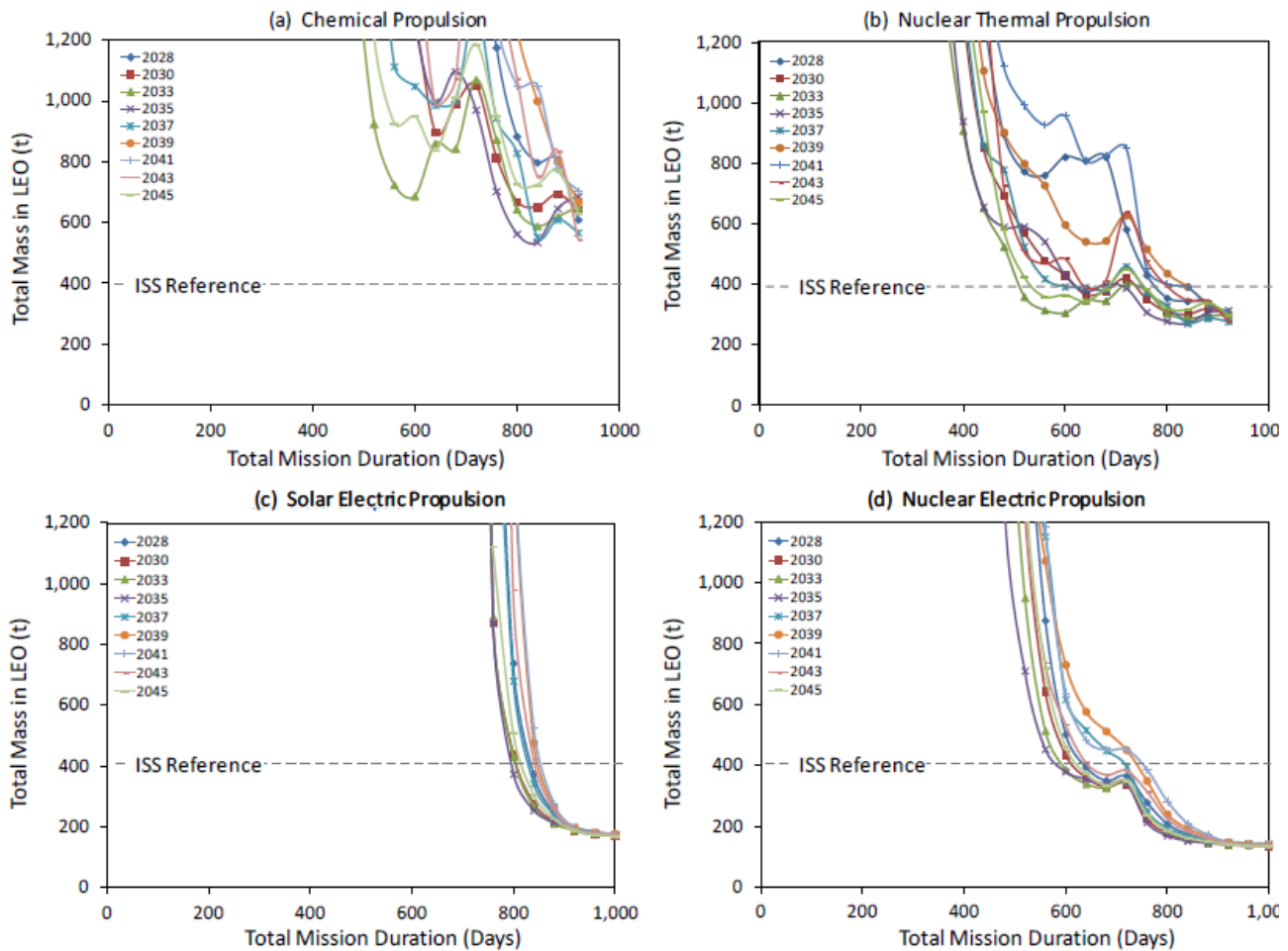


Figure 1. Sensitivity of the estimated crew vehicle mass as a function of round-trip mission duration for the short-stay, 60 days at Mars, opposition-class missions. short-stay, 60 days at Mars, opposition-class missions.

It would no doubt suffer from reliability, complexity and efficiency issues as well.

The total mass for the Mars mission is dependent on many factors, the primary ones being mission duration, length of stay, and type of propulsion employed. These are nicely summed up in Fig. 1, taken from the DRA-5 Addendum #2 [1]. It is evident from this analysis that long duration missions using either thermal or nuclear-electric propulsion can reduce the total mass in LEO to something that is less than the entire space station. It is also evident that these missions will be of long duration and will thus require the most protection from Galactic Cosmic Radiation as this is the greatest threat to the health of astronauts on long duration missions. Other major health hazards of prolonged spaceflight, in addition to the high levels of damaging radiation, include the absence of gravity, long periods of isolation and confinement, a closed and potentially hostile living environment, and the stress associated with being a long distance from Earth [2].

The obvious solution to these issues would be a propulsion system that can dramatically change the characteristic power and Isp that can be delivered for a given propulsion system mass (i.e., high specific power or α). It is abundantly

clear from Fig 1 that it will not be a chemical-based or solar electric propulsion system. A nuclear based system must do better than the current nuclear thermal and nuclear electric systems developed to date. As an analysis tool it is useful to consider what attributes would comprise the ideal nuclear propulsion system. Such a system would likely would satisfy the following criteria:

- (1) *all the nuclear energy is released into the propellant, not requiring conversion to electricity.*
- (2) *The propellant requires no significant tankage mass by being a solid, and where*
- (3) *the propellant is rapidly heated and accelerated to optimal exhaust velocity (> 20 km/s), with*
- (4) *no significant physical interaction with the spacecraft thereby limiting thermal heat load, spacecraft damage, and radiator mass.*

Such a system would drastically reduce the mass needed in LEO as well as operate at the highest α . It is clear that nuclear electric fails on criteria 1 and 2 primarily due to the thermal

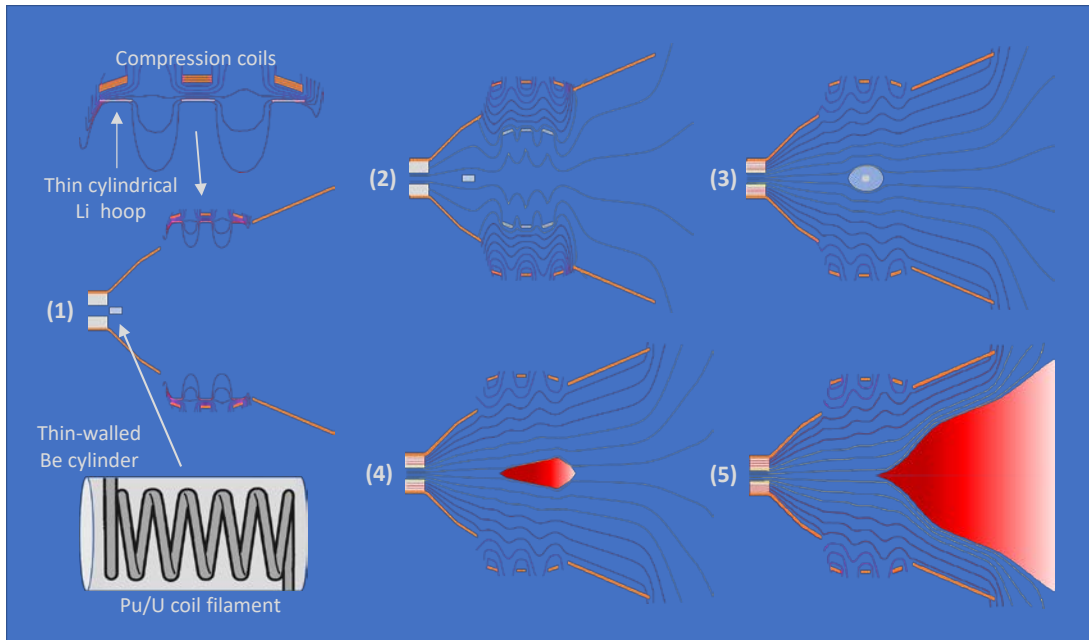


Figure 2. Cross-sectional schematic of the proposed Fission-Powered Pulsed Plasma Propulsion (F4P) thruster showing the key operational steps. The magnetic fields at each stage were calculated using the transient magnetic field solver from an 2D MHD code.

radiator mass penalty due to the reactor inefficiency. Other issues besides the tankage for a gaseous propellant is the lack of thrusters in the megawatt class that don't have significant wall erosion issues (criterion 3). Nuclear thermal fails on 2, 3 and 4. The Isp for even the light gases (H and He) is too low and creates serious tankage mass issues as well. There are also the problems associated with contact of the propellant and the heat exchanger walls (embrittlement and corrosion).

It is believed that, if successful, the F4P concept could actually achieve these four criteria for efficient, timely, and practical space transportation. The paper is thus divided as follows: section 2 presents a description of the F4P concept and the major challenges to its realization. Section 3 and 4 discuss a key element of the propulsion system, the metal liner driver and the compressional heating of the liner propellant from the nuclear fission process. Section 5 examines the spacecraft design and the manned Mars mission requirements assuming successful development of the F4P propulsion system. The summary section will include an outline for the further development of the F4P concept.

2. FISSION -POWERED PULSED PLASMA PROPULSION (F4P)

The Fission Powered Pulsed Plasma Propulsion concept (F4P) seeks to tap the tremendous energy gain from a nuclear fuel such as Uranium or Plutonium directly into the propellant, which would most likely will be Lithium metal. The process that will be used to perform this task has been evaluated through analytical and numerical modelling in both 1D and 3D calculations [3-5]. The results from several experiments as well as confirmatory experimental work at scale provided confirmation as to the soundness of the approach to the physical processes required to bring the both

the propellant and nuclear material to a point where one could expect significant energy gain.

The process for the F4P space-based propulsion system would proceed as follows: Three large-radius Lithium (Li) metal hoops (a.k.a. liners) are sequentially drifted into the chamber along the axis of symmetry until they are positioned directly underneath the fast electromagnetic coils (see Fig. 2 and YouTube video [6]). (1) These hoops (liners), are then rapidly driven radially and axially inward by magnetic compression from highly-coupled, pulsed coils. At the same time, a very small amount (grams) of fissile material, likely either Uranium (^{235}U) or Plutonium (^{239}Pu), positioned inside a Beryllium (Be) cylinder, is injected on-axis into the thruster chamber as depicted in Fig. 2-(1). This Be cylinder acts initially as a sabot, then as a containment vessel, and finally as a fission neutron reflector. The nuclear fuel is in the form of a coil filament and is attached at each end to the Be sabot forming a closed loop electrically. Unlike the illustration in Fig. 2, the fuel form would be more of a pancake coil like a Bitter magnet to better fill the space and increase the fissile mass inside the sabot. As the Li shells rapidly converge on the intersection point with the nuclear fuel, they continually thicken to finally form an impenetrable blanket several centimeters thick (see Fig 2-(2)). The kg mass Li shell now moving with a terminal velocity of ~ 3 km/s rapidly compresses the leakage magnetic flux inside the shell surrounding the sabot which in turn compresses the sabot and nuclear fuel (Fig. 2 (3)). The large radial magnetic gradient across the sabot and fuel induces a huge current flow first in the Be shell and then in the fissile fuel. Pu and U both possess a high resistivity (Pu is $\sim 3\times$ that of Tungsten) so that the fuel rapidly melts and then boils. The gas becomes partially ionized due to the large electric fields associated with the enormous dB/dt now experienced by the fuel. The Be shell

initially confines this ionized nuclear gas as it has a much lower resistivity and higher melting point. The Lithium shell at this point has also boiled and ionized as it has a much lower boiling point and is easily ionized. The picture for the fuel and Li propellant is now one of an ionized gas being compressed by megabar magnetic fields.

It is worth recalling at this point a few key nuclear fission concepts such as critical mass. This is the mass of fissile material at STP required to sustain the fission reaction indefinitely. For masses higher than the critical mass, the chain reaction becomes a runaway, and if the mass is kept confined during this process, it is a bomb. This is of course not quite what we are after here. For simplicity, the critical mass is characterized for a sphere by the product of the fissile fuel density, ρ_{ff} , and the critical radius R_c . For ^{239}Pu , R_c is 5 cm implying a critical mass, M_c , of 10 kg at STP. The idea here is to turn the fissile mass into at least a partially ionized gas and compress it to much a higher density, thereby considerably reducing the fissile mass required to obtain large energy gains. With nuclear reactivity scaling as the density squared, the compression of the nuclear plasma results in a large and rapid increase in the release of nuclear energy. As the fuel is a high Z atom at high density, the ionized gas temperature will be clamped by Bremsstrahlung and line radiation, as well as thermal conduction to the sabot and shell as it is compressed. For estimation purposes it will be assumed that the peak temperature, $T_{Pu} \sim 1 \text{ eV}$ (11,600 °K). The ionized gas pressure is: $P_{Pu} = n_{Pu} \cdot k_B \cdot T_{Pu}$ where k_B is Boltzmann's constant. Setting this pressure to the compressional pressure from a 600 T magnetic field, results in a peak Pu number density, $n_{Pu} = 8.9 \times 10^{23} \text{ cm}^{-3}$. The solid Pu number density at STP, $n_{STP} = \rho_{Pu} \cdot A_n / (239) = (19.85 \text{ g/cm}^3) \cdot 6.02 \times 10^{23} / (239) = 5.0 \times 10^{22} \text{ cm}^{-3}$ where A_n is Avogadro's number. The compressed density is a factor of 11 times greater. The critical radius is thus smaller by the same amount, or less than a centimeter, and the critical mass is

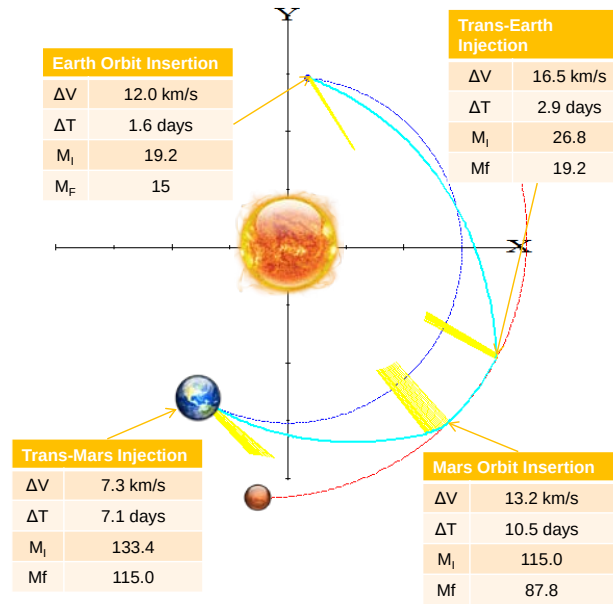


Figure 3. ΔV budget and mass parameters for a 210-day round trip manned Mars mission. The Copernicus software was employed to develop the mission architecture starting around a single 134 t launch to LEO. (See section 5).

reduced by the square of this ratio, i.e., a factor of 125 or just 80 grams.

The potential fission energy generated for even this small amount would be over 6,000 GJ! Clearly significantly less fissile material will suffice. Undoubtedly a detailed gain calculation must be part of the future effort to determine just what the required quantity is. A yield of at least 100 MJ per pulse is desired to provide the target jet power range of 10-30 MW, with a rep rate of several seconds. Assuming just a tiny fractional “burn” of the fissile material it is likely that the required fissile fuel can be reduced by a factor of 20 or more.

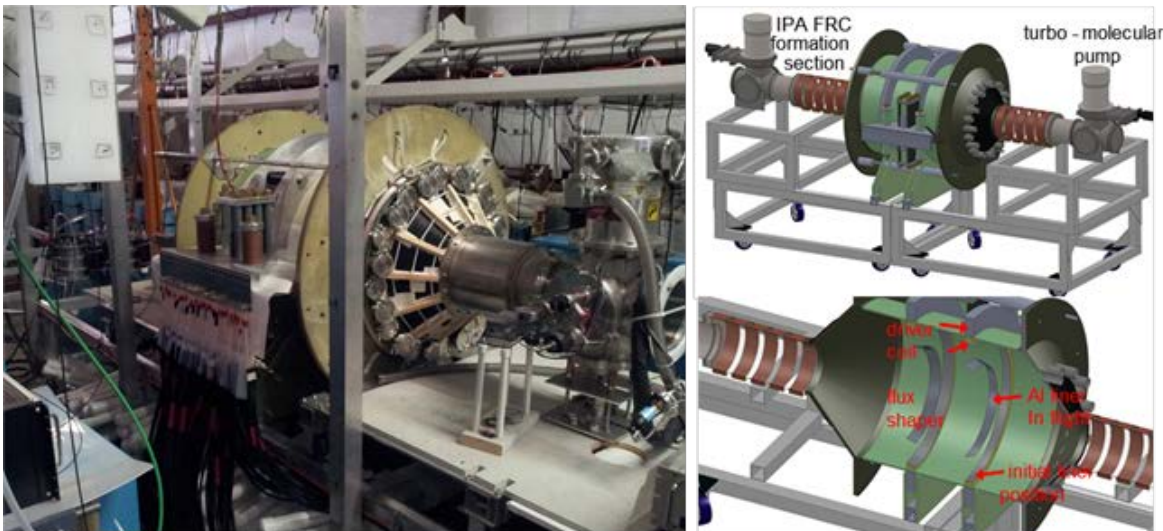


Figure 4. (Left) Picture of the Inductively Driven Liner (IDL) experiment at MSNW. (Right) CAD drawings of IDL showing the inner structures of device (Aluminum liners) and the external driver coils.

At 4 grams and a fractional burn of 0.00032 would yield 100 MJ of fission energy.

The propulsive energy required for a 90-day transit to Mars depicted in Fig. 3 would require roughly 10^7 MJ, or 10^5 pulses @ 100 MJ/pulse. At 4 g per pulse 400 kg of fissile material would be required. This is to be compared with RTGs that have flown in space where the nuclear fuel ranged from 4.8 to 32.7 kg generating at most a few hundred Watts of electrical power.

Virtually all of the energy released (plasma thermal, radiant, shell-ohmic, and fission-fragment) is absorbed by the Lithium shell which is subsequently vaporized and ionized. (4) It is this Li plasma that acts as the propellant in this scheme as its bulk mass is orders larger than that of the fission fuel. Lithium is very advantageous as a propellant as it is a malleable solid for easy onboard storage. It has a very low atomic mass, and as an alkali metal, it is easily vaporized and ionized. (5) The large external region filled with a divergent magnetic flux by the thruster coils now acts as a barrier and magnetic nozzle, which protects the spacecraft while expelling the Lithium and nuclear plasma. In this manner, both high exhaust velocity and thrust are achieved with only minor interaction with the spacecraft. The flux compression of the vacuum magnetic field by the expanding plasma plume generates a large back-emf on the thruster coils. This allows for a rapid recharging of the thruster capacitors [7,8]. With

this system, the F4P system could be self-powered requiring no electrical input from the spacecraft solar panels.

3. PHYSICS OF THE INDUCTIVELY DRIVEN METAL LINER IMPLOSION FOR MAGNETIC COMPRESSION

The implosion of magnetically driven thin metal hoops (liners) to achieve extremely high magnetic fields and thus extremely high compressional pressures on metal objects (> 1 Megabar) has a long history. It was first accomplished in the groundbreaking experiments performed by Cnare in the early 1960s [9]. It is also noteworthy that an inductively driven liner similar to those proposed here had been realized in previous experiments by Turchi et al. [10] at NRL in the mid-seventies. Large compression ratio convergence ($\sim 30:1$) was obtained employing a thin (1 mm), relatively large diameter (30 cm), but short (7 cm) aluminum hoop. Peak magnetic fields as large as 200 T were achieved with excellent symmetry and quality employing only a 540 kJ capacitor bank. More critically, it has been performed at the scale required at MSNW as part of a previous NIAC study [5,11] with an even smaller energy storage (see Fig. 4). Key results from these experiments are summarized in Figs. 5 and 6 with comparison to code results (to be discussed in the next section). It can be seen that there is very good agreement with the modelling. In order to carry out these tests, the device was

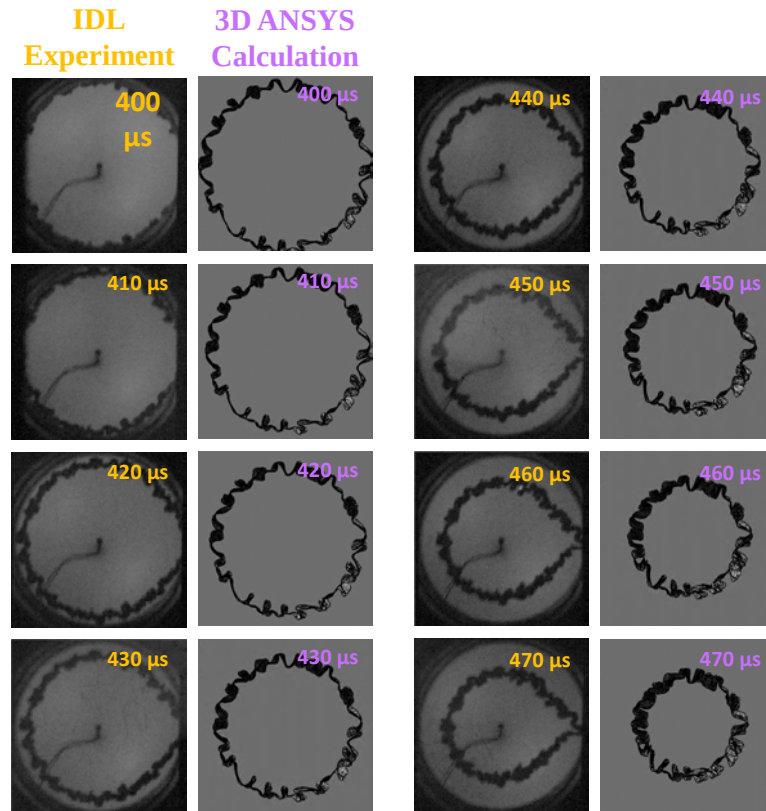


Figure 5. Aluminum Liner Testing at 1/4 power for code validation. Conditions: bank voltage – 20 kV, bank capacitance – 840 uF, Al liner thickness – 0.4 mm, Al initial diameter – 0.8 m

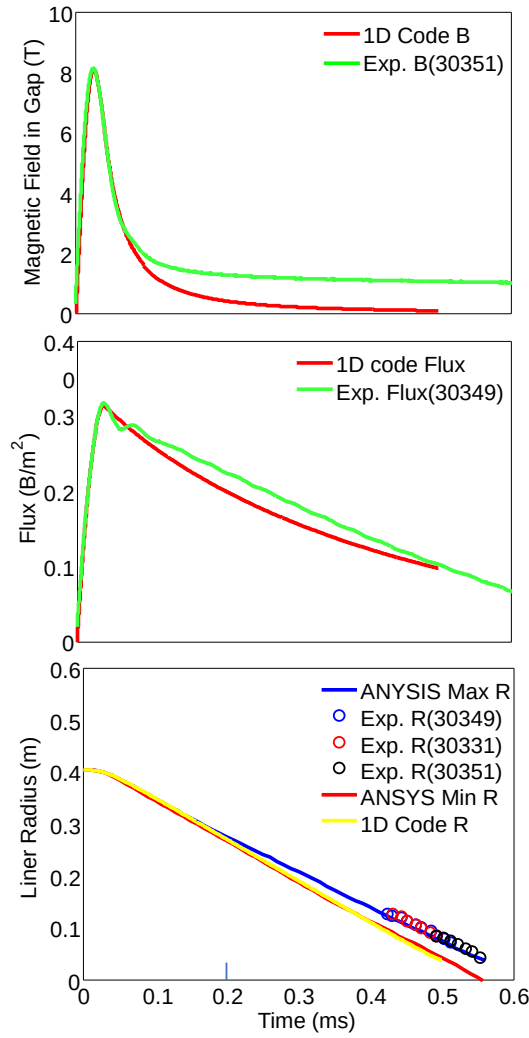


Figure 6. 1D Liner Compression code calculation of the 40 cm radius, 6 cm wide, 0.4 mm thick Aluminum liners at ¼ power (see Fig. 4). Numerical model shows good agreement with experimental results for flux, B field, and position. Discrepancy in position is due to loss of energy in internal stress and strains of the liner. This effect is not captured in the 1D liner code, but is predicted accurately via ANSYS simulations.

operated at one quarter power (170 kJ capacitor bank energy) to prevent post implosion damage. A vacuum environment was essential to avoid generating shock wave and drag as these liners propagated inward radially. Even at ¼ power, the peak magnetic field reached over 200 T with a magnetic pressure $P_B = 0.16$ megabar (2.4 million psi). The Aluminum liners melted and vaporized. Remarkably, there was no visible damage or Aluminum deposit on the outer Garolite vacuum wall. The Aluminum vapor jetted out along the system axis and terminated on 1 inch thick 12-inch square plates of polycarbonate, which showed considerable visible damage. The radial implosion must be highly inelastic and the evacuation of the Al vapor appeared to be guided by the axial magnetic field acting like a shaped charge.

The high energy liners were rapidly heated and melted by Ohmic currents flowing in the liner. As the liner collapses on axis, the magnetic field increases dramatically as the axial magnetic field, which had diffused through the liner during the initial acceleration, is rapidly compressed to very high values. The metal liner prohibits the diffusion of this flux on the short timescale of the final implosion, and the rapidly increasing and very large induced current quickly heats, melts and atomizes the liner.

It should be noted that magnetic flux compression of metal liners has also been used in a series of Japanese experiments to compress magnetic fields of up to 730 T [12,13]. These experiments were done at small scale using copper liners, but at high bank energies (multi-MJ). The magnetic pressure, P_B , ($= B^2/2\mu_0$) created by this field is 2.1×10^{11} Pa or 2.1 megabar. These results are encouraging in that it is very difficult to implode a liner as heavy as copper, but the inertia of the liner allowed for the use of a smaller driver coil. With no radial standoff, these experiments were destructive to both the coil and the containment vessel, but it did illustrate that the technique can be extended to very high compression pressures, which would be non-destructive to the thruster if the proper driver coil stand-off and magnetic insulation of the walls are employed.

4. NUMERICAL ANALYSIS OF THE INDUCTIVELY DRIVEN LINER COMPRESSION (IDLC)

The 1D IDLC Code

To optimize the system design of the Inductively Driven Liner Compression (IDLC) driver system, a model that captures the important physics of the liner implosion dynamics is required. At the same time, this model must also incorporate the driver circuit associated with the capacitor bank that is discharged in order to drive the liner compression. Furthermore, the model must also be computationally cheap, such that many parametric design points can be evaluated in relatively short amounts of time. This is necessary if the code is to serve as a first order design tool to aid in the future designs of the IDLC driver system. In order to meet these design goals, a pseudo-1D model for the liner implosion dynamics is coupled with a circuit solver to capture the full IDLC driver system behavior. The driver circuit, magnetic field, joule heating, and liner dynamics calculations are performed at each time step in sequence to couple these effects into the code. A 1D incompressible model is used for the liner dynamics to reduce the computational cost, while capturing the key dynamic behavior of the imploding liner. To obtain a more realistic magnetic field for a given driver coil geometry, 2D and 3D effects are incorporated into the 1D field calculation through the use of a correction factor table lookup approach.

A similar 1D approach to the modeling of the IDLC driver system has been used successfully in the past to analyze the behavior of the liner implosion process with a decent agreement to the experimental results [3,11]. However, in the previous work, an infinitely long cylindrical model was used. This is a stretch for a hoop-like liner with a large radius

compared with its length. The energy conservation was also not strictly enforced in the coupling of the circuit and the coil-liner system calculation in the previous work. The present model attempts to fix these shortcomings by enforcing a stricter energy conservation between the coupled solvers and by using a correction factor approach to the 1D calculation to retain the computational advantage of the 1D model, while correcting for some 2D and 3D geometric effects. This allows the code to not require a full 2D or 3D magnetic field calculation at each time step, which is computationally expensive since the domain must be re-meshed every time the liner geometry changes significantly.

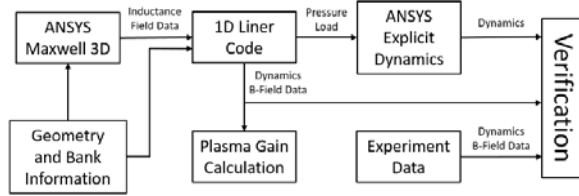


Figure 7. Coupling of the 1D liner code with external data and solvers

By pre-tabulating the approximate magnetic field correction required at each liner geometries, some geometric effects can be captured, while taking advantage of the fast 1D calculations. The 1D liner code is designed to take advantage of more advanced commercially available solvers to add 2D and 3D corrections and to serve as a verification tool. How the 1D liner code couples with the external data and external solvers are illustrated in Fig. 7. To obtain the field, flux, and inductance data for the coil-liner magnetic field calculation,

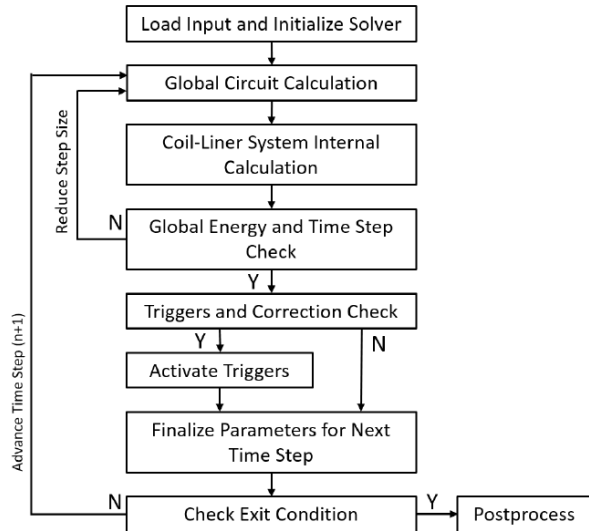


Figure 8. Flowchart of the 1D Liner Code.

ANSYS Maxwell 3D® solvers was used. To verify the liner dynamics, the pressure loads predicted by the 1D liner code is input into ANSYS Explicit Dynamics® Solver (AEDS) to verify that the liner dynamics computed by the 1D liner code agrees with the liner implosion dynamics computed by the AEDS, which accounts for the losses associated with the internal mechanical energy and the plastic deformations. A commercial SPICE solver is used to verify the 1D liner circuit

solver. The flowchart of the implemented 1D liner code is shown in Fig. 8. It is important to note that the 1D liner code is composed of two solvers coupled together: the circuit solver and the coil-liner system internal calculations. Two solvers are coupled together in an energy conserving manner to enforce a global energy conservation.

To test if the circuit model is solved correctly for the case of a dynamic inductance and resistance associated with the coil-liner system, an analysis with a dynamic liner was carried out in the 1D liner code. The resulting dynamic inductance and resistance profile was then exported to the SPICE solver to model case with a dynamic inductance and resistance in the SPICE solver. The profile of the dynamic resistance and inductance calculated by the 1D liner code and the fitted dynamic data imported into the SPICE solver is shown in Fig. 9. The agreement is excellent.

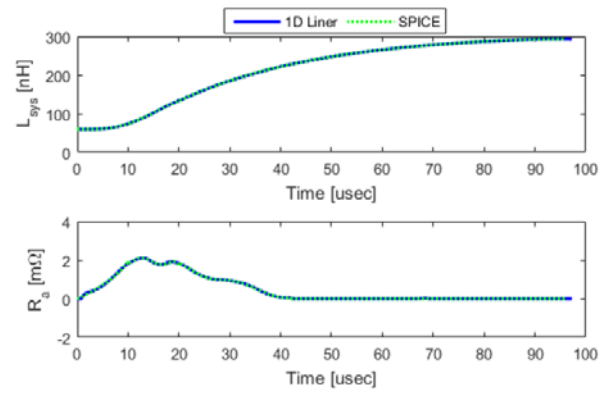


Figure 9. Profile of coil-liner system dynamic inductance and resistance.

A similar test was made for the 1D code and the 3D AEDS. The resulting kinetic energy of the liner and the time dependent liner radius is shown in Fig. 10. It can be seen that the liner dynamics predicted by the 1D liner code agrees in general with the more accurate result calculated by the Explicit Dynamics code, which accounts for true material property and internal mechanical energy losses. The time evolution of the liner radius computed from the Explicit

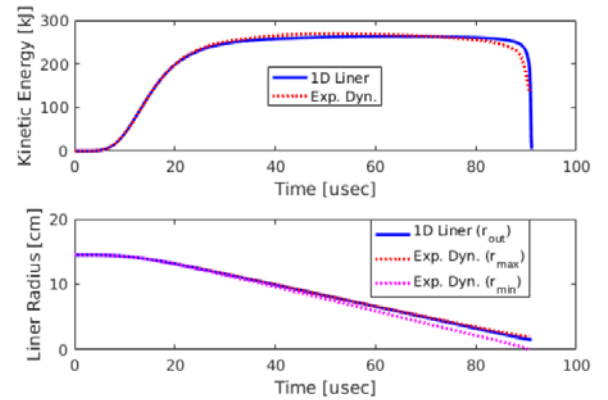


Figure 10. Comparison of liner radius and kinetic energy as function of time calculated by the 1D liner code and ANSYS Explicit Dynamics® Solver (AEDS)

Dynamics solver matches very well with that computed from the 1D liner code. The divergence in the data for the minimum radius computed from the AEDS is caused by the barrel distortion of the liner ends, where the ends collapse faster than the center of the liner due to the higher magnetic field near the coil wall at the ends of a flux-conserving cylinder. However, for majority of the liner, the result from the two solvers agree very well with each other. As it can be seen from Fig 10, the biggest divergence in the kinetic energy results occurs during the final deceleration phase of the implosion process. This is caused by the increasing loss from the plastic hardening of the liner, as the liner is forced to a smaller circumference.

The 1D liner code was employed to identify the optimum IDLC driver system configuration for the F4P rocket. In the parametric analysis, it was determined that the liner thickness of 40 mil (0.2 mm) corresponded to the optimum mass for a lithium liner that gives the desired IDLC driver system performance. It was seen that maximizing capacitor voltage was crucial in improving the performance of the driver system for a fixed capacitive energy stored in the capacitor bank. A minimization of the stray inductance and resistance was also seen to be important for the tested configuration. A slightly negative seed bias flux was also seen to improve the performance of the IDLC driver coil. It was seen that the negative bias also reduced the chance of the liner melting during the acceleration and the coasting phase of the implosion process. The bank voltage of 50 kV and the initial seed bias of -0.2 Wb was identified as an optimal design point that is still engineeringly feasible. The energy-coupling optimization was found to be 69.7 % for a fixed energy storage of 1.8 MJ which is the target size for the F4P IDLC prototype system.

The feasibility of the optimized design was also considered for experimental testing. Key solutions to ensure good liner implosion stability and symmetry were formulated, and will be discussed in the next subsection. Some design requirements for driver coils were analyzed, and it was seen that the requirement for the driver coil is feasible with today's technology. Use of a liner rotation and a liner thickness variation is most likely necessary in practice to ensure good liner stability to avoid these issues from being a significant road block to the further development of the F4P. While the high magnetic field and mega-ampere current flow do cause challenges to the design of driver coil circuit, it should be possible to design a functioning driver coil without use of any exotic materials. While the present model captures the initial acceleration phase of the liner compression well, it was seen that the deceleration phase of the implosion process is not captured as well in the model. In order to capture the deceleration phenomena, more advanced equation of state model for liner must be considered accounting for the liner compressibility, possible internal mechanical energy loss, and phase changes. Improvements to the liner code is currently planned to move toward a coupled a 1¼ D transport code that can handle a moving boundary to study the effect of target and liner interaction to further the understanding of the physics of the fission processes that will be the result of

IDLC and to obtain better estimate for the F4P rocket performance.

Optimization of the implosion geometry employing the 3D ANSYS Explicit Dynamics Solver (AEDS)

The AEDS was used to characterize the liner dynamics in 3D. These calculations were carried out with the same material, dimensions and magnetic pressure time history as the liners employed in the experiments. The near identical results validate the use of the AEDS code for accurately predicting liner dynamics in designing future liner implosion systems (see Fig. 5). A 3D solver is necessary to analyze and test desired liner properties such as liner stability, convergence, both radial and axial, as well as internal stresses and buckling. The buckling seen both experimentally and in the AEDS simulations are expected due to the extremely large ratio of liner diameter to thickness (4,000!). As this ratio increases the mode number of buckles increases. The Roark analytic value for the most unstable mode for this 0.8 meter diameter unsupported cylindrical hoop (liner) was 160-200, which was close to what was observed.

The buckling seen in both experiment and calculations occurs fairly early in the implosion. It appears that the early buckling serves a useful function in that it minimizes the energy lost to compressional internal energy in the liners. As the liners move inward, the buckles naturally merge together to form a thick, compact stable wall at peak compression as desired. At later times when the scale is small, the greatly thickened liner is now very stable to buckling but the thickness and surfaces can be lumpy. This is undesirable as it could lead to hot spots and jetting and thereby create problems during the transition to an ionized gas and further compression. The ability to form very uniform folds or buckles will be important in order to avoid this. It is best not to leave the potential buckling to underlying imperfections or lack of azimuthal symmetry when it would be only slightly more effort to form the liners with the locations for folding preset. This “pleating” will assure a symmetric folding as the liners converge radially

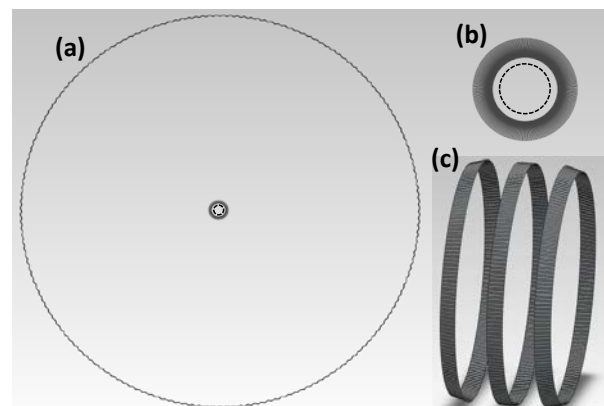


Figure 11. (a) End-on CAD drawing of an aluminum pleated liner with 180 folds dimensioned for the planned megabar experiment ($r = 40$ cm, $\delta = 0.2$ mm). (b) Pleated liner collapsed to inner fold contact point (dashed line is minimum radius at peak compression). (c) three liner assembly

inward with the added bonus of minimal internal stress buildup. The depth of the pleat can be set so that the fold depth is on the order of the final liner thickness minimizing the amount of liner deformation that must occur in compression (see Fig. 11).

Calculations with AEDS were performed for several different liner modulations. It was not difficult to find several that

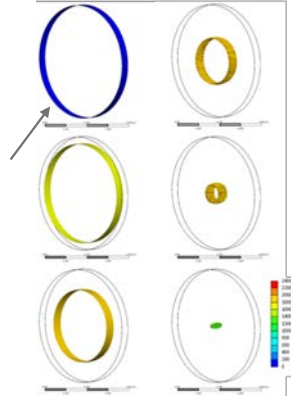


Figure 12. AEDS calculation of the 3D behavior of 5 cm wide, 0.2 mm thick Al liners. The color scale indicates liner speed from 2000 m/s to zero. (Left) No pleating. (Right) with pleating. In the pleated liner calculation, a test target the size of the F4P sabot can be seen. The times for the left liner plots are 0, 0.06, 0.12, 0.18, 0.22, and 0.27 msec.; and the right, 0, 0.04, 0.08, 0.12, 0.16, and 0.18 msec.

eliminated the arbitrary buckling thereby making the foldup uniform and with reduced internal stress. AEDS calculations for the two cases, with and without pleating, are shown in Fig. 12. It can be seen that the proper folding begins early and thereby avoids the eventual buckling instability observed in the case without modulation. The calculation shown had 200 azimuthal modes and was chosen to be similar to the natural buckling frequency. It can be seen that the pleated liner arrives faster and achieves a smaller final radius even with a solid target included at the axis. The reason for this is the elimination of both the internal energy sink from solid compression as well as plastic deformation energy losses during the compression as the foldup of the liner mitigates both.

Liner stability—There is also the possible issue of liner instability with regard to Rayleigh-Taylor (RT) modes during the deceleration and reversal that accompanies the magnetic compression and possible liner rebound. The initial results from the 1D calculation indicated that Ohmic heating from the large induced currents will rapidly bring the inner surface of the liner to the melting point. A liquid liner may relax internal stresses and form a better conducting surface but is unstable to RT. It has been shown by Turchi et al. however that rotation of the liner can completely stabilize these modes [10]. Adding a rotational momentum sufficient to accomplish this is actually not that difficult with the Inductively Driven Liner (IDL) compression method. It should be noted that the liner is free to rotate once it is driven away from contact with

the outer wall. With the large ratios of initial to final liner radius foreseen with the IDL, only a relatively small amount of rotational velocity, if induced early, to be sufficient for stabilization at the small radius where the melting will occur. It was found for a stability that one requires

$$\Omega_0^2 \geq \frac{2}{5\mu_0} \frac{r_0^2 B_0^2}{\rho_{Al} r_i^4 \ln(r_i / r_0)}. \quad (1)$$

For the F4P prototype IDLC parameters: $B = 400$ T, $r_0 = 0.01$ m, $r_i = 0.4$ m, $\rho_{Al} = 2700$ kg/m³ $\Omega_0 \geq 146$ rad/sec. The rotational energy acquired by the liner is 2.3 kJ which is negligible compared to the total liner kinetic energy. The rotation can be induced by simply imbedding a slowly rotating transverse magnetic field into the liner prior to implosion. From force calculations on the proposed aluminum liner, the Rotating Magnetic field (RMF) strength need be no greater than tens of gauss at the frequency of 4 rad/s. An example of the field line structure from a set of axial current rods placed radially outside the driver coils is shown in Fig. 13.

This same quadrupole field has been employed in many FRC

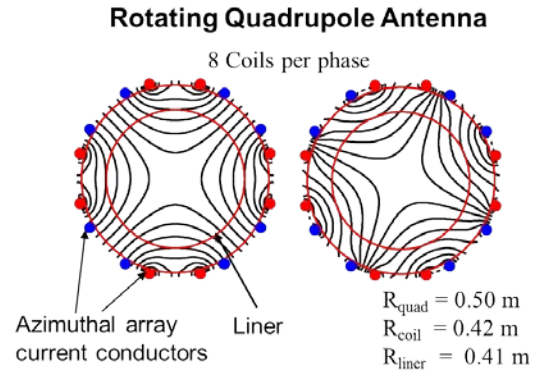


Figure 13. Two-phase coil configuration for rotational stabilization of the liner with a rotating quadrupole magnetic field.

experiments in the past [14] for stabilization and current drive and should work in a similar manner here as these fields will be frozen in the liner wall for the relatively short duration of the liner compression. The quadrupole field will act as a barrier field preventing unwanted contact between the plasma and liner both during and after peak compression.

5. SPACECRAFT AND MISSION ANALYSIS

Mission Analysis

As discussed earlier, the major limiting factors in manned space exploration missions are their long transit times and high costs. The F4P offers a solution that addresses both of these two hindrances by providing a propulsion method that has high specific power, α (engine power/spacecraft mass) in addition to high exhaust velocities (Isp). High specific power (α of order unity) significantly reduces trip times, and high Isp ($> 3,000$ s) substantially increases payload mass fraction, greatly reducing initial mass at LEO. Mission architecture

trade studies have been performed previously over a large range of nuclear gains (electrical energy in/propulsive energy out. and mission parameters [11]. It was found that a gain of 200 was sufficient to create a jet power of 36 MW at high α and an Isp in the range desired (3-5 ksec). Based on the hardware and propellant store needed to create this gain, a suitable spacecraft has been analyzed and designed (see next subsection). It turned out to be possible to plan a manned Mars mission architecture for a round trip mission time of 210 days using only a single launch from Earth. The details of this final mission architecture will now be described.

210 Day Round Trip Mission to Mars—For the final revision of the single launch manned mission to Mars, several small modifications were made to previous mission designs [5,11]. Most notably was the modification of the final mass returning to Earth. Originally, it was speculated that only 15 t would return to earth. However, after the initial investigation and a preliminary scaling of the spacecraft and all of its subcomponents, it was determined that the spacecraft dry mass alone would be about 11 t and that mass would not include a mass for the crewed vehicle. Therefore, an additional 19 t was added to the return mass making the total mass return to LEO from Mars to be 30 t. What this meant from a mission architecture point of view was the majority of the mass would stay in orbit and only a smaller payload would land on the surface of Mars. This seems reasonable in that the vast majority of the mission time (86%) will be spent in transit and only 30 days will be spent on the surface of Mars.

The Copernicus software was used to perform the mission analysis and determine the maximum Mars payload, assuming a limit of 134 t initial mass at LEO and a 30 t return mass. Confined by the ΔV budget of a 210-day round trip and a 30-day stay, it was found that a 30 t Mars lander was the maximum Mars payload. This mass was deemed reasonable considering a full Mars orbit insertion using the main F4P engines, which would minimize the thermal protection systems of the entry, descent, and landing systems. It may be worth noting for a sense of scale that the largest spacecraft to land on the surface of Mars was the Mars Science Laboratory, weighing 0.9 t. Also, it is possible to employ a precursor cargo mission to deliver the lander into Martian orbit prior to the manned mission. This could also include return propellant making the margin even greater, and the payload fraction much larger.

The final mission architecture, therefore, consisted of a trans-Mars injection maneuver with a ΔV of 7.6 km/s over a 7.2-day period. Details of the various mission parameters for the different stages of the mission can be found in Fig. 3. Total propellant used during this maneuver was 18 t. The spacecraft would then coast 63.6 days before performing a Mars orbit insertion burn to be captured into a 1-sol Martian orbit. This burn would last a total of 10.5 days and use 28 t of propellant to apply a ΔV of 13.8 km/s. This would result in a total outbound trip time of 73.6 days. The spacecraft, including the Mars Transit-habitat and the Fusion Driven Rocket, would remain in orbit for 30 days, while the 30 t Mars payload

would be delivered and left on the surface of Mars. Therefore, the initial mass of the returning spacecraft to Earth would be 53mt. The Trans-Earth maneuver to escape the gravity of Mars and set a trajectory back to Earth requires a ΔV of 16.2 km/s, requiring 15mt of propellant over 5.8 days. Again, the spacecraft coasts for 89.6 days before performing a final Earth orbit insertion maneuver to place the spacecraft back into its original circular 407 km orbit. These maneuvers require 11.9 km/s of ΔV and the remaining 8 t of propellant. With a burn time of 3.2 days, the total inbound transit time is 98.7 days, resulting in a total mission time of 210 days.

Spacecraft Analysis

Using a mission down design approach, the desire is to have a spacecraft that is less than 15 t with a rocket engine that produces at least 36 MW of jet power at an Isp of 3-5000 s. The challenge is to keep the spacecraft dry mass to 15 t or less, and to show how the design for the F4P can satisfy this design requirement.

One of the unique and major masses in the F4P concept will be the energy storage system required to drive the liners. As has been indicated earlier, this will entail high voltage, pulse capacitors. The assumption here is that the amount and distribution of nuclear material will be sufficient to yield on the order of at least 200 times the energy expended by the liner driver. It was found from the 1D liner calculations that at 2.8 MJ of stored energy will be required to reach the megabar pressures desired for adequate compression of the nuclear fuel. Given this yield, a nominal liner mass of 0.5 kg resulted in an Isp $\sim$ 5000 s as desired. To achieve an average jet power of 36 MW with 90% conversion efficiency will require a rep rate of $\sim$ 4/minute providing sufficient time for liner reload and recharging the energy storage systems. It will be assumed that the stored (bank) energy must be at 1.5 times the liner kinetic energy, reflecting the optimal coupling efficiency of 70% along with the calculated resistive circuit losses [4]. A 4.2 MJ bank of capacitors is thus required. Commercial high voltage, high energy density, pulse capacitors have demonstrated energy densities as high as 3 kJ/kg [15]. A somewhat lower value of 2.5 kJ/kg will be assumed here will be 2.5 J/g for margin, resulting in a total capacitor mass $\sim$ 1.7 t. Based on past experience, the energy delivery system (solar panels and converters, charging system, start and crowbar switches, housings, support structures, power feeds, cable, and stripline) will scale with the capacitor mass. The multiplier would certainly be less than half the capacitor mass. A very conservative value of roughly half however is assumed or 0.8 t.

The second major power system component that will be needed is the solar electric panels. Given that there is enough risk associated with the F4P concept, that at least for the initial analysis, the electrical energy will be powered by solar panels. There are several reasons for this. First, solar panels have a long flight heritage and a proven performance record in space. They have flown on over 99% of space missions launched to date. Second, while energy could certainly be obtained from the flux compression occurring during propellant expansion out of the rocket engine, this adds

complexity and risk. While this type of energy recovery may be incorporated later in a fusion rocket development program, it was not felt to be critical to the concept feasibility. The current state of the art solar panels are expected to generate over 200 W/kg, and this value was used in calculating the solar panel mass. It should be noted that the development of much higher density solar arrays are anticipated in the near future. At a fission-based gain of 200, the 36 MW of jet power needed for the 210-day Mars mission will thus require 180 kW of solar power. This is the solar power that will be needed throughout the mission; including Mars where the solar irradiance is 56% less. Since panel performance metrics such as specific mass are scaled for Earth irradiance, the solar panel system mass for FDR must be scaled up by a factor of 2.25 to 2 t for the aforementioned solar panels with an Earth-based α of 200 W/kg.

As was described earlier, the F4P Rocket is comprised of main liner compression coils as well as coils to create a divergent magnetic field at the exit. The mass of these coils is estimated based on them being high strength aluminum with the appropriate cross sections to be structurally sound, and capable of handling the required currents and impulsive loads. This resulted in a total mass of 0.4 t. These coils will certainly be fabricated taking advantage of modern, lightweight, high field structural materials such as SiC-SiC composites and other materials suitable for a neutron and space environment. Certainly, carbon composite structures would also be used to provide rigidity to the coil assemblies.

A thermal management system will also be needed. This system will most likely consist of a working fluid that will be pumped through the high heat flux components such as magnetic coils and power electronics. The fluid will be radiated through space radiator panels. Detailed MCMP calculations of the neutron, gamma and particle physics are needed with a more detailed understanding of the energy deposition in the various structures; then a more in-depth design of the thermal management system can be made. For now, it is assumed that a large fraction of the neutron energy that escapes the sabot will be absorbed and used in heating of the liner along with the fission fragments which comprise by far the largest energy output from fission. Given the large stand-off, along with the dimensional thickness of most of the structures, the energy deposition from the plasma propellant will be small. For high temperatures, space radiator specific mass of 1 kW/kg is quite conceivable. For this case study, due to the uncertain nature of the heat load to the engine subcomponents, a design margin of a factor of three was added. As a result 1.1 MT was allotted for the thermal control/heat rejection system.

The final two major spacecraft components relate to the propellant mass. The primary propellant for the F4P Rocket will likely be composed of the lithium in the form of liners. Since lithium is a solid at room temperature, a low mass propellant storage system can be employed with a minimal tank mass fraction. For the purpose of this study, it was assumed a constant tankage mass of 100 kg. The propellant feed mechanism on the other hand will require more mass.

This device is responsible for assembling and injecting the liners into the magnetic coils before each thruster firing. The optimum method is clearly one that requires the least amount of complexity on orbit, therefore it is assumed that the liner material will be completely fabricated to the required specification on earth and then loaded on to a roll. These rolls would then be used to form the liner hoops that would be guided mechanically, and injected electromagnetically at low velocity into position under the driver coils prior to driver coil activation. This method would also allow for possible layering of material within the liner to optimize for lower Ohmic losses, better neutron reflection and liner stability. While the envisioned process appears straightforward, a large mass budget of 0.7 t was specified due to the lack of a detailed assembly design at this point in the design.

The spacecraft structural components include all of the remaining subsystems normally included in a spacecraft design. This includes fairings, support structures, communication systems, data handling ACS, and battery power system. An important aspect of the support structure is a shock absorbing system that interfaces the FDR with the spacecraft structure. The total mass for the spacecraft structure is therefore given the largest mass budget of any subcomponent in this analysis at 2.7 t. As reflected in Table 1, all of these components result in a spacecraft dry mass of 11 t. The payload habitat and space exploration vehicle mass from the DRA5 was 40 t. The total lithium propellant mass

Table 1. Spacecraft Mass Budget

Spacecraft Component	Mass (mt)
Energy storage³	1.7
Solar Panels⁶	2.0
Pulse Electronics and cables	0.8
Spacecraft structure	2.7
Lithium containment vessel	0.1
Nuclear Materials and Delivery	1.2
Propellant Feed mechanism	0.7
Liner driver coils⁴	0.3
Thermal Management	1.1
Magnetic Nozzle	0.4
Spacecraft Mass	11
Crew habitat (DRA5.0)	33
Lithium Compound Propellant	56
Space Exploration Vehicle Mass	7
SUBTOTAL	107
Margin (25% of subtotal)	27
Total Mass	134

was 56 t for the total IMLEO mass is 107 t. In an attempt to be conservative, an additional margin of 25% for the entire spacecraft mass budget was included in addition to the margin and de-rating factor discussed for individual components for a grand total of 134 t.

Spacecraft Design—A schematic and rendering of the spacecraft can be found in Fig. 14. The lithium-based propellant is stored on 1.6-meter diameter spools, 9 m in

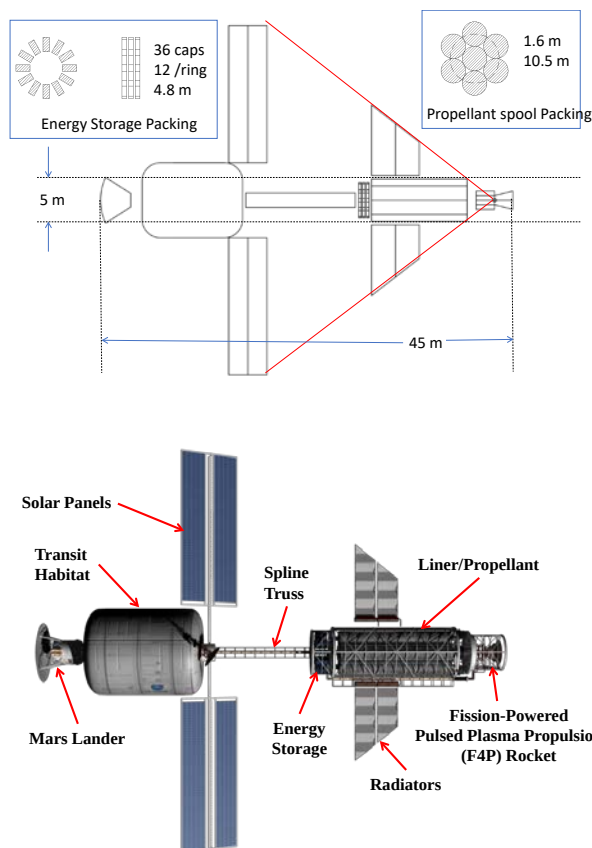


Figure 14. Schematic (top) and rendering (bottom) of the full 210-day F4P spacecraft. Total spacecraft length is 45 m with a diameter of 5 m. (Diameter does not include the solar and radiator panels)

length. Each spool Six spools are arranged azimuthally around a seventh central spool for a total propellant mass of 56 t of propellant. A good packing fraction is produced by this arrangement, and it also provides for all of the material to fit within a 5 m diameter saddle truss. This shield placement also acts as a shadow shield for all of the spacecraft components including solar panels. A saddle truss was employed to facilitate refueling and reuse of the F4P spacecraft. It also provides for the option of a propellant or equipment drop for an even faster 90 day round trip Marian mission (30 day transit with a 30 day stay). This mission would likely require fuel pre-deployment at Mars. A schematic of the device as well as a rendering is found in Fig. 13.

The heat rejection radiator system is also attached to the saddle truss. They are approximately 5.2 m wide and 7.7 m long with a surface area 120 m². The radiators have been profiled to stay within the shadow shield to limit neutron exposure. Located at the forward most end of the saddle truss is the energy storage and electronic components to power the main compression coils as well as the magnetic field coils. This compartment also contains all of the power conversion modules and capacitor charging supplies. The power is fed down along a box frame truss on the side of the saddle truss to the FDR in an insulated and cooled stripline configuration. In this configuration, the saddle truss and the attached

components can be classified as part of the propulsion system of the spacecraft. It includes the engine itself as well as all the power systems, cooling systems, and propellant. The propulsion system is attached to the rest of the space craft via a 9 m long spine truss. This truss allows more stand off from the propulsion system, reducing the size of the shadow shield if needed. The spine truss also acts as a good separation point to modularize the spacecraft if required and would provide a location for addition equipment and payload to be stowed or attached. At the forward most end of the spline truss is the main solar panels and the communication subsystem. The solar array is located here to place it well inside the shadow. Each solar panel is 4.2 m wide and 15 m long adding up to a total surface area of 126 square meters. The Mars Transhab is 8.2 m in diameter and 11 m long and would mostly likely be composed of an inflatable structure. At the very front of the vehicle is the Mars Lander. The spacecraft length is 34 m in total. The component design was carried out mindful of the diameter limits of future planned payload fairings. This of course requires the use of deployable solar and radiator arrays, inflatable habitat, and possibly a compressible spine truss section.

6. SUMMARY AND FUTURE PLANS

The nuclear propulsion concept, Fission-Powered Pulsed Plasma Propulsion (F4P) has been elucidated, and the key subsystems described and evaluated for feasibility. If successful, the most significant application for the proposed F4P concept would be to make routine; long-duration, exploratory deep-space missions with a rapid Mars mission as the logical first step. Current plans using either chemical-based or reactor-based (thermal or electric) propulsion have been stymied by enormous launch costs and extremely long mission times, which endanger the health of the astronauts, and increase the risk of failure due to the need for complex space rendezvous and assembly. All of this is avoided with a compact, high efficiency, single-launch spacecraft such as projected for the F4P. The mission architecture and spacecraft design for a single launch, a 210-day manned Mars mission has been analyzed for a fusion-based spacecraft similar to F4P in the power delivered by the nuclear fuel. Trip times as short as 30 - 90 days were envisioned with payload mass fractions of 36-65%. Unlike fusion however, the F4P can be realized with little extrapolation from known physics and current technology, and can operate at high specific power.

The experimental and theoretical results from research by the author and others have validated the metal hoop compression process needed to produce the desired nuclear energy gain. The major unknown to be resolved is the functional dependence of the fission energy yield on the several key parameters of the compression process. This must be fully understood and tested to prove the viability of the F4P concept. The future effort will focus on (1) a detailed physics model of the heating and compression of the fissile fuel from a solid to an ionized gas. (2) Calculations will also be carried out to determine the fission amplification and energy yield

throughout the compression cycle as a function of fissile mass and density, as well as Li mass and compression energy. From these models and calculations, the optimal parameters for fission energy gain and propulsion effectiveness will be determined. Based these results, a plan for testing of the F4P concept will be developed. Initially, surrogate materials with similar properties to the Li and $^{235}\text{U}/^{239}\text{Pu}$ would be used (e.g., Al and W). Further testing with the Li and suitable nuclear materials (e.g. LEU) to validate the fission energy gain scaling would follow.

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REFERENCES

- [1] Bret G. Drake and Kevin D. Watts, *Human Exploration of Mars Design Reference Architecture 5.0 Addendum #2* NASA/SP-2009-566-ADD2
- [2] Zarana S. Patel, Tyson J. Brunstetter, William J. Tarver, Alexandra M. Whitmire, Sara R. Zwart, Scott M. Smith and Janice L. Huff *Red risks for a journey to the red planet: The highest priority human health risks for a mission to Mars* Nature PJ - Microgravity (2020) 6:33
- [3] John Slough, Anthony Pancotti, David Kirtley, “*Analysis of Inductively Driven Liners for the Generation of Megagauss Magnetic Fields*”, 14th International Conference on Megagauss Magnetic Field Generation and Related Topics, paper 4056263, (2012).
- [4] Shimazu, A., Slough, J., and Pancotti, A., “*Parametric Optimization of the Fusion Driven Rocket Liner Compression Driver*,” 52nd AIAA/SAE/ASEE Joint Propulsion Conference, AIAA Propulsion and Energy Forum, July 2016.
- [5] Slough, J., Pancotti, A., Kirtley, D., and Votroubek, G., “*Electromagnetically Driven Fusion Propulsion*,” 33rd International Electric Propulsion Conference, October 2013.
- [6]<https://www.youtube.com/watch?v=tHSOmOu61b0&t=12s>
- [7] T.A. Oliphant, F.L. Ribe, and T.A. Coultas, “*Direct Conversion of Thermonuclear Plasma energy by High Magnetic Compression and expansion*”, Nuclear Fusion **13**, 529 (1973).
- [8] A.A. Harms, K.F. Schoepf, G.H. Miley, D.R. Kingdom, (2002) *Principles of Fusion Energy*, World Scientific Publishing, Singapore 912805, pgs. 267-277.
- [9] Cnare, E.C., “*Magnetic Flux Compression by Magnetically Imploded Metallic Foils*”, Journal of Applied Physics, Vol. 27, No. 10, pg. 3812, (1967)
- [10] P.J. Turchi, A.L. Cooper, R.D. Ford, D.J. Jenkins, and R.L. Burton, “Review of the NRL Liner Implosion Program”, Megagauss Physics and Technology, Ed. Peter Turchi, Plenum Press, New York (1980).
- [11] Pancotti, A., Slough, J., and Shimazu, A., “*Mars Mission Trade Studies and Technology Development of a 36 MW Fusion Rocket*,” Joint Conference of 30th International Symposium on Space Technology and Science, 34th International Electric Propulsion Conference, and 6th Nano-satellite Symposium, July 2015.
- [12] Y. H. Matsuda, F. Herlach, S. Ikeda, and N. Miura, “*Generation of 600 T by electromagnetic flux compression with improved implosion symmetry*”, Rev. Sci. Instrum. **73** 4288 (2002).
- [13] N. Miura, Y.H. Matsuda, K. Uchida, S. Todo, T. Goto, H. Mitamura, T. Osada, E. Ohmichi, “*New Megagauss Laboratory of ISSP at Kashiwa*”, Physica B **294-295** (2001) 562-567.
- [14] J.T. Slough and K.E. Miller, Phys. Rev. Lett. **85** 1444 (2000).
- [15] R. MacDonald, M.A. Schneider, J.B. Ennis, F.W. MacDougall, X.H. Yang, “High Energy Density Capacitors”, *IEEE Electrical Insulation Conference*, (2009).

BIOGRAPHY



John Slough received his B.A and PhD in Astrophysics from Columbia University. He has spent over 30 years working in the field of experimental physics. He founded MSNW in 1991, a small business with a long history in the research and development of advanced space propulsion and plasma physics technologies. He was the principal investigator on several large fusion related projects funded by the Dept. of Energy. Experiments performed by Dr. Slough led to advances in the formation and confinement of FRCs as a fusion concept as well as a better understanding of the relevant physics. Recently, he founded a spin-off company, Helion Energy, to develop a commercial fusion reactor which has received both DOE and venture capital funding. Dr. Slough was also on the faculty in the Astronautics and Aeronautics Dept. at the University of Washington for many years.