Electromagnetic propulsion technology has been thought to provide a potential form of future spacecraft propulsion for some time. In contrast to ion thrusters, which utilize the Coulomb force to accelerate positively charged species, electromagnetic propulsion systems utilize the Lorentz force to accelerate all species in a quasi-neutral state, providing significant technological benefits over ion thrusters. Several forms of electromagnetic propulsion have been researched and developed, such as the Variable Specific Impulse Magnetoplasma Rocket, pulsed inductive thrusters, and the electrodefluid plasma thruster. One of the most promising forms of electromagnetic propulsion, however, has been the magnetoplasmadynamic thruster. Whereas other electromagnetic propulsion systems provide high specific impulse values but low thrust capabilities, magnetoplasmadynamic thrusters have demonstrated the potential for both high specific impulse values and high thrust densities. However, magnetoplasmadynamic thrusters are not without drawbacks, and suffer from cathode erosion issues, among others. A proposed subtype of these thrusters, known as the Lithium Lorentz Force Accelerator, is thought to address some of these issues. As is demonstrated in this paper, Lithium Lorentz Force Accelerators may present a promising near-term form of spacecraft propulsion for rapid exploration of the solar system, however much more data is needed to analyze the efficacy of large-scale configurations.

Nomenclature

\[ \begin{align*}
  a & = \text{Acceleration, m/s}^2 \\
  a_0 & = \text{Ion sound speed} \\
  a_{sp} & = \text{Specific acceleration, m/s}^2 \\
  A & = \text{Area, m}^2 \\
  B_A & = \text{Applied magnetic field, T} \\
  E & = \text{Energy, J} \\
  E_{sp} & = \text{Specific energy, J/kg} \\
  g_0 & = \text{Earth’s gravity, m/s}^2 \\
  I_{sp} & = \text{Specific impulse, s} \\
  J & = \text{Current, A} \\
  m_b & = \text{Mass of batteries, kg} \\
  m_{dry} & = \text{Dry mass of spacecraft, kg} \\
  m_{prop} & = \text{Mass of propellant, kg} \\
  \dot{m} & = \text{Mass flow rate, kg/s} \\
  p & = \text{Pressure, Pa} \\
  P & = \text{Power, W} \\
  P_{sp} & = \text{Specific power, W/kg} \\
  P_T & = \text{Thrust power, N/W} \\
  r & = \text{Radius, m} \\
  T & = \text{Thrust, N} \\
  u & = \text{Energy density, W/m}^3
\end{align*} \]

* Chief Scientist, Department of Research & Development.
I. Introduction

SUBSTANTIAL research on Lithium Lorentz Force Accelerators (LiLFAs) has been conducted during the past several decades, in particular by Edgar Y. Choueiri and several colleagues at the Princeton University Electric Propulsion and Plasma Dynamics Laboratory (EPPDyL) [1-14]. As such, this paper will not serve as a detailed technical treatment of LiLFAs, but rather as a review of the current efficacy of their near-term implementation as a means of spacecraft propulsion.

LiLFAs utilize a multi-channel hollow cathode system as opposed to the solid cathodes used in traditional magnetoplasmadynamic thrusters (MPDTs). This serves to drastically reduce the issue of cathode erosion suffered by traditional MPDTs, which can be as high as 0.2 µg/C. LiLFAs also show increased efficiency, with values of applied-field (AF) systems exceeding 40%. These values are only present at power levels exceeding 500 kW; however, this survey will be restricted to LiLFA applications in the MW power range as these are more relevant to large-scale spacecraft [12]. This also allows the study of self-field (SF) applications, which can provide higher efficiencies than AF applications at sufficient power levels. The main issue regarding the implementation of LiLFAs has historically been the lack of sufficient power sources in spacecraft, which has been well-documented over the past several decades. In particular, there has been an absence of required steady-state power capabilities (in the MW range) as well as the required total energy capacity (on the order of 10^2 MWh for sufficiently rapid missions to Mars). Sankaran et al conducted a survey of LiLFAs and other propulsion technologies in 2003 and reported these issues. However, a general review of this technology has not been conducted in almost two decades, and another analysis is required to determine any changes in the feasibility of LiLFAs.

II. Astrodynamic Models

Kodys et al [12] have established thrust densities for MPDTs and LiLFAs, which are measured with respect to the exhaust area of the thruster, to be on the order of 10^3 N/m², and have determined specific impulse (I_{sp}) values to be in the range of 1500-8000 s. This allows for the introduction of a theoretical large-scale spacecraft for the purposes of conducting a feasibility analysis (see Fig. 1). This spacecraft has a dry mass (m_{dry}) of 60,000 kg, a payload volume of 500 m³, and a propellant mass (m_{prop}) of 100,000 kg.

Fig. 1 Theoretical spacecraft.
This theoretical spacecraft is powered by five LiLFAs that each have an exhaust area of $2 \text{ m}^2$, resulting in a thrust value ($T$) of 200 kN per thruster and a total thrust value of 1 MN. Note that the exhaust area $A_e$ is a function of both the anode radius, $r_a$, and the cathode radius, $r_c$. With this information, a transit time to Mars can be calculated. Note that this analysis assumes the scenario of a robotic mission in which the spacecraft does not return to Earth.

As this is a SF configuration, the two components from which thrust is generated are

$$T_{SF} = bj^2,$$

where $b$ is a geometric scaling factor defined by

$$b = \frac{\mu_0}{4\pi} \left[ \ln \left( \frac{r_a}{r_c} \right) + \frac{3}{4} \right];$$

and

$$T_{GD} = \dot{m} a_0 + p_e A_e,$$

where $a_0$ is the ion sound speed, $\dot{m}$ is the mass flow rate, and $p_e$ is the pressure at the exhaust exit [2]. LiLFAs of this scale have yet to be tested, so avoid complicating this discussion the total value of $\dot{m}$ for the LiLFAs our spacecraft uses was empirically determined to be 16.7 kg/s. Using Newton’s second law, this value was obtained from the expression $\dot{m} = (T - p_e A_e)/v_e$, using an exhaust velocity $v_e = 60,000 \text{ m/s}$ and neglecting the term $p_e A_e$ for simplicity (the thrust generated from this term would likely be negligible for large-scale SF configurations). The value of $v_e$ was obtained from an extrapolation of data from multiple sources and should provide a sufficient degree of accuracy for this analysis. To determine the transit time of our spacecraft, the General Mission Analysis Tool (GMAT), a high-fidelity mission analysis and trajectory optimization tool developed by NASA’s Goddard Space Flight Center, was used. GMAT allows for various types of mission simulations and allows users to determine unknown variables given a set of desired goals, as well as allowing users to optimize mission parameters. An electric thruster was used as the propulsion device using the parameters above and an $I_{sp}$ of 6104 s, which was obtained from the expression $I_{sp} = T/\dot{m}g_0$, where $g_0$ is the acceleration due to Earth’s gravity. This thruster was coupled with an electric tank having the previously defined propellant mass of $m_{prop} = 100,000 \text{ kg}$. The departure date was chosen to be 5/5/2018 to allow a comparison of our spacecraft’s mission to that of the InSight robotic lander, the most recent mission to Mars (the departure date was retroactively modified in order to converge on a solution for the new thrusting conditions). The trajectory was based on a B-Plane targeting of Mars, with the spacecraft decelerating before arrival to approximately 7000 m/s relative to Mars. The simulation produced a total transit time of 94 days, approximately 46% of the Insight lander’s transit time of 205 days. The simulation was performed for various values of $m_{prop}$, and the transit times for these values can be seen in Fig. 2.

![Fig. 2 Various transit times to Mars.](image)
The performance characteristics of large-scale SF configurations are therefore proven to be superior to that of current propulsion technology. As previously stated, the primary limiting factor regarding the implementation of these LiLFAs has been the lack of sufficient power in spacecraft. Although there have been significant advancements in spacecraft power over the past several decades, they still may not be enough to make large-scale SF configurations a feasible near-term form of spacecraft propulsion, as will be shown below.

III. Analysis of Energy Technology

Although the lack of sufficient power in spacecraft has long plagued the use of LiLFAs, improvements in energy technology by organizations in varying industries may have the consequence of changing the state-of-the-art in spacecraft propulsion. As such, we will now conduct an analysis of these advancements in order to better understand the efficacy of LiLFAs.

Of all the developments that have been made in a wide range of energy technologies during that past several years, some of particular relevance are the substantial advancements that have been made regarding lithium-ion (Li-ion) batteries. Companies such as Tesla, Inc. have made great improvements in the energy capacity and peak power delivery of these batteries for use in their electric vehicles (EVs). The United States Department of Energy (DOE) is also funding research in an effort to develop Li-ion batteries with a specific energy of $E_{sp} = 500$ Wh/kg [15]. However, the future of battery technology may be in the form of new solid-state batteries, Michael A. Zimmerman, a researcher at Tufts University, developed a solid polymer electrolyte to replace the liquid electrolyte in Li-ion batteries, and founded the company Ionic Materials, Inc. to research and develop this technology [16]. Additionally, the company Solid Power, Inc. is developing similar all-solid-state batteries (ASSBs), which replace the liquid electrolyte and plastic separators in Li-ion batteries. This provides more stability across a broad temperature range and allows for more efficient packaging, effectively increasing the energy density of the batteries. The company reports that their batteries will have a specific energy of $E_{sp} = 320$-700 Wh/kg, an energy density of $u = 700$-1100 kWh/m³, and a specific power exceeding $P_{sp} = 1$ kW/kg. With these values, we can determine the near-term feasibility of LiLFAs.

The approximate thrust to power ratio (referred to here as thrust power) of 25 N/MW for LiLFAs has been established for some time [11]. Using this ratio, we can determine the power requirement of the LiLFAs our spacecraft uses to be $P = 40$ GW. Then, using the value of the total operating time of the LiLFAs in the simulation discussed in Section II., as well as the upper limits of the ASSBs under development by Solid Power, Inc., we can determine that we would need $8.08 \times 10^7$ kg (80,800 t) and $5.14 \times 10^4$ m³ of ASSBs to meet the energy requirement, or $4.00 \times 10^7$ kg (40,000 t) and $2.55 \times 10^4$ m³ of ASSBs to meet the power requirement. This is a substantial technological requirement, especially considering that this analysis only accounts for a one-way robotic mission to Mars. Note that, in this case, it can be seen that the limiting factor is the total energy requirement.

IV. Analysis of Applied-Field Configurations

We will now repeat the calculations made in Sections II and III, but with a smaller spacecraft that utilizes a single small-scale AF configuration system. This new theoretical spacecraft will be designed to send a small rover to Mars, as opposed to humans or large equipment. This spacecraft serves as both an aeroshell and an integration structure for the LiLFA, as well as the propellant storage device. It has a dry mass of $m_{dry} = 2000$ kg, which includes the rover as well as 1000 kg of ASSBs. Using the upper limits of the ASSBs, we can determine that the spacecraft produces a power output of $P = 1$ MW with a total energy capacity of $E = 700$ kWh. Using the previously established thrust to power ratio of 25 N/MW, we can determine the thrust output of our LiLFA to be $T = 25$ N, which corresponds to an exhaust area of $A_e = 2.5$ cm². For reference, the AF thrust component is defined by

$$T_{AF} = kB_{A}\tau_{th},$$

where $k$ is a scaling constant and $B_{A}$ is the applied magnetic field [1]. However, here we use the thrust to power ratio for simplicity. Again, we can determine the value of $\dot{m}$ from the expression $(T - p_{A}A)/\nu_{e}$. Here we will choose a lower exhaust velocity of $\nu_{e} = 40,000$ m/s as this value is in the range of reported exhaust velocities for AF configurations, and again neglecting the term $p_{A}$ we obtain $\dot{m} = 0.625$ g/s, which corresponds to a specific impulse of $I_{sp} = 4077$ s. Our LiLFA will be limited to 2520 s of operation based on the total energy capacity of the ASSBs, which corresponds

---

† This data is according to the Solid Power, Inc. website as of the submission of this paper.
to a propellant mass of $m_{prop} = 1.58$ kg. The GMAT simulation was performed again with these new parameters, producing a total transit time of 204.8 days, nearly identical to the InSight lander’s transit time. To further investigate this result, the simulation was performed for masses of ASSBs. Transit times for various values of $m_b$ (mass of batteries) and $T$ can be seen in Fig. 3.

![Fig. 3 Transit times to Mars for various values of $m_b$ and $T$.](image)

It can be seen from the data in Fig. 3 that the curve is relatively flat due to the particular relationship between this set of parameters, with each value differing by less than $10^{-2}$ days. This is due to the relatively negligible effect that thrust has on the overall trajectory, which is a result of the specific power of the ASSBs. Further investigation leads to the conclusion that the specific power of the power source onboard a spacecraft has a substantial impact on the effectiveness of the propulsion system. This leads to the derivation of a new metric to assess the performance of electric propulsion designs, which is expressed as

$$a_{sp} = \frac{P_T P_{sp}}{m},$$

(5)

where $a_{sp}$ is the specific acceleration of the electric propulsion design, $P_T$ is the thrust power, and $m$ is the total mass of the spacecraft. Specific acceleration is the increase in acceleration per unit increase in battery mass for a given spacecraft mass and is in units of m/s²·kg. If we use the values for the AF configuration discussed above, $a_{sp}$ becomes $1.25 \times 10^{-5}$ m/s²·kg. This low value explains the lack of influence the LiLFA exhibited during the simulations with the AF configuration. Note that the low value of $P_{sp}$ for the ASSBs also means that the procedure used in Section II. is flawed and that adding the $8.08 \times 10^7$ kg of batteries would increase the transit time to a similar value of 204.7 days. Also note that Eq. (5) can be used to determine the specific acceleration of any electric propulsion device.

As it appears that battery technology is still insufficient for use with LiLFAs, the next logical candidate would be nuclear energy. Although governments have historically been prohibited from using nuclear power is space due to various legislation, the political climate regarding small-scale nuclear power systems has warmed in recent years, and governments are now beginning to research this technology. One device of particular interest is the Kilopower Reactor Using Stirling Technology (KRUSTY), a small fission reactor that is being researched by NASA and the DOE’s National Nuclear Security Administration. As the name would suggest, KRUSTY reactors use the Stirling cycle to generate electrical energy from the fission of uranium-235. The reactors are intended to be produced in four different sizes and will produce 1–10 kW, depending on the variant. They are also designed to be intrinsically safe and have a number of mechanisms to help reduce the risk of a nuclear meltdown, including a passive cooling system. Although the 10 kW variant’s mass of 1500 kg results in a specific power of $P_{sp} = 6.67$ W/kg, which is substantially lower than the specific power of the ASSBs, its estimated 12-15 year lifespan results in an effectively unlimited energy capacity during the course of an interplanetary mission, meaning that the duration of acceleration is limited only by the mass of propellant. This can be especially beneficial for robotic missions that explore deep in the solar system. Additionally,
NASA reports that four of the 10 kW variants are sufficient to provide In-Situ Resource Utilization (ISRU) on Mars by separating and cryogenically storing oxygen from the Martian atmosphere for use as propellant, and are also sufficient to support a crew of 4-6 astronauts after the ISRU phase is complete [17]. This represents promising progress regarding the feasibility of a manned mission to Mars, and further advancements of these reactors will improve their feasibility as form of power for LiLFAs.

It would appear from this analysis that LiLFAs still represent one of the most promising options for future spacecraft propulsion. However, substantial advancements in energy technology must be made in order for these devices to become better than current propulsion technologies, especially in the case of large-scale SF configurations. In particular, the specific power of spacecraft power systems must improve drastically. Battery technology is yet to be sufficient or powering these devices, and other power options, such as the KRUSTY reactor, may represent a better technological investment.

V. Conclusion

As can be seen from the values above, the current state-of-the-art in energy technology is still insufficient for the implementation of large-scale SF configurations by a considerable degree. Although significant advancements have been made over the past several years, and although research and development by organizations in various industries shows promising signs of continued advancement, an alternate form of energy may need to be considered for the successful implementation of these types of LiLFAs. The most obvious candidate would be nuclear energy. However, governments have historically been hesitant to use large-scale forms of nuclear energy in space for political reasons, and as a result current space-based nuclear power technology is small in scale and does not meet the requirements for large-scale LiLFAs. Until large-scale nuclear power systems are developed for space applications, or until the required advancements have been made in battery technology (which likely will not be for several decades), large-scale SF configurations will remain a long-term form of spacecraft propulsion, and manned missions will need to utilize other forms of propulsion. It is important to note that small-scale AF configurations still show promise as a near-term form of propulsion for small spacecraft, particularly for unmanned missions, and continued advancements in energy technology will bring these devices closer to reality.

At the very least, these results should warrant further investigation into the near-term readiness of this technology when coupled with new small-scale nuclear reactors. Historically, organizations such as NASA have been required to balance competing interests and limited resources. However, the technological advancements that LiLFAs present may justify a substantial increase in funding towards this technology and towards the supplemental energy technology. Regardless, as Princeton University’s EPPDyL and other organizations continue to make progress with this technology, a better understanding of LiLFAs will be gained and the state-of-the-art in the field will continue to advance.

Acknowledgments

The author would like to thank Joseph Dygert, Dr. Patrick Browning, and Dr. Christopher Griffin of the Department of Mechanical and Aerospace Engineering at West Virginia University for their valuable advice, as well as the NASA Goddard Space Flight Center for providing GMAT.

References

[1] Coogan, W. J., and Choueiri, E. Y., “Applied-Field Topology Effects on the Thrust of an MPDT,” 35th International Electric Propulsion Conference, Atlanta, GA, IEPC-2018-218, 2017.
[2] Coogan, W. J., and Choueiri, E. Y., “A Critical Review of Thrust Models for Applied-Field Magnetoplasmadynamic Thrusters,” 53rd AIAA/SAE/ASEE Joint Propulsion Conference, Atlanta, GA, AIAA 2017-4723, 2017. https://doi.org/10.2514/6.2017-4723
[3] Coogan, W. J., Helper, M. A., and Choueiri, E. Y., “Direct Measurement of the Applied-Field Component of the Thrust of a Lithium Lorentz Force Accelerator,” 52nd AIAA/SAE/ASEE Joint Propulsion Conference, Salt Lake City, UT, AIAA 2016-4537, 2016. https://doi.org/10.2514/6.2016-4537
[4] Helper, M. A., Coogan, W. J., and Iardi, B. L., “Liquid Metal Mass Flow Measurement by an Inductive Proximity Detector for Use in Conjunction with a J × B Pump,” 52nd AIAA/SAE/ASEE Joint Propulsion Conference, Salt Lake City, UT, AIAA 2016-4536, 2016. https://doi.org/10.2514/6.2016-4536
[5] Lev, D., “Investigation of Efficiency in Applied Field MagnetoplasmaDynamic Thrusters,” PhD thesis, Princeton Univ., Princeton, NJ, 2012.

[6] Lev, D., and Choueiri, E. Y., “Scaling of Anode Sheath Voltage Fall with the Operational Parameters in Applied-Field MPD Thrusters,” 32nd International Electric Propulsion Conference, Wiesbaden, Germany, IEPC-2011-222, 2011.

[7] Lev, D., and Choueiri, E. Y., “Scaling of Efficiency with Applied Magnetic Field in Magnetoplasmadynamic Thrusters,” 46th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Nashville, TN, AIAA 2010-2074, 2010. https://doi.org/10.2514/6.2010-7024

[8] Kodys, A. D., and Choueiri, E. Y., “A Critical Review of the State-of-the-Art in the Performance of Applied-field Magnetoplasmadynamic Thrusters,” 41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Tucson, AZ, AIAA 2005-4247, 2005. https://doi.org/10.2514/6.2005-4247

[9] Cassady, L. D., “Lithium-Fed Arc Multichannel and Single-channel Hollow Cathode: Experiment and Theory,” PhD thesis, Princeton Univ., Princeton, NJ, 2006.

[10] Sankaran, K., Jardin, S. C., and Choueiri, E. Y., “Investigation of basic processes in a lithium Lorentz force accelerator through plasma flow simulation,” Electric Propulsion and Plasma Dynamics Laboratory, Princeton Univ., Princeton, NJ, 2005. https://alfven.princeton.edu/research/Ifa

[11] Sankaran, K., Cassady, L., Kodys, A. D., and Choueiri, E. Y., “A Survey of Propulsion Options for Cargo and Piloted Missions to Mars,” Astrodynamics, Space Missions, and Chaos, Vol. 1017, No. 1, 2004, pp. 450-467. https://doi.org/10.1196/annals.1311.027

[12] Kodys, A. D., Emsellem, G., Cassady, L. D., Polk, J. E., and Choueiri, E. Y., “Lithium Mass Flow Control for High Power Lorentz Force Accelerators,” AIP Conference Proceedings, Vol. 552, No. 1, 2001. https://doi.org/10.1063/1.1358027

[13] Choueiri, E., “Scaling of Thrust in Self-Field Magnetoplasmadynamic Thrusters,” Journal of Propulsion and Power, Vol. 14, No. 5, 1998, pp. 744-753. https://doi.org/10.2514/2.5337

[14] Choueiri, E. Y., Chiravalle, V., Miller, G. E., Jahn, R. G., Anderson, W., and Bland J. “Lorentz Force Accelerator with an Open-ended Lithium Heat Pipe,” 32nd Joint Propulsion Conference and Exhibit, Lake Buena Vista, FL, AIAA 96-2737, 1996. https://doi.org/10.2514/6.1996-2737

[15] “2018 Annual Merit Review,” United States Department of Energy, November 2018.

[16] Sullivan, A., Saigal, A., Fragoudakis, R., Zimmerman, M., and Ahmadzadegan, A., “Quantifying the Directionality of Liquid Crystalline Polymers in Extrusion Processes Using an Order Parameter,” Emerging Technologies: Safer Engineering and Risk Analysis: Materials: Genetics to Structures, Vol. 14, March 2016. https://doi.org/10.1115/IMECE2015-52658

[17] Gibson, M. A., Oleson, S. R., Poston, D. I., and McClure, P., “NASA’s Kilopower Reactor Development and the Path to Higher Power Missions,” 2017 IEEE Aerospace Conference, Big Sky, MT, 2017. https://doi.org/10.1109/AERO.2017.7943946