

Review of Magnetoplasmadynamic Thrusters: An Advanced Electric Propulsion System

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Abstract: Magnetoplasmadynamic Thrusters (MPDT) represent a promising advancement in electric propulsion, offering high thrust and efficiency at low mass flow rates. This review explores the fundamental principles of MPDT operation, emphasizing that plasma acceleration results from coupled electric-field energy transfer, current flow, and magnetic-field shaping, with the $\vec{J} \times \vec{B}$ body force providing directed momentum exchange rather than being the sole source of kinetic energy. The classification of MPDTs into self-field and applied-field configurations is discussed, highlighting their distinct performance characteristics. The impact of different propellants on efficiency, thrust, and specific impulse is analyzed, with a focus on lithium and hydrogen, which have demonstrated thruster/discharge-level efficiencies of up to 69%. Key advancements in electrode design, propellant injection methods, and magnetic field optimization are examined to address challenges such as energy losses, plasma instabilities, and electrode erosion. A comparative evaluation with conventional MPDTs show advantages and limitations for deep-space missions and satellite applications. Based on the reviewed studies, the most critical future engineering direction is extending MPDT operational lifetime by reducing cathode/electrode erosion while maintaining stable and efficient high-current plasma acceleration. Therefore, future work should prioritize erosion-resistant electrode materials, improved thermal management, discharge-stability control, and magnetic-field optimization to improve long-duration reliability.

Keywords: Electromagnetic field, Lorentz force, MPD Thruster, Plasma acceleration, Specific Impulse.

1. INTRODUCTION

Electric propulsion has revolutionized space exploration by providing higher fuel efficiency and enabling longer missions with reduced propellant mass [1]. These systems are classified into electrostatic, electrothermal, and electromagnetic types based on the method of accelerating the propellant [2, 3]. Ion thrusters, an example of electrostatic propulsion, achieve high specific impulse, I_{sp} for precise and sustained thrust [4]. Electrothermal propulsion, including resistojets and arcjets, relies on heating the propellant, offering simplicity and reliability. Electromagnetic propulsion, such as Hall Effect Thrusters (HETs) and Magnetoplasmadynamic Thrusters (MPDTs), uses electromagnetic forces to accelerate ionized gases [2].

MPDT operation is governed by coupled electric and magnetic interactions. The discharge electric field and arc voltage supply energy to the charged particles, while the interaction between plasma current and magnetic field produces a $\vec{J} \times \vec{B}$ body force that redirects this energy into directed plasma momentum and thrust. This review discusses the working principles, classifications, and recent advancements in

MPDT technology. A comparative analysis highlights MPDT's advantages over other electric propulsion methods, emphasizing its potential for deep-space missions and heavy satellite applications [1-4].

2. FUNDAMENTALS OF MPDT

2.1. Basic Principle

MPDTs generate thrust by accelerating plasma using electromagnetic forces. A high-current discharge ionizes the propellant, creating plasma composed of free electrons and ions. When an electric current flow through the plasma in the presence of a magnetic field, the resulting Lorentz force propels the plasma, generating thrust [1]. This process enables efficient propulsion in space by leveraging electromagnetic interactions provides the necessary force to accelerate the plasma [2], [3]. Electromagnetic thrusters harness the Lorentz force, resulting from the perpendicular interaction of plasma current density, $\vec{J}$ and magnetic field, $\vec{B}$ in a control volume, V . This electromagnetic force, $\vec{F}_{em}$ is represented by the equation (1)

$$\vec{F}_{em} = \int_{V_1}^{V_2} (\vec{J} \times \vec{B}) dV \quad (1)$$

In MPDTs, the $\vec{J} \times \vec{B}$ force redirects plasma momentum without doing work, since magnetic forces act perpendicular to velocity. Actual energy input (ionization, heating, acceleration) comes from the

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discharge electric field and arc voltage. Thus $\vec{J} \times \vec{B}$ governs directional thrust generation, while electrical energy from the arc supplies the kinetic energy gained by the plasma. In electromagnetic thrusters, an axial Lorentz force is produced by two primary cylindrical configurations:

- a) Radial discharge currents interacting with azimuthal magnetic fields ($J_r B_\theta$).
- b) Radial magnetic fields ($J_\theta B_r$) interacting with azimuthal induced currents.

The pulsed inductive thruster (PIT) and the MPDT, respectively, serve as examples of these concepts [4]. The thruster consists of following key components:

- **Cathode:** The central electrode emitting electrons.
- **Anode:** The outer electrode collecting electrons to complete the circuit.
- **Magnetic Coils:** Generate the necessary magnetic field for plasma acceleration.
- **Propellant Feed System:** Introduces neutral gas into the thruster.
- **Power Supply:** Provides the electric current needed for ionization and acceleration.

Magnetic Coils are used to generate the magnetic field necessary for the Lorentz force generation as shown in Fig. (1). The cathode and anode are designed to withstand high temperatures and ion bombardment, often necessitating the use of materials that are both conductive and durable. The magnetic coils are strategically placed to optimize the magnetic field configuration, ensuring efficient acceleration of the plasma [4].

MPDTs typically operate at low discharge voltages but very high discharge currents. Reported 100-kW-class applied-field MPDT operation includes values up to V and A, while other 100-kW-class studies consider $\sim$ kA. Higher-power designs may reach approximately kA at V. These figures explain why MPDT power-processing units (PPUs) must manage low-voltage/high-current conversion, conduction losses, thermal control, and rapid protection. Two main PPU/power-supply approaches exist, depending on whether the thruster operates in steady or quasi-steady/pulsed mode.

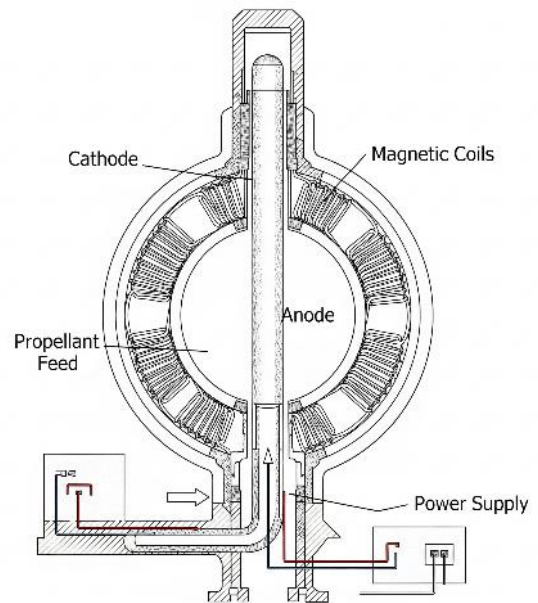


Figure 1: Cutaway schematic of a coaxial MPDT showing the central cathode, annular anode, surrounding magnetic coils, propellant feed, and power-supply connections

For quasi-steady/pulsed power, capacitor banks discharge through a pulse-forming network (PFN) to produce a near-rectangular current pulse. A typical NASA high-power application uses a multi-element Guillemin-type PFN of capacitors and inductors [94,95]. Every PFN includes: (a) a charging/energy-management stage, (b) a high-current closing switch [96], and (c) protection/diagnostics. Large energy stores and inductors needed for high pulse energy and peak current make PFNs heavy and bulky, with non-ideal efficiencies—system studies report losses in the capacitor store, charging unit, and switches [54].

For steady/continuous power, kA-range current with controlled ripple and fast protection is required. Laboratories achieve this by connecting industrial high-current supplies in series/parallel for $\sim$ kA at approximately (100 V), using separate supplies for the applied-field magnet [6,61]. Flight-relevant systems favor higher-voltage buses converted near the load via modular/interleaved converters with low-inductance buswork, addressing conduction losses, EMI/packaging, discharge stability near onset [30], and fault tolerance.

Regardless of topology, MPDTs need auxiliary supplies for cathode heating and arc ignition, with high-voltage pulses used in professional testing [6,61]. Consequently, the PPU/energy-storage subsystem

often becomes the heaviest element and reduces system efficiency relative to discharge-referenced efficiency, particularly for pulsed/PFN systems [54].

2.2. Plasma Formation

MPDTs form and sustain plasma via a high-current, relatively low-voltage arc discharge (typically tens to a few hundred volts), in which most of the voltage is dropped in the near-electrode regions. Ionization is dominated by electron-impact processes in the arc/plume core, fed by thermionic (and locally field-assisted) electron emission at cathode spots and ohmic/arc heating, rather than by “high voltage” breakdown per second [5-7]. Applied axial magnetic field increases electron magnetization and residence time, raising ionization fraction and improving plume collimation and efficiency [6-8]. Plasma instabilities in MPDTs may be grouped into three practical categories. The first category is discharge-attachment or onset-type instability, which appears through high-frequency voltage fluctuations, anode spot formation, unstable current attachment, increased anode fall, and enhanced chamber/electrode erosion. This type is mainly induced by excessive local current concentration, near-electrode voltage drops, unsuitable electrode geometry, insufficiently controlled propellant injection, and thermal non-uniformity near the anode or cathode [7, 9-12, 30]. The second category is electromagnetic or MHD-type fluctuation. In the deuterium MPDT experiments reviewed later in this paper, 300–500 kHz electromagnetic fluctuations were identified as fast Alfvén waves, with phase velocities of 40–80 km/s and a dependence on plasma current/input power [39]. These fluctuations are therefore linked to the discharge current level, applied/self-induced magnetic-field structure, and plasma magnetization. The third category is electrostatic drift-type instability. The 6–7 MHz electrostatic fluctuations observed in the same deuterium experiments were conjectured to be lower-hybrid drift instability [39]. Such drift-type behavior is especially relevant in magnetized MPDT plasmas where Hall effects, density gradients, electron-pressure effects, and cross-field current transport influence the current path [3, 5, 16-19].

The inducing factors for these instability classes are therefore not independent. They include discharge current/input power, applied magnetic-field strength and topology, local Hall parameter, propellant mass-flow rate and injection distribution, electrode geometry, anode/cathode heat loading, and density reduction near cooled anode regions [3, 5-8, 13-18, 26, 30].

Instability suppression should therefore be approached at both plasma and system levels. Existing control methods include optimizing the applied axial magnetic field and magnetic-nozzle configuration to improve confinement and plume collimation; selecting electrode geometries and propellant-injection schemes that avoid localized current attachment and reduce anode power deposition; controlling mass-flow rate to maintain a stable ionization and acceleration region; reducing power-supply ripple and using low-inductance, protected high-current power-processing architectures; and applying thermal-management and erosion-resistant electrode strategies to limit heat-flux-driven attachment changes [6-8, 13-15, 30-32, 42, 51]. In addition, MHD and PIC/MCC models that include Hall physics and instability-driven current-path changes are required to predict and control these modes more reliably [16-18].

3. CLASSIFICATION

3.1. Self-field MPDT

The self-induced magnetic field and the applied arc current combine to drive propulsion in a MPD thruster that does not have an externally applied magnetic field. A magnetic field is created when the current passes through the plasma, and this field interacts with the current to provide an electromagnetic acceleration force. The force that propels plasma can be broken down into two separate parts: $J_z B_\theta e_r$ and $J_r B_\theta e_z$. In this case, e stands for the unit vector, B for the self-induced magnetic field, and J for the applied arc current. The resulting Lorentz-force density can therefore be expressed as: $F = J \times B = -J_z B_\theta e_r + J_r B_\theta e_z$. The radial component, $-J_z B_\theta$, produces plasma pinching and compression. The resulting gas dynamic pressure distribution can contribute indirectly to the axial thrust. In contrast, the axial component, $J_r B_\theta$, directly accelerates the plasma in the axial direction and therefore provides a direct contribution to thrust [23, 112].

On a macroscopic scale, the thrust, T can be expressed as given in equation (2).

$$T = \frac{\mu_0 I^2}{4\pi} (C + \log \frac{r_a}{r_c}) \quad (2)$$

where r_a and r_c is the anode and cathode radius in m, respectively, I the total arc current and μ_0 is the magnetic permeability in N/A². C is the dimensionless constant depending on the current attachment patterns on the electrodes and is somewhat less than unity [20].

It is also referred as configuration factor of the electromagnetic acceleration caused by the self-field denoted by ' ζ ' [21].

Fig. (2), which depicts the gas injector plates and plasma jet, provides a more accurate 3D figure for better comprehension. Radial current is concentrated upstream and downstream end of the cathode [22]. Notably, the experimental findings show that the thrust depends more on the current distribution at the cathode surface than on the local plasma characteristics or the current distribution inside the plasma [23, 24].

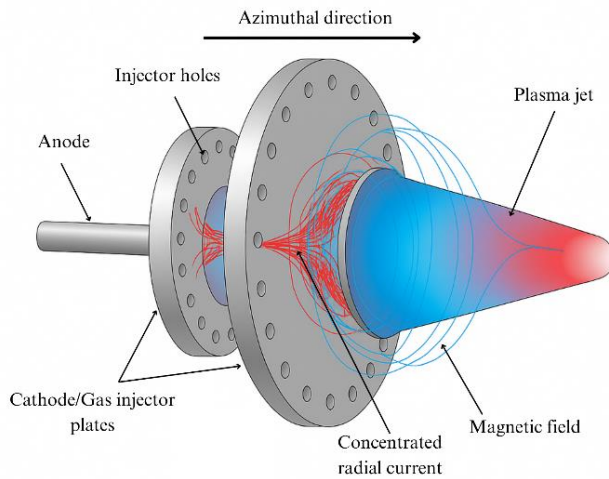


Figure 2: Self-field MPDT forming plasma jet.

In an MPD engine without an external magnetic field, electron motion is governed by the local Hall parameter ($\omega\tau$) in which ω and τ are the cyclotron frequency and successive collision time, respectively. Near the cathode tip, high Hall parameters cause electrons to spiral axially, while lower Hall parameters downstream allow collisional conduction across induced magnetic field lines. At low currents, the ion gyro radius (r_L) exceeds the electrode gap, preventing ion bombardment of the cathode. Ion orbit turning, aided by a large magnetic field, generates thrust through axial turning and outward spiraling of electrons, leading to current conduction across the gap. The arc voltage accelerates ions to their exit velocity, driving thrust through complex electromagnetic interactions [3, 25]. Heiermann and Auweter-Kurtz in 2005 [26], presents a finite volume code for predicting thrust in MPD self-field thrusters, verified by experimental data. It finds that electron pressure diffusion drives the arc out of nozzle-type thrusters, and density drops near water-cooled anodes cause instabilities. The code aids in optimizing thruster design for maximum thrust and efficiency.

3.2. Applied-field MPDT

Compared to running without an external magnetic field, a MPD engine working under a strong externally applied magnetic field usually needs a higher arc voltage and runs at a lower current. Fig. (3) defines the cylindrical coordinates and illustrates the decomposition of the electromagnetic force $\vec{J} \times \vec{B}$. The radial part improves confinement, the axial part produces thrust, and the azimuthal part generates swirl that can be converted to axial momentum in an applied-field configuration [1, 3].

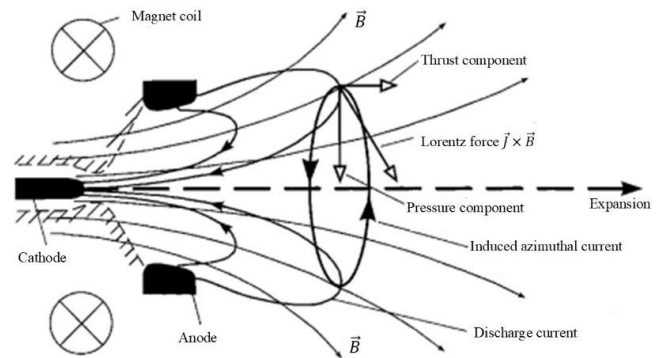


Figure 3: Cylindrical coordinate system and decomposition of the Lorentz force in an Applied-Field MPDT [3].

3.2.1. Acceleration Mechanisms of Applied-Field MPDTs

In an applied-field MPDT, the externally imposed magnetic field changes the current distribution and introduces additional acceleration mechanisms beyond the self-induced electromagnetic acceleration found in self-field MPDTs. The Lorentz-force components can be interpreted in cylindrical coordinates: the axial component contributes directly to thrust, the radial component supports plasma compression and confinement, and the azimuthal component produces plasma rotation. In applied-field operation, the most important acceleration mechanisms are Hall acceleration and swirl acceleration.

A. Hall Acceleration

Hall acceleration arises from the interaction between the applied magnetic field, the discharge electric field, and the induced cross-field motion of charged particles. In a magnetized plasma, electrons are more strongly affected by the magnetic field than ions, and their motion produces Hall currents. These currents interact with the applied magnetic field and generate an axial electromagnetic force component that contributes to thrust. The strength of this

mechanism depends on the electron Hall parameter, electrical conductivity, mass flow rate, current density, and magnetic-field strength. Hall acceleration is therefore important in determining the thrust and specific impulse of applied-field MPDTs, especially when the ionization fraction and magnetic-field configuration support efficient current coupling. The theoretical framework for Hall acceleration is grounded in the generalized Ohm's law, which for fully ionized plasma is expressed as given in equation (3) [3]:

$$\vec{J} = \sigma_0(E + (\vec{u} \times \vec{B}) - (\beta \vec{J} \times \vec{B}) + \beta \nabla P_e) \quad (3)$$

where, $\vec{J}$ is the current density (in A/m²), σ_0 is the electrical conductivity (in S/m), E is the electric field strength (in V/m), $\vec{u}$ is the plasma flow velocity (in m/s), β is the electron hall parameter ($\omega\tau$) and ∇P_e is the gradient of electron pressure (Pa). The Hall effect and diamagnetic effects dominate in this context, with their relative contributions governed by the electron Hall parameter. In MPD thrusters, the acceleration zone was typically modelled as a cylindrical region with radius, R and length, L by Sasoh and Arakawa [5].

B. Swirl Acceleration

Swirl acceleration occurs when the interaction between the discharge current and the applied magnetic field produces an azimuthal Lorentz force. This force gives the plasma rotational kinetic energy. In a diverging magnetic-nozzle configuration, part of this rotational energy can be converted into axial kinetic energy, thereby increasing useful thrust. The effectiveness of swirl acceleration depends on the magnetic-nozzle geometry, applied-field strength, current distribution, and the degree to which azimuthal motion can be converted into directed axial flow. In high applied-field regimes, swirl acceleration can become a dominant contribution to thrust because the azimuthal Lorentz force remains significant even when the Hall-current contribution becomes less effective [3, 21]. Fig. (4) represents the schematics to understand both the accelerations.

If the azimuthal kinetic energy is fully converted into axial energy in the nozzle's divergent section, the maximum thrust enhancement from swirl acceleration is given by equation (4).

$$T_{swirl} = BI \left(\frac{r_a}{r_c} - 1 \right) \sqrt{\frac{r_a^2 + r_c^2}{2}} \quad (4)$$

The swirl-thrust expression is applicable to applied-field MPDT operation in which the imposed magnetic

field produces an azimuthal Lorentz-force component and the resulting plasma rotation is partially converted into axial momentum by a diverging magnetic nozzle. This expression represents an idealized upper-limit estimate because it assumes efficient conversion of azimuthal kinetic energy into axial kinetic energy in the nozzle. It should therefore be interpreted as a mechanism-level estimate for swirl acceleration, rather than a complete thrust model for all MPDT operating regimes.

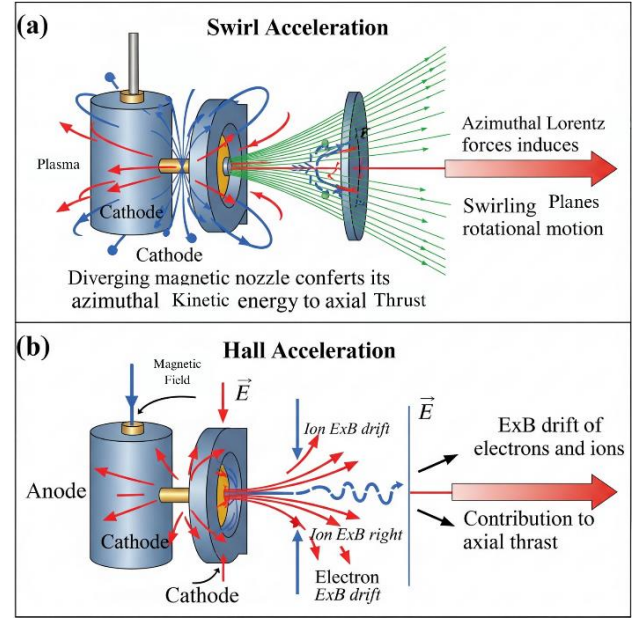


Figure 4: Applied-field MPDT acceleration mechanisms.

Besides Hall and swirl acceleration, Joule heating and self-field effects can also contribute to the overall thrust production. Joule heating increases plasma thermal energy, which may contribute to gas-dynamic expansion, while self-field acceleration remains present when the discharge current generates a significant magnetic field of its own. However, in applied-field MPDTs, these contributions are generally treated as secondary compared with the Hall and swirl mechanisms. Therefore, the applied-field acceleration process is best understood as a coupled mechanism in which electric-field energy deposition, Hall-current formation, plasma rotation, magnetic-nozzle conversion, and self-field effects act together to determine the final thrust and efficiency [3, 27].

Krulle *et al.* [3], finds Applied field MPDTs are suitable for large satellites due to their high specific impulse and compact design, and needs space-based experiments for qualification. The International Space Station offers a platform for this. Joint international

efforts are essential for their development and future space applications. Sasoh and Arakawa in 1995 [28], presented the derived general thrust formula for applied-field MPDT that links controllable operation parameters to thrust, highlighting the roles of generalized Hall, swirl, and self-magnetic accelerations.

Table 1 compares Self-Field and Applied-Field MPDT, showing Applied-Field MPDTs are more efficient (30%-60%) and use propellant better, making them advantageous for practical space applications.

3.3. Experimental Support and Limits of MPDT Thrust Scaling Laws

The thrust scaling laws of MPDTs are useful only when the discharge remains in a stable operating regime. For self-field MPDTs, the macroscopic thrust expression indicates that the electromagnetic thrust contribution scales approximately with the square of discharge current, $T_{self} \propto I^2$, when the electrode geometry, current attachment pattern, and configuration factor remain nearly constant. This relationship follows from the interaction between the discharge current and its self-induced magnetic field. However, the reviewed experimental observations show that the thrust is not governed by discharge current alone. The current distribution at the cathode

surface has a stronger influence on thrust than the local plasma characteristics or the current distribution inside the plasma [23, 24]. Therefore, the I^2 scaling should be interpreted as a stable-attachment scaling law. If the cathode attachment pattern, anode fall, or effective acceleration region changes, the configuration factor is no longer constant and the measured thrust may deviate from the ideal quadratic trend.

For applied-field MPDTs, the main experimental scaling trend reviewed in this paper is that thrust increases approximately with the product of discharge current and applied magnetic field, $T_{applied} \propto I^2$. This is consistent with the applied-field acceleration mechanism, in which externally imposed magnetic field changes the current distribution and supports Hall and swirl acceleration. Mantenieks *et al.* [7] reported that thrust was proportional to discharge current and magnet current in a 100-kW-class AF-MPDT, while Myers [6] reported scaling of 100-kW-class AF-MPDTs in which thrust followed the IB trend and applied-field operation increased efficiency compared with self-field operation. Thus, the reviewed experimental data support the IB scaling as a first-order engineering rule for stable applied-field operation.

These scaling laws break down near critical-current and onset operating conditions. In the pre-onset

Table 1: Comparison of the Key Characteristics of Both Configurations

Parameter	Self-Field MPDT	Applied-Field MPDT
Magnetic-field source	Self-induced magnetic field produced by the discharge current	External magnetic field supplied by magnetic coils or superconducting/HTS magnets
Typical power range	Generally high-power operation, commonly >1 MW for efficient self-field acceleration	Typically, lower power range, about 10–200 kW in 100-kW-class applied-field studies
Efficiency range	About 10–30%	About 30–60%, with higher values reported for optimized applied-field configurations
Thrust characteristic	High thrust capability at high discharge current	Moderate thrust with improved plasma confinement and propellant utilization
Main acceleration mechanism	Self-induced current–magnetic-field interaction, with plasma compression and axial acceleration	Coupled Hall acceleration, swirl acceleration, magnetic-nozzle conversion, and residual self-field effects
Typical propellants discussed in the present review	Argon, hydrogen, lithium, ammonia, and nitrogen, depending on the test campaign and mission requirement	Argon, hydrogen, lithium, ammonia, and other candidate propellants, with applied-field studies commonly emphasizing argon, hydrogen, and lithium
Applicable scenarios	High-power propulsion concepts, cargo/deep-space mission studies, and laboratory high-current MPDT investigations where high thrust is prioritized	Large satellites, deep-space propulsion, station-keeping/orbit-transfer concepts, and systems where improved efficiency and plume control are important
Major limitations	Requires very high discharge current and high power; lower efficiency; stronger electrode erosion risk; onset instability; greater sensitivity to current attachment and cathode/anode heating	Requires external magnet system and associated power/thermal support; added mass and complexity; net spacecraft-level efficiency may be lower when magnet, cryogenic, PPU, and auxiliary power are included
Engineering advantage	Simpler magnetic-field hardware because no external magnet is required	Better plasma confinement, higher efficiency, improved propellant utilization, and broader stable operating envelope

regime, increasing current or magnetic field improves acceleration in a predictable manner. Near onset, however, high-frequency voltage fluctuations, anode spot formation, unstable current attachment, enhanced chamber/electrode erosion, and increased anode power deposition appear [30]. Under such conditions, additional input power is increasingly converted into electrode heating, ionization/frozen-flow losses, plume divergence, or instability-driven current-path changes rather than useful directed kinetic energy. Consequently, the effective self-field configuration factor in $T_{self} \propto I^2$ and the effective coupling factor in $T_{applied} \propto IB$ are no longer constant. This causes the measured thrust and efficiency to depart from the ideal scaling laws.

The critical-current condition should therefore be treated as a practical boundary between stable scaling and unstable operation. Numerical studies around critical current and models including Hall physics, MHD behavior, and PIC/MCC transport indicate that current-path changes and plasma instabilities must be considered for reliable prediction near this boundary [16-18, 70]. In engineering terms, the I^2 and IB relations are most reliable below onset, where current attachment, electrode heat loading, propellant flow distribution, and magnetic-field topology remain controlled. Above or near onset, the thruster may still carry higher current, but the proportional increase in useful thrust is reduced and may be accompanied by stronger erosion and lower efficiency.

4. PERFORMANCE AND LIFETIME

4.1. MPDT Performance

MPDT are evaluated primarily through metrics such as thrust efficiency and specific impulse, which depend on accurate measurements of thrust, mass flow rate, discharge voltage, and current. Viscosity, velocity slip, ambipolar diffusion, and area variation significantly impact thruster performance [29]. Thrust efficiency is influenced by several factors including electrode power dissipation, frozen flow losses, and onset phenomena such as high-frequency voltage fluctuations, anode spot formation, and enhanced chamber erosion. Efficiencies for MPDT vary significantly with the type of propellant used. In the present review, literature values for efficiency are considered as the efficiency of the thruster unless the original authors clearly indicate otherwise. Typically, the term “thruster efficiency” refers in electric propulsion to the ratio of the jet power to the power supplied to the discharge as shown in equation (5).

$$\eta_{th} = \frac{P_{jet}}{P_{dis}} = \frac{\frac{1}{2}\dot{m}v_{eff}^2}{VI} = \frac{T^2}{2\dot{m}VI} \quad (5)$$

where $P_{dis} = VI$ is the discharge electrical power. For engineering trade studies, a broader system efficiency should be used as given in equation (6).

$$\eta_{sys} = \frac{P_{jet}}{P_{dis} + P_{aux}} \quad (6)$$

To quantify the impact of these auxiliary loads on net spacecraft performance, we can define an auxiliary power fraction, $\alpha = \frac{P_{aux}}{P_{dis}}$. Substituting this into Equation (6) yields a qualitative parametric relationship: $\eta_{sys} = \frac{\eta_{th}}{1+\alpha}$. Recent system-level AF-MPDT design work has considered auxiliary-power requirements as low as approximately 5% of the propulsion-system power for 20-kW-class HTS architectures and has used a 10% margin as a conservative preliminary design assumption [109].

Table 2: First-Order Sensitivity of Integrated Propulsion-System Efficiency to Auxiliary Power

α (%)	η_{th} for Lithium (%)	η_{th} for Hydrogen (%)
0	69	43
5	65.7	41
10	62.7	39.1
20	57.5	35.8
30	53.1	33.1

Note: The 5% and 10% cases are motivated by recent AF-MPDT system-level design assumptions, whereas 20% and 30% are included as parametric sensitivity cases and should not be interpreted as measured or typical auxiliary-power requirements.

Using 5%, 10%, 20%, and 30% auxiliary-power fractions as illustrative Table 2, the upcoming discussion which reports 69% lithium thruster efficiency identified by Sovey and Manteniaks [34] corresponds to approximately 65.7%, 62.7%, 57.5%, and 53.1% system efficiency. Similarly, the further reported 43% hydrogen efficiency corresponds to approximately 41.0%, 39.1%, 35.8%, and 33.1%. The results indicate that even moderate auxiliary-power requirements can produce a measurable reduction in the net system efficiency. These values are intended as first-order sensitivity estimates rather than universal penalties because the auxiliary-power requirement depends strongly on the magnet technology, cryogenic architecture, PPU efficiency, thermal-control requirements, and propellant-feed system. Recent experimental and system-integration studies of HTS and cryogenically cooled AF-MPDTs further

demonstrate that these subsystem requirements are strongly architecture-dependent [110, 111].

In P_{aux} the power demand of magnetic-field support equipment such as cooling systems for super-/HTS coils, magnet power electronics, propellant feed/control and other subsystems is included. As auxiliary power budgets are often incomplete and only discharge power is reported for efficiency in many experimental papers, η_{sys} cannot be consistently reconstructed across studies and hence is not claimed unless specifically mentioned. Throughout the present review, efficiency values reported for MPDTs are treated as thruster or discharge-level efficiencies unless the original source explicitly includes auxiliary power terms. For superconducting or HTS-assisted MPDTs, this distinction is especially important because cryocooler power, magnet-support electronics, thermal-control hardware, power-processing losses, and propellant-feed/control power can reduce the net flight-system efficiency. Therefore, HTS-assisted performance improvements discussed is not the net spacecraft-level efficiency gains unless these auxiliary power terms are included in the original analysis.

For instance, efficiencies between 10 to 35% have been observed with propellants like Argon, Ammonia, and Nitrogen, corresponding to specific impulses between 1000 to 4500 seconds. Higher efficiencies, reaching up to 43% and 69%, have been achieved with hydrogen and lithium, respectively, at around 5000 seconds specific impulse. These values should be read as reported thruster or discharge-level efficiencies unless the original source explicitly includes auxiliary power terms. Therefore, they should not be directly interpreted as net spacecraft-system efficiencies, because the latter may be reduced by PPU losses, magnet-support power, refrigeration or cryocooler demand, thermal-control power, and propellant-feed/control power. The largest power loss mechanism in thrusters operating below 2-3 MW is anode power deposition, which can account for up to 80% of input power in low-power steady-state thrusters. Techniques such as injecting propellant near the anode surface and applying magnetic fields are employed to reduce this power loss and enhance efficiency. Accurate performance evaluation requires high-fidelity diagnostics capable of determining local propellant species, charge-state, densities, and velocities. Test stands must handle currents exceeding 10 kA, and vacuum facilities must maintain operational pressures below 0.04 Pa to ensure reliable evaluation of thruster configurations [2, 30].

In the case of steady-state MPDT characterization, pendulum-type thrust stands are often the first choice since they allow separation of low thrust measurement from the high weight of the thruster; typical configurations are hanging pendulum, inverted pendulum, and torsional balances [101]. Inverted-pendulum configurations have been the most common for high-power MPDTs because of their stature in terms of high load capacity and high resolution, and the MPD thrust stands developed by NASA featured an inverted-pendulum structure with conduction-cooled electrical power flexures to eliminate thermal drift and current feeding tares [102]. One of the practical measures to lessen thermal errors in the course of long-duration firings is to utilize the combination of: (i) active thermal management of stand structural members and current leads such as water cooling/heat exchangers, (ii) remote leveling to regulate inclination drift under vacuum, and (iii) in-situ calibration before/during/after firing. The proposed calibration methods include applying known forces along the thrust axis with the pulley-and-weights method (remote weights/monofilament line) and observing deflection with displacement sensing like LVDT or non-contact optical sensors [57,102,103]. At last, null-type (force-rebalanced) operation with in-situ calibration is the best option when possible since it minimizes the effect of displacement-dependent parasitic forces and thus drifts better; the uncertainty studies of inverted-pendulum thrust stands have shown inclination drift and displacement drift to be the major contributors, highlighting the need for continuous monitoring, and repeatable in-situ calibration in order to conduct steady-state EP testing [101,104].

4.2. MPDT Lifetime

The lifetime of MPDT, particularly for long-duration missions to the Moon or Mars, is constrained mainly by cathode erosion. Estimates suggest that thrusters should last between 2000 to 8000 hours with total impulses ranging from 1×10^8 to 1×10^9 N-s. Historical cathode erosion rates have exceeded acceptable levels for sustained high-current operation, often surpassing $0.003 \mu\text{g/C}$. Recent advances in cathode composition, microstructural control, and magnetic-field/current-attachment management indicate significant potential for reducing both transient spot-driven erosion and steady-state evaporation. Under optimal conditions, cathode erosion rates have been reduced to nearly $4 \times 10^{-5} \mu\text{g/C}$, approaching the required levels for long-term missions. Long-duration tests of steady-state MPDT have shown promise, with

a notable 1969 test demonstrating 500 hours of operation at low power (33 kW), producing a total impulse of about 1×10^6 N-s. Further research and development aim to enhance the lifetime of steady-state MPDT by a factor of 700 through optimized design and operating conditions, making them more viable for long-term space missions. The primary life-limiting factor remains cathode erosion, particularly in pulsed thrusters where cathode mass loss is significantly higher than in steady-state thruster [2, 30].

Cathode lifetime in MPDTs is controlled by coupled plasma–material interaction and thermal effects. During ignition and transient operation, energetic ion bombardment and localized cathode-spot attachment can produce intense sputtering and localized material removal, while at elevated steady-state temperatures thermal evaporation becomes an important, and in some operating regimes dominant, mass-loss mechanism. Experimental and numerical investigations of tungsten MPDT cathodes have shown that start-up erosion can be substantially higher than steady-state erosion because of the stronger ion bombardment and transient cathode conditions [113]. Localized arc-spot attachment can further increase erosion by concentrating current over small surface regions and producing extreme local heat fluxes [114]. Repeated thermal loading can also modify the cathode microstructure and promote surface blistering, ablation, and other thermally induced damage [115]. Cathode-material selection therefore requires a balance between electron-emission capability and high-temperature durability. A low work function is desirable because it increases thermionic emission and can reduce the cathode temperature required to sustain the discharge, whereas a high melting point, low vapor pressure, resistance to ion bombardment, and microstructural stability are required to limit material loss during long-duration operation. Rare-earth-doped tungsten cathodes provide a practical compromise because the dopant can reduce the effective work function while retaining the refractory tungsten matrix. Recent MPDT studies of WCe and La–W cathodes demonstrate that both composition and microstructural control influence erosion resistance and cathode lifetime [116]. Therefore, future cathode development should optimize not only work function and electron-emission capability but also dopant stability, grain structure, resistance to sputtering and evaporation, and tolerance to localized thermal and arc-attachment stresses.

Literature also suggests some ideas to increase the lifetime of the thruster. Integrating cryogenic cooling

methods in MPDTs could mitigate heat-induced wear on thruster components, enhancing operational lifespan and maintaining performance stability during extended missions [31]. Jet impingement cooling techniques, extensively used in gas turbine blades to enhance thermal management and material lifespan, could be adapted to MPDT systems to mitigate thermal stresses on critical components and extend operational durability [32].

5. PROPELLANT SELECTION FOR MPDT

Propellant selection for MPDTs focuses on efficiency, lifespan, and storage, with lithium and alkali metals enhancing ionization and reducing degradation [10, 33]. MPDT usually use propellants like hydrogen, argon, or ammonia, accelerate ionized plasma by applying the Lorentz force [11]. Condensable propellants like ammonia reduce tank weight but risk electrode damage and spacecraft contamination from reactive vapors and exhaust. Condensable metal propellants such as lithium are a greater integration challenge compared to gas propellants: the spacecraft surfaces may be contaminated by the plume backflow and re-deposition. In MPD systems, the question of lithium deposition in the backflow region is often raised as a major integration concern because the layered material can cover solar arrays, optical payloads/star trackers, thermal radiators, and sensitive surfaces, thus [93] degrading power generation, optical throughput, and thermal control margins. These hazards mean penalties for the whole system beyond the thruster alone. Generally, lithium (and similar metals) must be kept and supplied in a heated state (liquid/vapor), so the propulsion system may need a heated reservoir and heated feed lines for the prevention of freezing/condensation and feed-line clogging, besides heating and insulation control—this raising power demand, complexity, and failure modes of the spacecraft [91, 92]. Therefore, the use of lithium might require additional spacecraft accommodations like thruster location restrictions, “keep-out” angles for arrays/optics, protective shielding/coatings, contamination budgeting, and operational constraints (duty-cycling, pointing restrictions). Thus, even though lithium can provide high thrust efficiency provided the right operating conditions, its use must be evaluated at the spacecraft-system level, thus trading propellant efficiency against contamination risk, additional thermal hardware, and mission/payload sensitivity [93].

While hydrogen can provide very high I_{sp} in MPDT operation, its cryogenic storage and low volumetric

density can impose a major spacecraft-integration penalty for size-limited platforms. Liquid hydrogen (LH_2) must be stored near its boiling point of about 20 K, which requires vacuum-compatible cryogenic tanks with multilayer insulation, careful thermal isolation, and boil-off management [105]. Even with modern approaches, long-duration cryogenic storage in space often requires active thermal control/zero-boil-off concepts, adding power and hardware mass [106]. From a packaging standpoint, LH_2 has a density of about 70.8 kg/m^3 near 20 K, which is nearly 6 times lower than liquid methane ($\sim 420\text{--}430 \text{ kg/m}^3$) and about 9–10 times lower than liquid ammonia ($\sim 682 \text{ kg/m}^3$) [107,108]. Therefore, for a fixed propellant mass m_p , the required tank volume scales as $V_t \approx m_p/\rho_p$, meaning LH_2 can demand substantially larger tanks and insulation volume than denser propellants, often offsetting part (or all) of the theoretical I_{sp} advantage on spacecraft that are constrained by envelope, packaging, and thermal design. In this review, hydrogen is therefore best interpreted as a high- I_{sp} option whose suitability depends strongly on mission duration, allowable tank volume, and the availability of cryogenic thermal management at the spacecraft system level. Table 3 discusses all the propellants discussed for its efficiency, I_{sp} , storage, integration risk, power, mission, and limitations.

From an engineering-selection viewpoint, lithium and hydrogen provide the highest reported discharge-level performance in the reviewed literature, but both introduce major spacecraft-integration penalties. Lithium offers high reported efficiency but requires heated storage/feed hardware and contamination-control measures. Hydrogen offers high specific impulse but requires cryogenic storage, large tank volume, insulation, and boil-off or zero-boil-off management. Ammonia offers easier storage and higher density than hydrogen, but dissociation/frozen-flow losses and reactive exhaust must be considered. Argon remains attractive for laboratory testing and cleaner handling, although its efficiency is lower than the best lithium and hydrogen cases. Therefore, the optimal propellant cannot be selected only from first ionization behavior; it must be selected from a coupled trade-off involving efficiency, specific impulse, feed-system complexity, spacecraft contamination, thermal-control burden, and mission duration.

Due to the lack of a comprehensive theoretical model for MPD engines, selecting the optimal propellant requires experimental comparisons. For

instance, hydrogen and ammonia exhibit varying degrees of ionization efficiency, with hydrogen potentially as low as 5% and ammonia ranging from 2% to 20%. Sovey and Mantenieks in 1991 [34] observed relatively high reported efficiencies of 0.43–0.69 with hydrogen and lithium propellants at specific impulses around 5000 s. In the present review, these values are treated as thruster/discharge-level efficiencies unless auxiliary power terms are explicitly included in the original source; therefore, they should not be used directly as net system-efficiency values for spacecraft-level design. However, these efficiencies should not be interpreted as fixed values determined only by the element itself. The propellant influences MPDT efficiency through coupled factors such as ionization behavior, molecular or atomic mass, dissociation and frozen-flow losses, electrode interaction, and compatibility with the discharge current and magnetic-field configuration. Hydrogen-storage materials and hydrogen isotopes should be interpreted as research-stage options rather than mature MPDT propellants in the present review. Lithium borohydride (LiBH_4) is attractive because of its high gravimetric and volumetric hydrogen-storage capacity [35, 36]. However, its direct MPDT application is limited by practical engineering bottlenecks, including the need to reduce hydrogen desorption temperature, control hydrogen release rate, prevent feed-line thermal instability, and demonstrate stable plasma operation using the released species. Destabilizing additives may improve desorption behavior [36], but direct MPDT performance data are still required before LiBH_4 can be treated as a validated propellant option. Ammonia, as a hydrogen-containing fuel is a potential propellant for MPDTs due to its combustion efficiency, storage advantages, and clean energy potential [37]. Ammonia, with its high energy density and ease of liquefaction at low pressures, offers potential as a sustainable MPDT propellant [38].

Chai *et al.* in 2024 [39], observed 300–500 kHz electromagnetic fluctuations in the MPDT with deuterium gas, identified as fast Alfvén waves at Korea Atomic Energy Research Institute (KAERI), Daejeon, South Korea. The phase velocities of these waves ranged from 40 to 80 km/s and were proportional to the plasma current/input power. Additionally, they observed 6–7 MHz electrostatic fluctuations, which they conjectured to be a lower-hybrid drift instability. These observations are important not only for deuterium or hydrogen-isotope propellant selection, but also for instability classification: the 300–500 kHz band

Table 3: Engineering Comparison of MPDT Propellants Discussed

Propellant	Reported Efficiency/ Performance in Reviewed Studies	Specific Impulse Range Discussed	Storage and Transportation Difficulty	Contamination / Integration Risk	Applicable Power Range / Operating Context	Suitable Mission Scenarios	Main Limitations
Lithium	High reported thruster efficiency; lithium has been reported with efficiency up to about 69% at around 5000 s specific impulse [34]	Around 5000 s in the reviewed high-efficiency results	High; lithium or similar metal propellants require heated reservoirs and heated feed lines to avoid freezing, condensation, and clogging [91, 92]	High; lithium backflow/re-deposition can contaminate solar arrays, optical payloads/star trackers, radiators, and sensitive spacecraft surfaces [93]	More suitable for high-efficiency applied-field or advanced MPDT concepts where spacecraft-level contamination control is acceptable	Deep-space or high-performance missions where high discharge-level efficiency is prioritized	Contamination control, heated feed system, added power demand, thermal hardware, duty-cycle/pointing restrictions, and spacecraft accommodation penalties
Hydrogen / LH2	Hydrogen has been reported with efficiency around 0.43 at approximately 5000 s specific impulse [34]	Around 5000 s in the reviewed high-efficiency results	Very high; LH2 must be stored near 20 K, requiring cryogenic tanks, multilayer insulation, thermal isolation, and boil-off management [105, 106]	Low chemical contamination compared with lithium, but high system-level thermal and packaging burden	Suitable where high specific impulse is required and cryogenic storage is acceptable	High-Isp deep-space or high-energy missions with available cryogenic thermal management	Low volumetric density; LH2 density near 20 K is about 70.8 kg/m ³ , requiring much larger tank volume than denser propellants; active thermal control may reduce net system benefit [105–108]
Ammonia	Ionization efficiency reported in the draft as 2–20%; identified as a hydrogen-containing candidate with storage advantages [37, 38]	Included in the general MPDT range of 1000–4500 s for lower-to-moderate efficiency propellants	Moderate; easier liquefaction and storage advantages compared with LH2	Moderate; condensable/reactive vapors and exhaust can cause electrode damage and spacecraft contamination	Candidate for systems where storage simplicity and hydrogen content are useful	Missions where easier storage, compact tanks, and reduced cryogenic burden are important	Dissociation and frozen-flow losses may reduce performance; reactive exhaust and electrode compatibility must be considered
Argon	Argon thrusters have demonstrated efficiencies of about 0.30–0.34 at specific impulses of 2400–2800 s	2400–2800 s in the reviewed argon results	Low to moderate; gas handling is simpler than heated lithium or cryogenic hydrogen	Low contamination risk compared with lithium and condensable/reactive propellants	Common in laboratory applied-field and self-field MPDT experiments, including 100-kW-class studies	Ground testing, baseline experiments, and missions where clean handling and stable operation are prioritized over maximum efficiency	Lower efficiency than lithium/hydrogen high-performance cases; lower propellant efficiency relative to optimized advanced propellants
Deuterium / hydrogen isotopes	Deuterium MPDT experiments showed 300–500 kHz electromagnetic fluctuations identified as fast Alfvén waves and 6–7 MHz electrostatic fluctuations conjectured as lower-hybrid drift instability [39]	Not established as a performance range in the reviewed draft	Similar hydrogen-isotope handling issues; practical storage and system integration require further evaluation	Main issue is not contamination, but plasma fluctuation/EMC coupling risk in kHz–MHz bands	Research-stage propellant option for studying hydrogen-isotope plasma behavior	Diagnostic and research-oriented MPDT studies; possible future hydrogen-isotope propulsion research	Instability behavior, conducted/radiated emissions, and lack of mature spacecraft-level performance data
Lithium borohydride, LiBH ₄	Discussed as a promising hydrogen-storage candidate due to high gravimetric and volumetric hydrogen storage capacity [35, 36]	Not reported as a demonstrated MPDT specific impulse in the reviewed draft	Potentially high; practical use depends on reducing desorption temperature and controlling release behavior	Not fully established in the reviewed draft; system-level contamination and by-product issues require further study	Research-stage storage/propellant concept rather than a demonstrated MPDT propellant in the present review	Future hydrogen-storage-assisted propulsion concepts	Application bottlenecks include desorption-temperature reduction, feed-system complexity, thermal control, and lack of direct MPDT performance validation

represents an electromagnetic/MHD-type mode, while the 6–7 MHz band represents an electrostatic drift-type mode. Therefore, propellant selection must be considered together with discharge current, magnetic-field strength, Hall effects, and PPU–thruster stability control. The reported electromagnetic (300–500 kHz) and electrostatic (6–7 MHz) fluctuation bands can also be interpreted from a spacecraft-integration standpoint

as potential sources of conducted and radiated emissions. Even when the instability is primarily a plasma phenomenon, the associated current/voltage oscillations can couple into the propulsion harness and structure through parasitic capacitance/inductance, producing spectral lines (and harmonics) in the kHz–MHz range. This is relevant because kHz–MHz emissions fall within common conducted-emission

verification bands used in aerospace EMC practice like 10 kHz–10 MHz in CE102-style testing and can also overlap with HF receiver bands (3–30 MHz), depending on the spacecraft communications architecture. Practical mitigation is therefore typically addressed at PPU–thruster integration level: (i) minimizing loop areas and using twisted/shielded conductors for high-current paths, (ii) feedthrough capacitors, LC/ π filters, and common-mode chokes at the PPU output to limit conducted noise, (iii) enclosure shielding and controlled bonding/grounding to reduce common-impedance coupling, and (iv) harness routing/separation from sensitive RF front-ends and sensor cabling. Space-system EMC requirements and verification are commonly handled within established standards. Prior electric-propulsion integration campaigns show that EMC testing and subsystem-level mitigation can be effective when treated early in the design loop; similar verification would be required for MPDT systems operating with strong instabilities or high-current switching [99,100].

Argon thrusters have demonstrated efficiencies of 0.30–0.34 at specific impulses of 2400–2800 seconds, while other propellants have shown lower efficiencies of 0.10–0.30 at specific impulses of 1000–4500

seconds. However, extended tests of MPDT have demonstrated total impulse capabilities far below the requirements for cargo vehicle propulsion. Haag *et al.* in 2006 [40], runs a simulation program for applied field MPDT at Institute of Space Systems (IRS), University of Stuttgart, Germany, which enables detailed analysis of argon plasma flows, utilizing a finite volume method on adaptive meshes. The program shows higher stability and better performance with anomalous diffusion. Additionally, new propellants are selected to address particular operating constraints such as ionization instabilities and temperature control [41]. Propellants with lower first ionization potentials are desirable to minimize ionization and dissociation losses, cooling losses, Frozen-flow losses, etc. although the choice depends on balancing reactivity, handling ease, and overall performance objectives. Lithium-based propellants, for example, have proven to have excellent cooling capabilities and work well with cutting-edge superconducting magnets, greatly increasing system reliability [42]. The molecular weight of the propellant also plays a crucial role, affecting the achievable specific impulse, especially under temperature constraints near the engine's physical limits [3]. Table 4 represents information on energy loss for the propellants.

Table 4: Available Energy-Loss Information for Propellants Discussed

Propellant	Quantitative Information Available in Reviewed Sources	Ionization-related Observation	Dissociation / Frozen-flow Consideration	Main Unquantified Loss Terms Requiring Further Data
Lithium	Reported efficiency up to about 69% at around 5000 s specific impulse [34]	Low ionization and alkali-metal behavior are beneficial for efficiency, but the reviewed draft does not provide a separate ionization-loss percentage	Not a molecular propellant; dissociation loss is not the dominant concern compared with feed/contamination issues	Electrode power deposition, plume divergence, frozen-flow/charge-state losses, contamination-related system penalties
Hydrogen	Reported efficiency around 0.43 at around 5000 s specific impulse [34]; ionization efficiency may be as low as 5%	Ionization behavior can strongly limit efficiency	Molecular/atomic processes and frozen-flow losses may be important, but the reviewed draft does not provide a separate dissociation-loss fraction	Ionization loss, dissociation loss, frozen-flow loss, cryogenic storage power, tankage and boil-off penalties
Ammonia	Ionization efficiency reported as 2–20%; identified as a storage-friendly hydrogen-containing candidate [37, 38]	Ionization efficiency varies widely	Dissociation and frozen-flow losses are expected to be important because ammonia is molecular and hydrogen-containing	Ionization/dissociation split, frozen-flow fraction, electrode interaction, reactive exhaust effects
Argon	Reported efficiency 0.30–0.34 at 2400–2800 s specific impulse	Noble-gas operation provides cleaner handling but does not reach the highest lithium/hydrogen efficiency values in the reviewed data	No molecular dissociation loss; frozen-flow and ionization/charge-state losses still require diagnostics	Ionization fraction, charge-state distribution, plume divergence, anode deposition
General MPDT operation	For thrusters below 2–3 MW, anode power deposition can account for up to 80% of input power	Ionization losses contribute to reduced efficiency	Dissociation and frozen-flow losses depend strongly on propellant species	Local species, charge-state, density, velocity, and electrode heat-flux measurements are required for full loss closure

Note: Exact per-propellant percentages to ionization, dissociation, and frozen-flow losses are not being assigned when such values are not reported in the reviewed sources. A complete propellant-resolved energy-loss budget would require diagnostics of local species, charge states, density, velocity, plume divergence, and electrode power deposition under the same operating conditions.

6. ADVANCEMENTS IN MPDT

1986–1991: Foundational performance, models, and diagnostics: Early theoretical and experimental work established baseline capabilities and limits: with argon propellant a theoretical maximum thrust efficiency of 50.65% was identified, and quasi-steady, multi-megawatt MPD engines demonstrated 38% thrust efficiency with ~ 4000 s specific impulse [43]. Temperature monitoring and control techniques were introduced to keep thrusters within safe operating ranges and to improve experimental precision [44]. High-power testing reported specific impulses of 2000–7000 s at nearly 40% efficiency and >500 kW power levels [45].

2006–2007: Hydrogen compatibility and system-level readiness: The materials challenges of using hydrogen in space power and propulsion plants—such as degradation and penetration—were analyzed, with recommendations for low-permeability materials and robust coatings to improve longevity and efficiency at extreme temperatures [46]. Building on accumulated progress in efficiency, empirical scaling laws, performance and lifetime datasets, numerical models, and erosion drivers, modern MPDTs were judged better suited for applications such as satellite station maintenance and deep-space exploration [47].

2020–2025: Contemporary advances in performance, scaling, thermal management, and small-sat adaptability: Recent work has improved anode geometries and propellant injection for performance gains; attention to plasma microturbulence has aided scaling; and enhanced simulation environments now provide more accurate predictions to guide design and operations. Erosion-reduction has been pursued via an axial converging/diverging magnetic field at the cathode. Voronov *et al.* [42] reported that 2G HTS coils can substantially improve MPDT performance, including increased thrust and discharge-level efficiency. However, these values should be interpreted as thruster/discharge-level performance improvements, not as net flight-system efficiency. Operation of superconducting or HTS-assisted MPDTs requires additional auxiliary power and hardware, including cryogenic refrigeration, magnet-support electronics, thermal-control systems, and related power-processing components. Since the reviewed studies do not consistently report cryocooler heat lift, coefficient of performance, magnet-support power, and complete auxiliary power budgets, the corresponding net system

efficiency cannot be reconstructed uniformly. Therefore, present review treats HTS-assisted MPDTs as promising for improving discharge-level performance, while noting that integrated flight-system efficiency would be lower and must be evaluated separately at the spacecraft-system level. Thrust of 850 mN and specific impulse of 3840 s were the results of the trials using hydrogen propellant. They also mentioned 7700 s specific impulse and 25% thrust efficiency at 400 A. This paper views these efficiency numbers as thruster efficiencies calculated on the basis of the discharge power and does not necessarily take into account the electrical power required by cryogenic refrigeration and other auxiliary systems that are essential for the operation of HTS coils. Therefore, the associated system efficiency would be lower and depend on the specific cryocooler heat lift and coefficient of performance, which are not uniformly reported across the experiments reviewed. In parallel, cross-disciplinary methods have informed MPDT subsystems: jet-impingement cooling strategies from turbomachinery for improved heat dissipation [32], unipolar voltage pulses and corona-discharge concepts from electrostatic precipitators for energy-efficient gas/plasma manipulation [48], microcombustion-based power generators (TEGs and thermophotovoltaics) for compact, efficient power and thermal-recirculation insights [49]; and ammonia / ammonia–hydrogen kinetics to inform cleaner, efficient propellant options [38]. Also high-speed combustion studies involving hydrogen can provide useful background on hydrogen mixing and thermal-flow behavior; however, these studies should be treated only as indirect supporting references and not as direct evidence of MPDT propellant performance [89, 90].

A feasible approach to modify the cooling system of a jet-impingement set up to a multi-pulse diode transistor is the concept treating the anode as a double-wall structure. Within the inner anode wall, the direct current coaxial (coaxial) discharge channel geometry is maintained, and the inner wall is a thin plenum/manifold distributing the coolant to an axisymmetrical arrangement of small jets that are directed upwards. The jets strike the back of the hot wall (the wall facing the plasma) and thus create very high locally heat-transfer coefficients; the water is then collected in the surrounding return passages and sent to the outlet. Also, importantly, the jets and the return flow of water are entirely inside the anode body. Consequently, the electromagnetic field distribution and plasma streamline will not be affected by the cooling

flow in a direct manner. The main design requirement is, therefore, to maintain axisymmetric and uniformly thick walls in the anode so that the current attachment and the corresponding $\vec{j} \times \vec{B}$ acceleration are not biased azimuthally. Water-cooled anode designs with internal cooling channels are already widely used in laboratory AF-MPDT hardware; for example, NASA's designs include water-cooling channels in the anode wall, which proves the possibility of embedding flow passages without changing the inner discharge geometry [97,98]. The concept of jet-impingement can be regarded as the improvement of such embedded cooling passages by distributing jets for higher local heat removal while maintaining the unchanged discharge channel. A schematic of the concept is shown in Fig. (5).

Efforts to scale MPDTs for smaller spacecraft include the Compact ECR Plasma Source (CEPS) at IIT Delhi (≈ 50 mN at 600 W) and a two-stage μ CAT+MPDT architecture for tasks such as orbit raising and interplanetary missions. For power-system testing and rapid transients, low-cost photovoltaic emulators (PVEs) with high dynamic response have been developed for small-sat applications [50]. On the cathode side, rare-earth-doped nano-cathodes have significantly improved lifespan and efficiency in applied-field MPDTs; accompanying theory clarified the dependence of thrust on applied-field strength and identified dominant thrust components across mass-flow regimes [51]. Finally, thermal homogenization

concepts, for example, a convergent-nozzle mixer assembly demonstrated in solar-air-tower systems, offer a path to uniform temperature fields; these principles can inform MPDT thermal conditioning to reduce localized thermal stresses [52]. Detailed summary on advancements is listed in Table 5.

7. TECHNICAL COMPARATIVE ANALYSIS OF MPDTs

MPDTs are renowned for having a high thrust-to-power ratio, which qualifies them for extended space missions. Conventional MPDTs, which usually achieve efficiencies of 30 to 50%, are constrained by energy conversion inefficiencies. Particle velocities range from 20 to 50 km/s, while specific impulse values fall between 1500 and 3000 seconds [79]. Typically, mass flow rates fall between 0.1 and 1 g/s. The main causes of these restrictions include ionization and dissociation losses, arcing-induced electrode degradation, and inadequate magnetic field designs that result in plasma leakage and decreased thrust.

Recent developments have considerably resolved these problems. For example, by enhancing ionization and temperature control, the introduction of sophisticated propellants like as lithium and noble gases has raised specific impulse and efficiency. Particle velocities of up to 80 km/s and reported thruster/discharge-level efficiencies of 60–70% have been presented for advanced configurations because

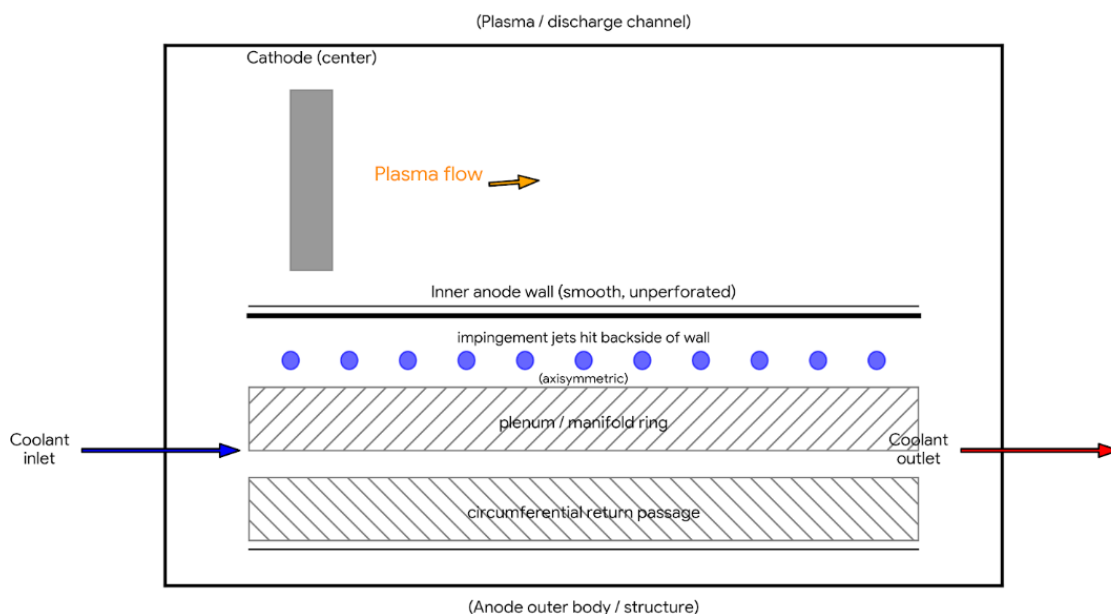


Figure 5: Conceptual jet-impingement cooling implementation within a coaxial MPDT anode.

Table 5: Streamlined MPDT Literature Summary Classified by Power Level and Technical Orientation

Power Level / Regime	Technical Orientation	Representative Studies	Evidence Type	Propellant / System Type	Core Quantitative Result or Conclusion Retained in this Review	Engineering Relevance
Early tens-of-kW applied-field MPDT	Lithium applied-field performance measurement	Fradkin & Roehling [53]	Experimental thrust-stand study	Lithium-fueled applied-field MPD arcjet	Early thrust-stand performance study of a lithium-fueled AF-MPD arcjet; detailed numerical metrics are not listed in the present manuscript	Retained as an early lithium AF-MPDT performance reference
Sub-kW to low-power concepts	Miniaturization and small-sat adaptability	Verma <i>et al.</i> [78]; Ganguli <i>et al.</i> [79]	Experimental / conceptual low-power development	Compact ECR Plasma Source / μ CAT-MPDT hybrid; argon-based concepts	CEPS reported about 50 mN thrust at 600 W; theoretical specific impulse up to about 4000 s and efficiency around 50.65%	Indicates possible MPDT-related scaling toward small spacecraft, but requires further validation for complete MPDT flight systems
10–200 kW steady MPDT	Performance, lifetime, and propellant trade assessment	Sovey & Mantenicks [11, 34]; Myers [62]	Experimental / review-level performance assessment	H ₂ , Li, Ar, and other propellants; self-field and applied-field MPDTs	H ₂ and Li showed reported efficiencies of about 0.43 and 0.69 at around 5000 s; lifetime requirement of about 2000–8000 h; Myers [62] is retained for applied-field MPDT performance with hydrogen and argon	Provides baseline efficiency, lifetime, and propellant trade data for engineering comparison
100-kW-class steady applied-field MPDT	Applied-field performance scaling and stable operating envelope	Mantenicks <i>et al.</i> [7]; Myers [6]; Myers <i>et al.</i> [59]; Boxberger <i>et al.</i> [71]; Albertoni <i>et al.</i> [72]; Boxberger & Herdrich [75]; Wang <i>et al.</i> [76]; Cong <i>et al.</i> [80]	Experimental characterization and operating-protocol studies	Mainly 100-kW-class applied-field MPDTs; argon and other test propellants depending on the campaign	Mantenicks <i>et al.</i> reported efficiency about 20–21%; Myers reported Isp about 1500–4000 s and efficiency about 20–25%, compared with about 5–10% for self-field operation; Wang <i>et al.</i> reported practical test limits including $B_z \leq 0.25$ T, $V \leq 130$ V, $I \leq 800$ A, mass flow rate < 40 mg/s, and vacuum pressure ≈ 0.01 Pa	Supports the engineering trend that applied magnetic field improves efficiency and expands the stable operating envelope; also provides practical 100-kW-class test-reference conditions
Quasi-steady applied-field MPDT	Applied magnetic-field operation and acceleration-process interpretation	Tahara <i>et al.</i> [58]; Tahara <i>et al.</i> [64]	Experimental applied-field / acceleration-process studies	Quasi-steady MPD arcjets with applied magnetic field	Retained for quasi-steady MPD arcjet operation with applied magnetic fields and for acceleration-process interpretation; detailed numerical values are not listed in the present manuscript	Supports the discussion of applied-field operation, near-Earth mission concepts, and acceleration mechanisms
0.5–multi-MW quasi-steady / high-power MPDT	High-power performance, feasibility, and pre-onset operation	Burton <i>et al.</i> [55]; King [56]; Myers <i>et al.</i> [45]; LaPointe <i>et al.</i> [30]; Auweter-Kurtz <i>et al.</i> [63]; LaPointe & Mikellides [67]; Miyazaki <i>et al.</i> [73]	Experimental / feasibility / high-power system studies	Multi-MW, high-power steady-state, quasi-steady, and self-field MPDT systems	Multi-MW operation reported Isp about 2000–7000 s and efficiency near 40%; NASA high-power argon tests approached about 35% efficiency with Isp above 6000–7000 s in pre-onset operation; Miyazaki <i>et al.</i> is retained as a 1-MW-class quasi-steady self-field MPDT study	Demonstrates high-power MPDT capability and highlights the need to distinguish stable pre-onset operation from onset-limited regimes
MW-class design studies	Thermal management and system design	Myers <i>et al.</i> [60]	Design study	2.5-MW applied-field MPDT design using LH ₂ and cryogenic magnet concept	Target values included about 10 kA, 250 V, Isp 3000–7000 s, efficiency goal above 50%, and about 233 kW anode heat load	Useful for system-level design because it shows that thermal management and anode heat removal dominate high-power MPDT feasibility
Permanent-magnet and multi-hollow-cathode MPDT	Magnetic-system simplification and cathode architecture	Koyama <i>et al.</i> [74]	Experimental-development / mission-oriented concept	Steady-state MPD thrusters with permanent magnets and multi-hollow cathodes	Retained for permanent-magnet and multi-hollow-cathode MPD concepts for in-space propulsion and Mars-exploration context; detailed numerical metrics are not listed in the present manuscript	Supports discussion of magnetic-system simplification, cathode architecture, and mission-oriented MPDT development
25–150 kW superconducting / HTS-assisted MPDT	Magnetic-field enhancement and high-efficiency operation	Voronov <i>et al.</i> [42]; Zheng <i>et al.</i> [81]; Zhou <i>et al.</i> [84]; Collier-Wright <i>et al.</i> [82]; Han & Rana [83]; Aftab <i>et al.</i> [86]	Experimental / review / numerical-experimental / system-level studies	Applied-field MPDT with HTS, LTS, or superconducting coils	Reported results include thrust up to 0.85 N and Isp up to 3840 s for 25-kW HTS MPDT; about 4 N thrust and Isp up to 5714 s for 150-kW superconducting systems; about 71% reported efficiency near $B_z \approx 0.5$ T in a 100-kW LTS system; review studies discuss TRL, HTS-module integration, onset/ionization theory, and 250–500 kW application context	Shows the benefit of stronger applied magnetic fields, but reported efficiencies should be interpreted as discharge/thruster-level values unless auxiliary cryogenic, magnet-support, and PPU power are included

Table 5 (contd....)

Power Level / Regime	Technical Orientation	Representative Studies	Evidence Type	Propellant / System Type	Core Quantitative Result or Conclusion Retained in this Review	Engineering Relevance
100-kW-class applied-field MPDT	Plume-current and acceleration diagnostics	Sun <i>et al.</i> [85]; Behnke <i>et al.</i> [87]; Myers <i>et al.</i> [59]	Experimental plume/current diagnostics	Applied-field MPDT plume and current diagnostics	Sun <i>et al.</i> reported 63–82% of discharge current in the outflow and self-field contribution below 5% of electromagnetic acceleration; Behnke <i>et al.</i> detected 70–80% of discharge current in the plume at 1000 mm and ≥90% at 300 mm; Myers <i>et al.</i> is retained for preliminary AF-MPDT plume characterization	Provides direct diagnostic evidence for current closure, plume structure, and acceleration-mechanism interpretation
Numerical modelling and critical-current behavior	MHD, PIC/MCC, onset-related modelling, and model–experiment comparison	Sankaran <i>et al.</i> [16]; Tang <i>et al.</i> [18]; Mayigué <i>et al.</i> [17]; Kubota <i>et al.</i> [70]; Sankaran <i>et al.</i> [68]; Haag <i>et al.</i> [69]; Mayigué [77]	Numerical modelling / model-experiment comparison	Gas-fed and applied-field MPDT modelling	MHD and PIC/MCC studies reproduce key applied-field acceleration behavior; critical-current modelling shows that plasma behavior changes near onset conditions; Sankaran <i>et al.</i> [68] is retained for simulated flowfields compared with experimental measurements; Haag <i>et al.</i> [69] is retained for AF-MPDT design supported by simulations; Mayigué [77] is retained for MHD/upwind numerical investigation	Supports interpretation of scaling-law limits, Hall physics, current-path changes, instability-driven deviations from ideal thrust scaling, and simulation-guided design
Quasi-steady and applied-field MPDT databases	Performance database and meta-level comparison	Choueiri & Ziemer [65, 66]; Chai & Parashar [88]	Database / meta-analysis	Quasi-steady and applied-field MPDT performance databases	Choueiri and Ziemer provide early and expanded quasi-steady MPDT performance databases; Chai and Parashar provide recent performance insights from the Princeton AF-MPD database	Retained as database-level evidence for performance trend comparison rather than as a single experiment
Electrode and lifetime improvement	Cathode material, erosion reduction, and lifetime extension	Zheng <i>et al.</i> [51]; lifetime data summarized from Sovey & Mantieniks [34]	Experimental / review-level lifetime assessment	Rare-earth-doped nano-cathodes and improved applied-field MPDT cathodes	Reported improvement in lifespan and efficiency; historical lifetime targets remain about 2000–8000 h for long-duration applications	Identifies electrode erosion as a major life-limiting issue and links material improvement to long-duration MPDT feasibility

Note: Efficiency values reported in this table are quoted as given in the reviewed literature. Unless explicitly stated by the original source, these values should be interpreted as thruster/discharge-level efficiencies rather than net spacecraft-system efficiencies. For superconducting, HTS-assisted, or advanced applied-field MPDTs, the listed efficiencies may exclude auxiliary power required for cryogenic refrigeration, magnet-support electronics, power-processing units, thermal-control hardware, and propellant-feed/control subsystems. Therefore, the corresponding integrated system efficiency would be lower and must be evaluated separately for a complete spacecraft design.

of improved magnetic-field configurations and better electrode geometries. However, these values (in Table 6 as well) should not be interpreted as full spacecraft-system efficiencies unless the associated PPU, magnet-support, refrigeration, thermal-control, and propellant-feed/control power terms are included. Additionally, longer operational lifespans have been made possible by advancements in materials, such as tungsten cathodes, which have decreased electrode wear [11, 36]. To improve plasma confinement and energy efficiency, modern systems also have magnetic nozzles and sophisticated cooling mechanisms. When combined, these advancements greatly increase the reliability of modern MPDTs.

In Table 6, the term ‘Conventional MPDT’ refers to the performance ranges reported for earlier laboratory MPDT configurations, including typical self-field and applied-field test campaigns. The term ‘Advanced MPDT’ refers to improved configurations reported in

the literature, such as optimized electrode geometries, applied-field and magnetic-nozzle designs, advanced thermal-management and material approaches, and superconducting/HTS-assisted systems. The values listed for advanced MPDTs are compiled from published experimental, theoretical, and model-based studies, depending on the data reported in each source. Therefore, the values in Table 6 are approximate ranges compiled from the studies reviewed in the manuscript. ‘Conventional MPDT’ values mainly represent earlier laboratory and review-level performance ranges, whereas ‘Advanced MPDT’ values combine experimental measurements, numerical/model-based results, theoretical estimates, and design projections depending on the source. Therefore, the table is intended for engineering comparison and trend identification rather than as a single uniform experimental dataset.

Table 6: Comparison between Conventional and Advanced MPDTs

Parameter	Conventional MPDTs		Advanced MPDTs		Implemented Changes
	Values	Data Basis / Confidence	Values	Data Basis / Confidence	
Mass Flow Rate	0.1–1 g/s	Compiled literature ranges from earlier laboratory MPDT campaigns; moderate confidence	0.2–3 g/s	Mixed experimental/design/scaling range; moderate confidence	Use of compact designs and advanced cooling
Particle Velocity	20–50 km/s	Literature/review-based range, not from a single uniform experiment; moderate confidence	60–70 km/s	Mixed experimental and model-based range for improved configurations; moderate confidence	Improved magnetic field configurations
Specific Impulse	1500–3000 s	Mainly experimental/review-level range from earlier MPDT studies; moderate to high confidence	3000–5000 s	Mixed experimental and model-based range; moderate confidence	Use of high-efficiency propellants like ammonia
Thruster Efficiency	30–50 %	Discharge-level efficiency range compiled from earlier experimental/review studies; moderate confidence	60–70 %	Mixed basis including optimized experimental systems, theoretical estimates, and model-supported studies; moderate confidence; not necessarily net spacecraft-level efficiency	Reduced ionization losses with refined designs
Operational Lifetime	2000–5000 h	Literature lifetime assessment and experimental endurance basis; moderate confidence	8000–10,000 h	Design target/extrapolated range based on improved materials and thermal management; lower confidence than measured performance data	Advanced thermal management and materials
Power Range	5–100 kW	Experimental/laboratory operating range; moderate confidence	0.5–50 kW (scalable systems)	Conceptual/scaling and small-system design basis; lower to moderate confidence	Introduction of miniaturized thrusters

8. CONCLUSION & FUTURE SCOPE

Present review demonstrates the potential of magnetoplasmadynamic thrusters (MPDTs) as high-power electric propulsion systems capable of producing high thrust and high specific impulse through electromagnetic plasma acceleration. The reviewed studies show that applied-field configurations can improve plasma confinement, propellant utilization, and discharge-level efficiency compared with self-field arrangements. However, practical MPDT application is still limited by ionization and dissociation losses, frozen-flow losses, electrode erosion, anode power deposition, plasma instabilities, power-processing complexity, and spacecraft-level integration challenges. Propellant selection also remains a critical design factor because efficiency, storage difficulty, contamination risk, and mission suitability vary significantly among lithium, hydrogen, ammonia, argon, and other candidate propellants.

The future development of MPDTs can be divided into three tiers: short-term engineering research, medium-term technological iteration, and long-term fundamental research.

In the short term, research should focus on engineering reliability. The most urgent requirements are reducing cathode/anode erosion, improving thermal management, stabilizing current attachment, and

operating the thruster below onset conditions. Possible implementation paths include erosion-resistant electrode materials, improved internal cooling or jet-impingement cooling, optimized propellant injection near the anode, reduced power-supply ripple, and calibrated thrust/lifetime testing with local plasma diagnostics.

In the medium term, development should focus on integrated applied-field and superconducting/HTS-assisted MPDT systems. Stronger applied magnetic fields can improve confinement and discharge-level performance, but their real spacecraft benefit depends on auxiliary power requirements for magnet support, cryogenic refrigeration, power processing, and thermal control. Therefore, future studies should report both thruster efficiency and system efficiency. Propellant selection should also be treated as a spacecraft-level trade-off involving performance, storage, contamination, feed-system complexity, and mission duration. Miniaturized or pulsed MPDT concepts for CubeSat-class and small-satellite platforms should be pursued only with proper validation of energy storage, thermal management, discharge stability, and lifetime.

In the long term, fundamental plasma-physics research is required to improve predictive design capability. Important topics include Hall-current transport, critical-current and onset behavior, Alfvén-wave-type and lower-hybrid-drift-type instabilities,

current-path changes, frozen-flow losses, plume divergence, and erosion mechanisms. High-fidelity MHD and PIC/MCC models should be validated using measurements of local species, charge state, density, velocity, current distribution, electromagnetic fluctuations, and electrode heat flux.

Therefore, future MPDT development should follow a staged roadmap: first, improve lifetime and stable operation; second, integrate applied-field, propellant, thermal, and power-processing technologies at the spacecraft-system level; and third, resolve the fundamental plasma-physics limits that control scaling, instability, and erosion. This structured approach can support the transition of MPDTs from laboratory and technology-demonstration systems toward reliable propulsion options for deep-space missions, high-power spacecraft, and next-generation satellite platforms.

COMPLIANCE WITH ETHICAL STANDARDS

On behalf of all authors, the corresponding author states that there is no conflict of interest.

DECLARATIONS

The authors have no relevant financial or non-financial interests to disclose.

NOMENCLATURE

Symbols and Units

B	= Magnetic Field (Tesla)
E	= Electric field (V/m)
e	= Unit vector (-)
T	= Thrust (N)
F	= Force (N)
I	= Electrical discharge current (A)
I_{sp}	= Specific Impulse (seconds)
J	= Current density (A/m ²)
$\dot{m}$	= Mass Flow Rate (kg/s)
P	= Power (W)
P	= Pressure (Pa)
r	= Radius (m)
U	= Plasma flow velocity (m/s)
V	= Control volume (m ³)
V	= Discharge voltage (V)
v	= Velocity (m/s)

Greek Symbols and Units

β	= Electron Hall parameter (-)
η	= Efficiency (-)
μ_0	= Magnetic permeability (N/A ²)
σ_0	= Electrical conductivity (S/m)
τ	= Collision time (s)
ω	= Cyclotron frequency (rad/s)
∇	= Gradient operator (-)

Subscripts

th	= thruster
em	= electromagnetic
sys	= system
swirl	= swirl
jet	= jet
dis	= discharge
aux	= auxiliary
self	= Self-Field
applied	= Applied-Field
z	= axial component
r	= radial component
θ	= azimuthal component
a	= anode
c	= cathode
L	= gyro
e	= electron
eff	= effective

CONFLICT OF INTEREST

The authors declare that they have no conflict of interest.

AUTHOR CONTRIBUTION

Shezan Ghamat: Conceptualization, literature review, data collection, analysis and interpretation, and writing—original draft.

Dinesh Kumar Bajaj: Conceptualization, critical review and synthesis of the literature, technical analysis, writing, review, editing and supervision.

Pandi Siddharth: Literature review, technical interpretation, writing and editing, and co-supervision.

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