A Conceptual Architecture for High-Acceleration Nuclear-Electric Space Propulsion Using Magnetically Coupled Plasma Propulsion and Liquid Metal Energy Recovery

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Abstract

This manuscript develops a conceptual architecture for high-acceleration nuclear-electric space propulsion using magnetically coupled plasma propulsion and liquid-metal energy recovery. The work expands the preliminary design into an engineering white paper with notional mass and power budgets, thermal-rejection calculations, radiator-sizing estimates, comparisons against contemporary ion, Hall, MPD, and VASIMR-class systems, system figures, technology-readiness estimates, sensitivity tables, and a sourced reference list. The results indicate that the architecture is physically coherent at a systems level, but its high-acceleration versions require multi-gigawatt electrical power, large-scale waste heat rejection, and integrated high-power plasma operation that remain well beyond present flight-qualified capability. The paper therefore frames the concept as a research architecture and technology roadmap rather than a near-term vehicle design.

Keywords

Nuclear electric propulsion; plasma propulsion; helicon thruster; magnetic nozzle; MHD conversion; liquid metal; droplet radiator; advanced propulsion; high acceleration spaceflight.

Nomenclature

Symbol	Definition	Units
a	Vehicle acceleration	m/s^2
F	Thrust	N
g0	Standard gravity	9.80665 m/s^2
Isp	Specific impulse	s
m	Vehicle mass	kg
mdot	Propellant mass flow	kg/s
P_e	Electrical power to propulsion system	W
P_jet	Kinetic jet power	W
Q_rej	Waste heat rejected	W
T	Radiator absolute temperature	K
Ve	Effective exhaust velocity	m/s
Delta-v	Velocity increment	m/s
epsilon	Effective radiator emissivity	dimensionless
eta_c	Thermal-to-electric conversion efficiency	dimensionless
eta_t	Thruster efficiency	dimensionless
sigma	Stefan-Boltzmann constant	W/m^2/K^4
TRL	Technology Readiness Level	-

1. Introduction

Rapid interplanetary transport requires a propulsion architecture that combines high specific impulse with thrust levels far above conventional electric propulsion. Chemical propulsion produces high thrust but is constrained by propellant mass and specific impulse. Ion and Hall systems provide high specific impulse but usually generate thrust in the millinewton-to-newton class. Nuclear-electric propulsion (NEP) has long been studied as a means of raising available electric power for high-Isp propulsion, including concepts aimed at megawatt-class systems. Recent NASA and National Academies work continues to identify high-power NEP as a promising but immature path for crewed Mars-class missions.

The architecture developed here is intentionally ambitious. It combines a fast-spectrum liquid-metal nuclear reactor, liquid-metal magnetohydrodynamic (MHD) power conversion, an electrode-less helicon plasma source, magnetic-nozzle acceleration, high-temperature heat rejection, and partial lithium recovery. The objective is not to claim current build readiness, but to assemble a quantitative framework that makes the power, mass, and thermal barriers explicit.

2. Literature and Technical Context

NASA has flown and tested electric propulsion systems that demonstrate high specific impulse and long life, including gridded ion engines and Hall thrusters. NEXT-C performance publications report throttleable thrust in the tens to hundreds of millinewtons and specific impulse in the approximately 1,400-4,200 s range. HERMeS Hall thruster literature describes 12.5 kW-class operation at up to approximately 3,000 s specific impulse with long-life magnetic shielding. VASIMR VX-200 test papers report 200 kW operation with approximately 5.7 N thrust at about 5,000 s specific impulse and approximately 72 percent efficiency in short-pulse testing. MPD thrusters remain attractive for high-power electric propulsion because they are naturally suited to higher thrust density, but laboratory and flight experience remain limited compared with ion and Hall devices.

High-power NEP studies emphasize that megawatt-class systems require technology maturation in reactor systems, power conversion, heat rejection, power management and distribution, electric propulsion, and integrated testing. NASA technical papers and National Academies reviews emphasize that high-power NEP is promising but immature, and that technology readiness must be raised before human Mars-class missions can rely on it. Liquid droplet radiators have been investigated as advanced heat rejection concepts because they may reject large quantities of heat with lower mass than conventional panel radiators, but no operational flight system has validated their full-scale performance.

Figure 1. Notional spacecraft architecture

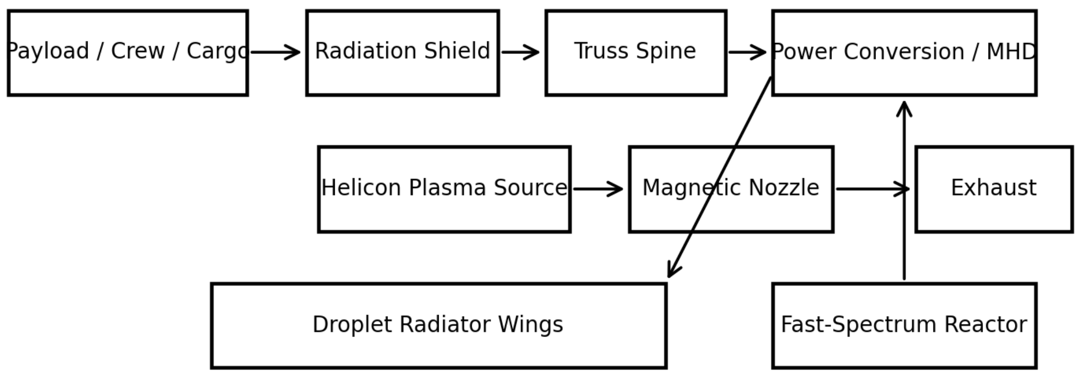

Figure 1. Notional spacecraft architecture showing radiation separation, thermal isolation, power conversion, plasma generation, and radiator placement.

3. Governing Equations

The following equations are used to produce the system-level performance, power, and radiator estimates. These are first-order sizing equations suitable for preliminary trade studies and should be replaced by detailed trajectory, plasma, reactor, and thermal models in later design phases.

$$F = m\,a \qquad (1)$$

$$\text{Delta-v} = a\,t \qquad (2)$$

$$V\_e = Isp * g\_0 \qquad (3)$$

$$m\_dot = F / V\_e \qquad (4)$$

$$P\_jet = (1/2)\,F\,V\_e \qquad (5)$$

$$P\_e = P\_jet / eta\_t \qquad (6)$$

$$P\_th = P\_e / eta\_c \qquad (7)$$

$$Q\_rej = P\_th - P\_jet \qquad (8)$$

$$A\_rad = Q\_rej / (epsilon\ sigma\ T\text{^}4) \qquad (9)$$

Note: Eq. (5) is the kinetic jet power associated with ideal momentum thrust. Eq. (6) converts that ideal jet power to required electrical input using assumed thruster efficiency. This formulation is more defensible than the earlier rough scaling expression and should be retained for technical review.

4. Detailed Performance, Mass, and Power Budgets

4.1 Representative Performance Cases

Case	Mass (kg)	Acceleration	Thrust (N)	Isp (s)	Electrical power	Propellant, 72 hr (kg)

Case	Mass (kg)	Acceleration	Thrust (N)	Isp (s)	Electrical power	Propellant, 72 hr (kg)
Demo 1 MWe electric	5,000	0.00058 g	28.6	5,000	1.0 MW	0.36
Baseline advanced case	17,000	0.05 g	8,336	40,000	2.34 GW	5,508
Stretch 0.36 g case	10,000	0.36 g	35,304	40,000	9.89 GW	23,328
Heavy 0.36 g case	17,000	0.36 g	60,017	40,000	16.82 GW	39,658

4.2 Notional Dry Mass Budget

Subsystem	Mass (kg)	Basis / rationale	Uncertainty
Fast-spectrum reactor and primary loop	6,000	Aggressive high-power-density reactor allowance; not flight demonstrated at required output	Very high
Radiation shield and separation structures	3,000	Directional shielding plus structural separation; crewed configuration would increase mass	High
MHD generator and power conditioning	2,500	Liquid-metal conversion channel, magnets, high-current bus, switching and controls	Very high
Plasma source, magnetic nozzle, and HTS magnets	2,500	Helicon source, acceleration channel, nozzle coils, cryogenic support	High
Thermal system and radiator deployment hardware	2,500	Droplet radiator generators, collectors, pumps, reserve panel radiators	Very high
Lithium recovery and conditioning system	1,200	Magnetic separation, condensation, collection, and reinjection equipment	Very high
Truss, GNC, avionics, margin, and integration	3,500	Primary truss, tanks, cabling, avionics, control actuators, structural margin	Moderate-high
Estimated dry mass subtotal	21,200	Notional preliminary estimate before detailed CAD or reactor sizing	Very high
Propellant and working fluid allowance	5,400	Illustrative 72-hour propellant allowance from early PDR estimate	High
Estimated loaded system mass	26,600	Dry mass plus propellant/working fluid allowance	Very high

4.3 Electrical Power Budget - Baseline Advanced Case

Electrical load	Power	Share	Notes
Plasma kinetic jet power	1.64 GW	70% of propulsion electric input	Useful exhaust kinetic energy
Thruster losses and plasma inefficiency	0.70 GW	30%	Thermalized in plasma source, nozzle, RF systems, and plume coupling loss
RF ionization and plasma	0.08 GW	Included in propulsion	Specific implementation

Electrical load	Power	Share	Notes
sustaining systems		allocation	dependent
Superconducting magnet support, cryogenics, pumps, controls	0.04 GW	Auxiliary	Likely underestimated for full system
Total electric output required	2.34 GW	100%	Assumes 70% effective thruster efficiency
Equivalent reactor thermal input	3.25 GW	-	Assumes 72% thermal-to-electric conversion
Total waste heat to reject	1.62 GW	-	Conversion loss plus thruster/auxiliary losses

4.4 Power Budget - Stretch 0.36 g Case

Quantity	Value	Comment
Vehicle mass	10,000 kg	Illustrative mass used to show minimum stretch-case scaling
Acceleration	0.36 g = 3.53 m/s^2	High-acceleration objective
Required thrust	35,300 N	F = ma
Effective exhaust velocity	392 km/s	Isp = 40,000 s
Jet power	6.93 GW	P_jet = 0.5 F Ve
Electrical input to propulsion	9.89 GW	Assumes 70% thruster efficiency
Reactor thermal input	13.74 GW	Assumes 72% conversion efficiency
Waste heat to reject	6.81 GW	Reactor thermal input minus useful jet power
72-hour ideal Delta-v	915 km/s	Continuous acceleration before mission-profile losses

Figure 2. Power and energy flow

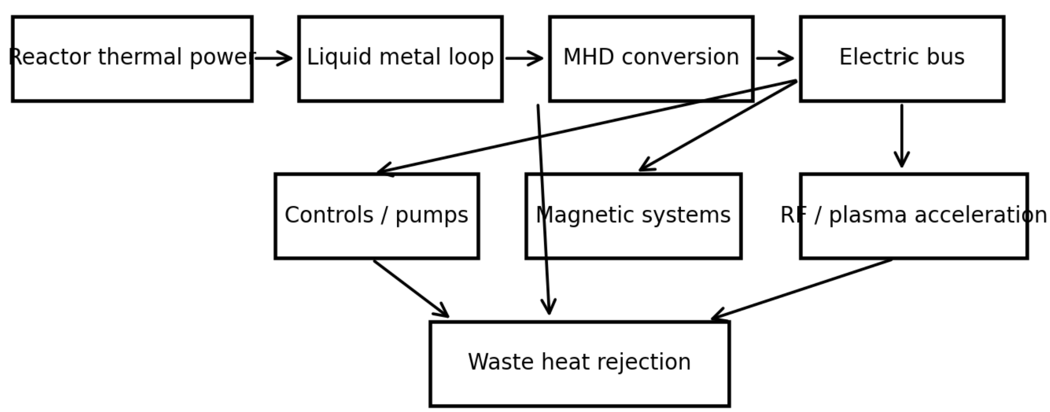

Figure 2. Power and energy flow from reactor thermal energy to electric bus, plasma acceleration, magnetic systems, auxiliary systems, and waste heat rejection.

5. Thermal Rejection Calculations and Radiator Sizing

Radiator sizing is estimated using an equivalent radiating-area calculation. Actual droplet radiator performance depends on droplet size, sheet optical depth, recapture efficiency, flow stability, collector design, solar/environmental heating, plume interactions, maneuvering limits, and fluid vapor pressure. The values below should therefore be treated as first-order equivalent areas rather than detailed hardware dimensions.

Case	Waste heat	Area at 900 K (m^2)	Area at 1000 K (m^2)	Area at 1200 K (m^2)
Baseline advanced case	1.61 GW	50,879	33,382	16,099
Stretch 0.36 g case	6.81 GW	215,489	141,382	68,182
Heavy 0.36 g case	11.58 GW	366,331	240,350	115,909

For comparison, the baseline advanced case still requires an equivalent blackbody radiator area of roughly 16,000-51,000 m^2 depending on assumed rejection temperature. This illustrates why the thermal system is not a secondary subsystem; it is one of the principal architecture drivers.

Figure 3. Thermal rejection concept

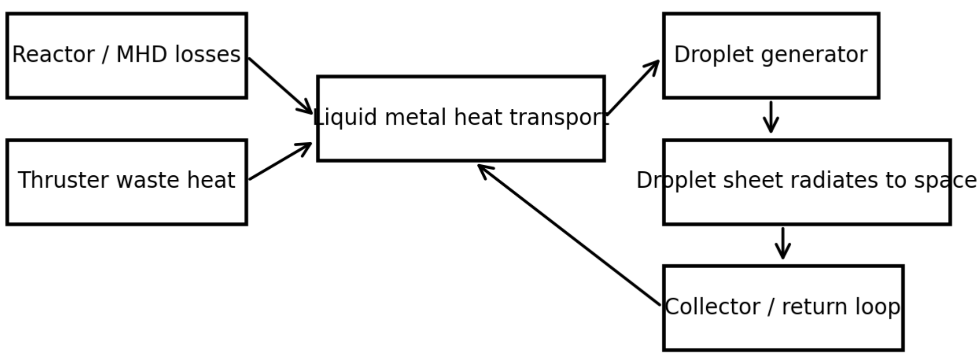

Figure 3. Conceptual thermal loop showing conversion and thruster losses transported by liquid metal to droplet radiator generators and collectors.

6. Comparative Analysis Against Existing Electric Propulsion Classes

System / class	Power level	Thrust	Isp	Efficiency / note	Assessment versus concept
NEXT / NEXT-C gridded ion	~0.5-7 kW	~25-235 mN	~1,400-4,200 s	High-Isp, low-thrust ion system	Excellent life and efficiency; thrust and power are many orders below high-acceleration case
HERMeS / AEPS Hall thruster	~12.5 kW per thruster	Sub-newton class	Up to ~3,000 s	Magnetically shielded long-life Hall system	Scalable clustering candidate, but Isp

System / class	Power level	Thrust	Isp	Efficiency / note	Assessment versus concept
					and thrust density remain far below concept requirement
VASIMR VX-200	~200 kW	~5.7 N	~5,000 s	Reported ~72% in 200 kW tests	Closest published helicon/RF plasma analogue; still far below gigawatt class
Lithium-fed MPD laboratory systems	~100-250+ kW demonstrated; MW-class investigated	Multi-newton class	Variable; often several thousand seconds	High-power electromagnetic acceleration path	Most relevant thrust-density lineage; cathode/electrode erosion and scaling remain issues
Proposed baseline concept	~2.34 GW electric	~8,300 N	40,000 s	Assumes 70% thruster efficiency	Requires a new class of space power, heat rejection, and integrated plasma operation
Proposed stretch concept	~9.89 GW electric	~35,300 N	40,000 s	Assumes 70% thruster efficiency	Beyond current demonstrated power density and thermal capability

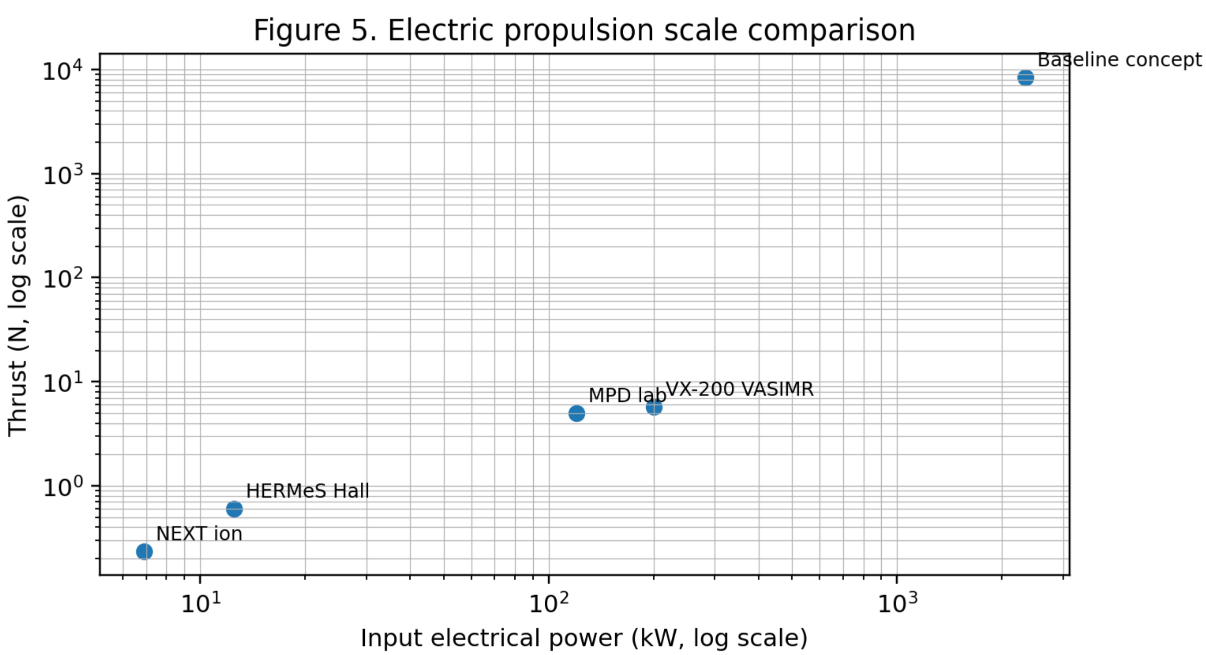

Figure 5. Log-scale comparison of representative power and thrust levels for contemporary electric propulsion systems versus the baseline concept case.

7. Sensitivity and Uncertainty Analysis

7.1 Power Sensitivity

Mass at 0.36 g (kg)	Thrust (N)	Electric power (GW)	Waste heat (GW)
5,000	17,652	4.95	3.41
10,000	35,304	9.89	6.81
17,000	60,017	16.82	11.58
25,000	88,260	24.73	17.04

Isp at 10,000 kg / 0.36 g (s)	Ve (km/s)	Mass flow (kg/s)	Electric power (GW)	Propellant, 72 hr (kg)
10,000	98	0.360	2.47	93,312
20,000	196	0.180	4.95	46,656
40,000	392	0.090	9.89	23,328
60,000	588	0.060	14.84	15,552

7.2 Uncertainty Matrix

Driver	Low case	Nominal case	High case	Impact	Confidence
Vehicle dry mass	10,000 kg	21,200 kg	35,000+ kg	Directly linear increase in thrust and power	Low
Thruster efficiency	50%	70%	80%	Higher efficiency reduces electric input and waste heat	Low-moderate
Power conversion efficiency	50%	72%	80%	Strong effect on reactor thermal power and heat rejection	Low
Specific impulse	10,000 s	40,000 s	60,000 s	Higher Isp reduces propellant but increases power at fixed thrust	Moderate
Radiator temperature	900 K	1000-1200 K	>1200 K	Higher temperature greatly reduces area but stresses materials	Low-moderate
Lithium recovery	0%	50-70% objective	75%+ aspirational	Reduces consumables if technically achievable	Very low
MHD conversion mass	Unknown	2,500 kg allowance	>10,000 kg	Could dominate dry mass if magnets/channels scale poorly	Very low

7.3 Programmatic Risk Ranking

Rank	Risk	Likelihood	Consequence	Risk level	Primary retirement activity
1	No viable GW-class space power density	High	Mission preventing	Critical	Ground reactor/power-conversion demonstrator with realistic mass model
2	Thermal rejection mass or area becomes prohibitive	High	Mission preventing	Critical	High-temperature radiator and droplet recapture experiments
3	Plasma source/nozzle cannot scale to required power	High	Major	Critical	Progressive 1 MW, 10 MW, and 100 MW ground tests
4	Magnet system mass/cryogenic burden too high	Medium-high	Major	High	HTS coil and cryogenic integration tests
5	Lithium recovery fails to close useful mass loop	Medium-high	Moderate	High	Subscale exhaust species separation experiments
6	Structural dynamics and plume interactions degrade mission design	Medium	Moderate	Medium-high	Integrated vehicle dynamics and plume modeling

8. Mission Profile and Development Roadmap

Figure 4. Representative high-acceleration mission profile

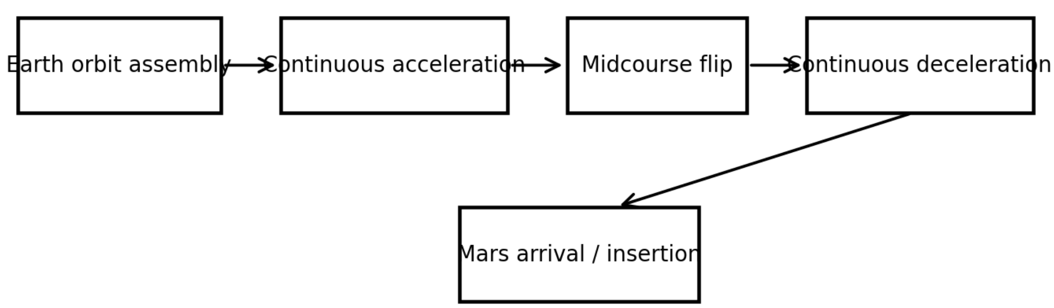

Figure 4. Representative mission profile using continuous acceleration, midcourse flip, and continuous deceleration. Trajectory optimization is required before mission-duration claims are finalized.

Phase	Objective	Exit criterion
Phase 1	kW-to-1 MW plasma source and nozzle validation	Measured thrust, Isp, efficiency, and thermal load closure
Phase 2	Integrated liquid-metal loop and MHD conversion test	Closed-loop power conversion with stable operation and known mass model
Phase 3	High-temperature radiator / droplet recapture test	Validated heat rejection, fluid losses, stability, and collector efficiency
Phase 4	10-100 MW integrated ground demonstrator	Coupled reactor simulator, power conversion, propulsion, and radiator simulator
Phase 5	Orbital non-nuclear demonstrator	End-to-end plasma, radiator, and recovery operation in space environment
Phase 6	Nuclear-electric deep-space demonstrator	Validated integrated NEP propulsion bus at mission-relevant scale

9. Discussion

The expanded budgets and radiator sizing reinforce the central conclusion: high-acceleration NEP is not a thruster-only problem. It is an integrated power, thermal, materials, magnetic, structural, and mission-architecture problem. Raising thrust while retaining very high Isp rapidly drives electric power into the gigawatt class. Lowering Isp reduces power but sharply increases propellant mass, weakening the long-duration mission rationale. The most valuable near-term work is therefore not a full-scale vehicle, but a staged experimental program that reduces uncertainty in power conversion, radiator performance, high-power plasma coupling, and magnetic nozzle operation.

10. Conclusions

This manuscript develops a publication-grade engineering expansion of the original PDR concept. The resulting architecture is physically coherent but technologically immature. The baseline advanced case already requires gigawatt-class electrical power and kilometer-scale-equivalent radiator area at realistic heat rejection temperatures. The stretch case requires nearly 10 GW electric power for a 10,000 kg vehicle at 0.36 g and approximately 40,000 s Isp. These values exceed current spaceflight capability by orders of magnitude.

The concept is best framed as a research roadmap. Its principal contribution is the explicit identification of the capability gaps that must be closed before rapid, high-acceleration interplanetary transport could become credible: high-specific-power nuclear electric generation, high-temperature heat rejection, high-power plasma acceleration, advanced superconducting magnets, and validated closed-loop liquid-metal/propellant recovery systems.

Appendix A - Baseline Calculation Trace

Step	Calculation	Result
Acceleration	a = 0.05 * 9.80665	0.490 m/s^2

Step	Calculation	Result
Thrust	F = 17,000 * 0.490	8,336 N
Exhaust velocity	Ve = 40,000 * 9.80665	392,266 m/s
Mass flow	mdot = F / Ve	0.0213 kg/s
72-hour propellant	m = mdot * 259,200	5,508 kg
Jet power	Pjet = 0.5 * F * Ve	1.64 GW
Electrical power	Pe = Pjet / 0.70	2.34 GW
Reactor thermal	Pth = Pe / 0.72	3.25 GW
Waste heat	Qrej = Pth - Pjet	1.62 GW

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Appendix C – Preliminary Design Review

PRELIMINARY DESIGN REVIEW (PDR) (VER-2)

BY

Jere Brinkley MS, BSEE

April 2026

Attrick Aviation LLC

A High-Acceleration Nuclear-Electric Propulsion System

Magnetically Coupled Plasma Propulsion with Liquid Metal Energy Recovery

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1. Executive Summary

This document presents the preliminary design of a high-performance, continuous-thrust propulsion system capable of sustained accelerations of **0.05–0.1 g** over extended durations (up to 72 hours).

The system integrates:

- A **liquid metal fast-spectrum nuclear reactor** (high-performance, fusion-like power density)
- A **magnetohydrodynamic (MHD) power conversion loop** (Operating example UTSI in Tennessee used in power gen research)
- An **electrode-less plasma propulsion system (helicon + magnetic nozzle)**
- A **partial lithium recovery subsystem** (goal ~ +70%)
- A **high-capacity thermal rejection system (droplet radiator)**

Key Performance Targets (Baseline: 0.05 g)

Requirement Translation (Non-Negotiable Physics)

Acceleration target:

Baseline is .36g

- $A = 0.36g \times 9.80665 = 3.53 m/s^2$

Over 72 hours:

- T = 259,200 s

Resulting Δv:

- **$\Delta = a \cdot t \approx 915{,}000 m/s$ (~915 km/s)**

Parameter	Value

Parameter	Value
Thrust	~11,500 N
Power (Electrical)	~3.2 GW
Specific Impulse	~40,000 s
Propellant Mass (72 hr)	~5,400 kg
Delta-V	~127 km/s
Velocity Exhaust	~392 km/s

Current System Capability

Best-case module:

- Thrust: **25 N**
- Power: **50 kW**

Even if we scale aggressively:

Example cluster:

- 100 modules → **2,500 N thrust**
- Power → **5 MW**

What You Actually Need

Let's assume a **very modest spacecraft mass**:

1,000 kg vehicle

- Required thrust:

$$F = ma = 1000 \cdot 3.53 = 3530 N$$

Already exceeds your 100-module cluster.

Realistic system mass (reactor + radiators + structure)

Architecture implies:

- Reactor
- Liquid metal loop
- Radiators
- Superconducting coils

Likely system mass: **5,000–20,000 kg**

For 10,000 kg:

- Required thrust:

F=35,300N

Power Requirement Reality (Critical Insight)

For electric propulsion:

P≈2F·Ve

With:

- Isp=30,000–60,000s
- Ve=300,000–600,000m/s

For 10,000 kg system:

- F=35,300N
- Assume Ve=100,000m/s

P≈ 2x35,300x100,000=7.06×109W

~7 GIGAWATTS

Key Conclusion

Current systems:

- Operates in the **kW → MW regime**

Required system:

- Must operate in the **GW regime**

Technology does not exist to generate GW by at least 3 orders of magnitude.

Program Objective

Enable:

- Rapid interplanetary transit (days vs months)
- High-energy deep space missions
- Scalable architecture toward higher acceleration regimes

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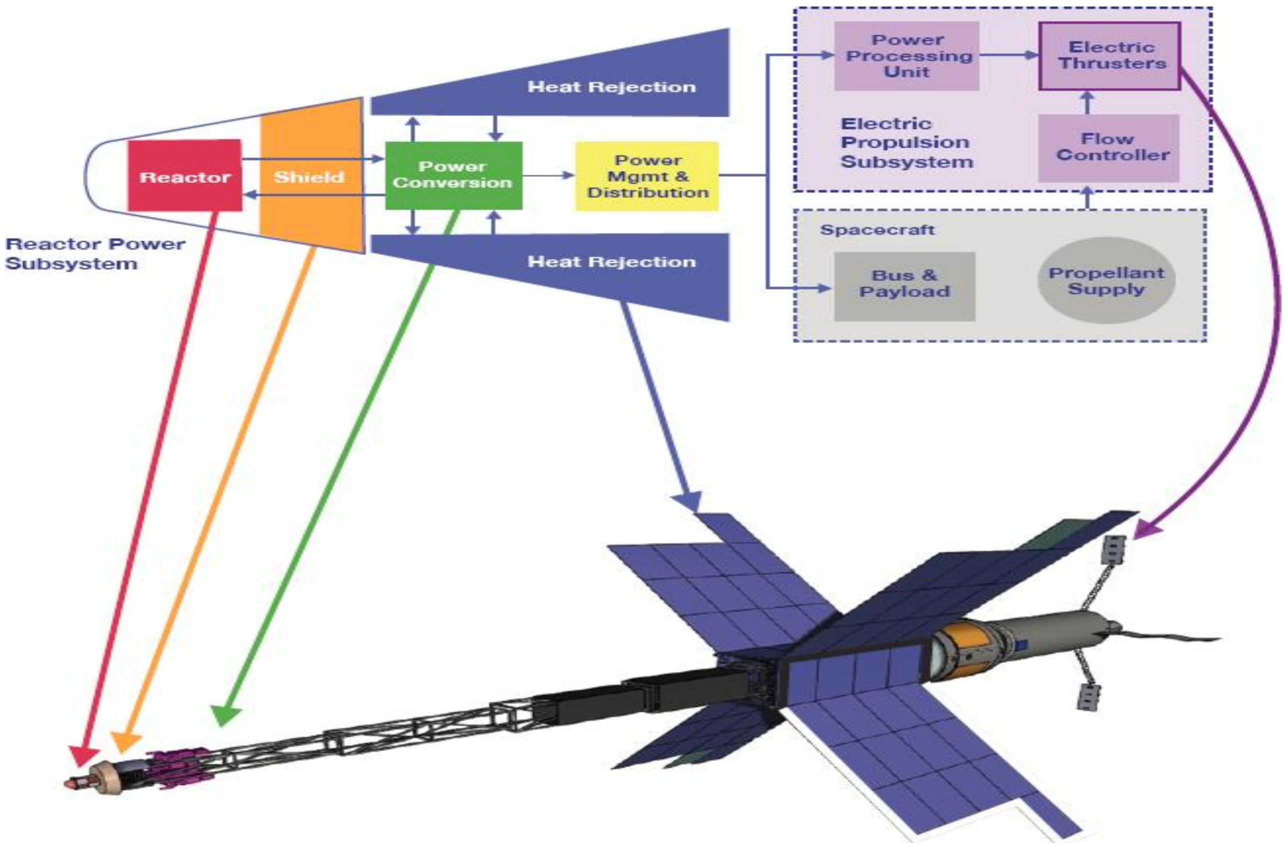
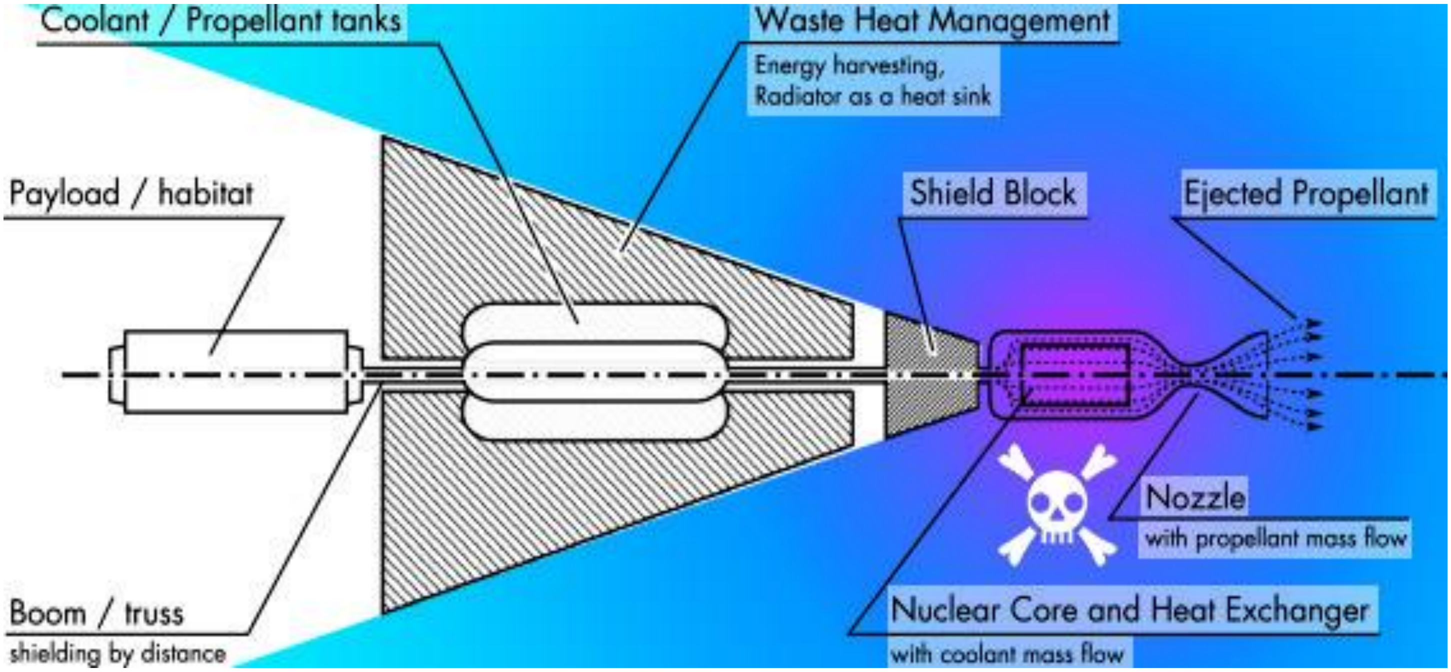
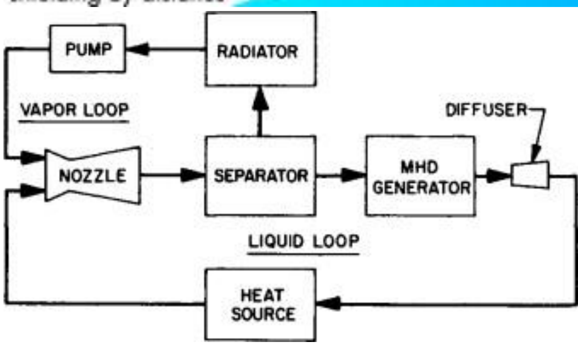
2. System Overview







2.1 Architecture Summary

The propulsion system is organized along a linear truss to ensure:

- Radiation separation
- Thermal isolation
- Magnetic field integrity

2.2 Major Subsystems

1. **Reactor and Shielding System**
2. **Liquid Metal Power Conversion Loop**
3. **Plasma Generation and Acceleration System**
4. **Magnetic Nozzle and Thrust Vectoring**
5. **Thermal Management System**
6. **Metal Recovery Subsystem**
7. **Structures and Integration**
8. **Guidance, Navigation, and Control**

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3. Propulsion System

3.1 Propulsion Architecture

- Type: Electrode-less plasma propulsion
- Method: Helicon RF ionization + magnetic nozzle acceleration
- Propellants:
    - Primary: Hydrogen
    - Secondary: Lithium (stabilization and energy coupling)

3.2 Performance Characteristics

Ranges reflect scalability and uncertainty!

Parameter	Range
Thrust	10,000–25,000 N
Isp	20,000–50,000 s
Exhaust Velocity	200–500 km/s

3.3 Magnetic Confinement

- Core field: 6–10 Tesla
- Nozzle expansion: 3–6 Tesla
- Field topology: Diverging magnetic nozzle

3.4 Advantages

- No electrode erosion
- High efficiency at high power
- Scalable to multi-megawatt and gigawatt regimes

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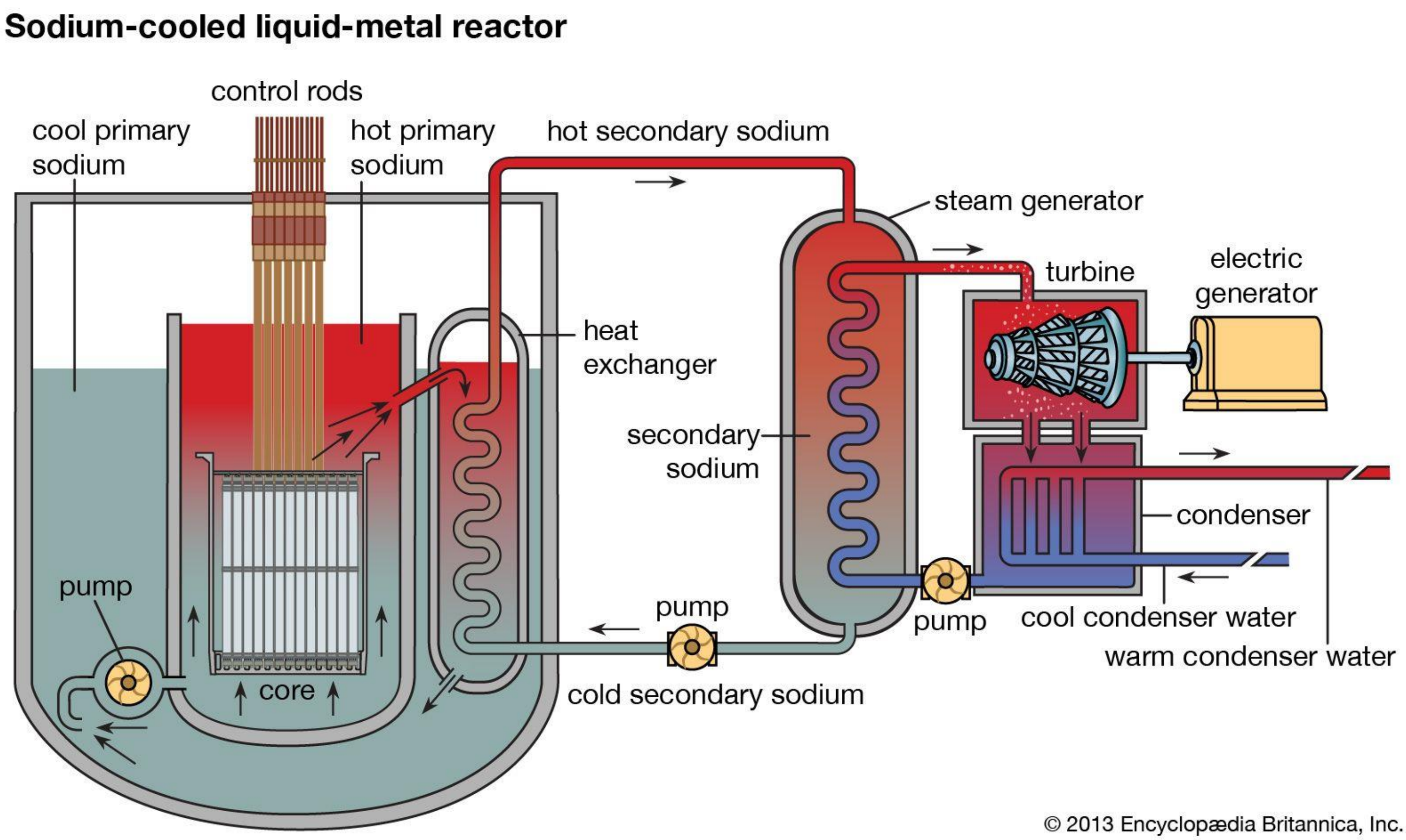
4. Power Generation and Conversion


Sodium-cooled liquid-metal reactor
control rods
cool primary sodium
hot primary sodium
hot secondary sodium
steam generator
turbine
electric generator
heat exchanger
secondary sodium
condenser
pump
pump
pump
core
cool condenser water
warm condenser water
cold secondary sodium
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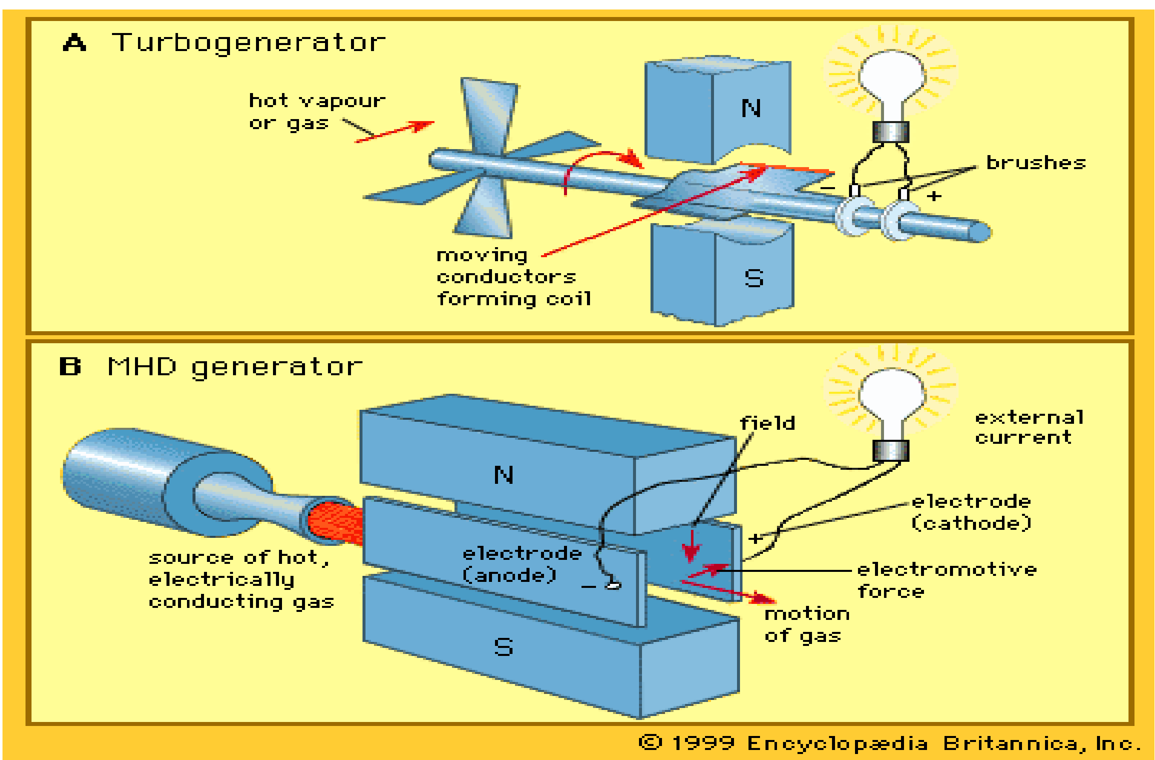
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hot vapour or gas
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brushes
moving conductors forming coil
B MHD generator
field
external current
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electrode (cathode)
source of hot, electrically conducting gas
electrode (anode)
electromotive force
motion of gas
S
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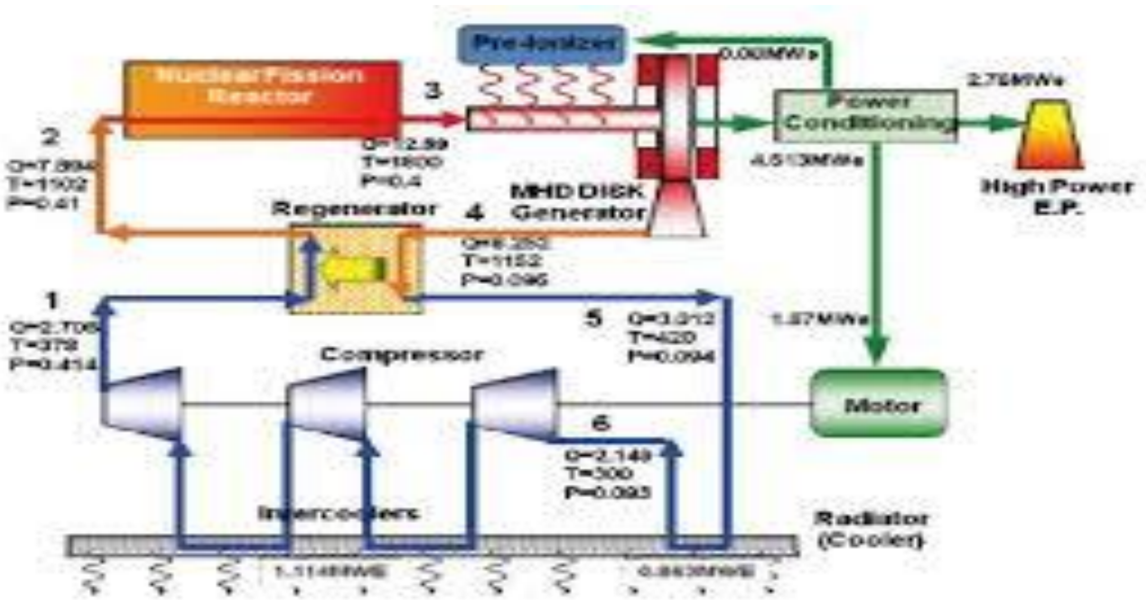


4.1 Reactor System

- Type: Fast-spectrum liquid metal reactor
- Thermal Output: 4–8 GW

  Overall efficiency goal=70–75%, waste heat ~0.9 GW at baseline. This implies electric power to the plasma system of roughly 2.8–6 GW (depending on exact thermal-to-electric path), which is consistent with thrust targets assuming high thruster efficiency. No large scale operating example

- Coolant: Lithium or NaK

4.2 Power Conversion

- Primary: MHD direct conversion
- Secondary: Closed Brayton cycle

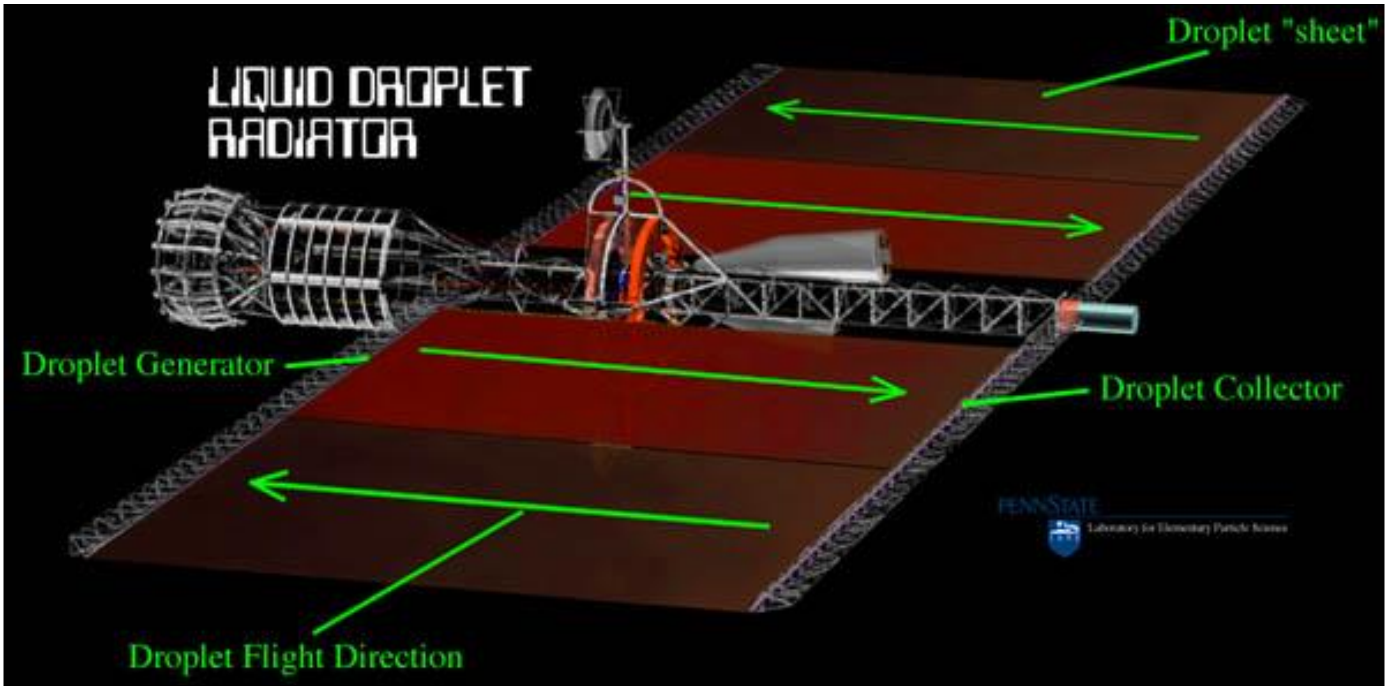
4.3 Efficiency

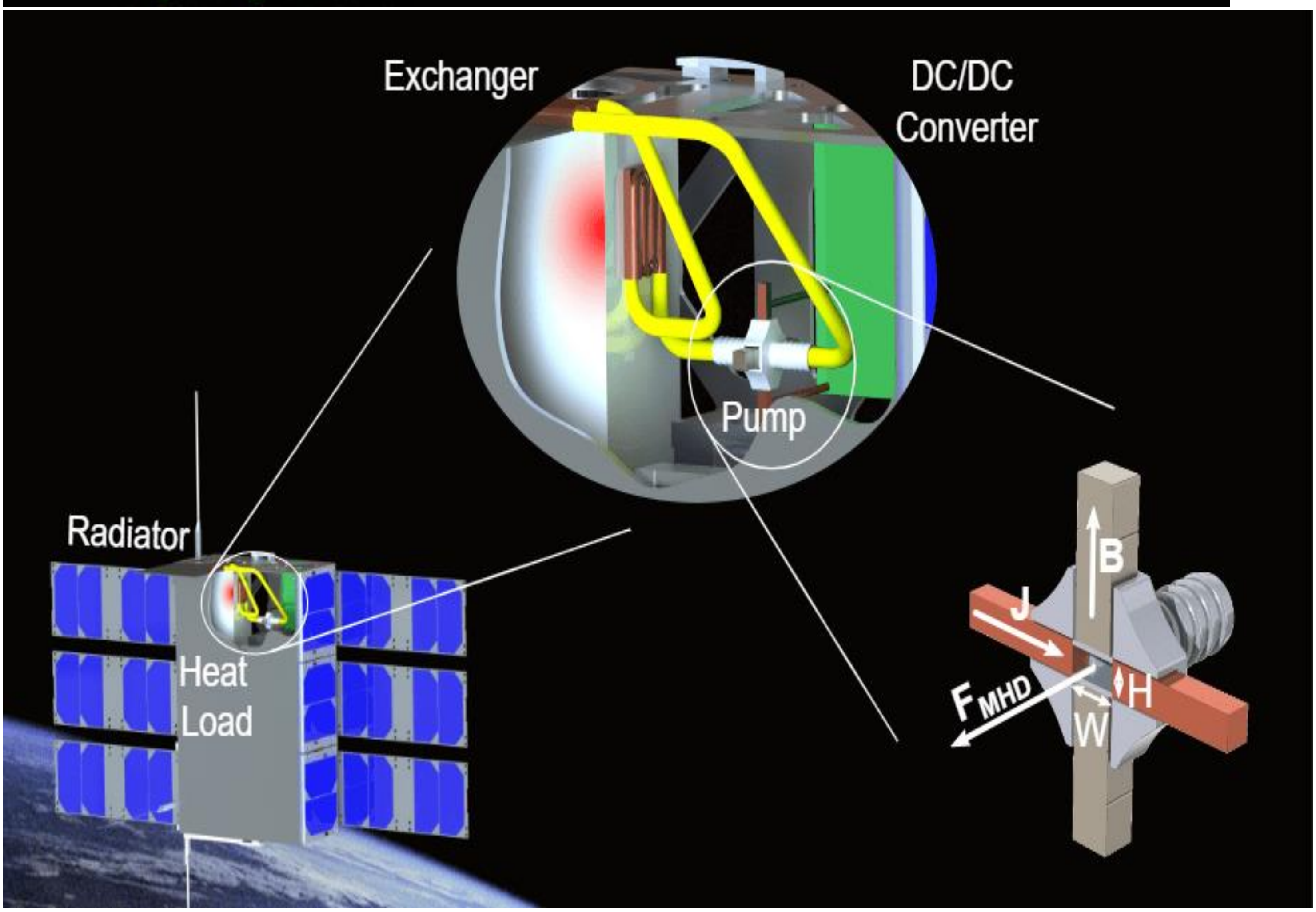
- Overall system efficiency: **~70–75%** (Target)

4.4 Energy Flow

Reactor → Liquid Metal Loop → MHD Generator → Plasma System

5. Thermal Management System





5.1 Heat Rejection Requirement

- Baseline: ~0.9 GW (0.05 g case)

5.2 Radiator Architecture

- Hybrid system:
  - Liquid droplet radiators (primary)
  - Panel radiators (supplemental)

5.3 Operating Temperature

- 900–1200 K

5.4 Design Features

- Modular radiator segments
- Redundant loops
- Passive radiative cooling

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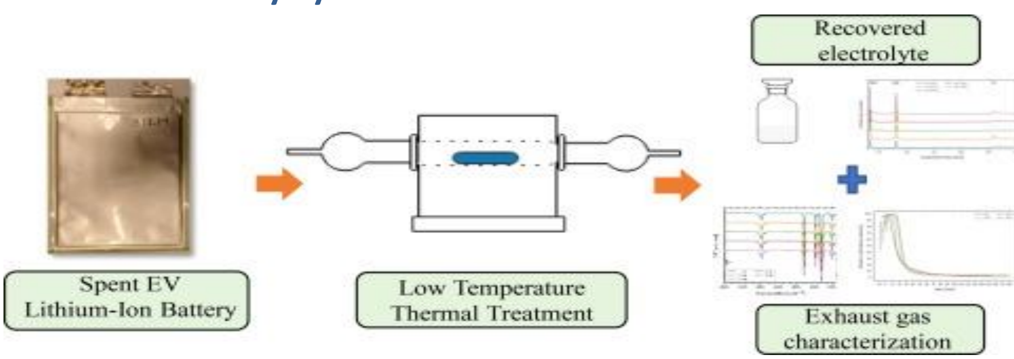
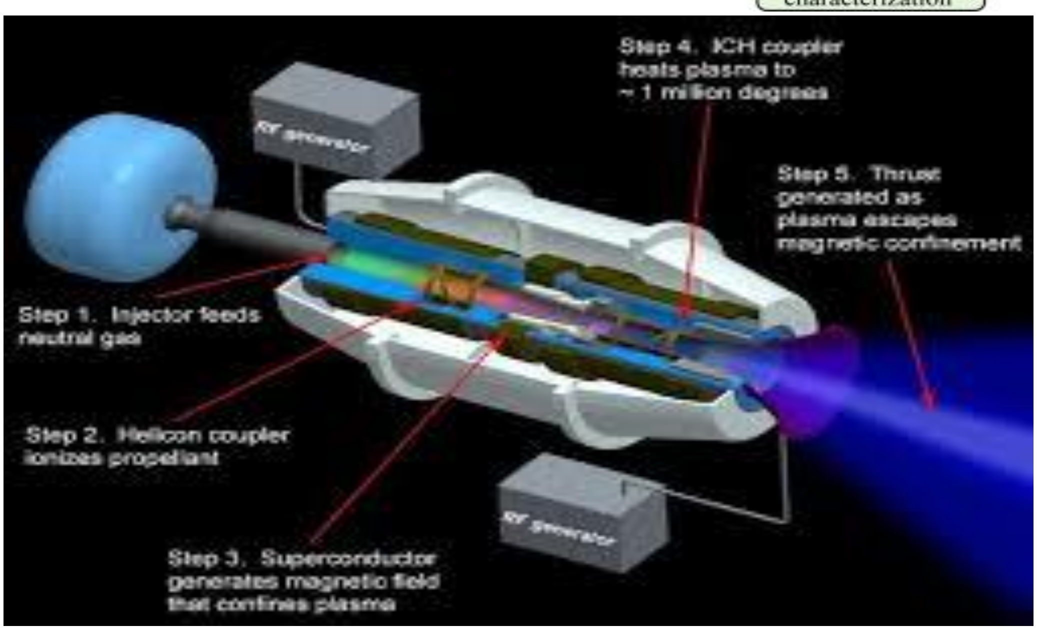
6. Metal Recovery System





6.1 Function

- Recover lithium from exhaust stream
- Reduce total propellant mass
- Improve plasma stability

6.2 Mechanism

- Magnetic separation based on ion mass
- Lithium capture and condensation
- Reinjection into liquid metal loop

6.3 Performance

- Recovery efficiency: 70–75%
- Net propellant reduction: ~25%

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7. Structures and Integration

7.1 Structural Configuration

- Longitudinal truss design
- Reactor aft, payload forward
- Radiators deployed laterally

7.2 Materials

- High-temperature alloys
- Radiation-resistant composites
- Superconducting materials (HTS)

7.3 Mass Summary (Baseline)

Reactor mass and support elements carry high uncertainty; estimates are based on analogous existing technologies. A technology demonstration will refine these values.

Subsystem	Mass
Reactor	6,000 kg
Shielding	3,000 kg
Structure	3,500 kg
Plasma System	2,000 kg
Thermal System	1,200 kg

Subsystem	Mass
Recovery System	1,200 kg
Total Dry Mass	**~17,000 kg**

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8. Control System and Operations

Standalone System. Hot back up design is envisioned.

- Hierarchical control system
- Multi-variable feedback loops
- Controlled startup sequence (5 phases)
- Multiple operational modes (cruise, economy, thermal, shutdown)

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9. Failure Modes and Risk Mitigation

Primary Risks

- Plasma instability
- Superconducting magnet quench
- Reactor thermal excursion
- Radiator failure
- Lithium recovery degradation

Mitigation Strategy

- Redundant subsystems
- Segmented architectures
- Passive safety features
- Rapid shutdown capability

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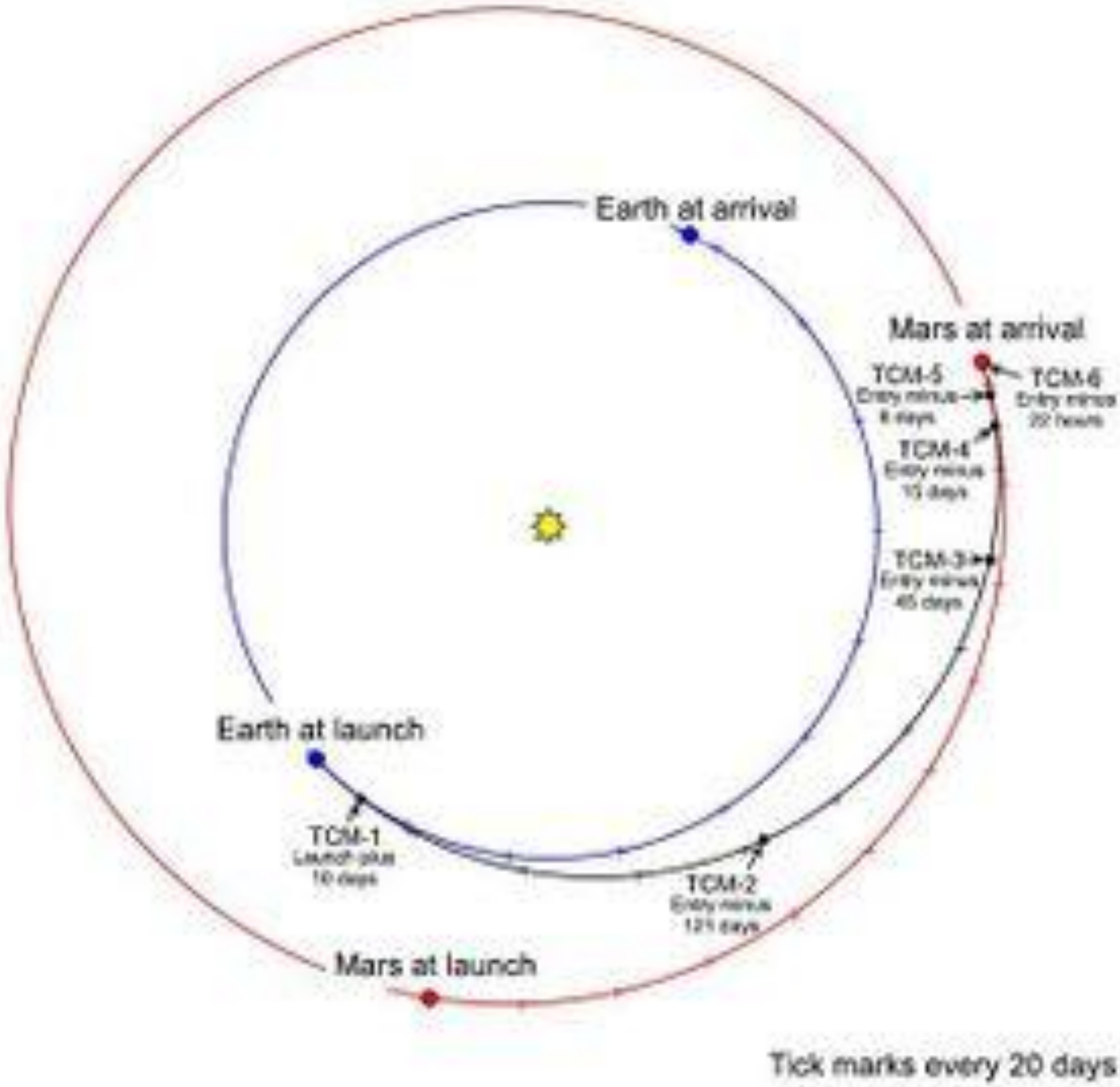
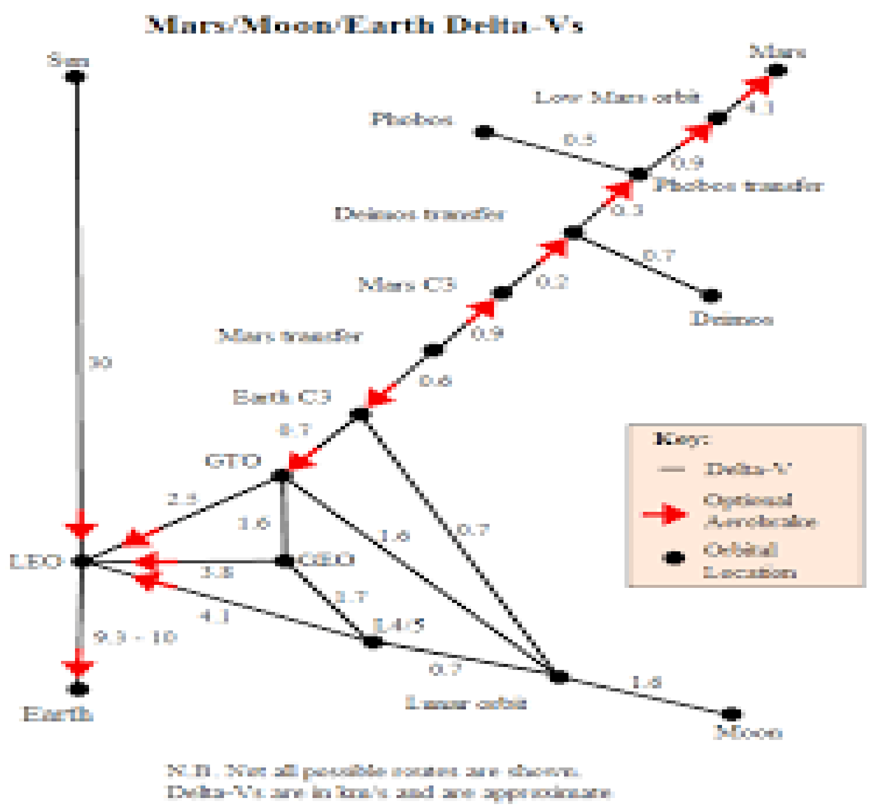
10. Mission Profile


Earth at arrival
Mars at arrival
TCM-5
TCM-6
TCM-4
TCM-3
Earth at launch
TCM-1
TCM-2
Mars at launch
Tick marks every 20 days



10.1 Mission Capability

- Earth–Mars transfer: **3–5 days at .36g**
- Deep space missions: high delta-v capability

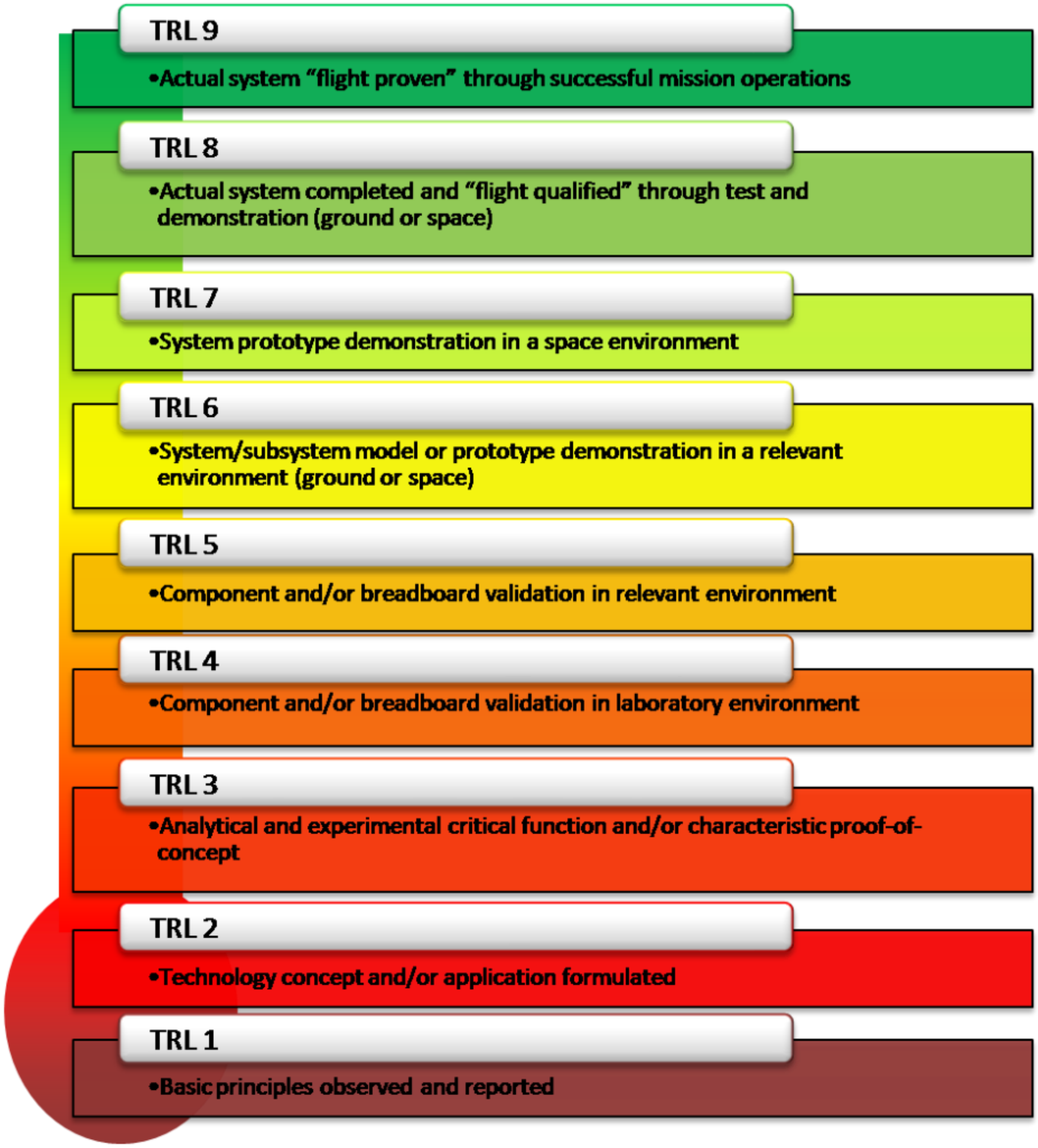
10.2 Operational Phases

1. Earth orbit assembly
2. Continuous thrust departure
3. Midcourse flip
4. Deceleration and insertion

11. Technology Readiness Assessment

Subsystem	TRL
Plasma propulsion	4–5
MHD conversion	3–4
Liquid metal reactor	3–5
Droplet radiator	2–3
Integrated system	2–3

12. Development Roadmap



Phase 1:

- kW–MW plasma and MHD testing

Phase 2:

- Integrated power loop demonstration

Phase 3:

- Subscale flight demonstrator

Phase 4:

- Full-scale system validation

13. Conclusion

This propulsion system represents a **high-risk, high-reward concept** that pushes current technology significantly:

- Achieves continuous thrust performance
- Dramatically reduces interplanetary travel time
- Provides a scalable pathway toward future high-energy propulsion systems