

Case Report

Electrical Power and Propulsion System Architecture for a 75 Kg Microsatellite Hall-effect Thruster

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Abstract

This paper presents a comprehensive system-level design for the electrical power and electric propulsion subsystems in microsatellites. The study begins with an overview of the subsystems typically integrated into microsatellite platforms, before focusing in greater detail on electrical power distribution and electric propulsion. Particular attention is given to the Hall Effect Thruster (HET), including its operating principle, advantages, and inherent limitations. A three-year mission scenario is considered to estimate annual velocity changes and corresponding power requirements, providing a realistic operational framework. The analysis incorporates orbital mechanics by examining the relationship between the Sun and satellite in terms of Earth's radius, gravitational constant, mass, and eclipse duration. Satellite velocity is calculated across different orbital geometries, with additional consideration of drag forces that may arise in low Earth orbit. Building on this foundation, the paper concentrates on the design of a miniaturized HET tailored for a 75 kg satellite operating in a 1000 km circular orbit. Key design parameters such as thrust requirements, power demands, propellant selection, and component sizing are systematically evaluated. The proposed system enables small-scale orbital maneuvers through continuous monitoring of orbital velocity and feedback-based corrections. Furthermore, the paper details strategies for power distribution among subsystems and identifies the fundamental components required for implementation. By integrating propulsion and power considerations at the system level, the study demonstrates a viable pathway for enhancing the autonomy and maneuverability of microsatellites in extended missions.

Keywords

Microsatellite, Hall Effect Thruster, Velocity Maneuvers, Eclipse Cycles, Xenon

1. Introduction

Introduction to microsatellite:

A 75-kg mini-satellite often called a microsatellite, typically includes all the standard spacecraft subsystems, just optimized for mass, power, and volume. Below is a high-level, non-sensitive overview of typical subsystems and what they do. The Structure & Mechanisms subsystem includes the

primary load-bearing frame, typically made from aluminum or carbon-fiber composites, along with mounting panels for payload and avionics, solar panel hinges or fixed body-mounted attachments depending on mission needs, and any required deployment mechanisms such as antennas or booms. The Electrical Power System (EPS) and Propulsion

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System consists of solar arrays, which may be body-mounted or small deployable, a battery pack such as Li-ion or Li-polymer, and power regulation electronics including Maximum Power Point Tracking (MPPT) and a Power Distribution Unit (PDU). [1]

For this class of microsatellite, peak power typically ranges from 50 to 150 W depending on the mission, and the propulsion system is used for orbit maintenance, deorbiting, or attitude desaturation. The Communication Subsystem (COMMS) includes UHF/VHF, S-band, or X-band transceivers depending on mission requirements, along with patch, monopole, or small parabolic antennas, providing telemetry, command links, and data downlink capabilities, with typical data rates ranging from low kbps to moderate Mbps for X-band systems. The Command & Data Handling (C&DH) subsystem consists of an on-board computer (OBC) with a radiation-tolerant processor, data storage such as flash memory, interfaces to payload and sensors, and

capabilities for handling housekeeping data. The Attitude Determination & Control System (ADCS) includes sensors such as sun sensors, magnetometers, gyroscopes/IMUs, and optionally a star tracker for high pointing accuracy, as well as actuators like reaction wheels and magnetorquers, providing pointing performance that ranges from coarse (degrees) to fine (arcminutes) depending on the mission requirements. The Thermal Control System (TCS) comprises passive thermal components such as multi-layer insulation (MLI), thermal coatings or paints, and conductive straps, and may include small resistive heaters for batteries or sensitive payload units. The Payload Subsystem depends entirely on the mission type and may include an Earth-observation camera, scientific instruments, technology demonstrators, communications payloads, or AIS/ADS-B systems, typically accounting for 10–30% of the total microsatellite mass, roughly 7–20 kg. [2, 3]

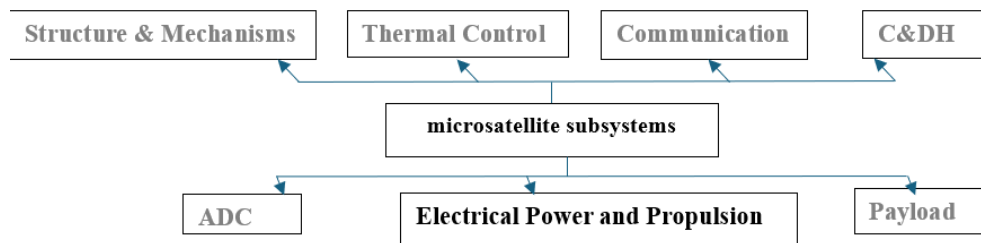


Figure 1. Microsatellite Subsystems.

Introduction to EPS and propulsion:

provides continuous and stable electrical power to all spacecraft subsystems and payloads throughout the mission lifecycle. EPS is typically composed of: The Power Generation Requirements for solar array panels, whether body-mounted or small deployable, are to provide sufficient end-of-life (EOL) power to support peak and average loads, operate efficiently under incidence angle variations of $\pm 25^\circ$ – $\pm 45^\circ$, and maintain performance despite radiation-induced degradation. [4]

The Solar Array Assembly (SAA) uses high-efficiency triple-junction GaAs cells and can be configured as simple body-mounted panels providing lower power (~ 40 – 80 W) or deployable panels for higher power (~ 100 – 150 W), with series and parallel strings arranged to meet the required bus voltage. The Energy Storage Requirements specify a rechargeable lithium-ion battery pack capable of supporting eclipse operations of up to ~ 35 minutes in low earth orbit (LEO), providing an adequate depth of discharge (DoD) ($\text{DoD} \leq 20$ – 30%) for long life, and ensuring safe charge and discharge performance under varying thermal environments. The battery system typically uses lithium-ion or lithium-polymer cells and is configured with 4–6 cells in series, depending on the bus voltage. Key parameters include a capacity range of 40–120 Wh, a DoD limit of 20–40% to ensure long life, and an integrated battery management system (BMS)

for cell balancing and protection. Power regulation requirements include the use of MPPT units, power conditioning stages, and appropriate converters to maintain bus voltage within acceptable tolerances (such as 12 V, 28 V, or regulated 5 V/3.3 V lines), while maximizing solar array output through MPPT control for each array or string and ensuring system protection through over-voltage, under-voltage, short-circuit, and over-current safeguards. The PCU is responsible for MPPT control for each string, routing solar power to the battery or load, managing the charging algorithm using a CC/CV approach, providing regulated DC-DC voltage conversion, and performing telemetry sensing. Power distribution requirements include using a PDU with protected outputs that provide independent, switchable loads for redundancy and safety, ensure all downstream lines are fused or current-limited, and supply power to the payload through dedicated regulation lines when necessary. The PDU uses solid-state relays to provide switchable loads, offers current-limited outputs, and includes dedicated power ports for the OBC and avionics, ADCS actuators such as reaction wheels and magnetorquers, communications transceivers, and payload modules, while also supplying housekeeping data including voltage, current, and temperature. Monitoring and telemetry functions include sensing voltage, current, and temperature. [5]

Table 1. Typical Performance Parameters.

Parameter	Typical Range
Average power generation	70–120 W
Peak power (sun-pointing)	100–150 W
Battery capacity	40–120 Wh
Main bus voltage	12 V or 28 V
Regulated lines	5 V, 3.3 V, payload-specific
Solar cell efficiency	28–32%
Power margin	20–30% recommended

Thermal and Environmental Considerations for the EPS require that components withstand temperature cycling from -20°C to $+50^{\circ}\text{C}$ during operation, radiation effects such as 2–3% annual degradation of solar cells in LEO, vacuum conditions using outgassing-compatible materials, and high vibration loads during launch, while thermal control ensures batteries remain within $0\text{--}30^{\circ}\text{C}$ for optimal performance and power electronics are kept within safe limits using conduction paths and radiators. EPS Fault Management typically includes strategies such as automatic load shedding during power deficits, battery over-temperature shutdown, redundant power converters for critical loads, solar string-level isolation for failure recovery, and a safe mode operation supplying only minimal essential loads like the OBC and COMMS. The Interfaces to Other Subsystems for the EPS include interaction with the OBC for command routing and telemetry reporting, with the ADCS to manage peak loads from reaction wheels and actuators, with the COMMS system to schedule power for high-power downlink sessions, with the payload to provide dedicated and stable regulated power, and with the Thermal subsystem to maintain proper battery and PCU temperature regulation. HET are a category of electrostatic plasma propulsion devices that utilize a cross-field discharge a combination of electric and magnetic fields to ionize and accelerate a propellant, producing thrust. They have emerged as one of the most successful and mature forms of electric propulsion for satellite station-keeping, orbit transfer, and deep-space exploration due to their efficiency, reliability, and relatively simple design. The operation of a HET is based on the Hall effect, discovered by Edwin H. Hall in 1879, which describes the generation of a potential difference (the Hall voltage) across a current-carrying conductor placed in a magnetic field perpendicular to the current. In the case of a Hall thruster, this principle is exploited to trap electrons and enhance ionization. A typical Hall thruster consists of an annular discharge channel, an anode, a cathode-neutralizer, and a magnetic circuit. The anode, located at the upstream end of the channel, also serves as the propellant inlet. A radial magnetic field (B) is generated across the channel using electromagnetic coils or permanent magnets, while an axial

electric field (E) is established between the anode and the cathode. [6]

When a neutral propellant gas most commonly xenon is injected into the channel, it is partially ionized by energetic electrons emitted from the cathode. These electrons are trapped by the magnetic field, causing them to drift azimuthally under the influence of the $E \times B$ field, forming what is known as the Hall current. The azimuthal electron motion increases the probability of collisions with neutral atoms, thereby maintaining a steady-state plasma and efficient ionization. The resulting positively charged ions are accelerated axially by the electric field to high velocities (typically 10–20 km/s). Upon leaving the channel, the ion beam is neutralized by additional electrons emitted from the cathode to prevent the spacecraft from becoming electrically charged. Thruster design incorporates several key components, including a discharge channel typically made of boron nitride or alumina ceramics to resist plasma erosion, an anode that functions as both the propellant distributor and positive electrode, a cathode-neutralizer that provides electrons for ionization and beam neutralization, and a magnetic circuit engineered to confine electrons and regulate plasma behavior. Hall thrusters typically exhibit specific impulses of 1200–2000 seconds, thrust levels ranging from 10 to 250 mN for most satellite-class systems, and efficiencies between 40% and 70%, depending on their design and operating conditions. Hall thrusters are widely used in space applications, including station-keeping and orbit correction on commercial geostationary earth orbit (GEO) satellites such as those using Space bus and Eurostar platforms, orbit transfer on all-electric satellites developed by companies like Airbus and Boeing where HET serve as the primary propulsion system, and deep-space missions such as ESA's SMART-1 lunar probe and NASA's DART and Psyche spacecraft, which demonstrate their long-duration operational capability and precise thrust control. Hall thrusters offer several advantages, including high propellant efficiency and specific impulse, a mature and flight-proven technology base, smooth and continuous thrust with precise control, and scalability from sub-kilowatt to multi-kilowatt power levels; however, they also face challenges such as discharge channel erosion from ion bombardment that limits operational lifetime, the need for complex magnetic field optimization to minimize electron losses and instabilities, and relatively low thrust levels that make them unsuitable for rapid maneuvers or heavy-lift missions. Recent developments in Hall thruster technology focus on alternative propellants such as krypton, iodine, and xenon–krypton mixtures to reduce cost and improve storage efficiency, magnetic shielding techniques to extend operational life by reducing wall erosion, high-power and clustered configurations for large spacecraft and interplanetary missions, and the advancement of miniaturized HET designed for CubeSats and small satellites. Hall Effect Thrusters represent a mature and efficient form of electric propulsion that bridges the gap between low-thrust, high-efficiency ion engines and

high-thrust, low-efficiency chemical rockets. By leveraging electromagnetic fields to generate and accelerate plasma, Hall Effect Thrusters have enabled new mission architectures and long-duration operations with minimal propellant consump-

tion. Continuous advancements in materials, magnetic topology, and power electronics are expected to further enhance their performance, making them a corner stone of next generation spacecraft propulsion systems. [7]

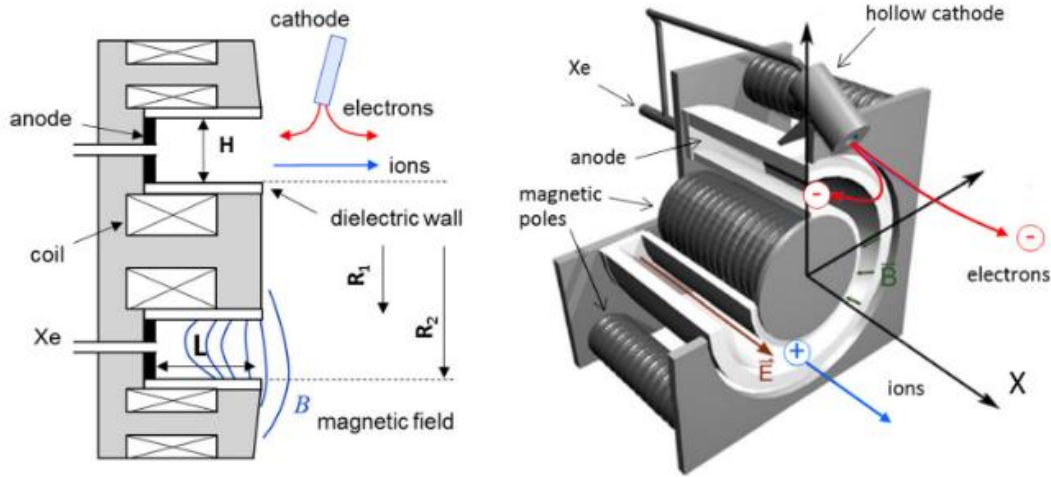


Figure 2. Hall Effect Thruster.

2. Methodology

2.1. Analysis

Let's calculate the orbital velocity of a satellite at an altitude of 1000 km above Earth's surface.

Altitude $h = 1,000 \text{ km} = 1.0 \times 10^6 \text{ m}$

Earth's radius $R_E = 6,371 \text{ km} = 6.371 \times 10^6 \text{ m}$

Gravitational constant $G = 6.674 \times 10^{-11} \text{ N} \cdot \text{m}^2/\text{kg}^2$

Earth's mass $M_E = 5.972 \times 10^{24} \text{ kg}$

Formula for circular orbital velocity

$$v = \sqrt{\frac{G \cdot M_E}{R_E + h}} \quad (1)$$

$v = 7356 \text{ m/s} \approx 7.36 \text{ km/s}$

let's find the gravitational acceleration (g_h) at 1000 km altitude above Earth's surface.

$g_0 = 9.81 \text{ m/s}^2$ (gravitational acceleration at Earth's surface).

Earth's radius (R_E) = $6.371 \times 10^6 \text{ m}$.

Altitude $h = 1.0 \times 10^6 \text{ m}$.

Gravitational acceleration decreases with the square of the distance from Earth's center.

$$g_h = g_0 \left(\frac{R_E}{R_E + h} \right)^2 \quad (2)$$

$g_h = 7.31 \text{ m/s}^2$

The drag force formula

$$F_D = \frac{1}{2} \cdot C_D \cdot \rho \cdot v^2 \cdot A \quad (3)$$

Were:

C_D = drag coefficient (≈ 2.2 for a typical satellite)

ρ = air density (kg/m^3)

v = orbital velocity (from previous calculation = $7,356 \text{ m/s}$)

A = cross-sectional area (m^2)

Atmospheric density at 1000 km, Typical density from the NRLMSISE-00 model is: $\rho \approx 10^{-15}$ to 10^{-12} kg/m^3

Let assume

$A = 0.5 \text{ m}^2$

$F_D = 3 \cdot 10^{-7} \text{ N}$

Table 2. Velocity Maneuvers for Different Operation Throughout Spacecraft Lifetime.

	Orbit Insertion ΔV	Station-Keeping / Orbit Maintenance ΔV	Attitude Control / Momentum Dumping ΔV	De-Orbiting ΔV
1 year	10 - 50 m/s	1 - 5 m/s per year	1 - 5 m/s per year	-
3 years	-	(3-15) m/s	(3-15) m/s	150 – 200 m/s.

Satellite operation time 3 years and consume (50–150 W),
took 150 W

Hall Effect Thruster Fundamentals

The main governing equations are:

Thrust (F):

$$F = \dot{m} V_e + (P_e - P_a) A_e \quad (4)$$

Specific Impulse (I_{sp}):

$$I_{sp} = \frac{V_e}{g_o} \quad (5)$$

Were:

V_e Effective Exhaust Velocity

g_o Standard Gravity

Power Requirement (P):

$$P = \frac{\dot{m} V_e^2}{2\eta} \quad (6)$$

Were:

η is the thruster efficiency.

$\dot{m}$ Mass flow rate (mass per unit time, e.g., kg/s).

V_e Exit velocity or jet velocity (velocity of the mass stream, e.g., m/s).

Determination of Required Thrust:

To maintain Orbit Insertion, station-keeping, Attitude Control and De-Orbiting the thrust must be overcome $\Delta V = 280$ m/s for 3 years.

$$T = F = \frac{m \cdot \Delta V}{t} \quad (7)$$

For a 75 kg satellite, and assuming $\Delta V = 280$ m/s:

$$T = F = \frac{75 \cdot 280}{365 \cdot 24 \cdot 3600 \cdot 3} = \frac{21,000}{94,608,000} \cong 2.22 \cdot 10^{-4} \text{ N} = 0.222 \text{ millinewtons (mN)}$$

Selection of Specific Impulse and Exhaust Velocity:

$$I_{sp} = 1,500 \text{ s}$$

$$V_e = I_{sp} \times g_0 = 1500 \times 9.81 = 14,715 \text{ m/s}$$

Propellant Mass Flow Rate:

$$\dot{m} = \frac{T}{V_e} = \frac{2.22 \cdot 10^{-4}}{14715} = 1.51 \times 10^{-8} \text{ kg/s}$$

Over three years:

$$m_{\text{prop}} = \dot{m} \times t = 1.51 \times 10^{-8} \text{ kg/s} \times (365 \cdot 24 \cdot 3600 \cdot 3) = 1.4285808 \text{ kg}$$

Power Requirement:

50% thruster efficiency:

$$P = \frac{\dot{m} V_e^2}{2\eta} = \frac{1.51 \times 10^{-8} \cdot 14,715^2}{2 \cdot 0.5} = 3.27 \text{ W}$$

The pressure inside the thruster is mostly dominated by plasma pressure, which can be estimated as:

$$P = \eta \cdot K_b \cdot T$$

Were:

η = plasma density (m^{-3})

K_b = Boltzmann constant (1.38×10^{-23} J/K)

T = plasma temperature (K)

Typical plasma densities in HETs are 10^{17} – 10^{19} m^{-3} , and electron temperatures are 10–50 eV (1 eV $\approx$ 11600 K).

$$n = 10^{18} \text{ m}^{-3}$$

$$T_e = 20 \text{ eV} \Rightarrow 20 \times 11600 \approx 2.32 \times 10^5 \text{ K}$$

$$P = \eta \cdot K_b T \approx (10^{18}) (1.38 \times 10^{-23}) (2.32 \times 10^5) = 3.2 \text{ Pa}$$

2.2. Component Design

2.2.1. Power Generation (Solar Array)



Figure 3. Triple-junction GaAs Solar Panel.

Table 3. Key Assumptions (Industry Typical) [8].

Parameter	Value (Typical)
Orbit	LEO (sunlit fraction ≈ 0.64)
Triple-junction GaAs efficiency	28–30%
Solar constant	1,358 W/m ²
Packing factor	0.9
Degradation	$\sim 2.5\%$ /year
Mission duration	3 years
Panel efficiency decay	$\approx 8\%$ total
Power budget margin	20–30%

Effective Solar Power per m²

Beginning of Life (BOL)

$$P_{\text{BOL}} = 1,358 \times 0.28 \times 0.9 \approx 342 \text{ W/m}^2$$

End of Life (EOL) (after degradation)

$$P_{EOL} = 342 \times 0.92 \approx 315 \text{ W/m}^2$$

Required Power from Solar Array

The satellite needs 150 W usable. let's apply a typical 25% power margin:

$$P_{req} = 150 \times 1.25 = 187.5 \text{ W}$$

$$\text{Required Solar Array Area } A = \frac{P_{req}}{P_{EOL}} = \frac{187.5 \text{ W}}{315 \text{ W/m}^2} = 0.595 \text{ m}^2 \approx 0.6 \text{ m}^2$$

This can be achieved with Two deployable wings of 0.30 m² each.

2.2.2. Electric Flow Control Valve

electronic switches and power-control devices that connect/disconnect the solar panel and battery to the Power PDB or PDU. Here are some of the common names and components used in spacecraft power systems. [4]

Table 4. Common Types / Names of Switches in Spacecraft Power Systems.

Remote Power Controller (RPC)	Solid-State Power Distribution Switch (MOSFET switch)	Battery Charge / Discharge Regulator	Main Bus Switching Unit (MBSU)
These are solid-state or electronic switches used in the PDU to turn on/off power lines. In NASA's modular power architecture, the PDU uses RPCs as the switching element + fault isolation.	Many satellites use high-side MOSFETs (or back-to-back MOSFETs) as their "switches" to connect/disconnect power. These switches can be designed with current-limiting, soft turn-on, and telemetry of load current.	These are power converters/regulators that manage how the solar array charges the battery (Battery Charge Regulator, BCR) and how the battery discharges to the bus (Battery Discharge Regulator, BDR) or PDU. These circuits also often include switching elements to isolate battery or solar power sources when needed.	On more complex spacecraft, the solar + battery bus is managed by a switching unit (e.g., MBSU) that controls which power source is connected to the main bus. These units can cross-tie different power strings or switch between power sources.

Components / Technologies to Use

MOSFETs: Very common for power switching in satellites, because they are light, efficient, and can be commanded electronically.

Protected MOSFET ("Smart Power Switch"): Integrated MOSFETs that include protection (overcurrent, thermal) and a gate driver.

Latching Relays: On some systems, mechanical relays or latching relays might be used, but more often now solid-state is preferred.

Current Limiters: Latching current limiters (LCL) are used in tandem with PDUs to protect lines. For example, in the PDU from TÜBİTAK UZAY, they use latching current limiters.



Figure 4. Rad-tolerant P-channel Mosfet for Space.

2.2.3. Rechargeable Lithium-Ion Battery Pack

Estimate eclipse time: At 1000 km altitude in LEO, an eclipse fraction might be 35–40% of the orbit (depending on inclination), [9] but for rough design let's say 35%.

For a circular orbit:

$$T = 2 * \pi \sqrt{\frac{r^3}{GM}} \quad (8)$$

Were

$$r = 6378 + 1000 = 7378 \text{ km}$$

$$GM = 3.986 \times 10^{14} \text{ m}^3/\text{s}^2$$

If orbit period is about 6300 sec, 105 minutes, 35% eclipse time is 36.75 min (0.6125 hr).

Energy needed during eclipse:

$$E_{\text{eclipse}} = 150 \text{ W} \times 0.6125 \text{ hr} \approx 91.875 \text{ Wh}$$

Add margin and depth-of-discharge (DoD):

You might design for a DoD of 30 to preserve cycle life.

With 30% DoD → need ~ 306.25 Wh of nominal capacity (because only ~30% of that is regularly used).

To convert watt-hours (Wh) to amp-hours (Ah),

$$Ah = \frac{Wh}{V} \quad (9)$$

Were

Ah: amp-hours

V: voltage

Let's take 12 V and Ah = 25.52 $\approx$ 26 Ah



Figure 5. Hermetic Sealed Battery AGM VRLA 12V 26Ah.

2.2.4. Propellant Feeding Components

Propellant Tank:

Mission Duration from Propellant Mass, $t = \frac{m}{\dot{m}} = \frac{1.4285808}{1.51 \times 10^{-8}} = 94,608,482.12 \text{ sec.} \approx 1095 \text{ days}$ of continuous operation. Decide Xenon Storage Pressure, Small systems commonly use supercritical xenon at: 150 bars (typical) at Xenon density $\approx 1.1 \text{ g/cm}^3$.

Calculate Required Tank Volume

$$V = \frac{m}{\rho} = \frac{1.4285808 \text{ kg}}{1100 \text{ kg/m}^3} = 1.3 \times 10^{-3} \text{ m}^3 = 1.3 \text{ L required}$$

xenon volume.

Choose Tank Geometry Spherical tank

$$\text{Volume formula: } V = \frac{4}{3} \pi r^3 \quad (10)$$

$$r \approx 0.067 \text{ m} = 6.7 \text{ cm} = \text{Tank diameter} \approx 13.4 \text{ cm}$$

Determine Wall Thickness

Use Ti-6Al-4V (space-standard xenon-compatible titanium): Allowable stress (with safety factor 2) $\sigma_{\text{allow}} \approx 450 \text{ MPa}$, Pressure: $P = 150 \text{ bar} = 15 \text{ MPa}$ and Radius: $r = 67 \text{ mm}$.

Thin-wall sphere formula:

$$t = \frac{pr}{2\sigma_{\text{allow}}} \quad (11)$$

$t = 1.12 \text{ mm}$; For manufacturing, 1.5 mm is typical.

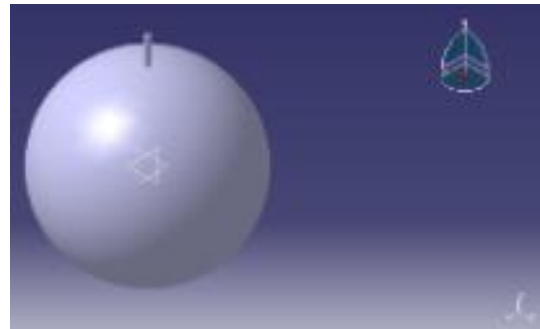


Figure 6. Propellant Tank with Extinction Pipe with the Same Thickness (1.5 mm) and 2.5 mm Hole.

Propellant Flow Pipe:

In a microsatellite electric propulsion system, (Hall thruster system), the xenon gas pipe is part of the propellant feed system. It is not a normal industrial pipe, spacecraft use small, ultra-clean, high-pressure-rated tubing built for reliability in vacuum and radiation. Typical Material, Xenon feed lines are usually made from stainless steel or titanium, chosen for strength, cleanliness, and compatibility with xenon: Common materials: 316 L stainless steel, 304 stainless steels and Titanium Grade 2 or Grade 5 (Ti-6Al-4V) for high strength and low mass. These materials: do not react with xenon, handle high pressure (100 – 300 bar) storage tanks, have extremely low outgassing and remain stable across wide temperature ranges (-60°C to $+120^\circ\text{C}$ or more). Dimensions, Xenon feed lines for microsatellites are small: Outer diameter (OD): 1–4 mm and Wall thickness: 0.1–0.5 mm. Thin-walled tubing keeps mass low while holding pressure. [10]

Propellant Flow Control Valve:

Precise Flow-Control Valve or Unit for $1.51 \times 10^{-8} \text{ Kg/S}$, Which Is $\sim 0.0151 \text{ Mg/S}$ Of Xenon (Assuming Xenon Density in Gaseous Flow Terms. That is very low compared to many typical Hall thruster controllers (which often operate in the mg/s range). So, you need a flow control unit (FCU) that supports very fine throttling or micro-flow regimes.

Table 5. Here Are Several Flow-control Devices or Systems That are (or could be) Suitable for Flow Rate, Plus Pros and Cons: [11].

AST μ FCU (Miniaturized Flow Control Unit)	CU Aerospace CAM Flow System	MEMS-Based Flow Control Module (ESA / Nano space)	SETS Xenon Feed System (XFS)
AST offers a μ FCU specifically designed for xenon.	CAM Flow is designed for very stable low flows and has been tested on a 600 W Hall thruster.	Under ESA's MEMS FCM project, they developed a MEMS flow control module that includes a micro proportional valve + flow sensor + pressure + temperature sensors.	SETS offers a xenon feed system with flow control.
According to their data, the μ FCU design can support a flow range from 0.01 sccm to 100 sccm.	Input pressure: up to 2,500 psi ($\sim 172 \text{ bar}$) supported, so high-pressure xenon tanks are pos-	The flow control range of their	Their XFS nominal flow rate: up to 3 mg/s for anode, 0.5 mg/s for cathode according to their
0.01 sccm of xenon is quite low; de-			

AST μ FCU (Miniaturized Flow Control Unit)	CU Aerospace CAM Flow System	MEMS-Based Flow Control Module (ESA / Nano space)	SETS Xenon Feed System (XFS)
pending on pressure you might be able to set up the system so that 0.015 mg/s is in its controllable region. Very lightweight: the EM model weighs ~62 g. Uses PWM (pulse-width modulation) solenoid for flow control. Has built-in filters (e.g., 5 μm) to protect against particulates. Limitation: their documented maximum “full scale” for some μFCUs is up to 100 sccm, but you might need to request a customized low-flow version tuned for your small flow.	sible. Flow control accuracy: $\pm 3\%$ demonstrated. Their architecture uses a “fixed frequency Boolean valve” (i.e., pulsing valves), which is very useful for long life and very fine mass control. This is likely a very good candidate, since your flow (~ 0.015 mg/s) is well within very low-flow regimes that CAM Flow might handle; but you will need to check with CU-Aerospace for the lower bound and any required calibration for that flow.	MEMS FCM is 1 to 18 mg/s for xenon, with high resolution. They report the ability to control very small flow changes — “changes around 50 μg/s” (which is 0.05 mg/s) demonstrated. Pros: very compact, integrated sensing + control, low power. Cons: might not go as low as 0.015 mg/s in standard versions; depends on the exact MEMS model. Also, EM-level, not necessarily fully flight-qualified for all uses (depending on your risk tolerance).	spec sheet. This is much higher than your 0.015 mg/s, so this might not be ideal unless you operate it in a very throttled regime or with a restrictor/orifice downstream. Advantages: mature system, likely robust for higher flow missions, but not optimal for ultra-low flow.

For any control valve, the required actuator power is:

$$P = F \cdot v \quad (12)$$

Were

F = force needed to move the valve (N)

v = actuator stem speed (m/s)

The force comes mainly from pressure acting on the valve area:

$$F = \Delta P \cdot A \quad (13)$$

Were

$\Delta P \approx$ line pressure (if no pressure balancing)

A = effective area the pressure pushes on

Estimate force for a 2.5 mm valve at 150 bar

Let's assume a small poppet/needle valve, orifice diameter = 2.5 mm.

Area, $A = \pi r^2 = \pi (1.25 \times 10^{-3})^2 \approx 4.9 \times 10^{-6} \text{ m}^2$

P = 150 bar = 15×10^6 Pa

Pressure force, $F = P \cdot A \approx (15 \times 10^6) (4.9 \times 10^{-6}) \approx 73.5 \text{ N}$

So the actuator must supply $\sim 75 \text{ N}$ of force (neglecting friction).

Estimate actuator power

Actuator speed depends on how fast the valve must move.

Example: if the valve stroke is 2 mm and needs to open in 50 ms:

$$v = \frac{0.002}{0.05} = 0.04 \text{ m/sec}$$

Power:

$$P = F \cdot v = 75 \times 0.04 = 3 \text{ W}$$

2.2.5. CPU for Microsatellite

A microsatellite CPU performs: Attitude control computation, Telemetry handling, Command & data handling, Payload control, Power system control, Fault detection & recovery (FDIR) and Communication protocol management. It must operate reliably in radiation, vacuum, thermal extremes, and sometimes with real-time constraints. CPU Requirements for Microsatellites, Microsatellite processors typically must have: Radiation tolerance: TID: 5–50 krad (minimum) and SEU/SEL mitigation techniques, Low power: 0.1–5 W typical, very limited cooling in space, High reliability: ECC memory, Redundant CPUs, Watchdog timers and Safe-mode processor, Communication interfaces: CAN, I²C, UART, SPI and Space Wire (some missions), Operating temperature: -40°C to $+85^\circ\text{C}$ or better. [12]

Table 6. Types of CPUs Used in Microsatellites, There are five Major Classes of CPUs.

Radiation-Tolerant Microcontrollers (most common)	Radiation-Tolerant Processors (high-end microsat)	COTS SBCs (Commercial ARM boards) – with protection	FPGA-Based Soft CPUs	Dual-CPU Architecture
Used in low-to-medium	Suitable for spacecraft with heavy ADCS or	Used in cost-sensitive missions (LEO, <3	Allows custom CPU logic with radiation	Common in mi

Radiation-Tolerant Microcontrollers (most common)	Radiation-Tolerant Processors (high-end microsat)	COTS SBCs (Commercial ARM boards) – with protection	FPGA-Based Soft CPUs	Dual-CPU Architecture
complexity missions. Examples: ARM Cortex-M7 / M4 / M33 (rad-tolerant versions) Microchip SAMRH71 (rad-hard ARM Cortex-M7) LEON3FT (fault-tolerant SPARC) Gaisler NOEL-V (RISC-V, rad-tolerant versions emerging)	payload demands. Examples: LEON4 / LEON3FT (Cobham/Gaisler) RAD750 (classic but high power) NanoXplore NG-Large / NG-Ultra FPGA CPU	years lifetime). Examples: ARM Cortex-A series Raspberry Pi Compute Module (with shielding & watchdogs) BeagleBone Black space-hardened versions	mitigation. Examples: Microblaze (soft-core) LEON3 soft-core RISC-V custom soft-core Used when payload requires flexible digital processing.	crossatellites: Examples: High-performance CPU (e.g., ARM A53) → main mission operations Low-power safe-mode MCU (Cortex-M or LEON3FT) → recovery, FDIR
Pros: Low power Safe for LEO microsatellites Easy software development	Pros: Very high reliability Used in many ESA/NASA missions	Pros: Very high CPU performance Low cost High memory Runs Linux easily		
Cons: Limited CPU power compared to SBCs	Cons: Expensive Higher power consumption	Cons: Low radiation tolerance (needs shielding + ECC + watchdog redundancy)		
Typical computing power: 100–2000 DMIPS				

Table 7. Example Microsatellite CPU Configurations.

Configuration A (Low-power microsat, ADCS + simple payload)	Configuration B (Medium-performance imaging satellite)	Configuration C (High-performance payload processing)
ARM Cortex-M7 rad-tolerant MCU 300–600 MHz 512 KB–2 MB RAM Redundant MCU + watchdog Bootloader in rad-hard memory	LEON3FT or LEON4 ECC SDRAM 256–1024 MB RTEMS or Linux OS Hardware redundancy (cold spare)	ARM Cortex-A53/A72 with radiation mitigation Runs Linux Accompanied by safe-mode microcontroller SDRAM with ECC Thick aluminum shielding

**Figure 7.** ARM Cortex-M7 CPU.

2.2.6. Power Distribution Board (PDB) for Microsatellite

A microsatellite PDB typically handles: Power routing (battery → loads), Voltage regulation (e.g., 3.3 V, 5 V, 12 V, unregulated bus), Protection (fuses, current limiting, latch-up protection), Power switching (on/off control to subsystems), Telemetry monitoring (voltage, current, temperature) and Communication with OBC (command + health data). [13]

Table 8. Key Subsystems on a Microsatellite PDB.

Power Switching Units (PSUs)	Voltage Regulation	Protection	Sensors	Communication Interface
Turn each subsystem on/off.	High-efficiency DC-DC converters: Synchronous buck converters Radiation-tolerant converters Efficiency 85–94%	Resettable electronic fuses Latch-up protection circuits Current limiting Reverse voltage protection Soft-start for sensitive loads	Each power line has: Voltage sense Current sense Temperature telemetry All sent to the OBC.	PDB normally communicates using: CAN bus (standard) I ² C (for CubeSat-type subsystems) RS-485 or UART (less common)

Table 9. Commercially-Available (or near-COTS) Power Distribution / PCDU (Power Conditioning & Distribution Unit) Boards for Microsatellites / CubeSats.

Product	Description / Key Features
GAUSS PCDU	From GAUSS S.r.l: a Power Conditioning & Distribution Unit with 4 regulated independent channels + 1 unregulated channel. PC/104 form factor. (Gaussteam) — Very configurable; each channel has switch, current limiter, telemetry (I ² C); up to 6 A on some outputs. (Gaussteam) — Flight heritage: used on UNISAT-7. (Gaussteam)
AAC Clyde Space – SmallSat PCDU (“STARBUCK-Mini”)	A very reliable, flight-proven PCDU. (AAC Clyde Space) They also have the STARBUCK-NANO EPS which includes power distribution modules with latching current limiters. (AAC Clyde Space) Provides regulated buses: 3.3V, 5V, 12V, etc., and supports up to 10 “switched” subsystems. (AAC Clyde Space)
SITAEL PCDU	Space-grade PCDU for small satellites / Microsats. (SITAEL S.p.A.) Designed for LEO, with power range ~300 W to ~1200 W. (SITAEL S.p.A.) Fully redundant for better reliability.
ISISPACE ICEPS2 (Compact EPS)	Their EPS is modular and has a separate PDU board (power distribution) in their architecture. (ISISPACE -) – Supports up to 6 voltage domains. – Very space-ready, off-the-shelf for small CubeSat missions.
GomSpace NanoPower P60 System	Modular power system: the “P60 Dock” motherboard plus PDU daughterboards. (GOMspace) Supports regulated 3.3 V / 5 V outputs (PC-104 compatible) for EPS and subsystem power.
2NDSpace SOLO EPS 8 + PCDU	The SOLO EPS 8 system includes 12 power outputs, configurable voltages (3.3, 5, 12V, etc.), latching current limiters, and multiple communications interfaces (I ² C, CAN, RS-422). (SatCatalog)
EREMS PDU / PCDU	EREMS offers a “PCDU NANO” unit for nano- / microsatellites. (EREMS) They also have modular converter boards based on GaN FETs. (EREMS)

2.2.7. Sensor

**Figure 8.** GAUSS PCDU PDB Board.**Figure 9.** Sensoron STIM300.

Orbital velocity must be *estimated* using: Accelerometers (IMU) → measure changes in velocity (Δv), GNSS/GPS receivers → give absolute velocity (when available) and Navigation filter (Kalman filter) → combines both. So what

you really need is a high-quality spacecraft accelerometer or IMU that can accurately measure *delta-V*. A high-quality IMU gives Δv accuracy of 1–5 mm/s for short burns. [14]

Table 10. High-Performance MEMS IMUs (Most Common for Microsats), These Measure Acceleration, Integrate to get Δv .

Sensoror STIM300 (Microsatellite-class, widely used)	SPP MRN-5X IMU	EMCORE / LITEF μ IMU series
Measures acceleration + angular rate	A MEMS IMU specifically designed for spacecraft	Low-drift MEMS IMUs
Very good for measuring small Δv	High stability accelerometers	Used for delta-V monitoring and navigation
Used in many small-spacecraft GNC systems	Good for precise Δv estimation	Suitable for microsats, low SWaP
Commercially available	Radiation-tested	

2.2.8. Electric Cable

In microsattellites, “electric cable” usually refers to space-grade wiring used for the satellite’s harness, connecting subsystems such as power, sensors, avionics, radios, and

actuators. Because satellites operate in vacuum, radiation, and large thermal swings, they use specialized materials and construction standards rather than ordinary commercial cables. [15]

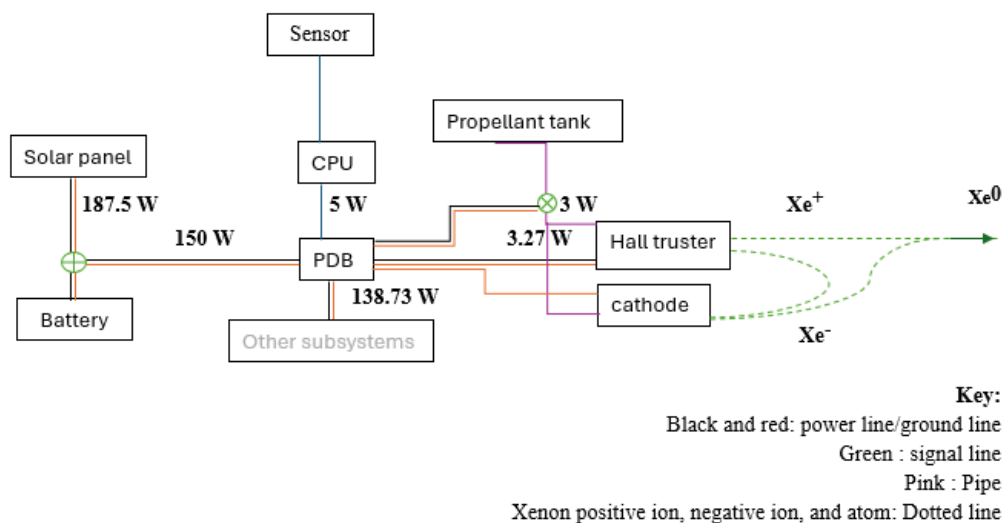


Figure 10. Electrical Power System (EPS) and Propulsion Subsystem.

Table 11. Common Types of Cable Used in Microsatellites.

Teflon-Insulated Wiring (PTFE, ETFE) these are the most common insulations for spacecraft harnesses:	Kapton-Insulated Wiring	Coaxial Cables for RF Links, used for antenna systems or high-frequency sensors:	Space-Qualified Flexible, Flat Cables (FFC/FPC)	Twisted-Pair or Shielded Cables Used for:	Power Harness Cables. Heavier gauge ETFE-insulated wires for:
PTFE (Polytetrafluoroethylene) excellent thermal stability (−65°C to +200 °C), radiation resistance. ETFE (Ethylene tetrafluoroeth-	Very lightweight Works across extreme temperature ranges	RG-316, RG-178 (fluoropolymer-insulated) High-reliability space-qualified coax with	Used in tight spaces or rotating mechanisms such as deploy-	Data buses (e.g., CAN, RS-422, Space Wire)	Battery power lines Solar panel connections

Teflon-Insulated Wiring (PTFE, ETFE) these are the most common insulations for spacecraft harnesses:	Kapton-Insulated Wiring	Coaxial Cables for RF Links, used for antenna systems or high-frequency sensors:	Space-Qualified Flexible, Flat Cables (FFC/FPC)	Twisted-Pair or Shielded Cables Used for:	Power Harness Cables. Heavier gauge ET-FE-insulated wires for:
ylene) – tough, lightweight, good abrasion resistance, often used in CubeSats. FEP another fluoropolymer with good thermal and chemical properties. Examples: MIL-W-22759 series (e.g., 22759/11, 22759/32) ESA ESCC 3901/002 wires	Popular for flexible cable assemblies Often used in flex harnesses or tape cables	silver-plated conductors	able arrays.	Noise-sensitive analog sensors Power lines requiring EMI reduction	Power distribution units (PDU)

Table 12. Component Order Specification.

No	Name	Model/Specification	Qty	Remarks (Customer /Cots)
1	Solar Array	Material: GaInP (Gallium Indium Phosphide) Panel Efficiency: 28–30% Panel Efficiency Decay: 8% Wing Area: 0.3 m ² Lifetime: 3 Years In-Orbit Thickness: 150 ± 20 µm	2	Customer
2	Lithium-Ion Battery Pack	Lithium-Ion: Rechargeable Energy Store Capacity: 306.25 Wh Discharge Capacity: 150 W Lifetime: 3 Years In-Orbit Working environment: Space Grade Brand: Hermetic sealed battery AGM VRLA 12V 26Ah	1	COTS
3	Hall Effect Truster	Thrust: 0.23 Mn Propellant Type: Xenon Lifetime: 3 Years In-Orbit Input Power: 3.27 W	1	Customer
4	Propellant Tank	Tank Shape: Sphere Inner diameter (tank): 6.7 cm Outer diameter (tank): 6.87 cm Thickness (tank): 1.5 mm Material (tank): Ti-6Al-4V Thickness of tank extinction: 1.5 mm hole of tank extinction: 2.5 mm	1	Customer
5	Flow Control Valves	Brand: CU Aerospace CAM Flow System Fully open diameter: 2.5 mm Withstand pressure: 150 bar	1	Customer
6	Valve Electricity	Brand: Rad-tolerant p-channel mosfet for space	1	COTS
7	Pipes	Material: Ti-6Al-4V		Customer

No	Name	Model/Specification	Qty	Remarks (Customer /Cots)
		Outer diameter (OD): 3.5 mm Inner diameter (ID): 2.5 mm Wall thickness: 0.5 mm		
8	Cable Line	Ptfe Wires Used For Power Lines Mil-W-22759/16 (Thicker Gauges, Awg 18–22)		COTS
		Ptfe Wires Used For Signal Lines Mil-W-22759/11 (Awg 24–30)		COTS
9	PDB	GAUSS PCDU	1	COTS
10	CPU	Arm Cortex-M7 (Rad-Tolerant Versions)	1	COTS
11	Sensor	Sensor Stim300	1	COTS

3. Result and Discussion

To correct the manipulate orbital velocity (7.36 km/sec) which is $\Delta V = 280$ m/s throughout the vehicle lifetime (3 years) 0.23 mN thrust force are generated from hall effect thruster. To fulfill such thruster xenon propellant must be proper 1.51×10^{-8} kg/s for 1095 days in 1.4285808 kg. From total power generated 187.5 W and the system need 150 W for continuous function from this only 11.27 (3 W for flow control valve, 3.27 W for hall effect thruster and 5 W for CPU) is used for propulsion system.

3.1. Limitation

This paper does not include the control section of the electric power and propulsion subsystem, nor does it address the design considerations for electronic components, engine size and mass, or thermal and environmental requirements. Additionally, it omits details regarding physical connectors and fittings between components.

3.2. Recommendation

Some Hall-effect thrusters with thrust levels below 1 mN are not readily available on the market, prompting ongoing efforts by researchers and companies to develop them. Further work is also needed on the control section, which should use the reference orbital velocity of the spacecraft, V_o (7.36 km/s), compare it with the actual velocity measured by onboard sensors, V_a , and compute the variation $V = V_o - V_a$; if V is equal to or greater than 4 m/s, the engine should activate until V becomes equal to or less than 0.

Abbreviations

HET	Hall Effect Thruster
EPS	Electrical Power System

MPPT	Maximum Power Point Tracking
PDU	Power Distribution Unit
COMMS	Communication Subsystem
C&DH	Command & Data Handling
OBC	On-board Computer
ADCS	Attitude Determination & Control System
TCS	Thermal Control System
MLI	Multi-layer Insulation
ADS-B	Automatic Dependent Surveillance–Broadcast
AIS	Automatic Identification System
BOL	Beginning of Life
EOL	End-of-life
SAA	Solar Array Assembly
LEO	Low Earth Orbit
DoD	Depth of Discharge
BMS	Battery Management System
GEO	Geostationary Earth Orbit

Conflicts of Interest

The authors declare that no conflicts of interest.

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Biography



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