

# Integration of Electric Propulsion Systems with Spacecraft An Overview

IEPC-2013-355

*Presented at the 33rd International Electric Propulsion Conference  
The George Washington University • Washington, D.C. • USA  
October 6 – 10, 2013*

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**Abstract:** The growing use of electric propulsion in the spacecraft community comes with the corresponding need for spacecraft designers and space systems engineers to understand how electric propulsion systems interact with the host spacecraft. A brief overview is presented of some key issues concerning the integration of electric propulsion systems onboard spacecraft.

## I. Introduction

Electric propulsion (EP) systems are increasingly used onboard spacecraft for exploration, scientific, military, and commercial missions. Recent high-profile exploration and scientific missions that have made use of EP include NASA's Dawn space probe exploring the asteroids Vesta and Ceres<sup>1</sup>, ESA's GOCE (Gravity field and steady-state Ocean Circulation Explorer) satellite providing high-resolution mapping of Earth's gravitational field<sup>2</sup>, and JAXA's Hayabusa sample return mission from the asteroid Itokawa<sup>3</sup>. For military missions, the first Advanced Extremely High Frequency military communications satellite successfully utilized EP for an unplanned orbit raising maneuver after failure of its liquid apogee engine in 2010.<sup>4</sup> Since Intelsat 502 launched in 1981 with hydrazine resistojets for North-South Station-Keeping (NSSK), over 200 commercial satellites have been launched with EP systems for station-keeping and attitude control applications<sup>5</sup>, and Boeing is presently building the 702SP, commercial industry's first all-EP satellite bus<sup>6</sup>. Around the world, various electric propulsion systems are currently being developed – ranging from high-power (*i.e.*, >10 kW) systems to support human and robotic solar system exploration efforts<sup>7-10</sup> to low-power (*i.e.*, <10 W) systems that provide nano-satellites<sup>11,12</sup> with primary propulsion and station-keeping capabilities.

Electric propulsion accelerates propellant via electrical heating, electrostatic forces on charged particles, or electromagnetic forces on plasmas.<sup>13</sup> Conventional chemical propulsion, whose energetics are limited by the propellant enthalpy, have specific impulses ( $I_{sp}$ ) on the order of several hundred seconds; in comparison, EP systems can generate specific impulses an order of magnitude higher. The resultant increase in propellant usage efficiency translates to less propellant needed to accomplish a desired propulsive capability by the spacecraft. This reduction in propellant needs translates to the following potential mission benefits:

- *Lower launch costs:* A smaller and cheaper launch vehicle may be utilized for the mission if sufficient propellant mass and volume savings are achieved on the spacecraft bus. Alternatively, multiple spacecraft may be stacked together onboard the original launch vehicle, thereby permitting increased payload to orbit and decreasing the per unit launch cost. This scenario is particularly attractive for commercial and Earth-observing satellite missions requiring the buildup of a constellation.
- *More useful payload mass per launch:* A spacecraft's design may trade propellant mass savings achieved with EP for additional payload capacity, thereby increasing data yield or revenue generation in the case of commercial missions.

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- *Increased on-orbit lifetime and operational flexibility:* Compared to conventional chemical propulsion, EP's ability to support more station-keeping and orbital maneuvers with the same propellant mass extends the on-orbit lifetime of the spacecraft and provides propellant margin for off-nominal or target-of-opportunity events.

For commercial satellites, an important consideration when using EP for primary propulsion (*e.g.*, orbit raising) is the longer time-to-orbit compared with conventional chemical propulsion. Satellites that utilize chemical propulsion to transfer from low Earth orbit (LEO) to geosynchronous orbit (GEO) can typically complete the maneuver in less than a month. However, the use of EP to accomplish the orbit transfer via a low-thrust, spiral trajectory can take several hundred days. This longer orbital transfer delays the onset of payload operations (*i.e.*, generation of data or revenue), and the spacecraft must also spend a longer duration at lower altitudes, with the accompanied greater risks of increased radiation exposure in traversing the Van Allen radiation belts and encounters with orbital debris at lower altitudes. To protect solar cells against the radiation environment, of which high-energy (*i.e.*, >10 MeV) protons trapped in the belts are the primary sources of damage, thicker cover glasses and oversized current and voltage capabilities must be used.<sup>14</sup> For other spacecraft components, greater shielding, more robust electronics, and additional protective structures must be used, all of which reduce the potential mass savings acquired by using EP. Nevertheless, net mass reductions of 25-35% are possible, as shown by Boeing's 702SP all-EP satellite compared with conventional satellite designs.<sup>6</sup> As onboard power availability continues to increase with developments in solar panel materials and architectures, the higher thrust capability afforded to EP systems makes them more desirable for primary propulsion applications.

These potential mission benefits have resulted in an increased pace of EP technologies being infused into space missions. Consequently, a growing need exists for mission planners, spacecraft designers, and space systems engineers to understand the interactions between EP systems and the other subsystems onboard the host spacecraft. While by no means comprehensive, the spacecraft integration issues presented in this paper highlight some key EP interfacial areas of which EP users should be aware.

## II. The Electric Propulsion System

As shown in Figure 1, a general EP system is composed of the thruster assembly, the power processing unit (PPU), and the propellant feed system.

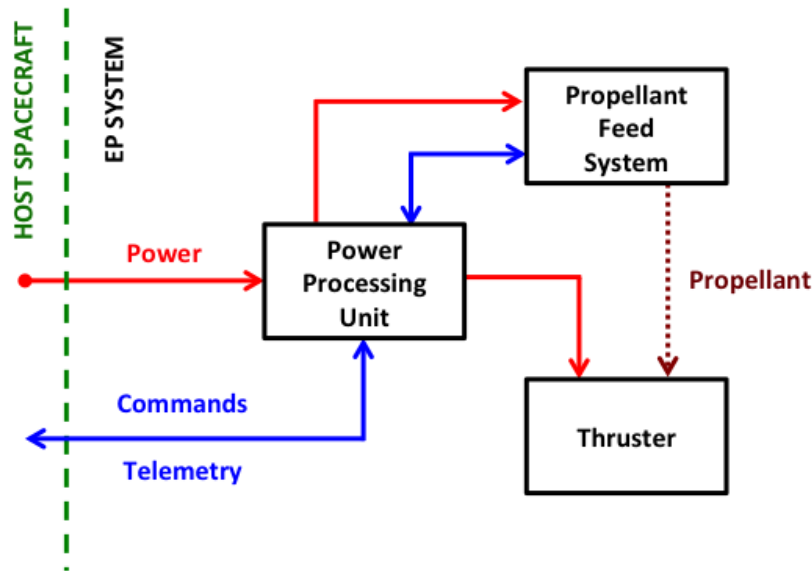


Figure 1: General EP system block diagram.

### A. Thruster Assembly

The thruster assembly, via electric and magnetic circuit elements that energize or manipulate the propellant flow, imparts kinetic energy to the propellant reactive mass for thrust generation. An overview of state-of-the-art thruster technologies is presented in Ref. 15. A mechanical gimbal assembly may be integrated with the thruster assembly to permit off-axis thrust generation; integration issues associated with the gimbal assembly are discussed in Section V.

Note that unlike electrothermal or electromagnetic thrusters, in which the thruster exhaust plume is charge-neutral (electrothermal) or quasi-neutral (electromagnetic), electrostatic thrusters generate thrust via an energetic beam of typically positively-charged particles. Consequently, electrostatic thrusters such as gridded ion engines and Hall effect thrusters require a neutralizer cathode to maintain overall spacecraft charge balance. Such cathodes are susceptible to contamination and generally require high-purity propellants with greater associated processing costs to avoid cathode poisoning; the long warm-up times needed to condition hollow cathodes prior to on-orbit thruster firing along with oscillations induced via plume interactions (Section III) must also be factored into mission operational plans.<sup>16</sup>

## **B. Power Processing Unit**

The PPU does the following:

- Converts incoming source power to the voltages and currents required for thruster operations
- Receives and relays thruster operational commands from the host spacecraft
- Provides operational telemetry to the host spacecraft
- Protects the power electronics from thruster-induced electromagnetic interference (EMI)

Conventional EP systems utilize source power from solar arrays, with power from onboard batteries used to meet thruster load demands during orbital eclipse periods. Some key spacecraft integration issues associated with the PPU are discussed in Section IV.

## **C. Propellant Feed System**

This system consists of the propellant tank, pressure regulators for maintaining proper tank and line pressures, and flow controllers for delivering the required propellant mass flow rates to the thruster. State-of-art ion and Hall effect thrusters are operated with xenon propellant, an inert gas with large atomic weight and ionization cross section. Xenon is costly – with typical prices in excess of \$4000/kg – and requires storage at either high-pressure or cryogenic conditions onboard the spacecraft. An ongoing effort is underway to investigate the feasibility of using cheaper, condensable propellant that may be stored in irregularly shaped reservoirs that maximize spacecraft volume utility.<sup>17,18</sup>

# **III. Thruster Plume Effects**

The near-field environment about an operating EP thruster consists of complex interactions among the thruster effluents, the spacecraft, and the local space environment. References 19 and 20 provide a comprehensive look at the particle and field effluent effects present in EP systems. The five major effluent types identified are summarized below along with their implications for spacecraft integration.

## **A. Primary Beam**

In electrostatic thrusters, the primary beam is composed of charged particles accelerated by the thruster to generate directed thrust. These charged particles, upon exiting the thruster, follow nearly straight-line trajectories as electromagnetic fields in the beam are generally too weak to perturb their path. Well-designed thrusters aim to minimize the expansion cone formed by the beam particles so as to reduce off-axis thrust losses. Typically, the divergence half-angle is ~15 degrees and >25 degrees for ion thrusters and Hall effect thrusters, respectively.

Spacecraft surfaces impacted by the beam cone can experience sputtering damage from the energetic particles. Typically, potential erosion damage is mitigated by shielding sensitive components from direct line-of-sight of the beam. Alternatively, the thrusters can be located or gimballed such that the plume's expansion cone does not intersect sensitive surfaces. This approach is of particular importance with regard to solar panels, as prolonged exposure to the thruster plume strips away the MgF<sub>2</sub> anti-reflective coatings on the cover glass, thereby decreasing array efficiency.<sup>21,22</sup>

In general, the high plasma densities in ion and Hall effect thruster plumes are sufficient to attenuate and refract transmissions below a few hundred megahertz.<sup>23</sup> However, most spacecraft communications and data frequencies are sufficiently high (*i.e.*, >1 GHz) such that these signals are generally unaffected other than experiencing small phase shift distortions due to the plasma.

## **B. Neutral Propellant Efflux**

Uncharged or unionized propellant leaves the thruster at thermal velocities corresponding to thruster wall temperatures. This neutral propellant is emitted in a cosine distribution in straight-line trajectories, and charge-

exchange (CEX) interactions with the beam particles can result in high-speed neutrals. In ion and Hall effect thrusters, hollow neutralizers provide an additional source of neutral propellant efflux. If the propellant is chemically reactive or electrically conductive, impingement or deposition of neutral propellant on sensitive spacecraft surfaces are undesirable due to possible chemical, metallurgical, or conductivity changes. This surface contamination is typically mitigated via line-of-sight shielding or appropriate thruster positioning.

### **C. Low-Energy Plasma Efflux**

Low-energy ions are formed in CEX interactions between beam particles and neutral propellant. In ion and Hall effect thrusters, CEX ions are generally formed just downstream of the thruster exit in a region of high beam and neutral particle density. These CEX ions readily respond to the local electric field and can cause backstream sputtering of ion engine grids, ion impingement on surfaces not in line-of-sight of the beam, and parasitic current losses to high-voltage solar arrays.<sup>24</sup> The expansion of CEX ions about the spacecraft can also lead to sputtering, deposition, and charging of sensitive spacecraft components upstream of the thruster. Understanding the distribution of CEX ions about a spacecraft is thus a critical matter in mitigating spacecraft contamination. In ion and Hall effect thrusters, the hollow cathode neutralizers provide an additional source of low-energy plasma efflux.

### **D. Non-Propellant Efflux**

Thruster materials ejected via beam or CEX sputtering compose the non-propellant efflux. These particles are generally uncharged and travel in straight-line trajectories, but a fraction may undergo CEX or electron impact interactions. Because of the material's low vapor pressure, impingement of non-propellant efflux on a spacecraft surface will generally result in material deposition, thereby affecting the surface's optical and electrical properties. Deposition is heaviest at angles of 30 to 70 degrees from thruster centerline; shallower angles experience less deposition due to sputtering effects from the primary beam.

### **E. Electromagnetic Field Efflux**

Steady-state or fluctuating electromagnetic fields, including optical emissions from the plasma plume, are present during EP thruster operations. Sensitive payloads (*e.g.*, magnetometers) must thus either be used in a time-share manner with thruster operations or placed on booms in regions of lessened field intensity so as to avoid interference. References 25 and 26 provide a comprehensive survey of electromagnetic emissions from electrothermal, pulsed plasma, ion, and Hall effect thrusters in flight, and on-orbit testing of spacecraft equipped with SPT-100 Hall effect thrusters have not shown discernable interference on the S, C, X, Ku, and Ka bands<sup>27</sup>. As higher-power EP thrusters become operational, the resultant higher-intensity electromagnetic fields will need to be evaluated. Potential issues can be mitigated by time-share operations or appropriate placement of antennas and the thruster to ensure a wide angular separation between the thruster exit and the antennas' lines of sight.<sup>25</sup>

## **IV. Power Subsystem Considerations**

All EP systems rely on external power sources to impart energy for propellant acceleration. For the near term, this external power source will primarily be in the form of solar arrays onboard the spacecraft, with emerging concentrator arrays projected to generate >100 kW at better than 7 kg/kW.<sup>28,29</sup> Because an active EP system can consume up to 90% of the total power generated onboard the spacecraft,<sup>14</sup> power allocation to support other spacecraft systems must be carefully managed. EP spacecraft with batteries supporting the payload must balance the need to meet propulsive requirements without unduly increasing battery capacity; this tradeoff is particularly relevant for near-Earth missions that experience regular eclipses, for which a LEO satellite may require 25-40% of the solar array output to recharge batteries.<sup>30</sup>

In a traditional EP system architecture, power from the solar array is conditioned by the PPU to match the thruster's load requirements. The PPU also provides timing and sequencing for thruster and propellant management commands, performs electrical fault protection for the thruster and spacecraft, conducts electrode-cleaning operations, and relays telemetry on the EP system's performance and health.<sup>31</sup> State-of-the-art PPU designs can achieve 95% efficiency, but as higher-power EP systems become operational, substantial waste heat from both the thruster and the PPU are generated and must be managed. To avoid impacting the thermal balance of the payload, dedicated radiators are frequently employed on GEO satellites for just the EP system.

### **A. Direct-Drive EP**

Some EP thrusters may be operated directly off the solar array voltage.<sup>32</sup> In these direct-drive EP systems<sup>33-35</sup>, the PPU can be simplified into a direct-drive unit (DDU). With emerging concentrator arrays capable of 300-V array

voltages without plasma-induced arcing in the LEO ionosphere<sup>36</sup>, direct-drive Hall effect thrusters become a possibility. However, other EP thrusters that require larger acceleration voltages (*i.e.*, >1 kV) such as ion or electrospay thrusters would not be usable as direct-drive systems.

The DDU for these Hall effect thrusters will not require voltage conversion for sustaining the discharge, thus decreasing subsystem mass and waste heat generation; as such, DDU designs are projected to be half the mass of conventional PPUs with up to 99% efficiency.<sup>33</sup> Not all voltage conversion can be eliminated in the DDU, however, as auxiliary power supplies are still needed to ignite the plasma discharge and operate the magnetic coils. Both PPUs and DDUs also require an input filter to isolate thruster discharge fluctuations from the solar array. Without a regulated discharge power supply, a direct-drive thruster's acceleration potential will change with the solar array voltage. Consequently, thruster performance will vary during the mission with distance from the sun along with off-nominal conditions<sup>37</sup>, resulting in added complexity to the mission planning process.

## B. EP System Grounding

Two architectures are used for placing the EP system's common voltage with respect to the spacecraft's structure ground.<sup>38</sup> Figure 2 shows the architectures as applied to a Hall effect thruster. In the fixed EP ground architecture shown in Figure 2(a), the EP system's common voltage is directly tied to the spacecraft's structure ground via a bonding strap. In this scenario, the spacecraft's structure ground has a higher potential than the neutralizer's cathode potential. Consequently, the cathode electrons not only neutralize the beam ions in the plume but are also lost to the spacecraft structure as leakage current that impacts spacecraft charging.

To mitigate these issues, Figure 2(b) shows an alternative grounding architecture in which the EP system's common voltage is kept floating above the spacecraft structure ground via a floating ground control element; this control element, being either a bleeder resistor, a coupling capacitor, or clamping diodes, is used to limit the range that the EP system's common voltage can float. A typical PPU is designed with the floating-ground architecture for PPU versatility, as converting the floating-ground PPU to a fixed-ground architecture if needed is much easier than doing the reverse.<sup>38</sup>

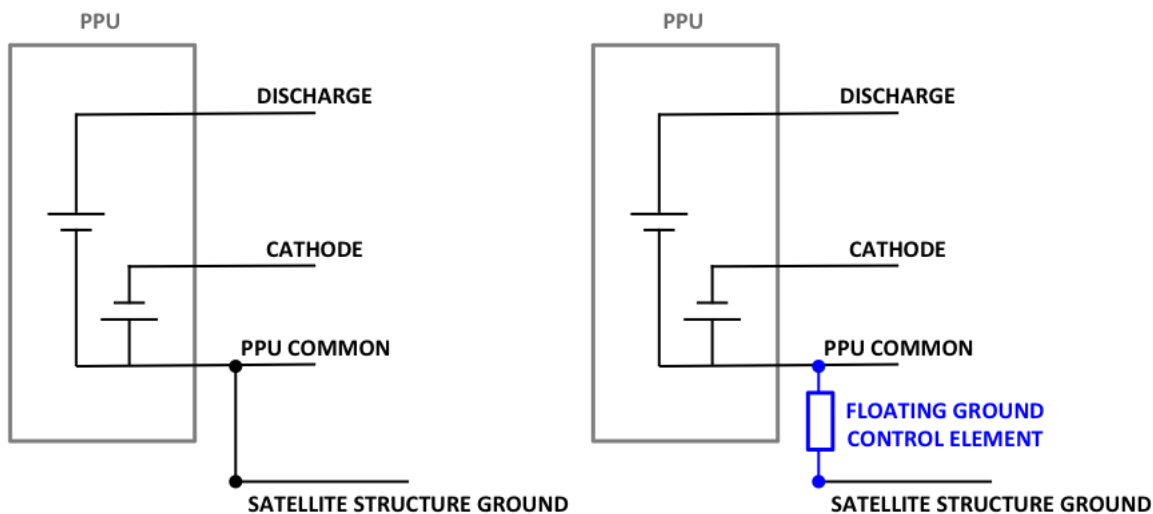


Figure 2: EP system grounding architecture for Hall effect thruster. (a) Fixed EP ground. (b) Floating EP ground.

## V. Thruster Gimballing

Because of their large surface area, long moment arms, and locations near the thruster plume, solar arrays can generate significant torques on the host spacecraft when impacted by energetic beam particles.<sup>27</sup> These torques must be compensated for by the spacecraft's attitude control system, with reaction wheels requiring eventual momentum offloading via thruster firings. Furthermore, spacecraft with ion and Hall effect thrusters experience inherent thruster swirl torque that must also be addressed with the spacecraft's attitude control system.

In the past, this operation was performed onboard EP-spacecraft with a separate cold gas or chemical propulsion system.<sup>39-42</sup> To perform the same operation using just the onboard EP system, thereby eliminating the complexity

and mass associated with an auxiliary attitude control thruster pack, multiple EP thrusters can be positioned about the spacecraft to provide control torques about all three axes. By providing a two-axis gimbal to the thruster, the number of EP thrusters needed for momentum control can be reduced, since each thruster can now provide control torques perpendicular to its nominal thrust axis. Randolph *et al.* has since proposed a simpler architecture in that a single EP thruster on a dual-axis gimbal, by using mirror image maneuvers, can be used to manage the angular momentum about all three axes.<sup>43</sup> Being able to dump the momentum about the nominal thrust axis in this manner also has the added benefit of being able to offset the swirl torque observed in Hall effect thrusters<sup>44</sup> gridded ion engines.<sup>45</sup>

Special care must be taken to minimize resistive torques from thruster wiring harnesses and propellant plumbing acting against the gimbal. For example in ESA's SMART-1 spacecraft, the xenon propellant lines routed through the gimbal mechanism to the Hall effect thruster were helical in shape to reduce resistive torque; however, the reduced stiffness in such a design necessitated a careful tradeoff against the increased susceptibility to vibrational damage during launch.<sup>46</sup> In order to protect SMART-1's thruster gimbal mechanism, with a maximum allowable temperature of 75 °C, from the thermal loads of the thruster, proper thermal management necessitated installation of a white-paint radiator on the gimbal mounting ring along with low thermal conductivity standoffs and goldized reflector.<sup>46</sup>

## VI. Conclusion

A brief overview of some key spacecraft integration issues with EP systems is presented above. Compared to conventional chemical propulsion systems, EP systems present a host of unique interfacial considerations. As operational EP systems become more prevalent in scientific, civilian, and military space missions, understanding such issues will become increasingly important to facilitate mission success. The discussion in this paper is by no means comprehensive, and the reader is invited to explore additional relevant issues such as those concerning the long operational lifetimes (*i.e.*, thousands to tens of thousands of hours) of EP systems and regarding the methodologies and facilities necessary for proper ground-based acceptance and qualification testing of EP systems.

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