Electric Propulsion

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Electric propulsion was first developed in the 1960's by both the United States and The Soviet Union. BY 1971 The Soviet Union were regularly using electric propulsion aboard their satellites. It took the United States until the 2000's to do so and electric propulsion system are now becoming industry standard for all communication satellites around the world. Currently the American, European, Indian, and Russian Space agencies have all placed satellites into orbit with functional electric propulsion systems

Nomenclature

A	=	Thruster open area fraction
$\mathbf{B}$	=	Magnetic Field
$\bar{c}$	=	Thermal gas velocity
d	=	Grid spacing
$\boldsymbol{E}$	=	Electric field
e	=	Electron charge
g_0	=	gravitational acceleration (9.806 m/s^2)
I_b	=	Ion current
Isp	=	Specific impulse
$\boldsymbol{J}$	=	Current density
J_i	=	Ion current density
k	=	Boltzmann's constant
k_e	=	Coulomb's constant
M	=	Ion mass
m_d	=	Delivered mass
m_p	=	Propellant mass
n	=	Number density (neutral gas)
N	=	Number of particles
n_i, n_e	=	Ion plasma density, electron plasma density
P	=	Pressure
P_0	=	Required power input to create thruster beam
P_b	=	Beam power
P_d	=	Ion production power
P_{in}	=	Input power
P_T	=	Total power
q	=	Ion charge
Q	=	Throughput
q_1, q_2	=	Charge
r	=	Radius
S	=	Pumping speed
T	=	Temperature
T	=	Thrust
$\mathbf{v}$	=	Velocity
V	=	Volume

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V_b	=	Acceleration voltage
v_e, v_i, v_{el}	=	Exhaust velocity, ion exhaust velocity, electron velocity
Δv	=	change in velocity, delta-v
α	=	Thrust correction factor
α_m	=	Mass utilization efficiency correction factor
ε_0	=	Permittivity of free space
η_c	=	Conductance corrective factor
η_d	=	Discharge loss
η_e	=	Electrical efficiency
η_m	=	Thruster mass utilization efficiency
η_T	=	Thruster efficiency (overall)
ρ	=	ion charge density

I. Introduction

Whenever electricity is used to in increase the exhaust velocity of propellant gases an electric propulsion system is being used. Electric propulsion (EP) relies on creating thrust with high exhaust velocity. Among the primary factors of concern are thrust, specific impulse, and overall efficiency. The efficiency of an electric thruster is the ratio of power from jet (which is created by the thrust beam) divided by the electrical power of the system. The main advantages of EP system are specific impulse values in the range of thousands to tens of thousands of seconds as well as the reduced propellant and thruster system mass. The decrease in thruster and propellant mass lead to smaller payload mass. This directly impacts the launch mass and the cost of launching payloads[1].

One brief subsection of EP to mention is when nuclear fusion is used as a power source aboard a spacecraft. SNAP-10A also known as SNAPSHOT was the only nuclear fission powered satellite launched in the United States. It was launched on April 3,1965 while Lyndon Johnson was President[2]. This system was designed to operate at an average of 500 W for slightly more than one year. However, 43 days into operation the voltage regulator malfunctioned and the fission system was shut down[3]. The most powerful nuclear rocket ever launched into orbit was the TOPAZ 1 reactor aboard the Cosmos 1818 satellite in February 1, 1987 and the Cosmos 1867 on July 10, 1987. Both satellites were launched in the Soviet Union under the leadership of Mikhail Gorbachev[2]. These satellites served as radar ocean reconnaissance satellites. There were a total of 33 RORSATs launched by the Soviet Union which used nuclear reactors as power systems[4]. Nuclear electric propulsion will no longer be referenced in the paper to focus on what traditionally are considered EP systems.

EP system can be generally categorized into three areas: electrothermal, electrostatic, and electromagnetic. In appendix A, Table 1 a list of EP thruster and operating parameters can be found. In an electrothermal system a traditional propellant is heated by an electric source, which increases the exhaust velocity of the propellants and provides a slight increase to the specific impulse. Electrostatic thrusters are defined as system which primary cause of acceleration is the Coulomb force. The Coulomb force between two particles is defined as $F = \frac{k_e q_1 q_2}{r^2}$ (1) This can also be interpreted to mean that a static electric field is applied in the direction of accelerations. Electromagnetic propulsion systems are those in which acceleration is attributed the Lorentz force. The Lorentz force is represented as $F = q\boldsymbol{E} + q\boldsymbol{v}\times\boldsymbol{B} = \boldsymbol{J}\times\boldsymbol{B}$ (2).[1]

The main class of electrothermal EP systems can be divided into two categories. These two categories are resistojets and arcjets. Resistojets are comprised of a resistively heated chamber through which propellant passes and is heated. This increase in heat transferred from the bed to propellants increases the exhaust velocity of the propellants. Resistojets are typically limited to specific impulse values of less than 500 s. In arcjets, propellants pass through high current arcs which are collinear with the nozzle feed system. A plasma is created where propellants are in the path of the arcs but leads to weak ionization and an insignificant effect which can largely be ignored. Arcjets are generally limited to specific impulse values less than 700 s.[1]

There are several types of electrostatic EP thrusters. The most common three are ion thrusters, hall effect thrusters (HETs), and electrospray thrusters. Ion propulsion systems ionize a propellant and accelerate the ions through electrically biased grids to very high exhaust velocity. HETs use the Hall Effect to generate a plasma. An electric field which is perpendicular to the applied magnetic field is the cause of ion acceleration in this case[1].

Electrospray thrusters typically produce less than 1 mN of thrust and can be subdivided into three categories. Colloid thrusters accelerate charged droplets, field emission electric propulsion (FEEP) thruster use liquid metals such as cesium or indium and create metallic ions, finally ionic liquid ion source (ILIS) thrusters use room temperature molten slats to produce ion salt beams. Electrospray thrusters are used for precision control of spacecraft and are a leading contender for CubeSat propulsion systems[1,8].

Electromagnetic thrusters the third main category of EP. Some types of electromagnetic thrusters are pulsed plasma thrusters (PPT), magnetoplasmadynamic (MPD) thrusters, electrodeless plasma thrusters, and the variable specific impulse magnetoplasma rocket (VASMIR). A PPT operates by using a pulsed electrical arc to ablate solid propellant which turns into a plasma. The plasma is then propelled between charged plates where the ions and electrons within the plasma complete the electric circuit between the charged plates. The current within the plasma induced a magnetic field and the Lorentz force is observed to accelerate the plasma to high exhaust velocities. The rate at which the pulse is repeated determines the overall thrust.[1,10] Simply, a VASMIR thruster uses two radio frequency couplers and a magnetic nozzle to create a plasma and direct it in a jet, this will be discussed in more detail later.[11] A MPD thruster uses a high current arc of electricity to ionize propellant and produce a plasma. The Lorentz force within the plasma discharge leads to an acceleration of the propellant. MPD thrusters are designed as a middle ground between traditional chemical rockets which have high thrust and low specific impulse and electric propulsion system which have high specific impulse and low thrust levels.[1] A MPD thruster developed at NASA Glenn which used hydrogen as a fuel source and operates at a power level of one MW, demonstrated exhaust velocities of 100,000 m/s and a thrust of 100 N.[12] While this MPD produces relatively high thrust compared to most electric propulsion systems and a specific impulse of over 10,000 s, it can easily be seen that power limitations are likely to limit the performance aboard a spacecraft. Another electromagnetic EP system is the electrodeless plasma thruster. This functions by using microwave radiation to accelerate propellant. This type of thruster was developed by Dr. Gregory Ensellem of The Elwing Company but is only in the prototype stage. It is known as the Electrodeless Ionization Magnetized Ponderomotive Acceleration Thruster (E-IMPAcT).[13]

II. A Brief History

The first EP systems were developed in the 1960s. Both the United States and the Soviet Union developed and launched EP systems into space to be tested in 1964. The U.S. tested an ion thruster while the Soviets tested a PPT. In 1995 Japan launched and tested their first successful EP system, however a launch vehicle failure limited the use of the ion thruster before the satellite went out of operation. Japan has since demonstrated use of ion propulsion during the JAXA Hayabusa asteroid sample return mission which launched on May 9, 2003. On September 27, 2003, the European Space Agency (ESA) launch SMART-1 to the moon which used a HET. On May 5, 2017, the Indian Space Research Organization (ISRO) launched (GSAT-9) aboard the Geosynchronous Satellite Launch Vehicle (GSLV) which will be the first satellite using an EP system for India. [1,15,16,17]

The Space Electric Rocket Test (SERT-1) was launched into space on July 20, 1964 by NASA. SERT-1 contained a cesium contact ion engine and a mercury electron bombardment ion engine which were successfully operated in orbit for 31 minutes[2]. The engine operated at 1.4 kW and produced a thrust of 28 mN. While operating at these specifications, the specific impulse of SERT-1 was approximately 4900 s[6]. In 1997 Hughes Aerospace launch a Xenon Ion Propulsion System (XIPS). This propulsion system was used aboard the PanAmSat-5 (PAS-5) which was a communication satellite.[1, 14] This marked the first commercial use of EP in the United States.[1] The next time an ion engine was by NASA, was aboard Deep Space 1 (DS1). The satellite launched on October 24,1998 with the primary mission of completing a flyby of asteroid 9969 Braille. DS1 had a NASA Solar Technology Application Readiness (NSTAR) electrostatic ion thruster. This marked the first time an ion propulsion system was used aboard a NASA spacecraft[6]. The NSTAR was designed to operate over a range of 500 W to 2300 W while providing a thrust of 19 mN – 92 mN. The specific impulse is 1900 s to 3100 s respectively. The burn time for the NSTAR for DS1 to flyby 9969 Braille occurred from October 24th 1998 to April 27th 1999 for a total of 1,764 hours. During the flight, the power level varied between 480 W to 1,940 W[7].

What inspired the United States to use EP systems on spacecraft can largely be linked to the Soviet Union. After 1991 when the Soviet Union collapsed, it was found that Russian scientists had been using HETs on satellites successfully since 1971[1]. However, the first known use of electric propulsion by the Soviet Union was a PPT aboard Zond 2 which launch November 30, 1964. In total six PPT were used aboard Zond 2 for attitude control and were used for a total of 70 minutes.[9,10] In Russia HETs are referred to as Stationary Plasma Thrusters (SPTs). The first test of an SPT in the Soviet Union occurred in 1971 or 1972 when one launched aboard the Meteor Satellite was operated successfully for over 170 hours. The delta-v provided by the SPT to the spacecraft allowed for an orbital change into sunsynchronous orbit.[21] For the next 20 years SPTs were used aboard Russian spacecraft. In 1991 with the fall of the Soviet Union the United States was exposed to the soviet history of successfully using HETs aboard spacecraft. It was then the project of Space Systems Loral to develop a model similar to the SPT-100 and provide enough reliability that it could be certified in western countries and installed on spacecraft and satellites.[21,22] The operating parameters of multiple SPTs developed in Russia can be found in Table 2 in Appendix A.[21]

Japan launched the Hayabusa asteroid sample return mission in 2003. The satellite used multiple cathode-less electron cyclotron resonance ion engines. By rendezvous with the Itokawa asteroid, a delta-v of 1,4000 m/s had been developed of a cumulative 25,800 hours. The system only required 22 kg of Xenon to achieve this delta-v on the outgoing mission to the asteroid. By the completion off the return mission, the ion engines aboard the Hayabusa had operated for a combined 35,000 hours and consumed a total of 40 kg of Xenon propellant.[18]

ESA launched SMART-1 in the fall of 2003. The result of the mission was a deliberate impact on the moon's surface on September 3, 20006. The primary propulsion system aboard the spacecraft was the PPS-1350G Hall Effect Thruster. This was the first instance of ESA using EP as a primary propulsion system. It is also the first documented case where EP was used to transfer out of geostationary orbit (GEO) into trans-lunar injection.[18]

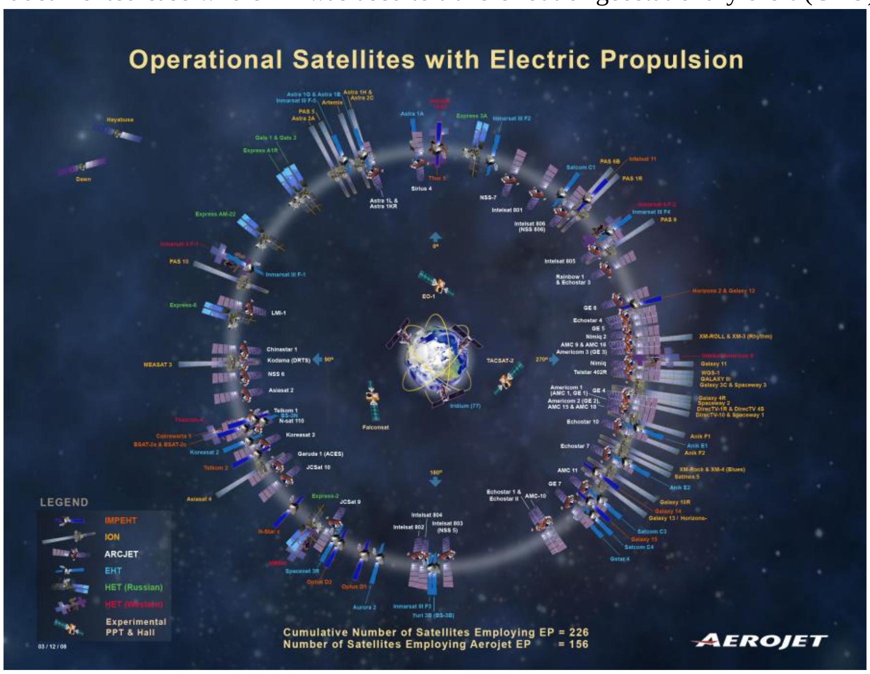

Figure 1: Operational Satellites using EP in 2008[19]

With the launch of GSAT-9 by ISRO in May 2017, India has joined the list of countries that have used EP aboard satellites. The GSAT-9 was able to launch with a total of 200-3000 kg of propellant which is an estimated 25% of the propellant that would have been required if a traditional chemical propulsion system were used. It also led to a decrease in payload mass from 5,000 kg to approximately 3,500 kg.[17] Using the lowest price per kg of launch of $2,800 for a Falcon 9 launch, the EP system on this satellite alone reduced the launch cost by $4.2 million.

In 2008 Aerojet Rocketdyne released Figure 1, which showed the current operational satellites which used EP systems. In total 226 satellites are shown in the figure. Of the 226 satellites listed, 156 are using EP developed by Aerojet. Among the types of EP currently in use are HETs both manufactured to western and Russian standards. Ion engines, arcjets, electrothermal hydrazine (EHT), improved electrothermal hydrazine (IMPEHT), experimental pulsed plasma thrusters, and experimental hall effect thrusters. It is easily observed by Figure 1; how much popularity electric propulsion system have gained in the United States and other western countries since the 1990's when the Russians were able to share their knowledge and history of successfully using HETs since the 1970's aboard spacecraft. [19]

III. Operational Principles

The thrust of a rocket engine can be generally described by $T = -v_e \frac{dm_p}{dt}$ (3). It can then be shown that the change in velocity which can be achieved by a spacecraft is $\Delta v = \left(I_{sp} * g_0\right) * \ln\left(\frac{m_d+m_p}{m_d}\right)$ (4) where the delivered mass is defined as $m_d = \left(m_d + m_p\right)exp^{-\Delta v/v_e}$ (5). Propellant mass and delivered mass then exhibit the following relationship

$m_p = m_d \left[exp^{-\frac{\Delta v}{v_e}} - 1 \right] = m_d \left[exp^{-\frac{\Delta v}{I_{sp} * g_0}} - 1 \right]$ (6), propellant mass must therefore increase exponentially with the delta-v requirements of a specific mission. From the second half of equation (6) we observe that the propellant mass can be limited with an increase in specific impulse. Ion thruster can achieve exhaust velocities of approximately 20-40 km/s while hall effect thruster can produce exhaust velocities in the range of 10-20 km/s.[1]

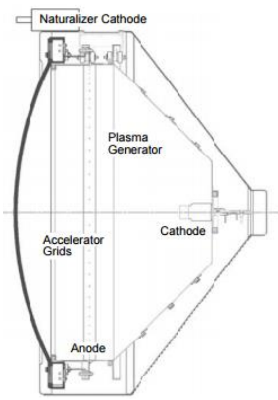

Figure 2: Ion Thruster Geometry[1]

An ion thruster has three primary subsystems: an acceleration grid, a plasma generation mechanism, and a neutralizing cathode. In Figure 2 a cross-section of an ion thruster can be seen. The central (discharge) cathode and anode are the plasma generator for this ion thruster. The ions produced in this interior region travel towards the grid where the ions are accelerated to create thrust. The plasma discharge creates a positive bias with respect to the spacecraft. The neutralizing cathode then supplies electron at the same rate as ion are ejected to neutralized the exhaust plume and avoids charge imbalance with the spacecraft. To model ion engines one begins with the one-dimensional Poisson's equation: $\frac{dE(x)}{dx} = \frac{\rho(x)}{\varepsilon_0} = \frac{qn_i}{\varepsilon_0}$ (7). Integrating eq (7) yield $E(x) = \frac{q}{\varepsilon_0} \int_0^x n_i(x')dx' + E_{screen}$ (8). E_{screen} is the electric field observed at the screen grid in an ion thruster. The force exerted on the screen grid is $F_{screen} = \frac{1}{2}\varepsilon_0 E_{screen}^2$ (9) while the force on the acceleration grid $F_{accel} = -\frac{1}{2}\varepsilon_0 E_{accel}^2$ (10). The thrust of an ion engine is demonstrated to be $T = F_{screen} + F_{accel} = \frac{1}{2}\varepsilon_0(E_{screen}^2 - E_{accel}^2)$ (11). The force exhibited on ion between the grid can be calculated as $F_{ion} = q\int_0^d n_i(x)\, E(x)dx$ (12). If one were to the solve equation (7) for ion number density and substitute into equation (12) it can be shown that $F_{ion} = \varepsilon_0 \int_0^d \frac{dE(x)}{dx} E(x)dx = \varepsilon_o \int_{E_{screen}}^{E_{accel}} EdE = \frac{1}{2}\varepsilon_0(E_{screen}^2 - E_{accel}^2)$ (13). Now from equations (10) and (13) is seen that the electrostatic force between the ions and polarized grids is the source of thrust in an ion engine.[1]

A HET has three main pieces: a cathode, a magnetic field generator, and a discharge region. In Figure 3, one can see a cross-section of a Hall Thruster. A cylindrical channel comprises the discharge region. A radial magnetic field is created between the central cylinder and outer channel wall. This gap is known as the flux return path. The discharge cathode is located at the top of the figure which is a hollow cathode. The anode is a ring located at the base of the cylindrical region. Electron initially travelling towards the anode become trapped by the transverse magnetic field. This leads to the electrons traveling around the thruster axis (which is the $\boldsymbol{E} \times \boldsymbol{B}$ direction). This spiraling motion is the Hall Effect the thruster is named after. Ions are generated by these trapped electrons which are accelerated by the outward electric field.[1,2]

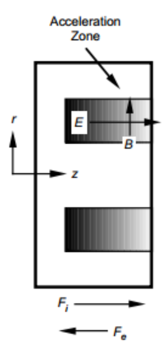

Figure 3: Hall effect thruster with shown electric and magnetic fields[1]

In a hall effect thruster ions are formed through the production of a plasma and subsequently accelerated by an electric field. The Hall current (which creates the transverse magnetic field) alters the force transfer. For purposes of simplification we also consider the plasma within a HET to be quasi-neutral which implies $qn_e \approx qn_i$. One also assume the Magnetic field and electric field within the cylindrical channel (Acceleration region) of the HET to be constant. The main influence on the ions is the electric field and is mathematically represented as $F_{ion} = 2\pi \iint q\, n_i \boldsymbol{E} r dr dz$ (14). The electrons in the Hall current are subject to a Lorentz force and move with speed $v_{el} = \frac{\boldsymbol{E} \times \boldsymbol{B}}{B^2}$ (15). The force exhibited on electrons $F_e = -2\pi \iint qn_e \boldsymbol{E} r dr dz - 2\pi \iint en_e \boldsymbol{v_{el}} \times \boldsymbol{B}\, r dr dz = 0$ (16) where the electrostatic force and Lorentz force on electrons cancel each other out. The force on the ions in a quasi-neutral plasma is $F_e = -2\pi \iint qn_i \boldsymbol{E} r dr dz + 2\pi \iint \boldsymbol{J_{Hall}} \times \boldsymbol{B}\, r dr dz = 0$ (17). The Hall current density is then defined as $\boldsymbol{J_{Hall}} = -en_e \boldsymbol{v_e}$. The force exerted on the ions is $\boldsymbol{F_i} = \boldsymbol{J_{Hall}} \times \boldsymbol{B}$ (18), which is the Lorentz force as we expected. The thrust on a hall thruster is transferred from the ions to the thruster body, mathematically that is $\boldsymbol{T} = -\boldsymbol{F_i}$ (19)**.** As the electric field is the driver of acceleration, HETs are referred to as a class of electrostatic thrusters. For equations (14) through (19) Figure 3 provides a simple schematic of a HET, the electric and magnetic field and the corresponding axes used for integration[1]

From equation (3) we have $T = v_e \frac{dm_p}{dt} = \dot{m}_p v_e = \dot{m}_\iota v_i$ (3), where propellant flow rate is the ion mass flow rate. The kinetic power of the ion beams is then $P_{jet} = \frac{1}{2}\dot{m}_p v_e^2 = \frac{T^2}{2\dot{m}_p}$ (20). Following the conservation of energy, the ion exhaust velocity can be expressed as $v_i = \sqrt{\frac{2qV_b}{M}}$ (21). Correspondingly the ion mass flow rate is shown to be $\dot{m}_\iota = \frac{I_b M}{q}$ (22). If one substitutes equations (21) and (22) into (3) the thrust is then $T = \sqrt{\frac{2M}{e}} I_b \sqrt{V_b}$ (23). Equation (23) represents the thrust in a unidirectional monoenergetic beam of ions which is singly ionized.

In real world applications equation (23) must be modified to adjust for non-singly charged ions and a diverging ion beam. To adjust for a diverging ion beam $F_t = cos\theta$ when θ is the half-angle divergence. For a nonuniform plasma or curved system the thrust correction must be integrate over the grid and beam geometries. In a cylindrical thruster (such as a HET) it is seen that $F_t = \frac{\int_0^r 2\pi J_i(r) cos\theta(r) dr}{I_b}$ (24). To account for singly and doubly charged ions the ion current becomes $I_b = I^+ + I^{++}$ (25). In this case the thrust is represented as $T_m = I^+ \sqrt{\frac{2MV_b}{e}} + I^{++}\sqrt{\frac{2V_b}{e}}$ (26). One can then define a thrust correction factor $\alpha = \frac{I^+ + \frac{1}{\sqrt{2}} I^{++}}{I^+ + I^{++}}$ (27). The overall correction which accounts for ion beam divergence and multiply charged ion species can then be expressed by the factor $\gamma = \alpha F_t$ (28). A more accurate model of thrust provided by equation (3) is then $T = \gamma \dot{m}_\iota v_i = \gamma \sqrt{\frac{2M}{e}} I_b \sqrt{V_b}$ (29).

Specific impulse measure the efficiency of a thruster. It is defined as $I_{sp} = \frac{T}{\dot{m}_p g_0} = \frac{v_e}{g_0}$ (30). If one substitutes earlier definitions, it can be shown that $I_{sp} = \frac{v_e \dot{m}_\iota}{g_0 \dot{m}_p}$ (31). The mass utilization efficiency of a thruster is $\eta_m = \frac{\dot{m}_\iota}{\dot{m}_p} = \frac{I_b M}{e\, \dot{m}_p}$ (32). Equation 32 holds for singly charged ion species and is a representation of the equivalency between ionized and unionized propellant. If doubly charged ions are present, then a correction must be made such that $\eta_{m^*} = \alpha_m \frac{I_b M}{e\, \dot{m}_p}$ (33). Here the mass utilization efficiency correction factor is defined as $\alpha = \frac{1+\frac{I^{++}}{2I^+}}{1+\frac{I^{++}}{I^+}}$ (34). Specific impulse in the case of a singly charged ions exhaust thruster to be $I_{sp} = \frac{\gamma \eta_m}{g_0}\sqrt{\frac{2eV_b}{M}}$ (35).[1]

There are several ways to classify the efficiency of an electric thruster. In the previous paragraph the thruster mass utilization efficiency was described. This is initial propellant mass which is ionized and accelerated by the electric thruster. The electrical efficiency, discharge loss, and thruster efficiency (sometimes called overall efficiency) can also be determined. The first to discuss is the electrical efficiency of the system. The electrical efficiency of an electric propulsion system is defined as the ratio between beam power and total power supplied to the system. The electrical efficiency is defined as $\eta_e = \frac{P_b}{P_T} = \frac{I_b V_b}{I_b V_b + P_0}$ (36).The discharge loss in a factor representing how efficiently ions are produced. Discharge loss is defined as $\eta_d = \frac{P_d}{I_b}$ (37) where P_d is the ion production power. Discharge loss is not dimensionless like most efficiency terms, but has units of eV/ion or W/A. The thruster efficiency or total efficiency is the ratio of power in the jet beam to the input power. This is shown as $\eta_T = \frac{P_{jet}}{P_{in}}$ (38). After substituting the definition in equation (20) one has $\eta_T = \frac{T^2}{2\dot{m}_p P_{in}}$ (39). An ion thruster demonstrated an overall efficiency of $\eta_T = \frac{\gamma \eta_m T v_i}{2\dot{m}_p P_{in}} = \frac{\gamma^2 \eta_m I_b V_b}{P_{in}}$ (40). To help simplify equation (40) the input power can be recursively defined as $P_{in} = \frac{P_b}{\eta_e} = \frac{I_b V_b}{\eta_e}$ (41). Substituting equation (41) into equation (40) yields an ion thruster total efficiency of $\eta_T = \gamma^2 \eta_m \eta_e$ (42). If one were to evaluate the thrust to power ratio and using equation (36) it is shown that $\frac{T}{P_T} = \frac{T\eta_e}{P_b}$ (43). Now if equation (29) is substituted for thrust and equation (35) are substituted in equation (43) the results is $\frac{T}{P_T} = \frac{2\gamma^2 \eta_e \eta_m}{g_0 I_{sp}} = \frac{2\eta_T}{g_0 I_{sp}}$ (44). From equation (44) one can directly observe that for a given power input and overall electric thruster efficiency, that an increase of specific impulse results in a lower thrust.[1]

To provide stable operation, the input power provided to an electric thruster that is not used to create thrust must be dissipated. Typically, excess power is radiated away from the system as heat. If a thruster has a demonstrated electrical efficiency, the power that must be dissipated away is $P_{dissipated} = P_{in}(1 - \eta_e)$ (45). If the thruster efficiency has not been demonstrated the power that must be radiated away can be determined by monitoring the thruster power supply. The input power can be regarded as

$$P_{in} = I_b V_b + I_d V_d + I_{ck} V_{ck} + I_{nk} V_{nk} + I_{A1}(V_b + V_a) + I_{A2} V_a + I_{DE1} V_b + I_{DE2} V_G \ (46).$$

In this setup 'b' indicates beam, 'd' implied discharge, 'ck' is cathode keeper, 'nk' is neutralizer keeper, 'A1' are beam ions colliding with the accel grid, 'A2' signifies charge exchange ions with grid potential V_a, 'DE1' represents decal grid current from beam ions, while 'DE2' represents decal grid for back streaming ions. In this case the dissipated power is all of that which is not part of the beam. That is

$$P_{dissipated} = I_d V_d + I_{ck} V_{ck} + I_{nk} V_{nk} + I_{A1}(V_b + V_a) + I_{A2} V_a + I_{DE1} V_b + I_{DE2} V_G \ (47).$$

This method is primarily used to imply power dissipated from ion thrusters and is much less effective in calculating the same thing accurately for Hall Effect Thrusters. If one can calculate the total beam power of a HET, while the input power is known, a rough estimate of the dissipated power can be made but is not a highly reliable method.[1]

The final aspect that fundamentally affects electric propulsion systems is flow rate at which the neutral gas is injected to the ionization chamber. The pressure observed in the system follows the standard gas law $PV = NkT_g$ (48). In this case $n = 9.66x10^{24} * \frac{P_{Torr}}{T}$ (49). Equation (49) provides the number of particles per cubic meter when pressure is calculated in Torr and temperature is in Kelvin. The gas flow into the chamber, referred to as throughput, is given by $P = \frac{Q}{S}$ (50). When gas is injected into the ionization chamber a negative pressure drop occurs and some gases backflow into the chamber. This backflow is referred to as ingested throughput. The backflow is calculated by the formula $Q_{ingested} = \frac{n\bar{c}}{4} A\eta_c$ (51) where $\bar{c} = \sqrt{\frac{8kT}{\pi M}}$ (52) is the thermal velocity of the gas. This makes the total throughput $Q = PS + Q_{ingested} = Q_{injected} + Q_{ingested}$ (53).[1]

IV. Deep Space Missions

Among the more recent spacecraft to use electric propulsion as a primary drive for deep space mission include Hayabusa, Dawn, and SMART-1.[7, 15, 16] From the brief descriptions below, it is easy to see the positive impact EP has had on robotic exploration missions. These missions also show the great potential in using EP to accelerate robotic satellites on future missions.

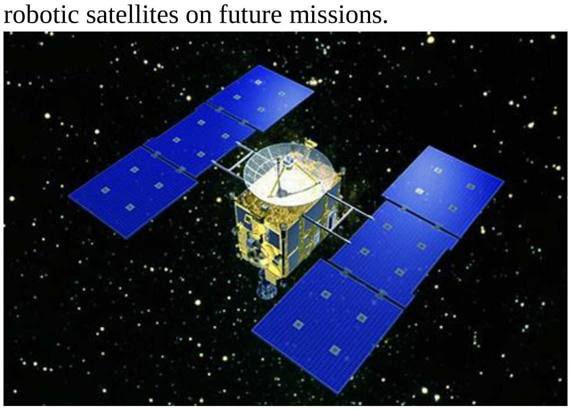

Figure 4: Hayabusa JAXA satellite[15]

Hayabusa is a spacecraft launched by JAXA that was launched May 2003. The satellite successfully intercepted the Itokawa asteroid in November 2005 and began the return trip April 2007. The spacecraft successfully arrived in June 2010 with the collected asteroid sample. Hayabusa relied on three ion thrusters known as µ10. The ion engine system (IES) had an inert (dry) mass of 59 kg. The total propellant mass for the mission 66 kg of Xenon. Each µ10 had a specific impulse of 3,000 s. 350 W power operating level, and average thrust of 8 mN. With one day of operation the spacecraft received a delta-v of 4 m/s. The main components of the IES included the four ion thrusters, Microwave Power Amplifiers (MPA), three Power Processing Units (PPU), a Propellant Management System (PMU) and an IES Pointing Mechanism which was a gimbal system.[15]

Dawn Launched in September 2007. The primary mission of Dawn was to visit Ceres and Vesta, which are protoplanets in the asteroid belt. Dawn entered orbit around both Vesta and Ceres. Dawn first orbited around Vista for 14 months before travelling to Ceres where it is currently orbiting. This is the first spacecraft to ever orbit two celestial bodies within our solar system.[23] Dawn has the NSTAR ion propulsion system onboard, which is capable of a specific impulse of 3,100 s and maximum thrust of 92 mN.[7,24] The NSTAR ion engine was first successfully tested aboard Deep Space 1 in 1998.[7]

On September 27, 2003 ESA launched the Small Missions for Advanced Research in Technology-1 (SMART-1) aboard an Ariane-5 from its launch site in French Guiana. The mission lasted until September 3, 2006 when the spacecraft was intentionally crashed into the lunar surface. The main propulsion system aboard SMART-1 was a solar Electric Propulsion (SEP) Hall-effect plasma thruster model PPS-1350G. The PPS-1350G was developed by SNECMA (which in 2016 was renamed Safran Aircraft Engines). The satellite was a combined 367 kg of which only 82 kg were propellant. SMART-1 used Xenon as a propellant for the PPS-1350G. This mission will be covered in greater detail in the next section.[16]

V. Analysis of SMART-1 mission

During the SMART-1 mission for ESA accomplished several milestones. This was the first time the space agency had used electric propulsion as a primary propulsion system. It also included the first-time EP was used for an ESA satellite to escape GEO. During the mission EP and gravity assist maneuvers were completed together for the satellite to take the preferred flight path. SMART-1 was also the first lunar orbiter for ESA. EP was used to guide the satellite within gravitational capture by the moon. Technologically, this marked the first successful use of a HET, first time for variable powered HET to be operated successfully under varying power conditions. During the mission the PPS-1350G was operated for 10 continuous days and collectively operated for more than 5,000 hours throughout the mission. [16]

While EP systems provide much lower levels of thrust, which is often viewed as having a negative impact, it was found to have an advantage for those directing the satellite. If an error was made in directing the spacecraft the propellant cost of course correcting was unimportant. This is a great benefit when one considers the fact that a fixed amount of propellant is aboard spacecraft. The storability of the propellants (typically Xenon) and the low operational cost of doing so suggests great advantages for long duration mission such as station keeping for GEO communications satellites.[16]

Figure 5: SMART-1 ESA Satellite[28]

The initial phase of the satellite was to escape LEO. SMART-1 was launch into a 3600 km LEO orbit by an Ariane-5. During a three-day phase of operation of the PPS-1350G. The thrust was turned on for approximately 12 hour periods as the perigee was approached. The thrust vector was aligned with the velocity vector during this period and the perigee height was increased to 13,600 km. This positioned the satellite above the Van Allen Belts and the harmful radiation exposure experienced in the region. The orbit was increased to 200,000 km where the gravitational field of the moon began influencing the flight path of SMART-1. To reach this height, the thruster was active for a total of 950 hours, at which point the satellite entered a polar orbit about the moon. The polar orbit was entered in February 2005, nearly 17 months after launch. In August 2005, a 340 burn of the PPS-1350G increased the orbit of SMART-1 allowing for an extra year of observation in lunar polar orbit.[16]

The EP system contained a 49 L propellant tank which stored 82 kg of Xenon. Two solar panel on individual arms were used for power production. At initial launch the solar panels were rated to deliver 1.85 kW of power. The EP system allowed for thrust vectoring so that thrust would continually be directed collinear with the center-of-mass of SMART-1. Gimballing also allowed for unloading reaction wheels aboard the spacecraft.[16]

The SNECMA designed PPS-1350G is a derivative of the SPT-100 design in Russia by Fakel.[16, 22] Ground testing of the PPS-1350G included 9,200 hours of operation over 7200 cycles. The total impulse delivered during ground testing is approximately 2.9 MN. This ground testing has qualified the thruster for 15 years of station keeping aboard communications satellites. This HET accelerates Xenon ions to speeds of roughly 16,650 m/s. The propellant storage tank was rated to store Xenon at a density of 1,700 kg/m^3 and pressures of up to 15 Mpa. The Xenon was reduced to .2 MPa before being fed into the ionization chamber by the Xenon Flow Controller (XFC). A Power Processing Unit (PPU) was included to control the operating conditions of the HET. In total 117 different operational power levels were chosen between .462 kW and 1.190 kW throughout the mission.[16]

Over the lifetime of the mission the EPS was operated for 4,958 hours. The total impulse was 1.2 MNs, where impulse is the force times burn time of the thruster. This compares with an impulse of about 1.2GNs for the first stage burn of a single Merlin 1D engine on a Falcon 9.[16,27] The longest continuous operation of the thruster was 240 hours which occurred shortly after launch to allow the satellite to escape the Van Allen radiation belts as quickly as possible. A total of 844 on/off cycles were executed during the mission and during each cycle ignition occurred with the first ignition impulse. Out of the initial 82 kg of Xenon launched, only .28 kg remained in the tank. This indicates a thruster mass utilization of 99.66%. The average thrust produced by the EPS was 67 mN and over one and a quarter million valve actuations were orchestrated. This was the first EP system used to escape Earth's gravitational field and exhibited significant reliability for the duration of operation and no major failures occurred. [16]

VI. Problems and Potential Solutions

Among the primary lifetime limitations for an EP system is cathode and channel wall erosion. One idea that has been developed and is currently undergoing testing is the E-IMPAcT. This thruster is designed to operate without the presence of a central cathode. As the central cathode is removed in E-IMPAcT the lifetime limitation of cathode erosion and corresponding performance loss is removed. If successfully tested and space certified it could bring even further possibilities to existing EP technology. There is ongoing experimental and theoretical research into the problems of hall effect thruster channel wall erosion. While there are no answers to directly mitigate the adverse effects currently, with more research we are coming closer to understanding the mechanism by which the erosion occurs. Once this mechanism is known it should be simpler to create solutions to the problem of channel wall erosion.[25]

Another main obstacle which exists for innovation in the realm of electric propulsion is the heavy dependency on plasmas and plasm physics. While we have certainly made global progress in understanding plasmas throughout the last century, testing involving plasmas is very expensive and simulation of plasma tend to be large and complex requiring long computation time or a loss of physical phenomena in a hope of manageable testing cost. There is no simple way around this barrier as experimentation requires vacuum chamber testing and simulations work can lead to integration calculations which much be carried out for individual particles which limits the amount of plasma which can be studied within a single simulation, again limiting the significant impact attainable from theoretical research of plasmas. While this research into plasma physics represents a difficult path, it may provide very good results if our understanding improves. While better techniques are developed to model plasmas and more plasmas are observed and tested in laboratory environments the potential for increased thruster technology and improved energy generation are likely to increase.

While high efficiency thruster with much greater thrust levels than ion thrusters and HETs exist, there historically has been a limit on the power systems that have been operational in space. As was mentioned earlier, a MPD thruster was developed at NASA Glenn which demonstrated exhaust velocities of 100,000 m/s and a thrust of 100 N.[12] While this sounds more beneficial than thrusters currently in use, the MPD thruster required 1 MW of power to operate at the conditions. This power requirement is an immense feat to have launched aboard a spacecraft. This would likely require a nuclear power source to be launched aboard a satellite using this thruster. Significant economic and political barriers likely stand in the way of this reality. Even if this system could be launch would also need to account for the large increase in mass of the energy production system and how much acceleration of the craft would be lost overall.

VII. Conclusion

Electric propulsion seems limited to high specific impulse and low thrust parameters for the near future. Hall Effect Thrusters have been growing in popularity throughout the last two decades. The Japanese, Indian, European, Russian, and American space agencies have all demonstrated EP systems successfully in orbit. We are likely going to see electric propulsion become industry standard for all communication satellites in the foreseeable future. They offer great advantages in more efficient propellant use, simple propellant storage procedures, reduced payload and therefore Launchpad mass, which directly decreases launch cost of satellites and allows for smaller launch vehicles to be utilized when appropriate. Since the new millennium we have observed three different countries and space agencies send deep space robotic exploration spacecraft which used electric propulsion systems for primary space propulsion. This is the greatest advantage I see from studying these thrusters. In the 21st century we now have well developed and sufficiently tested electric propulsion systems which aid in exploring the solar system. As the electric thrusters gain popularity among communication satellite systems, they will inevitable become cheaper and more reliable. The economic savings of launch long-lifetime satellite may also provide an additional boost to countries that are developing space economies and trying to launch satellites of their own for the first time. The low thrust limitations on EP systems do not allowed for their use in aiding spaceflight with humans aboard spacecraft, and strongly suggests that these limitations will not be overcome in the foreseeable future.

Acknowledgments

This paper uses various sources, including books, conference proceeding, journal articles, and websites. I have taken caution to thoroughly reference all source material throughout the paper. The data are equations included are used for educational purposes only and I believe the use satisfies the “fair use” clause.

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Appendix A

A1. EP Operating Parameters

Table 1. Operating Parameter Values by EP class[1]

Thruster	Specific Impulse (s)	Input Power (kW)	Efficiency Range	Propellant
Resistojet	300	0.5-1	0.65-0.90	N_2H_4
Arcjet	500-600	0.9-2.2	0.25-0.45	N_2H_4
Ion Thruster	2500-3600	0.4-4.3	0.40-0.80	Xenon
Hall Thruster	1500-2000	1.5-4.5	0.35-0.60	Xenon
PPT	850-1200	<0.2	0.07-0.13	Teflon

Table 2. Operating Parameters of SPTs developed in Russia[21]

Performance	Nominal Operating Power (kW)	Nominal Thrust (mN)	Specific Impulse (s)	Lifetime during ground tests (h)	Thruster Efficiency	Stage of Development
SPT-50	0.35	20	1100	1500	0.35	Flight Proven
SPT-70	0.7	40	1500	3000	0.45	Flight Proven
SPT-100	1.35	80	1600	9000	0.5	Flight Proven
SPT-140	5	300	1750	>7000 (likely)	>0.55	Under Qualification