

T.R.
GEBZE TECHNICAL UNIVERSITY
GRADUATE SCHOOL

**MATHEMATICAL MODELING OF SATELLITE'S ELECTRIC
PROPULSION SYSTEM (EPS) BASED ON HALL EFFECT
THRUSTER (HET)**

METE BİLGİR

**A THESIS OF MASTER OF SCIENCE
DEPARTMENT OF ENERGY TECHNOLOGIES
APPLIED PROPULSION SYSTEM DESIGN & ENGINEERING
FOR AEROSPACE TECHNOLOGIES PROGRAM**

ADVISOR: ASSIST. PROF. DR. SALİH ÖZEN ÜNVERDİ

JULY 2024

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T.C.
GEBZE TEKNİK ÜNİVERSİTESİ
LİSANSÜSTÜ EĞİTİM ENSTİTÜSÜ

UYDULAR İÇİN HALL ETKİLİ İTİCİ TABANLI
ELEKTRİKLİ İTKİ SİSTEMİ MATEMATİKSEL
MODELLEMESİ

METE BİLGER

YÜKSEK LİSANS TEZİ
ENERJİ TEKNOLOJİLERİ ANABİLİM DALI
HAVACILIK VE UZAY TEKNOLOJİLERİNDE
UYGULAMALI İTKİ SİSTEMİ TASARIM
MÜHENDİSLİĞİ PROGRAMI

DANIŞMAN: DR. ÖĞR. ÜYESİ SALİH ÖZEN ÜNVERDİ

TEMMUZ 2024



MASTER of SCIENCE JURY APPROVAL FORM

A thesis submitted by Mete BİLGER, defended on 10/07/2024 before the jury formed with the 08/07/2024 date and 2024/35 numbered decision of the GTU Graduate Administration Board, has been accepted as a MASTER of SCIENCE thesis in the Department of Energy Technologies, Applied Propulsion System Design & Engineering for Aerospace Technologies Program.

JURY

MEMBER

(THESIS ADVISOR) : Asist. Prof. Dr. Salih Özen ÜNVERDİ

MEMBER

: Prof. Dr. İlyas KANDEMİR

MEMBER

: Asist. Prof. Dr. Ali Uğur SAZAKLIOĞLU

APPROVAL

Gebze Technical University Graduate Administration Board
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ABSTRACT

Satellites have become an almost indispensable part of our lives today. All communication systems, smart devices, the Internet of Things (IoT), data reception or transmission, and many other scientific endeavors are carried out with the assistance of satellites. Upon separation from the launch vehicle and during all mission lifetimes, a satellite needs to perform many specific maneuvers. Although maneuver needs may vary depending on the mission's orbit, they are mainly as follows: orbit raising to increase its altitude from the launcher's initial orbit to the mission orbit; station acquisition to capture the mission window precisely; station keeping maintaining its position as desired against perturbations during mission life; and transferring to the graveyard orbit for de-orbiting at the end of mission life. The propulsion systems are an irreplaceable requirement for all satellites that aim to perform maneuvers.

In this thesis, first, electrical-based propulsion systems that have significant advantages for the satellite sector are to be examined. Currently, they are less preferred in the Turkish satellite sector due to their complex and unknown details. According to literature research, it is predicted that electric propulsion systems, which have been the trend of recent years and are widely used globally and increasing dramatically, will have a greater adoption in the Turkish satellite sector in the next decade.

The knowledge and experiences gained in national and international projects in the satellite sector are aimed at contributing to the literature within the scope of the near future of the Turkish satellite sector, considering the needs of the global market.

In this thesis, it is aimed to first determine which satellite platforms for the electric propulsion system are cost-effective and design-advantageous, and then to introduce the design solution that would emerge if it were applied to a sample communication satellite platform. Subsequently, it is aimed to focus on the development of the mathematical model of the electric propulsion system in the MATLAB-SIMULINK environment, which is rare (or even non-existent) in the Turkish industry. Finally, the developed mathematical model is to be verified in a sample maneuver scenario.

Keywords: Satellite Propulsion Systems, Electric Propulsion, Hall Effect Thruster, MATLAB-SIMULINK Model.

ÖZET

Uydular günümüzde hayatımızın vazgeçilemez bir bileşeni haline gelmiştir. Tüm haberleşme sistemleri, akıllı cihazlar ve nesnelerin interneti (IoT), veri alma-gönderme ve diğer birçok bilimsel faaliyetler uyduların yardımıyla gerçekleştirilmektedir. Bir uydunun fırlatıcıdan ayrıldıktan sonra görev ömrü boyunca birçok manevrayı icra etmesi gerekir. Manevra ihtiyaçları uydunun görev yapacağı yörüngeye göre değişkenlik göstermekle birlikte temel olarak şöyle özetlenebilir: ilk yörüngeden görevin idame edileceği hedef yörüngeye irtifa artırmak amacıyla yapılan yörünge yükseltme manevrası, görev penceresini hassas bir şekilde yakalamak için yapılan istasyon yakalama manevrası, belirlenen görev süresince görev yörüngesinde meydana gelecek bozulmalara karşı mevzi koruma manevrası, görev ömrü sonunda görevi sonlandırmak için mezarlık yörüngeye transfer manevrası. Bu manevraları gerçekleştirmek isteyen tüm uydular için itki sistemleri zaruri bir ihtiyaçtır.

Bu çalışmada uydu sektörü için önemli avantajları olan ve günümüzde Türkiye uydu sektöründe karmaşık ve bilinmeyen detayları sebebiyle daha az tercih edilen elektrik tabanlı itki sistemlerinin incelenmesi amaçlanmaktadır. Literatür incelendiğinde son yılların trendi olan, küresel kullanımı çok yaygın olan ve giderek artan elektrik tabanlı itki sistemlerinin gelecek 10 yıl içerisinde Türkiye uydu sektöründe daha fazla kullanılacağı öngörülmektedir.

Hali hazırda uydu sektöründe ulusal ve uluslararası projelerden elde edilmiş bilgi-birikimi ve deneyimlerin, küresel pazarın ihtiyaçlarını göz önüne alarak, Türkiye uydu sektörü yakın geleceği kapsamında literatüre kazandırılması hedeflenmektedir.

Bu çalışmada, öncelikle elektrikli itki sisteminin hangi uydu platformları için maliyet etkin ve tasarım avantajlı olduğu konularının tespiti ve ardından örnek bir haberleşme uydusu platformuna uygulanması durumunda ortaya çıkacak tasarım çözümünü tanıtmak amaçlanmıştır. Ardından Türkiye sektöründe nadir karşımıza çıkan (hatta üzerinde hiç çalışılmamış) elektrikli itki sisteminin matematiksel modellenmesi MATLAB-SIMULINK ortamında oluşturulacak ve tanıtılacaktır. Son olarak, tasarlanan matematiksel model örnek bir manevra senaryosunda doğrulanacaktır.

Anahtar Kelimeler: Uydu İtki Sistemleri, Elektrikli İtki, Hall Etkili İtici, MATLAB-SIMULINK Modeli.

ACKNOWLEDGEMENTS

I would like to express my deep and sincere gratitude to my supervisor, Assistant Prof. Dr. Salih Özen Ünverdi. His support, suggestions, and encouragement provided me with the motivation to complete this work.

Lastly, I am immensely thankful to my dear Sabiha Bilger, my significant other and life partner, who has consistently supported me without hesitation.



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LIST OF SYMBOLS AND ABBREVIATIONS

V_i	: Acceleration Voltage
$\dot{m}_a$	: Anode Mass Flow Rate
ΔV	: Delta-V, Velocity Changes
I_D	: Discharge Current
V_d	: Discharge Voltage
$P_{electrical}$	: Electrical Power
m_e	: Electron mass
e	: Elemental Electron Charge
v_{ex}	: Exhaust Velocity
g	: Gravitational acceleration
m_i	: Ion mass
$P_{ion\ jet}$	: Ion Jet Power
v_i	: Ion Velocity
q_i	: Ionization degree of atom
L_a	: Ionization Region Thickness
B	: Magnetic field strength
m_{ps}	: Mass of Payload
m_{sc}	: Mass of Spacecraft
I_{sp}	: Specific Impulse
T	: Thrust
η	: Thruster Efficiency
AC	: Alternative Current
AP	: Applied Field
CoG	: Center of Gravity
CPS	: Chemical Propulsion System
ComSat	: Communication Satellites
DART	: Double Asteroid Redirection Test
DC	: Direct Current
EEE	: Electrical, Electronic, and Electromechanical
EP	: Electric Propulsion
EPS	: Electric Propulsion System
ESA	: European Space Agency
ECR	: Electron Cyclotron Resonance
FEED	: Field Emission Electrostatic Propulsion
GEO	: Geostationary Orbit
GTO	: Geostationary Transfer Orbit
HALE	: Hall Thruster Propulsion System Development Project
HET	: Hall Effect Thruster
JAXA	: Japan Aerospace Exploration Agency
JPL	: Jet Propulsion Laboratory
LEO	: Low Earth Orbit
MATLAB	: Matrix Laboratory
MET	: Microwave Electrothermal Thruster
MPD	: Magneto Plasma Dynamic
NASA	: National Aeronautics and Space Administration
OBC	: On-Board Computer

PCS	: Power Control System
PCU	: Power Control Unit
PET	: Pulsed Electrothermal Thruster
PIT	: Pulsed Inductive Plasma
PPS	: Propulseur à Plasma Stationnaire
PPT	: Pulsed Plasma Thruster
PPU	: Power Processing Unit
PTFE	: Polytetrafluoroethylene
PVT	: Pressure-Volume-Temperature
RF	: Radio Frequency
RIT	: RF Ion Thruster
SF	: Self-Field
SPT	: Stationary Plasma Thruster
TAL	: Thruster With Anode Layer
TOM	: Thruster Orientation Mechanism
UK	: United Kingdom
USSR	: Union of Soviet Socialist Republics
U.S.	: United States
VASIMR	: Variable Specific Impulse Magneto Plasma Rocket
XFC	: Xenon Flow Controller
XSU	: Xenon Supply Unit

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1. INTRODUCTION

Propulsion systems are classified based on the source of energy used to expel the propellant gas through the nozzle. In chemical propulsion systems, a mixture of fuel and oxidizer reacts in the combustion chamber, releasing a large amount of energy and producing hot gases at very high temperatures and pressures. The thermal energy of the combustion products is then converted into kinetic energy as the gases expand and accelerate through the nozzle. Gases with high kinetic energy are expelled from the nozzle, creating thrust according to Newton's third law. Chemical energy stored in the fuel and oxidizer mixture ultimately creates thrust [1, 2].

In electric propulsion systems, the energy that generates thrust is not stored in the fuel. There is no energy limit, and the system is powered externally by solar radiation collectors, batteries, or nuclear sources.

The need for non-toxic next-generation green propellant systems is drawing attention to legal regulations concerning hydrazine-based propulsion systems. Non-toxic alternatives are being considered for applications in Moon and Mars missions, upper stages, interplanetary robotic missions, and launch vehicle applications due to their significant environmental and economic benefits [3].

To position a satellite on a target trajectory, it is necessary to perform the relevant thrusting maneuvers and maintain the required angular velocity. This process involves compensating for gravitational forces and atmospheric losses. Any deviation in the satellite's velocity from the required value will lead to distinct trajectories.

The required velocity for achieving a trajectory is provided by the thrust acceleration, which is in the opposite direction of the rapid ejection of a mass from the system. High thrust acceleration can be achieved by expelling a large amount of mass at high speed. Besides positioning the satellite on a trajectory, thrust systems are required after launch. Propulsion systems are typically implemented for three main reasons in satellites:

- **Orbit Raising:** The process of elevating, lowering, or changing the plane of a trajectory is referred as a maneuver. For instance, geostationary satellites must reach a significant altitude, and launch vehicles must leave the satellites in a

suitable geostationary transfer orbit (GTO) for subsequent maneuvers. The satellite is then guided to the target orbit using the thrust system on board. Table 1.1 shows an example of an initial orbit with a perigee of 250 km and an apogee on a geostationary earth orbit (GEO). Of course, there is an inclination of 27 degrees due to the latitude of the launcher side on Earth, but for GEO satellites, it needs to be eliminated as 0 degrees during the orbit-raising phase. The process of a trajectory maneuver can be observed in Figure 1.1, where a Hohmann transfer orbit is used to transition between two orbits in the same plane. This process involves two velocity changes (Δv_1 , Δv_2) using thrust [1, 2].

Table 1.1: Sample Initial Orbit.

	Initial Orbit	Target Orbit
Periapsis Altitude	250 km	35,786 km
Apoapsis Altitude	35,786 km	35,786 km
Inclination	27 deg	0 deg
Argument of Periapsis	178 deg	undefined
Right Ascension	0 deg	undefined
True Anomaly	180 deg	undefined

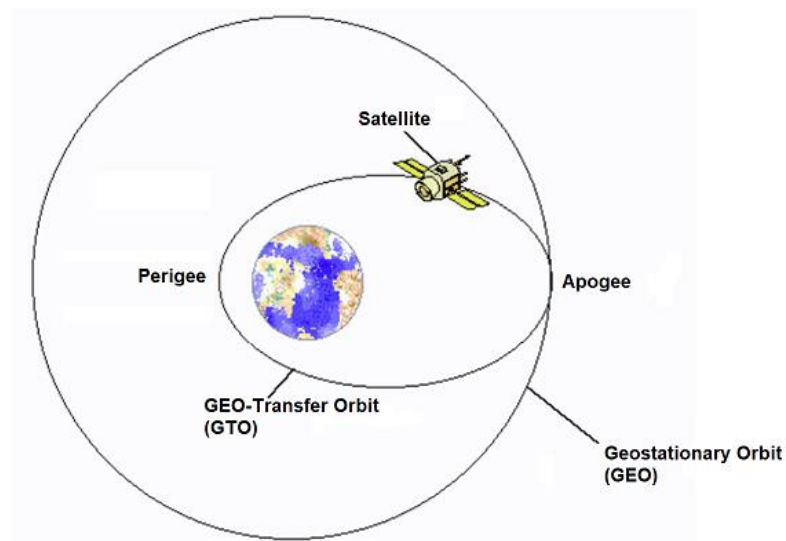


Figure 1.1: Geostationary Transfer Orbit (GTO).

- **Station Keeping:** Satellites are subjected to various impacts in space that can disrupt their trajectory. These include the uneven distribution of the Earth's mass, atmospheric drag, magnetic disturbances, and solar radiation, among others. To maintain and correct the trajectory or alignment of a single satellite or a group of satellites, thrust systems are used to counteract these effects at a certain level.
- **Attitude Control:** The orientation of satellites is controlled to ensure they face the desired direction in space. This can be achieved using six small thrust systems, one for each of the mutually perpendicular three axes.

Different propulsion systems exist to meet various propulsion requirements, and the selection of a suitable propulsion system for a satellite mission depends on various parameters, including the specific impulse (I_{sp}) values of the system, which indicate the efficiency of the thrust.

The upcoming sections will delve into a detailed examination of the various types of satellite propulsion systems and their comparative analysis. This exploration will involve evaluating different propulsion systems based on factors such as specific impulse, thrust-to-weight ratio, fuel efficiency, complexity, reliability, and cost.

2. PURPOSE OF THE STUDY

All developing nations with burgeoning advanced technologies are investing in space and satellite technologies, crucial for progressing into an information-driven society. Satellites have become an indispensable part of our lives, with all communication systems, smart devices, data acquisition or transmission, and many other scientific activities being carried out with their assistance.

Satellites and spacecraft require systems that can generate short-term thrust and torque to maintain their orbits and allow for long-term maneuvering. By applying force to create circulation around their orbits, space travel can be likened to a pencil motion. Satellites are launched into transfer orbits using rockets, but they must reach their final mission orbits, which are determined by their intended use. Satellites must maintain their position in their orbits despite the influence of planetary gravitational forces, Earth's irregular geoid shape, and other adverse space conditions, while also protecting their functions and orientations. To achieve this, they require their own propulsion (thruster) systems.

This study aims to define and examine the propulsion systems of satellites and spacecraft, as well as their functions and components, from a broad perspective. The information presented is intended to provide technical knowledge and fundamental concepts for designers in the field of satellite and spacecraft propulsion systems and to contribute to the development of technical infrastructure for training experts in Türkiye in this field.

By examining the application areas, main performance characteristics, advantages, and disadvantages of propulsion systems used in satellites and spacecraft and highlighting the determining factors in the propulsion system selection process, this study aims to introduce the necessary technical knowledge and basic concepts for designers. The fundamental principles of motion in space, based on Newton's basic laws of motion and Kepler's laws, have been used to determine the basic application areas and parameters of propulsion in the space environment.

This study has presented information on the operational activities of satellites, such as maintaining their orbital position, rising/descending from orbits, and transferring

between orbits, as well as the differences in propulsion systems. Propulsion systems have been classified based on the energy source of the propellant types, their storability, reliability, and performance requirements, as well as pre-flight ground tests and the equipment used in these tests. The study has also evaluated the positive and negative impacts of environmentally friendly, reliable, and efficient propulsion systems on current satellite and spacecraft users and manufacturers and discussed the potential for these systems to be preferred for space missions. Considering the future evolution of space systems, from reducing the cost of space missions to exploring the solar system, advanced propulsion systems play a crucial role in increasing the specific impulse (thrust per unit mass) and speed-up capabilities and performance of missions.

In addition to providing information on the critical components of propulsion systems, this study has also highlighted the functional effects and capabilities of spacecraft and satellites.

In this study, a mathematical model has been developed, and the results of this model on a sample satellite have been presented. The aim is to demonstrate the application of reliable simulation models in practice.

Developing a reliable mathematical simulation model for propulsion systems offers benefits such as speeding up development, cost reduction, problem anticipation, performance prediction, and creating a database for failure modes. However, due to system complexity, reliance on reduced-order models, and the need for extensive model parameters, obtaining reliable experimental data is crucial. Security issues can hinder data collection for each component, necessitating collaboration with research institutions for accurate system data. Despite the challenges, the increased interest in simulation programs highlights their importance in propulsion system development.

The subsequent sections of this study provide detailed information about the developed mathematical model, the results obtained, and the specifics of the example used in the research.

3. GENERAL OVERVIEW OF SPACECRAFT & SATELLITE PROPULSION SYSTEMS

Satellite propulsion systems are classified according to the type of energy source they use and their operating principles, as shown in Figure 3.1.

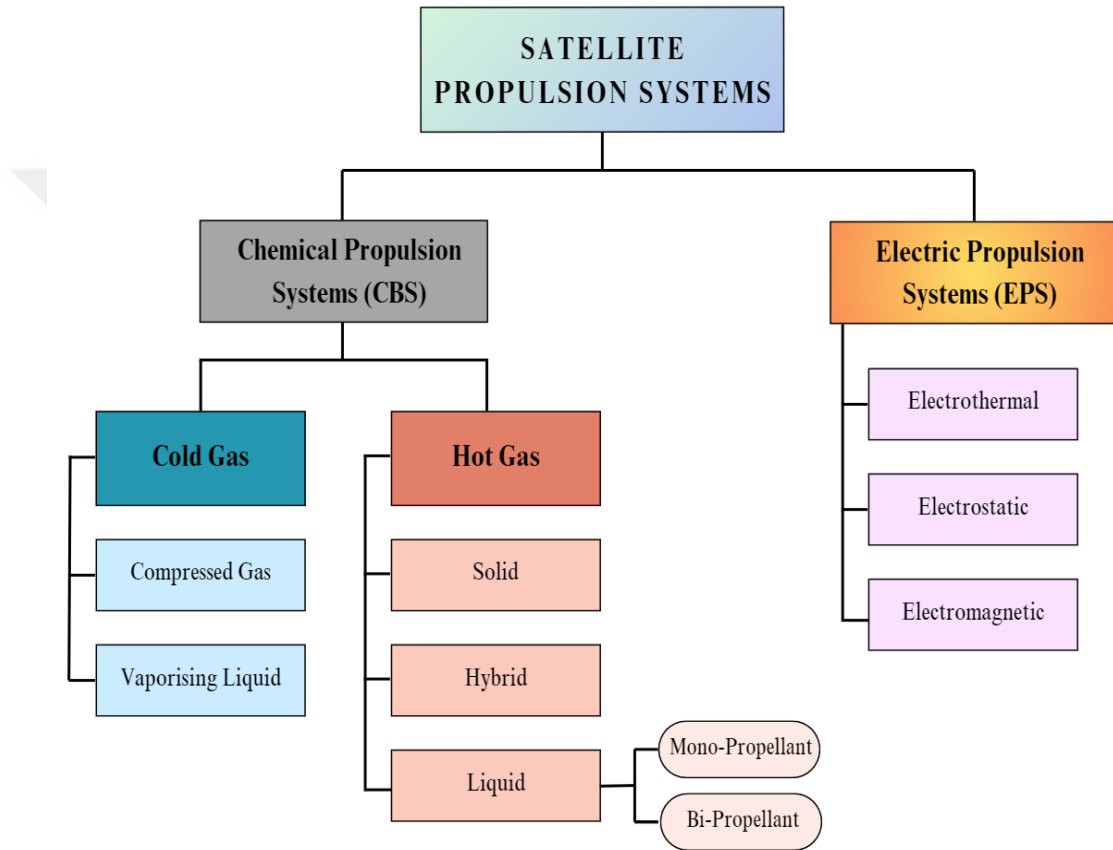


Figure 3.1: Classification of Satellite Propulsion Systems.

The main factor in the selection of propulsion systems is the Delta-V required by the satellite for the specified mission and the time needed to achieve this Delta-V. Figure 3.2 shows the Delta-V performance ranges of different satellite propulsion system concepts. In the Figure 3.2, m_{ps} represents the payload mass, m_{sc} denotes the satellite mass, and Delta-V indicates the increase in velocity in m/s [4].

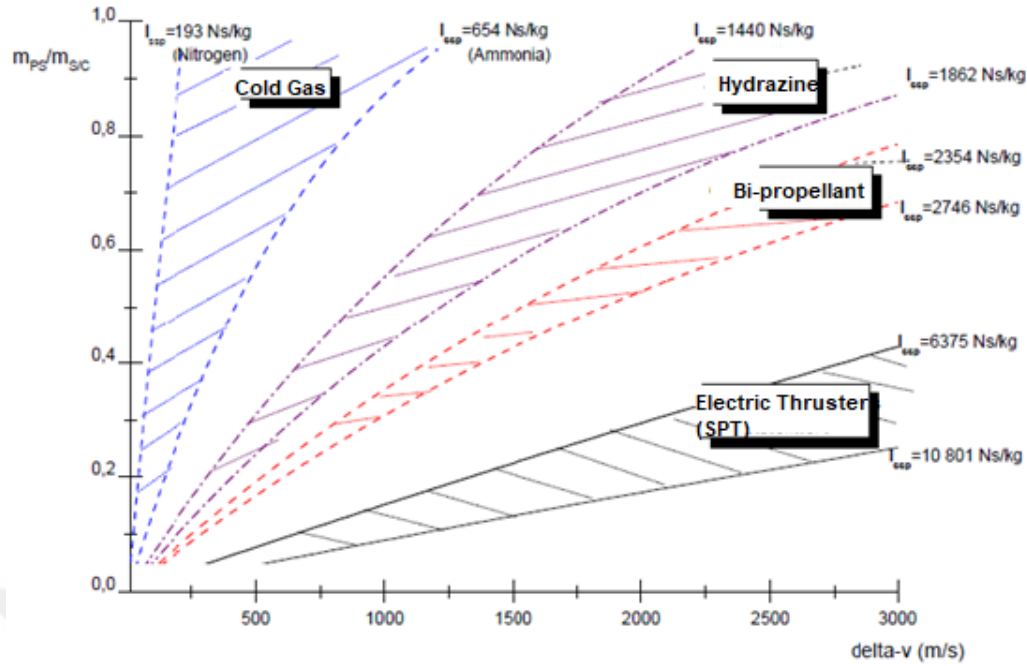


Figure 3.2: Delta-V Ranges of Different Type of Propulsion Systems [4].

From Figure 3.2, in a scenario where the m_{ps}/m_{sc} ratio is assumed to be less than 0.3, the following information can be obtained regarding the propulsion system that can be used:

- For a mission with a Delta-V requirement of $\Delta V < 150$ m/s, compressed cold gas and evaporating liquid propulsion systems are seen as the best solutions. This is based on the information presented in the figure, the requirements and cost-effectiveness make these the most preferred propulsion systems under these conditions.
- For a mission with a Delta-V requirement of $150 < \Delta V < 650$ m/s, a monopropellant hydrazine propulsion system is the most preferred one. Due to its simple structure, it is a safe system. Additionally, the requirements seen in the figure make this propulsion system the most preferred within this range.
- For a mission with a Delta-V requirement of $\Delta V > 650$ m/s, monopropellant chemical propulsion systems, monopropellant hydrazine resistojets, arcjet propulsion systems, and electrostatic/electromagnetic systems are the most suitable propulsion systems.

3.1. Literature Survey

3.1.1. The History of Spacecraft and Rocket Technology

The chronology of the development of rocket technology and space exploration can be traced back to ancient times with the invention of the steam-powered pigeon model by Aulus Gellius in the 360s before Christ, which laid the foundation for the development of rocket engines for spacecraft. The use of gunpowder rockets by the Chinese military in the 13th century and the scientific principles of rocket motion by Sir Isaac Newton in the 17th century also contributed to the development of modern rocket technology.

In the 19th and early 20th centuries, Russian scientist Konstantin Tsiolkovsky developed the fundamental principles of rocket science, including the Tsiolkovsky Rocket Equation, and proposed the use of liquid hydrogen and oxygen as rocket fuels. American scientist Robert Goddard conducted rocket experiments in the early 20th century and successfully launched a liquid-fueled rocket in 1926.

In the mid-20th century, the development of rocket technology accelerated with the V-2 rocket developed by German scientist Wernher von Braun during World War II, which contributed significantly to the development of civilian rocket technology after the war. The launch of Sputnik-1 by the Soviet Union in 1957 marked the beginning of the Space Age, followed by the launch of Explorer 1 by the United States military in 1958 [5].

In the late 1930s, a group of Caltech graduate students conducted an initial rocket test, marking the beginning of amateur rocket interest. The U.S. Department of Defense initiated technological research in rocketry and atmospheric sciences in the 1940s. In 1942, German rocket scientist Wernher von Braun's V-2 rocket was successfully launched from Peenemünde, causing concern in the U.S. In response, NASA, established in 1958, began developing rocket technology through the Jet Propulsion Laboratory (JPL), founded in 1944.

Before the U.S. entered WWII, there was a need for small rockets to facilitate the takeoff of heavy aircraft. This led to the production of jet-assisted takeoff rockets in August 1941, developed by the original "Rocket Boys" group, led by Frank Malina. After the U.S. entered WWII, guided missiles were developed in 1944, differing from JPL's previous rockets as they featured guidance systems for targeting.

JPL's first successful supersonic rocket test occurred on October 11, 1945, with the WAC Corporal (little sister), reaching a record-breaking 70 kilometers in altitude. JPL then began developing liquid-fueled missiles for the military. In the following years, JPL developed the simplified final rocket, "Sergeant" and solid-fueled systems. However, in 1959, the military transferred the project to the industrial contractor Sperry Corporation, with JPL providing long-term consulting.

The International Geophysical Year, which began in July 1957, aimed to observe various scientific phenomena worldwide. The Soviet Union's launch of Sputnik, the world's first artificial satellite, on October 4, 1957, shocked the scientific community. JPL was surprised by the Soviet Union's ability to launch both the satellite and its electronic systems. In response, the U.S. expedited the development of a four-stage rocket, placing the Explorer 1 satellite at its tip. On January 31, 1958, Explorer 1 became the first U.S. satellite, collecting data on Earth's radiation environment.

The U.S. space race against the Soviet Union began, and in December 1958, the military officially transferred JPL to NASA. In 1958, President Eisenhower allowed the U.S. Air Force and Army to develop spacecraft for landing on the Moon. Four Pioneer spacecraft were built, but the first three failed to reach their orbits. Pioneer-4, launched on March 3, 1959, reached a close distance to the Moon, paving the way for future deep space exploration.

The Jet Propulsion Laboratory played a crucial role in the development of rocket technology, leading to the creation of various spacecraft and the exploration of Venus and Mars. The Mariner 2 spacecraft, launched on August 27, 1962, successfully reached Venus, while the Mariner 4 spacecraft, launched on November 28, 1964, provided valuable information about Mars' surface and atmosphere. The Mariner 9 spacecraft, launched on May 30, 1971, became the first to orbit Mars, revealing the planet's geological features and evidence of past water activity.

The International Geophysical Year of 1957 marked a significant milestone in global scientific collaboration, during which the Soviet Union's launch of Sputnik on October 4, 1957, sent shockwaves worldwide. The USSR placed a small satellite equipped with a radio transmitter into orbit, marking the beginning of space history. The JPL community was taken aback by the Soviet Union's success in developing electronic technology for both a successful launch vehicle and a satellite.

The U.S. quickly responded by tasking the Naval Research Lab with developing a spacecraft, but their initial attempt was unsuccessful. However, within approximately 80 days, JPL and the U.S. Army's Ballistic Missile Agency collaborated to design a four-stage rocket, placing the Explorer 1 satellite at its tip. On January 31, 1958, Explorer 1 became the first American satellite in space, carrying only one piece of data equipment that began transmitting data on the radiation environment around the Earth. This success led to the U.S. initiating a "space race" with the Soviet Union, and by December 1958, the U.S. Air Force and U.S. Army had transferred JPL to NASA.

In 1958, President Eisenhower granted permission to the U.S. Air Force and U.S. Army to develop a spacecraft to be sent to the Moon. Four spacecraft named Pioneer were developed, but the first three failed to reach their orbits due to various reasons. On March 3, 1959, the 6 kg Pioneer-4 spacecraft was placed into a lunar orbit to track space objects near the Moon.

NASA's Cape Canaveral facility (now Kennedy Space Center) launched the first Mariner spacecraft towards Venus on July 22, 1962, which failed due to a guidance system failure. However, on August 27, 1962, JPL successfully completed the first planetary mission by sending Mariner 2 to Venus. Mariner 2 confirmed Venus's incredible heat, with surface temperatures exceeding 400 degrees Celsius and a surface pressure 300 times greater than Earth, making it inhospitable to any form of life.

Some Mariner spacecraft were sent to Venus, while others were directed to Mars. On November 28, 1964, Mariner 4 generated significant excitement among scientists about the possibility of discovering an Earth-like Mars. However, on July 14, 1965, the images received showed a cold, cratered, moon-like surface, disappointing expectations of finding an Earth-like planet.

Mariner 9, the first spacecraft to orbit a planet, reached Mars on November 14, 1971, and encountered a global dust storm upon arrival. After the storm subsided, Mars' surface was found to be the smoothest of any planet, and evidence confirmed that water had once existed on the Martian surface. Mars had hundreds of kilometers of large outflow channels, but scientists could not find water. Running out of fuel, Mariner 9 completed its mission on October 27, 1972. Subsequently, manned spacecraft such as Apollo 12 were sent into space, marking the beginning of a new era in space exploration.

In the following years, space exploration expanded rapidly with the launch of manned spacecraft, such as Apollo 12, and the development of various types of rocket engines, including chemical and electric propulsion systems. In Europe, the European Space Agency (ESA) was established in 1970 to develop an autonomous space industry, and various types of rocket engines, including liquid and solid-fueled engines, were developed by companies such as Messerschmitt-Bölkow-Blohm and the French National Space Research Center [3].

Today, chemical rocket engines are well-established and widely used, while significant efforts are underway to develop 'green propellants'—non-toxic alternatives. The most substantial technology investments, however, are currently focused on *electric propulsion (EP)* for deep space exploration.

3.1.2. The Development of Electric Propulsion Systems Worldwide

In the report published by NASA in 2004, detailed information on the designs and theoretical modeling of electric propulsion systems (EPS) up to 2004 is provided. The increase in the number of EPS-equipped satellites due to the loss of EPS-equipped satellites between 1996 and 1997 has continued with a higher linear slope (Figure 3.3). According to the NASA Space Propulsion Roadmap 2009, HETs are a near-term propulsion technology. In general, it is expected that the use of EPS in all satellite classes will increase. This increase is expected to be mainly in commercial applications, but there is also an increase in interest in military applications. In commercial applications, the main reason for this increase is the data traffic caused by the internet (such as high-resolution image demands). In military applications, smaller satellites are entering service due to electronic miniaturization. According to scientific studies in the United States, there is a belief that there is still room for development in Hall Effect plasma systems, and the institutions that support this research have budget allocations for this purpose [5, 6, 7, 8]. It is understood that NASA has provided funding to Busek with the view that new fuel gases (such as iodine) will improve performance. The interest in simulation indicates that more efficient and high-performance systems can be developed with the existing systems [5].

This funding is consistent with the basic topics in plasma that the National Science Foundation supports in fundamental research. Plasma is not yet fully understood in academic circles, and there is a belief that it needs to be expanded. In the field of EPS,

long-term development expectations are indicated by the interest in experimental studies and contributions to experimental studies. The interest in small satellite (micro and CubeSat) systems indicates that better modeling and more testing can be done due to size-related design changes. The CubeSat field is supported at the hobby level. Applications based on immediate needs (such as disaster, war, etc.) are also of interest. The trend towards the use of hybrid systems in EPS (such as the Venus program) indicates that there is a focus on specific uses in emergency situations. For example, there is a focus on the recovery of the European Artemis spacecraft [9].

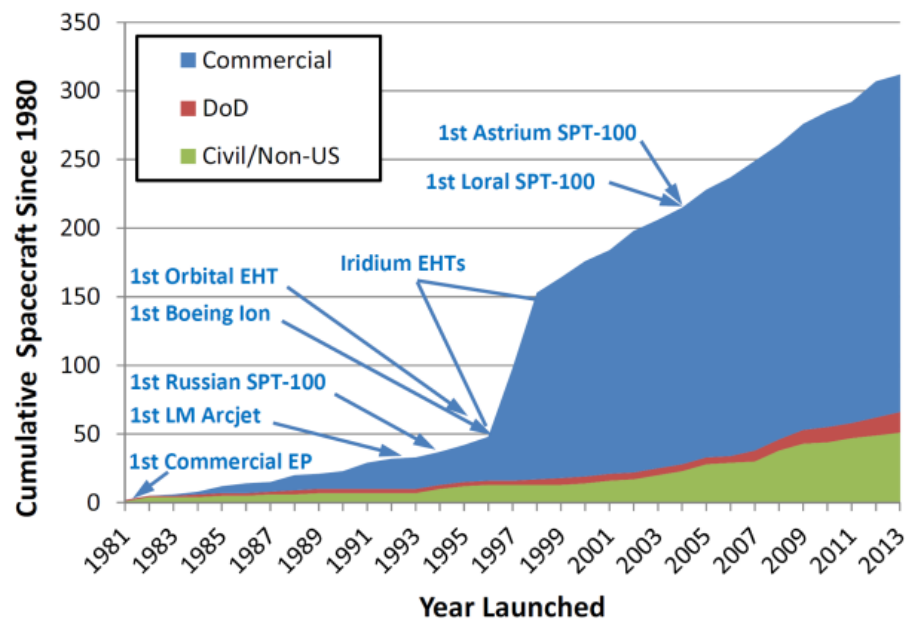


Figure 3.3: Change in the Number of Satellites using EPS over the Years [9].

In summary, if we were to outline the development of Hall Effect Thruster Systems worldwide, the concept of electric propulsion is not a new idea. The first basic concepts were proposed by Robert Goddard in the early 20th century. Serious research on electric propulsion began in the United States and the former Soviet Union in the 1950s. However, this technology transitioned from laboratories to space environments and started being used on satellites in the 1970s. This development was influenced by several independent factors. Over time, spacecraft capable of providing the power required for electric propulsion systems in orbit emerged, enabling high-power capabilities. Additionally, commercial competition among spacecraft manufacturers increased.

The Hall Effect is characterized by the voltage difference that develops across a conductor when a magnetic field perpendicular to the current direction is present. Discovered by Edwin Hall in 1879, it has been pivotal in the development of various technologies. Initial research on Hall Effect thrusters commenced in the 1960s, primarily in the United States and the former Soviet Union.

In the early years when Hall Effect thrusters emerged, the efficiency of EPS intended to be produced in the United States did not meet the desired values, leading to the cessation of Hall Effect thruster studies in the U.S. in 1970. The desired efficiency values could only be achieved with ion thrusters. Research on ion acceleration mechanisms in the United States led to improvements in thruster efficiency, and further research and development efforts paved the way for the development of ion thrusters. With the end of the Cold War, developments in Hall Effect thrusters created by the former Soviet Union began to flow to the outside world, leading to increased use and development of these systems in Western countries. In this context, the initial studies in the U.S. were focused on determining the performance of SPT and TAL thrusters. These studies were primarily conducted at the NASA Lewis Research Center and the Jet Propulsion Laboratory (JPL) to ensure flight reliability when using Russian-made SPT in Western spacecraft [8].

In the Soviet Union, high-power ion thrusters were used for interplanetary missions. Development efforts for low-power Hall Effect thrusters continued for satellite orbit insertion tasks. The first flight test of Hall Effect thrusters was conducted in February 1972 [10].

In this context, two types of Hall Effect thrusters were developed in the former Soviet Union [10]:

- Stationary Plasma Thruster (SPT), developed under the leadership of A.I. Morozov at the Kurchatov Institute.
- Thruster With Anode Layer (TAL), developed under the leadership of A.V. Zharinov at TSNIMASH.

Research efforts to improve power, performance, durability, and material quality in Hall Effect thrusters continue in the West and Russia.

3.1.3. Patent Research on HETs

The patent research on EPS has been conducted, and the current patents clearly show the level of current technology. Additionally, the patent applications related to the technology also reflect the global trend.

The fact that the most recent patents are considered indicates that the technology level being examined is also up to date. The information obtained from the patents during a technological research process provides a lot of information about the technology level and general direction. Patents:

- a) Provide comprehensive information about the patents that have been taken out in their entirety.
- b) The content of the patents includes a comprehensive discussion of the system.
- c) The technology to be developed is at the application level in the patents.

These features are of great importance in terms of providing significant information. In the evaluation process of patents and intellectual properties, it is important to consider the market structure of countries where patents are protected and the existence and protection of trade secrets. In the assessment of U.S. patents, the structure of import and export regimes and the structure of the manufacturing company's international trade plans become important. The location of the technology to be adapted, the related sub-technologies in the design, and the separate evaluation of each design and commercial situation are required. A general analysis of such complex structures can be difficult due to legal and commercial dimensions. Therefore, in such complex structures, developing practical systems is more possible by analyzing specific needs that have emerged because of clear definitions of specific goals rather than a general analysis [5].

The patent research results are presented in Table 3.1, which lists the patents held by organizations related to the subject. The entire patent study was conducted through the website [11], which is the Turkish extension of ESPACENET, an organization that collects and provides international patent data for statistical purposes.

When searching in the database, the keyword search included filters for "hall+effect+thruster" indicating that studies containing all these keywords in the title were compiled in Table 3.1.

Table 3.1: Patent Applications by Countries.

Year	Patent Owner	Country	Count	Patent No
2023	NASA	US	1	US11815074
	Orbion Space Tech Inc	US	1	US2023279845
	Inst. Nat. Polytech. Toulouse	FR	1	US2023193883
	Harbin Inst. Tech.	CH	1	CN115790932
	PIPL LTD	CY	1	WO2023057934
2021	Safran Aircraft Engines	FR	1	IL253659
2020	Univ Dalian Tech	CH	2	CN111852802 CN111219307
	Harbin Inst. Tech.	CH	2	CN111219307 CN111022276
	Shenzhen Magic Cubesat Tech	CH	1	CN210564917
	Guangzgou MFG	CH	1	CN110961637
	Orbion Space Tech Inc	US	1	WO2020005290
2019	Airbus Defence & Space	FR	1	US10718319
2018	Safran Aircraft Engines	FR	1	IL256754
	Airbus Defence & Space	FR	1	WO2018007542
2017	CALIFORNIA Inst Tech	US	1	US10480493
	Univ Beihang	CH	1	CN107165794
	Harbin Inst. Tech.	CH	1	CN106990085
	SNECMA	FR	1	UA114495
2015	SNECMA	FR	2	WO2015132534 US9234510
2014	SNECMA	FR	1	US9347438
	Harbin Inst. Tech.	CH	1	CN103596348
2013	Harbin Inst. Tech.	CH	1	CN103471738
	Private Individual	US	1	US2013026917
	SNECMA	FR	1	EP2564064

Table 3.1: Patent Applications by Countries. (continued)				
2012	SNECMA	FR	1	US8704444
	Private Individual	FR	1	US9127654
2011	BUSEK CO	US	1	US8550405
	SNECMA	FR	1	US8701384
2010 - 1997	Private Individual	US	1	US2010146931
	Space Power Inc	US	2	US5845880 CN1219279
	United Tech Corp	US	1	US7030576
	Centre Nat. Etd	FR	1	US7543441
	Aerojet General	US	1	US6982520
	BUSEK CO	US	1	US6834492
	Int Scient Products	US	1	WO9737126

Table 3.2 was created to provide a clearer distribution by country if Table 3.1 is summarized. It should be noted that studies from Ukraine and Russia are not included in this table. It is known that Russia, especially the FAKEL company, has many studies and patent applications in this field.

Table 3.2: Patent counts across countries.

Country	Number of Patent
France	14
USA	13
China	11

The given text discusses the patent applications related to HET made by major organizations from 1997 to 2023, with many of the patents being filed by U.S.-based companies. However, some Russian-based companies, such as Fakel, have also filed patents in the U.S. to expand their market presence.

From a technological perspective, there has been a significant increase in patent applications related to EPS in recent years. This trend can be interpreted as a shift in the U.S.'s focus from ion engines to HET technology. Additionally, when both

academic and patent research are considered, it can be observed that HET-related research is focused on developing smaller-scale propulsion systems. A notable example of this is Rafael (Israel) company's development of micro-drones that can perform long-term observation using HET technology.

The ranking of patent holders has the feature of pointing to the main topics of new technologies and related companies and institutions. The examination of patent protection areas reveals two main observations. Many Hall Effect Thruster (HET) designs have been patented, typically with a patent life of 20 or 17 years for U.S. patents. In recent years, specific ion components and system configurations of HETs have also been patented. For example, the detailed design and control of plasma and ion sources have been patented in large numbers in recent years. This situation indicates that patent protection areas have matured and that new patents must be detailed in areas based on experimentation and analysis. In addition, there are opportunities for improving efficiency, performance, and service life. Recent research articles (mostly simulation-based studies) also focus on these topics and indicate performance-efficiency improvement. Another important observation is that HET technology-related propulsion patents have started to emerge. The continuation of this trend is expected to continue in parallel with the trend of HET production technologies and the trend of smaller satellite production. In conclusion, the analysis of the patent field shows that companies wishing to enter this field must carefully analyze patents.

3.1.4. Heritage of Electric Propulsion Systems in Space

The increasing advancements in solar panel and thruster system technologies have made electric propulsion systems an appealing solution for the propulsion requirements of GEO satellites. The utilization of Hall thrusters and ion thrusters is becoming more widespread due to their provision of higher specific impulse (I_{sp}) levels compared to alternative propulsion systems. The recent trend in GEO satellites is towards fully electric satellites, which do not employ any chemical propulsion systems.

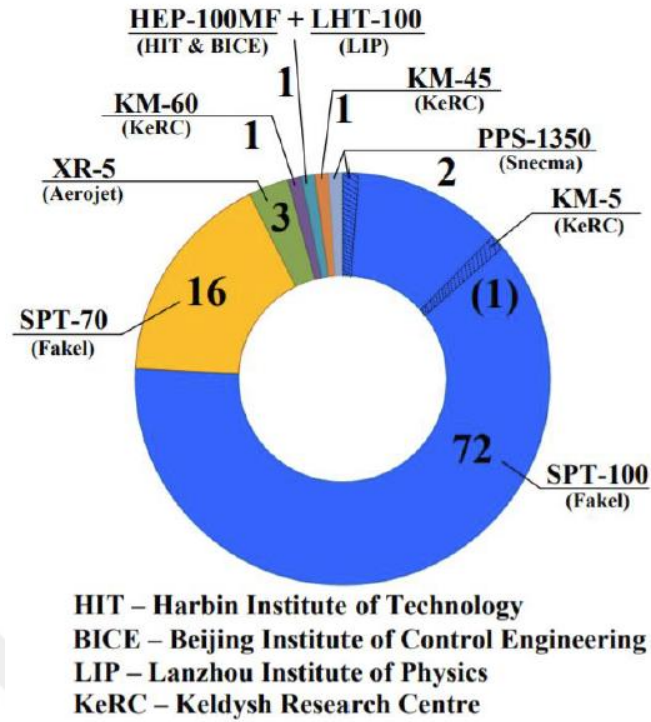


Figure 3.4: HETs used in GEO Satellites [8].

In 2015, the first all-electric satellite, the Boeing Eutelsat 115 West B, was successfully launched and completed its transfer from GTO to GEO using XIPS-25 thrusters. This marked the first time that two GEO communication satellites were stacked together for launch. In 2017, the Eutelsat 172 B, manufactured by Airbus, was launched by Ariane 5 into GTO and completed its transfer using Hall thrusters. Currently, Airbus is producing six all-electric satellites using the E300 EOR satellite platform. In Japan, the Japan Aerospace Exploration Agency (JAXA) plans to launch the Engineering Test Satellite, ETS 9, in 2021 to test their all-electric propulsion systems, utilizing SPT-140 Hall thrusters. In the United States, the XR-5 Hall thruster will be used for orbit-raising missions on the A2100 platform of Lockheed Martin and the GEOStar-3 platform of Orbital ATK. To date, electric propulsion systems, including Hall thrusters, ion thrusters, and arcjets, have been utilized in 208 satellites, with 46% of those employing Hall thrusters.

LEO satellites require different maneuvers than GEO systems, such as drag compensation. However, power limitations in smaller LEO satellites have constrained the use of electric propulsion in LEO systems. Another constraint for electric propulsion application in LEO is its low thrust capability for collision avoidance

requirements of LEO missions. LEO missions need collision avoidance maneuvers in frequently so this limitation coming from electric propulsion has negative impact on the LEO maneuver capabilities. To date, 19 LEO satellites have utilized electric propulsion systems, such as PPTs, arcjets, Hall thrusters, and ion engines. However, this situation may change in the future with planned satellite constellation projects, such as OneWeb and Starlink of SpaceX, which aim to provide internet access worldwide. Electric propulsion may enable maneuvers that are not possible with chemical systems, such as satellite servicing, repair, and refueling missions. Additionally, the development of air-breathing electric propulsion systems may make Very LEO missions possible.

Seven deep space and lunar missions have been performed using electric propulsion systems, with five of them utilizing ion thrusters. Several deep space electric propulsion missions are planned, including the European BepiColombo mission to Mercury, which launched in 2018 and utilizes gridded ion thrusters. The DART (Double Asteroid Redirection Test) mission of NASA, forecasted to be launched in 2021, will utilize a 7.5 kW NEXT-C gridded ion thruster and demonstrate an asteroid redirection mission for planetary defense. The spacecraft is designed to crash into the near-Earth asteroid Didymos.

In space research, development, and projects, the history of space plays a crucial role in the selection of products and systems. In this context, examining the history of electric propulsion systems is also important for demonstrating the capabilities of these systems. Table 3.3 lists the launch date, propellant/EPS type, and thruster primer country/company/comment for each space mission where EPS systems have been used. These missions demonstrate the capabilities of electric propulsion systems in space exploration.

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 [1,12].

Mission Name	Launch Date	Propellant/EPS Type	Thruster Primer Country/Company/Comment
Program 661A Flight A Flight B Flight C	1962 1964 1964	Cesium / ion engine	USA/EOS/experimental Flight A: no operation Flight B: Suborbital (19 min) Flight C: suborbital (4 min)

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)

Kosmos 21	1963	PTFE / PPT	USSR/Kurchatov Institute, OKB-1/Launch failure
Zond 1	1964	PTFE / PPT	USSR/Kurchatov Institute, OKB-1/partially tested
SERT 1	1964	Mercury / Ion Engine	USA/NASA-Hughes/First Ion engine demonstration (31 min)
Zond 2	1964	PTFE / PPT	USSR/Kurchatov Institute, OKB-1/first application of PPT in space (70 min)
Snapshot	1965	Cesium / ion engine	USA/EOS/experimental (1 hour)
Zond 3	1965	PTFE / PPT	USSR/Kurchatov Institute, OKB-1/thruster failed
Yantar-1	1966	Argon / ion engine	USSR//ballistic flight
Yantar-2	1968	Nitrogen / ion engine	USSR//ballistic flight
ATS-4	1968	Cesium / ion engine	USA/EOS/experimental (10 hours)
LES-6	1968	PTFE / PPT	USA/MIT Lincoln Lab./experimental
Yantar-3	1969	Air / ion engine	USSR//ballistic flight
ATS-5	1969	Cesium / ion engine	USA/EOS/experimentant
SERT 2	1970	Mercury / ion engine	USA/NASA/experimental (11 years and 3781 hours)
Meteor 1-10	1971	Xenon / HET (SPT-60)	USSR/OKB Fakel/orbit control
L-4SC-3	1974	PTFE / PPT	Japan/ISAS/experimental, launch failure
ATS-6	1974	Cesium / ion engine	USA/EOS/experimental (92 hours)
Meteor 1-18	1974	Xenon / HET (SPT-60)	USSR/OKB Fakel/orbit control
Kren-1	1975	Potassium / MPT	USSR/NIITP/ballistic flight
TIP-2	1975	PTFE / PPT	USA/Republic Aviation Co./experimental
Kren-2	1975	Potassium / MPT	USSR/NIITP/ballistic flight
Meteor 1-25	1976	Xenon / HET (SPT-60)	USSR/OKB Fakel/orbit control
TIP-3	1976	PTFE / PPT	USA/ Republic Aviation Co./experimental
Meteor 1-28	1977	Xenon / HET (SPT-50)	USSR/OKB Fakel/orbit control
Kust-1	1977	Air / MPT	USSR/ballistic flight
Kust-2	1978	Air / MPT	USSR/ballistic flight

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)			
Kosmos 1066	1978	Xenon / HET (SPT-50)	USSR/OKB Fakel/orbit control
Kust-3	1979	Air / MPT	USSR/ballistic flight
SCATHA	1979	Xenon / ion source	USA/Hughes/not a thruster, just ion beam generator
Intelsat-5 F-2	1980	Hydrazine / Resistojet	USA/ITSO,Inmarsat/Communication Satellite
MS-T4	1980	Ammonia / MPT	Japan/ISAS/experimental, for spin-up
ETS-4	1981	PTFE / PPT	Japan/AIST/experimental
Nova-1	1981	PTFE / PPT	USA/experimental, continuation of TIP missions
Meteor 1-31	1981	Xenon / HET (SPT-50)	USSR/OKB Fakel/orbit control
MDT-2A	1981	PTFE / PPT	China/Academia Sinica/ballistic
Kosmos 1366	1982	Xenon / HET (SPT-70)	USSR/OKB Fakel
ETS-3	1982	Mercury / ion engine	Japan/MELCO/experimental
Kosmos 1540	1984	Xenon / HET (SPT-70)	USSR/OKB Fakel
Nova-3	1984	PTFE / PPT	USA/Test
Kosmos 1700	1985	Xenon / HET (SPT-70)	USSR/OKB Fakel
Kosmos 1738	1986	Xenon / HET (SPT-70)	USSR/OKB Fakel
Kosmos 1818	1987	Xenon / HET (SPT-70)	USSR/OKB Fakel
Kosmos 1867	1987	Xenon / HET (SPT-70)	USSR/OKB Fakel
Start-1	1987	Air / MPT	USSR
Start-2	1987	Argon / MPT	USSR
Kosmos 1888	1987	Xenon / HET (SPT-70)	USSR/OKB Fakel
Kosmos 1897	1987	Xenon / HET (SPT-70)	USSR/OKB Fakel
Nova-2	1988	PTFE / PPT	USA/Test
Kosmos 1961	1988	Xenon / HET (SPT-70)	USSR/OKB Fakel
Kosmos 2054	1989	Xenon / HET (SPT-70)	USSR/OKB Fakel
Kosmos 2085	1990	Xenon / HET (SPT-70)	USSR/OKB Fakel

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)

Kosmos 2172	1991	Xenon / HET (SPT-70)	USSR/OKB Fakel
EURECA	1992	Xenon / RIT (RIT-10)	Germany-MBB/240 hours
Telstar 401	1993	Hydrazine / Arcjet	USA/Rocket Research Co./ communication Satellite
Gals-1	1994	Xenon / HET (SPT-100)	USSR/OKB Fakel/ communication satellite
ETS-6	1994	Xenon / ion engine	Japan/MELCO/experimental
Telstar 402	1994	Hydrazine / Arcjet	USA/Rocket Research Co. / satellite failed
Kosmos 2291	1994	Xenon / HET (SPT-70)	USSR/OKB Fakel
Ekspress 1	1994	Xenon / HET (SPT-100)	Russia/OKB Fakel
Luch 4	1994	Xenon / HET (SPT-70)	Russia/OKB Fakel
Space Flyer Unit (SFU)	1995	Ammonia / MPT	Japan/ISAS/experimental (43395 pulses)
Kosmos 2319	1995	Xenon / HET (SPT-70)	Russia/OKB Fakel
Telstar 402R	1995	Hydrazine / Arcjet	USA/Rocket Research Co. / communication Satellite
Luch 2-1	1995	Xenon / HET (SPT-70)	Russia/OKB Fakel
Gals-2	1995	Xenon / HET (SPT-100)	Russia/OKB Fakel/ communication satellite
Ekspress 2	1996	Xenon / HET (SPT-100)	Russia/OKB Fakel
A2100	1997	Arcjet	USA/Lockheed-Aerojet/ communication satellite
PAS-5	1997	Xenon / ion engine (XIPS)	USA/Hughes-EDD/first commercial satellite with XIPS
Kupon-1	1997	Xenon / HET (SPT-70)	Russia/OKB Fakel/ communication satellite
Galaxy 8i	1997	Xenon / ion engine (XIPS)	USA/Hughes-EDD
Astra 2A	1998	Xenon / ion engine (XIPS)	USA/Hughes-EDD
STEX	1998	Xenon / HET (TAL-D55)	Russia/TsNIIMash
Deep Space 1	1998	Xenon / ion engine (NSTAR)	USA/NASA
Satmex 5	1998	Xenon / ion engine (XIPS)	USA-Huhges EDD / propulsion failed after 12 years operation

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)			
PAS 6B	1998	Xenon / ion engine (XIPS)	USA/Hughes-EDD
ARGOS	1999	Ammonia / Arcjet	USA/Rocket Research Co. / experimental military
Orion 3	1999	Xenon / ion engine (XIPS)	USA/Hughes-EDD/injected into wrong orbit
Astra 1H	1999	Xenon / ion engine (XIPS)	USA/ Hughes-EDD
Yamal 101 and Yamal 102	1999	Xenon / HET (SPT-70)	Russia/OKB Fakel/ communication satellite
DirecTV 1R	1999	Xenon / ion engine (XIPS)	USA/ Hughes-EDD
Ekspress-A 1	1999	Xenon / HET (SPT-100)	Russia/OKB Fakel/satellite loss during launch
Galaxy 10R	2000	Xenon / ion engine (XIPS)	USA/ Hughes-EDD /malfunction in XIPS
Ekspress-A 2	2000	Xenon / HET (SPT-100)	Russia/OKB Fakel
SESAT 1	2000	Xenon / HET (SPT-100)	Russia/OKB Fakel
Galaxy 4R	2000	Xenon / ion engine (XIPS)	USA/ Hughes-EDD/ XIPS failed after 3 years.
Ekspress-A 3	2000	Xenon / HET (SPT-100)	Russia/OKB Fakel
Kosmos 2371	2000	Xenon / HET (SPT-70)	USSR/OKB Fakel
MightySat II.1	2000	PTFE / PPT	USA/NASA-Olin Aerospace
PAS 9	2000	Xenon / ion engine (XIPS)	USA/ Hughes-EDD
OSCAR-40	2000	Ammonia / Arcjet	Germany/University of Stuttgart/operated
EO-1	2000	PTFE / PPT	USA/Primex Aerospace
PAS 10	2001	Xenon / ion engine (XIPS)	USA/Boeing-EDD
Astra 2C	2001	Xenon / ion engine (XIPS)	USA/Boeing-EDD
Artemis	2001	Xenon / RIT (RIT-10 + UK-10)	Germany/Astrium
DirecTV 4S	2001	Xenon / ion engine (XIPS)	USA/Boeing-EDD
Ekspress-A 1R	2002	Xenon / HET (SPT-100 + KM-5)	Russia/OKB Fakel+Keldysh

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)

AsiaSat 4	2003	Xenon / ion engine (XIPS)	USA/Boeing-EDD
Hayabusa	2003	Xenon / ECR ($\mu 10$)	Japan/JAXA-ISAS
SMART-1	2003	Xenon / HET (PPS1350)	France/Snecma
Galaxy 13 (Horizons 1)	2003	Xenon / ion engine (XIPS)	USA/Boeing-EDD
Yamal 201 and Yamal 202	2003	Xenon / HET (SPT-70)	Russia/OKB Fakel
MBSat 1	2004	Xenon / HET (SPT-100)	Russia/OKB Fakel
Ekspress-AM 1 and AM 3	2004	Xenon / HET (SPT-100)	Russia/OKB Fakel
AMC 12	2005	Xenon / HET (SPT-100)	Russia/OKB Fakel
Ekspress-AM 2	2005	Xenon / HET (SPT-100)	Russia/OKB Fakel
Monitor-E	2005	Xenon / HET (SPT-100)	Russia/OKB Fakel
AMC 23	2005	Xenon / HET (SPT-100)	Russia/OKB Fakel
KazSat-1	2005	Xenon / HET (SPT-70)	Russia/OKB Fakel
ION	2006	Tungsten / Vacuum arc thruster (μ VAT)	USA/UIUC/satellite loss during injection
MEASAT 3	2006	Xenon / ion engine (XIPS)	USA/Boeing-EDD
TacSat-2	2006	Xenon / HET (BHT-200)	USA/Busek/military
ETS-8	2006	Xenon / ion engine (XIES)	Japan/MELCO
FalconSat-3	2007	PTFE / PPT	USA/Busek/experimental
Dawn	2007	Xenon / ion engine (NSTAR)	USA/NASA
Telstar 11N	2009	Xenon / HET (SPT-100)	Russia/OKB Fakel
GOCE	2009	Xenon / ion engine (T5)	UK/QinetiQ
Sirius FM5	2009	Xenon / HET (SPT-100)	Russia/OKB Fakel
STSAT-2A	2009	PPT	South Korea/KAIST-SaTReC /satellite loss during injection

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)

NSS-12	2009	Xenon / HET (SPT-100)	Russia/OKB Fakel
GSAT-4	2010	Xenon / HET (KM-45)	Russia-Keldysh/ISRO/launcher failure
STSAT-2B	2010	PPT	South Korea/KAIST-SaTReC /satellite loss during injection
AEHF-SV-1	2010	Xenon / HET (XR-5)	USA/Aerojet
FalconSat-5	2010	Xenon / HET (BHT-200)	USA/Busek/experimental
KazSat-2	2011	Xenon / HET (SPT-70)	Russia/OKB Fakel
Fobus-Grunt	2011	Xenon / HET (SPT-140)	Russia/OKB Fakel/failed to reach target orbit
SES-4	2012	Xenon / HET (SPT-100)	Russia/OKB Fakel
AEHF-SV-2	2012	Xenon / HET (XR-5)	USA/Aerojet
SES-5	2012	Xenon / HET (SPT-100)	Russia/OKB Fakel
Kanopus-V	2012	Xenon / HET (SPT-50)	Russia/OKB Fakel/earth observation satellite
TET-1	2012	Water / Resistojet	Germany/Aerospace Innovation/experimental
PROITERES	2012	PTFE / PPT	Japan/OIT/academic satellite, loss before propulsion tests
Shijian-9A	2012	Xenon / HET (HET-40) + ion engine (LIPS-200)	China/SISP+LIP
Yamal 402	2012	Xenon / HET (SPT-100)	Russia/OKB Fakel/ failed to reach target orbit
STRaND-1	2013	PPT	UK/SSTL/cubesat demonstration
Alphasat	2013	Xenon / HET (PPS1350)	France/Snecma
AEHF-SV-3	2013	Xenon / HET (XR-5)	USA/Aerojet
CUSat	2013	PTFE / PPT	USA/Cornell University
STSAT-3	2013	Xenon / HET	South Korea/KAIST-SaTReC
DubaiSat-2	2013	Xenon / HET	South Korea-Satrec Initiative +Japan-JAXA
Wren	2013	PPT	Germany/Stadoko/pocketcube
Ekspress-AM 5	2013	Xenon / HET (SPT-100)	Russia/OKB Fakel / EP is used for orbit raising
KazSat-3	2014	Xenon / HET (SPT-100)	Russia/OKB Fakel
Deimos-2	2014	Xenon / HET	South Korea/Satrec Initiative

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)			
Hodoyoshi 4	2014	Xenon / ion engine	Japan/University of Tokyo
Vernov	2014	PTFE / PPT	Russia/RIAME/ loss before thruster operation
Ekspress-AM 6	2014	Xenon / HET (SPT-100)	Russia/OKB Fakel / EP is used for orbit raising
Hayabusa2	2014	Xenon / ion engine	Japan/JAXA-ISAS
PROCYON	2014	Xenon / ion engine	Japan/University of Tokyo / >200 hours operation
Yamal 401	2014	Xenon / HET (SPT-100)	Russia/OKB Fakel
ABS-3A	2015	Xenon / ion engine (XIPS-25)	USA/L-3 ETI/Boeing 702SP
Satmex 7	2015	Xenon / ion engine (XIPS-25)	USA/L-3 ETI/Boeing 702SP
BRICSat-P	2015	Vacuum arc thruster (μ CAT)	USA/GW University/detumbling, spin and station keeping for cubesat
AeroCube 8A and 8B	2015	Ionic liquid / Electrospray Thruster	USA/MIT/Cubesat
KaiTuo-1A	2015	Xenon / HET	China/LIP+BICE
Lisa Pathfinder	2015	Ionic liquid / Electrospray Thruster	USA/Busek
Horyu-4	2016	Vacuum arc thruster	Japan/KIT
SES-9	2016	Xenon / ion engine (XIPS-25)	USA/L-3 ETI/Boeing 702SP
ABS-2A	2016	Xenon / ion engine (XIPS-25)	USA/L-3 ETI/Boeing 702SP
Satmex 9	2016	Xenon / ion engine (XIPS-25)	USA/L-3 ETI/Boeing 702SP
Shijian-17	2016	Xenon / HET	China/LIP
AeroCube 8C and 8D	2016	Ionic liquid / Electrospray	USA/MIT
AOBA-VELOX III	2016	PTFE / PPT	Singapore/NTU
Hispasat 36W-1 (SmallGEO)	2017	Xenon / HET (SPT-100)	Russia/OKB Fakel
SES-10	2017	Xenon / HET (SPT-100)	Russia/OKB Fakel

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)

Shijian-13	2017	Xenon / ion engine	China/LIP
GSAT-9	2017	Xenon / HET (KM-45)	Russia/Keldysh
SES-15	2017	Xenon / ion engine (XIPS-25)	USA/L-3 ETI/Boeing 702SP
Eutelsat 172B	2017	Xenon / HET (SPT-140)	Russia/OKB Fakel
PEGASUS	2017	PTFE / PPT	Austria/FH Wiener
Shijian-18	2017	Xenon / ion engine	China/LIP
VenµS	2017	Xenon / HET	Israel/Rafale
AsiaSat-9	2017	Xenon / HET (SPT-100)	Russia/OKB Fakel
SLATS	2017	Xenon / ion engine	Japan
Flock-3p	2018	Indium / FEEP	Austria/Enpulsion/First successful demonstration of FEEP
SES-14	2018	Xenon / HET (SPT-140)	Russia/OKB Fakel
SES-12	2018	Xenon / HET (SPT-140)	Russia/OKB Fakel
NovaSAR	2018	Xenon / Resistojet	UK/SSTL
BepiColombo	2018	Xenon / ion engine	UK/QinetiQ
2 Undisclosed mission	2018	Indium / FEEP	Austria/Enpulsion
FalconSat-6	2018	Xenon / HET (BHT-200)	USA/Busek
UWE-4	2018	Gallium / FEEP	Germany/Morpheus Space
AOBA-VELOX-IV	2019	PTFE / PPT	Singapore/NTU
Xiaoxiang-1	2019	Electrospray thruster	China/206 th Institute
Undisclosed mission	2019	Indium / FEEP	Austria/Enpulsion
PSN-6	2019	Xenon	USA/SSL
Undisclosed mission	2019	Indium / FEEP	Austria/Enpulsion
Starlink L0	2019	Krypton / HET	USA/SpaceX
Yamal 601	2019	Xenon / HET (SPT-100)	Russia/OKB Fakel
2 Undisclosed mission	2019	Indium / FEEP	Austria/Enpulsion

Table 3.3: Satellites with EPS Launched into Space between 1964 - 2023 (continued)			
JCSAT-18	2019	Xenon / ion engine	USA/Boeing-EDD
Shijian-20	2019	Xenon / ion engine	China/LIP
Eutelsat Konnect	2020	Xenon	
Turksat 5A	2021	Xenon / HET (SPT-140)	Russia/OKB Fakel
Turksat 5B	2021	Xenon / HET (SPT-140)	Russia/OKB Fakel
DART	2021	Xenon / ion engine	USA/Aerojet Rocketdyne
IMECE	2023	Xenon / HET (HALE)	Turkiye/Tubitak Uzay
MicroHETSat	2023	Xenon / HET	Italy/Sitael

The data presented in Table 3.3 does not encompass all the satellites, as it is known that there are more satellites using members of the Safran PPS thruster family. Considering the information from the paper related to “European Space Agency Electric Propulsion Activities” [13] that regularly shows the EPS activities in ESA and mentions a total of 100 PPS5000 orders with 47 deliveries, and assuming each satellite uses an average of 4 thrusters, this would imply around 25 historical satellites. This represents a much larger number of historical satellites, especially for France. The search results indicate that ENPULSION's FEED-type thrusters are present on over 100 spacecraft in orbit, with more than 200 orders. Assuming half of the orders are in orbit and each satellite uses 4 thrusters, this would mean these thrusters have been used in around 75 different missions, significantly adding to the historical data, particularly for Austria. These data points are not included in the table, and it is important to note that the table is incomplete in this regard. However, it would still be beneficial to group and present the results available in the table as a reference. In light of these additional data points, it is crucial to consider the big picture and evaluate the trends in electric propulsion technology.

The data from Table 3.3 has been compiled to show the distribution of EP history by country. This study can reveal which country stands out and dominates the EP sector. While there may be some missing numbers, the overall picture can be trusted based on the data in Figure 3.5. Two countries that have been competing in the space sector for many years stand out: the United States and Russia, with the European countries (the

European Union) following closely behind. Figure 3.5 has been created to better interpret this data.

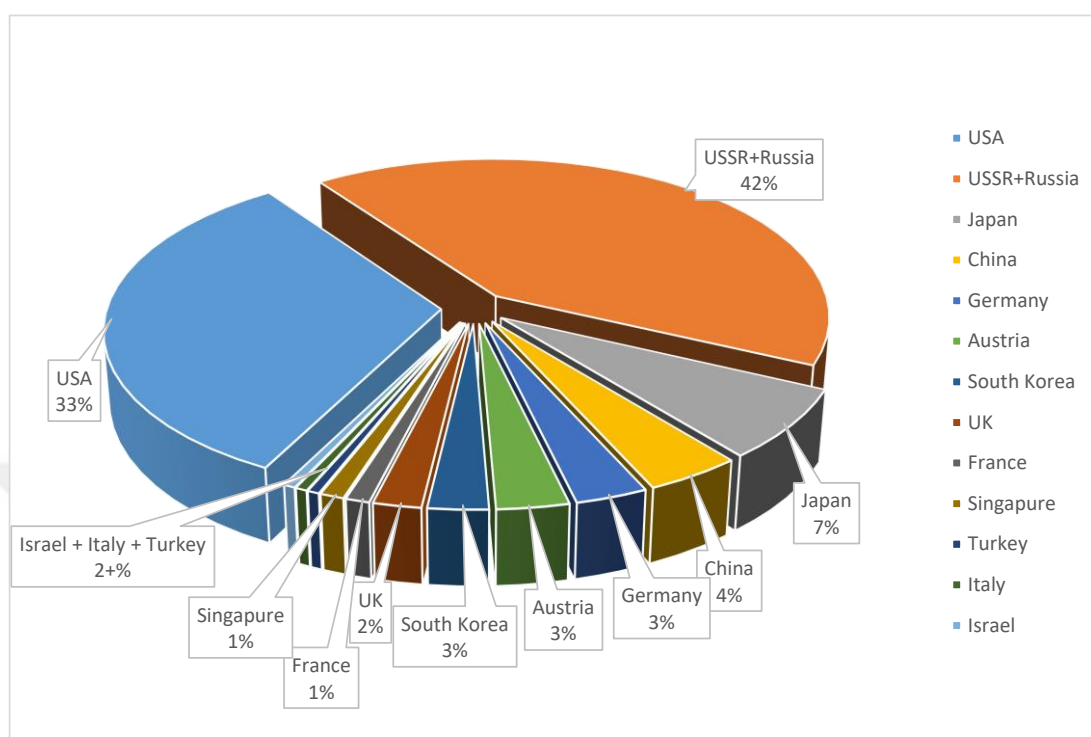
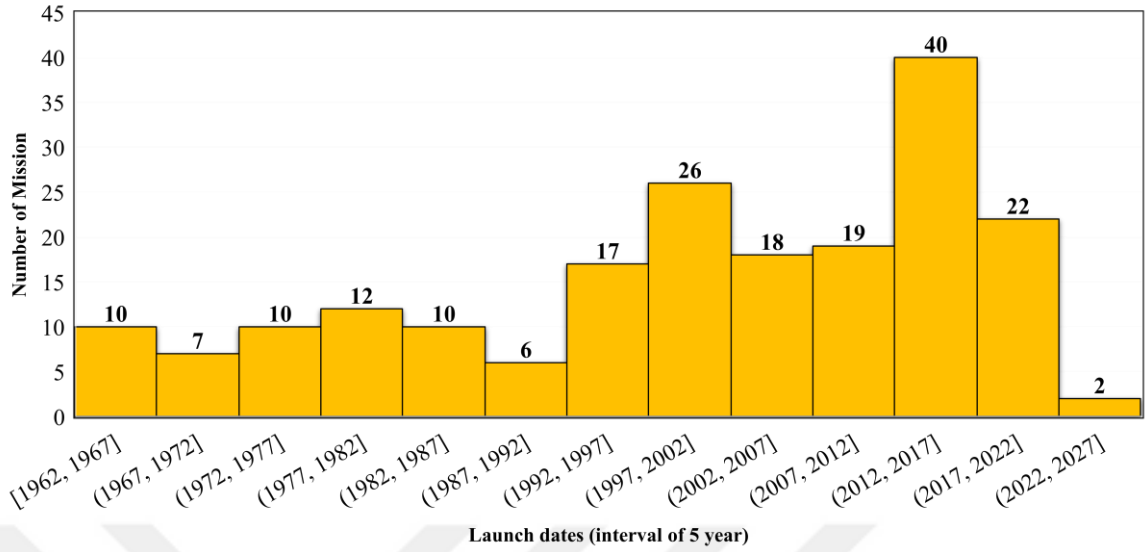


Figure 3.5: EP Heritage Distribution by Countries.

The detailed interpretation of the data in Table 3.3 is crucial to observing and analyzing the industry's trajectory. In this section, the aim is to extract the year-by-year trend of progress in the EP sector. The goal is to reveal the periods when investments were made in EP based on the year-by-year trend. The work conducted in 2023–2024 indicates a significant increase in EP usage and trend. However, the section covering 2022–2027 in this study is still ongoing and will be accurately interpreted in future studies. There is no doubt that between 2022 and 2027, more EP satellites will be in orbit than in the previous decade. Table 3.4 allows for the observation of the trend in EP usage over the years, divided into 5-year intervals.

Table 3.4: EPS Preference Trend by Years.



The data from the Satsearch website [14] provides valuable information on space equipment, including datasheets and detailed product information. This website is widely used by engineers for product selection on the market. Utilizing the dataset from satsearch, it is possible to determine the total number of propulsion models, the proportion that are electric thrusters, and specifically how many of those are Hall Effect Thruster (HET) models.

Among the electric thrusters, HET systems constitute 16% of the total, holding a significant share within the 53% segment dedicated to electric propulsion. This analysis is based on data compiled for a total of 224 thrusters, comprising 106 chemical thrusters and 118 electric thrusters, of which 36 are HET models.

Compiling these numbers and ratios of propulsion systems is crucial for understanding market dynamics and the transformation of investments into products. Interpreting the statistical information presented in Figure 3.6 provides insights into the sector's development direction.

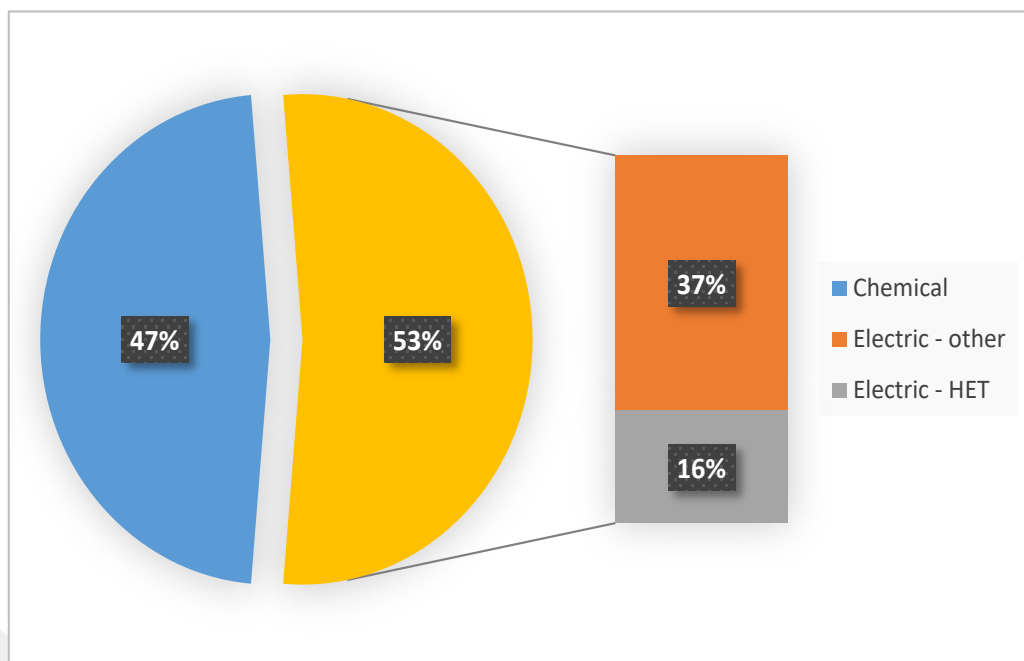


Figure 3.6: EPS and HET Proportions.

4. SATELLITE ELECTRIC PROPULSION SYSTEMS

Electric propulsion (EP) encompasses a wide range of strategies aimed at achieving high exhaust velocities to reduce the total fuel mass and corresponding launch weight of current and future space transportation systems. These strategies generally fall into three categories: electrothermal propulsion, where electrically heated propellant gas is thermodynamically expanded through a nozzle; electrostatic propulsion, where ionized propellant is accelerated through an electric field; and electromagnetic propulsion, where a current passing through an ionized plasma interacts with an external magnetic field to generate thrust in the opposite direction of the flow. These systems can achieve exhaust velocities and payload efficiencies significantly higher than those of the most advanced chemical rockets, enabling a wide range of compelling space missions and substantial advancements in space exploration.

Nonetheless, these systems have much lower attainable thrust consistence (thrust force per unit exhaust area), taking longer flight durations, and more complex charge circles. Also, these systems bear spaceborne electric power sources associated with applicable power processing outfits that are low-specific-mass and largely dependable. Thus, the optimization of EP systems involves multidimensional trade-offs among charge objects, energy and power factory mass, trip time, internal and external environmental factors, and overall system trust ability.

A persistent international effort in the development of practical electric thrusters has spanned decades, resulting in the increasingly widespread adoption of efficient EP systems in commercial and government space vehicles. Concurrently, advancements in EP technologies are pushing towards higher thrust capabilities for future missions.

Electric propulsion has proven to be a cost-effective and efficient solution for a wide range of space missions, offering increased operational flexibility and the ability to accomplish tasks traditionally performed by chemical propulsion systems while reducing spacecraft mass requirements. These propulsion systems have undergone rigorous testing both on the ground and in space since the 1960s, resulting in a diverse array of available system types. The emphasis is placed on systems compatible with

solar electric propulsion that are commercially viable, although other propulsion technologies are briefly mentioned for comprehensive coverage. The pursuit of perfection in propulsion technologies can be an ongoing endeavor, with certain methods like solar photon sailing, emulsion, railguns, and ray propulsion not extensively explored. This study provides concise summaries of fundamental propulsion system principles, benefits, drawbacks, testing experiences, spaceflight applications, and descriptions of the current electric propulsion systems available.

4.1. Classification of Electric Propulsion Types

There are various types of electric propulsion (EP) systems, which can be challenging to categorize comprehensively. Numerous researchers have endeavored to categorize these systems, but consensus remains elusive. For example, Hall effect thrusters are sometimes categorized as electromagnetic thrusters by some and as electrostatic thrusters by others.

The primary objective typically involves accelerating the propellant to achieve efficient operation. However, alternative methodologies may vary in ion generation techniques or fuel composition, deviating from conventional approaches. It is important to note that different approaches utilize different principles of force application. Despite these differences, the overarching goal remains efficiency.

In solar electric propulsion, where efficiency is a key consideration, it is essential to recognize that certain systems perform better with higher power inputs. This underscores the significance of exploring alternative power sources, such as nuclear or thermonuclear options, which have the potential to significantly enhance propulsion capabilities and mission efficiency. By harnessing the immense power output of nuclear or thermonuclear sources, spacecraft could achieve increased thrust, extended operational ranges, and improved maneuverability for complex space missions.

The utilization of these advanced power technologies opens new horizons for space exploration, enabling missions to reach farther destinations, carry heavier payloads, and operate more effectively in challenging environments. The integration of nuclear or thermonuclear power sources represents a promising avenue for advancing propulsion systems and pushing the boundaries of what is achievable in space exploration.

To categorize these systems, a conventional framework involves primary classification based on the method of accelerating the working fluid and secondary classification based on the method of generating the fluid (ionization in the case of plasma). Generally, electric propulsion systems are classified into three main types:

- Electrostatic
- Electromagnetic
- Electrothermal

These categories primarily differ in how plasma is generated (ionization) or how heat is applied to the propellant.

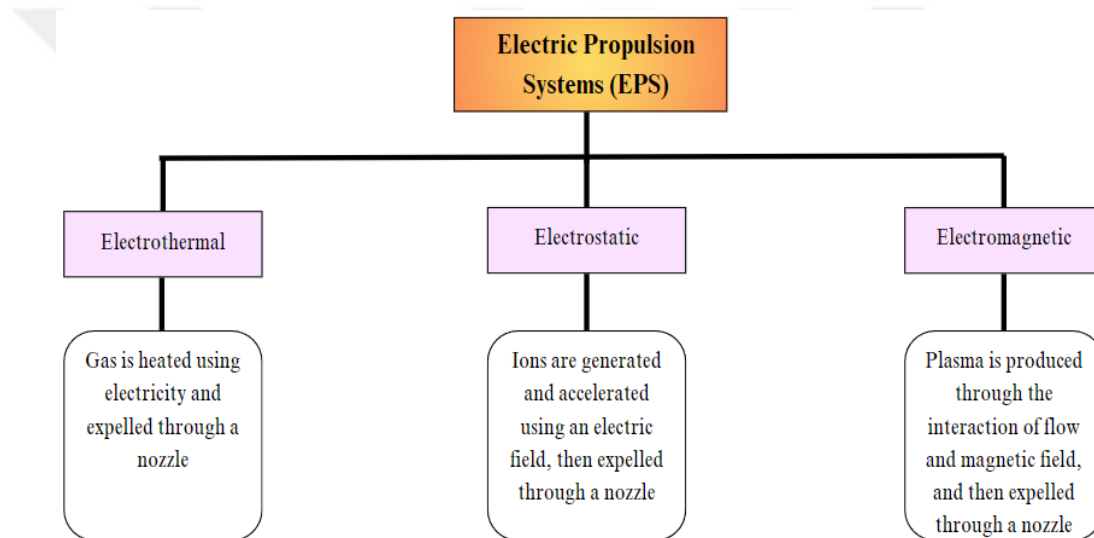


Figure 4.1: Main Types of EPS.

4.1.1. Electrostatic Propulsion

Electrostatic propulsion systems function by utilizing a powerful, high-voltage electrostatic field to drive ions towards achieving high exhaust velocities. Recent progress in this field has been directed towards systems that employ charged particles as the primary operational medium, integrating techniques to neutralize the ions upon reaching the exhaust. Almost all electrostatic propulsion systems feature a gridded layout at the exhaust outlet to confine and generate the intense electric field essential for ion acceleration, encompassing ion bombardment, RIT, and colloid thrusters. Typically, a gridded thruster operates by maintaining a DC potential difference of

approximately 1 kV between an inner grid anode located within the tube chamber and an exit-plane cathode, enabling effective ion acceleration and thrust production. At the exhaust port, several grids are employed to separate the functions of fuel confinement, ion acceleration, and beam divergence control. However, the effectiveness of conventional systems is limited by the radiative heat losses from the fuel heating and ionization chambers. Depending on the system used for fuel ionization, the power conditioning requirements can be complex. Due to grid and cathode erosion, grid electrostatic systems have had a limited lifetime. Advances in the materials used, particularly in the carbon-carbon composite grids, are currently being developed and are expected to improve the performance and life of these units.

Positive ions are produced by bombarding neutral fuel atoms with thermion-driven electrons in a discharge chamber. Typically, a cylindrical anode with a centrally located hollow axial cathode is used as the discharge chamber. They discharge towards the external anode with low current (1 to 15 amperes) and low voltage levels (25 to 30 VDC) thermionic electron emission by heating the axial cathode. To extend the trajectory and dwell time of the electrons and thus improve the probability of collision and the efficiency of fuel utilization, a magnetic field is implemented within the discharge channel. Furthermore, grids positioned at varying potentials near the exhaust port accelerate the plasma to achieve high exhaust velocities. To ensure that ions are not drawn back into the engine, an electron dispersion from a countercurrent cathode is used to dissipate ions in the exhaust.

4.1.1.1. Ion Bombardment

With more than 30 successful space missions and 40 ground tests, ion bombardment systems have a proven track record. During the 1970s, several low-earth orbit missions validated the design through ground testing and qualification. For example, north-south station-keeping maneuvers are performed by a 25 mN thruster UK-10 on the Intelsat VII mission. Since the annual ΔV requirement for east-west station keeping maneuvers is approximately 20 times that of north-south station keeping maneuvers at geosynchronous missions, these thrusters have been re-designed to a more comprehensive and qualified level, making them resistant to different mission phases and long mission lives.

Deep Space-1, which uses a 30 cm-diameter NASA/Hughes Xenon ion thruster, is the most notable example to date. The developed engineering models were tested with thrust levels greater than 1 N and power consumption ranging from 50W to 200kW. Several countries and organizations participated in the design development and flight unit production. Occasionally, as has been reported with recent systems, gridded acceleration can be hypersensitive. To remove contamination that formed a layer in the accelerator electrodes, the DS-1 engine needed to undergo hundreds of cycles before producing any thrust. Similar anomalies were encountered during the flight of the original ion bombardment-based spacecraft, which was the SERT 2 mission.

Gridded accelerators present a series of obstacles, while discharge chambers may face sputtering and spalling. Nonetheless, the appeal of these thrusters persists, owing to their remarkable performance, extensive developmental background, and inherent resilience. Ion propulsion systems demonstrate unparalleled cumulative thrust capabilities, positioning them favorably for both station-keeping and primary propulsion roles. There is widespread interest in the commercial utilization of this technology. Ongoing testing focuses on "bitsy" micro-thrusters featuring a hollow cathode design.

Ion engines, developed under the guidance of Dr. Harold Kaufmann at NASA's Lewis Research Center (currently known as John H. Glenn Research Center) in the 1950s, employed a fuel injection system that injects fuel into the engine's exhaust channel, where it is ionized by primary cathode-emitted electrons. The resulting ions are accelerated by a series of grids with applied voltage to achieve high velocity. A secondary cathode is present at the engine's output to supply electrons and neutralize the high-velocity, positively charged ions expelled into the external environment [2, 15].

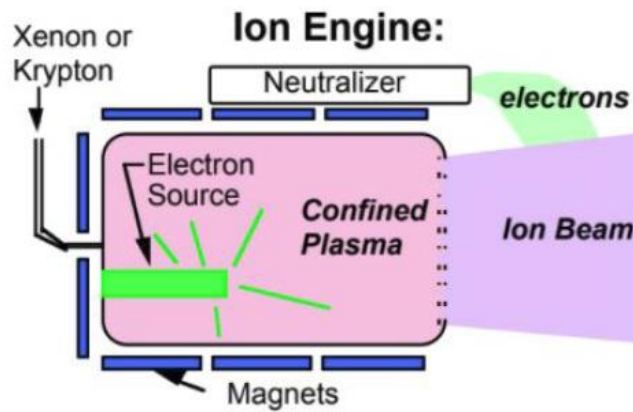


Figure 4.2: Ion Engine [1].

The initial models of ion engines primarily utilized heavy, solid fuels such as cesium and sodium in the 1950s [6]. These fuels provide high specific impulse while maintaining minimal neutral fuel loss. Additionally, their low ionization voltage increased the overall efficiency of the system [2]. However, in the 1980s, there was a shift towards using xenon and other noble gases as propellants in ion engines. This change was primarily due to the environmental concerns associated with cesium and sodium, which could cause atmospheric pollution during ground testing and unwanted buildup on spacecraft surfaces during in-flight [2]. Xenon became a popular choice due to its inert nature, which eliminated the risk of reacting with spacecraft materials. Furthermore, Xenon's high molecular weight provided sufficient momentum for the spacecraft, making it a suitable replacement for heavy, solid fuels. The first use of Xenon in ion engines was during the Deep Space 1 mission, which was part of NASA's New Millennium Program [16]. This mission successfully demonstrated the feasibility of using Xenon as a propellant in ion engines, paving the way for its widespread adoption in subsequent space missions.

4.1.1.2. Radiofrequency (RIT) & Microwave (MIT) Ion Thrusters

Radiofrequency ion thrusters represent a cutting-edge propulsion technology that utilizes radio frequencies to create highly efficient, long-lived, and environmentally friendly spacecraft propulsion systems. Unlike traditional chemical rockets or DC ion engines, RF ion thrusters do not require consumables, resulting in significantly lower mass penalties during mission planning. This innovative approach has led to their widespread adoption across diverse space exploration endeavors, from nanosatellite constellations to geostationary orbit maintenance operations.

In the RF (radio frequency) ion engine, a type of ion engine, fuel ionization occurs not through a hot cathode but via electromagnetic fields. This system includes an ionization channel made of insulating material, surrounded by an RF coil. The RF coil induces an axial magnetic field, creating a secondary circular electric field where free electrons gain energy through ionization collisions [17]. Positively charged heavy particles are then accelerated between grids via electrostatic force to generate thrust, similar to the principle employed in conventional ion engines [18]. The operational principle of the RF ion engine is illustrated in Figure 4.3.

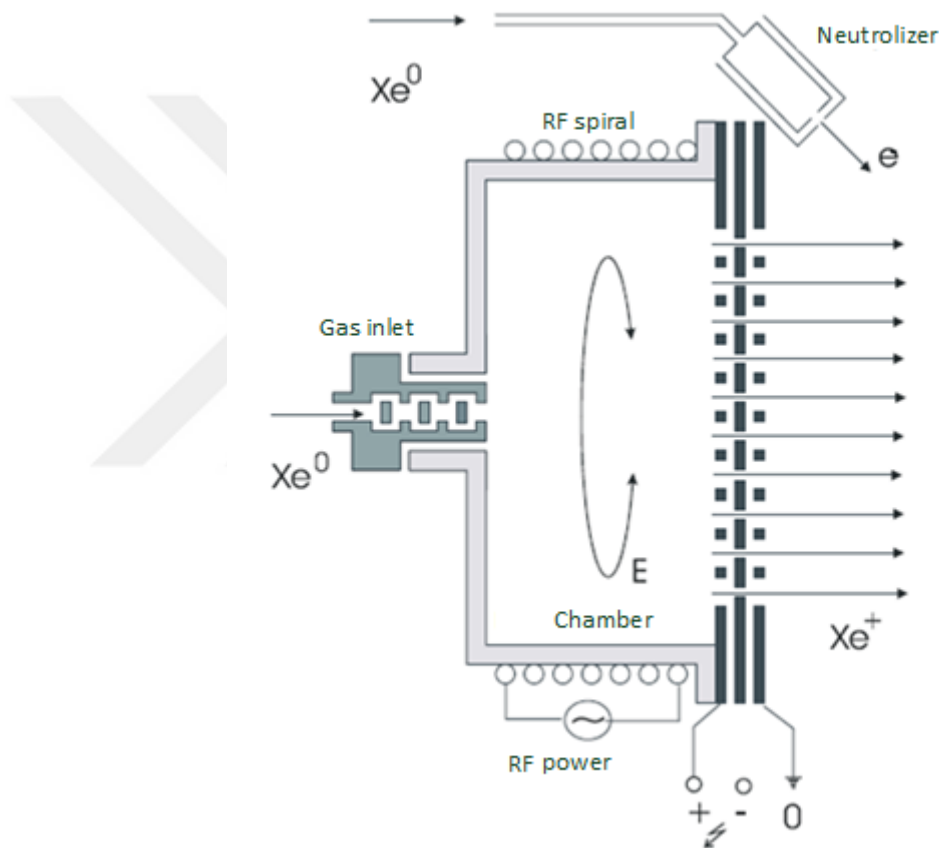


Figure 4.3: RF Ion Thruster [17].

The RIT model's successful demonstration paved the way for larger versions, showcasing the scalability of this technology. As the industry moves towards sustainable spaceflight practices, RF ion thrusters present a compelling alternative to conventional propulsion methods due to their inherent efficiency and minimal environmental impact. With ongoing research and development efforts, particularly at the European Space Agency, RF ion thrusters promise to revolutionize the future of

space transportation, enabling new possibilities for deep space exploration and Earth observation missions alike.

RF thrusters' unique advantages include their ability to operate continuously without degradation, providing consistent performance throughout extended periods. This characteristic aligns well with the demands of modern spacecraft requirements, where precision maneuverability and prolonged operation times are essential factors. Furthermore, the absence of moving parts in RF ion thrusters results in reduced wear and tear, leading to increased reliability and durability compared to other types of propulsion technologies.

As the global community strives to reduce its carbon footprint, RF ion thrusters emerge as a promising solution for reducing greenhouse gas emissions associated with space travel. By eliminating the need for toxic fuels and minimizing waste products, RF ion thrusters contribute positively to sustainability goals while promoting responsible space exploration practices.

4.1.1.3. Electrostatic Colloid Thrusters

Electrostatic Colloid thrusters utilize the principles of electrostatics to propel microscopic solid particles, colloidal beads, through space using electricity alone. Unlike traditional electric propulsion systems, colloid thrusters employ conducting liquids such as sodium and glycerol, which are pumped with a high voltage difference, typically around 5–10 kV. A negative extractor plate pushes the liquid into a stream until the positively charged side isolates from the stream, resulting in significant reductions in mass and complexity. However, despite their intriguing characteristics, colloid thrusters face several technical challenges, including low specific impulse, limited thrust capabilities, inconsistent particle production, and energy requirements for vapor condensation, which reduce efficiency.

Electrostatic colloid thrusters have been developed and refined over several decades, with initial studies dating back to the 1960s and 1970s. Researchers worldwide have dedicated considerable effort to addressing these limitations, aiming to optimize colloid thruster designs and improve overall performance. For instance, Busek Corporation's work on a 25 Micro-Newton class colloid thruster for the SBIR Air Force satellite system represents one such attempt to overcome existing obstacles. Similarly, Stanford University's contributions to the study of miniature colloid

thrusters highlight the growing interest in exploring novel approaches to enhance the viability of this emerging propulsion technology.

Colloid thrusters' unique attributes make them appealing candidates for specialized space missions, such as formation flying, rendezvous, and proximity operations. Due to their relatively simple construction and lack of complex mechanical components, they may prove advantageous when operating in harsh environments or demanding conditions. Moreover, colloid thrusters' compatibility with renewable energy sources further supports their role in fostering sustainable space exploration initiatives.

4.1.1.4. Contact Ion Motors

Contact ion motors function by transforming fluids such as cesium into ions through a process involving passing cesium fluid through a heated bed of permeable tungsten. The higher work function of tungsten compared to cesium facilitates electron exchange between cesium molecules and tungsten, causing cesium particles to vaporize from the tungsten surface. These vaporized cesium ions are then accelerated by an electrostatic field.

During the 1960s and '70s, the practical implementation of this type of motor faced significant challenges that limited their development beyond a 20-kilowatt power level. These challenges included complex structural designs to meet unique operational requirements, stringent thermal management to handle high operational temperatures, concerns over the toxicity risks of cesium, and the overall complexity of system integration. Additionally, a French program dedicated to contact ion motors was terminated in the mid-1970s following extensive testing conducted in the United States and in-orbit trials.

4.1.1.5. Field Emission Electric Propulsion

Field Emission Electric Propulsion (FEEP) is a propulsion system that utilizes a strong electric field, typically between 8 and 15 kilovolts, to ionize working fluids, often liquid metals. This innovative approach sets FEEP apart from traditional colloid ion thrusters, as it produces individual ions through apertures or pinholes approximately one micron in diameter, rather than ionized beads. FEEP achieves high efficiency due to its direct ionization process, which minimizes heat loss during ionization and allows for immediate acceleration of the ionized liquid.

From a spacecraft control standpoint, FEEP offers distinct advantages such as the consistency of pulse durations and precise impulse control, enhancing maneuverability and operational flexibility. The compact design of thrust units associated with FEEP not only provides relatively high specific impulse but also makes it an ideal choice for missions requiring precise control and efficient propulsion in the face of challenges like atmospheric drag and solar radiation pressure.

Organizations like Enpulsion have been at the forefront of developing flight units utilized in various satellites for test purposes. Their active promotion of these advanced propulsion systems underscores their potential for specialized station-keeping missions, emphasizing the critical role FEEP plays in advancing science-sensitive endeavors within the realm of space exploration.

4.1.1.6. Plasma Separator Ion Thrusters

The plasma separator ion thruster is a propulsion system that utilizes high-density cesium vapor actively circulated through a group of hollow cathodes featuring converging-diverging nozzles. This innovative design allows for the ionization of particles through discharge, which occurs between the anode ring positioned at the outport and the cathodes. After ionization, high nozzle exit velocities are achieved thanks to the acceleration system, enabling efficient propulsion firings.

During laboratory testing, it was noted that non-uniform plasma from the cathode cluster could impact the thruster's performance. Despite promising advancements in technology, development investments in plasma separator thrusters in the United States were suspended due to a lack of established spaceflight heritage. This pause in progress highlights the challenges faced in transitioning innovative propulsion concepts from the laboratory to practical application in space exploration missions. Further research and development efforts are essential to overcome these obstacles and unlock the full potential of plasma separator ion thrusters for future spacecraft propulsion systems.

4.1.1.7. Radioisotopic Ion Thrusters

Contrary to its nomenclature, thrust generation in this concept revolves around colloidal particles carrying an electric charge. During beta decay, radioisotopes emit beta particles, transforming into different elements and resulting in a net positive

charge on their surface. These charged particles expelled from the decaying radioisotope fuel contribute to creating an electric field that accelerates them, generating the necessary thrust to propel the spacecraft forward. This setup establishes a significant voltage difference, typically ranging from 500,000 to 1,500,000 volts. Electrons are released and collected by a protective barrier, enabling applications requiring high power.

This approach predominantly aims to utilize radioisotopes to institute a sizable voltage and propel a propulsion system, thereby acting as a power supply. Although theoretically sound, no functional prototype was constructed. Security considerations are likely to thwart further progress due to the formidable radiation risks and complexities associated with this concept.

Additionally, the potential benefits of this novel propulsion system include the ability to leverage abundant radioactive materials found in nuclear waste streams, potentially reducing environmental burdens and providing new opportunities for space exploration. Nevertheless, the daunting challenges posed by extreme radiation exposure and the need for rigorous safety protocols remain major roadblocks to realizing this vision.

4.1.1.8. Hall Effect Thrusters

R. G. Jahn's groundbreaking discoveries have illuminated diverse avenues for implementing the Hall Effect. Gridless Hall thrusters, characterized by their unique interaction of charges within intersecting electric and magnetic fields, represent a revolutionary leap forward in propulsion technology. The most popular arrangement comprises a radially symmetric chamber where electromagnets produce a spiraling magnetic field, guiding field lines either towards or away from the coil casing shell. Unlike traditional ion propulsion systems, Hall thrusters demonstrate exceptional efficiency by requiring less voltage to obtain specific impulses, similar to ion thrusters. This efficiency originates from the exclusive electromagnetic dynamics occurring within the system, warranting its classification as an electromagnetic propulsion system rather than merely electrostatic [19].

Global research and development initiatives centered on Hall thrusters span across Europe, Japan, and other preeminent regions in the field. These thrusters are routinely

implemented for crucial space missions involving orbit raising and station-keeping maneuvers owing to their dependability and proficiency.

Two main Hall Thruster categories have been identified:

- Stationary Plasma Thruster (SPT)
- Thruster with Anode Layer (TAL)

Each variation presents unique advantages in terms of specific impulse, efficiency, and mass reduction, making them increasingly appealing for spacecraft propulsion systems.

As advancements in Hall thruster technology continue to flourish, the prospect of revolutionizing spacecraft propulsion capabilities intensifies dramatically. Ongoing research and development efforts hold promise for improving mission success rates, decreasing operational expenses, and expanding new boundaries in space exploration. The potential for integrating Hall thrusters into next-generation spacecraft propulsion systems holds immense significance. By optimizing existing designs and exploring new possibilities, engineers and scientists worldwide strive to develop highly efficient, reliable, and cost-effective solutions for space travel and exploration.

4.1.1.8.1. Stationary Plasma Thrusters (SPT)

Stationary plasma thrusters, commonly referred to as 'closed-drift' thrusters, establish a precise electric field using direct current. In the context of SPTs, the positioning of the anode and cathode plays a significant role in the propulsion mechanism, facilitating efficient ion acceleration and thrust generation. The anode, acting as the positively charged electrode, attracts and propels ions towards the exhaust section. Conversely, the cathode, strategically positioned externally, serves as the negatively charged electrode. By emitting electrons (which undergo prolonged circulation due to $E \times B$ drift) into the chamber under the influence of the electric field, the cathode aids in ionizing the propellant and maintaining charge balance within the system. This process of ionization, acceleration, and charge neutrality is essential for the effective operation of the thruster.

FAKEL SPTs manufactured in Russia have successfully completed more than a few ten-thousands of hours of space operations, showcasing their robustness and reliability in actual space missions. Additionally, Safran's PPS®1350 and PPS®5000 thrusters

have demonstrated significant flight heritage and performance in various space applications. Safran thrusters are the European version of the Fakel thrusters and compatible with all European standards like harnesses, connectors, ECSS, etc. The PPS®1350 thruster has been pivotal in station-keeping operations and was notably utilized as the primary engine on ESA's lunar probe, Smart-1. On the other hand, Safran's PPS®5000 plasma thruster has been selected for integration into new Galileo satellites, highlighting its advanced capabilities and trustworthiness for cutting-edge space missions. These examples underscore the effectiveness and reliability of stationary plasma thrusters like FAKEL SPTs and Safran's PPS®1350 and PPS®5000 in supporting critical space endeavors with their proven flight heritage and operational excellence. Both are the most well-known manufacturers, dominating the market for SPT.

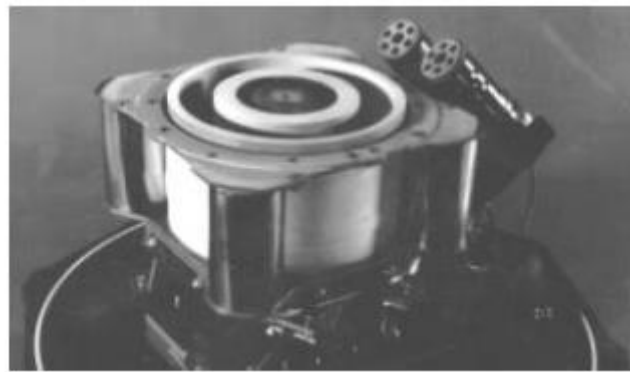


Figure 4.4: SPT-100 Russian Stationary Plasma Thruster.

The future of SPTs holds promise for further advancements in propulsion efficiency, thrust capabilities, and mission versatility. For example, a variety of SPT options are available in the global marketplace, catering to diverse mission requirements for small satellites and CubeSats. There is no doubt that continuing advancements in research and development are expanding the horizons of electric propulsion technology, positioning SPTs to be pivotal in defining the trajectory of space exploration in the years to come.

4.1.1.8.2. Thrusters with Anode Layer (TAL)

Thrusters with anode layer (TAL) engines present a unique design where ion generation occurs more externally compared to traditional setups like SPT units, with

the anode strategically positioned downstream. TAL thrusters are typically characterized by their compact size and lower erosion rates, making them efficient choices for various space missions.

One notable advantage of TAL thrusters is their superior thrust-to-power ratio, often boasting twice the efficiency of other common systems. Despite their high performance, TAL thrusters operate at relatively lower voltages, typically in the range of 300–400 volts, showcasing their efficiency and versatility in propulsion systems.

For instance, the utilization of TAL thrusters by the USSR in space missions has been extensive, with these engines deployed on a range of satellites, including Meteor weather satellites, military observation satellites, and Express communication satellites. The successful integration of TAL thrusters in diverse applications highlights their reliability and effectiveness in supporting critical space endeavors. Moreover, Russian Hall thrusters, including TAL variants, have been employed in collaborative U.S. military missions since October 1998. This cross-border utilization underscores international cooperation and the exchange of advanced propulsion technologies for space exploration and satellite operations. Additionally, the T5 Hall Effect Thruster developed by Aerojet Rocketdyne is another prominent example of a TAL thruster with significant heritage status. The T5 thruster has been utilized in numerous space missions, showcasing its reliability and effectiveness in providing propulsion for satellites and spacecraft. Its proven track record in space operations underscores its importance in this market.

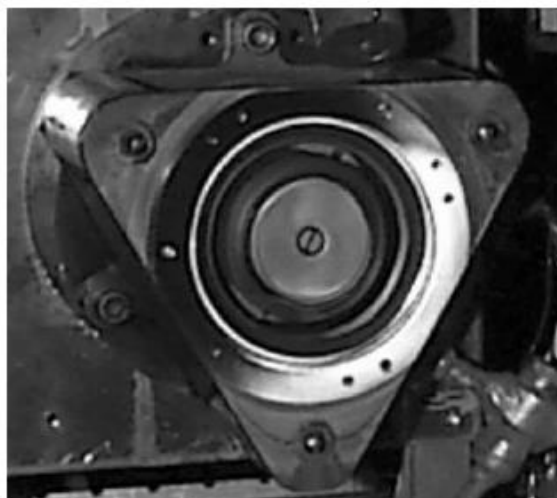


Figure 4.5: D-55 TAL Thruster with Anode Layer.

The continuous development and utilization of TAL thrusters exemplify the ongoing advancements in electric propulsion systems, paving the way for enhanced performance and efficiency in future space missions.

4.1.2. Electromagnetic Propulsion

Electromagnetic propulsion-based systems offer a notable benefit over their electrostatic counterparts in their ability to confine and direct gases with significantly higher temperatures and viscosities. This capability results in increased exhaust velocity and more efficient fuel mass utilization. In electromagnetic propulsion systems, propulsion involves accelerating a propellant through a nozzle by leveraging interactions with electric and magnetic fields within the chamber.

By effectively managing gases with elevated temperatures and viscosities, electromagnetic thrusters can achieve greater exhaust velocities, enhancing thrust performance and fuel efficiency. The acceleration of propellant through interactions with electromagnetic fields plays a vital role in generating thrust and propelling spacecraft in a controlled manner. The precise control over propellant flow enabled by electromagnetic fields contributes to optimized propulsion performance and improved mission capabilities.

Electromagnetic propulsion systems offer significant benefits in terms of fuel efficiency, thrust performance, and control over propellant flow compared to electrostatic systems. The ability to accelerate gases with advanced properties through electromagnetic interactions highlights the effectiveness and potential of electromagnetic propulsion technologies in advancing space exploration and satellite operations.

However, when comparing electromagnetic thrusters to electrostatic and electrothermal propulsion systems, electromagnetic thrusters present challenges due to their intricate designs, resulting in complexity and potentially higher energy consumption. Effective thermal load management is vital to prevent overheating, while limitations in propellant compatibility may constrain options. Despite their potential, electromagnetic thrusters may lack the established flight heritage and reliability of electrostatic and electrothermal systems. Therefore, selecting propulsion

systems for spacecraft missions requires careful consideration of trade-offs to optimize performance and efficiency.

4.1.2.1. Magnetoplasmadynamic (MPD) Arcjets

The evolution of MPD arcjets arises from advancements in arcjets and magnetogasdynamics, leading to the development of a propulsion system known as a Lorentz force engine or accelerator. These thrusters typically feature a radially symmetric discharge area housing a cathode spanning the plasma region and carrying substantial currents, often in the kiloampere range. By creating a potential difference between electrodes, incoming neutral gas is ionized, initiating propulsion through a combination of phenomena.

The first phenomenon is joule heating, also known as resistive or ohmic heating, where electric energy is converted into heat as current flows through a conductor—a principle governed by Joule's first law. The second phenomenon is electrodynamic forces, which involve interactions between electric charges at rest, in uniform motion, or under acceleration. The interaction between the current-carrying plasma and magnetic fields induces Lorentz acceleration, the primary mechanism for propelling the thruster.

The necessary magnetic field can be generated in two primary ways:

- Self-Field
- Applied Field

Two methods are employed for generating the magnetic field: self-field (SF) and applied field (AF). In both cases, the cathode-anode system is powered by a DC power source. The cathode, operating at high temperatures, emits numerous electrons through thermionic emission. The current flowing through the cathode generates a $\mathbf{J} \times \mathbf{B}$ force, which results in an azimuthal magnetic field. Figure 4.6 illustrates the configuration of a self-field (SF) MPD [20].

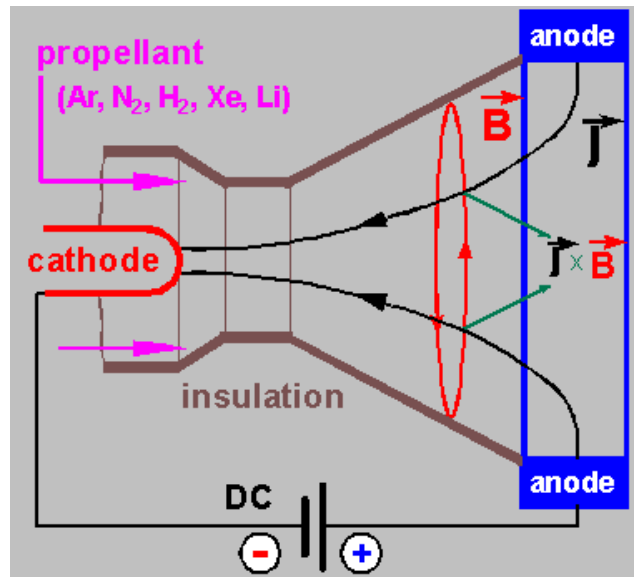


Figure 4.6: Self Field MPD Schematic [20].

In self-field systems, the magnetic field responsible for magnetizing the plasma is generated by the high current flowing from the cathode. In contrast, externally applied field systems involve the application of an external magnetic field by a surrounding coil, which enhances the confinement and acceleration of the plasma. Externally applied fields are preferred for high-temperature and high-energy plasmas because they enable the generation of high thrust with low plasma flow rates.

In both configurations, a DC power source energizes the cathode-anode assembly. The cathode, operating at high temperatures, primarily emits current through thermionic emission, which generates an azimuthal magnetic field along the cathode bar to initiate the $\mathbf{J} \times \mathbf{B}$ force on the plasma. Acceleration in the self-field configuration is facilitated by the magnetic field produced due to the high current flow along the cathode.

In contrast, an applied field configuration utilizes an external solenoidal magnetic field to enhance acceleration and plasma confinement, which is particularly advantageous for high-temperature and high-energy applications. Incorporating an applied field reduces the required plasma current to achieve equivalent propulsive power.

In high-power regimes, steady-state magnetoplasma dynamics thrusters (MPD) are employed, capable of operating at power levels ranging from several hundred kilowatts to megawatts. MPD thrusters utilize the Lorentz force acceleration principle, where the electric current flowing in an electric and magnetic field is perpendicular, and the resulting Lorentz force accelerates the propellant. Therefore, MPDs are also referred

to as Lorentz force accelerators. MPDs feature a symmetrical axis structure where a cathode is utilized to deliver large currents (typically in the kiloampere range) to the ionization region. The current-carrying flow interacts with the magnetic field, resulting in Lorentz forces that ultimately ionize the propellant. The Lorentz force plays a dominant role in the acceleration mechanism.

MPD thrusters exhibit versatility in operating modes, capable of functioning in steady-state or pulsed configurations with pulse durations as brief as a millisecond. The design flexibility of magnetoplasma dynamic thrusters allows for different operational modes:

- Steady State Operation
- Pulsed Operation

In pulsed operation scenarios, a capacitive system discharges to generate advanced currents and magnetic fields at specified power levels. However, pulsed arcjets are often prone to increased arc corrosion. Challenges persist in low-power operations where efficiency closely correlates with power input levels, highlighting cathode longevity as a critical concern, particularly in systems utilizing high-speed tubes.

Steady-state units have not yet been widely deployed due to their higher power requirements. MPD thrusters, however, offer significant capabilities, notably high specific impulse (I_{sp}) and thrust levels. The operational functionality of quasi-steady MPD thrusters has been extensively tested, particularly in Japan. These thrusters operate on principles similar to pulsed plasma thrusters but maintain a steady distribution of current and plasma flow over longer durations, typically around 200 μ s. They can operate either in steady-state mode for several hours or in a quasi-steady mode where the discharge is pulsed for 1 to 10 ms, depending on power requirements. Notably, only quasi-steady MPD thrusters have been flown, demonstrating efficiencies as high as 22% at a specific impulse of 2500 seconds.

In the early days of MPDs, lithium was used as fuel; however, due to environmental pollution concerns, this practice was discontinued in the Western world. Russian engineers continue to experiment with 500 kW lithium MPD thrusters, while Western countries are researching alternative fuels with minimal environmental impact [20].

Moreover, MPD thrusters are considered promising for interplanetary manned flights within the solar system, provided sufficient power is available.

4.1.2.2. Pulsed Plasma Thrusters (PPT)

Pulsed plasma thrusters operate by generating plasma through an arc discharge facilitated by a capacitor connecting two electrodes. The Lorentz force within the induced magnetic field further accelerates the ions in the plasma. These thrusters have a rich history, originating in the mid-1960s and experiencing a resurgence in popularity during the 1990s.

Pulsed plasma thrusters (PPT), belonging to the electromagnetic thrust class, operate at sufficiently low power levels. Typically, they function around 30 W and are primarily used for attitude correction in small spacecraft. In this system, a Teflon mold is compressed into a tight space between two electrodes using a piston mechanism. The piston continuously supplies new fuel as it is consumed. The Power Processing Unit (PPU) draws unregulated power from the spacecraft and charges the capacitor to voltages ranging from 500 to 2000 V. The PPU also provides a high voltage to the spark plug to trigger the discharge. When the discharge initiates, the energy stored in the capacitor (5–50 J) is released as a high-current, short-duration plasma discharge (30 kA, 5–10 μ s). This discharge ablates a small amount of material from the Teflon mold surface and ionizes it. The resulting ions are then accelerated by the Lorentz force to achieve high ejection velocities [21]. Figure 3.13 illustrates the fundamental components of a pulsed plasma thruster.

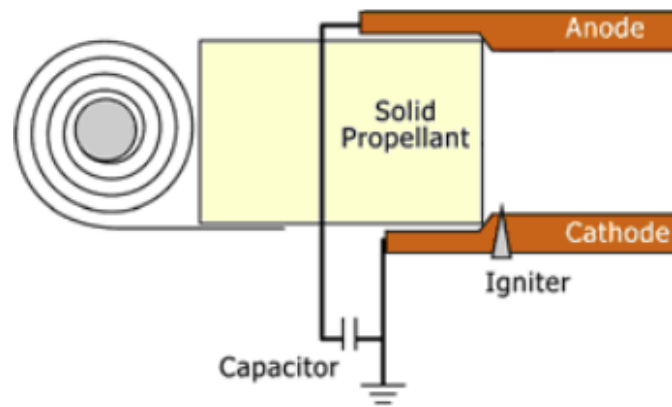


Figure 4.7: Pulsed Plasma Thruster Main Elements.

A notable example of advanced propulsion technology is the Teflon pulsed plasma thruster (PPT), which operates by ablating a solid block of Teflon propellant. These

thrusters employ a straightforward design for propellant delivery, utilizing a spring mechanism.

For instance, the Earth Observing-1 (EO-1) mission was pioneering in testing cutting-edge satellite technologies. EO-1 utilized Teflon PPTs manufactured by Primex Aerospace Company for precise attitude control and orbit maintenance, effectively showcasing the capabilities of PPT technology.

Similarly, the Deep Space-3 (DS-3) project, focused on an experimental space interferometer, utilized PPT thrusters from the BPT series. Developed by Aerojet (later part of Primex Aerospace Company) in collaboration with Busek Co., these thrusters enabled precise alignment and control, underscoring the versatility and reliability of PPT technology in demanding space missions.

Throughout their operational lifetime, these thrusters demonstrated the ability to execute all required mission maneuvers, solidifying their ongoing importance across various space missions [4].

Another promising development involves xenon or water, which can be used as fuel in PPT systems. Although this system has not yet been deployed in flight missions, it has demonstrated impressive durability, achieving over a few million cycles with a power input of 40 W. Particularly noteworthy is its ability to deliver 0.02-second pulses when using water.

4.1.2.3. Helicon Plasma Thrusters

Helicon plasma thrusters utilize the Lorentz force to propel plasma, akin to pulsed plasma thrusters but with a unique feature. They employ a helicon tube to generate a dynamic electromagnetic wave that travels through the plasma chamber core, preserving intense magnetic fields crucial for efficient propulsion. This innovative approach enhances plasma acceleration and regulation, thereby improving overall thruster performance and energy efficiency.

The utilization of a helicon tube in helicon plasma thrusters represents a cutting-edge advancement in electric propulsion technology. This method enables precise control over plasma dynamics and magnetic field strengths, leading to improved thrust generation and specific impulse. The ability to sustain high magnetic field strengths through the helicon tube's electromagnetic wave propagation is a key factor in

optimizing the efficiency and effectiveness of these thrusters for various space missions. The HIPATIA project, funded by the European Commission's H2020 program, seeks to develop helicon plasma thruster technology and related components for electric propulsion systems in non-geostationary satellite constellations and smaller spacecraft. Led by SENER Aeroespacial, the initiative involves collaborators such as the Universidad Carlos III de Madrid, Airbus Defense and Space, the Centre National de la Recherche Scientifique, and Advanced Space Technologies. Its objective is to validate the functionality and performance of helicon plasma thruster-based electric propulsion systems for a range of space applications [22].

4.1.2.4. Pulsed Inductive Thruster (PIT)

The Pulsed Inductive Thruster (PIT) is a type of propulsion system that utilizes a bank of high-voltage capacitors, typically operating at around 10 kV, to discharge energy into a coil that encircles a circular chamber. During this process, the high-voltage discharge generates a rapid and intense magnetic field.

Subsequently, neutral fuel is introduced into the chamber through a quick-valve system. As the propellant disperses in the chamber, the discharge pulse creates a powerful magnetic field that induces current in the coil, which in turn ionizes the fuel and accelerates the resulting plasma away from the coil.

In the design of the thruster, one end of the chamber is sealed, while the opposite end is open. This configuration allows the ionized fuel plasma to be expelled through the open end, producing thrust. Effective management of the magnetic and electric fields within the chamber, as well as proper shielding, are crucial for the efficient operation and safety of the thruster system.

Research and development efforts in this area have been pursued by organizations such as TRW, focusing on advancing technology and exploring its potential applications. However, despite significant progress in the development of PIT technology, actual spaceflight tests to validate its performance and reliability in the space environment remain to be conducted. These tests are essential for demonstrating the thruster's functionality and effectiveness for future space missions.

4.1.2.5. Variable-Specific Impulse Plasma Thruster

The Variable-Specific Impulse Plasma Thruster (VSIPT), featuring an electrodeless circular chamber design, stands at the forefront of advanced propulsion systems for space exploration. This sophisticated propulsion mechanism harnesses ionization, plasma heating, and controlled exhaust to efficiently generate propulsive thrust.

In the VSIPT, the process begins with the injection of hydrogen into the forward section of the chamber. High-voltage capacitors discharge energy into a coil surrounding the circular chamber, creating a rapidly changing magnetic field. This changing magnetic field induces electric fields that ionize the hydrogen, producing plasma composed of ions and electrons. The plasma is then further heated through several mechanisms as it moves towards the chamber's center, including electron cyclotron resonance heating, ion cyclotron resonance heating, and whistler wave heating. Electron cyclotron resonance heating uses microwave-frequency radio waves to energize electrons through resonance at the electron cyclotron frequency. Ion cyclotron resonance heating involves the resonant absorption of energy by ions at their specific cyclotron frequencies, typically achieved through targeted radiofrequency (RF) waves. Whistler wave heating involves the interaction of low-frequency whistler-mode waves with plasma to transfer energy and increase the plasma's thermal energy.

As the plasma progresses downstream towards the nozzle region, a critical transformation occurs where the thermal energy of the plasma is converted into kinetic energy for propulsion. This conversion is facilitated by precise adjustments in the shape and configuration of the magnetic fields within the chamber. To maintain optimal performance and prevent overheating, neutral hydrogen is strategically introduced near the nozzle to cool the chamber walls while simultaneously disrupting the plasma-magnetic field interaction to control specific impulse levels. For scenarios requiring a higher specific impulse, a high-frequency AC ripple field is introduced to further perturb the magnetic field-plasma relationship, enhancing the thruster's operational efficiency and adaptability across varying conditions.

Another prominent electromagnetic propulsion system is the Variable Specific Impulse Magnetoplasma Rocket (VASIMR). VASIMR utilizes radiofrequency (RF) waves to heat and ionize the propellant, achieving thrust by accelerating the resulting plasma with magnetic fields. The plasma heating methods employed in VASIMR, such

as radio-frequency (RF) heating, were developed due to the high-power demands of nuclear fusion research.

Each component of VASIMR is magnetically shielded to prevent direct contact with the plasma, which enhances the engine's potential longevity compared to traditional ion/plasma engines [7]. VASIMR can operate across a range of thrust and specific impulse levels by adjusting the energy supplied to the RF heater and the amount of fuel provided for plasma production. This capability allows VASIMR to function as both a low-thrust, high-specific-impulse engine and a high-thrust, low-specific-impulse engine [23]. Unlike traditional ion thrusters, VASIMR systems do not suffer from electrode erosion issues, and apart from a few minor components like gas valves, VASIMR engines have no moving parts, contributing to their durability.

However, the significant heat generated within the system poses a challenge for thermal management, and interactions with the magnetosphere can create unwanted torque. Figure 4.8 illustrates the VASIMR thruster.

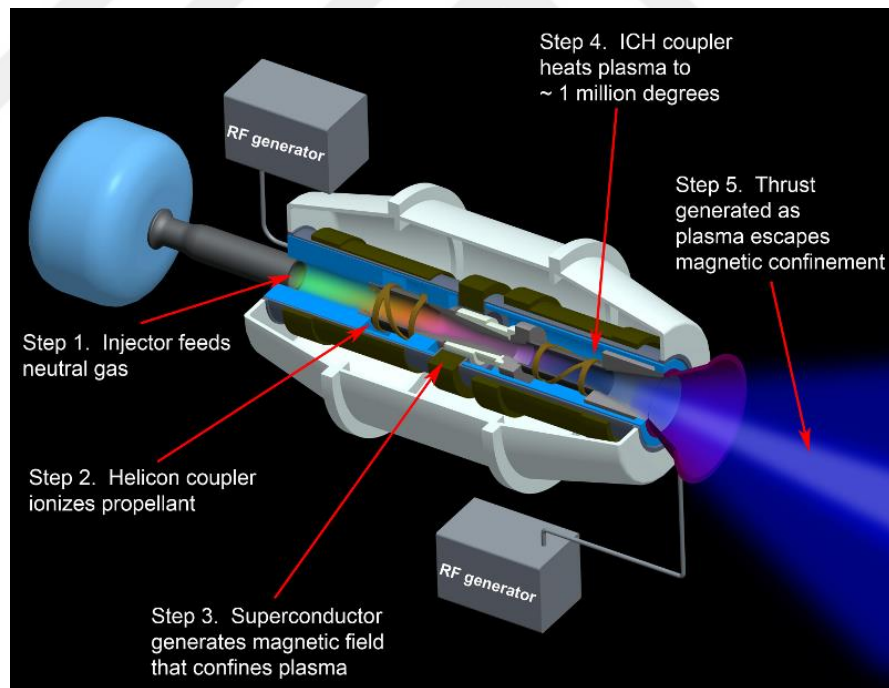


Figure 4.8: VASIMR Thruster Working Principle [23].

Currently undergoing rigorous laboratory testing and development, the VSIPT holds immense promise for future space missions. Institutions like the Johnson Space Center are actively exploring technologies such as VASIMR for potential integration into

manned space missions, reflecting a steadfast commitment to advancing space propulsion technology for enhanced exploration capabilities.

4.1.2.6. Electron-Cyclotron-Resonance (ECR) Thrusters

Electron cyclotron resonance (ECR) is a method used to heat and ionize gases to produce plasma for various applications, including space propulsion. In the ECR method, a microwave source generates electromagnetic waves at a frequency that matches the electron cyclotron frequency in a magnetic field. These waves are transmitted through a waveguide and directed into the plasma chamber.

The electromagnetic waves are in the transverse-electric (TE) mode, which creates a rapidly oscillating electric field. This field drives the electrons in the plasma to move in circular or cycloidal paths around the magnetic field lines. At the resonance frequency, electrons absorb energy from the microwave radiation, which increases their kinetic energy and contributes to the ionization of neutral gas molecules.

The ionized gas forms plasma, where the electrons and ions move under the influence of the magnetic field. The ECR technique leverages the fact that microwaves at the resonance frequency efficiently increase the plasma's density and temperature. In space propulsion systems, this high-energy plasma is then accelerated towards the exhaust port, generating thrust.

One notable example of ECR technology is the ECR ion thruster, which underwent extensive testing and successfully completed an 18,000-hour life test. This engineering model was part of the Japanese MUSES-C (Hayabusa) mission, launched in 2003 with the goal of returning samples from the asteroid Itokawa. This mission demonstrated the effectiveness of ECR-based propulsion systems for space exploration.

4.1.3. Electrothermal Propulsion

Electrothermal propulsion systems represent a class of propulsion technologies that generate thrust through the heating and expansion of a propellant. Unlike electrostatic and electromagnetic propulsion systems, which respectively utilize electric and magnetic fields for acceleration, electrothermal systems rely on the conversion of electrical energy into thermal energy. In these systems, various methods are used to heat the propellant to high temperatures, including resistive heating, arc heating, and microwave heating.

Once heated to high temperatures, the propellant expands rapidly as it exits through a nozzle, creating a high-velocity exhaust stream that generates thrust according to Newton's third law of motion. The efficiency of electrothermal propulsion systems is determined by factors such as the specific impulse of the propellant, the temperature achieved, and the design and efficiency of the nozzle. Specific impulse is a measure of how effectively the propulsion system converts propellant mass into thrust, with higher values indicating better performance.

Electrothermal propulsion systems are often compared to electrostatic, e.g., ion thrusters, and electromagnetic, e.g., MPD thrusters and VASIMR, propulsion systems. Overall, electrothermal propulsion systems offer a balance between simplicity and performance, making them suitable for a range of space missions, including satellite station-keeping and deep-space exploration.

4.1.3.1. Resistojets

Resistojet engines represent a well-established propulsion technology that utilizes primary ohmic heating to elevate the temperature of the propellant for thrust generation. This heating mechanism involves passing electrical current through a resistive heating element, such as a tungsten wire, situated within a specialized chamber. The propellant, which is in liquid form, flows over this heating element and absorbs the thermal energy produced, causing it to expand rapidly. This heated and expanding propellant is then expelled through a high-pressure nozzle to produce thrust according to Newton's third law of motion.

Unlike traditional chemical propulsion systems, resistojets use electrical energy to heat the propellant instead of relying on chemical reactions. This introduces a layer of complexity in the system design but also offers the advantage of greater control over the thrust and efficiency of the propulsion system.

In a typical resistojet system, as shown in Figure 4.9, the propellant passes over a tungsten heating element, which is electrically heated to high temperatures. The heated propellant then exits through a nozzle, creating a high-velocity exhaust stream that generates thrust.

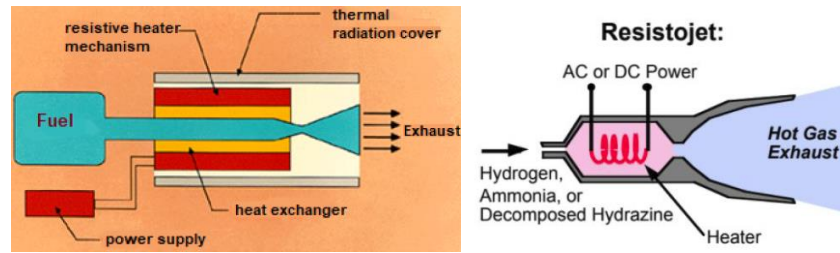


Figure 4.9: Resistojet Thruster Main Elements [1].

Resistojets are generally operated at voltages around 100V and require high current levels to achieve the necessary heating. Their design emphasizes compactness and lightweight characteristics, which make them particularly suited for space missions where efficiency and performance are crucial. For example, hydrazine resistojets have been effectively used during the orbit-raising phases of missions for satellites in the Iridium constellation, demonstrating their reliability and efficiency in space environments. Historically, resistojets have been employed on spacecraft like Vela, ATS-A, ATS-D, INTELSAT V, and RCA SATCOM G for tasks such as orbit adjustment, attitude control, and station-keeping. These missions highlight the versatility and proven reliability of resistojet propulsion systems.

Recent advancements in resistojet technology include the development of electrothermal microthrusters that use liquid ammonia as a propellant. These innovations showcase the ongoing progress in resistojet technology and its potential for micro-propulsion applications. By balancing power, efficiency, and reliability, resistojets continue to play a significant role in space exploration, with future developments promising even greater advancements in propulsion systems for aerospace applications.

4.1.3.2. Electrothermal Hydrazine

The electrothermally accelerated hydrazine thruster represents a sophisticated propulsion system that combines elements of traditional electrothermal thrusters with those of monopropellant chemical hydrazine thrusters. This advanced propulsion technology begins with the decomposition of hydrazine in a chemical reaction facilitated by a catalyst, which breaks hydrazine down into nitrogen gas and hydrogen gas.

In the secondary stage, the decomposed propellant gases pass through a heating chamber, where ohmic heating is employed to elevate their temperature to optimal levels. Ohmic heating involves passing an electrical current through a resistor, in this case, a tungsten filament, to convert electrical energy into thermal energy. The intricacies of this process require precise control of the heating duration and temperature to regulate and optimize the specific impulse of the thruster. Thermal energy increases the propellant's velocity as it exits through a nozzle, thereby generating thrust. This method of heating the propellant after chemical decomposition allows the thruster to achieve higher specific impulses compared to conventional monopropellant hydrazine thrusters.

One of the significant advantages of this electrothermal approach is its potential to increase the specific impulse of the thruster by up to 30% compared to standard hydrazine propulsion systems. This improvement is due to the additional thermal energy provided to the propellant, which enhances the efficiency of the thrust generation process. However, these advanced thrusters may exhibit lower thrust levels compared to traditional hydrazine thrusters, which is a trade-off for the increased specific impulse. Their increasing adoption as preferred replacements for station-keeping thrusters in space missions attests to their efficiency, reliability, and advanced performance characteristics.

The electrothermally accelerated hydrazine thruster has been utilized in various space missions for station-keeping and orbit maintenance tasks. A notable example of this technology in action is the deployment of electrothermal hydrazine thrusters on the Intelsat V satellite, where the advanced performance characteristics of these thrusters contributed to the satellite's operational success.

4.1.3.3. Arcjets

Arcjets represent a diverse category of propulsion systems that have undergone extensive development, testing, and fabrication. These electrothermal propulsion systems operate by generating thrust through the thermodynamic expansion of a hot gas or plasma expelled through a nozzle. Various methods are employed to heat the propellant, including DC, AC, RF, and laser-thermal heating. The design and operation of arcjets can involve both direct and interspersed current approaches, with direct

current setups being relatively straightforward but susceptible to issues such as corrosion.

In arcjets, the propellant is heated by passing through an electric arc before expanding through the nozzle. This heating can be achieved using either AC or DC power sources. However, AC-based arcjets have complex power conditioning systems and a shorter lifespan compared to DC-based arcjets, which is why they have not been developed or tested in space since the 1960s [20]. Figure 4.10 shows the schematic diagram of an arcjet.

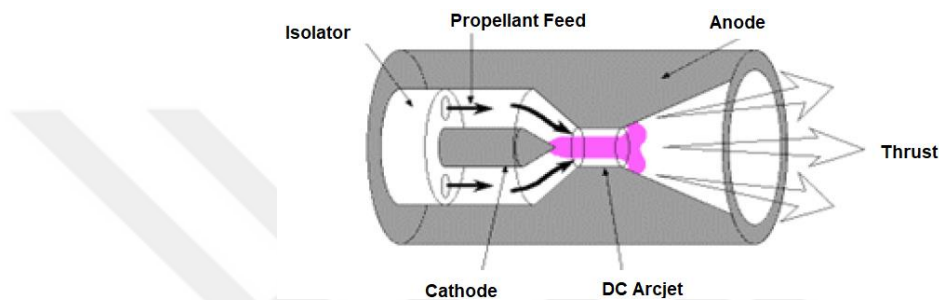


Figure 4.10: Arcjet Thruster Schematic.

Pulsed arc jets introduce a variation to traditional DC arcjet operation by utilizing pulsed current from charge and discharge cycles to generate rapid discharge pulses at specific breakdown voltages within the propellant stream. However, pulsed systems typically exhibit a lower specific impulse compared to their DC counterparts, which limits their practical applications.

Noteworthy applications of arcjet technology include the deployment of a 30 kW-class ammonia arcjet thruster on the ARGOS mission for orbit-raising operations in 1999. Additionally, lower-power ammonia arcjets find utility on AMSAT for tasks related to orbit-raising and inclination control, while hydrazine arcjets are utilized on Lockheed Martin geosynchronous satellites for station-keeping maneuvers.

Arcjets are simpler compared to other systems but are susceptible to erosion on both the anode and cathode. The propellants used are typically a mixture of hydrazine (N_2H_4) and ammonia (NH_3), chosen for their applications in chemical propulsion systems and minimal modifications to existing systems. Operational strategies often focus on minimizing on-off cycles to reduce component degradation. DC arcjets function by establishing a high current between the cathode and the anode nozzle. The

thrust output of DC arcjets is restricted by available power, while the specific impulse is influenced by the materials used in the construction of the nozzle.

4.1.3.4. Microwave Electrothermal Thruster (MET)

Microwave Electrothermal Thrusters (METs) operate by introducing microwave radiation from a waveguide into a plenum chamber through a dielectric window. Inside the chamber, the microwave radiation interacts with neutral propellant, accelerating electrons in high-field zones. This interaction causes collisions within the propellant, leading to its heating and resulting in exhaust generation. METs are versatile, capable of operating in both steady-state and pulsed modes.

An important advantage of METs is their electrodeless design, which minimizes erosion concerns typically associated with thruster systems. This design feature contributes to METs potentially having significantly longer operational lifespans compared to arc-type electrothermal systems. Experimental studies in laboratory settings have shown the feasibility of achieving specific impulses as high as 1300 seconds when using helium propellant. However, specific impulse values may vary with heavier particles.

Despite promising laboratory results, MET technology is still in the developmental phase, with no units having flown on space missions yet. Ongoing research and testing in laboratories aim to further refine MET design and performance for potential applications in space propulsion systems.

For more detailed information on microwave electrothermal thrusters, refer to studies by M. M. Micci, S. G. Bilén, and D. E. Clemens in EUCASS Proceedings and EUCASS, as well as resources from URA thrusters, Princeton University, and the Online Library. The unique characteristics and capabilities of METs position them as a promising technology with the potential to enhance propulsion efficiency and longevity in space exploration.

4.1.3.5. Pulsed Electrothermal Thrusters

Pulsed Electrothermal Thrusters (PETs) represent a sophisticated propulsion technology that operates by generating controlled bursts of plasma expelled from a specialized fuel chamber integrated as the anode within the system. The process of plasma formation within the fuel chamber is orchestrated through capacitor discharge

to an internal electrode, precisely heating the gas confined within the chamber. When molecular hydrogen serves as the propellant, PET systems are anticipated to achieve a theoretical specific impulse of approximately 2900 seconds, with specific impulse values for alternative propellants inversely correlated to their molar masses.

The operational longevity and performance efficiency of PET systems are closely tied to the erosion dynamics of electrode materials, which significantly affect the thruster's durability and reliability. Managing the gradual erosion of electrode materials is a critical challenge, requiring meticulous material selection strategies and advanced design considerations. These efforts aim to effectively mitigate erosive effects and ensure sustained performance over extended mission durations.

The intricate operational mechanisms and high-performance characteristics inherent in PET systems position them as cutting-edge propulsion technology with substantial potential to enhance propulsion efficiency, specific impulse capabilities, and operational endurance in space missions. PETs offer a promising avenue for advancing propulsion systems in future space exploration endeavors.

4.2. Comparison of Electric Propulsion Types

The list of electrical thrusters based on various technologies used in electric propulsion systems is presented in Table 4.1.

Table 4.1: Electric Propulsion Systems Overview.

ELECTROSTATIC	ELECTROMAGNETIC	ELECTROTHERMAL
-Ion bombardment -RF or Microwave -Colloid -Contact ion -Field emission -Plasma separator -Radioisotopic ion -Hall Effect: SPT / TAL	-Magnetoplasmadynamic -Pulsed plasma -Helicon plasma -Pulsed inductive plasma -Variable specific impulse -Electron-Cyclotron-Resonance	-Resistojet -Electrothermal hydrazine -Arcjet -Microwave -Pulsed

The utilization of some of the technologies mentioned above has been verified in space for propulsion systems developed based on limited technologies at the research level. Their applications primarily focus on orbit and position correction in satellites for stationkeeping and attitude control purposes. Table 4.2 presents the characteristics encompassing the performance parameters of electric satellite propulsion motor types verified in space.

Table 4.2: Performance Parameters of Electric Propulsion Systems [5, 6].

Type	Thrust Range (mN)	I_{sp} (s)	Thruster efficiency (%) **	Specific mass (kg/kW)	Propellant	Thrust-to-Power Ratio (mN/kW)**
Resistojet	200-300	200 - 350	65-90	1-1.6	NH ₃ , N ₂ H ₄ , H ₂	740-910
Arcjet	200-1000	400 - 1000	25-45	2.5-3.5	H ₂ , N ₂ , N ₂ H ₄ , NH ₃	110-135
Ion motor	0.01-200	1500-5000	40-80	9.1-23.7	Xe, Ar, Kr	25-41
Pulsed Plasma	0.05-10	600-2000	7-13	85-195	Teflon	16-21
Magneto Plasma Dynamic	0.001-2000	2000-5000	30-50	-	Ar, Xe, H ₂ , Li	-
Hall Effect	0.01-2000	1500-3000	35-60	6-7	Ar, Xe	55
Mono-propellant chemical ***	30-100,000	200-250	87-97	-	N ₂ H ₄	-

* Efficiency = thrust output power / electrical input power.

** This column is taken from reference [6].

*** Despite being a chemical propulsion system, it has been included in the table for comparative purposes.

Based on the recent academic research and patent studies, particularly in HET systems, advancements are anticipated in terms of performance and lifespan, despite the parameter ranges being standardized in current systems, as seen in Table 4.2.

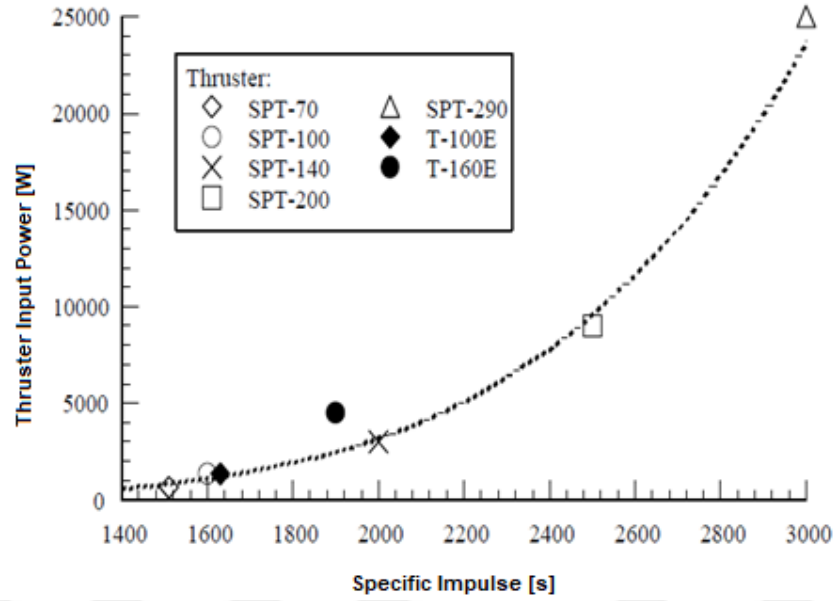


Figure 4.11: Effect of Input Power on Specific Impulse for Hall Effect Stationary Plasma Thrusters (SPTs) [25, 26].

The relationship between specific impulse (I_{sp}) and input power in Hall Effect Stationary Plasma Thrusters (SPTs) is presented in Figure 4.11 [15, 25, 26, 27, 28]. The rate of increase in the I_{sp} values with input power may decrease as input power is increased, but the trend in the graph suggests that I_{sp} increases with input power. This relationship is significant because it indicates that higher input power levels can lead to increased efficiency in terms of fuel consumption in space propulsion systems. However, the efficiency of these systems is also dependent on other factors, such as the mass utilization efficiency, which is the ratio of the ion mass flow rate to the input propellant mass flow rate. The mass utilization efficiency affects the thrust produced by the thrusters and the specific impulse, which is a measure of the efficiency of a thruster in terms of fuel consumption. High I_{sp} values are desirable for electric propulsion systems because they allow for the achievement of large spacecraft final velocities and large total impulses, provided that the thruster is fired for a sufficient

duration. However, the lifetime of the propulsion system, which is governed by the erosion and deterioration of components such as the plasma chamber, electrodes, and neutralizer, is also an essential parameter in the design and operation of electric propulsion systems. The technology must be highly reliable with a very low failure probability over a long-time span to ensure successful mission completion.

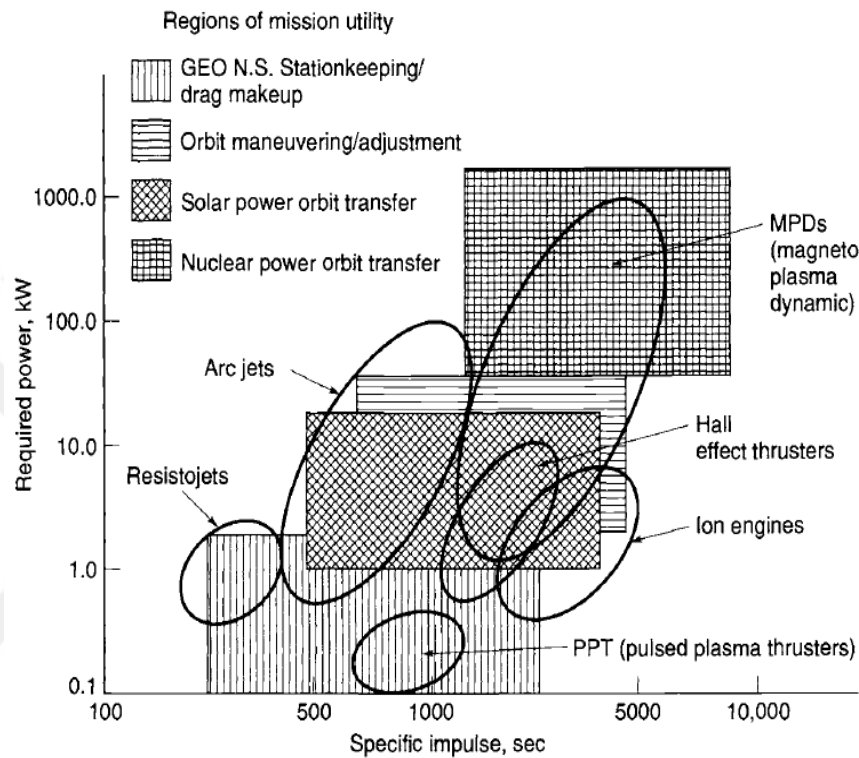


Figure 4.12: Power vs. Specific Impulse Ranges of EPS [2].

The application regions of required power and specific impulse for various electric propulsion systems are illustrated in Figure 4.12, along with operational ranges relevant to multiple missions. In the figure, Hall Effect Thrusters (HETs) typically operate at power requirements spanning from approximately 100 W to 25 kW, accompanied by specific impulse (I_{sp}) values ranging between 1500 and 3000 s. Recent advancements suggest ongoing development towards systems capable of handling up to 50 kW, with no definitive upper limit technologically imposed on Hall-effect thruster design. Consequently, Figure 4.12 suggests an expected increase in specific impulse values with higher power inputs in the future. Based on these capabilities, HETs offer broader utilization potential in geostationary Earth orbit (GEO)

applications such as station keeping, orbit correction and maneuvers, and in interplanetary missions, compared to alternative propulsion technologies.

The benefits and disadvantages of various electric propulsion systems are presented in greater detail in Table 4.3. This table provides a comprehensive overview of the features and performance of different electric propulsion technologies.

Table 4.3: Comparison of EPS Types [2].

Thruster Type	Advantages	Disadvantages	Comments
Resistojet (Electrothermal)	Simple design; Easy control; Simple power conditioning; Low cost; High thrust and efficiency; Compatible with many fuels including hydrazine	Low specific impulse; Heat loss; Gas dissociation; Gas indirect heating; Erosion; Use of toxic fuel	Operational
Arcjet (Electrothermal and Electromagnetic)	Direct gas heating; Low voltage; Simple design; High thrust; Catalyst compatibility; Use of byproduct fuels	Low efficiency; Erosion at high powers; Low specific impulse; High current; Heavy wiring; Heat loss; Complex power conditioning	Operational . Suitable for high-thrust units of 100kW or more.
Ion Thruster (Electrostatic)	High specific impulse; High efficiency; Use of non-toxic fuel (xenon)	Spacecraft weight limit; Complex power conditioning; High voltages; Only pure fuels; Low thrust per unit area; Heavy power supply; Low ion current and power density compared to HET	Suitable for small spacecraft
Pulsed Plasma Thrusters (Electromagnetic)	Simple design; Low power; Solid fuel; No gas or liquid fuel system; No zero-g effect on fuel	Low thrust; Teflon reaction products; Erosion; Inefficiency	Operational
Magnetoplasma Dynamics (MPD) (Electromagnetic)	Simple design; High specific impulse per unit area; High thrust per unit area	Difficulty in analytical simulation; High specific power; Heavy power supply	Operational
Hall Effect Thrusters (Electrostatic)	Desired specific impulse range; Compact; Simple power conditioning; Use of non-toxic fuel (xenon)	Use of only pure fuels; High plasma plume divergence; Erosion	Operational

The choice of electric propulsion systems is heavily influenced by the specific mission requirements of the spacecraft. As depicted in Figure 4.12 and detailed in Table 4.3, leading organizations in the US and Europe strongly favor Hall Effect thrusters (HETs) due to their technological superiority and design advantages. These include the capability for uninterrupted long-term operation, higher specific mass (kg/kW), thrust-to-power ratio (mN/kW), and superior specific impulse (I_{sp}), which results in reduced propellant consumption.

HETs are particularly preferred for large GEO satellites and medium- to high-delta-V missions, and they are increasingly being adopted for low Earth orbit (LEO) applications [29, 30]. Their high specific impulse and demonstrated reliability make them well-suited for space exploration. These thrusters deliver efficient performance across a range of power levels, from 100 watts to several kilowatts. Therefore, Hall Effect thrusters represent a suitable choice for missions where mission duration is a critical consideration.

5. AN INSIGHT INTO THE STUDIED SATELLITE PLATFORM AND ELECTRIC PROPULSION SYSTEM: STUDY INTRODUCTION

In recent years, studies and feasibility reports conducted worldwide indicate that the high cost of communication satellites (hereafter, Comsat) is greatly influenced by their higher volume and masses. Therefore, the industry has pursued an innovative approach here and sought more compact satellite product families. One of the innovative satellite concepts based on economic benefit is the emergence of Comsat with relatively small volume configurations. These satellites, known as "Small Concept Geostationary Communication Satellites, Small-GEO," stand out for their low production and launch costs compared to conventional Comsat and flexible payload solutions. Companies in the satellite industry believe that by combining future technological trends with cost advantages through the new small-concept satellite, significant commercial benefits can be obtained.

The favored satellite in this research is a small conceptual GEO satellite, designed as a telecommunications platform capable of supporting a diverse array of commercial payloads and missions. These missions encompass TV broadcasting, multimedia applications, internet access, and both mobile and fixed services in a wide range of frequency bands. Table 5.1 outlines the general specifications of small-concept GEO satellites. The specific example in this study is drawn from the GSATCOM Flexible High Throughput Satellites Family [31, 32].

Table 5.1: Small Concept GEO Satellite Specifications Summary.

Specification	Value
Orbit	Geostationary Orbit (GEO)
Launch Mass	~ 3000 kg
Mission Life Time	> 15 years
Propulsion Solution	Electric Propulsion System (Orbit Raising + Station Keeping)
Payload Frequency Band	Ka, Ku, X, C, S, L, UHF, and more

Of course, the mentioned satellite platform solution has its advantages and disadvantages. In this section, information regarding this will be provided. Below are some highlighted points provided for the sake of a summarized explanation.

- **Cost effective Comsat:** small concept Geostationary satellites usually do not use the chemical propulsion system. This allows the reduction of the satellite mass. Thus, the ratio of beneficial payload mass to the total mass can be improved.
- **Fast Product Line:** Small Concept Comsats can meet the necessary infrastructure for the communication requirements of countries with faster and more cost-effective solutions than traditional Comsats.
- **Protection of the orbit rights of the countries with cost-effective methods:** Small concept Geostationary satellites are an economic solution through which countries can protect their international orbit rights, which are critical for their communication infrastructure.

Table 5.2: Small Concept GEO Satellite Advantages and Disadvantages.

Advantages	Disadvantages
Low Launch Cost (Total Cost)	High-Cost Electrical Propulsion Thrusters and Propellant
More Precise Orbit Operations	Longer Orbit Raising (Orbit Transfer) Time
High Service Life	Relatively Narrow Scope of Space Heritage

5.1. Definition of EPS Main Parameters

EPS is a type of propulsion system used in spacecraft and artificial satellites. Unlike chemical propulsion systems, which rely on burning fuel to create thrust, EPS utilizes electrical or magnetic effects for propulsion. In chemical propulsion systems, the energy available is limited by the chemical bonds of the fuel. In contrast, EPS can generate thrust using electrical power supplied to the propellant, with the energy availability constrained primarily by the amount of electrical power and the operational characteristics of the propulsion system [6].

The use of electric energy in EPS offers advantages in terms of efficiency. Efficiency is defined as the ratio of the thrust power generated by the thruster to the electrical power consumed by it for operation, as expressed in Equation 5.1.

$$\eta = \frac{P_{ion\ jet}}{P_{electrical}} = \frac{\left(\frac{1}{2}\right) T^2 / \dot{m}_a}{V_D I_D} = \frac{\left(\frac{1}{2}\right) T I_{sp} g}{V_D I_D} . \quad (5.1)$$

In Equation 5.1, the efficiency of the thruster η is expressed in terms of generated thrust T , anode gas flow rate $\dot{m}_a$, discharge voltage V_D , discharge current I_D , specific impulse I_{sp} , and gravitational acceleration g . HETs, PPTs, arcjets, ion thrusters, and resistojets have thrust efficiency ranges of 35–60%, 7–13%, 25–45%, 40–80%, and 65–90%, respectively.

In propulsion systems, the thruster's specific impulse (I_{sp}), which increases with the exhaust gas exit velocity, is an important parameter calculated by Equation 5.2 [33].

$$I_{sp} = \frac{T}{\dot{m}_p g} . \quad (5.2)$$

The specific impulse is the thrust generated per unit weight flow rate of the propellant consumed. Here, T is thrust, $\dot{m}_p$ is propellant mass flow rate, and g is the gravitational acceleration.

Specific impulse fundamentally represents a rocket's fuel efficiency. Because thrust force T is the product of the propellant mass flow rate $\dot{m}_p$ and its exit velocity v_{ex} , specific impulse can also be expressed as

$$I_{sp} = \frac{v_{ex}}{g} . \quad (5.3)$$

Given that the gravitational acceleration is constant, the exhaust gas velocity of a rocket v_{ex} can also be considered as I_{sp} . As can be seen in Equation 5.3, specific impulse increases with exhaust gas velocity.

5.2. Working Principles of HET

Hall thrusters are electric propulsion systems that operate by accelerating positively charged ions. These ions are produced through collisions between electrons emitted from the cathode and neutral gas in the discharge channel. Acceleration occurs in the

axial direction of the electric field between the anode and cathode, resulting in thrust. Detailed explanations are provided in Section 4.1.1.8.

Hall thrusters consist of three main subunits:

- a) **Propellant Storage and Supply Unit:** This unit is responsible for transferring fuel from the tank to the anode and cathode at the required pressure and flow rate. It includes equipment such as high- and low-pressure solenoid valves, an orifice, and pressure and temperature sensors. Depending on the design, a pressure regulator may be included to reduce storage pressure (typically 180 bar) to the operational pressure suitable for Hall Effect Thruster (HET) operation (typically around 2 bar).
- b) **Power Processing Unit (PPU):** The PPU transfers power from the satellite's power bus to the electric propulsion system (EPS) at the necessary voltage and current levels for its subcomponents. It accommodates the varied power requirements of different components within the thruster. For instance, low-power equipment like the power supply unit (PSU) and high-voltage components such as the anode (>300 volts) are managed by the PPU. In this study, the PPU is designed to handle operational voltages between 300V and 450V, with a discharge power output of up to 4800 W. These specifications serve as the baseline for this thesis.
- c) **Thruster Unit (Anode + Cathode + XFC):** This unit comprises the anode, cathode, magnets, discharge channel, and Xenon Flow Controller (XFC). The anode supplies fuel to the discharge channel, while the magnets generate the necessary magnetic field. The cathode plays dual roles: ionizing the gas in the discharge channel and neutralizing the outgoing ions to form the thrust plume. The XFC includes valves that precisely control the flow rates to the anode and cathode elements. In some designs, the XFC is considered part of the thruster unit, as seen in this study.

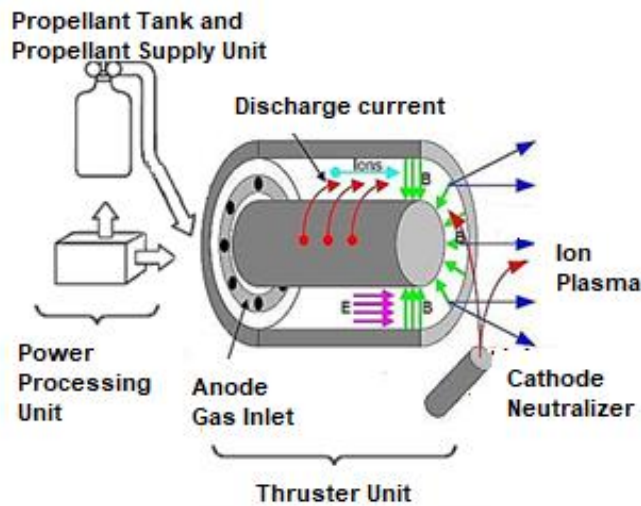


Figure 5.1: EPS Sub-units.

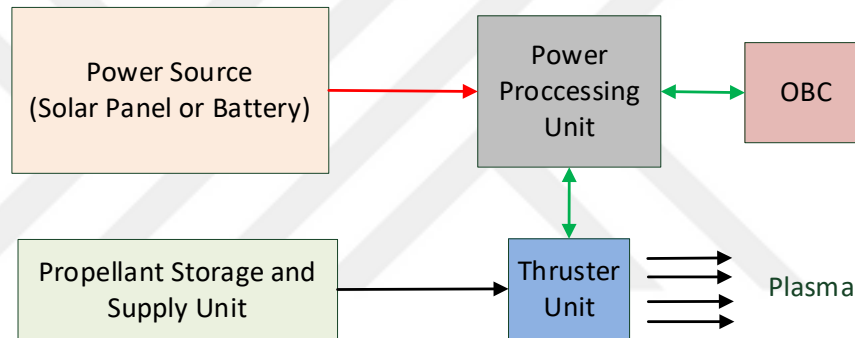


Figure 5.2: HET Operational Flow Diagram – Basic.

To operate, HETs require an electrical power source, typically obtained from solar panels or, in some cases, batteries within the Power Control Subsystem. The power generated by solar panels is in raw form and needs to be converted to the specific voltage and current values required for the HET's operation. This conversion task is carried out by the Power Processing Unit (PPU).

The PPU controls the thruster unit, as depicted in Figures 5.1 and 5.2. It regulates the electrical power supplied to the cathode and anode of the thruster, thereby governing the thrust output.

In Hall Effect Thrusters (HETs), Xenon gas is commonly used as a propellant (though other noble gases like Krypton or Argon can also be used), stored in a pressurized tank at approximately 180 bar. Before Xenon can be utilized as a propellant, it must be

introduced into the thruster's discharge channel. Transitioning the gas from a high-pressure environment to a vacuum without pressure reduction can cause flow issues, necessitating a reduction to a typical operational pressure of approximately 2 bar. This pressure reduction is achieved using a flow restrictor or regulator. The pressure-regulated Xenon gas is then supplied to the thruster's anode at a controlled flow rate through openings in the neutralizing hollow cathode, as illustrated in Figure 5.1.

The electrical power from the Power Processing Unit (PPU) connects to the anode of the Hall Thruster as the positive (+) pole and to the neutralizing hollow cathode as the negative (-) pole. Electrons emitted from the hollow cathode neutralize the ions in the exhaust gas produced by the thruster. Electrons, attracted from the cathode to the anode by the electric field, initially move in straight paths. However, their trajectories change radially due to the influence of the strongest magnetic field component at the channel exit, preventing direct movement towards the anode. Instead, these electrons become trapped around the axis of the thruster, forming a rotating region of trapped electrons [34, 35, 36, 37].

The Hall current is generated as electrons rotate around the motor axis in a region where perpendicular electric and magnetic fields confine them, giving rise to its name, 'Hall current.' When neutral xenon gas atoms collide with these electrons within the Hall current, they undergo ionization through elastic collisions. The resulting ionized xenon atoms are then accelerated by the electric field established by the potential difference between the anode and cathode. This acceleration occurs within the electric potential pit illustrated in Figure 5.3. Subsequently, ions are expelled from the thruster at speeds of approximately 15,000 m/s, generating thrust. Meanwhile, electrons emitted from the cathode neutralize some of these ions to maintain the overall charge balance within the system. The potential difference between the anode and cathode is responsible for creating this ion exit velocity, as described by Equation 5.4.

$$v_i = \sqrt{\frac{2q_i e V_i}{m_i}} . \quad (5.4)$$

Equation 5.4 expresses the ion velocity v_i in terms of the ionization degree q_i , the elementary charge e of charged particles, the acceleration voltage V_i , and the ion mass m_i . Meanwhile, electrons moving azimuthally collide with atoms of the propellant gas and the thruster's walls. These collisions result in the ionization of the propellant gas

and some electrons being absorbed by the walls. The remaining electrons are then directed towards the anode, completing the discharge current circuit.

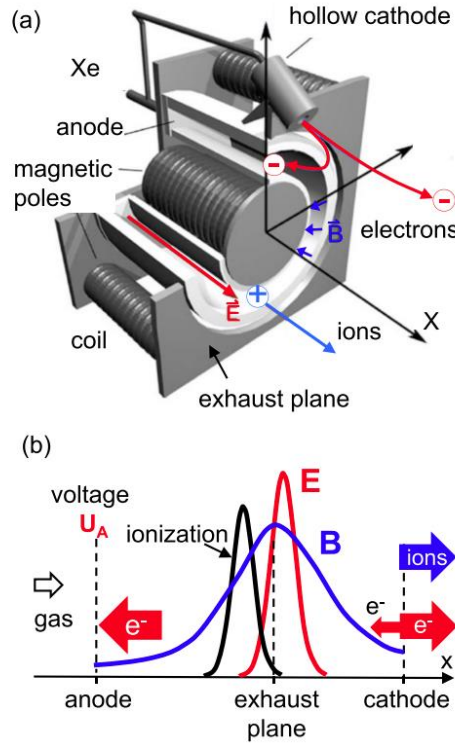


Figure 5.3: Magnetic and Electric Fields along the HET.

The thrust generated by a HET result from the momentum transferred by ions expelled from the thruster. Without neutralization, the spacecraft would gradually acquire a negative charge. To prevent this, electrons are emitted from a hollow cathode onto the ions, neutralizing them and maintaining the overall charge balance in the system.

The optimal operating conditions for a HET depend on the design of the magnetic field, as illustrated in Figure 5.3. The magnetic field strength must be sufficient to confine electrons within the discharge channel but not so strong that ions can escape [38, 39, 40, 41]. This balance is achieved by ensuring that the thickness of the ionization region, denoted as L_a , is much larger than the electron gyroradius yet much smaller than the ion gyroradius, as expressed in Equation 5.5.

$$\frac{\sqrt{2m_e V_d/e}}{B} \ll L_a \ll \frac{\sqrt{2m_i V_d}}{q_i e} \quad (5.5)$$

Relation 5.5 relates the ionization region thickness L_a , electron mass m_e , ion mass m_i , discharge voltage V_d , elemental electron charge e , number of ionizations per atom q_i ,

and magnetic field strength B . The left side of the equation represents the electron gyroradius, while the right side represents the ion gyroradius. The dimensions of HETs are critically dependent on Relation 5.5 [25, 26, 27, 28, 33, 35].

The perpendicular electric and magnetic fields induce a very small azimuthal velocity in the ions, making them primarily influenced by the axial electric field. Therefore, HETs are classified as electrostatic devices rather than electromagnetic devices.

To optimize the design of a Hall thruster for maximum thrust, efficiency, and minimal erosion over time, several key factors related to the magnetic field and operational principles should be examined in detail:

- Shaping the electric field distribution and controlling ion orbits (magnetic cusp effect) [42, 43, 44]
- Thrust direction control (magnetic steering) [45, 46]
- Techniques for controlling magnetic field strength and distribution [47, 48, 49, 50]

5.3. Studied EPS Architecture and Interfaces with Satellite

EPS has extensive interfaces with other satellite subsystems, as it is responsible for a variety of critical functions, such as propellant management, pressure control, and thrust provision. The specific interfaces of the EPS are detailed in Table 5.3, which provides a comprehensive overview of how the EPS connects and interacts with the other subsystems or driver units of the satellite system.

Table 5.3: EPS Interfaces with Other Satellite Subsystems.

Subsystem	Power Interface	Data Interface	Mechanical Interface	Thermal Interface
Power Control System (PCS)	X	-	-	-
Onboard Computer (OBC)	-	X	-	-
Thermal System	-	-	-	X
Structural System	-	-	X	-

Due to the stringent requirements for precise pressurized fuel storage and flow rate control in all propulsion systems, including EPS, meticulous thermal management is essential. This involves detailed analysis and testing of the placement of thermal control hardware and the temperature ranges necessary to maintain equipment integrity and ensure operational reliability, performance, and longevity of the systems.

Besides thermal control interfaces, EPS also includes mechanical interfaces. EPS is distributed across nearly all satellite panels, driven by the need to allocate thrusters for mission maneuvers, which results in the distribution of EPS units and tubing among the thrusters.

The propellant stored in the tank needs to be routed through piping to reach the anode and cathode elements of the thruster. Throughout this process, it is crucial to maintain the propellant in its desired supercritical phase and prevent phase changes, necessitating careful thermal control. Additionally, pressure drops can occur during propellant transfer due to tubing bends, long straight pipe sections, or contact with equipment along the route. Proper analysis of these factors is essential to establishing an accurate performance model and ensuring the fluidic system operates effectively.

In this system, high-pressure xenon gas stored in the tank is isolated using isolation valves for safety. Given that the thruster operates at a significantly lower pressure than the storage tank, a regulator is typically used to control the pressure before the propellant reaches the flow controller. Finally, at the flow controller stage, pressure-limiting equipment such as a thermothrottle ensures the propellant is delivered to the anode and cathode elements at the desired ratio and mass flow rates.

The flow diagram of the Xenon circuit developed in this thesis and utilized in the mathematical model is depicted in Figure 5.4.

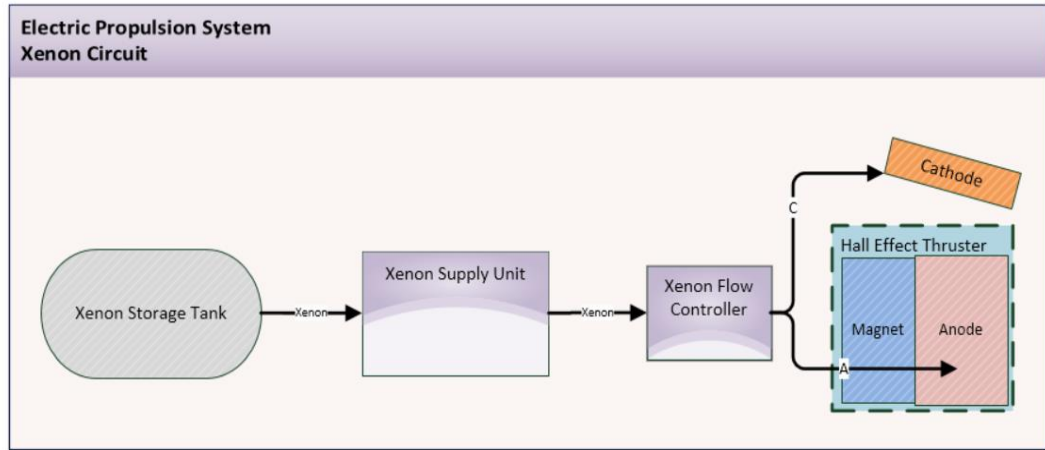


Figure 5.4: EPS Xenon Circuit Flow Diagram.

Although this section covers all EPS external and internal interfaces, including thermal and mechanical aspects, the primary emphasis in this study lies on the electrical interfaces. Specifically, the design and analysis of the electrical system have focused extensively on how these interfaces are represented in the mathematical model. The incorporation of electrical characteristics and interactions among various components into the modeling and simulation process was crucial as the study progressed.

The EPS electrical interfaces with driver units of PCS and OBC, as well as the EPS internal interfaces, mainly between PPU and the Thruster Module, are clearly presented in Figure 5.5.

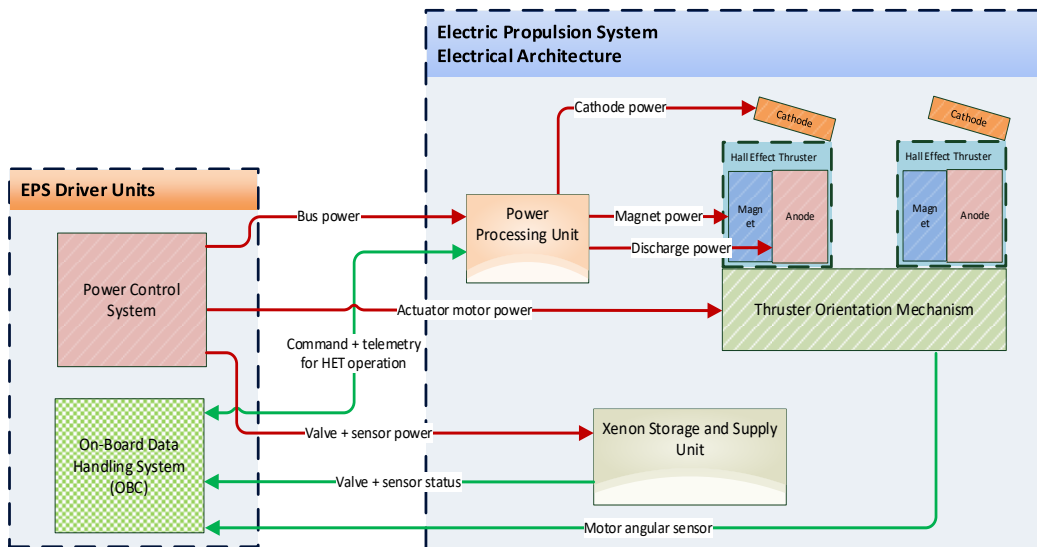


Figure 5.5: EPS Electrical Architecture.

As seen in Figure 5.5, EPS shares electrical external interfaces primarily with two main subsystems of a satellite. The functions and details of these drivers are summarized below.

- Power Control Subsystem (PCS) units: The Power Control Subsystem (PCS) interfaces with EPS to manage and provide power for various EPS units and equipment. It supplies main (100V) and secondary (28V) bus power directly to PPU's for the operation of the thruster module, including cathode, magnet, and discharge powers. PCS driver units also power the Xenon Storage and Supply Unit's pressure sensors and valve actuators. Additionally, PCS drivers control the motor actuators of the thruster orientation mechanism, which adjusts the thruster's attitude. If heaters are required for EPS thermal control, they are also powered by PCS drivers. Figure 5.5 illustrates these interfaces.
- On-Board Computer (OBC): The on-board computer (OBC) interfaces with EPS to command and monitor various EPS components and systems, ensuring overall status monitoring and control of the satellite. The OBC communicates via telemetry and telecommand with PPU's through a serial data bus interface to manage the thruster's operational parameters. Additionally, the OBC interfaces with the Xenon Storage and Supply Unit to command valve operations and monitor the telemetry of valve open or closed status, pressure, and temperature sensors. The OBC also controls the orientation of the thruster motor mechanism. It's important to note that any modification or reset of thruster operational parameters must be commanded by the OBC to the PPU's, since the OBC serves as the decision-maker brain for the satellite system. PPU's should not autonomously alter the operation of the thruster, as this can lead to issues. Figure 5.5 illustrates these interfaces.

6. MATHEMATICAL MODELLING

Mathematical modeling is the development of mathematical equations to accurately describe and analyze the behavior exhibited by physical systems based on a thorough comprehension of the fundamental laws, principles, and processes governing the underlying phenomena. A mathematical model should accurately represent the system's characteristics and behaviors, serving as the foundation for analyzing and controlling the system. By expressing physical laws, principles, and processes faithfully through mathematical equations that incorporate system parameters and input-output variables, a comprehensive and functional model is achieved.

Mathematical models are essential for the analysis and control of physical systems. They enable the understanding of complex systems by breaking them down into manageable components, which can be studied and manipulated using mathematical equations and principles. The models can be used to predict the behavior of the system under different conditions, allowing for the optimization of performance and the prevention of potential issues.

Mathematical models are also used in the design and development of new systems. By creating a model of the desired system, engineers can test and refine the design before implementing it in the physical world. This can save time, money, and resources, as potential problems can be identified and addressed in the design stage. Additionally, mathematical models can be used to simulate various scenarios, allowing for the evaluation of different design options and the selection of the most effective one.

In addition to their role in the analysis, control, and design of physical systems, mathematical models are also used in scientific research and education. They provide a framework for understanding and exploring complex phenomena, allowing researchers to make predictions and test hypotheses. In education, mathematical models are used to teach students about the behavior of physical systems and the principles of mathematical modeling.

6.1. Modelling Environment (MATLAB - Simulink)

MATLAB is a widely used software renowned for its flexibility and robust features. It functions not only as an application but also as a programming language with a matrix-based operational structure. MATLAB supports a wide range of computations, programming tasks, optimization, statistics, and the creation of 2D and 3D visualizations. With its focus on matrices, MATLAB offers a comprehensive platform for diverse computational needs and data visualization, serving as a valuable tool for professionals and researchers.

In the context of a thesis project, MATLAB's computational capabilities are combined with simulation, a significant aspect. Simulink, a part of MATLAB, specializes in multidisciplinary system simulations using block diagrams for model creation, aiding in simulation, code generation, and test system validation. Simulink's block library caters to various modeling requirements, providing a diverse set of components for system analysis and representation.

MATLAB is distinguished as a versatile and potent tool in engineering and scientific research. Its computational strength, coupled with simulation functionalities in Simulink, allows users to conduct complex analyses, build models, and validate system designs effectively. The software's intuitive interface, extensive function library, and visualization features make it an essential tool for professionals involved in intricate computational tasks and system simulations.

MATLAB - Simulink provides a powerful tool called S-functions to create custom Simulink blocks. An S-function is a computer language description of a Simulink block, and it can be written in MATLAB, C, C++, Ada, or FORTRAN. C, C++, Ada, and FORTRAN S-functions are compiled as MATLAB executable files.

S-functions can be used for a variety of applications, such as adding new general-purpose blocks to Simulink, incorporating existing C code into a simulation, or describing a system as a mathematical set of equations. An advantage of using S-functions is that a general-purpose block can be built and used many times in a model, with varying parameters for each instance of the block.

Simulink also provides a feature called masking, which enables the customization of the dialog box and icon for a subsystem. Masking allows simplifying the use of a model by replacing multiple dialog boxes in a subsystem with a single one. This provides a more descriptive and helpful user interface by defining a dialog box with user-specified block descriptions, parameter field labels, and help text. Masking also allows defining commands that compute variables whose values depend on block parameters, creating a block icon that depicts the subsystem's purpose, preventing unintended modification of subsystems by hiding their contents behind a customized interface, and creating dynamic dialogs.

As the model increases in size and complexity, subsystems can be used to simplify it. The use of subsystems provides the advantage of reducing the number of blocks displayed in the model window, keeping functionally related blocks together, and establishing a hierarchical block diagram where a subsystem block is on one layer and the blocks that make up the subsystem are on another.

These features are used to form the components for propulsion system modeling and simulation.

6.2. Description of Developed Mathematical Model

In this thesis study, the physical and avionics architecture of the system to be mathematically modeled is detailed in Sections 5.1 and 5.3. Now, almost the entire system has been transferred to the MATLAB Simulink program according to the defined avionics architecture, and simulations have been carried out here. The reason for saying "almost the entire" is that some thruster-specific parameters are only available to the thruster manufacturer, and it is almost impossible to model such parts. However, the remaining parts have been modeled using the equations provided earlier. The equations from Jahn, R. G. [19] and D. M. Goebel and I. Katz [38] have made very important contributions here. It should be noted that in addition to the equations of some variables presented in the previous sections, many more equations have been used in modeling. The governing equations and their derivatives from [19] and [38], especially applicable to HET-type thrusters, from and on the physics of electric propulsion have been used in the model produced in this study.

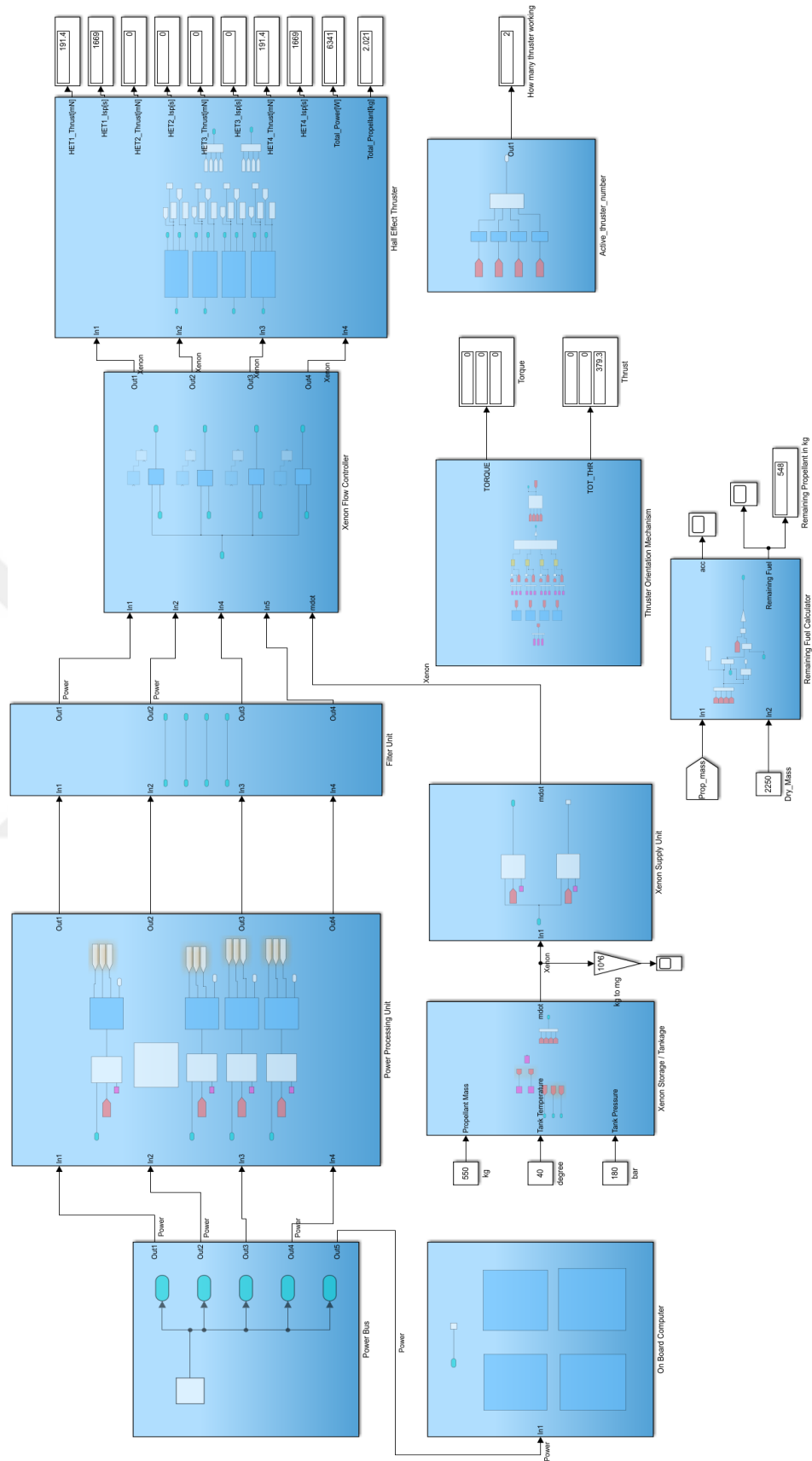


Figure 6.1: Mathematical Simulink Model General Overview.

The developed mathematical model not only offers an in-depth modeling capability specific to the thruster but also reflects the functions of other equipment such as the thruster orientation mechanism, Xenon flow controller, power processing unit, and OBC in terms of signal modeling (logic relations).

Since the model is extensive and all functions cannot be presented on a single screen, the functions of segmented model sections will be introduced in this section.

6.2.1. Power Bus (PCU Drivers) Block

The Power Bus block, as described in Section 5.3, incorporates PCU drivers for managing the 100V and 28V power interfaces. There are a total of four PPUs responsible for supplying the main 100V bus interface. Figure 6.2 illustrates a line connecting to the OBC, which supplies power as required. The design ensures reliable isolation of the 100V and 28V lines using relays and switches.

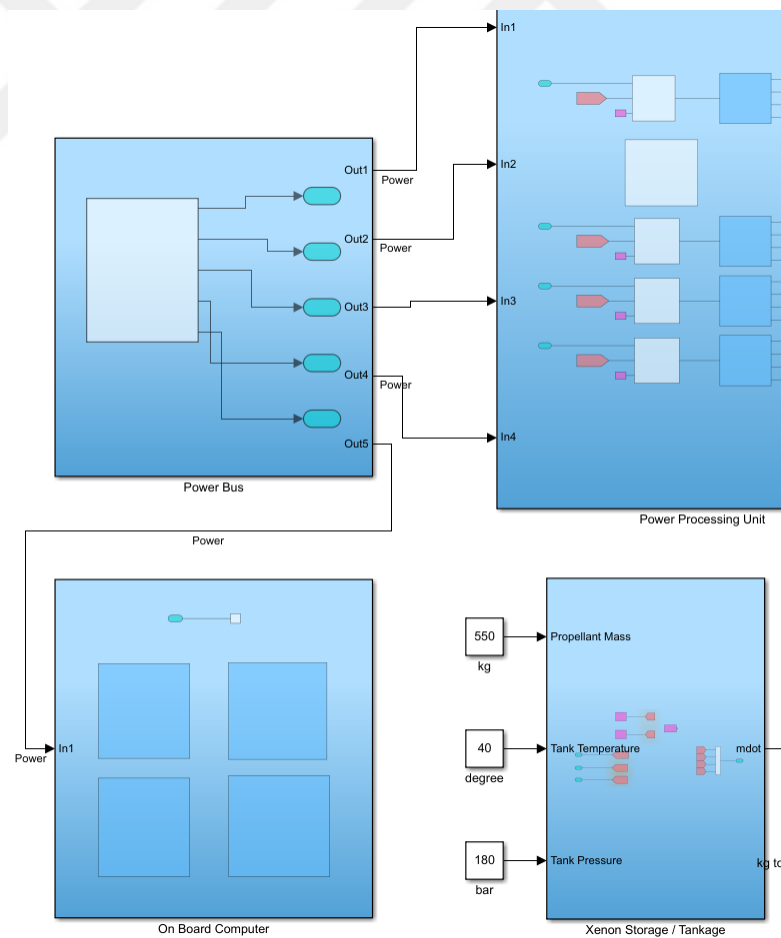


Figure 6.2: PCU Block.

6.2.2. OBC Block

On the other hand, the OBC block is responsible for commanding and controlling the entire system, as detailed in Section 5.3. It includes a processor that receives telemetry data, processes it to make decisions, and transmits these decisions to the relevant equipment. Although the block does not directly interface with other equipment via a signal interface in this mathematical model, wireless signal interfaces have been incorporated using Simulink's GoTo/From blocks.

Within the OBC block, the model encompasses control of PPU parameters, management of HET operations on a large scale, oversight of XSU valves and data, and adjustment of TOM motor orientations. These interfaces are illustrated in Figure 6.3.

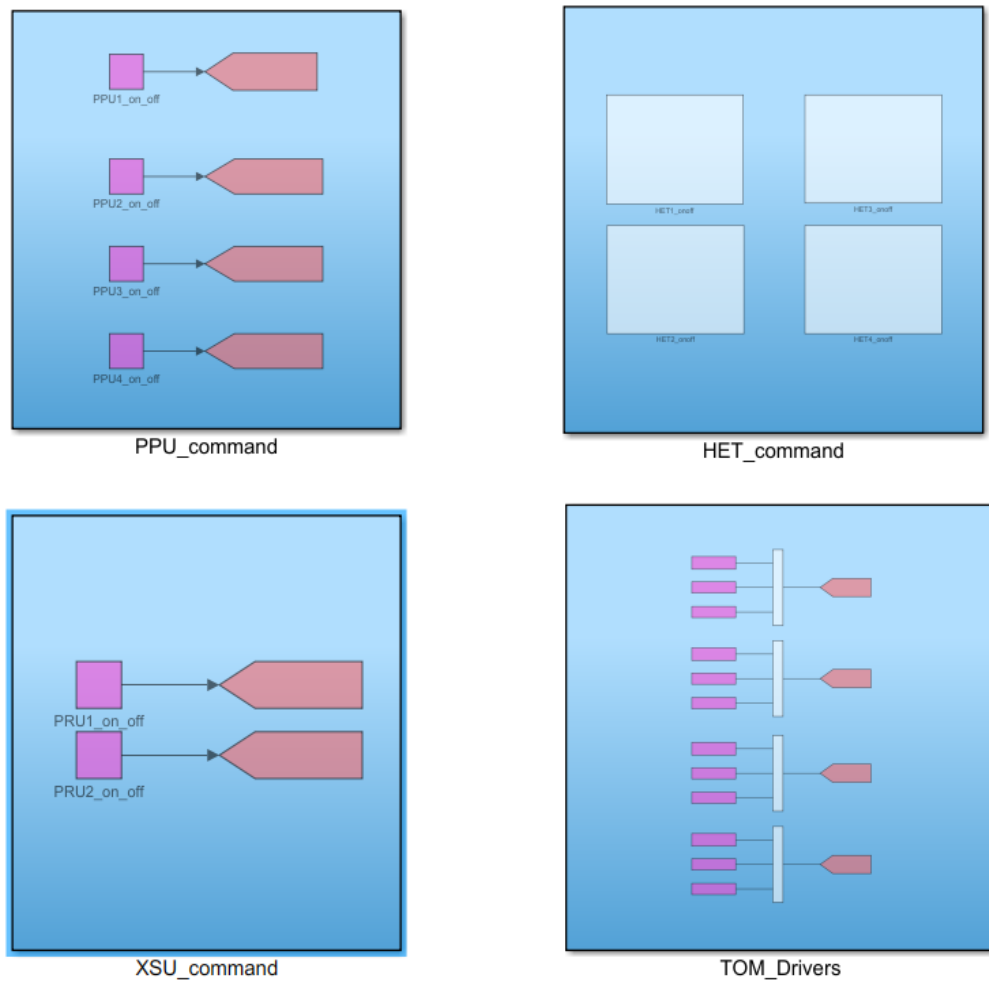


Figure 6.3: OBC Block.

The OBC block can control both the on/off status and operational parameters of the PPU equipment, as depicted in Figure 6.4. These operations are executed in accordance with commands received from the ground station throughout the mission. The PPU operational parameters ensure the thruster operates efficiently within the correct parameters. These include cathode power, magnet power, anode current, and anode voltage. Subsequent sections will elaborate on how changes in these parameters affect thruster performance.

Moreover, the OBC block is equipped to independently activate or deactivate all discrete power supplies to the thruster, as represented in Figure 6.4 of the model.

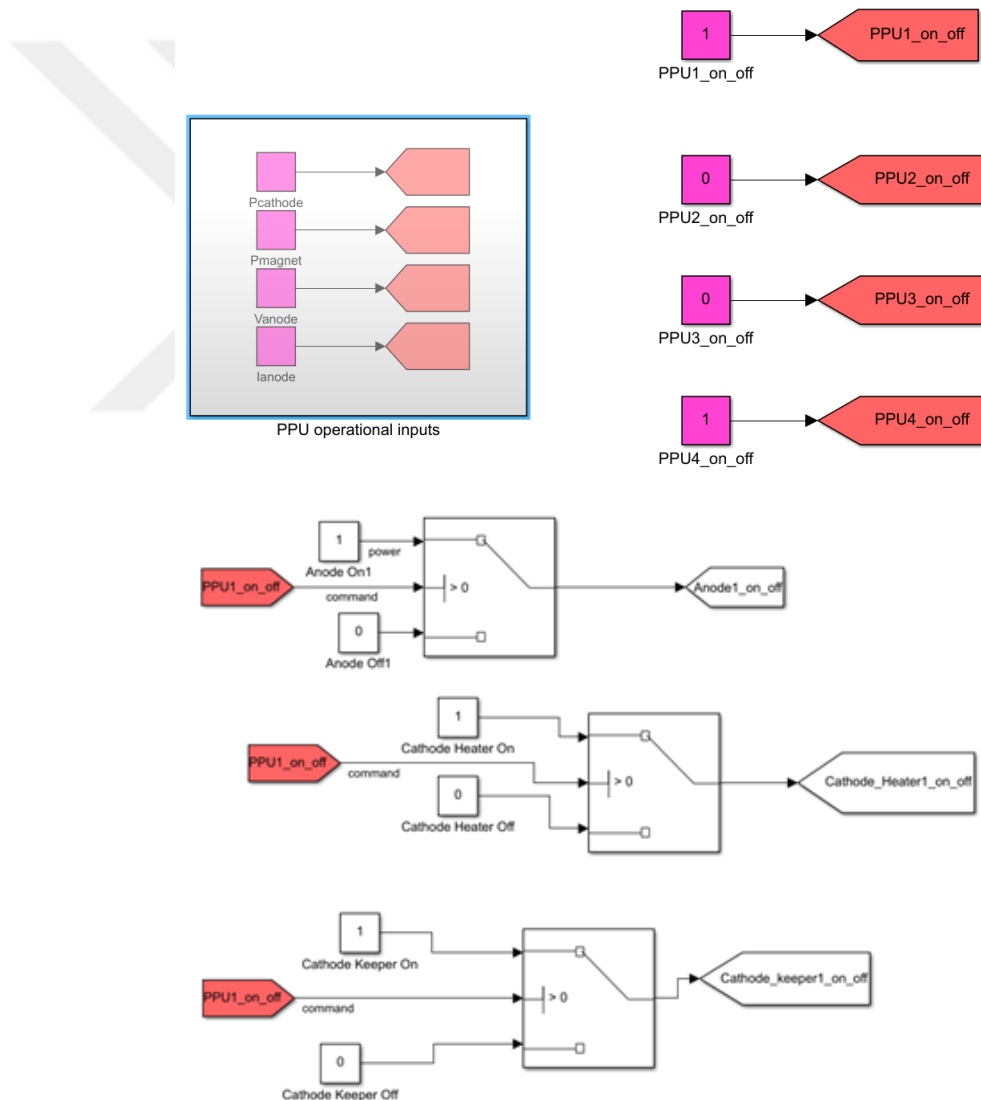


Figure 6.4: OBC - PPU Interfaces.

Another crucial function of the OBC is to ensure the accurate orientation of TOM equipment and verify this orientation using telemetry. This capability is illustrated in Figures 6.3 and 6.5 within the enhanced model. In this mathematical model, each thruster is assumed to be mounted on a separate TOM, allowing independent orientation along the XYZ axis. Typically, in practice, due to cost considerations, scenarios with two thrusters per TOM are more common.

Nevertheless, this model can be easily adapted to align with standard practices through minor adjustments. Users can input orientation information in XYZ coordinates for each TOM directly into the OBC interface, as demonstrated in Figure 6.5.

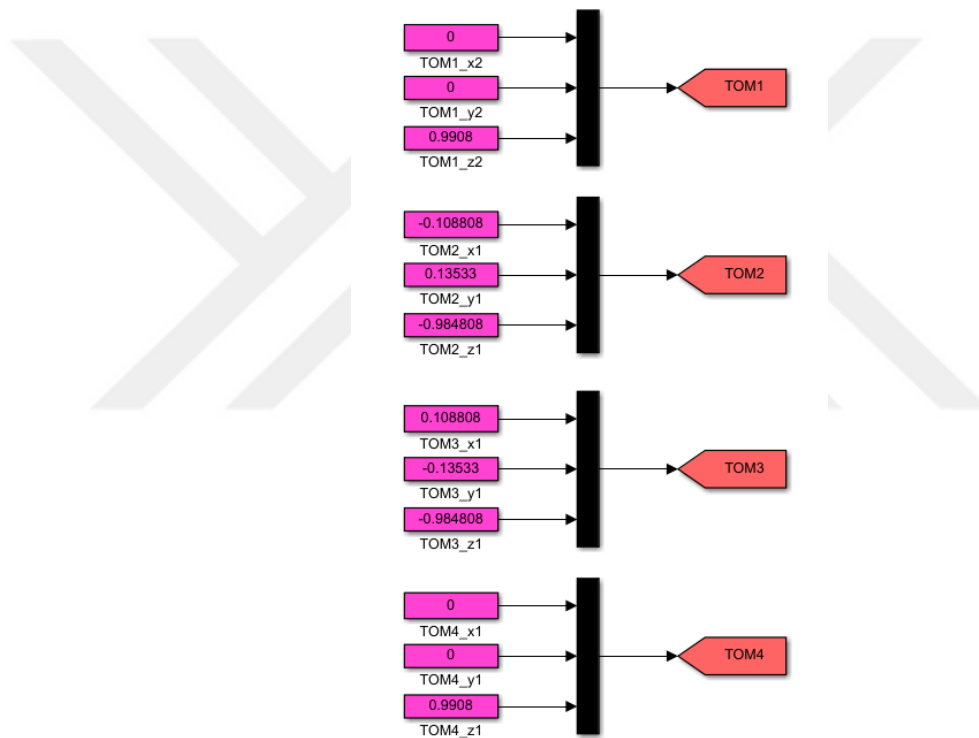


Figure 6.5: OBC - TOM Interfaces.

6.2.3. Xenon Storage and Supply Blocks

The Xenon Storage and Supply units aim to provide input to the Remaining Propellant Calculator based on the initial mass, temperature, and pressure of the propellant in the storage tank. In practical terms, this module can be thought of as the pressure and temperature sensors found on the satellite's tank. These parameters are user updateable. Simultaneously, this module contributes to calculating the total mass flow rate of the

propellant, utilizing the mass flow meter within the thruster block, which adjusts based on the number of active thrusters. The outputs of this module serve as the foundation for calculations in the Remaining Propellant Calculator module. The resultant output, the mass flow rate, interfaces with the display block to determine the present propellant mass flow rate from the storage tank to the thrusters.

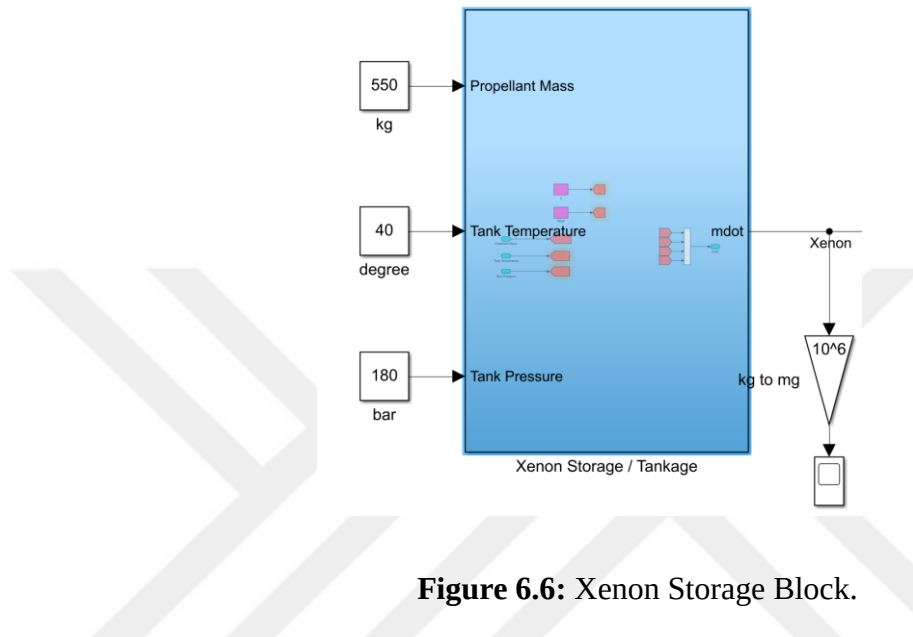


Figure 6.6: Xenon Storage Block.

Closing the isolation valves is one of the functions included in the XSU. This function is dedicated to ensuring the safe transmission of propellant. It requires a total of 3 inhibited uses in systems that could pose high-pressure catastrophic risks, aimed at mitigating hazards within the system. This block functions as a valve that the on-board computer (OBC) may regulate, but it has no direct effect on the model's performance. The Xenon Supply Block in the mathematical model is shown in Figure 6.7.

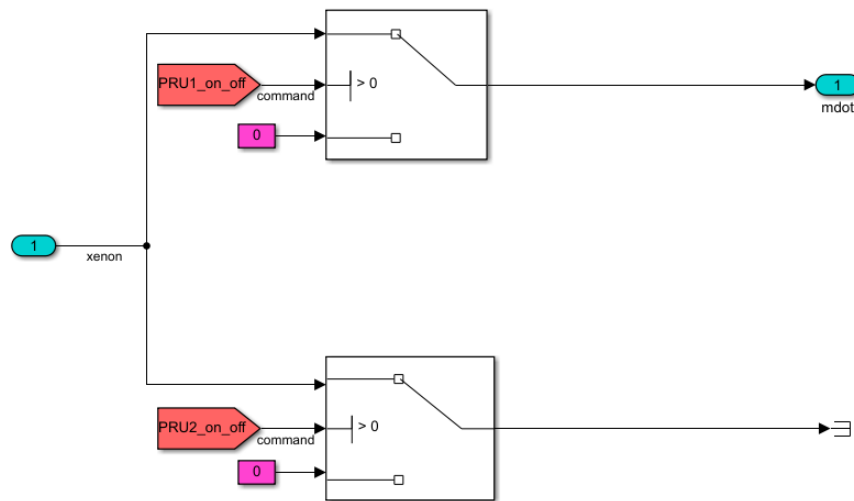


Figure 6.7: Xenon Supply Block.

6.2.4. Power Processing Unit and Filter Unit Blocks

The pivotal controller in the avionics architecture, detailed in Section 5.3 and depicted in Figure 5.5, is the Power Processing Unit (PPU) within this model. The PPU manages the entire startup sequence of the thruster, including individual equipment control requirements. Figure 6.8 provides a general overview of its functionality.

Initiating its operations upon commands from the on-board computer (OBC), the PPU oversees the activation of the cathode heater, igniter, and keeper within the thruster. Additionally, it regulates the valves and thermothrottle in the Xenon Flow Controller (XFC), transmitting pertinent data back to the OBC. The PPU also governs the magnet current crucial for ionization efficiency in the thruster, influencing the likelihood of collisions between electrons and neutral Xenon atoms.

Moreover, the PPU supplies the necessary voltage and current to the anode elements. A notable and challenging aspect of its role, not critical for performance simulation but significant for practical implementation, involves stepping up the 100V main bus input to the voltage levels required by the thruster—typically between 300V and 450V. This transformation is essential for high-efficiency HET operation and distinguishes the PPU in terms of complexity and cost compared to other hardware on the market.

Selecting Electrical, Electronic, and Electromechanical (EEE) components capable of enduring high voltage and radiation conditions over a satellite's operational lifespan (often around 15 years) is critical and costly.

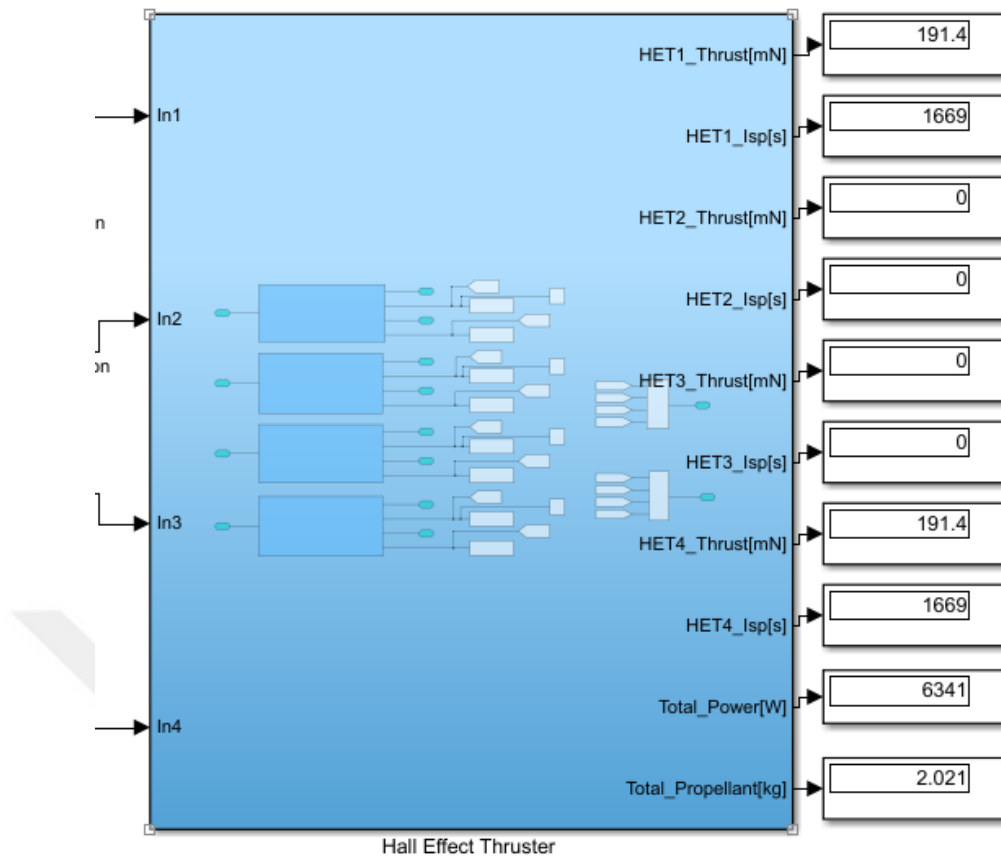


Figure 6.9: HET Block.

The provided sources detail the transition to the core operations of the model. This section is illustrated in Figure 6.9, presenting the inputs and outputs of the thruster block. The mathematical equations for this module are depicted in Figure 6.10. In the model represented in Figure 6.10, the Simulink Mask block feature has been utilized to incorporate inputs containing general parameters not directly visible to the user but essential for reflecting specific thruster details as required. This Mask feature ensures simplicity and clarity in user inputs.

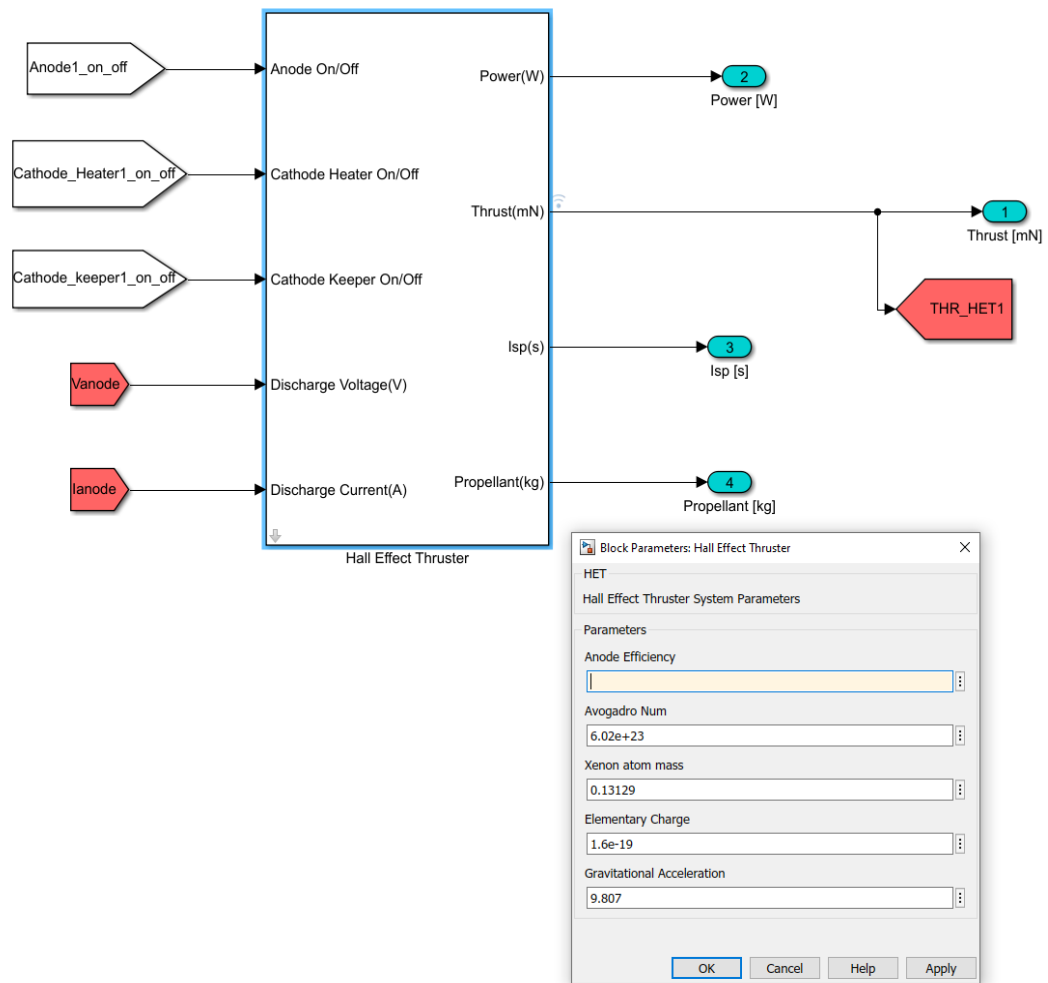


Figure 6.10: HET Block – Detailed.

The information presented in Figure 6.11 includes the equations for thrust and specific impulse (I_{sp}) calculations, along with validation factors specific to Hall Effect thrusters. This section also encompasses validation factors necessary for mass flow rate calculations, which will be further discussed in the XFC section. All power inputs from the PPU block are aggregated here to calculate the total power consumed by the system.

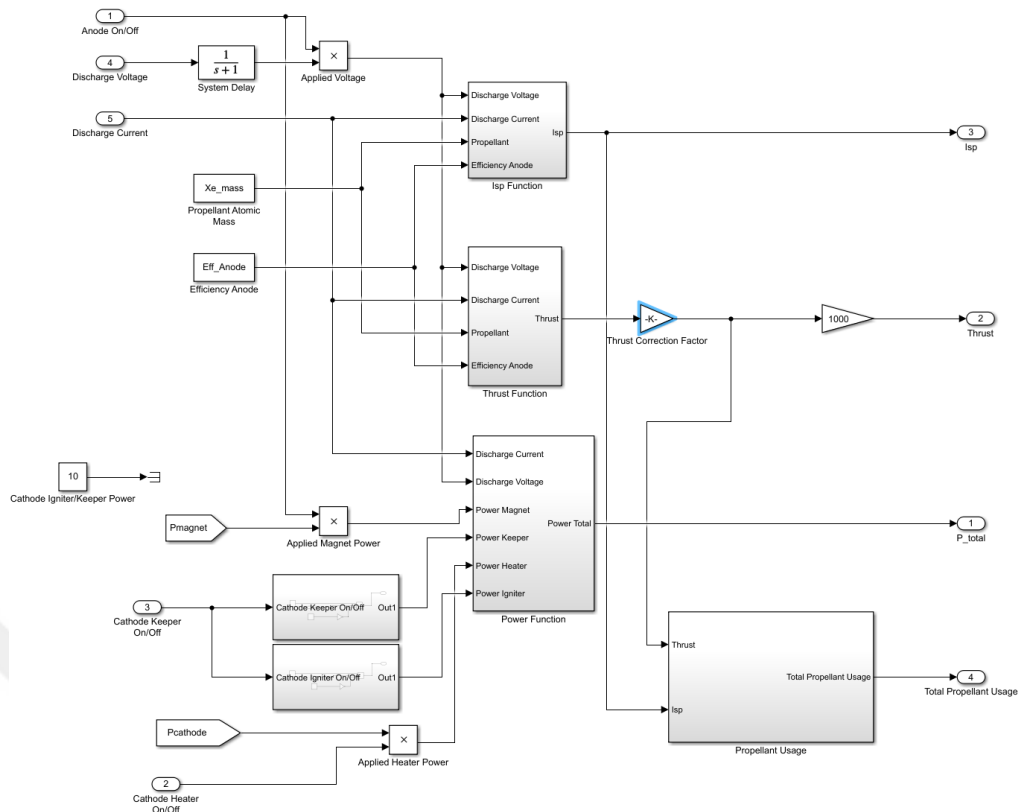


Figure 6.11: HET Block – Calculations.

The Xenon Flow Controller (XFC) ensures that xenon flow to the anode and cathode elements of the thruster is maintained at the correct flow rate and ratio using the thermothrottler and valves. An important factor to consider is the interdependence of mass flow rate, thruster current, and voltage. Increased anode current, for example, requires a higher mass flow rate. Therefore, a closed-loop controller managed by the Power Processing Unit (PPU) is crucial. The PPU dynamically adjusts the mass flow rate based on real-time thruster conditions.

Equation coefficients essential for accurate thrust and specific impulse (I_{sp}) calculations are derived from proprietary data obtained through ongoing projects and integrated into the model. This proprietary information, critical for precise performance predictions, is not publicly available worldwide. Details specific to the Safran PPS5000 thruster, a current supplier, have been incorporated into this model.

Due to confidentiality agreements, specific data cannot be openly shared. However, correction factors derived from studies on the PPS5000—such as the relationship between mass flow rate and power input and mass flow rate and thermothrottle—are embedded in the model with validated accuracy. This ensures the model accurately

reflects the power input vs. thermothrottle pattern required for optimal closed-loop operation.

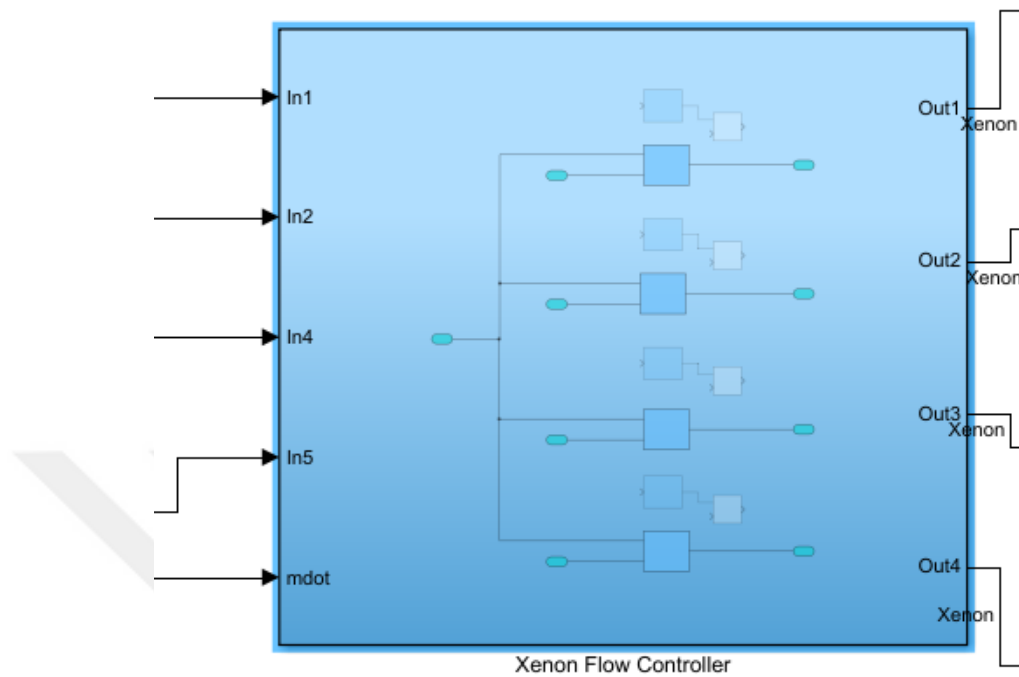


Figure 6.12: XFC Block.

6.2.6. Thruster Orientation Mechanism Block

This block encompasses equations that comprehensively calculate the external torque and force vectors generated by the satellite's thrusters, under the direction of the On-Board Computer (OBC). This is depicted in Figure 6.13.

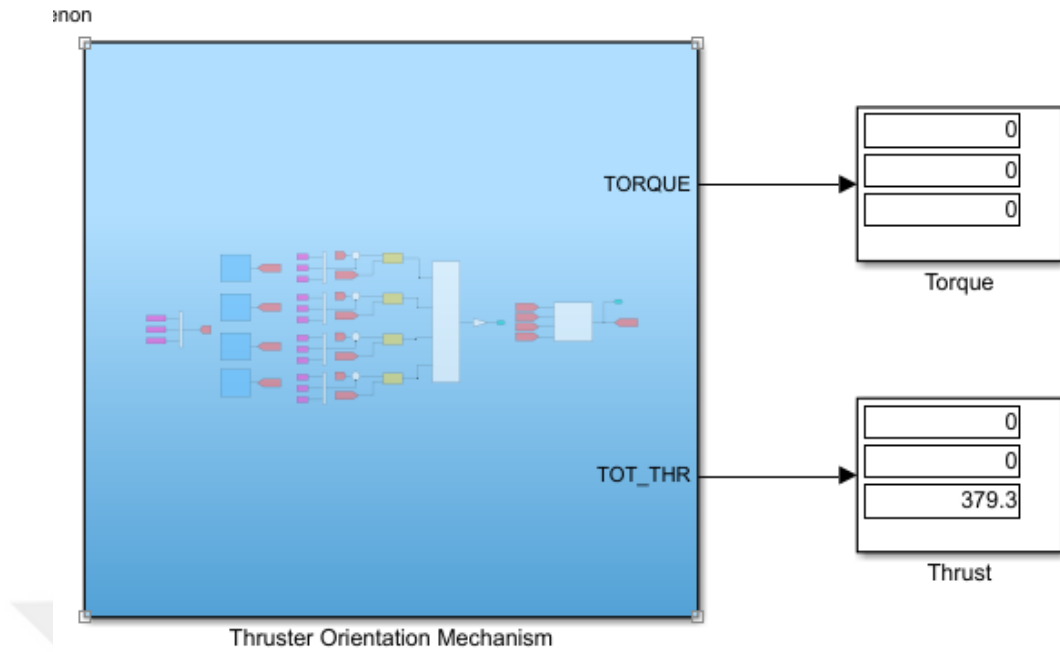


Figure 6.13: TOM Block.

The XFC block determines the thrust vector using satellite Center of Gravity (CoG) data, which users can update with their own satellite information at any time. With 4 Thruster Orientation Module (TOM) units corresponding to 4 thrusters, each thruster's orientation and position vary. Users can input thruster positions in this module, allowing observation of how different placements affect performance on the platform.

In this example platform, CoG values assume zero error in the x and y axes, with an exemplified CoG of +1200mm on the satellite's Z axis. The thrust vector results from multiplying the scalar-calculated thrust value from the HET block with TOM orientations determined by the OBC. Utilizing thruster position, thrust vector, and satellite CoG, this section also models external torque calculations during satellite attachment. Following vector cross operations, thrust and external torque values are output in vector form.

The external torque value derived here is crucial. In future studies, integrating external torque with data from the satellite's reaction wheels and current inertia information and incorporating a propagator will enable tracking of orbital changes induced by these factors. This study focuses solely on thruster mathematical modeling; future study content will be detailed in Section 8: Conclusion and Recommendations.

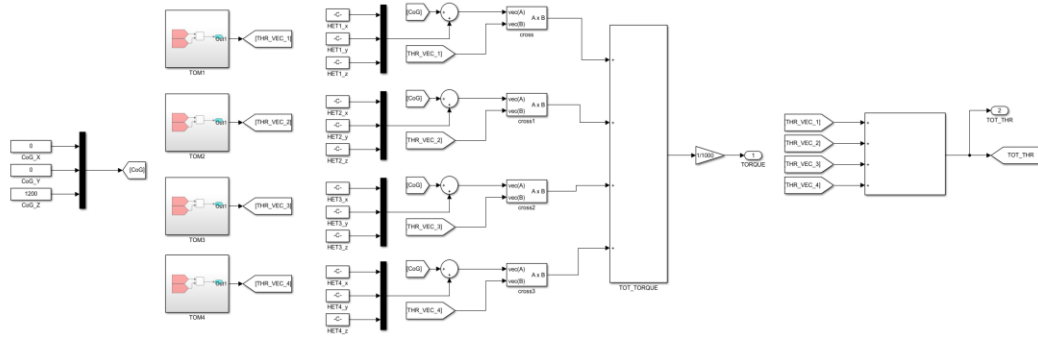


Figure 6.14: TOM Block – Calculations.

6.2.7. Remaining Fuel Calculator Block

When examining satellite propellant estimation methods, two commonly encountered approaches in the literature are the PVT method and the bookkeeping method.

As detailed in Section 4.1.3, the PVT method derives propellant estimation from temperature, pressure, and mass data provided by the user in the Xenon Storage Block. However, due to the unavailability of instantaneous sensor data in this study, the PVT method for propellant estimation was not preferred.

Instead, this study employs the bookkeeping method, which calculates propellant consumption per maneuver by recording thruster operation times and instantaneous propellant mass flow rates. This method, used extensively in real satellite models, accounts for details such as valve delays and command execution times, ensuring comprehensive estimation. Since this project's primary focus is on observing propellant consumption rather than modeling intricate operational details, a simplified calculation sufficed.

In Figure 6.15, the block acquires satellite propellant and dry mass information from the user to perform remaining propellant calculations (bookkeeping) based on recorded thruster operating times and instantaneous propellant mass flow rates. It also determines the total propellant expended during specified firing periods. Additionally, the module generates satellite acceleration information for potential use.

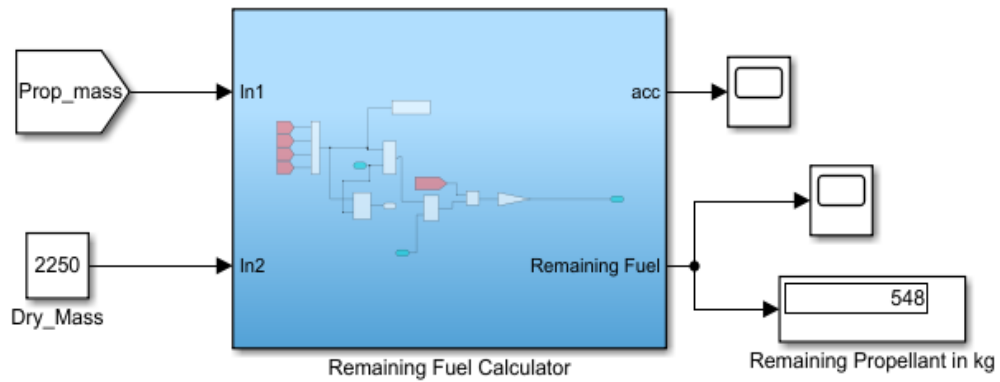


Figure 6.15: Remaining Fuel Calculator Block.

6.2.8. Number of Active Thrusters Block

In this model, the active status of thrusters commanded by the PPU can be communicated to the OBC, allowing determination of the number of thrusters operating simultaneously. This functionality is illustrated in Figure 6.16.

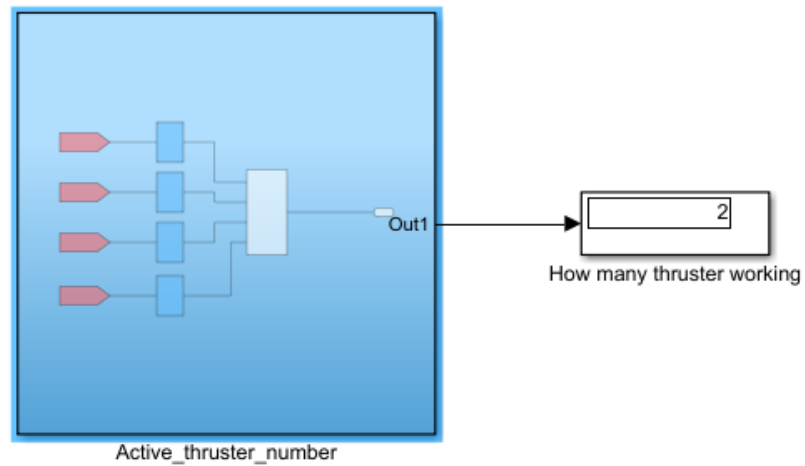


Figure 6.16: Number of Active Thrusters Block.

7. RESULTS OF THE DEVELOPED MATHEMATICAL MODEL

Within the scope of this thesis, results were obtained with various inputs to the mathematical model of a Hall effect thruster created in the MATLAB-Simulink program. Simulations have been conducted by altering thruster input parameters and operational mode. The results are presented and analyzed in this section.

7.1. Specified Model Inputs and Sample Firing Scenario

The previous sections extensively discussed the use of EP systems in satellite maneuvers, focusing particularly on crucial tasks such as orbit raising and station keeping. In this section, we will symbolically address a phase of the maneuver where the satellite, after separating from the launcher, stabilizes its orientation, determines the sun vector, and proceeds to raise to its final orbit for mission operations.

While some assumptions are hypothetical, certain inputs and assumptions are necessary to execute a sample maneuver. The assumptions made in this section and the user-provided inputs are listed below:

User inputs, assumptions, and determined parameters:

- Satellite dry mass: 2250 kg
- Satellite fuel mass: 550 kg
- Satellite launch mass: 2800 kg (2250 kg + 550 kg)
- Tank propellant temperature: 40 degrees Celsius
- Tank propellant pressure: 180 bar
- During firing, two thrusters operate simultaneously, as depicted in Figure 6.16. The number of operational thrusters is a variable in the model determined by the On-Board Computer (OBC).
- The thrust vector is generated using the Thruster Orientation Matrix (TOM) data provided in Figure 6.5.
 - During the orbit raising maneuver, it is essential to align the thrust along the satellite's Z-axis, which is the direction of acceleration. It is

assumed that the thrust vector has approximately 99% of its component along the Z-axis, although complete alignment may not be feasible due to mechanical constraints. In such cases, any torque generated is counteracted by an opposite torque from another thruster, preventing tumbling of the satellite during the maneuver.

- Initial torque can be applied to the satellite, but for this study, it is assumed that reaction wheels have nullified the satellite's angular velocity in all axes prior to firing.
- Important thruster parameters determined:
 - Anode (discharge) current: 10A
 - Anode (discharge) voltage: 300V
 - Magnet input power: 90W
 - Cathode supply power: 121W
 - Anode efficiency can be modified by the user for different thruster types.

While all outputs are significant, the primary objective of the model is to analyze parameters of the HET-based electric thruster. Figure 7.1 illustrates the observed I_{sp} and thrust values based on operational mode inputs. For instance, at a discharge voltage of 300V and discharge current of 10A, corresponding to a 3kW operational mode, the thruster generates 191.4 mN of thrust and achieves an I_{sp} of 1669 s. The figure displays outputs for the 1st and 4th thrusters, commanded open by the OBC.

In addition to the 3kW power input, the total instantaneous power consumption of 6341W is drawn from the platform, i.e., the PCS driver units, including the power consumption of the magnet supply and cathode supply. This relatively high instantaneous power consumption for satellites is used to shorten orbit-raising duration by obtaining more thrust, as electric thrusters are primarily used for the two most critical satellite missions: station keeping and orbital elevation. Typically, this high-power demand is met directly from solar panels, bypassing batteries. However, during eclipse periods when solar panels cannot supply sufficient power, thruster firing pauses until the satellite exits the eclipse.

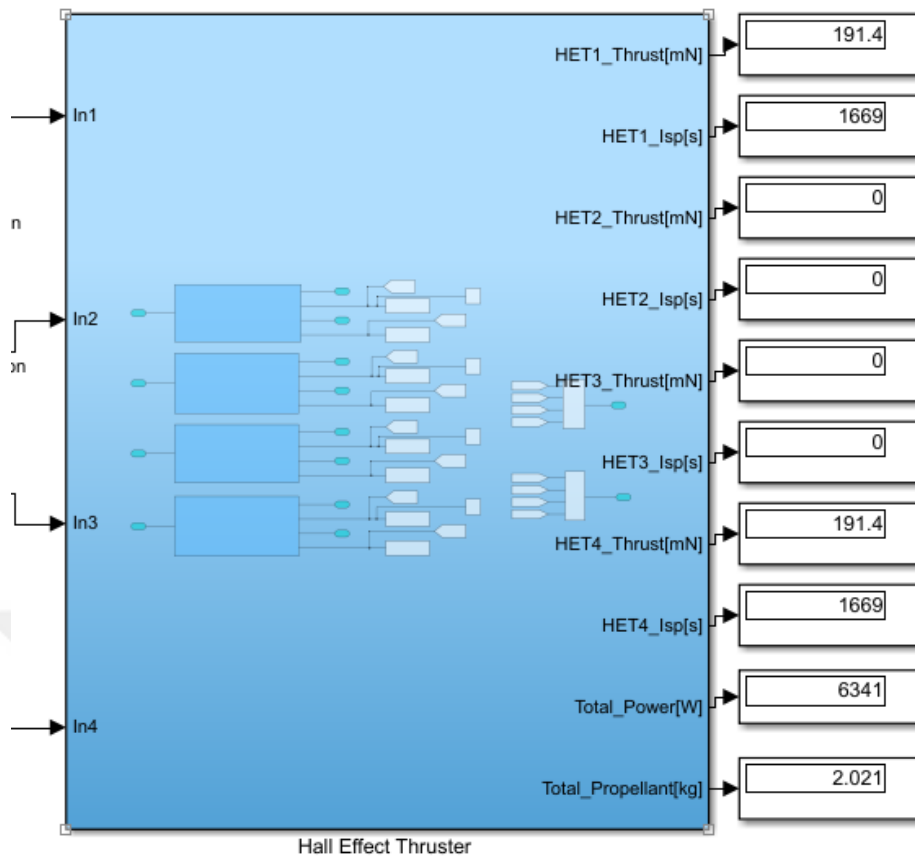


Figure 7.1: Results - Thruster Parameters.

All these inputs, along with a scenario involving continuous firing over one day, have been considered for the selected orbit-raising segment. In this scenario, two thrusters operate continuously for 84,600 seconds (1 day), and the total propellant consumption has been calculated accordingly. The results of this calculation are presented as both the total propellant consumed in Figure 7.1 and as a time-dependent graph showing remaining propellant in Figure 7.2.

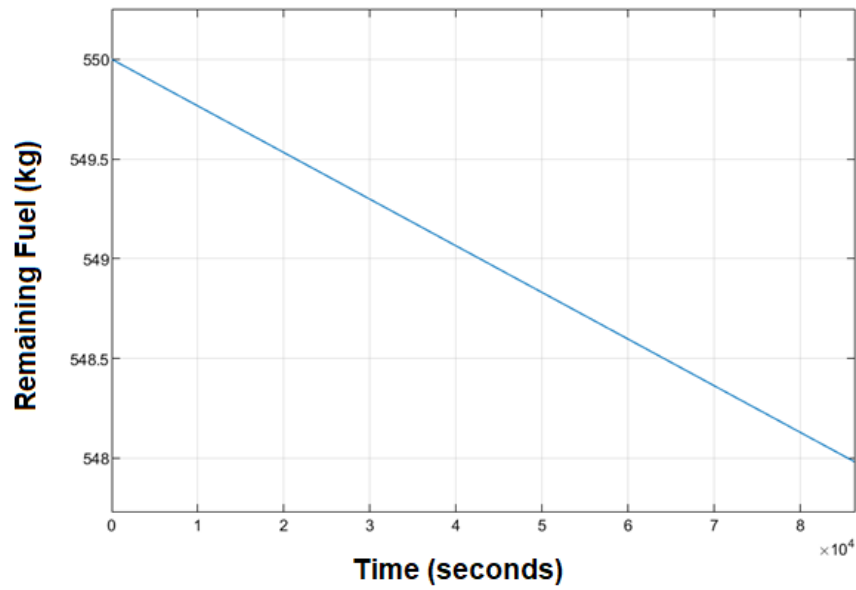


Figure 7.2: Results - Remaining Fuel.

7.2. Validation of the Developed Model

In general, there are two widely accepted approaches for model validation in engineering and scientific research:

- Verifying the model analysis results using test data.
- Comparing the model predictions with flight data from existing products documented in the literature or available on the market.

The first approach is often costly due to the requirement for vacuum chambers capable of simulating space conditions, which are typically available only to thruster manufacturers. This setup allows for realistic firing tests where propellant performance, thrust levels, and dispersion can be accurately measured. However, this method was impractical for our study, leading us to pursue the second approach.

We analyzed thrust and specific impulse (I_{sp}) data obtained from acceptance tests of the PPS5000 thruster, the preferred choice in current satellite platform designs. These values are detailed in Table 7.1 [51]. The data reveal significant standard deviation rates; even under identical conditions, there is an average fluctuation of 1% in both thrust and I_{sp} values, impacting their reliability.

Specifically, operational mode OP2 corresponds to conditions of 300V and 10A as listed in Tables 7.1 and 7.2. For OP2 in the thruster acceptance tests, average values were recorded as 193 mN for thrust and 1694 s for I_{sp} .

Table 7.1: PPS5000 Thruster Parameters [51].

	OP1		OP2		OP3		OP4		OP5		OP6		OP7		OP8		OP9		OP10	
	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)	F (mN)	I _{sp} (s)
Average	160	1634	193	1678	238	1694	286	1753	315	1771	143	1785	167	1805	242	1862	141	1853	163	1859
Std. dev.	0.6	21.5	1.1	31.1	1.4	20.4	1.9	28.7	2.4	23.1	1.9	31.1	0.9	23.9	1.4	22.4	1.0	28.6	0.9	25.1
Std. dev. (%)	0.4	1.3	0.6	1.9	0.6	1.2	0.7	1.6	0.8	1.3	1.3	1.7	0.5	1.3	0.6	1.2	0.7	1.5	0.6	1.4

More detailed thrust and I_{sp} graphs are presented in Figure 7.3.

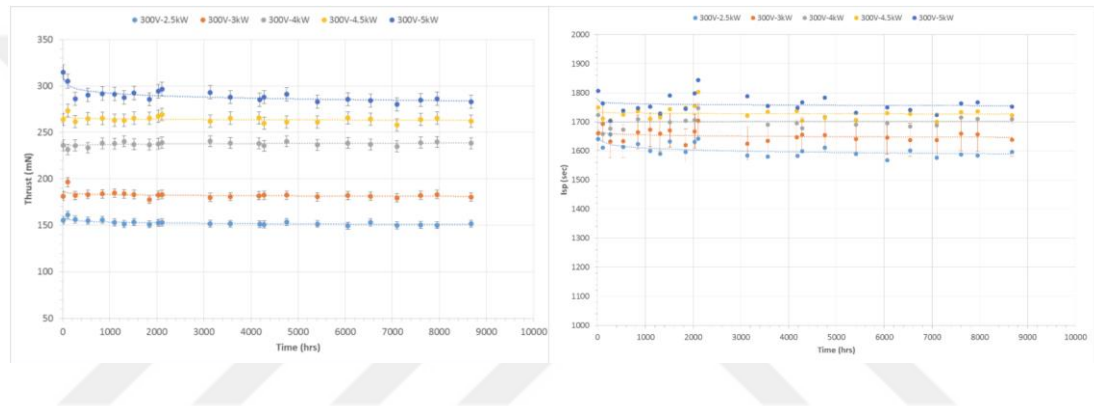


Figure 7.3: PPS5000 Thruster Parameters [51].

The orange dots in Figure 7.3 represent OP2, showing the ranges for thrust and specific impulse (I_{sp}). Another published study discusses the performance variation of the PPS5000 thruster's Engineering Qualification Models (EQM) in various test vacuum chamber setups. The primary distinction among these chambers lies in their background pressures, and Table 7.2 illustrates how these pressures affect performance parameters.

Table 7.2: Effect of Background Pressure on PPS5000 Thruster Performance Parameters [52].

Op. Point	Thruster @facility	Back. Pressure (mbar)	Ud (V)	Id (A)	Thrust(mN)	Isp(sec)	Difference (%)		
							Back. Pressure	Thrust	Isp
OP1	EQM1 @SAF LIC	8,20E-05	302	8,3	153	1632	-74%	0,7%	-1,7%
	EQM3 @AER LVTF2	2,10E-05	300	8,3	154	1604			
OP2	EQM1 @SAF LIC	9,37E-05	303	10,0	183	1670	-53%	0,5%	-1,0%
	EQM3 @AER LVTF2	4,41E-05	300	10,0	184	1654			
OP3	EQM1 @SAF LIC	1,16E-04	303	13,3	237	1699	-53%	2,1%	3,5%
	EQM3 @AER LVTF2	5,46E-05	300	13,3	242	1758			

The data in Table 7.2 shows that at a background pressure of $4.41\text{E-}05$ for the EQM PPS5000 thruster in OP2 mode, measured values were 184 mN for thrust and 1654 s for specific impulse (I_{sp}), showing a fluctuation of 0.5% to 2.1%. At nearly double the background pressure, $8.82\text{E-}05$, thrust measured 183 mN and I_{sp} 1670 s, with a fluctuation of 0.5% to 2.1% for thrust and 1% to 3.5% for I_{sp} , averaging 1% for thrust and 2% for I_{sp} . This analysis suggests the model results, with 1% thrust error and 2% I_{sp} error, are accurate and acceptable.

A study published in [52], which provides thrust and I_{sp} values, is considered less reliable due to its later publication date and focuses on acceptance tests directly on the Flight Model (FM) rather than the EQM. Therefore, comparing thrust and I_{sp} data based on values from [51] would be more appropriate. The standard deviation data in Table 7.2 for OP2 aligns with values in Table 7.3, indicating thrust fluctuation around 0.6% and I_{sp} fluctuation approximately 1.9%. Flight models generally provide more realistic results due to testing with flight components, offering closer approximations to actual performance.

In this section, acceptance test data from the literature is compared with model outputs developed in this study. Table 7.3 displays model-generated thrust and I_{sp} values alongside comparisons to experimental data in the literature.

Table 7.3: Mathematical Model Validation.

Parameter	Developed Mathematical Model	Thruster Qualification Campaign Results	Difference	Acceptable Difference*	Model Validation
Thrust (mN)	191.4	193	0,8%	1%	Good 
Specific Impulse (s)	1669	1678	0,5%	2%	Excellent 

* Defined based on Tables 7.1 and 7.2.

A comparison of the thrust and specific impulse (I_{sp}) predicted by the developed mathematical model with the test data presented in Table 7.3 demonstrates that the model predicts the thruster performance metrics within an acceptable error margin. In conclusion, the consistent alignment of the model performance predictions with the PPS5000 thruster qualification test campaign data validates the model and confirms its reliability.



8. CONCLUSION AND RECOMMENDATIONS

This study has compiled and presented information on various aspects related to the propulsion systems of satellites and spacecraft, which have become indispensable to our daily lives. These aspects include the history and evolution of propulsion systems, types of propulsion systems, patent studies on electric propulsion (EP) systems, the historical development of EP systems, comparisons between different propulsion systems, their working principles, and their areas of application.

Each satellite is equipped with a unique propulsion system tailored to the specific maneuver plans defined for its mission lifetime. This study discusses the advantages (and disadvantages) of satellites equipped with electric propulsion systems. It also examines the suitability of different thruster types for various maneuver types and satellite platforms.

Electric propulsion systems have become a trend in recent years, especially in new-generation small-size communication satellite platforms. These platforms utilize entirely electric propulsion systems, particularly those based on Hall Effect Thrusters (HET), to perform all required maneuvers, including station keeping and orbit raising.

The study has also introduced additional features of new-generation small-size satellites and provided information about their system architectures. Through the example of a specific platform, these details have been elucidated throughout the study. Such platforms offer advantages not only in terms of propulsion systems but also in terms of mass and size compared to their traditional counterparts.

Hall Effect Thruster (HET)-based systems are among the most preferred electric propulsion (EP) systems. The reasons for this preference have been thoroughly examined in the study, including detailed explanations of their working principles and the thruster parameter equations. Furthermore, the study introduces the avionics architecture required for a complex HET-based EP system, along with the satellite-level interfaces and other equipment necessary for its proper operation. Each component of the EP system is introduced, and their respective functions are examined.

The example platform and the detailed design of the HET-based propulsion system architecture throughout the study have been translated into a mathematical model developed in the MATLAB-Simulink environment, demonstrating the capabilities of MATLAB.

However, MATLAB excels in its ability to model systems using fundamental logical-mathematical relationships. Moreover, it offers a signal-based approach that can simulate behaviors akin to those observed in physical systems. This is achieved through feedback mechanisms where output variables within the same subsystem can influence input variables. Thus, MATLAB enables the creation of a modeling environment that bridges between signal-based and physical approaches. This does not diminish the utility of a physical approach; rather, it demonstrates that a signal-based tool can perform similar functions to those traditionally achieved through physical modeling.

Given the complexity of the EPS system, creating a mathematical model for it is inherently complex. The study introduces the importance and function of the EPS system, along with all the equipment that interfaces with the satellite within the mathematical model. Each Simulink block is introduced, and its specific purpose is explained.

In today's technology, medium-sized Hall Effect Thruster (HET) systems typically require a power input ranging from 3 kW to 5 kW. These thrusters are commonly chosen for medium-segment communication satellites. The PPS5000 thruster, designed for this scale, is a frequent choice for European-origin satellite platforms. Its typical operation at 10A and 300V, a standard qualified operational mode, is considered. The magnet and cathode power inputs of the thruster are critical for optimizing its performance, and these values have been determined.

Based on the specified inputs for the satellite and propulsion system and considering a continuous firing scenario lasting 1 day (84,600 seconds), the results of the maneuver have been calculated. This maneuver represents a segment of the 6-month electric orbit-raising phase typically undertaken by satellites after separation from the launcher to reach their final orbit. Within this scenario, the mathematical model outputs include a thruster thrust value of 191.4 mN and a specific impulse (I_{sp}) value of 1669 s.

To validate a simulation model, or parts thereof, it is crucial to utilize data from real-system experimental tests that accurately represent all components and physical phenomena involved in the model-system comparison. In this context, two reference studies (Tables 7.1 and 7.2) conducted by the manufacturer of the PPS5000 thruster have been consulted. These studies encompass measurements obtained from different test setups for both the Engineering Qualification Model (EQM) and the Qualification Model (QM) during their respective qualification and acceptance tests. The outputs of the mathematical model developed in this study, focusing on the Hall Effect Thruster (HET), have been compared with the findings of these reference studies, and the results have been presented.

The mathematical model developed in this study has demonstrated consistent and acceptable results compared to the reference sources. The model shows minor differences in thrust levels (0.8%) and specific impulse (I_{sp}) outputs (0.5%), validating with error rates of less than 1%. It is important to note that these variations are not errors within the model but rather natural fluctuations inherent to Hall Effect Thruster (HET) physics and the complex plasma environment. Factors such as production line errors or test conditions during each firing and setup can influence thruster performance, leading to these variations.

This study has developed and validated a mathematical model for an HET-based electric propulsion system (EPS) employed in satellites, an area with limited research in existing literature. This contribution is anticipated to make a substantial impact in the field.

Recommendations for future work:

- The HET mathematical model developed in this study, along with the thruster parameters it generates, can serve as a propulsion system model within more comprehensive satellite models in simulation environments like GMAT or STK. These environments allow for the integration of the propulsion model with the full mass properties of the satellite, enabling observation of changes in orbital Keplerian parameters after propulsion firings using a propagator. Furthermore, the model can simulate torque production necessary to neutralize angular velocities of reaction wheels upon reaching saturation, as well as model tasks such as tumbling damping using the propulsion system.

- A HET thruster simulator or emulator can be developed using the mathematical model and outputs generated in this study. This simulator can be used to test the propulsion systems of future satellite platforms under development. Alternatively, it can be utilized to perform end-to-end validation of driver units or PPU (Power Processing Unit) equipment developed in satellite projects.

It is anticipated that the information presented in this thesis will establish the foundation for electric propulsion systems, which are likely to be favored for future indigenous and national satellites planned for production in Türkiye. Additionally, it will illuminate the assessment of propulsion system performances.



REFERENCES

- [1] Sanchez M. M., (1998), "Spacecraft Electric Propulsion - An Overview", Journal of Propulsion And Power, Vol. 14.
- [2] Sutton G. P., (2001), "Rocket Propulsion Elements Seventh Edition".
- [3] Kozubsky K.N., Koryakin A.I., Murashko V.M., (2012), "History of space stationary plasma thruster's and their application in Russia, USA, and Europe. New challenges for stationary plasma thruster. On the 40-th anniversary of the first stationary plasma thruster space tests," Moscow Aviation Institute.
- [4] Wiley J. L., James R. W., (1995), "Space Mission Analysis and Design", Chapter 17: Spacecraft Propulsion Systems.
- [5] Wilbur P. J., Rawlin V. K., Beattie J. R., (1998), "Ion Thruster Development Trends and Status in the United States", Journal of Propulsion and Power, Vol. 14.
- [6] Jahn R. G., Choueiri E. Y., (2002), "Electric Propulsion in Encyclopedia of Physical Science and Technology", Third edition, Academic Press, Princeton University.
- [7] Billings L., (2009), "A Rocket for the 21st Century".
- [8] Lev, D. et al., (2017), "The Technological and Commercial Expansion of Electric Propulsion in the Past 24 Years", 35th International Electric Propulsion Conference, IEPC-2017-242.
- [9] Hoskins W. A. et al., (2013), "30 Years of Electric Propulsion Flight Experience at Aerojet Rocketdyne", IEPC-2013-439, 33rd International Electric Propulsion Conference.
- [10] Gusev Y. G., Pilnikov A. V., (2012), "The Electric Propulsion Role and Place within the Russian Space Program", Moscow Aviation Institute.
- [11] Web 2, (2024) https://tr.espacenet.com/searchResults?submitted=true&locale=tr_TR&DB=EPODOC&ST=advanced&TI=Hall+Effect+Thruster&AB=&PN=&AP=&PR=&PD=&PA=&IN=&CPC=&IC, (Access Date: 25/04/2024)
- [12] Web 1, (2024) https://en.wikipedia.org/wiki/List_of_spacecraft_with_electric_propulsion, (Access Date: 15/04/2024)
- [13] Gonzalez J. A., Di Cara D. M., (2022), "European Space Agency Electric Propulsion Activities", International Electric Propulsion Conference 2022.
- [14] Web 3, (2024) <https://satsearch.co/products/categories/satellite/orbit-determination-and-control/thruster?page=1>, (Access Date: 20/04/2024)

- [15] Space Power Inc., Keldysh Research Center, (1998), “Hall Effect Plasma Thrusters, Internationally Developed Technology: High Efficiency, High Specific Impulse System”.
- [16] Polk J.E., et al., (1999), “Validation of the NSTAR Ion Propulsion System on the Deep Space One Mission: Overview and Initial Results”, 35th Joint Propulsion Conference, AIAA 99-2274.
- [17] Hans. L. J., (2001), “Development of the Radio Frequency Ion Thruster RIT XT – A Status Report”, Germany.
- [18] Michael M., (2010) “Two-Dimensional Numerical Modeling of Radio-Frequency Ion Engine Discharge”, MIT.
- [19] Jahn R. G., (1968), “Physics of Electric Propulsion”, McGraw-Hill, 219-227.
- [20] Pisacane V., (2000), "Electric Propulsion: Which One For My Spacecraft?", Whiting School of Engineering.
- [21] Joseph C. R., Andrew H. W., (2000), “A Micro Pulsed Plasma Thruster (PPT) for the “Dawgstar” Spacecraft”, Primex Aerospace Company and University of Washington.
- [22] Mercedes R., Gómez V., Fajardo P., Navarro J., Albertoni R., et al., (2020), “HIPATIA: A project for the development of the Helicon Plasma Thruster and its associated technologies to intermediate-high TRLs”, 71st International Astronautical Congress (IAC).
- [23] Tim G. W., (2010), “Principal VASIMR Results and Present Objectives”, Space Technology and Applications International Forum.
- [24] Kybeom K., (2010), “A Novel Numerical Analysis of Hall Effect Thruster And Its Application In Simultaneous Design of Thruster And Optimal Low-Thrust Trajectory”, Georgia Institute of Technology.
- [25] Garner C.E., et al., (1994), “Cyclic Endurance Test of a SPT-100 Stationary Plasma Thruster”, 30th Joint Propulsion Conference, AIAA 94-2856.
- [26] Day M., Rogers W., Maslinekov N., (1994), “SPT-100 Subsystem Development Status and Plan”, 30th Joint Propulsion Conference, AIAA 94-2853.
- [27] Dubois L., Gaboriau F., Liard L., Harribey D., Hénaux C., et al., (2018), “a new double stage Hall thruster design”, Physics of Plasmas, 25 (9).
- [28] Clauss C., (1995), “High-Power Stationary Plasma Thruster Propulsion Systems”, Atlantic Research Corporation Memorandum.
- [29] Opall-Rome B., (2011), “The Prius of Spy Satellites Rafael's Ionic Thrusters May Extend Service in Low Earth Orbit”.

- [30] Eshel D., (2012), “Rafael Develops New Micro-light Satellite Concept”, Aviation Week and Space Technology.
- [31] Web 4, (2024), <https://www.itu.int/en/ITU-R/space/workshops/2019-SatSymp/Presentations/210%20Satellite%20Innovation%20INVAP-GSATCOM.pdf>, (Access Date: 20/04/2024).
- [32] Web 5, (2024), https://www.tusas.com/content/files/uploads/450/SmallGEO_low.pdf, (Access Date: 20/04/2024).
- [33] Gulczinski F.S., Spores R.A, (1996), “Analysis of Hall-Effect Thrusters and Ion Engines for Orbit Transfer Missions”, 32nd Joint Propulsion Conference, AIAA 96-2973.
- [34] Gulczinski F.S., (1999), “Examination of the Structure and Evolution of Ion Energy Properties of a 5 kw Class Laboratory Hall Effect Thruster at Various Operational Conditions", Department of Aerospace Engineering, The University of Michigan.
- [35] Zhurin V.V., Kaufman H.R, and Robinson R.S., (1997), “Physics of Closed Drift Thrusters”, IEPC 97-191, 25th International Electric Propulsion Conference.
- [36] Warner N.Z., (2007), “Theoretical and experimental investigation of Hall Thruster Miniaturization”, Department of Aeronautics and Astronautics, Massachusetts Institute of Technology.
- [37] McDonald M.S., (2012), “Electron Transport in Hall Thrusters", Department of Applied Physics, The University of Michigan.
- [38] Goebel D. M., Katz I., “Fundamentals of Electric Propulsion: Ion and Hall Thruster”, Jet Propulsion Laboratory California Institute of Technology, NASA.
- [39] Snyder J. S., Baldwin J., Frieman J. D., Walker M.R., Hicks N. S., Polzin K.A., Singleton J.T., (2013), "Flow Control and Measurement in Electric Propulsion Systems: Towards an AIAA Reference Standard", IEPC-2013-425.
- [40] Gallimore A.D., (1994), “Preliminary Characterization of a Low Power End-Hall Thruster”, 30th AIAA Joint Propulsion Conference..
- [41] Huismann T. D., (2011), “Improving Hall Thruster Plume Simulation through Refined Characterization of Near-field Plasma Properties”, The University of Michigan.
- [42] Daren Y., Jie L., Yong L., Binhao J., (2009), “Experimental Study on the Effects of Magnetic Field Configuration near the Channel Exit on the Plume Divergence of Hall Thrusters”, Plasma Science and Technology, Volume 11, Number 6, 714.
- [43] Zhurin V., Kaufman H., Robinson, R., “Physics of Closed Drift Thrusters”, Plasma Sources Sci. Technol. 8, R1–R20.

- [44] Valentian D., Bugrova A., Morozov A. “Development Status of the SPT MK II Thruster”.
- [45] Duchemin O., Illand H., Saverdi M., Cesari U., Signori M., Pagnon D., Estublier D., (2006), “Testing of a Thrust Steering Device on the PPS®1350 Hall Thruster”, AIAA 2006-4478 , 42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit.
- [46] Duchemin O., Prioul M., Illand H., Banetta S., Vicini A., Garrigues L., R. R., D. Estublier D., “Development of a Prototype Thrust Steering Device for Hall Effect Thrusters”.
- [47] Hofer R., (2004), “Development and Characterization of High-Efficiency, High-Specific Impulse Xenon Hall Thrusters”, Department of Aerospace Engineering, the University of Michigan.
- [48] Zurbach S., Lasgorceix P., Cornu N., “A 20kW High Power Hall Effect Thruster for Exploration”, IAC-10.C4.4.2.
- [49] Walker Q.E. and Cappelli M. A., Hargus W.A., Schilling J., (2001), “Preliminary Study of Arcjet Neutralization of Hall Thruster Clusters”, IEPC-01-63, Stanford University.
- [50] Henaux C., Vilamot R., Garrigues L., Harribey D., (2011), “A New Flexible Magnetic Circuit for a Hall Effect Thruster”, IEPC-2011-291, The 32nd International Electric Propulsion Conference, Wiesbaden, Germany.
- [51] Duchemin O. et al., (2019), “Qualification Status of the PPS5000 Hall Thruster Unit”, 36th International Electric Propulsion Conference University of Vienna, IEPC-2019-906.
- [52] Duchemin O. et al., (2018), “Development & Qualification Status of the PPS5000 Hall Thruster Unit”, AIAA Propulsion and Energy Forum, 2018 Joint Propulsion Conference.
- [53] Tata M., P-E, Frigot, Beekmans S., Lübberstedt H., Birreck D., Demaire A., Rathsman P., Rezugina E., Kutufa N., (2013), “SGEO Electric Propulsion Subsystem Development Status and Future Opportunities,” 33rd International Electric Propulsion Conference, The George Washington University, Washington, D.C., IEPC-2013-144.
- [54] Marchandise F., Vial V., Öberg M., Cornu N., Barbier D., Arcis N., (2013), “EPS-500 development status,” 33rd International Electric Propulsion Conference, The George Washington University, Washington, D.C., IEPC-2013-063.
- [55] Visee, M. De Jong, Timmerman J., (2013), “Miniaturized HV Power Supply,” 33rd International Electric Propulsion Conference, The George Washington University, IEPC-2013-258.

- [56] Saverdi M., M. Signori M., L. Biagioni L., (2007), "Experimental characterization of the HT-100 Hall thruster in twin engine cluster configuration," 30th International Electric Propulsion Conference, Florence, IEPC-2007-320.
- [57] Kim Y., Kang S., Jeong, Seon Y. J., Wee J., Yoon H., Lee J., Seo M., Choi W., (2009), "Development of Xenon feed system for a 300-W Hall-Thruster", 31st International Electric Propulsion Conference, University of Michigan Ann Arbor, Michigan, IEPC-2009-061.
- [58] Matticari G., (2014), "Major Elements/Units of an Electric Propulsion Sub-System", Laboratorio Ultrasuoni e Controlli Non Distruttivi.
- [59] Szabo J., Robin M., Paintal S., Pote B., Hruby V., Freeman C., (2013), "Iodine Propellant Space Propulsion," 33rd International Electric Propulsion Conference, The George Washington University, IEPC-2013-311.
- [60] Svelto F., Marcuccio S., Matticari G., (2002), "Status and Perspectives of Electric Propulsion in Italy", 53rd International Astronautical Congress the World Space Congress, IAC-02-S.4.04.
- [61] Corey R., Pidgeon D., (2009), "Electric Propulsion at Space Systems/Loral", 31st International Electric Propulsion Conference, University of Michigan Ann Arbor.
- [62] Kim V., (1998), "Main Physical Features and Processes Determining the Performance of Stationary Plasma Thrusters", Journal of Propulsion and Power, Vol. 14, No. 5.
- [63] Mazouffre, S., Gawron, D., Kulaev, V., Sadeghi, N., (2008), "Xe⁺ Ion Transport in the Crossed-Field Discharge of a 5-kW-Class Hall Effect Thruster," Plasma Science, vol.36, no.5, pp.1967-1976.
- [64] Uluşen D., Aydın B. Ç., Gülle İ. S., Bozkurt E., Özkaya H., Ontaç S., Türkmenoğlu M., (2013), "Overview of TUBITAK UZAY's electric propulsion development project (HALE)", 6th International Conference on Recent Advances in Space Technologies (RAST), pp. 643-647, TÜBİTAK Space Technologies Research Institute.
- [65] Aydın B. Ç., Gülle İ. S., Uluşen D., Karadağ B., "Magnetic Circuit Design, Manufacturing and Testing for a 1.5 kW Hall Effect Thruster".
- [66] Uluşen D., Gülle I. S., Aydın B. C., Türkmenoğlu M, Yüncüler Ç., Karadağ B., Yağlıoğlu B., Ontaç S., "Hall Effect Propulsion System Development Activities at TÜBİTAK UZAY".
- [67] Moghimi M. H., (2011), "Physical Features and Parameters Calculations of a Hall Thruster" IEPC-2011-002, 32nd International Electric Propulsion Conference.

[68] Hoskins W. A. et al., (2013), “30 Years of Electric Propulsion Flight Experience at Aerojet Rocketdyne”, IEPC-2013-439, 33rd International Electric Propulsion Conference.

[69] Myers R. M., (1993), “Electromagnetic Propulsion for Spacecraft”, 1993 Aerospace Design Conference.

[70] Choueiri E. Y., Ziemer J. K., (2001), “Quasi-Steady Magnetoplasmadynamic Thruster Performance Database”, Journal of Propulsion and Power 17(5), 967-976.

[71] Gonzalez J. A., Di Cara D. M., (2022), “European Space Agency Electric Propulsion Activities”, International Electric Propulsion Conference 2022.

[72] Web 4, (2024), <https://www.itu.int/en/ITU-R/space/workshops/2019-SatSymp/Presentations/210%20Satellite%20Innovation%20INVAP-GSATCOM.pdf>, (Access Date: 20/04/2024).

[73] Web 5, (2024), https://www.tusas.com/content/files/uploads/450/SmallGEO_low.pdf, (Access Date: 20/04/2024).

[74] Kim V., (2012), “Stationary Plasma Thrusters in Russia: Problems and Perspectives” Moscow Aviation Institute.

BIOGRAPHY

Mete BİLGER began his academic journey in 2014 at the University of Turkish Aeronautical Association (UTAA), where he pursued studies in the Faculty of Aeronautics and Astronautics. In 2019, he graduated as the top-ranking student in the Astronautical Engineering Department. His professional career started in 2018 at Turkish Aerospace Industries (TUSAŞ) as a long-term engineering intern, progressing to the role of Space Propulsion Engineer in 2019. Over the past six years, he has continuously advanced in this role. Notably, he served as an electric propulsion engineer on the Small-Size Communication Satellite Family Development Project, a collaborative effort between Türkiye and Argentina, during which he spent nearly two years at facilities in Argentina. Recently, he has expanded his expertise in avionic system engineering.

