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Design, Optimization and Sizing of a Water Electrolysis Propulsion (WEP) System for CubeSats

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Author: **Varun Chivukula**

Student ID: 10703423
Advisor: Prof. Luciano Galfetti
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Abstract

The present work focused on the upcoming technology of using water electrolysis to produce hydrogen and oxygen gas, which then are combusted in a bipropellant thruster to produce thrust. A comprehensive literature review was undertaken and the need for numerically modelling the Water Electrolysis Propulsion (WEP) system was identified. The process of conceptually designing the system was split mainly into 2 parts – the first was to develop a MATLAB®-Simulink® model of the entire system and the second was to use the Simulink® model for model-based optimization to achieve a 1 N thrust and highest possible specific impulse. The WEP system was mathematically modelled in Simulink® as a pressure-fed system which was tested with a few design choices taken from literature. The model was found to successfully simulate the working of the system. Chamber pressure, mass flow rate of propellant and cone half angle(s) of convergent and divergent section of nozzle were chosen as the design variables to minimize the cost function (a function of thrust and specific impulse). Genetic Algorithm was selected for the optimization and an average thrust of 1 N and a very good specific impulse of 338.9 seconds, with just 0.38 g of propellant per fire, were achieved. Also, the WEP system reported a Total-impulse-to-power ratio (TI/$\mathbf{P}$) of 1.19 N-s/W. The design variables were then used to size the entire system, which fit in a 2U volume, ideal for a 6U or a bigger CubeSat. In the future, this numerical model of the entire system can be used as it is, to conceptually design a WEP system. Modelling of the system could be made more realistic by considering real gas instead of ideal gas and also considering zero-g effects of the outer space.

Key-words: Water Electrolysis, CubeSat Propulsion, Optimization, Sizing

Abstract in italiano

Il presente lavoro si è concentrato sull'imminente tecnologia di utilizzo dell'elettrolisi dell'acqua per produrre idrogeno e ossigeno gassoso, che vengono poi bruciati in un propulsore bipropellente per produrre spinta. È stata intrapresa una revisione completa della letteratura ed è stata identificata la necessità di modellare numericamente il sistema WEP (Water Electrolysis Propulsion). Il processo di progettazione concettuale del sistema è stato suddiviso principalmente in 2 parti: la prima consisteva nello sviluppare un modello MATLAB®-Simulink® dell'intero sistema e la seconda consisteva nell'utilizzare il modello Simulink® per l'ottimizzazione basata su modello per ottenere un 1 N spinta e impulso specifico più alto possibile. Il sistema WEP è stato modellato matematicamente in Simulink® come un sistema alimentato a pressione che è stato testato con alcune scelte progettuali tratte dalla letteratura. Il modello è stato trovato per simulare con successo il funzionamento del sistema. La pressione della camera, la portata massica del propellente e i semiangoli del cono della sezione convergente e divergente dell'ugello sono stati scelti come variabili di progetto per minimizzare la funzione di costo (una funzione della spinta e dell'impulso specifico). L'algoritmo genetico è stato selezionato per l'ottimizzazione e sono stati raggiunti una spinta media di 1 N e un ottimo impulso specifico di 338,9 secondi, con solo 0,38 g di propellente per sparo. Inoltre, il sistema WEP ha riportato un rapporto impulso totale/potenza (TI/**P**) di 1,19 N-s/W. Le variabili di progettazione sono state quindi utilizzate per dimensionare l'intero sistema, che si adatta a un volume di 2U, ideale per un CubeSat da 6U o più grande. In futuro, questo modello numerico dell'intero sistema potrà essere utilizzato così com'è, per progettare concettualmente un sistema WEP. La modellazione del sistema potrebbe essere resa più realistica considerando il gas reale invece del gas ideale e considerando anche gli effetti a gravità zero dello spazio esterno.

Parole chiave: Elettrolisi dell'acqua, Propulsione per CubeSat, ottimizzazione, dimensionamento

Contents

1 Introduction

1.1. Background

Space missions are highly price and weight sensitive. Due to such sensitivity, and recent advancement in miniaturization, the same technical objectives like scientific experiments, reconnaissance, etc. can be performed by small satellites and so the demand for small satellites has drastically increased. One of the satellites in the small satellite category is the CubeSat, a stand-alone or a stack of 10x10x10 cm^3 cube(s). One such cube is called 1U and there are many CubeSats currently in-orbit with sizes ranging from 3U to 12U (see Figure 1). The advantages of having a fixed design of the satellite are a) standardization - easy integration onto a launch platform; (b) ease of off-the-shelf replaceability of its components. These advantages have meant an exponential increase in the number of CubeSat applications (see Figure 2). A satellite designed for a mission in a Low Earth Orbit faces a significant gravity decay and so, a propulsion system is needed to keep the satellite in its orbit after being placed there by a space launcher. In the recent years, the satellites are also expected to perform complicated maneuvers due to the advancement of live tracking and closed-loop guidance. Such maneuvers would need a propulsion system which can be switched on or switched off as per need, is throttleable and can provide the necessary Delta-v (Δv). Also, the propulsion system is needed to work under strict power restrictions (limited by the power generated by the solar panel or the size of battery that can be accommodated) and the propellants need to be safe for handling. The fixed volume of a CubeSat imposes a challenge in the design of a propulsion system in terms of size and capabilities. So, the challenges in designing a propulsion system for a CubeSat is two-fold: size-wise and performance-wise.
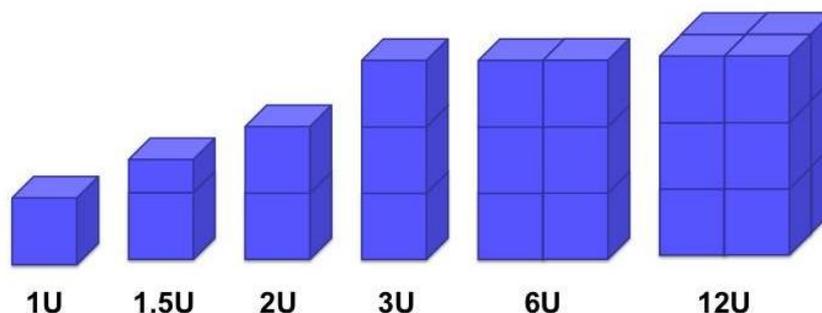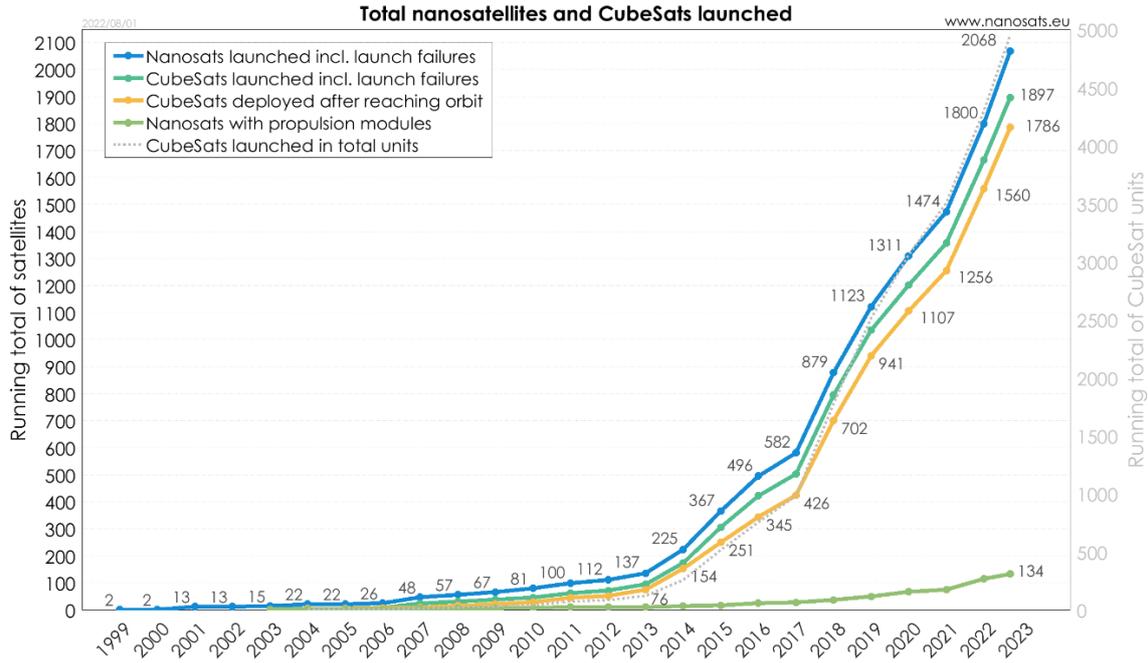
Figure 1: CubeSat configurations [1]

Figure 2: CubeSats launched by year [2]

1.2. Motivation of the thesis

Due to the imminent ban by the European Union (EU) on Hydrazine, used for higher thrust propulsion, due to its toxicity (for handling and for the environment) [3], there is a need to expedite a working technology based on green propellants. In this quest, water is an emerging green alternative due to its non-toxic nature, abundance, easy-to-handle, non-polluting aspects. In order to extract usable energy from water (H_2O), energy needs to be applied. A process of applying electrical energy to water to split it into its constituents, Hydrogen (H_2) and Oxygen (O_2) (usually in gaseous form) is called Electrolysis. Electric current needed for electrolysis can be generated by solar cells after the in-space deployment of solar panel [4]. The common electrolytes used are alkaline, solid-oxide and Proton Exchange Membrane (PEM). PEM is an upcoming technology due to its compatibility with hydrogen and oxygen, purity of H_2 it produces, low power consumption, etc., the perfect attributes for a satellite environment. The gaseous Hydrogen and Oxygen produced then can be used to produce thrust in various configurations such as Bi-propellant thruster, Cold gas thruster, Hall Effect Thruster (HET) [5], etc. Also, when the satellite is in the shadow, the H_2 and O_2 stored in separate pressurized tanks can be used in a Fuel Cell (FC), of the same apparatus as that of the electrolyzer, to produce electricity for the working of the satellite [6]. The Fuel Cell can also be considered as a secondary power source to the solar panels, improving redundancy.

1.3. Objectives

The present work will explore the concept of using water as a propellant (through electrolysis) in a small satellite environment like CubeSat and will be called a Water Electrolysis Propulsion (WEP) System. Although the concept of such a system is not new and can be attributed to be worked on since the early 20th century, a need was felt, for a numerically model the WEP system which can be used to perform a basic performance analysis and sizing. In the present work, a MATLAB® (version R2020b) - Simulink® model was built in order to simulate a real-time pressure-based WEP system. Later, Genetic Algorithm (GA) was used to optimize the system for 1N thrust and highest possible specific impulse. The performance data, specifically the pressure requirements were noted, and were used to size the entire system consisting of a water tank, electrolyzer, gas tanks and a thruster.

1.4. Plan of presentation

In order to achieve the aforementioned objectives, the following steps were followed:

- A thorough literature review of existing and developing WEP systems, sub-technologies such as types of electrolytes, etc. was carried out to understand and to define the technical requirements for a working numerical model.

- A Simulink® model was created to simulate the real time working of a pressure-fed WEP system consisting of a water tank, electrolyzer, pressure-based check valves, gas tanks, pressure-based trigger system and a thruster.

- An objective function was defined to obtain an initial technical requirement of 1N thrust and to maximize the specific impulse (for such a thrust).

- A Genetic Algorithm (GA) based optimization approach was implemented in a MATLAB® code which performed a model-based optimization using the developed Simulink® model to give the optimum performance parameters (such as Total impulse, Specific impulse, Thrust, etc.) and the optimum design variables (such as Chamber pressure, etc.)

- The design variables were then used to size the various components like the thruster, the gas tanks, the electrolyzer and the water tank to be easily accommodated in a CubeSat application.

2 Literature Review

2.1 General overview on water propulsion

Water Propulsion System (WPS) is a system, where the thrust demanded of it, is achieved by using water as the propellant. Water is a non-toxic, dense, easy-to-handle, stable, and can be found at different places in the solar system [7]. If found, refuelling is possible through a technique called the "In-Situ Resource Utilization (ISRU)" which can enhance the life-span of a mission. Different thrusters such as resistojets [8], plasma [9], thermal propulsion (steam) [10] can be used, replacing liquid hydrogen with water. A study [11] was carried out by considering 95 different propellants for a micro-resistojet and water emerged as the best alternative. In 2020, there were 2 flight tests of a WPS and a few issues were encountered, one of which was that water in its liquid form was expelled from the nozzle and an ice plug was formed as a result. The author recommends further research and recommends using a vaporization chamber as an alternative [12]. [3] performed an individual component analysis of a WPS and their impact on various missions. By studying the impact, an optimum sizing was performed for specific mission requirements. For missions in the Low Earth Orbit (LEO) which require deorbiting and station keeping, WPS must be chosen due to high specific impulse compared to electric propulsion. An innovative synergetic utilization of a WPS was suggested where 50 cells of an electrolyzer type would be providing oxygen for 6 crew on a 500-day mission; and the hydrogen produced amounting to around 315 kg could be utilized by an auxiliary propulsion device. Another innovative utilization suggested is to store the excess oxygen and hydrogen produced, to be used in a Fuel Cell when the satellite is in dark (out-of-sunlight condition) [13].

In recent times, the focus is on Electrolysis, a process of splitting water (H_2O) into Hydrogen (H_2) and Oxygen (O_2) using the electric current drawn from the satellite solar panel setup. Higher the current drawn, higher the rate of production of gases but power efficiency reduces, and so, the current drawn from the solar panels has to be managed accordingly [3]. Whenever thrust is needed, a bi-propellant thruster combusts H_2 and O_2 together, initially stored in their respective tanks. It fares on-par with the aforementioned thrusters in terms of Delta-v (Δv) and thrust. In a particular case, for a 1N thruster, a very high specific impulse of 350 seconds compared to conventional thrusters was achieved [14]. Such a propulsion system using electrolysis of water is referred to as Water Electrolysis Propulsion (WEP). As compared to micro-thrusters like ion engines, WEP performs better in terms of thrust per unit power, due

to the higher efficiency of electrolysis. Also, due to the high density of water, it allows for storing higher chemical energy in the same volume as compared to storing the propellant as gaseous hydrogen or oxygen, as they occupy more space for the same amount of chemical energy. The gaseous hydrogen and oxygen obtained on-demand by the process of electrolysis can be used by a bi-propellant thruster for high thrust requirements or by Hall Effect thruster for high specific impulse requirement. Such an application and versatility give greater mass budget for a small satellite mission [4]. [6] quantified the advantage given by WEP system at same thrust as power requirement of WEP is around 0.17 N/kW as compared to 0.08 N/kW for 2.2 kW arc-jets and 0.03 N/kW for 2.6 kW ion thrusters. This advantage is increased further at lower power where electric propulsion has lower efficiency. For a mission requiring higher number of burns by the thruster, WEP system ensures higher weight savings.

2.2   Tanks

Pressurized tanks for space applications have been a subject of research for many years. In the recent years, due to the demand of tanks for CubeSat propulsion systems, the focus has specifically shifted towards miniaturizing the tankage that ensure the same characteristics as the larger versions.

[15] commented about a compressed hydrogen tank in which a low-density hydrogen can be compressed and stored. As pressure increases the density of hydrogen gas, the storage pressures are anywhere between 35 MPa and 70 MPa. Very high pressure has its advantages as there is very high rate of emptying and compactness in terms of size, although filling up of the tank is difficult. Also, the increasing pressure does not increase the volumetric density. Around 2018, a pressurized compressed gas tank was built at Naval Research Laboratory (NRL) under the Ion Tiger program, using a carbon overwrapped aluminium which weighed 3.62 kg and stored 500 g of H_2 gas at 34 MPa.

[12] reported a tank design for storing 400 g of propellant. As the propellant was filled in the tank, it entered a rubber bladder inside the tank. And whenever the propellant was needed, the force of the bladder ejected the propellant out of the tank into the Vaporization chamber (combustion chamber) of the AQUARIUS, a Resistojet propulsion system.

[14] researched for a new water storage tank specifically to reduce mass and complexity of the diaphragm tanks. It was enquired if surface tension phenomena could be used. Also, the idea of integrating devices of propellant management directly within the tank, without post processing, was explored using additive layer manufacturing, which could also reduce the costs. To validate the manufacturing procedure, a raw unprocessed aluminium specimen was manufactured, also for its surface tension properties, by specific laser melting. It was found that without post processing, the 3D printed material showed higher surface tension and water adhesion, where contact angle of demineralized water was below 70° while typical

angle for aluminium lies between 70° and 90°. Preferably the contact angle of around 8° or lower achieved by hydrazine, which acts similar to water, was the target and so, the use of acids is suggested to reduce the contact angle, as they increase the roughness of the surface. Then, a new propulsion system based on water electrolysis, involved storing of the H_2 and O_2 gases produced by the electrolyzer at around 50 bar pressure in the respective tanks. For compactness, the storage tanks were integrated to the electrolyzer. The total volume of the propulsion system was around 1U of a CubeSat and volume of H_2 and O_2 gas tanks were reported to be 150 cm^3 and 75 cm^3 respectively. To ensure the same pressure of H_2 and O_2 entering into the thruster, a volume ratio of 2:1 of H_2 and O_2 is maintained, something that was used in the present work as well. Such high pressure ensured high thruster impulse and so, a lengthy time interval before the electrolyzer had to refill the gas tanks. Back flow from the thruster is avoided as the H_2 and O_2 gases are stored and fed separately.

[6] suggested that in a WEP system, water for the mission can be stored in thin-walled, light-weight tanks at ambient pressure as the vapor-feed electrolyzer itself pressurizes the propellant. A comparison was made between a mission which requires a single large Delta-v (Δv) burn and a mission requiring multiple burns. The mission needing a single burn led to a higher tank mass to store the required H_2 compared to the mission needing multiple burns. The propellant tanks were sized for largest burn which also can be used to accommodate multiple burns while the gas tank got refilled during the mission. A Performance-factor involving the burst pressure, internal volume and tank weight was suggested to be the figure of merit to design a light weight pressurized gas tank. A very high Performance-factor was found to be achievable as a thin bladder-liners overwrapped with T1000 carbon fiber composite was used. It was concluded that high performance factors were justified for smallest of tanks as the result was a significantly reduced weight as compared to conventional tankage.

Like [6], [3] and [16] also used the same 2:1 volume ratio of H_2 and O_2 gas since twice as much H_2 gas is produced than O_2 gas for every mol of water electrolyzed. So, same pressure of gases was ensured to the thruster, which ensured uniform combustion. [16] used 3D-printed titanium tanks for storing 150 cm^3 of H_2 gas and 75 cm^3 of O_2 gas.

[17] reports that sloshing of water weighing around 1 kg in the tank was found to be negligible for the entire mission.

2.3   Electrolyzer

Electrolysis of water occurs by the spontaneous redox reactions when the electrolyzer is supplied with electricity. A catalyst is needed as water is a poor conductor of electricity and a membrane is needed to separate the two electrodes.

In an alkaline electrolysis, the process occurs due to the transport of hydroxide ions (OH⁻) through the electrolyte from cathode to anode. Hydrogen is generated only on the cathode side. Liquid alkaline solution of sodium or potassium hydroxide are commonly used as the electrolytes. Until recently, solid alkaline exchange membranes as electrolytes are finding greater usage and traction but are not useful enough for industries. The usage is limited to laboratory scale. Solid oxide electrolysis uses solid ceramic materials as electrolytes leading to negatively changed oxygen ions (O2⁻) at high temperatures of around 700°- 800°C, making such a system inapt for the present work. In recent times, a lot of focus is being put to integrate compression of gases directly into the electrolyzer so as to save the complexity and cost of having an external compressor to compress the gases for storage [18].

Alkaline and steam electrolysis were used but recently the focus has shifted to Proton Exchange Membrane (PEM) electrolyzer because of high purity of H_2 gas at cathode, low power consumption, high safety and handling ease. It was first developed for the Gemini Space program by General Electric in 1960. PEM is a solid electrolyte and is simple in construction. Water is introduced at the anode (+) and is decomposed into protons and molecular oxygen. The molecular oxygen is evacuated by water circulation. Electrons follow the path of the current and protons migrate to the cathode (-) through the PEM under the effect of electric field and reduced to molecular hydrogen. The temperature of operation varies between 20°C and 100°C. There are some disadvantages like the high cost of components and low durability [19]. Figure 3 shows the schematic of the PEM and the associated reactions.
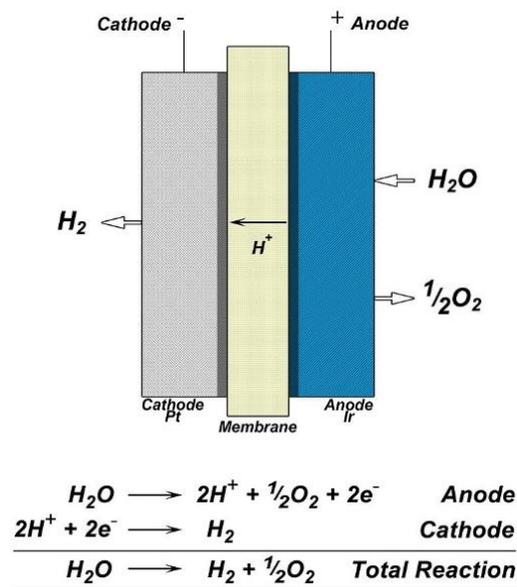

Figure 3: PEM Electrolysis [20]

The choice of materials [21] is based on high ionic conductivity; oxidative stability; surface area; mechanical, thermal and chemical stability; low permeability of gases and electrically insular. The followed materials are used for each of the components.

Anode (+): Iridium oxide, Iridium-tin oxide, Rhodium, Rhodium oxide

Cathode (-): Platinum, Platinum/Activated carbon

Electrolyte: PEM made of Nafion™ and Flemion™

Catalyst: Titanium-coated Iridium oxide, Molybdenum oxide with graphite, Tungsten trioxide nanorods, Palladium-Carbon nanotubes

[14] considered the usage of the electrolyzer in the zero-g environment of space and suggested feeding the water in vapor form to the cathode instead of anode as mentioned above. An electrochemical hydrogen pump was used to prevent backflow as the modern electrolyzers due to osmotic pressure are pumping out gases at higher pressure than inlet water. The pump using low voltage pumps any diffused hydrogen from water side back to the hydrogen side, preventing accumulation of hydrogen on the feeding membrane. If accumulated, the hydrogen masks the feeding membrane which would prevent evaporation of water, and so resulting in stopping of electrolysis.

[6], a NASA technical memorandum of 1997, reported that electrolyzers were able to electrochemically pump H_2 and O_2 gases from the electrolysis process from ambient pressure to pressures of at least 20 MPa. The electrolyzer framework was based on the Hamilton Standards' Solid Polymer Electrolyte (SPE) technology which used sulfonic acid PEM as the only electrolyte. A design perspective was also highlighted that the electrolyzer can be sized either based on available power or based on mission requirements. If based on former, the mission would be performed with the output propellant generation rate and if latter, the mission would define the required propellant generation rate and so, the power needed by the electrolyzer. Increase of input power to the electrolyzer led to increase in the propellant generation rate. Figure 4 showcases the average power required as a function of oxygen generation rate, which is approximately linear in nature, explained by the Faraday Law of Electrolysis, discussed later. In general, although higher current leads to increased gas generation rate, power efficiency drops and degradation of the inner materials increase. Sizing of an electrolyzer can be achieved through increasing the electrolysis area, which was used in the present work, or by stacking multiple cells [3].
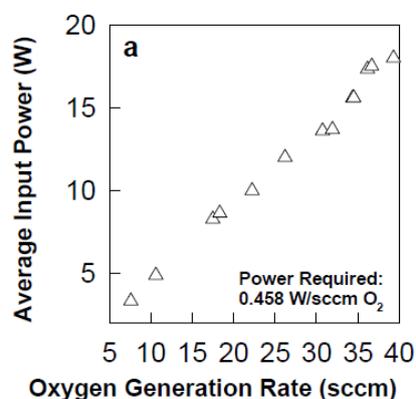

Figure 4: Power vs oxygen generation rate [6]

According to [22]'s experimental findings, electrolyzer operated at higher efficiency for higher temperatures. Figure 5 shows that higher the current drawn for the same dissociation voltage, higher was the gas production efficiency and closer it was to the ideal gas production rate.
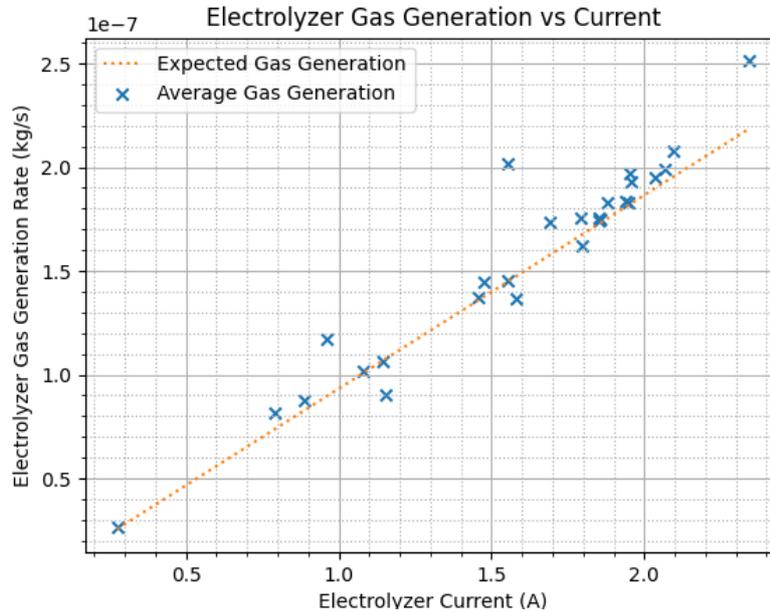

Figure 5: Electrolyzer gas generation rate vs Current [22]

[13] incorporated the multi-cell design approach for the electrolyzer sizing. The electrolyzer was designed as a redundant multiple cell device, which has multiple cells leading to higher gas generation rate with increased power consumption. Pressure of over 50 bar was recorded allowing for longer burn duration and a greater number of firings before gas tank had to be refilled. It is reported that electrolyzer had reached a Technological Readiness Level (TRL) of 6 during vacuum tests and in antagonistic gravity environment.

[17] dealt with the problem of zero-g environment on the physics of the system. Due to the zero-g environment, direction of the water flow is not fixed and so, the system was made to rotate to allow the water to contact the electrocatalyst. In the particular design, the electrodes were placed on the external edges of the electrolyzer, for the water to be driven onto the electrodes by centrifugal force due to rotation. At rotation rate of 1-20 RPM, sufficient centrifugal force was generated to allow for the electrolysis.

Similarly, [23] reported a spin rate of 2 rad/s for freeing up the gas from the electrolysis device. Additionally, such a rotation helped maintain the posture stability, momentum rigidity and eliminated errors in the thrust direction caused by mechanical misalignment. PEM electrolyzer was highly recommended due to its small size, high theoretical efficiency of 94% and high current efficiency of at least 90%. Existing commercial ultra-small PEM electrolyzer cells were used to calculate the efficiency (which came out to be 71-76%) and to predict the mission performance. The electrolyzer used had a contact area of 4 cm x 4 cm and the H_2 gas produced at the

anode was stored in a 294.5 mL stainless-steel chamber. When the electrolyzer was provided power between 3.955 – 9.7 W, the H_2 production rate was reported to be 0.02316 – 0.05302 mg/s. It was assumed that the satellite was able to generate solar power and perform electrolysis for 45 minutes in a 90-minute orbital cycle. It was found that on an average, 1.5 g of water was electrolyzed per orbital cycle with 75% current efficiency. With 76% system efficiency, 12.5 W of power is consumed in 45 minutes to decompose 1.8 g of water. When a power of 10 W was assumed in the conceptual design phase, it was concluded that water electrolysis was feasible and reasonable. Metal 3D printing was used to manufacture water tank, gas tanks and the thruster of Ti-6Al-4V material. The satellite was assumed to be of Al 6061-T6 aluminum alloy. The PEM electrolyzer was a commercially available Horizon PEM electrolyzer. The estimated weight of the propulsion system was approximated to be 2.3 kg [17].

Horizon Educational PEM Blue Electrolyzer (See Figure 6) is a commercially available PEM Electrolyzer with the following technical specifications:

- Dimensions (w x h x d): 5.4 cm x 5.4 cm x 1.7 cm
- Total weight: 69.7 g
- Input voltage: 1.8 V ~ 3 V (D.C.)
- Input current: 0.7 A
- Hydrogen production rate: 7 mL per minute at 1 A current
- Oxygen production rate: 3.5 mL per minute at 1 A current

The aforementioned specifications are perfectly suited for a CubeSat application due to the limitations in terms of power generation by the solar panels, the size constraints and the performance requirements. Another advantage is the above product is available for a price of around 54$ per unit making it affordable and usable for student projects [24].

Figure 6: Horizon Education PEM Electrolyzer [24]

2.4   Thruster

The propulsion mechanism in WEP involves an igniter, valves (for propellant introduction into combustion chamber and preventing reverse flow), a combustion chamber and a nozzle. A valve opens up the flow of H_2 and O_2 gases into the combustion chamber. [7] used catalytic igniters for WEP, with the prospect of laser ignition for future usage. Laser ignition can be useful in igniting the thruster multiple times without need of fuel-rich pilot flames. Such laser ignition could be placed in several thrust chambers using fiber optics while sharing one common source of laser. The valve downstream of the combustion chamber prevents the gases to escape whilst waiting for ignition. The gases are then ignited and the hot gases escape through the nozzle producing thrust. Compared to electric propulsion, required power per thrust (W/N) was similar because of electric power needed to start the electrolysis. However, the thrust was significantly greater than that of electric propulsion. For example, an electrolysis propulsion thruster requires 4.5 kW per N at 80% efficiency whereas NSTAR ion thruster which requires 23 kW per N. This is beneficial at very low-power scales, where ion thrusters do not scale down as efficiently [6]. Compared to conventional propulsion, long maneuvers may not be possible albeit a continuous electrical power for electrolysis and continuous supply of H_2 and O_2 gases at high pressure may allow long maneuvers needing continuous firing [7]. Also, there are a few challenges. As propellants are extracted from water, the only mixture ratio is stoichiometric. This leads to very high combustion temperature (see Figure 7) and non-ideal specific impulse as exhaust velocity is governed by temperature and molar mass. For this reason, a dedicated cooling mechanism (like film cooling) will be required for optimum performance [7]. Theoretically, number of molecules of Hydrogen will be twice as that of Oxygen and will be needed to be considered for tank design. One way of using excess oxygen is by incorporating cold gas thrusters, often used as secondary thrusters for vectoring and small maneuvers. Also, the gases produced might be moist, which when introduced in the piping might freeze, blocking the pipes [14]. The installation of propellant dryers based on a desiccant bed is a simple solution [6].
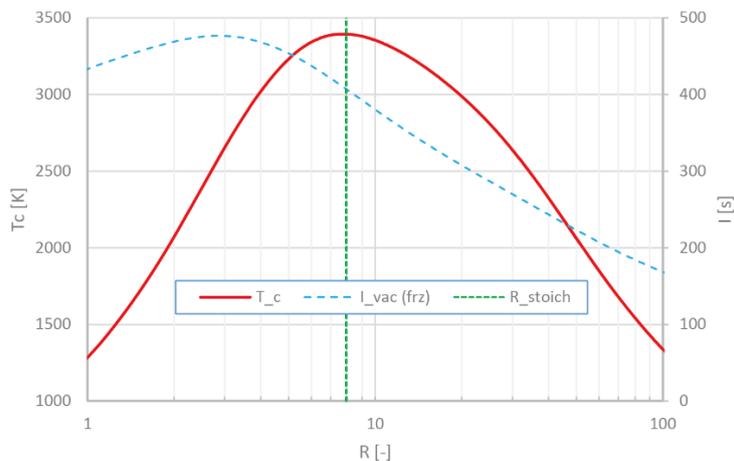

Figure 7: Combustion temperature vs Mixture ratio [7]

[14] proposed a propulsion system which used one or several 1 N thrusters. Stoichiometric mixture ratio was suggested for highest performance; however, it was noted that launcher bipropellant thrusters were often running on hydrogen-rich mixture ratio for higher specific impulse. In such a situation, the excess oxygen has to be released through a cold gas thruster or the main thruster itself. It has very low specific impulse (under 100 seconds). Due to stoichiometric mixture ratio, combustion temperatures over 3000 K were expected and it was suggested to focus on the cooling of the thruster in such a case. The system was fired and thrust was plotted against time. Initially, large fluctuations were noted and later the thrust settled around 0.45 N for burn time of around 25 seconds. It was concluded that the low thrust was due to a short combustion chamber which prevented proper mixing of the propellants and due to nozzle manufacturing by drilling. An updated version was later tested with hydrogen film cooled longer combustion chamber with 30° conical nozzle. It was recorded that maximum thrust achieved by this version was around 1.2 N while the constant thrust was around 0.65 N. The cooling led to very low temperatures of 140°C compared to 3000 K of the earlier version without cooling. To improve the design even further, it was recommended to increase the expansion ratio, the feeding pressure and better manufacturing of the throat.

[6] performed an experimental testing using a 1 N thruster working on H_2 and O_2 gas. It consisted of an ignitor, an injector, a chamber, a throat and a 23.3:1 area ratio nozzle. Earlier, it was reported that thrusters ranging from 0.5 to 22 N thrust were tested. The 0.5 N thruster demonstrated over 150,000 firings and total burn time of 10 hours with a specific impulse of 331 seconds. The tests reportedly showed that a high chamber pressure of more than 160 kPa was needed for optimal ignition leading the entire system to operate at a higher pressure. Materials like iridium-coated rhenium (Ir/Re) were suggested to be used to build the thruster, which give increased tolerance of 700 K temperature, for highest possible performance in the highly oxidizing combustion environment. Using such materials rendered film cooling unnecessary and resulted in higher performance. For the X-33 thruster, spark ignition had been used with emerging alternatives like laser, resonance and catalytic igniters. Duration of a typical experimental test was limited to 3-4 seconds, which was limited by the maximum pressure allowed by the electrolysis system and the volume of tanks. Once the chamber pressure dropped below a pre-defined value, the test was terminated. Such a pre-set cut-off pressure is defined in the present work.

In [25], a small spark plug driven by a capacitive ignition circuit was used for ignition of gases which were supplied to the combustion chamber when the H_2 and O_2 gases reached an imposed pressure in their respective feed lines through a solenoid valve. A commonly known nozzle called de Laval nozzle is ineffective in the vacuum environment as it is unable to expand the exhausted gases and so, conical nozzles with optimal expansion need to be designed based on the mission requirements.

[26] proposed a novel swirling combustion chamber for a NanoSat application. Dimensions of the combustion chamber was reported to be 6 mm x 6 mm (D x H); ducts had a diameter of 0.5 mm; the micronozzle had an expansion ratio of 49. Due to the position of inlet ducts, the O_2 gas entered tangentially to the cylindrical wall whereas H_2 gas was perpendicular. Assuming inlet pressure of 3 atm, mass flow rate of O_2 and H_2 gas of 0.008 g/s and 0.001 g/s respectively and a thrust of 0.03 N, exit velocity of 3300 m/s and specific impulse of around 350 seconds were reported. Due to the relative position of the inlet gas ducts, the gases take a helical path, there was presence of vortices and recirculation bubbles, which led to increase in the residence time to allow for complete combustion in such a small device. If 500 g of propellant is provided to the above device and a lower specific impulse of 300 seconds is considered, a Delta-v (Δv) of more than 400 m/s could be achieved. Such a Delta-v (Δv) allows for extension of satellite lifetime and also allows for forming constellations with other satellites at 200 – 300 km altitude from a 700 km deployment.

The case was made by [17] for using the thruster in a pulsed mode instead of continuous mode in order to decrease the thermal load caused by the stoichiometric combustion of H_2 and O_2 gases and to increase the lifespan. Various methods of ignition, like spark plugs, were discussed but catalytic igniter was chosen because it has a simple structure and there is no need for extra devices. This helped to achieve a compact and light weight design. The catalyst was created as an annular shape with an outer diameter of 19 mm and a 2 mm thickness to allow installation in a tight setup. H_2 and O_2 gases were supplied at 5 bar through separate feed lines and the reaction surface area could be varied by changing the height ranging from 8 mm to 32 mm. This premixed gas was fed to the catalyst bed which was fixed inside the igniter without preheating. It was reported that the conceptually designed system that operated in pulsed mode of 1 second, could electrolyze 1.8 g of water in every LEO period. It was earlier demonstrated that 1.5 g of water electrolyzed could extend the lifespan by 6 months at 250 km altitude and more than 3.5 years at 350 km altitude. The chamber pressure was fixed at 6 bar, the response time and lost pulse mode were ignored and a steady-state thrust was assumed. As the design was intended for a CubeSat, a nozzle throat diameter of 1 mm was fixed for ease of manufacturing, sufficient for a low target thrust. This led to a pressure of 3.41 bar at the throat, a temperature of 3027 K and flow rate through the throat was found to be 0.24 g/s. The characteristic length, defined as the ratio of combustion chamber volume and throat area, was fixed as 0.41 m and so, the length and diameter were found to be 1.4 cm and 1.45 cm respectively. The nozzle outlet diameter was found to be 1.4 cm and the expansion ratio was around 200. The thruster was meant for a 1 N thrust and a specific impulse of 430 seconds was achieved, twice as that of a hydrazine thruster.

For reducing the complexity of the design, [23] suggested that the gases could be produced by the electrolyzer on-demand and could be premixed and ignited at relatively low pressures, saving the requirement of a propellant tank. The premixed

propellant in gaseous state meant no need for an atomizer or an injector. Such a premixed propellant also meant higher overall efficiency as the combustion occurs as a mixture of gases and not as macroscopic droplets of liquid propellant. Once the combustion process is commanded, the gases are ignited by a spark, combusted and expanded through a small nozzle that is oriented along the spin axis, producing thrust. It was reported that in a 90-minute LEO, 1.5 g of water was combusted each time.

Contrarily, [16] suggested that the gases be injected without premixing to protect the platinum-doped alumina catalyst. The catalyst is only fed with hydrogen and subsequently, mixing and ignition occur directly in the combustion chamber.

2.5 Current trends and developmental status

HYDROS® propulsion system on NASA's Pathfinder Technology Demonstration (PTD)-1 mission is the first flight demonstration of a water electrolysis thruster which was launch in 2021. HYDROS electrolyzes liquid water into gaseous hydrogen and oxygen, which are then combusted in a bipropellant rocket nozzle. The prototype unit was called the HYDROS-C which was designed for a 2U volume of a CubeSat with an approximate size of 2U x 1U in the orthogonal direction to the thrust vector and 1U height aligned to the thrust vector. The water was stored in 2 identical tanks in a saddlebag configuration, as shown in the Figure 8, to accommodate the electrolyzer, gas management, thruster and avionics in 1U. The water is drawn from the water tank into the electrolyzer and current is provided to produce H_2 and O_2 gases until a target pressure is reached in their respective tanks. The operation was cyclic, in that, firstly gases were generated followed by combustion and expansion in the nozzle, and so the prototype acted like a bipropellant chemical pulsed-thruster. The combustion had a blowdown pressure profile to a chosen pressure floor and the pressures were configured as per design requirements. The results showed that the system delivered a relatively high thrust compared to electric propulsion and high specific impulse compared to a typical monopropellant thruster [22].
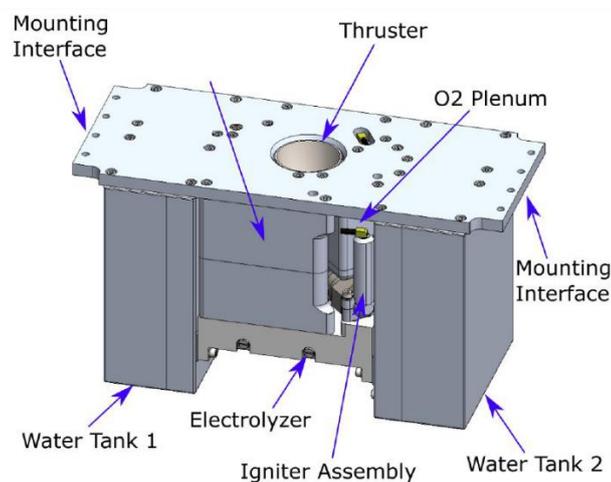

Figure 8: HYDROS-C [22]

[5] reported a Hall Effect Thruster (HET) that operated in the power range of 492 W to 2880 W and mass flow rate of O_2 ranging from 0.985 mg/s to 1.85 mg/s. The maximum thrust recorded was 38.63 mN which resulted in the specific impulse of around 4112 seconds.

In 2019, The "AQua Thruster-Demonstrator (AQT-D)" demonstrating the water resistojet propulsion system (propellant mass ~ 400 g was launched to rendezvous with the International Space Station (ISS) making it the world's first ISS-deployed CubeSat equipped with a WPS. The "AQUA ResIstojet propUlsion System (AQUARIUS-1U)" had one Delta-V-Thruster (F: 4 mN) for orbital maneuver and four Reaction-Control-Thrusters (F: 1 mN) for reaction control [12].

In Europe, ArianeGroup and Institute of Space Systems, University of Stuttgart are working towards proving the electrolyzer technology in order to flight-test the WEP with a 1 N thruster [14] and application of catalytic igniters, although analysis of characteristics of catalytic ignition according to supply conditions has not been performed [17].

[17] performed a study where a WEP system was used in a CubeSat operating in the LEO performing Earth observation and reconnaissance mission at an altitude of 300 km.

[17] also reported many examples of WPS developed in recent times, due to the abundance, feasibility and safety of water as a propellant. One way of using water is to initially heat it by electricity and to directly spray it using the thrusters, a technique demonstrated in thrusters developed by Purdue University, Tokyo University's AQUARIUS and Surrey Satellite Technology's water alcohol resistojet propulsion system. Another way of using water is through the water pulsed plasma thrusters, that accelerate the ionized water through electromagnetic force and produce thrust. Also, there are nuclear thermal propulsion systems, which use water instead of liquid hydrogen, as the fuel to produce heat which drives the vehicle.

Apart from the above methods (of using water directly), electrolysis of water is an indirect method to use water, that produces H_2 and O_2 gases. Imperial College, U.K., proposed a water electrolysis hall-effect thruster which uses the O2 produced as a result of the electrolysis [17].

Cornell University used a WEP system in a 3U CubeSat to overcome drag and reported an extended life-span of 6 months for an orbit at 250 km altitude and more than 3.5 years for an orbit at 350 km altitude [17].

NASA proposed the "Integrated Modular Propulsion and Regenerative Electro-Energy Storage System (IMPRESS)" which combined water electrolyzer and hydrogen fuel cell technology thereby maximizing the efficiency of the WEP system; and subsequently designed a 1 N thruster based on this concept [17].

The Arizona State University conceptually designed a WEP system for a 6U CubeSat intended to perform an interplanetary exploration such as Mars. It demonstrated that using lithium chloride aqueous solutions to prevent water freezing at low temperatures led to a great performance enhancement [17].

So, it can be concluded that WPS is a fully proven technology (TRL of 10).

3 Designing the WEP System

A model-based optimization and sizing of the WEP system requires a working model of the entire system which needs to be designed. The entire system consists of a water tank, an electrolyzer, gas tanks, a controller and a thruster. In the present work, a pressure-fed system has been designed in order to respect the space constraints imposed by the CubeSat architecture, as a turbopump-fed system occupies more space, is heavier and demands more power from the already limited power of a CubeSat. The pressure-fed system was implemented by selecting the Pressure in the combustion chamber (P_C) as the design variable. Once the P_C was fixed, the gas tanks were filled by the electrolyzer until the pressure in the gas tanks (P_{H2} or P_{O2}) is 5% more than the P_C, a value chosen to prevent possible backflow. Similarly, the water tank would be at a pressure 10% more than the P_C. Simulink® was chosen to perform a real-time transient analysis of the system, and so, real time pressure values were provided to check valves and in general to the control system, to stop the electrolysis process once the cut-off pressure value is reached.

The following block diagram (See Figure 9) describes the actual Simulink® model. For the detailed model designed in the present work, see Appendix A.
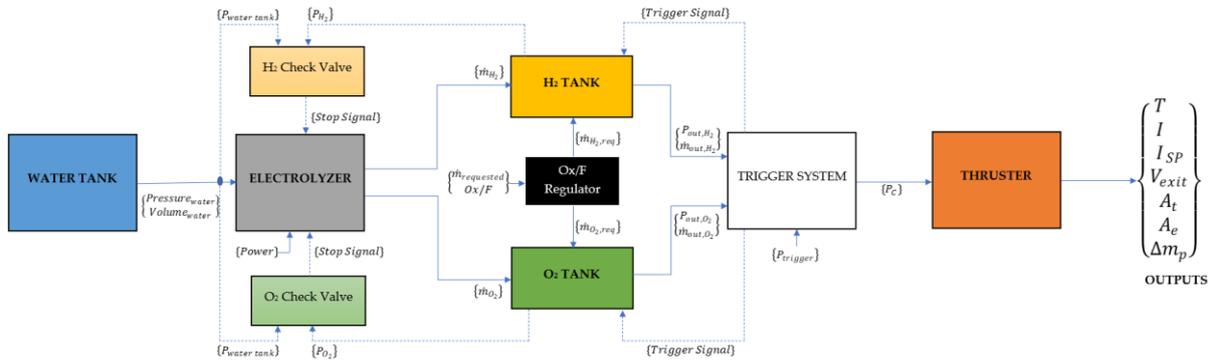

Figure 9: Block diagram of the Simulink® model

Water was provided to the electrolyzer and it draws power from the satellite's power system according to the requested mass flow rate by the thruster. Higher the requested mass flow rate, higher is the power drawn. H_2 and O_2 gas produced by the electrolyzer were filled into their respective tanks until the cut-off pressure was reached, which as described above, is 5% more than the P_C. It is important to note that the volume of H_2 tank is twice the volume of O_2 tank to allow the same outlet pressure into the thruster [7].

Once the tanks were filled up to the trigger pressure ($P_{trigger}$) which is the same as P_C, the trigger system was activated, which opens the feed line of the respective gas tanks to the combustion chamber. The trigger system, which receives the live pressure drop of the tanks, stopped the gases flowing into the combustion chamber when either gas tank has the gas at 0.01 bar pressure, which is defined as the cut-off pressure ($P_{cut-off}$). This value was chosen to mostly empty the gas tanks during a firing. The live pressure drop of the tanks is calculated based on the mass flow rate of the gas exiting the tank, which is exactly the mass flow rate of the gas requested by the thruster. The Ox/F regulator as seen in Figure 9 was input with the mass flow rate requested by the thruster and the mixture ratio, which in the present work was taken to be a stoichiometric mixture ratio of 8:1 (Oxygen: Hydrogen). Based on the mixture ratio, the propellant mass fraction was calculated and was used to calculate the individual gas mass flow rate requested. The thruster block consisted of the combustion chamber, the nozzle throat and the expansion section, all of which were mathematically modelled to give the output of Thrust (**T**), Total Impulse (TI), Specific Impulse (I_{SP}), Exit Velocity (v_{exit}), Throat Area (A_t), Exit Area (A_e) and the propellant consumed (Δm_P). In the actual Simulink® model, the mass of gas produced by the electrolyzer was continuously fed to a check valve which subtracted this value from the initial mass of water available (m_0). This is to stop the simulation once the result of the subtraction is zero, i.e., when there is no water left in the water tank.

3.1 Design choices

For modelling and testing the WEP system in the Simulink® platform, a few design choices were made either based on simple physics or based on literature and known values (of Specific heat ratio, etc.) from chemistry were taken. Table 1 lists out the design choices.

Table 1: Design Choices and Values from Chemistry

	Parameter	Value [units]	References
1	Volume ratio of H_2 and O_2 gas tank	2:1	[7]
2	Chamber Contraction Ratio (CR)	3.5	[27]
3	Temperature of all tanks (T_{tank})	293 K	-
4	Cut-off Pressure ($P_{cut-off}$)	0.01 bar	-
5	Oxidizer/Fuel (OF) Ratio	8:1 (stoichiometric)	-
6	Propellant mass fraction (MF_{prop})	1/9	-
7	Satellite altitude	500 km	-
8	Ambient pressure	~ 0 Pa	-

9	Combustion temperature (T$_C$)	3300 K (stoichiometric)	-
10	Specific heat ratio (γ)	1.32	[28]
11	Specific gas constant (R$_{gas}$) (water vapor)	461.5 J/kg.K	-
12	Voltage of dissociation of water (**V**$_{dis}$)	1.48 V	-
13	Molar mass of H$_2$ (M$_{H2}$)	0.002 kg/mol	-
14	Valency of H$_2$ (ν_{H2})	2	-
15	Molar mass of O$_2$ (M$_{O2}$)	0.032 kg/mol	-
16	Valency of O$_2$ (ν_{O2})	4	-
17	Universal gas constant (R)	8.314 J/mol.K	-

3.2   Mathematical equations

The following mathematical equations were used in the same order to model the WEP system in the Simulink® platform.

i)   <u>Mass flow rate of hydrogen/oxygen gas generated by the electrolyzer:</u>

$$\dot{m}_{gas} = \frac{I\,M}{\nu\,F} \tag{3.1}$$

where 'I' is the electric current (in A);

'M' is the molar mass of the gas (in kg/mol);

'ν' is the valency; 'F' is the Faraday constant (= 96485.332 C/mol)

ii)   <u>Rate of change of tank pressure (while emptying or filling):</u>

$$P\,V = n\,R\,T \tag{3.2}$$

$$P\,V = \frac{m}{M}\,R\,T \tag{3.3}$$

$$\dot{P} = \frac{\dot{m}_{gas}}{V_{tank}}\,R_{gas}\,T_{gas} \tag{3.4}$$

where '$\dot{P}$' is the rate of change of tank pressure;

'$\dot{m}_{gas}$' is the mass flow rate of gas;

'V$_{tank}$' is the volume of the tank (= V$_{gas}$ assumed to be completely filled with gas);

'R$_{gas}$' is the specific gas constant; 'T$_{gas}$' is the temperature of the gas

iii)   <u>Mass flow rate of gases</u>:

$$\dot{m}_{H_2} = MF_{prop}\, \dot{m}_{p_{req}} \tag{3.5}$$

$$\dot{m}_{O_2} = \left(1 - MF_{prop}\right) \dot{m}_{p_{req}} \tag{3.6}$$

where '$\dot{m}_{H2}$' is the mass flow rate of hydrogen gas;

'MF_{prop}' is the propellant mass fraction;

'$\dot{m}_{p,req}$' is the mass flow rate requested by the thruster

iv)   <u>Chamber pressure</u>:

$$P_c = \frac{P_{H2}\, \dot{m}_{H2} + P_{O2}\, \dot{m}_{O2}}{\dot{m}_{H2}\, \dot{m}_{O2}} \tag{3.7}$$

where 'P_c' is the chamber pressure;

'P_{H2}' and '$\dot{m}_{H2}$' are the pressure and mass flow rate of Hydrogen gas respectively;

'P_{O2}' and '$\dot{m}_{O2}$' is the pressure and mass flow rate of Oxygen gas respectively

v)   <u>Calculations at the nozzle throat</u>:

$$P_t = P_c \left(1 + \frac{\gamma - 1}{2}\right)^{-\gamma/\gamma-1} \tag{3.8}$$

$$T_t = \frac{T_C}{\left(1 + \frac{\gamma - 1}{2}\right)} \tag{3.9}$$

$$A_t = \frac{A_c}{CR} \tag{3.10}$$

where 'P_t' is the throat pressure; 'P_c' is the chamber pressure;

'γ' is the specific heat ratio; 'T_t' is the throat temperature;

'T_c' is the chamber temperature; 'A_t' is the throat area;

'A_c' is the chamber area; 'CR' is the contraction ratio

vi)   Calculations at the nozzle exit:

$$\frac{A_t}{A_e} = \left(\frac{\gamma+1}{2}\right)^{\frac{1}{\gamma-1}} \left(\frac{P_e}{P_c}\right)^{\frac{1}{\gamma}} \sqrt{\frac{\gamma+1}{\gamma-1}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} \tag{3.11}$$

where 'P$_e$' is the exit pressure

The above equation is solved using f-solve command in MATLAB® to get the ratio of pressure at exit and pressure at chamber (P$_e$/P$_c$).

$$A_e = A_t\ ER \tag{3.12}$$

$$P_e = \frac{P_e}{P_c}\ P_c \tag{3.13}$$

$$v_{exit} = \sqrt{\frac{2\gamma}{\gamma-1}\ R_{gas}\ T_c\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} \tag{3.11}$$

where 'A$_e$' is the exit area; 'ER' is the expansion ratio; 'v$_{exit}$' is the exit velocity

vii)   Thrust calculation:

An optimum expanded conical nozzle is assumed for the model.

$$\boldsymbol{T}_{id} = \dot{m}_p\ v_{exit} + (P_e - P_c)\ A_e \tag{3.12}$$

$$\lambda = \frac{1}{2}\ (1 + \cos\alpha) \tag{3.13}$$

$$v_{exit\_actual} = \lambda\ v_{exit} \tag{3.14}$$

$$C^*_{ideal} = \frac{P_c\ A_t}{\dot{m}_{prop}} \tag{3.15}$$

$$C_F = \frac{v_{exit\_actual}}{C^*_{ideal}} \tag{3.16}$$

$$\boldsymbol{T}_{actual} = C_F\ P_c\ A_t \tag{3.17}$$

where '$\boldsymbol{T}_{id}$' is the ideal thrust; '$\dot{m}_{prop}$' is the propellant mass flow rate;

'λ' is the correction factor for a conical nozzle; 'C$_F$' is the thrust coefficient

Equation (3.17) gives the actual thrust produced by the thruster.

viii)   <u>Limit mass flow rate for choking ($\dot{m}_{choke}$):</u>

$$\dot{m}_{choke} = A_t \, \frac{P_{tank}}{\sqrt{R_{gas}T_{tank}}} \, \sqrt{\gamma} \, \left(1 + \frac{\gamma - 1}{2}\right)^{\frac{\gamma+1}{2\,(1-\gamma)}} \tag{3.18}$$

3.3   Simulink® solver

It is of great importance to choose the right solver to solve the transient or the dynamic system designed and so, a robust reading of the Simulink® Documentation [29] was carried out.

For simulating/solving a dynamic system, the state of the system needs to be evaluated/computed at successive time steps, which are time intervals where computation of the system occurs. The size of such a time interval is known as the step size. Simulating/solving the system was performed by a solver in the Simulink®, which incorporates a chosen numerical method to solve the set of Ordinary Differential Equations (ODEs) that represent the system. The solver stops and the simulation ends once the convergence criteria pre-defined is met.

A variable time-step solver changes the time step according to the changing state of the system. It reduces the size when greater accuracy is expected, particularly when the gradient/change of the system is significant. It increases the size when the gradient/change of the system is slow/constant. Although a bit of computational time may be spent on calculating the step size during the simulation, overall, it reduces the total number of steps and so, the computational time. Such a solver captures the changes occurring in the system accurately. A continuous solver computes, through numerical integration, the continuous state of the system at the simulation time step, based on the state and it's derivates at the previous time step.

So, a variable time-step continuous solver was concluded to be the appropriate solver to simulate the dynamic system (WEP system) of the present work. Among the solvers available in Simulink®, "ode45" which is "medium-accuracy" Dormand-Prince method of Runge-Kutta ODE solvers family, was chosen for the simulation. The "4-5" represents the solver's ability to provide fourth- and fifth-order accurate solutions, specifically it is a fifth-order method that performs a fourth-order estimate of work. It uses a fourth-order interpolant for event location and smooth plots. These characteristics are necessary for the complex dynamics WEP system designed.

<u>Table 2</u> lists out the configuration parameters set for the highest possible accuracy and feasible simulation time for the Simulink® model designed.

Table 2: Configuration Parameters

	Parameter	Value [units]
1	End time of the simulation	3×10^7 sec
2	Max. time step	1 sec
3	Min. time step	0.001 sec
4	Relative tolerance	1×10^{-6}
5	Absolute tolerance	1×10^{-7}
6	Simulation mode	Accelerator

3.4   Testing the Simulink® model

The Simulink® model built for the WEP system was tested by selecting a few feasible and logical initial values, listed out in Table 3. The design choices and known values from chemistry are the ones reported in Table 1.

Table 3: Initial Values for Testing

	Parameter	Value [units]
1	Available power for electrolyzer (P_{EL})	10 W
2	Volume of water tank (V_{water})	1 L
3	Pressure of water tank (P_{water})	10 bar
4	Volume of H_2 tank (V_{H2})	300 mL
5	Volume of O_2 tank (V_{O2})	150 mL
6	Pressure of gas tanks (P_{gas})	10 bar
7	Mass flow rate requested by thruster	1.2×10^{-6} kg/s
8	Chamber pressure or trigger pressure (P_C)	6 bar
9	Chamber area (A_C)	6.6×10^{-4} m² (1.45 cm radius)
10	Burn time (t_b)	10 sec
11	Expansion ratio (ER)	200
12	Cone half angle (α)	15°

It is important to note that a few values in the above table may be perceived as infeasible, for example, a 1 L of water may not be the exact volume needed for a CubeSat mission, but in the present chapter, the interest was just to test the Simulink® model of the WEP system.

With the values reported above and using the mathematical equations from (3.1) to (3.18), a Thrust of 0.39 N, an exit velocity of 3275.8 m/s and a total impulse of 3.93 N-s

were achieved. More importantly, as shown in the Figures 10 to 12, the trigger system and the control mechanism developed for the model were found to be working as expected for a single firing.

As seen in Figure 10, H_2 was filled in the tank until the trigger pressure (of 6 bar) and was immediately emptied until the cut-off pressure (of 0.01 bar). Subsequently, the H_2 was refilled in the tank until the maximum pressure of the tank (10 bar).
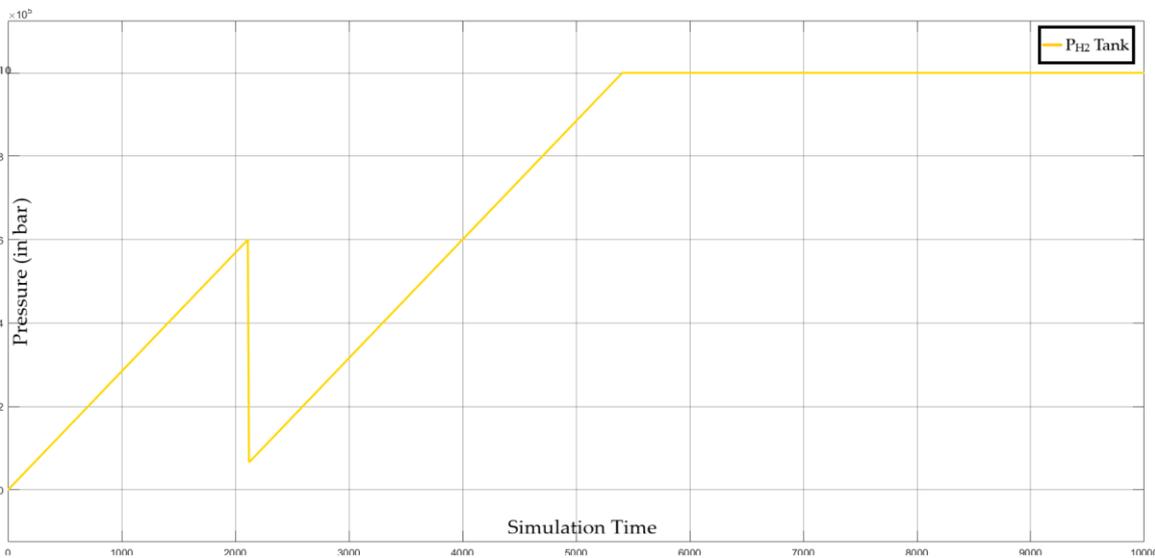

Figure 10: Pressure of H_2 tank vs time

Once the H_2 (and O_2) was refilled to the maximum capacity, the power to the electrolyzer was cut-off (See Figure 11).

Figure 11: Power used by the electrolyzer vs time

The thrust produced (maximum of 0.39 N) for the burn time (of 10 seconds) can be seen in Figure 12.
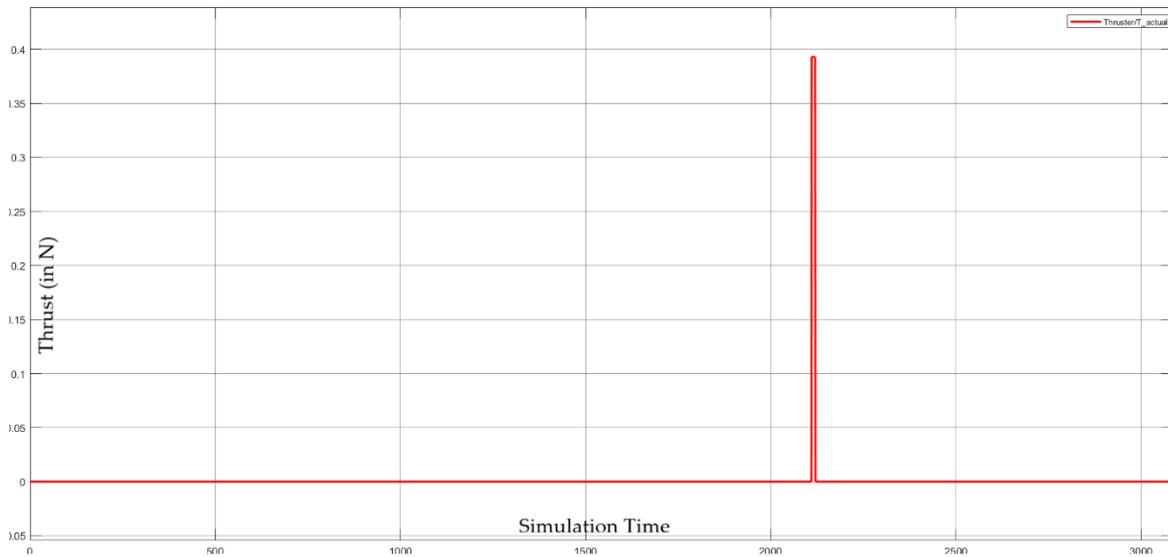

Figure 12: Thrust vs time

So, the MATLAB® (version R2020b) - Simulink® model of the WEP system was designed and tested. It was found that the controller mechanism designed for the model worked as expected, and this model could be used for optimization, which is dealt with in the next chapter.

4 The Optimizer

In the present chapter, the already tested MATLAB®-Simulink® model of the WEP system was used in the optimizer, a MATLAB® optimization code to perform a model-based optimization. A model-based optimization is a method of optimization where the design variables are iteratively fed into the Simulink® model until the global minima is found i.e., the optimization is complete.

4.1 Mission requirements

In the thorough literature review reported in Chapter 2, key focus was to identify the key requirements of a thruster in a CubeSat mission. Among the commercially available hydrazine thrusters, 1 N thrusters by Northrop Grumman in the United States, Aerojet, Snecma in France, European Aeronautic Defense and Space company ST in Europe, and RAFAEL in Israel have a length of 10-20 cm and a weight compatible with a CubeSat. So, a 1 N hydrazine thruster was targeted to be replaced by a 1 N WEP thruster [17].

For a WEP system of the present work, a thrust requirement of 1 N was the objective of many previous numerical and experimental works [17] [6] [14] [22] [16], and so, a thrust of 1 N was set as a target for the thruster. As per a detailed report [30] defining various possible mission requirements, a few Delta-v (Δv) values were noted to check the possibility of the present WEP system-powered CubeSat to carry out such missions. Table 4 lists the various maneuvers and (Δv) requirements in LEO.

Table 4: Delta-v requirements for LEO missions [30]

Manoeuvre	Value [units]
Changing orbit:	
Altitude change	0.6 m/s per km
Plane change	135 m/s per degree
Orbit control:	
Drag compensation @ 500 km altitude	4.4 – 25.8 m/s/year

4.2 The optimizer code

Optimization is the process of finding the best possible solution of a multi-variable problem. It involves defining a cost function, which is often minimized, and it is subjected to constraints. The design variables are the values which are varied to get the global minima of the cost function. The design variables, for which the cost function is minimized, are the "optimum design variables"; the performance parameters computed based on the optimum design variables are the "optimum performance parameters". Figure 13 showcases the steps followed to perform optimization in general. The same were followed in the present work.
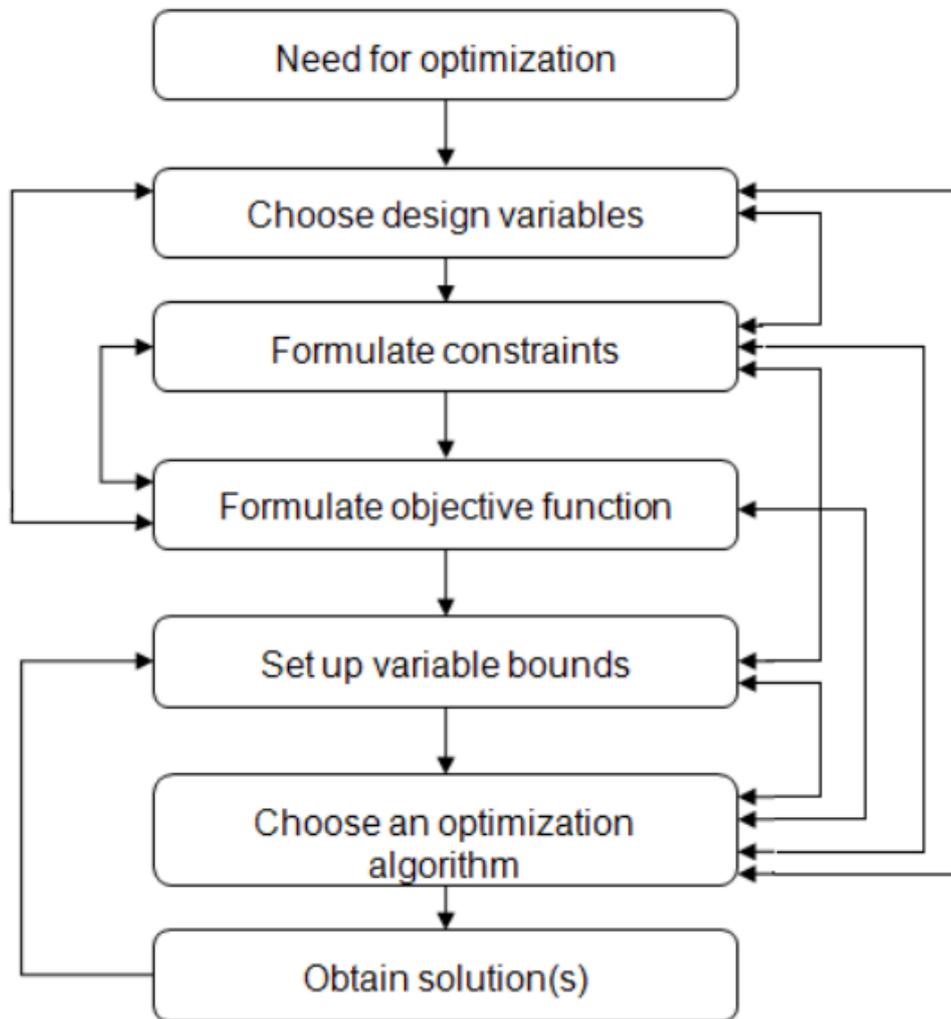

Figure 13: Flow chart - steps to get optimum solution [31]

The MATLAB® code for the optimization process consisted of 4 main functions, namely Cost function, Simulation function, Objective function and Non-linear constraint function, which are described in detail below.

4.2.1 Cost function

A cost function or an objective function is a mathematical equation which determines the objective of the optimization, whether to maximize or minimize something. For example, while using a material to build something, the objective may be to minimize the cost of the material used. The function chosen mandatorily needs to be continuous, needs to be differentiable and needs to be convex i.e., have a clear global minima.

In the present work, the objective was to achieve 1 N thrust (**T**) and have the best possible specific impulse (I_{SP}) for the WEP system. In order to select the best cost function, open-source websites like "Desmos" (Desmos, Inc.) [32] and "Math3D" [33] were used. To achieve 1 N thrust and best possible specific impulse, it was concluded that a two-variable cost function was needed, in which one variable has a minima at a magnitude of 1 (for 1 N thrust) as shown in Figure 14 and the other variable has a minima at infinity (for highest possible I_{SP}) as shown in Figure 15. The cost function selected is given by equation (4.1) and was minimized. Figure 16 represents the 3D cost function.

$$cost = \left[e^{\frac{1}{T}} + (1 - e) + e^{T} \right] \cdot \left(e^{\frac{1}{I_{SP}}} \right) \qquad (4.1)$$

4.2.2 Simulation function

In the optimizer MATLAB® code, a function called "Simulation function" was used to define the design variables as global variables. The "design" variables are those key variables which connect the performance and the design or size of the thruster. They are defined as global variables, for the Optimizer to vary them, in order to reduce the minimize the cost function. The design variables which lead to the minimum cost, were used to design the thruster.
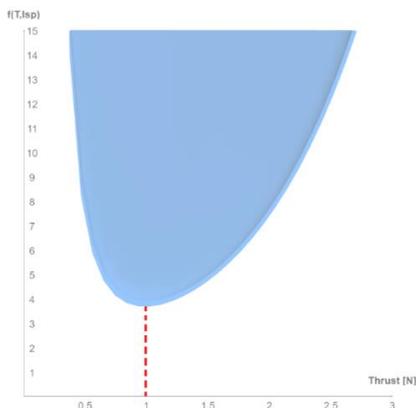

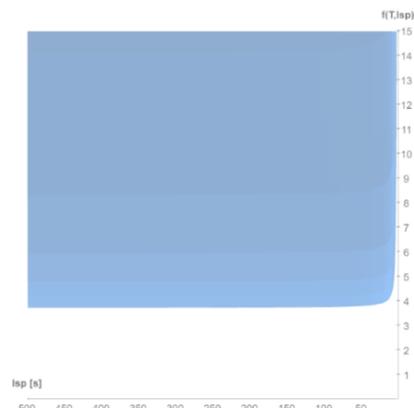
Figure 14: Cost function with minima at 1 N thrust

Figure 15: Cost function with minima at infinity to maximize I_{SP}
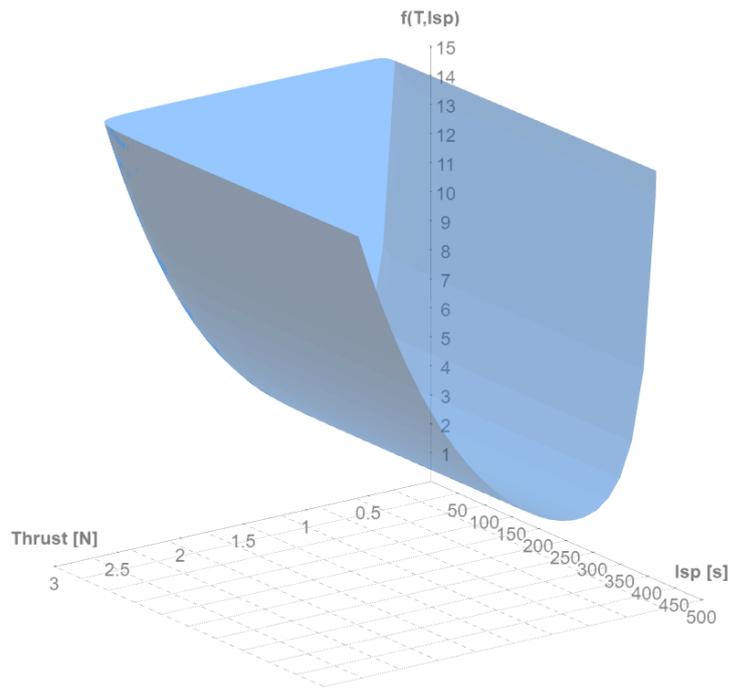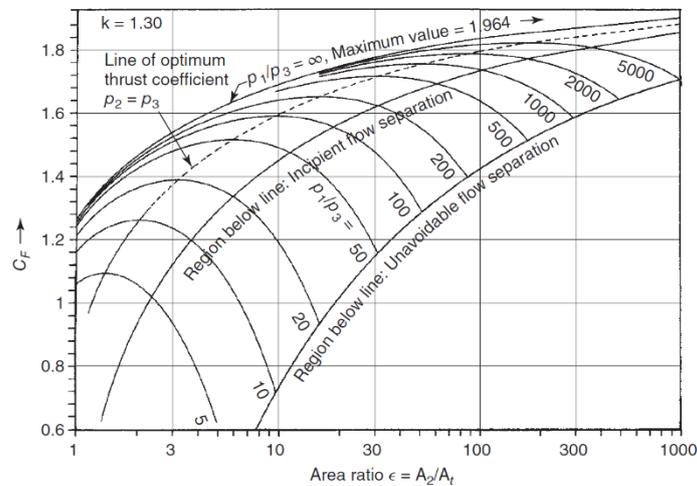
Figure 16: 3D view of the cost function

The design variables chosen were:

- Chamber Pressure (P_C)
- Mass flow rate requested by the thruster ($\dot{m}_{prop,req}$)
- Expansion cone Half Angle (α)
- Contraction cone Half Angle (α_C)

To ensure a compact and effective design, an expansion ratio (ER) of 500 was chosen. From the plot (See Figure 17) of thrust coefficient (C_F) vs expansion ratio (ER) available in [27], the corresponding C_F for ER of 500 is found to be around 1.85. The following equations were used to compute the exact values of Chamber Area (A_C) and Exit Area (A_e).

Figure 17: CF vs ER [27]

$$C_F = \frac{T}{P_C \, A_t} \tag{4.2}$$

For design thrust of 1N, $\qquad C_F = \dfrac{1}{P_C \, A_t} \tag{4.3}$

$$A_t = \frac{1}{P_C \, C_F} \tag{4.4}$$

$$A_{t,max} = \frac{1}{P_{C,min} \, C_F} \tag{4.5}$$

$$A_{t,min} = \frac{1}{P_{C,max} \, C_F} \tag{4.6}$$

$$A_C = A_t \, CR \; ; \; A_e = A_t \, ER \tag{4.7}$$

As mentioned in Chapter 3, a pressure-fed system was chosen in order to ensure a positive flow of the propellant i.e., from water tank to the thruster. To have the maximum pressure in the water tank ($P_{max \, w\_tank}$) and subsequently lower pressure in the gas tanks ($P_{max \, g\_tanks}$) and finally the lowest (of the system) in the chamber (P_C), a Pressure Factor ($PFactor_{gas \, tanks}$) of 1.05 was chosen for the gas tanks and a Pressure Factor ($PFactor_{water \, tank}$) of 1.10 was chosen for the water tank, which were multiplied with the chamber pressure to obtain the pressures of gas tanks and the water tank respectively. The same is clearly represented in the equations below.

$$P_{water \, tank} > P_{gas \, tanks} > P_C \tag{4.8}$$

$$P_{max \, g\_tanks} = PFactor_{gas \, tanks} \cdot P_C \tag{4.9}$$

$$P_{max \, w\_tank} = PFactor_{water \, tank} \cdot P_C \tag{4.10}$$

A few other variables were also defined as global variables namely:

- Maximum pressure of the water tank ($P_{max \, w\_tank}$)
- Maximum pressure of the gas tanks ($P_{max \, g\_tank}$)
- Trigger pressure ($P_{trigger}$) (= P_C), which commands the thruster to be used once the pressure in the gas tanks reach the required chamber pressure of the thruster
- Chamber area (A_C)
- Exit area (A_e)

The function then runs the Simulink® model of Chapter 3 with the configuration parameters listed in <u>Table 2</u> and gets the following outputs:

- Total impulse (TI)
- Average specific impulse (I_{SP})
- Average thrust (**T**)
- Propellant consumed (Δm_P)
- Burn time (t_b)

4.2.3   Objective function

The objective function first runs the "Simulation function" (<u>Section 4.2.2</u>) and then uses Average specific impulse (I_{SP}) and Average thrust (**T**) to run the "Cost function" (<u>Section 4.2.1</u>) to get the cost.

4.2.4   Non-linear constraint function (NonLCon)

The constraint (function) was created to design the convergent and divergent section. The following non-linear constraints were used.

$$d_c \leq 10 \; cm \tag{4.11}$$

$$d_e \leq 10 \; cm \tag{4.12}$$

$$d_t \leq d_c \tag{4.13}$$

$$d_t \leq d_e \tag{4.14}$$

$$L_{conv} = \frac{r_c - r_t}{\tan \alpha_c} \; ; L_{exp} = \frac{r_e - r_t}{\tan \alpha} \tag{4.15}$$

$$L_{conv} \leq L_{exp} \tag{4.16}$$

$$L_{conv} + L_{exp} \leq 9 \; cm \tag{4.17}$$

where 'd_c', 'd_t', 'd_e' are diameters of chamber, throat and exit respectively;

'L_{conv}' is the length of the convergent section;

'L_{conv}' is the length of the convergent section;

where 'r_c', 'r_t', 'r_e' are radii of chamber, throat and exit respectively;

The chamber and exit diameters were bound by the maximum of 10 cm, maximum side dimension of a CubeSat. The total length of the thruster is bound by 9 cm to allow a 1 cm catalyst bed, which is described in <u>Section 4.4.1</u>. The above inequality was coded as a 'nonlcon' function of MATLAB® for the Genetic Algorithm (GA).

4.3   The Genetic Algorithm (GA)

Having defined everything necessary for optimization in the previous sections, the present section discusses the various optimization techniques (see Figure 18) and the reason for choosing GA for the present work.

4.3.1   Why the GA?

In the previously done research activity of getting an optimum design for a thruster, barring a few research works, the most relied optimization technique was the gradient-based method to optimize the continuous variables. However, in the recent times, there has been more focus on Evolutionary Algorithm, one of which is the GA. It is a global search and optimization method that imitates naturally occurring biological evolution. It is advantageous over the traditional gradient-based methods as it has an ability to combine discrete, integer and continuous variables; it does not need an initial design; it can deal with non-convex, multi-modal discontinuous functions. It operates on a population of potential solutions of the cost function applying the principle of survival of the fittest i.e., it evaluates the function for every such value of the population to find the best set of solution. One such population is called a generation and the best set of solutions at the current generation are carried onto the next generation. At each generation of the subsequent generation, a new set of possible solutions is chosen along with the best of the previous generation. This allows for a better search as the algorithm does not fixate on an initial best guess. In the present work, GA was selected for the optimization for the advantages mentioned above and also because, the variation of the design variables (gradient) w.r.t the cost function was something that is difficult to know and mathematically code.

4.3.2   Design choices and GA options

The GA was run from the Global Optimization Toolbox® of MATLAB® and the following command syntax was chosen:

x = ga(fun,nvars,A,b,Aeq,beq,lb,ub,nonlcon,options)

where, 'x' represents the optimum obtained from the GA

'fun' is the objective function, specified as a function handle or function name

'A' are the linear inequality constraints, specified as a real matrix (Not used)

'b' are the linear inequality constraints, specified as a real vector (Not used)

'Aeq' are the linear equality constraints, specified as a real matrix (Not used)

'beq' are the Linear equality constraints, specified as a real vector (Not used)

'lb' is the Lower bound, specified as a real vector or array of doubles

'ub' is the Upper bound, specified as a real vector or array of doubles

'nonlcon' are the nonlinear constraints, specified as a function handle or function name
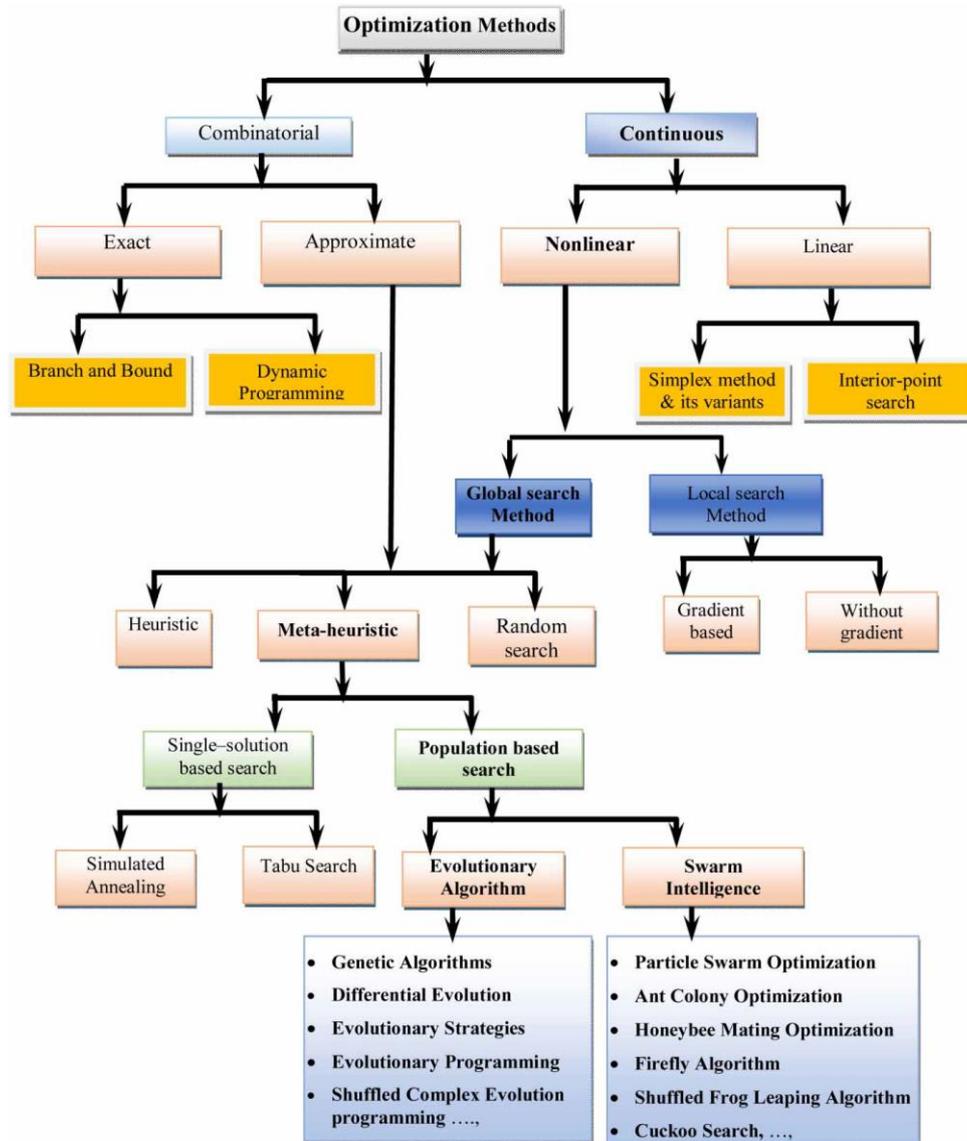
'options' are the optimization options

Figure 18: Optimization techniques [34]

Importantly, Table 5 lists out the lower and upper bounds of the design variables.

The values of the lower and upper bound of chamber pressure and mass flow rate requested by the thruster were based on design choice for the system and the choking mass flow rate respectively. Both cone half angles, α and α_c, were chosen as a design choice for the nozzle.

Table 5: Lower and Upper Bounds

	Parameter	Bounds	Units
1	Chamber pressure (P_C)	[1, 5]	bar
2	Mass flow rate requested by the thruster ($\dot{m}_{prop,req}$)	[1×10^{-5}, 0.001]	kg/s
3	Cone half angles (α) (α_c)	[15, 25]	deg

Table 6 lists out the GA options/parameters used in the present work.

Table 6: GA options

	Option	Value
1	Constraint tolerance	1×10^{-3}
2	Crossover fraction	0.5
3	Elite count	30
4	Max. generations	30
5	Max. stall generations	5
6	Function tolerance	1×10^{-6}
7	Population size	2000

The above listed GA options/parameters gave the best possible cost function value i.e., a thrust value closest to 1 N and the best possible specific impulse. The options chosen were arrived at, after a number of trial and errors which involved changing the population size and/or the elite count etc., and running the algorithm to check the cost function value. In order to get the best results from a GA, it is necessary to try out different combinations of the GA options.

4.3.3 Results from the optimization

After configuring the GA, the optimization was started. The MATLAB® code runs the GA, which runs the "objective function" (section 4.2.3) (constrained by the non-linear constraint (section 4.2.4)) which in turn runs the "simulation function" (section 4.2.2) and the "cost function" (section 4.2.1).

The convergence was achieved in 6.5 hours (simulation time) after 4 generations and Figure 19 shows the average distance between individuals, best fitness value and mean fitness value. The best fitness value nearly remains the same after the first generation; the mean fitness value and average distance between individuals showed a decreasing trend towards the end, which leads to convergence.
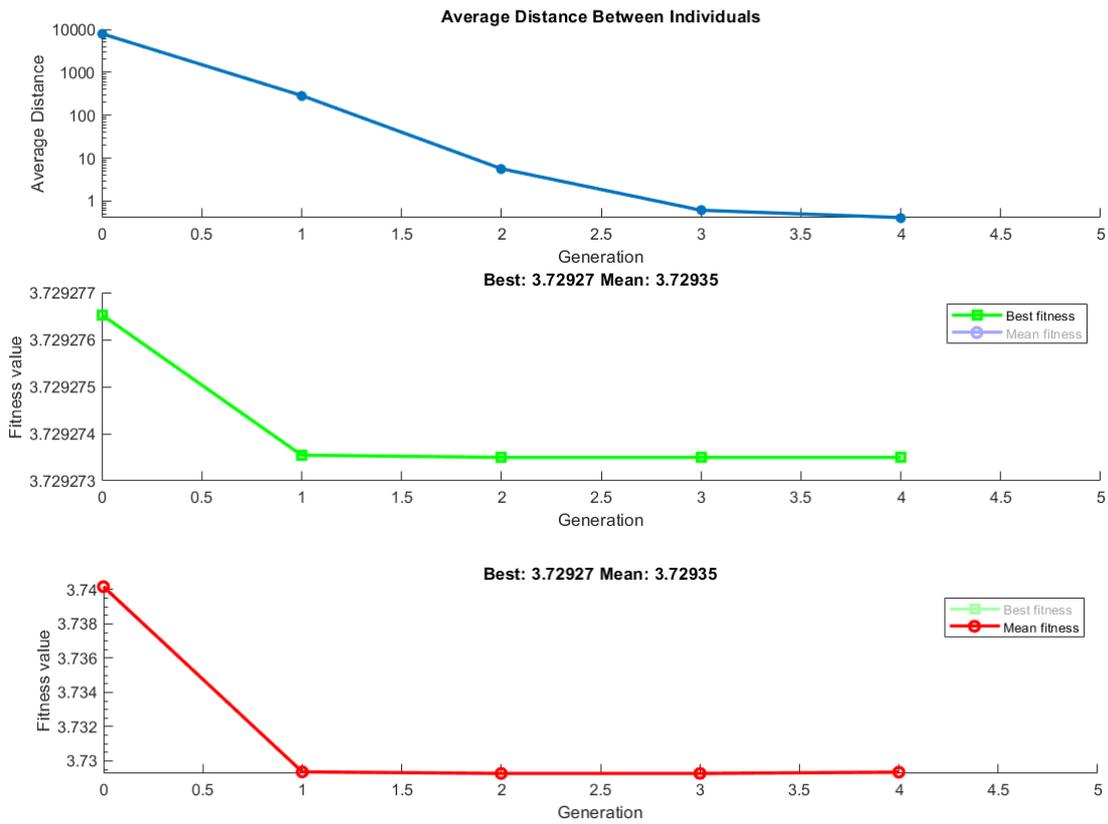
Figure 19: Plots from GA optimization

<u>Table 7</u> lists out the **Optimal design** and the **Optimal performance parameters** obtained after convergence.

Table 7: GA output

	Variable	Value [Units]
Optimal design parameters:		
1	Chamber pressure (P_C)	1.62 bar
2	Mass flow rate requested by the thruster ($\dot{m}_{prop,req}$)	0.301 g/s
3	Expansion cone half angle (α)	15.67°
4	Convergent cone half angle (α_c)	20.79°
5	Chamber area (A_C)	1.166×10^{-5} m²
6	Exit Area (A_e)	1.7×10^{-3} m²
7	Trigger pressure ($P_{trigger}$) (= P_C)	1.62 bar
8	Gas tanks max. pressure ($P_{max\ g\_tank}$)	1.70 bar
9	Water tank max. pressure ($P_{max\ w\_tank}$)	1.78 bar

Optimal performance parameters:

1	Total impulse (TI)	1.263 N-s
2	Specific impulse (I_{SP})	338.95 s
3	Thrust (**T**)	1.001 N
4	Propellant consumed (Δm_P)	0.38 g per fire
5	Burn time (t_b)	1.262 s
6	Exit velocity (v_{exit})	3325.1 m/s
7	Choking mass flow rate ($\dot{m}_{choking}$)	2.523×10^{-5} kg/s

5 Sizing the WEP System

The components of the WEP system namely the water tank, the electrolyzer, the gas tanks and the thruster were sized based on the design variables obtained in the previous chapter.

5.1 Sizing the thruster

The area of the throat (A_t) was computed using equation (4.4) and subsequently diameter of the throat (d_t) was calculated. The chamber area and exit area of the nozzle were subsequently computed using equation (4.7). The length of the convergent (L_{conv}) and length of the expansion (L_{exp}) sections were calculated using the equations (4.15).

A conical nozzle was chosen as suggested by [25]. A catalytic igniter was selected based on the experimental work of [17] as it has a simple structure. It was recommended to use a length of the catalyst bed between 8 mm and 32 mm. So, a length of 10 mm i.e., 1 cm was chosen for the catalyst bed (L_{CB}).

Table 8 lists the values related to the design of the thruster. Figure 20 and Figure 21 showcase the thruster designed.

Table 8: Thruster dimensions

	Design parameter	Value (units)
1	Length of the catalyst bed (L_{CB})	1 cm
2	Diameter of chamber (D_C)	0.38 cm
3	Length of the convergent section (L_{conv})	0.24 cm
4	Diameter of throat (D_t)	0.2 cm
5	Convergent cone half angle (α_c)	20.79°
6	Diameter of nozzle exit (D_e)	4.6 cm
7	Length of the expansion section (L_{exp})	7.84 cm
8	Expansion cone half angle (α)	15.67°
9	Expansion ratio (ER) (design choice)	500

SIDE VIEW

L_{exp} $\quad$ L_{conv} $\quad$ L_{CB}

Figure 20: Side view of the thruster

BACK VIEW

Chamber diameter $\quad \varnothing\, 0.38$

Throat diameter $\quad \varnothing\, 0.20$

Exit diameter $\quad \varnothing\, 4.60$

Electrolyzer

All dimensions are in cm

Figure 21: Back view of the thruster

5.2 Sizing the gas tanks

The chamber pressure was directly used to find out the maximum pressure of gas tanks using the PFactor$_{gas\ tank}$ (=1.05). This is the pressure until which the gas tanks are demanded to be filled by the electrolyzer.

A design choice was made w.r.t the shape of the tanks – both gases and water. A cuboidal shape of square cross-section was chosen and the reason was to utilize the already limited space of the CubeSat to best possible extent. A fillet was given at the edges of the tanks in order to completely empty the propellant in a zero-g environment. A fillet radius of 1 cm was chosen (See Figure 22).

The following equations (shown for H$_2$ gas tank) were used to size the both the gas tanks.

$$P_{gas\ tank} = PFactor_{gas\ tank}\ P_C \tag{5.1}$$

$$m_{H_2} = MF_{prop}\ \Delta m_p \tag{5.2}$$

$$V_{H_2} = \frac{m_{H_2}\ R_{H_2}\ T_{tank}}{P_{gas\ tank}} \tag{5.3}$$

$$h_{H_2} = \frac{V_{H_2}}{TBA} \tag{5.4}$$

$$m_{O_2} = (1 - MF_{prop})\ \Delta m_p \tag{5.5}$$

where 'm_{H2}' is mass of H$_2$ consumed per fire; 'm_{O2}' is mass of O$_2$ consumed per fire

'MF$_{prop}$' is the propellant mass fraction; 'Δm_P' is the propellant consumed;

'V_{H2}' is the volume of H$_2$ gas tank; 'h_{H2}' is the height of the H$_2$ tank;

'TBA' is the tank base area (cube of 10 cm x 10 cm minus the fillet radius of 1 cm);

Table 9 lists the various design and performance related values of the gas tanks and Figure 22 showcases the tanks design.

Table 9: Gas tank sizing

	Parameter	Value (units)
1	Pressure of gas in tanks (P$_{gas\_tank}$)	1.70 bar
2	Mass of hydrogen consumed per fire (m$_{H2}$)	0.0422 g
3	Volume of hydrogen gas tank (V$_{H2}$)	299.47 cm³
4	Height of hydrogen tank (h$_{H2}$)	3.02 cm
5	Mass of oxygen consumed per fire (m$_{O2}$)	0.3376 g
6	Volume of oxygen gas tank (V$_{O2}$)	149.74 cm³
7	Height of oxygen tank (h$_{O2}$)	1.51 cm

The volume of the H_2 tank is nearly twice the volume of the O_2 and so, the output given by the GA respected the volume and mass ratios of H_2 and O_2 combustion.
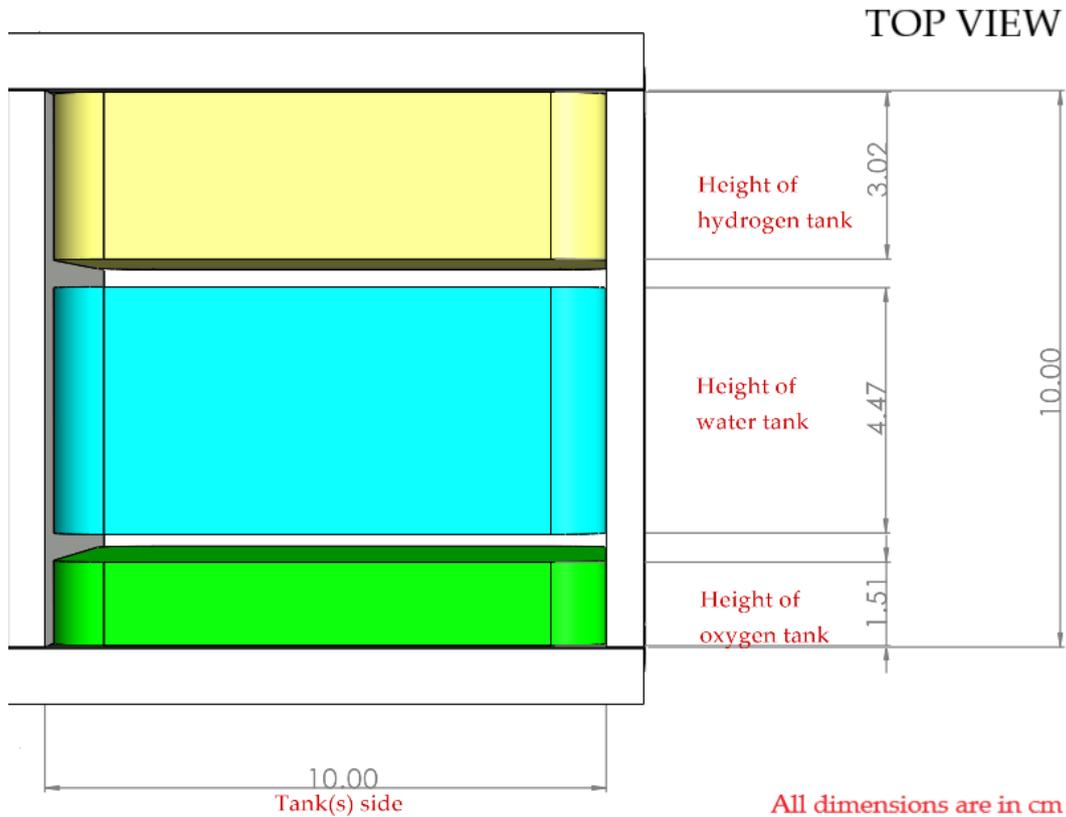

Figure 22: Tanks

5.3 Sizing the electrolyzer

The Horizon Education PEM electrolyzer reported in Chapter 2 was taken as a reference to size the electrolyzer assuming a constant current density (A/cm^2) of the electrolyzer.

To design the electrolyzer and the water tank, a CubeSat mission in a circular orbit at a 500 km altitude was considered. The satellite orbital period was calculated to be 94 minutes and it was assumed that the thruster was fired every time satellite reached the perigee. This gave the time between consecutive burns of 94 minutes.

The height of the electrolyzer was directly taken as 1.7 cm, same as that of the commercially available model.

The following equations were used to size the electrolyzer.

$$\dot{m}_{H_2} = {m_{H_2}}/{tbb} \tag{5.6}$$

$$I = \frac{\dot{m}_{H_2} \, \nu \, F}{M_{H_2}} \tag{5.7}$$

(Faraday Law of electrolysis)

$$Power = V_{dissociation} \, I \tag{5.8}$$

$$A_{EL} = \frac{I \, A_{ref}}{I_{ref}} \tag{5.9}$$

$$Side_{EL} = \sqrt{A_{EL}} \tag{5.10}$$

where '$\dot{m}_{H2}$' is the mass flow rate of rate of hydrogen consumed per fire;

'tbb' is the time between burns (= 94 minutes); 'I' is the current per fire;

'A_{EL}' is the surface area of the electrolyzer;

'I_{ref}' and 'A_{ref}' are references values of Horizon Education PEM electrolyzer;

'$Side_{EL}$' is the side of the square-shaped electrolyzer

A total-impulse-to-power ratio (TI/**P**) was defined as the ratio between Total Impulse (TI) and Power requested per fire (**P**$_{perfire}$) to showcase the capability of the present WEP system and to compare with existing systems.

Table 10 lists the various design and performance related values of the electrolyzer.

Table 10: Electrolyzer performance and sizing

	Parameter	Value [units]
1	Mass flow rate of rate of hydrogen ($\dot{m}_{H2}$)	7.482 x 10^{-6} g/s
2	Current per fire ($I_{perfire}$)	0.7163 A
3	Power per fire (**P**$_{perfire}$)	1.0601 W
4	Side of the electrolyzer (Side$_{EL}$)	5.46 cm
5	Total-impulse-to-power ratio (TI/**P**)	1.19 N-s/W

5.4   Sizing the water tank

The water tank was sized based on the height of the gas tanks. The tank base area is same as that of the gas tanks, and so, the height of the water tank was calculated using the equation (5.11). It is important to note that a 1 cm gap was considered for piping and other equipment which may be needed to be housed between the tanks.

$$h_{w\_tank} = (10 - 1) - h_{H_2 tank} - h_{O_2 tank} \tag{5.11}$$

The volume of the water tank was computed to be 441.87 cm³. With that the entire WEP system was sized (see Figure 23) in 2U i.e., 2 cubes of 10 cm x 10 cm x 10 cm dimensions.
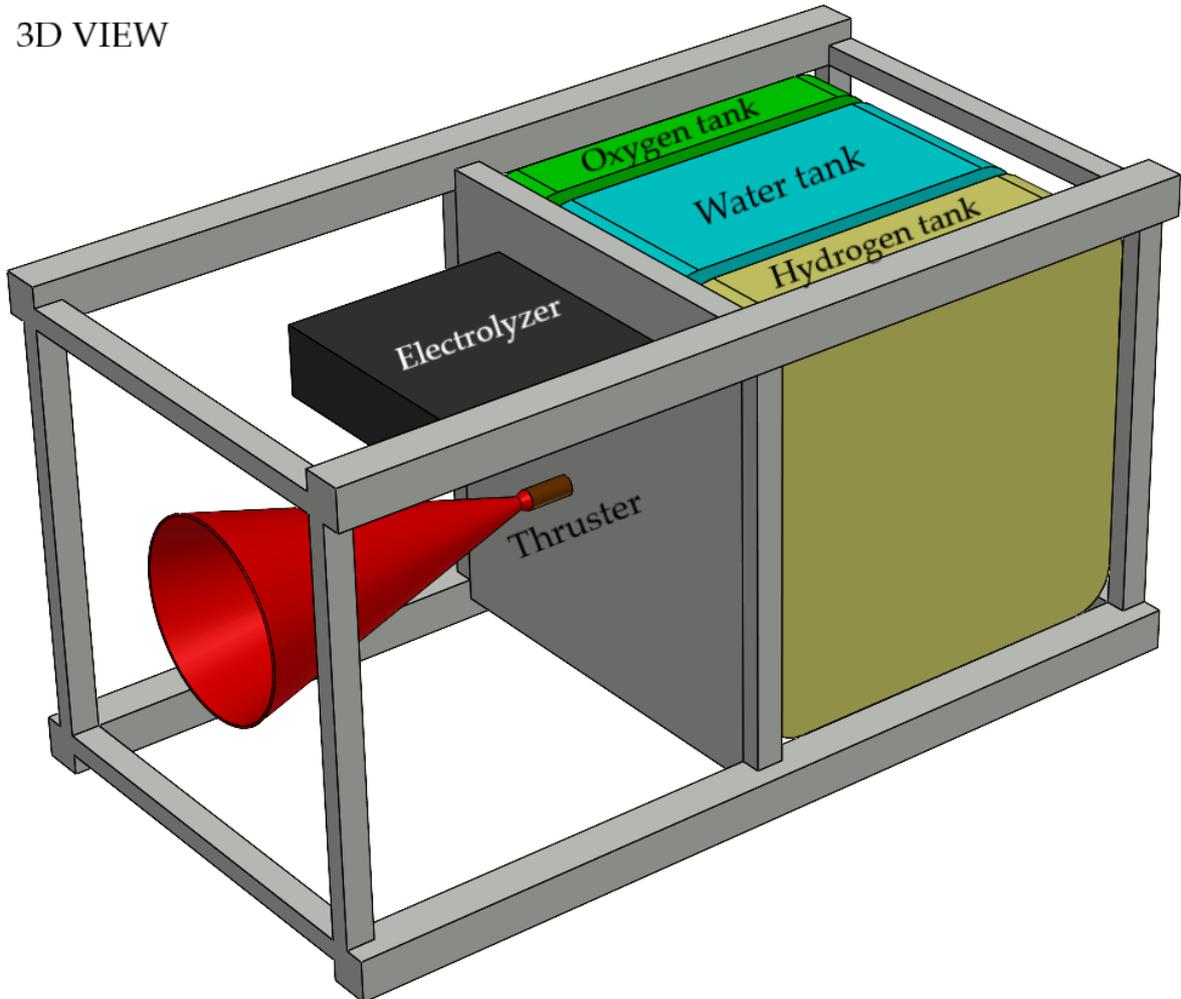
3D VIEW

Figure 23: 3D view of the entire WEP system

5.5   Mission design

Considering the mission selected in sub-chapter 5.4, this 2U WEP system can be used in a:

- 6U CubeSat in a circular orbit at an altitude of 500 km. Based on Table 4, for drag compensation to keep the satellite at the same altitude, the Delta-v (Δv) required is 15.1 m/s/year.

- 12U CubeSat in a circular orbit at an altitude of 500 km. Based on Table 4, the (Δv) required for drag compensation is 25.8 m/s/year. Two designs 'A' and 'B' were proposed for suiting such a mission. In design 'A', the existing 2U WEP system with ~442 mL water capacity is used. In design 'B', an additional 1 liter of water is added to the propulsion system making it a 3U propulsion system (2U WEP system + 1U of water = 3U of propulsion system).

The Delta-v (Δv) of the CubeSat was calculated using the equation below.

$$\Delta v = I_{sp}\ g_o \ln \frac{m_0}{m_f} \tag{5.12}$$

where 'g_0' is the 9.81 m/s^2; 'm_0' is the initial mass of the spacecraft;

'm_f' is the final mass of the spacecraft after entire propellant is consumed

Table 11: Mission-related parameters

m_0 (kg)	m_P (kg)	m_f (kg)	ΔV_{total} (m/s)	Δv (m/s/fire)	Δv req. (m/s/year)	Δv req. (m/s/orbit)	Life (years)
6U CubeSat (2U propulsion system)							
12	0.44	11.56	124.21	0.105	15.1	0.002696	8.23
12U CubeSat Design A (2U propulsion system)							
20	0.44	19.56	73.97	0.063	25.8	0.0046	2.87
12U CubeSat Design B (3U propulsion system incl. 1 L extra water)							
20	1.44	18.56	248.46	0.063	25.8	0.0046	9.63

So, the 2U WEP and the modified 3U WEP systems achieved significant values of Delta-v (Δv) and life of the satellite by compensating for drag:

- using just around 25% of the Δv per fire, provided by the thruster in case of 6U CubeSat
- using around 10% of the Δv per fire, provided by the thruster in case of 12U CubeSat, as listed in Table 11.

The precise required Δv per fire can be achieved by adjusting the burn time using a precise timer system. This mission design example is an indicator of the capabilities of the WEP system designed for a 6U or a 12U CubeSat.

6 Conclusions and Future Work

6.1 Conclusions

The present work focused on the upcoming technology of using Water as a propellant for in-space applications. One of the ways to utilize water is to split it into hydrogen and oxygen gas by electrolysis, which then can be combusted in a bipropellant thruster to produce thrust. A comprehensive literature review was undertaken to identify key research gaps or challenges faced, for the technology to reach its full potential. On completion, it was concluded that despite there being a lot of research, there was significant dependence on experimental methods to test the system. Although experimental testing has its own advantages, it may not be possible, both monetarily or logistically at the university level. Also, fixing a robust design before experimentally testing it, saves a lot of money and time. So, the need for numerically modelling a Water Propulsion System (WPS), specifically Water Electrolysis Propulsion (WEP) system was identified.

The present work is a conceptual design process that can be incorporated in the future for preliminary design of a WEP system. The process of designing the system was split mainly into 2 parts – firstly, developing a MATLAB®-Simulink® model of the entire system so as to perform a transient analysis, as in the case of an experimental setup and secondly, using the Simulink® model to perform a model-based optimization to achieve a target thrust and highest possible specific impulse.

The WEP system was mathematically modelled in Simulink® as a pressure-fed system and was tested with a few design choices taken from literature. The main focus was to test the control mechanisms implemented through mathematically modelled check-valves, which received the live pressure information of the various components of the system. The control mechanism was found to be accurately modelled as the electrolyzer stopped when the gases were filled in their respective tanks up to the pre-defined maximum pressure. Similarly, flow of the gases to the thruster was shut once the pre-defined minimum pressure of the tanks was reached. It was concluded that the model can be used for optimization.

A thrust of 1 N was fixed as the target for the WEP system. For any propulsion system, delivering a high specific impulse is desirable and so, maximizing the specific impulse for the current system was also affixed as an objective. A cost function that included thrust and specific impulse was selected. It was made sure that it met all the necessary

conditions, namely, being continuous, differentiable and convex. Chamber pressure, mass flow rate of propellant and cone half angle(s) of convergent and divergent section of nozzle were chosen as the design variables to minimize the cost function, which meant obtaining a 1 N thrust and the highest possible specific impulse. A MATLAB® code was created that included four functions which performed varying the design variables, running the Simulink®, running the cost function and constraining the function respectively. Genetic Algorithm was selected for the optimization as it has an ability to combine discrete, integer and continuous variables; it does not need an initial design; it can deal with non-convex, multi-modal discontinuous functions. After convergence, the average thrust achieved was exactly 1 N and a very good specific impulse of 338.9 seconds was achieved with just 0.38 g propellant consumption. Also, an excellent Total-impulse-to-power ratio (TI/**P**) of 1.19 N-s/W was achieved. Then the design variables were used to size the entire system, which fit in a 2U volume, ideal for any 6U or bigger CubeSats. A brief mission design was carried out to validate the performance of the present WEP system. For a 6U CubeSat in a circular orbit at an altitude of 500 km, a lifespan of 8.23 years was achieved due to the drag compensation made possible the WEP system. Similarly, for a 12U CubeSat, it was found that by carrying extra 1 L of water (1U capacity), making the WEP system occupy 3U, the lifespan was 9.63 years in a 500 km circular orbit. The model developed in the present work can directly be used for preliminary design and sizing of a WEP system for a CubeSat for any mission. Hence, all the objectives set out at the beginning of the present work, were successfully met.

6.2   Recommendations and future work

Having conceptually designed a WEP system for CubeSat, the present system could be tested experimentally. Also, the modelling of the system could be made more realistic by (a) considering the piping and accounting for losses in the piping; (b) considering real gas instead of ideal gas; (c) accounting for zero-g effects of outer space. As discussed, the very high stoichiometric temperature of combustion needs cooling mechanism which may be worked on in the future. Another way is to change the mixture ratio to use a fuel-rich mixture for combustion, and using the remaining oxygen in a secondary propulsion system. Lastly, continuous firing of the thruster could be explored, once the cooling mechanism is designed.
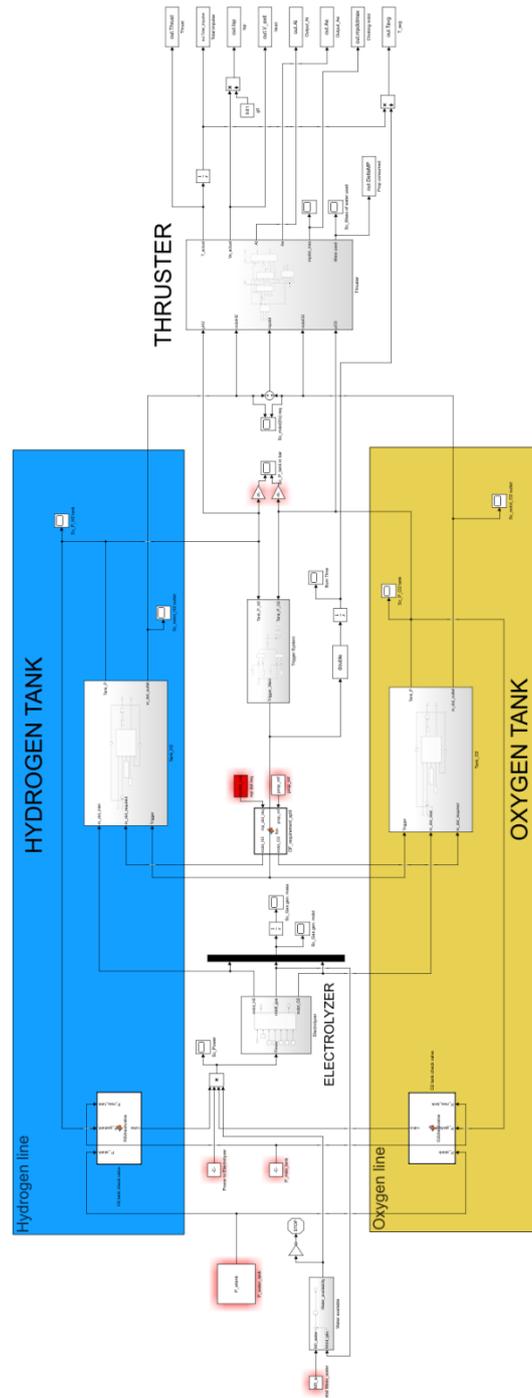
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A Appendix A: Simulink® model

List of Figures

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List of symbols

Variable	Description	SI unit
v	Velocity	m/s
Δv	Delta-v	m/s
P	Pressure	Pa
T	Thrust	N
TI	Total impulse	N-s
I_{SP}	Specific impulse	s
m	Mass	kg
T	Temperature	K
MF	Mass fraction	-
R_{gas}	Specific gas constant	J/kg.K
M	Molar mass	kg/mol
ν	Valency	-
R	Universal gas constant	J/mol.K
$\dot{P}$	Rate of change of pressure	Pa/s
$\dot{m}$	Mass flow rate	kg/s
CR	Contraction Ratio	-
ER	Expansion Ratio	-
α	Cone half angle	degree
L	Length	m
r	Radius	m
D	Diameter	m
V	Volume	m^3
h	Height	m
A	Area	m^2
TBA	Tank base area	m^2
P	Power	W
V	Voltage	V
I	Current	A
F	Faraday constant	C/mol
g_0	Gravitational acceleration	m/s^2
t	Time	sec (s)
TI/**P**	Thrust-to-power ratio	N-s/W

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