Development and Definition of a CubeSat Demonstrator for a Water Propulsion System

Entwicklung und Definition eines CubeSat Demonstrators für ein Water Propulsion Antriebssystem

Master Thesis of
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Master Thesis Work

of Mr. Colum Walter

Entwicklung und Definition eines CubeSat Demonstrators für das Water Propulsion Antriebssystem

Development and Definition of a CubeSat Demonstrator for a Water Propulsion System

Motivation:
At the site in Lampoldshausen, ArianeGroup develops, manufactures and tests orbital propulsion systems and equipment to cover the full range requirements for satellites and spacecraft. This includes chemical and electrical thrusters, pressure regulators and valves. Within the work of this thesis the propulsion system for a CubeSat based on the AGG Water Propulsion Technology shall be designed, a system architecture shall be developed and components and their mechanical interface shall be defined. Furthermore, the system's thermal design and a test environment for subsequent system tests shall be specified and designed.

Task description of the Master thesis work:

- WP1: Literature research on the current progress of the water propulsion system and on CubeSat platforms
- WP2: Development of a system demonstrator design
  - Definition of constraints and boundaries of the system demonstrator (specifications)
  - Definition, design and development of the system demonstrator
  - Identification / definition of required COTS components
  - Definition of a thermal design for the system demonstrator
- WP3: Definition and development of a test environment
  - Definition, design and development of a test environment for demonstrator tests
  - Preparing of specifications for COTS components
- WP4: System tests (dependent on progress of the water propulsion system)
  - System tests of demonstrator with available components of the water propulsion system
  - Evaluation and documentation of system tests
- WP5: Documentation
- The work shall also be documented in an internal report (Test & Design Report)

Objective:

- Overall system design of a flight demonstrator incl. a thermal design
- Development of a test- / simulation environment to complete system tests according to a predefined test plan and objective
- Performance of system tests according to predefined test plan and objective (TBC, dependent on available hardware)

The thesis will be accomplished at ArianeGroup GmbH in Lampoldshausen, Germany.

Internal supervisor:	Marius Wilhelm, M.Sc.
External supervisor:	Malte Wurdak, M.Sc.
Starting date:	01.07.2020
Submission until:	31.12.2020

Acknowledgement of receipt:
I hereby confirm that I read and understood the task of the master thesis, the juridical regulations as well as the study- and exam regulations.

Prof. Dr. S. Schlechtriem
(Responsible Professor)

External Supervisor

Signature of the student

Declaration

I, ***WALTER, Colum*** hereby certify that I have written this ***Master thesis*** independently with the support of the supervisor, and I did not use any resources apart from those specified. The thesis, or substantial components of it, has not been submitted as part of graded course work at this or any other educational institution.

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Abstract

ArianeGroup is currently developing the innovative semi-electric Water Propulsion System (WPS) based on the in-orbit production of the propellants, hydrogen and oxygen, by a polymer electrolyte membrane (PEM) electrolyzer. The propulsion system is composed of new technologies and components that shall be tested in their operational environment to verify their functionality in space. Therefore, a demonstrator system concept is developed which shall be tested on a CubeSat platform. An examination of the current developments of the WPS is performed in the first step followed by a description of the CubeSat mission by a project breakdown structure and a design and development plan for the Demonstrator Water Propulsion System (DWPS). The outcomes hereof combined with the results of a literature study on suitable CubeSat platforms result in the definition of technical requirements on the demonstrator system. These technical requirements form the foundation for the development of a concept of the DWPS which is analyzed by a Matlab calculation on the behavior of the gases produced by the electrolyzer. For the demonstrator propulsion system, a preliminary mission is defined in a last step. It provides an overview of the expected performance of the system, reviews orbit and launch possibilities and defines the operational procedure in space. Furthermore, a link budget is calculated that gives the data rate transmittable during a ground station flyby of the CubeSat.

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List of Symbols

a	Semi-major axis [m]
A	Surface area [m^2]
A	Empirical constant – isobaric heat capacity [J/(K·mol)]
A_s	Empirical constant – water vapor pressure
B	Empirical constant – isobaric heat capacity [J/(K^2·mol)]
B	Bandwidth [Hz]
$\vec{B}_{earth}$	Earth's magnetic field [T]
B_s	Empirical constant – water vapor pressure
B_V	Second Virial coefficient [m^3/ kg]
c	Velocity [m/s]
c	Speed of light [m/s]
C	Empirical constant – isobaric heat capacity [J/(K^3·mol)]
C_d	Discharge coefficient [-]
C_d	Drag coefficient [-]
c_p^0	Isobaric heat capacity of an ideal gas [J/mol·K]
C_s	Empirical constant – water vapor pressure
C_V	Third Virial coefficient [m^6/ kg^2]
D	Empirical constant – isobaric heat capacity [J/(K^4·mol)]
D	Data rate [bit/s]
d	Distance from satellite to ground station [km]
E	Empirical constant – isobaric heat capacity [J·K/mol]
E_b/N_0	Energy per bit to noise power spectral density ratio [dB]
F	Empirical constant – integration constant [J/mol]
F	Force [N]
$F_{G/T}$	Antenna gain-to-noise temperature - figure of merrit [dB/K]
$\boldsymbol{\mathcal{F}}$	Faraday Constant [A·s/mol]
f	Frequency [Hz]
g_0	Gravitational accelaration [m/ s^2]
G	Gravitational constant [m^3/(kg·s^2)]
G	Gain [dB]
H	Mass specific enthalpy [J/kg]
H	Orbit altitude [km]
I	Impulse [Ns]
I	Current [A]
I_{SP}	Specific Impulse [s]
L	Loss [dB]
m	Mass [kg]
$\dot{m}$	Massflow [kg/s]
M	Molar mass [kg/mol]
M_E	Earth's mass [kg]
$\vec{m}$	Magnetic moment [A·m^2]
n	Amount of substance [mol]
$\tilde{p}$	Average pressure [Pa]
p	Pressure [Pa]
P	Orbital Period [s]
P	Power [W]
$P_{[dBm]}$	Power [dBm]
q	Mass specific heat [J/kg]
$\mathfrak{R}$	Gas constant [J/(mol·K)]

R	Specific gas constant [J/(kg·K)]
R_E	Earth's radius [km]
SNR	Signal-to-Noise ratio [dB]
T	Time [s]
U	CubeSat Unit [-]
V	Volume [m^3]
v	Molar Volume [m^3/mol]
$\widetilde{w}$	Average velocity [m/s]
w	Velocity [m/s]
w_t	Mass specific technical work [J/kg]
Z	Real gas coefficient [-]
z	Charge number [-]

Greek Symbols

ε	Surface ratio [-]
φ	Relative humidity [-]
κ	Isentropic exponent [-]
λ	Wavelength [m]
η_{WT}	Water tank expulsion efficiency [-]
η_{ELY}	Electrolyzer power efficiency [-]
ρ	Density [kg/m^3]
τ	Torque [Nm]
τ	Time independent constant [-]

Indices

0	Stagnation point
1	State 1
2	State 2
a	Ambient
atm	atmosphere
CS	CubeSat
D	Drag
$[dBm]$	Unit of power
E	exit
$EIRP$	Equivalent isotropically radiated power
ELY	Electrolyzer
FS	Free space
GS	Ground station
GT	Gas tank
H_2O	Water
H_2	Hydrogen
id	Ideal
L,CS	Line of CubeSat
$MEOP$	Maximum Expected Operating Pressure
MOP	Minimum Operating Pressure
o	Orifice
Pol	polarisation
s	Saturated
sat	Satellite
tot	total
Tx	Transmitter
WT	Water tank
Man	Maneuver

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Abbreviations

ADCS	Attitude Determination and Control System
AG	ArianeGroup
AOCS	Attitude and Orbit Control System
AVUM	Attitude Vernier Upper Module
CDHS	Command and Data Handling system
CDR	Critical Design Review
COTS	Commercial Of The Shelf
CSD	Canisterized Satellite Dispenser
DAS	NASA’s Debris Assessment Software
DoD	Depth of Discharge
ECSS	European Cooperation for Space Standardization
ECU	Electrolyzer Control Unit
ELY	Electrolyzer
EOL	End-Of-Life
ESA	European Space Agency
FCV	Flow Control Valve
FDV	Fill and drain valve
GEO	Geosynchronous Equatorial Orbit
GNSS	Global Navigation Satellite System
GPS	Global Posistioning System
GTO	Geostationary Transfer Orbit
HPM	High Power Mode
IOD	In-Orbit Demonstration
IRS	Institute of Space Systems (Institut für Raumfahrtsysteme)
ITU	International Telecommunication Union
KSAT	Kongsberg Satellite Services AS
LEO	Low Earth Orbit
LPM	Low Power Mode
NCR	Non-Conformance Report
PDR	Preliminary Design Review
ROF	Ratio of Oxidizer to Fuel
SSC	Swedish Space Corporation
SSMS	Small Spacecraft Mission Service
SSO	Sun-Synchronous Orbit
STK	Systems Tool Kit
TEC	Total electron Content
THR	Thruster
U	CubeSat Unit

1 Introduction

A key subsystem on most satellites is its propulsion system. A propulsion system not only determines the satellite's mission capabilities but commonly defines the satellites lifetime in orbit. Most chemical satellite propulsion systems are based on hydrazine thrusters, a well proven and reliable technology. However, due to its toxicity and mutagenicity the space community is looking for alternative, green solutions

Aside from legislative uncertainties about a possible prohibition of the use of hydrazine as propellant in the future due to the European REACH regulation, propulsion systems based on so-called green propellants can offer performance and especially significant economic benefits. ArianeGroup, as a main European supplier for space propulsion systems, matures and researches different technologies for green propulsion systems, such as propulsion systems based on hydrogen peroxide (H_2O_2) or ammonium dinitramide (ADN) [1].

A novel technology with a high potential which is currently under development at ArianeGroup is the water propulsion technology - a technology based on the in-orbit production of oxygen and hydrogen by a water electrolyzer and the combustion by a catalytic thruster. A main advantage of such a propulsion system is the high performance that can be achieved with the propellant combination as well as the economic benefits because the system can be preloaded at the manufacturer's site and thus costly launch site operations can be minimized or even be eliminated.

The spacecraft industry especially with classical operators is very conservative: customers are reluctant to fly new technologies if they are not already flight proven. Therefore the motivation of ArianeGroup is to fly a demonstrator, to prove the function of this new technology in orbit and thus to pave the way for commercial applications.

A CubeSat with its small size, short lead times, reduced complexity and available commercial of the shelf components (COTS) represents the ideal platform for such an in-orbit demonstration (IOD). It is the objective of this thesis to define a demonstrator of the Water Propulsion System's (WPS's) key technologies, the electrolyzer, the thruster and a control unit, assembled to the Demonstrator Water Propulsion System (DWPS), that can be tested on a CubeSat platform.

The initial chapter describes the water propulsion technology, illustrates the current development of the actual WPS and gives an overview of required developments for the demonstrator propulsion system. Thereafter, chapter 3 summarizes general features of CubeSats and describes commercially available CubeSat platforms of three different suppliers. The overview of platforms allows the determination of limitations of a CubeSat platform for the demonstrator mission and the DWPS.

The next segment, chapter 4, defines the technical requirements on the demonstrator propulsion system which are derived on one hand from the actual WPS and on the other hand from the limitations, boundaries and properties of the CubeSat platform. These requirements result in the design of the DWPS which is described and analytically evaluated in the subsequent chapter.

Conclusively, this thesis describes the mission of the IOD in chapter 6, starting with the mission requirements, reviewing launch possibilities, defining an orbit for the mission, specifying a mission profile for the CubeSat and evaluating the ground operation from the ground station at the Institute of Space Systems at the University of Stuttgart.

2 Background & State of the Art

Before defining a CubeSat platform, designing DWPS components and defining a system architecture of the propulsion system, it is essential to attain an overview of the current state of development of the Water Propulsion System and its components.

2.1 Water Propulsion System State of the Art

During its nominal operational life the Water Propulsion System shall be capable of performing various maneuvers, including orbit raising, orbit and attitude control as well as the final de-orbit maneuver for a satellite. Therefore, the WPS produces gases with an electrolyzer that is fed with deionized water from a water tank. The gases are stored and pressurized by the electrolyzer in separate gas tanks, are then combusted by a catalytic ignition and accelerated by a nozzle to generate thrust. The main components of the WPS are illustrated in the following block diagram which gives an overview of the working principle.

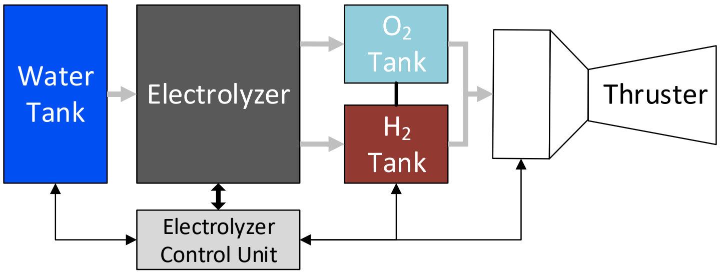

Figure 1: WPS block diagram of main components

This novel propulsion concept is based on technologies that have never been used in this way in space before. Especially the three components, the electrolyzer, its control unit and the thruster represent innovative subsystems that require a great amount of resources and effort for the design and development. Other components such as the fluidic components composed of the water and gas tanks and their interfaces generally require less effort during the development. Furthermore, commercially available gas and liquid propellant tanks for satellite propulsion applications are assumed to be compatible with the WPS and its fluids with the result that no new development for such a system is intended. At the current state of developments, preliminary models of these main three subsystems have been manufactured and tested. The following subchapters shall provide an overview of their composition and functionality.

2.1.1 Electrolyzer & Electrolyzer Control Unit

Conventional electrolyzers for earth-bound applications generate gases only up to a relative low pressure and contain a high amount of gaseous and liquid water. Therefore, in a second step the produced gases are pressurized by compressors and dried by using gravitational forces, e.g. within a centrifuge for the actual use. On a spacecraft without access to maintain or repair any defects, with strict volume and mass limits, the use of passive mechanisms is always preferred. Thus, a cell of the WPS' electrolyzer is based on the Proton Exchange Membrane (PEM) technology, a technology capable of producing hydrogen and oxygen with a high purity and pressurizing the gases decoupled from the actual water inlet pressure. Instead of using an alkaline solution as the ion conducting media, the PEM technology relies on a solid electrolyte conducting membrane. The solid electrolyte membrane, which is connected to a catalyst material on both sides, is placed between two electrodes which apply a potential across the membrane, as it is shown in the following scheme of the WPS' PEM electrolyzer in Figure 2. The water arrives at the membrane in its gaseous state, diffuses through to the anode side when a concentration gradient across the membrane is present and is split according to the reaction stated in equation (2.1) [6][7].

$$2H_2O \rightarrow 2H^+ + 2e^- + O_2 \tag{2.1}$$

The positively charged hydrogen ions conduct back through the membrane towards the negatively charged cathode. For this to happen, the presence of water is essential – the membrane must be wet. At the cathode side the hydrogen ions adjoin and form the resulting hydrogen according to (2.2) [6][7].

$$2H^{+} + 2e^{-} \rightarrow H_2 \qquad (2.2)$$

For the decomposition of water to the gaseous products hydrogen and oxygen, a minimum voltage of 1.48V, corresponding to the enthalpy of reaction for liquid water under standard conditions (1 bar and 298.15K), must be applied across the membrane [8].

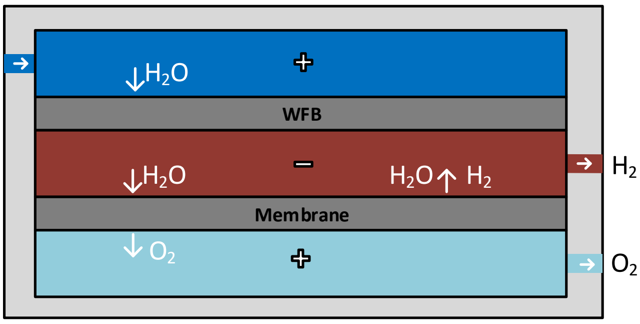

Figure 2: PEM electrolyzer cell scheme of the WPS' electrolyzer

An important element in the electrolyzer is a further semipermeable membrane, the Water Feed Barrier (WFB). The water enters the electrolyzer cell and diffuses through the membrane by an osmotic pressure which arises with a concentration difference across the membrane. This concentrating difference occurs as soon as the gas production is activated so that the supply of water is automatically controlled by the electrolyzer. Furthermore, the osmotic driven water feed even works when the hydrogen side of the WFB is under higher pressure than the water side, making it possible to pressurize the produced gases to a pressure which is decoupled of the actual water side pressure. However, with an increasing pressure on the hydrogen side, more and more hydrogen diffuses through the WFB to the water side resulting in bubbles at the membrane which obstruct the water feed and eventually dry out the membrane. This can be prevented by applying a voltage across the WFB and attaching a catalyst to the WFB. As a result, the molecular hydrogen diffusing through the WFB cleaves into hydrogen ions which are deflected back to the hydrogen side by the applied voltage[6][7]. The water feed barrier is therefore used for the passive feed of water by the osmotic pressure but also as an active hydrogen pump to prevent the backflow of gas to the water side.

Between the top plate and the WFB, the WPB and the electrolysis membrane and between the electrolysis membrane and the bottom plat, an essential component in this design, the so-called bi-polar plates, combine a number of functions. On the one hand, the bi-polar plates ensure that the fluids are distributed homogenously allowing a flow in radial direction for the in- and outlet and in axial direction towards the membranes. The loadable plate mechanically sustain the membranes on both sides and create flow fields within the plates with cavities arranged in certain patterns. On the other hand, these plates possess electrical connections, are in direct contact with the catalyst at the membrane and thereby act as the electrodes [6]. The bipolar plates are also shown in Figure 2 as dark blue, red and light blue plates.

The WPS electrolyzer shall be equipped with a stack of cells that can be operated in two modes, the Low Power Mode (LPM) at 50 watts and the High Power Mode (HPM) of 250 watts. These modes represent the gas production for two different types of maneuvers: Station keeping and orbital raise maneuvers. A satellite's attitude is influenced by various perturbations, such as atmospheric drag and gravity alterations. To counteract these perturbations, the thruster must perform short maneuvers requiring only little amount of fuel. For these station keeping maneuvers, the electrolyzer shall be operated in the LPM. The HPM, in which the electrolyzer shall be powered with 250 watts, shall be used preliminary to maneuvers that require a high amount of gases. These maneuvers include orbit raise or de-orbit maneuver.

The PEM technology has been researched and investigated since the beginning of the Water Propulsion project at AraineGroup in 2014 and a first electrolyzer was developed in cooperation with the Institute of Space Systems at the University of Stuttgart within the scope of a doctoral thesis. By now, various configurations have been developed and have led to a demonstrator model that is currently being tested.

For these tests the electrolyzer is operated by lab equipment, which regulates and controls the power supply. The actual WPS electrolyzer flight model will be attached to the Electrolyzer Control Unit (ECU) which main task it is, to automatically operate the electrolyzer in space. Different to conventional chemical bi-propellant thrusters a simple on and off command by the satellite's on-board computer (OBC) is not sufficient. Therefore, it is the aim to provide a "plug-and-play" design of the WPS to simplify its interface and reduce the complexity for a potential costumer. For this purpose, the ECU is equipped with a microprocessor including internal storage. The microprocessor interfaces to the satellite platform, receives simple commands by the OBC and the required power by the satellites power system. For the power supply of the electrolyzer, each cell must be attached to two different electrical circuits, one to power the hydrogen pump and a second to power the actual electrolysis process. For the power supply of the electrolysis process, the ECU receives the power by the satellite's electrical power system and then converts the power by a DC/DC converter to a controlled current. This makes it possible to preset the gas generation rate since it directly correlates to the current. The voltage depends on various influences, such as the pressure and temperature which would make it hard to determine the gas production rate for a voltage controlled power supply. On the other hand, the circuit powering the hydrogen pump shall be controlled by a DC/DC converter to a fixed voltage. The actual power required for the hydrogen pump process is once again dependent on the pressure of the hydrogen, however a constant voltage of around 0.8 volt is considered to be high enough to prevent the back flow of hydrogen through the WFB. Each cell shall be powered individually. A serial connection of cells was initially considered but then discarded to allow a redundant power supply for each cell.

A further task of the ECU is the assessment of various sensor and power measurements. An impedance measurement shall allow the evaluation of the membrane degradation. Temperature and pressure sensors provide the data required by the ECU to power heaters and to allow an assessment of the electrolyzer performance and behavior in space.

Even though the name of the ECU might lead to a different conclusion, the electrolyzer control unit is in charge not only of the regulation and control of the electrolyzer but also manages the temperature control and valve actuation for the entire assembled and compound WPS. Therefore, it possesses switches, which connect the satellite's electrical power system to heaters and valves which can be activated by a command from the ECU's microprocessor.

2.1.2 Thruster

A thruster for space propulsion systems is mainly characterized by its thrust and its efficiency. The efficiency of a thruster can be defined as the thrust F a propulsion system can produce per mass flow $\dot{m}$ of propellant, which is quantified by the specific impulse I_{SP} as shown in equation (2.3).

$$I_{SP} = \frac{F}{\dot{m}\, g_0} \tag{2.3}$$

Thus, a propulsion system with a high specific impulse can generate more thrust with less propellant. On the other hand, the thrust of a propulsion system based on the reaction thrust principle is calculated as follows.

$$F = \dot{m}\, \widetilde{w}_e + (\tilde{p}_e - p_a)\, A_e \tag{2.4}$$

The thrust is composed of the sum of the mass flow $\dot{m}$ multiplied by the average exit velocity $\widetilde{w}_e$ with the pressure difference at the nozzle outlet $(\tilde{p}_e - p_a)$ multiplied by the nozzle outlet area A_e [10].

For a more detailed view of the specific impulse, an ideal flow in the nozzle (unidimensional flow with isentropic expansion) and an adapted nozzle $(p_e = p_a)$ shall be assumed in the following. The ideal exhaust velocity can be calculated according to equation (2.5) by using the isentropic relations.

$$w_{e,id} = \sqrt{\frac{2\,\kappa}{\kappa - 1}\,\frac{\Re}{M}\,T_0\left[1 - \left(\frac{p_e}{p_0}\right)^{\frac{\kappa-1}{\kappa}}\right]} \tag{2.5}$$

In the next step, equation (2.5) shall be inserted in equation (2.4), which then again inserted into (2.3) results in the equation (2.6) for the specific impulse I_{SP}

$$I_{SP} = \frac{1}{g_0} \cdot \sqrt{\frac{2 \cdot \kappa}{\kappa - 1}\,\frac{\Re}{\boldsymbol{M}}\,T_0\left[1 - \left(\frac{p_e}{p_0}\right)^{\frac{\kappa-1}{\kappa}}\right]} \tag{2.6}$$

The WPS based on the thrust generation by the combustion of hydrogen and oxygen possesses a decisive advantage to classic hydrazine thrusters regarding the specific impulse in equation (2.6). Since the I_{SP} is proportional to $\sqrt{\frac{1}{M}}$, the molecular weight of the exhaust product plays an important role in the thrusters efficiency. Hydrogen as a molecule with the lowest possible molecular weight makes the propellant combination of oxygen and hydrogen very effective. For conventional hydrogen-oxygen propulsion systems a fuel-rich ratio of oxidizer to fuel is chosen so that a certain amount of hydrogen, with its low molecular weight, remains in the exhaust gas while enough hydrogen reacts with oxygen to achieve a sufficient temperature. A trade between the combustion temperature and the mean molecular weight of the exhaust gas mixture results in the highest specific impulse at a ratio of oxidizer-to-fuel (ROF) of around 3.5. The influence of the ROF on the specific impulse and the combustion chamber temperature is shown in Figure 3.

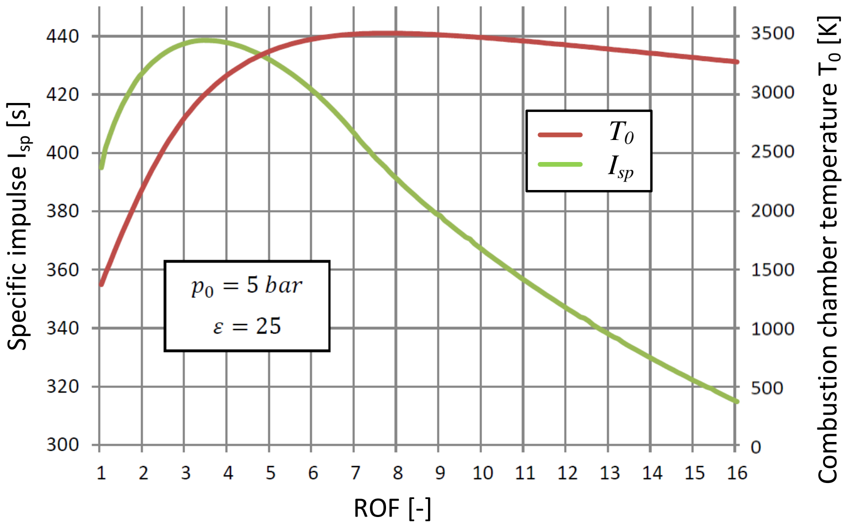

Figure 3: Behavior of the specific impulse and the combustion chamber temperature for the combustion of hydrogen with oxygen in dependency of the ROF [10]

This figure also shows the actual challenge of the WPS thruster design. The electrolyzer provides the gases in a stoichiometric ratio which is given by a ROF of 8. The stoichiometric combustion of hydrogen and oxygen results in a higher average molecular mass of the combustion gases thereby reducing the specific impulse. But, even more critical, the stoichiometric ratio results in the highest release of energy and generates the highest temperatures of around 3500 K, easily exceeding the maximum temperature of available combustion chamber and nozzle materials [11].

The thruster's chamber material, a PtRh alloy, can withstand temperatures up to 1600°C and must be actively cooled to be operated under the severe thermal loads. To reduce the maximum temperature at the combustion chamber wall, the gas flow is separated to allow the ignition and initial combustion in a pre-combustion chamber so that the resulting hot gases mix with the remaining gases in the actual combustion chamber. The hot gases are fed to the combustion chamber at the center while the cold gas enters the chamber close to the chamber wall. This is a cooling concept called film cooling, where the outer cold gas film creates a heat barrier and actively reduces the temperature at the combustion chamber and nozzle wall [10].

For the WPS thruster, the oxygen gas flow is split up, so that a small amount enters and reacts in the pre-combustion chamber while the rest is added to the actual combustion chamber separately. On the contrary, the entire mass flow of hydrogen is led through the pre-combustion. The ROF in the pre-combustion chamber is also strictly limited by the maximum temperature. This limit is set by the catalyst which reduces the reaction energy of oxygen and hydrogen and hence, initiates the combustion. The catalyst for the WPS thruster is composed of a porous carrier material coated with platinum. The maximum temperature the catalyst can withstand is limited to around 800°C to 1100°C and is achieved by a fuel rich ROF of 0.8 [10].

The mass flow of oxygen and hydrogen is predefined by orifice plates, which are often used for space propulsion systems. The gas flows from the gas tanks towards the thruster and is accelerated at the orifice due to the decreased diameter. If the pressure difference between up- and downstream of the gas is big enough, the gas reaches the speed of sound at the orifice, which decouples the flow from the downstream conditions. In this case, this means that the mass flow is only dependent on the inlet pressure and not on the pressure within the combustion chamber, which can oscillate especially during the phase of ignition. The mass flow through a choked orifice can be calculated as stated in equation (2.7).

$$\dot{m} = C_d\, p_{in}\, A_o\, \kappa \sqrt{\frac{\left(\frac{2}{\kappa+1}\right)^{\frac{\kappa+1}{\kappa-1}}}{\kappa\, R\, T}} \tag{2.7}$$

For the orifice to be choked, the pressure difference of inlet (gas tank) and outlet pressure (combustion chamber) must comply with equation (2.8) [12].

$$\frac{p_{out}}{p_{in}} < \left(\frac{2}{\kappa+1}\right)^{\frac{\kappa}{\kappa-1}} \tag{2.8}$$

Equation (2.7) shows, that the mass flow is only dependent on the gas properties, such as the heat capacity ratio κ and its gas constant R, the thermodynamic properties in the gas tanks including the inlet pressure p_{in} and the gas temperature T, as well as the geometry and properties of the orifice, given by the orifice area A_o and the discharge coefficient C_d. A Preliminary study on the WPS thruster, the bachelor thesis [10], assumes very low mass flow losses with a C_d value of 0.97. It shall be noted, that in other literatures the discharge coefficient is considered to influence the mass flow more considerably and is often stated to be around 0.6 for sharp edged orifice plates, as they are to be used within the thruster [12][13][12]. The exact flow behavior through the designated orifices will be determined in tests that will be performed in near future.

The consecutive combustion of the gases in two combustion chambers allows the reduction of temperature but also results in a decrease of specific impulse. The overall goal for the WPS's thruster is to achieve an I_{sp} of greater than 300s. This shall also be possible if the thruster is fed by wet gases, since the electrolyzer produces gases that contain a certain amount of vaporized water. The thruster is designed to generate a nominal thrust of 2N.

2.1.3 Current Development Status

Each novel technology developed for the use in the WPS is quantified by a maturity status which is defined by a Technical Readiness Level (TRL). A full mature technology possesses the highest level of TRL9. A TRL9 technology is distinguished by a successful mission operation and has fulfilled its service for an assigned mission. On the contrary, a maturity of TRL1 describes, that basic principles, that are to be used for a technology, have been observed by scientific research and can be translated into a target-oriented research and the initial development of a technology [2]. The definition for each TRL, published by the International Organization for Standardization (ISO), was adopted and slightly modified by the European Cooperation for Space Standardization (ECSS) and is reproduced in the following table.

Table 1: Definition of the Technical Readiness Levels (TRLs) as specified in the ECSS-E-AS-11C standard [3]

TRL	Definition
TRL1	Basic principles observed and reported
TRL2	Technology concept and/or application formulated
TRL3	Analytical and experimental critical function and/or characteristic proof-of-concept
TRL4	Component and/or breadboard functional verification in laboratory environment
TRL5	Component and/or breadboard critical function verification in a relevant environment
TRL6	Model demonstrating the critical functions of the element in the relevant environment
TRL7	Model demonstrating the element performance for the operational environment
TRL8	Actual system completed and accepted for flight ("flight qualified")
TRL9	Actual system "flight proven" through successful mission operations

As a first step for the classification of the subsystem's TRLs, the relevant environment must be specified. The relevant environment for a technology defines a set of operational environments that influence its function decisively so that it is generally required to evaluate the technologies behavior under these environments to achieve a higher maturity. The relevant environment for components can vary, even if the components are used in the same operational environment. This also applies for the WPS's subcomponents.

For the thruster, the relevant environment that influences its operation in space is the vacuum. Compared to the atmospheric pressure that surrounds the thruster in a laboratory environment, the thruster is exposed to a different ambient pressure and a drastically reduced heat transfer that is normally provided by the surrounding air. This environment can be reproduced in a vacuum chamber allowing the thruster to be tested in its relevant environment.

On the contrary, vacuum but also microgravity is considered to be the relevant environment for the electrolyzer operated in space. The vacuum influences the electrolyzer similar to the thruster by increasing pressure gradients and reducing heat transfer capabilities. Additionally, Forces and processes, which are usually mainly or partially masked by gravitation, e.g. surface tension and intermolecular forces become more dominant and can affect the function or performance of a technology. The microgravity is determined as a relevant environment for the electrolyzer, because especially the phase separation of water from its electrolysis gases, hydrogen or oxygen, might be influenced by the environment.

Initial tests of the thruster with its catalytic ignition have been performed and have verified the functional performance of its technology in a vacuum chamber so that the thruster technology can be classified with a maturity of TRL 5. Further tests shall show, that the thruster is also compatible with the gases produced by the electrolyzer which generally contain a certain amount of gaseous water. An assembled test shall make it possible to directly show the combined functionality and performance of electrolyzer and thruster.

As described above, the electrolyzer technology has been investigated thoroughly at ArianeGroup and the Institute of Space Systems (IRS). The results thereof have led to the design of a development model of the electrolyzer and a control unit at breadboard level for the WPS. Currently, these components are being manufactured and tests will follow in the next step. With the completion of the tests in the laboratory environment it is considered that both technologies achieve a maturity level of TRL 4.

2.2 CubeSat Technology Demonstration

Since the microgravity environment might influence the functionality and performance of the electrolyzer and consequently, of the entire WPS, an examination under this relevant environment is considered valuable for the development progress and the commercialization of the propulsion module. While it is comparatively simple to generate a vacuum to test the WPS, the capabilities for tests in a microgravity environment are limited. Aside from the actual in-orbit demonstration only two other test possibilities for microgravity examination, drop towers and parabolic flights, were identified. Drop towers, such as the 146 meters high Zarm drop tower in Bremen, create a microgravity environment within a capsule that is dropped from the top of the tower or, to increase the test duration, catapulted to the peak. Thereby, a maximum microgravity environment of not even 10 seconds can be created [4]. The second alternative is depicted by parabolic flights. An aircraft rising and then declining at a 45° angle can achieve a micro gravity environment for around 20 seconds [5]. These short durations cannot provide the microgravity environment for tests on the electrolyzer that produces gases at a rate of only a few grams per hour.

Therefore, the only possibility is an in-orbit examination of the WPS where the system can be tested under the real operational environment and its performance under these conditions can be determined. For this, a CubeSat platform as the host for a Demonstrator Water Propulsion System (DWPS) was identified as most suitable. The In-orbit demonstration with a CubeSat has the following advantages:

- Less complex, highly standardized satellite platform
- Reduced resources for CubeSat platform with available Commercial Of The Shelf (COTS) components
- Long term in-orbit testing – up to 5 years possible
- Reduced launch cost as secondary payload

On the contrary, it must be noted that the actual WPS, powered with up to 250 watts, generating a thrust of around 2N and supplied with the amount of water required for an entire mission of a satellite that may weigh more than a ton, cannot be implemented into a CubeSat. The in-orbit demonstration with the demonstrator propulsion system is not capable of increasing the actual WPS' TRL level. For an increase of the TRL level above TRL 5, the actual system must be tested under the relevant or operational environment. However, tests can still proof the functionality of the operating concept, will allow the determination of any performance deviations compared to ground tests and is considered a highly valuable feature to attract potential costumer. This is why a Demonstrator Water Propulsion System is currently being developed at the ArianeGroup parallel to the actual WPS. The DWPS shall contain the three main components, the electrolyzer, the ECU and the thruster, but scaled to fit the CubeSat envelope. Parallel to this thesis, three other students are working on the thruster design, the ECU and the electrolyzer design for the CubeSat propulsion system. Thereby, the electrolyzer design includes the implementation of the gas tanks which shall be directly attached to the electrolyzer to save space and reduce the mass of the design.

The subsequent paragraph shall give an overview of the entire CubeSat demonstrator project by identifying all relevant products within the project. It shall show the upcoming development tasks and provide the framework on which this thesis is based. The entire project can be split up in three different segments: a ground segment, a space segment and a launch segment. A project breakdown structure is shown in the following scheme, Figure 4, and is described in more detail thereafter.

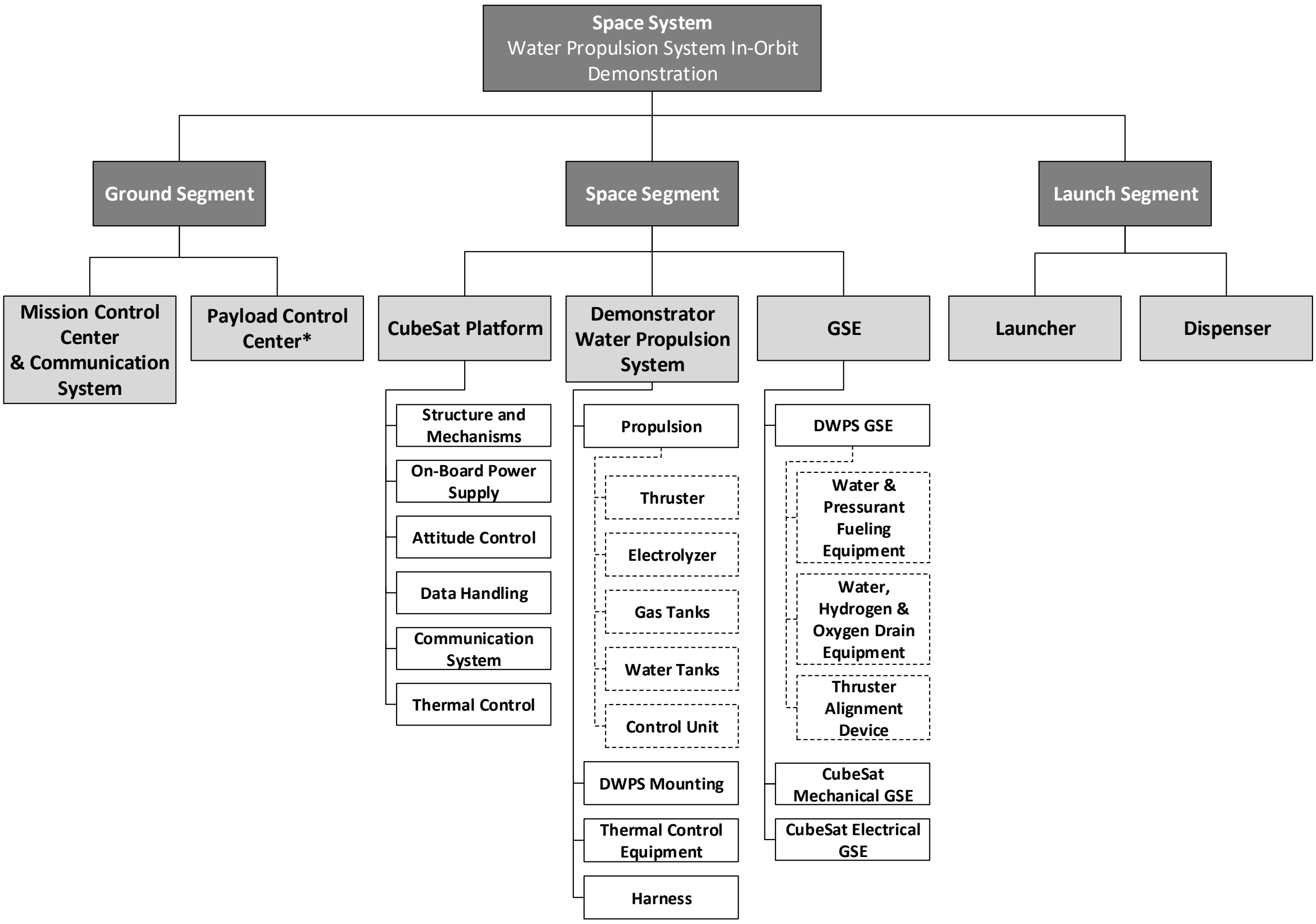

Figure 4: Water Propulsion System in-orbit demonstration project breakdown structure

The **Ground Segment** is composed of a mission control center with its communication system as well as payload control center. The task of the mission control center includes the operation of the CubeSat by tracking and communicating with the satellite. Payload data transmitted by the CubeSat and received by the mission control center will be sent to the payload control center, which is in charge of processing and evaluating the data, such as measurement data and status information of the DWPS.

The **Launch Segment** consists of the actual launcher and a dispenser. Various different launch systems provide the access to space for CubeSats. An overview thereof is shown in chapter 6.2. The dispenser is in charge of connecting the CubeSat to the launcher, protecting it during launch and releasing it once in the designated orbit.

The **Space Segment** contains the CubeSat platform, Ground Support Equipment (GSE) and the payload, the DWPS. The main subsystems of the CubeSat platform are the structure & mechanisms, the electric power supply system, the attitude control system, a data handling system and a communication system complemented by a thermal control system. The platform with its subsystems is described in chapter 3 in more detail. The DWPS consists of the actual propulsion system as well as the mounting structure, thermal control equipment and the harness. All components must be designed to suit the CubeSat structure, power supply and communication interface. On the other hand, the GSE provides the means to handle the CubeSat during ground operations.

ArianeGroup, as a specialist for space relevant propulsion systems possesses the resources and technical know-how to design and develop the DWPS. However, since the propulsion module will be used only this one time, COTS components are designated for the use whenever possible. This applies for any component other than the three relevant technologies: the electrolyzer, the thruster and the ECU. The design and development of a CubeSat platform, the operation of the satellite in space and the preparation and organization of the launch shall be conducted by an external supplier.

This means that, aside from a payload control center, which evaluates the DWPS performance data, the ground and launch segment shall be provided by an external partner. This also includes the CubeSat platform which is part of the space segment. Various CubeSat companies provide all-inclusive services including the design of a platform, the testing for acceptance of the final CubeSat, the arrangement of a launch slot and launch preparations as well as regulatory duties including the registration of the satellite and the filing for communication frequencies. A main part of this thesis focuses on the search and selection of such a supplier. The results thereof are summarized in chapter 3. Additionally, a possible cooperation with the Institute of Space Systems (IRS) at the University of Stuttgart was investigated to research the compatibility of the designated CubeSat communication system with the IRS ground station. The IRS, which is currently preparing for the operation of two CubeSats, could also provide the necessary ground control center and the ground station communication system for the CubeSat operation. The results are summarized in chapter 6.5.

As shortly depicted above, the DWPS is currently being designed and developed by four students in total. The subsystem and system design is thereby mainly influenced by the R&T activities on the actual WPS on the one hand, and by the CubeSat platform on the other hand. The design and development plan for the mission's space segment illustrating the influence of the R&T activities and the CubeSat platform on the DWPS is shown in Figure 5. It can be seen that the R&T results are used as input for the component design. The CubeSat design mainly influences the DWPS system design, which in return influences the component design. Requirements and capabilities of the platform and the DWPS are adjusted iteratively resulting in the preliminary design of the DWPS system.

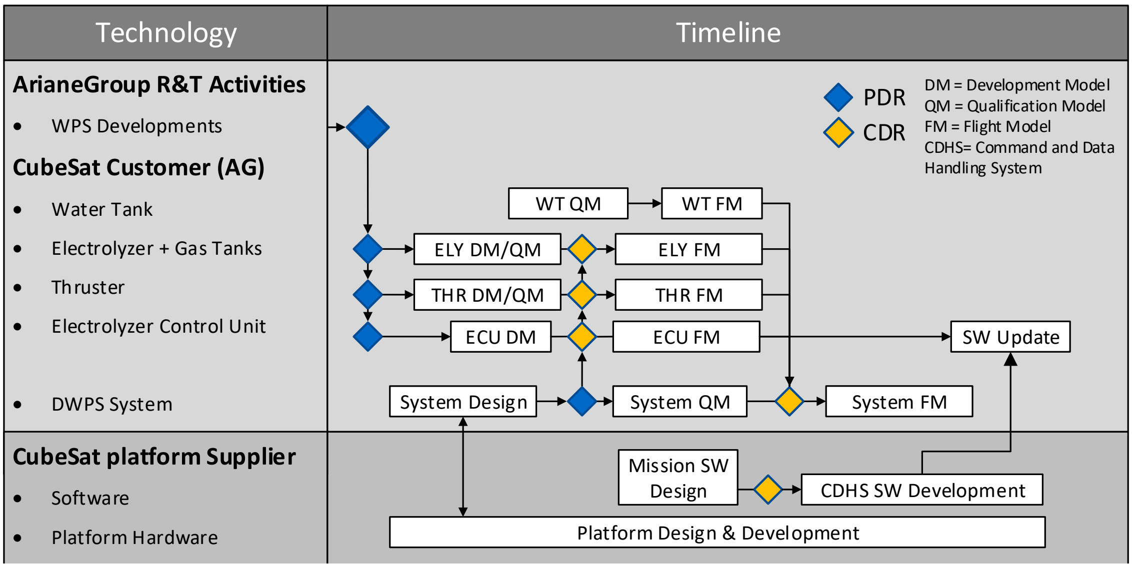

Figure 5: Design and development plan for the DWPS IOD space segment

Within the development of the DWPS, this thesis focuses on the system design of the propulsion module. Therefore, in the first step, a suitable CubeSat platform and provider is selected. Technical requirements defined on one side by the ArianeGroup's internal R&T developments and on the other side by the CubeSat platform are identified. With these requirements, given in the specifications described in chapter 4, each subsystem is defined and boundaries and requirements are specified (chapter 5). Thereafter, a system design is derived which includes the DWPS system architecture, its interfaces and measurement devices as well as an analysis of the DWPS' performance. In the last step, a preliminary mission definition is included which gives an overview of possible launch scenarios, defines a range of possible orbits and investigates the compatibility of the CubeSat communication system with the ground station of the Institute of Space Systems. The work flow of this thesis is illustrated in Figure 6. It must be noted, that the focus during the subsystem definition in this work is laid on the propulsion system rather than the thermal equipment and structure which must be defined in more detail once the main elements of the DWPS have been defined and designed.

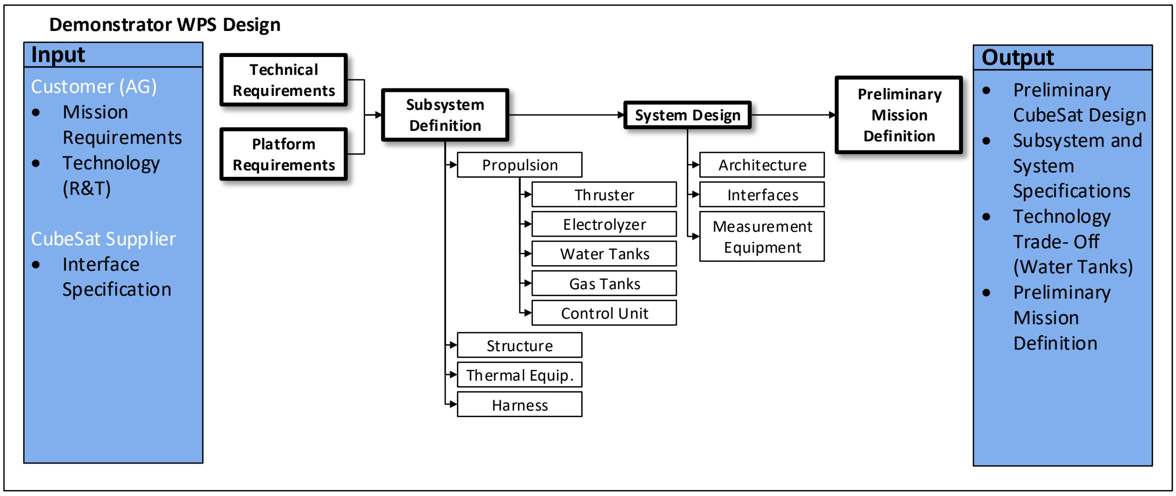

Figure 6: Framework of this thesis

2.3 CubeSat Propulsion Systems

Even though ten years ago not a single CubeSat had been equipped with a propulsion system, nowadays, a broad range of technologies have been developed and have reached high maturity levels. From conventional hydrazine propulsion systems to green propellant, cold gas, electrical propulsion systems and even unconventional systems such as sun sails and electromagnetic tethers have been developed. Before taking a closer look at CubeSat platforms and defining requirements on the DWPS the following chapter shall give an overview of possible CubeSat propulsion systems. For the purpose of this thesis only chemical propulsion systems shall be reviewed to limit the amount of information and to focus on the design of systems that are similar to the design of the WPS and the DWPS. It shall provide information on used components, show the current state of the art for a propulsion system at this size and shall highlight characteristics that are typical and important for the design of a CubeSat propulsion system.

2.3.1 Solid Propellant

A highly unconventional CubeSat propulsion system is constituted by the solid propellant based CubeSat Delta-V Motor CDM-1. This CubeSat propulsion system by Digital Solid State Propulsion Inc can provide a single high delta-V maneuver or a maneuver for the de-orbit of the CubeSat. With a burn time of 3 seconds the 450 gram motor generates a maximum thrust of 186 N to reach a total impulse of 226 Ns. With its cylindrical shape it fits into the deployment spring of conventional CubeSat dispensers and can be attached to the CubeSat's structure without taking up additional space in the actual CubeSat envelope. However, it must be noted, that at these high thrust levels a small thrust vector deviation can result in high rotation rates which might even bring the CubeSat off its nominal course. A very robust attitude control system must be implemented to prevent this. Furthermore, pyrotechnics, such as the HTPB based propellant of this propulsion system should generally be avoided since its use is prohibited by the CubeSat Design Standard (CDS) and will limit launch possibilities of the CubeSat [15][16][17].

2.3.2 Mono-propellant

Mono-propellants represent a propulsion system with a low complexity while still providing a high thrust at a high efficiency. Hydrazine mono-propellants make up the largest number of operated chemical propulsion system for satellites. The long-term stability and simple use make hydrazine so favorable. It can be assumed, that conventional propulsion systems used in large spacecraft for small pointing maneuvers, such as the 1N hydrazin thruster by the ArianeGroup, can be incorporated into a CubeSat platform as a main propulsion system [18]. The ArianeGroup 1N thruster operates with a nominal specific impulse of 220s [19]. Various other companies develop and design hydrazine thrusters at this thrust range, however, so far, only Rocketdyne has developed a hydrazine propulsion system specifically for the use in CubeSats. The CubeSat Hydrazin Adaptable Monopropellant Propulsion System (Champs) MPS-120 with its additively manufactured hydrazine piston tank can carry around 360 gram of hydrazine providing a total impulse of around 800 Ns. Each of the four thrusters is thereby capable of creating a thrust between around 0.25 and 1.25 N dependent on the pressure in the blow down operated tank with a maximum specific impulse greater than 225 s [15]. Figure 7 shows the 1U sized CHAMPS propulsion system which possesses a dry mass of one kilogram [20].

Figure 7: Rocketdyne's MPS-120 CubeSat High-Impulse Adaptable Modular Propulsion System (CHAMPS)[20]

Even though hydrazine propulsion systems possess extensive flight heritage and components for these propulsion systems are available, most CubeSat platform providers and propulsion system developer set their focus on other propellants and technologies. The carcinogenic, highly toxic characteristic and auto-combustion risks of hydrazine require significant safety measures that can easily take up a high proportion of the costs in the generally limited low budget of CubeSat missions [15]. The development of green propellant based monopropellant propulsion systems for CubeSats has increased. In these propulsion systems hydrazine was replaced by hydroxylammonium nitrate (HAN) or ammonium dinitramide (ADN). Aside from the reduced risk and safety precautions, these propellants also provide a higher density and specific impulse than hydrazine. An example thereof is the Green monopropellant Micro Propulsion System (MiPS) that is currently in production by VACCO Industries in cooperation with Bradford ECAPS. The propulsion model is also equipped with four individual thruster that generate a nominal thrust of 0.1 N with a specific impulse of around 200 s. It can be configured to either operate with the ADN based LMP-103S/LT or HAN based AF-M315E propellant and can carry 2 kilogram of propellant resulting in a wet mass of 5 kilogram and a total impulse of 3320 Ns [21]. A further green propellant system was developed by Busek Co. Inc.. The BGT-X5 system generates a nominal thrust of 0.5 N at a specific impulse of around 220 seconds. Within its envelope of one CubeSat unit (10x10x10 cm) plus an external volume equal to the volume of a “tuna can” (ø6.4cm x 3.6 cm) the fueled system weighs 1.5 kilogram providing a total impulse of 565 Ns [22].The MiPS green propellant propulsion system by Vacco and the HAN based propulsion system by Busek are shown in Figure 8.

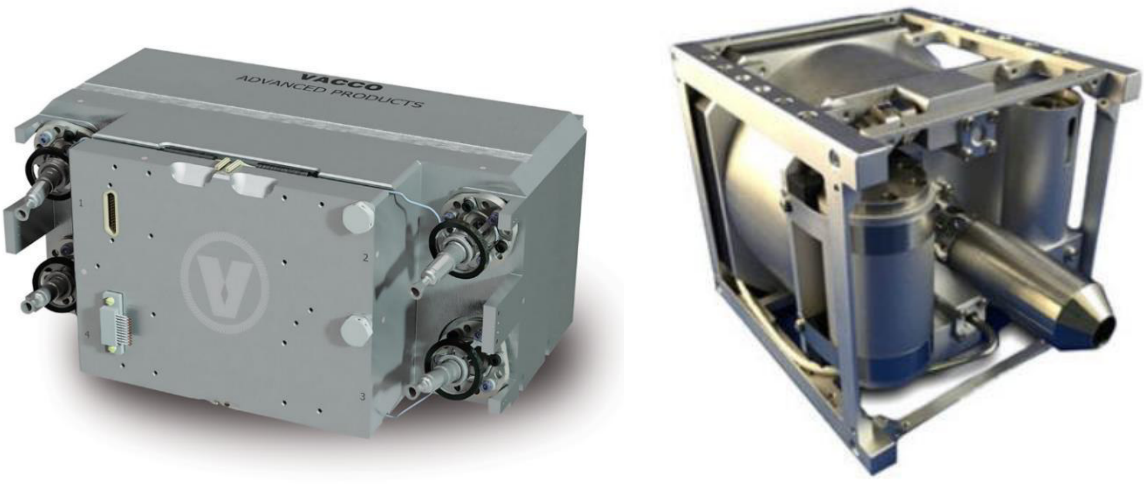

Figure 8: Vacco's MiPS green propellant CubeSat system (left) [21] and Busek's BGT-X5 HAN based propulsion system (right) [22]

A similar HAN-based propulsion system is the MPS-130 by Rocketdyne. It is similar to the hydrazine based CHAMPS MPS-120, uses the same 3D printed piston tank as fuel storage but is equipped with four AF-M315E thrusters instead of the four hydrazine thrusters which can reach a specific impulse between 206 and 235 seconds and a thrust of 0.25 to 1.25N each. The 1U configuration stores 500 gram of propellant and achieves a total impulse of 1200 Ns. All three of these systems are currently still being developed while it seems that the Vacco propulsion system will be the first to be used in orbit since the flight model is currently in production and it is to be flown on the Lunar Flashlight 6U that was planned to launch in November 2020 [21].

On the contrary, the CubeSat provider NanoAvionics has finalized its ADN based EPSS propulsion system. With an in-orbit demonstration in 2017 and a flight heritage from multiple CubeSat missions the propulsion system is fully developed and has achieved TRL 9. The system, with a size of 1.3 U, a dry mass of one kilogram and a propellant capacity of 200 gram can achieve a total impulse of 400 Ns at a nominal specific impulse of 213 and a thrust at the begin of life of one Newton. The EPSS system is shown in Figure 9: NanoAvionics ADN based EPSS propulsion system [24]Figure 9.

Figure 9: NanoAvionics ADN based EPSS propulsion system [24]

2.3.3 Bi-propellant

In general, the mission defines the use of either a mono- or bi-propellant system. Monopropellants are commonly used on small LEO satellites that require a thrust range between 1 and 20 N. The reduced dry mass and simple design eliminates the disadvantage of a reduced specific impulse of around 160 s to 220 s. Bi-propellant systems on the other hand achieve thrusts from 10 N as AOCS for GEO communication satellites up to 500 N for apogee motors. The increased dry mass and complexity is approved to achieve these high thrust levels and a more efficient propulsion system with an I_{sp} of around 250 s to 300 s [25]. This shows very obviously why the use of conventional bi-propellant systems for CubeSats can be considered less beneficial than the mono-propellant systems. An exception hereof is the water electrolysis based propulsion. As a rather unconventional bipropellant propulsion system it combines a few advantageous features that make the system able to compete with the given mono-propellants as a CubeSat propulsion system. The high density and almost zero cost of water, the reduced safety issues due to low pressures and inert properties of the propellant during ground handling and launch, as well as the high specific impulse greater than 300 seconds distinguish this concept from the mono-propellant systems.

Two of these systems have been developed and will each be part of CubeSat technology demonstrations in the near future. The first of such a propulsion system is developed at the Cornell University to fly aboard their 6U Cislunar Explorer CubeSat. The design is drastically simplified compared to the ArianeGroup WPS system. The task of storing water, producing and storing the electrolysis gases all takes place in a single tank. Therein, the electrolyzer generates the two gases which are then directly mixed and stored within the water tank. To separate the gases from the water the CubeSat is set in rotation.

The second water based propulsion system, HYDROS, has been developed by Tethers Unlimited. The first flight unit has been delivered which will be onboard of the Pathfinder Technology Demonstrator (PTD) CubeSat for its technology demonstration. The propulsion system is based on the gas production by a PEM electrolyzer and takes up more than two units with its dimension of 190x130x92 millimeters and a dry mass of 2.2 kilogram. With a maximum thrust greater than 1.2N, a maximum specific impulse greater than 310s and a propellant mass of 500 g it can deliver a minimum total impulse of 2150 Ns. During operation the propulsion system requires a power supply between 5 and 25 watts [26]. The concept of the HYDROS propulsion system seems to be similar to

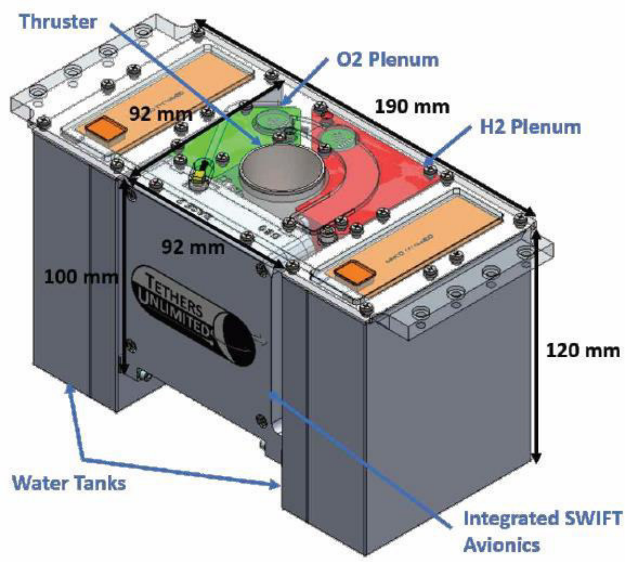

Figure 10:Hydros Propulsion System by Tethers Unlimited

the WPS by ArianeGroup. A main difference to the WPS is the design of the water and gas tanks. The HYDROS tanks are shaped so that thy use the available space optimally. The gas tanks (red and green) are placed around the thruster, to adapt to the space. The water tanks rectangular shapes allow a volume efficient storage for the use in a CubeSat. However, the maximum operating pressures of the tanks are not stated by the manufacturer.

2.3.4 Overview

This conclusion shall summarize and give and overview of the characteristics and performance of the CubeSat propulsion systems described within this chapter. Table 2 lists the main features of each propulsion system. It is noticeable, that the use of piston tanks for pressurized propellants is mainly used within the CubeSat propulsion systems rather than oval or spherical tanks which are often used for diaphragm or bladder tanks on larger satellites. Furthermore, it shall be noted, that all of the stated propulsion systems are connected to the CubeSat with a RS-422 command interface [20][21][22][23][24][26]. The RS-422 is a one-way communication channel that allows the OBC of the CubeSat to actuate the propulsion system.

Table 2: Performance and characteristics of CubeSat propulsion systems [17][18][19][20][21][23]

	MPS-120	MiPS	BGT-X5	MPS-130	EPSS	HYDROS
Propellant	Hydrazine	ADN	HAN	HAN	HAN	H_2O
Specific Impulse [s]	>225	200	220	206-235	213	>310
Thrust [N]	4x 0.25-1.25 N	4x 0.1	0.5	4x 0.25-1.25	1	>1.2 N
Total Impulse [Ns]	800	3320	565	1200	400	2150
Dry Mass [kg]	1.06	3	1.5 (wet mass)	1.06	1	2.2
Propellant Mass [kg]	0.36	2		0.5	0.2	0.5
Dimensions	1 Unit	22.5x15x10 cm^3	1 Unit + Tuna Can	1 Unit	1.3 Units	19x13x9 cm^3

3 CubeSat Platform for the In-Orbit Technology Demonstration

Since the development of the CubeSat standard in the late 90's around 1200 CubeSats have been launched and another 2500 CubeSats are forecasted to be launched within the next six years. Due to a short lead time, reduced sizing of electronics and highly reduced costs for components and launch, CubeSats have become popular for many different applications varying from educational projects to scientific experiments and commercial services. The initial standard describes a CubeSat as a cube with a 10 cm edge length. With the increasing demand for larger CubeSats this cube size was used to define a CubeSat unit, abbreviated to "U", so that adding units increases the CubeSat volume leading to configurations from 0.25 U to the largest current design of a 27 U CubeSat [29]. The CubeSat system described in this work will be used to demonstrate a newly developed propulsion system by taking advantage of the benefits of a CubeSat satellite, such as the reduced cost, complexity and lead time by using commercial of the shelf components (COTS) and the reduced cost for a rideshare launch [29][30]. Estimations regarding the size and weight of the Water Propulsion System lead to the required size of a 6U CubeSat for the IOD mission.

In the following three chapters, the selection process of a suitable CubeSat platform will be illustrated in three steps. The first step, summarized in chapter 3.1, includes the research on the current state of the art of CubeSat platforms and their composition. Thereafter, chapter 3.2 comprises an overview of the CubeSat platforms of three promising suppliers. It summarizes requirements and limitations of and on the DWPS and concludes with an evaluation of the proposed CubeSat platforms.

3.1 CubeSats – General Composition

In general, a CubeSat consists of more or less the same subsystems as any other Satellite. It requires a certain structure, an attitude control system, a data handling system, a communication system, a power system and includes a payload, which is the propulsion system in this case. These subsystems are connected via data bus for internal communication, via electrical harness for the supply and distribution of power and via mounting structures for electrical grounding of components, thermal distribution of heat fluctuations and of course to provide the mechanical stability [31]. During the launch the CubeSat is connected to the launcher by a dispenser, which holds the CubeSat throughout the launch and deploys the CubeSat once the designated orbit is reached.

This chapter shall give an overview and summary of the different subsystems and general architecture of a 6U CubeSat and its dispenser system. Each subsystem is described in detail and important properties for the in-orbit demonstration of a propulsion system are highlighted.

3.1.1 Dispenser

CubeSats are launched into space by a range of different launch vehicles and are most commonly launched as a secondary payload with a large satellite as primary payload or as one of many other small satellites. Further details on launch possibilities are given in chapter 6.2. CubeSats are usually attached to the launch vehicle by a dispenser which houses the satellite and protects it during the launch. Once the launch vehicle reaches the predefined destiny, the dispenser releases the satellite and separates it from other CubeSats, the primary payload and the upper stage of the launcher. For that, the launch vehicle sends a signal to the dispenser which activates a mechanism to open the dispenser's door and the CubeSat is then pushed out by a spring. Various companies have developed dispensers that all function with this principle with only one difference that has to be considered during the CubeSat Design. Most commonly the CubeSat is dispensed on a rail system. All four edges of the CubeSat interface with the dispenser which constrains the CubeSat in its position throughout the launch and allows a smooth ejection. A more unusual method is used by Planetary Systems Corporation's Canisterized Satellite Dispenser (CSD). Two thinner rails, often referred to as tubes, are clamped throughout the launch providing a firm attachment. A mechanism loosens the clamp during the opening of the deployer and allows the ejection of the satellite [17][32]. A scheme of both mechanisms is shown in Figure 11.

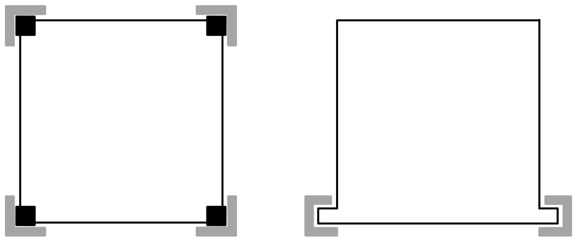

Figure 11: Constraint systems of CubeSat dispensers: standard rail system on the left and tube system by Planetary Systems Corporation on the right

3.1.2 CubeSat Subsystems

3.1.2.1 Structure

The CubeSat structure has to withstand the imposed loads and protect the interior subsystems. The size of the structure is specified by the CubeSat standard and predefines the available space for the CubeSat subsystems and payload. CubeSat components can be mounted to the load carrying frame directly or may be stacked on rails before the integration into the primary structure. The rails used for stacking the components generally conforms to the PC104 standard and can be used to stack any kind of PCB or any other flight modules. Furthermore, the structure has to comply with the specifications from the deployer system. Especially the rails of the satellite have to conform to a certain shape and surface roughness as specified in [17] to prevent wedging of the satellite during the deployment. Figure 12 shows a typical COTS 6U structure of ISISpace with the PC104 conform attachment rails on the inside.

Figure 12: COTS 6U CubeSat structure by ISISpace [2]

3.1.2.2 Communication System

The communication system is required for tracking of the spacecraft, to provide a downlink for telemetry and payload data and to provide an uplink for commands. Conventionally, radio frequency communication system consists of an antenna system and a compatible transceiver. A variety of frequency spectrums are available for CubeSat missions such as the S, X, Ku, Ka, VHF and, most common for CubeSats, the UHF band [33]. Highest data rates can be realized within the X, Ku and Ka bands that are used for missions that require a downlink of large files such as radar or optical measurements. In the case of the IOD mission the downlink data is limited to telemetry and sensor data from the payload and CubeSat platform so that only UHF, VHF and S Band communication will be considered in the following.

The antenna design depends on the frequency used for the communication. UHF and VHF antennas are commonly designed as deployable antennas and are attachable to the top or bottom surface of the CubeSat structure as shown in Figure 13. The S-Band antenna on the contrary is generally designed as a patch antenna and can be mounted onto any surface of the CubeSat as also shown in Figure 13[31][34]. A patch antenna can receive and transmit signals from and in one direction and therefore achieve an antenna gain for this direction of typically 6 dBi. However, the antenna has to be pointed at the ground station to achieve a communication link. The half power beam angle is around 100° and if the satellite's pointing is outside this range, the antenna's efficiency reduces drastically. The deployable UHF or VHF antenna on the other hand, is able to receive and transmit signals almost omnidirectional, meaning, that the antenna gain is almost similar in all direction. The gain is limited to values around 0 dBi and might be reduced in directions, where the view to the ground station is blocked by the satellite's body or solar panels. The omnidirectional nature of the deployable system allows communication between the satellite and the ground station, even if the satellite is in rotation or its attitude is out of bounds.

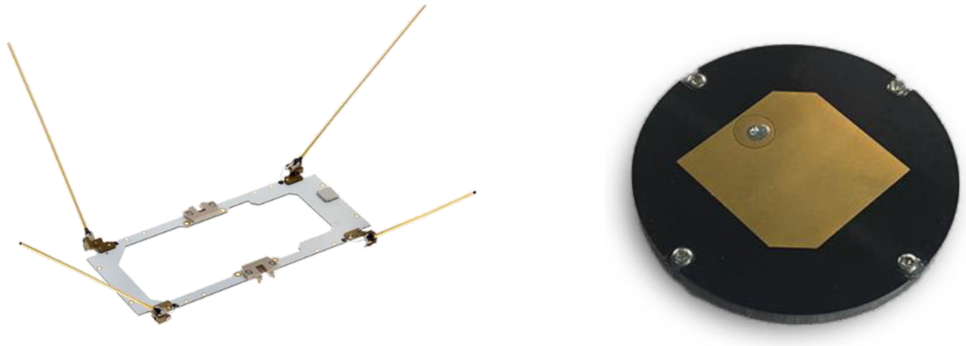

Figure 13: COTS CubeSat antennas: UHF/VHF antenna by NanoAvionics [34] (left) S-Band patch antenna by ISIS [31] (right)

The transceiver provides the interface between the CubeSat platform and the antenna system. Data is transmitted from the platform to the transceiver's microcontroller via data link such as I^2C, CAN or USART, it is modulated to a carrier frequency, sometimes amplified and transmitted to the antenna system. On the other side, the communication system receives data from the ground station and demodulates the input to a digital signal. Depending on the architecture, the system can either be full- or half-duplex. A full-duplex system can receive and transmit signals at the same time while a half-duplex system is limited to only one of these functions at a time. However, the advantage of a half-duplex system is, that it operates at only one frequency which simplifies the communication system design[35]. Main driver for designing or choosing a communication system for the IOD mission is the data down- and uplink rate. The amount of data to be sent to and from the satellite has to be evaluated and a required data rate can then be defined regarding the communication window and the link budget. The communication window depends on the CubeSat's orbit and the location of the ground station and can be determined with an orbital calculation software such as Systems Tool Kit (STK). The link budget can be calculated for a given CubeSat communication system and a ground station. A calculation of a link budget for the communication of the IOD CubeSat and the ground station at the institute of space systems in Stuttgart is described in chapter 6.5

Additionally, the use of a Global Navigation Satellite System (GNSS) antenna and receiver shall be considered for the IOD mission. It is a system to measure the position with a precision of around ± 1.5 m and the velocity with a deviation of around ± 0.03 m/s. The antenna receives signals by a constellation of satellites such as the Galileo or GPS satellites that contain information on the position of the respective transmitting satellite. With this information, the CubeSat is able to determine the distance to the GNSS satellite. With contact to three GNSS satellites, the CubeSat is in theory able to determine its exact location, a connection to a fourth satellite is necessary to eliminate inaccuracies [31][34][36].

3.1.2.3 Electrical Power System (EPS)

The Electrical Power System is in charge of generating, conditioning, storing and distributing electrical power. The following block diagram shows the process steps of the EPS from generation to usage of the electrical power.

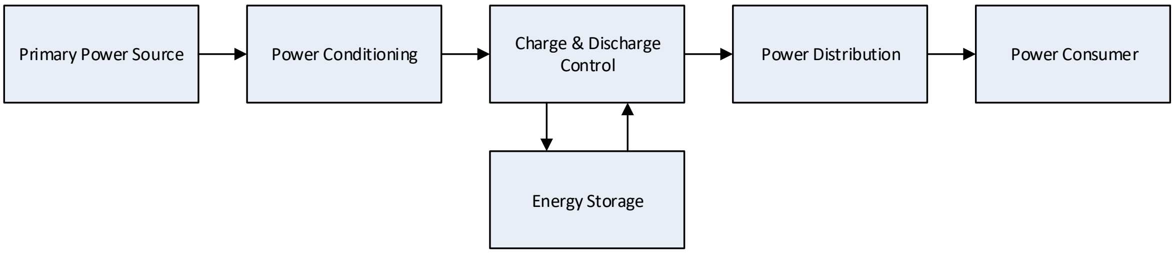

Figure 14: Block diagram of electrical power system

The primary power source for almost any satellite is solar power converted to electric energy by photovoltaic/ solar cells. The use of GaAs multi-junction solar cells is the current state of the art. These solar cells reach an efficiency of around 30% and are available in either body mounted configuration or as deployable panels. With a precise pointing towards the sun, the deployable configuration allows to use the entire solar array surface for high power generation, whereas the body mounted solar arrays have the advantage that they secure a power generation independent of the satellite pointing [37]. Solar cells are attached in series for a string. If a single cell fails a bypass diode allows the string to be still functional. A blocking diode prevents a reverse current into the string and thus protects the string in case of a failure within the Power Control and Distribution Unit (PCDU) [37].

The power produced by a solar array is regulated by a power conditioning unit. There are two types for the regulation of the power:

- **Direct Energy Transfer (DET):** Only the required power for given loads is provided by the regulator, the excess power is switched towards a resistor for heat dissipation.
- **Maximum Peak Power Point Tracking (MPPT):** The MPPT regulates the solar array to operate at a point, at which the generated power equals the required power and no excess power is produced. With this method, the solar arrays' voltage is decoupled from the main bus voltage and the MPPT requires a DC/DC converter.

The MPPT is very useful for varying conditions over an orbit and the satellite's lifetime, since it adjusts to changes and thereby can reduce the solar array size by 15%. The MPPT is the regulation method used on almost any COTS CubeSat platform [31][34][38][39]. The left graph in Figure 15 shows the current-voltage characteristic of a solar array for changing temperatures. Since the voltage is not fixed to the bus voltage, the MPPT can adjust to the operating conditions and allows an efficient power harvest. The maximum power can then be found by multiplying the current with the voltage as shown for a different solar array in the same figure on the right.

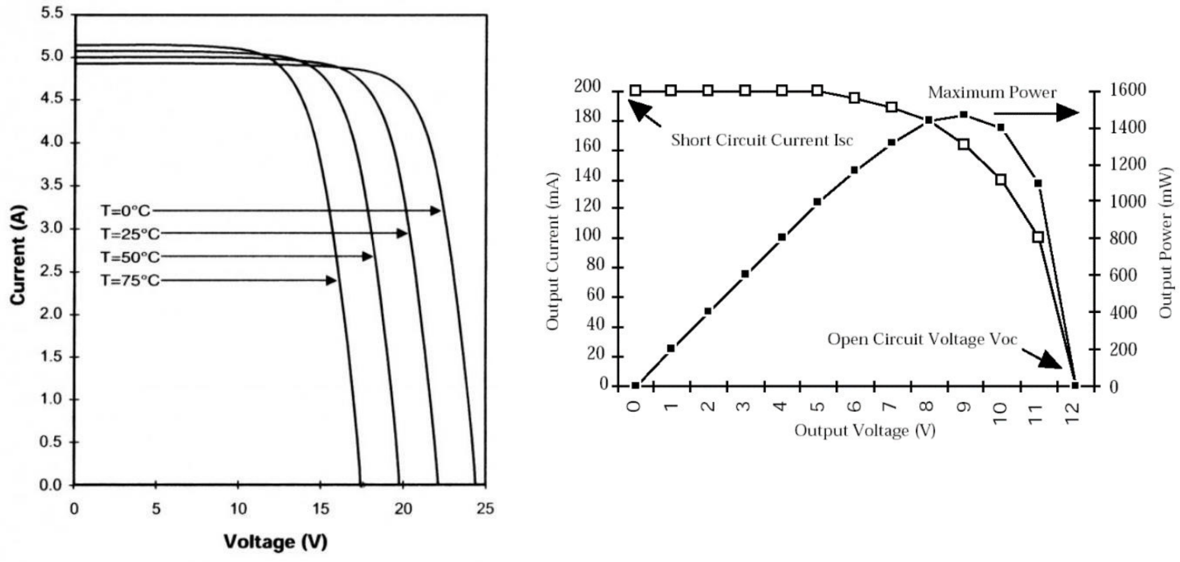

Figure 15: Current and voltage for a solar array for changing temperatures (left)[40] and power outcome for a solar array at a certain condition (right) [37]

The power provided by the solar arrays is conventionally stored by lithium ion batteries. These batteries are typically composed by a number of battery cells that are connected in series and parallel. A typical lithium ion battery cell used for CubeSats has a voltage of around 3.6V and can store around 3 Ah. The battery cells generally conform with the space proven 18650 form factor so that a battery cell has a length of 65 mm and a diameter of 18 mm [31][34][39]. The characteristics of the battery pack depends on the exact configuration of battery cells, e.g. attaching two cells in series increases the battery's voltage while attaching them in parallel increases the battery's current. For the selection of a suitable battery for a certain mission various aspects have to be taken into account. The battery's lifetime, especially the degradation of lithium ion batteries, is influenced by the battery's Depth of Discharge (DoD). The DoD is the discharge in percent of the satellites battery within a charge and discharge cycle, as shown in Figure 16. For a long lifetime with many cycles, the DoD must be kept low. A DoD of only 10% to 40% is suggested for mission lifetimes of 3 to 5 years if a charge and discharge cycle is performed within every orbit which would result in around 16,500 to 27,500 cycles. A trade between the lifetime, available volume and mass and the power requirements then results in the battery selection for the CubeSat [41]. Furthermore, for safety reasons, the energy stored in the CubeSat battery pack shall not exceed 100 Wh as stated in the CubeSat Design Specification (CDS).

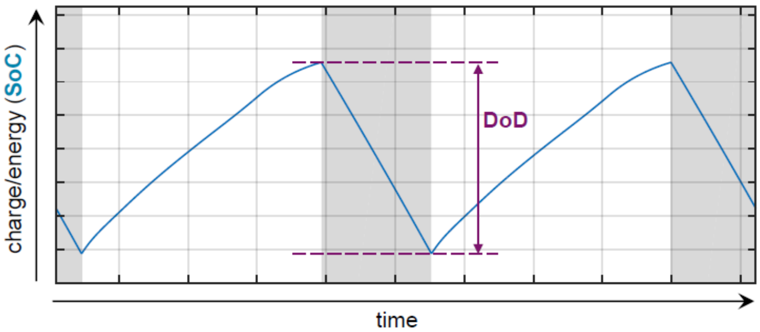

Figure 16: Depth of Discharge of a satellite battery for an orbital cycle. The grey shading implies the eclipse phase during the satellite's orbit [37]

The maximum power that a CubeSat can provide to a payload is also generally limited by the battery configuration since the highest power can be supplied on an unregulated line. The unregulated line provides the current and voltage dependent on the state of charge as well as dependent on the arrangement of cells in series and parallel. A battery pack by NanoAvionics for example can connect up to 14 individual cells to achieve a power of up to 175 W on an unregulated line by connecting 7 cells in series in two parallel strings [34].

The distribution unit is in charge of providing power to consumers at the desired voltages. Dependent on the requirements defined by the consumers, the power supply is either ensured by a regulated or an unregulated power bus. For an unregulated power bus, the voltage is driven by the battery voltage and can vary with the changing battery conditions. A regulated power line is attained by output converters that transform the incoming voltages by the solar array and battery to a defined voltage [37]. Typical power bus voltages for CubeSats are 3.3 V and 5 V with the possibility to add converters for costumer configured output voltages commonly around 3 to 24 V. The total power conversion is limited by thermal considerations which reduces the power throughput in many systems. Higher power throughputs can be achieve with unregulated power lines [34][39]. The distribution unit is equipped with overcurrent limiters that prevent latch-ups that might occur. Additionally, the interface between the battery and the distribution unit is linked by a kills switch. This kill switch is a necessary tool for almost any CubeSat and is also specified in the CDS [1]. For safety reasons, especially to protect other satellites and the launcher, CubeSats shall be unpowered during the launch and shall only be powered once they are deployed. The kill switch disconnects the battery with its stored power from the rest of the CubeSat and thereby deactivates all subsystems. Once deployed, a mechanism interfacing with the dispenser reactivates the system.

3.1.2.4 Attitude and Orbit Control System (AOCS)

An attitude and orbit control system is not essential for every CubeSat but is often required to control and stabilize the satellites attitude and position. The system consist of many different subcomponents all dependent on the requirements on the satellite and its mission. Once the CubeSat is deployed form the launcher, the AOCS takes over and is fully in charge of the positioning and attitude control of the satellite. The AOCS regulates translatory and rotatory motion of the satellite, which is often reduced to the regulation of the rotation for CubeSats, also referred to as Attitude Determination and Control System (ADCS). A translator movement requires a propulsion system which takes up a lot of the little volume and mass available in the CubeSat envelope. Consequently, this chapter will focus on the attitude control of CubeSats. The Water Propulsion System in this study shall not be viewed as part of the AOCS since its primary goal is not the orbit control of the CubeSat but the characterization of its performance.

The attitude control requires a sensor to determine the actual attitude, a computer to compare actual to nominal attitude and an actuator to adjust the satellites attitude. These systems cooperate in a closed control loop as shown in Figure 17.

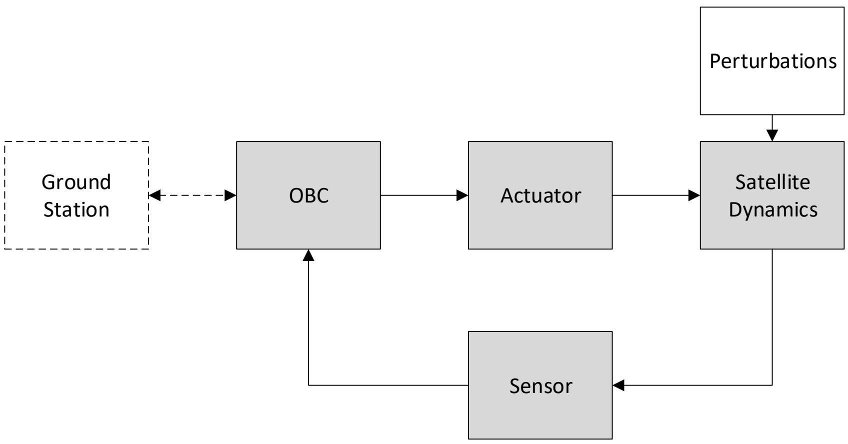

Figure 17: Control loop of an ADCS unit

While the Onboard Computer (OBC) will be discussed in chapter 3.1.2.5, the focus in this chapter will be on the different sensors and actuators for CubeSat systems.

Various sensors are used to determine the attitude of a satellite each measuring a different quantity resulting in some more and some less precise measurements. Each measurement technique has its advantages and disadvantages, might need complementing measurements by other sensors to allow the attitude determination or might function under conditions at which other sensors are useless and vice versa, so that generally more than one sensor type is used to support a satellite mission. The following listing gives an overview of applicable sensors and their respective properties.

- Magnetometer:
  A magnetometer measures the earth's magnetic field in 3 dimensions and allows the calculation of the satellite's rotation rate. It has a low accuracy for the determination of the satellite's attitude but has a spherical field of view and is simple and cheap measuring system.
- Earth sensor:
  It measures the direction to the earth by determining the propagation direction of the earth's infrared radiation. It has a hemispherical view factor with a low to mean accuracy and is also a low cost and simple device.
- Sun Sensor:
  The sun sensor operates similar to the earth sensor, it also detects the sun's radiation propagation direction to determine the attitude of a satellite in relation to the sun. It has the same properties as the earth sensor (hemispherical view factor, low/mean accuracy, low cost, simple design) and is often used in combination with the earth sensor. This type of sensor is then called Coarse Earth Sun Sensor (CESS).
- Gyroscope:
  Various Gyroscope systems are available. The highest precision can be achieved with Fiber-Optic Gyroscopes (FOG). The FOG sensor consists of a coiled light transmitting fiber, a light source and a light detector. The sensor detects the rotation of a satellite by determining the time a light pulse requires to travel through the fiber. It has a mean to high accuracy and has no limiting view factor. However, it is a complex device with higher costs than the previous described sensors. Simpler but less precise are the MEMS gyroscopes, which can detect the satellite's vibration and therewith calculate its rotation. The MEMS technology is widely used and is often preferred due to its small size and low expense.
- Star Sensor:
  The most precise attitude measurements can be performed with a star tracker which determines the satellite's attitude in reference to a pattern of stars. On the contrary, it can only perform these precise measurements once it is already coarsely pointed towards the star pattern since star sensors generally have a view factor of only 20° to 30°. It is a sensor system for high precision tasks with the highest costs.
- GNSS Sensor:
  A Global Navigation Satellite System (GNSS) sensor uses the information provided by a GNSS constellation as stated above and allows the determination of the satellite's speed and position relative to the earth [43].

The amount of possible actuators for satellites is limited and reduced for CubeSats especially due to the reduced mass and volume that can be accommodated. As already specified, propulsion systems for the orbit and attitude control will not be considered herein.

- Magnetorquer:
  A magnetorquer generates a magnetic field with a coil that interacts with the earth's magnetic field and thereby creates a force. The use of three coils allows the satellite to rotate in all directions. To use a magnetorquer, the earth's B-field at the satellite's position must be known, making magnetometers necessary for this method.
- Reaction Wheels:
  Reaction wheels rotate the satellite by accelerating and decelerating wheels according to the conservation of momentum. Similar to the magnetometer, a set of at least three wheels is required to rotate in all 3 directions. For the compensation of periodical perturbations the reaction wheels will be accelerated and decelerated equally. However, for random or one-sided perturbations the reaction wheels will be saturated after a certain time and have to desaturated by a different force for further operations [43].

The combination of these sensors and actuators allow the ADCS to be designed to be more or less accurate. The types of sensors and actuators that shall be used for the demonstrator mission is described in chapter 3.2.1

3.1.2.5 Data Handling System

A CubeSat's data handling is performed by an On-board Computer (OBC) which is in charge of the data handling functions, attitude and orbit control, thermal control and the management of the CubeSat platform and its payload. While in medium to large satellite system a Remote Interface Unit (RIU) offloads the OBC by acquiring sensor data and distributing non-safety critical commands, a RIU for the support of the OBC is generally not implemented in CubeSats [31][37][44]. However, the control and regulation of ADCS components is often outsourced to a second flight or ADCS computer similar to the actual OBC. This computer commonly has a 3-axis gyroscope and magnetometer directly attached to it [31][39].

The OBC's main component is the central processor, generally a 32-bit architecture with a focus on power saving features. An on-board timer generates the reference for clocked operations and might be synchronized by a GNSS system. Aside from a RAM as main memory, the OBC utilizes several non-volatile memory devices for the respective storage of a start-up software, a mission code and a safe mode reconfiguration software. The housekeeping data and mission timeline is stored on a mass memory allocated by SD cards for most CubeSats [31][34][37][38].

3.1.2.6 Interfaces

The interfaces between the CubeSat subsystems consist of logical and electrical connections. The electric linkage between the power system and a consumer is achieved with isolated, copper core cables that are twisted to compensate the magnetic momentum. The voltage on each line is either predefined and controlled by the EPS distribution unit for regulated lines or depends on the battery's properties and state of charge which then defines the voltage for unregulated lines.

The CubeSat internal communication between subsystems is achieved with signal lines. Various different standards defining the electrical interface such as RS232 or RS422 and communication protocols such as defined for UART are used for CubeSats. To reduce the amount of lines, CubeSats use serial data busses to allow the use of a single line for the communication of more than two subsystems. Data busses such as I²C, CAN or SPI are used by default for most commercial available CubeSats. If two components connected by a bus want to send a message at the same time only the dominant message will be transmitted. A dominant message must be defined by a certain method specified by the bus system. I²C and SPI use the master-slave concept where one component or subsystem is defined as a master while the other connected components are the slaves. The master can access the signal line at any time while the slave components require a request by the master to be able to send a signal. On the other hand, the CAN bus uses the multi-master concept where a potential signal collision is detected by a so-called CSMA/CR-Method. Thereby, a message with a higher priority is transmitted while the lower priority message is overridden.

3.2 CubeSat Platforms

A number of companies have specialized on developing CubeSat platforms within the last few years. More than 70 companies have been identified that are capable of providing a CubeSat platform with many more companies that are specialized on respective subsystems [30]. Many of these are new space companies, founded by students that gained their experience on university CubeSat projects. For the demonstration of the WPS' technology in space, a platform supplier is searched which can provide a suitable platform but is also capable of performing the necessary acceptance tests, preparing and organizing the launch, performing regulatory duties and operating the satellite in orbit. From these many companies, three companies were selected, ISIS, NanoAvionics and GOMSpace, since they offer a 6U CubeSat platform with components that possess sufficient flight heritage, as well as the infrastructure for such an all-inclusive support.

Even though a CubeSat platform is highly standardized, many components and systems have to be adapted for the respective purpose. To be capable of evaluating a platform of each supplier, a reference CubeSat platform with its performance and properties was provided by each company. These reference platforms shall be described and compared in the following to allow the estimation of the platforms performance and limitations. However, before defining these platforms in more detail, the requirements of the DWPS on the platform shall be summarized first.

3.2.1 CubeSat Platform Requirements

Size

It is estimated, that 4 units of a CubeSast should suffice the purpose of the demonstrator mission so that a 6U platform is expected to be the most promising size.

Communication

The CubeSat must be capable of sending measurement data acquired from the DWPS down to the ground station. This includes temperature, pressure and power supply information of the payload. In return, it must be capable of receiving commands from the ground station which shall activate certain maneuver sequences. The CubeSat platforms reviewed within this study either possess a UHF, VHF of S-Band communication system. The S-band communication system allows high data rates up to 1 Mbps but requires more power than the UHF and the VHF communication system and generally uses patch antenna which allow a beam gain but cannot operate omnidirectional as the conventional deployable UHF an VHF monopole antennas [31][34][39]. The amount of data that has to be transmitted during an operation defines the communication band but cannot be predicted yet. Therefore, the communication band should be chosen later during the development. It is assumed that an X-band communication system used for high data rates is not required [37].

Attitude Determination and Control

One of the most challenging tasks of the CubeSat platform is to keep the satellite stable during a maneuver. A minor thrust vector deviation at the comparatively high thrust maneuvers creates a momentum on the satellite that must be compensated. For this compensation the satellite can be equipped with two actuators, the magnetorquer and the reaction wheels. The magnetorquers of conventional platform providers create a magnetic moment of around 0.2 to 0.3 Am^2. This magnetic moment $\vec{m}$ creates a torque τ depending on the earth's magnetic field $\vec{B}_{earth}$ according to equation 2.8.

$$\tau = \vec{m} \times \vec{B}_{earth} \tag{2.8}$$

The strength of the earth's magnetic field strongly depends on the satellites position. It increases around the poles and is also influenced by deviations caused by external fields such as ionospheric plasma. However, in LEO the earth's magnetic field dominates other influences with a field strength of 0.25 to 0.5 Gauss [42]. The maximum achievable torque is thereby limited to a maximum of 15 µNm and is assumed to be insufficient for the stabilization of the CubeSat that is accelerated by the DWPS with a thrust of up to 1 N. Reaction wheels on the other hand can provide a much higher maximum torque of around 3 mNm, but cannot store the momentum unlimited and therefore have to be desaturated at some point. Therefore, the reaction wheels shall be used as primary actuator while the magnetorquers are used for the desaturation of the reaction wheels [31][34][38][39].

For the attitude measurement of the CubeSat, sun sensors and gyroscopes are typically used. The gyroscope is required during tumbling phases, in which the sun sensor is not functional. Once detumbled, the sun sensor allows the determination of the attitude referring to the suns position. This way, an attitude can be determined with a deviation of around 1-2°. For a more precise measurement, the CubeSat could be equipped with a star sensor to achieve a determination accuracy of around two orders of magnitude better. This however is not required for the DWPS mission. A magnetometer on the other hand is required. It determines the earth's magnetic field and is required to determine the torque that can be achieved by the magnetorquers and can complement the gyroscope in measuring the tumbling rate. Furthermore, the position of the satellite and the change of its velocity shall be monitored. This is possible with a GNSS antenna and receiver.

Power Supply

From the research on the current state of the art of CubeSats it has become clear that a CubeSat is not capable of supplying the electrolyzer with the high powers of up to 250 watts at which it shall be operated during its nominal operation. The maximum power supply that can be provided by the CubeSat depends on two factors: (1) the power generation and (2) the battery configuration. The power generation allows an estimation of the average available power, the so-called Orbit Average Power (OAP). It depends on the size of the solar panels and the orbit of the CubeSat. The battery configuration and its state of charge on the other hand define the maximum power that can be supplied for a shorter duration. This maximum power is defined by the amount of cells that are arranged within the battery configuration. Increasing the number of cells in series increases the voltage and increasing the number of cells in parallel increases the current that can be provided. If the energy required for a gas production cycle within the electrolyzer is below the energy that can be stored by the battery, it can be assumed that the maximum power is predefined by the battery configuration. For the mission, the battery size is mainly limited by the available space within the 6U envelope.

Considering these requirements, the CubeSat shall be assembled as shown in Figure 18. The heart of the CubeSat is the Onboard Computer. It is commonly divided into two computers, one for the command and data handling and one that is in charge of the attitude determination and control tasks. The OBC communicates with all components either via individual signal lines or via a data bus. Based on these platform requirements, the three CubeSat platforms by ISIS, NanoAvionics and GOMSpace are reviewed and their use for the demonstrator mission is evaluated in the following three chapters.

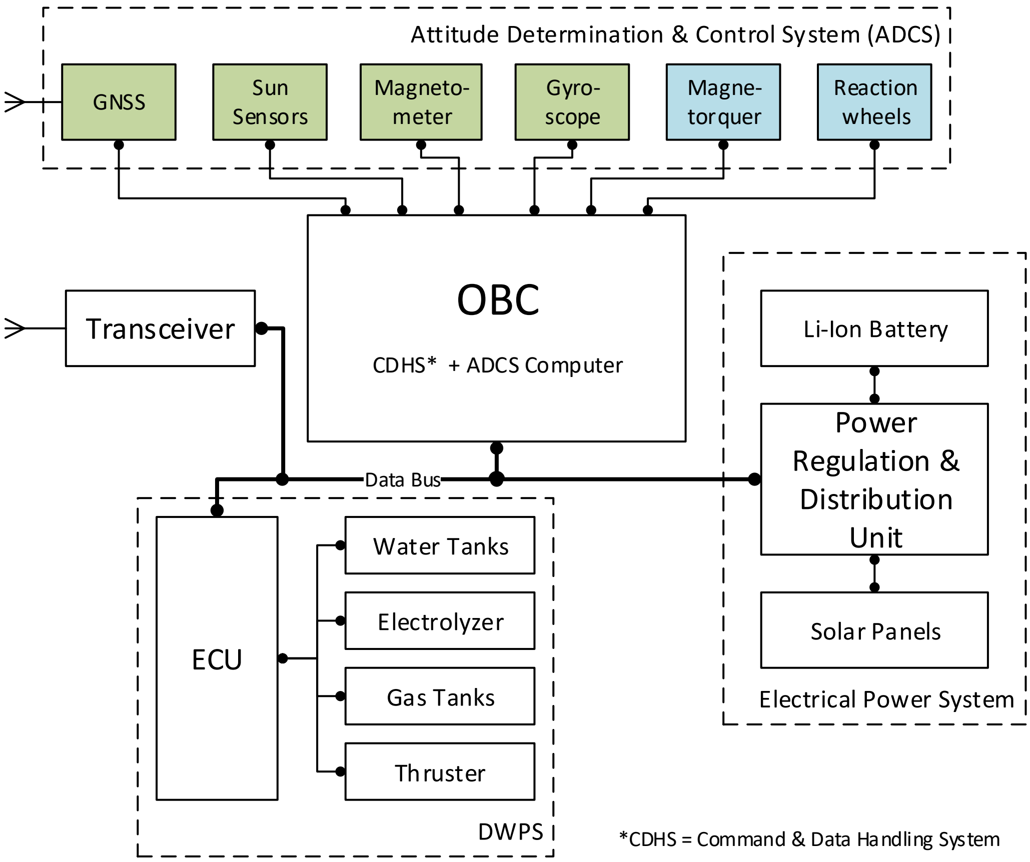

Figure 18: CubeSat architecture

3.2.2 ISIS

With more than 400 satellites launched since its foundation in 2006, ISIS possesses a broad experience with CubeSats. The reference platform suggested by ISIS is thereby almost fully equipped with components developed by ISIS.

The reference platform of ISIS represents an initial concept of a CubeSat platform for the company Space Norway. It is equipped with the necessary devices as shown in Figure 18 but also includes an electrical propulsion system, which is not required for the demonstrator mission. The general features of this platform are given in the following table.

Table 3: General features of the ISIS 6U platform [45]

Size	6U
Platform Mass	N.A.
Payload Available Volume	3U
Payload Available Mass	N.A.
Design Lifetime	N.A.

The power generated by the solar panels is regulated, controlled and distributed by the EPS. The properties of the EPS are summarized in Figure 1Table 4.

Table 4: Properties of the EPS of the ISIS 6U platform

Platform Power Usage	4.7 W (idle) / 12.3 W (peak)
Average Power Generation (OAP)	12 W*
Peak Power Generation	32 W
Energy Storage	90 Wh
Number of Battery Cells	8
Battery Voltage	14.4 V
Maximum Output Power Single Line ▪ **Regulated** ▪ **Unregulated**	 24 W (12V line) 86 W
Payload Regulated Voltages	3.3V / 5V / 12 V (2A max.)

* OAP @SSO 500 km, nadir pointing, LTAN 12h

The battery of the platform is composed by four cells in series on two parallel strings. With a nominal voltage of 3.6 volts of each cell, the battery provides a total voltage of 14.4 V, has a maximum voltage of around 16.8 V at full charge and a minimum voltage of 10 V if it is almost fully discharged. Using 2 battery pack's with 4 battery cells each, the maximum output power can reach 86 Watts [46].

An S-Band receiver developed by ISISpace is operated at a frequency between 2200 and 2290 MHz, receives at a rate of up to 1 Mbps and transmit at a downlink rate of up to 4.3 Mbps. A UHF / VHF full duplex transceiver by ISIS only achieves data rates of 9.6 kbps for receiving and transmitting data.

Information on the ADCS system is not publicly available.

3.2.3 NanoAvionics

The company NanoAvionics, based in Lithuania and expanded to the UK and the US has so far performed and supported more than 70 successful small satellite missions. Equal to the other two reviewed companies, it has specialized on CubeSat platforms with a 6U and a 12U platform as their flagship.

The reference platform of NanoAvionics is based on their standard M6P CubeSat platform. Its general features are summarized in the following.

Table 5: General features of the M6P platform by NanoAvionics [47]

Size	6U
Platform Mass	~ 5 kg
Payload Available Volume	Up to 5U
Payload Available Mass	7.5 kg
Design Lifetime	5 years

These information have to be viewed with caution. These values state the maximum possibilities within each category. A volume of 5U will most likely not be available for the payload if the platform consists of the required components for the demonstrator mission.

The main difference of the electrical power system to the power system proposed by ISIS is the battery's configuration. The battery consists of a minimum of two cells that are connected in series. Further battery cells can be connected in parallel in pairs of two up to a maximum amount of 14 cells which can provide a maximum power of 175 W. The reference platform is equipped with a battery pack of 8 cells which can provide a maximum output at full charge of around 100 W or a consistent maximum power of around 75 W. Further information on the EPS is provided in Table 6.

Table 6: Properties of the EPS of the M6P 6U platform by NanoAvionics

Platform Power Usage	N.A.
Average Power Generation (OAP)	10 W*
Peak Power Generation	N.A.
Energy Storage	75 Wh
Number of Battery Cells	8
Battery Voltage	7.4 V
Consistent Maximum Output Power Single Line ▪ **Regulated** ▪ **Unregulated**	 20W 75 W
Payload Regulated Voltages	3.3V / 5V / 3-18V

* OAP @SSO 500 km, nadir pointing, LTAN 12h

The communication system of the M6P is split up in two individual systems, one for the payload and one for the Telemetry, Telecommand and Control (TT&C) of the platform. The payload communication system is based on an S-Band full duplex transceiver allowing the transmission rate of around 200 kbps for up- and downlink. The TT&C communication is achieved within the UHF band. A transmission rate of 19.2 kbps is the maximum data rate that that is possible in both directions.

The ACDS is composed of the same sensors and actuators as the ISIS platform's ADCS. The properties are shown in the following table.

Table 7: Properties of the ADCS of the M6P platform by NanoAvionics

Pointing Accuracy	1° to 2.5°
Attitude Knowledge	0.8° to 2.3°
Angular Rate Knowledge	N.A.
Slew Rate	Up to 10 deg/s
Momentum Storage per Wheel	20 mNms
Maximum Torque per Wheel	3.2 mNm
Position Accuracy (GNSS)	2.5 m
Velocity Accuracy (GNSS)	0.1 m/s

The reaction wheels can store an impulse of 20 mNms so that a thrust maneuver with a thrust vector deviation from the center of gravity of around 0.02 m would allow a maximum impulse of around 0.75 Ns at a maximum thrust of 0.16 N.

3.2.4 GOMSpace

The third and last reviewed platform is the 6U CubeSat platform by GOMSpace. With 130 employees, the Danish company is the largest of the three CubeSat developers. The reviewed platform is shown in Figure 19.

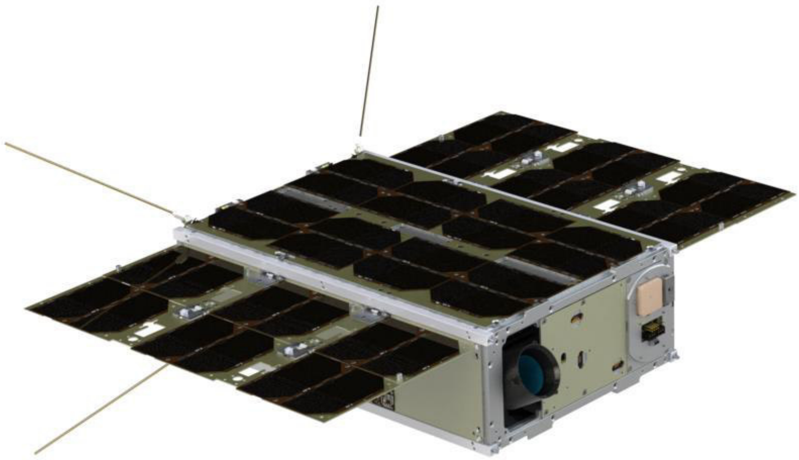

Figure 19: GOMSpace 6U reference platform [48]

It carries an electrical power system with a battery pack of 8 cells, an attitude control system based on a computer, fine sun sensors, magnetometers, a GPS system, reaction wheels and magnetroquers and an UHF or S-Band communication system for the data and command exchange. The general features of the system are summarized in the following [48].

Table 8: General features of GOMSpace's 6U CubeSat platform [48]

Size	6U
Platform Mass	6 kg
Payload Available Volume	4U
Payload Available Mass	6 kg
Design Lifetime	5 years

With the configuration of solar panels, the power generation is similar to the other two platforms. The performance of the EPS is summarized in table Table 9.

Table 9: Properties of the EPS of the GOMSpace 6U platform [48]

Platform Power Usage	N.A.
Average Power Generation (OAP)	15 W*
Peak Power Generation	29 W
Energy Storage	77 Wh
Number of Battery Cells	8
Battery Voltage	7.4 V
Maximum Output Power Single Line ▪ **Regulated** ▪ **Unregulated**	 24W (12V line) N.A
Payload Regulated Voltages	3.3 V, 5 V, 8 V, 12 V, 18 V, 24 V

* OAP @SSO 500 km, nadir pointing, LTAN 12h

Compared to the other two platforms the performance of the EPS is similar. A difference is given by the flexible design of the battery. It can be chosen between a 2s4p, 4s2p and an 8s1p configuration. The first number indicates the amount of cells in a string the second indicates the amount of parallel strings. With the 8s1p configuration, a voltage of up to 33.6 V can be reached.

The UHF communication system of the platform is capable of transmitting and receiving at data rates up to 38.4 Kbps. The upgrade to the conventional S-band system allows 115 kbps or to a high speed S-band system a downlink data rate of up to 6 Mbps.

The ADCS of the platform, equipped with the components listed above, can achieve the following performance.

Table 10: Properties of the ADCS of the GOMSpace 6U platform [48][49]

Pointing Accuracy	2.5° (in sunlight) 7.5° (in eclipse)
Attitude Knowledge	2° (in sunlight) 7.5° (in eclipse)
Angular Rate Knowledge	N.A.
Slew Rate	10 deg/s (peak)
Momentum Storage per Wheel	19 mNms
Maximum Torque per Wheel	2 mNm
Position Accuracy (GNSS)	1.5 m
Velocity Accuracy (GNSS)	0.03 m/s

The reaction wheels by GOMSpace are capable of storing a similar momentum as the NanoAvionic reaction wheels, the maximum torque on the contrary is not as high.

3.2.5 Conclusion

It is hard to perform a trade-off at the current state of the development, especially since the platforms possess similar performance and the information on the platforms can only be seen as initial characteristics and as a baseline for further discussions, developments and iterations. However, the presented platforms provide a great overview of the capabilities of a 6U CubeSat.

For the DWPS, an envelope of 4U and a mass of 6kg can be achieved with a 6U CubeSat platform. The mission can then be performed within a maximum duration of 5 years.

As expected, the power generation of the CubeSat is limited. However, if the DWPS requires only a certain amount of energy for a gas production cycle, the electrolyzer can be fed by the charged battery at power levels of around 75 W. If more energy is required, then the power budget of the satellite in its nominal operation has to be examined in more detail. Nevertheless, the gas tanks sizes, which determine the required power is also very limited, so that for now it shall be expected that a power supply of at least 50W should be possible for the production of the electrolysis gases. A higher power supply is generally possible but a further aspect, the heat dissipation, becomes more relevant and must also be considered.

A more restricting subsystem is the ADCS of the platform. With a momentum storage of around 20 mNms and maximum torques between 2 and 3.4 mNm, the precision of the thruster must be very high and the maximum firing duration is limited to only a few seconds.

For further developments, the following assumptions shall be considered.

Table 11: Estimated CubeSat performance for a 6U platform

Available Payload Volume	4U
Maximum Payload Mass	6 kg
Energy Storage	75 Wh
Maximum Power for Electrolysis	~50 W
Regulated Voltages	3.3 V, 5 V, 12 V
Maximum Torque per Wheel	3.4 mNm
Momentum Storage per Wheel	20 mNms

4 Technical Requirements on the DWPS

The first step of designing the propulsion system for the IOD, is the definition of requirements on the DWPS. These requirements are derived from the current R&D results of the WPS development, constraints that are predetermined by the CubeSat platform and in some cases from experience with former propulsion systems developed by the ArianeGroup. A functional analysis, described in chapter 4.1 is the foundation for the specification of requirements on the DWPS. Each function and constraint identified within this analysis defines certain requirements on the propulsion system. The generic requirements in the subsequent chapter state the mission objectives and generic design requirements. These generic requirements as top level, have to be adhered to in any case. They are based on general standards and considerations for the design of a CubeSat or a satellite in general and the mission it shall serve in space. A level lower, the DWPS System Specification contains specific requirements on the propulsion system as an assembled system. The lowest level specifications are the specifications of requirements on the different subsystems of the DWPS. The following scheme, Figure 20, shall show the dependencies of the requirements. The top level requirements influence the design of the system that is specified in the lower level.

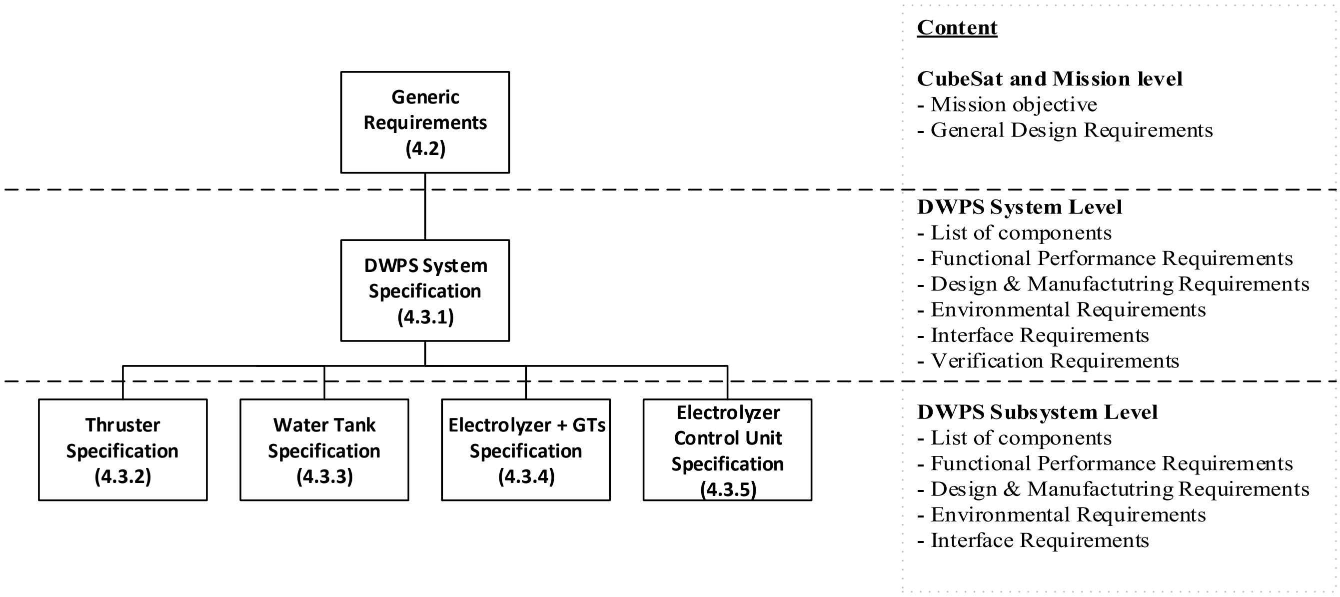

Figure 20: DWPS requirements and their dependencies

4.1 Functional Analysis

During the DWPS' lifetime, it will be exposed to many different environmental influences and must be capable of withstanding and functioning within these environments. Therefore, the DWPS must be adapted to each phase of the mission so that each mission phase influences the design of the propulsion system. Hence, before listing the DWPS functions and constraints that are required to operate the DWPS, Figure 21 shall give an overview of the DWPS' life cycle including all mission phases from the acceptance of the system, its integration into the CubeSat, the launch to the actual in-orbit demonstration in the orbital phase with the concluding passivation thereafter. Within each phase, the DWPS must perform certain functions and comply with constrains. These functions are defined as services provided by the DWPS to an external entity. A constraint is defined as the relationship between the DWPS and an external entity [50]. External entities for the DWPS are any regulations, standards or norms that need to be applied, the the external and imposed environments in each phase, facilities and equipment that interface with the DWPS, as well as the CubeSat and the subsystems of the DWPS.

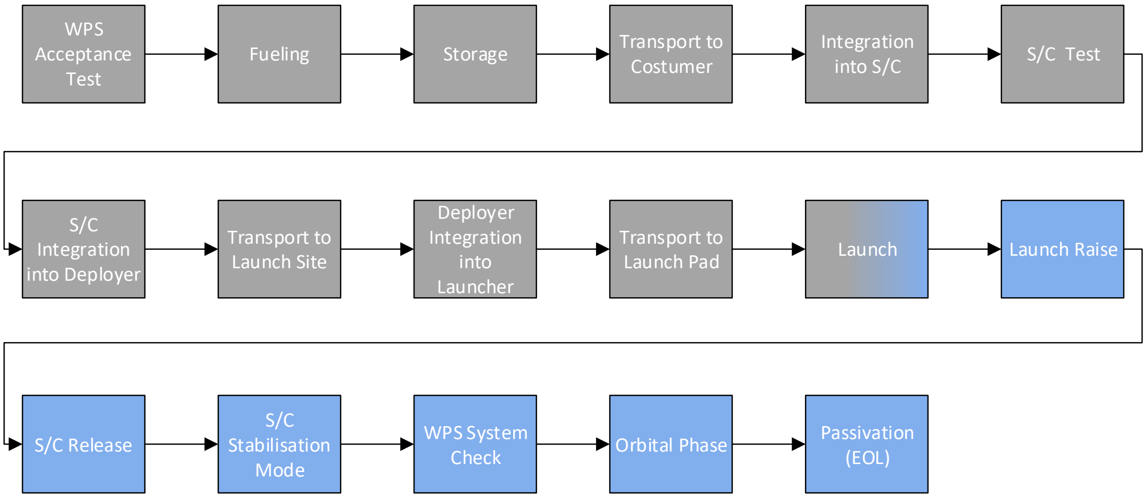

Figure 21: CubeSat DWPS life cycle for the IOD mission

The functions and constraints of the DWPS derived from each phase are summarized in the subsequent chapters 4.1.1 and 4.1.2.

During the actual operation of the DWPS in the orbital phase, the system is operated in a cycle with two phases, a loading and a firing phase. During the loading phase the electrolyzer produces the gases and pressurizes the gas until the gas tanks are pressurized to their nominal pressure. During the discharge cycle the valves to the thruster open and the gases are combusted. These procedures are performed consecutively and alternating. If one of these phases is active the second is passive.

4.1.1 Functions

The functions of the DWPS are similar to the actual WPS' functions. The water is stored in a reservoir, fed to an electrolyzer which produces the electrolysis gases hydrogen and oxygen in a stoichiometric ratio. These gases are stored in tanks and are then combusted and accelerated to generate thrust. However, aside from these main functions performed by the subsystems of the DWPS, the DWPS must be capable of delivering further services in orbit and on ground to be capable of serving its purpose. These secondary functions (Fs) are listed in the following:

Fs1: To follow commands from the OBC

Fs2: To acquire power from the CubeSat EPS and distribute it to the consumers

Fs3: To collect and process sensor data during the operation

Fs4: To maintain components within their temperature and pressure limits

Fs5: To prevent backflow of gases and flames

Fs6: To extract / drain gas out of the reservoir (e.g. passivation)

A more detailed overview of the main and secondary functions is given in the Appendix A.

4.1.2 Constraints

The design and operation of the DWPS is not just adapted to fulfill the functions but also to comply with constraints induced by the external entities. The resulting constraints that have been identified for the DWPS in every mission phase are summarized in Table 12.

Table 12: DWPS constraints

ID	Constraints
Fc1	To perform according to predefined test procedure and mission profile
Fc2	To withstand and function under environment imposed during storage, transport, testing, launch and operation
Fc3	To provide mechanical, electrical, thermal, hydraulic and logical interfaces
Fc4	To provide integration interfaces for storage and transport attachments
Fc5	To provide accessibility and mountability
Fc6	To perform maneuver at precise position without delay in orbit
Fc7	To be compatible with CubeSat software and power supply
Fc8	To comply with CubeSat standard, size, weight, max temperature, mechanical loads
Fc9	To withstand a phase in which the DWPS is electrically disconnected from power system for the launch in which CubeSat has to be in power off state
Fc10	To perform with power provided by CubeSat
Fc11	To allow outflow of air during launch raise phase
Fc12	To comply with safety regulations (toxicity, pressure)
Fc13	To comply with Launcher Regulations
Fc14	To comply with Dispenser Regulations
Fc15	To comply with debris mitigation guidelines
Fc16	To use materials that decompose entirely during atmospheric entry

Further evaluations on which function and constraint applies in which mission phase is given in the Appendix A.

4.2 Generic Requirements

4.2.1 Mission Objective

The mission purpose is the **demonstration** of the key technologies of the Water Propulsion System that is currently being developed by the ArianeGroup under its operational environment. This includes the operation of the catalytic thruster fed by a stoichiometric ratio of hydrogen and oxygen which are produced by a PEM electrolyzer in a vacuum, micro-gravity environment under the exposure of radiation. The propulsion system demonstrator shall consist of a control unit, a water reservoir, an electrolyzer, a gas tank for hydrogen and one for oxygen and a thruster. Thereby, the components shall either be down scaled or adapted to the CubeSat in a way that they operate and function in the same manner as within the actual WPS. For the demonstration, the electrolyzer depicts the most important technology. Whereas the other systems can be evaluated and tested under their relevant environments on ground, the electrolyzer's performance and functionality is influenced by the microgravity environment, which cannot be reproduced sufficiently in earth bound laboratories.

Aside from the functionality of the WPS in space, it is the mission goal to **verify and characterize the performance** of the DWPS in space and to compare its behavior to its behavior during ground based experiments. To evaluate the performance of the WPS the following quantities must be measured or recorded:

- Electrolyzer performance
- Hydrogen and oxygen filling level and condition
- Thruster performance
- CubeSat position and velocity

The electrolyzer's performance can be derived from the power supply parameters. The current and voltage allow the determination of the gas production rate and the power demand. On the other hand the degradation of the electrolyzer's membrane can be determined by measuring the impedance of the Membrane Electrode Assembly (MEA). The gas tanks filling level must be measured by a pressure and temperature sensors. As soon as the maximum pressure at the nominal temperature is reached, the gas production in the electrolyzer is terminated. The temperature and pressure values allow the determination of the electrolyzer's behavior on the one hand, and make it possible to characterize the thruster's performance on the other hand. In the nominal case, the pressure in both tanks should be equal so that the stoichiometrically produced gases are combusted stoichiometrically. The measurement of the thruster's temperature as well as the velocity and position measurement complements the performance evaluation of the DWPS.

Additionally, as a secondary mission goal, the WPS system shall be investigated for its **long-term behavior** in orbit. The main mission including the functional and performance verification shall be performed within the first 6 months. Thereafter, the WPS demonstrator shall be reactivated every year until the resulting water tank is drained. This shall allow a prediction of the WPS degradation under the severe thermal conditions in orbit. The exact mission is described in chapter 6.

4.2.2 General Design Requirements

This chapter contains the requirements that have to be adhered to for any DWPS component. These requirements originate from safety and redundancy considerations as well as the CubeSat Design Specification (CDS).

<u>Safety Considerations</u>

In general, it must be assured, that the CubeSat including the DWPS does not cause hazard to humans, the environment, other spacecraft and the launcher as well as other public and private property. The CubeSat WPS shall be designed and manufactured with compatible materials in such manner, that all hazards are eliminated, minimized and controlled. The use of a water based propulsion system reduces many risks:

- No high pressure in vessels on ground: The pressure required for the blow-down operated thruster is generated in orbit.

- No toxic fuel
- No hazard due to explosive/combustible fuel on ground

However, once in orbit, the electrolyzer produces gases up to a high pressure. The fluidic components exposed to this pressure must withstand these pressures during the entire operational time. Any rupture within the fluid system could cause the decomposition of the CubeSat, resulting in a large amount of uncontrolled space debris and must be prevented in any case. Moreover, a CubeSat orbiting the earth in a LEO will reenter the earth's atmosphere heading for the earth's surface at some point. It must be ensured, that the DWPS consists of materials that decompose during the atmospheric reentry to prevent any damage caused by components impinging on the earth's surface.

Redundancy Considerations

Once a satellite launches from earth it is inaccessible for any reparations and the failure of a component can only be compensated by a redundant architecture. Therefore, it is often preferred to avoid a single-string architecture where single point failures might lead to a loss of spacecraft and mission. On the contrary, it must be recognized that small satellite systems with only very little volume and mass available, cannot ensure a full redundant system. The emphasis for the DWPS project shall thus be to prevent the propagation of failure as commonly adapted by the ESA for their IOD CubeSat projects [52]. In the case of the CubeSat Water Propulsion System, especially the electrolyzer shall be designed to be redundant. This means, that the electrolyzer shall possess two full redundant segments with individual ports for water, hydrogen and oxygen. Furthermore, a failure in one segment shall not lead to a failure of the other segment. For the other systems, single point failures shall be prevented if possible. If this is not possible, the risk of failure shall be assessed, risk mitigation concepts shall be evaluated and implemented and the failure propagation shall be prevented.

CubeSat Design Specification (CDS)

The CubeSat Design Specification (CDS) was created in the late 90's by the California Polytechnic State University, San Luis Obispo and Stanford University's Space Systems Development Lab to facilitate CubeSat developments. While most requirements specified in the CDS only affect the design and architecture of the CubeSat platform, some requirements influence the design of the DWPS and are listed in the following. The requirements can be split up into general, mechanical, electrical and operational requirements. The identification number stated in brackets at the end of each requirement refers to the requirement's original number in the CDS [1].

1) General Requirements
    a. All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris will be created (3.1.1)
    b. No pyrotechnics shall be permitted (3.1.2)
    c. Any propulsion system shall be designed, integrated and tested in accordance with AFSPCMAN 91-710 Volume 3. The Air Force Space Command Manual on range safety requirements includes different safety requirements for hazardous materials, radiation sources, ground handling equipment and many more. For the DWPS the requirements on pressurized flight hardware is relevant. The standard defines a pressurized system with a pressure above 250 psig (17.24 bar) as hazardous flight hardware. Any component that is pressurized above this pressure during ground handling and launch must comply with extensive safety measures.
    d. The DWPS materials shall satisfy the following low out-gassing criterion to prevent contamination of other spacecraft. (3.17)
        - CubeSat materials shall have a Total Mass Loss $\leq$ 1.0%
        - CubeSat materials shall have a Collected Volatile Condensable Material $\leq$ 0.1%
    e. The CubeSat shall be designed to accommodate ascent venting per (ventable volume / area) < 5080 cm (3.1.9)

2) Mechanical Requirements

   a. The CubeSat shall use the coordinate system shown in Figure 22 (3.2.1)

   b. No components shall exceed the nominal 6U envelope by more than 10 mm normal to the surface. The nominal envelope for a 6U CubeSat is 100 x 226.3 x 326.5 mm which might vary in in z-direction (z=326.5 mm) depending on the dispenser [1][53][54]. (3.2.3)

   c. The maximum mass of a 6U CubeSat shall be 12.00 kg. A larger mass may be possible. (3.2.9)

   d. The CubeSat's center of gravity shall be located in its geometric center with the following maximum deviation thereof: (3.2.10)

      - X-Axis: ± 4.5 cm
      - Y-Axis: ± 2 cm
      - Z-Axis: ± 7 cm

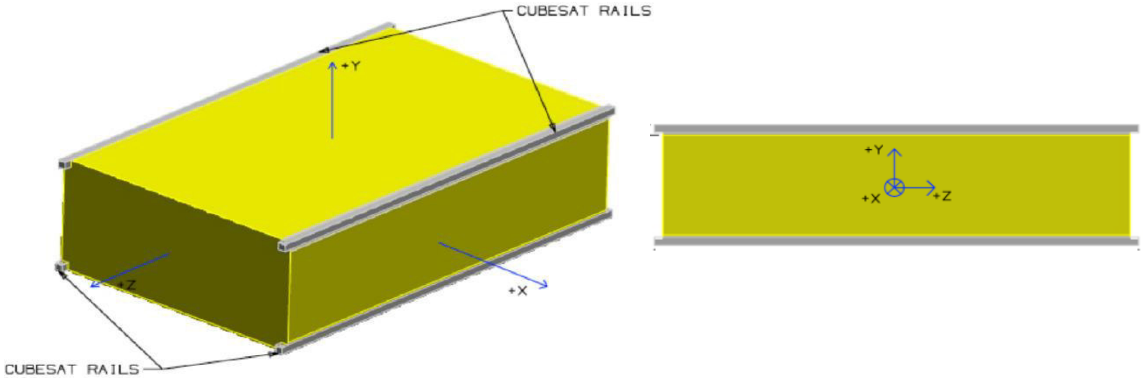

Figure 22: CubeSat reference coordinate system

3) Electrical Requirements

   The electrical requirements defined in the CDS influence the design of the CubeSat platform but barely have an impact on the DWPS. However, it shall be noted, that the CubeSat specification defines that a CubeSat power system must be at a power off state from the moment the CubeSat is integrated into the dispenser until it is released in space. Once deployed, the power system can be reactivated. The DWPS must be able to withstand such time frame without a power supply. (3.3.1)

4) Operational Requirements

   a. The CubeSat mission design and hardware shall be in accordance with NPR 8715.6, a Nasa Specification on procedural requirements for limiting orbital debris and evaluating meteoroid and orbital debris environments. In particular, the following requirements need to be adhered. (3.4.3)

      - It must be ensured, that no surviving debris of CubeSat components re-entering earth's atmosphere impacts on the surface with a kinetic energy greater than 15 Joules
      - It might be necessary to provide orbital debris mitigation data to a licensing agency or mission integrator who is responsible for all mission coordination duties. An analysis on orbital debris and atmospheric re-entry can be conducted with NASA DAS software.
      - A CubeSat, orbiting in a LEO, must re-enter the earth's atmosphere within 25 years after the satellite's end of life

   b. All deployables, such as solar panels must wait a minimum of 30 minutes before being deployed. This will reduce the amount of initially available electrical power.

Furthermore, the CubeSat shall not generate or transmit any RF signals within the first 45 minutes after deployment.

Additionally, the CDS document contains requirements on testing and verification of the CubeSat. The testing of the assembled CubeSat shall be performed by the platform provider and will, thus, not be discussed here.

4.3 Technical Requirements - DWPS System

The DWPS is composed of the following main components:

- Water Tanks (WTs)
- Gas Tanks + Electrolyzer (GTs + ELY)
- Thruster (THR)
- Electrolyzer Control Unit (ECU)

The design of the DWPS and its subsystems is mostly predefined by the actual Water Propulsion System and the R&D results of the developments on the thruster, the electrolyzer and the ECU. Hence, these three subsystems of the WPS must be similar for the equivalent subsystems of the DWPS. The technology and design of water and gas tank can be chosen for the DWPS independent of the water and gas tanks designated for the WPS. Thus, the gas tanks can be adapted to the electrolyzer to form an integral component with the electrolyzer. This is the reason why the electrolyzer and the gas tanks are viewed as a single subsystem of the DWPS. The other subsystems, on the contrary are only connected to each other by tubes and the electrical harness and shall be attached to the CubeSat structure with individual mounting structures.

4.3.1 Functional Performance Requirements

Performance Requirements

To be capable of defining the thrust generated by the DWPS, the specific impulse is estimated to be around 300 s for a quasi-steady state maneuver. With this estimation, an initial design of a thruster was developed and published in the thesis [10]. The thruster was designed to generate a thrust below 1N at the Maximum Expected Operating Pressure (MEOP). This thrust level is still very high when applied to a 12 kg CubeSat. Small thrust vector misalignments and deviations of the predefined center of gravity of the CubeSat can cause the CubeSat to start rotating. A further reduction of the thrust was also considered, however, since the design of the thruster is based on orifice plates which possess a very small orifice diameter of only a few micrometers, the minimum thrust is limited by the minimum possible orifice diameter that can be manufactured economically feasible. Further details on the orifice plates and their coherence to the thrust is described in chapter 5.1.1.

Table 13: Summary of the performance requirements on the DWPS

Specific Impulse	300 s
Thrust at MEOP	< 1 N

Lifetime

The CubeSat WPS shall be designed to withstand an operational lifetime of five years in orbit. Even though the main tests and maneuvers shall be performed within the first three to six months, the WPS shall withstand this long period in space to perform a few maneuvers after each year in orbit. Thereby, the degradation and long term performance in orbit shall be determined especially since the final WPS shall be functional around seven to eight years. Before the launch of the CubeSat, the WPS is integrated into the CubeSat, which then is integrated into the deployer, and the deployer is finally attached to the launcher. Within this time it is transported and stored various times. Hence, the CubeSat WPS system shall be capable of withstanding a duration of 2 years on ground before the actual operation. After the mission is completed and all necessary and possible tests were conducted, the CubeSat maintains its position within an orbit circling the earth. Due to residual particles in LEO the satellite will decelerated by the drag force and reenter the earth's atmosphere after a certain period. As stated in the CDS of the generic requirements, this shall happen within 25 years after the end of the mission. The CubeSat with its WPS must survive these additional 25 years in orbit without breaking apart. The fragmentation of the CubeSat before entering the atmosphere would cause unpredictable space debris that increases the risk of collisions and must be prevented.

Table 14: Lifetime of the DWPS

Storage Lifetime	2 years
In-orbit Lifetime	5 years
EOL Lifetime	25 years

Operational Media

The WPS system shall be compatible with various fluids for cleaning, testing and operating the system. As usual for space systems, either nitrogen or helium shall be used as pressurant. The inert characteristic of these gases prevent chemical interactions with the water or components of the fluid system. The water, used for the electrolysis process shall be of high purity. Impurities, such as ions dissolve at the electrolyzer's membrane and result in a higher membrane resistivity, reducing its efficiency. The purity can be determined by the waters resistivity. A high electrical resistivity indicates pure water while electrolytes and ions contaminating the water make it conducting and lower the water's resistivity [55]. The resistivity of the water used for the WPS shall be greater than $1\ M\Omega \cdot cm$ analog to a conductivity smaller than $1\ \frac{\mu S}{cm}$ which corresponds with the water requirements for commercially available PEM electrolyzers and is defined in ASTM-D-1193: "Standard Specification for Reagent Water" as type II [57][58]. The gases produced by the electrolyzer are hydrogen and oxygen. However, a certain amount of water vapor is able to pass through the electrolyzer and mixes with the respective gases. The electrolyzer for the WPS demonstrator has not been tested yet, but previous PEM electrolyzers tested for a WPS at the Institute of Space Systems (IRS) produced gases with a relative humidity of around 30% [56]. As a conservative approach, the WPS shall be fully functional with the gases fully saturated to a relative humidity up to 100%. Furthermore, it shall be possible to clean the WPS components with isopropyl alcohol (IPA), as defined in TT-I-735A or deionized water type II, as described in ASTM-D-1193, before the assembly of the WPS.

Pressure Requirements

The pressure within the WPS is a main quantity that defines the WPS performance. The water side and the gas side of the WPS operate at two different pressure ranges. The pressure ranges are defined by the Minimum Operating Pressure (MOP) and the Maximum Expected Operating Pressure (MEOP) as described in ECSS-E-ST-32C. The water side, pressurized by nitrogen or helium, shall maintain a pressure between a MOP of 2 bar and a MEOP of 12 bar. Generally, the elevtrolyzer creates a pull by the osmotic pressure and thereby realizes a water influx. The pressure range specified for the water side is required to determine the water fill level and was chosen after considering the following points. A higher initial pressure allows a reduced head space for the pressurant and, in addition, a greater range between maximum and minimum operating pressure allows a more precise measurement of the water fill level. On the contrary, a tank that is able to withstand a high MEOP requires a greater structural strength and thus, is heavier. The maximum pressure within the water side during the launch shall not exceed 250 psia, equal to around 17.2 bar, since components with a higher pressure are classified as hazardous flight hardware by the Air Force Space Command Manual (AFSCM) 91-710, volume 3. This manual shall be applicable for the DWPS to comply with the CubeSat Design Specification [17]. As defined by the manual, any hazardous flight hardware must undergo significant testing, inspection and certification and must comply with various design requirements, which shall be circumvented for the DWPS water tanks and the electrolyzer. Reviewing these aspects, a water side MEOP of 12 bar was considered as an appropriate pressure.

The WPS' gas side is only pressurized in orbit at its actual operation by the electrolyzer. While the liquid water possesses a high density, a relative large amount can be stored easily. The gases produced by the electrolyzer possess only fraction of the liquid water's density. To store a sufficient amount of hydrogen and oxygen in a small envelope, high pressures must be achieved. The PEM electrolyzer is capable of pressurizing the gases passively without any further equipment. A pressure of around 80 bar shall be achieved for the WPS. To reduce the complexity of the DWPS design, a MEOP of 50 bar was chosen. During a maneuver, the pressure difference between the gas tank and the combustion chamber drives the outflow of the gas towards the thruster. Thereby, the pressure within the gas tanks reduces continuously. At a certain point, the pressure difference is so low, that the risk of a backflow of hot combustion gases increases. The propulsion system shall therefore be operated with a minimum pressure of 10 bar.

To ensure the pressure resistance of pressure exposed components, they shall be tested at a higher pressure than the pressure they will be exposed to during the operation. A proof pressure shall be applied to verify the components integrity after the application of a higher pressure and a burst pressure, even greater than the proof pressure, shall verify that a certain pressure level can be achieved without the collapse or rupture of a component. The burst pressure, unlike the proof pressure, shall not be demonstrated on flight hardware, but rather on a qualification or demonstration model. The pressure level for proof and burst pressure testing is defined by a respective proof or burst factor which is multiplied with the MEOP. Proof and burst pressure factors depend on the particular component and its use within the system. For example, the proof and burst factor for a pressure vessel such as the water tank are 1.25 for the proof and 1.5 for the burst pressure. Resulting in a proof pressure of 1.25 x MEOP = 15 bar and a burst pressure of 1.5 x MEOP = 18 bar. These factors are defined in the ECSS- E-ST-32-02C standard on “Structural design and verification of pressurized hardware” and shall be determined for each pressure exposed component individually in the respective subsystem specification.

Table 15: Water and gas side MEOP and MOP at the nominal temperatures

Water Side Pressures		**Gas Side Pressures**	
MEOP	12 bar	MEOP	50 bar
MOP	2 bar	MOP	10 bar

It shall be noted, that to simplify the pressure definitions at this early phase, the pressures are only defined for the nominal temperature. However, at the maximum operating temperature, the pressure will increase depending on the volume, which should be considered during the design of the pressurized components.

Leakage

During the operation of the DWPS in orbit, the pressurized components will be exposed to a pressure difference to the vacuum environment or to connected components with different pressure levels. Any porosity, hole or crack will allow the stored gases to leak and to reduce the pressure difference. The small size of gas molecules make it impossible to fully prevent leakage, which is why a maximum allowable leakage rate must be defined for a pressurized system. The definition of a low leakage rate for the DWPS components is required due to two reasons. First of all, a high leakage rate reduces the amount of available propellant. Especially in the gas tanks, a leakage on only one gas tank could result in a pressure difference between the two tanks and would alter the ROF and performance of the propulsion system. And secondly, the leaking gases might accumulate and cause a hazardous gas mixture that could ignite and damage the CubeSat. To define a leakage rate for the DWPS, the leakage rate of former propulsion modules and of the actual WPS were reviewed. From these systems, a leakage rate for the leakage of gaseous helium was derived for internal and external leakage, which is assumed to be sufficient for the CubeSat mission. The external leakage constitutes the leakage from a component to the surrounding vacuum. The internal leakage is defined as the leakage between two components, e.g. the leakage through a valve seat. The defined maximum leakage rates for components of the DWPS for the operating pressure range as defined in Table 15 are given in Table 16.

Table 16: Maximum leakage of a DWPS component

External Leakage	1×10^{-6} scc/s GHe
Internal Leakage	1×10^{-5} scc/s GHe

The overall external leakage shall thereby not exceed the external leakage of 1.5×10^{-3} GHe as it is specified for the actual WPS.

4.3.2 Design & Manufacturing Requirements

Mechanical Requirements

As specified in the CDS, the maximum weight of a 6U satellite is defined by the dispenser and is generally limited to 12 kilogram. The CubeSat platform weighs around 6 kg, allowing the WPS to have a mass of 6 kg. To include any uncertainties into the mass budget, the CubeSat WPS shall be designed with a mass margin of 20% resulting in a design mass of 5 kg. The CubeSat platform components, including the OBC, EPS, ADCS and the communication system require an envelope of 2U so that a volume of 4U is available for the WPS. The WPS's component layout must be chosen in a manner to achieve the requirements for the CubeSat center of gravity (X-Axis: ±4.5 cm; Y-Axis: ±2 cm; Z-Axis: ±7 cm) as specified in the CDS and stated in chapter 4.2.2.

Each component of the DWPS that is mounted to the CubeSat must possess a certain stiffness which defines the natural frequency of the respective component. The required minimum natural frequency is generally defined by the launcher, since the vibration throughout the launch should be attenuated by the CubeSat and its components. At the current state of the development, a launcher has not yet been chosen. However, from former propulsion system developed by the ArianeGroup it can be derived that the natural frequency of a hard mounted component should be above 150 Hz.

Cleanliness Requirements

The WPS operates with small mass flows and thus, with a compactly designed fluid system. The diameter of various tubes and fittings is very small with a minimum diameter at the thruster's orifices of only a few micrometers. To prevent any damage or blockage of the orifices, the WPS must be clean and possess hardly any contamination. Therefore, the CubeSat WPS shall be measurably clean to a level equal to, or below the contamination stated in the following table

Table 17: Cleanliness requirement for the DWPS and its subsystems

Particle size	Max. allowable count of particles for entire WPS*	Max. allowable count of particles for a WPS subsystem*
>100 µm	0	0
51 – 100 µm	1	0
26 – 50 µm	5	2
11 – 25 µm	20	9
6 – 10 µm	140	60

** The particle count is related to a sample of 100 cm^3 of liquid or $1m^3$ of gas*

The cleanliness requirements of Table 17 are specified by the ECSS-E-ST-35-06 standard for propulsion systems with an external diameter bellow 20 mm and have been modified to reduce the particle count for the WPS. To further decrease the particle count at the orifice inlet, a filter shall be implemented that only permits particles with a size greater than 2 µm pass through.

The flushing media, IPA or deionized water, must be removed by a drying process after the cleaning procedure. During the drying process, non-volatile material or particles are deposited by the evaporating flushing media. To prevent the contamination with so-called non-volatile residue (NVR) the flushing media shall not contain more than 50 mg of NVR per liter flushing media, as also specified in the ECSS-E-ST-35-06 standard.

In addition, to prevent any kind of contamination, all operations on the WPS, including the cleaning, assembly and packaging, shall be performed under clean room conditions class 8, or better, as specified in EN ISO 14644.

4.3.3 Environmental Requirements

The environmental requirements state the conditions that the WPS must withstand on ground, during launch and in orbit. Each environment is influenced by natural and induced effects. An emphasis is set on the temperature and mechanical environments during these three phases. The influence of the radiation within a LEO is neglected within this thesis. A detailed overview of the radiation environment in a LEO can be found in the ECSS-E-ST-10-04C Rev.1 document.

DWPS Temperature Requirements

Especially the temperature, at which the WPS is handled or operated has to meet strict criteria. Generally, the WPS operates at two different nominal temperature. The water side, including the water tank and feed lines to the electrolyzer shall possess a nominal operating temperature of 20 °C, while the gas side, including the electrolyzer itself, the gas tanks plus the feed lines for the gasses from and to the gas tanks shall be kept at a nominal operating temperature of 70°C. For the water side, which shall be kept at 20 °C, a temperature range of 10° to 60° C shall be permitted. The main objective is to keep the temperature above the freezing point and below the boiling point.

The gas side's temperature must be kept in a much tighter range. The gases produced by the electrolyzer possess a temperature of 70°C and are mixed with a certain amount of gaseous water (around 30% of relative humidity). If the gas side temperature drops, the relative humidity rises until the gas is fully saturated and liquid water starts forming which in return reduces the thruster's performance. The gas side operation temperature range was therefore specified to a range between 65° and 75°C. When the electrolyzer is in the power off state and not producing any gases it shall be kept at a temperature above 10° and it must be ensured that it does not exceed a temperature of 80°C in any case. An overview of the temperature values are given in the following table.

Table 18: Nominal and operating temperatures for the water and gas side of the WPS

Water Side Temperatures		**Gas Side Temperatures**	
Nominal	20° C	Nominal	70° C
Operating	10° - 60° C	Operating	65° - 75° C

Ground Environments

The DWPS must be capable of withstanding the storage, transport and pre-launch environments defined hereafter. The environments were derived from former projects and are summarized in Table 19. Compared to conventional hydrazine propellant systems, a great advantage is the possibility to fuel the propulsion system at the manufacturing site. However, during storage and transportation the temperatures might drop down to -25°C so that the water carrying components must be thermally controlled throughout these phases. Aside from the thermal and climatic environments specified in the table, the DWPS will be exposed to mechanical influences during the transport. During air, sea and overland transportations, the system will be handled and moved with accelerations in vertical and horizontal directions. The maximum acceleration of around 4.5 g is expected during emergency situations in air transportations. The acceleration during hoisting, rolling, rotating and integrating of the DWPS as well as the ground transportation is expected to be around 2 to 2.5 g as a maximum.

Table 19: Climatic and thermal ground environments

Storage Environment	
Temperature	-25°C to +65°C
Relative Humidity	Max. 70 %
Pressure	970 to 1060 hPa
Cleanliness	Class 8 (EN ISO14644)
Transportation Environment	
Temperature	-25°C to +65°C
Relative Humidity	Max. 70 %
Pressure	700 to 1100 hPa
Rate of Pressure Change	3.5 kPa/sec
Pre-Launch Environment	
Temperature	+10°C to +50°C
Relative Humidity	Max. 70 %
Pressure	970 to 1060 hPa

Flight Environment

During the flight, the temperature will be controlled by the DWPS so that the exposed temperatures are expected to be equal to the operating temperatures. The external pressure drops from 1 atm to around 10^{-10} mbar in LEO during the launch. Furthermore, throughout the launch, the CubeSat and the DWPS has to withstand the maximum mechanical forces. To prove that the system is capable of withstanding these environments, each subsystem must be exposed to random and sinusoidal vibration, acceleration and shock loads during a qualification testing campaign. Typical loads for such tests are derived from former projects and are reproduced in the Appendix B –.

4.3.4 Interface Requirements

Hydraulic Interface

The tubes connecting the fluidic components shall be kept as small as possible. The mass flows of the fluids in the DWPS are very small and do not require a thick tube diameter. The minimum diameter is thereby limited by the welding possibilities at the ArianeGroup. The smallest diameter of tubes that can still be welded together is 1/8”. If possible, the use of welded connections shall be preferred to screwed connections, to reduce the leakage on the one hand, and to reduce mass and volume of large fittings on the other hand. If a welded interconnection is not possible, the screwed fitting shall comply with the AS4395-2 standard.

It shall be possible to fuel the DWPS once it is assembled and integrated into a CubeSat structure. Therefore, Fill and Drain Valves (FDVs) for the water and pressurant shall be accessible for the integrated system.

Electrical & Logical Interface

At the current state of the development it is impossible to accurately define the electrical and logical interface to the CubeSat. Main problem is the limited progress on the design of the CubeSat platform and the ECU which specifies the electrical and communication interface. Therefore, rather than defining the electrical and logical interface, possible connections to the CubeSat will be described in the following.

For the communication of the DWPS with the CubeSat a full or half duplex communication channel is required so that the OBC can send commands to the DWPS and the DWPS can send sensor data to the OBC in return. A possible and commonly used standard of a digital signaling circuit for CubeSat propulsion system is the RS422 standard. However, a communication channel based on this standard can only be used for simplex communication. A full duplex communication must therefore be assured by adding further wires. Other available channels are data buses such as the CAN-Bus, SPI, I2C or the RS-485 connection.

The electrical power supplied by the CubeSat’s EPS to the DWPS will be regulated and distributed by the ECU. This shall allow a “Plug & Play” utilization of the DWPS. However, as depicted, the ECU design has not yet been developed. The EPS can provide the ECU with unregulated power lines with a voltage dependent on the battery voltage and regulated power lines between around 3 and 18 V. The design of the ECU will then determine the respective power lines. Further information on the current state of the electrical architecture is described in chapter 5.1.4.

Mechanical Interface

Aside from the gas tanks and electrolyzer, each subsystem shall be mounted to the CubeSat structure individually. The mounting must be capable of safely fixing the component to the structure while being exposed to the mechanical loads that will be applied during the operation and most severely during the launch.

4.3.5 Verification Requirements

Each requirement defined herein must be verified by one or more of the following methods: review, inspection, analysis or test (including demonstrations). Each method is specified in the ECSS-E-ST-10-02C Rev.1 standard as described hereafter:

- Review: Verification by the review of a design shall be performed by issuing records or evidence such as technical descriptions, reports, design documents etc. to prove that the requirements are met.
- Inspection: Verification by inspection shall be performed by the determination of physical characteristics.
- Analysis: Verification by analysis shall be performed by using theoretical or empirical evaluation methods.
- Test: Verification by test shall be performed by measuring a systems performance and functionality under operating conditions.

Former projects have shown that improper ground testing increases the projects risks by resulting in the late discovery of defects and deviations or even the in-orbit failure of the system [59]. Thus, the emphasis in this chapter will be the verification by tests to accurately specify the test requirements and procedures for the development of a sufficient test campaign for the WPS and its subsystems.

Depending on the objective of a test, it can be assigned to one of four categories. Within the products life cycle, development tests are the first tests to be performed. They are performed to obtain necessary information for the design and manufacturing of the product. These test are used to aid the design process and to obtain a better knowledge of a systems operation and behavior so that safety margins, failure modes, and design parameters can be determined. Development tests do not verify the systems requirements and hence, do not require a formal documentation. Once a design has been implemented and the product was manufactured and assembled, qualification tests must be performed to demonstrate, that the product performs according to the specified requirements. The qualification tests shall be performed on a qualification model and any destructive tests, such as a burst tests shall be conducted at the end of a qualification program. The acceptance tests, on the contrary, shall provide the evidence that the flight model performs according to the requirements. The flight model will be exposed to a reduced load in the acceptance tests compared to the qualification model in the qualification tests. Thereby, the acceptance tests shall demonstrate the products acceptability for delivery.

Test Requirements

Any instruments used to measure the test parameter shall possess an accuracy that is an order of magnitude better than the tolerances for the measurement parameter. The measurement equipment's accuracy must be verified and calibrated with an approved calibration method. Any equipment that is expected to forfeit its required accuracy within the defined testing time shall not be used. Furthermore, the equipment that can affect the DWPS' cleanliness, shall also conform to the cleanliness requirements defined above. And at last, time shall be envisaged for the test equipment to reach the temperatures specified for the test. All temperature readings shall be within 3°C of the specified temperature.

Generally, if not otherwise stated, the test of the DWPS shall be performed under ambient conditions, as shown in Table 20. The actual ambient test condition that occur during the test shall be recorded and if the ambient conditions exceed the defined limits the responsible test operator shall decide if the test can be proceeded, must be paused or even be terminated. A sufficient evidence must be available, to ensure that there is no adverse influence on the components performance due to the off-limit ambient conditions.

Table 20: Ambient conditions for tests

Pressure [mbar]	944 – 1080
Temperature [°C]	22 ± 3
Relative Humidity [%]	40 – 65

After the tests, the investigated component shall be examined for damage. If any modifications or changes are performed on the component after the test, the test lapses and must be repeated with the modified component.

For acceptance and qualification testing several formal reports have to be issued. The record must contain the relevant test data, the equipment used, the predefined test procedures and any irregularities that occurred during testing. This information shall be provided within the four documents: a Test Plan, a Test Report, Test Procedures and a Non-Conformance Report (NCR), if any failures, defects or deviations occur. While the NCR strongly depends on the type of non-conformance and it fatality, the other three documents shall contain the information shown in the following chart, Figure 23.

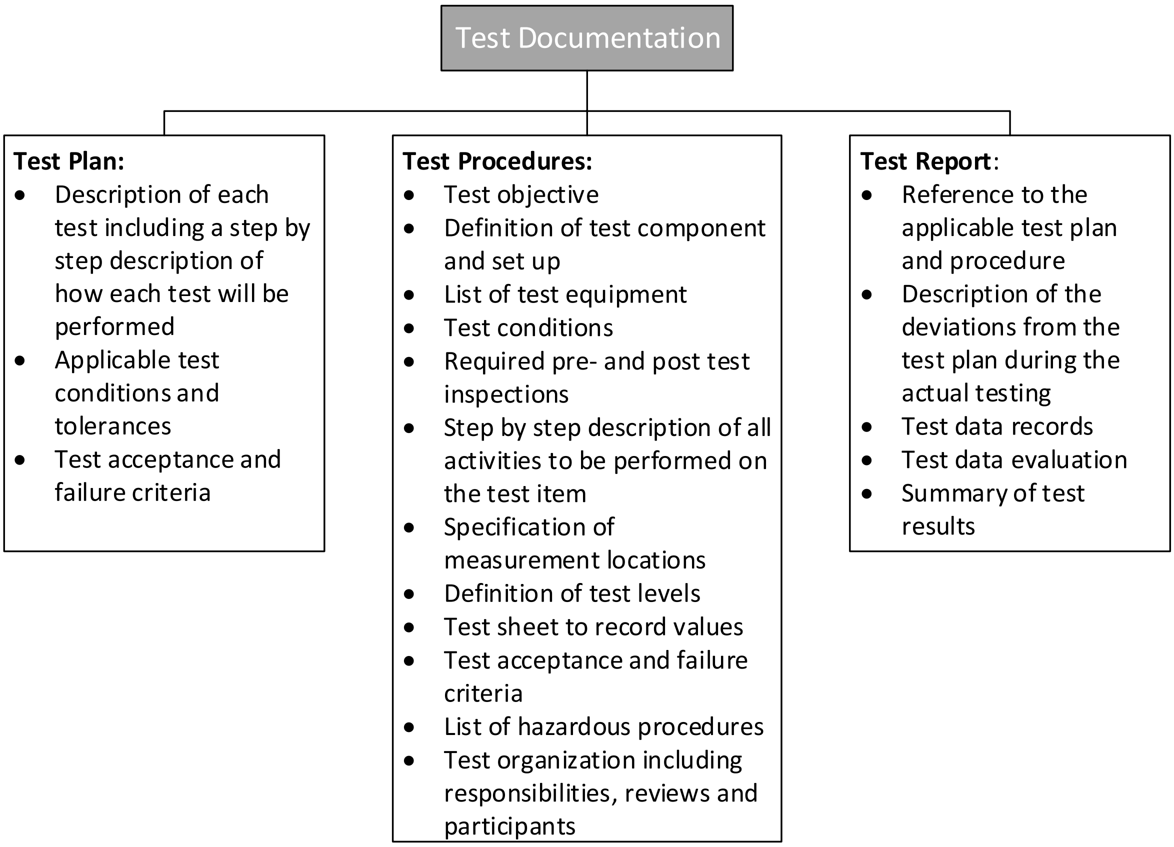

Figure 23: Required records for the test documentation and the respective content of each issue

Acceptance and Qualification Tests

The DWPS system and its subsystems shall be verified by completing a list of acceptance and qualification tests that are listed in the Appendix B. The qualification tests shall be performed on a qualification model, while acceptance tests shall be performed before the delivery of the DWPS or the integration and assembly of the subsystems. Destructive tests, such as burst pressure tests shall be performed on the respective subsystem's qualification model rather than on the assembled system.

4.4 Technical Requirements - Thruster

The Thruster burns the gas within its combustion chambers and accelerates the resulting gases in a nozzle. The design shall be derived from the actual WPS design. This includes the catalytic combustion in a pre-combustion chamber and the film-cooling by a separated oxygen flow in the main combustion chamber.

Therefore, the thruster shall be assembled with the following components:

- Heat barrier
- Decompositions chamber with a catalyst bed
- Catalyst
- Injector
- Nozzle including the combustion chamber
- Assembly flange
- 2x flow control valves (FCV)
- 2x particle filters
- 2x catalyst bed heater (CBH)
- 2x type S thermocouples
- 2x orifice plates

The assembly of the components must be performed accurately since small deviations might affect the thruster's alignment.

Some of the most important requirements have already been defined in the DWPS System Specification and are summarized in the following Table 21. The included orifice diameters were calculated according to equation 2.7 to allow a sufficient mass flow for the defined thrust. The values were determined in a previous study by the author of [10].

Table 21: DWPS system requirements that influence the thruster assembly design

Specific Impulse	300 s
Thrust at MEOP	1 N
External Leakage (Component)	1×10^{-6} scc/s GHe
Internal Leakage (Component)	1×10^{-5} scc/s GHe
MOP / MEOP	10 bar / 50 bar
H_2 – Orifice diameter	126 µm
O_2 – Orifice diameter (catalyst)	57 µm
O_2 – Orifice diameter (combustion chamber)	170 µm

4.4.1 Functional Requirements

Performance Requirements

Throughout the operation, the thruster shall demonstrate two different operating modes: continuous firing and pulse mode firing. These modes shall be performed according to the performance requirements defined hereafter at the nominal propellant feed temperature of 70°C and the pressure range between MEOP and MDP. Additionally, the thruster shall demonstrate the capability of being operated safely and without degradation at an off-limit temperature of 60C° to 80°C and pressure between 8 and 55 bar. Many requirements for the thruster's performance are derived from the requirements specified for the actual WPS including the oscillation, repeatability and reproducibility of the generated thrust or impulse and are summarized in the following table. It must be noted, that the firing durations of the DWPS will be limited drastically compared to the firing durations of the WPS. Within the short firing durations, a maneuver cannot be assumed as steady state firing. Thus, the thrust roughness and thrust repeatability will probably not be achieved during the actual operation with the nominal firing durations.

Table 22: Continuous firing and pulse mode firing performance of the CubeSat demonstrator thruster

Continuous Firing	
Thrust Roughness	For 90% of firing period, the thrust oscillation (peak to peak) shall not exceed ±5% at 50 bar (MEOP) and ±10% at 10 bar (MDP)
Thrust Repeatability	Thrust for two consecutive firing shall not deviate by more than 10% at MEOP and MDP
Pulse Mode Firing	
Minimum ON/OFF Time	Shortest pulse = 25 ms
Minimum Impulse Bit	< 1Ns at MEOP and 25 ms pulse on time
Impulse Bit Reproducibility	The thruster shall be capable of reproducing an impulse bit in quasi steady state conditions for a 2 second pulse within a range of ±10 % at MEOP and ±15% at MOP
Impulse Bit Repeatability	The total impulse of two consecutive pulse trains with identical conditions shall be equal with a maximum deviation of 5% at MEOP and 10% at MOP
Response Times	Rise ≤500 ms Decay ≤500 ms Centroid Delay ≤150 ms

The response times for an impulse are illustrated in the following graph, Figure 24 [61]. The stated response times apply for the preheated thruster.

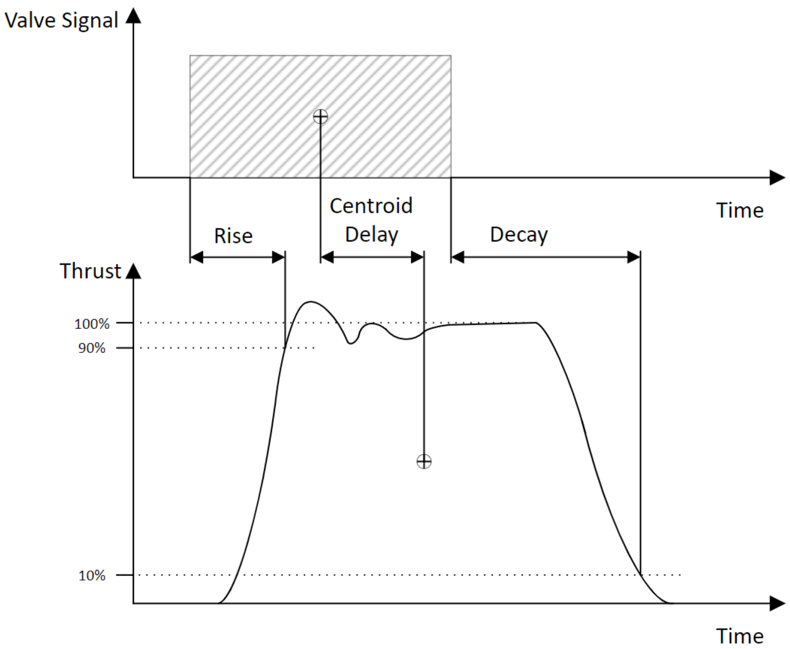

Figure 24: Firing response definitions

Operational Media

The thruster will be fed by the electrolyzer, which produces hydrogen and oxygen. However, the respective gases will also contain a certain amount of water vapor. Thus, it must be verified, that the thruster can operate with gases that possess a certain humidity. Former, similar PEM electrolyzers developed at the Institute of Space Systems have shown, that a relative humidity of around 30% can be achieved [10]. Nevertheless, the humidity might increase due to a deviation from the nominal temperature within the gas tanks or the tubes interconnecting the respective subsystems. Therefore, the thruster shall be capable of operating with gases that are fully saturated.

Pressure Requirements

The thruster assembly must withstand the same pressure loads as defined for the gas side of the WPS. This means that the Maximum Expected Operating Pressure (MEOP) is limited to 50 bar with a minimum operating pressure of 10 bar. During the operation of the WPS the tanks will be refilled after each maneuver. The thruster shall be capable of withstanding at least 1000 pressure cycles during the life cycle for ground tests and in orbit operation. Further pressure requirements are defined in Table 23.

Table 23: Pressure requirements for the thruster assembly

Proof Pressure FCV	1,5 x MEOP = 75 bar
Proof Pressure Thruster	1,5 x MEOP = 75 bar
Burst Pressure FCV	4 x MEOP = 200 bar
Burst Pressure Thruster	4 x MEOP = 200 bar
Reverse Differential Pressure FCV	1 bar

The proof and burst pressure for the thruster are defined for the configuration with open FCVs and a capped nozzle. The reverse differential pressure defines the pressure the FCV valves must withstand in reverse direction. This is important especially during the phases on ground, where the gas tanks are evacuated and the ambient pressure of 1 bar pressure must be withstood. Additionally, the thruster shall be capable of depleting the gas tanks from 10 bar down to 0 bar for the passivation of the WPS at the end of life.

4.4.2 Design & Manufacturing Requirements

Mechanical Requirements

The thruster assembly shall be designed so that it does not weigh more than 300 g and fits in a cylindrical envelope with a height of 10 cm and a diameter of 5 cm. The envelope does not include the mounting flange. Deviations from these parameters must be reviewed and might be acceptable.

The thruster shall generate a thrust with a thrust vector aiming through the satellite's center of gravity. The offset of the thruster's alignment must comply with the maximum torque that can be compensated by the reaction wheels. The thruster shall under no circumstances create a torque on the satellite that is greater than 3.4 mNm.

Flow Control Valve Requirements

The flow control valves (FCVs) initiate and terminate a thrust maneuver. They shall be designed in a way that they return to a closed state in the event of a loss of signal. Furthermore, to allow the firing of precise maneuvers, the FCVs shall be capable of opening and closing within a duration of 10 ms. Thereby, the response time between the valves shall not differ more than 1.5 ms. The FCVs shall be capable of being actuated with a voltage of 12 VDC and held open at a reduced holding voltage to ensure a reduced power usage and heat production during phases with long opening times.

Particle Filter Requirements

A particle filter shall be mounted before the FCVs on each gas line to prevent the decontamination of the FCVs and especially, to prevent a blockage at the orifice plates mounted just after the FCVs. The degradation, enlargement or blockage of the orifice would alter the performance or even obstruct the functionality of the thruster. Particles larger than 2 µm shall be filtered at the FCV with this additional filter.

Catalyst Best Heater Requirements

The catalyst test bed shall be heated with a test bed heater. The heater shall be capable of heating the catalyst test bed within 30 minutes to the required temperature to allow a sufficient ignition. Tests with the catalyst will be performed in the near future. Currently, it is estimated that the CBH must heat the catalyst to a temperature of at least 250°C and if possible to around 550° to 600°C. For redundancy reasons, the catalyst test bed shall be equipped with two heaters that can, if required, operate at the same time. The heaters, shall be wired individually with a voltage of 12 volt. On the contrary, the thruster shall be capable of withstanding up to 10 firing maneuvers at a temperature of only 4°C to ensure the functionality of the thruster even if the heater malfunctions and also to allow a maneuver on a short notice, e.g. to avoid a collision in space.

Temperature Sensor Requirements

The Thruster must be equipped with two temperature sensors, one at the catalyst test bed, and the second at the thruster's combustion chamber. Essentially, the catalyst test bed temperature must be measured, to ensure that the heater has achieved the required temperature to start a maneuver. The temperature sensor at the combustion chamber is used to characterize the thrust maneuver, to verify the functionality and to compare the temperature of the in-orbit tests to the ground tests performed in a vacuum chamber. In general, the reaction between oxygen and hydrogen produces high heat loads resulting in a maximum temperature of around 3500 K at the stoichiometric ROF. The maximum temperature at the combustion chamber wall or the catalyst test bed will be lower, since the mass flow of oxygen is split up resulting in a ROF below the stoichiometric ratio of 8 in the catalyst chamber and a ROF above 8 in the combustion chamber [10]. However, the temperature at the outer wall of the combustion chamber and the catalyst test bed will still be severe requiring temperature sensors that are able to withstand these high heat loads. The only sufficient temperature sensors are high temperature metal sheeted thermocouples that can resist temperatures of 1800 K, such as a type S Rhodium coated thermocouples [62]. A type S thermocouple is preferred compared to the type R thermocouple, which is capable of operating up to the same temperature but is less precise [63].

4.4.3 Environmental Requirements

Aside from the thermal and mechanical loads defined in chapter 4.3 the thruster will be exposed to a severe thermal load throughout the firing. The temperatures reached for the defined pulse and continuous firing must be determined and the thruster assembly must be capable of withstanding these loads for the entire lifetime without degradation or damage.

4.4.4 Interface Requirements

The thruster assembly shall be mounted to the CubeSat structure by a mounting flange. The flange must be designed to comply with the alignment accuracy described above and adjustments of the alignment must be possible once the thruster is mounted. To determine the alignment of the thruster before the launch, the assembly must be accessible for a suitable alignment measuring device.

During ground handling, storage and transport, the nozzle opening shall be covered by a protective cap to prevent any kind of contamination. Additionally, the nozzle shall be compatible with a leak check device which shall be attached to the nozzle and measure the leakage past the flow control valves.

4.5 Technical Requirements - Water Tank Assembly

The water tank assembly is responsible for the storage of the water in its liquid phase throughout the WPS' life cycle. It shall consist of the following components:

- 2x water tanks
- 2x heaters
- 4x temperature sensors
- 1x pressure transducer
- 1x fill and drain valve (FDV) for the water
- 2x fill and drain/vent valve (FDV) for the pressurant

Each water tank shall be equipped with a heater and two thermal sensors. Two temperature sensors make it possible to determine the temperature at two locations on the tank and thereby take temperature differences on the tanks surface into account. To reduce expenses and required resources, the water tanks shall be COTS components.

A few of the main requirements for the water tank are already defined by the DWPS system specification in chapter 4.3 and are therefore briefly summarized in the following table.

Table 24: Main WPS system requirements that influence the water tank assembly design

Pressurant	He, N_2
Water quality	Distilled, resistivity > 1 $M\Omega \cdot cm$
MOP / MEOP	2 bar / 12 bar
Nominal temperature	20°C
Operating temperature range	10° - 60°C
External Leakage (Component)	1 x 10^{-6} scc/s GHe
Internal Leakage (Component)	1 x 10^{-5} scc/s GHe

4.5.1 Functional Performance Requirements

Performance Requirements

The amount of water that can be carried aboard the CubeSat is mainly limited by the reduced available space. The volume efficiency shall therefore be a main driver in the design of the water tanks. With the MOP and MEOP values for the water tank, the ullage is defined to 1/6 of the tanks available volume. Furthermore, the water tanks shall be capable of draining as much water out of the tank as possible. A minimum of a 95% expulsion efficiency shall be acceptable.

Pressure Requirements

The water tank assembly is exposed to the water side pressure between 12 and 2 bar throughout the entire DWPS lifetime and must be capable of storing the water and pressurant leak tight at these pressures. To verify the water tanks and its connected components capabilities under these condition it shall be tested up to a proof and burst pressure as listed in Table 25.

Table 25: Proof and Burst Pressure requirements of the water tank assembly's components

Proof Pressure FDV	1.5 x MEOP = 18 bar
Proof Pressure Water Tank	1.25 x MEOP = 15 bar
Proof Pressure Pressure Transducer	1.5 x MEOP = 18 bar
Burst Pressure FDV	4 x MEOP = 48 bar
Burst Pressure Water Tank	1.5 x MEOP = 18 bar
Burst Pressure Pressure Transducer	4 x MEOP = 48 bar

These safety margins conform to the margins defined in the ECSS-E-ST-32-02C standard on structural design and verification of pressurized hardware.

4.5.2 Design & Manufacturing Requirements

Mechanical Requirements

The water tank assembly with all components defined in the components list above excluding the FDVs and filled with water and the pressurant shall not weigh more than 700g. Within the CubeSat, the water tank assembly shall be placed left and right of the thruster within an envelope of around 75x80x100 mm on each side. The fittings for the tanks must be included within this envelope. To prevent the shift of the center of gravity, it must be ensured, that the water tanks deploy the water at the same rate.

Fill and Drain Valve Requirements

The fill and drain valves for the pressurizing gas and the deionized water must be accessible even when integrated within the CubeSat to allow any refueling or tests in the assembled configuration. The performance of the valves are specified by the leakage and pressure defined for the water tank assembly in the previous segment. Since the actuation of the FDVs is only required during fueling, they shall not be actuated by the ECU. Thus, they shall either be actuated manually or by an external power supply.

Pressure Transducer Requirements

A pressure transducer shall be capable of measuring a pressure up to the tank's MEOP of 12 bar. In combination with a temperature sensor, the measurement shall provide the means to calculate the water tanks filling level. Since the mass of water used to fill up the gas tanks is very low, a pressure difference cannot resolve the pressure drop that occurs for a single maneuver. The possibility of complementing the filling level of the water tank by a position sensor will be evaluated in chapter5.2.

Temperature Sensor Requirements

The temperature of the water tank shall be measured by type K thermocouples which can easily operate in the temperature range between 0°C and 100°C. In combination with the heaters, the thermal sensors allow the control and regulation of the water's temperature. In combination with the pressure sensor, the filling level can be determined. Two sensors shall be used on each water tank to allow a precise and redundant measurement. One of the sensors shall be placed on the side facing the thruster. This will allow the determination of the temperature at the water tank's surface during a maneuver.

Heater Requirements

The temperature shall be adjusted by a heater attached to each water tank. It must ensure, that the temperature of the water stays above the freezing temperature. A voltage of 12V is commonly applied for COTS heaters which complies with the capabilities of the CubeSat's EPS. The power must be determined once a thermal analysis has been performed for the CubeSat.

4.5.3 Environmental Requirements

The water tanks will be exposed to high thermal loads during the firing of the thruster. Since the heat will be transferred by thermal radiation, the exact heat load is dependent on various influences (Surface of the water tank, distance between thruster and tank, temperature and emissivity of the thruster etc.). Thus the exact heat load must be examined and the effect on the water tank investigated. If the heat load results in a temperature exceeding the maximum temperature for the water tank, a heat shield must be implemented. Further environmental influences expected during launch and the phases that are not influenced by the firing are described in the WPS system specification of chapter 4.3.

4.5.4 Interface Requirements

Each water tank shall be mounted to the CubeSat structure individually. The FDVs for the pressurant and the water must be accessible even in the integrated state to allow refueling of the DWPS. The FDVs shall be attached to the tubes by electron beam welding if possible, to prevent additional leakage. Additionally, the water fill and drain valve shall be attached to an interface panel which shall is placed on a side of the CubeSat which allows access to the hydraulic components.

4.6 Technical Requirements - Electrolyzer and Gas Tanks Assembly

The electrolyzer, as the main and novel technology to be tested in orbit, shall be operated redundantly. Therefore, two individual segments shall be designed for the electrolyzer that each possess a separate water inlet and gas outlets. Each segment shall consist of a double cell and each cell shall consist of three bi-polar plates, a Water Feed Barrier and a Membrane Electrode Assembly as it is defined for the WPS' electrolyzer and described in chapter 2.1.1.

Additional to the two double cells the following components are required for the operation of the electrolyzer:

- 2x heaters
- 2x temperature sensors
- 2x water drain ports
- 6x check valves (CV)
- 2x latch valves (LV)

To reduce the mass and volume of the DWPS, the electrolyzer and gas tanks shall form an integral component. Thus, the following specification also defines the requirements on the gas tanks and their components which are listed in the following:

- 1x hydrogen tank
- 1x oxygen tank
- 2x heater
- 2x temperature sensor
- 2x pressure transducer
- 2x fill and drain ports

Some of the main requirements derived from the requirements specified for the DWPS system in chapter 4.3 that decisively influence the electrolyzer's and gas tanks' design are summarized in the following table

Table 26: Main DWPS system requirements that influence the electrolyzer and gas tanks assembly design

Water quality	Distilled, resistivity > 1 $M\Omega \cdot cm$
Water Side MOP / MEOP	2 bar / 12 bar
Gas Side MOP/MEOP	10 bar / 50 bar
Nominal temperature	70°C
External Leakage (Component)	1 x 10^{-6} scc/s GHe
Internal Leakage (Component)	1 x 10^{-5} scc/s GHe

4.6.1 Functional Performance Requirements

Performance Requirements

An essential requirement is the available volume within the tanks. The electrolyzer produces the gases, according to the conservation of elements in a stoichiometric ratio. Considering the ideal gas law,

$$pV = n\Re T \tag{4.1}$$

and the electrolysis reaction of water to hydrogen and oxygen,

$$2H_2O \rightarrow 2H_2 + O_2 \tag{4.2}$$

it can be seen, that two molecules of H_2O result in two molecules of H_2 but only a single molecule of O_2. Thus the amount of substance n of hydrogen and oxygen are produced in a ratio of 2:1. Regarding the ideal gas law and targeting the same pressure and temperature in the hydrogen as in the oxygen tanks, the volume of the hydrogen reservoir must be double the size of the oxygen reservoir. This includes not just the tank volume but also the additional volume added by tubes and other components. Real gas effects for this considerations can be neglected for oxygen and hydrogen since they would barely influence the ratio at these temperatures and pressures [68][69]. The volume of the gas tanks shall be limited by the available volume within the CubeSat.

For a maneuver, the gas tanks feed the gases through the respective orifice to the thruster. The mass flow is directly dependent on the pressure within the gas tanks. Therefore, it is important, that the pressure within one tank does not differ from the pressure in the other tank. A high pressure difference and the arising deviation of the nominal mass flow would result in a different ratio of oxidizer and fuel as initially designated. This could lead to unforeseen consequences, such as a fluctuating performance, the degradation or even the damage of the thruster. Thus, the difference between the hydrogen and oxygen tank pressure shall be below 5 bar before and during the firing.

The performance of the DWPS' electrolyzer depends on the power that is supplied by the CubeSat platform. A HPM and LPM which is used for the actual WPS is not feasible since the maximum power is limited. However, a power of around 50 W is expected to be feasible for the amount of gas produced within a gas charging cycle.

Pressure Requirements

From Table 26 it can be seen, that the water feed barrier must withstand the pressure difference between the water and gas side of the DWPS which can reach a difference of 48 bar in the worst case at the end of the mission lifetime when the water tank is empty and the gas tanks are fully loaded for the final maneuver. All pressurized components of the electrolyzer and gas tanks assembly must be capable of functioning after being exposed to a pressure of 1.5 x MEOP equal to 75 bar. Furthermore, all components must withstand a burst pressure of 4 x MEOP, equal to 200 bar, without rupture. The only exceptions are the latch valves which are required to prevent water entering the electrolyzer in the case of a malfunctioning WFB. Thus, the latch valve in its closed state will only be exposed to the water inlet pressure of maximum 12 bar. Therefore, the proof pressure of the latch valves can be reduced to 15 bar and the burst pressure to 48 bar. The system architecture described in chapter 5.1.4 gives a better overview of the arrangement of the respective valves and explains the task of each valve more precisely.

4.6.2 Design & Manufacturing Requirements

The electrolyzer and gas tanks assembly shall be placed in the center of the CubeSat because it is expected that it possesses the highest mass of all WPS components so that it can be ensured that the center of gravity is close to the geometric center of the CubeSat. The assembly shall weigh less than 3 kg and fit into an envelope of around 85 mm by 100 mm by 100mm. More volume is available if the design of the assembly is adapted to the 6U CubeSat structure.

The membranes of the electrolyzer, which allow water to migrate from one to the other side, must always be wet. If the membranes dry out, the further feed of water is obstructed. Hence, it is important, that the membranes are soaked with water throughout the entire mission

Check Valve Requirements

Six check valves shall be mounted to the electrolyzer, one at each of the respective water inlets and one at each of the respective hydrogen and oxygen outlets. These check valves shall prevent any back flow of gas from the gas tank into the electrolyzer or from the electrolyzer into the water tanks. These check valves shall close automatically as soon as a pressure difference occurs in reverse flow direction to prevent the outflow of any gas in case of a leak within the electrolyzer or the backflow of gas into the water tank. The reseal pressure, defined by the difference between inlet and outlet pressure at which the seal changes from open to closed state, shall therefore be 0 bar or even defined by a certain inlet pressure higher than the outlet pressure to prevent the back flow in any case. This is required especially for the case of a leakage within an electrolyzer double cell. A minor leakage would not create a large pressure difference which would keep most check valves that only seal with and reverse pressure difference. The cracking pressure on the other hand shall be kept as low as possible so that the water flow into the electrolyzer or the gas flow into the tanks is not prevented [65].

Latch Valve Requirements

The latch valves shall be actuated in the case either of a leakage of water in an electrolyzer double cell or in the case of a ruptured WFB. For both cases, the latch valve shall prevent the water from entering the defect double cell for the rest of the mission lifetime. Therefore, the latch valve shall consume only little or no power at all in either the open or the closed state since both states shall be held for a long duration. A bi-stable latch valve is preferred which must only be actuated and powered to flip its state from open to close and remains unpowered in both states. The latch valves, attached to each of the electrolyzer's water inlet tubes, shall be operated at a nominal voltage of 12V.

Temperature Sensor Requirements

The electrolyzer and gas tanks assembly will be operated at similar temperatures as the water tanks. Therefore, the same type K thermocouples shall be used which can easily monitor the temperatures between 0°C and 100°C. The measurement of the temperature, especially on the gas side is very important, since it indicates the humidity in the gas. If the temperatures drop, the gases' humidity rises until, in the worst case, the water condensates and accumulates within the tank or the electrolyzer. Thus, the thermocouples are an essential component in the temperature control circuit. The monitored temperature also indicates the gas tanks filling level if complemented by a pressure measurement. In total, at least four thermocouples shall be attached to the assembly, with one at each gas tank and one close to each water inlet.

Heater Requirements

The temperature control circuit's actuator is the heater. The heaters shall keep the electrolyzer and gas tanks at the nominal temperature within the operating temperature range. At least two heaters shall be attached to the respective gas tank. The same voltage of the water tanks heater of 12 volts is assumed to suit the application of heating the electrolyzer and the gas tanks.

4.6.3 Environmental Requirements

The electrolyzer and gas tanks assembly shall be capable of withstanding the induced environment during storage, transport, launch and in orbit as defined for the assembled WPS described in chapter 4.3.

4.6.4 Interface Requirements

For the fueling of the water tanks an outlet is required which allows remaining air or other gases to be pushed out by the inflowing water. In the fueled state the water tanks and the water feed lines must be gas free including the water chamber up to the WFB of the electrolyzer. To achieve this, the water must flow through the electrolyzer to push the gases out on the other side, requiring two water outlets at the electrolyzer which are connected to two water outlet ports. These outlet ports must be accessible by GSE for the fueling of the assembled and integrated DWPS and allow the venting during fueling.

4.7 Technical Requirements - Electrolyzer Control Unit

The control unit is in charge of operating the entire DWPS with a main focus on the challenging regulation and operation of the electrolyzer. It shall actuate the valves, power the heaters, gather and process sensor data and supply the electrolyzer with the power required during the operation. Thereby, the control unit constitutes the main interface for electrical power and logical signals between the DWPS and the CubeSat platform.

The ECU shall be composed of the following components

- Microprocessor with internal storage
- AD-converters
- DC/DC converters
- Fast switch

4.7.1 Functional Performance Requirements

Software Requirements

The control unit software must be compatible with the commands of the OBC. The CubeSat OBC is most likely to be operated by a FreeRTOS operating system and can connected to the control unit by various communication interfaces, such as CAN, SPI, I2C or UART/USART. The OBC shall thereby perform only two logical operations. It shall decide if the batteries are charged enough to activate a gas charging cycle and shall activate a firing sequence only if the reaction wheels are desaturated. The remaining logical operations shall be performed by the ECU's software. The ECU's software must be capable of transmitting the data of the temperature and pressure sensors, as well as the impedance and power supply information of the electrolyzer. A higher rate of sensor data acquisition is thereby required in phases where the electrolyzer is producing gases or the thruster is firing the gases to generate thrust. These information shall be processed and if any failure is detected, the ECU shall be capable of automatically switching the DWPS into a safe mode and sending an error message to the OBC.

An operation logic for all relevant mission scenarios shall be implemented so that a simple command by the OBC can start a predefined operation sequence. The two main sequences are the loading cycles and the firing cycles that shall be implemented in various configurations. A more detailed description of an in-orbit operation and a generic operational cycle is given in chapter 6.4 which shall give a brief overview of the communication of the ECU with the CubeSat's OBC.

For the health monitoring of the DWPS, the ECU shall possess closed-loop control circuits, so that the heaters can be activated depending on the pressure and temperature values of the respective systems. Thus, the ECU must be capable of combining temperature and pressure values to define if a heater actuation is required.

Hardware Requirements

The power supply for the electrolyzer is the main challenge of the ECU. The ECU shall provide each MEA with a current controlled power via DC/DC converters. Generally, more power can be provided on unregulated lines directly connected to the CubeSat battery, which is why the high power demand by the electrolyzer is most likely to be realized by an unregulated line. Thus, the DC/DC converter must be capable of operating at varying input voltages which depend on the batteries current state of charge.

The hydrogen pump on the contrary shall be powered at a constant voltage of around 0.8 V as it is performed for the WPS. It must be ensured that the voltage does not exceed the thermoneutral voltage of 1.4 V which could result in the gas production at the WFB and would obstruct the feed of water. Thus, a voltage of 1.0 V shall be the maximum occurring voltage across the WFB.

To allow the evaluation of the MEA's degradation, a fast switch shall allow the measurements of the impedance. A fast switch allows a short interruption of the power supply, usually within the range of microseconds. The impedance can then be calculated from the voltage and current recordings.

Including the remaining DWPS components, the ECU must provide the following components with power:

- 2x electrolyzer double cells
- 6x heater
- 2x flow control valves
- 2x bi-stable latch valves
- 3x pressure transduce

The number of heaters shall state the minimum required amount. If required, the implementation of further heaters shall be possible.

For the use of these component, the CubeSat's EPS can provide either regulated voltages of typically 3.3 VDC, 5 VDC or 12 VDC or an unregulated voltage that must be regulated by further DC/DC converters of the ECU. The input power supply therefore depends on the design of the ECU and shall not be defined as a requirement at the current state of the development. Further

On the other side, the control unit must be capable of operating the DWPS' sensors and of receiving and processing their data. The following types of sensors are used for the WPS.

- 3x pressure transducer
- 2x type S thermocouples
- 8x type K thermocouples

For the thermocouples, the ECU must be capable of detecting minor voltage changes. A change of temperature of 1°C at temperatures around 100°C creates a change of the thermoelectric voltage by the type K sensor of around 40 µV. The type S sensors will operate at higher temperatures. At around 1000°C the thermoelectric voltage detected by the sensors will only change 11 µV for a temperature change of 1°C [63]. The AD-converter processing the sensor data, must be capable of detecting such precise changes of the voltage.

4.7.2 Design & Manufacturing Requirements

Mechanical Requirements

The control unit shall be mounted to the electrolyzer and fit into the envelope of 96 x 90 x 50 mm which complies with the PC104 form factor for PCBs and including the maximum possible height of 50 millimeter. The mass of the controller including the mounting structure shall not weigh more than 500 grams.

4.7.3 Environmental Requirements

Each power conduction device will be heated in proportion to its resistance and the power level that is conducted. Since only a small space is available for the ECU, the heating and heat conduction of each component must be considered within a thermal design. Thereby, the proximity of the ECU to the electrolyzer, which is nominally heated to 70°C during its operation, must be considered.

Furthermore, the control unit must be capable of withstanding the environments during storage, transport and launch as defined in chapter 4.3.

4.7.4 Interface Requirements

Each component that is supplied with power is attached to the ECU. Thus, the ECU specifies the DWPS's electrical architecture, which is described in chapter 5.1.4 in more detail. For the electrical connection to the hydrogen pump a redundant power supply shall be designated to prevent the loss of a complete double cell by the failure of the electric power supply of a single hydrogen pump. Furthermore, each cell's MEA shall be powered individually. This allows the operation of one cell within the double cell design even if the second's cell power supply fails.

5 Demonstrator Water Propulsion System Design

5.1 Subsystem Definition

The technical requirements, specified for the DWPS and its subsystems, are used in the following chapter to define main parameters of each subsystem. Dimensions and technologies to be used are defined herein to allow the design in a next step. A focus will be set on the definition of water and gas tanks since the designs for the thruster and electrolyzer are mostly derived from the WPS. An overview of the current ECU is given thereafter.

5.1.1 Thruster

The DWPS thruster design shall mainly be derived from the actual WPS thruster, which is described in more detail in chapter 2.1.2. Especially the double combustion in a pre-combustion and a main combustion chamber shall be implemented. The same catalyst shall be used, heated by a heater to the same temperature as in the WPS thruster. The initial thruster design for the DWPS has been developed by the author of thesis [10] which is shown in Figure 25. This design was developed for the implementation of the thruster into a 3U CubeSat so that the actual combustion chamber could be placed within the "tuna can" envelope.

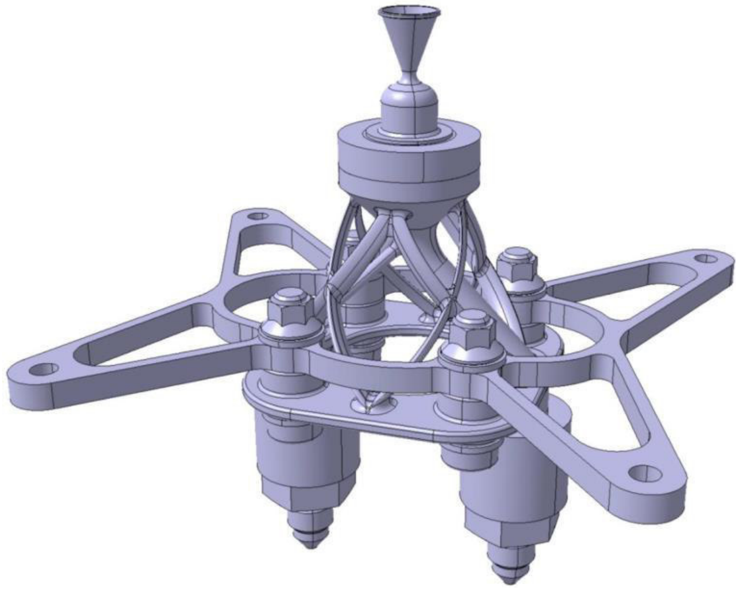

Figure 25: Initial DWPS thruster design developed for the implementation into a 3U rather than a 6U structure [10]

The main difference is given by the targeted thrust of the DWPS. Compared to the nominal thrust of 2N of the actual thruster, the DWPS thruster shall be reduced by at least 1 N. Considering the orifice diameter from [10] in Table 21, an inlet pressure of 50 bar and a C_D value of 0.6, the maximum mass flow of gases through the thruster can be calculated with equation 2.7, resulting in a mass flow of 0.195 $\frac{g}{s}$. The thrust, calculated by equation 2.3 with a specific impulse of 300, results in 0.58 N. A new design is required so that it can be mounted to the 6U structure. Thereafter, tests must be conducted with the thruster to determine its performance.

5.1.2 Electrolyzer and Gas Tanks

As stated in the specification, the electrolyzer shall be designed hot redundant so that two individual segments of the electrolyzer can operate independently and at the same time. Each segment shall be composed of two cells combined to a double cell and each double cell shall possess a single water inlet and a hydrogen and oxygen outlet each. The water within a double cell is fed to two Water Feed Barriers (WFB) which allow the necessary amount of water to pass through. Hydrogen and Oxygen are then produced and separated by a Membrane Electrode Assembly (MAE) in each of the cells. The hydrogen and oxygen gas of both cells are recombined and channeled towards the respective gas tanks. The schematic to the right, Figure 26 shows one of the two double cells with its single inlet for water and single outlet for each gas. This double cell design reduces the mass and required volume of the electrolyzer especially by eliminating the need of a bottom and top plate for each cell which presses the layers together and thereby seals the electrolyzer and withstands the pressure loads. The outlet ports shall lead directly into the tanks so that no additional fitting and tubes are necessary. The water outlet on the right of Figure 26 is the water outlet for the venting of the electrolyzer during fueling.

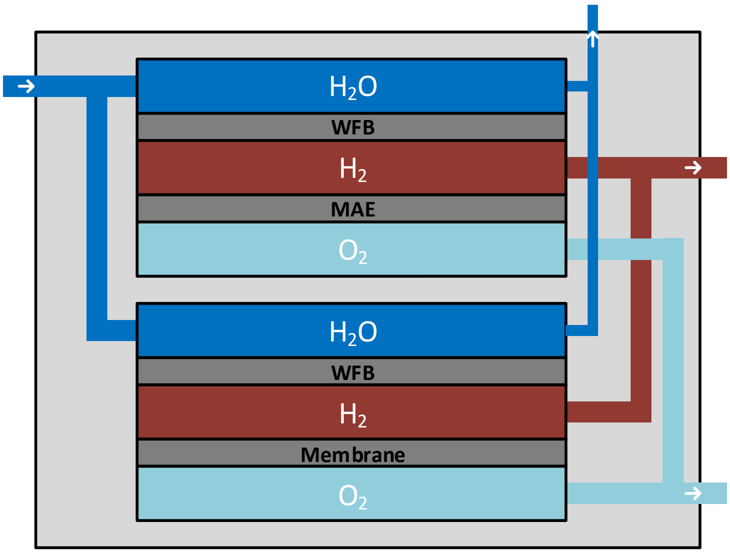

Figure 26: Electrolyzer double cell

The materials used for the CubeSat electrolyzer shall be equal to the materials used in the actual WPS. The main difference of the DWPS electrolyzer to the actual shall be the rectangular shape rather than a cylindrical geometry to use the cubic space within the electrolyzer as good as possible. A further decisive difference to the WPS' design is the design of gas tanks that are directly attached to the electrolyzer. Such a design has been developed before at the IRS and shall be reconstructed for the DWPS CubeSat mission. Figure 27 shows this design, which was adapted to the size of 1U (10x10x10 cm^3). This design resulted in gas tanks with a size of 75 ml for oxygen and 150 ml for hydrogen. The DWPS system must be adapted to a CubeSat 6U structure so its size must be adapted to the inner geometry of the structure and results in a smaller envelop than the one unit. However, a similar gas tank size is expected to be possible.

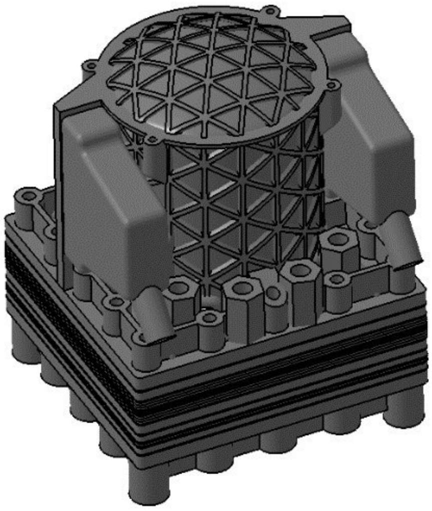

Figure 27: Electrolyzer with attached gas tanks [7]

The actual size of the gas tanks defines the maximum burn time of the thruster. The average mass flow through the orifices on the other hand is only dependent on the initial and terminating pressure of a maneuver. For the determination of the average mass flow the gas in the tank is assumed to be isothermal. The pressure change within the tank is then calculated by using equation (2.7) and the isentropic process correlations, resulting in the following equation:

$$p_{in} = p_{MEOP} \exp\left(-\frac{t}{\tau}\right) \tag{5.1}$$

Thereby, τ depicts a time independent constant derived from equation (2.7) and is compound according to equation (5.2).

$$\tau = \frac{V_{Tank}}{C_d A_o} \sqrt{\frac{\left(\frac{\kappa+1}{2}\right)^{\frac{\kappa+1}{\kappa-1}}}{\kappa\, R\, T}} \tag{5.2}$$

The pressure determined over the time can then be inserted into equation (2.7) to calculate the mass flow. For this calculation, the inlet pressure must always exceed the outlet pressure according to equation (2.8) [60]. Figure 28 shows the mass flow of hydrogen and oxygen combined for a full blow down maneuver from 50 to 10 bar with the tank sizes of 75 ml for oxygen and 150 ml for hydrogen. As it would be expected, the mass flow decreases fast for a high pressure and then slowly approaches towards a mass flow of zero. Since it is assumed that the specific impulse is around 300 s for a maneuver, the thrust can be shown by the right y-axis. The average mass flow is calculated to 97.15 $\frac{mg}{s}$ resulting in an average thrust of 285.91 mN. With a burn time of 38.9 seconds for a complete MEOP to MOP blow down, the thruster would create an impulse of around 11.1 Ns.

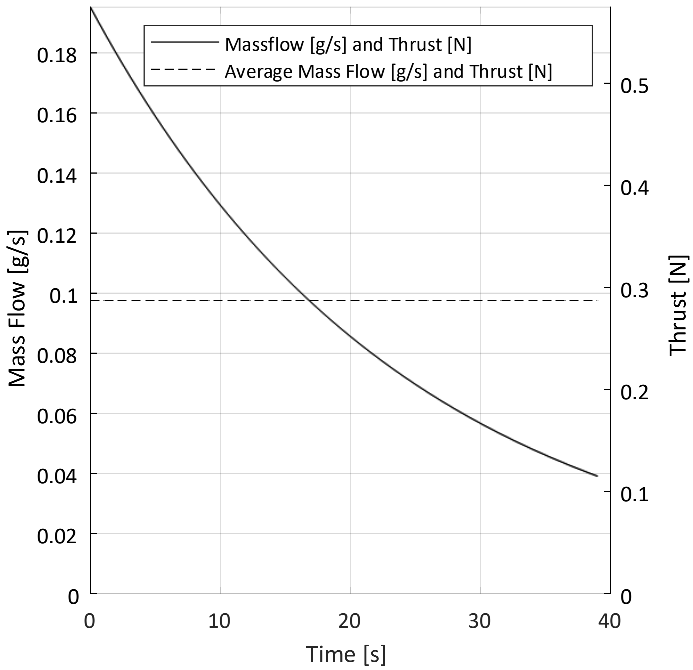

Figure 28: Mass flow of gases through the thruster for gas tanks of 75ml and 150 ml volume

These calculations can of course only be seen as rough and preliminary estimation to present achievable impulses. Tests with the orifice plates will allow the determination of the mass flow and hot firing tests will make it possible to specify the impulse of a certain maneuver more precisely.

The main limitation for a maximal firing duration is given by the CubeSat attitude control system. The reaction wheels compensate center of gravity deviations and thrust vector misalignments during a maneuver and can only prevent the satellite from rotating for a certain duration. An impulse of 11 Ns exceeds the maximum momentum storage of the reaction wheels by far.

According to the calculation above, the mass flow and average thrust for a full blow down maneuver (MEOP to MOP) is constant and independent of the gas tank size. Therefore, the duration and the specific impulse for such a maneuver is directly proportional to the gas tank size. This proportionality is highlighted by Table 27. For now, it shall be assumed that the size of 75 ml and 150 ml gas tanks fits within the envelope designated for the gas tanks and the electrolyzer.

Table 27: Max firing duration and impulse for varying gas tank sizes

Tank Sizes	Max. Maneuver Duration	Max. Maneuver Impulse	Mass of water used per maneuver
H_2 – 50 ml O_2 – 25 ml	13.0 s	3.72 Ns	1.262 g
H_2 – 100 ml O_2 – 50 ml	26.0 s	7.43 Ns	2.524 g
H_2 – 150 ml O_2 – 75ml	38.9 s	11.1 Ns	3.780 g

5.1.3 Water Tank

The water tanks must store a sufficient amount of water for the technology demonstration mission. The expulsion of water for an earth-bound fuel tank is pretty simple. The tank outlet is placed at the bottom of the tank which is where the fuel accumulates due to gravity. However, without the earth's gravitational force the water starts floating and other concepts must be designed to extract it.

The water tank for the actual WPS has not been designed nor defined yet. Thus, the initial objective was to find an available COTS component for the DWPS instead of developing a new water tank. This would drastically reduce the resources required for the water tank and allow to put a focus on the main subsystem developments of the electrolyzer and thruster. However, research has shown that deionized water compatible tanks are currently not available at the required size.

On the one hand, conventional ground-operated tanks such as accumulators that could be converted to operate in a microgravity environment are generally too large and too heavy. Accumulators prevent pressure oscillations within a pressurized fluid system and are designed to absorb pressures of 250 bar or more, requiring a thick tank wall and making them pretty heavy and hence, unusable for the CubeSat [80].

Figure 29: Valcor CubeSat propellant tank [81]

On the other hand, space COTS propellant tanks at this size are currently not available. Since CubeSats have only been equipped with propulsion systems within the last 10 years and the focus was initially put on the rather simple cold gas thrusters, not many individual components for propulsion systems are available on the market [15]. Conventionally, as seen on most chemical propulsion systems described in chapter 2.3, the design of a propellant tank is directly adapted to the propulsion system since the CubeSat limits the available volume so drastically. Rocketdyne for example developed an additively manufactured piston tank that also acts as the direct mounting structure for the four individual thruster in their CubeSat propulsion system CHAPMS [23]. The only tank that was considered suitable is a piston tank specially designed for CubeSat applications by Valcor Engineering Cooperation. It's a lightweight fiber overwrapped tank, as shown in Figure 29. With a length of 13 cm and a diameter of 4.5 cm it possesses a nominal volume of 131 milliliters. This tank, however, was initially developed for the use with hydrazine. No information is available on the compatibility with deionized water. Additionally, the reinforcing fiber structure is prone to high heat loads. Since the tank shall be placed close to the thruster, temperatures might rise above the maximum tank surface temperature of 145 °C. And at last, the tank has not yet been qualified for space applications. The possible use of this tank was no longer pursued due to these disadvantageous and also due to the reduced amount of information on the tank provided by the technical support of Valcor. Consequently it was decided, that an individual water tank shall be designed and developed to satisfy the DWPS requirements.

Within this chapter, information on possible tank concepts with their advantageous and disadvantageous were gathered and summarized. The information results in a trade-off and selection of a water tank technology which shall be used for the DWPS water tank design. Various different tank concepts are currently being used on satellites which can be assigned to two groups of tanks: Positive Expulsion Device (PED) and Propellant Managements Device (PMD) tanks.

5.1.3.1 Propellant Management Devices (PMDs)

Without the force of gravity, the surface tension becomes dominant in defining the location of the propellant within a tank. Within a PMD tank, liquid propellant and the gaseous pressurant are stored without any barrier for separation. To prevent the flow of the pressurant out of the tank, PMD components are integrated into the tank which control the location of the liquids and provide a sustaining, gas free flow of liquid propellant. The main parameter that influences the functionality of PMDs is the wetting behavior of the propellant with the PMD material's surface. This parameter can be derived from the contact angle of a propellant drop on the respective surface. A contact angle above 90° is considered to be non-wetting whereas a contact angle bellow 90° defines a wetting composition of surface and propellant. In a low gravity environment, a wetting liquid will flow towards corners and crevices, an effect that can be utilized with various PMD types. For a PMD tank to operate sufficiently, a contact angle of maximum 30° must be achieved. Conventional propellants have a contact angle of almost zero (<5°) with titanium. For the use in propellant tanks, the most basic types of PMDs are vanes, galleries, traps and sponges [70][72][71][76].

Vanes are thin metal plates that are connected to the tank wall and create a path along which the propellant can flow towards a designated location such as the outlet of the tank. The intersection between the tank and a vane creates a corner, an area where the liquid accumulates. Alternatively to vanes, galleries can be used to create such a flow path. Galleries are tube-like porous channels that are generally attached to the tank only at distinctive locations guiding the propellant towards the outlet. Traps and sponges on the other hand are PMDs placed at the outlet of the tank and are in charge of keeping the outlet covered with liquid. Conventional sponges consist of planar panels arranged rotationally symmetric around the outlet. A wetting liquid clings to the crevices created between the sponge plates and during depletion of propellant, the sponge creates a pull toward its center. Traps that fulfil the same task as sponges are used if high accelerations are expected. They conventionally consist of a solid wall container that contains the propellant even at high accelerations. A porous element of a trap allows further liquid to enter the container. Both, sponges and traps, prevent gases from entering and thereby provide a gas free flow of the liquid propellant. An example of these four PMD types is shown in Figure 30 [70][71][72][73].

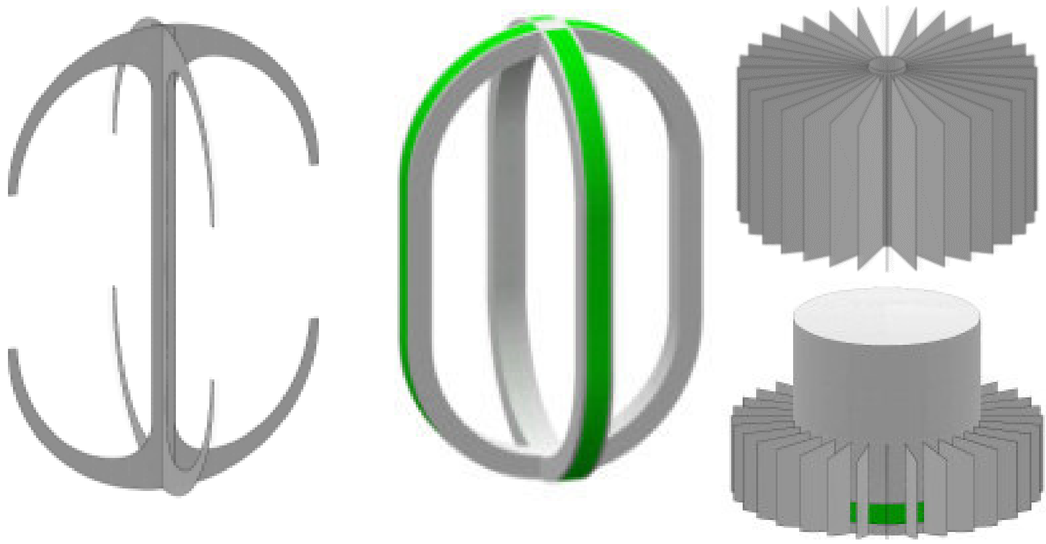

Figure 30: Overview of PMDs: vanes (left), galleries (middle) sponges (top right) and a sponge combined with a trap (left bottom)

PMD tanks are commonly used for satellite applications and are compatible with most propellants such as the conventional monopropellant hydrazine or the bi-propellants MON and MMH. These fuels possess perfect wetting behaviors with low contact angles on titanium PMDs and tank walls [74][75]. On the contrary, water, with its high surface tension is characterized by a rather low wettability in combination with titanium. Investigations by the author of [76] have shown that the contact angle of deionized water on aluminum (Al6061) and titanium (Ti6Al4V) varies strongly with angles between 50° and 90° for the aluminum and 25° to 80° for the titanium. The conventional propellants MMH, MON or hydrazine generally have a contact angle of almost zero (<5°) [74]. Further research has shown that the surface wettability can be increased by applying a silicon dioxide surface coating onto the titanium. Contact angles between 10° and 20° were achieved. However, the long term stability of the surface coating was not satisfactory since the contact angles increased by 50% within two weeks [76]. As a first conclusion, it can be drawn that the PMD technology is not yet mature enough for the use within the WPS. Further investigations on the long term stability must be performed to assure a low contact angle of the water on the tank's surface during the entire mission lifetime. Ongoing research on different surface coatings and treatments for the use as water PMD is being conducted at the Institute of Space Systems in Stuttgart (IRS) and seems to generate promising results.

An additional and decisive disadvantage of a PMD tank for the DWPS CubeSat mission is the risk of pressurizing gas entering the electrolyzer. The DWPS and the CubeSat will be exposed to various accelerations during transport, launch and in operation. During each of these processes the system would have to be handled very carefully, ideally always with the same acceleration orientation to prevent the gas from exiting the tank. If gas enters the electrolyzer it masks the water feed barrier and dries it out which would make the electrolyzer unusable.

5.1.3.2 Positive Expulsion Devices (PEDs)

In Contrast to PMDs, PEDs possess a barrier between the pressurant and the propellant. The PED tanks are based on a either flexible or movable barrier which pushes the fuel out of the tank while the pressurant expands. The use of such a barrier poses various advantageous. It prevents the outflow of the pressurizing gas. Additionally, a loss of propellant due to its evaporation and mixing with the pressurant can be prevented. And at last, the barrier allows a better control of the center of gravity since it prevents the sloshing of the liquid propellant. Conventional PED tanks possess either a diaphragm, a bladder or a piston as barrier between the two fluids. All three types have different properties that shall be summarized hereafter [77][78].

A **bladder tank** contains a balloon-like membrane which is either filled with the propellant or the pressurizing gas. In the former case the bladder is surrounded by the pressurant gas and contracts whereas in the latter case the bladder expands during the propellant expulsion. When the propellant is stored fully within the bladder, the tank material will only be in contact with the inert pressurizing gas. The tank material can then be chosen without considering compatibility issues with the propellant. For both configurations, the bladder must be adapted to the tank's shape. Bladders that are smaller than the tankage may be exposed to intensive stretching while on the contrary, larger bladders may be damaged by creases. Since the elasticity of materials is not sufficient to allow bladders to be collapsed fully without a remaining internal volume, the bladder will start folding at some point. Therefore, the durability and efficiency of a bladder tank is strongly affected by the bladders folding pattern. A further important factor during the design of bladder tanks is the compatibility of the bladder material with the propellant. Studies on the compatibility of typical propellants and oxidizers with possible bladder materials are summarized in the ECSS standard ECSS-E-ST-35-10C but unfortunately don't include any information on deionized water. The bladder for the DWPS tank must possess a low permeability and high resistance to hydrolysis with water. Such a water tank has been developed by MT Aerospace with a bladder made of Fluorethylenpropylen (FEP), however, the size of the tank exceeds the size of the entire 6U CubeSat [78].

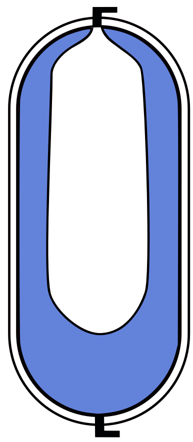

Figure 31: Concept of a bladder tank

The second type of PED tanks are the **diaphragm tanks**. Instead of a balloon-like membrane, the diaphragm separates the liquid propellant from the gaseous pressurant into two compartments by a flexible membrane. Therefore, the tank material is generally in contact with both fluids on each side of the diaphragm. The principle is easy: A diaphragm reverses itself by the expansion of the gas until it adapts to the tank's internal shape. Thus, the diaphragm does not fold as intensely as a bladder, simplifying the design of the membrane. On the contrary, however, the diaphragm is conventionally attached to the tank at its widest inner diameter resulting in a complex assembly. Aside from the reduced tension on the diaphragm due to reduced intensity of creases, the diaphragm material must also be compatible to water just as the bladder material.

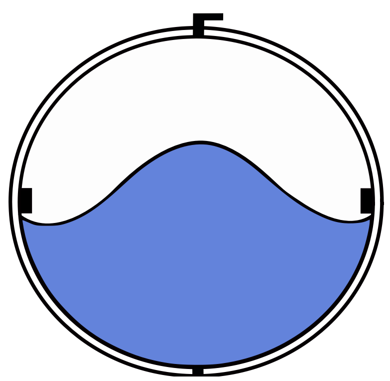

Figure 32: Concept of a diaphragm tank

The **piston tank,** different to the other two PED tanks, is based on a solid piston as a barrier that slides along the entire length of the tank. The pressurizing gas can thereby be sealed from the propellant by two different methods, either by sealing rings that slide along with the piston or by bellows that extent or contract with the movement of the piston. A further decisive difference of this type of PED is the shape of the tank. While diaphragm and bladder tanks are generally adapted to the shape of the expanded membrane or bladder, the piston can be shaped freely to fit a cylindrical tank with flat, concave or convex ends. For the cubic shape of a CubeSat this makes it possible to use more of the available volume compared to spherical or elliptical shaped tanks. To prevent cocking of the piston during the expulsion of the propellant the piston must either be assisted by some guiding mechanism or possess a certain length. Cylindrical pistons with a length of 5/8 of their diameter are considered to be stable without any guidance which is why the design of rather elongated cylindrical tanks is beneficial [77].

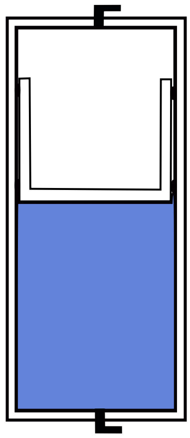

Figure 33: Concept of a piston tank

5.1.3.3 Trade-off

At first, it shall be noted, that the PMD concept will not be reviewed within the following trade study. Especially the risk of gas entering the electrolyzer and thereby damaging the system as well as the reduced maturity of this technology in combination with water as a propellant, makes this technology undesirable for the DWPS water tank. Thus, the trade-off focuses on the different PED types and highlights their advantageous and disadvantageous.

For the trade-off each tank type is assessed by different criteria which are weighed depending on their importance. Each tank's performance or its features are graded for each of these criteria. The following criteria have been identified and are weighed according to the percentage stated for each criteria.

- Complexity (25%):
  A high complexity shall be assigned to a system with a high number of components, which must be assembled in various steps. It shall also include the complexity of the barrier component and of the mechanism it uses during the expansion of the pressurizing gas. A high value in the grading scheme indicates a low complexity.
- Volume Efficiency (35%):
  The space available for the water reservoir is limited to two rectangular envelopes left and right of the thruster. A high volume efficiency shall be assigned to a technology capable of using this space most efficiently.
- Susceptibility to contamination (5%):
  A low reliability shall be indicated by the susceptibility to contamination.
- Reusability (5%):
  A high reusability shall be assigned to a system that can be used for multiple in- and outflow cycles without being damaged.
- Compression Ratio (20%):
  The compression ratio is the ratio of the gas chamber at the beginning of the expulsion compared to the final gas chamber size. This ratio thereby defines possible start and end of life pressures for the water tank. A high value shall be assigned to a system that can compress the gas to the lowest possible size.
- Weight (10%):
  The weight can be derived from the comparison of weight of ground-operated accumulators..

To be able to assess and compare the different types of PED tanks, the comparison of different types of hydraulic accumulators was reviewed and applicable parameters were derived for satellite propellant tanks [79]. The criteria of each PED type was graded on a scale from 1 to 5. The grade 5 indicates that a PED performs better or possesses better properties for the given criteria than a PED graded with a lower value.

Table 28: Water tank technology trade study

PED	Bladder	Diaphrag m	Piston	Weighting
Complexity	3	4	2	25%
Volume Efficiency	2	1	4	35%
Susceptibility to Contamination	5	5	1	5%
Reusability	3	3	3	5%
Compression Ratio	1	3	5	20%
Weight	4	5	4	10%
Overall Rating	**2,27**	**2,31**	**2,6**	100%

It can be seen, that according to the stated criteria and the respective weighting, the piston tank seems to be the most suitable candidate. Especially, the favorable volume efficiency that can be achieved by the cylindrical shape as well as a high compression ratio of this technology reflect the decisive advantageous of this technology. But it can also be seen that due to the sliding piston the assembly and design is very complex. Cocking must be prevented in any case. Especially if a guiding mechanism is required the complexity increases a lot. The sliding piston design is also prone to contaminations. Any particles that adhere to the tanks wall can damage the piston seal and might cause its failure.

The diaphragm tank scores in the category of complexity and weight of the design. The simple mechanism of the diaphragm reversing itself to allow the gas expansion simplifies the design a lot. Since the diaphragm adapts to a spherical like shape, the geometry of the tank must be similar. This can be seen as advantage and disadvantage. A spherical tank can withstand pressure loads better than any other shape, allowing thin tank walls and a light design. On the contrary, however, this design provides a very inefficient shape for the use in the rectangular area within the CubeSat.

The bladder tank is evaluated to be the most unsuitable tank for the CubeSat application of these three PED type tanks. A main disadvantage of this design is the compression ratio which is mainly influenced by the functional mechanism of this tank. The bladder, either filled with the pressurant or the propellant, expands or contracts during the expulsion. For a high compression rate, which defines the ratio of the initial gas volume to the final gas volume within the tank, the bladders internal volume must change a lot throughout the expulsion. In accumulators, for example, the bladders are conventionally filled with the expandable gas. During the filling of the accumulator, the gas volume should general not contract lower than ¼ of its initial size, since creases occur and might damage the bladder. For larger satellite tanks, the bladder is often designed to fold itself according to a predefined pattern [77]. This might make it possible to achieve higher compression ratios but would definitely increase the complexity of this design which is weighed more in the given trade study.

A main parameter, the reusability of such PEDs is very hard to predict. On one side, the bladder and diaphragm materials are prone to temperature changes. Temperatures above or below the tolerances can cause brittleness and lead to ruptures. On the other hand, the piston seal that slides along with the piston might be damaged over time.

The compatibility of these three tank technologies with water and the pressurant has also been reviewed. The piston within the piston tank can be sealed by piston sealant rings which are complemented by wear rings. The sealant ring seals the barrier between the two fluids, the wear rings are used as guiding rings to prevent the metal on metal contact. Conventional sealant companies such as Trelleborg produce such seals that are compatible with water. The bladder and diaphragm tank technologies are also compatible with water. The company Holscot Fluoropolymers Ltd. for example, is capable of producing fluorinated ethylene propylene bladders and diaphragms that have been used with water for space applications before. An example is the bladder developed for an MT Aerospace water tank that was carried on the European ATV to the ISS [82][83].

All in all, the trade study shows that PED tanks have advantageous and disadvantageous for the utilization in the CubeSat DWPS that counterbalance so that the overall rating results are very similar. However, from all three types, the piston tank seems to be the most compatible. This also complies with tanks developed and used for other CubeSats, as shown by the CubeSat tank by Valcor or the tanks used for the CubeSat propulsion systems described in chapter 2.3.

A general advantage of the piston tank is the capability of using measuring devices that provide information on the position of the piston within the water tank to allow a more precise determination of the water tank filling level. Such measuring devices can either be implemented within the tank or use a technique to measure it exteriorly. There are many different types of sensors for the determination of distances and position all using measurements based on different physical and electrical properties. Eddy current, capacitive, laser, confocal, inductive (LVDT), magneto-inductive, ultrasonic and draw wire sensors have been reviewed for this purpose. However, since the water tank's size is limited intensely by the available space within the CubeSat, it was decided, that a further measuring device shall not be used. All measuring devices that can be implemented within the water tanks, take up a lot of volume, reducing the amount of water that it can carry. External measuring devices such as magneto-inductive sensors also take up a lot of space so that the overall tank size would have to be reduced. The use of sensors that can detect if the piston has reached a certain position, has also been reviewed. An external attachable sensor is the magnetic field sensor, which measures the magnetic field of the cylindrical tank. A magnetic piston travelling inside the tank creates a remanence magnetic field within the tank walls which changes its polarity as soon as the piston travels by. However, for this, the cylindrical tank must be made of a ferromagnetic material and would rule out the use of aluminum and titanium. Thus, for the use in the DWPS system, the water tanks shall be equipped only with pressure transducer and temperature sensors for the determination of the filling level, even if the measurement becomes less accurate at decreasing pressures. Since the amount of water that can be carried aboard the CubeSat is already limited, it is far more important to increase the amount of water rather than to be capable of measuring the filling level of the tank [84][85].

5.1.4 Electrolyzer Control Unit

Each subsystem possesses components that are actuated by the ECU. An overview of all powered components of the DWPS is given in the power breakdown of Figure 34.

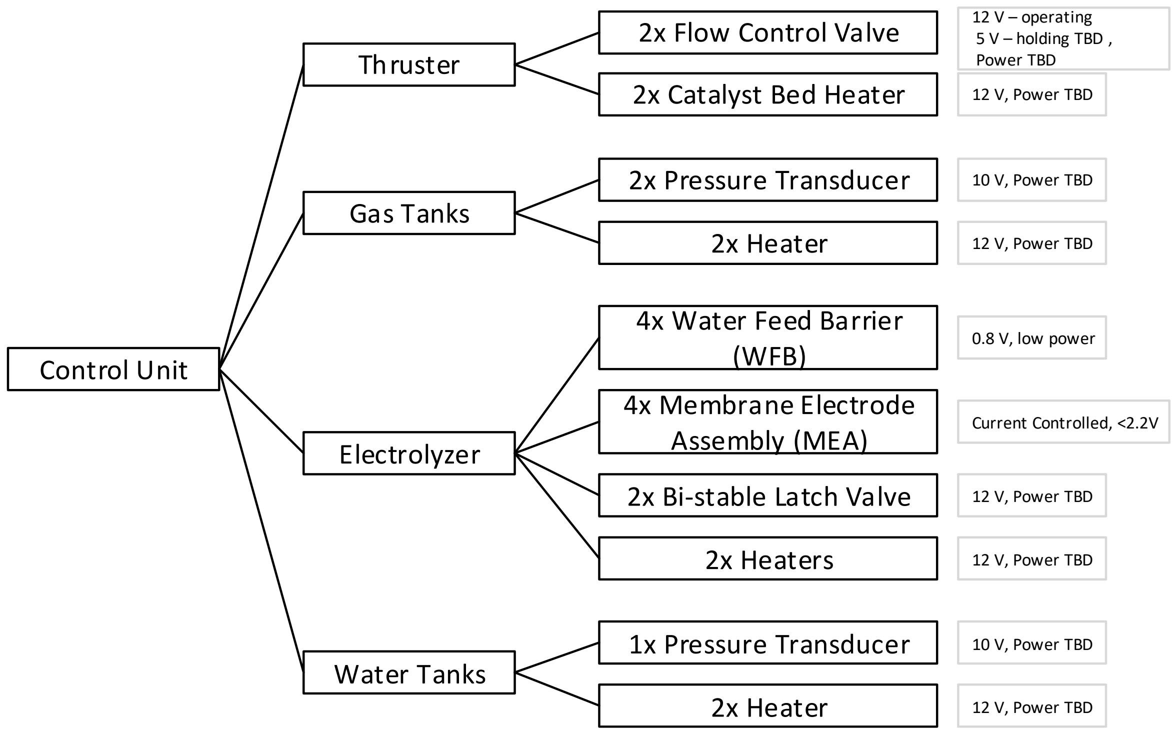

Figure 34: DWPS power distribution breakdown

This figure contains the absolute necessary power supply interfaces. Additional to the 6 heaters defined in the figure, it might be necessary to equip further components, such as long tubes or valves with more heaters. Furthermore, the hydrogen pump shall be provided with a redundant power supply. This way, if the power supply of one hydrogen pump fails, the failure of the entire double cell can be prevented. A connection of the power supply of the MAE and hydrogen pump in series, as it is shown in Figure 35, was regarded initially. It can reduce the amount of DC/DC converters on the ECU and thus simplify its design. However, with a single failure within one of the circuits, the entire double cell would fail. Furthermore, the characterization of the performance derived from the current and voltage within each cell is less precise and much more challenging. Thus, each MEA and hydrogen pump within a cell shall be powered individually and not in series as it is shown in the following scheme.

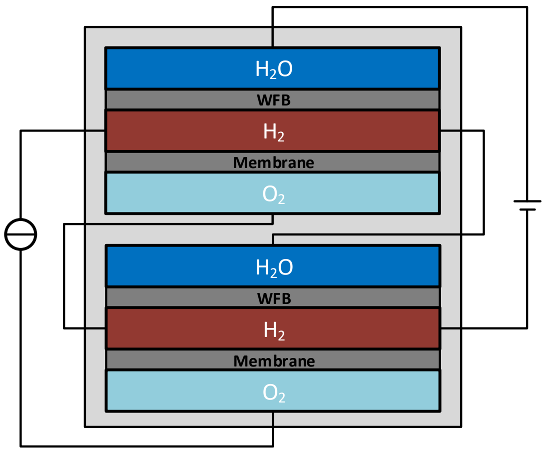

Figure 35: Series connection of an electrolyzer double cell

Additional to the power lines, the ECU is connected to various components for the acquisition of data, which is shown in the scheme of Figure 36. Similar to the power supply, it shall be possible to add further equipment. Especially the amount of type K thermocouples shall be increasable. If further heaters are to be applied they must be controlled by these additional heaters.

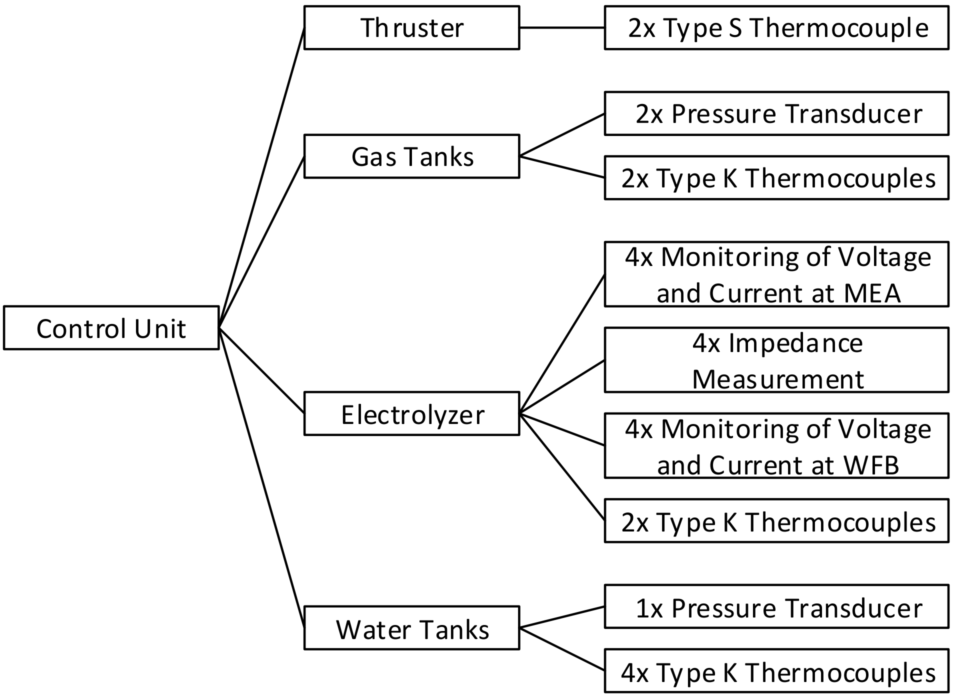

Figure 36: DWPS data acquisition breakdown

The power supply for the ECU by the CubeSat's EPS is yet to be defined. In general, there are two possibilities. To allow a full flexibility, the ECU could be supplied only by the unregulated lines. The ECU would then have to regulate the power by DC/DC converters to the respective voltage of each component. The ECU could then be designed to allow all types of voltage and current supplies for the components. On the other hand, to simplify the ECU design, the ECU could be supplied mainly by regulated power lines. As a result the ECU would just have to be equipped with MOSFET switches to activate or deactivate the power supply. This for example can be applied for all components that operate at a single voltage, such as the heaters, the pressure transducers or the latch valves. The power supply for components such as the electrolyzer or the FCVs which require the regulation of the voltage during the operation could be provided from the unregulated power lines that are regulated by the ECU. Since the design of the ECU of the DWPS shall be equal or at least similar to the WPS' ECU, it shall be adapted and designed accordingly.

The following table defines the required input voltages for the case where the ECU design is simplified and regulates only the power for the FCVs and the electrolyzer components.

Table 29: Control Unit - CubeSat power interface

ECU Voltage Input	
Voltage	**Component**
2x 12 V / 5V or unregulated	FCV
2x 12 VDC	Catalyst Bed Heater
3x 5 VDC	Pressure Transducer
2x 12 VDC	Bi-Stable Latch Valve
6x 12 VDC	Heater
4x 3.3 VDC / or unregulated	WFB
4x unregulated	MAE

These input parameter only include the required power lines for the redistribution of power to the DWPS. The components of the control unit itself also require power, such as the microprocessor, which in return powers the MOSFET switches and signal amplifiers. These power lines must be included once the CubeSat DWPS control unit has been designed.

5.2 DWPS System Design

As shown in previous chapters, the DWPS is composed of the 5 major subsystems: the electrolyzer, the water tanks assembly, the gas tanks, the thruster assembly and the control unit. Assembled to a system they form the DWPS that will be aboard the 6U CubeSat. This chapter describes the system architecture and discusses design decisions.

5.2.1 Assembly and Arrangement

The 6U CubeSat platform provides a volume of 4U for the DWPS. The estimation of required space for the individual components has led to the layout of the respective subsystems in the CubeSat as shown in Figure 37.

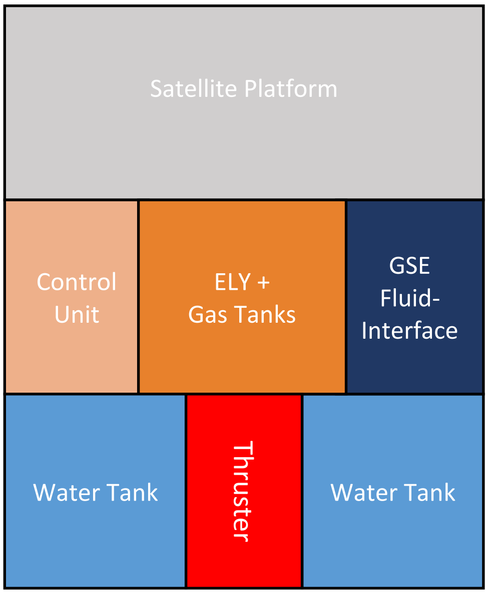

Figure 37: 6U CubeSat layout

The GSE Fluid interfaces thereby includes a filling port for the water tanks, two drainage ports for the release of excess air trapped in the water lines and two vent ports to release remaining hydrogen and oxygen from the gas sides without guiding them through the thruster during ground operations. Furthermore, this space shall be used for the pressure transducers for the gas tanks and the latch valves.

The thrust vector of the thruster must aim through the center of mass of the CubeSat. Since it is assumed that the electrolyzer with the gas tanks is the heaviest component of the entire DWPS it is placed in the center of the CubeSat.

To assure that the center of mass stays within the thrust vector axis (z-axis), throughout the entire mission, the use of propellant cannot shift the center of mass in x- or y-direction. Therefore, it was decided to use two water tanks, placed on each side of the thruster that shall be emptied at the same pace. This will make the CubeSat's center of gravity move along the z-axis but will not shift it in x or y direction. Furthermore, it is assumed, that short high heat loads, radiated by the thruster during firing can be absorbed better by the water tanks than by any other component.

An even more detailed overview of the DWPS is given in Figure 38. It separates the propulsion system in 5 elements that are reviewed within the definition of the system. The first and most obvious element is the propulsion system itself, consisting of the major subsystems (water and gas tanks, electrolyzer and thruster) and further components such as tubes, valves, sensors etc. The second element is the CubeSat platform which has a main influence on the propulsion system design. Additional elements of the DWPS, are its thermal equipment, structural attachments to the structure and the electrical harness. The latter three elements, especially the structural mountings and the thermal equipment have to be investigated in further studies once the development of a detailed design of the propulsion system is implemented. A more detailed definition and design of these elements cannot be performed at the current state of the development.

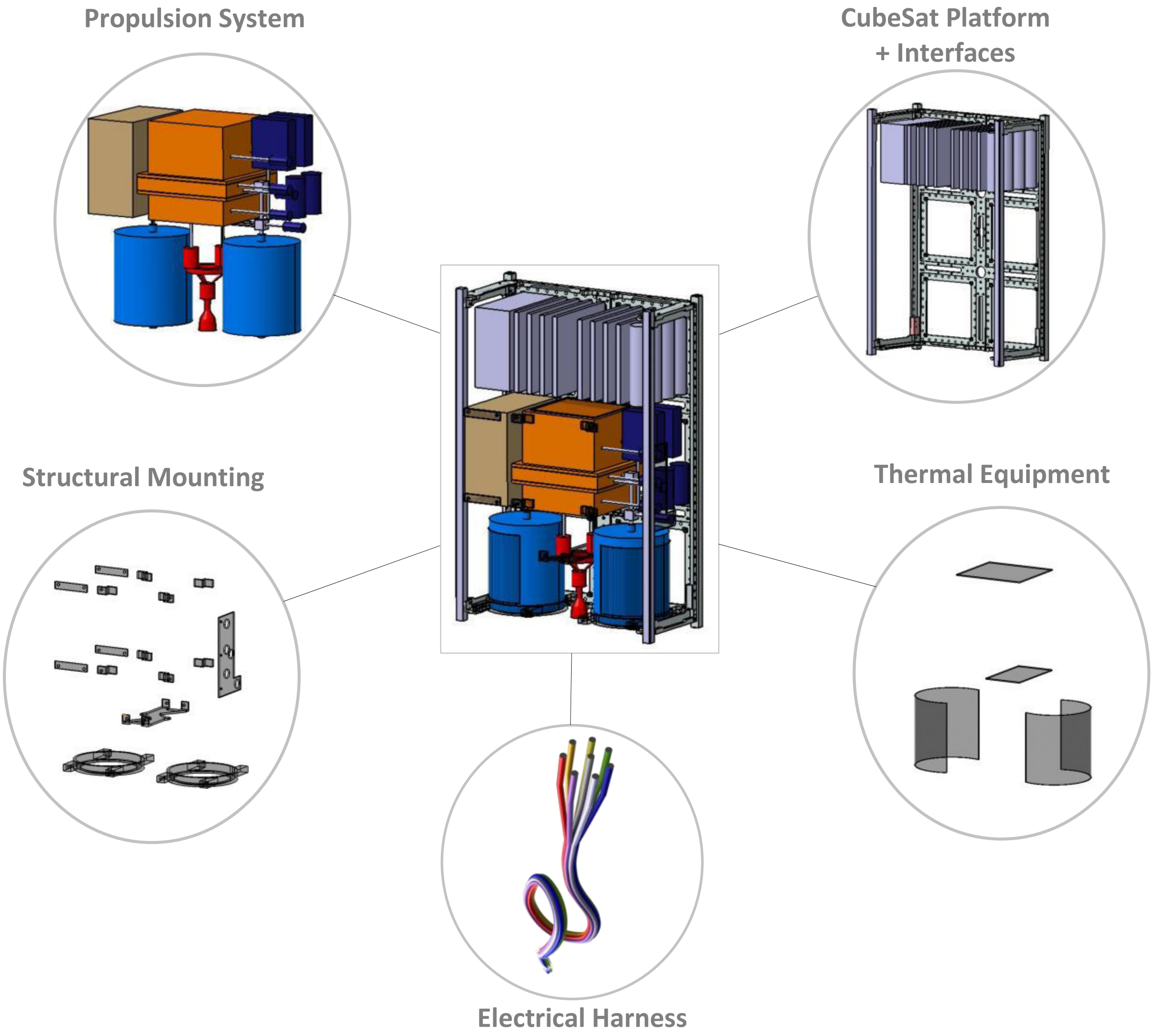

Figure 38: Composition of the Demonstrator Water Propulsion System

An initial rough estimation of the mass of each element is shown in Table 30. This mass estimation results in an overall mass of 5 kg. Since the maximum mass of a 6U CubeSat is specified to 12 kg and the platform weighs around 6 kg a margin of 20% is available.

Component	Mass
Control Unit	500 g
Electrolyzer + Gas Tanks	3000 g
GSE Fluid Interfaces + Tubing	500 g
Water Tank Assemly	700 g
Thruster Assembly	300 g

Table 30: Mass breakdown of the DWPS

5.2.2 Fluid System

The fluid system is composed of the main components, the thruster, the electrolyzer, the gas and water tanks, and their connections. Figure 39 shows the flow schematic of the DWPS.

Since a main emphasis is set on the in-orbit investigation of the electrolyzer's performance, a redundant architecture was developed to ensure the operation of the system even if one electrolyzer double cell fails. The event of a rupture in the WFB is assessed to be the most critical failure. The WFB is the barrier within the electrolyzer between the gas and water side. The pressure on both sides keeps varying throughout the mission exposing the barrier on one hand to a pressure gradient from 12 bar on the water side and vacuum on the gas side and on the other hand to a pressure of 2 bar on the water side and 50 bar on the gas side. In the case of a rupture, it shall be possible to isolate the electrolyzer double cell from the fluid system and to operate the second without any perturbations. Thus, the electrolyzer is equipped with two latch and six check valves. Once the WFB breaks, these valves on one side prevent the flow of water through the electrolyzer and into the gas tanks and on the other side prevent the flow of gas from the gas tanks or the other electrolyzer double cell back through the water feed barrier and into the water feed line. For the case, where the pressure is greater on the gas side rather than on the water side, the check valves prevent the backflow of any gas. The check valves (NRV1 & NRV2) are required to prevent the backflow of gases that are trapped within the electrolyzers. Even though this might just be a small amount of gas, it is still capable of creating a flammable mixture in the water tank or can mask the WFBs of the intact double cell and must be prevented. The back flow of gas from the gas tanks or the second, intact double cell is prevented by additional four check valves (NRV 3-6) on the gas side. These check valves don't require any external actuation. Once a reverse pressure differential occurs, these valves seal and prevent the back flow. To prevent the flow of water through the electrolyzer in the case, where the water side pressure is higher than the gas side pressure, an additional latching valve is required. These latch (LV1 & LV2) valves are actuated as soon as the failure is detected by the control unit and prevent a further flow of water. Since the detection of the rupture is slightly delayed and a delayed actuation cannot sufficiently prevent the back flow of gas, the check valves (NRV1 & NRV2) are still required. Furthermore, the latch valve cannot prevent the flow of water that is present between the valve and the WFB. However, it is assumed that the gas side and the thruster will not be damaged by a small amount of water. This way, the latch valves can be placed at the inlet rather than at the outlets which allows to reduce the amount of valves.

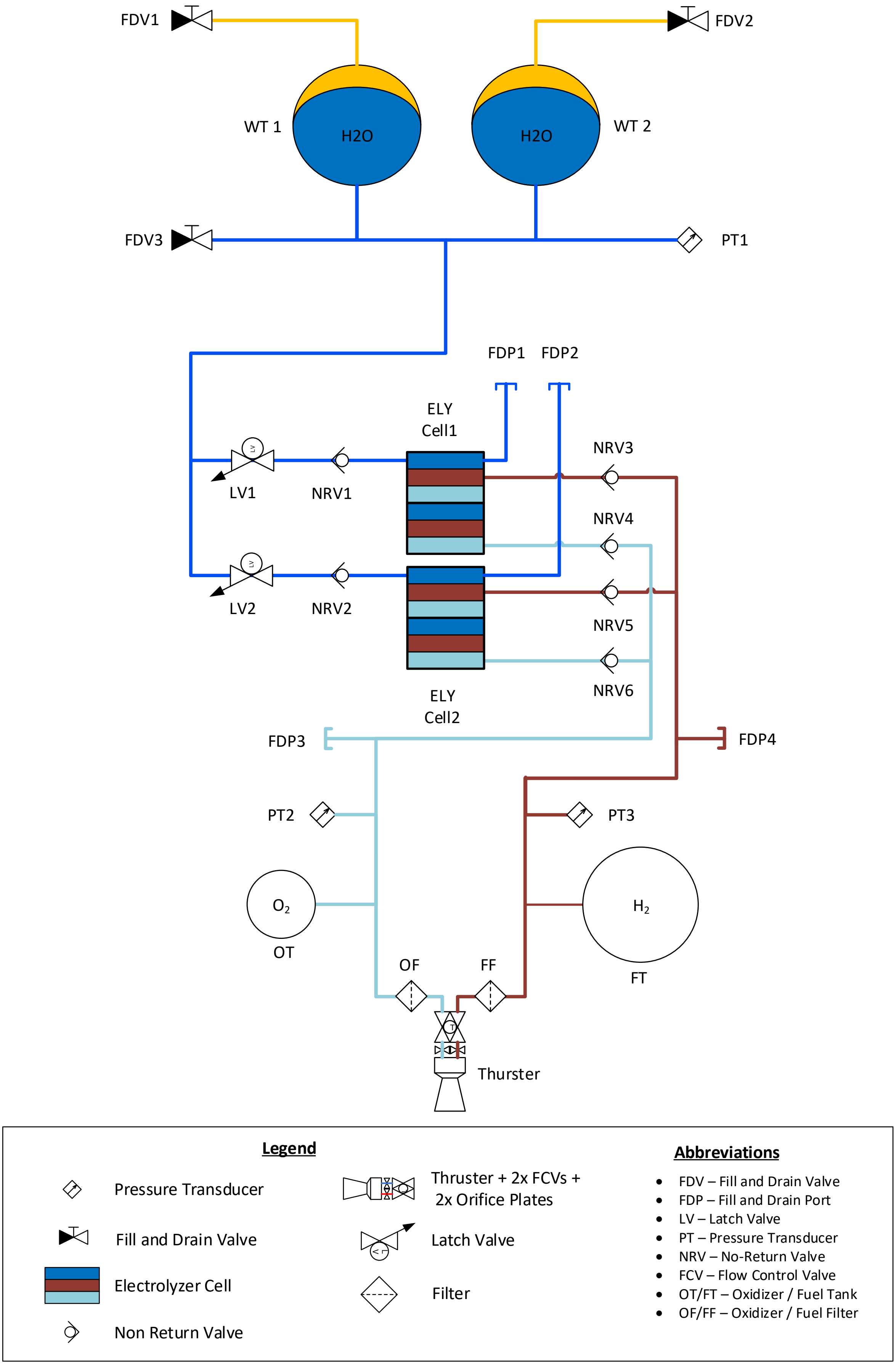

Figure 39: DWPS Flow schematic

Aside from the arrangement shown in the flow schematic, it was also reviewed if each water tank should be connected to only a single electrolyzer double cell. This would make it possible to directly compare the amount of water used on each double cell. The performance of both segments could be compared to each other since it directly correlates to the amount of used water. However, this configuration would require an additional pressure transduce and two extra FDVs for the pressurant and water. Furthermore, if one of the double cells fails early during the mission, only half of the water could be used and would thereby shift the center of gravity. Hence, it was decided to connect the water tanks before separately supplying the electrolyzer with water.

5.2.3 Gas Flow Analysis

Before defining a mission profile and evaluating the CubeSat's operation in space, the behaviour of the oxygen and hydrogen gas shall be examined. The gases produced by the electrolyzer contain a certain amount of vaporized water. Thus, a change in pressure and temperature of the gas can result in the condensation and accumulation of liquid water especially at cold areas of the DWPS.

If the temperature of the gases sinks, the humidity rises until the gases are saturated and liquid water starts to form. This is why the gas tanks, the electrolyzer and the tubes shall be equipped with heaters, so that no liquid water can accumulate at cold areas of the DWPS. However, a further phenomenon, the Joule-Thomson Effect must also be considered during the firing of the gases. The Joule-Thomson Effect occurs during the throttling of real gases at the orifice plates of the propulsion system and results in an increase or decrease of the temperature of a gas during its expansion.

To model the gas behaviour during the throttling the initial and final properties of the gas mixtures must be described. Therefore, the mixture of gaseous water with either hydrogen and oxygen shall be approximated as a mixture of ideal gas, as it is often done to predict the state of water in air [86]. As stated by the Dalton's Law, each gas within a mixture of ideal gases behaves as if it was present independent of the other gases. This allows the statement of the ideal gas law for each gas i individually.

$$p_i \mathrm{V} = n_i \Re \mathrm{T} \tag{5.3}$$

The overall pressure of the gas mixture results from adding up the partial pressures p_i of each gas. The same applies for the overall amount of substance n.

$$\sum_{k=1}^{N} p_k = p\,;\ \sum_{k=1}^{N} n_k = n \tag{5.4}$$

To define the partial pressure of the water vapor within the oxygen and hydrogen tanks, the relative humidity shall be used as the input parameter. It allows the determination of the concentration of the water within the gas mixture and also indicates its state. It is calculated with the water vapour partial pressure p_{H2O} and the water vapor pressure p_s as shown in equation (5.5) [86].

$$\varphi = \frac{p_{H2O}}{p_s} \tag{5.5}$$

The water vapor pressure is the pressure at which the water vapor starts forming liquid water. It is a parameter strongly dependant on the temperature and can be described by the Antoine equation in bar:

$$p_s = 10^5 \cdot 10^{A_s - \frac{B_s}{T + C_s}} \tag{5.6}$$

The empirical values A, B and C variate for different gases and can be found in the NIST database. For water at a temperature between 255.9 K and 373 K the values of A, B and C are given in the following [87].

$$A_s = 4.6543$$
$$B_s = 1435.264$$
$$C_s = -64.848$$

Figure 40 shows a plot of equation (5.6), of the water vapor pressure in dependency of the temperature between 273 and 373 K. Once the partial pressure of water reaches the vapor pressure at a given temperature, the water starts condensing which in return keeps the partial pressure at the vapor pressure even if the gas mixture pressure rises. The partial pressure of water within a gas mixture can therefore not exceed the grey area marked in Figure 40.

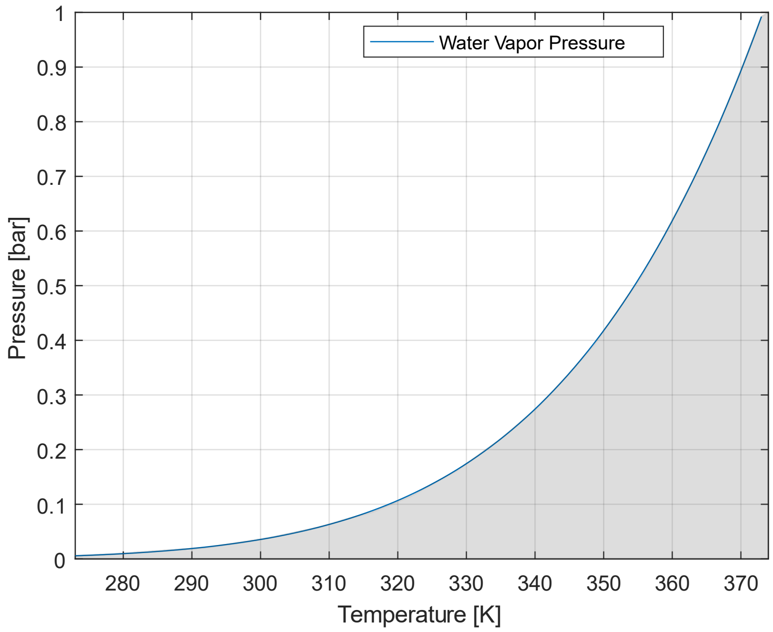

Figure 40: Water vapor pressure between the temperature of 273 an 373 K

During the throttling process, the temperature of a gas changes depending on the initial gas temperature and pressure (state 1) as well as the pressure that occurs after the throttling (state 2). It is the objective of the following calculations, to determine the temperature after the throttling process. On one hand the temperature can be used to calculate the water vapor pressure but also to define the partial pressure of the water after the throttling.

The partial pressure of the water after the throttling process can be derived from the ratio water partial pressure to the oxygen or hydrogen partial pressure by using the Dalton's Law (5.3).

$$\frac{p_{H2}}{p_{H2O}} = \frac{n_{H2}}{n_{H2O}} \tag{5.7}$$

If the gas is mixed ideally, the ratio of amount of substance within the gas mixture does not change across the throttling process from state 1 to state 2. This shows that the ration of partial pressure for the gaseous water and hydrogen or oxygen of state 1 is also equal to the ratio at state 2. Of course, this is only applicable if no liquid water is formed in state 2. The relative humidity can then be calculated at state two according to equation (5.8) by using the relation of (5.4).

$$\varphi_2 = \frac{p_2}{p_s(T_2)\left(1 + \frac{p_{H2}}{p_{H2O}}\right)} \tag{5.8}$$

The pressure at state 2 is mainly dependent on the processes occurring within the combustion and pre-combustion chamber. For the following calculation it shall be assumed that p_2 is known by approximating it to a constant value. To evaluate the relative humidity at state two, the temperature

T_2 must be determined in the next step by using the first law of thermodynamics which is shown in the following for a stationary flow process.

$$0 = h_1 + \frac{c_1^2}{2} - h_2 - \frac{c_2^2}{2} + q + w_t \tag{5.9}$$

It shall be assumed that the gases pass through the orifice without a significant transfer of heat to the walls and vice versa which allows the assumption of an adiabatic process (q = 0). Furthermore, no work is extracted from the system (w_t = 0) and it shall be assumed that the change in velocity can be neglected since the kinetic energy is low compared to the enthalpy. With these assumptions the throttling is defined as an isenthalpic process and equation (5.9) results in the following equation.

$$h_1 = h_2 \tag{5.10}$$

For an ideal gas, the enthalpy is a function only of the temperature. Since the throttling process is reviewed as isenthalpic process, an ideal gas does not change its temperature. For real gases, on the other hand, the enthalpy is a function of temperature and pressure. A throttling from state 1 with a pressure p_1 to a state 2 with a lower pressure p_2 requires a change in temperature in most cases to fulfil equation (5.10). The difference between a real gas and an ideal gas are the intermolecular forces that occur between the gas molecules in a real gas which are neglected for ideal gases. These molecular forces are either repulsive or attractive depending on the type of gas and its state. Hydrogen molecules for example repel each other at atmospheric pressure and 300 K. By expanding the gas at these conditions, the thermal potential energy is reduced which creates an increase in the thermal kinetic energy of the molecules which indicates an increase in temperature. On the other hand, hydrogen molecules at atmospheric pressure and a temperature below 200 K attract each other. The expansion forces the molecules apart which reduces the thermal kinetic energy and cools the gas. These intermolecular forces have to be included in the gas law which is achieved by various approaches, such as the Van-der-Waals equation or the Virial equation which is used in the following.

To calculate the change in temperature, the gas mixture shall in this step be assumed as a pure gas of either oxygen or hydrogen. The amount of water within the gases even at a high relative humidity is comparatively low, so that it shall be assumed that the gaseous water does not influence the throttling process significantly.

The specific enthalpy of a real gas can be derived from its total differential.

$$dh = \left(\frac{\partial h}{\partial T}\right)_p dT + \left(\frac{\partial h}{\partial p}\right)_T dp \tag{5.11}$$

Based on equation (5.11) the enthalpy can be defined in dependence of T and p according to the mathematical conversion shown in [88], resulting in equation (5.12)

$$h(T,p) = h_0 + \int_{T_0}^{T} c_p^0\,(T)dT + \int_0^p \left[v - T\left(\frac{\partial v}{\partial T}\right)_p\right] dp \tag{5.12}$$

Thereby, the specific heat capacity c_p^0 is the specific heat capacity of an ideal gas which can be approximated by a polynomial equation and integrated to equation (5.13) as given in the NIST database in [87].

$$\int_{T_0}^{T} c_p^0\,(T)dT = \frac{1}{M}\left(AT + B\frac{T^2}{2} + C\frac{T^3}{3} + D\frac{T^4}{4} - \frac{E}{T} + F\right) \tag{5.13}$$

And the empirical coefficients for oxygen applicable for temperatures between 100 and 700 K and hydrogen applicable for temperature between 298 and 1000K as given in the following table [88].

Table 31: Coefficients for the approximation of the specific enthalpy of a real gas [88]

Coefficient	O_2	H_2
A [J/(K·mol)]	31.32234	33.066178
B [J/ (K²·mol)]	-20.23531 · 10^{-3}	-11.363417 · 10^{-3}
C [J/(K³·mol)]	57.86644 · 10^{-6}	11.432816 · 10^{-6}
D [J/(K⁴·mol)]	-36.50624 · 10^{-9}	-2.772874 · 10^{-9}
E [J· K/mol]	-0.007374 · 10^{6}	-0.158558 · 10^{6}
F [J/mol]	-8.903471 · 10^{3}	-9.980797 · 10^{3}

The first part of equation (5.12) consists of the enthalpy of an ideal gas and can be approximated as shown in equation (5.13) with the listed coefficients. The real gas effects are implemented by the last term of equation (5.12) which is a little more complicated to determine. The first step therefore is the determination of the real gas factor Z which is derived from the Virial equation of (5.14).

$$pv = ZRT \tag{5.14}$$

The Virial equation approximates the real gas behavior of a gas by a serial expansion in the form of the real gas factor Z [88].

$$\frac{pv}{RT} = Z = 1 + \frac{B_V}{RT}p + \frac{\left(C_V - {B_V}^2\right)}{(RT)^2}p \tag{5.15}$$

The coefficients B_V and C_V are stated as the second and third virial coefficients and vary for different gas species and are functions of the temperature. Similar to the heat capacity, the virial coefficients can be approximated by polynomial equations according to (5.16a) and (5.16b) [88].

$$B_V = a_1T^2 + b_1T + c_1 \tag{5.16a}$$

$$C_V = a_2T^2 + b_2T + c_2 \tag{5.16b}$$

For hydrogen the values for a_1, b_1, c_1, a_2, b_2, c_2 are taken directly from [88], are reproduced in Table 32 and are applicable for temperatures between 288.15 K and 373.15 K. For the coefficients of oxygen, the values were calculated using the values of the Virial coefficients experimentally determined in [89].

Table 32: coefficients for the calculation of the Virial coefficients of hydrogen and oxygen [88]

Coefficient	O_2	H_2
a_1 **[m³/ K²·kg]**	-3.0000 · 10^{-8}	-5.25 · 10^{-8}
b_1**[m³/ K·kg]**	2.5719 · 10^{-5}	4.425 · 10^{-5}
c_1 **[m³/ kg]**	-5.5063 · 10^{-3}	-1.35 · 10^{-3}
a_2 **[m⁶/ K²·kg²]**	2.4404 · 10^{-11}	-1.25 · 10^{-10}
b_2 **[m⁶/ K·kg²]**	-1.9165· 10^{-8}	-1.625 · 10^{-7}
c_2 **[m⁶/ kg²]**	4.5326 · 10^{-6}	1.575 · 10^{-4}

Thus, the real gas factor Z is calculated as shown in equation (5.17).

$$Z = 1 + \frac{a_1}{R}Tp + \frac{b_1}{R}p + \frac{c_1}{RT}p - \frac{{a_1}^2}{R^2}T^2p^2 - \frac{2a_1b_1}{R^2}Tp^2 - \frac{2a_1c_1 + {b_1}^2 - a_2}{R^2}p^2 + \frac{b_2 - 2b_1c_1}{R^2T}p^2 + \frac{(c_2 - {c_1}^2)}{R^2T^2}p^2 \tag{5.17}$$

By combining the constant coefficients, equation (5.17) can be simplified to (5.18).

$$Z = 1 + k_1 Tp + k_2 p + k_3 \frac{p}{T} - k_4 T^2 p^2 - k_5 T p^2 - k_6 p^2 + k_7 \frac{p^2}{T} + k_8 \frac{p^2}{T^2} \tag{5.18}$$

The coefficients k_1 to k_8 are defined as follows.

$$k_1 = \frac{a_1}{R} \qquad k_5 = \frac{2a_1 b_1}{R^2}$$

$$k_2 = \frac{b_1}{R} \qquad k_6 = \frac{2a_1 c_1 + {b_1}^2 - a_2}{R^2}$$

$$k_3 = \frac{c_1}{R} \qquad k_7 = \frac{b_2 - 2b_1 c_1}{R^2}$$

$$k_4 = \frac{{a_1}^2}{R^2} \qquad k_8 = \frac{(c_2 - {c_1}^2)}{R^2}$$

The real gas factor can now be used to eliminate the enthalpy's dependency of the variable v, by using equation (5.15) to transform the terms of v and $\left(\frac{\partial v}{\partial T}\right)_p$ of equation (5.12).

$$v = \frac{ZRT}{p} \tag{5.19a}$$

$$\left(\frac{\partial v}{\partial T}\right)_p = \frac{R}{p}\left(T\left(\frac{\partial Z}{\partial T}\right)_p + \mathrm{Z}\right) \tag{5.19b}$$

Inserting equations (5.19a) and (5.19b) into the last term of (5.12) and integrating, results in the following equation.

$$\begin{aligned}\int_0^p \left[v - T\left(\frac{\partial v}{\partial T}\right)_p\right] dp \\ = R\left[(k_3 - k_1 T^2)p + \left(2k_4 T^3 + k_5 T^2 + k_7 + \frac{2k_8}{T}\right)\frac{p^2}{2}\right]\end{aligned} \tag{5.20}$$

In conclusion, assembling equation(5.20) and (5.13), the equation to determine the enthalpy of an ideal gas results to the following expression.

$$\begin{aligned}h(T,p) = h_0 + \frac{1}{M}\left(AT + B\frac{T^2}{2} + C\frac{T^3}{3} + D\frac{T^4}{4} - \frac{E}{T} + F\right) \\ + R\left[(k_3 - k_1 T^2)p + \left(2k_4 T^3 + k_5 T^2 + k_7 + \frac{2k_8}{T}\right)\frac{p^2}{2}\right]\end{aligned} \tag{5.21}$$

With the enthalpy only dependent on the pressure and temperature of the gas, the change in temperature after the throttling process can be determined if the gas pressure and temperature within the gas tank and the pressure at state 2, after the throttling, is known. Therefore, equation (5.21) must be inserted into equation (5.10) and solved for the temperature T_2.

These calculations were performed with a Matlab code that is attached in the appendix. The results are illustrated in Figure 41 to Figure 44 for different variations of the pressure and temperature at the initial state 1 and final state 2 of the gases. Thereby, Figure 41 and Figure 42 show the change in the temperature of the resulting temperature as well as the resulting relative humidity at state 2 for a varying input pressure p_1. The left graph of the respective figures illustrates the temperature increase for the hydrogen and decrease for the oxygen gas from an initial temperature T_1 of 343.15 K in Figure

41 and a temperature T_1 of 303.15 K in Figure 42 and shows the outcome of the Joule-Thomson Effect. The drop or rise of temperature from T_1 to T_2 increases with an increasing pressure drop from p_1 to p_2. On the right side of the respective figures, the relative humidity that occurs at state 2 is shown in dependence of the initial pressure p1 with an initial relative humidity of 30% in state 1. It can be seen, that, even though the temperature of the oxygen is reduced, the relative humidity is also reduced. The reason therefore is that the pressure and hence the partial pressure of the water is reduced more than the vapor pressure of the water which declines due to the declining temperature in state 2, after the throttling process.

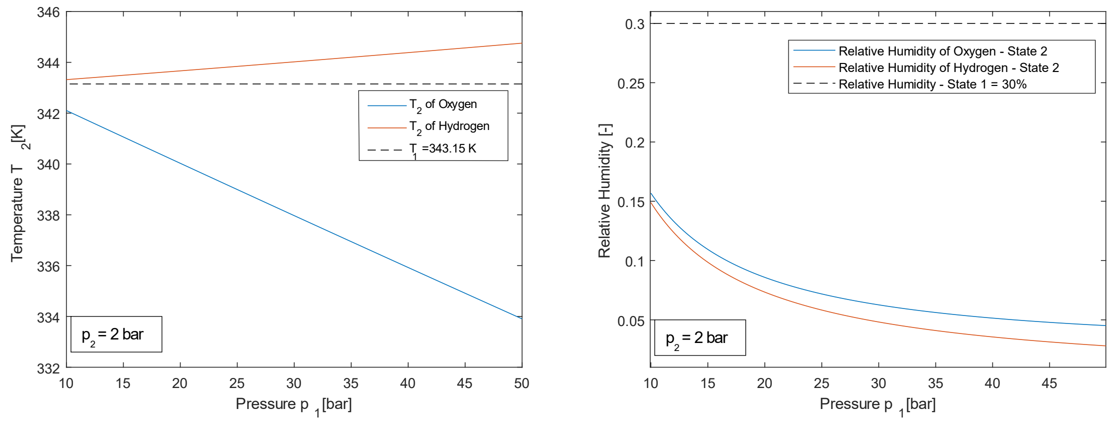

Figure 41:Temperature (left) and relative humidity (right) of oxygen and hydrogen gas after the throttling process for a varying initial pressure p_1, with a constant pressure p2 of 2 bar, an initial temperature T1 of 343.15 K and an initial relative humidity of 30% in state 1

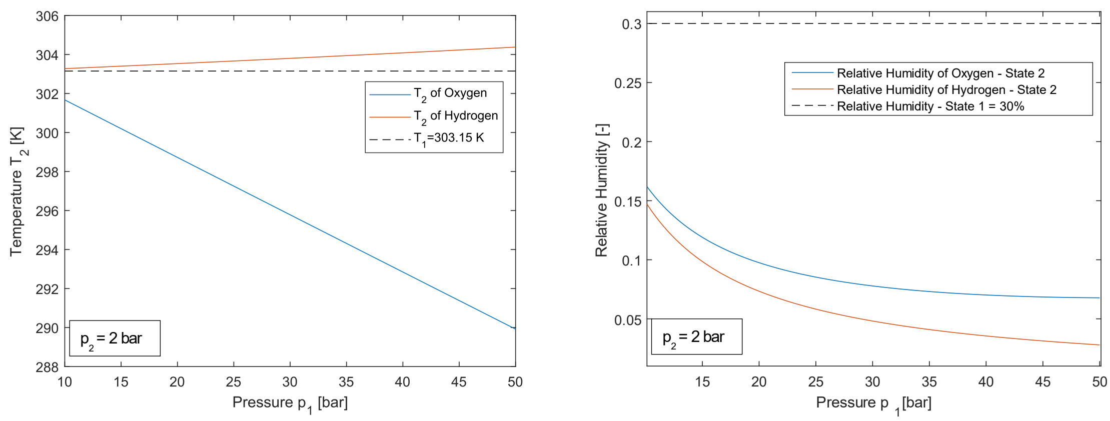

Figure 42: Temperature (left) and relative humidity (right) of oxygen and hydrogen gas after the throttling process for a varying initial pressure p_1, with a constant pressure p_2 of 2 bar, an initial temperature T_1 of 303.15 K and an initial relative humidity of 30% in state 1

Furthermore, Figure 43 and Figure 44 illustrate a similar behavior for a varying initial temperature T_1 rather than a varying initial pressure p_1. Thereby, Figure 43 shows the result of the calculation for a pressure drop from 50 bar to 2 bar and an initial relative humidity of 30% whereas Figure 44 shows the results for a pressure drop from 20 bar to 2bar with the same relative humidity. Once again, it can be seen that the temperature for the hydrogen gas increases while the oxygen gas decreases. As shown in the figures above, the Joule-Thomson effect is more dominant on the oxygen gas within the viewed temperature and pressure range, which is even increased for lower initial temperatures T_1. However, the decrease of temperature created by the Joule-Thomson effect does not increase the relative humidity. Even at 273 K the relative humidity of the oxygen water mixture is below the initial value of 30%.

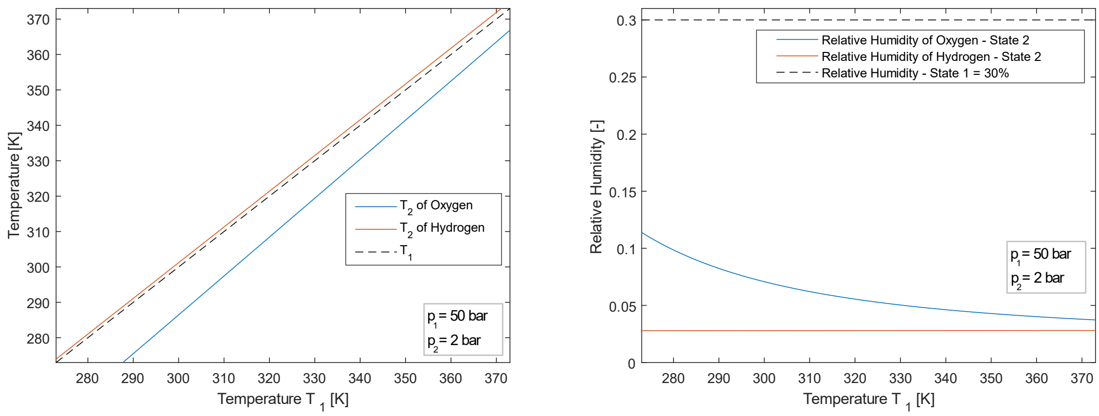

Figure 43: Temperature (left) and relative humidity (right) of oxygen and hydrogen gas after the throttling process for a varying initial Temperature T_1, with a constant pressure p_2 of 2 bar, an initial pressure p_1 of 50 bar and an initial relative humidity of 30% in state 1

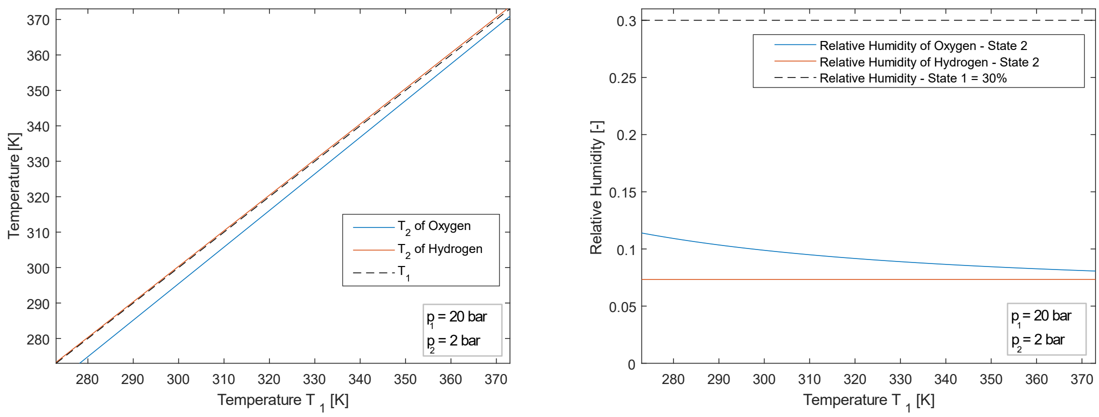

Figure 44:Temperature (left) and relative humidity (right) of oxygen and hydrogen gas after the throttling process for a varying initial Temperature T1, with a constant pressure p2 of 2 bar, an initial pressure p1 of 20 bar and an initial relative humidity of 30%

The trend of the oxygen temperature decrease and increase of relative humidity at lower temperatures was further investigated and the results are shown in Figure 45. It can be seen, that the relative humidity starts to increase at lower temperatures. At an initial temperature T1 of 225 K the relative humidity in state two stays equal to the initial humidity of 30 % and a further decrease of the initial temperature increases the relative humidity after the throttling until the gas is saturated and liquid water starts forming with an initial temperature of around 210 K.

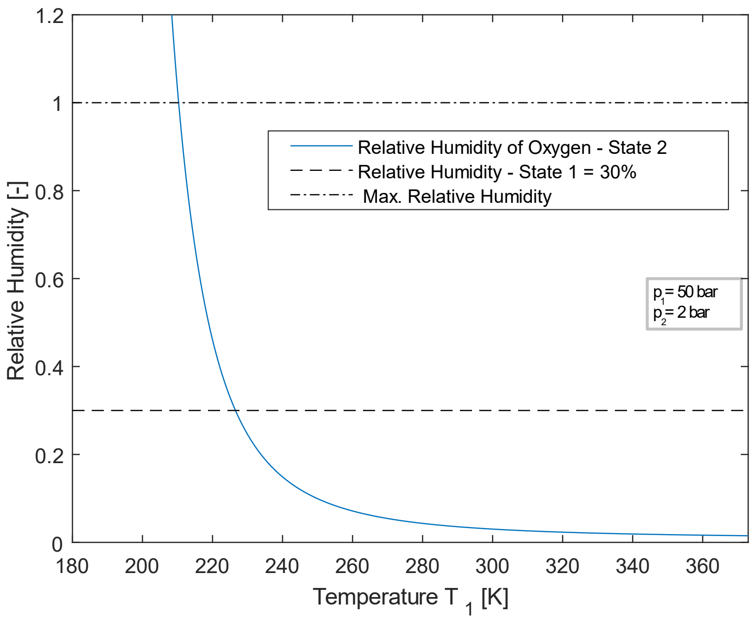

Figure 45: Relative humidity of an oxygen – water gas mixture after the throttling from 50 bar to 2 bar in dependency of the initial gas mixture temperature T_1

In conclusion, the calculations have shown, that the Joule-Thomson effect has only a minor influence on the relative humidity of the electrolysis gases produced by the DWPS' electrolyzer. Only if the temperatures within the gas tanks drop to a significant low level, the joule Thomson effect would result in the increase of the relative humidity and the formation of liquid water. However, these low temperatures are highly unlikely to occur and would result in far more damage to the electrolyzer and other systems of the DWPS than the formation of liquid water.

6 Mission

6.1 Mission Requirements

The main mission requirement is the fulfilment of the mission objectives these are described in chapter 4.2.1 and summarized in Table 33.

Table 33: Mission objectives of the IOD of the water propulsion technology

Ref.	Type	IOD Mission Objectives
OB-1	Primary	Demonstration of the functionality of the WPS' main components technologies (electrolyzer, thruster and control unit) in its operational environment
OB-2	Primary	Characterization and evaluation of the DWPS performance by measuring following parameters and transmitting them to the ground station: ▪ Water tank pressure and temperature ▪ Electrolyzer current, voltage and impedance ▪ Hydrogen and Oxygen tank pressure and temperature ▪ Thruster temperature ▪ CubeSat position and velocity
OB-3	Secondary	Evaluation of the long-term performance and degradation of the DWPS in its operational environment

To accomplish the first mission objective, OB-1, the CubeSat must be equipped with an electrolyzer, thruster and a control unit that are assembled and function just as the equal components in the actual WPS. Their functionality must be verified in a microgravity, vacuum and radiation exposed environment such as given in the LEO. In this environment, the electrolyzer shall be powered by a satellite platform and produce the gases that are combusted in the thruster. The thruster shall be tested in two firing modes, continuous firing and pulse firing, which allow the evaluation of the thruster's behavior in its operational environment.

To verify and characterize the exact performance of each component, the DWPS is equipped with various measuring devices that collect sensor data. The data is processed by the control unit and sent to the OBC which forwards it to the communication system for the transmission to the ground station. In return, the ground station generates and transmits commands to the CubeSat to control and operate the satellite. For a successful mission it is essential that all necessary data can be transmitted within the fly-by maneuvers of a satellite. A sufficient link must be provided for the communication of the ground station with the CubeSat. Therefore, the data transmission rate, the fly-by time as well as the amount of fly-bys must be high enough to transmit the required amount of data.

In general, the exact orbit of the CubeSat for the IOD of the Water Propulsion System is irrelevant. The functionality of the DWPS can be approved in LEO, GEO or any other orbit which provides the microgravity environment which is so hard to copy within a ground based testing facility. Two aspects, however, decisively influence the selection process of an orbit. First of all, the CubeSat shall comply with the Space Debris Mitigation Guidelines of the Inter-Agency Space Debris Coordination Committee which limits the end of life duration of satellites in certain orbits such as the LEO or GEO. LEO satellites shall possess a lifetime limit of 25 years after the termination of their operation. GEO satellite must be removed from the orbit before the end-of-life. To reduce the mission's complexity, the orbit shall be chosen from which the CubeSat will reenter earth's atmosphere within the stated 25 years without any additional maneuvers. This is only possible in LEO where the residual atmospheric gases create a drag force on the satellite and thereby reduce its altitude. On the other hand, the CubeSat must be capable of being operated in space long enough to fulfil the objective OB-3. Conventionally, the lifetime of CubeSats is limited to five years. These five years shall be fully exploited to allow the examination of the degradation behavior of the DWPS in a space environment.

A CubeSat generally has various launch opportunities, as it is shown in the chapter 6.2 hereafter. To reduce costs, the CubeSat shall be launched as a secondary payload.

6.2 Launch

Before defining the actual mission for the IOD it is necessary to research launch possibilities to be able to determine accessible orbits. As shown in the following, it won't be possible to predetermine the exact orbit for the CubeSat since it is generally defined by a primary payload or the highest bidder. Hence, the research on launch possibilities stated in this chapter shall not lead to one single launch option for the IOD mission but rather give an overview of possibilities.

A CubeSat is generally delivered to orbit via a rideshare launch. The orbit is then mostly determined by a primary, much larger payload which takes up most of the space and mass available for the payloads on the launch system. The available surplus of space on the launcher can be used by secondary payloads such as CubeSats which are therefore delivered to the same orbit as the primary payload.

A second option allowing access to an orbit is the deployment from the ISS. The CubeSat is brought to the ISS by one of the cargo spacecraft's delivering supplies to the station on a regular basis. It is then integrated into the Small Satellite Orbital Deployer of the Japanese Experiment Module (JEM) and deployed into the required direction with a robotic arm. The regular delivery of supplies to the ISS allows a very short turnover time. However, once on the ISS it might take another three to six months until the CubeSat is deployed. The Orbit of the CubeSat is fixed to the ISS's orbit, dependent on the actual position of the ISS, with an altitude between 400 to 420 km and an inclination of 51.6° [93].

Within the last few years further launch possibilities for CubeSats have appeared. The demand for launchers for smaller payloads has increased leading to initiatives such as the European Small Innovative Launcher for Europe (SMILE) project for the development of a launcher for Satellites up to 50 kg [90]. Various companies have recognized the potential of launches for small satellites and have made a lot of progress. An example of this progress is the Electron launcher, developed by Rocket Lab and initially launched in 2017 which is able to deliver a payloads of 200 kg to sun-synchronous orbit [91]. Other launch providers have adapted to the small satellite market and customized their launch vehicles to host CubeSats. The VEGA launcher for example was equipped with a newly developed modular carbon fiber dispenser SSMS (Small Spacecraft Mission Service) launched for the first time in the beginning of September 2020 and carrying more than 50 satellites, most of them CubeSats. The SSMS is part of the upper stage which is maneuvered by the liquid-fuel propulsion system AVUM (Attitude Vernier Upper Module). The restartable propulsion system makes it possible to deliver the satellites to different orbits with a single launch [92].

By April 2020 around 1200 CubeSats have been launched. While there have been CubeSats launched to mars, GTO and a few are planned for lunar orbits or other extraterrestrial destinations, most CubeSats circle the earth on a LEO orbit between 200 and 800 km. The vast majority of CubeSats are used in a SSO Orbit or in the ISS's Orbit. Launches were provided by Antares, Atlas V, Delta II, Dnepr, Electron, Falcon 9 & Falcon Heavy, H-IIA, H-IIB, Long March, Minotaur, PSLV, Soyuz and Vega rockets. The following table shall give an overview of the different launchers and possible orbits. Delta II and Dnepr will not be included since these two launchers are not being developed any more.

Table 34: Data on possible launch vehicles for CubeSat launch (September 2020) [95][96][97][98].

Launcher Family	Active Version	CubeSat Launches	Remarks
Antares	230+ - 233+	147	Cygnus as primary payload for ISS Resupply; no SSO Launches
Atlas V	all	140	Most CubeSats delivered to ISS or deployed from Cygnus
Electron	-	37	
Falcon	9	135	High inclination launches possible from Vandenburg
Falcon	Heavy	13	High inclination launches possible from Vandenburg
H-IIA	202	9	
H-IIB	304	47	HTV as primary payload for ISS Resupply; no SSO Launches
Long March	2D	19	Other configurations have carried CubeSats in very small numbers
Minotaur	4	6	
PSLV	-	271	
Soyuz	2.1a	102	
Soyuz	2.1b	10	
Soyuz	ST	3	ArianeSpace launch in Kourou
Vega	-	76	

It should be noticed, that this table includes only the currently available launchers. Further developments of small satellite launchers might be available by time of the CubeSat launch. Launchers such as Firefly Alpha, Vector-H and Vector-R, Vega-C, Orbex or LauncherOne could provide access to space for CubeSats since their maiden flights are all planned for the next two years.

Regarding the number of CubeSats launched on previous but also upcoming, already planned missions, the chances of a launch with a Falcon 9, PSLV, Vega, Antares, Atlas V or Soyuz rocket are the highest. The orbits with these launchers are either fixed to the ISS Orbit (Antares) or can vary between low inclination degrees to polar orbits with an SSO as the most common orbit [29][94].

6.3 Orbit

The satellite’s orbit has various influences on the mission. First of all, depending on the orbit a satellite spends more or less time in the earth’s shadow. During the eclipse phase the satellites energy supply is based solely on the stored energy and reduces the average available power for the CubeSat. On the contrary, the satellite is exposed to higher thermal loads and an eclipse phase is an effective way of keeping components temperatures within their limits. The chosen orbit also defines the possibility to communicate with a ground station. Dependent on the inclination of the satellite and the latitude of the ground station it might be possible to exchange data with the satellite either a few times a day or never at all. Additionally, the lifetime of a CubeSat in Orbit is strongly dependent on the orbit altitude. Within the LEO residual particles from earth’s atmosphere cause an atmospheric drag which slows the satellite and lowers its altitude. The density increases with decreasing altitude resulting in a faster orbital decay over the CubeSats lifetime.

As the research on previous CubeSat launches has shown, most CubeSats are either launched to an SSO or to the ISS orbit. The ISS orbit has an altitude of around 400 kilometer and an inclination of 51.6°. At this altitude the mission lifetime is limited to a maximum of around 24 months before the CubeSat reenters the earth’s atmosphere. However, changes in the atmospheres density can result in a faster orbital decay resulting in an end of life within 9 months after deployment.

An SSO has many advantageous compared to the ISS orbit. The SSO is a particular kind of polar orbit with an inclination greater than 90° degrees. Due to the earth’s oblateness, an orbiting satellite is exposed to a gravitational force which increases with a decreasing altitude and results in a gyroscopic precession. Within a SSO the inclination is chosen so that this force is synchronized to the earth’s rotation around the sun. Thus, the orbital plane is always aligned to the sun in the same way. This simplifies the mission design extensively. The eclipse and sun phases are the same during the entire mission and, if the launch is conducted at a certain time, the satellite can be set into an orbit without any eclipse at all. Furthermore, the satellite flies over the earth’s surface at the same local time from south to north and a local time of 12 hours later from north to south. The orbit of an individual satellite is therefore named after the time at which it travels from south to north, also called Local Time of Ascending Node (LTAN). This characteristic simplifies the mission operation since the satellite will pass the ground station always at the same local time. With a typical inclination between 97° and 98° degree at an altitude between 470 and 650 km for CubeSats in an SSO, the satellite will pass a ground station situated in the polar region every single orbit. Ground stations in polar regions such as operated by the Swedish Space Corporation (SSC) or the Norwegian company Kongsberg Satellite Services AS (KSAT) make it possible to contact the satellite very frequently. Even if the Satellite shall be operated by a different ground station it can be an option to acquire the service and support by SCC and KSAT for critical phases during the mission.

To protect and preserve near earth space, satellites orbiting in crowded orbits such as the LEO or GEO should be removed from these orbits at the end of life (EOL). To limit the long term presence of satellites after their EOL in LEO, the Inter-Agency Space Debris Coordination Committee defined a limit of 25 years within which spacecraft shall reenter the earth’s atmosphere. As shortly depicted, a satellite orbiting the earth in LEO is exposed to a drag resulting from the residual atmosphere at this height. The drag force depends upon the satellites mass and cross-sectional surface in the direction of travel, the density of the atmosphere and the orbital parameters. The density of the upper atmosphere can be calculated by various empirical models that have been postulated in the last few decades. In general, the upper atmosphere’s density is dependent on various influences including the solar radiation flux and is not easily predicted. For the following calculations, the NRLMSISE-00 model was used assuming mean solar activity [99]. This model allows a simple approach to the orbital decay calculation and only requires the altitude as orbital parameter. After the actual operation, the CubeSat will not be controlled anymore. Since most CubeSats are built to only survive a maximum of around 5 years in orbit, the CubeSat’s attitude control will stop functioning at some point and the CubeSat attitude will be random. Therefore, the cross-sectional surface A_{CS} of the CubeSat shall be averaged over all possible viewing angles using a flat plate model as shown in [100] and calculated as follows.

$$A_{CS} = \frac{1}{2}(S_1 + S_2 + S_3)$$

With the CubeSat cross-sectional surface and the density model a drag on the satellite can be calculated:

$$F_D = \frac{1}{2}\rho v^2 A_{CS} C_D$$

The drag coefficient C_D of a satellite is a further parameter that is hard to determine. Commonly, the drag value is assumed to be equal to two but can vary widely. For long-duration orbit lifetimes the variation of the $C_D$can be ignored and an average value of two can be assumed [99][101].

From Kepler's second law we can derive the orbital period P as function of the satellite's orbit semi-major axis:

$$P^2 G M_E = 4\pi^2 a^3$$

Including energy considerations for a circular orbit results in the reduction of the period by:

$$\frac{dP}{dt} = -3\pi a\rho \frac{A_{CS} C_D}{m_{sat}}$$

With these formulas a simple Matlab code was written to determine the orbital altitude at which the CubeSat will reenter the earth's atmosphere within the required timespan of 25 years. Figure 46 shows the orbital decay of a 6U CubeSat with a mass of 12 kg in a circular orbit up to an altitude of 700 km. The orbital decay which is solely dependent on the atmospheric drag in this calculation was calculated for three different atmospheric densities since the density of atmospheric gases varies strongly throughout within the LEO. The main influence thereof is the solar and geomagnetic activity given by the F10.7 parameter for the solar flux and the geomagnetic activity described by the geomagnetic indicia a_p. At a high solar and geomagnetic activity atmospheric gases reach higher altitudes increasing the density in LEO. The resulting higher drag on the satellite reduces its altitude faster reducing the in-orbit lifetime. The influence thereof on the lifetime can be seen very well in Figure 46. The lifetime of a 6U CubeSat at an altitude of 700 km can vary between 10 and 1000 years depending on the solar and geomagnetic activity. However for a long duration in space, it can be assumed that the solar and magnetic activity changes constantly which is why the density for a mean solar and geomagnetic activity can provide the most accurate lifetime prediction. The Matlab code can be found in the appendix.

Regarding the lifetime for mean solar and geomagnetic activity in Figure 46, it can be seen that the lifetime in orbit increases drastically with the altitude. The decay from 700 to 600 kilometers takes around 75 years while on the contrary it takes the satellite only 25 years for the orbital decay from 600 kilometers to the earth's surface. The calculation terminates at an altitude of around 200 km since the orbital decay from this height occurs within a few hours [101]. From this calculation the upper limit for the CubeSats orbit can be defined to 600 km. With a desired minimum lifetime of 5 years the satellite should on the other side be deployed at an orbit above 500 km.

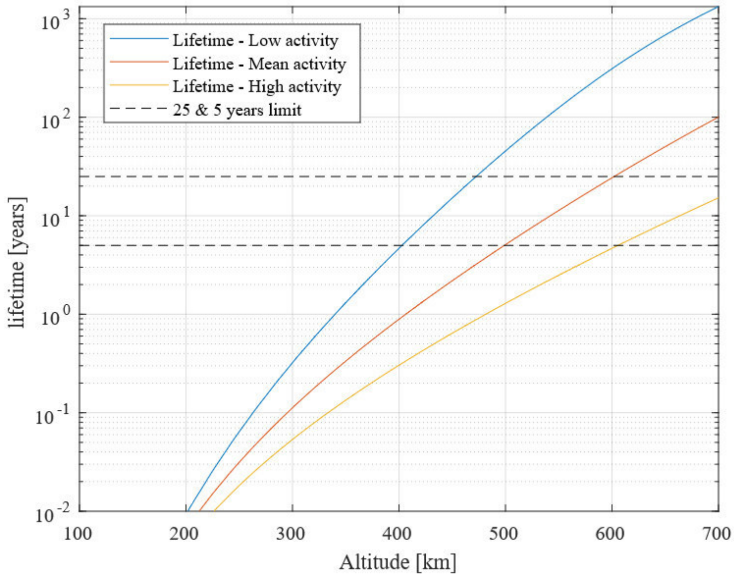

Figure 46: Orbital decay and lifetime of a 6U CubeSat with a mass of 12 kg in LEO

With these limitations the nominal orbit with an altitude of 550 kilometers was chosen. An orbital lifetime of around 10 years was calculated for this altitude providing a sufficient margin, making it possible to operate the satellite for at least 5 years while assuring an atmospheric decomposition within the required 25 years. The inclination changes with the altitude as shown in Figure 47. At the altitude of 550 kilometers the inclination for an SSO is calculated to 97.6° [29].

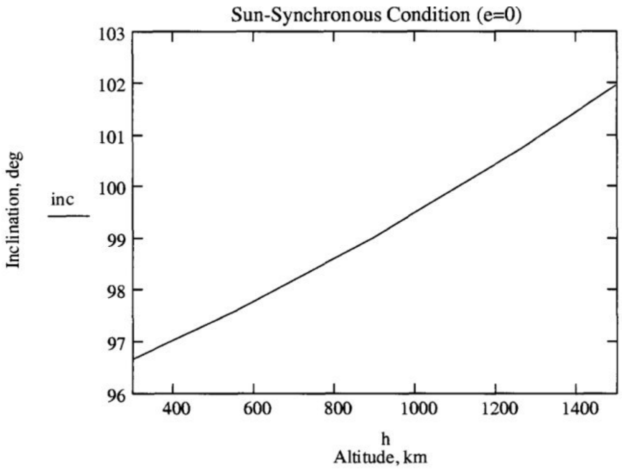

Figure 47: Altitude versus inclination for an SSO [102]

More detailed orbit calculations should be performed in the future. Tools such as the Systems Tool Kit (STK) by Analytical Graphics Inc or NASA's Debris Assessment Software (DAS) allow a more detailed calculation of the orbital decay and the fly by times in which the satellite can communicate with a given ground station. However, for now, the Matlab calculation makes it possible to define a nominal orbit altitude at which the in-orbit demonstration shall be performed.

Using the orbit evaluation tool SPENVIS the orbital parameters where calculated and are shown in the following table:

Table 35: Designated Orbit for IOD mission

Orbital Parameter	Value
Altitude	550 km
Inclination	97.6°
Period	1.59 h
Number of Orbits per day	15.09

6.4 Mission Profile

Since it is only possible to communicate with a satellite in an SSO for short periods around 4 times within 24 hours, the operation of the DWPS must be automatized and every required task within a process must be considered and preplanned. Within this chapter, the on-orbit operation of the DWPS shall be illustrated. At first a maneuver sequence will be described followed by the required input parameter for the mission design and the performance of the DWPS for a generic maneuver.

6.4.1 Operational Sequence

For safety reasons, CubeSats are launched in their power off state. All components must be disconnected from the electrical power supply until the satellite is deployed from the launcher [17]. Once the power supply is reactivated, the CubeSat starts switching its components on, checks their functionality and performs the detumbling of rotation caused by the deployment. After this initial phase, the DWPS can be activated and the in orbit demonstration of the propulsion system can start. This section aims to describe the sequence of tasks that are performed by the DWPS in combination with the CubeSat platform during the in-orbit operation. This scheme of the operation starting with the activation and ending with the passivation of the DWPS is shown in Figure 48. Each task listed in the scheme shall be encapsulated in the following.

I. The CubeSat platform activates the DWPS by supplying the ECU with power
II. The ECU then starts with its health monitoring activities, which include the measurement of temperature and pressure data. This health monitoring process is performed in regular time steps continuously throughout the entire mission
III. The information on the temperature of the DWPS' subsystems is processed by the ECU's microprocessor which activates respective heaters if the temperature of individual components drops below its limit.
IV. In the meantime, the microprocessor sends the accumulated data to the OBC which stores the information and prepares the data ton transmit it during the next ground station fly-by maneuver.
V. The fifth step is the initial task for the nominal operation of the DWPS. For the purpose of supplying the DWPS with the necessary power, the platforms battery must be charged sufficiently. If its charge is below a predefined value, the operation of the DWPS must be delayed until the solar panels have generated enough power.
VI. Once the battery is charged sufficiently, the CubeSat activates the electrolyzer by commanding the ECU to start the gas production.
VII. The ECU then must activate the hydrogen pump in the first cycle. Once it is active it must be active as long as hydrogen is within the hydrogen gas compartment of the electrolyzer. Otherwise the pressurized hydrogen could migrate through the water feed barrier, mask the membrane and prevent the further flow of water.
VIII. The actual gas generation starts. The electrodes are powered current controlled by the ECU until the pressure in the gas tanks reach the defined value at a controlled temperature.
IX. The power supply of the electrolysis electrodes is deactivated as soon as the gas tanks are full
X. Subsequent to the deactivation of the power supply of the electrodes, the ECU notifies this information to the CubeSat's OBC.
XI. Preliminary to an actual maneuver, the reaction control wheels must be desaturated, to make sure, that any rotation caused by a DWPS thrust maneuver can be compensated.
XII. If the reaction wheels are not sufficiently desaturated, the magnetorquers must be activated to perform the desaturation
XIII. On the contrary, if the reaction wheels are sufficiently desaturated, the OBC can start a firing maneuver by calling for a predefined maneuver.
XIV. The ECU receives the command by the OBC and starts by activating the catalyst bed heater to heat the catalyst to its operational temperature. If a few maneuvers are fired within a short duration, this step might be unnecessary, since the catalyst could possesses a high enough temperature from previous firings.

XV. Since the firings are performed within only a few seconds or even milliseconds, the data acquisition rate of pressure and temperature data must be increased during the firing duration.

XVI. The ECU activates and opens the FCVs and holds them open at two different voltages

XVII. By deactivating the power supply, the FCVs close after a predefined time, which is dependent on the desired maneuver. The FCV can be activated to allow either a continuous firing or a firing of short pulses.

XVIII. The measurement data collected by the ECU is sent to the OBC which forwards them to the CubeSat's communication system for the downlink during the next ground station fly-by.

XIX. If the gas tanks still carry a sufficient amount of gas, the next firing can be prepared by starting at XI again. On the other hand, if the gas tanks are empty, the filling level of the water tanks must be checked.

XX. If the water tanks contain water, the gas generation process is restarted by returning to step V. However, If the water tank is fully depleted, the DWPS must be passivated

XXI. Therefore, the gases that remain in the hydrogen and oxygen tank shall be drained separately until the pressure drops to zero bar.

XXII. The ECU then deactivates the power supply for all components and is then deactivated itself by the OBC in the last step.

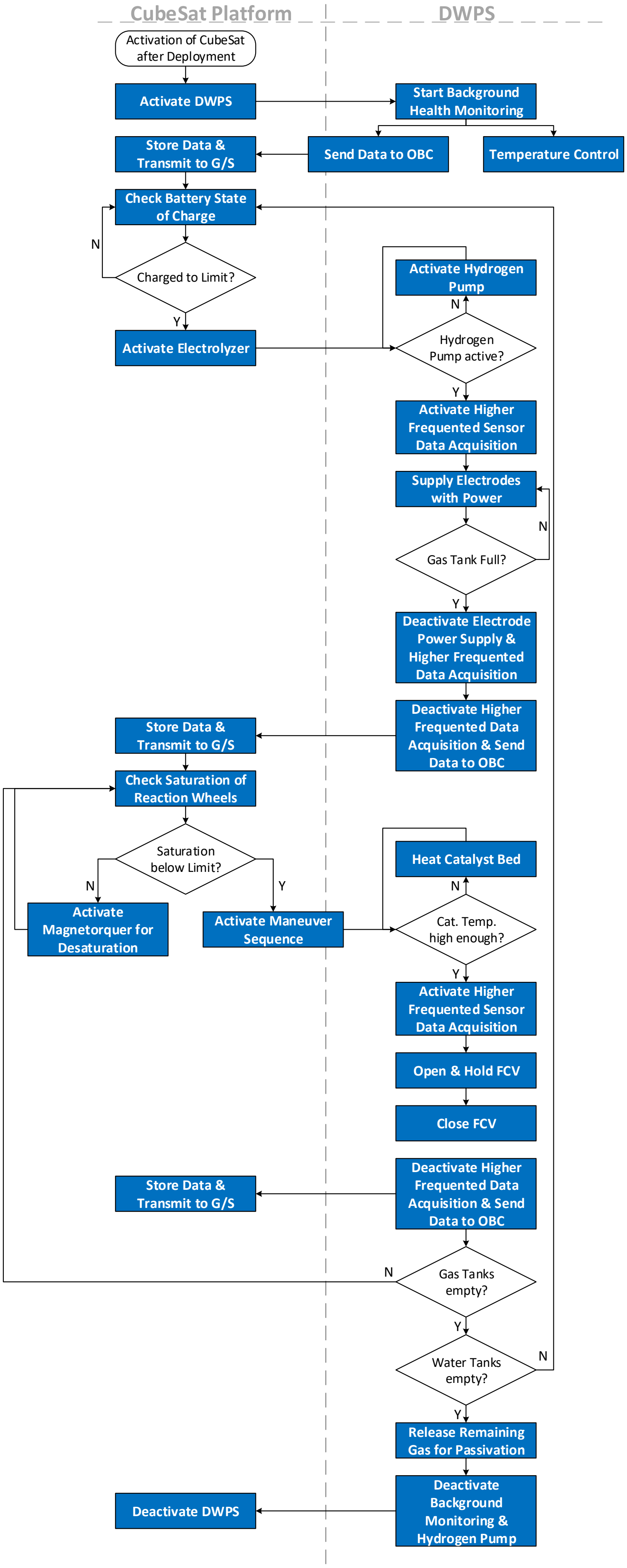

Figure 48: Mission Sequence of the DWPS from its activation to passivation

6.4.2 Input parameter

The mission profile is determined by various influences. To allow an estimation of the performance and the illustration of certain characteristics, the first step is to define input parameters that are required for the calculations.

The first of these parameters is the water tank size. Its volume specifies the amount of water that can be aboard the CubeSat and thereby presets the possible operation duration of the DWPS. Since a water tank has not yet been designed it is hard to predict the amount of water that can be carried within the CubeSat envelope. Thus, for an initial mission profile, the propellant tank size of the Valcor piston tank shall be used as a reference. A single piston tank possesses a volume of 131 ml. With a maximum pressure of 12 bar at the beginning and a final pressure of 2 bar within the water tank, the nitrogen pressurant takes up $^{2}/_{12} \approx 16.67\ \%$ of the tank volume which is equal to 21.8 ml in each tank. The amount of water is therefore limited to two times 109 ml resulting in an overall mass of 218 gram of water available for the CubeSat mission.

The second decisive parameter is the volume of the gas tanks. It defines the gas production time and thereby the required electrical energy required within a gas production cycle. For the gas tanks, a volume of 75 ml for the oxygen and 150 ml for the hydrogen tank was derived from a former similar design.

The power available for the gas production is estimated to be around 50 W. Deviations thereof shall be evaluated by including further power supplies at 40W and 60 W to see the dependency of a change in the input power. The efficiency of the DWPS electrolyzer depends on various influences, such as the temperature, pressure or concentration of the gases, the resistance of the conducting materials and losses for the activation of the electrolysis process [9]. The overall power efficiency has not yet been determined for the DWPS electrolyzer which is why a value for η_{ELY} equal to 88% shall be used as reference which is the efficiency published by Tether's Unlimited for their Hydros CubeSat propulsion system's PEM electrolyzer [26]

With this amount of gas that can be stored within the tanks, the impulse is very high and is expected to exceed the momentum storage capacity of the reaction wheels. Therefore, the depletion of the tanks cannot be done at once. With a maximum thrust of around 0.6 N and a maximum torque of 3.4 mNm, the maximum off-center deviation of the thrust vector is calculated to 5.6 mm. With this deviation and the maximum momentum storage of 20 mNms, the maximum impulse that can be fired before the reaction wheels are saturated can be calculated to 3.5 Ns.

Table 36 depicts further assumptions required for the dimensioning and summarizes the input parameters. Table 37 thereafter states physical constants and chemical properties required for the calculations.

Table 36: Input Values for the performance calculation

Parameter	Variable	Value
Water Mass	$m_{H2O,tot}$	218 g
Expulsion Efficiency WTs	η_{WT}	95%
Usable Water Mass	m_{H2O}	207 g
Water Temperature	T_{H2O}	293.15 K
H2 / O2 Tank Volume	V_{H2} / V_{O2}	150 ml / 75 ml
GTs' MEOP / MOP	$p_{GT,MEOP}$ / $p_{GT,MOP}$	50 bar / 10 bar
Gas Temperature	T_{GT}	343.15
Available Power for Electrolysis	P_{ELY}	40 W / 50 W / 60 W
ELY Power Efficiency	η_{ELY}	88%
Specific Impulse	$\mathbf{I_{sp}}$	300 s
Max. Maneuver Impulse	$\mathbf{I_{Man}}$	3.5 Ns
CubeSat Total Wet Mass	$m_{CS,wet}$	12 kg
CubeSat Dry Mass	$m_{CS,dry}$	11.782 kg

Table 37: Physical constants and chemical properties of H2 and O2 [87]

Parameter	Variable	Value
Molecular Mass	M_{H2}	2 g·mol^{-1}
Molecular Mass	M_{O2}	32 g·mol^{-1}
Charge Number	z_{H2}	2
Charge Number	z_{O2}	4
Heat Capacity Ratio for H2 & O2	κ	1.4
Faraday Constant	$\mathcal{F}$	96485.33 A·s·mol^{-1}

6.4.3 Performance

This section shall provide an overview of the DWPS' expected performance in orbit, calculated with the input parameters defined in chapter 6.4.2

With these parameters, the performance of the DWPS can be characterized. At first, the production of the gases within the electrolyzer shall be reviewed. At the current state of the development the exact performance cannot be given yet. However, a simple way of calculating the rate at which the gases are produced, is given by the Faraday's Law. According to the Faraday's Law, the amount of substance that yields at the electrodes is proportional to the electric charge across the membrane. This relation can be used to determine the rate at which the mass of hydrogen and oxygen are produced and is given for hydrogen in equation (6.1).

$$\dot{m}_{H2} = \frac{M_{H2}\, I\, \eta_{ELY}}{z_{H2}\, \mathcal{F}} \tag{6.1}$$

It shows that the gas production rate is dependent on the gases' molecular mass (M_{H2} or M_{O2}) and charge number (z_{H2} or z_{O2}), the Faraday constant ($\mathcal{F}$) and the current (I) that is supplied to the electrodes. Additionally, the efficiency factor (η_{ELY}) includes losses. In the ideal case, the electrolysis process is activated once the electrodes provide a voltage of around 1.48 V which corresponds to the enthalpy of reaction of water [7]. For the actual electrolyzer, losses must be considered so that higher voltages must be applied. For the following calculation, 2 V is assumed as the average required voltage for the electrolysis process. With these parameters, the production mass rate of oxygen and hydrogen can be calculated dependent on the available power as shown in Table 38.

Table 38: Production rate of electrolysis gases at different power levels

	40 W	50 W	60 W
$\dot{m}_{H2}$	$0.657\,\frac{g}{h}$	$0.821\,\frac{g}{h}$	$0.985\,\frac{g}{h}$
$\dot{m}_{O2}$	$5.253\,\frac{g}{h}$	$6.567\,\frac{g}{h}$	$7.880\,\frac{g}{h}$

To fill the gas tanks at this rate from zero to 50 bar, the electrolyzer must produce 4.21 gram of oxygen and 0.53 gram of hydrogen according to the ideal gas law. From 10 bar to 50 bar which is defined as the nominal operation range, 3.36 gram of oxygen and 0.42 gram of hydrogen must be produced. The required time for this process is stated in Table 39.

Table 39: Production duration to fill gas tanks from 0 or 10 bar up to 50 bar

	40 W	50 W	60 W
$t_{0-50\,bar}$	0.80 h	0.64 h	0.53 h
$t_{10-50\,bar}$	0.64 h	0.51 h	0.42 h

If these production rates can be maintained throughout the entire mission, the overall time, the electrolyzer requires to produce hydrogen and oxygen from the 207 grams of water, can be derived from equation 6.1. The results are shown in Table 18.

Table 40: Overall duration for the conversion from 207 grams of water to hydrogen and oxygen

	40 W	50 W	60 W
t_{total}	35.02 h	28.02 h	23.35 h

Since the electrolyzer produces around 3.79 gram of electrolysis gases within each gas production cycle, the electrolyzer can be operated for 55 cycles. Of course, the gas tanks don't always have to be fully filled during an operational cycle. If required, the gas tanks can be filled to any pressure between 10 and 50 bar.

Once a sufficient amount of gas is stored, the actual in orbit maneuver can take place. Therefore, as shown in Figure 48, the CubeSat's reaction wheels must be desaturated and the catalyst must be heated to its operational temperature in the initial step. The FCVs can then be opened to initiate the firing and closed after a certain duration, to terminate it. The firing duration is limited by the maximum possible impulse. For high pressures, the mass flow is higher so that the limiting impulse of 3.5 Ns is reached within a shorter firing duration whereas at lower pressures, the lower mass flow allows a longer firing. If the specific impulse can be assumed to be constant throughout the entire blow down pressure spectrum, then the maneuver impulse can be derived directly by the pressure differential. This is possible since the impulse of a maneuver $I_{tot,man}$ is proportional to the mass of propellant used during a maneuver (equation 6.2).

$$I_{tot,man} = \Delta m_{H2O}\, I_{sp}\, g_0 \tag{6.2}$$

The mass per maneuver is proportional to the pressure difference within the tanks before and after the maneuver. To show this, equation (2.7) is rearranged as shown in equation (6.3) and integrated as shown in (6.4).

$$\int dm_{gas} = C_d\, A_o\, \kappa \sqrt{\frac{\left(\frac{2}{\kappa+1}\right)^{\frac{\kappa+1}{\kappa-1}}}{\kappa\, R\, T}} \int_{t_1}^{t_2} p_{in}\, dt \tag{6.3}$$

By inserting equation (5.1), (6.3) can be integrated resulting in equation (6.4). The time constant term on the right side of equation (6.3) is abbreviated to the constant K.

$$\Delta m_{gas} = -\tau\, K\, [\, p(t_2) - p(t_1)] \tag{6.4}$$

This Δm calculation must be executed for both gases separately, since τ and K are two gas dependent constants. Adding the Δm values of the hydrogen and oxygen thereby results in the Δm_{H2O} value.

$$\Delta m_{H2O} = \Delta m_{H2} + \Delta m_{O2} = (\tau_{H2}\, K_{H2} + \tau_{O2}\, K_{O2})\, [\, p(t_1) - p(t_2)] \tag{6.5}$$

By inserting (6.5) into (6.2), equation (6.2) can be solved for the maximum pressure between the pressure at the start to the pressure at the end of a maneuver, which would result in a maneuver that achieves an impulse of 3.5 Ns. For the defined input parameters and the gas tanks sizes of 75 ml and 150 ml, the pressure within the tanks must drop by 12.57 bar to achieve the 3.5 Ns impulse. This calculation uses a lot of assumptions, such as the assumption that the temperature within the tanks stays constant or that the specific impulse is 300s in average for any maneuver. The calculation can therefore just give the order of magnitude rather than the exact value.

With these calculations, the longest and shortest maneuver that achieve an impulse of 3.5 Ns can be calculated. At lower pressures the outflow of gases is slower, reducing the average thrust and making it possible to fire a longer maneuver. Using the maximum pressure difference of 12.57 bar, the maximum and minimum maneuver time for this pressure drop are given on Table 41.

Table 41: longest and shortest maneuver to achieve 3.5 Ns

Maneuver	Gas Tank Pressure	Maneuver Duration
Shortest 3.5 Ns Maneuver	50 bar → 37.43 bar	7.01 s
Longest 3.5 Ns Maneuver	22.57 bar → 10 bar	19.71s

It can be seen that the firing duration can be tripled by using different inlet pressures. The maneuver from 22.57 bar down to 10 bar thereby represents the longest possible firing for the DWPS.

Furthermore, the thruster shall also be operated in pulse mode. Therefore, the thruster is fed with hydrogen and oxygen for very short durations. The minimum on/off time that can be achieved by the FCVs sets the minimum impulse bit that can be achieved by the thruster. The FCVs' minimum on/off time is defined to 25 ms. The specific impulse at such short pulse firings might vary significantly since a lot of energy can dissipate as heat into the cold structures. Tests on the thruster shall verify the minimum impulse bits achievable.

For an SSO, the CubeSat would pass a ground station, such as the ground station at the IRS in Stuttgart, around 4 times a day. Two times while traveling from south to north in two subsequent revolutions and another two time around 12 hours later while traveling from north to south. For an orbit with an LTAN of 6 am, the first two flybys will occur at around 6 am and the other two flybys will take place at around 6 pm. Therefore, in the first flyby, the CubeSat shall send the health monitoring data of the CubeSat and the DWPS to the ground station. Within one orbital revolution, the data must be assessed and approved so that for the second flyby, the command, to start a maneuver can be uploaded. The DWPS system then has around 12 hours till the next flyby, to perform the sequence it was commanded to do. Thus, the first 12 hours can be used for the gas production and gas storage, the second 12 hours for the firing of the gases. The duration of the in-orbit operation can then be directly derived from the maximum possible charging cycles. The water tanks supply 207 g of water that are transformed to hydrogen and oxygen within 55 cycles to fill the 25ml and 50 ml oxygen and hydrogen gas tanks to a pressure of 50 bar. The actual operation of the DWPS could therefore be conducted within 55 days. However, since the long term stability and functionality of the DWPS shall be investigated, a certain amount of water must be saved so that after each year, the CubeSat spends in space, around 2 or 3 charging and firing cycles can be performed in orbit. Therefore, the main characterization of the DWPS' performance can be done within the first 2 months if everything works out as planned. Thereafter, the DWPS can be operated for further cycles each year.

Overall, within this duration and considering the input parameter, the DWPS's electrolyzer produces hydrogen and oxygen from 207 grams of water. The gases are then combusted by the thruster, to achieve a total impulse of 609 Ns at an average specific impulse of 300 s. With an initial CubeSat wet mass of 12 kg, the DWPS can provide a total Δv of $51.21\ \frac{m}{s}$ according to the Tsiolkovsky rocket equation.

6.5 Ground Station

For the operation of the satellite, regular communication with a ground station is required. The communication of the satellite with the ground station is possible when the satellite is in sight of the ground station which is therefor only possible for a certain, in a LEO very short period of time. Within this short time the satellite has to send all the necessary housekeeping and payload data to the ground station while in return the ground station sends commands to the satellite. The time frame in which the contact between satellite and ground station is possible is predefined by the CubeSat's Orbit. To increase the amount of data that can be sent in this time frame, the bit rate can be increased. However, the achievable bit rate is dependent on the communication link between the satellite and the ground station and their respective communication system. E.g. a satellite, transmitting with a higher power or a ground station with a higher antenna gain can achieve a higher bit rate allowing more data to be sent. To give an overview of possible bit rates, a link budget was calculated and will be described in the following.

6.5.1 Link Budget

The link budget was calculated for the ground station at the institute of space systems (IRS) at the University of Stuttgart. The institute implemented the infrastructure for the satellite communication for the satellite "Flying Laptop" and will also operate the CubeSats "CAPE" and "Source" for the student small satellite group KSat at the University of Stuttgart. The experience of the IRS working on CubeSat projects in cooperation with students make the institute a suitable and desirable partner for the in-orbit demonstration of the DWPS. However, the link budget calculated in this chapter can be easily adapted to other ground stations if required.

6.5.1.1 Downlink

The downlink is established by the satellite emitting radio frequency waves at a certain frequency. Most communication systems of CubeSats use a frequency in either the VHF, UHF or S Band. Since it is estimated that only low data rates are required, the link budget was calculated using the UHF transceiver system of ISIS,GOMSpace and NanoAvionics The link budget starts with the transmitter power of the satellite which is generally assumed to be around $P_{Tx} = 1\ W$ for CubeSats [31][34][35]. The unit of the power is given in dBW or dBm, as shown in the following equation (6.6).

$$P_{Tx,[dBm]} = 10log_{10}\left(\frac{P_{Tx}}{1dBm}\right) \tag{6.6}$$

So that the transmitting power results in $P_{Tx,dBm} = 30dBm$ emitted at the transmitter outlet [103]. The signal travels through the signal line to the antenna with a line loss of around $L_{L,CS} = 0.5dB$ and is then emitted towards earth surface by a deployable antenna system with an almost omnidirectional emission. Hence, the antenna gain is assumed to be $G_{CS} = 0\ dB$. The equivalent isotropically radiated power (EIRP) can then be calculated by equation (6.7) [103]:

$$EIRP = P_{Tx,[dBm]} + G_{CS} - L_{L,CS} = 29.5dBm \tag{6.7}$$

The signal emitted by the satellite will be exposed to various effects and losses along its path to the ground station antenna. Thus, the signal power at the ground station is strongly dependent on the path length. For the link budget calculation the longest possible distance is used which is derived from the lowest elevation α at which the ground station can receive the satellite's signal as shown in Figure 49. The distance d can be calculated according to equation (6.8).

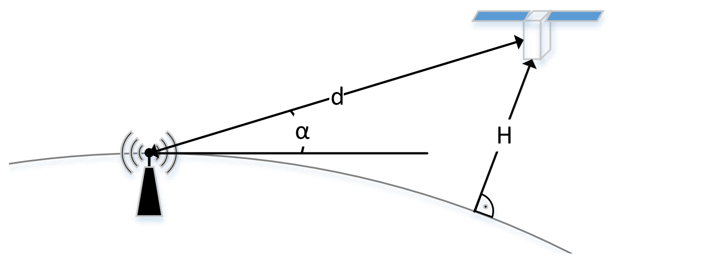

Figure 49: Slant range d for an orbiting CubeSat at the elevation α and an altitude of H

$$d = -R_E \sin(\alpha) + \sqrt{(R_E \sin(\alpha))^2 + H^2 + 2R_E H} \tag{6.8}$$

Resulting in a slant range d of 1570 km with the earth's radius R_E of 6371 km and a minimum elevation angle α of 10° [103]. The elevation angle is defined by the position of the ground station and its antenna. 10° is the typical value used for the IRS ground station, as stated by the ground station operator at the institute.

Along this slant range the signal is influenced by various losses especially due to the divergence of the signal. Electromagnetic waves spread out as defined by the inverse square law which leads to a reduction of the signal power proportional to the square of the distance. The so called free-space path loss can be calculated by the Friis transmission formula [104]109[105]:

$$L_{FS} = 20log_{10}\left(\frac{4\pi d}{\lambda}\right) \tag{6.9}$$

The wavelength λ can be calculated by the given frequency f and the speed of the wave, the speed of light in vacuum: $\lambda = \frac{c}{f}$

Additional losses along the signal's path occur due to the interaction between the electromagnetic wave and particles of the earth's atmosphere. In the upper atmosphere the radio waves are influenced by ionized media which is generated by solar radiation. The ionosphere in general causes four types of different power attenuation:

1. Faraday rotation: Rotation of the wave polarization due to the total electron content (TEC) along the satellite's transmission path.
2. Local ionospheric irregularities causing random rotation and time delay of the propagating wave
3. Dispersion and group velocity distortion due to non-linear frequency dependent interaction of ionospheric electrons and electromagnetic wave
4. Scintillation due to localized ionospheric irregularities acting like convergent and divergent lenses for the electromagnetic wave

These effects occur at wave frequencies bellow around 12 GHz and become severe at frequencies below 1 GHz to a point where the ionosphere cannot be penetrated anymore and the radio wave is fully reflected. In the lower earth atmosphere, the particle density increases, increasing the interaction between the electromagnetic waves and uncharged molecules such as atmospheric gases, clouds and rain. These effects become relevant at a frequency above 1 GHz and are partly neglected even up to a frequency of 10 GHz. The exact overall atmospheric attenuation can be calculated according to the International Telecommunication Union (ITU) statements as specified in the recommendations ITU-R P.531-14 and ITU-R P.618-12. For the link budget calculation within this work an estimation considering similar communication systems of CubeSat projects shall be sufficient especially since the attenuation is dependent on various factors such as solar flux, total electron content or weather effects along the satellites transmission path that are not easily predicted. The atmospheric attenuation L_{atm} shall be similar to the attenuation stated in [35] and [103] and was estimated to 2 dB after consultation with the ground station operator at the institute of space systems [106][107].

The last transmission path loss is defined by the respective antenna systems. Apriori, the CubeSat antenna has no exact pointing direction but sends the signal at a constant power for a great beam angle. Therefore, a pointing loss of the CubeSat antenna can be neglected. The CubeSat antenna sends the signal with a circular polarization. This makes it possible for the ground station to receive data even if the CubeSat is spinning. However, the receiving antenna is usually designed to receive linear polarization signals resulting in a polarization loss of 50% of the transmitted power. For the link budget the loss is once again transformed to the unit dB and results in a loss L_{Pol} of 3 dB. The Isotropic Signal Level that arrives at the ground station can then be calculated by adding up all the losses.

$$P_{GS,[dBm]} = EIRP - L_{FS} - L_{atm} - L_{Pol} \tag{6.10}$$

The satellite's signal at the ground station is then received by an antenna system with a given pointing loss and antenna gain. At the IRS in Stuttgart, the antenna has a pointing loss $L_{P,GS}$ of less than 0.5 dB and an antenna gain of 14 dBi which could be increased to 16 dBi if required. The cable from the antenna to the ground station receiver has a length of around 30 meters with a transmission line loss of 0.08 dB per meter. The effective noise temperature of the ground station was approximated by the ground station operator of the IRS to a maximum of 750 K. The ground station figure of merit can then be calculated as follows:

$$F_{G/_T} = G_{GS,[dB]} - 10log_{10}(T_{N,GS}) \tag{6.11}$$

From the figure of merit the ground stations Signal-to-Noise power density can be calculated.

$$\left(\frac{SNR}{B}\right)_{[dB/Hz]} = P_{GS,[dBm]} - L_{P,GS} + F_{G/_T} - 10log_{10}\left(\frac{k}{1000}\right) \tag{6.12}$$

With the Bandwidth B and the Boltzmann constant which has to be adjusted to the unit $\left[\frac{dBm \cdot s}{K}\right]$.

For the next step the desired data rate D has to be defined. The a typical data rate for UHF CubeSat transmitter systems is around 9.6 kbps which equals 39.82 $\frac{dB}{Hz}$. Dividing the signal to noise power density by the data rate, as shown in equation (6.13), results in the E_B/N_0 parameter which defines the available signal power per bit relative to the noise power in a 1 Hz bandwidth. This parameter is used to define the error probability during the transmission process. A higher value of E_B/N_0 implies that a higher amount of energy is available for the transmission of a single bit, making it easier for the receiver to differ between signal and noise.

$$(E_B/N_0)_{[dB]} = \left(\frac{SNR}{B}\right)_{[dB/Hz]} - D_{[dB/Hz]} \tag{6.13}$$

In the next step, an acceptable bit error rate must be defined. It defines the maximum allowable bit reading errors that can be accepted for the respective data. A bit error rate of 10^{-5} was suggested by the IRS ground station operator to be sufficient.

Each received signal is demodulated from its carrier signal to a digital signal that can be processed by a computer. Various different demodulation methods can be used. The reviewed CubeSat communication system utilize the GMSK or FSK method. The demodulation method decisively affects the bit error rate. The FSK method requires a much higher input signal to be capable of differing between noise and the actual signal. An E_B/N_0 threshold defines the minimum E_B/N_0 value that is required for the demodulation method to operate sufficiently. Values for the threshold are taken from different literature [35][108]. By dividing the E_B/N_0 value by the E_B/N_0 threshold, the link margin is calculated. The link margin has to be positive to allow a stable communication between the satellite and the ground station. In general, a positive value between 6 and 10 dB is preferred, as specified by the IRS ground station operator. This ensures that additional effects and losses that have not been considered in the margin calculation will not inhibit the satellite communication. Link margins above 10 dB indicate an oversized communication link which allow even higher bit rates than assumed in the calculation.

The downlink budget is shown in the following table, Table 42. It can be seen, that the highest data rates achievable with the GOMSpace communication system. The downlink of the ISIS system is not fully utilized since the margin is above the specified 10 dB. However, the communication system itself is limited to a data rate of 9.6 kbps.

Table 42: Downlink Budget

Downlink Parameter	ISIS	NanoAvioncs	GOMSpace	Unit
Spacecraft				
Duplex	full duplex	half duplex	half duplex	-
Tx Frequency	267-273	395-440	430-440	MHz
Ref. Frequency	267,0	430,0	430,0	MHz
Tx Output Power	0,501	1	1	W
- in dBm	27	30	30	dBm
Tx Antenna Gain	0	0	0	dBi
Spacecraft Total Transmission Line Loss	0,5	0,5	0,5	dB
Polarization	circular	circular	circular	-
EIRP	26,5	29,5	29,5	dBm
Downlink Path				
Downlnk Antenna Pointing Loss	0	0	0	dB
S/C-to-Ground Antenna Polarization Loss	3	3	3	dB
Slant Range	1571,8			km
Free Space Loss	144,9	149,0	149,0	dB
Atm. Attenuation (estimation)	2			dB
Signal level at G/S	-123,4	-124,5	-124,5	dBm
Ground Station				
Ground Station Antenna Pointing Loss	0,5			dB
Ground Station Antenna Gain	14			dBi
Ground Station Total Transmission Line Loss	2,4			dB
Ground Station Effective Noise Temperature	750			K
Ground Station Figure of Merrit (G/T)	-14,75			dB/K
G/S Signal-to-Noise Power Density	59,9	58,8	58,8	dBHz
Maximum possible Data Rate	**9600**	**9600**	**19200**	bps
- in dBHz	39,82	39,82	42,83	dBHz
Telemetry System Eb/No for the Downlink	20,12	18,98	15,97	dB
Demodulation Method	GMSK	GFSK2 / GFSK4	GFSK / GMSK	-
System Allowed or Specified Bit-Error-Rate	1,0E-05			-
Eb/No Threshold	7,8	13	7,8	dB
System Link Margin	12,32	5,98	8,17	dB

6.5.1.2 Uplink

The uplink is generally less problematic than the downlink. While the CubeSat transmitter power is limited to the available power of around one or two watts, the transmission power of a ground station can easily be increased by adding an amplifier. The calculation is similar to the downlink calculation only with a different noise temperature at the receiving CubeSat. Since the antenna of the CubeSat is pointed towards earth, it will receive the electromagnetic waves that are emitted due to the surface and atmosphere temperature. The noise temperature received by the antenna is therefore 290 K which is often assumed to be the average surface temperature of the earth [103]. This noise temperature is normally increased by an amplifier and modified by losses in the transmission line between antenna and receiver. Since the signal line in a CubeSat is very short and the antenna provides no gain, the noise temperature shall be assumed to be equal to the antenna noise temperature of 290 K which represents a conservative assumption compared to the link budget in [35].

The uplink budget is shown in Table 43. The link margins calculated for the uplinks are much higher than for the downlink and illustrates the advantage of a signal sent from a ground station rather than the signal from a CubeSat. The power can easily be amplified to 100 W with reduced resources.

Table 43: Uplink budget

Uplink Parameter	ISIS	NanoAvioncs	GOMSpace	Unit
Rx Frequency	312-322	430 - 440	430-440	MHz
Ref. Frequency	312	430	430	MHz
Rx Sensitivity	-104	-120	-137	dBm
Ground Station				
Ground Station Transmitter Power Output	100			W
- in dBm	50			dBm
G/S Total Transmission Line Losses	2,4			dB
G/S Antenna Gain	14			dBi
G/S EIRP	61,6			dBm
Uplink Path				
G/S Antenna Pointing Loss	0,5			W
G/S-to-S/C Antenna Polarization Loss	3			dBm
Free Space Loss	146,259012	149,0452892	149,0452892	km
Atmospheric Losses	2			dBi
Signal level at G/S	-90,159	-92,945	-92,945	dBm
Spacecraft				
S/C Antenna Pointing Loss	0			0
S/C Antenna Gain	0			0
S/C Total Transmission Line Losses	0,2			0,2
S/C Effective Noise Temperature	290			290
S/C Figure of Merrit (G/T)	-24,62			-24,62
S/C Signal-to-Noise Power Density (S/No)	83,82	81,03	81,03	dBHz
Maximum possible Data Rate	**9600**	**38400**	**38400**	bps
- in dBHz	39,82	45,84	45,84	dBHz
Command System Eb/No	43,99	35,19	35,19	dB
Demodulation Method	FSK	GFSK4	GMSK	-
System Allowed or Specified Bit-Error-Rate	1,0E-05			-
Eb/No Threshold	13	13	7,8	dB
System Link Margin	30,99	22,19	27,39	dB

7 Conclusion and Outlook

The Demonstrator Water Propulsion System (DWPS) described and defined within this thesis is designated to test and characterize the functionality of the Water Propulsion System (WPS) in its operational space environment. The WPS is an innovative semi-electric propulsion system based on the in-orbit production of hydrogen and oxygen by a PEM electrolyzer. Such a propulsion system has never been used nor tested in space, so that an In-Orbit Demonstration (IOD) is seen as an important appliance to characterize its behavior in space, to reduce the risk of failure of the WPS and is considered as a highly valuable measure to convince potential costumers. The demonstrator propulsion system thereby consists of adapted replicas of the WPS' main subsystems: A catalytic thruster fed by oxygen and hydrogen, produced in a PEM electrolyzer that are scaled in size and performance to fit onboard a CubeSat.

The first step in the definition and the design of the DWPS is the review of suitable CubeSat platforms. Therefore, chapter 3 summarizes CubeSat subsystems and gives an overview of reference platforms of the three satellite developing companies, ISIS, NanoAvionics and GOMSpace. All three of them are specialised on the design and development of CubeSat platforms and offer all-inclusive support from the development and assembly of the CubeSat platform to the organization of regulatory duties, conducting launch preparations and performing the actual operation of the satellite in space. The size of 6U was recognized as large enough to carry the DWPS for the mission. The research on the platforms has shown that the in-orbit demonstration of the propulsion system aboard a CubeSat at this size implicates a few restrictions that must be considered during the DWPS development.

- The available and maximum power is limited due to limited power generation, storage and distribution possibilities on the CubeSat
- The maximum mass and size of the DWPS is limited by the CubeSat standard
- The maximum thrust and impulse generated by the thruster must be limited so that the ADCS can stabilize the CubeSat sufficiently during a maneuver

The DWPS and its subsystems are then defined in detail in chapter 4. The technical requirements defined herein include the requirements and limitations of the CubeSat platform but are also based on the results from the R&D activities on the actual WPS. These requirements include the functional demands on each subsystem, an overview of the necessary components and their limitations, environmental influences that must be withstood throughout the entire mission and definitions of interfaces between the individual subsystems and to the structure.

These specifications on the technical requirements form the foundation for the development and design of the DWPS. They are and were used by three other students that work on the design and development of a thruster, an electrolyzer with included gas tanks and a control unit - the ECU (electrolyzer control unit), for the DWPS. Additional to these three major subsystems, the DWPS requires tanks to store the water in a space environment

For the choice of the water tanks a trade study resulted in piston tanks as the preferable tank technology. Within a space environment, a piston tank can ensure a steady supply of water to the electrolyzer. A main advantage of the piston tank is the flat cylindrical ends of the tank which allow the most volume efficient design for a water tank within a box shaped CubeSat. Other technologies such as Propellant Management Device (PMD) tanks or bladder and diaphragm tanks were also reviewed. The use of PMD tanks initially seemed very promising. These tanks, which are based on the expulsion of liquids by its surface tension, possess a high potential for a lightweight and less complex design. However, the liquid water is not separated from the gaseous pressurant by a barrier so that the pressurant could enter the electrolyzer and thereby damage it. Diaphragm or bladder tanks on the other hand constitute sufficient alternatives to the piston tanks.

The fluid system is based on a water side, which is composed of the water tanks, connecting tubes and a part of the elctrolyzer, and a gas side, which contains the gas side of the electrolyzer and the gas tanks. The electrolyzer is thereby the barrier between the water and gas side and is exposed to fluctuating pressure gradients up to almost 50 bar. The electrolyzer, which shall be composed of two individual double cells, represents the most challenging component of the DWPS. It is the component with the lowest maturity and the highest complexity and is the main element that is to be tested in orbit. A leak or rupture within a cell can thereby decontaminate the entire system and must therefore be prevented. Thus, an arrangement of check and latch valves is designed to prevent the failure of the entire electrolyzer in the case of the failure of only one cell.

Furthermore an analysis was performed to examine the gases produced by the electrolyzer. Generally, the oxygen and hydrogen contain a certain amount of gaseous water. The gases, which are fed to the thruster through orifices undergo a throttling process which changes the gas temperature of real gas by the so-called Joule-Thomson effect. This cooling could result in the formation of liquid water which would influence the propulsion systems performance negatively. The results of a calculation performed with a Matlab code however, demonstrate that the Joule-Thomson effect has an insignificant impact on the gases. The relative humidity of the gas mixtures is generally reduced rather than increased by the throttling within the range of the viewed inlet temperature and pressure so that no further design implementations have to be considered to prevent the condensation of water.

The concept of the DWPS results in a propulsion system with the size of four CubeSat units that can carry at least around 220 millilitres of water to achieve a Δv of 51 m/s. The next step in the development will require a more detailed design of the electrolyzer control unit (ECU) which defines the electrical interface to the CubeSat platform. Additionally, the mounting structure of the DWPS to the CubeSat must be investigated. It must be capable of withstanding the mechanical loads occurring throughout the entire mission. At last, the thermal design of the CubeSat and the propulsion system must be reviewed. So far, the temperature limits for each subsystem have been defined. A simulation of the in-orbit thermal environment and heat transfer conditions must be established to show the temperatures that occur during the operation. If necessary, heat conducting or insulating devices must be added to prevent overheating of certain components.

The actual mission is then described in the last chapter of this thesis. The CubeSat shall be launched to an sun-synchronous orbit of around 550 km. At this orbit, it is possible to contact a ground station situated in central Europe at least four times a day, for a ground station closer to the polar region it is possible to contact the satellite even more often. The operation of the propulsion system is generally dividable in three phases: The production of hydrogen and oxygen, the actual maneuver and the desaturation of the reaction wheels. Before the beginning of a cycle of these three phases, the battery must be charged sufficiently. The first contact is thus mainly used to check the battery state. If the batteries are sufficiently charged, the gas production can be initiated by the second contact. The third contact is then used to inform the ground station of the state of the DWPS so that the fourth communication link to the satellite can be used to start the firing of the gases and the subsequent desaturation of the reaction wheels thereafter. A single cycle is therefore performable within 24 hours. A more automatized approach might be applicable after the functionality of the propulsion system in orbit has been verified sufficiently. The minimum expected amount of water stored within the water tanks of around 220 grams in combination of the specified gas tank sizes, allows the system to be operated for at least 55 cycles in orbit.

In conclusion, it can be stated, that the initial step for the demonstration of the Water Propulsion System is completed. The concept for the DWPS has been defined in detail and components for the use within the CubeSat have been chosen. An overview of CubeSat platforms shows the limitations for the propulsion system but also shows the possibilities. The available power supply as well as the envelope of a 6U CubeSat suffice for the operation of the propulsion system in orbit. A main challenge for the design and especially the assembly of the DWPS will be the precise alignment of the thruster to conform with the strict thrust vector misalignment limitations. A preliminary mission design shows that the main mission objectives can be realized comfortably within the first 6 months in orbit.

In the next step, the design of the propulsion system concept must be realized and a demonstrator model must be developed and tested. Thereafter, the thermal and mechanical loads can be defined in detail, to investigate the structural integrity of the system and to develop a thermal control system for the CubeSat model. Further tasks will include the development of the ECU software which must be conducted in close cooperation with the development of the Software for the CubeSat OBC.

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Appendix

Appendix A – Functional Analysis

A detailed overview of the main and secondary functions is shown in the following two tables, Table 44 and Table 45. The allocation of which function is performed and which constraint is applicable in which mission phase is shown in the Operational Matrix in Table 46 thereafter.

Table 44: Detailed main functions of the DWPS

Main Functions	
Fm1	To store water
Fm1.1	To store water in liquid phase
Fm1.2	To store water leak-tight
Fm2	To conduct water from a water reservoir to the electrolyzer
Fm2.1	To provide pure, pressurant free water
Fm2.2	To provide a sufficient mass flow
Fm3	To produce electrolysis gases
Fm2.1	To supply the electrolyzer with water
Fm2.2	To supply the electrolyzer with power
Fm2.3	To perform electrolysis
Fm2.4	To extract the gases separately from the electrolyzer
Fm4	To store the electrolysis gases
Fm3.1	To feed the electrolysis gases to tanks by increased pressure
Fm3.2	To store the electrolysis gases separately
Fm5	To generate thrust
Fm4.1	To feed the gases separately to a combustion chamber
Fm4.2	To mix the gases
Fm4.3	To ignite the gases
Fm4.4	To combust the gases
Fm4.5	To guide the combustion gases in thrust opposing direction
Fm4.6	To provide the mechanical & thermal strength to transfer thrust to S/C

Table 45: Detailed secondary functions of the DWPS

Secondary Functions	
Fs1	To follow commands from CubeSat OBC
Fs1.1	To receive commands
Fs1.2	To process commands
Fs1.3	To power actuators (valves, heaters, electrolyzer)
Fs2	To acquire power from CubeSat
Fs2.1	To receive power
Fs2.2	To regulate power
Fs2.3	To distribute power
Fs3	To provide sensor data
Fs3.1	To acquire data
Fs3.2	To process data
Fs3.3	To transmit sensor data
Fs4	To accommodate water in a pressurized reservoir
Fs4.1	To allow water to enter the reservoir
Fs4.2	To store water under pressure
Fs5	To extract/drain gas out of the reservoir (e.g. passivation)
Fs6	To maintain components within their temperature and pressure limits
Fs6.1	To insulate components
Fs6.2	To heat components
Fs6.3	To conduct and radiate heat from high temperature components
Fs6.4	To open valves for safety release of pressure
Fs7	To prevent backflow of gases and flames

Table 46: Operational Matrix of the DWPS main (Fm) and secondary (Fs) functions and constraints (Fc)

				FM					FS						FC															
		Phase		1	2	3	4	5	1	2	3	4	5	6	1	2	3	4	5	6	7	8	9	##	##	12	13	14	15	16
1		DWPS Acceptance Test			X	X	X	X	X	X	X	X	X	X	X	X	X		X		X	X		X		X				
2		Fueling							X	X	X	X	X				X		X							X	X	X	X	
3		Storage										X				X	X	X								X				
4		Transport to Customer										X				X	X	X								X				
5		Integration into S/C															X		X		X	X				X		X	X	
6		Acceptance Test S/C							X	X	X	X	X		X	X	X		X		X	X				X	X	X		
7		Integration S/C into Deployer															X		X			X	X			X		X		
8		Transport to Launch Site										X				X	X						X			X	X			
9		S/C Integration into Launcher																					X			X	X	X		
10		Transport to Launchpad										X				X	X						X			X	X	X		
11		Launch										X				X						X	X			X	X	X		
12		Launch Raise										X				X							X		X	X	X	X		
13		S/C Release							X	X	X	X				X	X					X	X			X	X	X		
14		S/C Stabilisation Mode							X	X	X	X				X	X													
15		PS System Check																												
	I	Health Check							X	X	X	X			X	X	X				X			X						
	II	Drain Storage Gas							X	X	X	X	X	X	X	X	X			X	X			X						
	III	Production of Propellant Gas for Checking		X		X	X		X	X	X	X	X		X	X	X				X			X						
	IV	Stop Propellant Gas Production				X	X		X	X	X	X	X		X	X	X				X			X						
	V	WPS in Stand-by					X		X	X	X	X	X		X	X					X			X						
	VI	Thruster Check (firing)						X	X	X	X	X	X		X	X	X			X	X			X						
16		Orbital Phase																												
	VII	Production of maximum Propellant Gas		X		X	X		X	X	X	X	X		X	X	X				X			X						
	VIII	Stop Propellant Gas Production				X	X		X	X	X	X	X		X	X	X				X			X						
	IX	WPS Operational (Subsystem in Stand-by)					X		X	X	X	X	X		X	X	X				X			X						
	X	WPS active firing						X	X	X	X	X	X		X	X	X			X	X			X						
	XI	WPS in Stand-by					X		X	X	X	X	X		X	X					X			X						
	XII	a) Refilling of Propellant and back to VIII		X		X	X		X	X	X	X	X		X	X	X				X			X						
		b) Start Passivation Sequence							X	X	X				X	X					X			X						
17		Passivation																												
	XIII	Drain Propellant Gas							X	X	X	X	X	X	X	X	X			X	X			X					X	
	XIV	Production of Propellant Gas		X		X	X		X	X	X	X	X		X	X	X				X			X					X	
	XV	Stop Propellant Gas Production (when WT empty)					X		X	X	X	X	X		X	X	X				X			X					X	
	XVI	Stop Draining Propellant Gas							X	X	X	X	X	X	X	X	X				X			X		X			X	
	XVII	Switch Off Electrolyzer							X	X	X	X			X	X	X				X			X		X			X	
	XVIII	Switch Off Valve Driver and WPS Electronics							X	X	X	X			X	X	X				X			X		X			X	
	XIX	Passivation of Power Supply							X							X	X				X			X		X			X	
18		Uncontrolled De-Orbit													X		X									X			X	X

Appendix B – Test Requirements

Appendix B1

The following tables specify the vibrational loads that have to be applied during qualification testing of the DWPS and its subsystems.

Table 47: Sinusoidal vibration loads for qualification and resonance search

Axis	Test	Frequency [Hz]	Acceleration [g] (0-Peak)	Sweep Rate
all (x,y,z)	Qualification	5 to 20	± 11.0 mm (max Shaker Travel)	2 oct/min
		20 to 100	± 20	
all (x,y,z)	Resonance Search	5 to 2000	0.5 or 1.0	2 oct/min

Table 48: Random vibration loads for qualification testing

Axis	Acceptance		Qualification	
	Frequency	Single Amplitude	Frequency	Single Amplitude
all (x,y,z)	20 - 50	+3dB/oct	20 - 80	+3dB/oct
	50 - 800	0.32	80 - 400	0.7
			400 - 560	-3dB/oct
			560 - 800	0.5
	800 - 900	0.082	800	0.35
	900 - 2000	-9dB/oct	800 - 2000	-3dB/oct
Total		60 s/axis		120 s/axis
		16.9 g_{rms}		27.71 g_{rms}

Table 49: Shock vibration loads for qualification testing

Frequency [Hz]	Acceleration [g]
100	20
2000	2000
10000	2000

Appendix B2

The following tables define the required qualification and acceptance test sequence for the DWPS system and its respective subsystems thereafter.

Table 50: Qualification test sequence for the assembled DWPS

No.	Qualification Test
1	Acceptance tests
2	Visual inspection
3	Proof pressure test
4	External leak test
5	internal leak test
6	Electrical checkout (Coil / heater resistance, insulation resistance, response times, coil inductance) for all electrical connected components (Valves, Sensors, Heaters, Harness)
7	Resonance search
8	Qualification sinusoidal vibration
9	Resonance search
10	Qualification random vibration
11	Resonance search
12	Shock
13	Resonance search
14	Functional verification according to No. 4-6
15	Thermal Vacuum test
16	Functional verification according to No. 4-6
17	Gas flow test
18	Performance qualification hot firing & gas production rate
19	Gas flow test
20	Functional verification according to No. 4-6
21	Alignment verification
22	Final Examination

Table 51: Acceptance test sequence for the assembled DWPS

No.	Acceptance Test
1	Visual inspection (without mass measurement)
2	Alignment verification
3	Proof pressure test
4	External leak test
5	Internal leak test
6	Electrical checkout (coil/heater resistance, insulation resistance, response times, coil inductance) for all electrical connected components (Valves, Sensors, Heaters, Harness)
7	Gas flow test
8	Vibration test - Resonance search - Acceptance Random Vibration - Resonance search
9	Acceptance Function tests: hot firing test & electrolyzer gas production tests with mission duty cycles
10	External leak test
11	Internal leak test
12	Electrical checkout (see 6.)
13	Gas flow test
14	Alignment verification & mass measurement
15	Final examination

Table 52: Qualification test sequence for the thruster assembly

No.	Qualification Tests
1	Acceptance tests
2	Visual inspection
3	Proof pressure test
4	External leak test
5	internal leak test
6	Electrical checkout (Coil / heater resistance, insulation resistance, response times, coil inductance)
7	Resonance search
8	Qualification sinusoidal vibration
9	Resonance search
10	Qualification random vibration
11	Resonance search
12	Shock
13	Resonance search
14	Functional verification according to No. 4-6
15	Thermal Vacuum test
16	Functional verification according to No. 4-6
17	Gas flow test
18	Performance qualification hot firing
19	Gas flow test
20	Functional verification according to No. 4-6
21	Alignment verification
22	Radiographic inspection of thrust chamber
23	Burst pressure test
24	Final Examination

Table 53: Acceptance test sequence for the thruster assembly

No.	Acceptance Tests
1	Visual inspection (without mass measurement)
2	Alignment verification
3	Proof pressure test
4	External leak test
5	Internal leak test
6	Electrical checkout (coil/heater resistance, insulation resistance, response times, coil inductance)
7	Gas flow test
8	Vibration test - Resonance search - Acceptance Random Vibration - Resonance search
9	Acceptance hot firing test with mission duty cycles
10	External leak test
11	Internal leak test
12	Electrical checkout (see 6.)
13	Gas flow test
14	Alignment verification & mass measurement
15	Final examination

Table 54: Acceptance test sequence for the water tank assembly

No.	Qualification Tests
1	Acceptance tests
2	Visual inspection
3	Proof pressure test
4	External leak test
5	internal leak test
6	Electrical checkout (pressure transducer & heater)
7	Resonance search
8	Qualification sinusoidal vibration
9	Resonance search
10	Qualification random vibration
11	Resonance search
12	Shock
13	Resonance search
23	Burst pressure test
24	Final Examination

Table 55: Acceptance test sequence for the water tank assembly

No.	Acceptance Tests
1	Visual inspection
2	Proof pressure test
3	External/internal leak test
4	Mass measurement
5	Final examination

Table 56: Qualification test sequence for the electrolyzer assembly

No.	Qualification Tests
1	Acceptance tests
2	Visual inspection
3	Proof pressure test
4	External leak test
5	internal leak test
6	Electrical checkout (heater resistance, insulation resistance, response times)
7	Resonance search
8	Qualification sinusoidal vibration
9	Resonance search
10	Shock
11	Functional verification according to No. 4-6
12	Thermal Vacuum test
13	Functional verification according to No. 4-6
14	Gas and water flow test
15	Performance qualification of gas production
16	Gas and water flow test
17	Functional verification according to No. 4-6
18	Burst pressure test
19	Final Examination

Table 57: Acceptance test sequence for the electrolyzer assembly

No.	Acceptance Tests
1	Visual inspection
2	Proof pressure test
3	External leak test
4	Internal leak test
5	Electrical checkout (heater resistance, insulation resistance, response times)
6	Gas and water flow test
7	Vibration test - Resonance search - Acceptance Random Vibration - Resonance search
8	Acceptance gas production functional test with mission duty cycles
9	External leak test
10	Internal leak test
11	Electrical checkout (see 6.)
12	Gas and water flow test
13	Mass measurement
14	Final examination

Appendix C – Throttling Calculation

```
%% Intro
%The following code calculates the temperature T2 and relative humidity of
%hydrogen and oxygen after a throttling process in which the Joule-Thomson effect
%results in the heating or cooling of the gas
clc;
close all;
clear all;

%% Input
%Gas parameters
T1 = 343.15;
p1 = 50* 10^5;
p2 = 2*10^5;

%relative humidity
rh_1 = 0.3;

%% Constants
%Molecular Mass & Gas Constants
R_u = 8.3144626181532;
M_o = 32 * 10^-3;
M_h = 2 * 10^-3;
R_o = R_u/M_o;
R_h = R_u/M_h;

%Heat Capacity Constants - NIST
%Oxygen
A_o = 31.32234;
B_o = -20.23531*10^-3;
C_o = 57.86644*10^-6;
D_o = -36.50624*10^-9;
E_o = -0.007374*10^6;
F_o = -8.903471*10^3;

%Hydrogen
A_h = 33.066178;
B_h = -11.363417*10^-3;
C_h = 11.432816*10^-6;
D_h = -2.772874*10^-9;
E_h = -0.158558*10^6;
F_h = -9.980797*10^3;

%Virial coefficientes
%Oxygen
a_1_o = -3 * 10^-8;
b_1_o = 2.5719 * 10^-5;
c_1_o = -5.5063 * 10^-3;
a_2_o = 2.4404 * 10^-11;
b_2_o = -1.9165 * 10^-8;
c_2_o = 4.5326 * 10^-6;

% Hydrogen
a_1_h = -5.25 * 10^-8;
b_1_h = 4.425 * 10^-5;
c_1_h = -1.35 * 10^-3;
a_2_h = -1.25 * 10^-10;
b_2_h = -1.625 * 10^-7;
c_2_h = 1.575 * 10^-4;


%Water Vapor Pressure
As=4.6543;
Bs=1435.264;
Cs=-64.848;
```

```matlab
%% Calcualtion

% Partial pressures
p_s_1    = 10^5*10^(As-(Bs/(T1+Cs)));
p_H2O_1 = rh_1 * p_s_1;
p_o_1    = p1 - p_H2O_1;
p_h_1    = p_o_1;

pr       = p_o_1/p_H2O_1;

% Temperature T2 for Oxygen (o) and Hydrogen (h)

T2_o = T2(T1,N,p1,p2,R_o,M_o,A_o,B_o,C_o,D_o,E_o,F_o, ...
          a_1_o,b_1_o,c_1_o,a_2_o,b_2_o,c_2_o);
T2_h = T2(T1,N,p1,p2,R_h,M_h,A_h,B_h,C_h,D_h,E_h,F_h, ...
          a_1_h,b_1_h,c_1_h,a_2_h,b_2_h,c_2_h);

% Relative Humidity at state 2

rh_2_o = p2 /(10^5*10^(As-(Bs/(T2_o+Cs)))*(1+pr));
rh_2_h = p2 /(10^5*10^(As-(Bs/(T2_h+Cs)))*(1+pr));

%% T2 Function
function[T2]=T2(T1,N,p1,p2,R,M,A,B,C,D,E,F,a_1,b_1,c_1,a_2,b_2,c_2)
k_1=a_1/R;
%k_2=b_1/R;
k_3=c_1/R;
k_4=(a_1^2)/(R^2);
k_5=(2*a_1*b_1)/R^2;
%k_6=(2*a_1*c_1+b_1^2-a_2)/R^2;
k_7=(b_2-2*b_1*c_1)/R^2;
k_8=((c_2-c_1^2))/R^2;


j_1=D/(4*M);
j_2=C/(3*M)+R*k_4*p2^2;
j_3=B/(2*M)+R*0.5*k_5*p2^2-R*k_1*p2;
j_4=A/M;

h1= 1/M*((A*T1) + (B*0.5*T1^2) + (C*T1^3/3)...
    + (D*T1^4/4) - (E/T1) + F)...
    +R* ((k_3-k_1*T1^2)*p1 + (2*k_4*T1^3 + ...
    k_5*T1^2 + k_7 + (2*k_8/T1)) * (p1^2/2));

j_5=F/M+R*(k_3*p2+p2^2*k_7/2)-h1;
j_6=R*k_8*p2^2-E/M;


pol=[j_1 j_2 j_3 j_4 j_5 j_6];
T2_all=roots(pol);

T2_all(find(imag(T2_all)~=0))=-1000;
T2_all(find(T2_all < 0))=-1000;
T2_all(find(T2_all > 400))=-1000;
T2=max(T2_all);
end
```

Appendix C – Orbital Decay Matlab Code

```
%% Intro
%The following calculation uses the atmospheric density data provided in
%the ECSS-E-ST-10-04C to calculate the drag and the resulting orbital
%decay of a 6U CubeSat. To include the variation in solar flux and
%geomagnetic activity, the orbital decay is calculated three times (first
%loop with j=1,2,3)  with different density values given by the NRLMSISE-
%00 model.In the first loop (j=1) the density is given for a low solar
%and geomagnetic activity (F10.7=F10.7_avg=65, Ap=0), in the second (j=2)
%for mean activity(F10.7=F10.7_avg=140, Ap=15) and in the third loop %(j=3)
for high activity(F10.7=F10.7_avg=250, Ap=45). The density values %of the
ECSS standard are stored in a seperate Excel file %("NRLMSISE_00.xlsx")
that is invoked in at the beginning

%% Calculation
data = xlsread('NRLMSISE_00.xlsx');      %ECSS density values
j=1;                         %loop parameter
for j=1:3
    % Input values

    m_s = 12                 %Satellite Mass[kg]
%Satellite effective surface (Average surface exposed to atmosphere)
[m^2]:
    A_s = 0.5 * ((0.2*0.3)+(0.2*0.1)+(0.3*0.1));
    Cd = 2;                  % Satellite Drag Coefficient
    R_e = 6378000;           % [m] earth's radius
    M_e = 5.98*10^24;        % [kg] earth's mass
    G = 6.67*10^-11;         % [m^3/(kg*s^2)]Universal constant of gravity
    dt =0.1 * 3600 * 24;     % [s]time increment

    i=1;                     %loop parameter
    clear H R P t;
    H(1) = 700*10^3;                                   %[m] Satellites Altitude
                                                       %max=900 km
    R(1) = R_e + H(1);                                 % [m]Orbital Radius
    P(1) = 2 * pi * sqrt((R(1)^3)/(M_e*G));            % [s] Orbital period
    t(1) =0;
    while (H(i) > 100*10^3) & (H(i)<=H(1))

        [rho] = atmosphere(H(i),j,data);               %Density at current orbit
                                                       %altitude
        dP(i) = - 3 * pi * R(i) * rho * ...            %Change of Orbital period
           A_s * Cd / m_s  * dt;
        P(i+1) = P(i) + dP(i);                          %Reduced orbital period
        R(i+1) = (P(i+1)^2 * G *M_e /(4 * pi^2))^(1/3);
        H(i+1) = R(i+1) - R_e;
        t(i+1) = t(i) + dt;
        i = i + 1;
    end
%Lifetime calculation:
    if j==1
        lifetime_low=fliplr(t)/(3600 * 24 * 365.25);
        Altitude_low=H/1000;
    elseif j==2
        lifetime_mean=fliplr(t)/(3600 * 24 * 365.25);
        Altitude_mean=H/1000;
    elseif j==3
        lifetime_high=fliplr(t)/(3600 * 24 * 365.25);
        Altitude_high=H/1000;
    end
end
```

```
%% Plot
figure
% semilogy(Altitude,lifetime);
semilogy(Altitude_low,lifetime_low);
hold on;
semilogy(Altitude_mean,lifetime_mean);
semilogy(Altitude_high,lifetime_high);
% semilogy(Altitude,max);
xlabel('Altitude [km]');
ylabel('lifetime [years]');
semilogy([100 700],[25 25],'k--');
semilogy([100 700],[5 5],'k--');
legend('Lifetime - Low activity','Lifetime - Mean activity','Lifetime -
High activity','25 & 5 years limit')
axis([100 700 0.01 inf]);


%% NRLMSISE-00 density (max.=900km) from ECSS-E-ST-10-04C

% Function "atmosphere" calculates the density at the satellites current
% height

function [rho]= atmosphere(H,j,data)
    H_km = H/1000;
    alt_dis = data(:,1);

    if j==1
        rho_dis = data(:,2);        %Low solar and geomagnetic activity
    elseif j==2
        rho_dis = data(:,3);        %Mean solar and geomagnetic activity
    elseif j==3
        rho_dis = data(:,4);        %High solar and geomagnetic activity
    else
        error('j=/1,2,3')
    end

    % Interpolation
    rho = interp1(alt_dis,rho_dis,H_km);
end
```