μHoubolt Technical Report

Team 25 Technical Report to the 2022 EuRoC

TU Wien Space Team

2022

Contents

1 Abstract

μHoubolt is a bi-liquid propelled rocket project by the TU Wien Space Team. Designed and manufactured almost exclusively in-house, with the needed Know-How slowly built up over precursor projects and the projects life cycle, the team has managed to create a robust and lightweight rocket together with the necessary Testing and Ground Support Equipment to safely operate it.

The rocket is designed to fly to an altitude of 3 km - thus being in the L3 launch category - using its engine powered by ethanol and liquefied nitrous oxide and to then safely return to the ground thanks to a two stage recovery system. The engine design has been iterated upon over years with many improvements on every part from the injector over propellant feed pressures and igniters to combustion chamber concepts. Throughout the entire project special care has been put into safety, ranging from refining checklists to be as clear as possible, over a remote controlled oxidizer loading system to bleed orifices and burst discs for passive depressurization. As liquid rockets need a lot more and more complex Ground Support Equipment than a typical solid propelled rocket, considerable amount of time has also been invested in simple and easy to use Mission Control software that can control both the rocket while on the pad and the GSE.

The payload flown will be the EDU subsystem from the STS1 (SpaceTeamSat1) project from Space Team. This gives the cubesat project the opportunity to flight test parts of their hardware.

2 Introduction

TU Wien Space Team is a student organization engaging in various projects in aerospace engineering. Our mission statement is to foster the Know-How and enthusiasm for aerospace technologies in our peers by providing an accessible entry into rocketry and giving members the opportunity to learn. The team is working on a number of different projects ranging from solid propelled two staged rockets, which can reach the edge of space to hydrogen powered autonomous airplanes and from cubesats to rockets with liquid propulsion systems.

The μHoubolt project has its origin as a small scale technology demonstrator for our large rocket concept, Houbolt. The much reduced scale has allowed for faster and cheaper iteration in development and testing and has given us the opportunity to gather experience and work on our testing equipment and ground infrastructure. The goal of the project is to build and launch the first bi-liquid propelled rocket of the Space Team. The design should provide a solid foundation which can be further developed and used in more advanced projects. If regulatory limitations allow it, we would like to make the design open-source after a successful launch at EuRoC.

Our mission objectives are as follows:

- Flight to target altitude of 3 km
- Successful 2-stage recovery of vehicle
- Working telemetry and data recording during the whole mission
- Secondary mission: test platform for the CubeSat project STS1 from Space Team

2.1 Academic Program

The TU Wien Space Team is a student association with about 250 members. Due to the affiliation to the TU Wien and the technical focus of the projects, the team consists largely of students from the TU Wien and the fields of study Mechanical Engineering, Electrical Engineering, Technical Physics and Computer Science. However, efforts are being made to make the team more diverse and to include students from other universities and fields of study. As a result, we now have students from e.g. the University of Vienna, the Vienna University of Economics and Business Administration, the University of Applied Sciences St. Pölten, the University of Applied Arts Vienna and University of Applied Sciences Wiener Neustadt in our team.

2.2 Stakeholders

Apart from universities, which gain visibility and reputation through TU Wien Space Team, and students, who have the opportunity to deepen their knowledge and apply it in practice, there are other parties that are essential for the organisation. Industry sponsors provide resources and mentor the team members, for example by sharing manufacturing techniques or holding workshops. In return, the project provides visibility both internally to highly motivated students with practical experience and with that to possible future employees and externally to potential customers.

2.3 Team Structure

TU Wien Space Team is divided into eight active projects:

- The Hound: Two staged solid rocket slated to reach the Karman Line
- μHoubolt : Bi-liquid propelled rocket
- SpaceTeamSat1: Fully in-house developed and built cubesat
- GATE: Large scale liquid propellant engine
- AcrossAustria: Hydrogen-powered long range autonomous drone
- Penrose: Hybrid rocket
- CanSat: Space Team as launch provider for student-built can-sized satellites
- FIRST: Onboarding project for new Space Team members as an introduction to rocketry

In addition, there are organisational and cross-project departments, those being IT, HR, finance and marketing.

2.3.1 μHoubolt Team Structure

Currently, about 25-30 people are working on project μHoubolt . Although the structures are flat and exchange across the team is encouraged, there are clearly assigned tasks and module leads who are responsible for ensuring that the goals in their respective areas – these being Recovery, Propulsion, Avionics, Ground Systems, Aerostructure and Software – are achieved. Additionally, internal module meetings are held weekly to assign and discuss tasks.

The workload of the team members is rather diverse: while a minimum of 5 hours per week is expected of everyone, active members devote all their available time into the project at stressful phases. Being able to choose the effort so freely allows us to have a diverse team, whereby the willingness to put time and energy into the project is naturally reflected in the weight of the tasks and the position in the team structure. Apart from that, the team can draw on the experience of former members, who are consulted when questions arise, that have already been dealt with in previous projects.

In addition to the members who work on the rocket directly, there are people within the team who take care of the project's presence on social media, photography and legal matters.

2.4 Technical Challenges

A considerable challenge was to not only build a liquid propulsion system from the ground up, but to also make it as small and lightweight as we did. Eliminating parts is always good for eliminating risks and reducing weight is obviously a positive impact for rocket performance, but that is easier said than done. Additionally we've put a lot of effort into the safety of our design, both by designing systems that are easy to operate and having as little interaction with hardware that is in a potentially harmful state as possible by remote controlling large parts of the ground support equipment. A compounding factor for all this is that the entire software stack has been designed and developed completely in-house without using external software solutions (excluding CAN driver and the used Operating Systems) - the same goes for mechanical components, which we tried to manufacture in our own workshop safe for the aluminium propellant tanks, some of the plumbing & valves and the main carbon fiber body tube.

2.5 Vision

The team's vision is that μHoubolt will become a benchmark for bi-liquid rockets in the field of student rocketry. The entire project should go open source, and thus provide the foundation for further development. The advantages μHoubolt has already had for us – fast and cost-efficient iteration thanks to small scale and well thought-out designs – should be of advantage to future rocket enthusiasts.

3 System Architecture

3.1 Overview

3.1.1 Physical Architecture

Figure 3.1 shows a rendering of the μHoubolt CAD model. The vehicle consists of the airframe, the propulsion system, the recovery system and avionics. The propulsion system structure, which also contains the avionics components, is self-contained and can be slid into the airframe from the bottom. The recovery system is mounted to the top of the airframe. The custom pressure bearing components were validated using FEM simulations as well as pressure proof tests to 1.5x of maximum pressure.

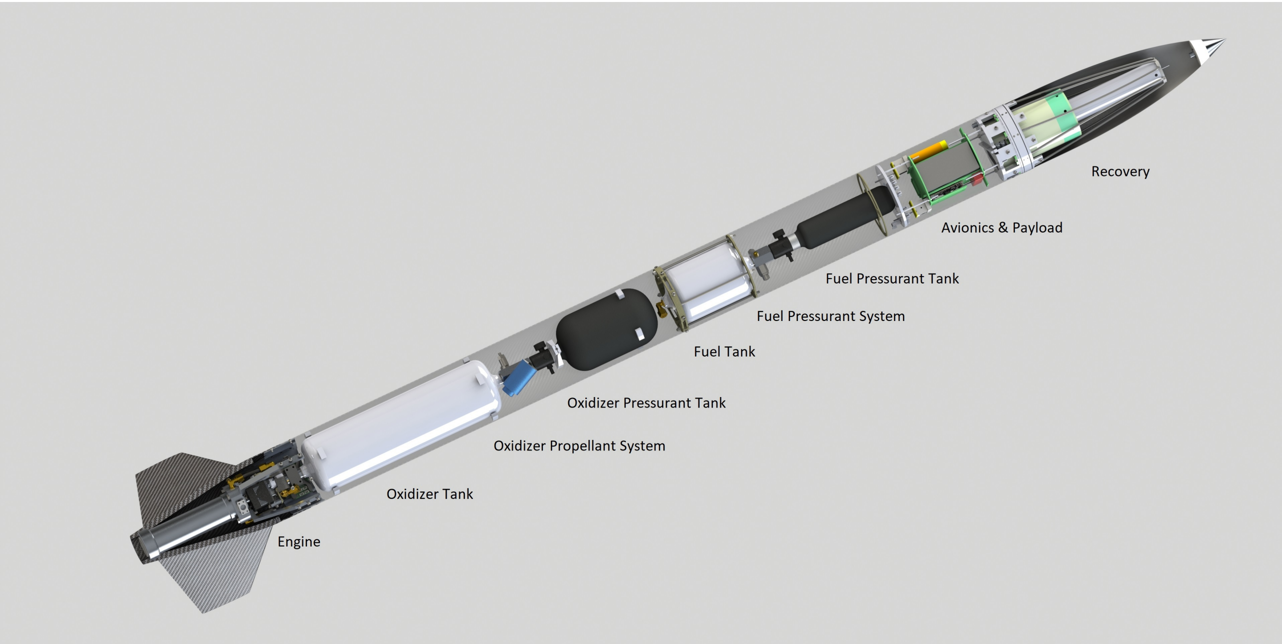


Figure 3.1: μHoubolt physical architecture

3.1.2 Fluid System

Visible here is the PnID for the complete fluid system. This includes the rocket itself, as well as the Ground Support Equipment which encompasses the Pressurant and Ox Fill System. The three connections between the rocket and the GSE are quick connect fittings which are connected to a movable arm to automatically disconnect them shortly before launch. See section 3.7 for more info on the GSE and section 3.7.5 for the umbilicals specifically.

3.1.3 Mass Budget

Table 3.1 show the current mass budget for μHoubolt .

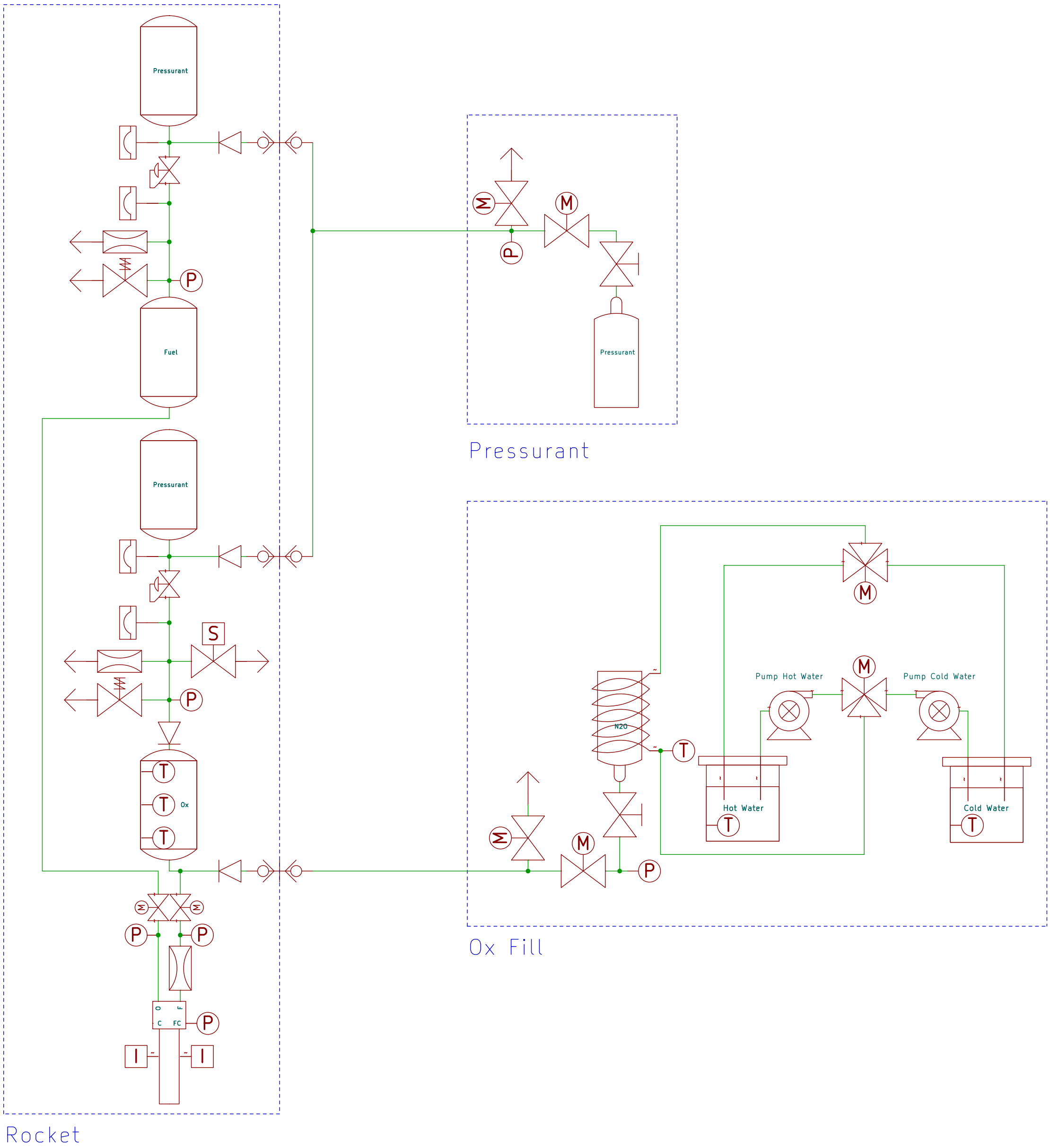


Figure 3.2: μHoubolt Piping and Instrumentation Diagram (PnID)

Component	Mass in g
Airframe	1900
Recovery	800
Avionics	400
Payload	1000
Oxidizer System	2600
Fuel System	1600
Engine Section	650
Total Mass Dry	**8950**
Oxidizer	2100
Fuel	700
Pressurant	290
Total Mass Wet	**12040**

Table 3.1: General System Data

3.1.4 Flight Simulation

A flight simulation was performed with OpenRocket [4], using a model of the vehicle closely matched to the CAD model and mass budget. The OpenRocket engine configuration (representing the whole propulsion system) was generated using ORLEG [1], which, in addition to changes in mass, also incorporates changes of the center of gravity location due to emptying propellant tanks and flowing pressurant gas as well as the effect of the changing ambient pressure on the engine performance. Assuming medium wind speeds and a launch downwind, the simulation yields the following results:

- Apogee: 3558 m
- Maximum velocity: $302\,\mathrm{m\,s^{-1}}$ (Mach 0.91)
- Maximum acceleration: $51.5\,\mathrm{m\,s^{-2}}$
- Flight time to apogee: 27.8 s
- Total flight time: 217 s

Assuming an effective launch rail length of 6 m (total length of 7 m minus the distance between the rail buttons of about 1 m) the vehicle has a velocity of $22.2\,\mathrm{m\,s^{-1}}$ when leaving the rail and is aerodynamically stable.

3.2 Propulsion

Thrust is provided by a SRAD pressure fed bi-propellant liquid propulsion system. It utilizes nitrous oxide and ethanol as propellants and is pressurized by two nitrogen tanks through mechanical pressure regulators. The system is optimized for simplicity, low mass and small size. It can roughly be divided into three main subsystems which are described in detail below: the propellant tanks with piping, the pressurization systems and the engine with its valves and fill connections. A diagram of the whole fluid system, including the connected ground systems, can be found in Section 3.1.2.

The entire propulsion system, which is shown in Figure 3.3 is assembled separately from the rest of the vehicle and can be tested standalone without it. It is installed into the airframe by sliding it into the body tube from the rear as one integrated component, only requiring two electrical connectors to be plugged in and a few screws to be installed.

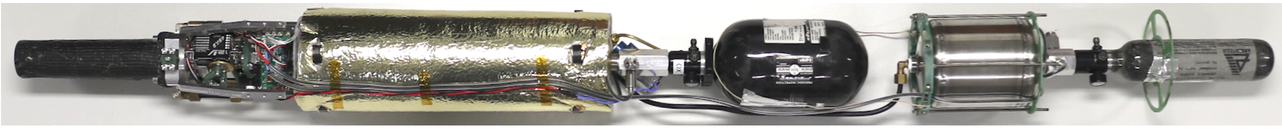


Figure 3.3: Complete propulsion system, ready to be installed into the airframe

3.2.1 General Parameters

The thrust is chosen to be 600 N at liftoff as a trade-off between sufficient launch acceleration for aerodynamic stability and the capacity of the engine testing infrastructure. The chamber pressure and oxidizer/fuel mass flow ratio (O/F ratio) is chosen as a trade off between engine efficiency, combustion temperature, propellant mass flows and feed pressures. As a compromise between high efficiency, low heat transfer into the thrust chamber and low feed pressures, a chamber pressure of 15 bar is chosen. Figure 3.4 shows various engine parameters as a function of the O/F ratio.

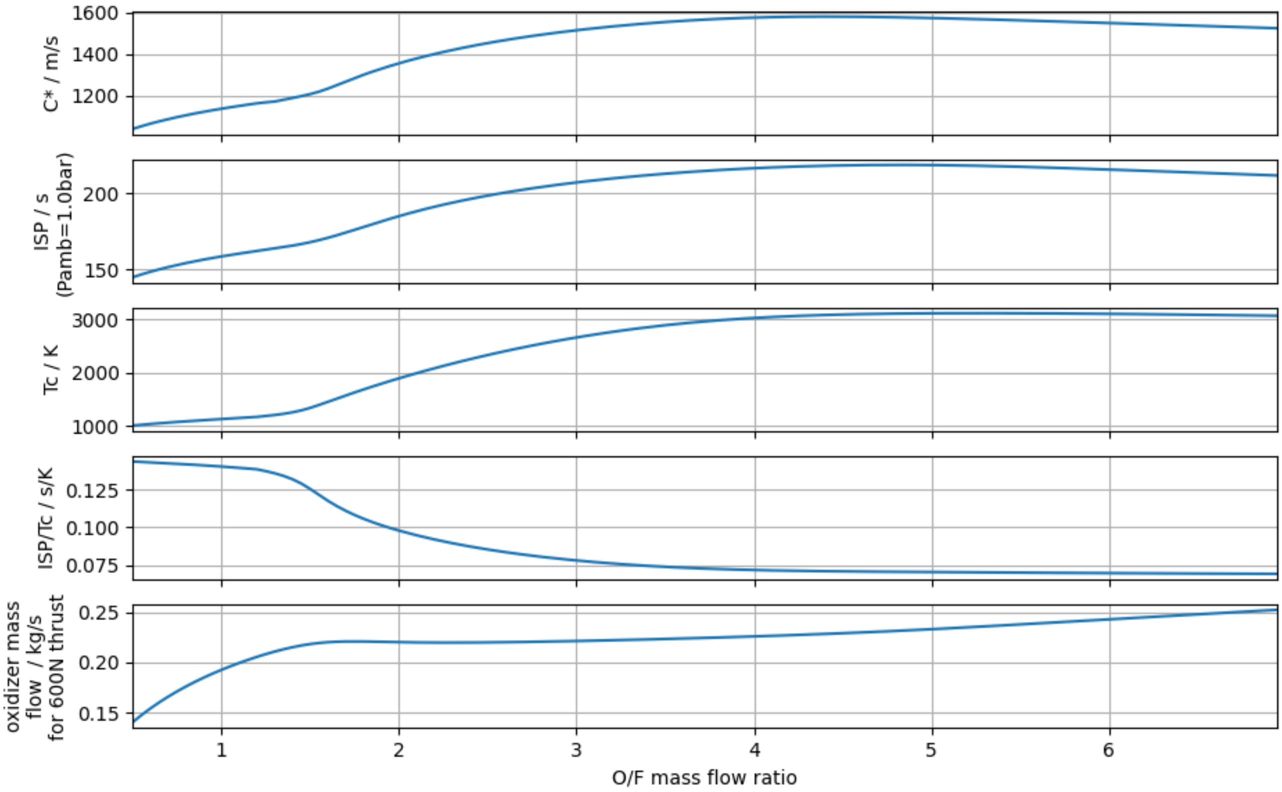


Figure 3.4: Ideal engine performance parameters as a function of the oxidizer/fuel mass flow ratio at a chamber pressure of 15 bar

Since the oxidizer mass flow rate is nearly constant at ratios above 1.5 and smaller ratios lead to unacceptable efficiency, this metric can be ignored. The fuel mass flow rate is not plotted but favors high ratios. The efficiency to combustion temperature ratio is highest at low O/F ratios but the efficiency is estimated to impact the vehicle mass more than the combustion temperature so a higher mass flow ratio of 3.0 is chosen based on this data.

As the vehicle will not reach high altitudes, the expansion ratio is sized for an ambient air pressure of 1000 mbar. When assuming a 93 % C* and ISP efficiency (based on previous engine testing data) and a thrust of 600 N, the following engine parameters were calculated using the self developed Python program ORLEG [1] which uses the CEA [2] Python wrapper RocketCEA [3]:

- throat diameter: 19 mm
- nozzle exit diameter: 32 mm
- nozzle expansion ratio: 2.7
- combustion temperature: 2632 K
- c*: 1405 $\mathrm{m\,s^{-1}}$
- ISP: 192 s
- total mass flow rate: 318 $\mathrm{g\,s^{-1}}$
- fuel mass flow rate: 79 $\mathrm{g\,s^{-1}}$
- oxidizer mass flow rate: 238 $\mathrm{g\,s^{-1}}$

Considering the complete vehicle design, a burn time of about 8 s is necessary to reach the intended target altitude of 3 km. The fuel tank is sized to a volume of 900 mL and the oxidizer tank to 2400 mL, fitting about 700 g of fuel at ambient temperature and 2085 g of oxidizer at a temperature of 0 °C (see section 3.2.5.2) and an assumed fill level of 95% due to boil-off between filling and launch. This gives a maximum burn time of up to 8.8 s, when a constant thrust of 600 N is assumed. As the tank pressures and therefore mass flow rates and thrust drop off during the burn due to non-ideal pressure regulators, the burn time increases slightly.

3.2.2 Propellant Tanks and Piping

Figure 3.5: Aluminium propellant tanks

The propellant tanks, shown in Figure 3.5 are manufactured from aluminium in three parts each that are welded together, the two end-caps with threaded connections and a tube in between. They were designed by the team but are one of the few components manufactured externally. This allowed for a more optimized end-cap design to be manufactured and additionally the welds were done by a professional welder, assuring consistency.

Sharing the end-cap design, the fuel and oxidizer tanks have a diameter of 101 mm. A length of 192 mm for fuel and 387 mm for oxidizer gives the tanks a volume of 900 mL and 2400 mL and a mass of 594 g and 1170 g.

To minimize mass, a heat treated high strength aluminium alloy is used, which at the same time has to be weldable. The choice fell on EN AW-6082. As there were uncertainties regarding the strength of the welded joints, which is crucial for this application, representative samples of the material were cut in half and welded back together by the designated tank manufacturer using the process planned for the tanks. After heat treatment for hardening, the samples were professionally tested for tensile strength, yielding good results. Based on those, the tanks were designed for a burst pressure of 125 bar and manufactured before receiving the same heat treatment as the samples.

The maximum nominal operating pressure of the tanks is about 50 bar for oxidizer and 40 bar for fuel. As under some circumstances, the tanks could be exposed to up to the 60 bar set pressure of the safety valves, the pressure used for proof testing was based on this value. With a 50% safety factor, the tanks were hydrostatically tested at 90 bar for an extended amount of time, without showing leaks or other abnormalities. The detailed test report can be found in Section 6.2.6.

The oxidizer tank is axially supported by its connection to the engine assembly via the oxidizer tube and in turn axially supports the oxidizer pressurant system. Radial support against the body tube is provided by 3D-printed plastic spacers. As the pressurant fill connection needs to be aligned with the other fill connections, the connection between the oxidizer tube and the oxidizer valve (which is rigidly mounted to the engine) can be mounted with arbitrary rotation, as described in Section 3.2.4.1. To reduce heat flow into the tank, which causes the liquid oxidizer to boil off, the tank is wrapped in thermal insulation.

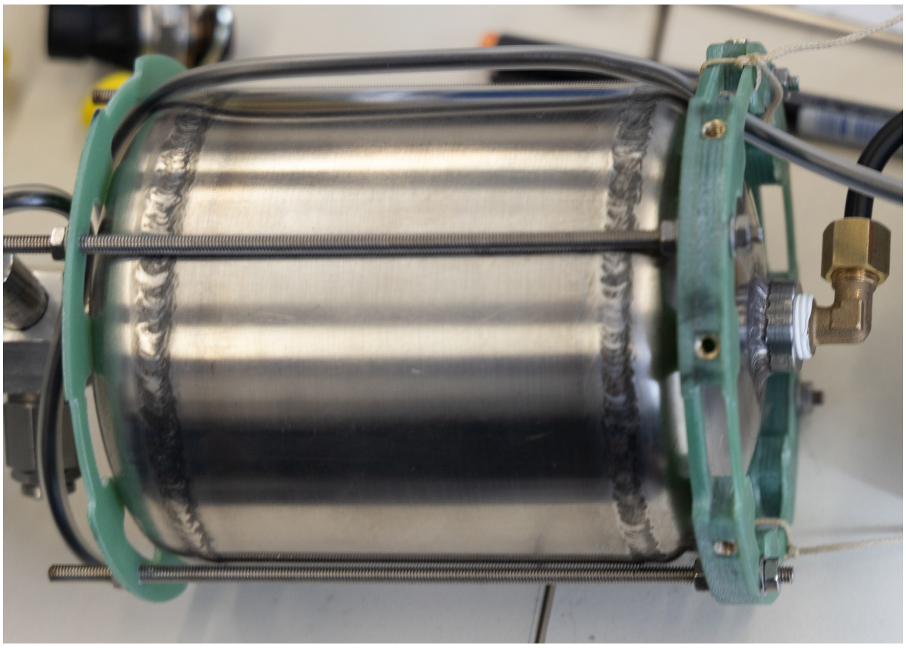


Figure 3.6: Steel Prototype Fuel Tank in its mounting cage

The fuel tank also supports the fuel pressurization system but it is not rigidly connected to the rest of the propulsion system. Instead, the fuel tank is clamped in a cage made of GFRP plates and threaded rods (shown in Figure 3.6), which in turn is attached to the body tube via radial screws. During installation of the propulsion system into the body tube from the rear, the fuel system is pushed in by the rest of the propulsion system. It can then be manipulated from the front to seat the radial screws. As the fuel tank can be rotated within its mounting cage with a bit of force, the fuel systems rotation can then be finely adjusted to align the pressurant fill connection with the associated opening in the body tube. While the process may sound tedious, it is actually pretty simple and quick. To avoid the fuel system getting pulled out by the flexible fuel tube connecting it to the engine when pulling out the propulsion system, a piece of string connects the two subsystems and removes any tension from the fuel tube.

The connection between the fuel tank and engine is made with an aluminium-poliamide composite tube with an outside diameter of 6 mm and a wall thickness of 1 mm. It is flexible, light weight and compatible with standard compression ring fittings, making it well suited for this application. While the pressure rating given by the manufacturer is not sufficient for the application, it is given with a higher than needed safety factor, and the tube was successfully pressure tested to twice its maximum operating pressure.

3.2.3 Pressurization System

The fuel and oxidizer tanks are pressurized by nitrogen from two COPVs via mechanical pressure regulators. The COPVs with max. pressure of 300 bar and volumes of 800 mL and 250 mL for the fuel and oxidizer systems as well as the pressure regulators are COTS components, originally intended for pressurizing Paintball equipment.

The regulators exhibit less than ideal performance, with the fuel pressure dropping from 40 bar static pressure to 30 bar at startup due to the mass flow demand and then progressively down to 25 bar during the burn due to decreasing pressurant tank pressure. The oxidizer regulator behaves similarly with a pressure drop from 50 bar static to 40 bar at startup and down to 35 bar during the burn. While this behaviour is far from ideal and additionally varies a bit from run to run, it is good enough for this vehicle. The additional influence of the acceleration on the pressure differential between the top of the propellant tanks and the injector can be ignored due to the vehicles small size.

Active pressure control using electrically actuated valves and a software control loop was evaluated with good preliminary results but got abandoned in favor of the simple solution using mechanical regulators. For larger vehicles or more stringent pressure accuracy requirements, active control would be the preferred method though.

No pressurization valves are used, so the propellant tanks get pressurized as soon as the pressurant tanks get filled (the regulators include fill connections and check valves), which (apart from a partial pre-pressurization before oxidizer filling) only happens shortly before liftoff, see Section 3.2.5.3.

(a) Fuel (b) Oxidizer (c) Oxidizer

Figure 3.7: Pressurization manifolds with connected components

The output sides of the pressure regulators are screwed into the top of aluminium manifolds, which connects multiple components together. In case of the oxidizer, an inline check valve is integrated inside the manifold after the regulator to prevent any oxidizer flowing back into the pressurization system, this is not needed for the fuel side. The bottom side of the manifolds is screwed into the top of the propellant tanks. Connected on the sides of each manifold are a tank pressure sensor, a safety valve and a bleed orifice. The assemblies are shown in fig. 3.7.

The safety valves are set to their maximum of 60 bar, well under the tanks proof and burst pressures. The bleed orifice consists of a 0.1 mm 3D-printing nozzle that is used to slowly depressurize the system over time. This helps to safe the vehicle in case of a complete avionics failure and does not negatively impact normal operations.

On the oxidizer manifold, a solenoid vent valve is installed as well, to allow for active regulation of the tank pressure and thus oxidizer temperature during and after filling, see Section 3.2.5.2. The output of the solenoid valve flows through a tube to an opening in the body tube, to not cause pressure spikes within the airframe that could trigger the recovery electronics, and to make the vented plume visible, which helps recognize the completion of the filling process.

The regulators include two burst discs, one with a 517 bar burst pressure to protect the pressurant bottle and one with a 124 bar burst pressure on the output. Since this output burst disc's burst pressure is much greater than the opening pressure of the safety valves, this burst disc is not expected to be needed unless the safety valves fail and an overpressure event is caused. In case of the oxidizer system, the burst disc can only protect against overpressure caused by a regulator failure, as overpressure originating from the oxidizer tank can't flow back to the burst disc through the check valve.

Both pressurization system assemblies are axially supported by the connection to their respective propellant tank. Radial support against the body tube for the pressurant tanks is provided by a GFRP spacer plate for the fuel tank and 3D-printed plastic spacers for the oxidizer tank.

3.2.4 Engine, Valves and Fill Connections

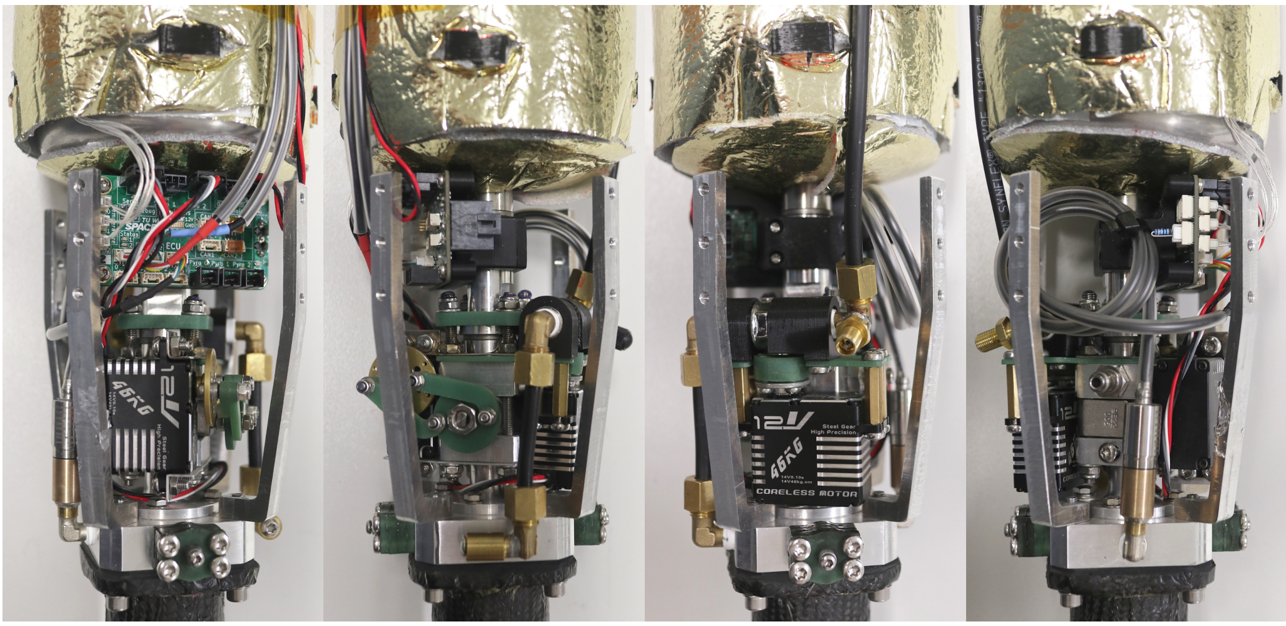


Figure 3.8: Engine assembly, shown from multiple sides

The engine assembly, shown in Figure 3.8, consists of the engine with injector and thrust chamber, the main propellant valves with actuators, a redundant integrated ignition system, propellant fill connections, pressure sensors, propulsion avionics and structural components. The assembly features a high integration density and low mass, which is achieved mainly by the use of a semi-custom oxidizer valve, custom manifold components and fill connections with integrated check valves. The avionics are described in detail in Section 3.6, the other components in this section.

The central component of the assembly is the aluminium engine head. It sits atop the thrust chamber (Section 3.2.4.4) and is mounted to the airframe (Section 3.3.2.3) via four aluminium struts that transfer most of the thrust force during powered ascent and carry the engine and oxidizer assemblies during descent. On the four sides of the engine head, the ignition system (Section 3.2.4.3) is integrated, as are the fuel connection, a venturi for fuel flow regulation (Section 3.2.4.2) and a chamber pressure sensor connection via an elbow adapter.

Mounted to the top of the engine head is the injector (Section 3.2.4.2). Together they form an internal annular volume which is used for distributing fuel to the fuel injection orifices. The injector doubles as a part of the main oxidizer valve, whose other components are mounted on top. The propulsion avionics are mounted to the oxidizer tube above this valve, which doubles as a structural component. The main fuel valve and the actuators for both valves sit to the sides of the oxidizer valve assembly and are mounted to it.

3.2.4.1 Main Valves and Fill Connections

The main fuel valve is a small COTS ball valve, mounted to the side of the oxidizer valve. An elbow fitting connects the valve to a short tube going down into the engine head. The associated fitting on the engine head also includes a connection to attach an injector pressure sensor. While the small sensors used in other parts of the system would fit here, this connection is only used for ground testing with larger sensors (that only fit with the Fincan removed), as the availability of the small type is limited and the measurement is not needed for flight. Another elbow fitting connecting to the pipe going up to the fuel tank also has the fuel fill connection attached. It consists of a repurposed tyre fill connector with integrated check valve. While normally used for pressures below 10 bar, testing has shown it to still work reliably at pressures of 60 bar or higher. Connecting it close to the valve ensures that the tube leading up to the fuel tank is always filled with fuel and doesn't trap any air, assuring consistent engine startup behaviour. The valve is actuated by a standard size high torque RC servo motor mounted co-axially.

The main oxidizer valve consists of the body and inner components of a COTS Swagelok 3-piece ball valve (with upstream vented ball) in combination with custom aluminium flanges. The lower flange is part of the injector, avoiding an additional connections and setting the orientation of the valve assembly and all connections mounted to it while also saving space and mass. The component also includes a connection to attach a pressure sensor, which – for the same reason described for the fuel side in the previous paragraph – is currently only used for ground testing. The upper flange includes the connection for filling the oxidizer tank, which uses a repurposed Paintball fill connector with included check valve. It is rated for up to 300 bar and is – apart from the o-ring which was replaced by a FKM one – made from materials compatible with nitrous oxide. Apart from making for a very compact assembly, this design also avoids any gas to be trapped between the fill port and valve during filling, which might introduce variability to the startup behaviour. A piece of fine stainless steel mesh is added inside the fill connector to act as a filter. The main valve is actuated by a standard size high torque RC servo motor mounted to the side of the valve, with the servo axle and valve stem coupled via a pair of push/pull rods.

The four threaded rods holding together the oxidizer valve are welded to plates in pairs to lock their rotation, simplifying assembly. One of the plates is also used as a mounting point for the oxidizer valve servo. The rods extend up above the valve and are additionally used to connect the oxidizer tube to the upper flange, with a bonded seal in between. This arrangement allows adjustment of the rotation of the tube and the oxidizer tank and pressurization system connected to it. As all fluid connections above are threaded and sealed with bonded seals, this feature is needed to correctly set the direction of the oxidizer pressurant fill connection to be the same as the oxidizer fill connection.

As similarly light weight pressure regulators with adjustable fill port angle have become available, this feature can be removed when upgrading to one of these regulators. In that case, the extended rods are not needed anymore, which would allow for a simpler design. The upper flange and tube could be made as one part and the oxidizer valve could be held together by four bolts screwed from the top into threaded holes in the lower flange / injector. The oxidizer servo mount would consist of plates screwed to threads on the side of the flanges, and to the servo via threaded spacers. This modification was not made to the EuRoC 2022 design due to time constraints but is recommended for future builds.

3.2.4.2 Injector and Mass Flow Regulation

The aluminium injector, shown in Figure 3.9, is a quadruple unlike-doublet impingement type, with four radial fuel jets impinging with four axial oxidizer jets. It is relatively simple to manufacture and has delivered good reliability and adequate performance in hot fire testing.

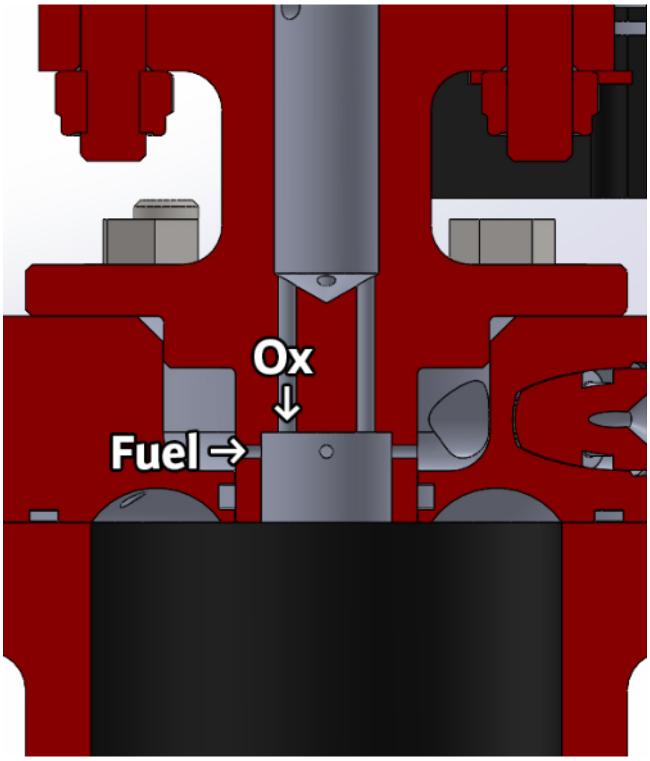


Figure 3.9: Cross section of the injector

The fuel mass flow is regulated by a cavitating venturi integrated into the engine head, which is shown in Figure 3.10. Its throat diameter A_t is 1.4 mm and the experimentally determined discharge coefficient C_d is 0.89. The mass flow through it for a given density ρ, vapor pressure p_{sat} and inlet pressure p is governed by the following equation, as long as the pressure drop is large enough to lead to cavitation at the throat:

$$\dot{m} = C_d * A_t * \sqrt{2 * \rho * (p - p_{sat})} \tag{3.1}$$

The four fuel injector orifices with a diameter of 1 mm are sized large enough to avoid cavitation and thus choking and small enough to cause a high injection velocity for good atomization and mixing.

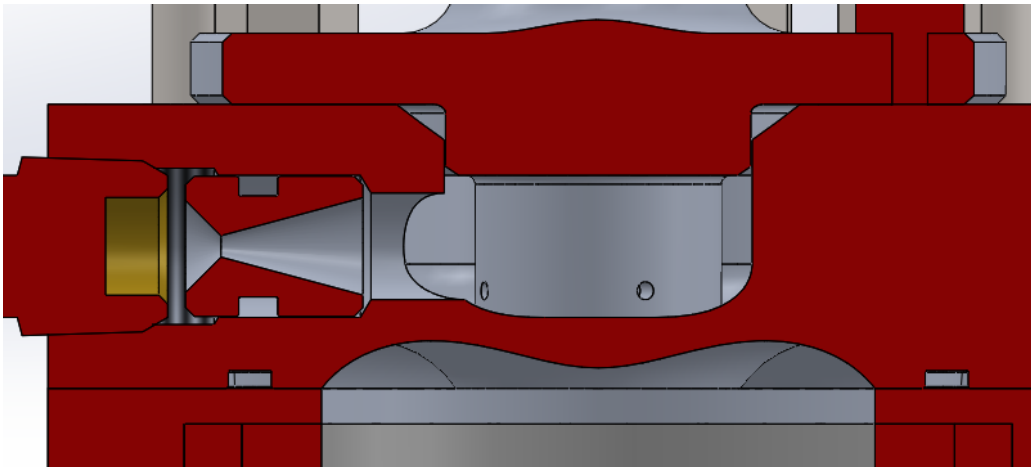


Figure 3.10: Engine cross section showing the fuel cavitating venturi. The brass fitting on the left is simplified.

The liquid oxidizer with its vapor pressure significantly higher than the chamber pressure does not require a separate flow regulation device. Instead, it flashes to steam in the four injection orifices, which have a diameter of 1.6 mm and a length of 10 mm. This setup shows a similar behaviour of the mass flow only depending on the orifice geometry and inlet fluid properties for a sufficient pressure drop, as is shown in [6]. A model (included in [1]) of this flashing flow behaviour was implemented in Python, based on the iterative approach described in [5, L2.4/4.2]. It was experimentally verified to be accurate during cold flow and hot fire testing with carbon dioxide and nitrous oxide, and then used to size the oxidizer injector orifices.

With this approach and fixed orifice geometries, the propellant mass flows depend only on the fluid conditions (i.e. temperatures and pressures) at the engine inlets. The mass flows therefore being decoupled from the chamber pressure makes feed coupled instability unlikely and simplifies the feed system design.

3.2.4.3 Ignition

Ignition is accomplished by a redundant pyrotechnic ignition system integrated into the engine head. While this design is more complex than traditional ”stick through the nozzle” ignition systems, it has proven to be less troublesome, as the igniters can’t be ejected prematurely. Additionally, wiring the igniters to the vehicle avionics is simplified.

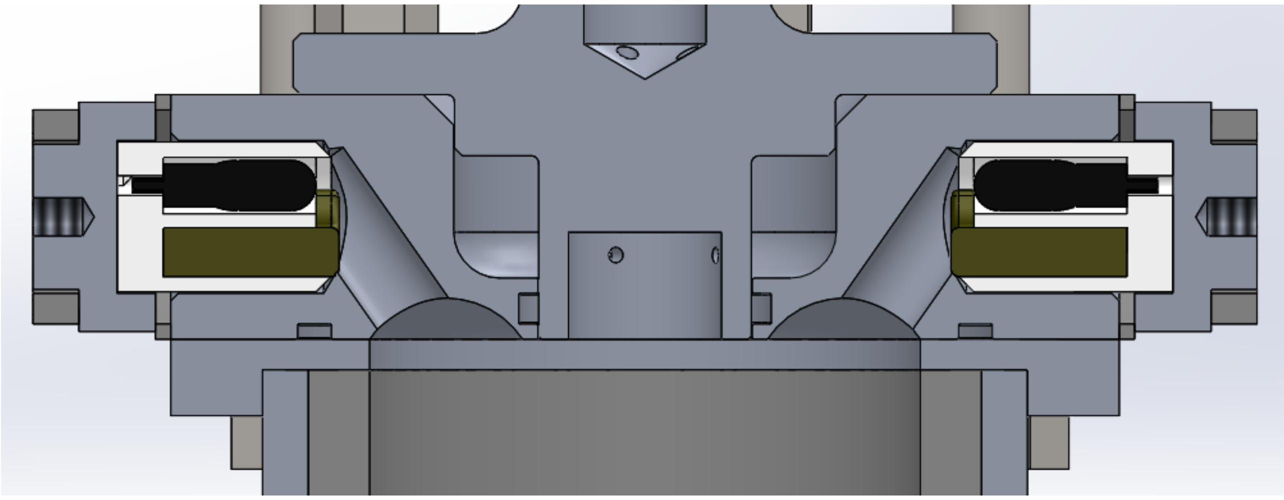


Figure 3.11: Engine cross section showing the ignition system.
cartridges: white, e-matches: black, pyro mixture: brown

Two 3D printed plastic cartridges with a diameter of 10 mm and a length of 12 mm, containing an e-match and about 0.7 g of a mixture of potassium nitrate, sugar and magnesium (i.e. rocket candy with extra sparks) are installed in openings on opposing sides of the engine head, as shown in Figure 3.11. Breeches are mounted on top with four screws each, which not only act as gas tight covers but also as electrical contacts to the cartridges. The positive leads of the two igniter outputs of the ECU are connected to the breeches while the negative leads are connected to the engine head. While this arrangement can be a bit annoying at times due to bad electrical contact between the cartridge and engine head and/or breech, it avoids the need for gas tight wire feedthroughs, which caused problems in past designs.

After the e-matches ignite the mixtures, the igniters burn for roughly 5 s. The ignition flames with hot sparks are led into the combustion chamber by angled holes, meeting in the center, below the injector.

The igniter cartridge installation takes about 10 minutes when done by an experienced team member, and is planned to be done in the prepping area.

3.2.4.4 Thrust Chamber

The ablatively cooled thrust chamber consists of a casing and a liner with a cylindrical section and a converging-diverging nozzle, bots seen in Figure 3.12. The conical nozzle geometry is not the most efficient, but allows for simple manufacturing.

(a) Casing (b) Liner

Figure 3.12: Thrust chamber

The cooling method was chosen as regenerative cooling is not feasible at this scale (fuel mass flow too low compared to heat flux, cooling with oxidizer is considered complex and risky due to possible decomposition) and film cooling is more complex to implement and increases propellant consumption.

The casing is manufactured from aluminium and consists of a tube section with a flange on one end and an external thread on the other. The flange is mounted to the engine head and sealed with an axial o-ring. After installing the liner, which is sealed to the casing using two radial o-rings, it is held in place by a retainer ring screwed onto the casing.

The ablative chamber liner and nozzle is combined into one component and consists of phenolic resin cotton fabric composite. It is turned out of commercially available round stock, making it a simple, affordable and safe option.

Laminating the thrust chamber from carbon fibre, glass fibre or cotton fabric and liquid phenolic resin was the initial plan, but the procurement of the resin proved to be very challenging. This, together with safety concerns lead to the abandonment of this approach. Using epoxy resins as widely available and safe replacement for phenolics was evaluated, but abandoned after repeatedly delivering unsatisfactory results in hot fire testing.

Cost and manufacturing time could be reduced by switching to a two part ablative design, this was however not pursued for the EuRoC 2022 design due to time constraints. Phenolic paper tubes are commercially available with custom dimensions and lower cost than solid phenolic cotton stock. The manufacturing of the chamber liner would then be as simple as cutting it to length. The nozzle could still be turned from phenolic cotton, but a more viable approach might be to use graphite instead. Using a chamber liner with sufficient thickness and a graphite nozzle would allow the assembly to be used for multiple firings of the engine, while the current one-piece phenolic cotton approach is limited by the high regression rate at the throat.

3.2.5 Propellant Loading and Offloading

3.2.5.1 Fuel

Fuel filling is done manually using a big syringe with a tyre fill adapter connected via a hose, allowing for a precise volume of fuel to be added. The fill adapter opens the check valve completely, allowing the fuel tank to also be drained using the same method. As the ullage gas in the tank needs to flow through the small bleed orifice during filling and draining, significant force on the syringe is needed to finish quickly.

3.2.5.2 Oxidizer

Connected to the fill port is the remotely controlled oxidizer fill system described in section 3.7.3. Oxidizer loading starts by activating the vent valve pressure regulation and setting it to 30 bar. Pressurant is then carefully filled until the vent valve gets activated, after which pressurant filling is stopped. The oxidizer tank is now filled with nitrogen at 30 bar. Now, the oxidizer fill valve is opened, connecting the oxidizer bottle (cooled down to about 0 °C to reach a vapor pressure of approximately 30 bar) to the fill port on the vehicle. The pre-pressurization with nitrogen prevents oxidizer from quickly rushing in at this moment, which would be a safety hazard considering the possibility of the nitrous oxide explosively decomposing. Also, the gas phase of the oxidizer in the tank is diluted by the nitrogen, further lowering the risk of a decomposition event. Next, heat is introduced into the oxidizer bottle, leading to a slight rise in vapor pressure. This, in combination with the vent solenoid valve still regulating the tank pressure to about 30 bar (and thus 0 °C) leads to liquid oxidizer flowing from the bottle into the tank. The displaced gas in the tank gets vented while the heat used in the bottle for evaporation gets replenished by the external heat source. The completion of the filling process is seen by liquid oxidizer being vented, creating a visible plume, and by the venting frequency changing suddenly. The filling process is stopped by closing the fill valve and deactivating the bottle heating. If desired, the tank can be topped off by opening the fill valve again (and enabling the bottle heating if necessary), but this is rarely needed as the tank is insulated well and the boil-off rate is thus quite low.

During filling and until pressurization, the oxidizer vapor pressure is regulated to 30bar by the solenoid vent valve. The valve is then kept closed and the pressurant tank filled. The regulator is set to provide a static pressure of 50bar, which drops to 40bar operating pressure as soon as the main valve opens. This active pressurization (“supercharging”) speeds up the launch preparations and prevents cavitation in the feed system.

3.2.5.3 Pressurant

Pressurant in the form of nitrogen at up to 300 bar is loaded via the fill ports integrated into the Paintball pressure regulators. During normal flight preparation, the tanks are first partially filled to aid in the oxidizer fill procedure as described above. As the pressurant fill system is shared between the fuel and oxidizer systems, the fuel pressurant tank also gets partially filled in this step. This does not matter though and the system depressurizes within a few minutes through the bleed orifice.

Shortly before liftoff, the pressurant tanks are fully filled after the oxidizer vent solenoid gets switched from pressure control to closed. This is done by opening the fill valve before waiting for the propellant tank pressures to stabilize and then some more to ensure the pressurant tanks are completely filled. After finishing the pressurant fill process by closing the fill valve, liftoff should occur soon to avoid too much pressurant escaping via the bleed orifices. If there is a delay, pressurant can be topped off as long as the fill umbilicals have not yet been disconnected.

3.3 Aerostructure

The airframe consists of three main parts: the nosecone, the body tube and the fincan. These are connected by aluminium coupler rings, whereby the one that connects the nosecone and body tube is described in detail in the recovery section (section 3.4). Most of the parts are made in-house, so besides the idea behind the design, the process of manufacturing is outlined.

3.3.1 Nosecone

Based on the Von Kármán shape, the nosecone (shown in Figure 3.13) optimizes the airflow around the rocket and minimizes drag. Due to being made of glass fiber, it is transparent to electromagnetic radiation, making it the RF window of the rocket.

The glass fiber mats were wet laminated to a positive 3D-printed mold with epoxy resin in three layers. The surface was then sanded and touched up evenly in several passes before the 3d-printed core was removed. This 360 mm long center part is enclosed by the 42.25 mm lathed aluminum tip that is screwed into a mounting piece at the top and a aluminium coupler ring serving as a recovery separation mechanism connecting it to the body tube at the bottom, which is described in more detail at section 3.4.

Figure 3.13: Different versions of the nosecone, improvements from back to front (final version). The black paint was removed after a test flight and replaced with a white coat.

3.3.2 Body Tube

The body tube has an outer diameter of 123 mm and a wall thickness of 1.5 mm. It consists of radially wound carbon fiber inner layers and weaved outer layers and was manufactured to spec externally.

3.3.2.1 Umbilical Feedthroughs

To make fueling, arming and setting up pad-communication convenient, easily accessible connectors and mechanisms are necessary while the rocket is on the launchpad. For this purpose, openings for the connections are provided on the side of the airframe.

3.3.2.2 Launch Pad Mechanical Interface

The vehicle is connected to the launch rail using two rail buttons made of brass. The location of the upper rail button influences both the stability on the rail as well as the effective length of rail available for stabilization during launch. It is mounted with the screw that is also used to hold the fuel tank. The bottom rail button is used to support the weight of the vehicle while on the launch pad and to hold it down until successful engine ignition is confirmed. It is screwed to the airframe fincan coupler.

3.3.2.3 Airframe-Fincan Coupler

Fincan and body tube are joined using an aluminium coupler ring that provides a tight fit and is held in place via radial screws. This ring is also where the thrust from the engine is introduced into the airframe, using four aluminium struts that connect the coupler ring to the engine head. This is explained in more detail under Section 3.2.4.

3.3.3 Fincan

To bring the center of pressure well below the center of gravity and thus ensure sufficient static stability, we opted for four fins and a lightweight carbon fiber construction. The fillets between fins and center piece are designed rather extensive, so that those areas are resistant enough.

The Fin profile is based on a symmetrical airfoil profile and was then adapted so that a positive mold could be 3D-printed in house. This mold was sanded and treated with several thin layers of coat to seal pores and after that with release agent. Then a four-part negative mold consisting of high-temperature epoxy tooling gelcoat and high-temperature epoxy moulding paste was taken. This was then used to laminate with pre-preg carbon fiber. For the fin cores Rohacell, a closed-cell rigid foam, was chosen. Those inlays are both essential for the stiffness of the fins and ensure that enough pressure is exerted on the laminate during the curing process. The Boards of Rohacell foam were CNC-milled to the correct form and then placed on the four layers of carbon fiber mats before the negative mold was assembled. The inside of the tincan was strengthened with another two layers of carbon fiber. After being sealed in a vacuum bag and cured by by being gradually heated to 135°C in the oven, the fincan was sanded, cut down to the lengh of 300 mm and prepared for painting (shown in Figure 3.14)

Figure 3.14: Surface sanding after curing of the Fincan.

3.3.4 Livery Design and Surface Finish

To accomplish a flawless look as well as create a surface that mitigates some of the solar heating experienced in the EuRoC launch environment, the airframe is painted white. The decision was made in favor of lacquer and against adhesive foil, since the latter has proven to be both unsightly and less resistant. Furthermore, painting allows to not have solid black markings, but to leave the carbon fiber underneath visible, which displays the lamination work and the texture of the airframe.

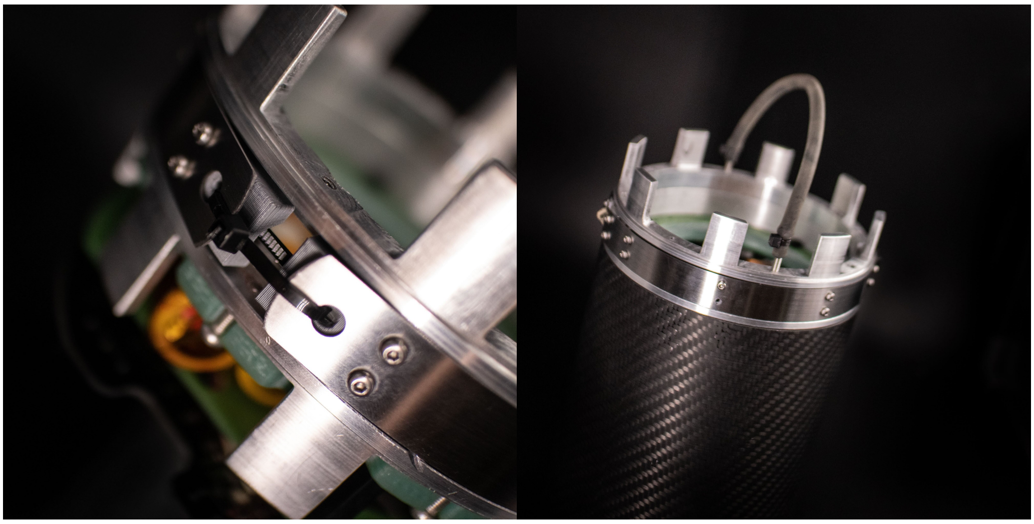
On the body tube the team name, the project name, the academic affiliation and the sponsor logos are arranged into a minimalistic design. The fins all feature the Team ID and the Austrian flag. Each of them additionally displays a unique but simple graphic pattern of black and white to allow ground-based observers to track and record the launch vehicle's altitude.

To achieve a satisfactory result, first the flaws were repaired with epoxy resin, then the whole airframe was sanded and coated with transparent primer. After drying, stencil stickers were applied in the places of the markings. The white paint was then sprayed on and allowed to harden. The stickers were then carefully removed and the airframe covered with a clear, matte two-component coat.

3.4 Recovery

3.4.1 Deployment System

Recovery is accomplished by a two stage parachute system. At apogee, the nose cone is separated from the body tube via our custom designed clamp band mechanism. It consists of two coupler rings (one mounted to the body tube and the other one to the nose cone) tightly clamped together via a spring steel band.



(a) Clamp band Connection (b) Clamp band coupled to body tube

Figure 3.15: Clamp band Assembly

The clamp band is released by one of two redundant pyrotechnic line cutters severing a line that connects the two ends of the clamp band. The spring steel band separates from the coupler rings (still connected to the main tube by a short line) and releases the connection between them. The coupler on the nose cone is then catapulted away from the bottom coupler ring. This is accomplished by four strong slingshot rubber tubes, ensuring a clean and reliable separation even at high airspeeds (and thus high drag force on the nose cone). Along with the nose cone, a small cup, housing a drogue parachute, is pulled away, releasing the drogue chute.

3.4.2 Parachute

Drogue Chute Diameter	340 mm
Drogue Chute Type	Round
Main Chute Diameter	1400 mm
Main Chute Type	Pull-down Apex

Table 3.2: Chute Specs

During the drogue phase, the vehicle descends quickly (about $35\,\mathrm{m\,s^{-1}}$) and in a controlled manner. The nose cone with the rubber band mechanism and the drogue cup is suspended directly from the drogue with a line shorter than the drogue line to avoid collisions between the nose cone and the body tube. The main parachute and its lines are packed into a deployment bag, which is housed in a cup mounted to the lower section. A few hundred meters above ground, a second set of line cutters severs the connection between the drogue line and the vehicle and another line that keeps the main deployment bag from sliding out of its cup. This puts tension on the line connecting the drogue line and the main deployment bag, allowing the drogue to pull out the main deployment bag. Tension on the main line opens the deployment bag, allowing first the lines and then the parachute to be pulled out of it in a controlled manner. The rocket now descends at about $7.6\,\mathrm{m\,s^{-1}}$. In this final landing configuration, the drogue and nose cone assembly stay connected to the deployment bag and main parachute. The system has been successfully validated in multiple ground tests, and in a test launch to 700 m using a mockup airframe propelled by a solid motor. Small modifications are planned to increase reliability and to allow for easier assembly/preparation. Additionally, the parachute sizes will be adapted to the increased vehicle mass.

3.4.3 Failure during Test Flight

On our most recent test flight of the rocket, the recovery has failed by the drogue chute line ripping during the initial drogue deployment. With the rocket then being in free fall the main chute line never had a chance of holding. After carefully studying the received data and video footage of the test flight, the failure has been identified to most likely be caused by the following: The wind speeds encountered at launch day were higher than expected, leading to a higher horizontal velocity at apogee. This of course means that the airspeed, when deploying the drogue, was higher, increasing the load on the lines. Additionally, quite thin drogue lines were chosen based on pre-existing knowledge and experience within the team, which in conjunction with a drogue that was too large and a non-functional shock absorber lead to the lines snapping. These issues will be rectified in the new recovery system for EuRoC. Since all of the identified causes of the recovery failing have been eliminated and a successful recovery test flight (not using our liquid propulsion, instead having a dummy solid booster) was conducted, we are confident that the recovery system will perform nominally for our flight at EuRoC.

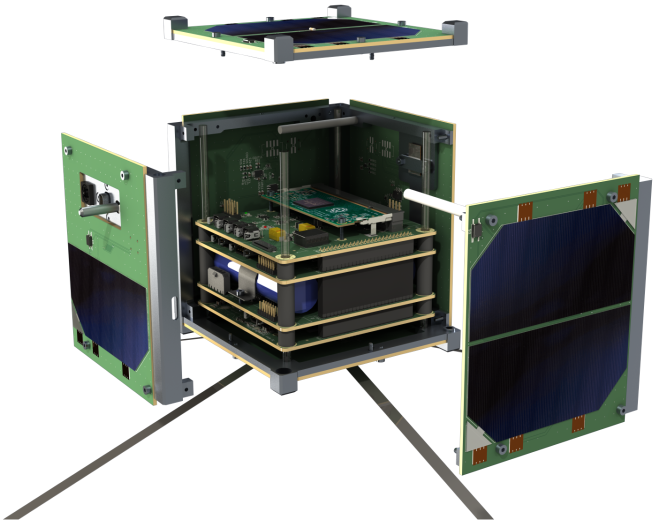


Figure 3.16: Payload: STS1 Render

3.5 Payload

μHoubolt 's payload is part of STS1 (SpaceTeamSat1), a different TU Wien Space Team project. SpaceTeamSat1 is a 1U cubesat, its primary mission is to design and build a cubesat completely in-house that will eventually fly orbit in space and survive the conditions there. The secondary mission of STS1 is to give students the opportunity to run their own Python scripts in space. The scripts are executed on a Raspberry PI CM 3+, which is connected to several sensors via an I2C bus. The sensors include temperature, magnet, gas, acceleration and more.

This inter-project collaboration offered itself as the μHoubolt launch is a great opportunity to flight test some of the hardware for the cubesat and because μHoubolt needs to fly a payload anyways. Since the full 1U cubesat wouldn't fit inside the μHoubolt airframe only the EDU subsystem is going to be flown.

To fulfill the internal goals for the payload the education subsystem of STS1 and a power supply is required. The whole assembly is installed in a 100 mm x 100 mm x 50 mm housing with a total mass of 1 kg, this meets the requirement of four PocketSats with a mass of 250 g each.

3.6 Avionics

It is advised to read section 6.7 in advance to fully comprehend the following content.

3.6.1 Subsystem Block Diagram (SSBD)

The avionics subsystem is divided into multiple different modules as shown in fig. 3.17. Those include the three custom designed ARM based units: ECU (engine control unit), PMU (power management unit) and RCU (radio control unit), two COTS Altimax G4 altimeter as well as the mandatory Eggtimer TRS.

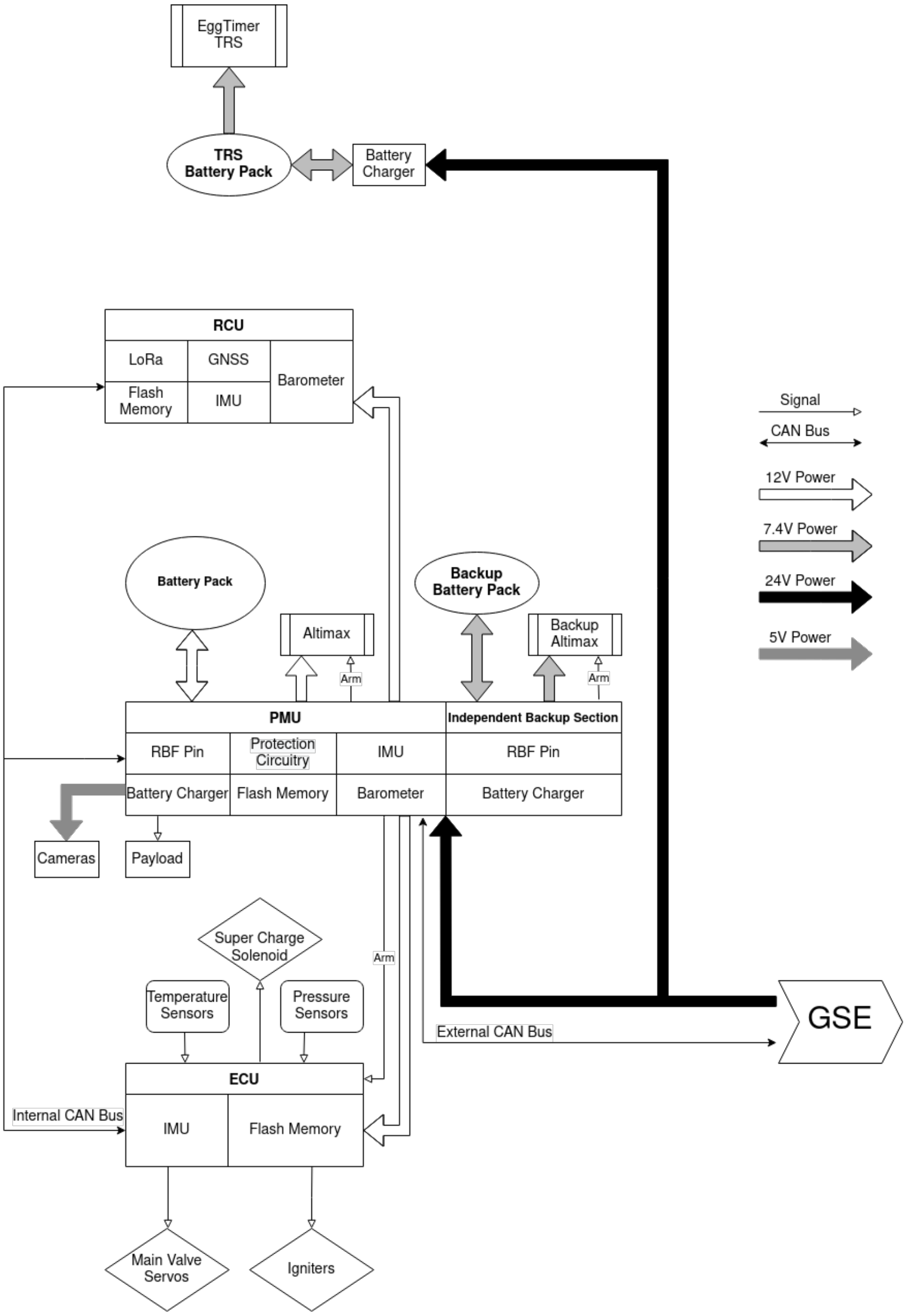


Figure 3.17: Subsystem Block Diagram, see fig. 3.20 for the block diagram for the GSE

3.6.2 Physical Architecture

The following section describes our three custom units, all equipped with an ARM processor and two CAN-FD interfaces: **ECU**, **PMU** and **RCU**.

The ECU (engine control unit) contains all the electronics necessary for the liquid propulsion system. This includes circuitry for controlling the main valve servos, the supercharge solenoid as well as the igniters. It also has inputs for the chamber and tank pressure sensors, for the tank temperature sensors and current and servo position feedback. A 32 Mbit external NOR-flash-chip allows for the logging of all data packets sent to mission control in full sampling speed. This is especially useful during flight, where the data transmission speed is limited by the LoRa communication.

The PMU (power management unit) is powered by a 3S1P Li-Ion 18650 and features over-current protection, voltage and current monitoring. It also contains an IMU and a barometric pressure sensor for redundancy reasons, and logs all its data to flash memory. The PMU interfaces with all the other avionics, supplies them with power and bridges between the two CAN-FD busses. Located on a separated part of the PCB, an independent backup section powered by a backup battery pack charges and arms the backup Altimax Altimeter. Moreover, the PMUs RBF (Remove before Flight) pin circuit switches on the igniter voltage, carried to the ECU on its own power cable, separately, allowing an ECU power-up without ignition capability and, thus, ensuring that the ECU can not activate the igniters in operation modes, where the pad crew is present.

The RCU (radio communication unit) consists of communication and sensing hardware. A GNSS module, an IMU and a barometric pressure sensor collect important data for recovery and post processing. Again, all data is logged to external flash memory. A 433 MHz LoRa module sends data to the ground station.

3.6.3 Operational modes

Safe: Once the rocket is mounted the RBF pin is pulled halfway, which powers on all avionics onboard. At this point data is redundantly sent via 433 MHz LoRa and the rocket can be fully controlled by our Mission Control software via a 2.4 GHz directed radio link, the only exception being the ignition, which is still physically disconnected from power by the RBF pin. Thus, the final pad preparations including fueling and preparing oxidizer filling are done in this mode.

Armed: Once the on-pad preparations are done and the pad area is vacated, the RBF pin is pulled out completely, which exposes the ignition circuitry to power. The whole system is now operational. When the launch window opens oxidizer filling begins, marking the last event before launch.

Internal Control: After oxidizer loading is completed the rocket is put into internal control, where the ECU begins its internal sequence.

Holddown: If neither hold nor abort is commanded by the operator or the ECU, the ignition sequence is automatically started. After ignition the holddown clamps are still engaged holding the rocket firmly on the launchpad. The rocket stays in holddown mode until sufficient engine performance is detected by measuring combustion chamber pressure.

Lift-off: If suitable chamber pressure is detected by the ECU, it commands the launch pad to release the holddown clamp. The rocket leaves the launchpad, disconnecting the electrical umbilical cord, and, thus, the power and CAN connection to the launch pad. This leaves us with only the unidirectional LoRa communication. The rocket is now fully controlled by the ECU.

Powered Ascent: During the powered ascent phase, data is being sent to Mission Control at all times. At the end of the planned burn time, the ECU closes the main valves to shut down the engine, also known as Main Engine Cutoff (MECO).

Unpowered Ascent: With the ECU having finished it's internal sequence, the SRAD avionics only remain active to transmit data, shifting focus towards the COTS altimeters detecting apogee.

Recovery: Once apogee is detected, the two-phase recovery begins with the ejection of the nose cone and drogue chute. This marks the last operation mode of the rocket. See section 3.4 for more details on the recovery process.

3.6.3.1 Pre-Launch Sequence

The Pre-Launch Sequence is conducted on our self developed Web-Client. It contains a dynamic and interactive Piping and Instrumentation Diagram (P&ID/PnID) for monitoring and controlling all components, the tanking system as well as avionics inside the rocket. A checklist is carefully being processed that states each step needed for preparing and tanking the rocket for launch. Each step is announced by the Launch Commander and is re-validated and executed manually by Mission Control. At T-10 seconds internal control is activated and the rocket self-checks for any malfunctions during engine startup and launch.

3.6.3.2 Abort Sequence

In case of any malfunctions during engine startup either the internal control system can abort or Mission Control can send an abort to the rocket manually. In either case the main valves are instantly closed to prevent any thrust generation. After that a separate checklist is conducted manually to revert the system back to a safe state.

3.6.3.3 CAN protocol

To establish a steady communication with ground systems as well as the rocket, the CAN-FD protocol is used. The whole setup is split into three busses that are all connected to an Ubuntu server. Since the CAN bus is disconnected from the rocket at lift-off, a separate internal CAN bus is used to maintain communication between ECU, RCU and PMU. The PMU is able to bridge CAN messages between the internal and external CAN busses. A self developed protocol is used on top of the CAN-FD protocol to establish the controllability and flexibility needed for Mission Control.

3.7 Ground Systems

3.7.1 Launch Pad

For the launch pad, the newly constructed Space Team launch pad will be used. It is large enough to fit all ground system components and with a new rail extension the now 7 m launch rail is long enough to allow the rocket to take off at a decent speed. Since it is already built, it saves time and resources which would be required to construct a new one, specific to μHoubolt . fig. 3.18 shows the launch pad in usage.

Figure 3.18: Launch Pad with μHoubolt during lift-off on first test flight

3.7.1.1 Rail

The launch rail consists of pieces of 30x30 mm aluminium extrusion profile, each mounted to one corner of a section of triangular aluminium truss. Pipe clamps screwed into one side of the profiles are used to connect the profile to the truss sections. The whole rail consists of three truss sections, two of them measuring 2.5 m, one at 2 m, which are connected by slotting the ends of one section into the ends of the previous section and securing them with cross bolts. The extrusion profiles on top are then aligned using long sliding blocks in the lateral grooves, resulting in a seven meter long continuous launch rail. The rail buttons on the rocket ride in the dorsal groove. One of the truss sections is mounted to the launch pad trailer using hinges. This allows the lowest rail section to be folded down during transport. For a launch the other two rail sections are added and the whole rail, once in the upright position, is braced against the trailer as well as guyed to the ground. See fig. 3.18.

3.7.1.2 Holddown

While the rocket will be guided along the rail using rail buttons its vertical movement will be restricted by an additional holddown system. Two bars are placed on the top and bottom of the lower rail button. The top bar is connected to a pivoting mechanism, while the bottom bar is connected to a plate with a force measurement system. By measuring the weight of the rocket while tanking the fill level of the oxidizer tank can be estimated. During the ignition sequence, the upper bar of the holddown system prevents a premature liftoff. As soon as the chamber pressure (and thus the thrust) is high enough, the system unlocks the pivoting mechanism with a COTS RC servo motor and releases the rocket. This ensures that a liftoff can only happen when the engine performs as expected and accelerates the vehicle to a velocity sufficient for aerodynamic stability when leaving the launch rail. It also offers an additional abort scenario in case of a loss of control over the fuel or oxidizer valves by constraining the rocket to the launch rail for the entire burn duration.

3.7.2 Tanking Infrastructure

The tanking infrastructure (containing oxidizer and pressurant loading system) is mounted on the trailer next to the launch rail. It contains all the necessary plumbing and electronics to remotely tank oxidizer and pressurant and disconnect the umbilicals before the launch.

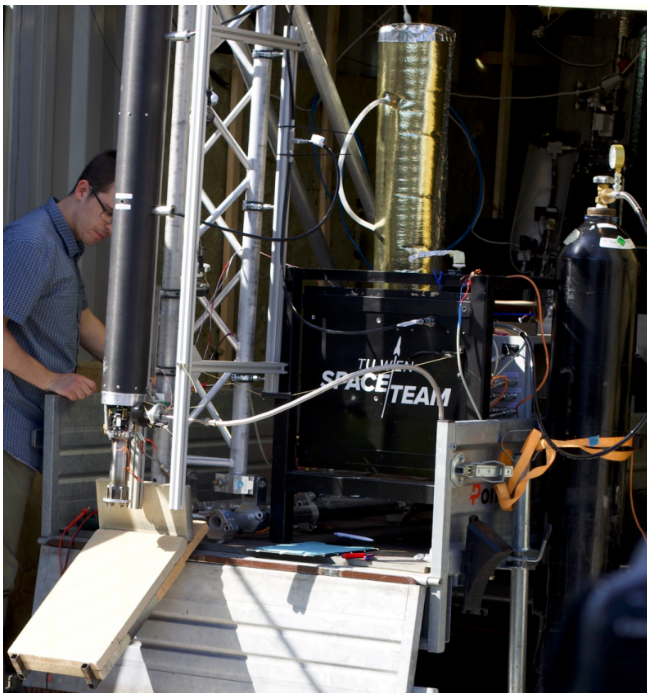


Figure 3.19: Rocket on rail (left) and oxidizer & pressurant loading system (right)

3.7.3 Oxidizer Loading System

See section 3.2.5.2 from the Propulsion chapter for a more complete description of the process. This chapter mostly focuses on the physical setup. For a piping diagram see fig. 3.2.

Nitrous Oxide is supplied from a standard gas cylinder which is mounted upside down in the propellant loading system. A heat exchanger jacket is placed over the cylinder in order to be able to control its temperature. In cooling mode the jacket is supplied with water slightly above 0 °C, while in heating mode the water has 30 °C. The hot and cold water is stored in two insulated tanks on the back of the propellant loading system.

This heat exchanger jacket mostly consists of a modified PVC drainage pipe. To install the gas cylinder the bottom flange of the heat exchanger jacket is placed upside down on the bottle neck, with the valve sticking through a hole in the flange and secured using a nut which fits the threads normally used for the bottle cap. The cylinder is then turned on its head and the bottom flange inserted into a cradle on the propellant loading cart where it is secured using cross pins. Then the jacket is installed and fastened to the bottom flange using three toggle latches around its outside. Lastly the cooling/heating system is connected to the heat exchanger using two hoses equipped with quick-connect fittings. The feed line enters the heat exchanger on the bottom while the return line exits it on the top.

At this point the nitrous oxide cylinder can be connected to the oxidiser loading system and the oxidiser and pressurant umbilicals can be connected to the rocket. Before the start of actual oxidizer loading the tank inside the rocket needs to be pre-pressurized with nitrogen which uses the same pressurization system as the final pressurization before launch via the pressurization umbilicals.

After completion of the filling process (indicated by liquid being vented through the solenoid valve instead of gas), the fill valve is closed. Opening a vent valve in the oxidiser loading system depressurizes the umbilical in preparation for its remote controlled detachment. This is needed as the quick disconnect fittings in use for the umbilicals can't be remotely disconnected under pressure. The entire oxidizer loading process starting from opening the fill valve to the oxidizer tank being full takes about one minute.

The pressurant fill system then supplies nitrogen from a standard 300 bar cylinder to the two pressurant tanks in the vehicle. Same as the oxidizer loading system it also uses two valves for filling and umbilical venting, but it does not need a temperature management system.

The individual steps that need to be executed during the filling procedure are listed in section 4.2.6.

3.7.3.1 Temperature Control

Switching between the heating and cooling cycle is facilitated by actuating two 3-way valves, one in the feed line, one in the return line and only turning on the hot or cold water feed pump.

The water in the heating cycle is brought to temperature using a 3 kW water heater controlled by a temperature sensor inside the hot water reservoir, while the cooling cycle temperature is maintained by periodically dumping dry ice into the cold water reservoir. Although water ice can also be used if dry ice is not available, using dry ice is the preferred option, since it doesn't influence the water level in the system.

3.7.3.2 Electronics Components

The Ground Support Equipment, including the oxidizer loading system uses a modified version of our Engine Control Unit PCB, with the outputs for the igniter being repurposed for driving the water pumps through relays. The nitrogen pressurization also takes care of the oxidizer tank pre-pressurization and utilizes two SRAD Turboservos which replaced COTS servo motors as those couldn't provide a high enough torque for actuating the valves at such high pressures, which also have their accompanying BLMB (Brushless Motor Board) PCB connected. All of these components are connected via CAN to Mission Control.

The Turboservos are SRAD servo motors which utilize a brushless DC motor connected to a servo gearbox and a rotary encoder. The control PCB is connected directly to the CAN bus and takes position commands which it translates into movement for the motor to reach the desired position which is then checked in a control loop with the rotary encoder.

The water temperature is being measured with two submersible NTC Resistors which can measure the temperature accurate to +/-0.5 %.

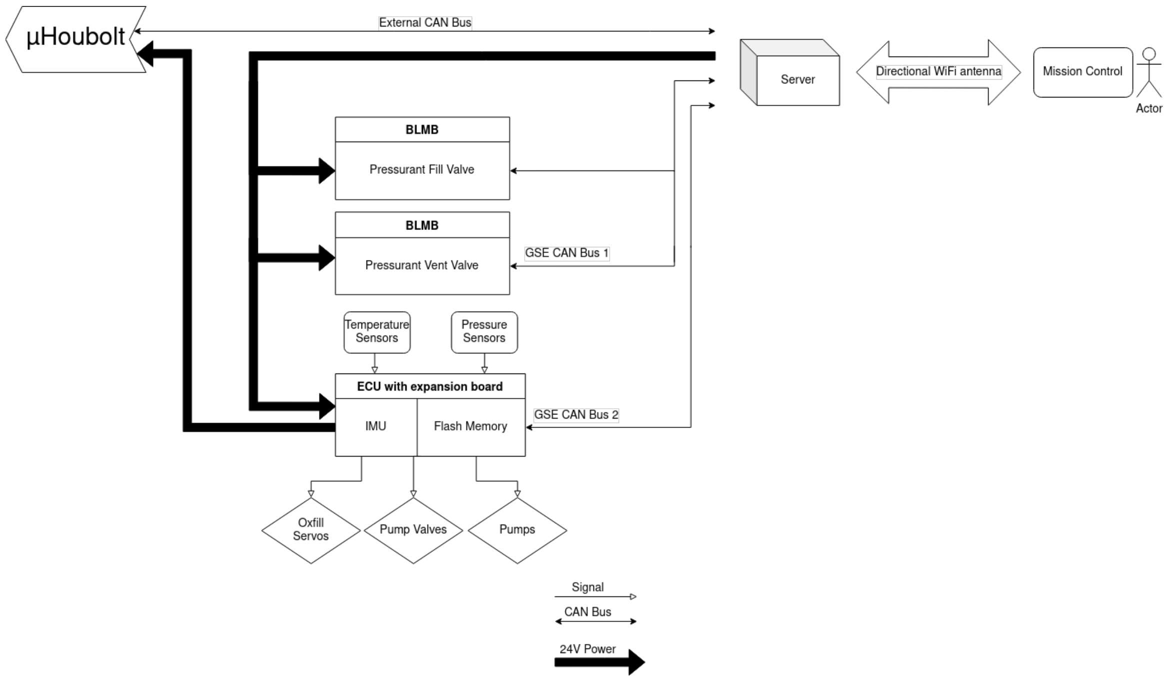


Figure 3.20: Subsystem Block Diagram, see fig. 3.17 for the block diagram for the Avionics.

3.7.4 Pressurant Loading System

3.7.4.1 Overview

The pressurant tanks inside the rocket, which are actually paintball tanks rated for 310 bar are pressurised to 300 bar through two seperate umbilicals which are fed from a nitrogen gas cylinder, as can be seen in fig. 3.2. Since the target pressure in the rocket's tanks is the same as the pressure inside the full nitrogen cylinder a pressure regulator is not required. Pressurisation of the rocket as well as depressurisation of the oxidiser loading system can be remotely controlled using two motorised valves.

3.7.5 Umbilicals

3.7.5.1 Electric Connection

The electrical umbilical is magnetically connected to a port at the top of the vehicle. During lift-off the electrical umbilical will automatically disconnect as the rocket moves along the launch rail.

3.7.5.2 Fluid Umbilicals

The fluid umbilicals are connected at three points along the side of the rocket. The upper two supply the pressurant gas while the lower one provides the oxidiser. Thus the two pressurant umbilicals consist of flexible high pressure gas line that is able to withstand the 300 bar inside the pressurant system. The oxidiser umbilical is a piece of flexible tubing compatible with nitrous oxide, which is reinforced with metal braiding along the outside. All three fluid umbilicals terminate in quick-disconnectors that have their counterparts on the rocket. The ends of the umbilicals are connected to the strongback that is pulled away from the rocket before the launch. In order to not apply any unnecessary torque to the rocket that could damage the rail or the railbuttons umbilical disconnectors (see fig. 3.21) are used

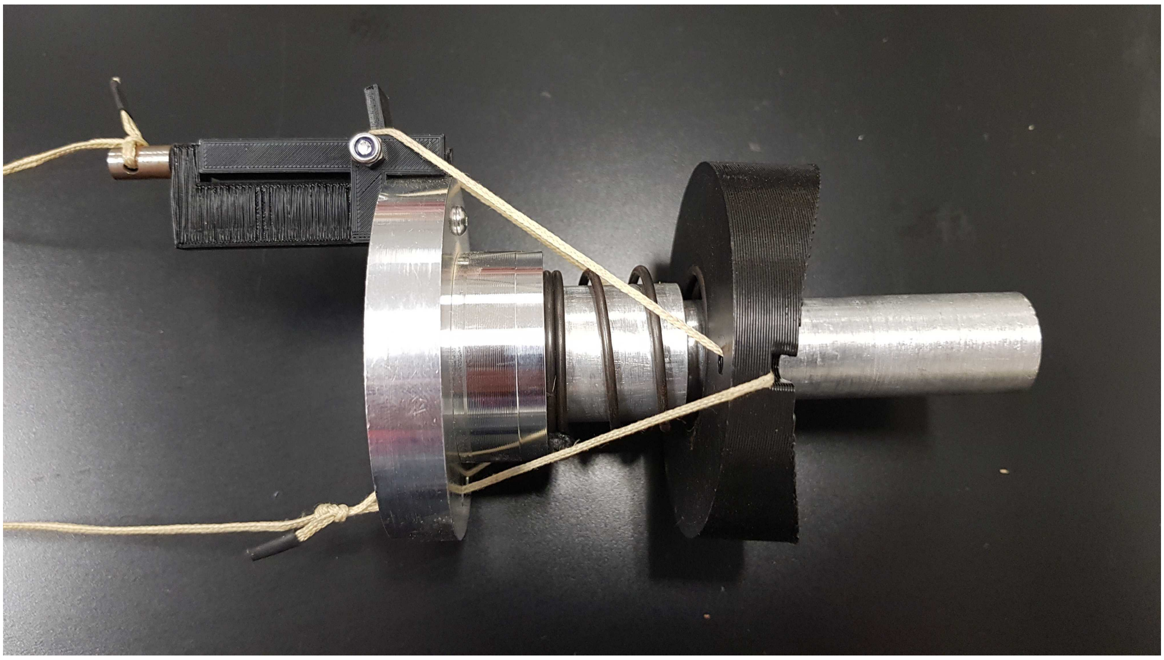


Figure 3.21: Umbilical Disconnector

to separate the lines from the rocket. They are spring loaded devices that are placed over the quick-disconnects. When the strongback is pulled back it first pulls a pin (uper left) out of the umbilical disconnector that releases the spring, which presses against the airframe using a pusher plate (black). A lip at the front of the tube engages with the quick-disconnect and releases it. The strongback, which is connected to the umbilical and disconnector through the line in the lower left then pulls the device away from the rocket.

In order for this system to work reliably the oxidiser and pressurant lines first have to be depressurised.

3.7.6 Ground Support Equipment Electronics

This section has overlap with information on the Mission Control Software, so if interactions are unclear it is advised to read section 6.7 from the appendices which covers the the software architecture.

Similarly to the rocket itself, the electronics in the launch pad are connected to the main Ubuntu Server via CAN Bus and is controlled via the same Low Level Server and Mission Control data flow as the rocket. There are three components in the GSE: A modified ECU (Engine Control Unit) board whose igniter outputs are wired to the cooling and heating pump of the oxidizer loading system, and two BLMB (Brushless Motor Board).

The launch pad is wired to the Ubuntu Server in a mobile rack with a CAN cable and the server rack is connected to the Mission Control with a directed radio link. In the server rack, connected directly to the server via LAN, a Raspberry Pi single board computer with a LoRa shield is used for the 433 MHz radio connection to the rocket during flight.

The entire GSE is powered using mobile power generators and during launch preparations the electronics umbilical also provides power to the rocket, keeping the internal batteries charged until automatic disconnect at lift-off.

4 Concept of Operations

4.1 Rocket lifecycle during EuRoC

1. Rocket gets presented at the exhibition.
2. On the day before the launch, the rocket gets assembled, the recovery section prepared and the electronics checked.
3. On the launch day we bring the rocket to the launch site and start going through the launch day checklists.
4. After landing we recover the rocket and bring it back to the launch.
5. After inspection of the rocket, we bring it back to the exhibition area.

4.2 Launch Procedure

4.2.1 Vehicle Assembly

1. Pulling the RBF.
2. Testing electronics - running test and launch sequence.
3. Inserting the RBF halfway.
4. Starting fuel loading process (see chapter ref 4.2.4).
5. Checking the water state (cooling and heating cycle).
6. Checking that the igniter electronics are deactivated.
7. Installing the igniters.
8. Mounting the Fincan to the body tube.
9. Vacating the launch area.

4.2.2 Mission Control Setup

1. Connecting directed radio link and Raspberry Pi with LoRa shield to server.
2. Powering on server, Mission Control PC and monitors.
3. Connecting Mission Control PC to server via LAN.
4. Opening Mission Control Web-Application on Web-Browser.

4.2.3 Launch Pad Setup

1. Mounting flame diverter.
2. Connecting GSE to directed radio link.
3. Filling water reservoirs.
4. Flipping, installing and sealing oxidizer bottle inside temperature exchange mantle.
5. From now on continuously check and add ice to cold water reservoir.
6. Installing pressurant bottle.
7. Sliding rocket onto launch rail.
8. Mounting T-nut underneath lower rail button.
9. Securing and closing holddown above lower rail button.
10. Connecting oxidizer, pressurant and electrical umbilicals.
11. Pulling RBF pin halfway to power on avionics.
12. Checking sensors and actuators, verifying movement and calibration, via Mission Control.

4.2.4 Fuel Loading

1. Closing fuel main valve.
2. Mounting the tyre fill adapter connected to the syringe filled with ethanol to the fuel inlet section to open the check valve completely.
3. Applying force to the syringe to fill the tank with a precise amount of fuel and enabling the ullage gas to exit the tank through the small bleed orifice.
4. Removing the syringe from the rocket after filling.
5. Covering the rocket's fuel inlet section.

4.2.5 Final Pad preps

1. Opening oxidizer bottle and checking for leaks.
2. Opening pressurant bottle and checking for leaks.
3. Staring pad cameras.
4. Pulling RBF pin to connect power to ignition circuitry.

From this point onwards the rest of the preparations until launch can be done completely remotely.

4.2.6 Oxidizer Loading

Pressure and temperature data is closely monitored throughout the whole process.

1. Closing oxidizer main valve.
2. Activating and setting the vent valve pressure regulation to 30 bar.
3. Closing pressurant vent valve.
4. Opening pressurant tanking valve for filling pressurant until vent valve gets activated. Then closing pressurant tanking valve.
5. Closing oxidizer vent valve.
6. Opening oxidizer fill valve to start the oxidizer loading.
7. Activating the heating cycle in the oxidizer bottle.
8. Quickly closing oxidizer fill valve when plume (venting liquid oxidizer) is visible.
9. Deactivating the heating cycle/activating the cooling cycle in the oxidizer bottle.
10. Opening oxidizer fill valve again after a stabilization of the venting frequency.
11. Quickly closing oxidizer fill valve when plume is visible.
12. Opening oxidizer vent valve.

4.2.7 Pressurant Loading

1. Deactivating vent valve pressure regulation.
2. Opening pressurant tanking valve.
3. Waiting for stable pressurization.
4. Closing pressurant tanking valve.
5. Opening pressurant vent valve.
6. Activating umbilical retract of pressurant and oxidizer tanking lines and verifying clean separation.

4.2.8 Internal Countdown and Launch

1. Activating the rockets internal control via Mission Control after Go/NoGo.
2. The rocket does a continuity check on the igniters, if continuity is given, start internal countdown.
3. The rocket checks for proper engine performance after ignition. This is measured by a proxy measurement of combustion chamber pressure. The threshold is set beforehand by Mission Control and is usually set to around 10 bar.
4. If proper engine performance is detected by the rocket, it sends a signal to the Launch Pad to release the holddown.
5. Lift-Off is achieved once the electrical umbilical that is magnetically held in place gets disconnected by the rocket moving out of reach.

Until lift-off there is still a possibility for manual abort from Mission Control. Beginning with lift-off and the electrical umbilical disconnecting the rocket is monitoring itself and no manual abort is possible. The rocket is now in powered ascent phase.

The entire engine burn duration is 8 s long, after which the main valves are closed and we have achieved MECO (Main Engine Cut Off). Since the propellant tanks are sized for this 8 s burn duration, alternatively the engine could also just be left open to run out of propellant (which should happen roughly 8.8 s after ignition). This approach has the advantage of not needing to open the main valves again for complete depressurization.

From then on, the rocket is in unpowered ascent until apogee is detected and recovery is triggered.

4.2.9 Recovery

1. Opening vent valve pressure regulation.
2. Closing vent valve pressure regulation when the tanks are fully depressurized.
3. Opening fuel main valve for remaining fuel unloading.
4. Separation of the nose cone from the body tube at apogee.
5. Drogue chute release at apogee.
6. Main chute release 250 m altitude. Backup Altimax triggers at 200 m.
7. Recovering the rocket after landing.

4.2.10 GSE Safing

1. Stopping all cameras.
2. Closing the oxidizer bottle.
3. Closing the pressurant bottle.
4. Vacating the pad area.
5. Opening oxidizer fill valve.
6. Opening pressurant tanking valve.
7. Checking all pressure data to verify a full depressurization of the system.
8. Announcing ”safe state”.

5 Conclusion and Outlook

Our motivation back when starting the liquid propulsion project within the Space Team was both to be able to work on extremely cool technologies and learn a lot while trying to develop our own rocket engines, as well as pushing the boundaries of what is thought to be possible for a student team to achieve. Throughout the over three years since the first endeavours in liquid engines, we have learned so much about the technology and its limits, developed extensive testing infrastructure and built a staggering amount of prototypes.

The project we have actively worked on has slowly morphed from the gigantic Houbolt rocket to the small scale technology demonstrator μHoubolt and we also branched out into sub-projects like GATE, a Houbolt-sized engine concept to go along with our large engine test stand - but the main goal has stayed the same: pushing boundaries. We have managed to build and operate the largest rocket engine in Austria, built a reliable small scale engine to be used in μHoubolt and now have a rocket to demonstrate these capabilities.

However, all of this would be in vain if all this progress gets lost once the project concludes. No one can be part of such an ambitious project during their free time forever and as such the probably most important part of the project is the final documentation of it. Once all is said and done and the pressure of internal goals and milestones has passed, we plan to clean up and organize our project files; from CAD models to PCB designs, software infrastructure and engine designs. When that is done it is planned to release everything needed to make a ”μHoubolt 2.0” to open source - but that’s not all! μHoubolt has been a tremendous undertaking, but we do have plans for the future internally as well: Having now proven we can build a reliable engine, it’s time to do more interesting things with it than just aimlessly shooting a rocket into the sky. The next twinkle in our eyes is μHouverer, a craft that can propulsively take off and land powered by a bi-liquid rocket engine.

6 Appendices

6.1 System Data

Mass (Dry)	8950 g
Mass (Wet)	12 040 g
Tank Volume (Fuel)	900 mL
Tank Volume (Ox)	2400 mL
Length	2100 mm
Diameter (Body)	123 mm
Diameter (Fins)	383 mm
Pressurant Pressure	300 bar
Oxidizer Pressure	40 bar
Fuel Pressure	30 bar
Pressurant Pressure	300 bar
Nominal Thrust	600 N
Combustion Chamber Pressure	15 bar
Burn Duration	8 s
Max Speed	307 m s^{-1} (Mach 0.92)
Apogee	3 km
Descent Rate (Drogue)	39 m s^{-1}
Descent Rate (Main)	7.6 m s^{-1}
Altitude Main Chute Deployment	Main Altimax: 250 m, Backup Altimax: 200 m
RF (LoRa) Frequency	433 MHz

Table 6.1: General System Data

6.2 Detailed Test Reports

6.2.1 Ground Test Demonstration of Recovery System

µHoubolt Ground Recovery Test Protocol

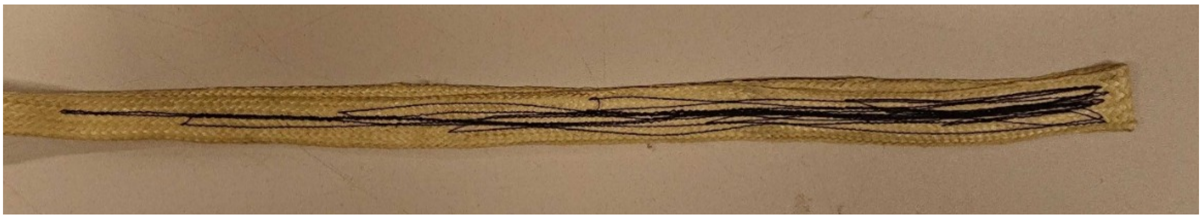
28.9.2022

TEST BY
ROTH MARCUS, BREYNER JOHANN, DELLEKART FLORIAN

TESTTYPE	Recovery System, Shock Absorber and Clean Separation
TESTGOAL	Verification of significant shock reduction on main chute lines and assurance of clean separation of newly manufactured recovery system
CHANGES	• Shock-absorbers sewn into main chute lines • Reinforcement of chute line mounting points with aluminium to ensure better distribution of the line shock and pressure on the GFRP main plate of the recovery system.
FAILS AND LEARNINGS	• Main chute line failure during test flight. • Crap-Band absorbers not strong enough to absorb impact of vehicle • Different wind conditions in different altitudes can lead to differing vehicle velocities at the point of chute ejection. • Shock absorbers have a very high impact on the maximum load the equipment experiences.
ADDITIONAL INFO	The design of our recovery system was tested in flight two times before: 1. A mock-up rocket (steel tube equipped with original nosecone and recovery system propelled by a solid fuel motor). The test was successful, the nosecone separated cleanly, and the full system was recovered using the two-stage chute system 2. Fully configured test flight. In this case the nosecone also separated cleanly, and the drogue was ejected, however, due to a higher velocity of the vehicle than in the previous test, the shock on the lines of the main chute upon deployment stressed them to much and led to a subsequent failure of mentioned lines, which lead to a ballistic landing of the overall system.

Test summary

The shock absorbers consist of a cord, which is sewn into the main chute lines as shown below. The working principle of this absorber is, that the individual cords break when the load caused by the deceleration of the vehicle hits the line and by that dissipate energy which would fully be transmitted into the vehicle structure without them.



Tests were conducted by dropping a mass, connected to the line on the one end, from a height of approximately two meters.

Multiple tests were conducted:

1. No absorbers as a reference.
2. Line with moderate sewing tightness of absorber cords.
3. Line with high sewing tightness of absorber cords.

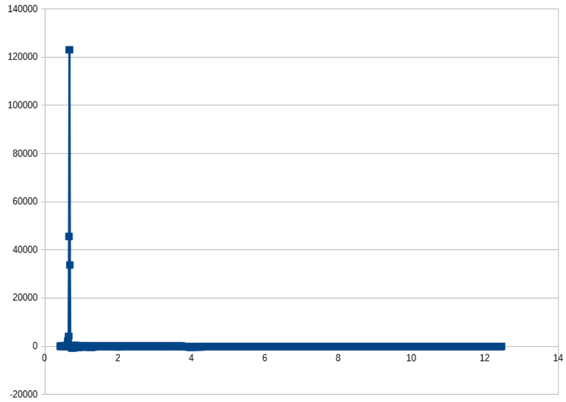
NO ABSORBERS

The test setup was dropped from a height of approximately two meters multiple times with increasing masses until the line broke. This took place at a total mass of 5kg. Subsequent tests were conducted with this mass.

Assuming that in the test setup, which was mounted onto a steel beam on top and is therefore very rigid, the mass needs around 20 cm to decelerate from a speed of approximately 6 m/s after 2 m free fall, the acceleration acting on the system is of around 90 m/s².

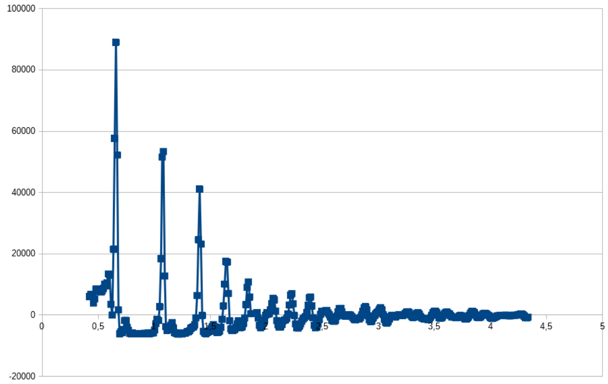
Comparing this to simulation results which yield a maximum acceleration of around 80 m/s² during main parachute deployment, whilst the vehicle at that point has a mass of around 10 kg, it becomes clear that a sufficient shock absorbing mechanism must at least mitigate the shock by a factor of 0.5 to ensure safe operation.

The load cell measured a maximum load 120 kg for this test. The values on the plot axis are shown in grams.



MODERATE SEWING TIGHTNESS

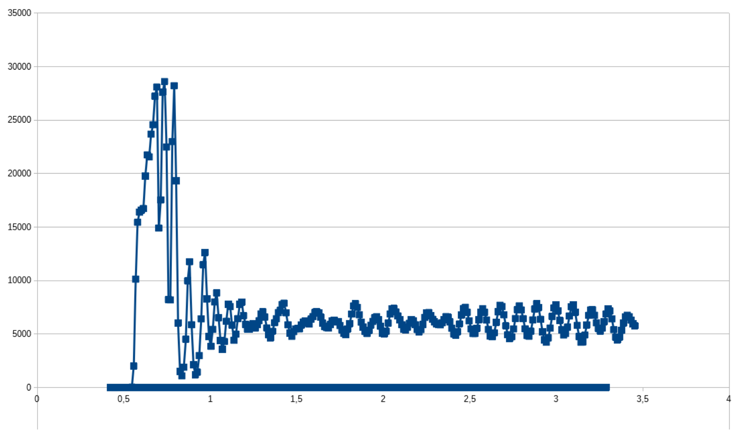
The moderate sewing tightness absorber reduced the shock load on the line around 90 kg as it can be seen in below figure.



The moderate sewing tightness was certainly not enough to properly absorb the shock acting on the vehicle, therefore sewing tightness was further increased.

INCREASED SEWING TIGHTNESS

The last test with increased sewing tightness reduced the shock load further to below 30 kg as shown in the next figure.

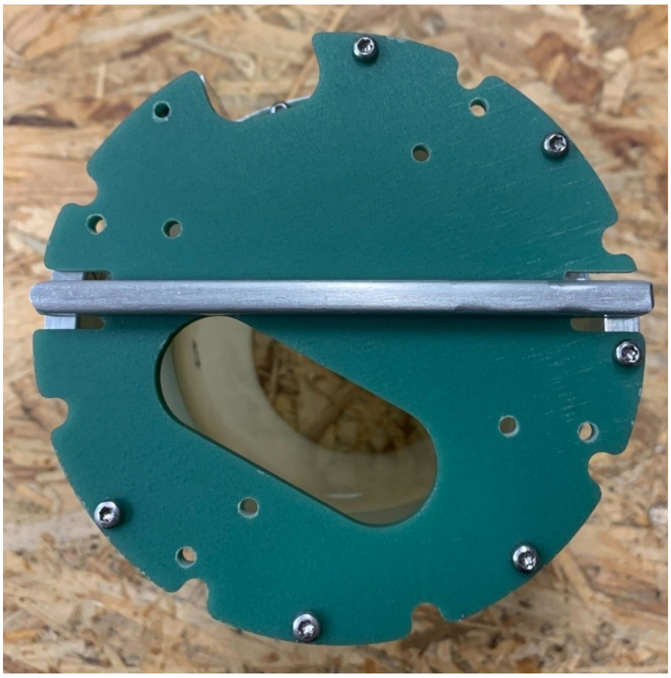


It is also clearly visible, that the shock time was spread out to a longer duration, which further reduces the load on the system. This was reached by a special sewing pattern which ensures that the absorber first breaks in less robust regions and later gradually increases the load. A sketch of the sewing pattern which has shown to yield best results after some iterations is shown below.

The used shock absorber, in combination with the test setup consisting of mass and load cell is shown in the following figure. The disintegration of the absorber cords is clearly visible.

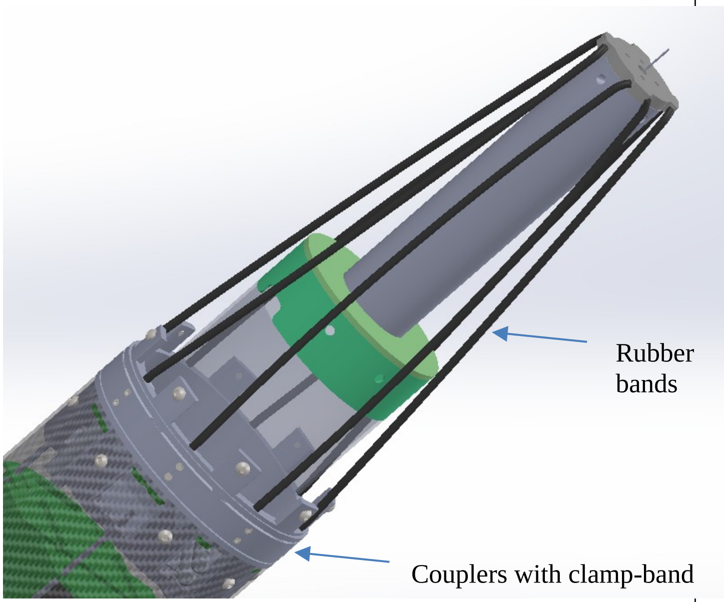
As the results from simulations and tests yielded higher values than expected, also the chute mountings were reinforced to ensure safe operation during future deployments.

The aluminium beam on the mounting points which reinforces the GFRP mountings and distributes the parachute load onto a greater area of the recovery base plate is shown in the following picture. For orientation, note that if this part would be inside the vehicle, the shown picture would be taken from the side of the engine towards the nosecone. The lines are mounted on the left and right of the aluminium bar.



Lastly, it was tested if the newly manufactured deployment system, whose design was not modified since previous tests apart from the shock absorbers and the base plate reinforcement, separates the nosecone in a clean way from the vehicle.

During ascent, the nosecone is mounted to the main airframe by couplers, which are held together by a sheet-metal clamp-band. On first stage recovery deployment, a line which holds the clamp-band together is cut open, which leads to the clamp-band falling off. The nosecone is pre-loaded by rubber bands, which push it away from the main vehicle during deployment.



The used line cutters during flight are human rated, expensive and testing the electronic control of them is in the scope of electronics testing. Therefore, the clamp-band line was cut manually, and a clean separation of nosecone and couplers was verified.

Signature 1

Marcus Roth

Signature 2

Johann Breyner

Signature 3

Florian Dellekart

6.2.2 Flight Test Demonstration of Recovery System

µHoubolt Recovery Flight Test Protocol

18.03.2022

LEAD	MISSION CONTROL		RANGE SAFETY	
DANIEL FRANK	Markus Pinter		Reinhard Rath	

TESTTYPE	Recovery Flight Test
TESTGOAL	Verify correct function of recovery system
CHANGES	None, Initial test
FAILS AND LEARNINGS	Altimax light sensitivity, Railbutton fit, Premature logging end

Test summary

On 18.03.2022, a flight test of the designated µHoubolt recovery system was conducted in Straubing, Germany.
The goal of the test was to verify the correct function of the recovery system, namely the separation of the nosecone and drogue deployment at apogee and the main chute deployment at a set altitude. Both the function of the COTS Altimax recovery altimeter and the SRAD mechanical design was tested.
The exact performance of the system with regards to shock loading resistance and descent speed was not evaluated however, as the maximum altutude was limeted, the mass of the test vehicle was lower than the planned mass of µHoubolt and the parachute size was differrent.

Previously, the mechanical design of the system was verified by separation and deployment tests on the ground (where the function of the linecutters was imitated by using sidecutters) and the altimeter and its configuration was tested using by simulating the air pressure profile of a flight in a vacuum chamber. All tests were sucessfull.

The vehicle (aptly named „Ofenrohr" as its body tube was made from oven exhaust pipes) used for the test was a simple one stage rocket propelled by a COTS solid rocket motor. It was built specifically for this test using cheap and fast manufacturing methods (Except for the nose cone and the recovery system itself, which were designated to fly on µHoubolt). The dimensions of the vehicle were a close match for µHoubolt but the mass was lower at around 7kg.

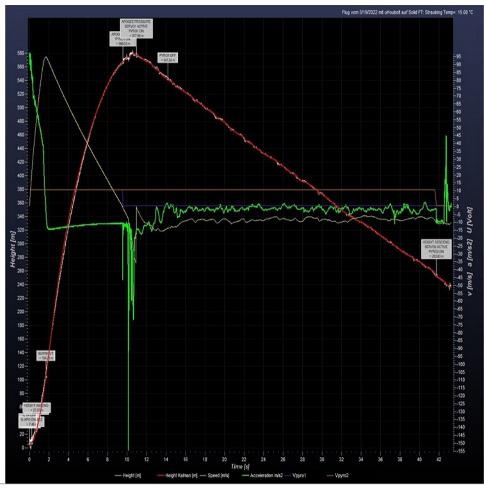
After stressfull preparation days, where the airframe was built in record time, the launch day was fairly uneventful, except for two mishaps during preparation:

- During preparation of the recovery system the altimeter was powered on and armed for testing and software configuration. While this was not expected to cause problems, the nosecone separation line cutter, normally to be fired at apogee, was triggered. While this released the clamp band and required the single use line cutter to be replaced, this did not cause any further problems nor danger, as the powerul nose cone ejection mechanism was not primed.
  The suspected cause is, that sunlight directly hitting the barometric pressure sensor caused false readings due to the photoelectric effect, which is a known problem with th type of sensor used.
  To mitigate this issue, dark open cell foam will be placed over the altimeter pressure sensors to block any light from hitting them.
- The vehicle was for the first time assembled fully and installed on the launch rail at the launch site. This revealed a problem with the dimensions of the railbuttons, which had to be replaced.
  Such problems shall be avoided for future flights by completing the hardware and doing fit checks on the rail well in advance of launch day.

The flight itself went well, with the recovery system performing exactly as planned, bringing the vehicle safely back to the ground and allowing the nose cone and recovery system to be used for µHoubolt. An apogee of around 580m AGL was reached, close to the predicted value. One problem was still discovered though:

- The altemeter stopped recording data after triggering the main chute deployment. It is unclear why this happened, as the logging should have continued until landing gets detected. One possible explaination is an intermittent loss of electrical power due to the deployment shock.

Diagrams and Pictures



Signature 1

Daniel Frank

Signature 2

Markus Pinter

Signature 3

Andreas Ungersböck

6.2.3 Static Hot-Fire

µHoubolt Test Protocol

[1ST STATIC FIRE]

[30.04.2022]

LEAD	MISSION CONTROL	PAD	RANGE SAFETY	FIREFIGHTER
DANIEL FRANK	MARKUS PINTER	GEORG MIKULA	ANDREAS UNGERSBÖCK	MARCUS ROTH

TESTTYPE	First static fire of µHoubolt
TESTGOAL	Successful tanking, Ignition and utilization of our new checklist
CHANGES	-
ADDITIONAL INFO	-

Fails and Learnings

1. Minor leaks in the tanking system. Fixed by retightening pipe connections.
2. Combustion chamber burnt through. Other methods need to be considered.
3. Some checklist items were out of order or at the wrong location. They were corrected during the test procedure.

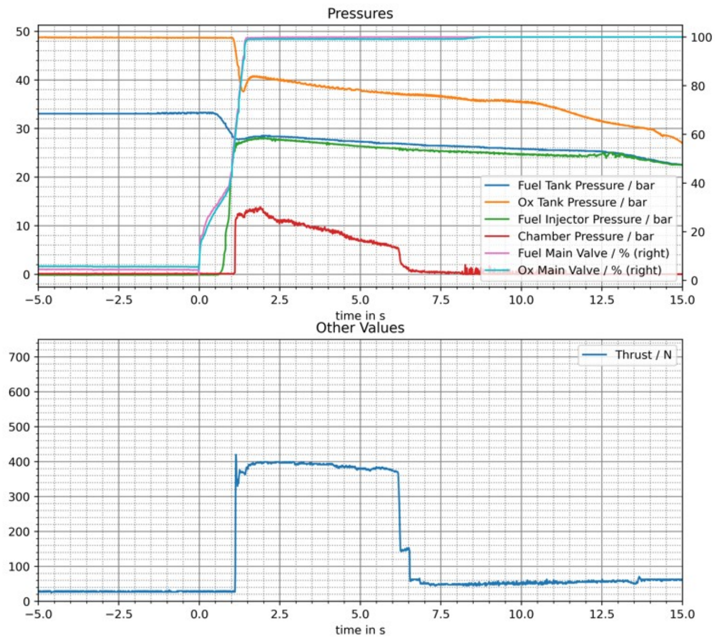
Test summary

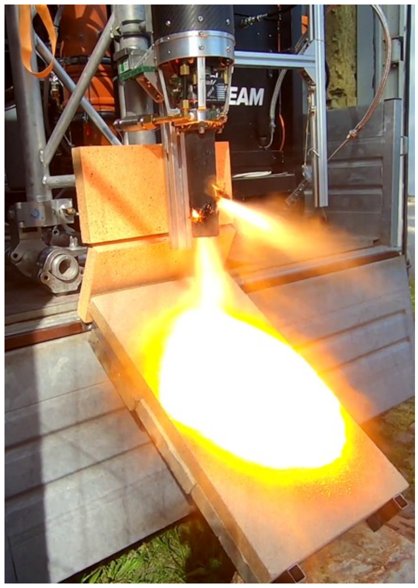
We conducted the first static fire test with an ablative combustion chamber and nozzle made of carbon fibre and epoxy resin. Our focus lied on verifying the adapted checklists previously used for hot fires on our TS02 test stand. A few items were out of order or at the wrong position, but it was recognized and corrected during the test procedures.

The tanking system had minor leaks, so the tanking was aborted. After that we filled the rocket without any issues. The umbilical-retract worked flawlessly.

Engine start-up worked as expected, but after a few seconds of burn time the ablative combustion chamber failed which resulted in a major pressure drop as can be observed in the diagram. The main valves were kept open to simulate a full duration burn until the tanks were empty.

Diagrams and Pictures

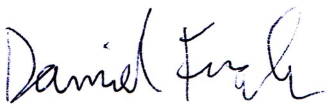




Signature 1

Georg Mikula

Signature 2



Daniel Frank

Signature 3

Markus Pinter

µHoubolt Test Protocol

[2^{ND} STATIC FIRE]

[14.05.2022]

LEAD	MISSION CONTROL	PAD	RANGE SAFETY	FIREFIGHTER
DANIEL FRANK	GEORG MIKULA	ANDREAS UNGERSBÖCK	TARAS WEINL	MICHAEL POHN

TESTTYPE	Second static fire of µHoubolt
TESTGOAL	Successful test of new combustion chamber
CHANGES	New combustion chamber, New batch of igniters
ADDITIONAL INFO	-

Fails and Learnings

1. Igniter failed at first try. New igniters were installed, and a second test has been conducted on the same day.
2. Hard start of engine. Probably due to underperforming igniters. Further inspection and comparison to old batches will be done.
3. Due to the explosion of the combustion chamber, the new prototype couldn't be tested. Further tests will be carried out.

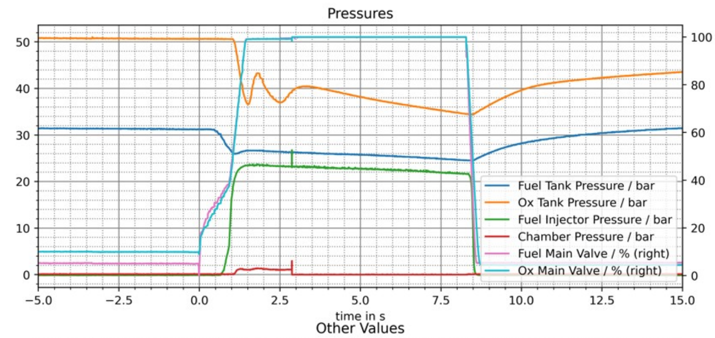
Test summary

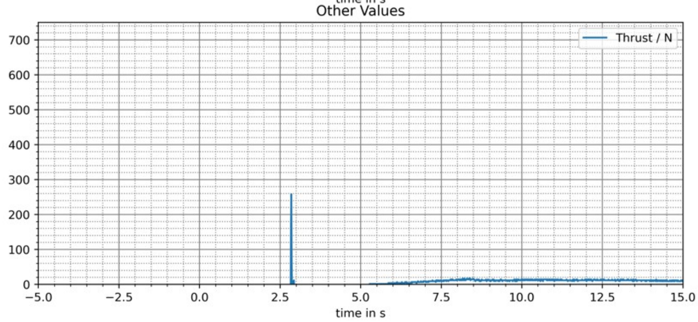
For the second test we used a similar ablative combustion chamber made of carbon fibre and epoxy resin to exclude any manufacturing flaws of the previous chamber.

Tanking went flawlessly but engine start-up failed due to a malfunction of the igniters. The test was aborted, and the oxidizer tank was emptied.

After making a second igniter batch the test samples behaved as expected. Tanking went as expected and igniters ignited but they didn't produce a hot enough flame because they were installed upside down, which resulted in a hard start and in an explosion of the combustion chamber.

Diagrams and Pictures

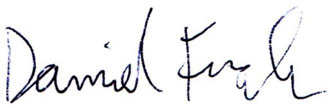




Signature 1

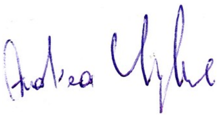
Georg Mikula

Signature 2



Daniel Frank

Signature 3



Andreas Ungersböck

µHoubolt Test Protocol

[3RD STATIC FIRE]

[28.05.2022]

LEAD	MISSION CONTROL	PAD	RANGE SAFETY	FIREFIGHTER
GEORG MIKULA	MARKUS PINTER	ANDREAS UNGERSBÖCK	DANIEL FRANK	MICHAEL POHN

TESTTYPE	Third static fire of µHoubolt
TESTGOAL	Successful test of new combustion chamber, successful internal control operation
CHANGES	New combustion chamber, new batch of igniters, new firmware code for internal operation
ADDITIONAL INFO	-

Fails and Learnings

1. For simplicity, a debugging and flash cable was connected to the ECU inside the rocket. Due to a large fireball after burnout, it was destroyed, and the ECU stalled until a reset.

2. The combustion chamber failed at the throat but held up at the walls.

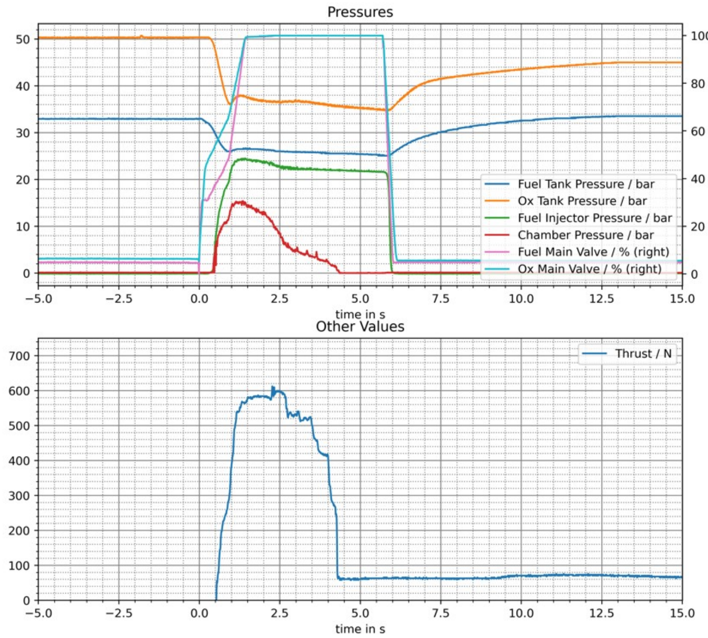
Test summary

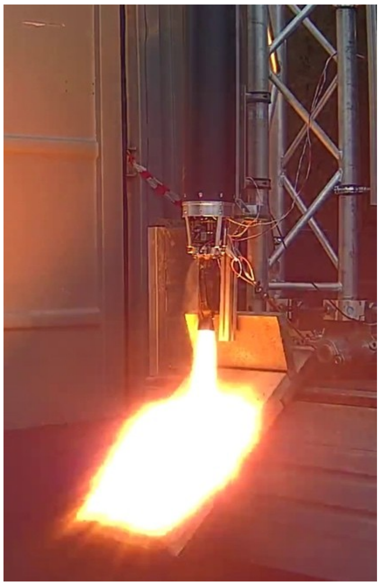
For the third test we used a similar ablative combustion chamber made of carbon fibre and epoxy resin to exclude any manufacturing flaws of the chamber used for the first static fire.

Tanking went flawlessly. Internal control operation was successful. Engine start-up was nominal. After approximately two seconds the combustion chamber broke at the throat which led to a rapid loss of pressure inside the chamber. Interestingly the produced thrust stayed above 400N even with the missing throat.

Due to the new internal control sequence, the main valve opening curves where modified to achieve a softer engine start-up which worked perfectly.

Diagrams and Pictures





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Georg Mikula

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Markus Pinter

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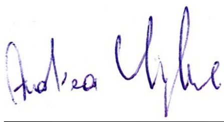


Andreas Ungersböck

µHoubolt Test Protocol

[4TH STATIC FIRE]

[31.05.2022]

LEAD	MISSION CONTROL	PAD	RANGE SAFETY	FIREFIGHTER
GEORG MIKULA	MARKUS PINTER	DANIEL FRANK	ANDREAS UNGERSBÖCK	MAX SPANNRING

TESTTYPE	Fourth static fire of µHoubolt
TESTGOAL	Successful test of new combustion chamber, successful internal control operation
CHANGES	New combustion chamber, new batch of igniters
ADDITIONAL INFO	-

Fails and Learnings

1. Internal control operation didn't work as expected in a dry run. We reverted to the external test sequence.
2. The new combustion chamber held up with the temperature at the wall, but the throat diameter enlarged very rapidly which resulted in minor decreased performance.

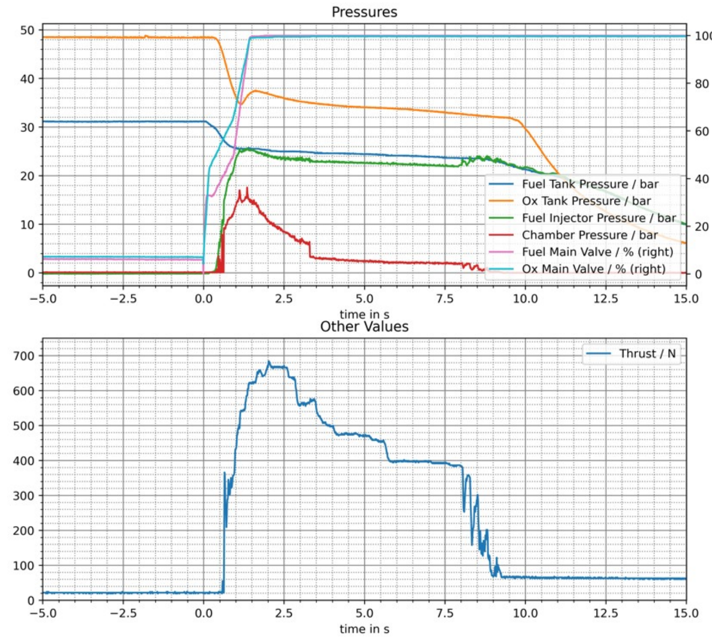
Test summary

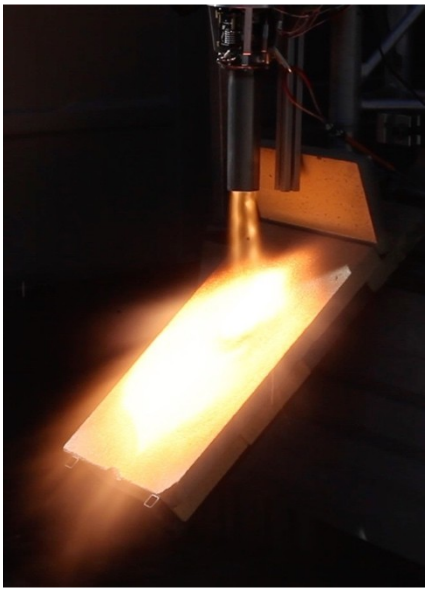
For the fourth test we used a new ablative combustion chamber design made of phenolic resin cotton fabric composite as the liner surrounded by a steel casing, since we had no aluminium tube at hand. Our focus lied in testing the liner.

Tanking went flawlessly. Internal control operation didn't work so we used the external test sequence instead. Engine start-up was nominal. Slower ramp up was clearly visible again.

At start we achieved our highest measured thrust of 680N. After an expanded throat it decreased to 400N. Overall it was our best static fire yet.

Diagrams and Pictures





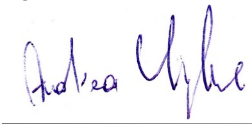
Signature 1

Georg Mikula

Signature 2

Markus Pinter

Signature 3



Andreas Ungersböck

µHoubolt Final Static Fire Test Protocol

30.09.2022

LEAD	MISSION CONTROL	PAD	RANGE SAFETY	FIREFIGHTER
Georg Mikula	Markus Pinter	Daniel Frank	Laurens Tanzer	Bernhard Hansemann

TESTTYPE	Static Fire Propulsion test
TESTGOAL	Verify correct function of finalized propulsion system
CHANGES	Rebuild of flown system, no changes
FAILS AND LEARNINGS	Propulsion performance sufficient, thrust measurement faulty

Test summary

On 30.09.2022, a static fire test of the final propulsion system was executed. The design of the system was before proven in numerous static fire tests and a test flight, but the test flight led to a destruction of the vehicle. The goal of this test was to verify the performance of the newly built propulsion system, which aims to closely resemble the old one.

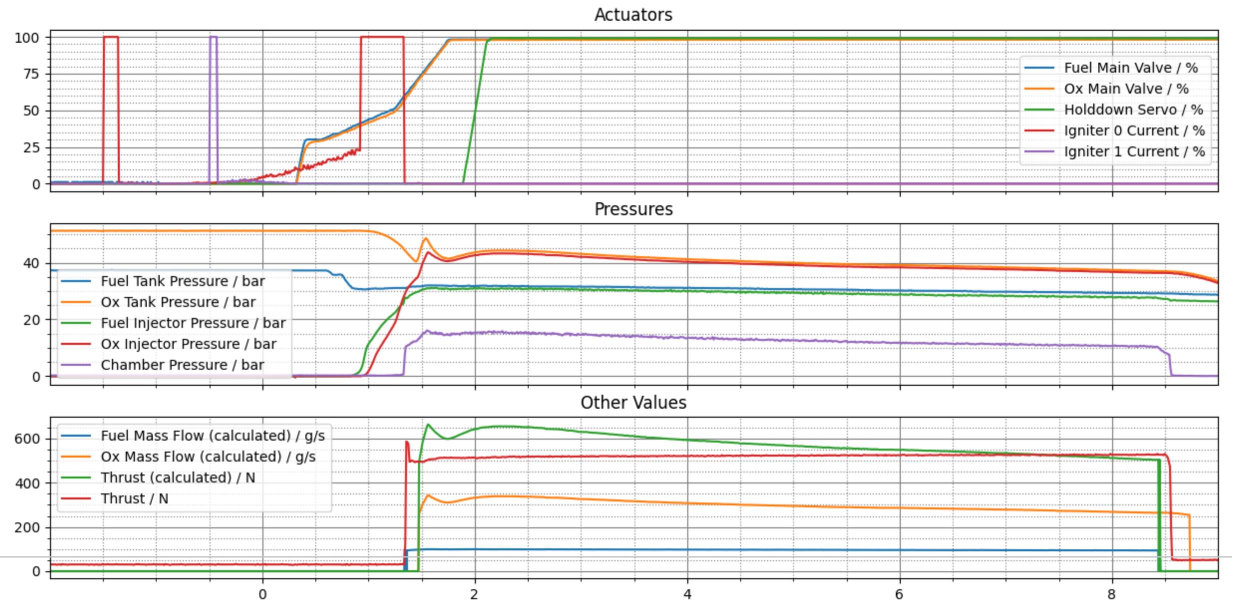
The Propulsion system was integrated into the body tube, so the parts missing for a complete vehicle were the fincan, the payload, avionics and recovery stack and the nosecone. In addition to the three propellant tank and chamber pressure sensors that will be used in flight, two additional injector pressure sensors were installed, they can be seen sticking out in the images.
One of them interferes with the automatic disconnection of the oxidizer fill umbilical, it was therefore disconnected manually in this test. The two pressurant fill umbilicals were remotely disconnected, as per the standard launch procedures.

On the launch pad, the holddown system was replaced with a thrust measurement system and the vehicle was additionally secured by two more methods to make a flyaway impossible. The servo actuating the holddown was placed in the cameras view to verify its function.

The launch preparation followed the standard procedures and the propulsion avionics were running flight ready firmware, leading to a realistic test.

The test went well and the data shown below looks mostly nominal.
The igniters fire 2-3s before the main valves open to get them to burn well. The main valves cross the opening point slowly to avoid hard starts. Shortly after, the holddown servo moves to release the rocket if the chamber pressure is nominal.
The pressure values look as expected with the injector pressures rising to almost the tank pressures when the main valves open. The chamber pressure rises to slightly above 15bar before slowly dropping as the ablative throat burns away.
The injector pressures are used to calculate propellant mass flows using models refined in previous tests, the calculated total ejected propellant mass correlates to the calculated propellant mass based on tank volumes and the propellant’s density in the tank after filling. The mass flows are then used to calculate an expected thrust.
The measured thrust does not quite meet the expectations and oddly rises during the burn, but this is due to a problem with the thrust measurement system that has been seen before. Even when assuming an engine efficiency much lower than in previous tests (done for the plot below), the thrust is calculated to have been over 600N in the beginning.

Diagrams and Pictures



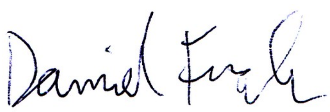
Signature 1

Georg Mikula

Signature 2

Markus Pinter

Signature 3



Daniel Frank

6.2.4 Liquid Propellant loading and off-loading

Liquid propellant loading and offloading test

[TEST START: 15:46]   [TEST END: 16:23]   [TEST DATE: 13.05.2022]

LEAD	MISSION CONTROL	PAD	RANGE SAFETY	FIREFIGHTER
GEORG MIKULA	Markus Pinter	Johann Breyner	Daniel Frank	Andreas Ungersböck

TESTTYPE	Cold Flow
TESTGOAL	Demonstrate that the rocket can be fueled and defueled in a controlled manner using the prepared systems and procedures.
CHANGES	-
FAILS AND LEARNINGS	Coupling of fuel pump difficult → reengineered connection
ADDITIONAL INFO	

Test summary

For this test the liquid propellants were loaded into the rocket as they would be before launch. First the fuel tank was filled with ethanol using a custom made handheld pump. Connecting the pump directly to the rocket via a short section of pipe with a threaded adapter at the end turned out to be possible but impractical. The pipe has since been replaced by a piece of flexible hose with a union nut at the end.
After fuel loading the fuel pump was removed and the oxidiser and pressurant loading equipment connected to the rocket via the strongback. Propellant and pressurant loading was conducted according to the checklist without difficulties.

For the offloading test, the propellant tanks were brought to nominal pressure. First the oxidiser main valve was opened, allowing the nitrous oxide to vent through the engine out of the rocket. After waiting five minutes, in order to give the oxidiser time to completely evaporate and the fumes to clear, the fuel main valve was opened, also emptying the fuel tank through the engine. After making sure that the system was completely depressurised the "firefighter" applied a generous amount of water to the launch pad directly below the engine, in order to dilute the ethanol to a non-flammable concentration.

This test proved that the system can be completely vented from a ready to launch condition without causing any damage to the engine or other systems and without causing the nitrous oxide to decompose or react with the combustion chamber liner.

Signature 1	Signature 2	Signature 3
Georg Mikula	Markus Pinter	Andreas Ungersböck

6.2.5 Combustion Chamber Pressure

µHoubolt Combustion Chamber Test Protocol

00:49 00:56 27.09.2022

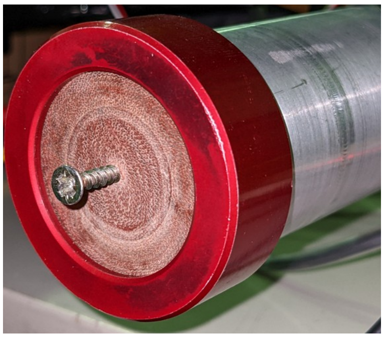
TEST BY
DANIEL FRANK ANDREAS UNGERSBÖCK LUIS BÜCHI

TESTTYPE	Proof Pressure Test
TESTGOAL	Testing the combustion chamber with a maximum static pressure
FAILS AND LEARNINGS	Test passed
ADDITIONAL INFO	Over 28bar for 2.5min

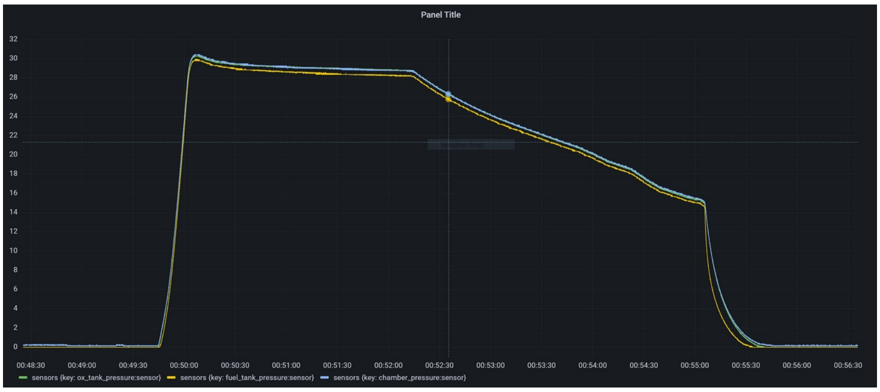
Test summary

General Description:
To ensure that the combustion chamber pressure can withstand the pressures it is exposed to during engine operation, it was statically pressure proofed. The maximum expected chamber pressure is 16bar, so a proof pressure of at least 24bar is needed to satisfy the safety factor of 1.5. The engine nominally operates for 8s, and the pressure test needs to last at least twice as long, so 2.5 min were chosen as testing duration.

Test Setup:
The test was conducted as part of a leak check of the whole propulsion system. The main propellant valves were opened, and the system was pressurized using nitrogen. The pressure was measured using three independent pressure sensors, two of them connected to the two propellant tanks and one connected to the combustion chamber. Instead of a liner (which is not pressure bearing), a sealed endcap was installed in the combustion chamber casing and held in place by the usual retainer cap.



Test Outcome:
A pressure of over 28bar was maintained in the system for about 2.5min, before the system was depressurized again. The pressures of the three sensors confirmed the value within a range of 1bar. No leaks could be detected, and the combustion chamber withstood the pressure without any problems and damages.



Signature 1	Signature 2	Signature 3
Daniel Frank	Andreas Ungersböck	Luis Büchi

6.2.6 Proof Pressure Testing Pressure Vessels

µHoubolt Test Protocol – Proof Pressure Testing of Fuel System Pressure Vessel

[TEST START: 16:30] [TEST END: 19:30] [TEST DATE: 31.08.2022]

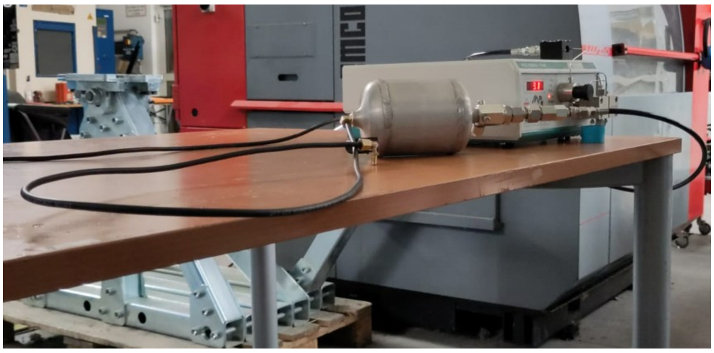
TEST BY
GEORG MIKULA PAULA-MARIA HANDLE LUIS BÜCHI

TESTTYPE	Proof Pressure Test
TESTGOAL	Testing the fuel tank with a maximum static pressure

Test summary

General Description:
To ensure the tank's functionality under high pressure conditions, the fuel system was statically pressure proofed. As the safety relief valve works in a range of 30 to 60 bar and the fuel tank pressure is defined as 40 bar, the relief valves were set to 60 bar. The conducted test required a pressure 1.5 times larger than the maximum operating condition, so the test was conducted with 90 bar and water as operating fluid due to safety and expedience reasons. Previously performed static fire tests proved the maximum system working time after pressurizing to be around 30 minutes. Thus, the testing duration was expanded to 2 hours (4 times larger than the maximum system working time) to guarantee that the tank withstands the 90 bar pressure over a long period of time.

System Setup:
The system was pressurized using the HPLC Compact Pump which was connected via Swagelok connections to the fuel manifold. To ensure that the overpressure doesn't damage any active component, none of them were screwed into the manifold. Instead, all the ports were closed off with fitting plugs. The fuel manifold itself was in turn connected to the top of the tank, to which below the flight configuration was applied, ending the testing system with the fuel main valve.




Test Outcome:
The fuel vessel endured the static pressure of 90 bar over a 2-hour duration without any negative effect on the material or components. Gentle cracklings occurred during the pressurizing processes, but no damages or leakages were found during or after the testing duration.

Signature 1

Georg Mikula

Signature 2

Paula-Maria Handle

Signature 3

Luis Büchi

µHoubolt Test Protocol – Proof Pressure Testing of Oxidizer System Pressure Vessel

[TEST START: 13:00] [TEST END: 16:00] [TEST DATE: 31.08.2022]

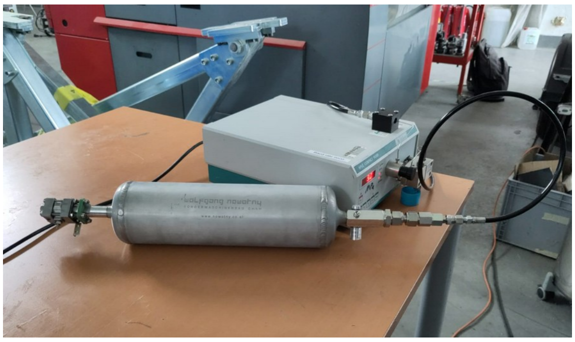
TEST BY
GEORG MIKULA PAULA-MARIA HANDLE LUIS BÜCHI

TESTTYPE	Proof Pressure Test
TESTGOAL	Testing the oxidizer tank with a maximum static pressure

Test summary

General Description:
To ensure the tank's functionality under high pressure conditions, the oxidizer system was statically pressure proofed. As the safety relief valve works in a range of 30 to 60 bar and the oxidizer tank pressure is defined as 50 bar, the relief valves were set to 60 bar. The conducted test required a pressure 1.5 times larger than the maximum operating condition, so the test was conducted with 90 bar and water as operating fluid due to safety and expedience reasons. Previously performed static fire tests proved the maximum system working time after pressurizing to be around 30 minutes. Thus, the testing duration was expanded to 2 hours (4 times larger than the maximum system working time) to guarantee that the tank withstands the 90 bar pressure over a long period of time.

System Setup:
The system was pressurized using the HPLC Compact Pump which was connected via Swagelok connections to the oxidizer manifold. To ensure that the overpressure doesn't damage any active component, none of them were screwed into the manifold. Instead, all the ports were closed off with fitting plugs. The oxidizer manifold itself was in turn connected to the top of the tank, to which below the flight configuration was applied, ending the testing system with the oxidizer main valve.



Test Outcome:
The oxidizer vessel endured the static pressure of 90 bar over a 2-hour duration without any negative effect on the material or components.

Signature 1

Georg Mikula

Signature 2

Paula-Maria Handle

Signature 3

Luis Büchi

6.2.7 Burst Pressure Testing Pressure Vessels

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6.2.8 Test of SRAD flight computers with capability of actuating the recovery systems

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6.3 Hazard Analysis Report

6.3.1 Liquid Propulsion System

The liquid propulsion system contains by definition several hazardous materials and components:

- Igniter
- Ethanol
- Nitrous Oxide

In the following the hazards related to these materials and the mitigations for them are laid out.

- Transport
  - **Igniter**

    Hazard: As the igniter is unsurprisingly flammable it poses a risk of starting to burn at an unwanted time.

    Mitigation: The igniter mixture is put together on premise shortly before needing it, during transport the individual components are not dangerous.
  - **Ethanol**

    Hazard: While not burning quickly, Ethanol is flammable and evaporates quickly.

    Mitigation: The ethanol gets transported and stored in their original packaging (plastic bottles).
  - **Nitrous Oxide**

    Hazard: Tipping over could break the bottle open.

    Mitigation: Only gets transported with the safety lid on.
- Usage
  - **Igniter**

    Hazard: Igniter going off after being installed on the combustion chamber or while being manufactured.

    Mitigation: The ignition system is only armed once the RBF pin is fully removed from the PMU, which happens just before launch preparations are done and the team vacates the launch pad. Before then, even on a misfire in the electronics, the igniter voltage to start the burning is physically disconnected. The person manufacturing the igniters wears eye protection and gloves. The heating plate is closely monitored to ensure proper heating temperature. All other people not involved hold their distance.
  - **Ethanol**

    Hazard: Ethanol spills during handling could pose a fire risk on and around the pad.

    Mitigation: The ethanol gets filled into a large syringe away from the pad and then pushed from the syringe into the fuel tank. This removes potential spills from the pad.

– **Nitrous Oxide**

  Hazard: The nitrous oxide can explosively decompose.

  Mitigation: All parts that get in touch with the N2O get rigorously ox-cleaned. Filling the N2O tank in the rocket is one of the last steps of the launch checklist just before launch. Just one experienced member of personnel is at the launch rail. He/She opens the N2O gas cylinder. After some checks the last person leaves the launch rail. Only then the tanking of the N2O begins, controlled remotely from Mission Control. See section 4.2.6 for details on the oxidizer loading procedure.

6.3.2 Slingshot Mechanism in Recovery System

Hazard: The slingshot mechanism of the recovery system stores considerable potential energy when loaded (See section 3.4). During launch preparations this has to be done a while before the pad crew vacates the premises and as such the slingshot mechanism could (mis)fire when personnel is around. This could happen if the knot loosens or the recovery system fails and prematurely detects apogee on the pad.

Mitigation: As soon as the mechanism is loaded a cable tie gets wrapped around the clampband which stops the mechanism from firing until removed right before the crew leaves the launch pad.

6.4 Risk Assessment

FAILURE MODE	MISSION PHASE	FAILURE PROBABILITY	MISHAP SEVERITY	CRITICAL RANKING	TEAM'S COMMENTS AND JUSTIFICATION
Connection loss to the pad via radio link	Before the flight	1	2	2	Numerous test shown that the radio link is reliable
Igniter material looses quality due to air humidity	Before the flight	1	1	1	Every component gets transported in air tight containers
Altimax fires recovery at the pad	Before the flight	1	1	1	Altimax will be armed shortly before the launch
Ox cleaned components get contaminated	Anytime	1	2	2	Components can be cleaned at EuRoC
Cooling/Heating system pump failure	Before the flight	1	1	1	Leads to slower ox filling
Automatic umbilicals disconnect doesn't work properly	Before the flight	1	2	2	Leads to delays, but can be recycled
Running out of ice due to launch delay	Before the flight	2	1	2	Leads to slower ox filling
Inefficiency of oxfilling due to sun/heat	Before the flight	2	1	2	Leads to slower ox filling
Malfunctioning of our custom made turbo servos due to sun/heat	Before the flight	1	2	2	Backup turbo servos available, leads to delays
Igniters get installed the wrong way round	Before the flight	1	2	2	Led to hardstarts in past tests, special care during installation should mitigate this.
ECU does not detect ignition properly	Ignition phase	2	1	2	Numerous test shown that the ECU can detect ignition reliably
Electrical umbilical gets seperated to early due to vibrations - holddown wont open automatically	Ignition phase	1	2	2	Manual release can be triggered
Insufficient ignition leads to hard start	Ignition phase	1	2	2	Numerous test shown that our ignitions is reliable
Malfunction of the combustion chamber - nozzle gets ejected	Ascent phase	1	2	2	Leads to slightly lower thrust, and would only decrease the apogee
Wrong connector to oxidizer or pressurant bottle provided at EuRoC	Before the flight	1	2	2	Many different adapters are prepared in the case the bottle has not the expected connector
Faster wear down of the combustion chamber	Ascent phase	1	2	2	Leads to slightly lower thrust, and would only decrease the apogee

Human error	Anytime	2	1	2	The whole system is designed so that a single human error can not cause any major impact on the mission
Recovery failure, because the parachute is not ejected	Descent phase	1	3	3	Both recovery flight tests showed that the system is reliable, even if the drogue chute rips off
Recovery failure, because parachute rips off	Descent phase	1	3	3	Shockabsorbers are installed to minimize the risk of the lines ripping, in addition extra thick lines are used
Manual holddown releases to early	Ignition phase	1	2	2	Manual release can only be triggered after 2.5s so that an experienced team member has enough time to decide
Filter of the pump could get dirty - no cooling/heating system	Before the flight	1	1	1	Leads to slower ox filling
Flame diverter gets destroyed	Ignition phase	2	1	2	Can only happen if holddown doesn't release, and has no significant impact
Server overheats and shuts down	Before the flight	1	3	3	Server temperature has to be monitored, if the server is not operational the launch has to be aborted
Eggtimer TRS cannot bulit up connection	Before the flight	1	2	2	Numerous test shown that the TRS connection is reliable

6.5 Checklists

Lead

Name: .... PTT ID: .... Date: ....

> **Important Note:**
> Coordination with Mission Control is required. Tasks for Mission Control are underlined. Mission Control does not have a dedicated checklist and therefore needs to be told every underlined instruction.
> Ensuring that the tasks on your checklist are done is your responsibility.
> Mission Control is just here to help.

L1. Assign roles (PTT Adapter ID) ☐

- Mission Control: ....
- Rocket Prep 1: .... (   )
- Rocket Prep 2: .... (   )
- Pad 1: .... (   )
- Pad 1: .... (   )
- Pad 2: .... (   )
- Pad 3: .... (   )
- Range Safety: .... (   )
- Documentation: ....

L2. Request Launch Clearance ☐

L3. Instruct documentation and pad personal to start with prep checklist ☐

Preps

L4. Pull RBF ☐

L5. Run test sequence ☐

L6. Run launch sequence ☐

L7. Insert RBF halfway ☐

L8. Set Holddown Settings ☐

L9. <u>Close fuel main</u> ☐

L10. Request Fueling ☐

L11. Check water state ☐

L12. Check igniters deactivated ☐

L13. Install igniters ☐

- Use fresh PTFE seal
- E-match side up
- Note which igniter (0/1/2/...) installed where (A/B)

- Fully tighten screws

L14. Connect Fincan to Body Tube ☐

L15. Take team photo ☐

L16. Vacate area ☐

L17. Everyone to their post ☐

Filling Preps

L18. Close ox tanking ☐

L19. Open ox vent ☐

L20. Close pressurant tanking ☐

L21. Open pressurant vent ☐

L22. Request Final Preps ☐

L23. Verify igniter continuity ☐

L24. Request Ox Filling clearance from LCO ☐

Terminal Launch Preparation

L25. Go/NoGo Poll ☐

L26. Arm Altimax ☐

Filling
Monitor Ox bottle, ox tank, pressurant, fuel tank pressures

L27. Close ox main ☐

L28. Set Supercharge 30bar, Hysteresis 1bar ☐

L29. Enable Supercharge ☐

L30. Close pressurant vent ☐

L31. Pressurize tanks ☐

L31.1. Open pressurant tanking

L31.2. Quickly close pressurant tanking after first plume

L32. Tare thrust ☐

L33. Close ox vent ☐

L34. Slowly open ox tanking ☐

L35. Activate heating cycle ☐

L36. Quickly close ox tanking, after first plume ☐

- L37. Activate cooling cycle ☐
- L38. Wait for stable vent frequency ☐
- L39. Slowly open ox tanking ☐
- L40. Quickly close ox tanking, after first plume ☐
- L41. Open Ox vent ☐
- L42. Start remote cameras ☐
- L43. Enable internal cameras ☐
- L44. Set Supercharge 60bar, Hysteresis 1bar ☐
- L45. Open pressurant tanking ☐
- L46. Wait for stable pressures ☐
- L47. Close pressurant tanking ☐
- L48. Open pressurant vent ☐

Launch

- L49. Go/NoGo (TRS camera points to receiver) ☐
- L50. Activate umbilical retract ☐
- L51. Verify clean separation ☐
- L52. Switch to internal power ☐
- L53. Launch ☐

Safe GSE and Rocket after Abort

Rocket is fully pressurized

- L54. Open supercharge ☐
- L55. Close supercharge after pressure in tank is zero ☐
- L56. Verify no heat sources ☐
- L57. Open fuel main OR Wait for fuel bleed to vent ☐

Safe GSE

- L58. Stop remote cameras ☐
- L59. Stop pad cameras ☐
- L60. Instruct Pad to close Ox bottle ☐
- L61. Instruct Pad to close Pressurant bottle ☐
- L62. Vacate area ☐

L63. Open ox tanking ☐

L64. Open pressurant tanking ☐

L65. Verify all pressures are zero ☐

L66. Announce "safe state" ☐

Pad Preperation

Name: .... PTT ID: .... Date: ....

- P1. READ THE WHOLE CHECKLIST BEFORE STARTING ☐

Pad preparation

- P2. Pack tools ☐
- P3. Place trailer in front of container ☐
- P4. Connect GSE to Server ☐
- P5. Fill hot water ☐
- P6. Fill cold water ☐
- P7. Weight Ox bottle and check if enough ☐
- P8. Install Ox bottle ☐
- P9. Instruct MC to activate cooling cycle ☐
- P10. Add ice to cold water, regularly check and add more ☐
- P11. Check sensors
  - P11.1. Hot water temperature ☐
  - P11.2. Cold water temperature ☐
  - P11.3. Mantle water temperature ☐
  - P11.4. Pressurant pressure ☐
  - P11.5. Ox pressure ☐
- P12. Check actuators (verify movement and calibration)
  - P12.1. Ox tanking valve ☐
  - P12.2. Ox vent valve ☐
  - P12.3. Pressurant tanking valve ☐
  - P12.4. Pressurant vent valve ☐
  - P12.5. Umbilical retract ☐
  - P12.6. Hot water pump ☐
  - P12.7. Cold water pump ☐
  - P12.8. Holddown ☐
- P13. Move trailer at test position ☐
- P14. Install Pressurant bottle ☐
- P15. Mount flame diverter ☐

Rocket mounting (Static Fire)

- P16. Slide rocket into rail ☐
- P17. *Mount T-Nut underneath lower rail button ☐
- P18. *Mount scale above lower rail button ☐
- P19. *Secure rocket with plate on top ☐
- P20. *Secure rocket with steel cable ☐
- P21. *Secure rocket with "loose" strap ☐
- P22. Connect Ox umbilical ☐
- P23. Connect Ox pressurant umbilical ☐
- P24. Connect Fuel pressurant umbilical ☐
- P25. Connect Electrical umbilicals ☐
- P26. Pull RBF Pin halfway ☐
- P27. Vacate area, everyone to their position. ☐

Rocket mounting (Launch)

- P28. Slide rocket into rail ☐
- P29. Mount scale underneath lower rail button ☐
- P30. Secure holddown above lower rail button ☐
- P31. Ensure holddown locked ☐
- P32. Connect Ox umbilical ☐
- P33. Connect Ox pressurant umbilical ☐
- P34. Connect Fuel pressurant umbilical ☐
- P35. Connect Electrical umbilicals ☐
- P36. Pull RBF Pin halfway ☐
- P37. Vacate area, everyone to their position. ☐

Fueling

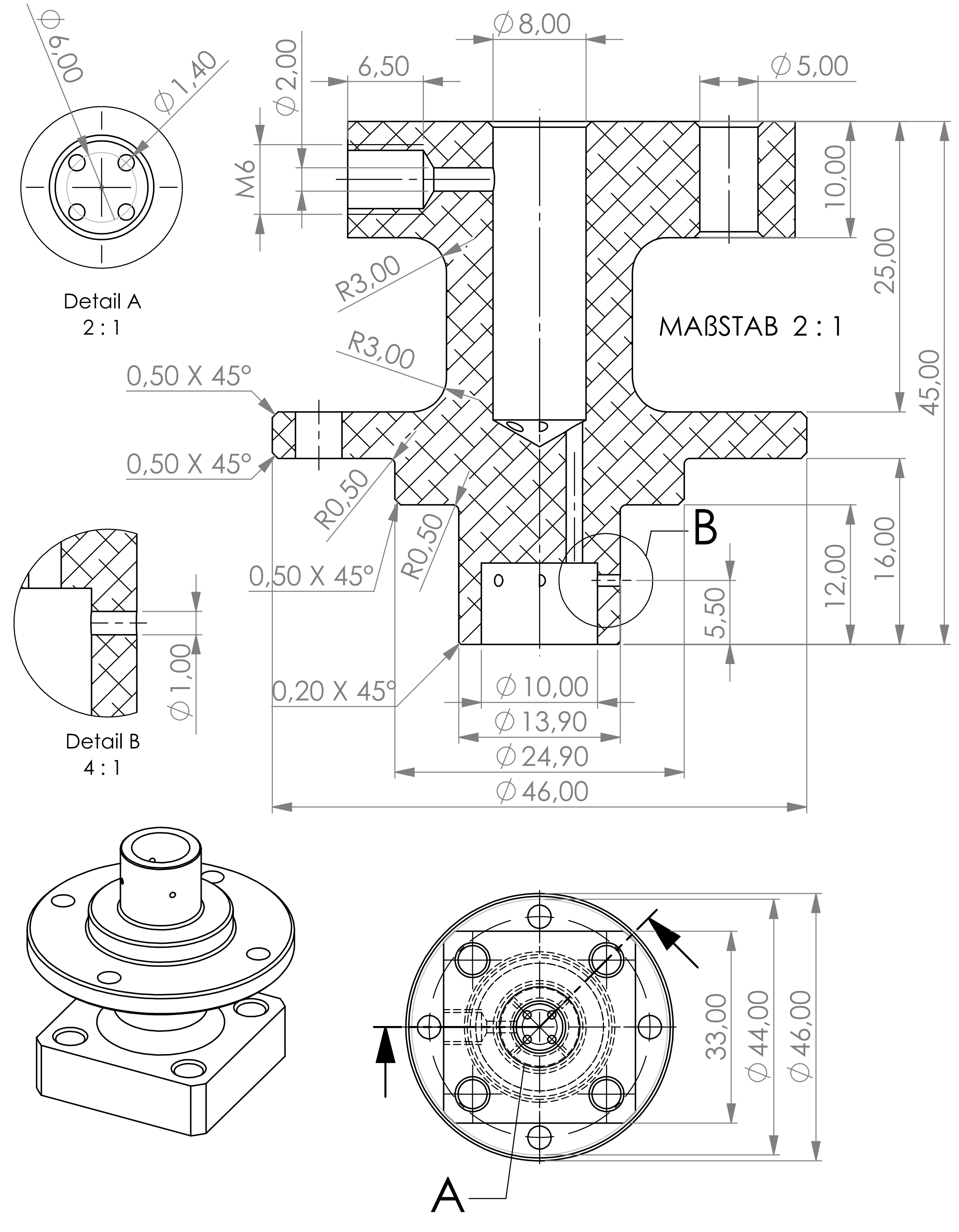
- P38. Fill fueling syringe with ethanol ☐
- P39. Fuel ☐
- P40. Clean up spills ☐

Final Preps

Constant communication to MC required

- P41. Verify holddown closed ☐

6.6 Engineering Drawings



ENTGRATEN UND SCHARFE KANTEN BRECHEN

TU Wien Space Team

	NAME	SIGNATUR	DATUM
GEZEICHNET			
GEPRÜFT			
GENEHMIGT			
PRODUKTION			
QUALITÄT			

BENENNUNG:

P37 Lower Ox Valve Flange

WERKSTOFF:

7075-T6

ZEICHNUNGSNR.

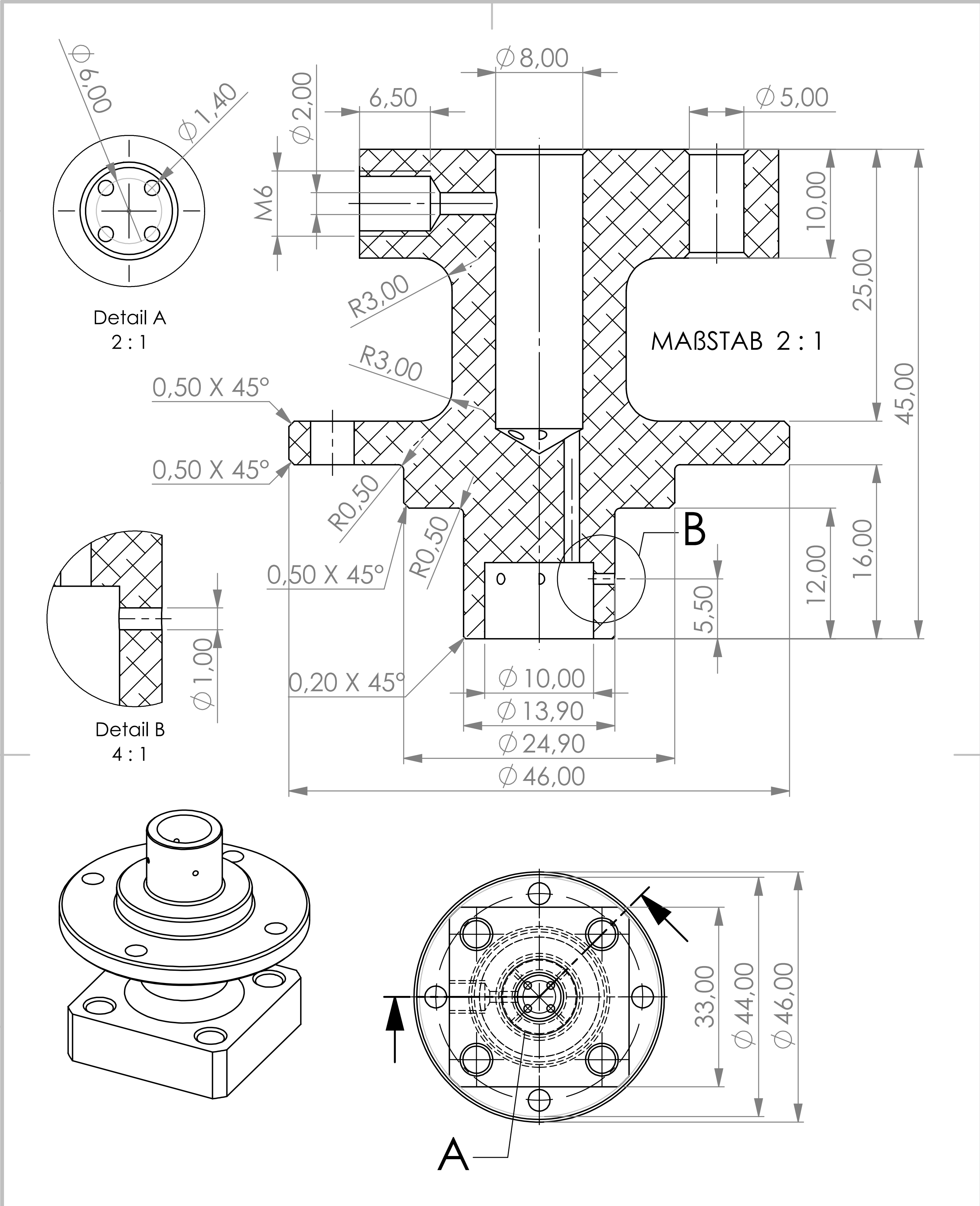
Nr.: 1.1

A4

GEWICHT:

MASSSTAB:1:1

BLATT 1 VON 1



ENTGRATEN UND SCHARFE KANTEN BRECHEN

TU Wien Space Team

	NAME	SIGNATUR	DATUM
GEZEICHNET			
GEPRÜFT			
GENEHMIGT			
PRODUKTION			
QUALITÄT			

BENENNUNG:

P37 Lower Ox Valve Flange

WERKSTOFF:

7075-T6

ZEICHNUNGSNR.

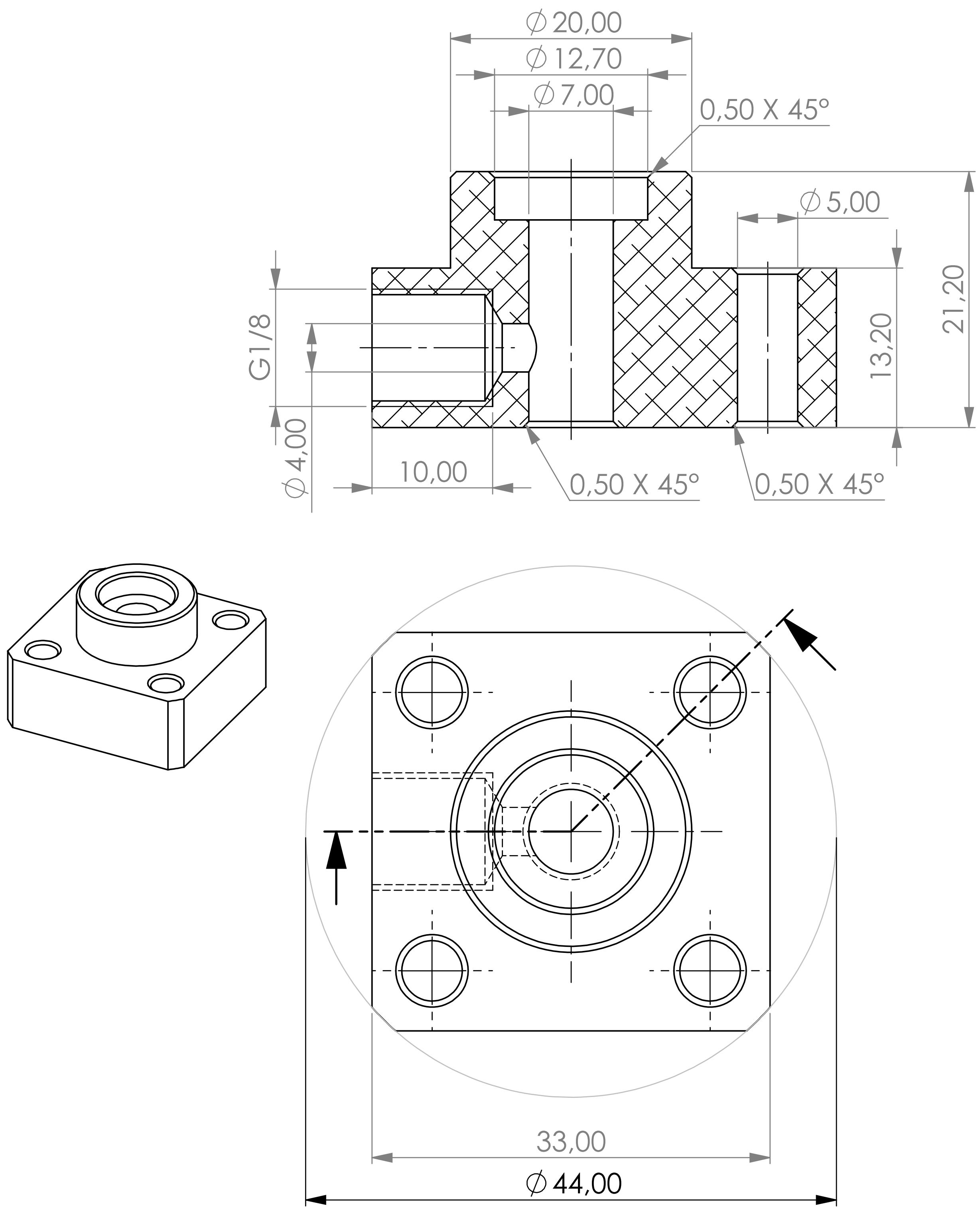
Nr.: 1.1

A4

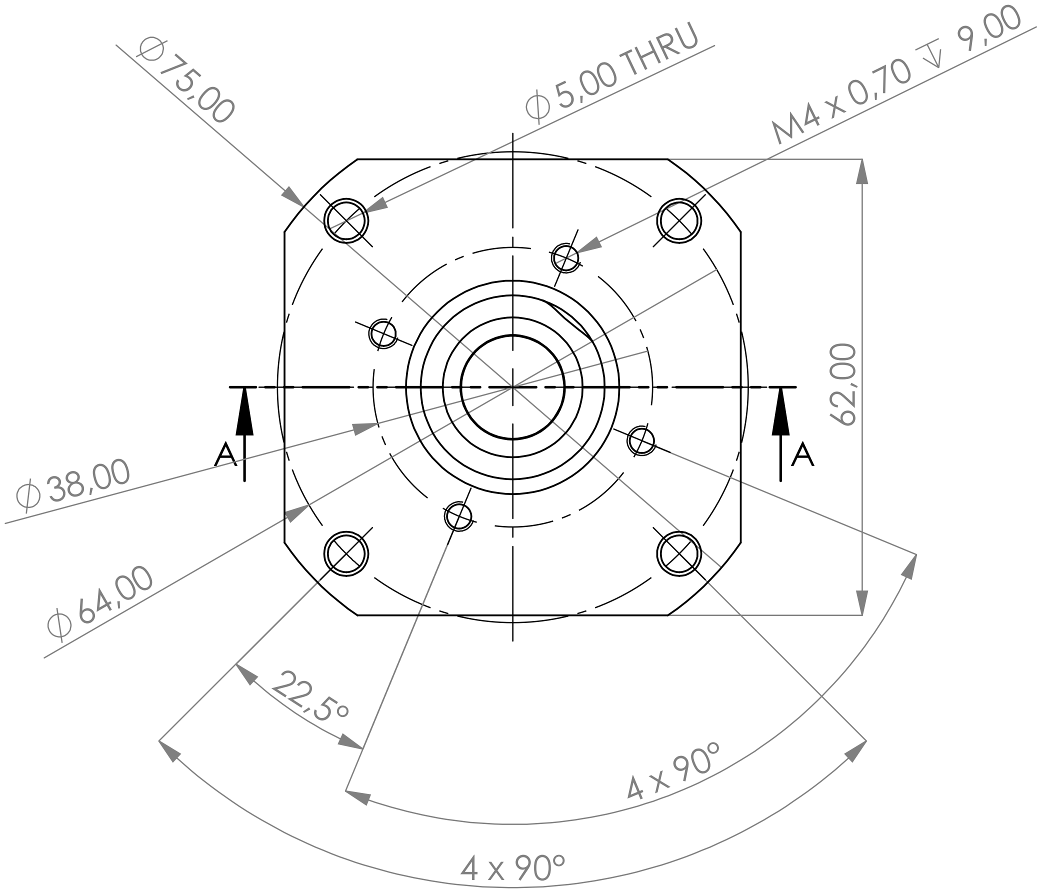
GEWICHT:

MASSSTAB:1:1

BLATT 1 VON 1



		ENTGRATEN UND SCHARFE KANTEN BRECHEN	TU Wien Space Team	
	NAME	SIGNATUR	DATUM	BENENNUNG: P40 Upper Ox Valve Flange
GEZEICHNET				
GEPRÜFT				
GENEHMIGT				
PRODUKTION				
QUALITÄT			WERKSTOFF: 7075-T6	ZEICHNUNGSNR. Nr.: 1.1 — A4
			GEWICHT:	MASSSTAB: 2:1 — BLATT 1 VON 1



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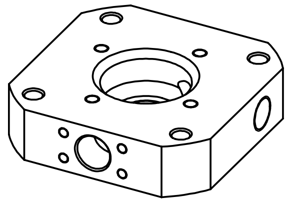

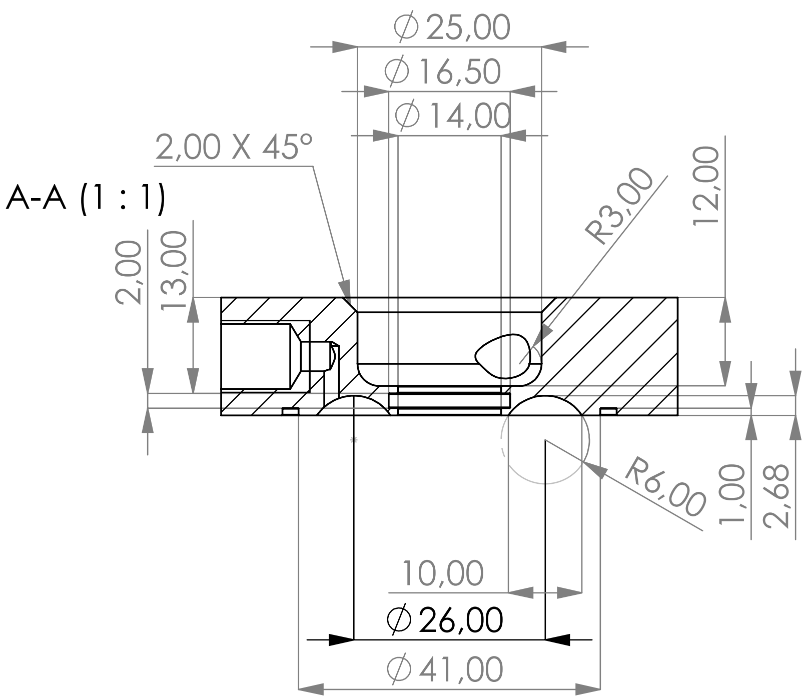

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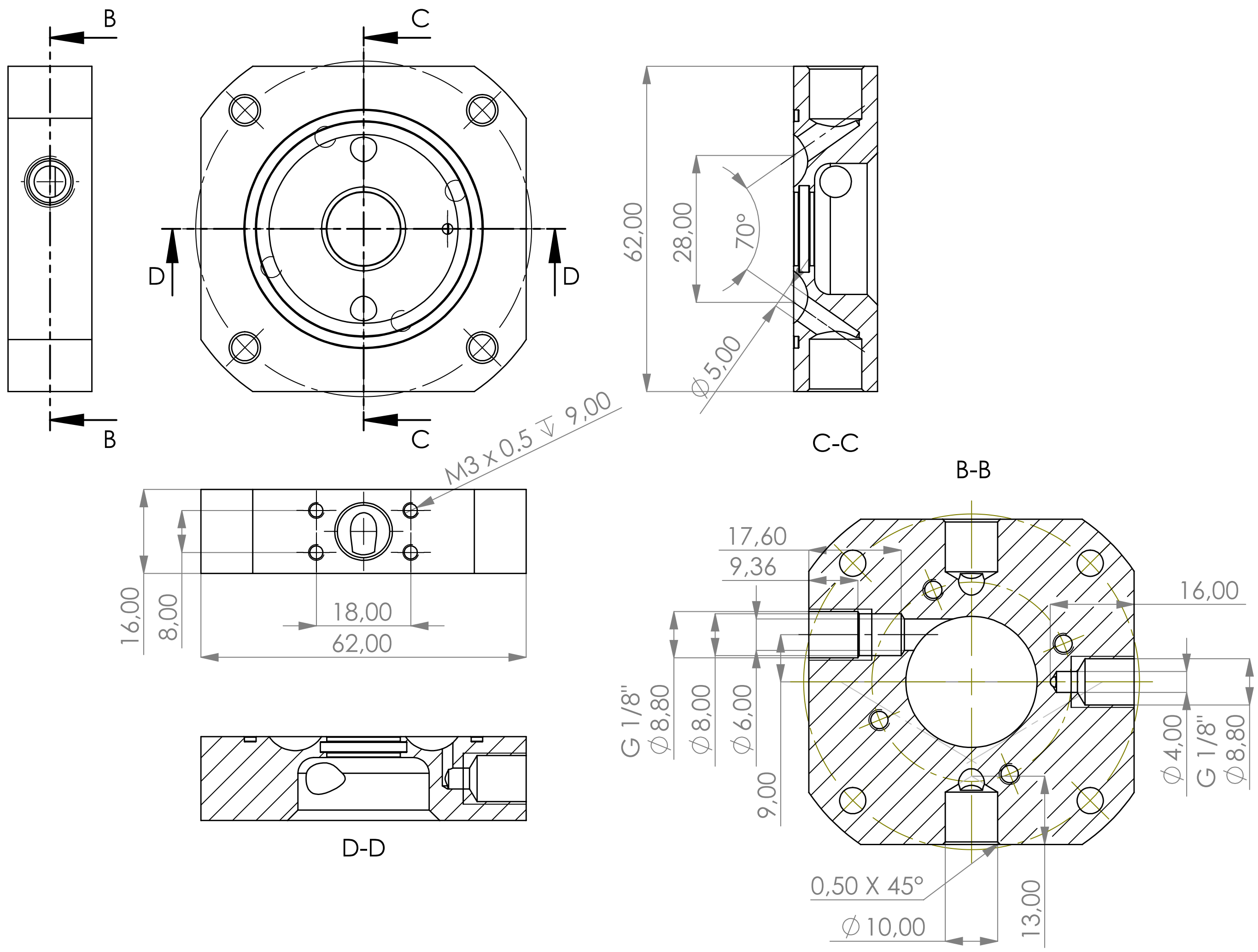

B
C
D
D
B
C
62,00
28,00
70°
⌀5,00
C-C
M3 x 0.5 ↧ 9,00
B-B
17,60
9,36
16,00
16,00
8,00
18,00
62,00
G 1/8"
⌀8,80
⌀8,00
⌀6,00
9,00
⌀4,00
G 1/8"
⌀8,80
D-D
0,50 X 45°
⌀10,00
13,00

6.7 Detailed Software Architecture

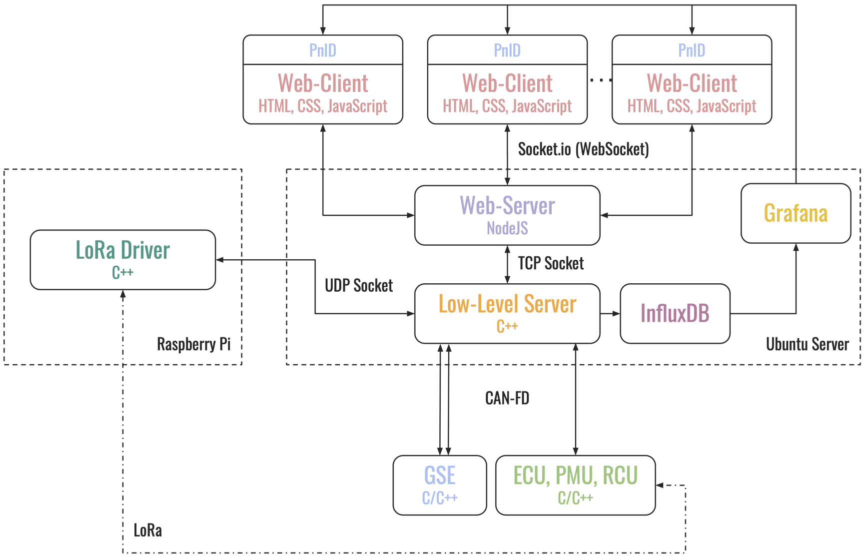

Figure 6.1: Software Architecture

6.7.1 Mission Control

Mission Control consists of a PC or Laptop running a Web-Application (Web-Client) inside a Web-Browser. It manages the communication between the Operator and the Rocket as well as the Ground Support Equipment. The Web-Client displays all measured data and actuators in a self developed interactive Piping and Instrumentation Diagram (P&ID/PnID). Every value for each P&ID element gets validated. When out of range, it is signalled by changing colour. This way, the operator doesn't have to check each number in detail but rather only has to watch for color changes, which is much more apparent.

6.7.2 Ubuntu Server

The Ubuntu Server uses a PCIe CAN bus extension card with four CAN bus ports. It is responsible for communication with the Hardware, i.e. GSE, ECU, RCU and PMU. Mission Control interfaces with the Hardware via a self developed software as depicted in figure 6.1. The connection from the server to Mission Control can either be made by a long Ethernet cable or a directed radio link depending on the necessary safety distance of MC from the pad.

6.7.2.0.1 Low-Level Server Written in C++ it is responsible for managing CAN and other time critical tasks such as logging and processing sensor data.

6.7.2.0.2 Web-Server Uses NodeJS to host the Web-Clients and synchronize data between them.

6.7.2.0.3 InfluxDB and Grafana InfluxDB is a time series based database for logging sensor data and user inputs. Grafana is used for real time plots and gauges on the Web-Client. It is also used for post-launch procedures.

6.7.3 LoRa Raspberry Pi

To communicate with the rocket during flight, LoRa is used. A self developed LoRa transceiver shield is connected to a Raspberry Pi which processes and transmits the messages over UDP to the Ubuntu Server. There it gets processed like a normal CAN Message.

References

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