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Development History and Verification of the Flight Model of a 500 N Ethanol/LOX Rocket Engine

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Abstract

In this paper we present the flight model development of our 500 N Ethanol/LOX rocket engine. It covers results from first firings of open combustions and test engines in previous test campaigns to the finalization of the flight model design and its verification this year. Performance characteristics obtained from our test rocket engines (e.g. thrust, specific impulse and characteristic velocity) are presented. The latest results of the flight model engine tests are presented to verify the previously obtained results. Furthermore, we show the application potential of oxide-oxide ceramic matrix composites for rocket engines.

1. Introduction

The SMART Rockets Project (SRP) is a student education programme at the Technische Universität Dresden (TU Dresden) with the ambitious goal of developing, testing and flying a liquid propelled sounding rocket. This project is embedded within the nationwide education programme "Studentische Experimentalraketen" (STERN), initiated and conducted by the German Space Administration (DLR) and funded by the Federal Ministry of Economics and Technology (BMWi). The main purpose of the STERN programme is the promotion of young professionals for launcher systems and a practical education of students in the field of aerospace engineering.[1]

While most universities use a hybrid engine with solid fuel and liquid oxidants, our development focuses on the implementation of a liquid-propellant rocket engine (LPRE). The rocket motor uses the "green" propellant combination of liquid oxygen (LOX) as oxidizer and ethanol as fuel.

While previous works have presented the development of the used coaxial swirl injector[4] and the utilisation of oxide-oxide ceramic matrix composites (OCMCs) in the thrust chamber design,[2] this paper will focus on the design and first experimental verification results of the engine's flight version. Therefore, section 2 summarises the previously conducted test campaigns on the ground versions of the engine (EM), before section 3 will introduce the flight model (FM) design and explain additional sensor placements for evaluating the engine's performance. Finally, the according tests are presented in section 4, including a discussion of their results. This contribution closes with section 5, which summarises the lessons learned and gives an outlook on future tests and developments.

2. Previous test campaigns

Between summer 2014 and the end of the year 2016, we conducted in total eight test campaigns with actual combustion. In this campaign series, we conducted 122 runs to ignite our fuel mixture of liquid oxygen and denatured ethanol - less than a dozen test runs did not ignite properly. Approximately half of them were combustion tests with an open flame and the other half with a variety of test combustion chambers (see also[3,6]).

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2.1 Open combustion

In our first combustion tests, we conducted two types of experiments: analysis of the effect of the recess length of the injector and different ignition systems. The recess position of the injector describes the axial distance between the LOX injector outlet plane and the one for ethanol (see also[6]). In combination with the individual spray angles, this recess length affects the mixing and atomization of the propellants. Figure 1 shows the effect of different recess positions on an open combustion flame. From these pictures, we derived the suitable recess length for our injector design. Furthermore, we experimented with different mass flows and mixture ratios. We assumed the combustion to be optimal, when a long oxygen rich (blue) flame is starting from the injector and a secondary (ethanol) spray cone - for film cooling - is still visible.

In order to find a reliable solution, we experimented with a variety of ignition systems. Neither with model rocket motor igniters nor with model rocket motors itself (type D-9) a successful ignition could be realized. But we successfully ignited the propellants with an 25 kW propane-oxygen-burner, a sparking plug, an electric heater and - which became our favorite and final system - a sparkler. Currently, we use three sparklers which are ignited by an electrically controlled bridge wire detonator enhanced by three short powder fuses.

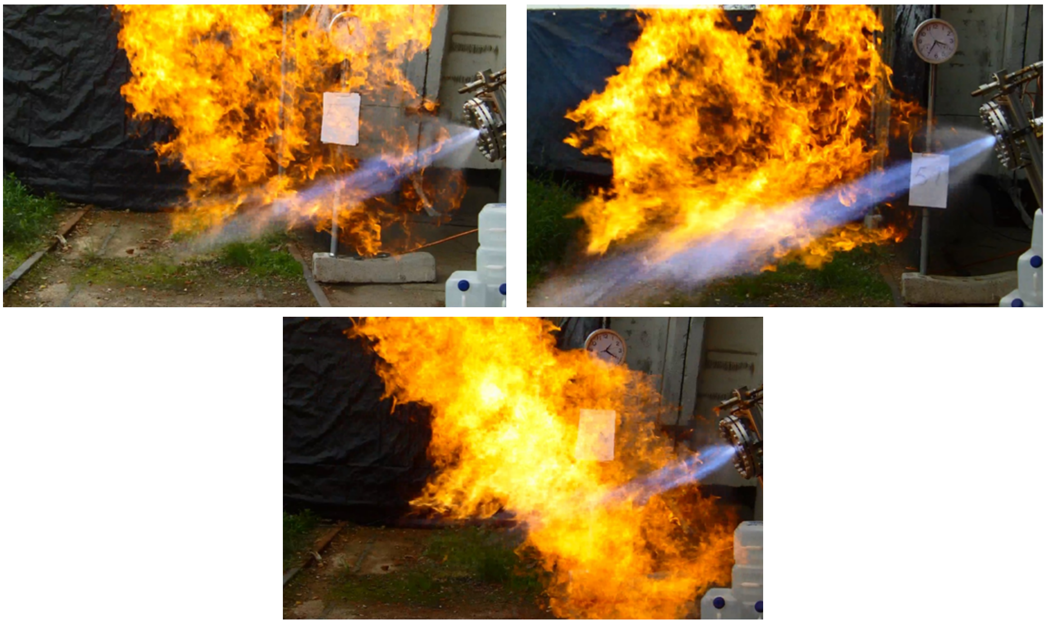

Figure 1: Open combustion tests with different recess positions at comparable mass flows [recess lengths: 5.3 mm (top left), 6.0 mm (top right) and 7.2 mm (bottom); eth. mass flow: 0.080 – 0.089 kg/s; LOX mass flow: 0.076 – 0.089 kg/s]

2.2 Test combustion chamber - steel

Also in September 2014 we had our first successful combustion test with a complete rocket engine. The engine is quite simple and consists of the load bearing steal corset and a graphite inlay in which the chamber and nozzle contour is incorporated (see fig. 2). The engine parameters are are presented in detail in the following section 3. In 2014 we reached with an oxygen-fuel-ratio ROF of 1.0 a maximum thrust of 344 N. In later tests we were able to deliver up to 440 N with slightly higher tank pressures and an increased ROF of 1.1.

2.3 Test combustion chamber - OCMC

In continuation of the rocket engine development, we made the next step towards an engine capable of flying. We replaced the heavy steel corset with one made of a oxide-oxide ceramic matrix composite (OCMC) (see fig. 3). In our OCMC, aluminum-oxide is used as the main ingredient for fiber and matrix.[2] This engine configuration with the same engine parameters as the steel version was first tested in may 2015 and reached a thrust of 370 N. Following tests in this year, the highest thrust so far realized was measured in test #122 in which 470 N were reached with mass flows of 0.120 kg/s and 0.135 kg/s ($ROF = 1.1$) for ethanol and LOX respectively.

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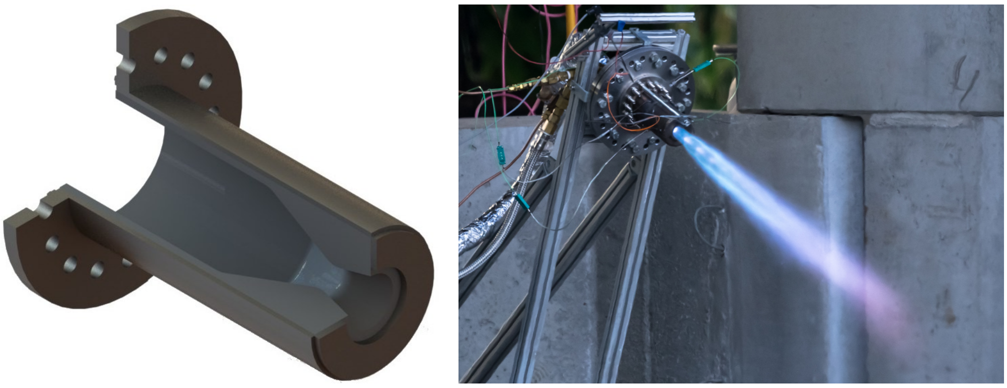

Figure 2: Test combustion chamber with steel corset (photo: ©J. Dziewiencki)

Figure 3: Test combustion chamber with OCMC corset

3. Flight model design

Based on the previous test campaigns, we made minor changes to our rocket engine engineering model (EM) in order to optimize the flight model (FM) design with respect to the target performance. We kept our concept the same, using a metallic injector head with a graphite chamber and nozzle material which is supported by an OCMC corset (see fig. 4). For detailed specifications of the EM see.[6]

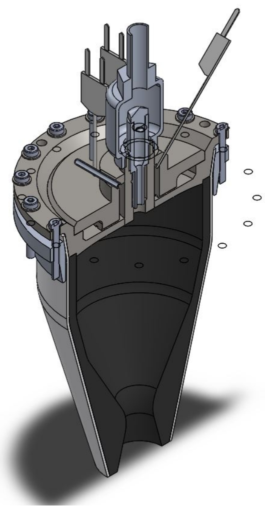

Figure 4: CAD model of the FM design

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The first and most obvious design change is the thickness and diameter of the injector head. It is made out of stainless steel 1.4301, its thickness has been reduced to 4 mm and the diameter has been reduced to the envisaged rocket diameter of 120 mm. The relevant liquid flow cross sections of the outer and inner ring collectors and the cooling channels remained the same in order to keep the FM as close as possible to the EM. The single pieces of the injector head are brazed together with a nickel brazing paste. Only the inner LOX injector and its surrounding collector are connected via threads.

One slight adaption regarding the combustion process itself has been done. Regarding the results of the previous test campaigns it became obvious that we could reach the desired thrust more easily with a less fuel rich mixture. So we decided to raise the ROF from 1.0 to 1.2 by increasing the mass flow coefficient of the LOX injector. Under the same conditions (pressure drop over the injector and back pressure from the combustion chamber) the new injector would deliver 0.15 kg/s of liquid oxygen. The new conditions we expected can be seen in table 1. All perfomance data were derived using the software "Rocket Propulsion Analysis v.1.2 Lite Edition" (RPA) based on the original code of Gordon and McBride.[5]

Table 1: Design parameters for the 500 N-engines

Engine parameters	**EM design**	**FM design**
Nominal thrust [N]	500	520
Chamber pressure [MPa]	1.5	1.6
Adiabatic chamber temperature [K]	2350	2810
Specific impulse (I_{sp}) [s]	205	217
Characteristic Velocity [m/s]	1527	1615
Oxidizer mass flow [kg/s]	0.125	0.133
Fuel mass flow [kg/s]	0.125	0.111
Oxidizer-fuel-ratio (ROF) [-]	1.0	1.2
Characterisitc chamber length [m]	1.5	1.5
Nozzle throat diameter [m]	0.018	0.018
Nozzle expansion ratio [-]	3	3

Even though the rocket engine design is in a state of a flight model, a lot of sensors were included in the injector head. So we added several temperature and pressure measuring points in the injector head.

At first, a pressure sensor for the combustion chamber has been inserted into the injector plate, measuring the chamber pressure from the downstream direction. Since the velocity in the chamber itself is rather low, the measurement error due to the dynamic pressure is expected to be less significant. Furthermore, a perpendicular chamber pressure measurement would be more of a challenge to manufacture, due to the brittle graphite.

In order to continue the characterization of the coaxial swirl injectors, pressure measurement points near the injectors were inserted. In combination with the chamber pressure measurements, the detailed injector analysis from Fiore et al[4] can be extended based on measurements during a combustion process.

Two thermocouples are used to determine the temperature of the front plate towards the combustion chamber of the injector head in the area of the cooling channels. Both are situated in the middle of a cooling rib. The first point of measurement is in the back plane of the plate and the second half way in chamber direction. This allows a first estimation of the wall temperature on the hot gas side.

Additional three thermocouples are included to measure the temperature change of the ethanol during the flow through the injector head. The first one measures the initial temperature near the feeding connector. The second one is situated at the opposite side in the outer ring collector and the third in the inner ring collector. With this data, the thermal load on the injector head can be estimated and further numerical analysis with actual measurement data supported.

4. Test campaign for the 500 N flight model

The thoroughly designed rocket engine was already tested on several campaigns throughout the last years. The latest campaign was conducted in February and March 2017. The objective of this campaign was to prove the ability of the engine to produce the desired thrust and show that the feeding system works as desired.

4.1 Campaign Overview

The campaign begun in the second week of February during which the newly acquired testing ground had to be set up for testing, including the lab and the test bench. Thereafter, the first flow tests were performed. They were necessary to

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get the parameters in the desired regions and check for any possible leaks throughout the whole feeding system. These tests showed that the desired mass flow can be achieved. Subsequently the first open combustion tests were conducted, which verified that the ignition mechanism works and can be operated with the control software. Additionally, these tests showed that it is challenging to get the desired mass flow when both propellants are used. During the separate flow tests were 0.135 kg/s mass flow for Ethanol achieved with around 1.0 MPa in the feeding lines. Within the open combustion test the flow dropped to around 0.105 kg/s although the pressure was raised to 1.5 MPs. The reason for this characteristic is assumed to be a combination of two effects. The interacting fluid flows on the one hand, and on the other side the resistance due to the pressure at the injector outlet created by the open combustion. Resulting from this behavior, the pressures for the following tests had to be adjusted to higher levels for a stable propellant mass flow near the design point. With this knowledge the first test with a mounted combustion chamber was conducted. This first test is numbered as #130 and the recorded data is listed in table 2. Because the recorded data shows a clear deviation from the design point, the pressure had to be increased for the next test. Unfortunately, during test #131 a problem with the used clamping ring occurred and the combustion chamber separated from the injector plate. As it hit the ground, parts of the OCMC corset broke and the test campaign had to be put on hold till a new one was produced. Unfortunately the software experienced a malfunction and no data was recorded for this shot. In the meantime, a new clamp was designed which offers a more solid connection. Furthermore, the data acquisition software was optimized for error-free operation.

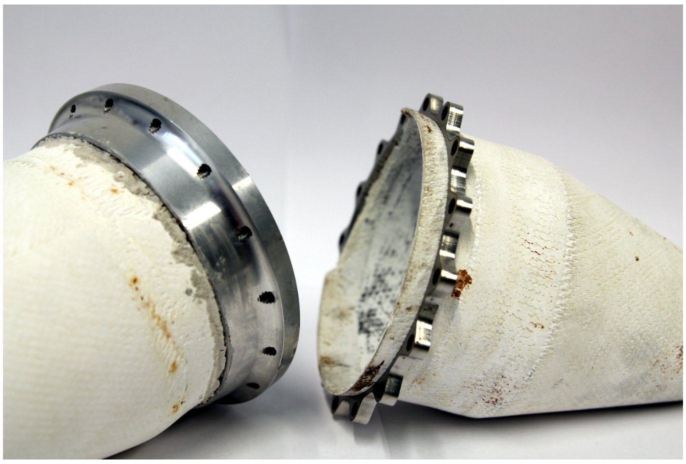

Figure 5: Comparison of new (left) and old (right) clamping ring

A week later the campaign could be continued and more tests were conducted. In the beginning of this second part we started with a mounted combustion chamber. After first attempts to reproduce test #130, the goal was to achieve higher thrust through increasing the system pressures.

Table 2: Measured and calculated chamber data

Parameter	**FM Design**	**#130**	**#135**	**#136**
Thrust [N]	520	235	320	424
Time of constant thrust [s]	20	6	9	17
Oxidizer mass flow [kg/s]	0.133	0.086	0.107	0.125
Fuel mass flow [kg/s]	0.111	0.081	0.086	0.102
Specific impulse (I_{sp}) [s]	217	144	169	190
Oxidizer-fuel-ratio (ROF)	1.20	1.06	1.26	1.23
Area-specific mass flow[a] [kg/(m^2 s)]	959	655	758	893
Chamber pressure[b] [MPa]	1.6	1.05	1.28	1.5
Adiabatic chamber temperature[b] [K]	2810	2497	2681	2645
Characteristic velocity[b] [m/s]	1615	1608	1684	1679

[a] At the nozzle throat
[b] Derived values with RPA

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4.2 Detailed Analysis of test #136

In the following our last test (#136) will be explained in detail. This test had results closest to the design values and can therefore be a reference for future campaigns. Figure 6 shows a diagram of the thrust and mass flow over time.

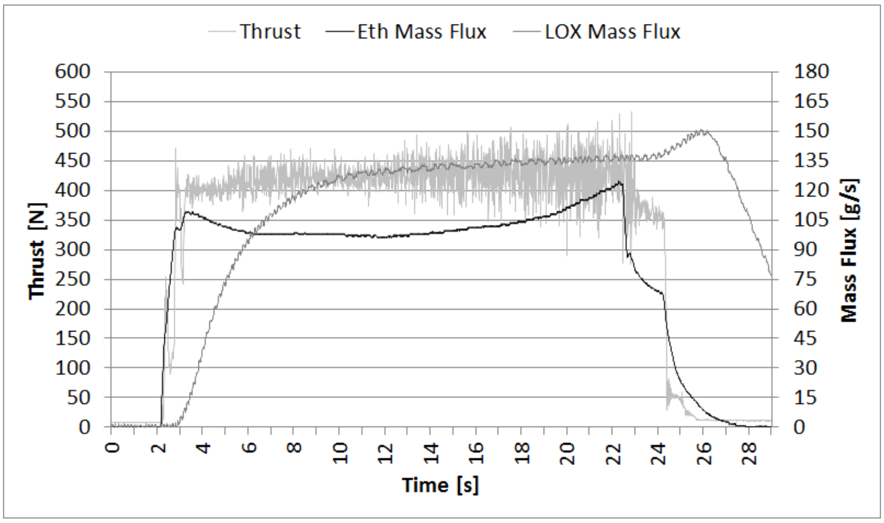

Figure 6: Thrust, LOX and ethanol mass flow - test #136

There are a couple of notable things in this depiction. First of all it can be observed that the LOX mass flow does not react as quickly as the Ethanol mass flow. In the beginning of the shot the LOX does not react at all. This dead-time can be explained with regards to the measurement system, a Coriolis force sensor. These sensors have a reaction time of one second or more and therefore explain this behavior. Next up it can be noted that the curve of the LOX mass flow does not rise quickly but rather slowly approaches its maximum. This behavior is known as a delay element of first order in control engineering. On the other side of the spectrum does the ethanol flow react instantly to the opening of the valve. Here a completely different behavior can be observed. It reacts instantly but overshoots at the beginning. After the first seconds, it reaches a semi stable state and starts to rise continuously after the first half of the shot. The first overshoot is explainable again with regards to the measurement system. In the ethanol system, the flow is checked with a measuring aperture. Under the sudden shock the aperture bends itself. This leads to a bigger cross-section which results in a rising volume flow and therefore the mass flow rises too. The ethanol flow itself drops after this first shock to a semi stable value. This could also be explained with the pressure which comes from the combustion chamber. The pressure drop over the injector sinks because the pressure in the chamber rises as a result of the combustion process. The explanation for the rise of the flow after around half the shot can be found in figure 7 which shows the pressure in the combustion chamber and the thrust over the time. This shows another important observation made after the test. As the image shows, the combustion started with a pressure at around 0.9 MPa and dropped afterwards down to around 3 MPa. In comparison with the estimated 15 MPa which came from RPA analysis the starting point was already way to low. A closer look on the video material made of the shot gives an insight why the pressure developed like this.

These pictures show that the combustion for this test was way hotter than previous ones which led to a red glowing graphite inlay. It can be safely assumed that the O/F ratio for the last tests was too high. At the very beginning of the campaign the used graphite inlay had a nozzle throat diameter of around 0.018 m. After test number 136 it had a diameter of 0.023 m (see fig. 8).

Our assumption is that the previous tests already damaged the graphite and slightly enhanced the diameter of the throat. This would explain the low pressure at the beginning of the shot. Over the course of the shot for test #136 the high pressures in the feeding line in combination with the higher ROF resulted in a higher combustion temperature. The oxidizer rich content of the chamber then damaged the inlay and widened the throat. In the wake of this the ethanol mass flow rose because the pressure in the combustion chamber dropped. This resulted in a reduced ROF and led to a higher thrust which was generally more unstable because of the low pressure. Again, this are only assumptions which could explain the results. A definite explanation without further investigation in form of further tests or a simulation can not be made.

The following diagram shows the thrust and line pressures over the time. First it is visible that the ethanol curve oscillates more than the LOX curve. This can be linked to the measuring system. The sensors are connected to the system by capillaries. The capillary for the LOX measurement is longer than the ethanol capillary which leads to a

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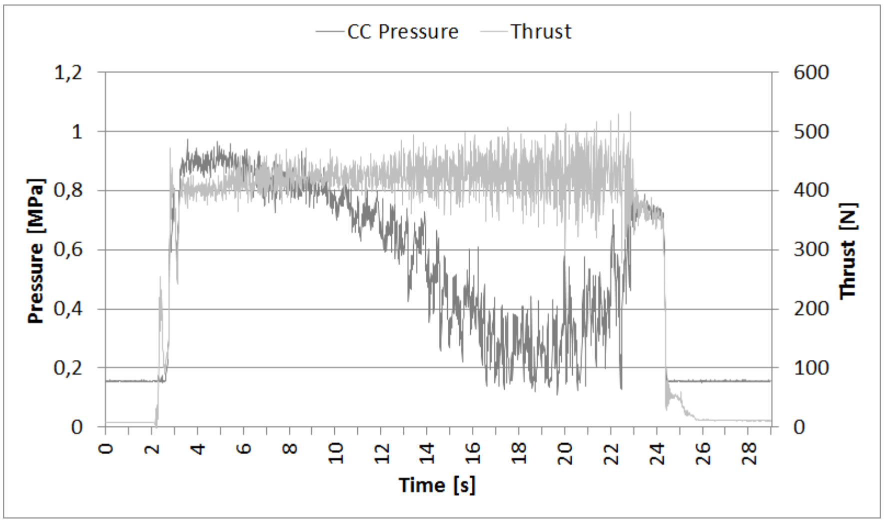

Figure 7: Thrust and chamber pressure - test #136

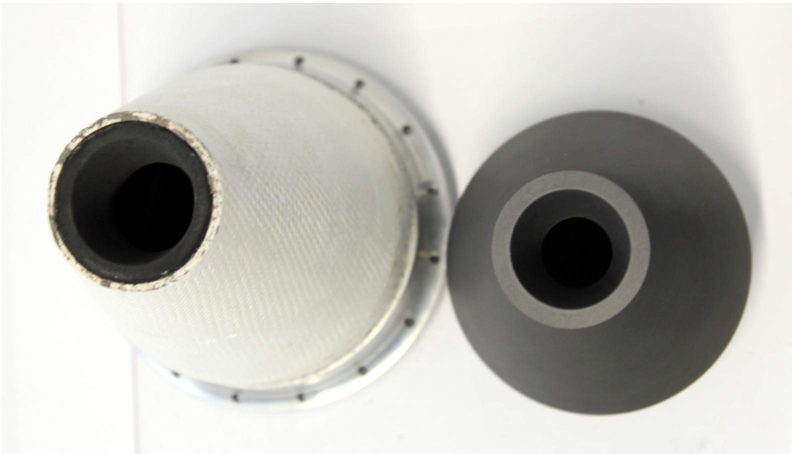

Figure 8: Comparison of used (left) and brand new (right) graphite inlay

bigger amplitude under the vibrations of the working engine. Whether these oscillations of the actual sensor have an influence on the measurement itself or not should be determined by further tests.

4.3 Summary of the Campaign

Overall the campaign was a partial success. We did not reach the designed values but learned the lessons to achieve this objective. In future campaigns the pressure should be at this level or even higher to get the desired thrust. On the other hand, the ROF should be under close surveillance and should be brought to around one to reduce the heat load on the combustion chamber. Moreover, the ethanol line pressure sensor should be fixated to reduce possible influences on the measurement.

5. Conclusion

In our last test campaign, we could demonstrate, that we were able to derive a flight model version from our past test rocket engines. In different tests, general functionality of the FM was proven, minor drawbacks were resolved. Even though the nominal thrust of 500 N was not reached yet, we are confident to reach that goal with the next test campaign. The next major step is the development of a more powerful rocket engine with 700 + N of thrust. This will be necessary to improve the thrust-to-weight-ratio of our rocket in order to leave the launch rail with enough speed.[3] From the last experience, this engine will again have an ROF of 1.0.

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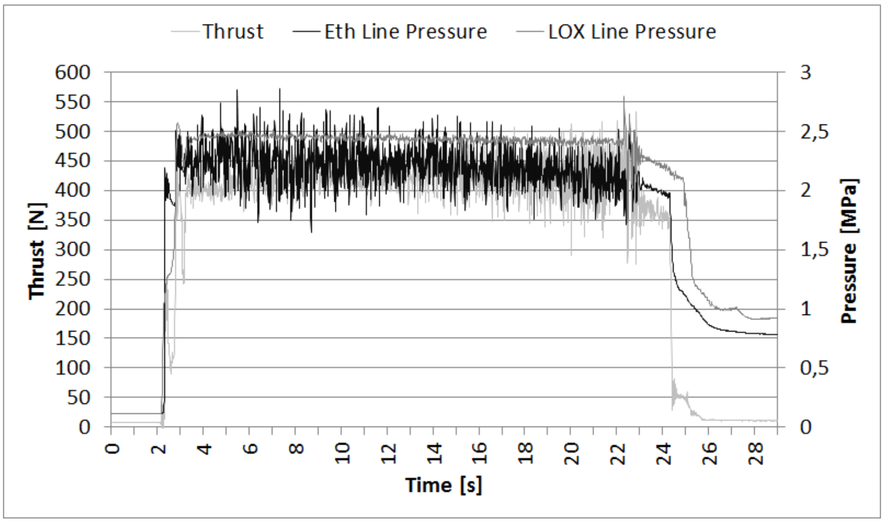

Figure 9: Thrust, LOX and ethanol injector pressure - test #136

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