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Development of a 10 kN LOX/HTPB Hybrid Rocket Engine through Successive Development and Testing of Scaled Prototypes

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Abstract

The development of a hybrid flight engine for a sounding rocket requires intensive preparation work and pretests. To obtain better understanding of the difficulties in design and operation that occur with LOX/HTPB rocket engines, four rocket engines with increasing size and thrust level are successively developed and manufactured. Listed chronologically, those engines are the demonstrator engine, the sub-scale engine with a cylindrical grain geometry and later a star shaped geometry, the full-scale test engine and the full-scale flight version. At first a small-scale technology demonstrator was developed. It produced a mean thrust of 160N for 5s and had the purpose to validate the design calculations and to gain first experiences with the use of cryogenic propellants, especially regarding cooldown procedures.

Based on this test results the so-called sub-scale engine was developed and tested. The engine can be operated with two grain configurations, using either a cylindrical grain or a star shaped grain. The star shaped grain geometries are realized, using an additive manufactured core, outside of which the HTPB is casted. In the low thrust configuration, the engine is operated using a cylindrical single port HTPB grain and it produces a thrust of about 540 N for a duration of 10 s. In the high thrust configuration, the engine is operated using a star shaped HTPB grain and delivers a thrust of 1600 N, with an operation time of 5 s. This grain geometry is a downscaled version of the actual full-scale grain geometry, which provides the opportunity to gain insight on the regression behavior and to validate and improve the performance calculations. The test engines are built in a differential design, so various injector types, as well as pre- and post- combustion chamber geometries can easily be tested. The nozzle segment can be exchanged in order to find a nozzle design that can withstand the hot exhaust gases, while still remaining lightweight.

The next development step is the design of the 10 kN full-scale test engine, which shows the same combustion and performance characteristics as the flight version, but is designed with very high safety margins. The results from this test campaign will be used to build a high performance and lightweight 10 kN LOX/HTPB engine which will power the Warr-Ex3 rocket, the largest sounding rocket ever planned by the WARR (Wissenschaftliche Arbeitsgemeinschaft für Raketentechnik und Raumfahrt) student rocketry group.

This study will discuss the design, construction and testing of the sub-scale engines, as well as the subsequent design decisions that went into the development of the full-scale engine.

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Nomenclature

Roman Symbols

$\dot{m}$	Total mass flow	[kg/s]
$\dot{m}_{\mathrm{fuel}}$	Fuel mass flow	[kg/s]
$\dot{m}_{\mathrm{ox}}$	Oxidator mass flow	[kg/s]
O/F	Oxidizer to fuel ratio	[-]
A_{th}	Throat area	[m^2]
c^*	Characteristic velocity	[m/s]
c_{e}	Effective exhaust velocity	[m/s]
c_{F}	Thrust coefficient	[-]
F	Thrust	[N]
g_0	Acceleration of gravity	[$\mathrm{m/s^2}$]
I_{sp}	Specific impulse	[s]
p_{c}	Chamber pressure	[Pa]
r_{c}	Chamber radius	[mm]
r_{ds}	Downside radius	[mm]
r_{l}	Liner radius	[mm]
r_{p}	Port radius	[mm]
r_{us}	Upside radius	[mm]
t_{b}	Total burn time	[s]

Greek Symbols

Δm_{fuel}	Burned fuel mass	[kg]
η_{Isp}	Total efficiency	[-]
η_{c^*}	Combustion efficiency	[-]
$\eta_{c_{\mathrm{F}}}$	Nozzle efficiency	[-]

Acronyms

HTPB	Hydroxyl-terminated polybutadiene
LN2	Liquid nitrogen
LOX	Liquid oxygen
N2O	Nitrous oxide
WARR	Wissenschaftliche Arbeitsgemeinschaft für Raketentechnik und Raumfahrt (Scientific work-group for rocketry and spaceflight)

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1. Introduction to the WARR Student Organization and its Rocket Development Program

The Scientific work-group for rocketry and spaceflight, or short WARR (German:"**W**issenschaftliche **A**rbeitsgemeinschaft für **R**aketentechnik und **R**aumfahrt") is a group composed of mainly students, working in various aerospace related projects. A short overview on other non-rocketry related projects involves a successful participation in the Hyperloop competition, the design of a cube-satellite in the MOVE-II project and coordinating and participating in space elevator design challenges.

Since the foundation of the WARR in 1962, the development of rockets and rocket engines has played a major role in the history of the club, which launched the first German hybrid rocket Barbarella in 1974.[6] While the work on hybrid rocket engines and corresponding sounding rockets was always one of the core ideas of the WARR propulsion group, activities in the field of hybrid rockets had stopped up until the mid 2000's due to a lack of club-members. In 2007 the development efforts towards a hybrid sounding rocket using the fuel combination N2O/HTPB restarted and culminated in 2015, when the Warr-Ex2 rocket was successfully launched in Brasil(Natal) reaching a maximum altitude of 4.3 *km*.

This leads to "Cryosphere", the group's current rocket development project, with the main goal of developing and launching the "Warr-Ex3" rocket, with a cryogenic 10 kN LOX/HTPB hybrid-rocket engine. Creating this engine requires big steps technology-wise for the rocketry group. A few of these technology steps are:

- Increasing the maximum achieved thrust by one order of magnitude, from about 1 kN to 10 kN.
- Exchanging the relative easy to handle and well known N2O/HTPB fuel combination by the more efficient but harder to handle LOX/HTPB combination.
- Increasing the surface area of the fuel grain, by implementing star shaped grains.
- Safe Handling of Liquid Oxygen(LOX) and Liquid Nitrogen(LN2)
- Development of safe and robust test procedures.

In order to reduce the complexity of the problems at hand, it was decided to break them down by designing and testing four rocket engines increasing in size and thrust. This work aims to give an overview on the design and test results of the various hybrid-engines, the experiences and lessons learned, that went into the design of the final fullscale 10 kN engine. Figure 1 shows a size comparison of the developed test bench engines (fullscale flight version not shown).

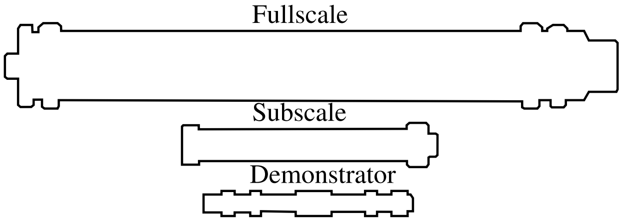

Figure 1: Relative size comparison of the developed engines. Outer Dimensions in mm(Length/Diameter): Battleship 1900/200; Subscale 790/90; Demonstrator 645/45

First, an overview on the fundamental principles of rocket propulsion will be given in Sec. 2, especially regarding the difficulties when dealing with hybrid engines. The following sections will give an overview on the rocket engines developed during the "Cryosphere" project , starting with a short description of the "Hy800" engine in Sec. 3, a LOX/HTPB technology demonstrator engine and the first cryogenic engine that was ever tested by the WARR rocketry group. Next, is a more thorough view on the "HyperLOX" engine in Sec. 4, explaining the design and presenting the coldflow and hotfire test results. Concluding, Sec. 5 will give an outlook on the "Battleship" full-scale test engine and the design choices that were derived from experiences with the sub-scale engine tests.

2. Overview on the Fundamentals of Hybrid Propulsion and Important Performance Parameters

This section aims to give a short overview on the fundamental principles and governing equations of hybrid rocket propulsion, focusing on basic calculation methods of important performance parameters, using hotfire test measurements. Detailed explanations to the theory laid out in this section can be found in Sutton and Biblarz,[7] which also

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contains some theory regarding hybrids. A more thorough look on hybrid propulsion theory is given by Schmucker[5] as well as Altman and Holzman.[1]

One of the most important performance parameters for comparing rocket engines is the specific impulse I_{sp} which basically describes the fuel efficiency of a rocket engine in achieving a given total impulse. Using the effective exhaust velocity c_{e} and the acceleration of gravity $g_0 = 9.81\ m/s^2$ it can be expressed as

$$I_{\text{sp}} = \frac{c_{\text{e}}}{g_0} . \tag{1}$$

By measuring the thrust F and the total massflow $\dot{m} = \dot{m}_{\text{ox}} + \dot{m}_{\text{fuel}}$ it is possible to determine c_{e} using

$$c_{\text{e}} = \frac{F}{\dot{m}} . \tag{2}$$

Measuring the oxidator massflow $\dot{m}_{\text{ox}}$ is possible using a massflow meter or a turbine (measuring the volume flow) coupled with a pressure and temperature sensor, but measuring the fuel massflow $\dot{m}_{\text{fuel}}$ during a hotfire test can be quite difficult. Since the main interest lies in determining mean values of the engine performance parameters, the problem can be reduced to determining a mean fuel massflow $\dot{m}_{\text{fuel,mean}}$ by weighing the solid grain before an after the hotfire test and dividing the resulting Δm_{fuel} by the burn time t_{b}:

$$\dot{m}_{\text{fuel,mean}} = \frac{\Delta m_{\text{fuel}}}{t_{\text{b}}} . \tag{3}$$

In order to maximize the specific impulse of an engine, an optimal Oxidizer to Fuel ratio O/F $= \dot{m}_{\text{ox}}/\dot{m}_{\text{fuel}}$ should be maintained during engine operation. Heavy shifts in the O/F ratio are a typical characteristic of hybrid engines, as even with a constant oxidizer supply, the fuel massflow can shift quite heavily due to changes in grain area and regression rate during combustion. While this effect is rather insignificant in cylindrical grain configurations, it becomes a major source of performance loss in other configurations, like the star-shaped grain used in the fullscale engine.

The combustion efficiency $\eta_{c^*} = c^*/c^*_{\text{id}}$ and nozzle efficiency $\eta_{c_{\text{F}}} = c_{\text{F}}/c_{\text{F,id}}$ can be determined using the characteristic velocity $c^* = (p_{\text{c}} * A_{\text{th}})/\dot{m}$ and thrust coefficient $c_{\text{F}} = F/(p_{\text{c}} * A_{\text{th}})$,with measurements of the chamber pressure p_{c} and with a known throat area A_{th}. Here, the ideal characteristic velocity and ideal thrust coefficient are determined using the CEA software by Gordon and McBride.[3]

3. The "Hy800" Technology Demonstrator

As a first step to gain knowledge in operating cryogenic fueled engines, the technology demonstrator engine "Hy800" was designed and tested. Here a basic single port grain design was used to verify the performance predictions and especially the regression rate law. Also the chamber pressure was fixed for all further engines at 20 bar to minimize the changes between the different engines. Further information about the design of the "Hy800" are described by Lungu and Haidn.[4]

The fuel combination LOX/HTPB was chosen for several reasons. First of all the WARR rocketry group has experience in processing and casting HTPB. Predecessor engines which were developed by WARR also used this rocket fuel. The oxidizer was changed from N2O to LOX due to a higher theoretical I_{sp} of the combination LOX/HTPB. The theoretical I_{sp} and c^* values for LOX/HTPB and N2O/HTPB can be seen in Tab. 1, calculated for a chamber pressure of 20 bar.

Table 1: Theoretical performance parameters

Combination	I_{sp} [s]	c^* [m/s]
LOX/HTPB	257.4	1782.5
N2O/HTPB	226.9	1613.4

Based on that pre-selection, the optimum O/F required for the highest I_{sp} was identified by using the CEA2[3] tool. With this software tool the curve in Fig. 2 was generated to identify the real optimum. The computation was done with frozen reactions after throat. As a result of these calculations the O/F value was set to 2. Note that this value is only a guide number. In hybrid rocket engines the ratio shifts during operation as mentioned in Sec. 2.

The engine was designed to produce a total impulse of 800 Ns with a burn duration of 5 s which results in a thrust of 160 N. The design parameters of the engine are listed in Tab. 2.

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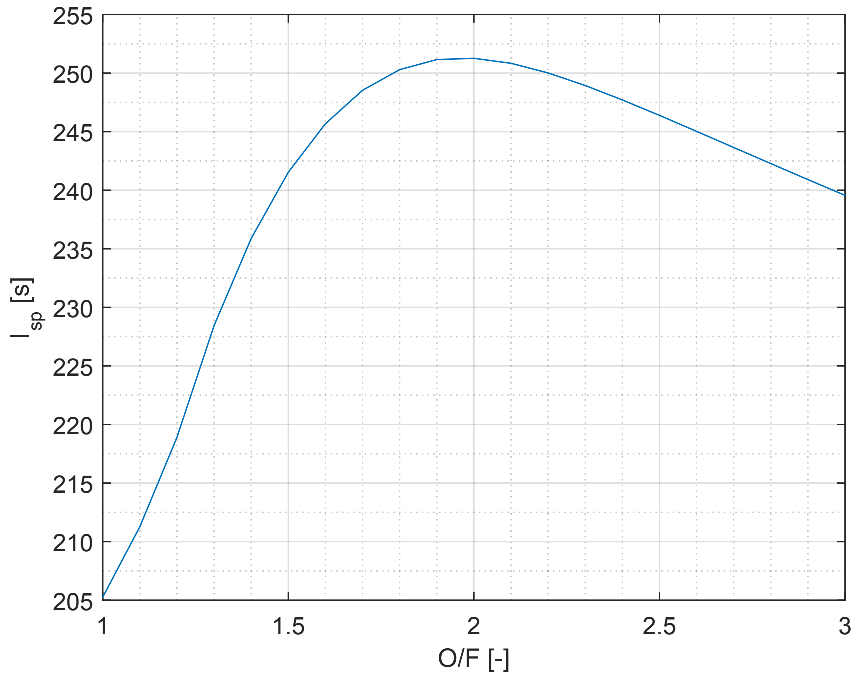

Figure 2: Dependency of the sea-level I_{sp} to the O/F-ratio. The computation was done with frozen reactions after throat.

Table 2: Theoretical performance parameters

Parameter	Value
Chamber Pressure	20 bar
Pre Injector Pressure	25 bar
Mean O/F-ratio	2
Nozzle expansion ratio	3.4
Total burn time	5 s
Mean thrust	160 N
Mean total mass flow	0.0794 kg/s
Characteristic velocity (O/F=2)	1514.956 m/s

4. The "HyperLOX" Subscale Engine

The subscale HyperLOX engine is a modification of the Hyper engine (The N2O/HTPB engine powering the Warr-Ex2 rocket mentioned in Sec.1). To reduce manufacturing costs, many components of the original Hyper engine were either re-used or slightly modified to withstand greater loads.

4.1 Mechanical Design and Performance Calculations of the "HyperLOX" Engine

This section explains the considerations that went into the mechanical design of the HyperLOX engine, especially on the nozzle, a critical component because of the high thermal loads when using LOX. In the second part the performance values and engine dimensions are summarized in Tab. 3.

Figure 3 is a schematic of the HyperLOX engine with an overall chamber length of about 700 mm and an chamber diameter (insulating liner thickness included) of 73.5 mm. From left to right the components are:

1. Injector flange

2. Showerhead injector

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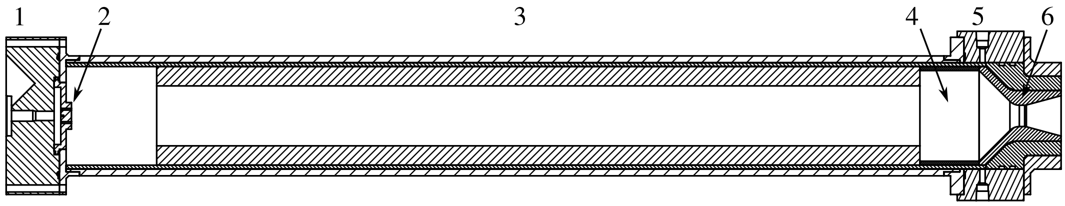

Figure 3: Model of the HyperLOX engine.

3. Combustion chamber with an inserted cylindrical grain
4. Additional insulation in the post combustion chamber
5. A post combustion chamber attachment to incorporate pressure and temperature sensors
6. The graphite/cotton laminate composite nozzle.

The nozzle proved to be an critical component as it had to withstand ten seconds of operation, so double the time of the Hy800 demonstrator engine and higher thermal loads than previous nozzles in N2O/HTPB engines. At first, the nozzle design of the original Hyper engine was reused, which consisted of a full graphite nozzle. This design failed during the second test (see Sec. 4.4), making a redesign necessary.

To combine the positive characteristics of graphite at high temperatures, with the insulating properties of cotton laminate, a hybrid concept was developed. The graphite was used as an inlay to withstand the thermal loads of the exhaust gas, while the cotton laminate thermally insulated the nozzle and the aluminum nozzle flange. Overall, three different composite designs were built and tested (Design "A","B", and "C"), with only type "C" being able to withstand a hotfire test for the full duration of 10 seconds. Figure 4 shows a drawing of the nozzle, which can also be seen in Fig. 3.

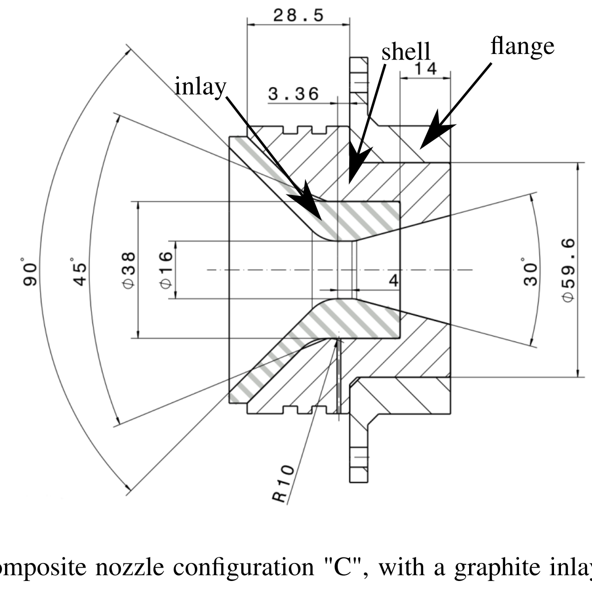

Figure 4: Drawing of the composite nozzle configuration "C", with a graphite inlay, a cotton laminate shell and the nozzle flange.

Initial performance calculations were conducted using a Matlab routine that was created during the Hy800 development. The software can simulate the engine ballistics taking regression of the grain and therefore changes in grain area into account. As the HyperLOX is a modification of a N2O/HTPB engine, the chamber and grain length (original Hyper casting molds were reused) constrain the possible pre- and post combustion chamber lengths. This in turn causes a significant reduction in combustion efficiency, therefore the combustion efficiency was assumed to be of a low value at around 0.85 in the initial design calculations. Table 3 gives an overview on the HyperLOX engine geometry and lists the assumed efficiency parameters. Remaining performance parameters can be found in Tab. 4 with the actual test data.

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Table 3: Geometry and assumed efficiency values of the design calculations.

Parameter	Value
Grain length	572 mm
Engine length	700 mm
Start of burn chamber pressure	20 bar
Mean specific Impulse	198 s
Characteristic velocity	1463 m/s
Total mass flow	0.277 kg/s
Mean thrust	540 N
Burn duration	10 s
O/F ratio	2.8 (mean)
Combustion efficiency	0.85
Nozzle efficiency	0.96

4.2 Precooling Sequences

In order to get the engine started with full oxidizer massflow from the beginning it is necessary to precool the whole system. This is important because with a warm infrastructure the LOX starts to boil inside the main line and the gaseous phase blocks the injector bore holes. To prevent this, a set of precooling sequences was designed to get the system to the desired condition. The sequences are:

- **Sequence A**: Precooling of the main oxidizer line with LN2 through a drain valve with about 1.5 bar; pressurized with air.

- **Sequence B**: Precooling of the massflow sensor and the liquefier piping with LOX through a drain valve with about $5 - 10$ bar; pressurized with gaseous oxygen.

- **Sequence B Dome**: Precooling of the massflow sensor and the liquefier piping with LOX through a drain valve at operation pressure; pressurized with gaseous nitrogen.

- **Sequence C**: Precooling of the injector with LOX through a drain valve with about $5 - 10$ bar; pressurized with gaseous oxygen.

- **Sequence C Dome**: Precooling of the injector with LOX through a drain valve at operation pressure; pressurized with gaseous nitrogen.

- **Sequence PUMP**: After switching the pressure source from gaseous oxygen to gaseous nitrogen, this sequence is used to settle the new pressure level inside the system.

First, Sequence "A" is used to precool the main oxidizer line at low pressure with liquid nitrogen (LN2). Parallel to all sequences the injector is cooled with LN2 using a separate cooling circuit. The Sequence "A" precooling is done by injecting LN2 behind the LOX liquefier piping system, where it is cooling down the main line and bypassing the engine through the drain valve. This sequence is used to save up on LOX. Following Sequence "A", Sequence "B" is used to precool the liquefier piping, especially the massflow sensor. This is achieved by pumping LOX from the main tank through the whole system and again bypassing the injector with the drain valve. The bypass is used to prevent the igniter and grain from cooling down. The LOX is pressurized with gaseous oxygen from the same line while the gas is inserted for liquefaction. This method of pressurization cannot be used to operate the engine during hotfire tests due to massflow limits of the pressure reducer. Therefore, for engine operation, a dome pressure reducer is used.

The switching of the pressurization gas is done as late as possible to prevent impurities in the LOX from nitrogen. After precooling the whole system (with the injector still warm) , the pressurization is switched from oxygen to nitrogen through a dome pressure reducer which can handle the increased massflow during engine operation. Then the "PUMP" sequence is activated to level out the pressure difference caused by the new pressure source. When the system has

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leveled out, Sequence "B Dome" is activated, which is basically the same as Sequence "B", with the new pressure source. This Sequence is repeated until the system is precooled up to the drain valve. Next, the Sequence "C Dome" is used. Now the system operates like it does during a coldflow test, except for a shorter operation time to save LOX. This sequence is repeated until the system is ready for the hotfire test. The system readiness is indicated by the pressure in the LOX tank, the temperature at the faceplate of the injector as well as the massflow and density readings of the LOX coriolis sensor. After finishing the precooling process, which can be seen in Fig. 5, the system is ready for the test sequences.

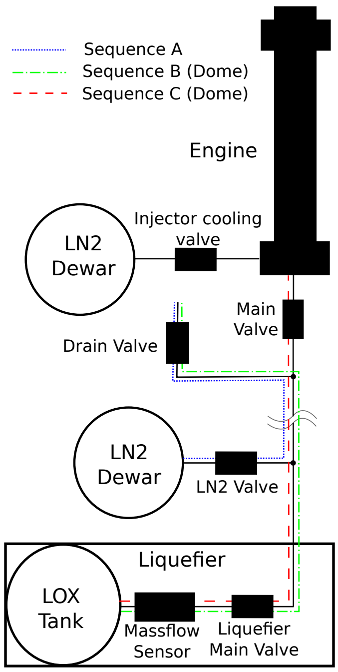

Figure 5: Scheme of the precooling sequences. The liquid phase side of the test rig is simplified.

4.3 Evaluation of Coldflow Test Results

In order to give a quick overview on the injector design, the design criteria and the test results are briefly discussed. First of all the pressure drop over the faceplate of the injector is fixed in dependency on the expected chamber pressure. The pressure drop has to be large enough to avoid a feedback to the fluid system. Some design rules are described by Sutton and Biblarz.[7] The massflow is predetermined by the internal ballistics calculation. So based on the equations written in von Böckh,[8] the area of the injector orifices can be determined. The only uncertain part is the discharge coefficient which can be estimated with empirical formulas.

To validate the calculation and to adjust the discharge coefficient an injector characterization is done. Therefore during several coldflow tests, different system pressures are pre set. So the massflow at different injector pressure drops can be measured and the data can be fitted to recalculate the discharge coefficient. This measurements are used to determine the exact system pressure which is needed to operate the engine with the desired combustion chamber pressure. In Fig. 6 such a fit can be seen. The injector in use is a flat showerhead injector intended for usage with a star shaped grain. It is very similar to the one depicted in Fig. 3. Here, three different system pressures are used and at each point three measurements were taken.

4.4 Evaluation of Hotfire Test Results

Overall, six hotfire tests were performed using a cylindrical grain. The goal was to verify the performance calculations and the mechanical design of the engine. Note that no data of the first hotfire test is available, as the data aquisition

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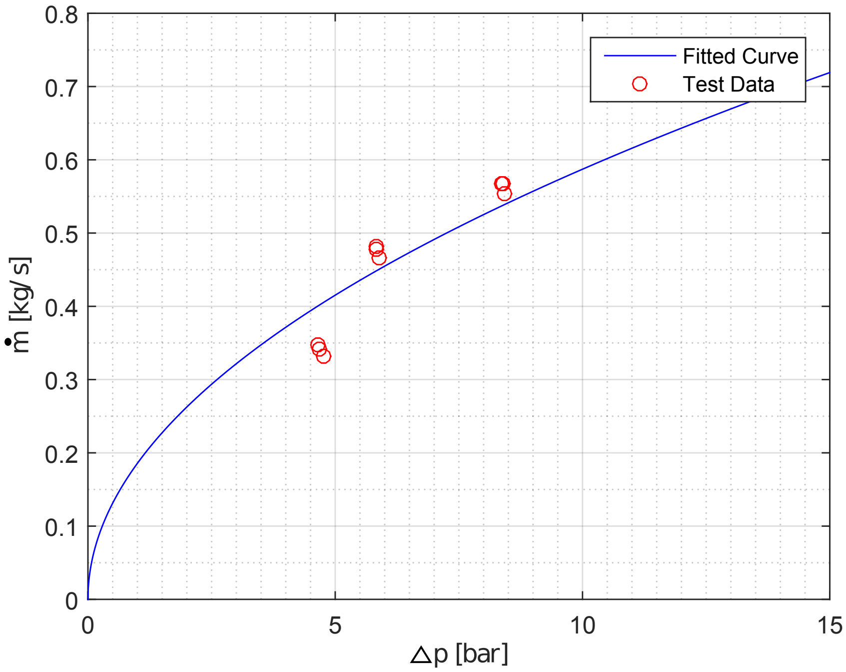

Figure 6: Curve fit of the injector characterization.

system malfunctioned. In this first test the engine was successfully fired for a duration of five seconds, using a graphite nozzle.

The Coriolis mass flow meter used in the oxygen liquefier, showed a strong transient behavior in some of the tests. This was a design error and has since been solved by implementing a new sensor. Nevertheless, to counter that effect for the test evaluation, about 2 seconds of mass flow sensor data is cut off at the beginning of each test and the mass flow is averaged for the remaining time. The results can still be considered quite accurate with margins on the oxidator massflow of about $\pm 10\%$.

Table 4 lists the averaged performance parameters of tests two to six and compares them with the design values. Note that for test 2 no data on the oxidator massflow is available, making subsequent performance calculations impossible. Detailed figures on thrust, pressure and the relationship between injector pressure drop and oxidizer massflow can be found in Fig. 8, Fig. 9 and Fig. 10. All tests had the goal of reaching the operating chamber pressure of 20 bar and a thrust of about 540 N. Additionally, the full graphite nozzle had to be redesigned after test two, as the aluminum flange that held the nozzle in place failed due to the high thermal loads (see Sec. 4 for details on the mechanical construction).

Another phenomenon that could be observed in many tests, where combustion instabilities at a frequency of about 400 Hz. While a more in depth analysis is still to be conducted, a first estimation showed that these instabilities can be linked to the first longitudinal eigenfrequencies of the cylindrical combustion chamber.

The following list is a short summary of the test results for tests 2-6:

- **Test2 :** Due to high thermal stresses the nozzle failed at about 7.7 seconds, which is shown by a spike in thrust seen in Fig. 8(a).

- **Test3 :** Composite nozzle concept "A" (see Sec. 4.1) failed and a emergency stop of Test 3 was initiated. A design flaw in the data aquisition software caused data loss of about 5 seconds.

- **Test4 :** A full cotton laminate nozzle was tested (no graphite inlay), which remained intact for a reduced test duration of five seconds. As expected the regression of material at the throat was too strong to be considered for a fullscale use. Because of the increasing throat diameter, the chamber pressure is decreasing (see Fig. 9(c)), this causes an increased pressure drop at the injector and therefore an increase in massflow and thrust, as seen in Fig. 10(c) and Fig. 8(c).

- **Test5 :** Composite nozzle concept "C" is tested (see Fig. 4). The test was a success as the nozzle remained intact

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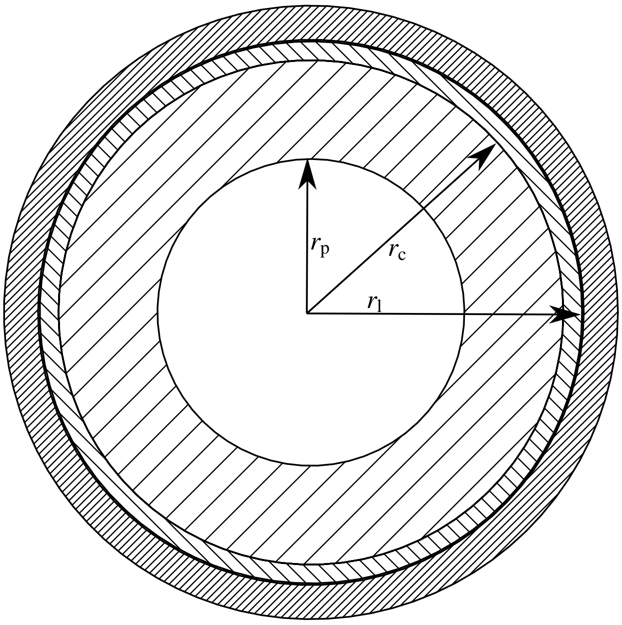
for the whole test duration. Also the operating point of the engine was reached very well, closely matching the expected performance values.

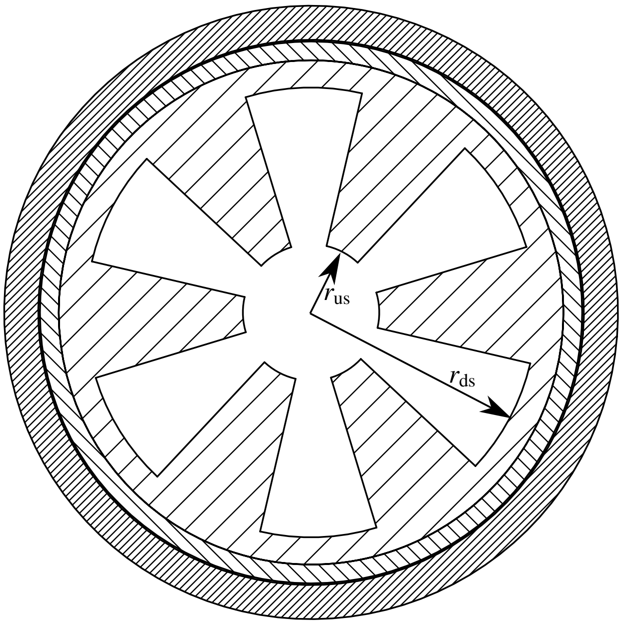
- **Test6 :** A slightly varied version of the composite nozzle design "A" (from Test 3) was tested but failed at the throat. The thermal load on the liner in the fore-combustion chamber was higher than expected. Direct exposure with the oxidizer caused the liner to burn away almost entirely, which exposed the combustion chamber material to the high temperature gases inside the chamber.

While the test campaign can be considered a success, it also showed the complexity and difficulties when testing cryogenic hybrid engines. One notable result was the lower than expected combustion efficiency η_{c^*}. Here, the previous tests with the Hy800-engine achieved a high combustion efficiency of about 95%, so the rather low values of in average 82% were unexpected at first. There are many factors that contribute into lowering η_{c^*} and further studies are needed to quantify their effects, but two main factors are the injector type and the length of the pre- and post- combustion chamber. Here, while the HyperLox had the advantage of a six borehole injector to the one borehole injector of the Hy800 engine. The pre- and post combustion chamber of the HyperLox engine had a length of 55 mm and 70 mm respectively, whereas the Hy800 utilized 100 mm each. While the required length of the pre-combustion chamber can be decreased by using a more efficient (in terms of spray atomization) injector, the post-combustion chamber length is a crucial factor in hybrid engines as it enables a better mix of fuel and oxidizer gases.

Following the hotfire tests with a cylindrical grain, two tests using a star shaped grain were conducted. The tests were performed at a nominal chamber pressure of 20 bar with a goal of producing about 1.6 kN of thrust for a duration of five seconds. An adapted version of nozzle Type "C" was used for these tests. While the tests were an overall success as the nozzle did not fail and the design operating point was reached quite well, the high thermal stresses possibly inflicted over the course of the first six tests, caused the engine to buckle in test seven. Fig. 7 shows the star geometry that was used. Details on the test evaluation for test seven and eight and star-grain manufacturing can be found in Bauer et al.[2]



(a) Cylindrical grain



(b) Star grain

Figure 7: Cylindrical and star shaped grain geometry, used in the HyperLOX hotfire tests. With r_p = 22.5 mm, r_c = 36.8 mm, r_l = 40 mm, r_{us} = 10 mm and r_{ds} = 33 mm.

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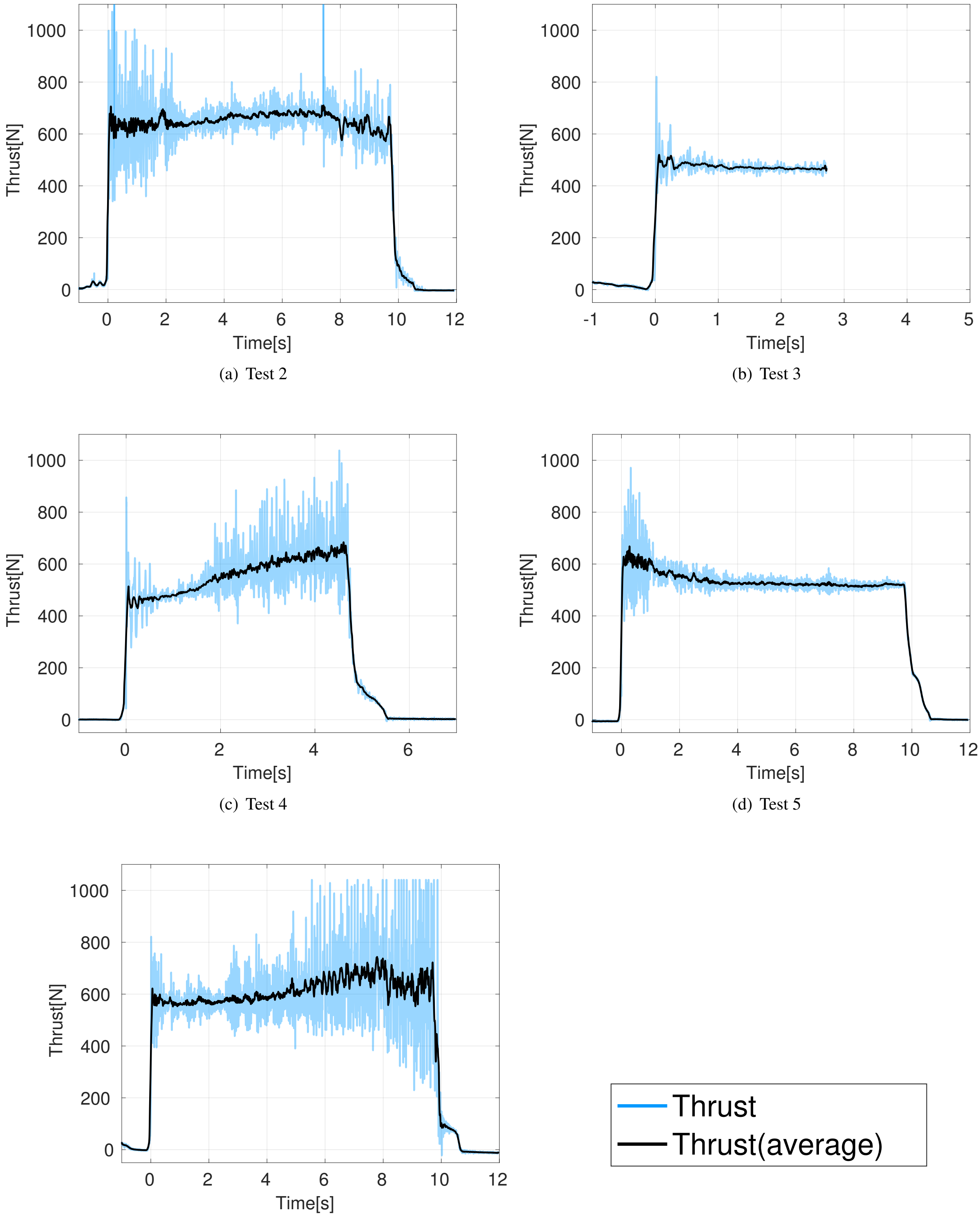

(a) Test 2

(b) Test 3

(c) Test 4

(d) Test 5

(e) Test 6

(f)

Figure 8: Thrust measurements of the hotfire-tests 2-6 with the HyperLOX engine, using a cylindrical grain.

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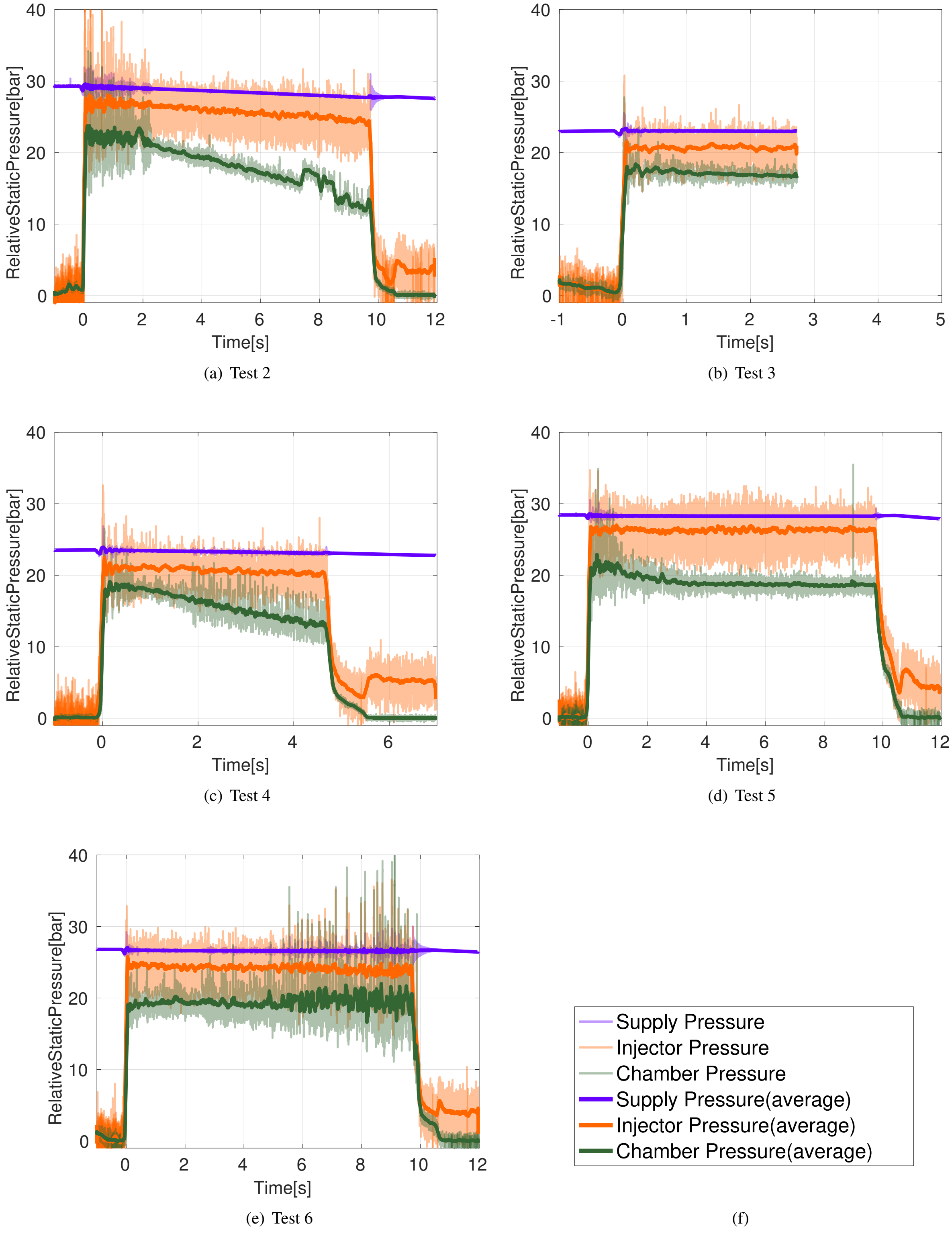

Figure 9: Pressure measurements of the hotfire-tests 2-6 with the HyperLOX engine, using a cylindrical grain. From top to bottom the pressures are listed as the supply pressure from the LOX tank, the pressure at the injector and the pressure inside the combustion chamber.

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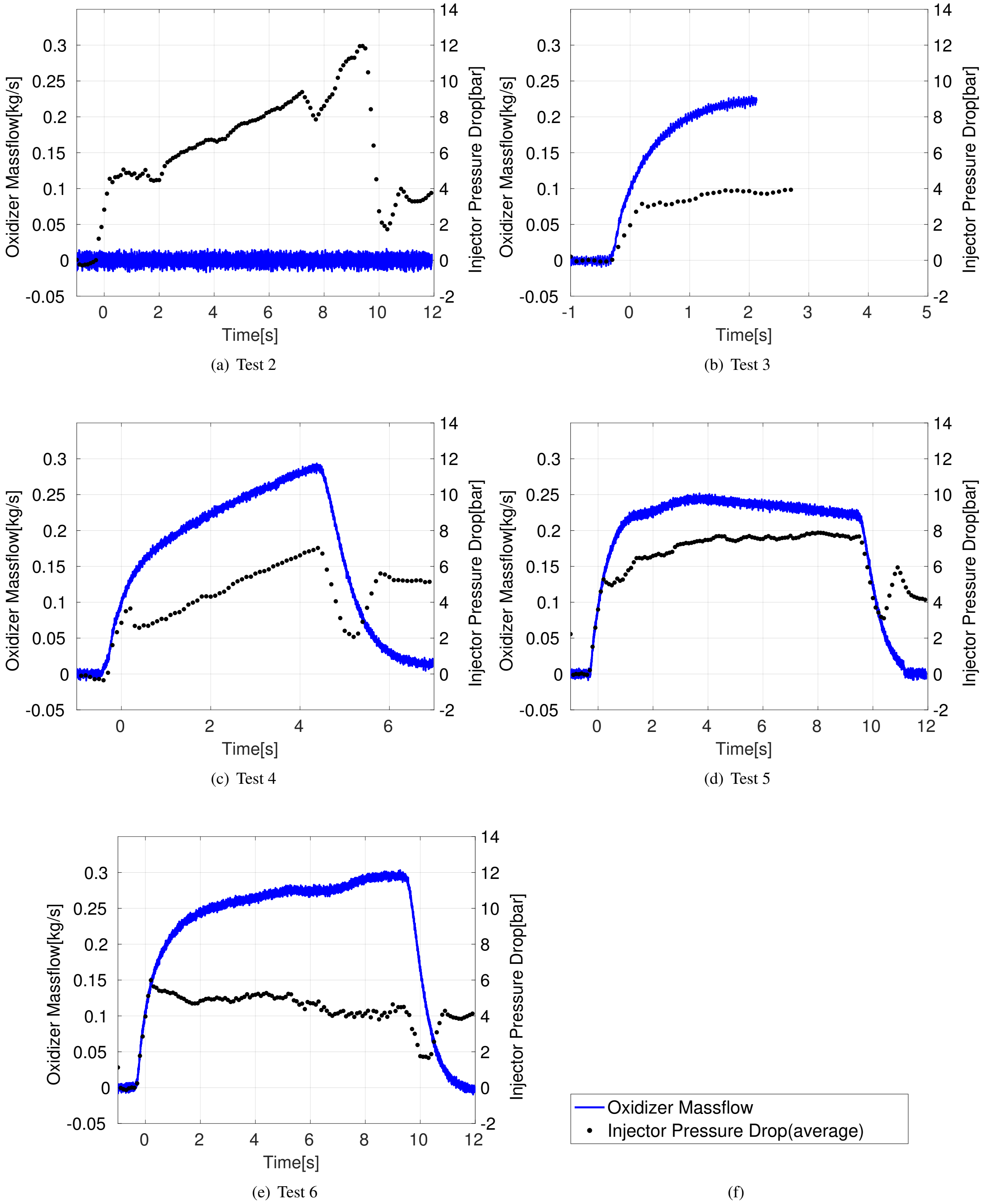

(a) Test 2
(b) Test 3
(c) Test 4
(d) Test 5
(e) Test 6
(f)

Figure 10: Injector pressure drop and massflow measurements of the hotfire-tests 2-6 with the HyperLOX engine, using a cylindrical grain.

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Table 4: Summary of the hotfire test-data.

	Design	**Test 2**[a]	**Test 3**	**Test 4**	**Test 5**	**Test 6**
p_{inj} [bar]	25	26.8	21.7	21.7	26.3	25.2
p_{c} [bar]	20	19.3	18.1	16.8	20.1	20.5
F [N]	540	654	473	565	534	615
I_{sp} [s]	198	-	165	184	177	181
$\dot{m}_{\text{ox}}$ [g/s][b]	205	-	215	242	234	270
$\dot{m}_{\text{fuel}}$ [g/s]	72	66	77	71	73	77
O/F [-]	2.82	-	2.8	3.1	3.2	3.5
η_{c^*} [-]	0.85	-	0.74	0.87	0.81	0.88
η_{Isp} [-]	0.816	-	0.68	0.8	0.73	0.75
$\eta_{c_{\text{f}}}$ [-]	0.96	-	0.91	0.92	0.9	0.85

[a] No oxidator massflow data available.
[b] Variations/fluctuations in data possible due to flawed $\dot{m}_{\text{ox}}$ sensor.

5. Outlook on the "Battleship" Fullscale Engine

Based on the data gained during the demonstrator and subscale test campaign, a fullscale rocket engine is currently under development. The performance calculations are based on the results of the hotfire tests done with the smaller engines. The fullscale is a 10 kN hybrid rocket engine with a burn duration of 15 s. It is a heavy test stand engine, designed with large safety factors and additional measurement ports for a wide variety of performance measurements. These test results will be used to build a lightweight flight engine with the exact same internal ballistics to propel the sounding rocket Warr-Ex3. Detailed performance calculations can be found in Lungu and Haidn.[4]

The calculated performance data of the fullscale rocket engine are listed in Tab. 5. This data was computed with the same software as the smaller engines except for some corrections done on the efficiencies which were measured during the subscale hotfire test campaigns.

Table 5: Theoretical performance parameters

Parameter	Value
Start of burn chamber pressure	20 bar
End of burn chamber pressure	15.5 bar
Throat diameter	68.88 mm
Mean specific Impulse	223 s
Total Impulse	135 kNs
Total mass flow	4.1 kg/s
Initial Thrust	10 kN
Mean thrust	9 kN
Burn duration	15 s
Mixture ratio	2.25 (mean); 3.23 (max.)
HTPB grain mass	21 kg

6. Conclusion and Outlook

In order to design and build a 10 kN class cryogenic hybrid rocket engine for a flight application different downscaled rocket engines were designed and tested. The design was always based on the precursor engine, so the complexity and the thrust level could be increased in controlled steps. This was done to avoid mistakes in the design of the larger and more expensive engines. First of all the demonstrator engine was used to gain knowledge on the fuel combination and the regression rate laws. Then the thrust was increased with the subscale engine, using a cylindrical single port grain. Next step was to change the subscale grain geometry to a star shaped grain. This configuration is basically a downscaled version of the fullscale engine which will be used in the flight configuration. In addition, the test procedures and especially the precooling procedures were developed and refined. Without these procedures it would not be possible to operate the larger engines properly. Another important achievement was the improvement of the measurements, the test rig operation and the whole data evaluation process in general.

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