

DEVELOPMENT AND TESTING OF A CARBON FIBER OVERWRAPPED ABLATIVELY COOLED LIQUID ROCKET ENGINE USING GREEN PROPELLANTS

Ben Oker*, Benjamin Peter*

* Hybrid Engine Development - HyEnD e.V., Student Group of the University of Stuttgart, Pfaffenwaldring 29, 70569 Stuttgart, Germany

Abstract

HyEnD, a student rocketry group from the University of Stuttgart, is developing high-performance liquid rocket engine as part of the BLAST project, aiming to surpass the altitude record for student-built liquid rockets. A performance optimized, lightweight variant of this liquid rocket engine utilizing nitrous oxide and ethanol as propellants, features an ablative combustion chamber liner enclosed within a carbon-fiber-reinforced polymer (CFRP) composite pressure casing. The manufacturing process employs an in-house developed, 4-axis NC-controlled filament winding machine, enabling rapid prototyping of optimized fiber layups and winding patterns. The development cycle involved extensive ignition characterization using an ablatively cooled 'battleship' engine to establish a reliable startup sequence and characterize the thermal behavior of the ablative material. The system was iteratively scaled and hot-fire tested at thrust levels ranging from 200 N to 2 kN. The technology achieved flight-proven status during the 2025 European Rocketry Challenge (EuRoC), where a 1 kN variant successfully propelled the LUMINA rocket to a category win. This work demonstrates the feasibility of utilizing automated composite manufacturing for high-pressure rocket engine architectures in student-led aerospace research.

Keywords

Rocket Propulsion; Green Propellants; Carbon Fiber Reinforced Polymer

1. INTRODUCTION

Traditionally, space propulsion systems have often relied on toxic propellants such as hydrazine or dinitrogen tetroxide. Current research increasingly focuses on non-toxic alternatives, commonly referred to as green propellants, to reduce the environmental impact of space travel. This shift is also relevant to student rocketry, where low toxicity and ease of availability are crucial, making nitrous oxide as an oxidizer particularly appealing in both contexts [1].

HyEnD, a student group at the University of Stuttgart, has a notable history of employing nitrous oxide in hybrid rockets. Their HEROS 3 rocket set an altitude record for student hybrid rockets in 2016 [2, 3], nearly doubled to 64.4 km in 2023 by the N₂ORTH rocket [4]. Following N₂ORTH, the BLAST project was initiated to develop a high-performance liquid rocket engine using nitrous oxide and ethanol to set new altitude records in the student built liquid rocket category [5]. A key design driver is minimizing structural mass while safely containing the required high internal pressures. To this end, a combustion chamber architecture combining an ablative liner with a carbon-fiber-reinforced polymer (CFRP) composite pressure casing was developed and manufactured using an in-house built 4-axis NC-controlled fila-

ment winding machine. Reliably igniting a liquid bi-propellant engine without damaging hard starts is a non-trivial challenge in its own right; an ablatively cooled "battleship" engine with simplified, damage-tolerant geometry was therefore used to characterize ignition behavior and establish a reliable startup sequence ahead of the flight-representative CFRP engine design.

Propulsion systems combining nitrous oxide and ethanol remain uncommon, with publicly available research primarily limited to Tokudome et al. [6] and Youngblood et al. [7]. This paper contributes to that body of research by presenting the development and testing of a carbon fiber overwrapped, ablatively cooled liquid rocket engine using these propellants, iteratively scaled and hot-fire tested from 200 N to 2 kN. The technology achieved flight-proven status at the 2025 European Rocketry Challenge (EuRoC), where a 1 kN variant propelled HyEnDs LUMINA rocket to a category win.

The paper first introduces the fundamental equations governing rocket engine performance, injector design and heat transfer, before outlining the engine design methodology, manufacturing process, and finally the experimental setup, test campaigns and the LUMINA test flight.



FIG 1. LUMINA Launch at EuRoC2025

2. GOVERNING EQUATIONS

This section gives an overview of the basic equations applied in this work. For more details the reader is referred to the reference literature [1, 8].

2.1. Rocket Performance

To evaluate engine performance, measurable properties like propellant mass flow rates, thrust and chamber pressures are normalized to allow comparison across propellant combinations and propulsion systems.

$$(1) \quad I_{sp} = \frac{F_T}{\dot{m} \cdot g_0} = \frac{\dot{m} \cdot u_e + (p_e - p_a) \cdot A_e}{\dot{m} \cdot g_0}.$$

The weight specific impulse I_{sp} is proportional to the characteristic velocity c^* , which describes the thermochemical merit of a propellant combination [1, 8]. Since real engine performance is non-ideal, the characteristic velocity efficiency $\eta_{c^*} = \frac{c_{exp}^*}{c_{id}^*}$ is defined as the ratio between the experimental and ideal characteristic velocity, with

$$(2) \quad c_{exp}^* = \frac{p_{t0,exp} \cdot A_{th}}{\dot{m}_{exp}},$$

where $p_{t0,exp}$ is the total chamber pressure, A_{th} the throat area and $\dot{m}_{exp}$ the combustion gas mass flow. Ideal performance parameters are obtained via chemical equilibrium calculations using NASA CEA [9] for a given chamber pressure, propellant combination and mixture ratio $OF = \frac{\dot{m}_{ox}}{\dot{m}_f}$. Comparing η_{c^*} across tests allows for evaluating combustion performance between engine configurations, chamber geometries and injector designs [1, 8].

2.2. Equations for Injector Sizing

The injector atomizes the propellants during injection, influencing droplet formation, vaporization, mixing,

combustion efficiency, stability and wall heat flux [10]. Common types include shower head injectors (axial holes, high flow speeds), impinging jet injectors (angled jets colliding to form droplet sheets), and centrifugal swirl injectors (helical high-velocity flow forming thin sheets upon exit for fine atomization). Injector design is based on the discharge coefficient C_d , giving the propellant mass flow rate [10, 11]

$$(3) \quad \dot{m}_{Prop} = C_d A_{inj} \sqrt{2 \rho_{Prop} \Delta p_{inj}},$$

where A_{inj} is the critical injector area, ρ_{Prop} the propellant density and Δp_{inj} the injector pressure drop. The injector is sized for a target pressure drop of 20–25 % of the chamber pressure, providing adequate decoupling between chamber and feed system to avoid coupling instabilities [10].

3. ENGINE DESIGN PROCESS

This chapter details the design methodologies and considerations for the main subsystems of the rocket engine.

3.1. Ignition System Design

The selected ignition system uses a non-metalized ammonium perchlorate composite propellant based pyrotechnic charge, glued directly into the combustion chamber and initiated with an e-match. The charge is designed to burn fuel-rich, providing the vaporization and decomposition enthalpy required to reliably initiate bi-liquid combustion of the nitrous oxide and ethanol. Details on the ignition sequence and its characterization are provided in section 5.2.1.

3.2. Injector Design

The choice of injector depends on the propellant combination, oxidizer-to-fuel (OF) ratio, propellant state at injection, combustion chamber geometry and manufacturability. Given their major influence on overall engine performance, significant development effort was invested in designing and testing the injectors.

Impinging jet injectors were selected for the demonstrator chamber, as they offer good atomization, repeatable droplet sheet formation and comparatively simple injection hole and manifold geometries. These injectors are classified by the number of impinging jets (typically 2 to 4–5, with higher counts causing droplet agglomeration and reduced atomization) and by whether the impinging propellants are Like-on-Like or Like-on-Unlike [10].

The injectors were sized using the iterative discharge coefficient algorithm described by Bazarov [10], which accounts for wall friction, inflow and constrictor losses as a function of hole geometry and Reynolds number. Fluid properties at the expected injection pressure and temperature were obtained from CoolProp [12, 13].

The injection angle strongly influences atomization and droplet sheet formation, alongside manufacturability, manifold placement and hole pattern. For a planar doublet element, the resulting spray angle θ , relevant for droplet flow direction and recirculation zone formation, follows from the momentum balance [1]:

$$(4) \quad \tan \theta = \frac{m_1 u_1 \sin \gamma_1 - m_2 u_2 \sin \gamma_2}{m_1 u_1 \cos \gamma_1 - m_2 u_2 \cos \gamma_2},$$

where m_1, m_2 are the mass flow rates, u_1, u_2 the jet velocities and γ_1, γ_2 the jet angles of the individual jets.

Two impinging jet injector types were developed and tested; coaxial swirl injectors were additionally developed for future testing.

3.2.1. Like-Like Nitrous Oxide Doublet + Ethanol Like-Like Triplet Injector

This injector combines nitrous oxide doublet elements arranged in a concentric ring (90° opening angle, 15° resulting droplet sheet angle) with a central ethanol triplet element. The inward-angled nitrous oxide droplet sheet creates a secondary impingement point with the fuel droplets at roughly 0.3 times the chamber length.

3.2.2. Like-Like-Unlike Triplet Injector

The second injector uses Like-Like-Unlike triplet elements on a concentric ring, where two nitrous oxide jets and one ethanol jet impinge at a single point (Figures 2). The resulting 8° droplet sheet angle produces a secondary impingement point at 0.4 times the chamber length. Since the propellants already mix during impingement, combustion can begin closer to the injector plate, providing a longer effective reaction length and promising higher combustion efficiency.

3.2.3. Coaxial Swirl Injectors

Preliminary coaxial swirl injectors, following the design theory in [10], were also designed and manufactured using both conventional and additive methods, with internal, external and recessed swirl chamber variants (Figure 3). In-depth testing of ignitability, combustion stability and efficiency is planned for an upcoming test campaign.



FIG 2. Like-Like-Unlike triplet injector



FIG 3. coaxial swirl injector assembly

3.3. Combustion Chamber Design

The combustion chamber employs a cylindrical geometry combined with a simple conical converging and diverging nozzle, chosen over more complex contours to enable fast machining and rapid prototyping during the iterative development process. The internal volume was sized using a characteristic chamber length of $L^* = 0.8$ m, derived from the characterization tests discussed in chapter 5.2. Figure 4 shows an exemplary section view of a 1000 N CFRP engine after a successful hot-fire test; the 200 N and 2000 N engines are conceptually similar.

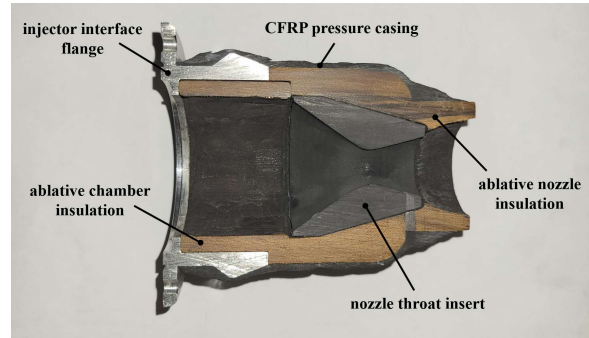


FIG 4. 1000N CFRP engine section

3.3.1. Ablative Cooling

Combustion gas temperatures significantly exceed the melting point of any conventional chamber material. An ablative liner therefore protects the pressure-load-bearing structure, both the metallic casing of the battleship engine and the CFRP overwrap of the flight-representative engine, as shown in Figure 4.

The liner surface undergoes pyrolysis when exposed to the hot combustion gases, absorbing energy and forming a porous, low-conductivity char layer. This layer recedes as the pyrolysis front propagates into the virgin material, with the regression rate governed by the net heat flux into the wall and the material's thermochemical properties.

The convective heat flux is estimated using the semi-empirical Bartz correlation for the gas-side heat transfer coefficient $h_{g,0}$ [1]:

$$(5) \quad h_{g,0} = \frac{0.026}{D_t^{0.2}} \left(\frac{\mu^{0.2} c_p}{Pr^{0.6}} \right)_0 \left(\frac{p_{t0} g_0}{c^*} \right)^{0.8} \left(\frac{D_t}{r_c} \right)^{0.1} \left(\frac{A_t}{A} \right)^{0.9} \sigma,$$

where D_t is the throat diameter, r_c the throat radius of curvature, A_t/A the local area ratio, σ a boundary-layer property correction factor, and subscript 0 denotes stagnation conditions. This correlation was originally derived for smooth, non-blowing chamber walls and therefore overestimates the convective heat flux at an ablating surface, where pyrolysis gases are continuously injected into the boundary layer, thickening it and displacing the hot combustion gases from the wall.

To account for this effect, the baseline coefficient $h_{g,0}$ is commonly corrected using a blowing reduction factor derived from turbulent boundary layer theory with mass transfer [14]:

$$(6) \quad \frac{h_g}{h_{g,0}} = \frac{\ln(1+B)}{B},$$

where B is the dimensionless blowing parameter, defined as the ratio of the pyrolysis mass flux injected into the boundary layer to the mass transfer conductance of the undisturbed boundary layer. Higher blowing rates B , resulting from higher local ablation rates, therefore lead to a progressively larger reduction in convective heat flux compared to the uncorrected Bartz prediction, providing a self-limiting effect on the ablation process.

The ablative surface regression follows from a surface energy balance, where the total heat flux into the wall, consisting of convective and radiative heat transfer, is balanced by conduction into the virgin material and the decomposition enthalpy of the ablative material:

$$(7) \quad \dot{q}_{\text{conv}} + \dot{q}_{\text{rad}} = \dot{q}_{\text{cond}} + \rho \dot{r} \Delta h_{\text{decomp}},$$

where $\dot{q}_{\text{cond}}$ is the conducted heat flux into the virgin material, ρ the virgin material density, and Δh_{decomp} the effective decomposition enthalpy of the phenolic material. Applying this theory to the cotton phenolic combustion chamber yields a theoretical regression rate of 2.48 mm/s, which was used to size the battleship chamber for the ignition tests in chapter 5.2. These tests instead found an experimental rate of 0.3 to 0.45 mm/s, roughly 6–8 times lower, likely due to simplifications in the decomposition enthalpy that neglect effects such as char layer porosity and secondary decomposition gas reactions. The experimental values were therefore used to size the ablator thickness in the CFRP chambers.

3.3.2. Material Selection

The combustion chamber represents a highly demanding environment, characterized by a high-enthalpy, oxidizing gas flow and, in the nozzle expansion section, elevated shear forces. A cotton phenolic composite was selected as the ablative material for the chamber and nozzle insulation, as shown in Figure 4.

Cotton phenolic, along with other ablative composites such as carbon-phenolic or silica-phenolic, has been successfully applied as a thermal protection system on previous HyEnD engines, including the HyFIVE and HyLIGHT hybrid rocket engines [4, 15]. While silica-phenolic offers better ablation characteristics and a lower regression rate, cotton phenolic was ultimately chosen for its superior machinability and lower cost.

Cotton phenolic combines low thermal conductivity with favorable ablative properties, and forms a stable

char layer during pyrolysis of the phenolic binder that sustains the ablative protection over the burn duration. Its relevant thermal properties are summarized in Table 1.

λ [$\frac{\text{W}}{\text{mK}}$]	T_{decomp} [$^{\circ}\text{C}$]	ε [-]
0.15	545	0.9

TAB 1. Thermal conductivity λ , decomposition temperature T_{decomp} and radiation emissivity ε of the cotton phenolic ablative material.

Since the thermal load peaks close to the nozzle throat, a graphite throat insert is used there, offering higher temperature and oxidation resistance than the ablative materials. The phenolic insulation, together with the glued-in graphite throat insert, is contained within a 7075 aluminum interface flange, which connects to the injector and injector dome (Figure 4).

3.4. Composite Casing

The ablatively cooled chamber assembly is reinforced with a carbon fiber winding designed to withstand internal pressures of up to 180 bar. A CFRP overwrap was chosen over a metallic casing for its higher specific strength and resulting mass savings, building on HyEnD's extensive prior experience with composite pressure vessels from previous hybrid rocket motor casings and propellant tanks.

3.4.1. Outer Contour

The casing's outer contour is tailored to the requirements of filament winding and the surrounding engine architecture, with a single polar opening on the nozzle side. This opening uses an isotenoid contour, which produces optimized load paths and a near-ideal stress distribution, while also enabling improved fiber placement since the fiber paths follow geodesic curves without slippage.

On the injector side, no polar opening is needed; instead, a larger diameter accommodates the aluminum interface flange and the cylindrical chamber section. Between the two ends, the contour transitions into a cylinder featuring a specially shaped circumferential bump, over which the fibers are wound to mechanically lock the layers in place and prevent axial slippage.

3.4.2. Layer Buildup

Classical laminate theory is used for the strength calculation of the composite pressure vessel, based on UD laminate properties evaluated at the Institute of Aircraft Design at the University of Stuttgart. All strength parameters are multiplied by a reduction factor of 0.8 to cover manufacturing errors and are shown in Table 2.

With a design chamber pressure of 60 bar and an ignition overpressure of up to 1.5 times nominal, typical for hard starts in liquid rocket engines, a design ignition pressure of 90 bar is obtained. A safety factor of 2

Parameter	Value	Unit	Parameter	Value	Unit
E_1	118000	MPa	R_{11}^+	1500	MPa
E_2	6470	MPa	R_{11}^-	-900	MPa
G_{12}	3150	MPa	R_{21}^+	1500	MPa
ν_{12}	0.347	[-]	R_{21}^-	-900	MPa
ν_{21}	0.021	[-]	-	-	-

TAB 2. UD laminate test data.

on this value yields a design burst pressure of 180 bar. The table 3 shows the exemplary layer buildup for the 1000N CFRP engine shown in figure 4.

Layer	1	2	3	4
Angle [deg]	$\pm 55^\circ$	$\pm 55^\circ$	$\pm 55^\circ$	$\pm 55^\circ$
Thickness[mm]	0,25	0,25	0,25	0,25

TAB 3. Layer distribution of the 1000N combustion chamber overwrap

The resulting overwrap has an overall thickness of 1.05 mm. For the 180 bar design burst pressure and a 70 mm casing diameter, the resulting flux forces on the winding are

$$N_{\text{tangential}} = 630 \frac{\text{N}}{\text{mm}} \quad N_{\text{axial}} = 315 \frac{\text{N}}{\text{mm}}$$

Structural safety is evaluated using the ZTL failure criterion by Schürmann, applied to the ply stresses from classical laminate theory and the UD material data [16]. Distinguishing fiber and inter-fiber failure, the criterion yields a fiber failure factor of 3.42 and an inter-fiber failure factor of 1.46, confirming the structure can safely withstand the applied loads.

4. ENGINE MANUFACTURING

The ablative liner and nozzle geometry are produced using conventional machining. The nozzle throat insert is bonded into the liner before the final nozzle contour is machined in a single in-situ operation spanning both parts, ensuring concentricity and avoiding any disturbance of the thrust vector. The finished liner is then bonded to the aluminum interface flange, forming the interface to the injector and injector dome.

The assembled motor is mounted on the in-house developed 4-axis NC-controlled filament winding machine (Figure 5) and wound using a wet filament winding process, in which the carbon fiber roving is infiltrated with resin via a resin bath immediately before being wound onto the mandrel. Winding patterns are generated using Taniq Wind software, enabling complex path generation with adaptable fiber tension and simultaneous 4-axis motion to accommodate the varying diameter and local thickening of the casing, particularly around the isotenoid dome and the circumferential bump on the injector side. As specified in Table 3, only $\pm 55^\circ$ helical layers are wound with an adaptive fiber angle up to $\pm 65^\circ$ to pass over the aluminium bump.

**FIG 5. Wet filament winding of the 1000 N CFRP engine pressure casing.**

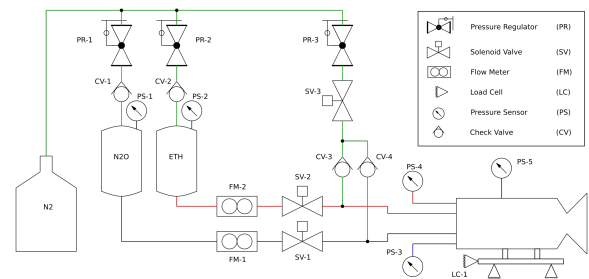
To avoid fiber slippage, the winding pattern's dwell angle was set to greater than 90° , improving fiber placement around the aluminum bump. The Taniq Wind software's layer buildup prediction further allowed toolpaths to be adapted to the casing's varying geometry and layer thickness, ensuring consistent fiber placement. This manufacturing process proved repeatable and reliable for all thrust classes of CFRP liquid rocket engines.

5. ENGINE TESTING

This chapter details the experimental campaign conducted to validate the design and performance of the developed and manufactured CFRP engines.

5.1. Test Bench Setup

HyEnD operates a mobile test bench for solid, hybrid and liquid engine tests of up to 2000 N, located at a certified explosives bunker at the University of Stuttgart, as well as a large-scale test bench at the M11.5 test site at DLR Lampoldshausen capable of handling up to 10 kN. Both test sites were used in the development of the CFRP overwrapped liquid rocket engine. Although differing in size and complexity, the two test benches share a similar overall principle of operation and layout, shown in figure 6.

**FIG 6. HyEnD mobile liquid rocket engine test bench fluid plan**

Ethanol and nitrous oxide are each held in separate intermediate tanks, isolated from the pressurization system by check valves (CV-1, CV-2) and independently pressurized via dedicated regulators (PR-1, PR-2) to allow operation at different OF -ratios. Pres-

surization is supplied by 200 bar nitrogen gas, either from a bottle or, at DLR, large high pressure local storage tanks. Two independently actuated solenoid main valves (SV-1, SV-2) enable variable ignition timing, with flowmeters (FM-1, FM-2) measuring propellant mass flow downstream of the run tanks. A third regulator and solenoid valve (PR-3, SV-3) provide a nitrogen purge to clear the feed lines and cool the engine after each test. The engine is mounted on a thrust structure with a spring-pretensioned load cell (LC-1) on a linear rail, enabling axial thrust measurement while eliminating static friction effects. Figure 7 shows a 1000 N CFRP engine mounted on the thrust structure at the M11.5 test site.

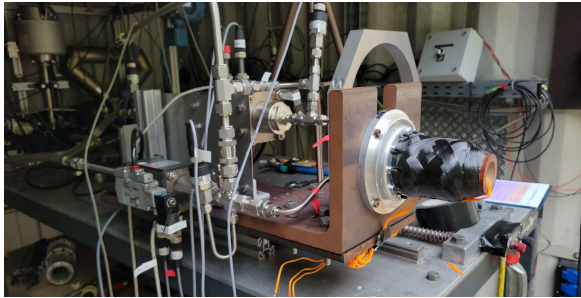


FIG 7. 1000N CFRP engine on testbench

5.2. Battleship Engine Test Results

Rocket engine testing involves inherent risks, particularly from hard starts, ignition failures and detonations. A conservative approach was adopted by performing initial tests with a battleship combustion chamber featuring ablative cooling and a simplified geometry. This battleship test engine can be manufactured using only conventional machining techniques and allows for easy internal chamber geometry modifications [5, 17]. Such geometric freedom combined with a damage-tolerant design facilitates less expensive and rapid testing, enabling fast design iterations.

5.2.1. Ignition Testing

The initial testing campaigns aimed to establish a reliable ignition sequence for the engine, as detailed in section 3.1. Various ignition methods, including steel wool and a solid rocket motor igniter, were evaluated and are discussed in detail in [5]. The most reliable method identified was a pyrotechnic charge based on a non-metalized ammonium perchlorate composite propellant, glued directly into the combustion chamber, which provides the vaporization and decomposition enthalpy required to initiate reaction of the nitrous oxide and ethanol. Metal additives were deliberately avoided in the charge composition to prevent clogging of the small injector holes with metal oxide residues during startup.

The charge is ignited with a prerun time of 1.5 s to ensure sufficient igniter surface is burning before propellant injection begins, as shown in figure 8. An oxidizer

lead is used to allow for complete decomposition of the nitrous oxide ahead of ethanol injection, as seen in figure 9. After a delay of 0.2 s following the nitrous oxide lead, the ethanol main valve opens, injecting ethanol into the hot, decomposed nitrous oxide. This reliably initiates bi-liquid operation of the engine (Figure 10). This ignition method proved to be both scalable and reliable across the different thrust levels of CFRP liquid engines presented in this work.



FIG 8. ignition sequence: igniter at $t = -1.5$ s



FIG 9. ignition sequence: N_2O valve open at $t = 0$ s



FIG 10. ignition sequence: Ethanol valve open at $t = 0.2$ s

5.2.2. Injector Characterization

To evaluate the performance of the different injectors, hotfire tests with the like-like and like-like-unlike impingement injectors were conducted. Figure 11 shows the injection and chamber pressure and Figure 12 the thrust and propellant mass flow measurements of the like-like-unlike injector test. The chamber pressure data indicates steady-state operation after an ignition transient of ≈ 0.3 s, with the 0.2 s oxidizer lead also visible in the mass flow rate measurements. In this test, the average chamber pressure of 46.5 bar and OF -ratio of 3.1 were slightly below the target operating point due to a lower than expected oxidizer mass flow rate. Statistical average values of the measured pressures, thrust and mass flow rates were used to calculate the injector discharge coefficients according to Equation 3, and I_{sp} and c^* according to Equations 1 and 2.

Table 4 summarizes the results of both injector tests, which operated at comparable, slightly below-target chamber pressure and OF -ratio. No I_{sp} could be computed for the like-like test due to a thrust measurement error. The discharge coefficients agree well between both tests, with the nitrous oxide injector holes showing a notably lower discharge coefficient than the ethanol holes. The like-like-unlike injector achieved a $\approx 7.4\%$ higher c^* -efficiency than the like-like injector, indicating higher combustion gas temperature and combustion completeness.

	p_{cc} [bar]	OF -ratio	$C_{d,ox}$	$C_{d,fuel}$	η_{c^*}	I_{sp} in s
Like-Like	43.7 ± 0.1	3.1 ± 0.08	0.442	0.722	0.883 ± 0.011	-
Like-Like-Unlike	46.5 ± 0.1	3.03 ± 0.08	0.427	0.749	0.958 ± 0.011	218 ± 3.4

TAB 4. average test results of different impingement injector tests

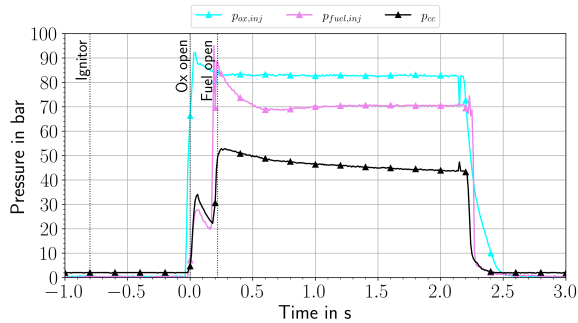


FIG 11. injection- and chamber pressure of like-like-unlike impingement injector test

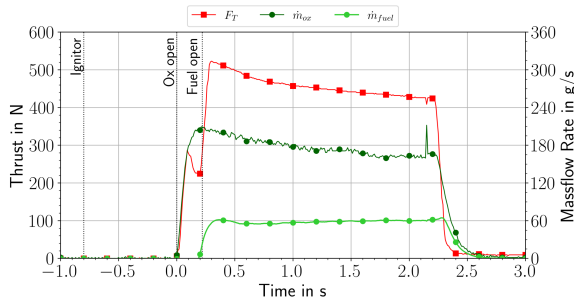


FIG 12. thrust and propellant mass flow rates of like-like-unlike impingement injector test

This is further confirmed by the exhaust plume color: the like-like injector test (Figure 13) glows bright orange, characteristic of unburnt carbon and carbon monoxide radicals reacting with atmospheric oxygen, while the like-like-unlike injector test (Figure 14) glows pale blue, indicating the absence of unburnt carbon and confirming its higher combustion efficiency. These results demonstrate the overall superior performance of the like-like-unlike impingement injector, though further refinement of the injector hole geometry is still required to reach the design operating point.



FIG 13. like-like-unlike impingement injector test



FIG 14. like-like-unlike impingement injector test

Based on the measurements of the injector characterization tests using the battleship engine, the like-like-unlike impingement injectors were scaled to different thrust classes and hole sizing was adjusted to hit the target operating point and target mixture ratio successfully, which is visible by the summarized data in table 5.

5.3. Composite Casing Pressure Testing

To validate the structural integrity of the composite pressure casing prior to hot-fire testing, hydrostatic pressure testing of the combustion chamber was performed. For this test, the nozzle throat was sealed using a PTFE plug that seals against the conical surface of the converging section. The chamber and injector dome assembly was then pressurized with water to twice the nominal operating pressure of 60 bar, reaching a test pressure of 120 bar. Even after five minutes at this test pressure, no leakage through the chamber wall was detected, validating the composite casing design for hot-fire operation. Upon inspection of the material and the engine after the hydrostatic test, no soaking of water into the material was observed and no further impact of these tests on the structural and mechanical properties of the CFRP casing and the ablator were observed.

5.4. 200N Hotfire Test Campaign

Following the successful validation of the ignition method, the like-like-unlike impingement injector, and hydrostatic pressure testing of the composite casing, the first hot-fire test of a CFRP overwrapped 200 N demonstrator engine was conducted.

The primary objective was to validate the structural integrity of the composite pressure casing under representative loads, study the ablative material's regression behavior, and demonstrate stable engine operation. No chamber pressure sensor was installed at this early stage, so η_{c^*} could not be evaluated directly.



FIG 15. 200N CFRP engine hotfire test

The hotfire test data is shown in 16. After the ignition transient, the engine reached stable operation with no observable combustion instabilities. The measured thrust exceeded the nominal design thrust of 200 N, reaching an average of 258.8 ± 1.5 N, consis-

tent with the above-design propellant mass flow rates. The ablated insulation mass, determined by weighing the chamber before and after the test, was added to the fuel mass flow to compute the average mixture ratio, which remained close to the design point at $OF = 4.35 \pm 0.08$.

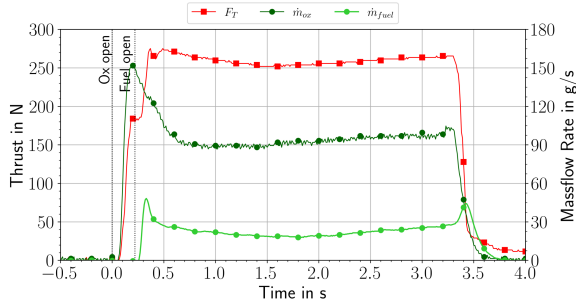


FIG 16. 200N thrust and mass flow test data

While the absence of chamber pressure data prevented calculation of the characteristic velocity efficiency, the thrust and mass flow measurements yielded an average specific impulse of 230.6 ± 2.4 s, corresponding to an I_{sp} efficiency of $89.5 \pm 1.1\%$ relative to the ideal NASA CEA performance at the measured mixture ratio. These results, summarized in Table 5, validated the overall engine concept and provided the basis for scaling towards the subsequent 1000 N engine.

	OF -ratio [-]	F_t [N]	I_{sp} in [s]	$\eta_{I_{sp}}$ [-]
200 N	4.35 ± 0.08	258.8 ± 1.5	230.6 ± 2.4	0.895 ± 0.01
1 kN	3.68 ± 0.25	877.7 ± 3.5	221.3 ± 5.2	0.874 ± 0.02
2 kN	4.87 ± 0.06	1892.2 ± 5.5	231.9 ± 2.4	0.893 ± 0.01

TAB 5. average test results of hotfire tests

5.5. 1000N Hotfire Test Campaign

Multiple hot-fire tests of the 1000 N CFRP engine were conducted over the course of this campaign. Following the successful validation of the ablative chamber and CFRP pressure casing during the 200 N hot-fire campaign, these tests were used to iteratively evaluate and improve the engine design, including analysis of nozzle erosion and ablation behavior. The test discussed here reflects the final geometry resulting from these previous tests. No chamber pressure was measured during this test, and an incorrect setup of the coriolis mass flow sensors limited the mass flow measurements to a low resolution of approximately 2 Hz.

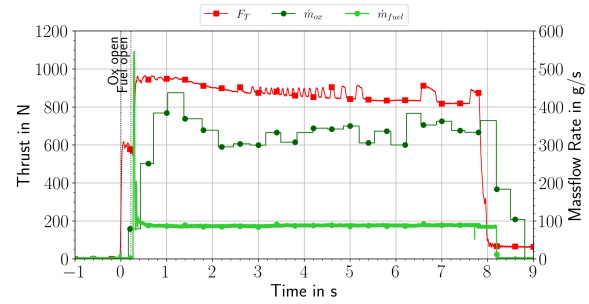


FIG 17. 1000N testdata

With the final geometry, the target thrust of 1000 N was reached, with only a slight reduction in thrust over the duration of the hot-fire test resulting from a decrease in nitrous oxide mass flow. Slight oscillations are visible in the thrust measurement, indicating a possible instability. Their frequency is far too low to correspond to an acoustic instability of the combustion chamber; a shock detachment and reattachment oscillation within the nozzle is therefore considered a potential candidate to explain the observed variance.



FIG 18. 1000N hotfire test

No visible sparks and an almost clear exhaust plume confirm a low and even ablative material and throat erosion rate without any fragments of the char layer breaking loose. The section view shown in figure 4 after this hotfire test confirms the even erosion and in combination with the test data a satisfactory performance was reached to employ this engine in its first flight mission.

6. LUMINA TESTFLIGHT

Following the successful qualification of the 1 kN engine, this thrust class was selected to power the LUMINA rocket, HyEnD's flight demonstrator for the CFRP overwrapped ablative engine architecture. Figure 19 shows a horizontal cross-section view of the LUMINA rocket. LUMINA employs a pressure-fed propulsion system utilizing nitrous oxide and ethanol as propellants, with a mixture of helium and nitrogen used as pressurant gases. The 1000 N combustion chamber is operated at a target chamber pressure of 60 bar with a target injection pressure of 75 bar.

The nitrous oxide tank is a type 3 pressure vessel with an aluminum liner, ensuring compatibility with the nitrous oxide, and is additionally employed as a structural tank with an internal passageway carrying the ethanol. It holds a volume of 3.41 l of nitrous oxide corresponding to 2.7 kg at 20 degC and 55 bar. The ethanol tank holds up to 11 of fuel and is a type 4

pressure vessel with an internal polymer liner. Both propellant tanks are operated at a target pressure of 100 bar. The pressurant is stored in a commercial off-the-shelf (COTS) composite overwrapped pressure vessel (COPV), filled to a pressure of 300 bar. This pressurant gas is regulated down to the propellant tank operating pressure using a mechanical pressure regulator.

The flight itself proceeded nominally, with the engine igniting successfully and operating on target throughout the burn. The lift-off from the launch rail is shown in figure 1. The rocket reached an altitude of 2910 m and recovery was executed successfully. This flight performance ultimately led to LUMINA winning the liquid L3 category at EuRoC 2025 and proving the flight performance of the CFRP overwrapped liquid rocket engine architecture.

7. 2000N HOTFIRE TEST CAMPAIGN

The 2 kN thrust class represents the next scale-up step of the technology and corresponds to the thrust level required for the AURORA rocket, with which HyEnD aims to participate in EuRoC 2026.

7.1. Battleship Engine Test

Before testing the flight-representative CFRP engine at this thrust level, a battleship engine test was conducted again to evaluate the test bench pressure set point, to test the newly scaled like-like-unlike impingement injector, and to verify the scalability of the previously established ignition method.

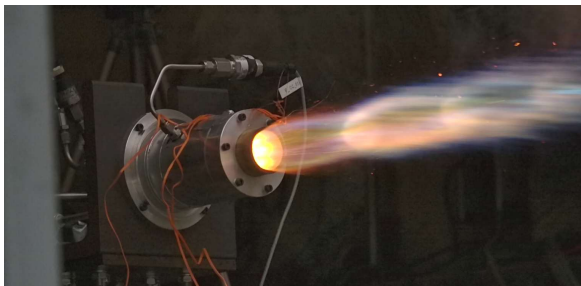


FIG 20. 2000N battleship engine hotfire test

The test was successful, with the ignition method showing reliable ignition and the engine reaching the target operating pressure, OF-ratio and thrust. The successful ignition test can be viewed in figure 20.

7.2. 2000N CFRP Engine Test

Building on the results of the battleship test, the simple but reliable CFRP engine design discussed in section 3.3 was scaled up to the 2000 N thrust level. To accommodate the longer burn durations of more than 15 s targeted for this engine, the insulation thickness of the ablative liner was increased accordingly. Additionally, the length of the nozzle throat insert was increased so that it extended to cover larger expansion ratios, addressing the erosion issues previously

observed in the expansion section downstream of the throat insert.



FIG 21. 2000N CFRP engine hotfire test

The first hot-fire tests were conducted in August 2026 (figure 21) to evaluate the engine's suitability for a potential flight application at EuRoC 2026. The engine was successfully fired for 17 s, with the target thrust and operating point reached demonstrating its capabilities at higher thrust levels.

8. CONCLUSION AND FUTURE WORK

This paper presented the development of a carbon fiber overwrapped, ablatively cooled liquid rocket engine using nitrous oxide and ethanol as green propellants, covering the ignition system, injector characterization and composite casing testing that led to a simple, scalable and weight-optimized engine design. The engine was successfully scaled from 200 N through 1000 N to 2000 N, with Figures 15, 18 and 21 illustrating this progression. The architecture achieved flight-proven status with the 1000 N variant, which propelled the LUMINA rocket to a category win at EuRoC 2025, while the 2000 N variant is a promising candidate for HyEnD's AURORA rocket at EuRoC 2026. Future work will focus on alternative ablative materials and ceramic matrix composite chamber liners to improve regression resistance, as well as scaling the engine to higher thrust levels.

ACKNOWLEDGEMENTS

The authors would like to thank their fellow propulsion team members at HyEnD, Felix Klein, Maximilian Mehne, Markus Luft, Hannes Ganssauge, Tobias Laschefski and Vincent Kirchner for their contributions to the development and testing of this liquid rocket engine. The authors are also grateful for the testing opportunities and guidance from the M11 at the DLR Lampoldshausen as well as continuing support from the Materialprüfanstalt (MPA) and Institute of Space Systems (IRS) at the University of Stuttgart.

Contact address:

ben.oker@hyend.de

References

- [1] G. Sutton and O. Biblarz. *Rocket Propulsion Elements*. Wiley, 9 edition, 2017. ISBN: 978-1-118-75365-1. DOI: [10.1007/978-3-030-26965-4](https://doi.org/10.1007/978-3-030-26965-4).

- [2] Mario Kobald, Ulrich Fischer, Konstantin Tomilin, Anna Petrarolo, Paula Kysela, Christian Schmierer, Andreas Pahler, Jonas Gauger, Jonas Breiting, Ferdinand Hertel, and Benjamin Hochheimer. Sounding rocket "heros" - a low-cost hybrid rocket technology demonstrator. 07 2017. [DOI: 10.2514/6.2017-4902](https://doi.org/10.2514/6.2017-4902).
- [3] Mario Kobald, Christian Schmierer, U. Fischer, Konstantin Tomilin, and Anna Petrarolo. A record flight of the hybrid sounding rocket heros 3. In *31st International Symposium on Space Technology and Science (ISTS)*, 06 2017.
- [4] Indira Keserovic, Colin Barth, Julian Dobusch, Marc Gritzka, Marc Hazenbiler, Philipp Jochum, Florian Merz, Robert Rödinger, and Maximilian Oechsle. Design and launch of the hybrid sounding rocket n2orth. In *26th ESA Symposium on European Rocket and Balloon Programmes and Related Research*, 05 2024.
- [5] Laurence Hoffelder, Ben Oker, Adrian Matuschik, and Felix Klein. Development and testing of a 550 n additively manufactured liquid rocket engine using green propellants. In *11th European Conference for Aeronautics and Space Sciences (EUCASS)*, 07 2025. [DOI: 10.13009/EUCASS2025-643](https://doi.org/10.13009/EUCASS2025-643).
- [6] Shinichiro Tokudome, Tsuyoshi Yagishita, Ken Goto, Naohiro Suzuki, Takayuki Yamamoto, and Yasuhiro Daimoh. An experimental study of an n_2O / ethanol propulsion system with 2 kn thrust class bbm. *Transactions of the Japan Society for Aeronautical and Space Sciences, Aerospace Technology Japan*, 19(2):186–192, 2021. [DOI: 10.2322/tastj.19.186](https://doi.org/10.2322/tastj.19.186).
- [7] Stewart H. Youngblood. Design and testing of a liquid nitrous oxide and ethanol fueled rocket engine. Master's thesis, New Mexico Institute of Mining and Technology, Socorro, New Mexico, August 2015.
- [8] D Mishra. *Fundamentals of Rocket Propulsion*. CRC Press, 2017. ISBN: 978-1-4987-8535-8.
- [9] Sanford Gordon and Bonnie J McBride. Computer program for calculation of complex chemical equilibrium compositions and applications. part 1: Analysis. Technical report, 1994.
- [10] V. Yang, M. Habiballah, J. Hulka, and M. Popp. *Liquid Rocket Thrust Chambers: Aspects of Modelling, Analysis and Design*. American Institute of Aeronautics and Astronautics, 2004. ISBN: 1-56347-223-6.
- [11] V. Bazarov, V. Yang, and P. Puri. Design and dynamics of jet and swirl injectors. *Liquid Thrust Chambers: Aspects of Modeling, Analysis and Design*, 200(19-103), 2004. [DOI: 10.2514/4.866760](https://doi.org/10.2514/4.866760).
- [12] Ian H. Bell, Jorrit Wronski, Sylvain Quoilin, and Vincent Lemort. Pure and pseudo-pure fluid thermophysical property evaluation and the open-source thermophysical property library coolprop. *Industrial and Engineering Chemistry Research*, 53(6):2498–2508, 2014. [DOI: 10.1021/ie4033999](https://doi.org/10.1021/ie4033999).
- [13] E. W. Lemmon, I. H. Bell, M. L. Huber, and M. O. McLinden. NIST Standard Reference Database 23: Reference Fluid Thermodynamic and Transport Properties-REFPROP, Version 10.0, National Institute of Standards and Technology, 2018. [DOI: https://doi.org/10.18434/T4/1502528](https://doi.org/10.18434/T4/1502528).
- [14] William M. Kays, Michael E. Crawford, and Bernhard Weigand. *Convective Heat and Mass Transfer*. McGraw-Hill, New York, 4 edition, 2005. ISBN: 978-0-07-246876-1.
- [15] Maximilian Oechsle, Julian Dobusch, Marc Gritzka, Philipp Jochum, Florian Merz, and Robert Rödinger. Design and launch of the hybrid rocket demonstrator compass. In *2nd International Conference on Flight Vehicles, Aerothermodynamics and Re-entry Missions & Engineering (FAR)*, 2022, 06 2022.
- [16] Helmut Schürmann. *Konstruieren mit Faser-Kunststoff-Verbunden*. Springer, Berlin, Heidelberg, 2 edition, 2007. ISBN: 978-3-540-72189-5. [DOI: 10.1007/978-3-540-72190-1](https://doi.org/10.1007/978-3-540-72190-1).
- [17] T Kahn and I Qamar. Factors affecting characteristic length of the combustion chamber of liquid propellant rocket engines. *Mehran University Research Journal of Engineering and Technology*, 38(3), 2019. [DOI: 10.22581/muet1982.1903.16](https://doi.org/10.22581/muet1982.1903.16).

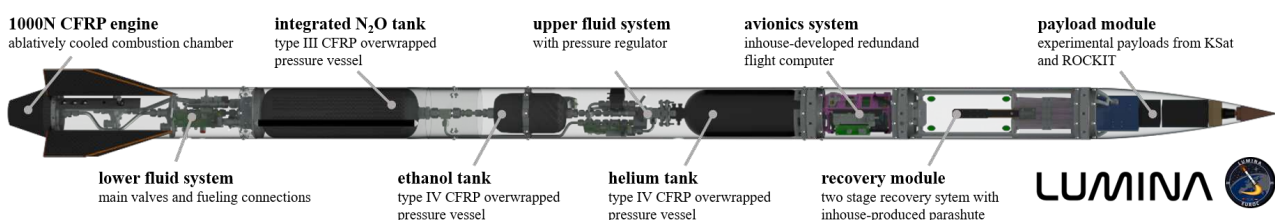


FIG 19. LUMINA rocket section view