

HOT FIRING OF A N₂O/C₂H₄ PREMIXED GREEN PROPELLANT: FIRST COMBUSTION TESTS AND RESULTS

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ABSTRACT:

Today hydrazine is the commonly used monopropellant for attitude- and orbit control of satellites and for powering probes or landers. Due to changing political and economic framework, (e.g. the REACH regulation in Europe) different propellants for replacing hydrazine are currently under development or qualification. The Institute of Space Propulsion of the German Aerospace Center (DLR) at Lampoldshausen is focusing on two different propellants to replace hydrazine: ADN-based monopropellants and mixtures of nitrous oxide with hydrocarbons. The latter are so called premixed monopropellants: oxidizer and fuel are stored in a premixed state in one tank. Thus the simplified propulsion system of a monopropellant can be combined with the high I_{SP} of a bipropellant. To gain experience with a propellant mixture consisting of nitrous oxide (N₂O) and ethene (C₂H₄), DLR is conducting hot gas combustion tests with an experimental combustor. The paper summarizes the results of combustion tests conducted with the premixed propellant injected in gaseous state. Calculated and measured performance (c^* and η_{c^*}), depending on mixture ratio and chamber pressure is shown and discussed.

1. INTRODUCTION

Since the early days of spaceflight hydrazine is used as a monopropellant to power rockets, satellites or probes. During the 50s and 60s of the 20th century a large number of different propellants were tested to be used as a monopropellant [1]. Among those, hydrazine offered a good performance, long term storability, handling without the danger of explosions and relatively low costs. Space flight and the operation of satellites are a business which is strongly focused on reliability, thus development and qualifications of new propellants and thrusters consume lots of time and money. These constrain make hydrazine the most established monopropellant to this day.

During the last decade several things changed. The high toxicity of hydrazine became a point of concern. In the EU the so-called REACH-Regulation [2] came into effect, here hydrazine was set on the list of substances of very high concern (SVHC). Thus it becomes more and more likely that the use of hydrazine will be limited or prohibited in future, even though exceptions for the space industry might be given [3]. To compensate a possible prohibition of hydrazine, across the globe several so-called green propellants are under development or being qualified. Among those alternatives, the ADN-based monopropellant LMP-103S seems to be a promising candidate. The propellant is currently under qualification by ESA, two 1N thrusters were already tested in space on the PRISMA satellite

[4]. Nevertheless other propellants or propellant mixtures may offer significant advantages and are therefore under thorough investigations.

The following chapter will give a short overview of some alternatives. The propellant mixture focused on in this paper is a mixture of nitrous oxide and ethen.

1.1. Green propellants overview

Ionic liquids: ADN based propellants:

ADN based monopropellants consist of at least an energetic salt (ammonium dinitramide) a fuel component and water. The most known propellants are LMP-103S, invented by the Swedish company ECAPS and FLP-106, developed by FOI (Swedish Defense Research Agency). LMP-103S offers a 6% higher specific impulse (253 s) than hydrazine and a 30% higher density impulse while being less toxic and not carcinogenic. Storability was already tested for more than 7 years and the propellant can easily be ignited by using a preheated catalyst (approx. 350°C) [5]. FLP-106 offers an even slightly higher specific impulse (about 260 s) coming along with higher combustion temperatures [6]. The main difference of FLP-106 in comparison to LMP-103S is the usage of a less volatile fuel. Detailed information about ADN based monopropellants can be found in [6–8].

Ionic Liquids: HAN based propellants

HAN (hydroxyl ammonium nitrate) based propellants are another class of ionic liquids. Those propellants were intensively investigated by the United States Air Force Research Laboratory (AFRL) to be used as liquid gun propellant. HAN has been studied since the 1960s, e.g. during the 1980s the liquid gun propellant LP1846 was developed. Due to the increasing toxicity concerns of hydrazine, the HAN based propellants were also considered to be used as a monopropellant in space applications. Thus the AF-315E propellant was developed. This propellant was selected for the Green Propellant infusion mission (GPIM) [9]. AF-M315E may offer an I_{sp} of 257 s and a 45% higher density than hydrazine. The thrusters can be ignited by a preheated catalyst. According to the higher I_{sp} the combustion temperature of AF-M315E exceed the combustion temperatures of conventional hydrazine [10].

Hydrogen Peroxide, H_2O_2

Hydrogen peroxide is another well studied green propellant alternative. The main advantages of

H_2O_2 are a negligible toxicity, easy ignitability via catalyst and relatively low decomposition temperatures (up to 1275 K). The drawbacks of H_2O_2 are a lower I_{sp} than hydrazine (up to 196s, depending on concentration) and the incompatibility with several materials (e.g. copper, iron, magnesium alloys, titanium) [11–14]. Due to the availability of high concentrated hydrogen peroxide, well studied catalysts and a lot of experience in handling the substance, hydrogen peroxide seems to be a very promising alternative for several monopropellant applications; e.g. H_2O_2 was under consideration for the A5ME upper stage attitude control system [15].

Water electrolysis propulsion

Another possible - definitely green - alternative is water electrolysis propulsion. The idea is to have a satellite equipped with a water tank, two gas tanks and an electrolyser. By using the electrolyser the water is decomposed to gaseous hydrogen and oxygen which are stored in separate tanks. The satellites thrusters are then powered by gaseous H_2 and O_2 at a low mixture ratio to assure modest combustion temperatures. Remaining oxygen can be used e.g. in a cold gas propulsion system. Ignition of the H_2/O_2 can be achieved by using a platinum catalyst. Easy handling of the purified water, no safety and toxicity concerns and available technology (electrolysers) are the main advantages. Drawbacks are a quite complex propulsion system with three tanks, necessary pressure regulators, valves, bipropellant injectors and a corresponding high weight of the whole system. Additional the avoidance of H_2 leakage in a later propulsion system might be challenging. Airbus D&S is recently studying this kind of propulsion system [16, 17].

Mixtures of hydrocarbons with nitrous oxide/nitrous oxide fuels blends

Another class of green propellants is the so called nitrous oxide fuel blends, mixtures of hydrocarbons with nitrous oxide. Those mixtures are no single species monopropellants as e.g. hydrazine or hydrogen peroxide. The oxidizer (N_2O) and a fuel (e.g. C_2H_2 , C_2H_4 or C_2H_6) or a fuel combination are stored premixed, i.e. monopropellant-like in one tank. In comparison to a classical bipropellant system, only one tank, one feeding line and one injection system is needed. Thus these propellants are sometimes called "premixed monopropellants", offering a monopropellant like system while having a bipropellant performance (I_{sp} approx. 320 s).

Mainly the components are cooled down ($< 253\text{ K}$) and mixed. The high vapor pressure of the components can offer a self-pressurizing propulsion system without any external pressure supply. Beside the mentioned advantages, nitrous oxide fuel blends provide some non-minor challenges: Very high combustion temperatures ($> 3000\text{K}$) require an active cooling of the nozzle and combustion chamber. Furthermore the propulsion systems needs proper flashback arrestors as well as newly designed ignition and injection systems. The most known nitrous oxide fuel blend is NOFBX from Firestar [18, 19]. Recently several explosions occurred during DARPA/Boeing's work on a nitrous oxide acetylene propellant mixture called NA-7 [20]. The German Aerospace Center (DLR) in Lampoldshausen is working on a dinitrogen monoxide/ethene mixture [21–24].

1.2. N_2O & C_2H_4 propellant mixture (“HyNOx”)

The German Aerospace Center (DLR) in Lampoldshausen has chosen a mixture consisting of dinitrogen monoxide and ethene to be used as a premixed monopropellant. The mixture was named “HyNOx” (Hydrocarbons mixed with nitrous oxide). Ethene was chosen as fuel due to its not to different vapor pressure compared to nitrous oxide (vapor pressure at 273 K : C_2H_4 : 41 bar; N_2O : 31.2 bar [25]). The similarity concerning the vapor pressures should assure good miscibility and simultaneous evaporation in a propellant tank. Furthermore ethene is quite safe to handle, so compared to acetylene self-decomposition hazards can be avoided. Though

the theoretical vacuum specific impulse is lower in comparison to a mixture of nitrous oxide and acetylene (approx. 319 s for $\text{N}_2\text{O}&\text{C}_2\text{H}_4$ to 330 s for $\text{N}_2\text{O}&\text{C}_2\text{H}_2$ [13]).

At the beginning of DLR's research activities a cooling and liquefaction setup for the N_2O & C_2H_4 mixture was assembled [26]. After some preliminary tests it was demounted. In 2016 an improved system will be set up.

To gain experience with the propellant mixture and to conduct the first tests, DLR chose to mix the oxidizer and fuel in their gaseous state upstream the injector. This offers several advantages:

- The mixture ratio can be adjusted easily via exchange of orifices and/or adjusting the feeding pressure;
- Common gas bottles can directly be connected to the setup;
- Easy comparability to CFD simulations is possible; no evaporation effects of the liquefied propellant have to be considered;
- The general performance of the propellant for different mixture ratios can be adjusted easily;
- If a hard ignition or a flashback across the injector occurs, due to lower density and propellant mass in the feeding lines less damage will be caused;
- Flashback across the injector is a very critical issue, the gaseous mixture is easier to ignite, and thus flashback is more likely. So by using a gaseous mixture a “worst case” approach is achieved. If the flashback can be avoided with gaseous mixtures, it seems to be very likely that it can also be avoided by using the liquid, cooled

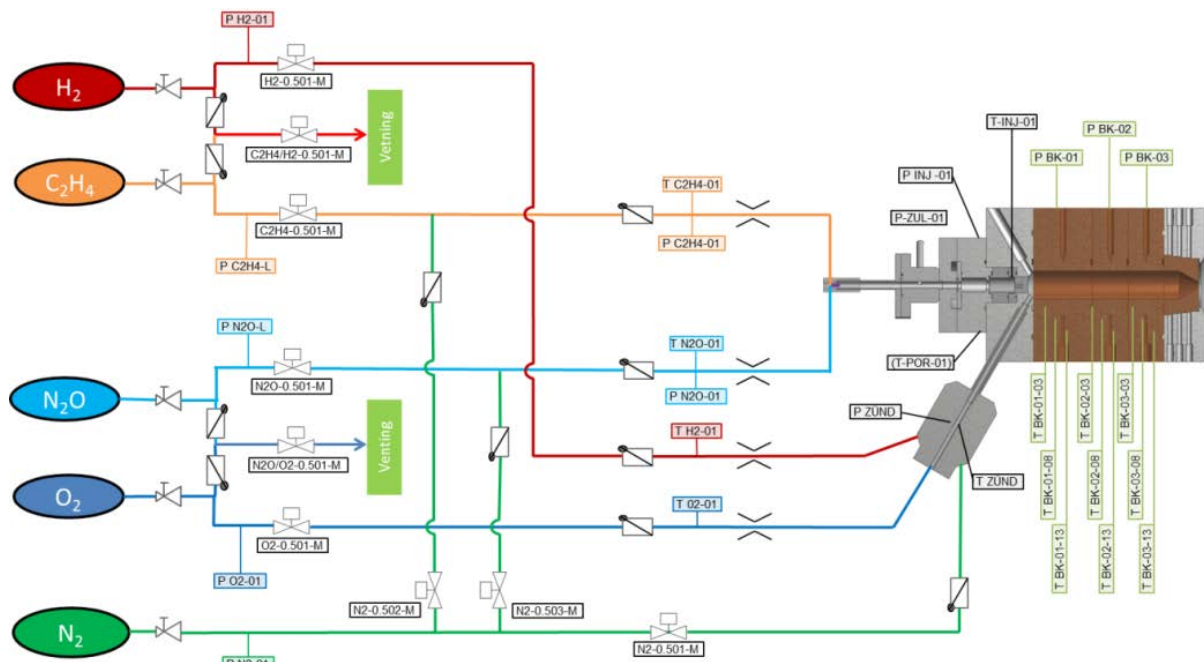


Figure 1: Simplified P&ID of the test setup

propellant;

g) During the later use in a propulsion system 2-phase or gaseous flow of the propellant cannot be avoided for all situations. Especially during ignition or shut down of a thruster, gaseous propellant will flow through the injector. By using gases, the occurring pressure drop as well as the ignition and shutdown behavior at this operation points can be studied in advance.

All described combustion tests were performed with gaseous propellants changing the mixture ratio by using different feeding pressures and a calibrated set of orifices.

2. TEST SETUP AND COMBUSTOR DESIGN

The combustion tests were conducted at DLRs M11 test complex. A green propellant test bench was assembled at the test bench M11.5 [21, 27].

2.1. Test setup

A simplified sketch of the test bench's fluid system can be seen in *Figure 1*.

As previously described, the propellant is mixed in its gaseous state upstream the injector of the combustor. Nitrous oxide and ethene are stored in 50l pressure tanks outside of the test container. The pressure, mixture ratio and the resulting mass flow are adjusted via pressure regulators and calibrated orifices. Additionally the test bench is equipped with H₂ and O₂ feeding lines to supply the torch igniter. For flushing of the combustor, to realize redline shutdown sequences and for operation of the pneumatic valves nitrogen supply is needed. Each feeding line is equipped with at least one pressure transducer upstream the main valve. At the N₂O and C₂H₄ lines upstream each orifice a pressure sensor and a thermocouple are mounted, corresponding measurement data are used for mass flow calculation. Downstream the tube junction where the nitrous oxide and the ethene are mixing, another pressure transducer is situated. This sensor is delivering the inlet pressure to the corresponding injector.

2.2. Combustor design

A sectional view of the combustor can be seen in *Figure 2*. On the left hand side upstream the combustor a Plexiglas tube is mounted. Thus a flashback during a test run can be observed optically via camera.

The Plexiglas tube is connected to the injection element of the combustor. These elements are designed to be exchanged easily, so different injector types or injector geometries can be tested. All the tests described in this paper were

conducted with a showerhead injector. The injector consisted of 17 boreholes with a diameter of 0.65 mm. *Figure 3* shows a photo of the injector.

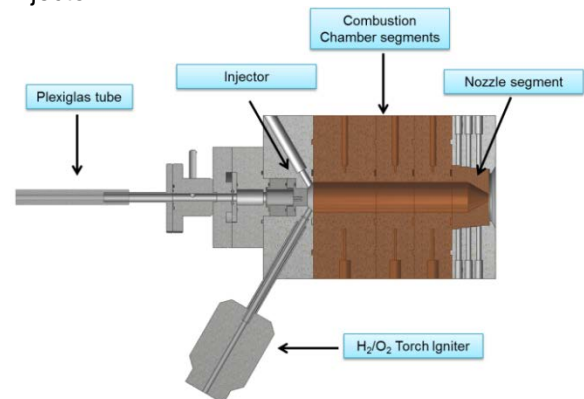


Figure 2: Combustor design

Two of the originally foreseen boreholes were blocked due to a manufacturing failure, so only 17 of 19 injector holes were completely drilled.

Upstream the injector a pressure transducer (P-INJ) is mounted, thus the pressure drop across the injector can be derived.



Figure 3: Showerhead Injector

The igniter is a H₂/O₂ torch igniter commonly used for different research activities at various test benches. The igniter is equipped with hydrogen and oxygen feeding lines in which calibrated orifices are mounted. The orifices assure an oxygen/hydrogen mixture ratio of about 1.5 in case of a sonic flow. Due to the design of the igniter during all operation modes sonic speed was reached in the orifices. By using a big excess of hydrogen relatively low temperatures (1277 K [13]) in the igniter were achieved. The overall igniter mass flow at its nominal operation point is about 2.6 g/s. In the frame of the later described tests, the igniter's mass flow was reduced to about 1.3 g/s. The excess of hydrogen and the torch's flame significantly influence the combustion process of the N₂O/C₂H₄ propellant

mixture during the first 1.5 s of a test run. At the igniter a pressure sensor (P-ZÜND) and a thermocouple (T-ZÜND) are installed. During the ignition sequence the proper function of the ignitor can be supervised by those sensors.

The HyNOx combustion chamber itself consists of Elbrodur (CuCr1Zr) segments with different axial length. The recent configuration's overall length is 110 mm with a combustion chamber diameter of 24 mm; the first segment is 50 mm, the other two 30 mm long. At the center of each segment three thermocouples (T-BK-X-Y) and a pressure sensor (P-BK-X) are installed. The thermocouples are placed in 3 mm, 8 mm and 12 mm radial distance from the inner combustion chamber wall.

The combustion chamber is completed by a nozzle segment. Here CuCr1Zr nozzles with different throat diameters and expansion ratios can be used. During the described tests a truncated nozzle ($\epsilon = 1$) with a throat diameter of 5mm was used.

3. EXPERIMENTAL PROCEDURE AND TEST RESULTS

All combustion tests were conducted by using the gaseous premixed propellant. Nitrous oxide and ethene were mixed about 0.3 m upstream the injector at a tube junction. The mixture ratio and the mass flow rates were controlled by using calibrated orifices and adjusting the feeding pressure of the corresponding gases.

3.1. Preliminary tests

Prior to the combustion tests, several sets of orifices were calibrated. The effective diameters of the orifices were derived by using an experimental setup equipped with a Coriolis mass flow meter, pressure and temperature sensors and assuring sonic flow through the orifice. Equation (1) and (2) were used to determine the effective diameter of the orifice for a chosen pressure. The density upstream the orifice (ρ_0) and the heat capacity ratio κ for a given pressure (p_0) and temperature were taken from the REFPROP database [25]. For each combustion test the diameter was used to calculate the mass flow rate (equation (3)).

$$\psi = \left(\frac{2}{\kappa + 1} \right)^{\frac{1}{\kappa - 1}} \sqrt{\frac{\kappa}{\kappa + 1}} \quad (1)$$

$$d_{eff} = \sqrt{\frac{4 * \dot{m}}{\pi * \psi * \sqrt{2p_0\rho_0}}} \quad (2)$$

$$\dot{m} = \frac{\pi}{4} d_{eff}^2 * \sqrt{2p_0\rho_0} * \psi \quad (3)$$

3.2. Test preparation and sequence

In preparation of each combustion test the whole setup was pressurized and leakage tested, the pressure transducers and thermocouples were checked and the test sequence was programmed. At the beginning of the test sequence, the whole setup was purged with nitrogen from -8 to -5 s to assure identical start conditions (see *Figure 4*). After nitrogen purging, all valves stayed closed for additional 5 seconds. The H₂/O₂ igniter was lighted at 0 s, N₂O and C₂H₄ valves opened at 0.5 s and the propellant mixture was ignited. The H₂ and O₂ valves closed at 1.0 s, the propellant continued to burn for 10 seconds until the main valves were shut. After closing the N₂O and C₂H₄ propellant lines, all valves stayed closed for additional 10 seconds, then post test run nitrogen purging started.

3.3. Measurement data

Figure 4 shows the pressure curves of a test run with a mass flow of about 12 g/s and an oxidizer to fuel ratio of about 10.

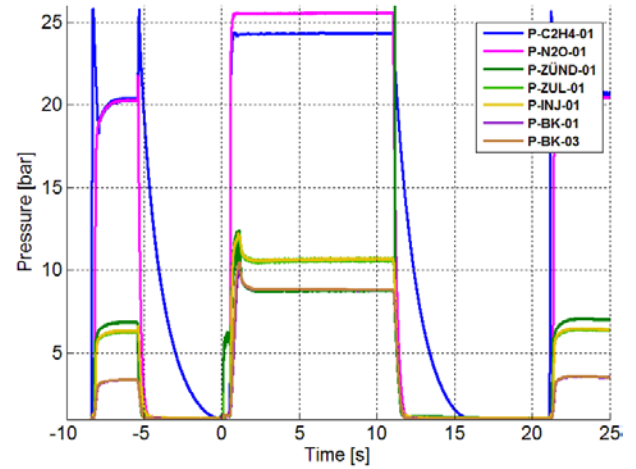


Figure 4: Pressure data during combustion test V_1 212

In the diagram, P-C2H4-01 (blue) marks the ethene pressure upstream the orifice, respectively P-N2O-01 (magenta) names the nitrous oxide pressure. P-ZÜND-01 (dark green) is the pressure in the torch igniter, P-ZUL-01 (light green) the pressure of the mixture in the combined N₂O/C₂H₄ feeding line, P-INJ-01 (yellow) the pressure directly upstream the injector and P-BK-01 names the combustion chamber pressure.

The diagram (Figure 4) shows the nitrogen purging at the start of the test run, the ignition at 0s, the pressure rise due to the torch igniter and the stable combustion regime of the N₂O/C₂H₄ after turning off the igniter.

The propellant valves were closed 11 s after the torch igniter lit up and 10 seconds after the H₂/O₂ valves were closed. 10 s after all valves were closed, nitrogen purging started again (at 22s).

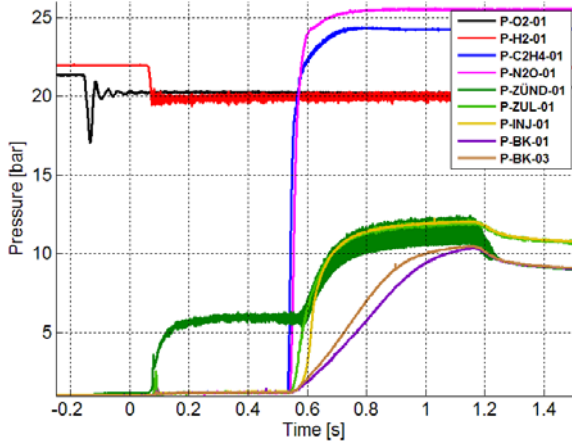


Figure 5: Ignition of N₂O/C₂H₄ mixture, V_1 212 Detail

Figure 5 shows in detail the pressure development during ignition. The igniter started working at approx. 0.1 s, raising the pressure in the igniter to 6 bar. At 0.5 s the N₂O/C₂H₄ mixture is injected. Due to the igniter and the corresponding mass flow of the N₂O/C₂H₄ mixture, the chamber pressure rises to more than 10 bar. Turning off the igniter is leading to a decrease of the combustion chamber pressure to the value of around 8.8 bar. The oscillations of the igniter's pressure (green line) are characteristic for the used torch igniter. They are caused by the H₂/O₂ combustion in combination with the coaxial injector of the igniter.

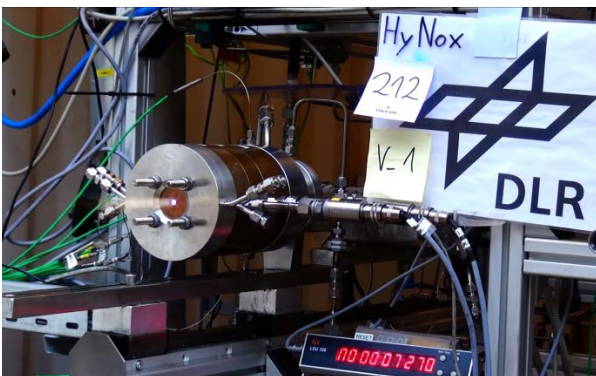


Figure 6: Screenshot of test video

3.4. Mass flow rates and mixture ratio

The mass flow rates, the average chamber pressure and the oxidizer to fuel ratio of the conducted tests are shown in Table 1. The test sequence was conducted as described above. The mixture ratio and mass flow were derived from the pressure drop across the orifices. During test V_1 201 to V_2 209 the N₂O and C₂H₄ valves stayed open for 5.5 seconds, for the tests V_3 209 to V_1 212 the opening time of the valves, respectively the N₂O/C₂H₄ combustion time was increased to 10.5 s.

To ignite the mixture during each test the torch igniter was used. The total mass flow of H₂ and O₂ was 2.6 g/s for the tests V_1 201 to V_3 206. For the following runs (V_1 207 to V_3 207) the igniter's mass flow was reduced to 1.95 g/s, followed by a reduction to 1.3 g/s during startup of the runs V_1 208 to V_1 212. The H₂/O₂ mass flow was reduced to avoid high thermal loads on the combustion chamber and to keep the influence of excess hydrogen during the ignition process as low as possible.

Table 1: Test results, mass flow, mixture ratio and chamber pressure

Test-No.	Average mass flow [g/s]	Mixture Ratio (O/F)	Average chamber pressure [bar]
V_1 201	4,35	9,04	3,05
V_2 201	4,33	8,88	3,08
V_1 202	6,56	10,02	4,7
V_2 202	6,59	10,1	4,75
V_1 203	9,55	10,39	7,01
V_2 203	9,68	10,49	7,02
V_1 204	12,07	9,25	8,90
V_2 204	12,16	9,26	8,90
V_1 205	13,6	8,62	10,1
V_2 205	13,8	8,75	10,19
V_1 206	16,64	9,02	12,49
V_3 206	17,07	9,15	12,64
V_1 207	16,91	9,05	12,53
V_2 207	16,96	9,1	12,5
V_3 207	16,39	8,78	12,42
V_1 208	16,59	8,98	12,72
V_1 209	4,46	10,82	3,15
V_2 209	4,44	10,8	3,14
V_3 209	4,46	10,83	3,14
V_1 210	6,76	10,3	4,89
V_2 210	6,75	10,3	4,88
V_1 211	9,54	9,45	7,02
V_2 211	9,61	9,46	7,05
V_1 212	12,06	10,09	8,83

The prefix of the test number (V_1, V_2, and V_3) indicates a repetition of the test under identical conditions. Those tests were conducted directly after each other, only separated by nitrogen purging. Due to the short time intervals in between the tests, the capacitively cooled combustion chamber heats with each test run.

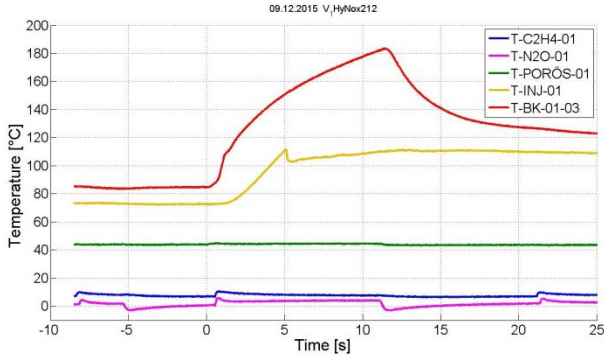


Figure 7: Temperatures during test run V_1 212

This results in high combustion chamber temperatures at the start of the subsequent tests (approx. 100°C). Figure 7 shows the temperature course at several positions of the combustion chamber and setup. T-BK-01-03 marks the temperature at the first chamber segment in 3 mm distance to the inner combustion chamber wall. T-INJ-01 is the temperature in 3 mm distance from the hot wall at the faceplate. The sudden decay at a test time of 5 s is caused by a lost contact of the thermocouple to the stainless steel surface.

T-PORÖS marks a thermocouple in the N_2O/C_2H_4 propellant feeding line, directly upstream the injector. The temperatures of the gases upstream the orifices are named T-N2O-01 and T-C2H4-01. A detailed analysis of the occurring heat flux and the temperature development during the described tests can be found in [28].

3.5. Flashback challenges

In several tests flashback into the feeding line was observed. The occurrence of flame propagation across the injector was recorded via video camera, focused on the Plexiglas tube [23]. For those events, a distinct pressure and temperature rise in the feeding line could be detected. For the conducted tests immediate propagation of the flame across the injector was found only if a hard ignition of the propellant occurred. It is assumed that a delayed ignition of the mixture leads to an explosive decomposition. Detailed analysis of the flashback and ignition events will take place in future.

4. DISCUSSION

With the results for the chamber pressure, the mass flow and the corresponding mixture ratio, c^* is derived. The experimental c^* is compared to the theoretical performance calculated with NASA CEA [13]. Furthermore possible measurement errors and deviations of the theoretical and experimental performance will be described.

4.1. Theoretical and experimental c^*

Figure 8 shows the absolute c^* above the mixture ratio for the described test runs. The experimental c^* was derived using equation (4). The black squares mark the experimental c^* for the corresponding test number.

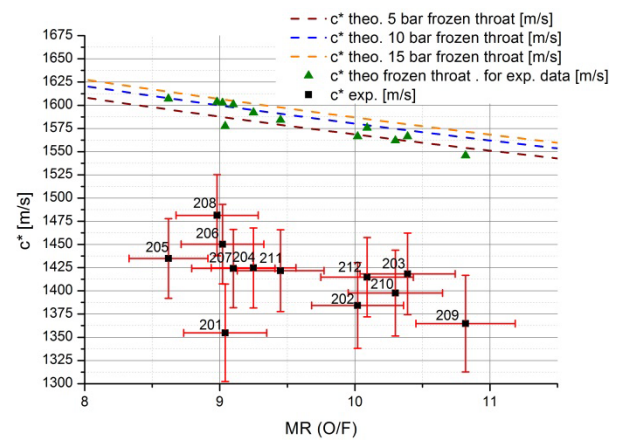


Figure 8: Experimental and theoretical c^* for different chamber pressures and mixture ratios

During the tests, the mixture ratio was varied in between 8.5 and 11. Different results for c^* at the same mixture ratio are caused by different combustion chamber pressures. The green triangle above each experimental value (black square) indicates the corresponding theoretical c^* . The theoretical c^* was calculated via NASA CEA, by using the reaction model "frozen at throat". This results in an equilibrium reaction in the combustion chamber. The dashed curves indicate the theoretical c^* for a variation of the chamber pressure (5, 10 and 15 bar). c^* rises with increasing combustion pressure. The increase of the experimental and theoretical c^* with increasing chamber pressure is caused by an increase of the combustion temperature (T_c), a decay of the isentropic coefficient (κ) and small changes in the specific gas constant (R_s) due to changes in the combustion products composition.

$$c^* = \frac{p_c \cdot \pi \cdot d_t^2}{4 \cdot \dot{m}} = \frac{\sqrt{\kappa R_s T_c}}{\kappa \sqrt{\left(\frac{2}{\kappa + 1}\right)^{\frac{\kappa + 1}{\kappa - 1}}} \quad (4)$$

Equation (4) shows the relationship of c^* and the mentioned thermodynamic variables. Here p_c is the combustion chamber pressure, d_t the nozzle throat diameter and \dot{m} the overall mass flow.

$$\eta_{c^*} = \frac{c^*_{exp.}}{c^*_{theo.}} \quad (5)$$

The results for the characteristic velocity ($c^*_{exp.}$) were compared to the theoretical results. The ratio $c^*_{exp.}/c^*_{theo.}$ indicates the combustion efficiency for the chosen chamber, injector and nozzle geometry (see Equation (5)).

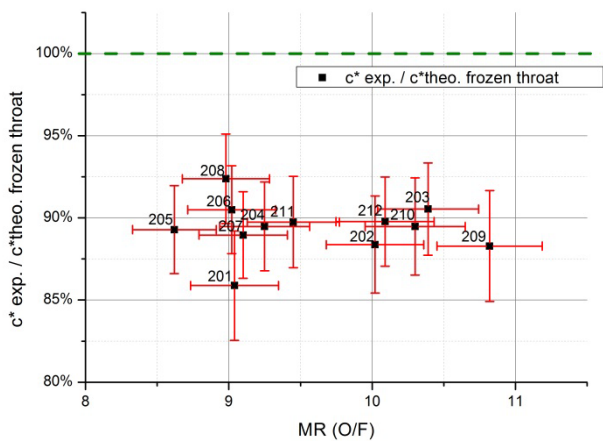


Figure 9: Combustion efficiency for the conducted test runs

The results for η_{c^*} depending on the mixture ratio are seen in Figure 9. For the test series the combustion efficiency was in between 86% and 92%. When plotting the absolute c^* above the chamber pressure for the tests runs, the described increase of c^* with rising chamber pressure can be seen in Figure 10.

Both, the theoretic and the measured c^* increase with rising chamber pressure. Here Figure 10 shows the theoretical characteristic velocity for reactions frozen at the nozzle throat (green) and for reactions frozen at the combustor's end (blue) assuming an infinite area combustor. The solid lines are linear fits through the calculated and measured data. The lines for the theoretical c^* (colored green and blue) show nearly identical gradients, while the experimental value's fit seem to get closer to the theoretical values with increasing chamber pressure. This effect can also be seen in Figure 11.

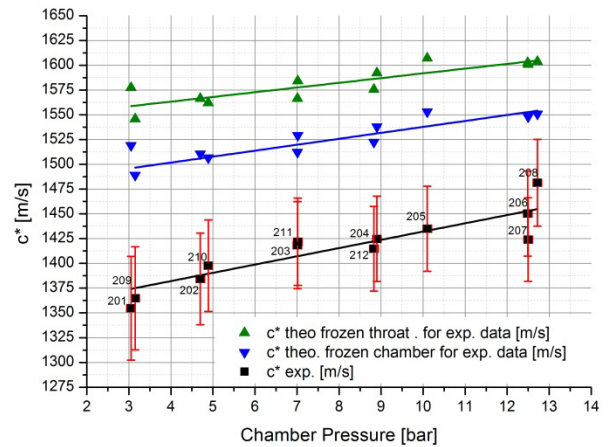


Figure 10: Experimental and theoretical c^* depending on chamber pressure

The fit of the combustion efficiency increases from about 88% to 91% with a rise of the chamber pressure from 3 to 12.5 bar.

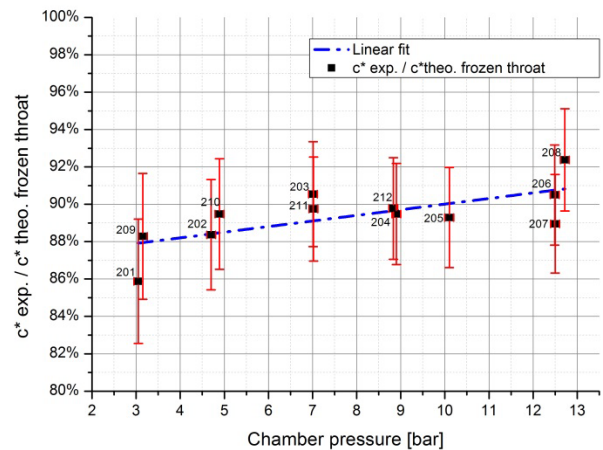


Figure 11: Combustion efficiency depending on chamber pressure

The effect of rising combustion efficiency with increasing combustion chamber pressure is mentioned in the literature [29]. With higher chamber pressure the reaction paths shift more to the equilibrium conditions, which are calculated in NASA CEA. Another effect additionally increases the combustion efficiency: with rising chamber pressure the released energy per volume increases. The rise of additional energy exceeds the energy losses via heat conduction at the chamber walls. So the heat generation due to the chemical reactions grows stronger than the rise of heat conduction due to higher chamber pressure. Thus more energy for acceleration of the exhaust gases is available and the combustion efficiency increases.

4.2. Measurement error

To calculate the measurement uncertainty and derive the error bars, the deviation of each sensor was calculated or compared to calibrated devices. After deriving the sensor's deviation, equation (6) was used to calculate the uncertainty of the experimental c^* , the pressure and the mixture ratio.

$$u_y = \sqrt{\left(\frac{\partial y}{\partial x_1} * u_1\right)^2 + \left(\frac{\partial y}{\partial x_2} * u_2\right)^2 + \dots} \quad (6)$$

In equation (6) u_y marks the overall deviation, depending on the derivative of equation y (in our case c^*) to the single variables x_1 , x_2 , etc. (here p_c , d_t , \dot{m}).

The maximum error for the coriolis mass flow sensor (Emerson CMF025) which was used to calibrate the orifices is 2.4%. This deviation occurs only when the mass flow is at the far end of the measurement range. Due to the uncertainty of the mass flow sensor, the mixture ratio's deviation was derived correspondingly. Furthermore the pressure sensors of the combustion chamber were pressurized parallel with a calibration module (Beamex MC5). The resulting data recorded by the measurement system was compared to the measured values by the calibration module. As result of this end to end comparison a maximum sensor deviation of +/- 0.1 bar was found. The nozzle's throat diameter was measured under ambient temperatures before each test run and in hot conditions after the test runs. Due to heating of the nozzle segment the throat diameter varied by 0.08 mm (4.92 mm up to 5 mm). The calibration of the orifices (see chapter 3.1) took place some time ago. Thus a cross check of the mass flow calculation via the earlier determined orifice diameter and an additional measurement with the coriolis sensor took place. This comparison showed deviations of smaller than 0.5% in between the calculated and the measured values, thus these errors were neglected. Furthermore the contraction ratio of the combustion chamber is quite high (23.41), this results in low Mach numbers (approx. 0.025). With this the deviation in between the static and absolute pressure is approximately 0.3%. Thus for the c^* calculation the experimental measured static pressures were used.

5. SUMMARY AND OUTLOOK

5.1. Summary

DLR's institute of Space Propulsion set up a green propellant test bench to analyze the combustion and ignition behavior of a so called premixed green propellant. The propellant mixture consists of nitrous oxide (N_2O) and ethene (C_2H_4) and was called "HyNOx" – hydrocarbons with nitrous oxide. First combustion tests with gaseous N_2O and C_2H_4 were conducted. During a test campaign the mass flow and mixture ratio was varied. The results of the conducted tests are:

- a) Pressure and temperature data at different positions of the combustor were collected. A set of 24 hot runs with different mass flow and mixture ratios was conducted.
- b) The combustor is equipped with thermocouples at different axial and radial chamber wall positions; the heat fluxes for the test runs were derived (see [28]).
- c) The mixture's theoretical c^* reaches up to 1650 m/s. During the test campaign a maximum combustion efficiency of 92% could be observed, reaching a c^* of about 1480 m/s.
- d) The combustion efficiency and the absolute c^* of the propellant mixture increases with rising chamber pressure
- e) The appropriate design of flashback arrestors is essential for the use of the propellant mixture
- f) The setup will be modified to conduct tests with a wider range of mixture and mass flow ratios. Other injection and ignition methods as well as flashback elements will be tested.

5.2. Modifications of the setup

Due to test results, several modifications at the setup will take place. A wider range of mixture ratios and mass flow variations will be investigated. Further new injectors and flashback arrestors will be tested [30].

To increase the accuracy of the mass flow and mixture ratio data, the feeding lines of N_2O and C_2H_4 will be equipped with a coriolis mass flow meter each. One main benefit of this modification is that the effective diameter of the orifices does not have to be derived prior to the tests. Additionally no sonic speed in the orifices is needed to increase the accuracy of the measured data.

Furthermore the test bench will be equipped with an optimized thrust measurement. So in addition

to the c^* values, the thrust of the combustor can be derived. With the thrust measurement the I_{sp} under ambient conditions can be derived. With the measured I_{sp} the resulting vacuum I_{sp} will be predicted.

5.3. Testing Flashback arrestors

Critical for a safe use of the premixed propellant mixture in a combustor is the avoidance of flashback during start up, stationary operation and shut down. To analyze and design appropriate flashback arrestors, a measurement section for flashback analyses will be set up. Here two chambers with the propellant mixture in a defined condition will be separated by a permeable wall of a porous material. By igniting one of the chambers, propagation of the flame to the other chamber or its extinction will be investigated. Different porous materials will be tested; furthermore the occurring pressure drop of the elements will be measured and calculated [31].

5.4. Liquefaction facility

For final tests with liquid propellants, a liquefaction setup will be reconstructed. The setup will produce small batches of liquefied N_2O and C_2H_4 to conduct tests with the premixed propellant. The mixture will be produced with different oxidizer to fuel ratios. The resulting performance of the liquid propellant will be compared to the performance of the gaseous mixture.

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