

# Experimental Investigation of Heat Flux in a Small-Scale Oxygen-Hydrogen Rotating Detonation Rocket Combustor

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## 1 Introduction

In recent years, an ever increasing amount of research has focused on the field of Rotating Detonation for combustion applications, because of its promise to provide an efficiency advantage compared to deflagrative combustion. The Rotating Detonation combustion is a pressure gain combustion process and utilizes the Fickett-Jacobs thermodynamic cycle in contrast to the commonly used deflagrative combustion utilizing the isobaric Brayton cycle. Most of the past work on detonative combustion was conducted in the field of energy generation with airbreathing Rotating Detonation Combustor (RDC) to investigate the increase of overall efficiency. Recently, the interest in RDC for rocket applications increased and a lot of research in this area is being done. As the detonation process in RDCs is a highly unsteady and fast process, many challenges need to be overcome. One of these challenges is the high heat load and the related high heat flux at the combustion chamber wall, which occur due to the higher combustion temperatures and the more compact combustion regime of the detonation. Knowledge about the heat loads and heat fluxes and their spatial distribution in the combustion chamber are key information for the design of the combustor's cooling system, necessary to enable longer test durations.

To also contribute to this research activities, Technische Universität Dresden (TUD) conducted a test campaign to measure heat fluxes in a small-scale GOX/GH<sub>2</sub> Rotating Detonation Engine (RDE) at German Aerospace Center's (DLR) Institute of Space Propulsion in Lampoldshausen. The focus was on heat flux measurements through the outer combustion chamber wall of a capacitively cooled combustor, developed by Armbruster et al. [1]. For reasons of simplicity, for this study a method using clusters of thermocouples to implement a gradient heat flux measurement methodology was used and integrated in a small-scale RDC powered by gaseous hydrogen and gaseous oxygen [1].

## 2 Experimental Set-up

The value of interest is the heat flux through the outer wall of the combustion chamber of an RDC powered by gaseous hydrogen and gaseous oxygen.

### Test Specimen

The combustor was originally designed by the German Aerospace Center (DLR) [1] and has already been tested successfully [1–3]. The combustor is of the annular type and has an outer channel diameter of 68 mm, the combustion chamber inner diameter is 59 mm, resulting in a 4.5 mm channel width. Its length is 50 mm. The material used for the inner and outer body is a copper-alloy to allow for a fast heat distribution in the material and thus implementing a capacitive cooling. Fig. 1 shows a cross-cut view of the combustor. An Aerospike Nozzle with a contraction ratio of 2 can be mounted optionally (part 5 and 6 in Fig. 1). The injector plate utilizes 72 pairs of unlike impinging injectors with a diameter of 1.5 mm for the oxygen orifice and 1.0 mm for the hydrogen orifice.

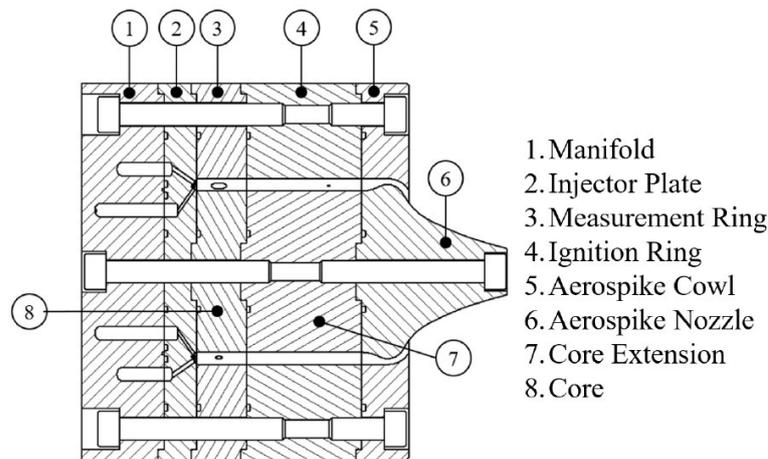


Figure 1: Cut-view of the used RDRC with installed Aerospike nozzle (parts 5 and 6) [Armburster'2023]

### Test Bench

The test campaign was conducted at the test facility M3 at DLR. This facility has been utilized for more than 30 years for research and technology development in the field of cryogenic rocket propulsion. Its research focus is on key processes within rocket combustion chambers and supply systems, which include propellant conditioning, transient flow, injection, ignition sequencing, and combustion. The M3 facility features three distinct test positions: M3.1, M3.3, and M3.5, from which M3.1 was utilized for the test campaign. The M3.1 test position is specifically designed for injection, ignition, and combustion testing. It offers the flexibility of variable ignition sequencing, by fast-reacting test bench valves capable of sequencing propellant flows in the order of milliseconds. Currently, the setup supports testing with liquid oxygen/hydrogen and liquid oxygen/methane at subcritical pressures. Further information regarding the test bench M3.1 can be found in [4].

### Heat Flux Measurement Method

In this test campaign, a gradient method to measure heat flux was implemented. This method has already been used to measure heat fluxes in a rocket combustor experiment at DLR [5]. The basis concept of this method is to determine the heat flux by measuring temperature at different distances to the hot gas wall side. Via the differences in temperatures at the different positions, i.e. the temperature gradient in the material, the heat flux can be calculated using Eq. 1.

$$\dot{q} = \lambda_{cc} \frac{(T_1 - T_2)}{\ln(r_2/r_1) \cdot r_{cc,o}} \quad (1)$$

This equation is valid for stationary heat flux in cylindrical tubes. In this equation,  $\lambda_{cc}$  represent the thermal conductivity of the chamber material,  $r_{cc,o}$  represent the radius of the outer hot gas wall side of the combustion chamber.  $T_1$  and  $r_1$  are the values for temperature and radius for the temperature measurement position at lower radial position. Respectively,  $T_2$  and  $r_2$  represent the values at the higher radial measurement position.

To measure the temperature at the different positions, in this campaign thermocouples of the type K with a 0.5 mm diameter have been used. The data was acquired with a National Instruments NI-9214 card with a sample rate of 30 Hz. Unique to this method is, that the thermocouples are mounted with a spring preloading a radial force to the thermocouple to ensure that the thermocouple is always in contact with the combustion chamber material. Fig. 2 shows a scheme of the general mounting situation of the thermocouples. The used thermocouples are off- the-shelf thermocouples which were modified in-house at DLR.

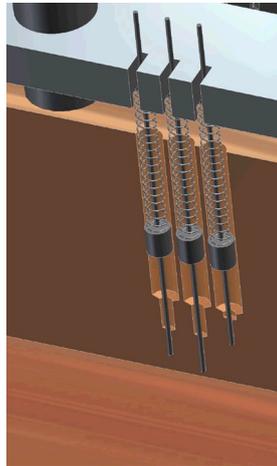


Figure 2: Scheme of general installation situation, modified [6]

Usually, this method is implemented to determine heat fluxes in combustors with thermal equilibrium, i.e. steady heat fluxes. As the possible run times with this capacitive cooled RDE are limited to the order of 1s, uncertainties are expected. Nevertheless, this test campaign is being used to gain data for the development of a second demonstrator, aiming for longer test runs and thus reducing the uncertainties.

### Implementation of the Method

The test specimen was modified to fit 11 more pre-loaded thermocouples. Tab. 1 gives an overview of the thermocouple positions.

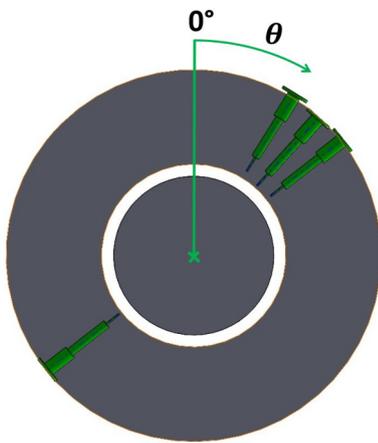
A schematic cross-cut view of the combustor showing the thermocouples' positions is presented in Fig. 3a. With this set-up, heat fluxes can be measured at 3 axial positions allowing for an axial resolution of the heat flux. With the two opposing thermocouples at the azimuthal position  $\theta = 235^\circ$ , a check for symmetry of the temperature in the combustor can be performed.

### Test Matrix and Testing

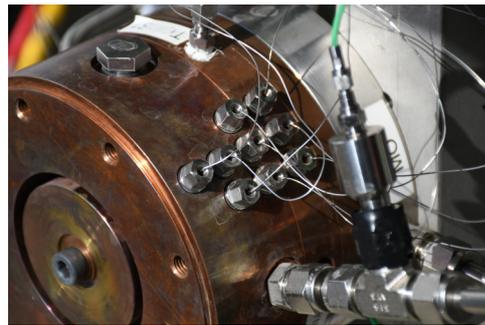
The aim of this test campaign was to measure heat fluxes into the outer combustor hot gas side wall at different operation modes. Therefore, the ratio of oxidizer to fuel (ROF) was varied between 5.8 and 7.5. Moreover, mass flow rates were varied between  $(45-70) \text{ g s}^{-1}$  resulting in mass fluxes varying between

Table 1: Overview of thermocouple positions

Nr.	Axial Position $x$ / mm	Azimuthal Position $\theta$ / °	Distance to hot gas side wall $wd$ / mm
1	8	35	3
2		45	1
3		55	2
4		235	1
5	22	35	3
6		45	1
7		55	2
8		235	1
9	35	45	1
10		55	2
11	48	45	1



(a) Thermocouple positions at axial position  $x = 8$  mm distance from the injector plate, thermocouple azimuthal positions: 35°, 45°, 55° and 235°



(b) Combustor with install thermocouples

Figure 3: Position of thermocouple cluster

(50–80)  $\text{kg s}^{-1} \text{m}^{-2}$ . Additionally, test with and without an Aerospike nozzle with a contraction ratio of 2 were conducted. For most of the test a run time of 0.7 s was realized, some selected operation points were tested at elongated run times of 1.5 s. All in all, a total number of 25 tests have been conducted.

### 3 Preliminary and Planned Data Analysis

As example for the data analysis, preliminary results are shown in Fig. 4. Data is taken from a test with an aimed total mass flow of  $45 \text{ g s}^{-1}$  at an ROF of 5.8. The Aerospike nozzle with a contraction ratio of 2 was mounted. The run time was 0.7 s.

From the temperatures at the axial position of 8 mm distance to the injector plate, the heat fluxes can be calculated by using multiple combinations of the difference in temperature at the varying radial positions, as shown in Fig. 4b. For this test, at this axial position the calculated heat fluxes provide

similar values.

Fig. 4a shows the temperature at 11 mm distance to the hot gas side wall for different axial positions at azimuthal positions of  $45^\circ$ . As it can easily be identified, the measured temperatures decrease with increasing axial position, except for the temperature at 48 mm distance to the injector plate, temperature is increasing again, most likely caused by the contraction from the Aerospike nozzle, with its smallest cross-sectional area at 51.3 mm.

Also notable is the much higher axial temperature gradient between axial position 8 mm and 22 mm compared with the temperature gradients between the other axial positions, which can be identified in Fig. 4a.

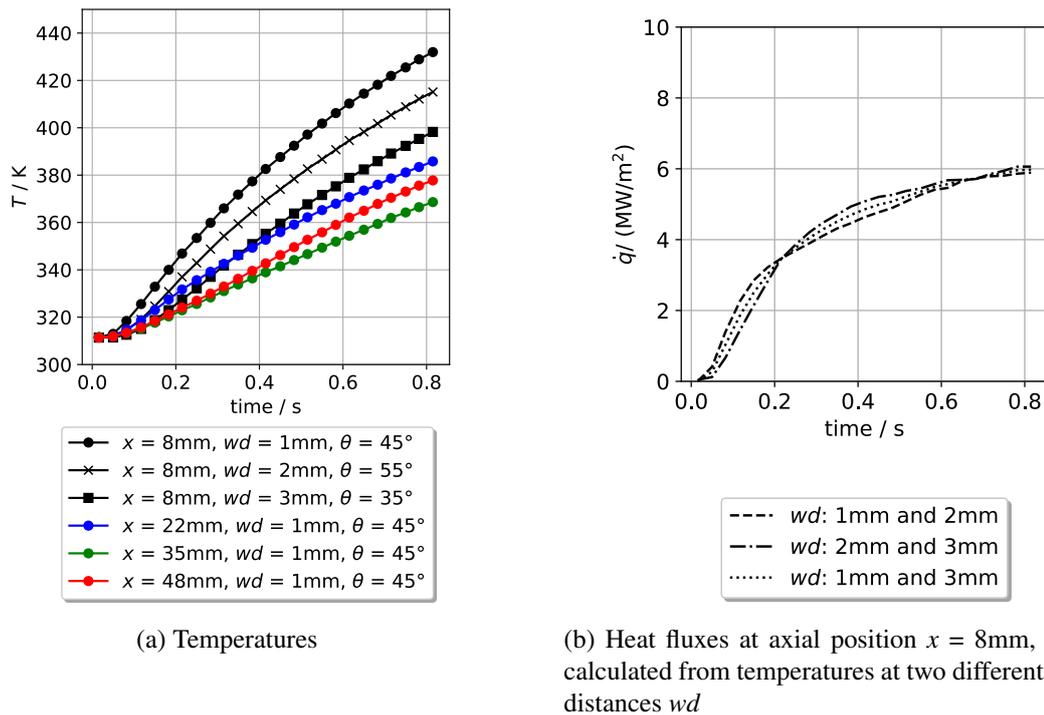


Figure 4: Temperatures and heat fluxes from test with:  $\dot{m}_{aim} = 45 \text{ g s}^{-1}$ , runtime of 0.7 s,  $ROF_{aim} = 5.8$

The final paper will present a more detailed analysis of the thermocouple data acquired during the test campaign. Heat fluxes for different axial positions are calculated based on the temperature measurements at varying distances from the hot gas side wall. Moreover, a comparison of the obtained results will be presented, giving further insights into the heat loads and their distribution during the operation of an RDE. Finally, an error estimation of the utilized evaluation technique will be incorporated.

#### 4 Project Overview of NEDSERD

This test campaign was conducted within the project ‘Numerical and Experimental Demonstration Study for Engines using Rotating Detonation’ (NEDSERD). In this project, TUD researches Rotating Detonation Engines (RDEs) in the context of space propulsion applications. NEDSERD is funded by the Federal Ministry for Economic Affairs and Climate Action (BMWK) and is conducted in cooperation with ArianeGroup GmbH and DLR Institute of Space Propulsion. During the project, Computational Fluid Dynamics (CFD), experimental work and a system analysis are being conducted to increase TRL of RDEs in Europe and gain understanding how to design and operate RDEs.

Thus, a subgoal in this project is the development of a verified numerical model of an RDE to predict

the flow field and associated values. At the beginning, a simplified numerical model in a 2D domain will be used to gain first insights on how to build up a capable numerical model of a continuously traveling detonation wave and how to initiate a self-sustaining detonation wave in the numerical domain. Based on the findings from the development of the 2D model, a 3D model will be established.

Verification and validation data will be provided by an initial experimental test campaign at DLR Institute of Space Propulsion using their small-scale RDE run by gaseous hydrogen and gaseous oxygen, [1, 2]. With the verified numerical model and the experimental experience gained in the initial test campaign, an updated design of the demonstrator with increased TRL will be developed. Experimental data from a second test campaign performed with the second demonstrator will provide further reference data for the verification and validation of the numerical model. With the verified and validated numerical model parameter studies will be conducted to identify critical parameters and start deriving guidelines for scaling RDEs.

The presented data was gained during the initial test campaign at DLR. The second test campaign with an improved combustor design is planned to be conducted in the test facility M11 [7] at DLR Institute of Space Propulsion, allowing for testing at higher mass flow rates.

In parallel to the numerical and experimental work, a system analysis is being conducted. The aim of this analysis is to investigate the influence of RDEs on a system level in comparison to conventional space propulsion. During this analysis, a reference configuration will be selected and the integration of this configuration into the spacecraft system will be investigated.

## 5 References

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