Impact of heat transfer on a 1 inch rotating detonation rocket engine

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1 Introduction

The race to expand surveillance capabilities in cislunar space has revealed a pressing need for more versatile and highly efficient satellite propulsion systems. Detonation has emerged as a promising mechanism for such applications, with rotating detonation rocket engines (RDREs) showing significant potential for aerospace propulsion. RDREs operate using continuous, self-sustaining detonation waves that propagate around a cylindrical or annular chamber to generate thrust and are understood to be up to 10% more efficient than conventional rocket engines [1]. Additionally, when operating in stable, locked detonation modes, RDREs are less likely to experience unwanted oscillations in combustion chamber pressure—a phenomenon that conventional rocket engines require complex tuning to avoid [2].

The study of RDREs dates to the 1960s, following Voitsekhovskii's first observation of spinning detonations [3]. Recently, Nagoya University and JAXA performed the first in-space demonstration of an RDRE, with good agreement in key performance metrics between the in-flight and previous ground testing [4]. Other notable achievements have been NASA's hot fire testing of an RDRE with additively manufactured injector components and combustion chamber [5]. This campaign aimed to demonstrate continuous detonation modes with liquid propellants and achieve long-duration firings exceeding 100 seconds. NASA achieved a 133-second hot fire test that maintained detonation, and during a full-power test, the RDRE generated a measured thrust of 4171 lbf at a maximum chamber pressure of 622 psia [5]. These are key advancements towards demonstrating the feasibility of detonation for in-space propulsion.

However, despite extensive research, RDREs are still considered an emerging technology due to the need for further verification and validation of existing results. These devices are highly sensitive to stochasticity and inlet conditions, and varying conditions can greatly impact performance and detonation behavior [2]. Several factors contribute to the complexity of RDREs, including injection dynamics, heat loss, propellant mixing, and the control of operating modes [6].

Heat transfer is a critical concern in RDRE development, as excessive overheating of hardware can result in catastrophic failures. Laboratory RDREs are often constrained to very short run times due to significant heat flux through device walls. In fact, a recent study on a 6-inch RDRE found detonation wave-generated heat flux spikes exceeding 25 MW/m² [7]. These findings highlight the need for advanced thermal management strategies to mitigate these extreme thermal loads. Furthermore, studying detonation on the small scale introduces a range of additional complexities, including the consideration

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of curvature effects [1] and the replenishment rate of fuel and oxidizer [8]. Thus, investigating heat transfer in small-scale RDREs is a key step towards the development of RDREs for small spacecraft and satellite propulsion.

This numerical study seeks to broaden understanding of detonation in small-scale RDREs by analyzing heat transfer effects in a 1-inch outer diameter combustor, which has been less explored in the literature. Using a compressible reacting flow solver on the OpenFOAM [9] platform, reacting flow of a lean mixture of hydrogen and oxygen is simulated in the small-scale (25 mm, i.e. 1 inch) RDRE geometry. This paper begins with an overview of the numerical frameworks used in this work and outlines the simulation configuration. Current results are then presented with a focus on validation against experiment and the impact of heat transfer.

2 Numerical methods

The reactive Navier-Stokes equations for mass, momentum, species, and total energy (Equations 1- 4) are solved:

$$\partial_t \rho + \nabla \cdot (\rho \mathbf{u}) = 0 \ , \tag{1}$$

$$\partial_t (\rho \mathbf{u}) + \nabla \cdot (\rho \mathbf{u}\mathbf{u}) = -\nabla p + \nabla \cdot \boldsymbol{\tau} \ , \tag{2}$$

$$\partial_t (\rho Y_k) + \nabla \cdot (\rho \mathbf{u} Y_k) = -\nabla \cdot (\rho D_k \nabla Y_k) + \dot{\omega}_k \ , \tag{3}$$

$$\partial_t (\rho E) + \nabla \cdot (\rho \mathbf{u} E) = -\nabla \cdot (\mathbf{u} p) + \nabla \cdot (\mathbf{u} \cdot \boldsymbol{\tau}) - \nabla \cdot \mathbf{q} + \dot{q}_c \ , \tag{4}$$

where ρ is the density, $\mathbf{u}$ is the fluid velocity vector, p is the pressure, Y_k is the mass fraction of a species k, D_k is the mixture-averaged diffusion coefficient for a species k, $\dot{\omega}_k$ is the reaction rate of a species k, and $\mathbf{q}$ is the heat flux vector. The viscous stress tensor, $\boldsymbol{\tau}$, is defined as

$$\boldsymbol{\tau} = 2\mu[\mathbf{S} - \frac{1}{3}(\nabla \cdot \mathbf{u})\mathbf{I}], \tag{5}$$

where $\mathbf{I}$ is the identity matrix, and $\mathbf{S}$ is the rate of strain tensor $\frac{1}{2}[\nabla \mathbf{u} + (\nabla \mathbf{u})^T]$ [10]. The total energy E (excluding chemical contributions) is defined as $e_s + \frac{1}{2}|\mathbf{u}|^2$, where e_s is the specific sensible internal energy. The heat release rate $\dot{q}_c$ is defined by $\sum_{k=1}^{N_s} \dot{\omega}_k h^0_{f,k}$, where $h^0_{f,k}$ is the enthalpy of formation for a species k.

Convective fluxes are computed using a second-order central-upwind scheme [11]. A van Leer limiter is used to extrapolate cell average quantities to cell faces. Diffusive fluxes are computed using a second-order central difference scheme, and time integration is performed using an explicit second-order multi-state Runge-Kutta scheme [12].

The solver employed in this study applies an analytical Jacobian to compute the chemical source terms. The Courant-Friedrichs-Lewy (CFL) number is kept under 0.25 using fixed time steps to ensure a high level of numerical stability [12]. An 11-species version of the Foundational Fuel Chemistry Model Version 1.0 (FFCM-1), a well-validated hydrocarbon fuel reaction model, is used for hydrogen/oxygen combustion [13]. Dynamic load balancing is applied to ensure efficient use of computational resources [12]. The simulation is ignited by imposing an initial ZND wave traveling in the azimuthal direction. This initial wave spans 20 mm in the axial direction, from the injection plane towards the exhaust plenum.

3 Simulation configuration

The RDRE geometry considered in this study was provided by experimental collaborators at the Advanced Propulsion, Energy and Combustion Science Laboratory (APECSLab) at the University of Alabama in Huntsville (UAH). This RDRE is similar to the 25-mm device investigated by Knowlen et al. in a study that demonstrates that regular detonation could be achieved at this scale [1]. Like the RDRE described in [1], the device in this investigation contains 48 flat-faced injector pairs equally spaced around the injection plane. However, a centerbody is not used in this configuration and the combustor is left as a hollow cylinder with no annulus.

The computational domain for this study includes the fuel and oxidizer intake plenums, injectors, and the exhaust expansion region in addition to the combustor itself, which has an outer diameter of 25 mm (nominally 1 inch) and length of 31.25 mm. The exhaust region is a cylinder with a diameter of 50.8 mm and length 46.7 mm. A schematic of the geometry is presented in Figure 1. The geometry is currently discretized with a polyhedral grid with 731770 cells. Results reported here are obtained using slip adiabatic walls, and simulations with nonslip non-adiabatic (500 K isothermal) walls are currently underway.

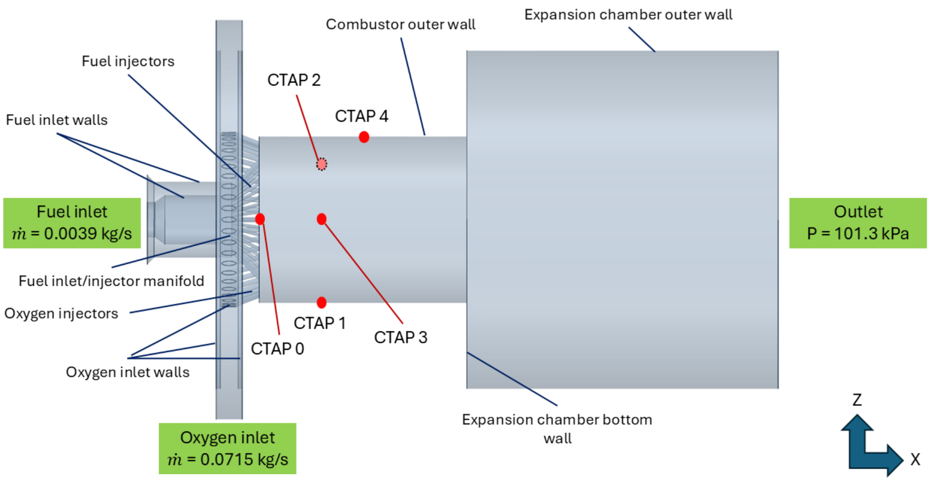


Figure 1: Transparent view of RDRE fluid volume. Capillary Tube Averaged Pressure (CTAP) probe locations are indicated. CTAP 2 is located on the opposite side of the combustor.

The current configuration targets a fuel lean hydrogen/oxygen system, with a global equivalence ratio of 0.43 and a total mass flow rate of 0.0742 kg/s. To achieve this, fuel (H_2) and oxidizer (O_2) inlet boundary conditions enforce a mass flow rate of 0.0039 kg/s and 0.0715 kg/s, respectively. The inlet temperature for both streams is 298 K. At present, a zero gradient pressure boundary condition is applied at both inlets, and the outlet is non-reflecting at atmospheric pressure. For ambient pressure conditions, these parameters yield a Chapman-Jouguet (CJ) speed of 2247 m/s, a von Neumann pressure of 3.69 $\times 10^6$ Pa, and a von Neumann temperature of 1621 K, obtained using the Shock & Detonation Toolbox within the Cantera framework [14].

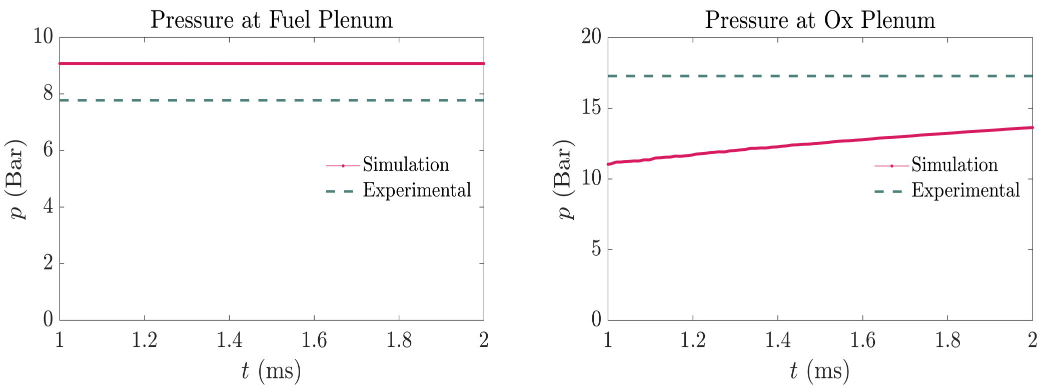
4 Results and discussion

The static pressure, thrust, plenum pressure, and wave speed is first compared with experimental measurements. Subsequently, the impact of heat transfer through the outer wall is qualitatively studied by varying the outer combustor wall boundary from adiabatic condition to 500 K isothermal condition.

Table 1: Average pressure measurements from simulation compared to experimental CTAP measurements. Pressure is measured from simulation at the location of the CTAP port and averaged over 1 ms of simulation time.

Probe	Simulation (bar)	Experiment (bar)
CTAP 0	2.0814	1.9594 ± 0.0146
CTAP 1	1.7308	1.6789 ± 0.0146
CTAP 2	1.5857	1.7894 ± 0.0132
CTAP 3	1.7457	1.8212 ± 0.0123
CTAP 4	1.6381	1.5272 ± 0.0138

Averaged pressure measurements from the adiabatic simulation are compared to the experimental CTAP measurements in Table 1, obtained from the experiments described in [15]. The pressure is generally slightly under-predicted within the combustor. Regarding plenum pressures, the fuel plenum pressure is steady for times $t > 1$ ms and approximately 10% greater than experiment (Fig. 2a). Due to the substantially larger volume, the oxygen plenum does not reach a steady pressure, with pressures increasing towards the experimental value for the duration of the preliminary simulation (Fig. 2b). Nevertheless, the thrust remains steady for times $t > 1$ ms, and (consistent with the CTAP measurements) is approximately 10% less than the experimental results (Fig. 3).



(a) Fuel plenum pressure, adiabatic simulation. (b) Oxygen plenum pressure, adiabatic simulation.

Figure 2: Evolution of fuel plenum (a) and oxygen plenum (b) pressures over the period of steady operation ($t > 1$ ms). The mean plenum pressure from experiments is plotted as the dashed blue line.

Wave speed is extracted from the simulation by averaging the time intervals between successive pressure peaks in the pressure trace at a fixed location. Measured at the outer diameter of the combustor, the wave speed is 3534.8 m/s for the adiabatic case. Preliminary results from the non-adiabatic case show slight decrease of the wave speed in comparison. Both are higher than the experimental value. The relatively coarse mesh used in this study may contribute to the higher wave speed, especially for the isothermal boundary condition case. A grid refinement study is being conducted to further understand wave speed and its variation with boundary condition.

Overall, the predicted RDRE performance using the adiabatic boundary and the isothermal boundary is quite similar, indicating that the effect of wall loss may be small compared to the overall heat release from the combustor. However, the low resolution employed in this study can significantly underpredict the wall loss effect. Future study will investigate whether a larger wall loss term has a greater impact on

RDRE performance.

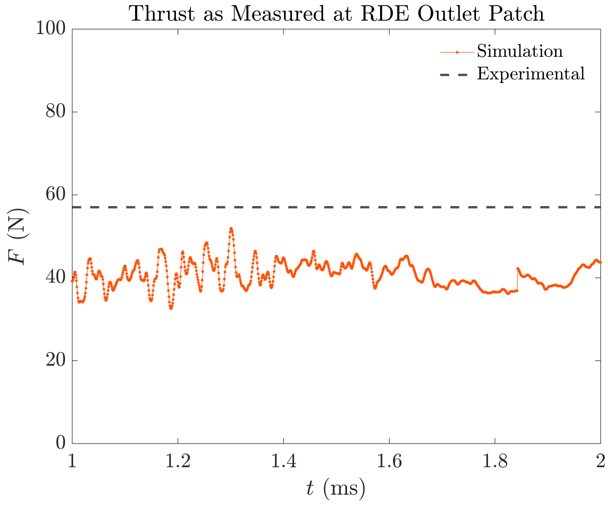


Figure 3: Thrust over time, adiabatic simulation over the period of steady operation ($t > 1$ ms).

5 Conclusion

Simulations of a 1-inch rotating detonation rocket engine are conducted for a lean hydrogen/oxygen mixture. The computational geometry replicates an experimental configuration, including fuel and oxygen plenums and injectors, 1-inch diameter combustor without centerbody, and a large exhaust plenum. Outer combustor wall boundary condition is varied to explore the impact of heat loss on the performance of the small RDRE. Static pressure at several locations along the injector plane and above it is compared with experimental measurement. Plenum pressure is also compared against experimental measurement.

Overall, the combustor pressure is under-predicted by the simulation, and longer duration of simulation may be necessary to further pressurize the combustion chamber and oxygen plenum. Consequently, the thrust is also under-predicted by the simulation by approximately 10%. Wave speed is over-predicted, which may be a consequence of the relatively coarse mesh. Preliminary simulations with isothermal boundary conditions suggest that the detonation is largely insensitive to wall heat losses, likely due to the small heat loss area relative to the volume of the detonation mixture. Future work will focus on achieving higher resolution or improved wall modeling to further understand the impact of wall heat transfer on detonation performance and limit.

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