Experimental Testing of a Rotating Detonation Engine Coupled to Nozzles at Conditions Approaching Flight

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1 Introduction

Rotating detonation engines have been shown to be viable pressure gain combustion devices. Resent experimental thrust stand testing at the Detonation Engine Research Facility of the Air Force Research Laboratory has focused on how best to harness the potential thermodynamic advantages of this novel cycle through various nozzling configurations. The successful conversion of the work-potential provided by rotating detonation engine systems to usable thrust is a key aspect in the future application of pressure gain combustion technology to aerospace propulsion.

Operation of rotating detonation engines of different nominal detonation channel diameters has been reported by Shank et al. [1], Naples et al. [2], Fotia et al. [3,4], Russo et al. [5] and Dyer et al. [6]. Shank et al. presented the initial experimental development effort on a nominal six-inch diameter rotating detonation engine with a focus on mapping the mass flow and equivalence ratio operating space of this new device. Naples et al. examined this device further through the use of a quartz outer-body and high-speed chemiluminescence imaging to provide basic data on the various angles present in the flow structure for use in the validation of modeling efforts.

Fotia et al. [3] reported the effects of exhaust flow nozzling on the measured thrust production of a nominal six-inch diameter rotating detoantion engine. A discussion on appropriate stagnation states is used to identify the effects of pressure gain in the system, while examining the comparative effect on performance of both bluff-body and plug nozzle exhaust schemes. The different regimes of ignition observed in this scale rotating detonation engine are detailed by Fotia et al. [4], where an air ejector configuration is used to provide independent backpreesure to the detonation channel. Dyer et al. tested a larger diameter device with a twenty-inch detonation channel diameter and found that for rotating detonation devices there is a critical interaction between the fuel/air mixing and detonation propagation. The detonation wave/fuel plenum interaction and the potential coupling between the two were further examined by Fotia et al. [7] where a two-dimensional test-section was used in a single-pass operation configuration.

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The modeling and simulation of rotating detonating engines have been more recently conducted by Schwer and Kailasanath [8, 9], Paxson [10] and Davidenko et al. [11]. Schwer and Kailasanath [8] developed a numerical procedure for modeling the flow field of rotation detonation engines which showed good structural agreement with experimental observations, as well they provided indicators for the potential for losses in the combustion of fuel outside of the detonation wave and the post-detonation shocking of combustion products. These same authors [9] later used this algorithm to examine the effects of various engine size parameters that included the nominal diameter, length and depth of the detonation channel, as well as the area ratio of the propellant injection scheme. Paxson proposed a model that allowed for the reduction of the computational time and resources required to simulate a rotating detonation engine by considering a periodic two-dimensional computational space.

A parametric study of flow field parameters was conducted by Davidenko et al., in which the injection total pressure and the spatial period of device operation are identified as scaling factors for the geometry and reactive flow pressure respectively. Average injection mass flux was found to be a factor in driving these two parameters. Rankin et al. [12] showed through simulation that a reduction in exhaust plume unsteadiness in the rotating detonation engine is possible as it transits a converging-diverging nozzle arrangement. This natural reduction in flow turbulence has implications when considering the efficiency of energy conversion into thrust by the nozzle.

2 Experimental Set-up

The data that will be discussed here was collected through the use of a thrust stand installed at the Air Force Research Laboratory's Detonation Engine Research Facility on Wright-Patterson AFB. The testing was conducted on the six-inch diameter modular research (SIMR) rotation detonation engine test-section at air flow rates from 0.30 to 1.82$\pm$0.03 kg/sec, and global equivalence ratios between 0.6 and 1.35$\pm$0.02. The fuel and air mixing scheme at the base of the detonation channel is similar to that used by Shank [1]. Test results from experiments utilizing both gaseous hydrogen and ethylene fuels was be discussed. The SIMR rotating detonation engine was configured to have a 138.6 mm (5.46 inch) center body diameter and an annular detonation channel with a 7.62 mm (0.3 inch) radial gap for the present study. The current configuration of the SIMR test-section is operated as a uncooled heat-sink during operation.

The various nozzle configurations that have been tested are shown in Figure 1. It was shown by Fotia et al. [3] that the bluff-body exhaust schemes achieved fair performances at sea level conditions as compared to the aerospike, or plug, nozzle arrangements. However, the present work is mainly concerned with performance at flight conditions, under which the bluff-body configurations are not suitable.

The three aerospike nozzle configurations were tested, which involve an identical central spike, with two designed to provide a 40% and 20% area constriction of the detonation channel and the other no constriction. The addition of an area constriction allows for the quality of the exit choke on all of the examined cases to be assessed and conclusions made as to the state of the flow at the exit plane of the device. The classical converging-diverging nozzle configuration is used as a baseline to better understand the relative merits of the other configurations considered. Experimental data from all of the above mentioned configurations will be presented.

3 Brief Results

Experimental thrust stand measurements for the converging-diverging nozzle configuration are shown in Figure 2, where various mass flows and equivalence ratios are given for the same combustor geometry. A

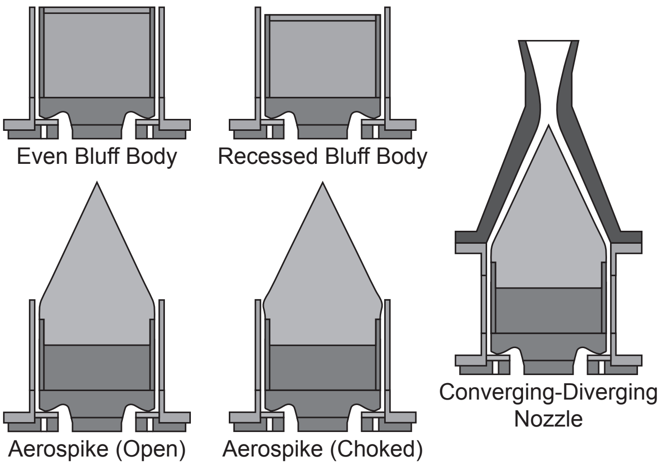

Figure 1: Nozzle configurations tested on a six-inch rotating detonation engine.

couple of trends are readily observed. The first is that for a set geometry there are diminishing returns on attained performance as the mass flow rate through the engine is increased. This reduction indicates that there will be vehicle flight conditions that will dictate either a smaller or larger engine be implemented based of the flight profile, and under which conditions maximum efficiency is required.

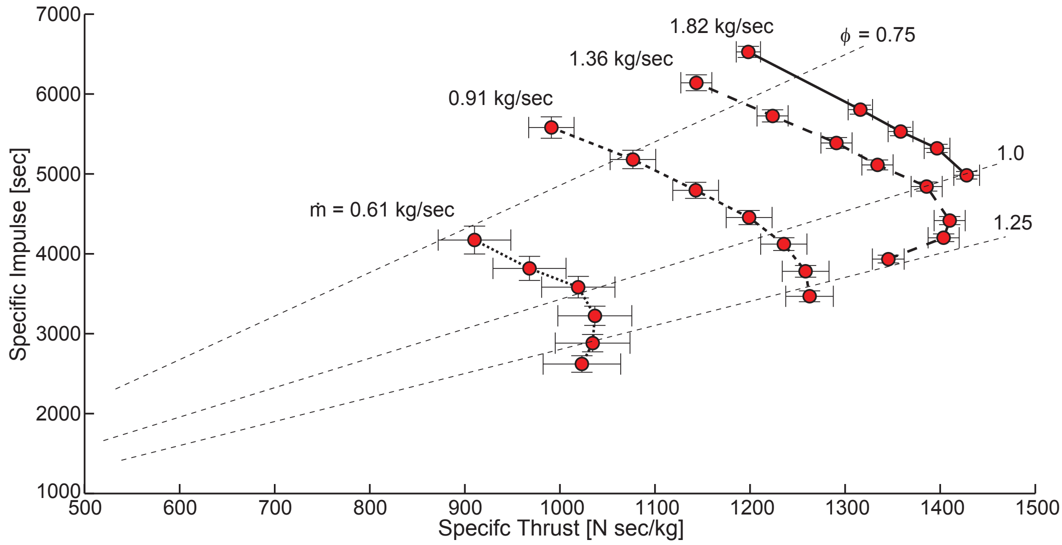

Figure 2: Performance of a six-inch rotating detonation engine coupled to a converging-diverging nozzle, A/A^* = 1.25, shown in terms of specific impulse as a function of specific thrust for various air mass flow rates and equivalence ratios.

As might be expected, the specific thrusts are seen to trend with the addition of stagnation temperature at each of the test conditions. The peak values occur at equivalence ratios just above stoichiometric, which correspond to the maximum adiabatic flame temperature. For a given mass flow rate, the rotating detonation engine can be seen to be capable of being throttled by approximately 30%, with no alteration to the geometry. The wave mechanics present in the detonation channel provide adequate adjustment of the injected reactants to allow this to be possible.

The observed performance is encouraging that a viable aerospace propulsion device can be based on this pressure gain technology. However, testing to this point has been conducted on a limited range of flight conditions and enthalpies, as shown in Figure 3 where the flight altitudes and Mach numbers that correspond to the test conditions shown earlier are given.

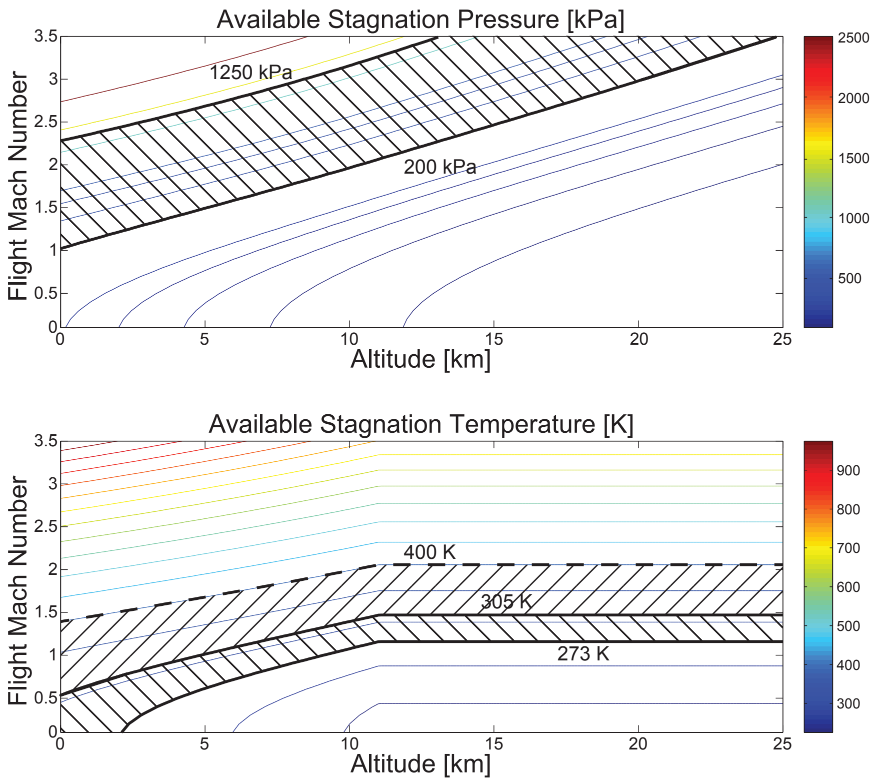

Figure 3: Stagnation pressure and temperature conditions as a function of flight Mach number and altitude, with hatched regions denoting current test conditions.

The currently available test data can be seen to represent two different Mach-altitude regions, depending on if stagnation pressure or stagnation temperature at the devices inlet are to be matched. If some liberty is allowed with regard to the state of stagnation temperature at the inlet, shown in the figure by the 400 K line, the current data is can begin to describe the engine operation around Mach 1 and at altitudes below 5 km. It is clear that further testing is required to better understand the flight envelope behavior if an aerospace propulsion system is to be developed, rather than a laboratory test apparatus. From an application point of view, it is promising that a novel pressure gain combustion device such as a rotating detonation engine is even capable of operating under conditions approaching those required for flight applications.

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