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Summary of Pressure Gain Combustion Research at NASA

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April 2018

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Summary of Pressure Gain Combustion Research at NASA

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Abstract

NASA has undertaken a systematic exploration of many different facets of pressure gain combustion over the last 25 years in an effort to exploit the inherent thermodynamic advantage of pressure gain combustion over the constant pressure combustion process used in most aerospace propulsion systems. Applications as varied as small-scale UAV's, rotorcraft, subsonic transports, hypersonics and launch vehicles have been considered. In addition to studying pressure gain combustor concepts such as wave rotors, pulse detonation engines, pulsejets, and rotating detonation engines, NASA has studied inlets, nozzles, ejectors and turbines which must also process unsteady flow in an integrated propulsion system. Other design considerations such as acoustic signature, combustor material life and heat transfer that are unique to pressure gain combustors have also been addressed in NASA research projects.

In addition to a wide range of experimental studies, a number of computer codes, from 0-D up through 3-D, have been developed or modified to specifically address the analysis of unsteady flow fields. Loss models have also been developed and incorporated into these codes that improve the accuracy of performance predictions and decrease computational time. These codes have been validated numerous times across a broad range of operating conditions, and it has been found that once validated for one particular pressure gain combustion configuration, these codes are readily adaptable to the others.

All in all, the documentation of this work has encompassed approximately 170 NASA technical reports, conference papers and journal articles to date. These publications are very briefly summarized herein, providing a single point of reference for all of NASA's pressure gain combustion research efforts. This documentation does not include the significant contributions made by NASA research staff to the programs of other agencies, universities, industrial partners and professional society committees through serving as technical advisors, technical reviewers and research consultants.

Introduction

Pressure gain combustion (PGC) is a generic term used to describe a family of physical processes and configurations which provide an increase in total pressure during the combustion process within a fixed volume combustor.[1] The most commonly studied types of pressure gain combustors are those associated with pulsejets, wave rotors, pulse detonation engines (PDEs) and rotating detonation engines (RDEs). Each of these combustors uses gasdynamic waves to confine the combustion process (all but the pulsejet also pre-compress the fuel/air mixture) to achieve an approximation to constant volume combustion, which results in a higher thermodynamic cycle efficiency than standard constant pressure combustion (Brayton cycle) found in typical gas turbines, ramjets and rockets (Ref. 1). The use of gasdynamic waves in the combustion process results in combustor outflows that vary either in space or time (or both), with the amount of variation proportional to the strength of the gasdynamic waves used. Stronger waves lead to higher unsteadiness, but also provide higher levels of pressure gain.

Research into the potential application of pressure gain combustion devices to aerospace propulsion systems has been undertaken sporadically since the 1940's by various universities and government laboratories, but a consistent research effort was not pursued until the late 1980's. The exception to this is

[1]This is in contrast to the pressure rise which occurs during the combustion process in the Otto cycle used in internal combustion engines, where combustion occurs in a variable volume cylinder.

the pulsejet, which was developed as a stand-alone propulsion system starting early in the 20th century, culminating in the German V-1 "Buzz-Bomb" of World War 2. However, the low performance of the pulsejet as a stand-alone system, despite its simplicity of design and low cost, resulted in its being set aside as a candidate for further development. In the early 1990's NASA Glenn Research Center (then Lewis Research Center) began investigating gasdynamic PGC devices for use in aircraft engines, starting with the Wave Rotor configuration. In subsequent years, investigations into PDEs (airbreathing, rocket, and combined cycle), pulsejets (for gas turbine hybrids) and RDEs have been pursued in projects led by NASA and also in partnership with other organizations. The purpose of this paper is to document the broad scope of these research activities over the past quarter century. These efforts have resulted in approximately 100 technical publications by NASA lead authors, in addition to approximately 25 publications with NASA contributing authors and 50 publications by authors whose work on the subject was financially supported by NASA through grants or contracts. The majority of these publications are referenced herein.

Before looking at the NASA contributions to PGC research, it should be noted that a number of survey papers covering specific PGC concepts and/or organizational interests are already available in the literature. Akbari and Nalim (Ref. 2) presented a review of wave rotor technology in 2009, while Kailasanath (Ref. 3) in 2003 and Dean (Ref. 4) in 2007 each provided a summary of PDE development at key points in time during the peak years of PDE research in the United States. Additionally, Tang, et al. (Ref. 5) provided a summary of PDE development in China as of 2008. More focused reviews are also available, such as Kailasanath's summary of PDE nozzle development as of 2001 (Ref. 6) and Williams' review of detonation chemistry as of 2002 (Ref. 7). An example of an organizational research review would be the review by Bussing, et al. (Ref. 8), of the Pratt & Whitney/Boeing Phantom Works team's collective PDE research activities. In the area of RDE research, an overall review was provided by Kailasanath (Ref. 9) in 2017, and an organizational review was provided by Heister, et al. (Ref. 10), for RDE work performed at Purdue University. Lastly, a series of summary reports have been generated by the JANNAF PGC Working Group (Refs. 11 to 15) on a periodic basis that provide a brief synopsis of progress in the research community as reported by the member organizations. Collectively, these survey reports provide a valuable history of PGC development.

Despite the plethora of PGC survey reports available, it was deemed worthwhile to provide a summary of NASA's PGC research efforts for several reasons. First, NASA is one of the only organizations to have worked extensively across all the different PGC concepts that have been studied. This breadth of experience allows NASA to have a unique perspective on the merits and challenges associated with each of the different PGC concepts. NASA is also not associated with any single PGC configuration and can therefore act as something of a neutral party in evaluating the various technologies. Second, NASA has worked on PGC concepts continuously throughout the modern era of PGC research, maintaining the same key personnel throughout the era. Lastly, NASA's research efforts have been somewhat under-represented in the previously mentioned survey reports. This is due to the fact that much of the research has been conducted at the fundamental level, while the focus of the community at the time tended to be on higher profile development activities culminating in large scale test programs. These factors of experience with a wide variety of concepts and applications and also continuous effort in fundamental research has put NASA in a unique position to benefit the PGC community on current and future research efforts.

Wave Rotor Research

Beginning in the early 1990's, research into the use of wave rotor based devices in gas turbines for performance enhancement was undertaken at NASA Glenn (then Lewis) Research Center. The wave rotor had initially been developed in the 1940's and 1950's (Ref. 2), but had been superseded by rapidly improving aerodynamic compressors for gas turbines. However, by the 1990's the rate of improvement in gas turbine overall pressure ratio (OPR) had diminished significantly due to limiting factors in materials and blade height. It therefore seemed plausible to return to the wave rotor as a means of improving gas

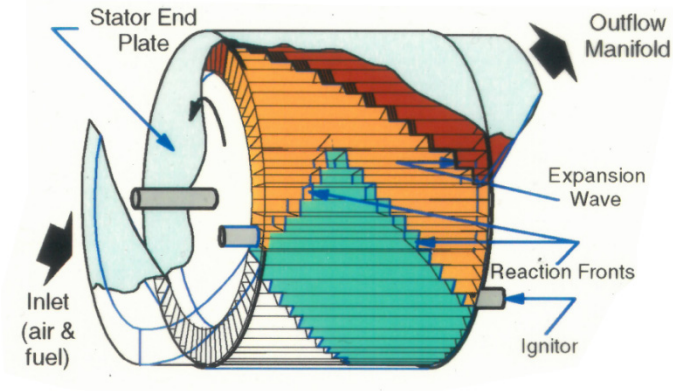

Figure 1.—Wave rotor combustor.

turbine performance beyond the capabilities of the established configuration. Furthermore, it was proposed that the wave rotor could be used not only to increase OPR, but it could also be used to replace the combustor, further increasing the thermal efficiency of the cycle since each channel could function as a constant volume combustor. Figure 1 shows the basic configuration of a wave rotor used as a combustor. A complete description of the device and its operation can be found in Reference 2.

The wave rotor research program consisted of complementary modeling and experimental studies, augmented by preliminary engine cycle and integration studies to determine the practicality and benefits of the wave rotor topped gas turbine. Early on in the program, a 1-D numerical tool was developed to analyze the wave cycle of the device (Ref. 16). While this tool was found to qualitatively follow the wave cycle as given in previously published experiments, there were quantitative differences with the experiments. Loss models for finite valve opening time, viscous effects (friction, heat transfer), and leakage were added as the code was further developed, and acceptable quantitative agreement with published experiments was achieved (Ref. 17). An in-house experiment using a 3-port divider cycle wave rotor (Ref. 18) was also used to validate these loss models by systematically varying the geometry of the wave rotor to provide multiple validation points for each loss model (Ref. 19). It should be mentioned that matured versions of the 1-D code (Refs. 20 to 22) developed for the wave rotor, with the addition of area variation (quasi-1D), have been used with great efficacy for the analysis of PDE's and pulsejets and their derivatives – any device that can be construed to be primarily quasi-1D. The loss models incorporated into the code have also been found to be valid across the whole family of gasdynamic devices discussed within this paper. The initial wave cycle design and analysis performed with the quasi-1D code was supplemented with more detailed Computational Fluid Dynamic (CFD) analyses to study specific flow issues within the wave rotor and its inlet and exhaust ducts, such as the effects of gradual tube opening and closing and rotation (Refs. 23 to 27). The in-house wave rotor experiment was again used to validate these simulations (Ref. 28).

Following these initial fundamental studies on wave rotor design and simulation, work was undertaken at NASA GRC to look at applications for the wave rotor as a topping cycle in either a turboshaft or a turbofan. Significant decreases in specific fuel consumption and increases in specific power were calculated in these studies (Refs. 29 and 30), with the greatest benefit for lower overall pressure ratio applications, such as small turboshaft engines. However, the smaller benefits calculated for large scale turbofans were still quite significant, particularly considering the longer missions flown by the aircraft for which these engines would be intended. Bolstered by these benefits assessments, a new 4-Port wave rotor experiment was designed and implemented that represented a potential configuration for use in a gas turbine topping cycle application (Ref. 31) and which would be consistent with the on-rotor constant volume combustion configuration. This new test rig was designed using the numerical tools validated using the original experiment, and improvements to the design, such as passage height variation (Ref. 32), were evaluated. Alternate methods for wave rotor design and analysis were also pursued during this time period (Refs. 33 and 34).

While the wave rotor itself was shown to provide significant performance improvement for the gas turbine cycle, consideration was given to other possible wave rotor-based configurations in the late 1990's. One option considered was the addition of curvature to the wave rotor channel to allow for power extraction directly from the wave rotor (Ref. 35), such that the wave rotor's rotation could be "self-powered" instead of being driven by an external turbine, or to meet other power extraction requirements. Another configuration studied involved bypassing some of the air that would typically have gone to the combustor through a bypass duct, which was then mixed back in with the effluent from the combustor, effectively lowering the wave rotor exhaust gas temperature while achieving approximately the same pressure gain as the conventional cycle (Ref. 36).

As in-house wave rotor research wound down in the mid-2000's, several improvements were made to the 4-port wave rotor experiment designed to allow the rig to demonstrate the full performance shown in previous analysis. These improvements included transition duct geometry variations to reduce total pressure losses in the flow exiting the wave rotor (Refs. 37 and 38), as well as improved end wall sealing methods (Refs. 39 and 40). A parallel effort to reduce end-wall leakage for wave rotors, with a particular emphasis on on-rotor combustion, was pursued under NASA contract with Allison Advanced Development Corporation (Ref. 41) in the same time frame. While the NASA wave rotor program had demonstrated significant potential for improving gas turbine performance, concerns over leakage, structural durability and packaging within a flight-rated gas turbine led to a withdrawal of funding support in favor of other pressure gain combustion options. The problems perceived in wave rotor design and application were not fundamental ones, but represented significant development challenges that the agency was not at that time prepared to undertake.

While the prospects for the application of wave rotors to commercial aircraft dimmed somewhat in the late 2000's, it only took a few years for a new application to develop. The emergence of small-scale drone aircraft has provided a new field for wave rotor application. Because traditional aerodynamic compressors perform poorly at small scale, the Air Force Research Laboratory (AFRL) began to consider wave rotors as the compression system of choice for hydrocarbon fueled micro-aircraft. Given NASA GRC's experience with wave rotors, they sought out NASA expertise and simulation capabilities to assist in this research area. Under AFRL funding, NASA has contributed significantly to developing a small scale wave rotor that has been tested with internal combustion engines (Refs. 42 to 44), Brayton cycle configurations (Refs. 45 to 47), and even diesel engines (Ref. 48). The principal simulation tool used for all these configurations has been the quasi-1D code developed in the early 1990's under the initial NASA GRC wave rotor program.

Pulse Detonation Research

Along with many others, NASA researchers turned their attention to pulse detonation based concepts in the mid-1990's. The basic configuration of a PDE is shown in Figure 2. A synopsis of the history of pulse detonation research prior to this time can be found in the review by Kailasanath (Ref. 3) and its references, along with details of research conducted by a number of government, university and private industry organizations up to the time of that publication. Further updates can be found in later summaries, such as Reference 4.

The scope of PDE research performed by NASA and the breadth of personnel involved, as well as the multiple applications considered, requires that this section be organized somewhat differently than the previous wave rotor section. Ten sub-sections representing different categories of research, such as acoustics, ejectors, and materials and structures, etc., will be presented. The order in which these sections are presented has no relation to their relative importance or any other particular scheme. Programmatically, the majority of the work described was conducted under the Pulse Detonation Engine Technology (PDET) project at NASA GRC and the Constant Volume Combustion Cycle Engine (CVCCE) project at NASA GRC. However, other programs also contributed support, such as the Small Business Innovative Research (SBIR) program, and various fundamental research support programs.

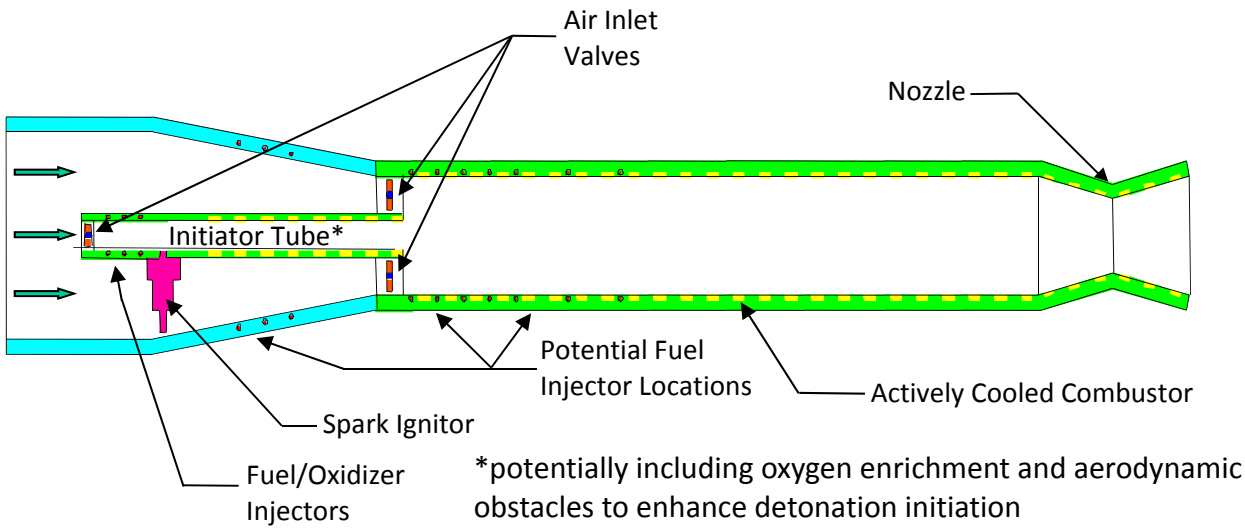

Figure 2.—Basic features of a pulsed detonation engine.

Computational Methods for Detonation

Many of the combustion and gasdynamic codes available to NASA in the 1990's had not been previously used for detonation problems. For example, the previously developed in-house Conservation Element/Solution Element (CE/SE) code was modified to include finite-rate chemistry and demonstrated in both 2-D and 3-D to accurately calculate detonation structure and propagation (Refs. 49 to 52). The NASA LaRC SPARK CFD code initially developed for hypersonics was modified with an updated kinetics solver and demonstrated for detonation problems (Ref. 53). The National Combustion Code (NCC), developed for gas turbine combustor analysis, was shown to adequately model detonative combustion, and was used as part of a study to develop a reduced mechanism for Jet-A/air for detonative combustion (Refs. 54 and 55). Lastly, a 1-D/2-D implicit totally variation diminishing (TVD) based in-house code, 2nd order accurate in space and time, developed for hypersonic ram-accelerator and other hypersonics studies, was extensively tested on detonation and related problems (Refs. 56 to 59). This code became the workhorse multidimensional CFD code used at NASA GRC for studying PDE performance, nozzles, ejectors and combined cycle configurations. NOx kinetic mechanisms for both hydrogen and hydrocarbon fuels were added to the finite-rate kinetics in the TVD code, validated against experimental data, and then used for a number of NOx minimization studies in PDE's (Refs. 60 to 62).

Cycle Performance Calculations

The uniqueness of the PDE cycle and the lack of familiarity among analysts with unsteady flow systems led to significant variation in published performance values in the 1990's and on into the early 2000's. The validity of common cycle analysis simplifications such as ideal gas or calorically perfect gas, isentropic inlet and exhaust flow, etc., was unknown. A number of performance calculation studies were therefore undertaken at NASA. Starting with a basic analytical cycle analysis such as is performed for Otto and Diesel cycle engines (Ref. 1), CFD was used to look at potential corrections, such as real-gas effects due to the very high peak temperatures found in and immediately after the detonation wave, and the decrease in recombination rates during both gasdynamic expansion within the tube and the physical expansion of the flow to atmosphere (Refs. 63 to 69). Another area of difference was found to be the treatment of the subsonic flow exit boundary condition during the blowdown phase of the detonation tube cycle (Refs. 70 and 71), with the greatest impact found for shorter detonation tubes which were shown to be still impacted by the pressure field of the blast wave despite its having propagated a significant

distance from the detonation tube exit. Yet one more area of study was the effect of heat transfer on PDE performance. Many performance calculations assume adiabatic walls, but a significant performance penalty, up to 50 percent of the expected benefit, could be incurred in a PDE with cooled walls (Ref. 72). Lastly, the effects of fuel/air mixing were studied (Ref. 73), and it was determined that as long as a detonation was maintained, the effects of fuel/air equivalence ratio gradients was minimal.

For many applications, a PDE could serve as the combustor in a larger propulsion system, such as a gas turbine. For such an application, it is common to represent a complex component, such as a compressor or turbine, with a performance map that provides the output and efficiency of the component as a function of operating condition for use in an overall propulsion system cycle deck. Toward that end, PDE performance maps were generated using quasi-1D CFD (Refs. 74 to 76), again showing the utility of the reduced order approached developed under the wave rotor program. Another approach for cycle analysis is to use an analytical model of the different flow processes within a PDE to come up with the overall time averaged performance (Ref. 77). This approach was used to perform PDE performance analysis using the NPSS cycle code that has been developed for propulsion system cycle analysis by NASA and a number of industry partners. Given the different approaches available for calculating PDE performance, either through cycle decks or directly from CFD, it became necessary to reconcile the somewhat different results that were obtained with the experimental data available, and with each other. Toward that end, NASA undertook a concerted effort at such a reconciliation (Refs. 78 and 79), which was then followed by several efforts from JANNAF (Refs. 80 to 82). It should be noted that a 0-D performance code that was developed as part of the efforts documented in Reference 82 was later released as NASA code LEW-17670-1 and has been used for PDE benefits analyses by other government agencies.

Combustor Characterization

In the course of its PDE research, NASA dedicated limited time and resources to actual detonative combustion experimentation, as it was felt that other organizations were already covering that area of endeavor adequately. Therefore, the bulk of NASA resources were spent looking into areas associated with PDE system integration—inlets, nozzles, ejectors, materials, et al. However, there were a few particular experiments that were of special interest to NASA. An early demonstration of basic operation of a PDE was performed under a Small Business Innovative Research (SBIR) contract (Ref. 83). In this program a multicycle PDE was built using multiple spark ignition points to generate a detonation wave. This engine was successfully run at frequencies up to 100 Hz for short durations.

The second test program was a benchmark PDE code validation test. While many experiments had been run on PDE combustors prior to this test, almost all these experiments were sea-level-static tests. Therefore, it was important to get PDE performance data with elevated inlet temperatures to represent supersonic flight and/or post-compressor conditions in a hybrid configuration. This test was performed at an AFRL test facility using Boeing PDE hardware under NASA contract (Ref. 84). This test data was subsequently used for both performance code and CFD validation.

A third experiment of particular interest to NASA was to determine that possibility of sensitizing hydrocarbon fuels for improved detonability through the use of nano-particle aluminum mixed with the fuel. It was found that while JP-8 alone required 30 percent O_2 enriched air to detonate in the NASA PDE testbed, the same fuel mixed with nano-particle aluminum could be detonated in regular air (Ref. 85). Lastly, in an update to the previously referenced NASA Q1D gas-dynamic code, parametric data from an AFRL test rig using various deflagration-to-detonation (DDT) obstacles was used to develop an empirical loss model for DDT obstacles, taking into account both total pressure loss and heat loss from the obstacles (Ref. 86). It was noted that the heat transfer effect was significantly larger than the aerodynamic total pressure loss for the cases studied.

Acoustics

As NASA has an interest in utilizing detonation for civilian propulsion application, reduced acoustic signature was of concern. It was therefore of interest to get a set of benchmark data in an acoustic facility of a PDE to help quantify the problem, and potentially to aid in determining appropriate attenuation strategies. A comprehensive dataset of both near field and far field measurements was therefore taken of the same Boeing PDE rig that was referenced in the previous section as having been run at elevated inlet temperatures at AFRL. This time the rig was run in the Aeroacoustic Propulsion Laboratory at NASA GRC (Ref. 87). A number of different operating points were run with different nozzle configurations. This dataset was later supplemented by additional test data taken at the University of Cincinnati (Ref. 88). All of the above data was taken with a PDE combustor by itself or with a number of different nozzles that were used to demonstrate the effect of tube exit geometry on noise signature, but NASA's primary interest was by this time a gas turbine hybrid configuration. Therefore, a CFD study of the amount of noise attenuation to be expected from a single high pressure turbine stage was undertaken (Ref. 89). This study showed that significant broad band attenuation could be expected, and therefore the use of a pulse detonation combustor in a gas turbine hybrid would have little effect on the acoustic signature of the overall propulsion system. This result was later corroborated by experimental measurements (Ref. 131).

Inlets

Intrinsic to a basic PDE configuration is a time-varying inlet flow caused by the opening and closing of the combustor tube valves (see Fig. 2). At the combustor assembly face, the flow can be expected to vary both in space and time as the valves are opened and closed in sequence. This flow characteristic would be expected to lower the overall effective inlet total pressure recovery, and perhaps impact basic operability for a supersonic inlet. This situation was recognized by the hypersonic systems group at NASA LaRC who were considering the use of a PDE as the low speed system for a hypersonic vehicle. A CFD study was therefore initiated, and it was concluded that an inlet system could be designed that was operable and adequately efficient for that application (Ref. 90).

A more generalized experimental and computational research effort was undertaken by NASA GRC. An existing small scale mixed compression inlet test rig was modified to allow periodic disturbances at the exit of the diffuser consistent with those expected from PDE combustor air valves. Some differences were found between the experiment and the high-fidelity CFD used to model the experiment which demonstrated that additional development of the CFD techniques available at that time needed to be pursued in order to accurately model these flows (Ref. 91). This inlet model was later used for a series of studies that examined ways to offset the impacts of unsteady flow on operability and performance of a mixed-compression inlet over a wide range of conditions (Ref. 92). Additional supporting mixed-compression inlet experiments were run at the University of Florida using a different flow disturbance method that utilized a number of flow visualization techniques to examine the effect of oscillatory flow on the inlet flow structure (Ref. 93).

Nozzles

One key area of research for the basic cycle PDE was the nozzle. Given the wave reflections that would be generated by a nozzle, it was unclear how a nozzle would affect the operability of the combustor. However, without a nozzle it would be impossible to operate the PDE at altitude, and the efficiency would remain low. Therefore, NASA partnered again with Boeing to provide an initial test of a PDE at altitude with a nozzle (Ref. 94). These tests demonstrated both operability and performance at simulated flight conditions, and also that the performance could be predicted from the analytical tools available at that time. To aid in the understanding of how to design a PDE nozzle, a number of contract and grant efforts were supported by NASA. A team at Wayne State University performed a computational study of the exit flow of a PDE to help understand the structure of the exit flow (Refs. 95 and 96). A team

at the Applied Physics Laboratory undertook an effort to develop a design methodology for unsteady flow nozzles using multiple combustors (Ref. 97). This effort used a combination of 1-D, 2-D and 3-D CFD simulations to develop a nozzle design. Lastly, the University of Cincinnati performed basic exhaust flow studies (Ref. 98) and then nozzle geometry studies (Ref. 99) using a combination of experimental and computational methods. In-house, NASA GRC performed computational nozzle design studies using its quasi-1D code as part of an optimization routine (Ref. 100), as well as multidimensional CFD (Ref. 101).

Ejectors

In order to detonate hydrocarbon fuels, an equivalence ratio near 1.0 is required for standard air. It is therefore necessary to mix bypass/cooling air into the exhaust from a PDE in order to feed a turbine to avoid exceeding the turbine's temperature limit. One of the most efficient ways to mix in this extra air is with an ejector. An ejector also can be used to "smooth out" the flow from the highly impulsive PDE exhaust to provide a more acceptable flow for the turbine. Led by this reasoning, NASA GRC undertook an extensive study of unsteady ejectors, both in-house and through a series of university grants. In addition to studying actual PDE-driven ejectors, a number of unsteady ejector drivers were used in order to gain a better understanding of what pulse characteristics provide the most efficient ejector operation, as well as to be able to expedite the use of a variety of research test facilities, some of which would not have been compatible with a detonative combustion driver.

Initial CFD calculations of a pulse detonation combustor driving a converging-diverging ejector showed significant potential for thrust augmentation of a PDE by the ejector, with a dependence on the position of the ejector throat relative to the exit of the PDE (Ref. 102). Initial unsteady ejector experiments were performed with surrogate drivers, such as a resonance (Hartmann-Sprenger) tube, which also showed significant thrust augmentation. In this experiment, the ejector length, diameter and flow inlet geometry were varied and it was found that the maximum thrust augmentation occurred when the diameter of the ejector matched the diameter of the starting vortex ring generated by the ejector driver (Refs. 103 and 104). This finding led to efforts to better characterize and control the vortex structure generated by the driver (Ref. 105) in order to better optimize the performance of the ejector. These findings were later corroborated using a speaker-driven jet as the surrogate unsteady ejector driver (Ref. 106).

The above ejector experiments were done with subsonic driver flow. In order to study the effect on ejector performance of a supersonic driver flow, an experiment using a specially designed 4-port rotary valve feeding a Mach 3.7 supersonic nozzle was constructed. This experiment showed similar trends as the subsonic driver experiments, although the driver pulse had a very different waveform (Ref. 107). To be able to analyze the data collected from this experiment, a data reduction technique for the specific situation involved had to be developed (Ref. 108). The experiments done with detonation tube surrogates were then repeated and expanded upon using a PDE itself (Refs. 109 and 110). The ejector optimization methods developed in the earlier experiments were found to be effective with the PDE, and a maximum thrust augmentation in excess of 2.0 was eventually achieved.

To supplement the in-house research on unsteady ejectors, grants supporting both computational and experimental studies were funded. A CFD study was undertaken by the University of Tennessee Space Institute to examine the physics of thrust augmentation for a PDE-driven ejector (Ref. 111). The positive impact of the blast wave in transferring energy to the secondary flow is particularly noted in this work. A CFD study performed at Purdue University helped to elucidate aspects of the resonance tube experiment performed earlier at NASA GRC (Ref. 112). A series of experiments on PDE-driven ejectors were conducted at the Pennsylvania State University. PDE thrust augmentations of over 2.0 were also measured in this experiment (Refs. 113 to 115), consistent with the NASA in-house results. Lastly, additional PDE ejector data was collected by the University of Cincinnati using a 2-D parametric ejector (Refs. 116 and 117). The results from this rig were consistent with other reported data, despite the difference in the ejector cross-section shape. The University of Cincinnati also performed a parametric study of the effect of an ejector on the acoustic signature of a PDE (Ref. 118), which found that the ejector significantly reduced the sound pressure level measured in the far field.

Materials and Heat Transfer

Some effort was made by NASA GRC researchers to consider how a gas turbine integrated pulse detonation combustor would be built. Toward that end, a typical combustor material, Haynes 188, was tested under pulsed thermal loading similar to what would be experienced in a PDE (Ref. 119). Mechanical loading (4-point bend) was also applied to the test specimens after completing the thermal cyclic load testing (Ref. 120). It was found that surface cracking could be a significant issue for the material studied. In an effort to mitigate potential combustor material damage, a study was also conducted on potential thermal barrier coatings for the combustor (Ref. 121). These materials were found to perform their function well, but their performance did degrade somewhat with time.

In addition to materials studies, air-cooling was considered for the combustor. Both backside cooling with an ejector configuration (Ref. 122) and film cooling (Refs. 123 and 124) were considered. The film cooling design was also evaluated experimentally and found to be effective despite the pulsed flow present in the detonation tube. Lastly, a set of experiments were run at AFRL that demonstrated the effectiveness of the heat transfer model previously incorporated in NASA's quasi-1D performance code (Ref. 125). This model validation was important in allowing the designers to determine approximate heat loads as a function of operating condition for the combustor.

Combustor/Turbine Interactions

For a gas turbine/pulse detonation hybrid configuration, the ability to efficiently expand the unsteady combustor effluent through a turbine is a critical element of a successful system design. Traditional turbine designs allow for low levels of variation in inflow properties, so there was no available information on how a turbine would perform with a pulsed combustor. There was also a concern that the turbine wouldn't be able to structurally survive. While some exploratory modeling of this problem was undertaken (Ref. 126), it was not considered viable to use CFD as the primary tool for this study due to the complexity of the geometry along with the unsteady flow field (including rotation). Another issue with using CFD was that there was no benchmark data set against which a model could be validated. To address both life and performance issues, a contract was entered into with GE Global Research Center to perform a series of experiments that would provide a basis for determining the viability of unsteady inflow turbines.

The system assembled by GE as a PDE-turbine testbed consisted of eight 2 in. detonation tubes firing into a single stage axial flow turbine. The PDE exhaust was mixed with bypass/cooling air prior to entering the turbine. A series of test and analysis reports were produced using this rig that examined performance, durability, structural response and flow structure (Refs. 127 to 130). The nonoptimized performance of the PDE-turbine system was not found to differ substantially from that of the steady flow baseline. This program was summarized in a final contractor report (Ref. 131) and a detailed journal article (Ref. 132).

In addition to the GE study, a parallel study was performed at the University of Cincinnati, where a 6-tube array of pulse detonation combustors was integrated with an axial flow turbine (Refs. 133 and 134). This study also found comparable performance between the PDE-driven turbine and the same turbine driven by steady flow. Lastly, a small effort was undertaken at the University of Toledo to develop an analysis methodology to determine the aero-elastic response of a turbine rotor to pulsed flow (Refs. 135 and 136). While progress was made in the development of the methodology, its application to a PDE-hybrid was not fully realized.

System Design Studies

System integration and system level benefit studies are an important part of any propulsion system develop activity. These studies help to determine the most likely flight application for a propulsion system concept which in turn serves to provide sub-system performance goals that feed down to

component performance goals that drive research tasks. System studies also provide the basis for program level funding decisions. Starting in 1999, NASA sponsored The Boeing Phantom Works to perform a comprehensive study of potential applications for PDE technology (Ref. 137). A broad range of potential flight applications were considered, from airbreathing launch vehicles to unmanned aerial vehicles (UAV's) to tactical aircraft to commercial transports. The propulsion cycle options that were considered were basic cycle PDE, rocket PDE (PDRE), PDE/ramjet combined cycle, PDE/PDRE dual mode operation and gas turbine hybrid. After an initial screening phase of the study, four vehicles were selected for detailed study: a tactical aircraft, a strike missile, a supersonic UAV and an airbreathing single-stage-to-orbit (SSTO) vehicle for which the PDE serves as the low speed accelerator propulsion system. Potential benefits were found for each of the applications in the detailed study. Technology development roadmaps for the four propulsion cycles from the detailed studies were provided.

As later NASA activities focused primarily on the gas turbine hybrid configuration, an engine conceptual design study was funded with Pratt & Whitney Aircraft to look at a full inlet-exhaust configuration of a PDE-Hybrid (Ref. 138). This concept engine was for a subsonic commercial transport application, baselined against the P&W 2040 production engine at a nominal maximum sea-level-static (SLS) thrust of 40,000 lbf. A full engine concept utilizing a rotating PDE "tube pack" was created with a specific fuel consumption improvement of 12 percent predicted over the baseline. This design study was followed by an experimental program aimed at demonstrating the performance of the PDE core of the engine. Total pressure losses of several components within the test hardware were higher than predicted, leading to lower overall performance. Further development was expected to reduce these losses. A second PDE-Hybrid design study was funded with General Electric (Ref. 129). The GE study was baselined against its CF34-10 engine, which generates 18000 lbf of thrust at SLS conditions. The GE concept engine was not estimated to significantly improve SFC over the baseline due to a number of performance penalties assessed against individual engine components. However, since a conservative approach was taken in assessing the losses due to unsteady flow during the design process, it was viewed as likely that the performance of this concept engine would improve with continued research and development.

Pulse Detonation Rocket Research

In the late 1990's, NASA Marshall Space Flight Center (MSFC) undertook a comprehensive research program aimed at determining the applicability of a pulse detonation rocket engine (PDRE) for an upper stage application. The Revolutionary Rocket Propulsion System (RRPS) program included computational studies of basic PDRE configurations aimed not only at determining performance in and of itself, but also determining the relative applicability of 1-D, quasi-1D and 2D simulation tools for determining that performance (Refs. 139 to 143). Additional modeling studies looked at more complex geometries, such as two tubes firing into a common nozzle (Ref. 144), and at wall heat transfer effects (Ref. 145).

A complementary interest in PDRE's beyond improved performance was the possibility of directly extracting power from the flow via magnetohydrodynamics (MHD). The high peak temperature of an H_2-O_2 detonation was known to significantly ionize the flow, giving rise to the possibility of utilizing MHD. An initial computational study also included cesium injection to increase the ionization within the detonation tube, and found that sufficient energy could be extracted to operate the detonation tube's detonation initiation system (Ref. 146). A later study also looked at using the power extracted through MHD to power a boost pump to increase the fill rate of the detonation tubes, increasing the overall thrust density of the system (Ref. 147). In parallel with the MHD simulations, a series of experiments that demonstrated both PDRE performance and MHD power extraction were performed (Refs. 148 to 150). Predicted values for I_{sp} and MHD power extraction were in line with predictions. A second generation PDRE was constructed in order to work at a larger scale and to be more flexible in its configuration (Ref. 151). Performance of the larger PDRE was as anticipated, but MHD power extraction was below expectations due to physical limitations on the magnets used (Ref. 152).

Building on previous airbreathing PDE development efforts, NASA MSFC entered into two contracts to develop larger scale PDRE's with traceability to a future upper stage engine. Adroit Systems, Inc., was able to demonstrate low pressure operation of a multitube, rotary valve, common nozzle concept engine at sea-level-static conditions. Performance of this engine met design expectations based on CFD predictions (Ref. 153). Pratt & Whitney demonstrated baseline operability of a large, single tube engine concept with a large bell nozzle under altitude conditions. This engine had a rotating valve located midway down the tube, and used additional propellant injection at the end of the tube in an effort to use the momentum of the injected propellants to elevate pre-detonation fill pressure (Ref. 154). Issues with the rotating valve limited the operability of this concept and reduced performance.

Pulse Detonation Rocket-Based Combined-Cycle Research

The first proposed use of a PDE as part of an airbreathing launch vehicle propulsion system was by Cambier, Adelman and Menees (a NASA Ames research scientist) (Ref. 155). This system involved the mounting of a series of PDE tubes around a central ramjet/scramjet (dual-mode) flowpath, with the flow direction of the PDE tubes the same as the flow direction of the ramjet. In this configuration, the PDE would serve a number of purposes, and would be operational throughout the SSTO trajectory. At very low speeds, the PDE's, in airbreathing mode, would provide primary propulsion (they could be used in rocket mode initially for additional thrust if needed). As sufficient ram inlet pressure was established the ramjet flowpath would be fueled, and a small amount of the pulsed flow from the PDE's would flow through sidewall slots into the ramjet flowpath to greatly enhance mixing, allowing for a much shorter flowpath than is typical for a dual-mode ramjet. In ramjet mode, it was also expected that ejector action from the PDE's would improve the propulsive efficiency of the combined system. As with the ramjet mode, the PDE's would be used to provide mixing enhancement in scramjet mode as Mach number increased. When airbreathing propulsion ceased to be viable above Mach 10, the PDE would be operated in rocket mode (PDRE) for final ascent into orbit. Viability of this concept was supported by detailed CFD simulations.

The second proposed use of a PDE for an airbreathing launch vehicle was as a PDE/ramjet combined cycle where the PDE tubes doubled as the combustors for a ramjet (Ref. 156). At the Mach number at which ramjet propulsion provided higher performance than the PDE cycle (around Mach 2.5 - 3.0), the PDE inlet valves would be left open with the tubes serving as steady flow combustors. A full vehicle performance assessment was performed using the PDE/ramjet engine as the low speed propulsion system in place of a turboramjet.

Lastly, an ejector ramjet configuration consisting of a central PDRE surrounded by a ramjet/scramjet duct (Refs. 157 and 158), Figure 3, was studied. In this setting, the PDRE would provide primary propulsion at low speeds, followed by both the PDRE and ramjet operating in tandem up to full ramjet take-over (at around Mach 3.0). The PDRE would then be turned back on above Mach 8.0 for final ascent into orbit. It was found during simulations of this concept that the ejector mode could continue to be used above Mach 8 since the PDRE shock wave would still transfer energy to the secondary stream and no viscous mixing was required, as in traditional ejector-ramjet cycles. In another permutation on the

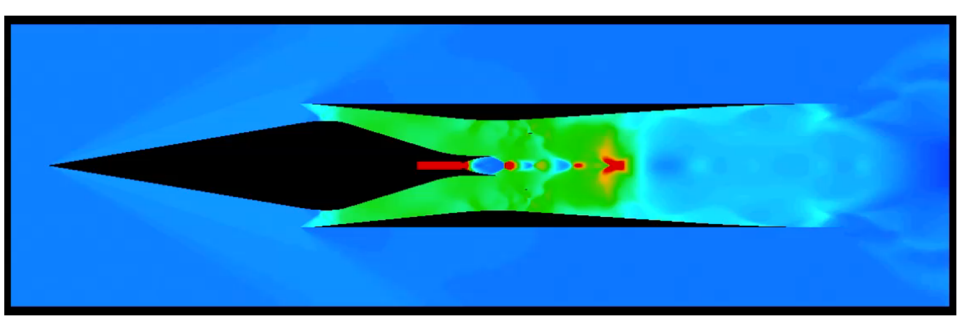

Figure 3.—CFD temperature contours of PDRE ejector-ramjet cycle engine.

operation of this cycle, the fuel could be pulsed in the ramjet duct during low speed operation and a pulse detonation operational mode established in the ramjet duct, initiated by the PDRE. The performance of this PDRE/PDE combined cycle was significantly above that of the previously simulated PDRE/ramjet mode (Refs. 159 and 160). The basic PDRE-ejector configuration was experimentally studied under a grant to the Pennsylvania State University (Ref. 161). This experiment provided primarily code validation data, as it was not capable of operating at the same conditions as the proposed propulsion system.

Pulsejet/Pulsed Resonant Combustor Research

In parallel with looking at detonative systems for pressure gain combustion, NASA, uniquely in the recent PGC community, considered pulsejet derived combustors as well. The pulsejet is a well-understood and proven propulsion system in this class, but with much lower performance than an ideal PDE. However, the maturity of the concept, the lower thermal and mechanical stresses, and the much more benign exhaust flow makes the concept attractive. In order to smooth out the exhaust flow of the pulsejet and to bring the exhaust temperature to an acceptable level, bypass air must be mixed with the pulsejet exhaust, as with the PDE. A series of in-house experiments were run using a model-scale pulsejet driving an ejector in order to determine an optimum ejector geometry and placement relative to the pulsejet exit (Ref. 162). In a follow on experiment, the pulsejet and ejector were enclosed in a duct and a turbocharger was run off a portion of the pulsejet/ejector exhaust flow. The turbocharger supplied pressurized air which was run through a heated pipe to bring the air inlet temperature up to typical compressor exit temperatures. Using this heated, high pressure air, the pulsejet/ejector showed a pressure gain of 3.7 percent with an exhaust to inlet temperature ratio of 2.3, which is typical for many gas turbine applications (Refs. 163 and 164). This closed loop test represented the only demonstration to date of actual pressure gain in a PGC system in the modern era of PGC research and did so at operating conditions consistent with gas turbine operation. This work was supplemented by collaborations with AFRL (Refs. 165 and 166) and North Carolina State University (Ref. 167) to look at scaling effects, operational issues and flow characteristics related to pulsejets.

While the experiment cited above in References 163 and 164 ran at representative compressor exit temperature, the low pressure ratio of the turbocharger available for the experiment resulted in a pulsejet/ejector inlet pressure that was significantly below compressor exit conditions, though still elevated above atmospheric pressure. As there was limited ability to test the model scale hardware at the required inlet pressure and temperature simultaneously, this work was performed using high-fidelity simulations. First, studies on appropriate fuel injector placement and timing were performed at high pressure, high temperature conditions (Refs. 168 and 169). Next, the pulsejet had to be placed inside a shroud/ejector as it would be in a gas turbine configuration (Refs. 170 and 171), as shown in Figure 4. This step also involved optimization of the ejector/shroud geometry, including the addition of an ejector exhaust nozzle. When this geometry was optimized, it was considered too long for practical integration into a gas turbine. Therefore, a study aimed at shortening the pulsejet, the shroud and the ejector was undertaken (Ref. 172). Initial studies showed that the configuration could be shortened by at least 30 percent, and it was unexpectedly found that this new configuration provided an approximately 30 percent increase in pressure gain over the previously optimized full-length system. This work is still ongoing.

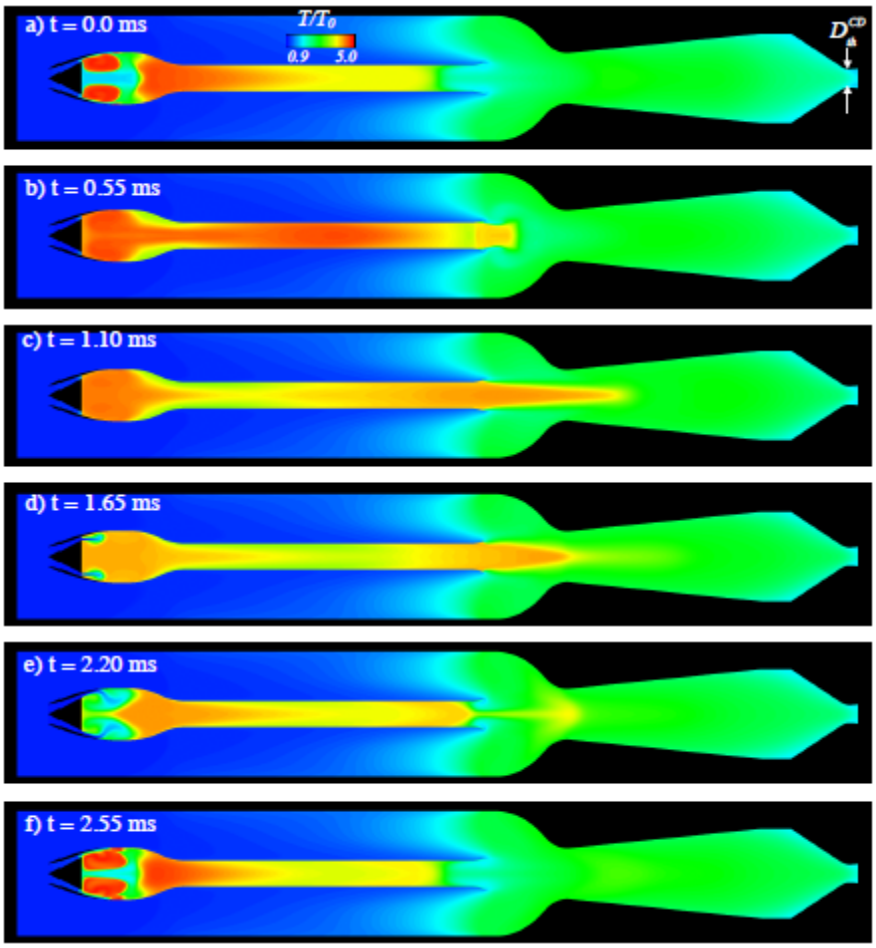

Figure 4.—Time series of temperature contours of pulsed resonant combustor.

Rotating Detonation Engine (Airbreathing and Rocket)

The NASA Quasi-1D CFD code referenced numerous times in the previous sections could not simulate the flow field in a rotating detonation engine (RDE), which has in inherent dimensionality of at least 2-D. Figure 5 shows the basic configuration of an RDE, in this case a rocket, with its annular combustor. While NASA has not as yet had an experimental program for RDE's, a 2-D version of the previous Quasi-1D code, with the same level of fidelity and incorporating many of the same sub-models, has been developed. In cooperation with AFRL, this code has been validated against experimental data (Refs. 173 and 174), and been applied to studying RDE heat transfer (Ref. 175) and nozzle configurations (Ref. 176). Additional validation was obtained by comparing the results of the lower fidelity code with high fidelity simulation results using detailed chemical kinetics from the in-house code described in Reference 56. A typical high fidelity result is shown in Figure 6. The limited ability to make meaningful measurements in an RDE has made this 2-D code very valuable in helping to interpret experimental results from the limited data sets that are typically available. Examples of the use of the lower fidelity code in this manner was the analysis of a coupled RDE/ejector/turbine experiment performed at AFRL (Ref. 177) and an analysis of the lower than ideal wave speeds commonly observed in RDE's (Ref. 178).

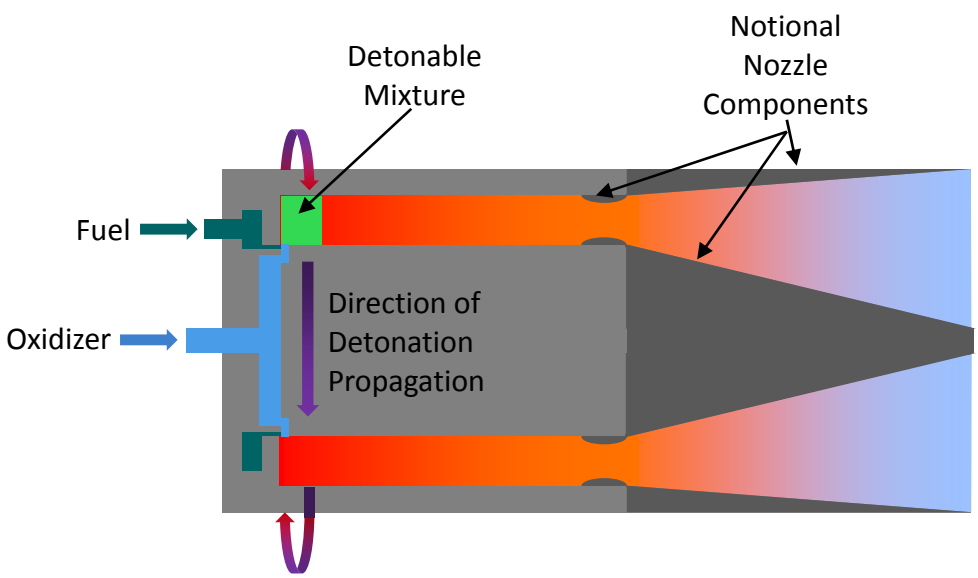

Figure 5.—Notional rotating detonation rocket engine.

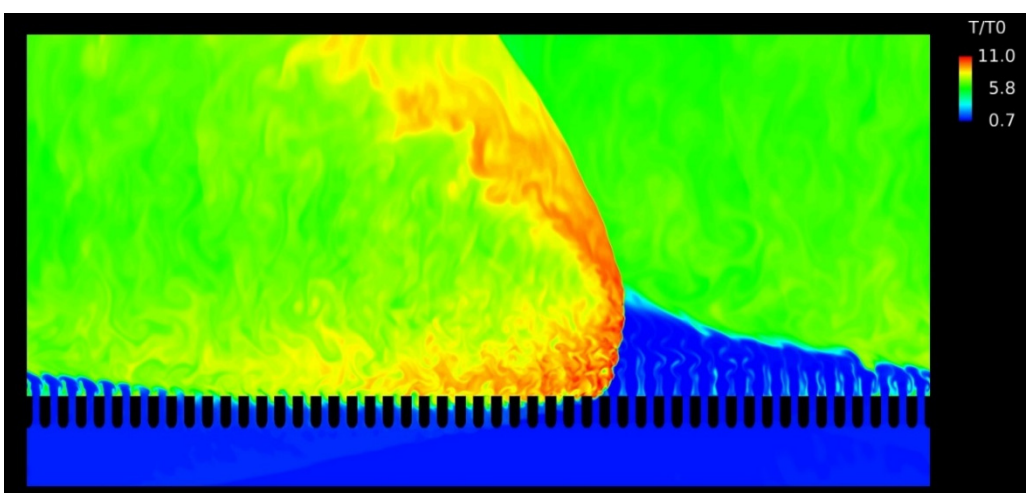

Figure 6.—High fidelity CFD contours of normalized temperature in a premixed, inviscid "Unwrapped" RDE at a moment in time.

Other recent NASA RDE work has included contributions to the study of an experimental RDE rocket utilizing hypergolic propellants at Purdue University (Ref. 179) and grant support for the development of a "nested" quasi-1D code for use with RDE's (Ref. 180), also at Purdue. This proposed code will do quasi-1D sweeps in the azimuthal direction, the results of which will provide source terms to a series of lumped parameter zones in the axial direction.

Multiconfiguration Studies

A number of studies contributed to or performed by NASA have not been restricted to a single type of PGC device. For instance, an experimental study that compared the results from a test of a large scale pulsejet with a pulse detonation engine was reported in Reference 181. In another case, a study was done on methodologies for calculating average turbine performance in a system using an unsteady flow combustor (Ref. 182). This methodology was demonstrated using an idealized pressure gain combustor, but can be applied to any type of device. Yet another case was the development of a generic PGC element to be used in gas turbine performance calculations (Ref. 183) which was incorporated into the NPSS cycle analysis code. This element could also be used in concert with the turbine performance calculation methodology in Reference 182 to give a more accurate system performance estimate. Another performance calculation methodology was developed and presented in Reference 184 for the correct accounting of the presence of unsteady flow elements in otherwise steady flow propulsion systems. Specifically, the correct method was given for incorporating unsteady flow into the conservation equations. Lastly, it was noted that the development of the many different variations of PGC propulsion system architecture found in the literature presented a challenge in nomenclature and engine station numbering. Under the auspices of JANNAF, a uniform set of standards was developed and published (Ref. 185).

Summary and Conclusions

NASA has undertaken a systematic exploration of many different facets of pressure gain combustion over the last 25 years in an effort to exploit the inherent thermodynamic advantage of pressure gain combustion over the constant pressure combustion process used in most aerospace propulsion systems. Applications as varied as small-scale UAV's, rotorcraft, subsonic transports, hypersonics and launch vehicles have been considered. In addition to studying pressure gain combustor concepts such as wave rotors, pulse detonation engines, pulsejets, and rotating detonation engines, NASA has undertaken the study of inlets, nozzles, ejectors and turbines which must also process unsteady flow in an integrated propulsion system. Other design considerations such as acoustic signature, combustor material life and heat transfer that are unique to PGC's have also been addressed in NASA research projects.

In addition to a wide range of experimental studies, a number of computer codes, from 0-D up through 3-D, have been developed or modified to specifically address the analysis of unsteady flow fields. Loss models have also been developed and incorporated into these codes that improve the accuracy of performance predictions and decrease computational time. These codes have been validated numerous times across a broad range of operating conditions, and it has been found that once validated for one particular PGC configuration, these codes are readily adaptable to the others.

All in all, the documentation of this work has encompassed approximately 170 NASA technical reports, conference papers and journal articles to date. This documentation does not include the significant contributions made by NASA research staff to the programs of other agencies, universities, industrial partners and professional society committees through serving as technical advisors, technical reviewers and research consultants. Despite this volume of research, many challenges remain before any one of the family of PGC devices can be put into service on a modern aerospace vehicle. Inclusion of a PGC device changes every aspect of the propulsion system design and therefore cannot be treated as a "drop-in" replacement component. It is because of the potential for large performance benefits across a broad range of propulsion applications that research continues on this class of combustion devices despite these challenges.

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