**pulse detonation engine full report for gentle landings.**

The engine operates on pulses, so controllers could dial in the frequency of the detonation in the "digital" engine to determine thrust. Pulse detonation rocket engines operate by injecting propellants into long cylinders that are open on one end and closed on the other. When gas fills a cylinder, an igniterâ € such as a spark plugâ € is activated. Fuel begins to burn and rapidly transitions to a detonation, or powered shock. The shock wave travels through the cylinder at 10 times the speed of sound, so combustion is completed before the gas has time to expand. The explosive pressure of the detonation pushes the exhaust out the open end of the cylinder, providing thrust to the vehicle.

A major advantage is that pulse detonation rocket engines boost the fuel and oxidizer to extremely high pressure without a turbo pumpâ € an expensive part of conventional rocket engines. In a typical rocket engine, complex turbo pumps must push fuel and oxidizer into the engine chamber at an extremely high pressure of about 2,000 pounds per square inch or the fuel is blown back out.

The pulse mode of pulse detonation rocket engines allows the fuel to be injected at a low pressure of about 200 pounds per square inch. Marshall Engineers and industry partners United Technology Research Corp. of Tullahoma, Tenn. and Adroit Systems Inc. of Seattle have built small-scale pulse detonation rocket engines for ground testing. During about two years of laboratory testing, researchers have demonstrated that hydrogen and oxygen can be injected into a chamber and detonated more than 100 times per second.

NASA and its industry partners have also proven that a pulse detonation rocket engine can provide thrust in the vacuum of space. Technology development now focuses on determining how to ignite the engine in space, proving that sufficient amounts of fuel can flow through the cylinder to provide superior engine performance, and developing computer code and standards to reliably design and predict performance of the new breed of engines.

A developmental, flight-like engine could be ready for demonstration by 2005 and a full-scale, operational engine could be finished about four years later. Manufacturing pulse detonation rocket engines is simple and inexpensive. Engine valves, for instance, would likely be a sophisticated version of automobile fuel injectors. Pulse detonation rocket engine technology is one of many propulsion alternatives being developed by the Marshall Centerâ„¢s Advanced Space Transportation Program to dramatically reduce the cost of space transportation.

2. DIFFERENCES COMPARED TO OTHER ENGINE TYPES

The main differences between the PDE and the Otto engine is that in the PDE the combustion chamber is open and no piston is used to com- press the mixture prior to ignition (and also that no shaft work is extracted).

Instead the compression is an integral part of the detonation, and two of the main advantages of the PDE - the efficiency and simplicity - can be explained by the fact that the combustion occurs in detonative mode. The efficiency of the cycle can be explained by the high level of precompression due to the strong shock wave in the detonation.  Also, the simplicity of the device is a result of the fact that the shock wave - responsible for this compression â€œ is an integrated part of the detonation. Therefore, pre-compression through mechanical devices (e.g., a piston) is not necessary. In this sense the PDE is similar to both the pulse-jet (e.g., the engine used for propulsion of the V-1) and the ram jet engine. But in those two cases the mechanism behind the pre-compression is completely different:

Â¢ For the pulse-jet the pre-compression is a result of momentum effects of the gases, and is a part of the resonance effects of the engine. The resonance effects are influenced strongly by the external conditions of the engine, and the thrust is drastically reduced at higher speeds (approaching speed of sound). Furthermore, both the specific impulse and the specific thrust are significantly lower than for turbo-jet or turbo-fan engines. This is due to the fact that the levels of preconditioning that can be obtained through the resonance effects are rather low.  Â¢ In the ramjet, pre-compression is obtained through the ram effects as the air is decelerated from supersonic to subsonic. The major drawback with this concept is that the engine is ineffective for speeds lower than around Ma=2. Fig. 1 Main chamber with pressure transducers (used to detect the detonation), Shchelkin Spiral (used to enhance the transition from flame to detonation), spark plug and central body

2.1 EXPERIMENTAL SET-UP

One example of a PDE is shown in Fig 1. This particular engine - which was assembled at one of FOI's (the Swedish Defence Research Agency) departments, Warheads and Propulsion - runs on hydrogen and air and is capable of reaching  frequencies up to 40~Hz. The experimental set up is rather simple, basically consisting of a straight tube (in this case with a length of about one metre) in which hydrogen and air is injected, and ignited by an ordinary spark plug. In this experimental engine, the pressure transducers are only used to find out whether the engine operates successfully in detonative mode. This can be seen both by the level of pressure and the speed of propagation of the wave (a detonation in hydrogen air reaches pressures over 20 bar and propagates at around 2,000 m/s). That is, the pressure transducers are used just for the experiments and are not necessary for the operation of the engine.  Also shown is a spiral, which, since it helps to induce turbulence in the flow field is known to speed up the transition from flame to detonation. The hydrogen enters the engine through twelve holes of 1 mm. diameter at the edge of a 72 mm. diameter disk at the right end of the engine. The air enters between the central body through which the hydrogen is emerging and the interior walls of the tube.

3 PRE-COMPRESSION AND DETONATION

In the PDE the pre-compression is instead a result of interactions between the combustion and gas dynamic effects, i.e. the combustion is driving the shock wave, and the shock wave (through the increase in temperature across it) is necessary for the fast combustion to occur. In general, detonations are extremely complex phenomena, involving forward propagating as well as transversal shock waves, connected more or less tightly to the combustion complex during the propagation of the entity.  The biggest obstacles involved in the realization of an air breathing PDE are the initiation of the detonation and the high frequency by which the detonations have to be repeated. Of these two obstacles the initiation of the detonation is believed to be of a more fundamental character, since all physical events involved regarding the initiation are not thorough- ly understood. The detonation can be initiated in two ways; as a direct initiation where the detonation is initiated by a very powerful ignitor more or less immediately or as a Deflagration to Detonation Transition (DDT) where an ordinary flame (i.e. a deflagration) accelerates to a detonation in a much longer time span.

Typically, hundreds of joules are required to obtain a direct initiation of a detonation in a mixture of the most sensitive hydrocarbons and air, which prevents this method to be used in a PDE (if oxygen is used instead of air, these levels are drastically reduced). On the other hand, to ignite an ordinary flame requires reasonable amounts of energy, but the DDT requires lengths on the order of several meters to be completed, making also this method impractical to use in a PDE. It is important to point out that there are additional difficulties when liquid fuels are used which generally make them substantially more difficult to detonate. A common method to circumvent these difficulties is to use a pre-detonator - a small tube or a fraction of the main chamber filled with a highly detonable mixture (typically the fuel and oxygen instead of air) - in which the detonation can be easily initiated.

The detonation from the pre-detonator is then supposed to be transmitted to the main chamber and initiate the detonation there. The extra component carried on board (e.g. oxygen) for use in the pre-detonator will lower the specific impulse of the engine, and it is essential to minimize the amount of this extra component.

4. PRINCIPLE OF THE ENGINE

As the name implies the engine operates in pulsating mode, and each pulse can be broken down to a series of events. The time it takes to complete each of these events puts a limit to the performance of the engine, and the thrust can be shown to be proportional to the frequency and volume of the engine. The events in one cycle are shown schematically in Fig 2, where p0 is the ambient pressure, p1 represents the pressure of the fuel and air mixture, p2 is the peak pressure of the detonation and p3 is the plateau pressure acting on the front plate. As stated above, the thrust of the engine is proportional to the frequency of the engine, and in order to reach acceptable performance levels the indicated cycle has to be repeated at least 50 times per second (depending on the application and the size of the engine).

4.1 STATUS

The first experiments on the PDE were done in the beginning of the 1940s, and since then several experiments and numerical calculations have been done. No flying applications have been reported in the open literature, and doubts have been expressed regarding the claimed success of some of the earlier experiments. However, in recent years the PDE has received a renewed interest, and especially in the US work in many different fields related to the PDE has been initiated. One of the most promising efforts is pursued at the Air Force Research Lab (AFRL) at Wright Patterson's Air Force Base headed by Dr. Fred Schauer In that group successful operation of a PDE using hydrogen and air at frequencies at least up to 40 Hz has been demonstrated. In a series of experiments, the proportions between air and hydrogen have been varied from stoichiometric (i.e., where in an ideal combustion process all fuel is burned completely) to lean mixtures. Even at rather lean mixtures the engine is reported to operate in detonative mode and to deliver the expected performance.

This is an indication that the engine could operate on liquid hydrocarbon fuels since those fuels (in a stoichiometric mixture with air) and lean hydrogenair mixtures have similar properties regarding the initiation of the detonation. The PDE at FOI described earlier, did not produce clean detonations propagating over the whole length of the engine. In an effort to improve the situation several parameters were varied: Â¢ The length of the mixture chamber. Â¢ The shape of the contraction section connecting the air supply to the rest of the engine. Â¢ The separation between the contraction section and the beginning of the tube. Â¢ The position where hydrogen is introduced. Â¢ The position of the spark plug. Â¢ In four of the geometries a reed valve was also used, in an attempt to uncouple the engine from the supply systems during the initiation of the detonation. In these cases hydrogen was introduced either upstream or downstream relative to the valve. These changes did not result in a successful, detonative operation of the engine. However, localized peak pressures well above those obtained in detonations, and valuable insight regarding detonations were obtained.

For example, it was concluded that a valve controlling the inflow of hydrogen and air is a critical component in the engine. This is also the most significant difference between the engine at FOI and the successful one at AFRL described above. This issue is addressed in the ongoing research at FOI, whose goal is to obtain better understanding of the physical processes involved, and thereby providing efficient design strategies for the PDE.

5. COMBUSTION ANALYSIS

While real gas effects are important considerations to the prediction of real PDE performance, it is instructive to examine thermodynamic cycle performance using perfect gas assumptions. Such an examination provides three benefits. First, the simplified relations provide an opportunity to understand the fundamental processes inherent in the production of thrust bythe PDE. Second, such an analysis provides the basis for evaluating the potential of the PDE relative to other cycles, most notably the Brayton cycle. Finally, a perfect gas analysis provides the 0framework for developing a thermodynamic cycle analysis for the prediction of realistic PDE performance.

The present work undertakes such a perfect gas analysis using a standard closed thermodynamic cycle. In the first sections, a thermodynamic cycle description is presented which allows prediction of PDE thrust performance. This cycle description is then modified to include the effects of inlet, combustor and nozzle efficiencies. The efinition of these efficiencies is based on standard component performance. Any thermodynamic cycle analysis of the PDE must begin by examining the influence of detonative combustion relative to conventional deflagrative combustion. The classical approach to the detonative combustion analysis is to assume Chapman-Jouget detonation conditions after combustion.

The Chapman-Jouget condition is merely the Rayleigh line analysis limited to sonic velocity as the outlet condition, Shapiro4. Detonation is the supersonic solution of the Chapman-Jouget limited Raleigh analysis, Figure 1. The subsonic Chapman-Jouget solution represents the thermally choked ramjet. To insure consistent handling of the PDE and ramjet, this paper uses Rayleigh analysis for both cycles. A comparison of the ideal gas Rayleigh process loss was made for deflagration and Chapman-Jouget detonation combustion, Figure 2. The comparison was made for a range of heat additions, represented here by the ratio of the increase in total temperature to the initial static temperature. Four different entrance Mach numbers were also considered. The figure of merit for the comparison is the ratio of the increase in entropy to specific heat at constant pressure. The results show that at the same heat addition and entrance Mach number, detonation is consistently a more efficient combustion process, as evidenced by the lower increase in entropy. This combustion process efficiency is one of the basic thermodynamic advantages of the PDE.

6. IDEAL CYCLE ANALYSIS

The following sections examine the relative performance of the PDE, ramjet and Brayton cycles along a representative 500 psf dynamic pressure trajectory for a range of realistic fuel-toair ratios, Figure 3. The upper fuel-to-air ratio is representative of Boeingâ„¢s missile integration study results for the maximum power heat addition required. Also shown is a reduced fuel-to-air ratio that is more indicative of cruise power settings. These fuel-to-air ratios relate to the heat added to the cycle. The mass addition related to the fuel is neglected in this analysis. The perfect gas analyses use a ratio of specific heats of 1.4, a specific heat at constant volume of 0.240 BTU/lbm, and a fuel heating value of

19000 BTU/lbm.

This first section presents a perfect gas, ideal cycle comparison to assess the identified combustion process efficiency influence on thrust production efficiency. The methodology used provides inclusive, first principals performance analysis that effectively captures the unique combustion process of the PDE. The analysis begins with the familiar expression for conservation of energy which describes the increase in both kinetic and thermal energy through the engine as developed by the heat of combustion, qadd:

The change in velocity from the freestream, V0, to the exit, V10, is computed by comparing the amount of energy added, qadd, with the amount of thermal energy rejected in the exhaust stream, qrej; the difference being the energy available for thrust. Equation ( 3 ) presents the definition of thermodynamic efficiency based on the energy available from the fuel, qsupp, which comprises the lower heating value and the fuel-to-air ratio: And inserting these velocity relationships into the basic thrust equation,

The basic relationship between the thermodynamic efficiency and the thrust is found. This energy-based thrust equation is the form described in detail by Heiser and Pratt3:

The sequence, 0-3-4-10 along the dashed lines in Figure 4 illustrates the energy-entropy states experienced by a fluid element moving through a ramjet. The dotted line 10-0 closes the cycle by a virtual process returning the exhaust (state 10) to free stream conditions (state 0). The corresponding states of a fluid element being processed by a detonation wave are given as 0-3- 4-10-0 along the solid and dotted lines. The change from state 3 to state 4 is described by the classical Zelâ„¢dovich-von Neumann-Doe ring (ZND)5 model of the detonation which consists of a close coupled shock compression and combustion zone. As experienced by the fluid element, the leading shock wave of the detonation does work on the fluid element, raising its energy state, 3-3â„¢. After this compression the total enthalpy includes the kinetic energy associated with detonation velocity. It is at this elevated energy state that the heat of combustion is released, 3â„¢-4. Thus, at the end of the Chapman-Jouget (CJ) detonation (state 4 on the solid line), the static enthalpy of the fluid element is higher than the combined entrance total enthalpy and heat release (state 4 on the dashed line). Thus, the unsteady wave motion provides a local boost in kinetic energy in the detonative system. This kinetic energy boost is paid back, however, during the expansion process coupled to and following the detonation wave, which brings the fluid back to rest relative to the detonation tube. This expansion process causes the fluid element to expand, doing work by pushing the detonation along, and thus losing energy. The fluid state that results from this expansion lies between station 4 and 10 on the solid line. For the initial comparisons, all component processes are assumed perfect, with no losses. The only cycle loss mechanism is the loss due to heat addition included in the ramjet and PDE analyses. The heat addition losses are the sole difference between the Brayton cycle and ramjet performance shown, Figure 5. Those heat losses are most significant at lower speeds, where the largest differences between the Brayton cycle and the ramjet are shown. The PDE has higher thermodynamic efficiency, Figure 6, and reduced fuel consumption at all Mach numbers relative to both the Brayton cycle and the ramjet. This is, of course, a direct result of efficient combustion process, figure 2

7 LOSSES

7.1 INLET LOSSES

To understand the relative importance of each component efficiency to the ideal cycle analysis, component efficiencies were added one at a time. The first component efficiency added was inlet total pressure recovery. For the inlet component efficiency model, MIL STD 5007D total pressure recoveries were used. To use total pressure recovery as an efficiency index, ideal gas relationships were used to transform the total pressure recovery into its associated process temperatures. These process temperatures were then used to compute a compression efficiency for use in the cycle analysis. The resultant fuel consumption comparison is shown in Figure 7. As both the ramjet and the PDE are experiencing the same component efficiency through the same compression process, no change occurred to the relationship between the cycles. The PDE still exhibits reduced fuel consumption at all Mach numbers.

7.2 COMBUSTOR LOSSES

The next step in the cycle comparison is to introduce degraded combustor component
efficiencies. In this step, a nominal 90% heat release efficiency was used. The results, Figure 8, are similar to the inlet degraded results in that the PDE still exhibits reduced fuel consumption. As before, both the ramjet and PDE are experiencing similar component losses, so no significant relative change in performance occurs.

7.3 NOZZLE LOSSES

For nozzle loss modeling, the generally accepted nozzle gross thrust coefficient, CV, is used. Gross thrust is obtained from the equation: The ideal gross thrust, Fg,i, is derived from ideal expansion to ambient pressure: Where VY is the ideal velocity of the flow expanded to ambient pressure with no losses. To use nozzle gross thrust coefficient, the energy based thrust equation ( 5 ) must be combined with the basic thrust equation ( 4 ). Substituting the definition of the gross thrust coefficient ( 6 ) results in an expression of the actual exit velocity including losses:  In this formulation, the thermodynamic efficiency must not include any nozzle thermodynamic losses as they are included in the CV. Using the above formulation for thrust, the fuel consumption for a real engine was computed, as shown in Figure 9, using a CV of 95%. Once again, the PDE sustains its performance advantage at all Mach numbers. This result differs from the previous work of Heiser and Pratt3. To understand the apparent discrepancy with previous results, an examination of the momentum and energy forms of the nozzle efficiency (CV and e respectively) is necessary. The two nozzle efficiencies are directly related, equation ( 10 ). The state indicated by subscript Y, represents the isentropic expansion from state 4 to ambient pressure. Expansion losses result in the actual nozzle exit velocity, V10, being lower than that possible with isentropic expansion, VY. The lower exit velocity equates to lower kinetic energy and higher temperature in the exhaust stream. The 95% value of CV used for this study equates to an e of 90%.

The momentum form of the nozzle efficiency, CV, operates only on the ideal thrust.
The ideal thrust, in turn, is solely dependent on the post-combustion fluid entropy, state 4, and ambient pressure, which together define the ideal, isentropic expansion to state Y, Figure 10. Therefore, the momentum form, CV, is only dependent on the post-combustion entropy state of the fluid, and independent of the postcombustion fluid energy level, state 4. On the other hand, the energy form of the nozzle efficiency, e , operates directly on the energy of state 4, as can be seen by comparing Figure 4 with Figure 10. Expanding from state 4 incurs a significant loss. However, as previously explained, state 4 includes the kinetic energy of the detonation wave which is paid back when the gases expand back to static conditions. Therefore, a state reflective of the actual available energy is required. The state reflective of the actual energy available to the system can be determined by considering the instant the detonation arrives at the end of the chamber. At this instant, the entropy of the system is known to be the same as the CJ entropy because the gases expand to rest isentropically. The known CJ entropy, combined with the known system enthalpy (h0 + qadd), defines state 4â„¢. In this way, energy conservation is assured. This analysis assumes a thin detonation wave. When the energy form of nozzle efficiency is applied to the cycle, the difference between state 4 and 4â„¢ becomes important, Figure 10. Counter intuitively, use of the higher energy state, 4, results in lower performance. This occurs because the expansion from the higher energy state leads to higher entropy generation and lower performance. Use of energy state 4â„¢, rather than state 4, is a more representative cycle point for accurate PDE analysis, as it more appropriately represents the available energy: The performance implications of properly conserving energy in the nozzle expansion are significant, Figure 11. Because energy state 4 is not available to produce thrust, itâ„¢s use in the computation of the nozzle loss, labeled PDE from CJ, over penalizes the PDE cycle. The higher losses result in the PDE performance dropping below that of the ramjet around Mach 4. Analyses based on the conserved energy state, 4â„¢, labeled Energy Conserved PDE, results in an advantage for the PDE through Mach 5. It also can be shown that the energy conserved e analysis of Figure 11 is in excellent agreement with the CV analysis of Figure 9, further corroborating the methodology.

7.4 CRUISE POWER COMPARISONS

To complete this study of PDE performance, reduced power levels meant to represent cruise conditions were evaluated. The cruise fuel-to-air ratios of Figure 3 were used. The resulting fuel consumption is presented in Figure 12. The energy conserved PDE maintained its fuel consumption improvement over the ramjet, although its margins of improvement have diminished. These diminished margins are a direct result of the diminished heat addition. Since the heat addition, i.e. combustion, phase of the cycle provides the PDE its efficiency advantage, its advantage reduces as the heat addition reduces This effect is illustrated in more detail in Figure 13, where the fuel consumption at Mach 3 is examined over a range of heat additions. At the higher heat additions, represented here by the higher levels of specific thrust, the PDE enjoys its highest fuel consumption benefit. As heat addition and specific thrust are reduced, the PDE advantage is reduced until at the low power settings the ramjet enjoys the better fuel efficiency. It should be noted, however, that these power settings are not representative of sustained flight, as vehicle drag will far exceed engine thrust.

7.5 DYNAMIC CONSIDERATIONS

In order to make a first order comparison between the PDE and ramjet cycles, the present analysis conserved global enthalpy and tracked the entropy generated by the detonation and by process inefficiencies. However, in order to gain further insight into the detailed operation of the PDE cycle and to apply more suitable component efficiencies, the different phases of PDE operation must be carefully described. Such a thermodynamic description will carefully apply conservation of energy and conservation of enthalpy respectively to the imbedded constant mass and steady flow processes which occur during the PDE cycle. For example, the preceding discussions have used global enthalpy considerations to examine the most appropriate state against which to levy nozzle losses. A similar result can be reached through consideration of the wave dynamics in the chamber: After being processed by the detonation wave, each fluid element is brought to rest relative to the closed end of the detonation tube by an isentropic expansion. The coupled expansion is an inherent part of the detonation which burns the mixture in the chamber in a constant mass process. In contrast, the subsequent blow-down of the detonated gas from the chamber is a quasisteady flow process. It is the blow-down process which generates thrust and is directly related to the classical nozzle flow.  Thus, again, it is appropriate to keep the detonation-coupled expansion process (4 - 4â„¢) separate from the flow expansion through the nozzle (4â„¢ - 10â„¢) and to assess nozzle losses against this last expansion. The simplified analysis given in the previous sections made use of constant nozzle thrust coefficients. Real systems, however, have fixed nozzles, and their performance is a function of nozzle pressure ratio. Nozzle pressure ratio can vary an order of magnitude during the dynamic chamber blow-down process. Detailed integrations of the blowdown process show that energy is conserved.  To correctly enter the dynamic blowdown calculation, the system energy, rather than enthalpy, must be evaluated. A state 4 is defined to be the conserved energy and post-CJ entropy, Figure 10. In the dynamic calculation the energy exiting the PDE can be shown to conserve global enthalpy exactly as state 4â„¢. However, since impulse is a function of velocity and kinetic energy is a function of velocity squared, the dynamically calculated impulse is slightly lower than the effective steady state computation.

8 COMPUTATIONAL AND EXPERIMENTAL STUDIES OF PULSE DETONATION ENGINES

Recent research on pulse detonation engines has revealed their potential as an efficient, low cost propulsion system. Conceptually, pulse detonation engines (PDE) offer few moving parts, high thrust to weight ratios, low cost, and ease of scaling. The fact that a PDE does not require a high pressure-ratio compression cycle eliminates the necessity for heavy and expensive compressor and turbine units. Due to the obvious potential advantages of the pulse detonation cycle, a PDEbased propulsion system is an attractive alternative to conventional propulsion systems. The flight operating range of a PDE ranges from static conditions to hypersonic flight Mach numbers. As of today, there are no single-cycle propulsion systems available with such potential for a broad range of operability. A recent review on PDE research efforts has been given by Kailasanath [1]. Researchers have discovered that one method of controlling thrust and specific impulse of a pulse detonation engine is to alter the pressure relaxation rate by adjusting the percentage of the tube that is filled (fill fraction) with a detonable mixture [2]. Fill fraction is defined as the ratio of the tube volume that is initially filled with a fuel/air mixture to the overall tube volume.

At fill fractions greater than 1.0, i.e. the detonable mixture over-fills the tube, no increase in performance was observed because the external detonation process has no thrust surface upon which to act. Conversely, if the tube is underfilled, essentially acting as a straight nozzle with either a purge cycle or previous cycle exhaust products filling the remainder of the tube, significant performance gains were observed. It is believed that the unfilled portion of the tube acts to increase the blow-down relaxation length and time scales [2].  Alternatively, converging and/or diverging nozzles can be used to alter the blowdown process. Eidelman and Yang [3] have shown that various nozzle geometries dramatically affect the performance of a PDE. Another conventional method of increasing thrust from an engine is by augmenting the engine with an ejector. The ejector is typically used to direct the entrainment of ambient air into the exhaust plume and promote the mixing of the hot combustion products with the relatively cooler ambient flow, whereby an increase in exhaust mass flow rate is obtained. Due to the increased exhaust mass flow, thrust augmentation can be achieved with an appropriately contoured ejector. Previous work by Bernardo and Gutmark [4] have shown the potential of using ejectors to entrain mass flow for steady-flow supersonic jets. However, the benefits of pulsed vs. steady supersonic ejectors, as a function of frequency, is not known. The successful extension of this concept to pulse-detonation engine applications hinges upon rapid mixing between the primary (detonation) stream and the secondary (ambient) stream over a short distance. Due to the unsteady nature of the detonation, strong vortical structures can develop which can aid in the mixing of the two streams thus providing additional thrust augmentation over steady-flow ejectors [5]. In addition to the thrust augmentation due to mass entrainment, an ejector for a PDE can provide thrust by favorably altering the blow-down process. Preliminary investigations of pulse detonation driven ejectors have been performed computationally by Allgood et al. [6] and experimentally by Hoke et al. [7]. Insight into how devices such as nozzles and ejectors affect the thrust production of a PDE can be obtained by visualizing the exhaust jet. Development of design guidelines for optimization of PDE systems requires a complete understanding of this complex and dynamic jet flow and the ability to predict its behavior. Although separate experimental and computational studies on PDEâ„¢s have been performed and are published in the literature, there has been a lack of direct comparison between data produced from experimental flow diagnostic techniques, such as shadowgraph/schlieren visualizations, and computational modeling predictions.

The current work is presented as a preliminary contribution in this area. The first objective was to develop a shadowgraph/schlieren system that was capable of capturing the fast moving detonation wave without any visible smearing or distortions of the shock waves. This required nanosecond duration of light pulses from the shadowgraph light source that must be synchronized with the detonation event. The second objective was then to compare these results to direct numerical simulation predictions. This direct comparison would provide guidance and validation towards the development of accurate tools for modeling pulse detonation engines. Experimental Facility The experimental data presented here was obtained through the use of the research pulse detonation engine PDE) at the Air Force Research Laboratory located at Wright-Patterson Air Force Base.
The PDE test facility was located in a 750,000 cubic feet test cell that was enclosed by 2ft reinforced concrete walls. The valve system of the PDE was constructed from a modified automotive cylinder head. Hydrogen and air were metered through choked flow nozzles and premixed before being injected into the PDE tube. The pressure and flow rate data was collected via a remote 5MHz 16-channel ADC system. The PDE was mounted on a 1000 lbf maximum damped thrust stand. However, during these tests the thrust stand was prevented from moving to allow accurate and repeatable shadowgraph visualizations of the exhaust flow to be obtained. For a more detailed description of the PDE test facility the reader is referred to the recent paper by Schauer et al. [8].  A detonation tube of 2-inch diameter and 40-inch length was used in the current work. The PDE was operated at stoichiometric conditions of premixed hydrogen/air and the tube was completely filled with a detonable mixture before being ignited. Two pressure transducers were mounted 6 inches apart to monitor the speed of the detonation wave. The measured wave speed was confirmed to be approximately the Chapman- Jouget wave speed of 1966 m/s. One of the two transducers were mounted 12 inches upstream of the exit of the PDE and was used to trigger the shadowgraph system. In addition to the straight tube experiments, the effects of exit-area reduction on the structure of the detonation wave and dynamic blow-down process was visualized. A standard 1.84:1 diameter ratio pipe reducer (3.5 inches long) served as a converging nozzle and was mounted at the end of the detonation tube.

8.1 NANOPULSER SHADOWGRAPHSYSTEM

A shadowgraph imaging system for visualizing pulse detonation engine flowfields was designed and constructed at the University of Cincinnati. This unique imaging system utilizes a nanopulser arc light source (XENON CORP.) that delivers 10 nanosecond duration light pulses. Light pulses of this time scale were necessary to capture traveling detonation waves without excessive smear of the leading shock wave. For example, for a Chapman-Jouget wave speed of 2000 m/s, the shock wave will only have moved 0.02 mm in the image. Figure 1 shows a schematic of the UC shadowgraph system aligned to visualize the exhaust flow of the AFRL PDE test rig. The light was collimated by 6 f/8 parabolic mirrors and refocused onto a high-speed CCD imager using a modified z-configuration. This arrangement was selected due to physical limitations of the laboratory. To avoid CCD saturation, neutral density filters were used to attenuate the light before entering the camera. During the testing, both the camera and the light source were triggered using a pressure transducer mounted 12-inches upstream of the exhaust of the PDE. When the pressure transducer recorded the event of a passing detonation wave (i.e. the voltage exceeded a given threshold) a TTL signal was sent to both the camera and the light source.

By varying the delay of the light source, successive images could be captured to reconstruct the dynamic blow-down process of one complete detonation cycle from separate detonation events. The repeatability of the detonations was evident in the consistency of the images that will be shown in the experimental results section of this paper. Description of Computational Simulations A two-dimensional Navier-Stokes solver was written in generalized coordinates (figure 2) at the University of Cincinnati. The numerical method for solving the governing equations was the explicit 2nd-order MacCormack predictorcorrector technique with the nonlinear monotonicity preserving 4th-order FCT (Flux- Corrected Transport) scheme [9].

The artificial damping and anti-diffusion coefficients used in the FCT algorithm were those recommended by Boris and Book [9] for minimum residual diffusion. The chemical reactions are represented by the parametric Korobeinikov model [10] consisting of a two-step reaction mechanism: 1) a non-exothermic irreversible induction reaction where the progress variable () changes from 1 to 0, and 2) an exothermic reversible recombination reaction with its progress variable () changing from 1 to eq (0.23). The rates of change of these variables were expressed in the Arrhenius form given in figure 2. This two-step reaction model has been used successfully in the past to address two-dimensional unsteady detonation problems [11-13]. The parameters of the model were chosen to represent a stoichiometric hydrogenoxygen mixture diluted with argon (2H2+O2+7Ar) [10]. This mixture is known to generate a wellbehaving detonation with a Chapman-Jouguet (CJ) detonation wave Mach number of 4.8. The computational grid used in the simulations is outlined in figure 3. A uniform mesh with a grid spacing of 0.02933 cm (0.011547 inches) was selected. This resolution provided at least 10 points within the induction zone of a CJ detonation wave, which has been shown in previous studies to be more than adequate for resolving the correct cell size of the regular twodimensional cellular detonation wave [14]. The time-step was chosen based on a CFL number of 0.2.

Due to the near symmetry of the problem, symmetry-boundary conditions were imposed at the bottom boundary of the 6 computational domain (see figure 3). This allowed a smaller grid size to be used in the simulations. The top and bottom of the domain were modeled as exit boundaries with 1st order extrapolations being imposed. The left boundary was modeled as a reflective wall. The pulse-detonation engine tube and head wall were also modeled as reflective free-slip walls. Wall boundary layer effects were not modeled in the simulations. The diameter of the PDE tube was matched to the 2-inch diameter tube used in the experiments. However, due to computer memory restrictions, a shorter detonation tube of 10 inches was used in the modeling.

Thus, only the data for the initial blow-down process was compared to the experiments. The detonation was initiated in the simulations by an initial condition of two small high pressure/temperature regions near the head end of the tube. The energy level and size of the ignition regions were chosen to produce a fast and stable cellular detonation wave. The entire domain was modeled as a stationary uniformly filled mixture of combustible gases. This uniform mixture distribution was selected since the tube was being generously overfilled during the experiments, and because the level of overfilling could not be accurately determined, the entire computational domain was initially filled with a detonable mixture. Future computational and experimental efforts will investigate the effects of  overfilling/underfilling the tube on the structure of the PDE exhaust flow. Experimental Results Two PDE configurations were tested in the current work. The first configuration was a 2- inch diameter straight tube of 40-inch length with no exit nozzle present. The tube was completely filled with a detonable mixture of H2 and air (fill fraction 1). Only single shot detonations were run in these experiments. The second configuration had the same geometry as the first except for a 1.84:1 diameter ratio converging nozzle at the exit of the PDE. Sample pressure traces recorded by the pressure transducer are given in figure 4. The figure shows that the converging nozzle maintained the pressure in the PDE tube for a longer time period. By restricting the mass flow and thereby delaying the blowdown process, a converging nozzle improved the thrust production of a PDE but could limit its maximum operating frequency due to the increased blow-down time. Figures 5a and 5b are series of shadowgraph images for configuration 1 with no exhaust nozzle present. The numbers below each image correspond to the time delay in milliseconds that was added to the light source. For example, the first image in Figure 5a shows the detonation wave exiting from the detonation tube 0.3125 ms after the wave had passed the pressure transducer.

This image also shows that very strong gradients were produced by the leading shock wave as indicated by the thick black band in front of the leading shock wave. As the detonation front exited from the tube, the shock wave wrapped itself around the tube and traveled both upstream and downstream of the PDE. Behind this leading shock wave was a small zone of gas that separated the shock front from the combustion products (see the 0.45ms image). This is believed to be the well-known induction region of a detonation wave. As the combustion products expanded out of the PDE into a large plume, a well-defined vortex formed and a strong Mach disk developed approximately 1 diameter downstream of the exit as shown in the 0.51ms image.

The continued decay in tube pressure to atmospheric pressure caused the Mach disk to weaken into oblique shocks, which at this time have now propagated upstream closer to the exit of the PDE (t=0.62ms). At a delay of 0.7ms, the oblique shock waves were no longer visible. At 3ms, the pressure recorded by the upstream transducer had dropped to atmospheric pressure and the exhaust flow resembled that of a turbulent jet. When the pressure inside the tube over-expanded belowatmospheric pressure, the jet was observed to contract in diameter resembling a venturi effect. Finally, 6.5 ms later no combustion gases were seen to exit from the tube.

The converging nozzle was then placed at the exit of the PDE tube to observe the effects of the reduction in area on the exhaust flow. Figure 6 is a collection of shadowgraph images obtained for this configuration. Several differences in the flow structures were observed. First, the detonation wave exiting from the converging nozzle appeared to have a more oblong shape, in that the leading shock wave moved much more rapidly along the centerline of the PDE than in the radial direction along its outer periphery.

This was a result of the constriction accelerating the detonation wave before exiting the PDE. Second, the vortex produced by the detonation wave was initially smaller than in the case without the constriction. This change in vortex size was expected since the exit jet diameter was smaller with the converging nozzle. However, the size of the vortex is an important parameter in the operability of a PDE driven ejector [6]. The ejector should be properly sized so as to be able to ingest the vortex plume. Thus, optimum ejector dimensions will be a function of the PDE exit nozzle geometry. The final observation that can be made from these results is that Mach wave radiation during the blow-down process was more visible with a converging nozzle at the exit of the PDE. This might suggest that a converging nozzle would produce more turbulent jet noise.

In future PDE noise studies, detailed acoustic measurements will be obtained to quantify any difference in the noise levels due to exit nozzle geometry and correlate these differences to the observations made in the shadowgraph visualizations. Computational Results The results obtained from the computational modeling were processed to visualize the gradients in density and species (mass fraction) concentrations. This provided a means for qualitatively comparing the simulations to the shadowgraph data shown previously. It should be noted that shadowgraph images are a result of line-of-sight integration. No corrections in the data were made for this difference. Also, due to the limitation of the computational grid size and the small viewing range of the shadowgraph system, only the initial exiting of the detonation wave was compared. Figure 7 is a collection of images of experimental (top-half of each image) and computational (bottom-half of each image) visualizations for the straight tube configuration. The first image in figure 7 corresponds to a delay of 0.325ms from when the detonation wave passed the upstream pressure transducer.

Both the experimental and computational results showed qualitatively the same structure and size of the exiting detonation wave â€œ a strong leading shock wave closely followed by a region of hot combustion products. The leading shock wave of the detonation wave was observed to be nearly planar near the centerline of the PDE tube and more spherical near the outer periphery of the tube. Furthermore, both results showed a strong backward propagating shock wave. exited the tube, the leading shock increased its detachment distance from the trailing combustion gases. This separation distance of the leading shock from the combustion gases was accurately predicted by the modeling. Inside the detachment region, both the experiments and modeling visualizations depict the formation of randomly spaced compression waves.

The final observation that can be made from this comparison is that the computations model correctly the relative rate of growth of the leading shock wave and the formation and size of the trailing vortex. Future studies will investigate the ability of the modeling in simulating the latter stages of the blow-down process for this case and other exit nozzle configurations.

9. CONCLUSION

A first principal’s comparison was made between the performance of the ramjet and PDE cycles. The PDE was found to out perform the ramjet through Mach 5 for the ideal cycle and for representative component efficiencies. Component efficiencies were applied as inlet recovery, combustion heat release efficiency, and nozzle velocity coefficient. It was shown that conserving global energy provided a more representative basis for the assessment of nozzle loss. Application of the energy form of nozzle performance coefficient to the local CJ state resulted in over prediction of entropy generation during the nozzle expansion process. Finally, the effect of throttle setting was examined. It was shown that at high flight speeds and very low throttle settings, the thermodynamic advantage of the PDE is lost.  Shadowgraph visualizations of a pulse detonation engine exhaust flow field were performed using a new 10 nanosecond duration light source. The complete blow down cycle was reconstructed by synchronizing the shadowgraph system with the detonation event. Images of the highest quality were obtained with this new visualization system due to the short pulse of light produced by the source. No smear or distortion of the detonation front and shock waves was observed. The experimental visualizations were then compared to preliminary computational modeling results obtained from a newly developed code at the University of Cincinnati. Very good agreement on the structure and development of the exiting detonation wave was obtained.