AIAA 2003-0714

THE PERFORMANCE OF STEADY DETONATION ENGINES

E. Wintenberger and J.E. Shepherd
Graduate Aeronautical Laboratories,
California Institute of Technology, Pasadena, CA 91125

41st Aerospace Sciences Meeting and Exhibit
January 6-9, 2003, Reno, NV

THE PERFORMANCE OF STEADY DETONATION ENGINES

E. Wintenberger and J.E. Shepherd
Graduate Aeronautical Laboratories,
California Institute of Technology, Pasadena, CA 91125

Propulsion systems driven by steady normal detonations are studied. The practical difficulties associated with stabilizing a detonation wave are highlighted. The requirement on the freestream total enthalpy is considered in parallel with effects such as condensation or auto-ignition of the fuel-air mixture and limitations associated with fuel sensitivity to detonation. A criterion for detonation stabilization is formulated and applied to an analytical treatment of a detonation ramjet and a detonation turbojet, which are the respective analogs of the ramjet and the turbojet using detonative combustion. The performance of these engines is evaluated and compared to the ideal ramjet and turbojet models for hydrogen and JP10. A limitation is placed on the maximum total temperature allowed in the combustor, based on material considerations. The results show that steady detonation engines have a small thrust-producing range, due to the requirements for detonation stabilization. Their performance is always lower than that of the conventional ramjet and turbojet because of the total pressure loss across the detonation.

Nomenclature

c	speed of sound
C_p	specific heat at constant pressure
d	detonation tube diameter
f	fuel-air mass ratio
F	thrust
h_t	total enthalpy per unit mass
L	detonation chamber length
$\dot{m}$	air mass flow rate in engine
M	Mach number
P	static pressure
P_t	total pressure
q	non-dimensional heat release parameter for detonation wave
Q	heat release per unit mass of fuel
R	gas constant
T	static temperature
T_c	fuel or oxidizer condensation temperature
T_{ign}	auto-ignition temperature of fuel-air mixture
T_{max}	maximum temperature at combustor outlet
T_t	total temperature
$TSFC$	thrust specific fuel consumption
u	flow velocity
U_{CJ}	Chapman-Jouguet detonation velocity
w	channel width
X_{O_2}	oxygen mole fraction
Δ	reaction zone length
γ	ratio of specific heats
λ	cell width
η_0	overall efficiency
η_p	propulsive efficiency
η_{th}	thermal efficiency
ϕ	equivalence ratio
π_c	compression ratio
τ_{ign}	ignition time

Subscripts

0	freestream
2	state downstream of inlet
4	state upstream of detonation wave
5	combustor outlet
8	state downstream of turbine
9	nozzle exit
CJ	Chapman-Jouguet

Introduction

The recent interest in pulse detonation engines has focused the attention of researchers on applying detonations to propulsion. Pulse detonation engines use unsteady detonations to generate thrust. However, in order to gain a better understanding of detonation applications to propulsion, it is instructive to consider propulsion devices using steady detonation waves.

The idea of using steady detonation waves for propulsion applications is not new and started in the 1950s when Dunlap et al.[1] studied the feasibility of a reaction engine employing a continuous detonation process at the combustion chamber. Early studies[1,2] showed that no thrust was produced below a minimum flight Mach number and that the subsonic burning ramjet always has better performance, although the differences are minor at some flight regimes. These analyses assumed that the flow is slowed down or accelerated to the Chapman-Jouguet (CJ) conditions just upstream of the detonation; hence, the configurations studied included a nozzle upstream of the combustor. Dabora[3] considered a hypersonic, detonation-driven ramjet consisting of an inlet, a wedge onto which a normal or oblique detonation wave can be stabilized, and an expanding nozzle. Dabora showed that the performance of normal or oblique detonation-driven ramjets was lower (by at least a factor of 2) than that of the ramjet. Rubins and Bauer[4] experimentally studied combustion behind a normal shock generated by oblique shocks induced by wedges. They described the phenomenon observed as shock-induced combustion rather than detonation because the normal shock wave was not directly affected by the combustion. Applying these ideas to a hydrogen-fueled high-altitude scramjet concept, they calculated fuel-based specific impulses of 1000-1200 s. However, all the previous studies were conducted without placing a limit on the stagnation temperature at the combustor outlet, which creates a more realistic upper bound on the performance of any propulsion system.

The idea of using steady detonations as the main combustion mode in an engine is attractive because of the rapid energy release occurring in detonations. Since detonations are characterized by higher temperatures and pressures than deflagrations, steady detonation engines may offer performance gains over usual air-breathing engines at high flight Mach numbers. They also offer other advantages in terms of simplicity (for the detonation ramjet), higher pressure rise in the combustor which facilitates the exhaust of burned gases, and shortened combustion chamber due to a smaller reaction zone. In this paper, we looked only at normal detonation waves, although other research efforts have studied oblique detonation wave engines.[5,6] We first consider the issues associated with detonation wave stabilization and propose some criteria for the generation of stabilized normal detonations. Limitations associated with fuel sensitivity to detonation are presented. Then, we apply our solution to an analytical treatment of a detonation ramjet and a detonation turbojet, where detonative combustion replaces the usual deflagrative subsonic combustion. We show that detonation waves can be stabilized only for a limited range of initial conditions. Unlike previous studies, we place a limitation on the maximum temperature in the combustor due to material considerations. Performance figures of merit of steady detonation engines are derived using an ideal model and the results are compared with the analogs that use the standard deflagrative combustion mode.

Stabilized normal detonations

A propulsive device using a steady detonation wave is constrained by the consideration that the wave be stabilized within the combustor. Propagating detonation waves in hydrocarbon fuel-air mixtures typically move at a Mach number on the order of 5, which requires that the flow Mach number upstream of a combustor with a stabilized, steady detonation be at least this value. It is, therefore, clear why experimentally stabilizing a detonation wave may be difficult.

The first reported works on stabilized detonation waves were those of Nicholls and Dabora[7] and Gross and Chinitz.[8] Nicholls proposed some criteria for the establishment of standing detonation waves based on hydrodynamic considerations, the second explosion limit and ignition delay time considerations. The key result is that the freestream total temperature has to be high enough so that CJ detonations can be established. Gross and Chinitz[8] studied stabilized detonation waves using a normal shock generated by the intersection of two oblique shocks created by wedges in a Mach reflection configuration. Although the phenomena obtained in the experiments of Nicholls and Dabora[7] and Gross and Chinitz[8] were originally described as standing detonations, the influence of the combustion on the shock wave was very limited and these phenomena are better described as shock-induced combustion.[4] Propagating detonations are characterized by a strong coupling between the shock and the reaction zone, and by a cellular instability, which were not observed in these experiments.

The primary difficulty in creating standing detonation waves is to obtain a mixture with a total enthalpy that is high enough to stabilize the detonation without igniting the mixture upstream of the shock. On the other hand, if the total enthalpy of the flow is too low, the low post-shock temperature will result in a wider induction zone and a decoupling of the shock and the reaction zone. Shepherd[9] estimated the necessary total enthalpy by considering the stagnation states upstream of a CJ detonation. A minimum total enthalpy of 2 MJ/kg is required for hydrogen-air mixtures.

Detonation stabilization condition

We propose to study analytically the problem of generating a stabilized normal detonation wave using a flow isentropically expanded from a reservoir at a total temperature T_{t0}. This situation is analogous to the experimental setup of Nicholls et al.[10] A schematic of the problem considered is shown in Fig. 1. Air is accelerated to a supersonic velocity from a reservoir of total temperature T_{t0} through a converging-diverging nozzle. Fuel is injected at some location downstream of the nozzle throat. We assume that fuel and air mix homogeneously in an instantaneous fashion without total pressure loss. In order to stabilize a normal detonation, the flow has to be accelerated to a velocity greater than or equal to the CJ velocity through the converging-diverging nozzle. For flow velocities higher than U_{CJ}, overdriven detonations are possible but, as discussed later, the requirements for a minimum total pressure loss across the detonation in an engine make them undesirable. Hence, we will consider only CJ detonation waves. In practice, the situation described above, with a detonation wave stabilized in a nozzle, might be unstable to flow perturbations and the wave might tend to move upstream or downstream.

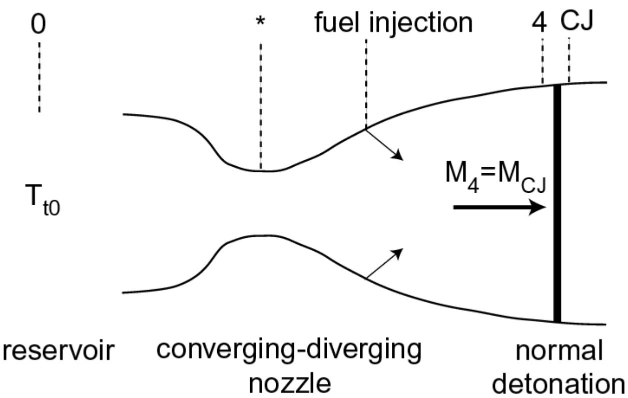

Fig. 1 Standing detonation generated by the isentropic expansion of an airflow from a reservoir of total temperature T_{t0}, with fuel injection downstream of the nozzle throat.

Assuming steady, adiabatic and inviscid flow of an ideal gas, the detonation stabilization condition can be written as $M_4 = M_{CJ}$, where station 4 corresponds to the location just upstream of the detonation wave. The detonations are modeled as hydrodynamic discontinuities, using a one-γ model.[11] This simple model does not include any considerations of the detonation wave structure. The influence of chemical kinetics and the reaction zone structure have to be considered in order to get a more realistic idea of the flow. However, the one-γ model is a useful approximation for studying the thermodynamic aspects of performance. The equation $M_4 = M_{CJ}$ can be solved analytically for the temperature upstream of the detonation wave T_4. Two solutions are obtained, only one of which is acceptable since M_{CJ} has to be greater than 1. The solution of this equation is

$$T_4 = 2\frac{\gamma-1}{\gamma+1}T_{t4}\left(\frac{1}{\gamma-1} - q - \sqrt{q(1+q)}\right) \quad (1)$$

where q is a non-dimensional heat release parameter defined by $q = fQ/(C_pT_{t4})$. Once T_4 is calculated, the properties downstream of the detonation wave can be computed using the one-γ model.

General limitations

Detonations cannot be stabilized for arbitrary values of the governing parameters. In particular, there are restrictions on the allowable values of T_4. Since the flow is accelerated through the nozzle up to a Mach number of about 5, the static temperature drop can become significant and condensation of some components of the mixture can occur in the nozzle. Hence, T_4 has to stay above a limiting temperature T_c corresponding to fuel or oxidizer condensation. Condensation is actually determined by the value of the gas-phase fuel or oxidizer partial pressure relative to its corresponding liquid-phase vapor pressure, which depends only on temperature. In order to simplify the problem, we assume the fuel or oxidizer condenses below a condensation temperature T_c constant throughout the range of pressures encountered in the nozzle. This condition imposes a restriction on the total enthalpy of the reservoir. This condition is much more stringent for liquid fuels such as Jet A or JP10 which condense below 450 K than for hydrogen, for which the oxygen in the air will condense first at 90 K.

Another issue is the location of fuel injection. The flow at the nozzle throat is hot and the fuel-air mixture must be prevented from pre-igniting before the conditions for the stabilized detonation are encountered.[4] It is better to locate the fuel injection system further downstream from the throat where the flow is cooler. However, in practice, there is a compromise with the length necessary for supersonic mixing of the fuel and air. The influence of the upstream conditions can be studied by considering the simple criterion that $T_4 < T_{ign}$, assuming the auto-ignition temperature of the fuel-air mixture is independent of pressure in the range considered.

The temperature upstream of the detonation wave T_4 has to be above the condensation temperature T_c and below the fuel-air mixture auto-ignition temperature T_{ign}: $T_c < T_4 < T_{ign}$. This condition can be solved using Eq. 1, yielding a criterion for the upstream total temperature

$$f(T_c) < T_{t0} < f(T_{ign}) \quad (2)$$

where $f(T)$ is defined by

$$f(T) = \frac{\gamma+1}{2}T + \frac{\gamma^2-1}{2}\frac{fQ}{C_p}\left(1+\sqrt{1+\frac{2}{\gamma+1}\frac{C_pT}{fQ}}\right). \quad (3)$$

We applied this criterion to hydrogen-air mixtures, for which T_c=90 K and T_{ign}=793 K.[12] It is then possible to determine the values of the reservoir total temperature for which a stabilized detonation is obtained, as a function of the fuel-air mass ratio (or, equivalently, the total heat release per unit time). Fig. 2 shows the allowable domain. Below the lower curve, T_{t0} is too low and condensation of the oxygen occurs inside the nozzle; above the upper curve, T_{t0} is too large and the fuel and the air will start combusting ahead of the detonation. Comparisons with data obtained by Nicholls et al.[10] in their hydrogen-air open jet experiments resulted in good agreement for cases where they observed stabilized detonations or pre-ignition of the mixture.

The restrictions on the allowable domain for liquid hydrocarbon fuels are more severe since fuel condensation occurs at much higher temperatures, and the auto-ignition temperature is lower than that of hydrogen. For example, JP10 has a boiling point[13] of 455 K

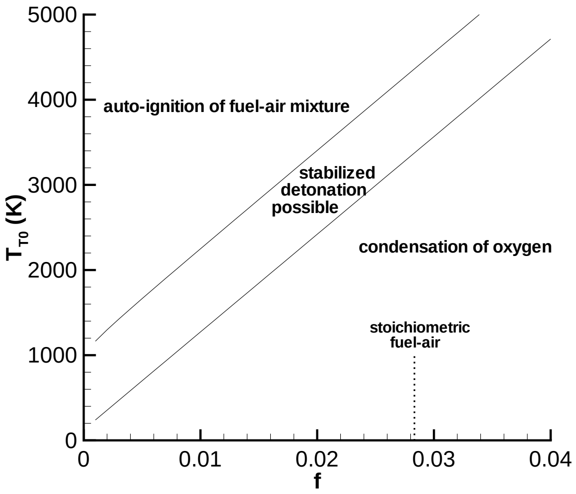

Fig. 2 Allowable domain for the generation of a stabilized detonation in hydrogen-air as a function of the reservoir total temperature T_{t0} and the fuel-air mass ratio f. Q=120.9 MJ/kg for hydrogen.

and an auto-ignition temperature[13] of 518 K. Therefore, a much smaller region exists where stabilized detonations can be established using liquid hydrocarbon fuels. However, detonations can be obtained with liquid hydrocarbon fuels at temperatures below their boiling point as long as the vapor pressure of the fuel at the temperature considered is high enough. For example, for a stoichiometric mixture of JP-10 and air at atmospheric pressure, the temperature has to be above 330 K for complete vaporization of the injected fuel.[14] Detonations in low vapor pressure liquid fuel aerosols are possible, but harder to establish because aerosols have much higher ignition energies and larger reaction zones than vapor phase mixtures. Hence, condensation (even partial) of the fuel can be very penalizing for the practical establishment of a stabilized detonation wave. For liquid hydrocarbon fuels, the threshold based on the boiling point will be used as a conservative zero-order criterion.

Detonation-related limitations

Up to now, we have modeled detonations as hydrodynamic discontinuities. The simplest model that includes chemical kinetics consists of a shock wave followed by a reaction zone referred to as the ZND model. In this model, the leading shock front is followed by an induction zone, through which the thermodynamic variables remain relatively constant while free radicals, such as OH, are produced. Significant energy release occurs at the end of the induction zone and corresponds to a rapid rise in temperature and a decrease in pressure accompanied by the formation of the major products. The length scale associated with the induction zone is called the reaction zone length Δ. Another length scale associated with detonations is the cell width λ. The cell width is a characteristic length scale corresponding to the intrinsic instability and the structure of propagating detonation waves. Attempts to correlate the cell width with the reaction zone length showed that λ is between 10 and 50 times Δ for stoichiometric mixtures and between 2 and 100 times Δ for off-stoichiometric mixtures.[15]

Simulations of steady, one-dimensional detonations were performed with a code developed by Shepherd[15] based on a standard gas-phase chemical kinetics package.[16] The code solves the one-dimensional, steady reactive Euler equations of the ZND model. The chemical reaction model of Konnov[17] and standard thermochemistry were used to calculate reaction zone lengths for hydrogen-air at various initial conditions. Validation of this mechanism against shock tube induction time data is given in Schultz and Shepherd.[18] The reaction zone length was calculated as the distance from the leading shock to the point of maximum temperature gradient. For hydrogen-air mixtures, the scaling relationship $\lambda = 50\Delta$ gave the best agreement with experimental cell size data.[19–21]

Cell widths were also estimated for JP10-air mixtures, since JP10 is a fuel of interest to propulsion applications because of its high energy density. The reaction zone lengths for JP10-air mixtures were estimated from the ignition time correlation of Davidson et al.,[22] who carried out shock tube measurements of JP10 ignition. The correlation they obtained is

$$\tau_{ign} = 3.06 \cdot 10^{-13}\ P^{-0.56}\ X_{O_2}^{-1}\ \phi^{0.29}\ e^{52150/RT}\ . \quad (4)$$

The ignition time was multiplied by the post-shock velocity, which was calculated[23] for a non-reactive shock with realistic thermodynamic properties, to obtain the reaction zone length. The relationship $\lambda = 10\Delta$ gave good agreement with the JP10 cell width data of Austin and Shepherd[14] and is used to predict JP10-air cell widths.

Limitations on detonation chamber dimensions

The characteristic detonation length scales, the reaction zone length and the cell width, impose constraints on the geometry and size of the combustor. The usual rule of thumb for propagating detonations is that the channel width has to be greater than the detonation cell width for the detonation to propagate. The limit for detonation propagation in cylindrical tubes of diameter d is usually taken to be determined by the criterion $\lambda \approx \pi d$[24,25] and for two-dimensional planar channels of width w by $\lambda = w$.[25] The problem of detonability limits for propagating detonations does not have a single definitive answer and, at present, there are no data at all for stabilized detonations. For the purposes of the present study, we adopt the criterion $\lambda = d$ as the detonability limit for a stabilized detonation wave in a given channel of dimension d.

It has been claimed[1,2,26] that using detonations in ramjet-like engines would enable reductions in the length of the combustor. In practice, the CJ state has to be achieved inside the combustor for maximum efficiency and to isolate the detonation from potential perturbations in the flow downstream of the combustor. If the combustor is too short, the combustion process inside the combustor is incomplete and part of the energy released by the combustion is lost to the surroundings. The detonation can also become unstable if flow perturbations penetrate the subsonic region between the detonation front and the CJ plane. Hence, the location of the CJ surface is critical for the design of the detonation chamber. Experiments[27] using different techniques resulted in measurements of $6\lambda - 17\lambda$ and $1.5\lambda - 5\lambda$ for the distance between the detonation front and the CJ surface. Other studies[28,29] reported measurements of $3\lambda - 7\lambda$ and $0.2\lambda - 0.6\lambda$. Although there is a wide range of values for the sonic surface, it apparently lies within 5λ of the front for propagating detonation waves and no measurements have been made for stabilized waves. For the purposes of the present study, we propose to use a criterion for the minimum length of the detonation chamber $L > 5\lambda$.

In configurations close to the detonability limits, it may not be necessary to have $\lambda < d$ if viscous effects can be used to stabilize the flow. The detonation velocity can be substantially lower than the CJ velocity (as low as 50% of U_{CJ}) for detonation propagation in small-diameter tubes or at low pressures.[30,31] These situations may significantly extend the regime of operation of a steady detonation engine. The idea of being able to stabilize a detonation wave at a velocity lower than U_{CJ} is attractive, since it reduces the requirements on the total temperature of the flow. However, propagating detonations at near-limit conditions generally have an unstable behavior,[30] characterized by a strong oscillation of the detonation velocity, which makes them totally inadequate for detonation stabilization. The possibility of stabilized detonations with velocities substantially less than the CJ value is highly speculative and will not be considered any further. In the present study, we adopt the requirement $u \geq U_{CJ}$ for stabilizing a detonation in a combustor.

Application to hydrogen-air and JP10-air stabilized detonations

The previous criteria impose some severe restrictions on the dimensions of the detonation chamber of a steady detonation engine. Table 1 lists the minimum requirements for the diameter and length of a detonation chamber at various initial conditions. The minimum temperature chosen for JP10 was 350 K based on vapor pressure considerations.[14] The minimum dimensions vary by several orders of magnitude with equivalence ratio and initial pressure. Typical air-breathing engines run at an equivalence ratio substantially less than one in order to limit the maximum temperature in the combustor due to material considerations. The same approach with a steady detonation engine leads to impractical minimum dimensions when the equivalence ratio is decreased to 0.5. The claim that using steady detonations in propulsion devices might allow a reduction of the combustor length is not justified, as the detonation chamber length has to be at least five times the minimum chamber diameter. Finally, the difficulties associated with the use of liquid fuels, which are insensitive to detonation and have large cell sizes, are highlighted in Table 1.

Fuel	T_0(K)	P_0(atm)	ϕ	d(mm)	L(mm)
H_2	300	0.1	1	84.9	424.5
H_2	300	1	1	8.9	44.5
H_2	300	10	1	11.2	56
H_2	300	0.1	0.5	214.5	1072.5
H_2	300	1	0.5	271.5	1357.5
H_2	300	10	0.5	289	1445
H_2	500	0.1	1	98.1	490.5
H_2	500	1	1	10.4	52
H_2	500	10	1	13.1	65.5
H_2	500	0.1	0.5	213	1065
H_2	500	1	0.5	215.2	1076
H_2	500	10	0.5	247.9	1239.5
JP10	350	0.1	1	369.4	1847
JP10	350	1	1	55.2	276
JP10	350	10	1	9.2	46
JP10	350	0.1	0.5	8185	40925
JP10	350	1	0.5	2100	10500
JP10	350	10	0.5	561.5	2807.5
JP10	500	0.1	1	165.8	829
JP10	500	1	1	25.5	127.5
JP10	500	10	1	4.3	215
JP10	500	0.1	0.5	1557.5	7787.5
JP10	500	1	0.5	384.8	1924
JP10	500	10	0.5	100.8	504

Table 1 Minimum detonation chamber length and diameter for a range of initial conditions for hydrogen- and JP10-air.

Detonation ramjet

A detonation ramjet, or dramjet, is a steady propulsive device using the same principle as a ramjet, except that the combustion takes place in the combustor in the form of a steady detonation wave instead of a bluff-body stabilized flame. First, we will briefly describe the ideal ramjet since it has many components in common with the dramjet, and this will be used as a performance standard. Second, we will discuss the portions of the dramjet model which are different from the ramjet. Third, the performance of both engines will be compared. Finally, limitations are considered due to detonation stabilization requirements, ignition limits, and fuel and oxidizer properties.

Ramjet

A standard ramjet consists of an inlet diffuser through which the air flow is decelerated to a low subsonic Mach number and mixed with the fuel, a combustor where the mixture is burned, and an exit nozzle through which the hot products are expelled due to the pressure rise in the diffuser.[32] The simplest performance model of an ideal ramjet is derived assuming steady, inviscid, and adiabatic flow of an ideal gas. Products and reactants are assumed to have the same heat capacity and γ. The compression and expansion processes are assumed to be isentropic and the combustion process takes place at constant pressure and very low Mach number. The flow through the exit nozzle is assumed to be isentropically expanded to ambient pressure and the fuel-air mass ratio $f \ll 1$. These assumptions are, of course, not realistic due to the presence of irreversible processes such as shocks, mixing, wall friction, and heat transfer. This model also does not take into account the dissociation of the combustion products. It is possible to make the model much more realistic but for the present purposes, these idealizations are adequate since we are primarily interested in performance comparisons rather than absolute performance. The performance characteristics of an ideal ramjet are derived assuming a maximum temperature T_{max} at the combustor outlet due to material limitations.[32] This maximum temperature implies a limitation on the total temperature at the combustor outlet T_{t5} since it is the temperature of a stationary material element in the flow.

Detonation ramjet

A detonation ramjet has to accommodate a stationary detonation wave in the combustor. The flow must be accelerated or slowed down to a velocity higher than or equal to the CJ detonation velocity. For flow velocities higher than U_{CJ}, overdriven detonation waves could be stabilized. However, overdriven waves are not desirable in order to avoid excessive total pressure loss across the detonation. We consider only CJ detonation waves. A dramjet has to include a general converging-diverging nozzle between the inlet diffuser and the combustor inlet in order to bring the flow to the CJ velocity. It will be shown later that a converging inlet section is actually more appropriate. A schematic of a dramjet is given in Fig. 3.

In our performance analysis of the dramjet, we make the same assumptions as for the ramjet, except for the combustion process. Instead, the stabilization condition for the detonation wave (Eq. 1) and the one-γ model for the detonation are used. The detonation wave is assumed to be stable with respect to flow perturbations. All these assumptions are used to derive simple performance estimates of an ideal dramjet, which can be used as the detonative combustion analog of the ideal ramjet. We apply a limitation on the total temperature at the combustor outlet similar to the ramjet case. The flow evolves isentropically through the inlet and the converging-diverging nozzle. Hence, $T_{t0} = T_{t2} = T_{t4}$ and $P_{t0} = P_{t2} = P_{t4}$. The detonation stabilization condition is that the flow at station 4 has a Mach number $M_4 = M_{CJ}$.

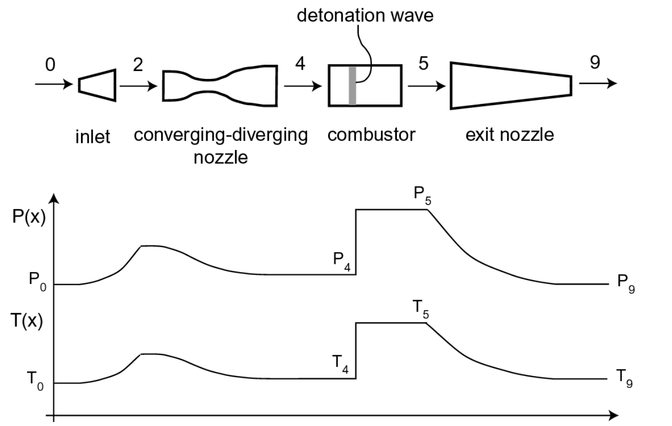

Fig. 3 Schematic representation of a detonation ramjet (or dramjet). The pressure and temperature profiles through the engine are shown.

The fuel-air ratio f is determined by the maximum temperature condition. The flow properties at the combustor outlet are dictated by the CJ conditions. The flow through the exit nozzle is considered isentropic and the exit velocity u_9 can be calculated assuming the flow at the nozzle exit is pressure-matched

$$u_9 = \sqrt{2C_pT_{max}\left[1 - \frac{2}{\gamma+1}\frac{T_0}{T_4}\left(\frac{\gamma+1}{1+\frac{2\gamma}{\gamma-1}(\frac{T_{t0}}{T_4}-1)}\right)^{\frac{\gamma-1}{\gamma}}\right]} \tag{5}$$

where T_4 is given by Eq. 1. The values of the various performance parameters can be deduced from the value of u_9.

Performance comparison

The specific thrust, thrust specific fuel consumption ($TSFC$), and efficiencies of the dramjet were calculated for a set of initial conditions corresponding to flight at 10,000 m altitude using a fuel of heat release per unit mass Q=45 MJ/kg (typical of hydrocarbon fuels) and a maximum allowable temperature in the combustor T_{max}=2500 K. These parameters are compared to their ramjet analogs in Figs. 4, 5, and 7. The $TSFC$ numbers given in Fig. 5 are obtained using the one-γ model and the assumptions discussed before and are therefore very optimistic figures.

The ramjet performance at subsonic flight Mach numbers is poor due to the small amount of ram compression at low Mach numbers. As the flight Mach number increases, the overall efficiency of the ramjet increases (Fig. 7) due to the higher theoretical ram compression. However, the specific thrust decreases for Mach numbers higher than 3 because of the lower amount of fuel injected due to the limitation on the total temperature at the combustor outlet. The dramjet does not produce any thrust below a flight Mach number of about 5 for the initial conditions considered due to the stabilization condition for a detonation wave. The freestream Mach number is almost always higher than the CJ Mach number when the detonation wave is stabilized (except for $M_0 < 5.1$). This means that the supersonic flow between stations 2 and 4 has to undergo a deceleration through the inlet and only a converging section is required, unlike the situation depicted in Fig. 3. The pressure and temperature would then increase continuously from station 0 to station 4.

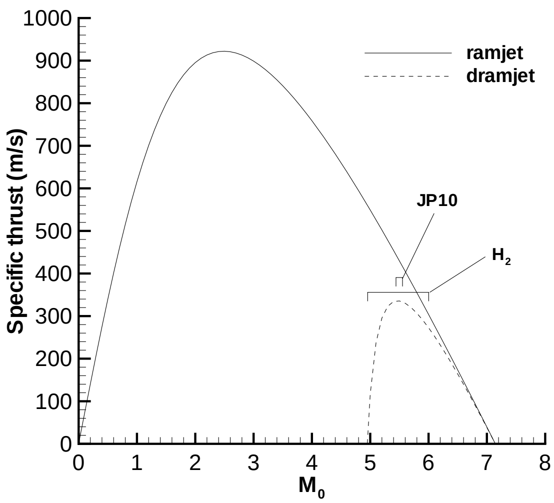

Fig. 4 Specific thrust of ramjet and dramjet. T_0=223 K, P_0=0.261 atm, Q=45 MJ/kg, T_{max}=2500 K. The limits for effective detonation stabilization are shown for hydrogen and JP10.

The specific thrust for the dramjet (Fig. 4) shows a maximum near $M_0 = 5.5$. The performance of the dramjet then decreases with increasing M_0 due to the maximum temperature limitation in the combustor. As M_0 approaches its upper limit, the amount of fuel injected decreases and the CJ Mach number approaches 1. The combustion process becomes, in theory, closer to a constant-pressure heat addition as in the case of the ramjet, which explains why the two curves match at high Mach numbers. However, as the amount of fuel is reduced, the CJ Mach number will decrease and, therefore, the reaction zone length will strongly increase until it exceeds the physical dimension of the combustor, and incomplete reaction is obtained in the combustor. Below a minimum fuel-air ratio, the mixture will not be flammable and combustion will not be obtained. For this reason, the actual maximum flight Mach number will be lower than the ideal value.

As the flight Mach number decreases, the performance of the dramjet drops sharply. This can be

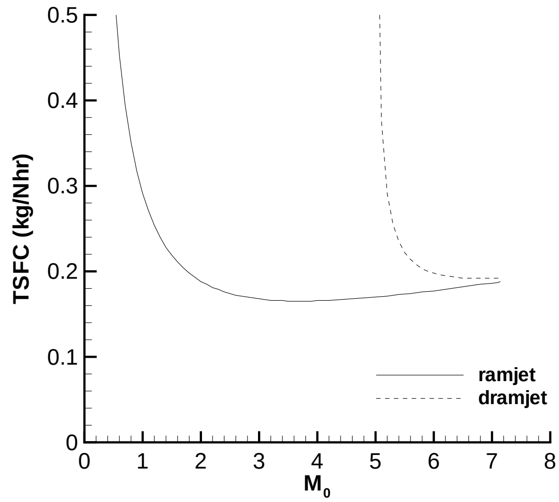

Fig. 5 Thrust specific fuel consumption of ramjet and dramjet. T_0=223 K, P_0=0.261 atm, Q =45 MJ/kg, T_{max}=2500 K.

explained by the very substantial total pressure loss across a detonation wave. The total pressure ratio across a CJ detonation was computed as a function of the CJ Mach number and is shown in Fig. 6. The total pressure ratio decreases rapidly as M_{CJ} increases. CJ detonation waves have very high total pressure losses; for example, the total pressure loss across a detonation wave with $M_{CJ} = 4$ is 88%, and the total pressure loss across a wave with $M_{CJ} = 5$ is greater than 94%. Total pressure losses are penalizing for air-breathing engines because the exit velocity u_9 and the thrust F decrease with decreasing total pressure P_{t5}. The dramjet performance is strongly penalized compared to the ideal ramjet, for which there is negligible total pressure loss across the combustor.

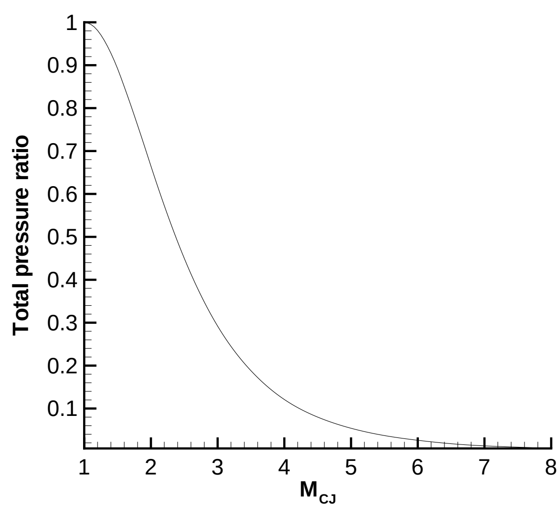

Fig. 6 Total pressure ratio across a Chapman-Jouguet detonation wave. γ=1.4.

As seen in Fig. 5, the thrust specific fuel consumption increases sharply for both engines as the flight Mach number decreases due to the decreasing specific thrust while the fuel consumption rate remains finite. At higher Mach numbers, the $TSFC$ remains finite as both the fuel-air mass ratio and the specific thrust decrease, and the process approaches constant-pressure combustion. The thermal efficiency of the ramjet and the dramjet increases as M_0 increases. The freestream total pressure increases with M_0, and adding heat at higher total pressure is thermally more efficient since the exit velocity u_9 is higher. The overall efficiency follows a similar behavior, showing that both engines are more efficient at higher flight speeds. A more realistic approach would take into account irreversible processes such as inlet losses. These losses would, in general, increase with increasing Mach number, making for a more rapid decrease in performance at high Mach numbers for both ramjet and dramjet. However, our goal here is to compare ideal models whose characteristics can be used as performance goals of realistic engines.

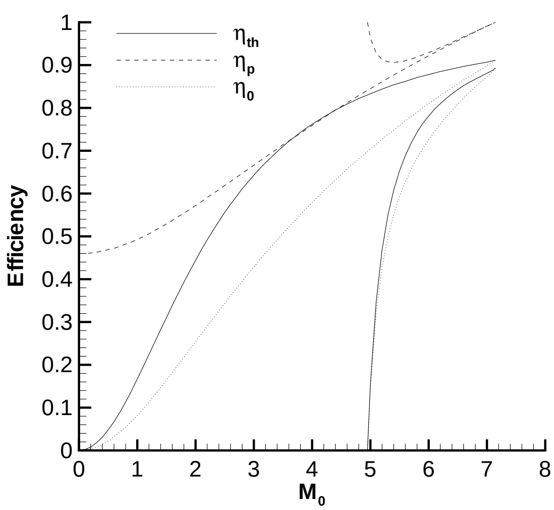

Fig. 7 Thermal, propulsive, and overall efficiencies of ramjet and dramjet. T_0=223 K, P_0=0.261 atm, Q=45 MJ/kg, T_{max}=2500 K. The dramjet curves are the ones extending only from M_0=5 to 7.

Dramjet limitations

Effects such as fuel condensation, mixture pre-ignition, and reaction zone thickness have to be considered when looking at the dramjet performance curves.

As M_0 gets close to the lower limit of the dramjet thrust-producing range, such effects as fuel or oxidizer condensation are going to take place as described in the previous section on standing normal detonation waves. The static temperature at the nozzle outlet is, in general, higher than the freestream temperature, but it is lower for $M_0 < 5.1$. Although this is not an issue for a fuel such as hydrogen, it is definitely a problem for liquid hydrocarbon fuels which have a boiling point above 450 K. Near the upper limit of the thrust-producing range of M_0, the static temperature T_4 becomes very high because of the strong flow deceleration from a high freestream Mach number to a low M_{CJ} due to low fuel input. Pre-ignition of the fuel-air mixture is then expected for $M_0 > 6$. For hydrogen, $5 < M_0 < 6$ is necessary for steady detonation generation. For a representative liquid hydrocarbon fuel such as JP10, $5.45 < M_0 < 5.55$ is necessary for detonation stabilization. These limits are shown in Fig. 4 for mixtures with hydrogen and JP10. If, instead of using the condensation temperature criterion for JP10, we consider vapor pressure requirements so that the amount of fuel injected is totally vaporized, then $5.25 < M_0 < 5.55$ is required for effective detonation stabilization. The difficulties associated with generating steady detonations using liquid hydrocarbon fuels are readily apparent.

Both the ramjet and dramjet have been modeled so far without considering any total pressure loss other than across the detonation wave. There are obviously total pressure losses across the inlet during supersonic flight, but both engines would suffer a similar decrease in performance. However, the performance of a realistic dramjet is handicapped compared to the ramjet due to the mixing requirements ahead of the combustion chamber. In a ramjet, mixing and combustion occur at $M \ll 1$ where losses are minimal. In a dramjet, mixing has to take place at supersonic speeds. Supersonic mixing is associated with total pressure losses which penalize the performance. Supersonic mixing studies[1] have predicted total pressure losses on the order of $10-40\%$ that could result in thrust losses[33] of $30-50\%$. This effect could have a significant impact on the dramjet performance compared to the ramjet.

The limitations associated with detonation reaction zone structure impose further constraints on the performance of the dramjet. The corresponding cell widths for the mixtures can be estimated[16,22] at the flight conditions considered. The initial conditions upstream of the detonation were given by T_4 and P_4. Fig. 8 displays the cell width estimates as a function of the flight Mach number. The mixtures are all very lean and the fuel-air mass ratio decreases with increasing M_0 because of the maximum combustor outlet temperature condition. The pressure P_4 and temperature T_4 increase very rapidly with increasing M_0. The cell width is sensitive to the changes in pressure and temperature and decreases by many orders of magnitude with increasing Mach number. For conventional applications, the corresponding cell width λ probably has to be below 1 m which requires that $M_0 > 5.6$ for both fuels. The range of applicability of hydrogen-fueled dramjets is now reduced to $5.6 < M_0 < 6$. For a JP10-fueled dramjet, there is no practical range of applicability due to the lower auto-ignition temperature of the fuel. This illustrates clearly the strong influence of the fuel properties and the characteristic detonation length scales on the use of detonations in steady-flow engines.

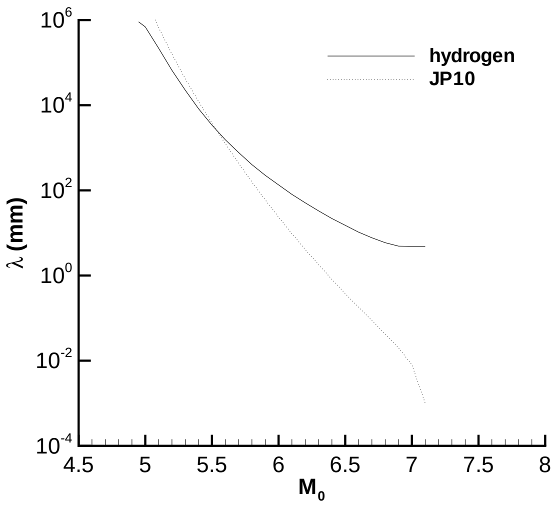

Fig. 8 Cell width λ versus flight Mach number M_0 for a dramjet operating with hydrogen and JP10. T_0=223 K, P_0=0.261 atm, T_{max}=2500 K.

Detonation turbojet

The turbojet engine includes an additional compressor, characterized by a compression ratio π_c, and a turbine driving the compressor. The turbine blades are highly sensitive to high temperatures, and a limitation is usually placed on the temperature at the combustor outlet due to material considerations. The ideal turbojet is analyzed in the same fashion as the ramjet, with the addition of the power balance between the compressor and the turbine:

$$(1 + f)\dot{m}(h_{t8} - h_{t5}) = \dot{m}(h_{t4} - h_{t2}) \ . \qquad (6)$$

Assuming $f \ll 1$ and that the specific heat capacity of the products is the same as that of the reactants, the equation simplifies to $T_{t8} = T_{t5} + T_{t4} - T_{t2}$. After its passage through the turbine, the flow is expanded through an exit nozzle into the atmosphere.

The detonation turbojet, or turbodet, has the same components as the turbojet engine, except that it requires an additional nozzle between the compressor and the combustor in order to accelerate the flow to the CJ velocity, as depicted in Fig. 9. Unlike the ramjet, the turbodet must have a converging-diverging nozzle since the flow exiting the compressor has a subsonic Mach number and must be accelerated to a supersonic velocity. The sonic flow exiting the combustor has to be decelerated before entering the turbine in order to minimize losses associated with shock waves.

The specific thrust, thrust specific fuel consumption, and efficiencies of the turbojet and turbodet engines

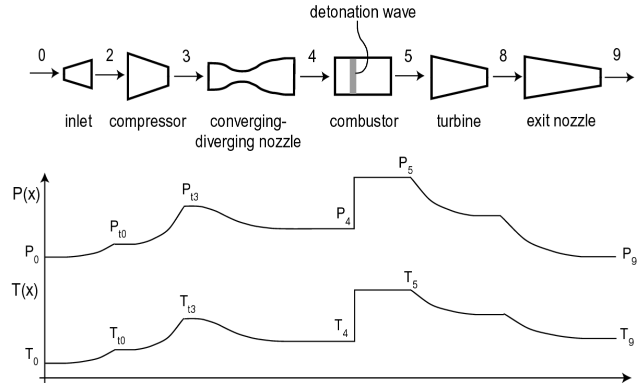

Fig. 9 Schematic of a detonation turbojet, including the variation of pressure and temperature across the engine.

are plotted on Figs. 10, 11, and 12, respectively. The turbojet engine shows a high specific thrust at low Mach numbers due to the work supplied by the compressor. Its specific thrust then decreases due to the limitation on the turbine inlet temperature. Even at very low Mach numbers, the temperature at the combustor inlet is already high due to the work input by the compressor. On the other hand, as can be seen in Fig. 10, the turbodet engine shows relatively poor performance compared to the turbojet. It does not produce thrust below a Mach number of 1.75 for the case considered here (the value of the limiting Mach number depends on the compression ratio at fixed flight conditions) due to the detonation wave stabilization conditions. The drastic total pressure loss across the steady detonation causes the specific thrust to fall off at lower flight Mach numbers while the maximum temperature condition causes its decrease at higher flight Mach numbers. The limits corresponding to condensation and pre-ignition conditions are pointed out on Fig. 10 for hydrogen and JP10. Hydrogen can be used for $1.75 < M_0 < 2.6$, and JP10 for $2.2 < M_0 < 2.3$ using the condensation temperature criterion, or $2 < M_0 < 2.3$ using vapor pressure considerations. The $TSFC$ of the turbojet, Fig. 11, is about 0.9 kg/N.hr and does not vary much with M_0. The $TSFC$ of the turbodet is higher at all Mach numbers and peaks at low values of the thrust-producing range because the specific thrust vanishes. The thermal efficiency of the turbojet, on Fig. 12, increases with the flight Mach number due to the higher efficiency of heat addition at higher stagnation conditions but already has a high value at zero Mach number due to the compression work. The thermal and overall efficiencies of the turbodet increase with M_0 but have a lower value than those of the turbojet.

Cell width estimates corresponding to the flight conditions are very large due to the low fuel input of a temperature-limited turbodet engine. The scaled cell widths are less than 1 m only for $M_0 > 2.8$ for both fuels. However, the static temperature upstream of the

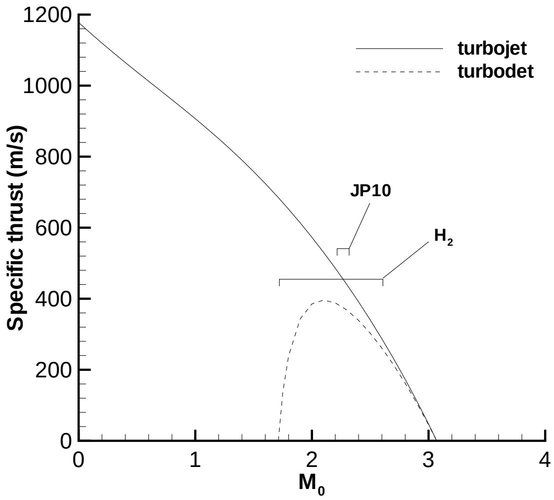

Fig. 10 Specific thrust of turbojet and turbodet engines. π_c=30, T_0=223 K, P_0=0.261 atm, Q=45 MJ/kg, T_{max}=1700 K. The limits for effective detonation stabilization are shown for hydrogen and JP10.

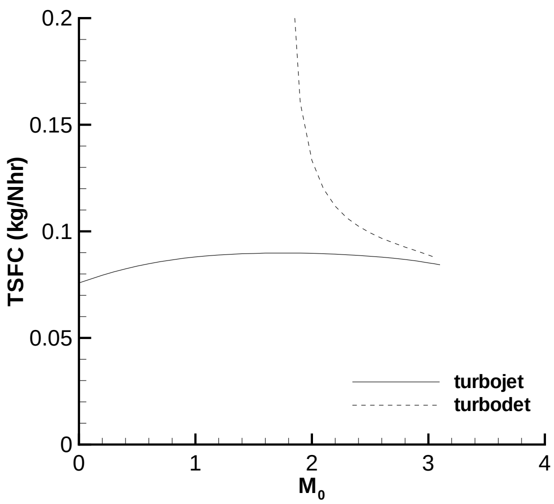

Fig. 11 Thrust specific fuel consumption of turbojet and turbodet engines. π_c=30, T_0=223 K, P_0=0.261 atm, Q=45 MJ/kg, T_{max}=1700 K.

detonation T_4 is already higher than the auto-ignition temperature of the mixture for this case for both hydrogen and JP10. Consequently, there is no useful range of Mach numbers for practical applications of the turbodet engine.

Conclusions

The performance of steady detonation engines was estimated and compared with the ideal ramjet and turbojet models. A normal detonation wave ramjet does not appear as an attractive alternative to the conventional ramjet. The performance of the dramjet suffers

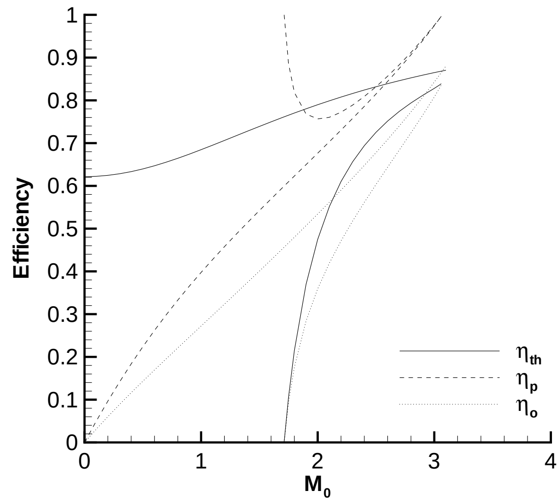

Fig. 12 Thermal, propulsive, and overall efficiencies of turbojet and turbodet engines. π_c=30, T_0=223 K, P_0=0.261 atm, Q=45 MJ/kg, T_{max}=1700 K. The turbodet curves are the ones extending only from M_0=1.75 to 3.1.

from two problems: the stabilization of the detonation wave, which reduces the thrust-producing range (between M_0=5 and 7 for flight conditions at 10,000 m) and the drastic total pressure loss across a normal detonation wave. Moreover, the use of stabilized detonations imposes an additional set of constraints related to condensation and pre-ignition phenomena and to the characteristic length scales of the problem. All these considerations strongly reduce the useful operating range of a dramjet, which is $5.6 < M_0 < 6$ for a hydrogen-fueled dramjet. Liquid hydrocarbon fuels such as JP10 have an even smaller range of application due to their lower auto-ignition temperature. The detonation turbojet suffers from the same drawbacks and generates thrust only for $1.75 < M_0 < 3.1$ for a compression ratio of 30. Moreover, if the various limitations associated with detonations are taken into account, it turns out that there is no Mach number for which a steady detonation can effectively be stabilized in a reasonable-size combustor without pre-ignition. This result may vary with the value of π_c, but it shows that the presence of a compressor and a turbine in the turbodet does not contribute to any performance gain over the dramjet.

In order to calculate the performance of steady detonation engines from their thermodynamic cycle, it is necessary to correctly treat the kinetic energy of the flow. A correct cycle analysis for high-speed propulsion systems has to use correct total pressure and energy balance statements based on an open-system control volume approach that includes the kinetic energy terms. This requirement translates, in practice, into the detonation stabilization condition and the non-zero (sonic) flow velocity just downstream of the detonation. It is not possible to evaluate these specific cycles on a purely static thermodynamic basis. Unlike the ramjet or turbojet cases, the dramjet and turbodet performance depends directly on the specific details of the engine.

The strong penalty created by the continuous presence of the detonation in the flow path and the problems associated with practical detonation stabilization are difficult obstacles to the feasiblity of steady detonation engines. The implications of our analysis are that using a detonation wave in a steady engine is not practical, but this clearly does not apply to the unsteady case. In fact, it suggests that unsteady detonation wave engines, such as the pulse detonation engine, are the only useful way to apply detonations to propulsion.

Acknowledgments

This work was supported by a Stanford University Contract PY-1905 under Dept. of Navy Grant No. N00014-02-1-0589 *"Pulse Detonation Engines: Initiation, Propagation, and Performance"*.

References

[1]Dunlap, R., Brehm, R. L., and Nicholls, J. A., "A Preliminary Study of the Application of Steady-State Detonative Combustion to a Reaction Engine," *Jet Propulsion*, Vol. 28, No. 7, 1958, pp. 451–456.

[2]Sargent, W. H. and Gross, R. A., "Detonation Wave Hypersonic Ramjet," *ARS Journal*, Vol. 30, No. 6, 1960, pp. 543–549.

[3]Dabora, E. K., "Status of Gaseous Detonation Waves and Their Role in Propulsion," *Fall Technical Meeting of the Eastern States Section of the Combustion Institute*, Pittsburgh, PA, 1994, pp. 11–18.

[4]Rubins, P. M. and Bauer, R. C., "Review of Shock-Induced Supersonic Combustion Research and Hypersonic Applications," *Journal of Propulsion and Power*, Vol. 10, No. 5, 1994, pp. 593–601.

[5]Cambier, J.-L., Adelman, H., and Menees, G., "Numerical Simulations of an Oblique Detonation Wave Engine," AIAA Paper 88-0063, 1988.

[6]Menees, G. P., Adelman, H. G., Cambier, J.-L., and Bowles, J. V., "Wave Combustors for Trans-Atmospheric Vehicles," *Journal of Propulsion and Power*, Vol. 8, No. 3, 1992, pp. 709–713.

[7]Nicholls, J. A. and Dabora, E. K., "Recent Results on Standing Detonation Waves," *Proceedings of the 8th Symposium on Combustion*, 1962, pp. 644–655.

[8]Gross, R. A. and Chinitz, W., "A Study of Supersonic Combustion," *Journal of the Aero/Space Sciences*, Vol. 27, No. 7, 1960, pp. 517–524.

[9]Shepherd, J. E., "Detonation Waves and Propulsion," *Combustion in High Speed Flows*, Kluwer Academic Publishers, 1994, pp. 373–420.

[10]Nicholls, J. A., Dabora, E. K., and Gealer, R. L., "Studies in Connection with Stabilized Gaseous Detonation Waves," *Proceedings of the 7th Symposium on Combustion*, 1959, pp. 766–772.

[11]Thompson, P. A., *Compressible Fluid Dynamics*, Advanced Engineering Series, Rensselaer Polytechnic Institute, pp. 347–359, 1988.

[12]Kuchta, J. M., "Investigation of Fire and Explosion Accidents in the Chemical, Mining, and Fuel-Related Industries - A Manual," Tech. rep., Bureau of Mines, United States Department of the Interior, 1985, Bulletin 680.

[13]*Aviation Fuel Properties*, Coordinating Research Council, 1983, CRC Report No. 530.

[14]Austin, J. M. and Shepherd, J. E., "Detonations in Hydrocarbon Fuel Blends," *accepted to Combustion and Flame*, 2002.

[15]Shepherd, J., "Chemical Kinetics of Hydrogen-Air-Diluent Detonations," *Progress in Aeronautics and Astronautics*, Vol. 106, 1986, pp. 263–293.

[16]Kee, R., Rupley, F., and Miller, J., "Chemkin II: A Fortran Chemical Kinetics Package for the Analysis of Gas-Phase Chemical Kinetics," Tech. Rep. SAND89-8009, Sandia National Laboratory, 1989.

[17]Konnov, A. A., "Detailed Reaction Mechanism for Small Hydrocarbons Combustion," Release 0.4. http://www.homepages.vub.ac.be/∼akonnov.

[18]Schultz, E. and Shepherd, J. E., "Validation of Detailed Reaction Mechanisms for Detonation Simulation," Tech. rep., GALCIT, 2000, FM 99-5.

[19]Stamps, D. W. and Tieszen, S. R., "The Influence of Initial Pressure and Temperature on Hydrogen-Air-Diluent Detonations," *Combustion and Flame*, Vol. 83, 1991, pp. 353–364.

[20]Ciccarelli, G., Ginsberg, T., Boccio, J., Economos, C., Sato, K., and Kinashita, M., "Detonation Cell Size Measurements and Predictions in Hydrogen-Air-Steam Mixtures at Elevated Temperatures," *Combustion and Flame*, Vol. 99, 1994, pp. 212–220.

[21]Guirao, C. M., Knystautas, R., Lee, J., Benedick, W., and Berman, M., "Hydrogen-Air Detonations," *Proceedings of the 19th Symposium on Combustion*, 1982, pp. 583–590.

[22]Davidson, D. F., Horning, D. C., Herbon, J. T., and Hanson, R. K., "Shock Tube Measurements of JP-10 Ignition," *Proceedings of the 28th Symposium on Combustion*, 2000, pp. 1687–1692.

[23]Reynolds, W., "The Element Potential Method for Chemical Equilibrium Analysis: Implementation in the Interactive Program STANJAN," Tech. rep., Mechanical Engineering Department, Stanford University, 1986.

[24]Dupre, G., Knystautas, R., and Lee, J. H., "Near-Limit Propagation of Detonation in Tubes," *Progress in Aeronautics and Astronautics*, Vol. 106, 1986, pp. 244–259.

[25]Lee, J. H. S., "Dynamic Parameters of Gaseous Detonations," *Annual Review of Fluid Mechanics*, Vol. 16, 1984, pp. 311–336.

[26]Dabora, E. K. and Broda, J. C., "Standing Normal Detonations and Oblique Detonations for Propulsion," AIAA Paper 93-2325, 1993.

[27]Vasiliev, A. A., Gavrilenko, T. P., and Topchian, M. E., "On the Chapman-Jouguet Surface in Multi-Headed Gaseous Detonations," *Astronautica Acta*, Vol. 17, 1972, pp. 499–502.

[28]Edwards, D. H., Jones, A. T., and Phillips, D. E., "The Location of the Chapman-Jouguet Surface in a Multi-Headed Detonation Wave," *Journal of Physics D: Applied Physics*, Vol. 9, 1976, pp. 1331–1342.

[29]Weber, M., Olivier, H., Grönig, H., and Biegling, J., "On the Location of the Chapman-Jouguet Surface in Gaseous Detonations Close to the Limit of Propagation," *Proceedings of the 18th ICDERS*, 2001.

[30]Lee, J. J., Dupre, G., Knystautas, R., and Lee, J. H., "Doppler Interferometry Study of Unstable Detonations," *Shock Waves*, Vol. 5, 1995, pp. 175–181.

[31]Manzhalei, V. I., "Low-Velocity Detonation Limits of Gaseous Mixtures," *Combustion, Explosion and Shock Waves*, Vol. 35, 1999, pp. 296–302.

[32]Hill, P. G. and Peterson, C. R., *Mechanics and Thermodynamics of Propulsion*, Addison-Wesley, Second Edition, 1992.

[33]Papamoschou, D., "Thrust Loss Due to Supersonic Mixing," *Journal of Propulsion and Power*, Vol. 10, 1994, pp. 804–809.