Theoretical Analysis of a Rotating Two Phase Detonation in a Rocket Motor

by

I-wu Shen
T. C. Adamson, Jr.

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R. J. Priem, Project Manager

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THE UNIVERSITY OF MICHIGAN
Department of Aerospace Engineering
Ann Arbor, Michigan
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ABSTRACT

Tangential mode, non-linear wave motion in a liquid propellant rocket engine is studied, using a two phase detonation wave as the reaction model. Because the detonation wave is followed immediately by expansion waves, due to the side relief in the axial direction, it is a Chapman-Jouguet wave. The strength of this wave, which may be characterized by the pressure ratio across the wave, as well as the wave speed and the local wave Mach number, are related to design parameters such as the contraction ratio, chamber speed of sound, chamber diameter, propellant injection density and velocity, and the specific heat ratio of the burned gases. In addition, the distribution of flow properties along the injector face can be computed. Numerical calculations show favorable comparison with experimental findings. Finally, the effects of drop size are discussed and a simple criterion is found to set the lower limit of validity of this "strong wave" analysis.
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NOMENCLATURE

A area
a speed of sound
B function of Mach number, \( M_0 \), defined by Eq. (5)
c\(_p\) specific heat at constant pressure
d diameter
f oxidizer to fuel mass flow ratio (= \( \dot{m}_0/\dot{m}_f \))
h specific enthalpy
H total enthalpy in the wave fixed coordinate system
l thickness
L length and wave length
\( \dot{m} \) mass flow rate
M Mach number
p pressure
q velocity
Q heat release (= \( h_d - h^0 \))
R mean radius of the annular chamber
\( R_0 \) gas constant
t time
T temperature
u velocity component in the axial direction
v velocity component in the tangential direction
\textbf{Symbols:}

- $V_w$ wave propagation velocity
- $x$ axial coordinate
- $x_d$ detonation length
- $x_j$ jet break-up length
- $y$ tangential coordinate ($= R\theta$)
- $z$ parameter defined in Eq. 11
- $\gamma$ specific heat ratio

\[ \Gamma = \sqrt{\gamma} \left( \frac{2}{\gamma + 1} \right) \]

- $\rho$ specific density
- $\theta$ tangential coordinate
- $\eta$ non-dimensional tangential coordinate ($= y/x_d$)
- $\tau$ period
- $\Omega$ angular velocity

\textbf{Subscripts:}

- $0$ condition in front of the wave
- $1$ condition behind the wave
- $b$ breakup
- $c$ chamber
- $cr$ critical
- $d$ droplet
- $f$ fuel
j  jet
o  oxidizer
l  liquid
p  propellant
s  condition behind the leading shock

Superscripts
  o  condition at absolute zero
  bar (—) average value
  star (*) non-dimensional quantities
I. INTRODUCTION

Randomly distributed small explosions, sometimes described as "popping"\(^1\), often occur in liquid propellant rocket motors. Since the energy released per unit volume near the injector plate is high, these small explosions may often be amplified through detonation-like processes, and finally lead to steep fronted and self sustained pressure waves. This phenomenon is sometimes called "resonant combustion"\(^2\). The pressure wave resembles the detonation wave in fundamental respects; it propagates supersonically in the circumferential direction at a nearly constant frequency, and the pressure ratio across the wave front has been recorded as high as thirty.

The main objective of recent research in non-linear instability, theoretical as well as experimental (e.g. Refs. 2, 3, 4) is to understand the cause of the finite disturbances and their subsequent amplification. However, should the amplification not be successfully subdued, it becomes imperative that the engine designer know the effects of various design parameters on the final amplitude of the wave. Then the amplitude of the detonation-like wave can at least be reduced to a minimum strength in the design stage. Hence, in this analysis, it is assumed that the non-linear wave is fully developed at its final strength, a two phase detonation wave; the details of its evolution are set aside.
By studying the steady state problem, one can determine the weakening and strengthening effects of various design parameters on the wave amplitude, and thus indicate the proper direction for a stable wave design.
II. PHYSICAL PICTURE OF ANALYTICAL MODEL

There are several facts which have emerged from experimental studies of the strong wave instability studied here (e.g. Ref. 2); they lead directly to basic assumptions for the analytical model to be described.

First, it is known that the waves rotate at a constant angular velocity, and this is thus taken to be the case in the analysis. A somewhat subtle result of this observation is the condition of periodicity in the fluid properties. For example, since the wave velocity doesn't increase or decrease, the local properties immediately in front of and behind the wave are unchanged with time. In fact, at a given point on the chamber wall, an instrument would always measure the same distribution of pressure (for example) with time, between and across each succeeding wave.

Next, it is observed that the rotating waves are strong near the injector plate, but very weak near the nozzle, and that a wave doesn't "wind up" on itself, that is that even though the strength of the wave varies in the axial direction, the wave angular velocity is the same at each axial location. This is not to say that the whole wave structure is in one plane, for it isn't; it simply means that the weak and strong parts of the wave each travel with the same angular velocity, no matter what the wave shape may be. This is also assumed in the model.
Finally, it is observed that the propellant mass flow rates are not significantly changed from their equilibrium running values when the rotating wave motion occurs, and this is also assumed in the model.

The geometry of the combustion chamber under consideration is shown schematically in Fig. 1. In order to avoid the complexity arising from the radial variations in the governing equations, the analysis is limited to an annular chamber with a thin annulus. However, in a cylindrical chamber, because the wave travels tangentially at a constant angular velocity, it is only near the periphery of the chamber that the wave is strong enough to sustain reaction, and only near the injector plate that significant amounts of propellant droplets exist. Hence, there exists an annulus in which the assumptions made in the present analysis are clearly approximated. Just as in the familiar equilibrium rocket motor performance calculations, viscous forces are taken to be small compared to pressure forces and the heat transferred through the walls is considered to be small compared to the stagnation enthalpy of the gases in the chamber; thus, it is assumed that the chamber walls are frictionless and adiabatic. As a result, between waves, the gas flow is isentropic, and homoenergetic.

A fully developed two phase detonation wave (or waves), rotating at a constant velocity, \( V_w = R\Omega \), in the negative \( \theta \) direction, is considered. It is assumed that this wave front is planar and parallel to the axial direction. It extends an axial distance \( x_d \) from the injector plate, where \( x_d \)
is the distance to which the propellants penetrate between successive waves. Since all propellants are consumed within $x_d$, for $x > x_d$, the wave is a shock rather than a detonation wave. In fact, as will be seen, the detonation branches into two oblique shocks at $x = x_d$. For simplicity, the detonation is treated as a discontinuity, and all reaction is assumed to take place and be completed inside the wave. This assumption is valid as long as the time for droplet breakup and combustion behind the shock part of the detonation wave, is very small compared to the time between waves (or from another viewpoint as long as the wave thickness is small compared to the circumferential distance between waves). It is this condition which sets a limit on the analysis, as will be seen.

Since the wave propagates supersonically, the period of the wave is extremely short (for example the typical wave period of an 11 in. OD chamber is of the order of 470 $\mu$sec). It can be shown for droplets of moderate size (for example 100 microns in diameter) which are typically found in present day rocket motor combustion chambers, that in one wave period the changes in droplet mass, momentum, and temperature are small compared to their initial values. Hence, it may be assumed that the droplets are mechanically and thermally frozen at their injected conditions in one wave period. In addition, the effects of the droplets on the gas motion can be shown to be negligible. Hence,
to good approximation, the two phases near the injector plate may be considered as two independent systems, with phase change occurring only within the detonation wave.

As a result of the side relief offered to the gases immediately behind the detonation wave, in the axial direction, the detonation wave is assumed to be a Chapman-Jouget wave. That is the velocity of the gases leaving the wave, relative to the wave, is sonic, so no more pressure pulses can travel upstream to weaken the wave further.

Finally, it is assumed that the spray is dilute enough that droplet interactions may be ignored, that the gases follow the perfect gas law and are calorically perfect (constant specific heats), and that the energy available through reaction is not changed significantly by changing the mode of reaction. The first two of these assumptions are made for simplicity; the last is justifiable as long as the concentrations of the various species obtained at the end of wave reaction do not vary markedly from the corresponding values which the gaseous products would have in reaction taking place at the design condition. Certainly, as a first approximation, this variation in species concentrations may be neglected from the viewpoint that the propellant mixture ratio and hence the basic energy available is the same; only the mode of reaction is changed during transition from the design condition to the wave running condition. It should be noted that the main effects of different fuels and mixture ratios are introduced through this energy of reaction.
The analytical model described above has emerged as a result of efforts made to simplify the calculations, while still retaining and emphasizing the essential mechanisms. At the nozzle end of the chamber, it resembles a pure acoustic model, because the waves are very weak waves which must travel therefore at essentially acoustic velocities for the mode considered, through gases which are essentially at conditions corresponding to those in the equilibrium running engine. This, plus the fact that all parts of the wave travel at the same angular velocity, explains the success of acoustic theory in predicting wave speeds. At the injector end of the chamber, however, the structure of the model has no relationship whatsoever to an acoustic model. Here, the waves, both detonations and shock waves are strong waves with pressure ratios as high as thirty. The properties of the flow field through which the waves travel are not even closely approximated by the properties in the equilibrium running engine. Typically, the high pressure and temperature immediately downstream of the detonation wave are rapidly decreased by strong expansions from values far above to values below the equilibrium running values. Hence the waves travel through "cold", low pressure regions, compared to the design conditions; even though the velocity of the detonations is nearly the same as that of an acoustic wave in an undisturbed combustion chamber, it is highly supersonic compared to the actual cooler gases through which it passes. Typical pressure distributions, illustrating this point, are shown later.
III. ANALYSIS

The problem, in view of the rotating wave system, is basically unsteady in terms of a coordinate system fixed to the engine. This complexity can be removed by adopting a coordinate system fixed to the waves and thus rotating at a constant tangential velocity, $V_w$. Such a system is illustrated in Fig. 2, where it is seen that in this case, the waves are stationary and the wall and droplets are moving at a velocity $V_w$. The periodicity found in the original fixed coordinate system is made manifest here by the repetitive nature of the flow picture; e.g., a wave is seen each time a distance $L$ has been traversed in the $y$ direction in Fig. 2, with exactly the same conditions on either side of it as seen on the preceding wave. Since no frictional forces or wall heat transfer is considered, the motion of the wall presents no computational problems. Because the two phases may be considered as separate systems, the governing equations for the gases can be studied independently of the droplet motion. Finally, since radial variations are neglected, the conservation equations for the gases may be finally reduced in this system to conventional two dimensional steady state equations.

It is seen in Fig. 2, that the effects of the side relief are accounted for by an expansion fan which is centered at the triple point. The strength of the fan must be such as to bring the gas conditions back from their values immediately downstream of the detonation wave to the conditions
which existed immediately upstream of the detonation wave thus satisfying the periodic condition. The strength of the oblique shocks must be such that the pressure must be continuous across, and the velocity vectors must be parallel on either side of, the slip line (the line which separates the flow which goes through the detonation wave from that which passes instead through the shock waves at the triple point). Downstream of the weak wave (characteristic) reflected at the intersection of the last wave of the expansion fan with the wall, the flow is parallel to the wall in the region bounded by the wall and the slip line.

**Governing Equations and Method of Solution**

As a result of the previously mentioned assumptions, it is easily shown\(^5\) that the specific entropy, and total relative enthalpy which is defined as

\[
H = h + \frac{1}{2} (u^2 + v^2 - V_w^2)
\]

must be constant along every streamline in the gaseous flow field between any two consecutive waves (see Fig. 2). However, since the detonation wave itself is one dimensional, there is no variation in the specific entropy and total relative enthalpy normal to the streamlines. Consequently, between the waves, the gaseous flow field is isentropic and homoenergetic. Furthermore, in order to satisfy the periodic condition, these two properties must reach the same values in front of the wave in each and every
cycle. Therefore, it is concluded that there should be no net change of the specific entropy and total relative enthalpy of the gases across the detonation wave itself. This in no way violates the Second Law of Thermodynamics, since the entropy of the mixture of gases and droplets does increase across the wave.

The mixture entering the wave consists of burned gases and unburned propellant droplets, while the flow leaving the wave consists only of burned gases. The following non-dimensional quantities are defined:

\[ P_1^* = \frac{P_1}{P_0} \quad \rho_1^* = \frac{\rho_1}{\rho_0} \]
\[ Q^* = \frac{Q}{C_p T_0} \quad v_1^* = \frac{v_1}{\sqrt{\gamma R_o T_0}} \]
\[ V_w^* = \frac{V_w}{\sqrt{\gamma R_o T_0}} \quad \rho_d^* = \frac{\rho_d}{\rho_0} \]
\[ T_1^* = \frac{T_1}{T_0} \quad M_o = \frac{v_o}{\sqrt{\gamma R_o T_0}} \]

where the subscripts "o" and "1" refer to conditions upstream and downstream of the wave, respectively, and \( Q \) denotes the energy released across the wave per unit mass of propellants; that is

\[ Q = h_d - h^0 \]

\( \rho_d \) is the spacial density of the propellant droplets. It can be shown that there are eight equations (three from the jump conditions across the two phase detonation wave, two from the condition that the specific entropy and total relative enthalpy of the gaseous phase must be conserved
across the wave, two from the equations of state, and one from the
Chapman-Jouguet condition) for the nine parameters associated with the
detonation wave, i.e. $P_1^*, \rho_1^*, T_1^*, v_1^*, V_w, T_0, P_0, \rho_0$, and $v_0$.
These equations may be written as explicit equations for the wave param-
eters in terms of $M_o$, the local wave Mach number, as follows:

\[ T_1^* = B \] (4a)
\[ P_1^* = B^{\gamma/(\gamma-1)} \] (4b)
\[ \rho_1^* = B^{1/(\gamma-1)} \] (4c)
\[ v_1^* = B^{1/2} \] (4d)
\[ V_w^* = \frac{\left(\frac{\gamma+1}{\gamma}\right) B^{\gamma/(\gamma-1)} - (M_o^2 + 1/\gamma)}{B^{(\gamma+1)/2(\gamma-1)} - M_o} \] (4e)
\[ Q^* = T_c^* = \left(\frac{\gamma+1}{2}\right) B - \left(\frac{\gamma-1}{2}\right) \left\{ \frac{\left(\frac{\gamma+1}{\gamma}\right) B^{\gamma/(\gamma-1)} - (M_o^2 + 1/\gamma)}{B^{(\gamma+1)/2(\gamma-1)} - M_o} \right\}^2 \] (4f)
\[ \rho_d^* = \frac{\left\{ B^{(\gamma+1)/2(\gamma-1)} - M_o \right\}^2}{\left(\frac{\gamma+1}{\gamma}\right) B^{\gamma/(\gamma-1)} - (M_o^2 + 1/\gamma)} \] (4g)
\[ \frac{V_w}{a_c} = \frac{\left\{ \left(\frac{\gamma+1}{2}\right) B^{\gamma/(\gamma-1)} - (M_o^2 + 1/\gamma) \right\} \left\{ B^{(\gamma+1)/2(\gamma-1)} - M_o \right\}^{-1}}{\left(\frac{\gamma+1}{2}\right) B - \left(\frac{\gamma-1}{2}\right) \left[ \frac{\left(\frac{\gamma+1}{\gamma}\right) B^{\gamma/(\gamma-1)} - (M_o^2 + 1/\gamma)}{B^{(\gamma+1)/2(\gamma-1)} - M_o} \right]^2 \right\}^{1/2} \] (4h)

where $B$ depends only on $M_o$ and $\gamma$. 

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\[ B = \frac{2}{(\gamma + 1)} \left[ 1 + \frac{(\gamma - 1)}{2} M_o^2 \right] \]  

\( T_o \) and \( \rho_o \) can be obtained from Eqs. (4f) and (4g) respectively, and \( v_o \) and \( P_o \) are calculated using the definition of \( M_o \) and the equation of state respectively; thus,

\[ v_o = M_o \sqrt{\gamma R_o T_o} \]  
\[ P_o = \rho_o R_o T_o \]

It should be noted in Eq. (4f) that \( T_c^* \) is the dimensionless chamber temperature, \( T_c^* / T_o \), and that the fact that \( Q^* = T_c^* \) is the direct consequence of the fundamental assumption that the energy release is independent of the wave motion. Typical variations of \( T_1^*, P_1^*, V_w^*, T_c^*, \rho_d^*, \) and \( V_w / a_c \) with \( M_o \), are shown in Figs. 3, 4, and 5, for a representative value of \( \gamma = 1.26 \). In addition, \( P_s^* = P_s / P_o \) and \( T_s / T_o \), the pressure and temperature ratios respectively across a shock wave at the same Mach number, \( M_o \), are shown for comparison with the corresponding ratios across the two phase detonation wave.

Equations (4) do not afford a complete solution for the parameters of the problem, because, as noted previously, there are so far, only eight relations for nine unknowns. This is seen in Eqs. (4) by the fact that \( M_o \) is needed to calculate the remaining parameters. The final necessary relation is found by considering the flow through the nozzle and relating this flow to that from the injector plate.
It is seen, in Fig. 2, that the flow field between the first slip line and the nozzle end of the chamber is very complicated due to the existence of oblique shock waves induced by the detonation wave. However, if these waves become sufficiently weak at the nozzle end, as assumed, the flow through the nozzle may be considered as a mean flow which satisfies the conventional one dimensional flow equations plus two dimensional small disturbances. To the degree of approximation used in this analysis, one may neglect these small disturbances, and employing a control volume which extends from the injector plate to the nozzle entrance and is bounded by the chamber walls, derive conservation equations (for mass, axial momentum, and total relative enthalpy) which relate conditions at the injector plate to the mean flow conditions at the nozzle end. Thus, the difficulties inherent in the irreversible flow region, where many oblique shocks with finite strength exist, can be avoided. As a result, and because the mass rate of propellant flow with waves is unchanged from its design, wave free value, one can relate the average pressure on the injector plate to the design chamber pressure, \( P_c \); that is,

\[
\frac{1}{L} \int_0^L P \, dy = P_c \left( \frac{2\gamma}{(\gamma - 1)} \left[ 1 - \frac{\Gamma^2}{2} \frac{A_t}{A_c} \right]^{1/\gamma} - \left( \frac{\gamma + 1}{\gamma - 1} \right) \left[ 1 - \frac{\Gamma^2}{2} \frac{A_t}{A_c} \right] \right)
\]

(7)

The average pressure on the injector plate may be written as
\[
\frac{1}{L} \int_{0}^{L} P \, dy = P_0 \left\{ 1 + \frac{x_d}{L} J_2 \right\} \quad (8a)
\]

\[
J_2 = P_1 \ast \int_{0}^{\eta_0} \frac{P}{P_1} \, d\eta - \eta_0 \quad (8b)
\]

where \( \eta_o = y_o/x_d \) is the non-dimensional distance along the injector plate from the wave front to the point where the flow has expanded back to those conditions which exist upstream of the wave front, satisfying the periodic condition. The pressure distribution along the injector plate, \( P/P_1 \) can be calculated using the method of characteristics since the flow field between consecutive waves is irrotational. It is seen from Fig. 2 that the flow pattern resembles that of half of an under-expanded two-dimensional nozzle with the throat at the Chapman-Jouguet plane and centerline along the wall. The pressure distribution, \( P/P_1 \), and Mach number, \( M \), along the wall are plotted in Fig. 6. Finally, the pressure distribution can be integrated numerically to obtain the average pressure; curves of \( \eta_o, \int_{0}^{\eta_0} (P/P_1) \, d\eta, \) and \( J_2 \) are illustrated in Figs. 3 and 4 for \( \gamma = 1.26 \). It should be noted that after \( P_1 \) is obtained, a dimensional pressure distribution along the wall (pressure-space trace) can be obtained easily from Fig. 6. A typical wall pressure trace is shown later (Fig. 8).
From Eqs. (7) and (8) one can show that

\[
\frac{P_C}{P_o} = P^*_C = \frac{1 + \frac{x_d}{L} J_2}{\left(1 - \frac{\gamma}{2} \left(\frac{A_t}{A_c}\right)^2\right)^{1/\gamma} \left(\frac{\gamma + 1}{\gamma - 1}\right) \left[1 - \frac{\gamma}{2} \left(\frac{A_t}{A_c}\right)^2\right]^{1/\gamma}}
\]

(9)

Also, since the total propellant mass flow, which comes from the injectors and passes through the (choked) nozzle, must also enter the detonation wave,

\[
A_t \Gamma \frac{P_C}{\sqrt{R_o} T_c} = \rho_d V_w A_c \frac{x_d}{L}
\]

(10)

Finally, Eqs. (9) and (10) may be combined to give,

\[
\frac{x_d J_1}{L} \left(1 + \frac{x_d}{L} J_2\right) = Z
\]

(11)

where \( J_1 \) and \( Z \) are defined as follows:

\[
J_1 = \rho_d^* T_c^* \frac{V_w}{a_c^*}
\]

(12a)

\[
Z = \frac{A_t \Gamma}{A_c \sqrt{\gamma}} \left(\frac{2\gamma}{\gamma - 1}\right) \left[1 - \frac{\gamma}{2} \left(\frac{A_t}{A_c}\right)^2\right]^{-1/\gamma} \left[1 - \frac{\gamma}{2} \left(\frac{A_t}{A_c}\right)^2\right]^{1/\gamma}
\]

(12b)
Using Eqs. (4), one can write \( J \), in terms of \( M_o \); this function is shown in Fig. 4 for \( \gamma = 1.26 \).

Equation (11) is the desired relation to give \( M_o \), as a function of \( \gamma \), \( A_t/A_c \), and \( x_d/L \). Since it is not possible to write an explicit functional form for \( M_o \), a numerical solution is necessary. A typical plot of a solution to Eq. (11) is shown in Fig. 7, where \( x_d/L \) is plotted vs \( M_o \) for various constant values of \( Z \) for \( \gamma = 1.26 \). Since \( \gamma \) is a given value, \( Z \) varies only with the area ratio \( A_t/A_c \). The curves are found by setting \( Z \) equal to a given value, choosing a value for \( M_o \), calculating \( J_1 \) and \( J_2 \) (or using Fig. 4) and then calculating \( x_d/L \) using Eq. (11). This is done for several values of \( M_o \), and repeated for several values of \( Z \), leading to the curves shown in Fig. 7.

Although a typical numerical calculation will be illustrated later, it is worthwhile to point out that at this stage of the calculations, all desired parameters and fluid property variations could be calculated if the following engine design parameter are known:

1. \( A_t/A_c \)
2. \( \gamma \)
3. \( P_c \)
4. \( T_c (a_c) \)
5. \( R_o \)
6. \( x_d/L \)
where, again, $P_c$ and $T_c$ (or $a_c$) are those values associated with the equilibrium running engine, and $R_o$ is the specific gas constant and thus equal to the universal gas constant divided by the molecular weight of the exhaust gases in the equilibrium running engine. The procedure is as follows:

1. Knowing $A_t/A$ and $\gamma$, calculate $Z$, using Eq. (12b).

2. Knowing $x_d/L$ and $Z$ find $M_o$, using Fig. 7. This presupposes a plot such as Fig. 7 for the given $\gamma$.

3. Knowing $M_o$, calculate $B$ (Eq. (5)) and then $T_1^*$, $P_1^*$, $\rho_1^*$, $v_1^*$, $V_w^*$, $T_c^*$, $\rho_d^*$, and $V_w/a_c$, using Eqs. (4).

4. Knowing $a_c$ and $T_c$, as well as $V_w/a_c$ and $T_c^*$, calculate $V_w$, and $T_o$.

5. Calculate $P_c^*$, using Eq. (9), and knowing $P_c^*$, calculate $P_o$.

6. Knowing $P_o$, $T_o$, and $R_o$, calculate $\rho_o$, using Eq. (6b).

7. Knowing $P_o$, $\rho_o$, $T_o$, and $R_o$, as well as $T_1^*$, $P_1^*$, $\rho_1^*$, $v_1^*$, $\rho_d^*$, and $M_o$ calculate $T_1$, $P_1$, $\rho_1$, $v_0$, $v_1$, and $\rho_d$.

The pressure distribution can be obtained by finding $\eta_o$ using Fig. 3, and then calculating $y_o/L = \eta_o x_d/L$. Then one can use Fig. 6, since $P_1$ is known, to plot $P$ vs $y/L$ up to $y_o/L$, after which $P = P_o = \text{constant}$ up to $y/L = 1$ where a new wave exists. Thus, all quantities of interest can be obtained. It should be noted that knowing both $R_o$ and $a_c$ implies that one knows the fuel-oxidizer mixture ratio.
Calculation of Wave Width, $x_d$

In the previous section, it was shown that in order to obtain a general solution one must know not only the familiar design parameters of a rocket engine, but also $x_d/L$. It is this latter parameter which contains all the parameters associated with the injectors; one must be able to estimate this important parameter for various injector systems.

$x_d$ is taken to be the furthermost penetration distance of spray for the given injection system in the time between successive waves; i.e., it is assumed that no droplets exist beyond $x_d$. For simplicity, it is assumed that the droplet distribution is uniform over $x_d$, that is, the wave is one dimensional.

In the simple case where droplets are formed at the plane of the injector plate, $x_d$ may be defined as the distance over which droplets move in one wave period, that is,

$$
\frac{x_d}{L} = \frac{u_d}{V_w} = \frac{a_c}{V_w}
$$

where $u_d$ is the average droplet axial velocity, and would be considered to be known. However, it is noted that Eq. (13) generally underestimates the penetration of the spray for those injectors presently in use. Firstly, the spray tends to form at a finite distance from the injector plate, since a finite time is required for the liquid jets to be disintegrated. Secondly,
droplets in the spray are not uniform in size and thus are not uniformly distributed; larger drops penetrate further downstream. Hence, the penetration distance of the actual spray tends to be longer than the penetration distance calculated on the basis of the mean drop size.

There is no known exact theory to predict the penetration distance of the spray during the rotating wave motion. The following analysis is an approximate calculation, wherein the difficulties of the non uniform distribution of droplets in the spray are ignored and a simple expression derived using the equations expressing the conservation of mass and axial momentum of the injected liquid jets. It is assumed that the liquid jets are disintegrated at some distance, $x_i$, from the injector plate, and the jet breakup points are viewed as uniformly distributed sources, from which droplets issue. In the case where the jets are injected into the high speed transverse flow, $x_i$ may be expressed by

$$x_i = u_j t_b$$

(14)

where $u_j$ is the velocity of the liquid jet and $t_b$ is the jet breakup time given by

$$t_b = 4 \frac{d_j}{|q - q_j|} \sqrt{\frac{\rho_j}{\rho}}$$

(15)

In Eq. (15), $d_j$ is the jet diameter, $q - q_j$ is the slip velocity between the jet and the surrounding gases, $\rho_j$ is the density of the liquid jet, and $\rho$ is the density of the surrounding gases. In the case of impinging jets,
the impingement distance may be used for $x_i$, provided that the impingement distance is less than the length calculated using Eq. (14).

By using the conditions that the mass and axial momentum of droplets after breakup must be equal to the corresponding properties of the liquid jets coming into the point sources, a simple expression for $x_d/L$ can be derived:

$$\frac{x_d}{L} = 2 \frac{x_i}{L} + \frac{u_j}{V_w}$$

where $u_j$ is the mean jet velocity. In the case where bipropellants are used, then,

$$u_j = \frac{1}{(1+f)} u_f + \frac{f}{(1+f)} u_o$$

where $f$ is the oxidizer to fuel ratio, and $u_f$ and $u_o$ are the axial velocity components of the fuel and oxidizer, respectively. Due to the difficulty in assessing the effects of the rotating wave motion on the jet breakup, $x_i$ is approximated here by the predicted wave free jet breakup distance for the given injection system.

**Example Calculation; Comparison with Experiment**

A series of experiments on detonation-like wave motions were conducted at JPL for various design conditions which are listed in Table I. In order to test the validity of the present analysis these design conditions were used to calculate various wave and flow properties for
comparison with experimental results. A typical calculation is presented here to illustrate the method of solution in detail. It should be noted that all the design conditions and experimental data presented in Table I are for cylindrical chambers; as mentioned previously, however, there is an annulus in the cylindrical chambers, where the assumptions used here are met. Only data from those transducers located in this annulus, i.e. near the corner made by the injector plate and the chamber wall, are used for comparison. Some are located on the injector plate and some on the wall.

For the calculation to be made now, the design parameters of Ref. 11 (Table I) are used. In addition, γ is chosen to be 1.26. Thus, $A_i/A_c = 1/2$, $a_c = 3530 \text{ ft/sec}$, $P_c = 300 \text{ psia}$, $x_i = 0.75 \text{ in.}$, $L = 34.54 \text{ in.}$ (for one wave), $f = 2.8$, $q_f = 148 \text{ ft/sec}$, $q_o = 86 \text{ ft/sec}$, $\theta_f = 28^\circ 18'$ and $\theta_o = 15^\circ 42'$. Here $q_f$ and $q_o$ are the velocities of the fuel and oxidizer liquid injection jets respectively, and $\theta_f$ and $\theta_o$ are the corresponding angles made by these jets with respect to the axial direction. The injectors are of the impinging type and so $x_i$ is taken to be the axial impingement distance from the injector plate.

From Eq. (12b), $z$ is calculated to be 0.2796. Next from Eq. (17), $v_j$ is calculated to be 93.6 ft/sec. Then, from Eq. (16),

$$x_d/L = 0.0432 + 0.0265(a_c/v_w)$$
where $V_w/a_c$ is given in terms of $M_o$ by Eq. (4h). If Eq. (18) is plotted on Fig. 7 (dashed line) it is found to intersect the $z = 0.2796$ curve at $M_o = 2.73$, and $x_d/L = 0.0575$. With this value of $M_o$, $V_w/a_c = 1.847$ (from Fig. 3 or Eq. (4h)), $P_1* = 14.74$ (from Fig. 5 or Eq. (4b)), $J_2 = 18.1$ (from Fig. 4, defined by Eq. (8b)), and $P_c* = 1.94$ (Eq. (9)). Other desired parameters such as $T_1*$, etc., could be found in the same way by using Figs. 3 and 4 or Eqs. (4). Since $P_c = 300$ psia, then $P_o = P_c/P_c* = 155$ psia, and $P_1 = P_o/P_1* = 2280$ psia. Since $a_c = 3530$ ft/sec, $V_w = a_c (V_w/a_c)$ = 6520 ft/sec. The calculated and experimental results are both shown in Table II for this and other cases. It is seen that the analytical results compare quite favorably with the experimental data, especially in the case of Ref. 11.

After $P_1$ is obtained, the dimensional pressure distribution along the injector plate can be obtained easily from Fig. 4, after determining $\eta_o$ from Fig. 3. In this case, for $M_o = 2.73$, $\eta_o = 4.63$, so $y_o/L = \eta_o (x_d/L) = 0.266$. For $y > y_o$, of course, the pressure is constant at $P_o$. In Fig. 8, the calculated pressure distribution is shown and compared with an experimentally measured pressure trace obtained from Ref. 11. The agreement is seen to be excellent.
IV. EFFECT OF DESIGN CHANGES ON WAVE STRENGTH

It is of interest to use the results given by the preceding analysis to predict the effects of various design parameters on the wave strength which is characterized here by the detonation wave pressure ratio, \( P_1^* = P_1/P_0 \). In each case, the effects are illustrated in the form of a plot of \( P_1^* \) versus the design parameter in question, other parameters being held constant at those values listed under Ref. 11 in Table I and used in the example problem, except where noted.

1. Contraction Ratio, \( A_t/A_c \)

A plot of \( P_1^* \) vs \( A_t/A_c \) is shown in Fig. 9, for \( \gamma = 1.26 \) and for \( x_d/L \) constant. A point may be calculated by choosing an \( M_o \), and finding the corresponding \( P_1^*, J_1, J_2, \) and \( V_w/a_c \) points from Figs. 3, 4, and 5. \( x_d/L \) is here chosen to be the value calculated in the example calculations, \( x_d/L = 0.0575 \). With \( x_d/L, J_1, \) and \( J_2 \) known, \( z \) may be calculated using Eq. (11), and the corresponding \( A_t/A_c \) value may be found using Eq. (12b). By choosing several values of the parameter \( M_o \), the \( P_1^* \) and corresponding \( A_t/A_c \) values can be calculated and plotted as shown in Fig. 9. In addition, a few points with \( x_d/L \) varying as in Eq. (18) were calculated and are shown for comparison; this variation has a minimal effect.

It is seen that as \( A_t/A_c \) increases, the pressure ratio, \( P_1^* \), increases; the wave becomes stronger. Conversely, decreasing the contraction ratio has a stabilizing effect (wave becomes weaker). It can also be shown,
using Eqs. (4) that the wave speed increases or decreases the same as \( P_1^* \). This result is also found experimentally\(^10\).

2. Chamber Radius, \( R = L/2\pi \)

For a given \( x_d \), \( x_d/L \) increases as \( R \) decreases, and it is seen in Fig. 10 that for other design parameters being held constant (most importantly a constant contraction ratio), decreasing \( R \) results in decreasing \( P_1^* \). Thus as engine size increases, for geometrically similar engines, the wave strength increases. Again, \( V_w \) varies in the same manner as \( P_1^* \).

The curve of \( P_1^* \) versus \( R \) can be calculated as follows. For \( A_t/A_c = 1/2 \) and \( \gamma = 1.26, \ z = 0.2796 \). Then by choosing various values of \( M_0 \), and finding the corresponding \( P_1^* \) and \( x_d/L \) values from Figs. 5 and 7, one can calculate \( L = x_d / (x_d/L) = 2\pi R \) for each \( M_0 \) and thus for each \( P_1^* \). \( x_d \) was chosen to be the value found in the example calculation, \( x_d = 0.0575 \cdot 11 \cdot \pi = 1.99 \) in. (Note that Eq. (11) may be used rather than Fig. 7, with \( J_1 \) and \( J_2 \) given in Fig. 4.)

3. Droplet Injection Velocity, \( u_j \)

In this case, the mass rate of flow of propellants is held constant and \( u_j \) is varied by changing the injector areas. In addition, impinging jets are assumed, so Eq. (16) is employed. It is seen in Fig. 11 that
as \( u_j \) increases (which corresponds to an increase in \( x_d \)), \( P_1^* \) decreases. The effect of increasing the jet velocity is to weaken the detonation wave. \( V_w \) varies in the same way as \( P_1^* \).

The computations for Fig. 11 are performed as follows. Equation (16) is rearranged to give,

\[
\frac{u_j}{c} = \left(\frac{x_d}{L} - \frac{2x_i}{L}\right) \frac{V_w}{a_c} \cdot \frac{1}{a_c}
\]

For these calculations, \( 2x_i/L = 0.0432 \) and \( a_c = 3530 \) ft/sec, the values from the example problem. Again, for \( A_t/A_c = 1/2 \) and \( \gamma = 1.26 \), \( z = 0.2796 \). The points are found by choosing values of \( M_o \), finding corresponding values of \( V_w/a_c \), \( x_d/L \), and \( P_1^* \) from Figs. 3, 5, and 7, and calculating the corresponding value of \( u_j \) from the above equation.

4. Chamber Pressure, \( P_c \), and Chamber Speed of Sound, \( a_c \)

\( P_c \) and \( a_c \) are considered simultaneously since they are intimately related by the mass flow relation, i.e., \( \dot{m}_p \propto P_c/a_c \). In the first case considered here \( \dot{m}_p \) is held constant. In Fig. 12, \( P_1^* \) is plotted vs \( a_c \) for the case of impinging jets, so Eq. (16) is used for \( x_d/L \); since \( \dot{m}_p \) is constant, \( P_c \) varies also. \( P_c \) is plotted vs \( a_c \) in Fig. 13. This case corresponds to changing propellants or mixture ratios, holding other parameters constant. It is seen that as \( a_c \) increases, \( P_1^* \) and \( P_c \) both increase. Moreover, since \( P_c^* \), plotted also in Fig. 12, remains
relatively constant, then $P_o$ increases as $P_c$ increases. Thus, not only the pressure ratio, $P_1^*$, but also the pressure levels (i.e., $P_o$ and $P_1$) increase. Since $V_w/a_c$ increases or decreases when $P_1^*$ increases or decreases, and $P_1^*$ increases as $a_c$ increases, $V_w$ increases even more rapidly than $a_c$ (i.e., $V_w = (V_w/a_c) a_c$).

The curves in Figs. 12 and 13 may be computed as follows. From Eqs. (16), (9), (12b), and (10), one can show that

$$a_c = \frac{u_j}{\left(\frac{x_d}{L} - \frac{2x_1}{L}\right)} \frac{a_c}{V_w}$$

$$P_c^* = \left(1 + \frac{x_d}{L} J_2\right) z \frac{\sqrt{\gamma}}{\sqrt{r}} \frac{A_c}{A_t}$$

$$P_c = \frac{\dot{m}_p}{A_c} \frac{A_c}{A_t} \frac{a_c}{r \sqrt{\gamma}}$$

For these calculations, the values employed in the example calculations were used. Thus,

$$A_t/A_c = 1/2 \quad \gamma = 1.26 \quad z = 0.2796$$

$$u_j = 93.6 \text{ ft/sec} \quad 2x_1/L = 0.0432 \quad \dot{m}_p/A_c = 1 \text{ lbm/in.}^2 \text{ sec}$$

where the value of $\dot{m}_p/A_c$ is that corresponding to the other parameters in the example calculation as seen in Column 1 of Table I. As in previous
calculations, various values of the parameter $M_0$ are chosen and corresponding values of $x_d/L$, $V_w/a_c$, $J_2$, and $P_1^*$ are found from Figs. 7, 3, 4, and 5 respectively. Then the above equations may be used to calculate the corresponding values of $a_c$, $P_c^*$, and $P_c$.

It should be noted that if an injection system where $x_d/L$ is a constant were considered, for this case where $\dot{m}_p$ is held constant, then from Fig. 7 it is seen that since $z$ is also a constant, $M_0$ would be a constant. Hence, from Fig. 5, $P_1^*$ would also have a constant value as $a_c$ and $P_c$ increased. However, since $P_c^*$ (see Eqs. (9) and (12b)) would also remain constant, $P_o$ would increase as $P_c$ increased. Likewise $P_1 = P_o \cdot P_1^*$ would increase as $P_c$. The result would be a case where the pressure levels increased in such a way that the pressure ratio, $P_1^*$, remained constant.

In addition since from Fig. 3, $V_w/a_c$ would have a single value (for one value of $M_0$), $V_w$ would vary directly as $a_c$.

The second case to be considered here is that where $a_c$ is held constant and $P_c$ is varied, so that the propellant mass flow rate, $\dot{m}_p$, varies. However, $u_j$ is held constant so that the $\dot{m}_p$ variation is due entirely to variations in the cross sectional area of the injectors. Again impinging jets are assumed so that Eq. (16), holds, written again here as

$$u_j = \left(\frac{x_d}{L} - \frac{2x_i}{L}\right) \frac{V_w}{a_c}$$
Now, \( x_i/L, u_j, \) and \( a_c \) are constants, and it is seen from Figs. 7 and 3 that since \( z \) is a constant, \( x_d/L \) and \( V_w/a_c \) depend only on \( M_o \). That is, the above equation could be written with \( M_o \) as the only unknown parameter and thus it could be used to calculate the value of \( M_o \) for the case in question. That is, \( M_o \) is a constant. Hence, from Fig. 5, \( P_1^* \) is constant and the plot of \( P_1^* \) vs \( P_c \) would be simply a straight line, \( P_1^* = \text{constant} \).

Thus, from Figs. 3 and 7, or Eqs. (4), and from Eq. (9), \( V_w/a_c, x_d/L, \) and \( P_c^* \) have constant values. Then, as \( P_c \) increases, with \( a_c \) constant, \( P_o = P_c/P_c^* \) increases and \( P_1 = P_1^* \cdot P_o \) increases; the pressure levels increase such that the pressure ratio remains unchanged. The wave speed, 

\[ V_w = (V_w/a_c) a_c \]
remains unchanged.

5. Droplet Penetration Distance, \( x_d \)

The droplet penetration distance is a parameter controlled by the type of injector system employed and is thus indirectly a design parameter. In Fig. 10, the graph of \( P_1^* \) vs \( x_d \) indicates that as \( x_d \) increases, the wave strength decreases. In this case it can also be shown easily, from Figs. 7 and 3 that as \( x_d \) increases, \( V_w \) decreases.

Calculations for the curve of \( P_1^* \) vs \( x_d \) in Fig. 10 are made as follows. \( A_t/A_c = 1/2 \) and \( \gamma = 1.26 \) are assumed so \( z = 0.2796 \). Then, various values of \( M_o \) are chosen and corresponding values of \( P_1^* \) and \( x_d/L \) are found from Figs. 5 and 7. Finally \( x_d = (x_d/L) \cdot L \) and \( L \) is chosen to be the value in the example calculation, \( L = 11 \cdot \pi = 34.54 \) in.
V. LIMIT OF THE ANALYSIS

In Ref. 10, it is shown experimentally that an increase in drop size has a stabilizing effect, i.e. that the wave strength is decreased. On the basis of the present analysis, this effect may be explained by noting that the larger droplets tend to penetrate farther downstream, leading to a larger $x_d$, and hence a smaller $P_1^*$. However, the drop size effect is not formally incorporated into the present theory due to the fundamental assumption that the wave is treated as a discontinuity. Now it has been shown, that the reaction zone length in a two-phase detonation wave increases with drop size; hence losses due to lateral expansion occurring inside the reaction zone must become important at some point as the drop size is increased, and the thin wave assumption breaks down. Evidently the theory cannot be valid for droplets of all sizes. Although a detailed analysis of the structure of the reaction zone is not available, it is possible to set at least a rough limit on the drop sizes for which the analysis is valid.

The structure of a spray detonation has been observed experimentally. The study indicates that much of the mass transfer from the droplet is associated with a local explosion during droplet breakup. Hence, it is believed that a great portion of the energy released from the droplet may be associated with a blast wave initiating from the explosion site,
and that the interaction between the blast wave and the leading shock part of the detonation wave, may be mainly responsible for the energy transmission. On the basis of this observation, a simple criterion for the existence of a strong thin wave may be formulated. That is, the time required for a sound wave initiating at the explosion site of the disintegrating drop to propagate upstream to the leading shock must be equal to or less than the time required for the same wave to propagate laterally the distance to which the detonation extends. If conditions immediately behind the shock are used to estimate the sound velocity and known expressions for droplet breakup times are used, this criterion finally leads to calculation of the critical drop diameter for a wave of given strength. A detailed derivation is given in Ref. (5); the resulting critical drop diameter is,

\[ d_{cr} = \frac{1}{4} \left[ \frac{x_d}{L} A c \frac{\dot{m}_p}{\rho_p} \right]^{1/2} J_3 \]  

(19a)

\[ J_3 = \left( 1 - M_s \right) \left( 1 - \frac{v_s^*}{v_w^*} \right) \left[ c \left( \frac{T}{\rho_s^*} \right)^{1/2} \right] \left( \frac{\rho_s^*}{\rho_d^*} \right)^{1/2} \]  

(19b)

where the subscript \( s \) refers to conditions behind the leading shock and \( \dot{m}_p/\rho_p \) is the average volumetric flow rate, \( \rho_p \) being the average liquid density. \( J_3 \) is plotted in Fig. (14) for \( \gamma = 1.26 \). If \( \bar{d} \) is the average drop diameter calculated for the given injection system, then Eq. (19a)
leads to the following conclusions:

(1) The strong wave analysis is valid if \( d \leq d_{cr} \).

(2) Weak waves, which may not be properly described as detonations, are likely to occur if \( \overline{d} > d_{cr} \).

There are experimentally correlated equations to calculate \( \overline{d} \) for different types of atomization\textsuperscript{(18)}. For the case of impinging jets, for example, the following equation may be used to calculate the mass median drop diameter\textsuperscript{(19)}

\[
\overline{d} = 6.92 \times 10^4 \left[ \frac{d_f^{0.27} d_o^{0.023}}{q_f^{0.74} q_o^{0.33}} \right]
\]  

(20)

where the diameters of the fuel and oxidizer injectors, \( d_f \) and \( d_o \) respectively, are in inches and the jet velocities of the fuel and oxidizer, \( q_f \) and \( q_o \) respectively, are in ft/sec.

If \( \overline{d} \) and \( d_{cr} \) are calculated for each of the experimental conditions covered in Table I, it can be shown\textsuperscript{(5)} that only in the case of Ref. 11 is \( \overline{d} \) less than \( d_{cr} \); it is for this case also that the comparison between theory and experiment is most satisfactory. Evidently the waves in the remaining cases (see Table III) do not satisfy a strong wave criterion.
VI. CONCLUSIONS

This study is concerned with tangential mode detonation-like phenomena in liquid propellant rocket motors. It is found that the amplitude of the wave, measured by the pressure ratio across it ($p_1^*$), may be reduced by reducing the contraction ratio ($A_t/A_c$), and chamber diameter, and by increasing the spray penetration distance during the wave running condition ($x_d$). Expressions for $x_d$ are provided. It is found that $x_d$ increases as the jet velocity and projected jet breakup distance increase. In general, the variation of the wave speed follows the same trends as the wave pressure ratio.

For a given contraction ratio, the effects of the chamber speed of sound and chamber pressure depend on the type of injector system used, and upon the propellant mass flow rate. In general, as $a_c$ and $p_c$ increase the pressures immediately upstream and downstream of the wave, and the wave velocity, increase. In some cases, the pressure ratio across the wave also increase, and in others it remains unchanged.

Finally, the limit of the strong wave analysis due to the finite drop size is discussed. A critical drop size, above which a strong wave probably cannot occur is derived, to set a lower limit of validity on the strong wave analysis.
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Table I. Geometric and operating parameters of the equilibrium running chamber \( \frac{A_t}{A_c} = \frac{1}{2} \).
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<th>Run</th>
<th>$V_w$ (ft$^2$/sec)</th>
<th>$P_1$ (psia)</th>
<th>$P_0$ (psia)</th>
<th>$P_1/P_0$</th>
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Table II. Comparison of experimental and analytical results.
REFERENCES


Fig. 1. Geometry of the Annular Chamber.
Fig. 2. Flow Pattern Between Two Consecutive Waves in the Wave-Fixed Coordinate System.
Fig. 3. $V_w^*, T_c^* (= Q^*), T_s^*, T_1^* (1 + \eta_0 / 10), V_w / a_c$, 
and $1 + \int_0^{\eta_0} \left( \frac{P}{P_1} \right) d\eta$ versus $M_0$ for $\gamma = 1.26$. 

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Fig. 4. $\rho_d^*$, $J_1$, and $J_2$ versus $M_0$ for $\gamma = 1.26$. 
Fig. 5. $P_1^*$ and $P_s^*$ versus $M_0$ for $\gamma = 1.26$. 
Fig. 6. Distribution of $P/P_1$ and $M$ Along Injector Plate.
Fig. 7. $x_d/L$ versus $M_o$ for given values of $z$ and $\gamma = 1.26$. Dotted Lines Show Intersection of Eq. (18) with Curve for $z = 0.2796$, giving $M_o = 2.73$ for Example Calculation.
Fig. 8. Comparison of Actual and Calculated Pressure Distributions Along the Injector Plate for Run 956 of Ref. 11.
Fig. 9. $P_1^*$ versus $A_t/A_c$ for $x_d/L = \text{constant} = 0.0575$ and $T = 1.26$. The Points Indicated by • Show the Very Small Changes which Occur when an $x_d/L$ which Varies According to Eq. (18) is employed.
Fig. 10. $P_1^*$ versus $x_d$ and $R$ for $\gamma = 1.26$ and $A_t/A_c = 1/2$ ($z = 0.2796$). Calculations for $x_d$ were made assuming a 5.5 in. radius motor, as in Ref. 11, so that $L = 34.54$ in. Calculations for $R$ were made assuming $x_d = 1.988$ in., corresponding to the numbers in the Example Calculation in the Test. ($x_d/L = 0.0575$)
Fig. 11. $P_j^*$ versus $u_j$, for $\gamma = 1.26$, $x_j/L = 0.75/34.54$
$= 0.0432$, and $A_c = 3530$ ft/sec (Re. Eq. (16)).
Fig. 12. $P_1^*$ and $P_C^*$ versus $A_c$ for $\gamma = 1.26$, $A_t/A_c = 1/2$, 
$(z = 0.2796)$, $u_j = 93.6$ ft/sec, and 
$x_1/L = 0.0432$. 
Fig. 13.  $P_c$ versus $A_c$ for Same Conditions as in Fig. 12
and $m_p/A_c = 1$ lbm/in.$^2$ sec.
Fig. 14. $J_3$ versus $M_0$ for $\gamma = 1.26$. 
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Theoretical Analysis of a Rotating Two Phase Detonation in a Rocket Motor

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Washington, D.C. 20546

Project Manager, Richard J. Priem, Chemical Propulsion Division
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Tangential mode, non-linear wave motion in a liquid propellant rocket engine is studied, using a two phase detonation wave as the reaction model. Because the detonation wave is followed immediately by expansion waves, due to the side relief in the axial direction, it is a Chapman-Jouguet wave. The strength of this wave, which may be characterized by the pressure ratio across the wave, as well as the wave speed and the local wave Mach number, are related to design parameters such as the contraction ratio, chamber speed of sound, chamber diameter, propellant injection density and velocity, and the specific heat ratio of the burned gases. In addition, the distribution of flow properties along the injector face can be computed. Numerical calculations show favorable comparison with experimental findings. Finally, the effects of drop size are discussed and a simple criterion is found to set the lower limit of validity of this "strong wave" analysis.

Detonation
Rocket Motor Combustion
Combustion Instability

Unclassified - unlimited

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