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EVALUATION OF THE OBLIQUE DETONATION WAVE RAMJET

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EVALUATION

OF THE

OBLIQUE DETONATION WAVE

RAMJET

BY

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SUMMARY

The potential performance of oblique detonation wave ramjets is analyzed in terms of multishock diffusion, oblique detonation waves, and heat release. Results are presented in terms of thrust coefficients and specific impulses for a range of flight Mach numbers of 6 to 16.

1.0 Introduction

Interest in ramjet propulsion began toward the end of World War II at a time when the turbojet was being accepted as an effective means of obtaining higher flight speeds. In this time period it was also felt that ramjet propulsion would be a next step for attaining still higher flight speeds. Ramjets were also envisioned as vehicles for delivering warheads over intercontinental distances either by cruise or by skip flight trajectories. All of these proposed techniques involved subsonic combustion for supersonic vehicles operating at low supersonic speeds.

The competition of the rocket and the ramjet for payloads to be delivered over intercontinental distances was awarded to the rocket in the 1950's and interest in subsonic combustion ramjet propulsion waned to very low levels. Interest in "supersonic combustion ramjets" and "standing detonation wave ramjets", however, increased during this period. Numerous ramjet configurations and techniques were offered that utilized these latter concepts, however, only one persisted and survived, that of the diffusive burning scramjet. With this singular exception it appears most of the interest in hypersonic airbreathing propulsion ceased in the 1965 to 1970 time period.

The use of oblique detonation waves is a more recent consideration for use in ramjets that operate at hypersonic speeds. This concept differs markedly from the diffusive burning scramjet.

The diffusive burning scramjet compression process is carried out to high pressures and to the high temperatures required of the process.

The compression process of the oblique detonation wave ramjet is moderate and carried out to relatively low pressures and temperatures. The shock component of the detonative process supplies an additional large compression as well as corresponding high temperatures required for rapid combustion.

This concept, which has received little attention in the past, is analyzed to place in perspective its position in the flight regime of airbreathing propulsors.

2.0 Literature Search

To determine what research has been conducted on oblique detonation wave ramjets, a thorough literature search was considered appropriate.

The initial literature search was conducted through the Defense Documentation Center (DDC). Short abstracts of some 400 documents were provided by DDC, and of these 30 were requested for detailed perusal. Considering the mass of apparently pertinent documentation available through DDC there was surprisingly little really useful information[1] on research of multi-shock detonation phenomenon for achieving hypersonic performance in ramjets. This lack of information was, by itself, useful in that it indicated the minimum effort that had been directed toward this phenomenon.

Following the initial DDC literature search additional documentation was obtained through the Johns Hopkins Applied Physics Laboratory, the Air Force Office of Scientific Research, NASA and -- of particular interest -- the Navy intelligence system.

A partially annotated bibliography of the most useful documentation resulting from the literature search will be provided NASA, Langley, as a separate document independent from this report.

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1 A notable exception is a Royal Aircraft Establishment Technical Report (Ref. 1) of 1970 which describes analytical studies of multi-shock hypersonic ramjet intake flows matched, to strong or Chapman-Jouguet detonation waves. The results "emphasise (sic) the need for research on oblique rather than normal detonations..."

3.0 New Concepts for Hypersonic Propulsion

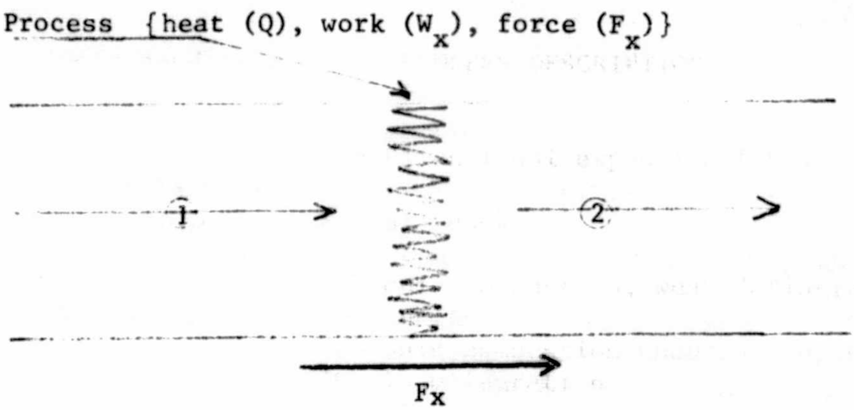
3.1 General

New concepts of hypersonic propulsion seem to have been singly phenomenological in nature rather than the symbiosis of the seemingly unrelated phenomena into a complete and integrated propulsion system. Most of these concepts also focus upon new uses for well known phenomena and many times a well understood phenomena. In this regard recourse is usually made to an analysis of the thermodynamic aspects of the process, with attention being focused on state changes that occur in the process. The dynamic aspects of the process are de-emphasized by elimination of velocity terms in the conservation equations thereby rendering indistinct the functional relationships that exist between the thermodynamic properties and dynamic properties of high speed combusting mixtures.

Characteristically the above analyses utilize the Hugoniot equations and the corresponding pressure vs. volume plots to describe combustion processes and phenomenological changes that take place in the flow. The means for description and classification exist and one of these concepts follows.

Consider, as in Figure 3.1.1, the flow of a gas in a one dimentional duct wherein the flow undergoes a process change from ① to ②. Solution of the one dimensional conservation equations of mass, momentum, and energy in terms of Mach number yields:

FIGURE 3.1.1 Constant Area Duct



$$\frac{M_1^2\left(1+\frac{\gamma-1}{2}M_1^2+\frac{Q-W_x}{C_pT_1}\right)}{\left(1+\gamma M_1^2+\frac{F_x}{P_1A}\right)^2}=\frac{M_2^2\left(1+\frac{\gamma-1}{2}M_2^2\right)}{\left(1+\gamma M_2^2\right)^2} \tag{3.1.1}$$

Where:

M = Mach number

γ = Ratio of specific heats

P = Pressure

A = Area

F_x = Body force (i.e., drag etc.)

Q = Heat added to the system

W_x = Work done by the system

C_p = Specific heat at constant pressure

T = Temperature

$\Phi(M)$ is defined as:

$$\Phi(M) \equiv \frac{M^2\left(1+\frac{\gamma-1}{2}M^2\right)}{(1+\gamma M^2)^2} \qquad 3.1.2$$

A graph of $\Phi(M)$ vs. Mach number for air ($\gamma = 1.4$) is shown in Figure 3.1.2. The various processes associated with the thermodynamics of combustion in a constant area duct can be described in terms of $\Phi(M)$ and the flow Mach number. An adiabatic process, such as a shock, would possess the same value of $\Phi(M)$ and proceed from some point such as C or D to its corresponding companion point A or B. Subsonic combustion would take place from some point A to a point B depending upon the heat release.

FIGURE 3.1.2 Graph of $\Phi(M)$ vs. Mach Number, M, for a Constant Area Duct and $\gamma = 1.4$ (air)

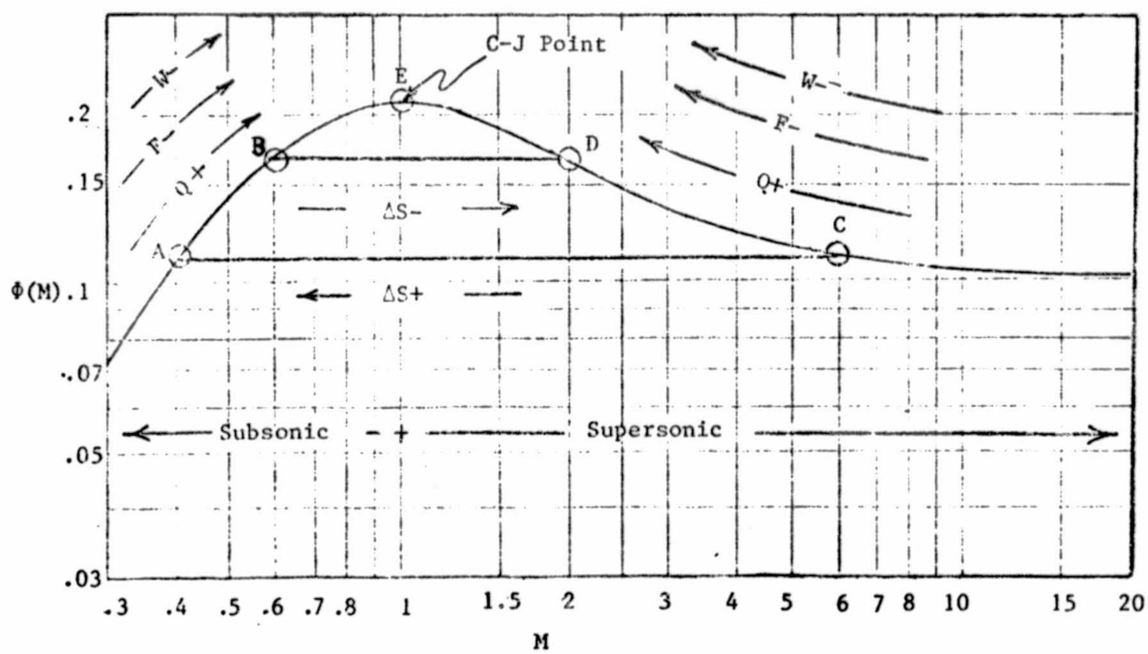


TABLE 3.1.1 TABLE OF ONE DIMENSIONAL PROCESS DESCRIPTIONS

(Referencing Figure 3.1.2)

PROCESSES FOR Q POSITIVE	PROCESS DESCRIPTION
B to D or A to C	One Dimensional expansion ($\Delta S-$) impossible
C to A or D to B	Normal Shock
A to B	Subsonic combustion, weak deflagration
A or B to E	Subsonic combustion Chapman-Jouguet (C-J) deflagration
C to A to E or D to B to E	Chapman-Jouguet (C-J) detonation
C to D or D to E	Supersonic combustion
A to B to D	Strong deflagration, impossible
C to A to B to D	Weak detonation, impossible
PROCESSES FOR Q NEGATIVE	**PROCESS DESCRIPTION**
D to C	The aerothermopressor
PROCESSES FOR F NEGATIVE	**PROCESS DESCRIPTION**
A to B or C to D	Friction tube

Table 3.1.1 is a summary description of the possible processes that encompass many of the new concepts involving hypersonic propulsors. These new concepts fall in the following categories:

Supersonic combustion techniques	3.1.1
Aerothermopressor	3.1.2
Normal and Oblique detonation waves	3.1.3
Rotating detonation waves	3.1.4
Miscellaneous Concepts	3.1.5

3.1.1 Supersonic Combustion

The main focus of supersonic combustion concepts center upon variable area ducts in which combustion is accomplished by diffusive burning of a fuel injected into high temperature supersonic airstreams. Diffusive combustion being characteristically slow requires that fuel injection techniques be employed that greatly enhance the mixing processes of the air with the fuel, i.e., the utilization of multiple injection ports and turbulent mixing processes.

The requirements on the geometry of the combustion chamber necessitate that the areas be matched to the heat release in order to control the pressure, and that thermal choking be avoided for heat addition at low supersonic flow velocities.

The requirements of a suitable fuel include low molecular weight for high diffusion rates and a high heat of combustion. Hydrogen is the preferred fuel. Pyrofluoric fuels have been suggested as one means for enhancing combustion rates.

It is of interest that foreign technology seems to be concentrating upon the detailed chemical kinetics of combustion in supersonic flows. Whereas the United States is primarily concerned with hydrogen as a fuel, foreign technology is considering hydrocarbon fuels as well as hydrogen. The structure and stability of combustion processes are also pointed up in their research.

New concepts on supersonic combustion have concentrated upon such details as local geometries, injection techniques, boundary layer interactions, local flow dynamics, ignition and pre-ignition, and other related topics.

3.1.2 The Aerothermopressor

The aerothermopressor is a theoretical technique to provide cooling of the intake airflow with cryogenic fuels such as hydrogen and to provide supersonic/hypersonic cooled compression of improved intake efficiency.

The indicated process is shown in Figure 3.1.2 and Table 3.1.1. Although this concept is not new its use for hypersonic propulsor could provide a means of alleviating cooling problems and providing increased performance capability.

3.1.3 Normal and Oblique Detonation Waves

The concept of detonative combustion orginated in the latter part of the nineteenth century when French physicists Vielle, Berthelot, Mallard, and Le Châtelier noted, in the course of their investigations with combustible mixtures, that under certain conditions combustion waves developed which possessed the extraordinary velocities of thousands of feet per second. About 1900 Chapman (Ref 2) and Jouguet (Ref 3) independently advanced the explanation that such phenomena could be accounted for if this "detonation wave" were treated as a shock wave followed by combustion -- the combustion, in turn being initiated by the high temperature accompanying the shock rather than the diffusion processes usually associated with deflagrations. With the exception of minor alterations and elaborations the theory remains unchanged up to the present.

Bone, Frazer, and Wheeler (Ref 4) noted in their experimental investigation that, under certain conditions, the

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2 Chapman, D. C., Philosophical Magazine, (5), Vol 47, 1890 p. 90

3 Jouguet, E., J. Math, 1905, p 347, 1906 p. 6; "Mechanique de Explosifs", Paris 1907; "La Theorie Thermodynamique de la Propagation des Explosions", Proceedings of the 2nd International Congress for Applied Mechanics, Zurich, 12-17 Sept. 1926, pp. 12 to 22

4 Bone, W. A., Frazer, R. P., and Wheeler, W. H., Philosophical Transactions of the Royal Society of London, A, Vol 235, 1936 p. 29

detonating front propagated in a cork screwing spiral fashion down tubes of various cross-sectional shapes. This phenomenon, called "spinning detonation", commonly exists as the lean limits of detonability are approached, a region, wherein reaction rates are sluggish and the combustion zones are greatly enlarged.

Photographic observation of the spinning detonation reveals a complex of oblique shock waves in a regular and seemingly repetitive manner. The concept of spinning detonation waves has not been exploited, even though the concept could well be utilized to explain the stability of detonative combustion and to determine flow geometries that produce stable detonative combustion.

Markstein and Polyani noted in the late 1940's that very dilute combustible mixtures could be made to produce "cellular flames" which stabilized in an array of cell sizes. Flow velocities were very low and vertically oriented with resultant very small gradients in the flow field. It would appear that this phenomenon bears some relation to granular structures observed for detonations wherein large gradients exist in the flow field.

Foreign technology is expending a large effort to understand the detailed structure and stability of detonation waves.

Initial concepts for the use of detonative combustion for propulsion purposes considered the following:

- Normal (to the flow) detonations at flight Mach number
  See Figure 3.1.3

- Oblique detonations at flight Mach number

  See Figure 3.1.4

These initial concepts were followed by more advanced concepts i.e.:

- Normal detonations at below flight Mach number

  See Figure 3.1.5

- Oblique detonations at below flight Mach number

  See Figure 3.1.6

All the above concepts were rejected for hypersonic propulsion systems except that of oblique detonations at below flight Mach number.

The reasons for rejection were:

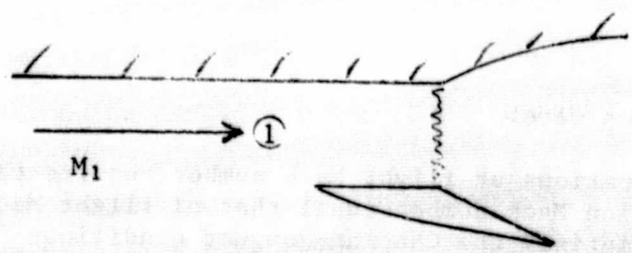
- Normal detonations at flight Mach number require that the detonation Mach number equal that of flight Mach number and satisfy the Chapman-Jouguet conditions. At high flight Mach numbers the heat addition requirements and the stagnation pressure losses would be excessive.

- Normal detonations at below flight Mach numbers would likewise require that the Chapman-Jouguet conditions be satisfied and that the heat release be identical to that required to stably maintain the Mach number of detonation exactly equal to the stream Mach number. In addition the increase of stream static temperature from the free stream static temperature places the requirement of higher heat releases for maintaining the detonation. Excessive stagnation pressure losses, however, are somewhat ameliorated.

- Oblique detonations at flight Mach number are stable and do not require that exact heat addition to match flow Mach numbers as in the case for maintaining the normal detonation. The cycle efficiency, however, is somewhat low making this option unattractive.

Oblique detonations at below flight Mach number emerges as the new concept with excellent promise for the use of detonative combustion. Stagnation pressure losses for the propulsion unit can be made low for good cycle efficiencies. The complexities associated with maintaining a stable normal detonation wave are eliminated giving flexibility for heat release under stable operating conditions.

FIGURE 3.1.3 Normal C-J Detonation at Flight Speed



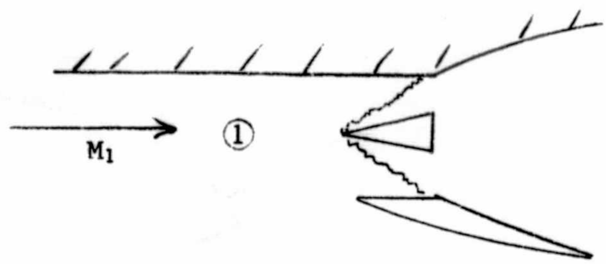
PROCESS
- NORMAL SHOCK COMPRESSION
- COMBUSTION
- EXPANSION

ADVANTAGES
- SIMPLE GEOMETRY

DISADVANTAGES
- HIGH Po LOSSES
- STABILITY OF DETONATION AT $M_{DC-J} = M_1$
- FUEL INJECTION AT FLIGHT SPEED
- ONLY USEFUL AT C-J SPEED
- THRUST CONTROL LACKING

ESTIMATED POTENTIAL INTERNAL PERFORMANCE CHARACTERISTICS FOR STOICHIOMETRIC HYDROGEN-AIR AT $M_1 = 5$
Isp = 2916 sec. $C_T = 0.495$ Po LOSS RATIO = 0.054

FIGURE 3.1.4 Oblique Detonation at Flight Speed



PROCESS

1 OBLIQUE SHOCK COMPRESSION
COMBUSTION
EXPANSION

ADVANTAGES

SIMPLE GEOMETRY
STABILITY OF DETONATION WAVE IMPROVED OVER NORMAL C-J DETONATION
THRUST CONTROL

DISADVANTAGES

RELATIVELY LARGE Po LOSSES
FUEL INJECTION AT FLIGHT SPEED
LIMITED RANGE OF USEFUL FLIGHT SPEEDS

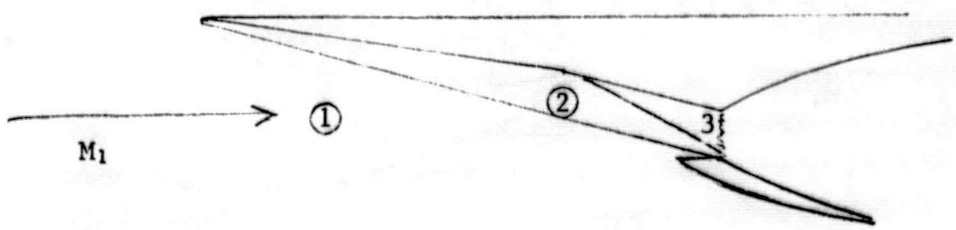
ESTIMATED POTENTIAL INTERNAL PERFORMANCE CHARACTERISTICS

C_T = .17 C_T IDEAL = .25

Isp = 2100 sec. Isp IDEAL = 3000

FOR M_1 = 10 & Q = 3489 J/g

FIGURE 3.1.5 Normal C-J Detonation Under Diffused Conditions



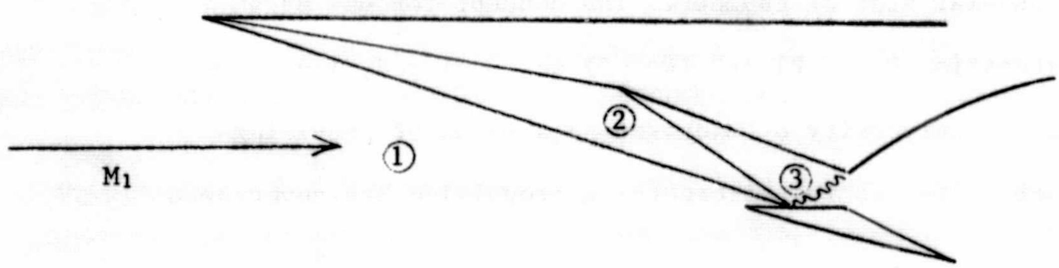
2 SHOCK DIFFUSER CONFIGURATION

PROCESS
- 2 OBLIQUE SHOCKS + 1 NORMAL SHOCK COMPRESSION
- COMBUSTION
- EXPANSION

ADVANTAGES
- RELATIVELY SIMPLE GEOMETRY
- MODERATE Po LOSSES

DISADVANTAGES
- STABILITY OF DETONATION AT $M_{DC-J} = M_3$
- THRUST CONTROL LACKING

FIGURE 3.1.6 Oblique Detonation Under Diffused Conditions



2 SHOCK DIFFUSER CONFIGURATION

PROCESS
- 3 OBLIQUE SHOCK COMPRESSION
- COMBUSTION
- EXPANSION

ADVANTAGES
- RELATIVELY SIMPLE GEOMETRY
- THRUST CONTROL
- STABILITY OF DETONATION
- WIDE RANGE OF USEFUL FLIGHTS SPEEDS

DISADVANTAGES
- NON OPTIMUM PERFORMANCE

3.1.4 Rotating Detonation Waves

A rotating detonation wave is a detonation wave that rotates around an annular slot or chamber. The concept for use as a "Rotary Detonation Power Plant" (Ref 5) was explored, inconclusively, at the University of Michigan as a means of providing rocket power. Its use for airbreathing propulsion has never been explored.

Rotating detonation waves have, however, received a very significant foreign attention particularly in the USSR, wherein Voytskhovskiy received a Lenin prize for his work in this subject.

There are several properties to the rotating detonative phenomenon that could be utilized for propulsive purposes, namely:

- The highly convective flow in the wake of both the shock and combustion zones is subjected to large centrifugal forces which tend to stabilize the detonation process.
- The annular chamber may be regarded as an ever increasing sloped ramp which will "overdrive" an initial Chapman-Jouguet detonation into a strong detonation. This would provide wider detonation limits.
- Multiple waves can be employed in the annular chamber. The USSR scientists are reported to have simultaneously stabilized several waves in the same chamber.

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5 Morrison, R. B., et al US Patent 3,240,010

- The pressures behind the "overdriven" detonations are higher than for Chapman-Jouguet detonations.

The concept for the utilization of the rotating detonation wave in hypersonic and/or rocket propulsors requires further analysis before its pertinence may be determined.

3.1.5 Miscellaneous Concepts

There are many miscellaneous concepts related to the improvement of airbreathing propulsors, i.e.:

- Regenerative techniques
- Thermodynamic cycle improvements such as; more efficient compression (diffusers), more efficient expansion (nozzles) and more efficient combustion.
- Fuel injection techniques that promote mixing and/or reduce flow stagnation pressure losses
- Pyrofluoric fuels
- Combustion stabilization techniques
- Special fuels such as those which should combine with atmospheric nitrogen.

3.2 Analysis of the Potential Interior Performance of Oblique Detonation Wave Ramjet Propulsion

The one dimensional compressible flow processes, discussed in Section 3.1, can be further characterized and usefully classified for the case of normal detonation waves (Ref 6) by the introduction of the functions:

$$f \equiv \frac{2(\gamma+1)}{(M_1{}^2-1)} M_1{}^2\left(\frac{Q}{C_pT_1}\right) \qquad \text{(Definition)} \tag{3.2.1}$$

and

$$F \equiv 1+\sqrt{1-f} \qquad \text{(Definition)} \tag{3.2.2}$$

into the conservation equations, subscript (1) denoting conditions immediately in front of the normal detonation wave. This results in the following equations for pressure, density, temperature, and downstream Mach number of:

$$\frac{P_2}{P_1} = 1+\frac{\gamma F}{\gamma+1}(M_1{}^2-1) \tag{3.2.3}$$

$$\frac{\rho_2}{\rho_1} = \frac{1}{1-\dfrac{F}{\gamma+1}\dfrac{(M_1{}^2-1)}{M_1{}^2}} \tag{3.2.4}$$

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6 Adamson, T.C. and Morrison, R.B, "On the Classification of Normal Detonation Waves" Jet Propulsion, August 1955

$$\frac{T_2}{T_1} = \left\{1+ \frac{\gamma F}{\gamma+1} (M_1{}^2-1)\right\} \left\{1- \frac{F}{\gamma+1} \frac{(M_1{}^2-1)}{M_1{}^2}\right\} \qquad 3.2.5$$

$$M_2 = \sqrt{\frac{(\gamma+1-F)(M_1{}^2-1) + (\gamma+1)}{\gamma F(M_1{}^2-1) + (\gamma+1)}} \qquad 3.2.6$$

Subscript (2) denotes conditions immediately downstream of the normal detonation.

For adiabatic flows, $Q = 0$, and $f = 0$, making $F = 2$; i.e., the case of a normal shock wave. For non-adiabatic flows and Q positive, $0<f<1$, defines the range of possible solutions consistent with the one dimensional conservation equations. $f = 1$ further defines the case of limiting heat addition (thermal choking) and the Chapman-Jouguet (C-J) conditions. Values for f between 0 and 1.0 represent cases of non-limiting heat addition and strong detonations.

The above equations for pressure, density, temperature, and downstream Mach number can be applied to oblique waves if these equations are written in terms of the normal components to the wave. For a wave angle of β and deflection angle of Θ these equations are then:

$$\frac{P_2}{P_1} = 1+ \frac{\gamma F}{\gamma+1} (M_1{}^2\sin^2\beta-1) \qquad 3.2.7$$

$$\frac{\rho_2}{\rho_1} = \frac{1}{1 - \frac{F}{\gamma+1}\frac{(M_1{}^2\sin^2\beta - 1)}{M_1{}^2\sin^2\beta}} \qquad 3.2.8$$

$$\frac{T_2}{T_1} = \left\{1 + \frac{\gamma F}{\gamma+1}(M_1{}^2\sin^2\beta - 1)\right\}\left\{1 - \frac{F}{\gamma+1}\frac{(M_1{}^2\sin^2\beta - 1)}{M_1{}^2\sin^2\beta}\right\} \qquad 3.2.9$$

$$M_2\sin(\beta - \Theta) = \sqrt{\frac{(\gamma+1-F)(M_1{}^2\sin^2\beta - 1) + (\gamma+1)}{\gamma F(M_1{}^2\sin^2\beta - 1) + (\gamma+1)}} \qquad 3.2.10$$

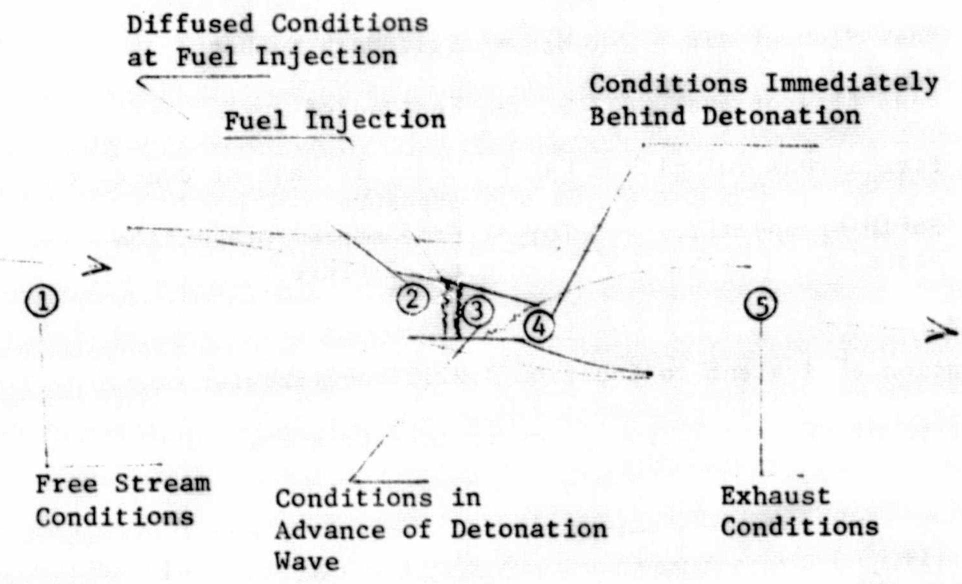
$$\tan\Theta = \frac{\frac{F}{\gamma+1}\left(1 - \frac{1}{M_1{}^2\sin^2\beta}\right)\tan\beta}{1+\tan^2\beta\left[1 - \frac{F}{\gamma+1}\left(1 - \frac{1}{M_1{}^2\sin^2\beta}\right)\right]} \qquad 3.2.11$$

The above oblique wave equations are classified in the identical manner of normal waves i.e., $F = 2$ describes oblique shock waves, $F = 1$ describes oblique Chapman-Jouguet detonation waves, and $1<F<2$ strong detonation waves.

The interior performance of the oblique detonation wave ramjet can be perspicuously analysed in terms of the stagnation pressure losses incurred for each part of the thermodynamic cycle, i.e., diffuser, fuel injection, detonation, and nozzle stagnation pressure losses.

The nomenclature used in the analysis that follows is shown in Figure 3.2.1.

FIGURE 3.2.1 Nomenclature for Oblique Detonation Wave Analysis



3.2.1 Overall Thrust Coefficient

The equation for the overall thrust coefficient, C_T, is derived from the conservation equations in terms of the entrance and exit Mach Numbers. For a ramjet that exhausts to atmospheric pressure:

$$C_T \equiv \frac{Fx}{\dot{M}aV_1} = \left\{1+\frac{\dot{M}a}{\dot{M}f}\right\} \frac{M_5}{M_1} \left\{1+\frac{Q}{CpTo_1}\right\}^{\frac{1}{2}} \left\{\frac{1+\frac{\gamma-1}{2} M_1{}^2}{1+\frac{\gamma-1}{2} M_5{}^2}\right\}^{\frac{1}{2}} -1 \qquad 3.2.12$$

Where:

Fx	=	Thrust	M_1	=	Entrance Mach number
$\dot{M}a$	=	Mass Flow of air	M_5	=	Exit Mach Number
$\dot{M}f$	=	Mass flow of fuel	Q	=	Heat added
V_1	=	Free stream velocity	Cp	=	Specific heat at constant pressure
γ	=	Ratio of specific heats	To_1	=	Free stream stagnation temperature

An equation equivalent to Equation 3.2.12 can likewise be derived in terms of the stagnation pressure loss to give:

$$C_T = \left\{1+\frac{\dot{M}f}{\dot{M}a}\right\} \left\{\frac{\left(\frac{Po_5}{P_5}\right)^{\frac{\gamma-1}{\gamma}}-1}{\left(\frac{Po_1}{P_1}\right)^{\frac{\gamma-1}{\gamma}}-1}\right\}^{\frac{1}{2}} \left\{1+\frac{Q}{CpTo_1}\right\}^{\frac{1}{2}} \left\{\frac{Po_1}{Po_5}\right\}^{\frac{\gamma-1}{2\gamma}} -1 \qquad 3.2.13$$

Where:

P_1 = Free stream static pressure

P_{o1} = Free stream stagnation pressure

P_5 = Exhaust static pressure

P_{o5} = Exhaust stagnation pressure

Equations 3.2.12 and 3.2.13 define the thrust coefficient in terms of the captured incoming momentum.

The thrust coefficient is also commonly defined in terms of the free stream dynamic pressure and a characteristic cross sectional area, i.e., as:

$$\frac{Fx}{\frac{\rho_\infty}{2} Ac V_\infty^2}$$

Where:

Fx = Thrust

ρ_∞ = Free stream static density

Ac = Characteristic area of combustor/inlet

V_∞ = Free stream velocity

$\frac{\rho_\infty}{2} V_\infty^2$ = Free stream dynamic pressure

Thrust coefficients defined by equations 3.2.12 and 3.2.13 are converted to thrust coefficients based upon free stream dynamic pressure by multiplying by two and by the ratio of the true capture area to the combustion chamber characteristic area.

The overall stagnation pressure ratio loss is then:

$$\frac{Po_1}{Po_5} = \underbrace{\frac{Po_1}{Po_2}}_{\text{Diffuser}} \times \underbrace{\frac{Po_2}{Po_3}}_{\text{Fuel Injection}} \times \underbrace{\frac{Po_3}{Po_4}}_{\text{Detonation Wave}} \times \underbrace{\frac{Po_4}{Po_5}}_{\text{Nozzle}} \qquad 3.2.14$$

3.2.2 Diffuser Losses

Diffuser stagnation pressure losses are based on multi-shock intake flows that consist of equal strength oblique shocks.

3.2.3 Fuel Injection Losses

Fuel injection stagnation pressure losses depend strongly upon injection techniques.

Injection perpendicular to the flow will induce losses from two causes a) the induced shocks from the injection jet and b) the stream energy required to bring the fuel mass flows up to stream velocity. This technique does, however, promote good mixing.

Injection downstream parallel to the airflow alleviates the problems associated with perpendicular injection, this, however, introduces a new problem, mixing lengths are long.

Injection downstream at small oblique angles appears to be the best compromise from the consideration of overall system performance.

The aerothermopressor technique of injecting cryogenic fuels far upstream and from the diffuser walls offers a technique that could provide:

- Thermal protection
- Long mixing lengths
- Air compression
- Boundary layer control

Inasmuch as these stagnation pressure losses incurred from fuel injection could vary from large values to negligible, or possibly gains (aerothermopressor technique) this loss/gain was taken to be 0.

3.2.4 Detonation Losses

Detonation stagnation pressure losses were evaluated from the oblique wave equation:

$$\frac{Po_4}{Po_3} = \left[1+ \frac{\gamma F}{\gamma+1} (M_3{}^2 \sin^2\beta - 1)\right] \left[\frac{1+ \frac{\gamma-1}{2} M_4{}^2}{1+ \frac{\gamma-1}{2} M_3{}^2}\right]^{\frac{\gamma}{\gamma-1}} \qquad 3.2.15$$

Where: F = Wave classification number

$F = 1$ Chapman-Jouguet detonation

$F = 2$ Normal shock

$1 < F < 2$ Strong detonation

β = Wave angle

M_3 = Mach number forward of the detonation

M_4 = Mach number after the detonation

$\frac{Po_4}{Po_3}$ = Stagnation pressure loss across the oblique detonation

The wave angle, β, and M_4 were obtained from the oblique wave equations:

$$\tan\Theta = \frac{\frac{F}{\gamma+1}\left(1-\frac{1}{M_3^2\sin^2\beta}\right)\tan\beta}{1+\tan^2\beta\left[1-\frac{F}{\gamma+1}\left(1-\frac{1}{M_3^2\sin^2\beta}\right)\right]} \qquad 3.2.16$$

and

$$M_4 = \frac{1}{\sin(\beta-\Theta)}\sqrt{\frac{(\gamma+1-F)(M_3^2\sin^2\beta-1)+(\gamma+1)}{\gamma F(M_3^2\sin^2\beta-1)+(\gamma+1)}} \qquad 3.2.17$$

Where: Θ = Deflection angle

For Chapman-Jouguet detonations where F = 1 Equation 3.2.6 reduces to:

$$M_4 = \frac{1}{\sin(\beta-\Theta)}$$

Inasmuch as:

$$M_3\sin\beta = M_{DC-J} \qquad 3.2.18$$

where: M_{DC-J} = the Mach number of a Chapman-Jouguet detonation, all that is required for solution of the above set of equations are the appropriate values for the Mach number of Chapman-Jouguet detonations. A semi-empirical equation which best correlated experimental data for eight fuel-oxygen combinations for a wide range of mixture ratios is:

$$M_{DC-J} = 2.37\Phi^{\frac{1}{3.05}} \qquad 3.2.19$$

Where:

$$\Phi \equiv \frac{M_3}{M_4}\frac{Q}{C_pT_3} \qquad 3.2.20$$

and:

M = Molecular weight

Q = Heat of combustion at 20^0C

Subscripts (3) and (4) denote conditions before and after the detonation front.

Equations 3.2.19 and 3.2.20 check well with the experimental detonation data for hydrogen-air mixtures.

3.2.5 Nozzle Expansion Losses

The losses associated with nozzle expansion are twofold:

- Losses from overexpansion or underexpansion
- Thrust vector losses resulting from non axial exhaust flow

For this analysis it was assumed that these losses were negligible and that the expansion to atmospheric pressure from station 4 to station 5 was isentropic.

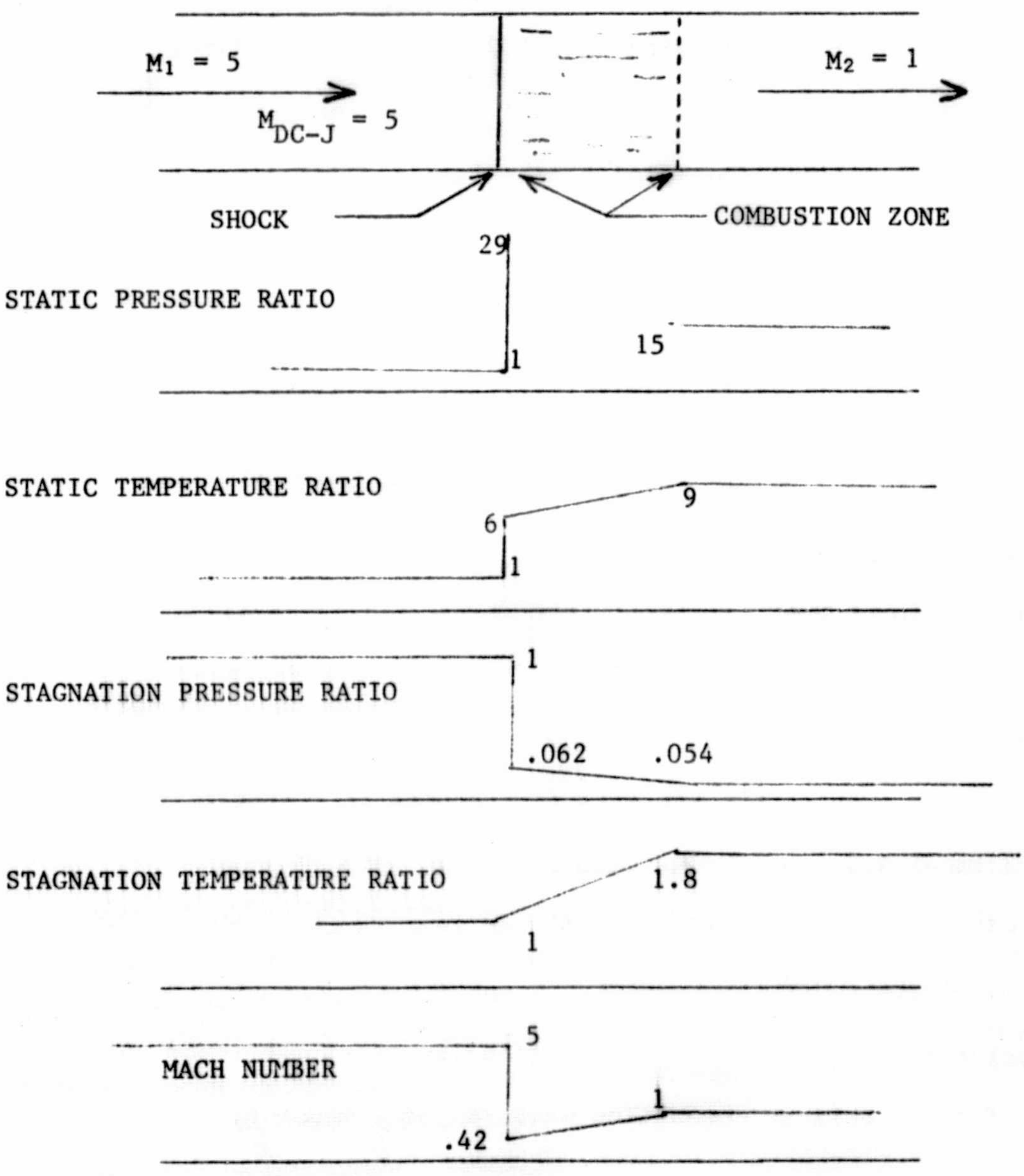
3.3 Multiple Oblique Shock Analysis

The analysis of the oblique detonation wave ramjet combustion differs in some respects from the analysis of subsonic combustion ramjets and the supersonic diffusive burning ramjets. In the latter cases, the compression or diffusion process is commonly treated separately from the combustion process. The combustion process, in turn, is treated as a heat addition to the airstream under controlled conditions of area ratio and pressure throughout the length of the combustion process.

In the case of detonative combustion it is impossible to separate the compression supplied from the shock component of the detonation from that of the compression supplied by the diffuser. Furthermore the heat addition to the airstream is confined to a very short combustion length of essentially constant area, and decreasing static pressure.

Figure 3.3.1 illustrates the variation of various flow parameters for a stoichiometric hydrogen-air Chapman-Jouguet detonation wave. It is noted that the normal shock wave component of the detonation provides a static pressure compression ratio of 29 to 1 prior to combustion. The combustion process takes place subsonically reducing the static pressure ratio from 29 to 15. It is noted further that the stagnation pressure ratio loss associated with the shock component of the detonation is much more significant than the loss incurred for the combustion process. Reduction of shock losses can be attained by use of oblique shocks for the diffuser and oblique detonation waves for the combustor.

FIGURE 3.3.1 Variation of Pressure, Temperature, & Mach Number through an Idealized Chapman-Jouguet Detonation Wave of a Stoichiometric Hydrogen-Air Mixture.



The compression to be realized from a number of equal strength oblique shocks is shown in Figure 3.3.2. The case of N = 1 is one oblique shock that corresponds to a normal shock possessing the normal component of that flight Mach number. N = 2, 3, 4, etc., represents the number of oblique shocks wiht a normal component of the flow Mach number equal to Mn. A compression ratio of approximately 20 could for example be attained from one oblique shock with Mn = 4.2, two oblique shocks with Mn = 2.0, three oblique shocks with Mn = 1.57, four oblique shocks with Mn = 1.4, or five oblique shocks with Mn = 1.3.

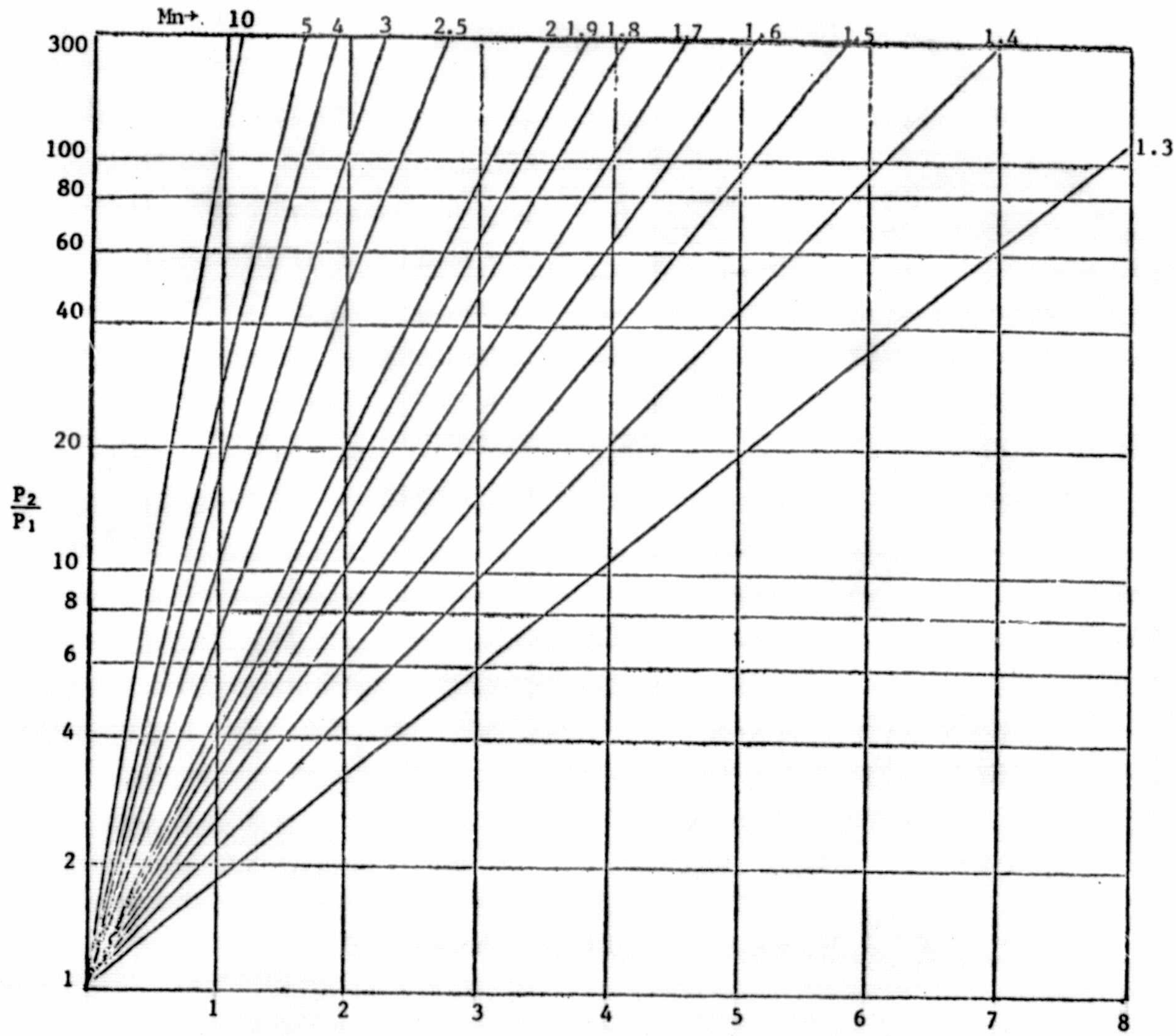
The stagnation pressure ratio losses sustained from multiple oblique shocks are shown in Figure 3.3.3. Corresponding to the example above for a compression ratio of approximately 20, one shock with Mn = 4.2 would entail a stagnation pressure loss ratio of 0.16, two oblique shocks with Mn = 2 a loss ratio of 0.51, three oblique shocks with Mn = 1.57 a loss ratio of 0.75, and four oblique shocks with Mn = 1.4 a loss ratio of 0.82.

The static temperature ratios realized from multiple oblique shocks are shown in Figure 3.3.4. Again, for the example of a static pressure ratio of 20, one shock with Mn = 4.2 results in a static temperature ratio of 4.5, two shocks with Mn = 2 a ratio of 2.9, three shocks with Mn = 1.57 a ratio of 2.6, and four shocks with Mn = 1.4 a ratio of 2.5.

The temperature ratio resulting from diffusion is a most important consideration for the oblique detonation wave ramjet combustor.

FIGURE 3.3.2 Compression Ratio as a Function of Mn & Number of Equal Strength Oblique Shocks

Mn = Mach Number Normal to the Oblique Shocks



N = Number of Equal Strength Oblique Shocks

FIGURE 3.3.3 Stagnation Pressure Ratio Loss as a Function of Mn & Number of Equal Strength Shocks

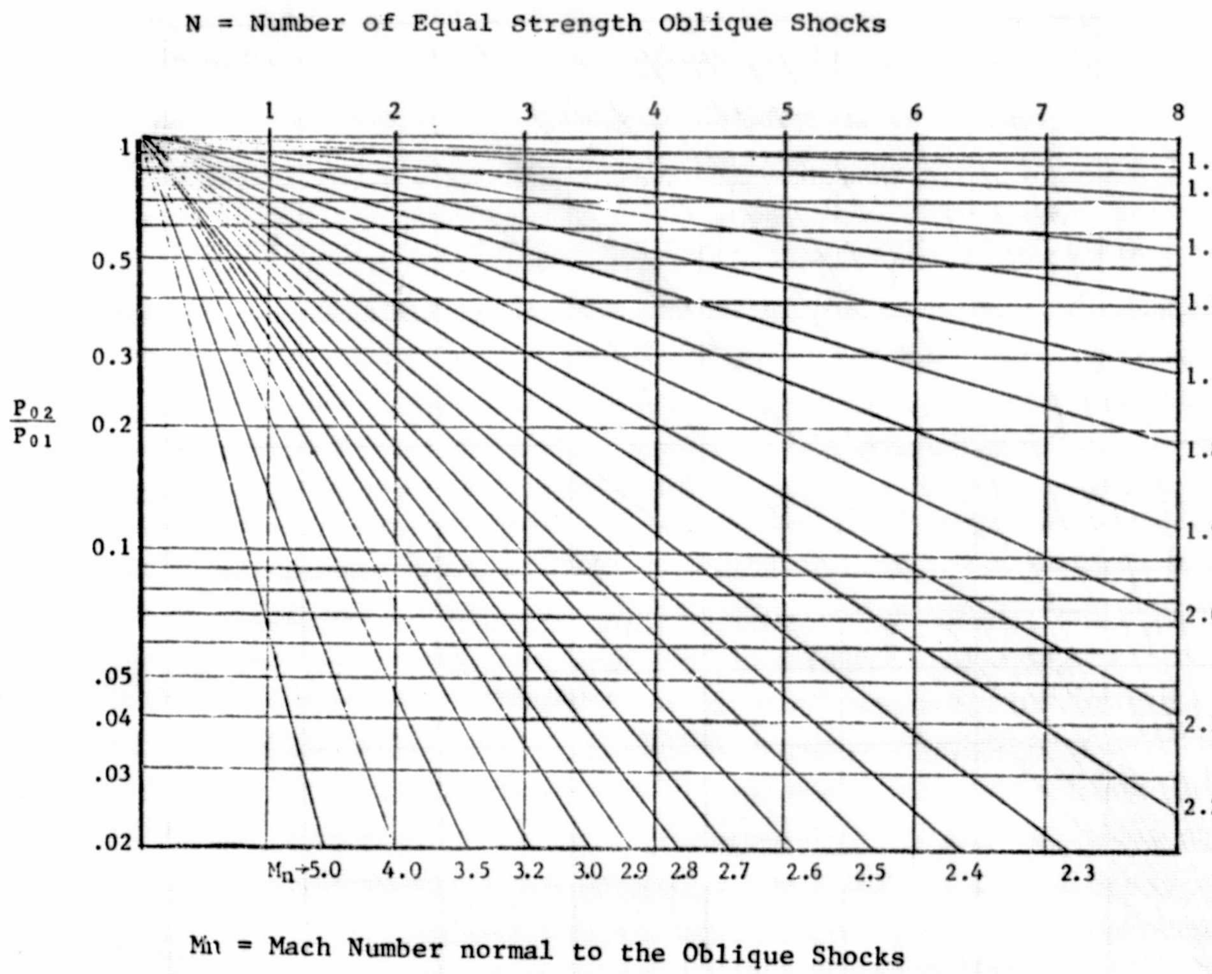
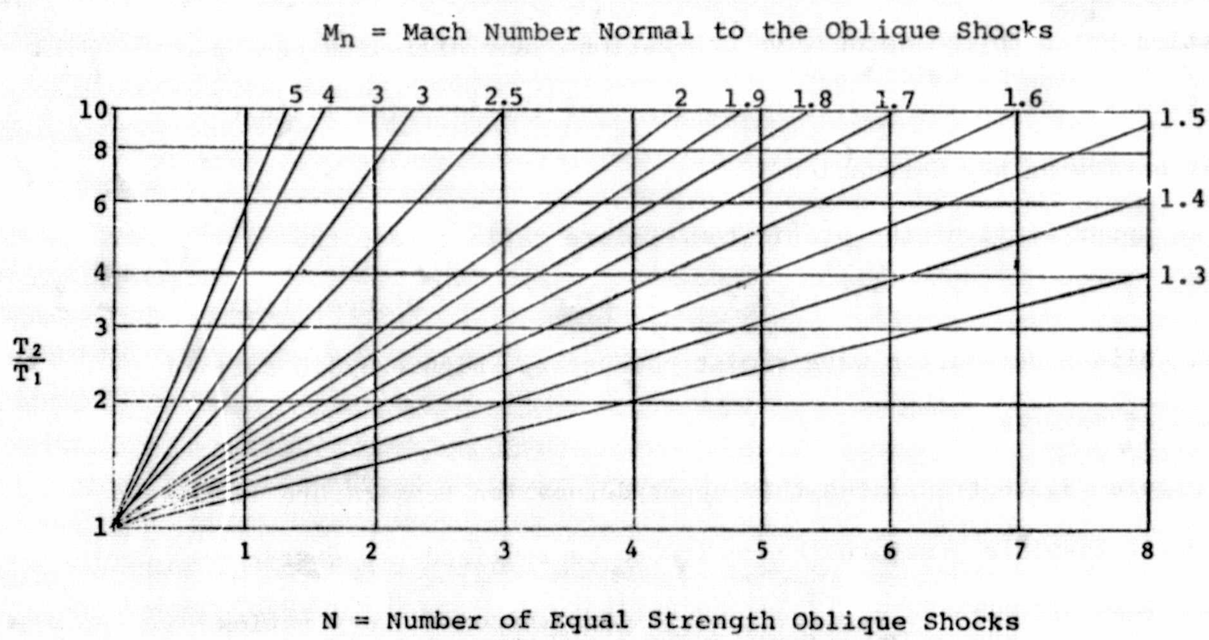


FIGURE 3.3.4 Static Temperature as a Function of Mn & Number of Equal Strength Oblique Shocks



Two interrelated performance factors must be considered, i.e.:

- The limits of detonability became narrower at higher temperatures and at some high temperature the fuel-air mixture cannot be detonated.

- The Mach number of detonation is decreased at higher temperatures resulting in decreased stagnation pressure losses.

In addition to the above two performance factors, there is the practical consideration of the flammability limits of the fuel, which cannot be greatly exceeded without incidence of diffusive combustion prior to the detonative combustion. Any diffusive combustion likewise raises the static temperature of the incoming mixture thereby narrowing the detonation limits.

An upper limit of the static temperature ratio across the diffuser of four to five is indicated to define zones of operation for the oblique detonation wave ramjet combustor. Figure 3.3.5 defines the zone of interest.

Figure 3.3.6 translates this upper detonation temperature limit in terms of possible pressure recoveries to be realized and static pressure recoveries to be realized. Static pressure recovery ratios of 60 to 80 appear to be realizable within the detonability limits of most fuel-air mixtures.

The effect of the upper detonation temperature limit on the stagnation pressure loss ratio is shown in Figure 3.3.7. It is to be noted that the region of interest lies above the temperature limit line.

FIGURE 3.3.5 Indicated Upper Temperature Limit for Detonation

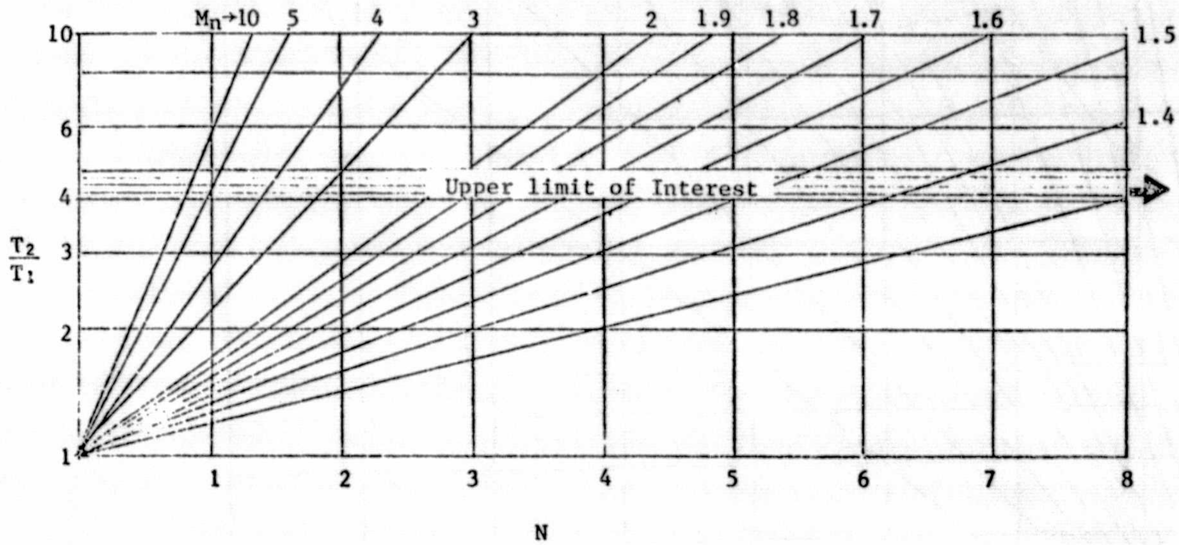


FIGURE 3.3.6 Effect of Detonation Temperature Limit Upon Possible Pressure Recovery

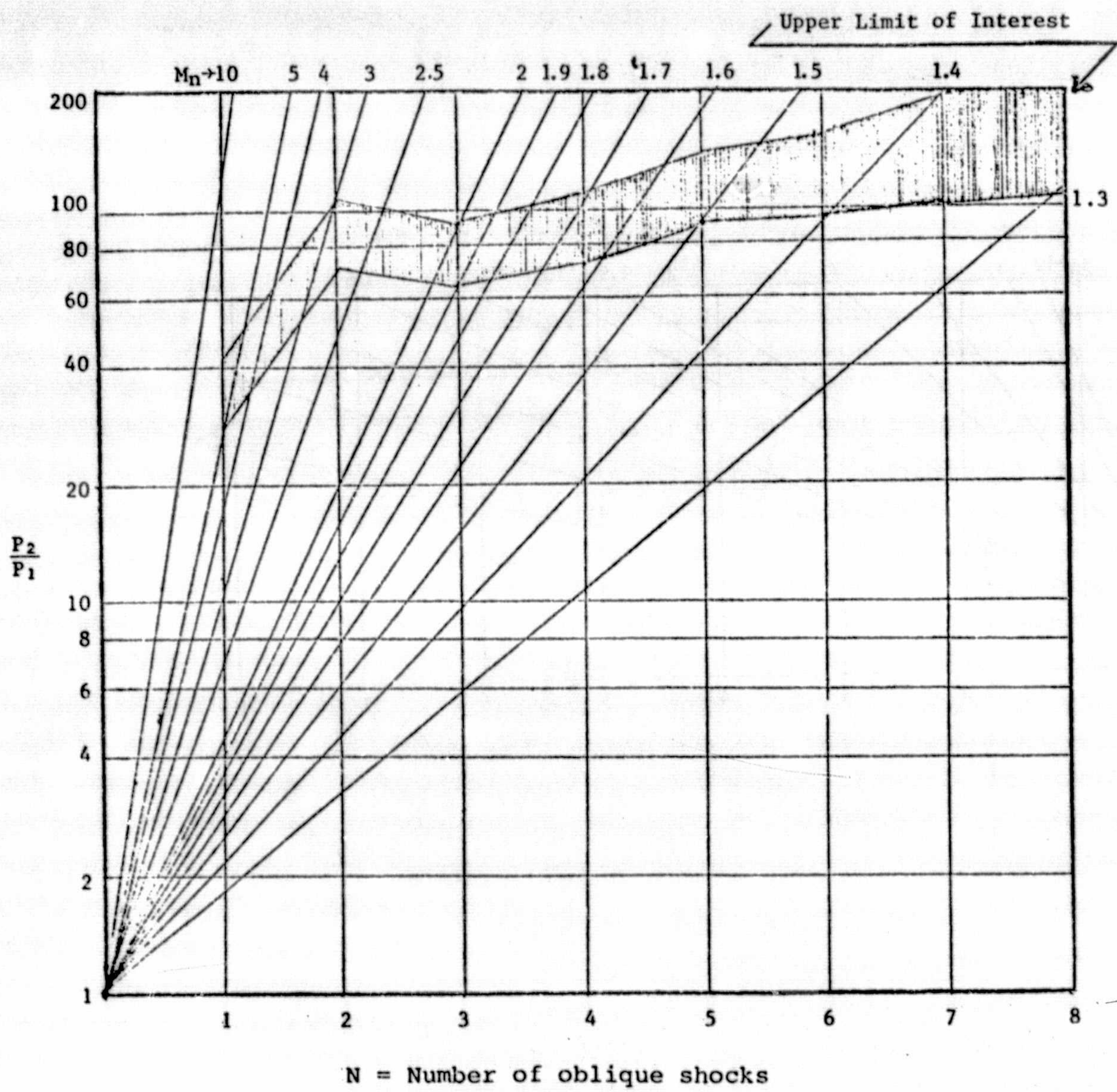


FIGURE 3.3.7 Effect of Temperature Detonation Limit Upon Required Multi-Shock Performance

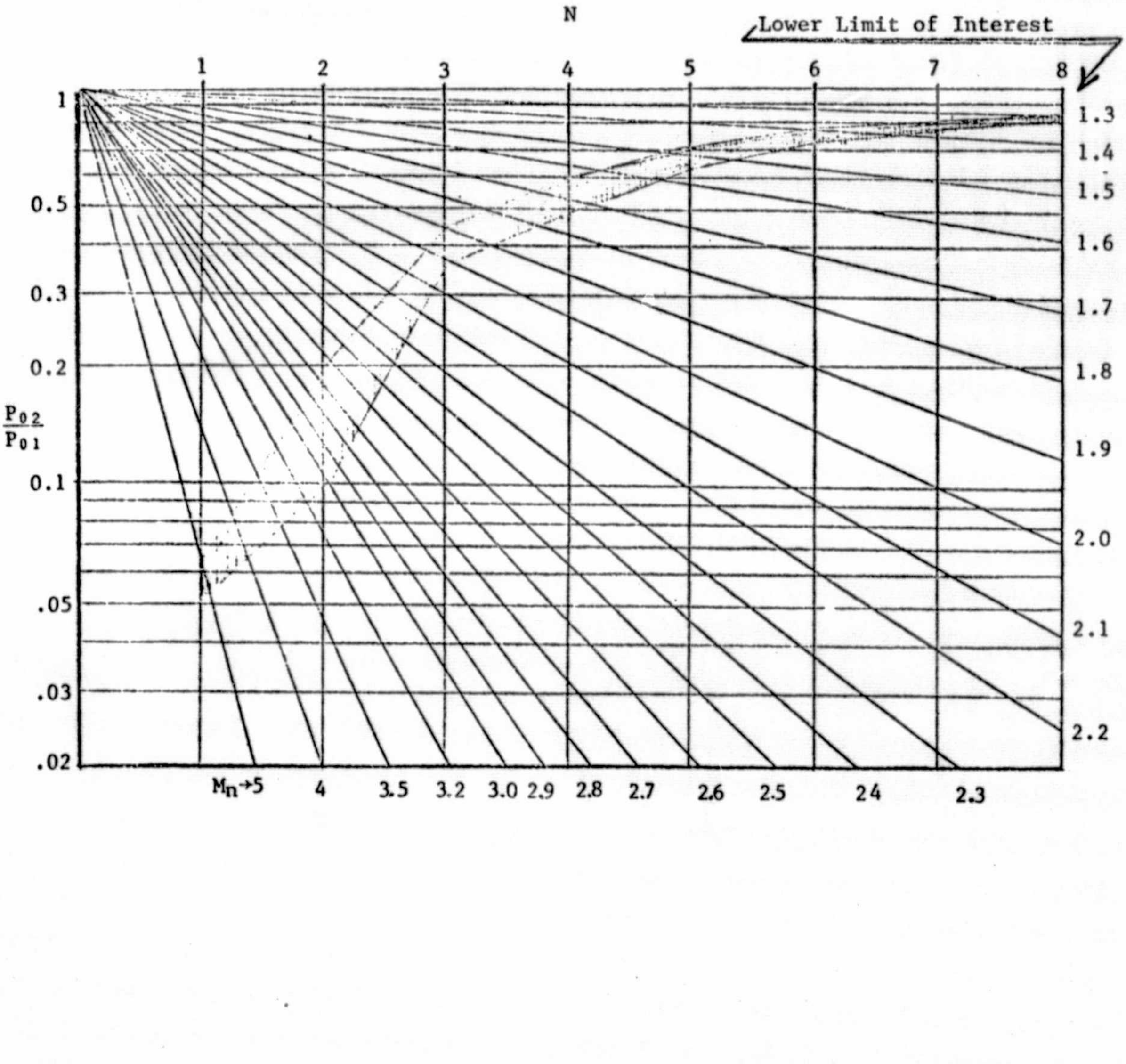


Figure 3.3.6 and 3.3.7 delineate a region of prime diffuser interest, that is the best compromise of simple geometries, acceptable stagnation pressure losses and good pressure recovery, that is in the neighborhood of three equal strength oblique shocks.

4.0 Performance Evaluation of the Oblique Detonation Wave Ramjet

The technique of analysis for the oblique detonation wave ramjet is discussed in section 3.2. The evaluation of potential performance is made in terms of thrust coefficients and specific impulses both of which are functions of (1) the flight Mach number, (2) the heat release and (3) the stagnation pressure losses of the propulsion system.

In general it may be stated that the deleterious effects of stagnation pressure losses upon Isp and C_T are magnified at low supersonic flight Mach numbers, circa $M_1<5$. At high supersonic flight Mach numbers, circa $M_1>8$, these deleterious effects are ameliorated.

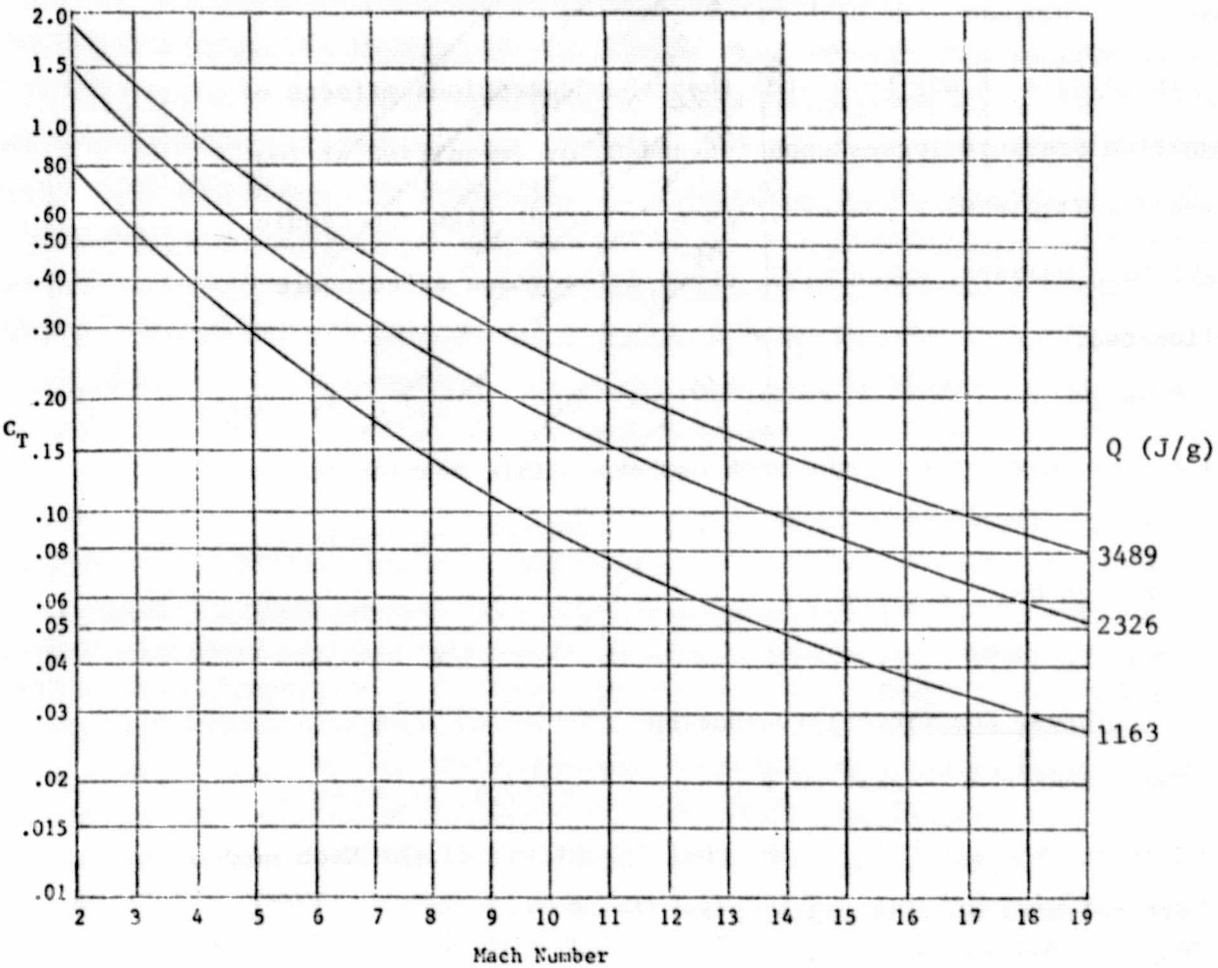
Likewise the effects of heat release upon Isp and C_T are magnified at low supersonic flight Mach numbers. This degradation of Isp and C_T is accentuated for small heat releases and conversely are lessened for high heat releases.

4.1 Thrust Coefficient Evaluation

The potential or ideal performance of a ramjet is shown in Figure 4.1.1 as a graph of thrust coefficient vs. flight Mach number for heat releases of 1163, 2326, 3489 J/g of air.

These curves do not incorporate the correction factor for fuel-air ratio of $(1+\dot{M}_f/\dot{M}_a)$ in the equation for thrust coefficient

FIGURE 4.1.1 Ideal Thrust Coefficient vs. Mach Number for Various Heat Additions



(Equation 3.2.1). This correction factor, depending upon the fuel and its heat release for a specific fuel-air ratio, is a small positive correction that is neglected in the interest of preserving generality.

The ideal thrust coefficient for the same heat release decreases with increasing flight Mach numbers. At the higher flight Mach numbers, circa $M_1 > 5$, this rate varies approximately as $M_1^{(-1.7)}$.

The effects of heat release are shown in Figure 4.1.2, the ideal thrust coefficient being a linear function of the heat release for fixed flight Mach numbers.

Figures 4.1.3, 4.1.4, and 4.1.5 show the effects upon performance of stagnation pressure loss for heat releases of 1163, 2326, and 3489 J/g of air.

The importance of maintaining high diffuser efficiencies at the lower flight Mach numbers is evidenced by the sharp dropoff in performance that occurs at flight Mach numbers slightly lower than those of the maxima for the thrust coefficients. The performance zone at flight Mach numbers larger than those for the optima is the region of stable and efficient ramjet operation.

Figures 4.1.6, 4.1.7, and 4.1.8 demonstrate more clearly the effects upon performance of stagnation pressure losses at low flight Mach numbers.

A ramjet, to operate at $M_1 = 3$, requires that a Po_5/Po_1 in excess of 0.5 be maintained to accomodate a range of heat releases larger than 1163 J/g of air. At $M_1 = 4$ a Po_5/Po_1 in excess of 0.1 would be required.

FIGURE 4.1.2 Ideal Thrust Coefficient as a Function of Heat Release for Various Flight Mach Numbers

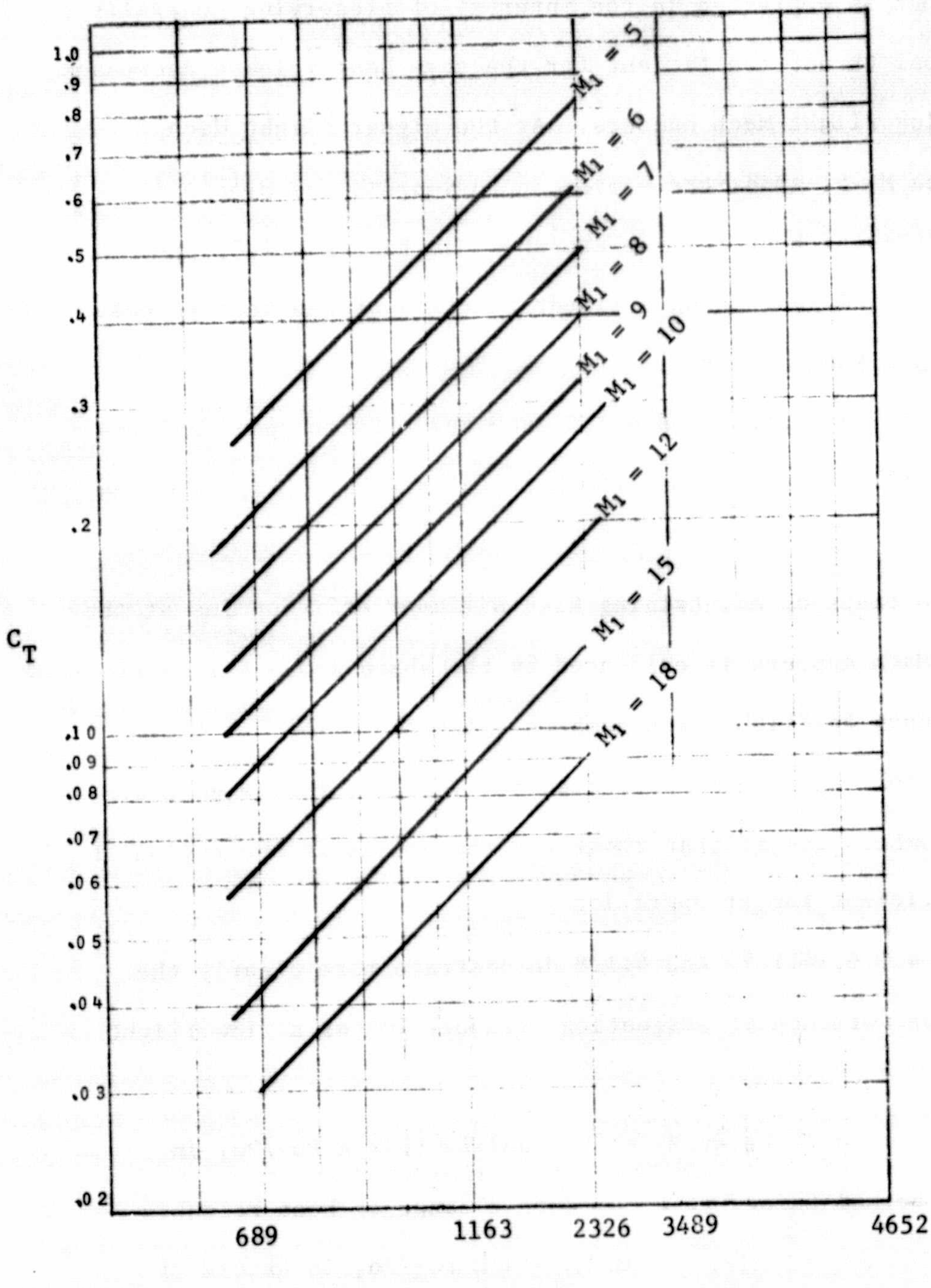


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FIGURE 4.1.3 C_T vs. Mach Number for Various Values of Po_5/Po_1 & Q = 1163 J/g

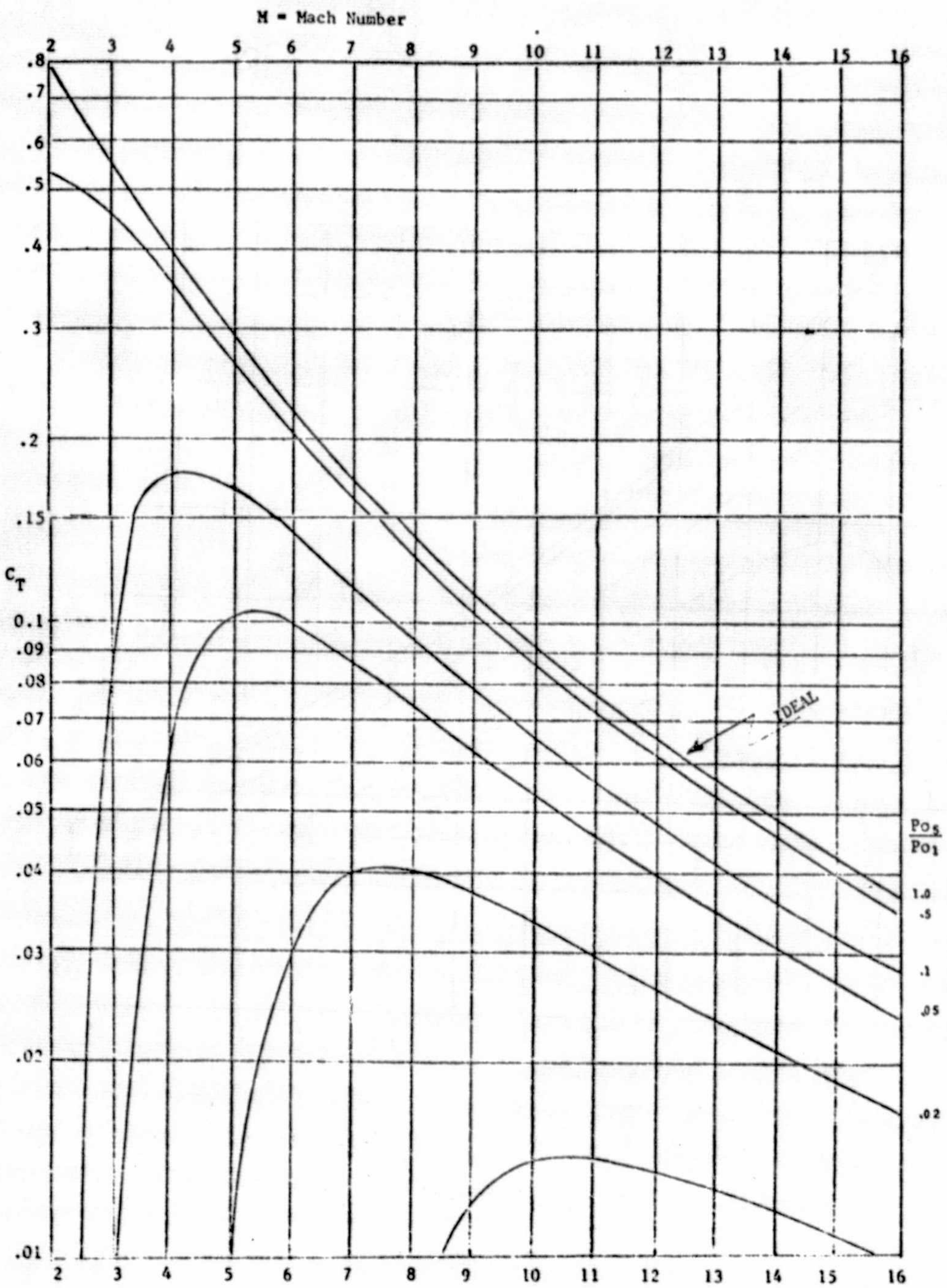


FIGURE 4.1.4 C_T vs. Mach Number for Values of Stagnation Pressure Losses for Q = 2326 J/g

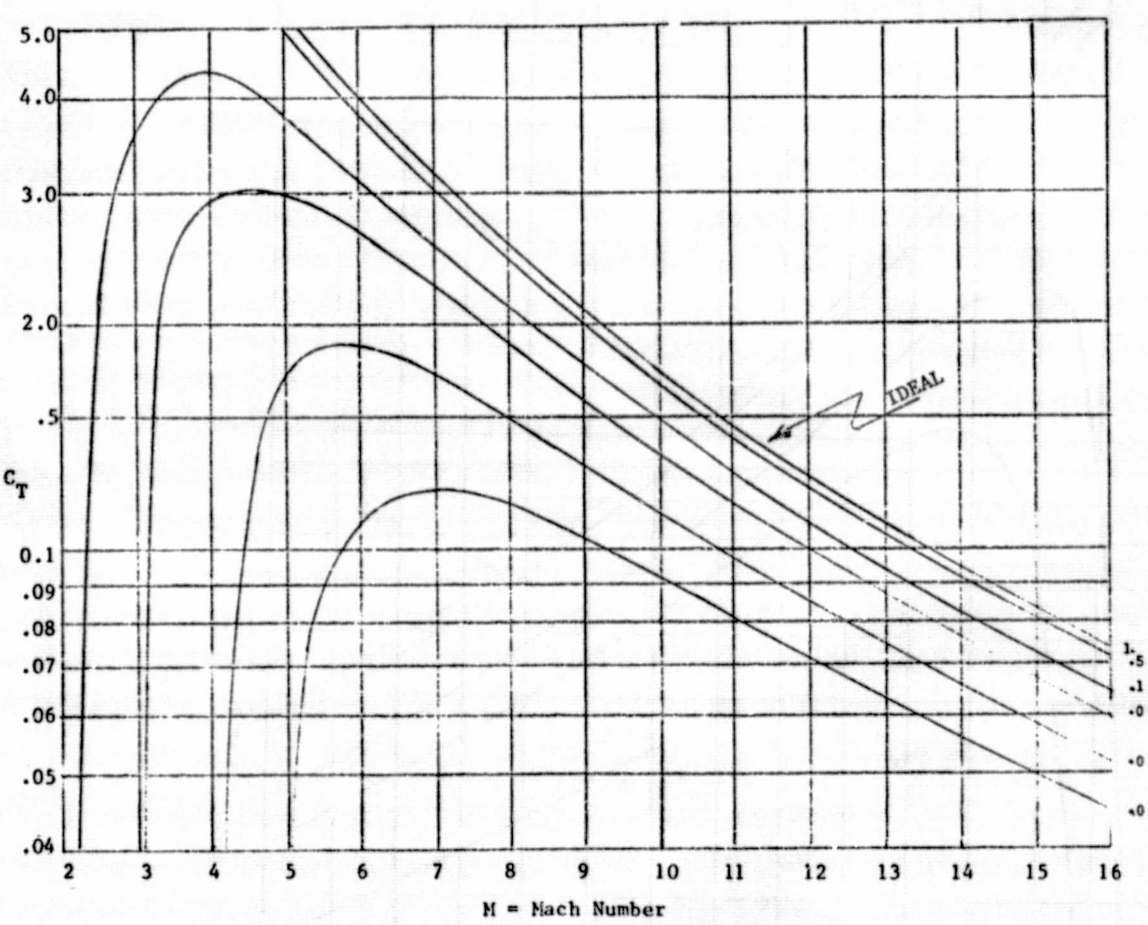


FIGURE 4.1.5 Thrust Coefficient, C_T, as a Function of Mach Number for Various Overall Stagnation Pressure Loss Ratios for Q = 3489 J/g

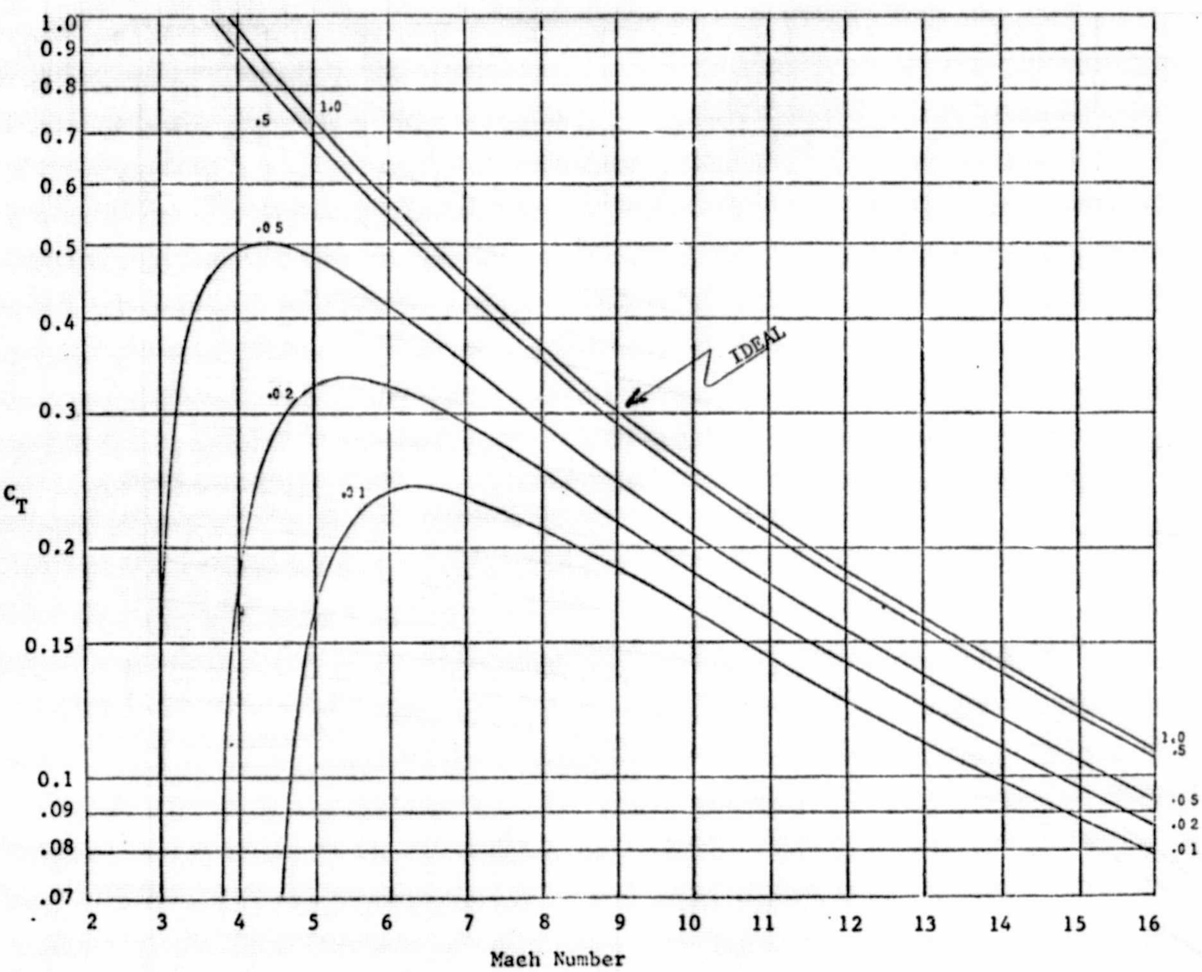


FIGURE 4.1.6 Thrust Coefficient vs. Stagnation Pressure Ratio Loss for Various Flight Mach Numbers and Q = 1163 J/g Air

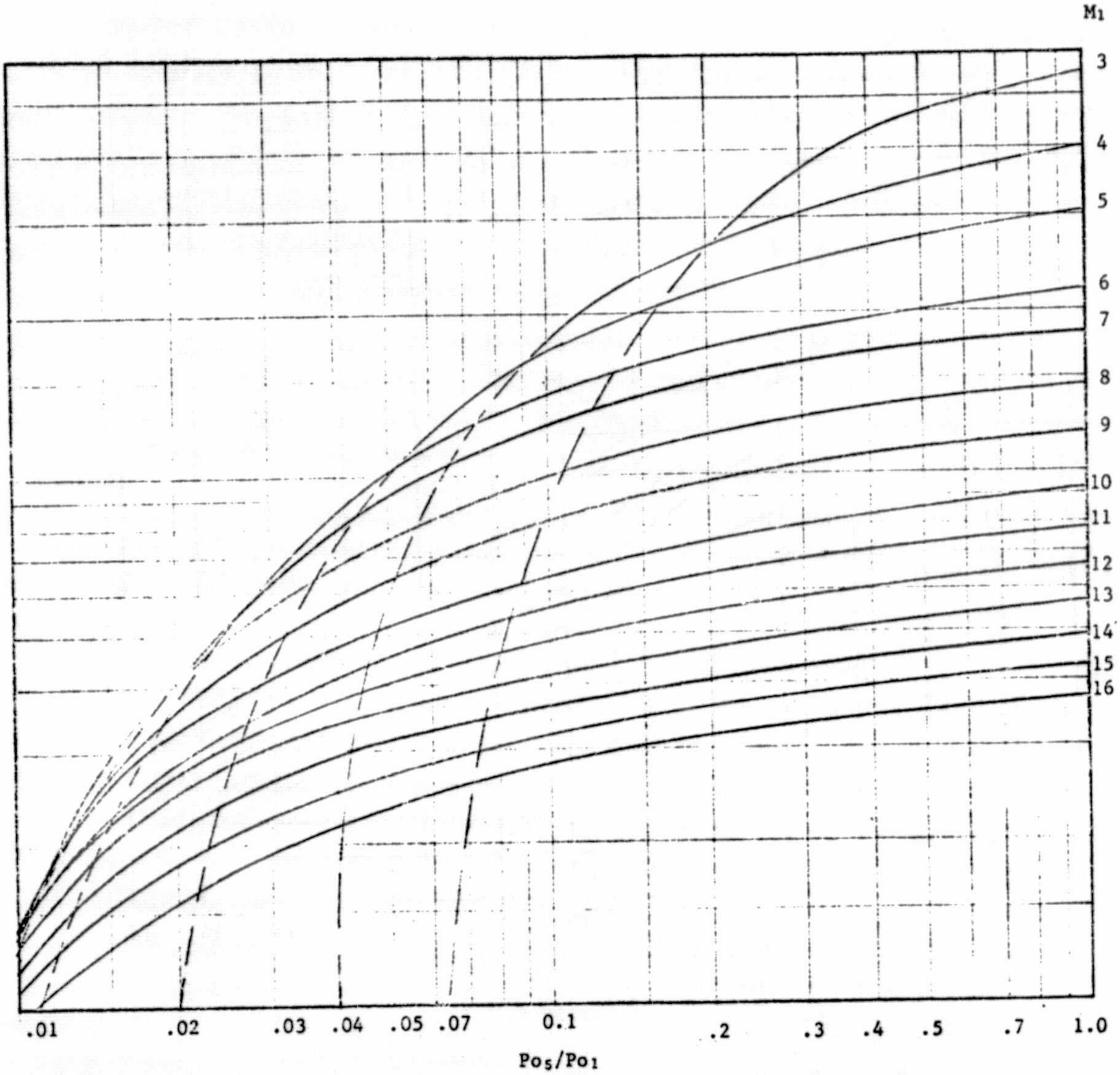


FIGURE 4.1.7 Thrust Coefficient vs. Stagnation Pressure Ratio Loss for Various Flight Mach Numbers and Q = 2326 J/g Air

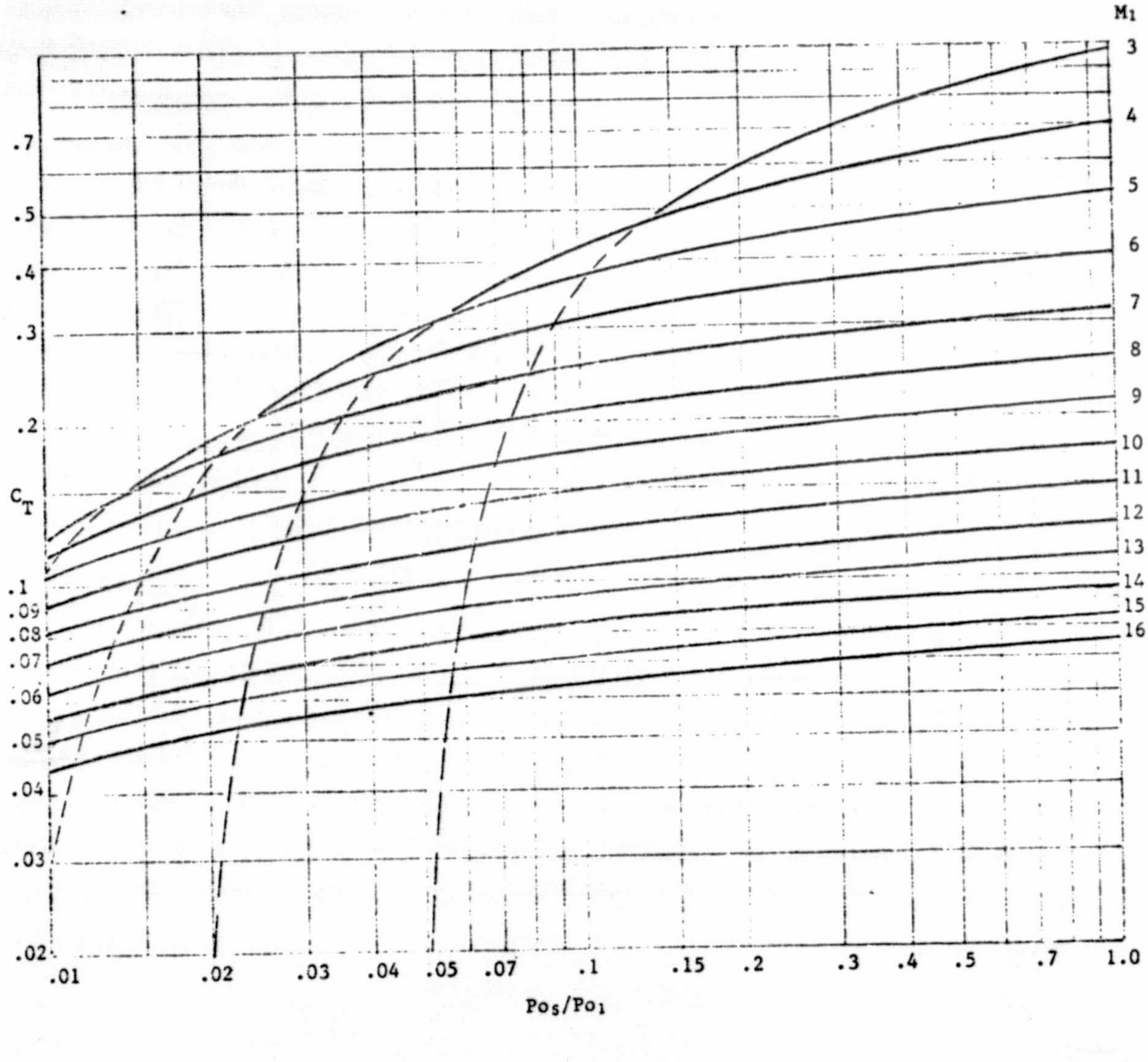
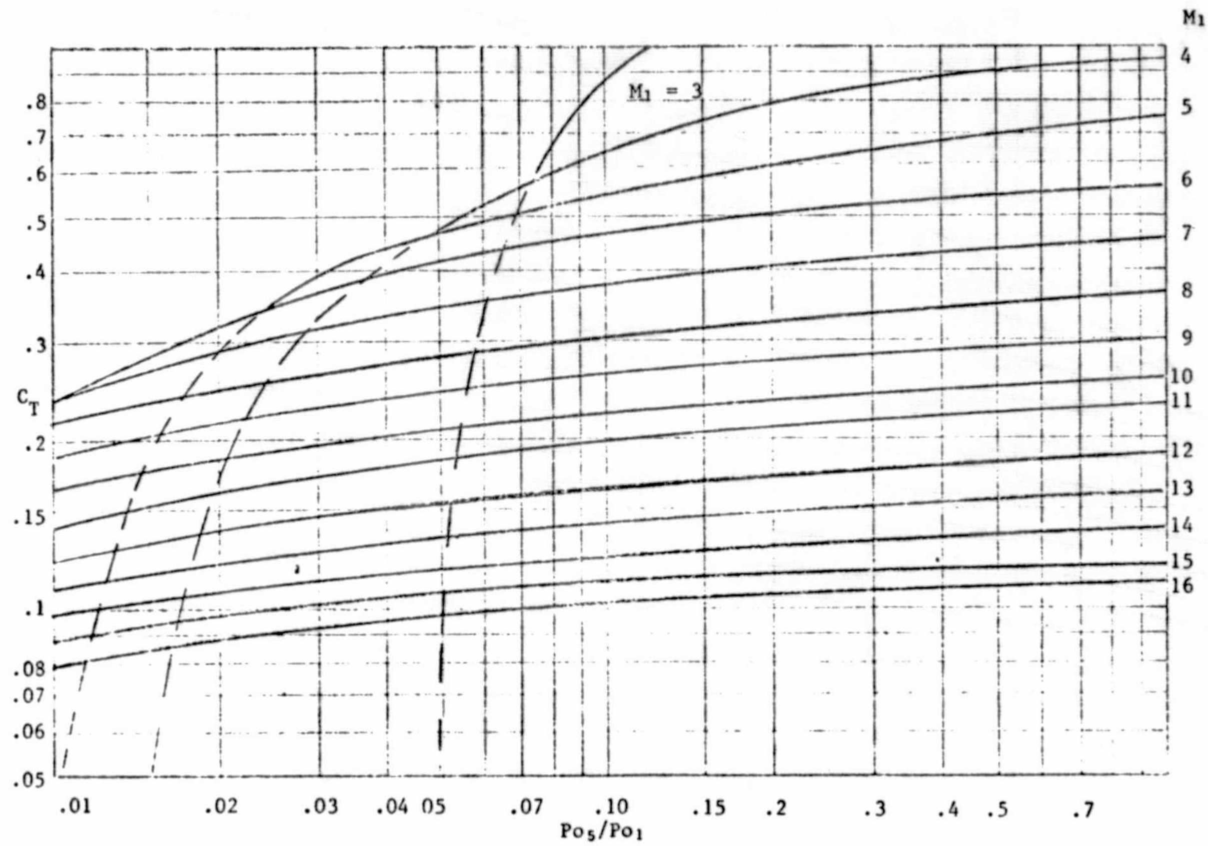


FIGURE 4.1.8 Thrust Coefficient vs. Stagnation Pressure Ratio Loss for Various Flight Mach Numbers and Q = 3489 J/g Air



4.2 Specific Impulse Evaluation

Figures 4.2.1, 4.2.2, and 4.2.3 show the effects of stagnation pressure loss and heat release upon the specific impulse; these effects being the same as for the thrust coefficient.

It is noted that stagnation pressure losses are most critical for the case of low flight Mach numbers with small heat releases.

Proper balancing of the diffuser shock losses against those of the oblique detonation wave losses should however, result in propulsion systems conservatively yielding specific impulses between two thirds and three quarters of the ideal specific impulse; this throughout a range of flight numbers extending from M_1 = 6 to 8 to M_1 = 16 to 18.

FIGURE 4.2.1 Specific Impulse in Seconds vs. Flight Mach Number for Various Overall Stagnation Pressure Ratio Losses for Q = 1163 J/g

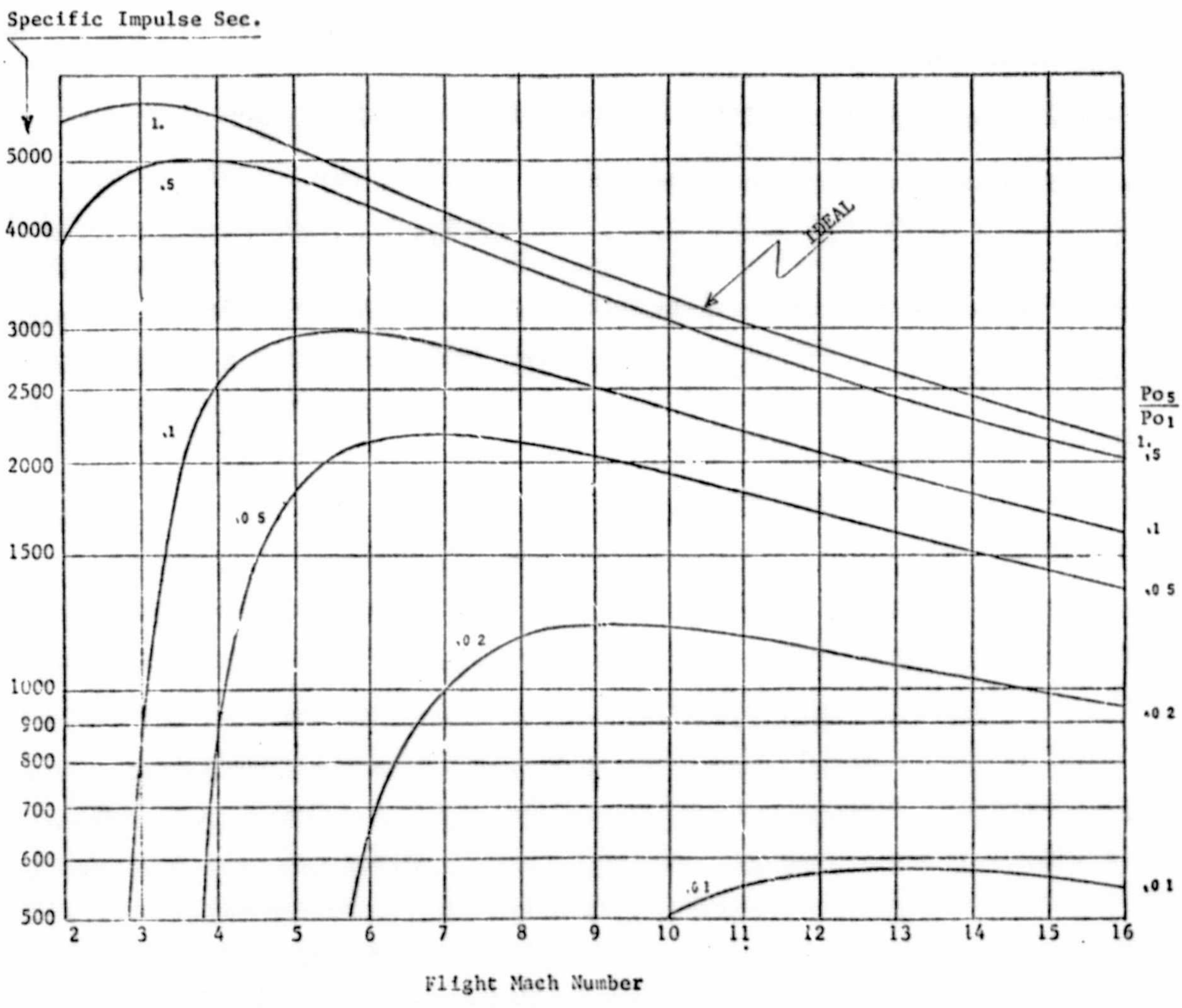


FIGURE 4.2.2 Specific Impulse in Seconds vs. Flight Mach Number for Various Overall Stagnation Pressure Ratio Losses for Q = 2326 J/g

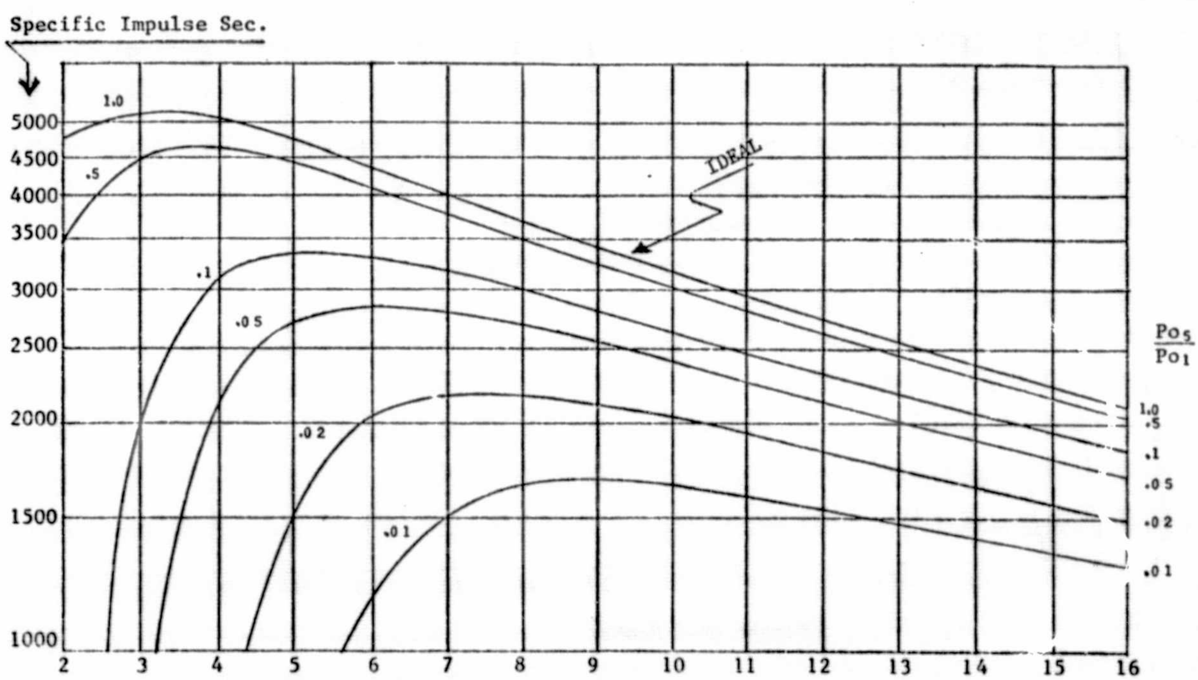
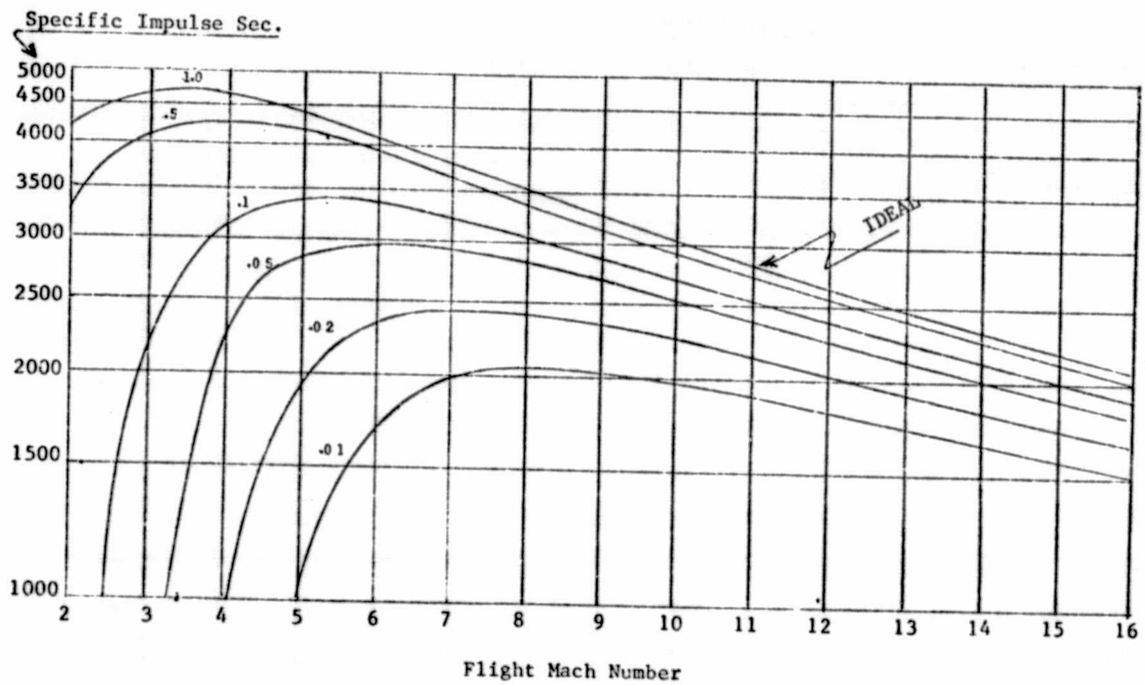


FIGURE 4.2.3 Specific Impulse in Seconds vs. Flight Mach Number for Various Overall Stagnation Pressure Ratio Losses for Q = 3489 J/g



4.3 Performance Summary

The performance to be obtained from the oblique detonation wave ramjet depends upon the following three elements:

- Diffuser performance, i.e., the diffuser stagnation pressure losses and static temperatures produced by compression.
- Oblique detonation wave or combustion performance, i.e., the stagnation pressure losses produced by the oblique detonation waves.
- The detonation limits of the fuel-air mixtures.

The above three elements are intimately connected as follows:

- High compression in the diffuser produces high static temperatures as inlet conditions to the detonation waves.
- High static temperatures reduce detonation Mach numbers thereby reducing the stagnation pressure losses of the detonative combustion.
- High static temperatures narrow the limits of detonation, and if sufficiently high will result in diffusive combustion in place of a detonation.

The above effects are illustrated in Figures 4.3.1, 4.3.2, and 4.3.3. In Figure 4.3.1 an upper temperature detonation limit of $T_2/T_1 = 4$ is imposed which leads to the diffuser configurations shown in Table 4.3.1.

Each of the configurations in the Table 4.3.1 would provide inlet temperatures of approximately 1,111 K to the oblique detonation wave.

FIGURE 4.3.1 Effect of a $T_2/T_1 = 4$ Limit Line Upon the Performance of Multi-shock Diffuser Configurations

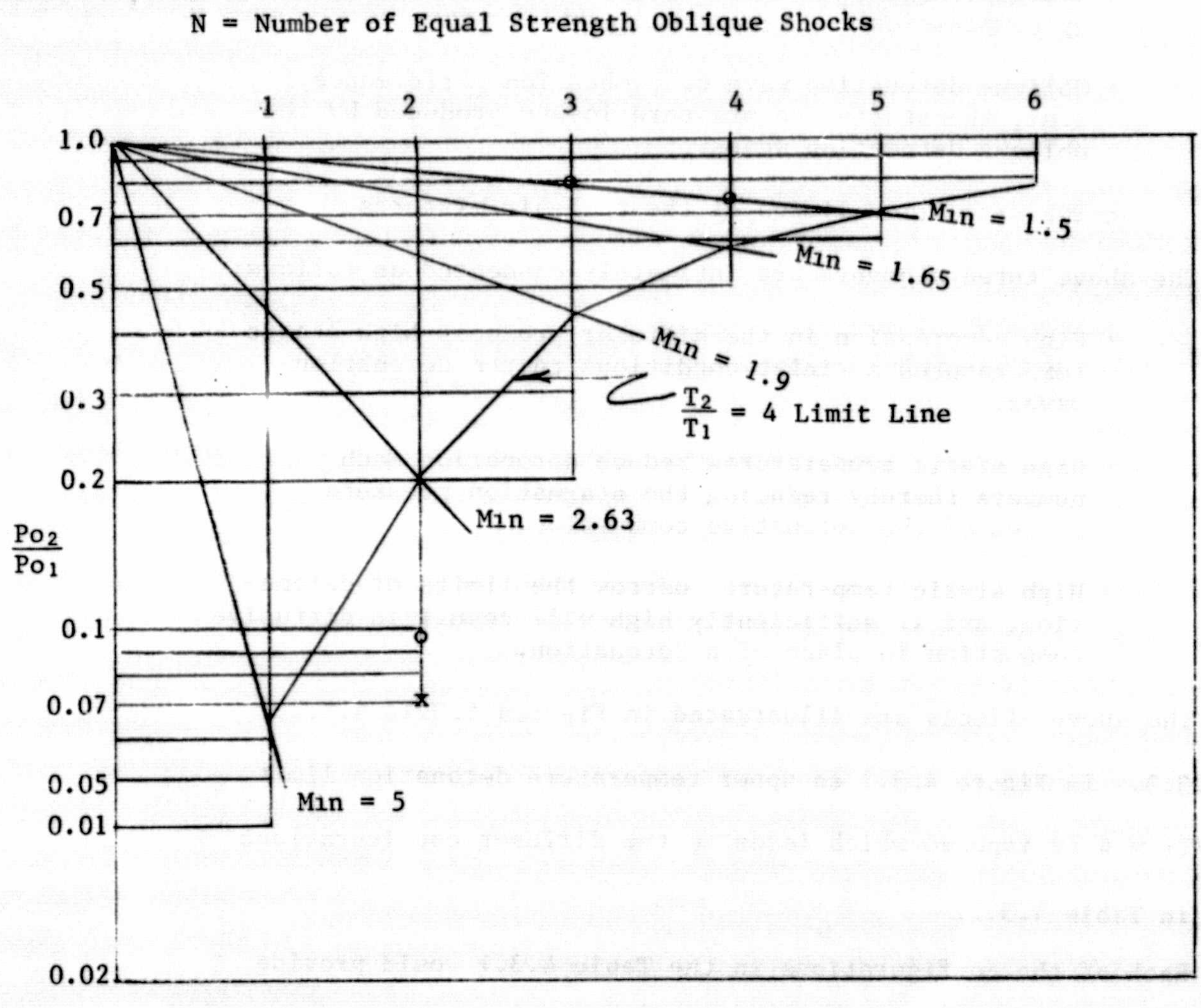


TABLE 4.3.1 POSSIBLE DIFFUSER CONFIGURATIONS FOR A $T_2/T_1 = 4$ LIMIT

Number of Oblique Shocks	1	2	3	4	5
Mn-Oblique Shock Strength	5	2.63	1.9	1.65	1.5
Po_2/Po_1 - Stagnation Pressure ratio loss	0.065	0.2	0.44	0.58	0.7

For stoichiometric hydrogen-air mixtures at this temperature the $M_{DC-J} \cong 3.4$. The stagnation pressure losses associated with the oblique detonation wave, depending upon the Mach number of the flow at the combustor inlet, require that the flight Mach number be specified.

Considering the case of 5 oblique shocks (reference Figure 4.3.1) with $M_{1n} = 1.5$ at a flight Mach number of $M_1 = 8$ the flow Mach number prior to detonation is approximately 3.5. This would result in a stagnation pressure ratio loss of approximately 0.2, and an overall pressure ratio loss of about 0.14.

The specific impulse associated with the above is 3100 seconds. This is to be compared to the ideal specific impulse at this flight Mach number of 3500 seconds.

In the above case the Mach number of detonation of $M_{DC-J} \cong 3.4$ is very close to the flow Mach number of 3.5 indicating a narrow margin of stability. Reducing the number of oblique shocks of M_{in} = 1.5 to 3 would alleviate this condition as well as reducing the temperature limit to T_2/T_1 = 2.3.

The performance of the latter configuration operating on stoichiometric hydrogen-air mixtures is an overall stagnation pressure ratio loss of approximately 0.06 and a specific impulse of 2860. This is to be compared to the 5 oblique shock case with a specific impulse of 3100 seconds. The small performance degradation is attributable to the larger stagnation pressure losses of the detonation process associated with the detonation Mach number of 4.1.

The three oblique shock configuration of M_{in} = 1.5 applied to a flight Mach number of 16 would possess the following performance characteristics; a specific impulse of 1700 seconds, an overall stagnation pressure loss ratio loss of approximately .07, and a T_2/T_1 = 2.3. The ideal specific impulse is 2100 seconds.

The effect upon diffuser configuration of T_2/T_1 limits of 3 and 2 are shown in Figures 4.3.2 and 4.3.3 respectively. Decreasing of T_2/T_1 ratio limit has the effect of reducing the number of oblique shocks used in the diffusion process and therefore a reduction of diffuser efficiency.

To summarize, multi-shock diffuser performance in the neighborhood of 0.5 stagnation pressure ratio loss can be attained for a range of configurations and flight Mach numbers. Oblique detonation

FIGURE 4.3.2 Effect of a $T_2/T_1 = 3$ Limit Line Upon the Performance of multi-shock Diffuser Configurations

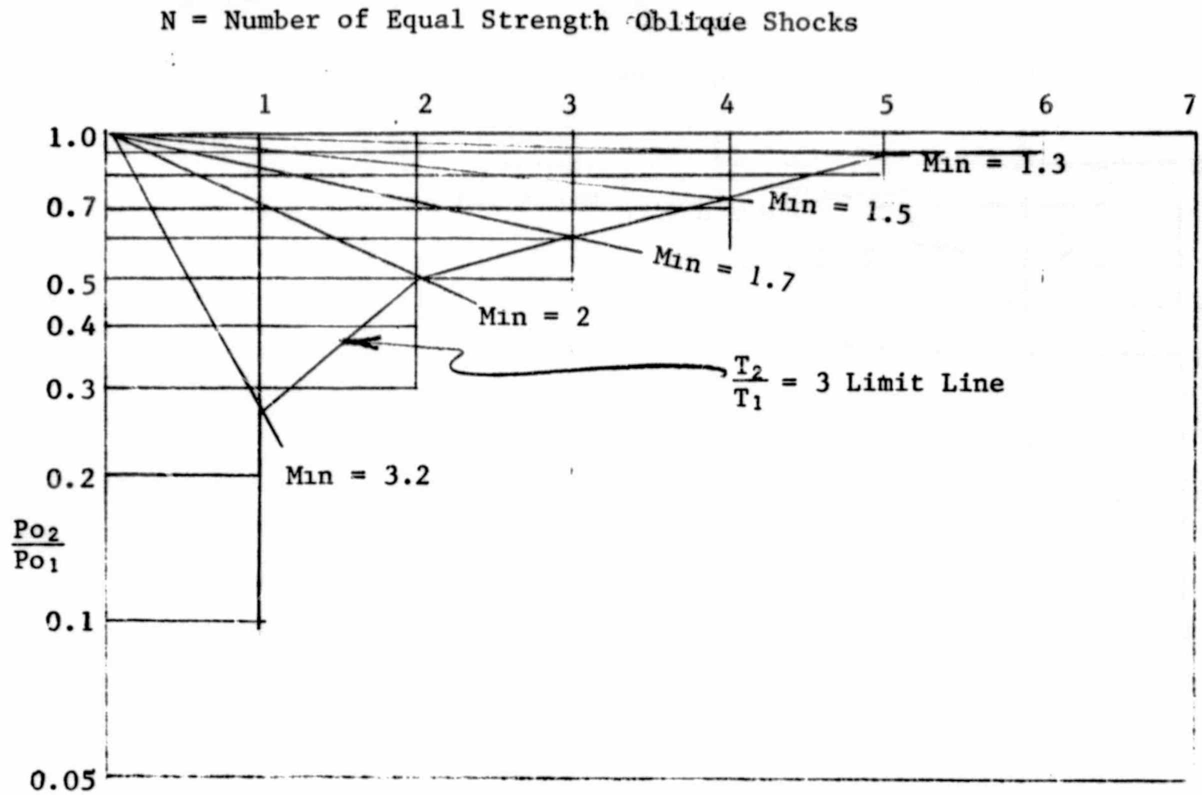
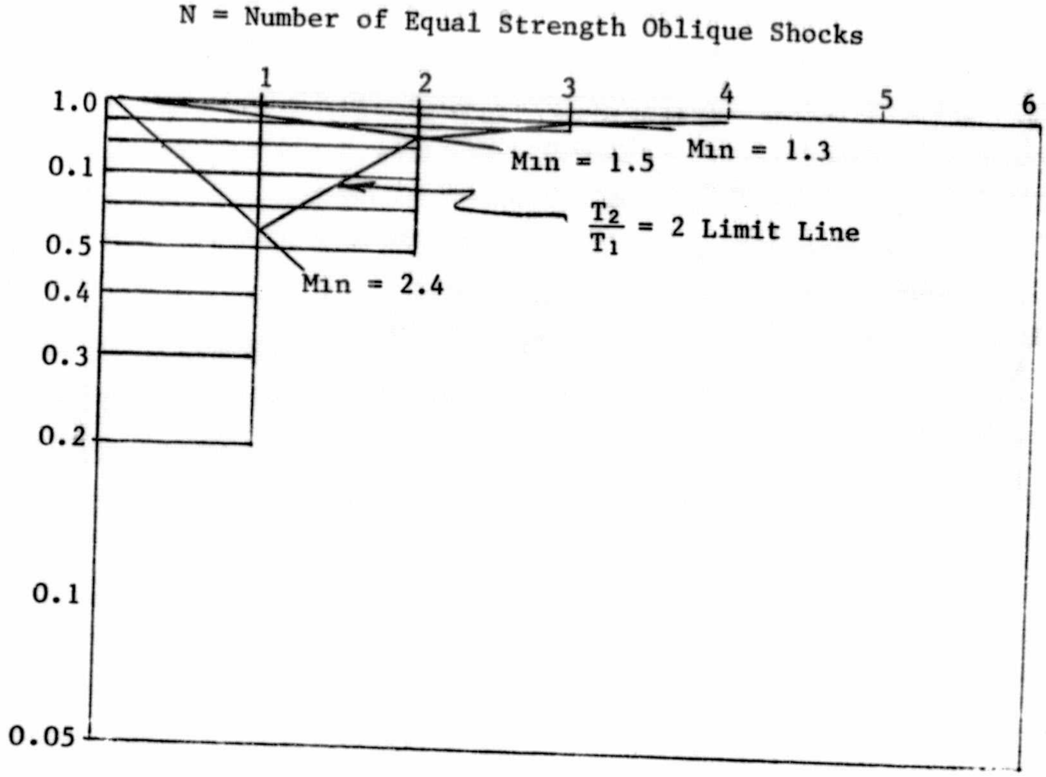


FIGURE 4.3.3 Effect of a $T_2/T_1 = 2$ Limit Line Upon the Performance of Multi-shock Diffuser Configurations



wave performance in excess of a 0.1 stagnation pressure ratio loss can be obtained for a large range of configurations and flight Mach numbers. An overall stagnation pressure ratio loss of .05 is representative of the loss to be expected as characteristic throughout a flight Mach number range of 6 to 16.

The expected performance to be obtained from the oblique detonation wave ramjet is summarized for thrust coefficients and specific impulse in Figures 4.3.4 and 4.3.5. It should be noted in these figures that are plotted for a constant overall stagnation pressure loss ratio of .05 that this value is representative, only, of expected losses and that losses will generally be a function of flight Mach number.

FIGURE 4.3.4 Thrust Coefficient as a Function of Flight Mach Number for Various Values of Heat Release and for a Stagnation Pressure Loss Ratio of 0.05

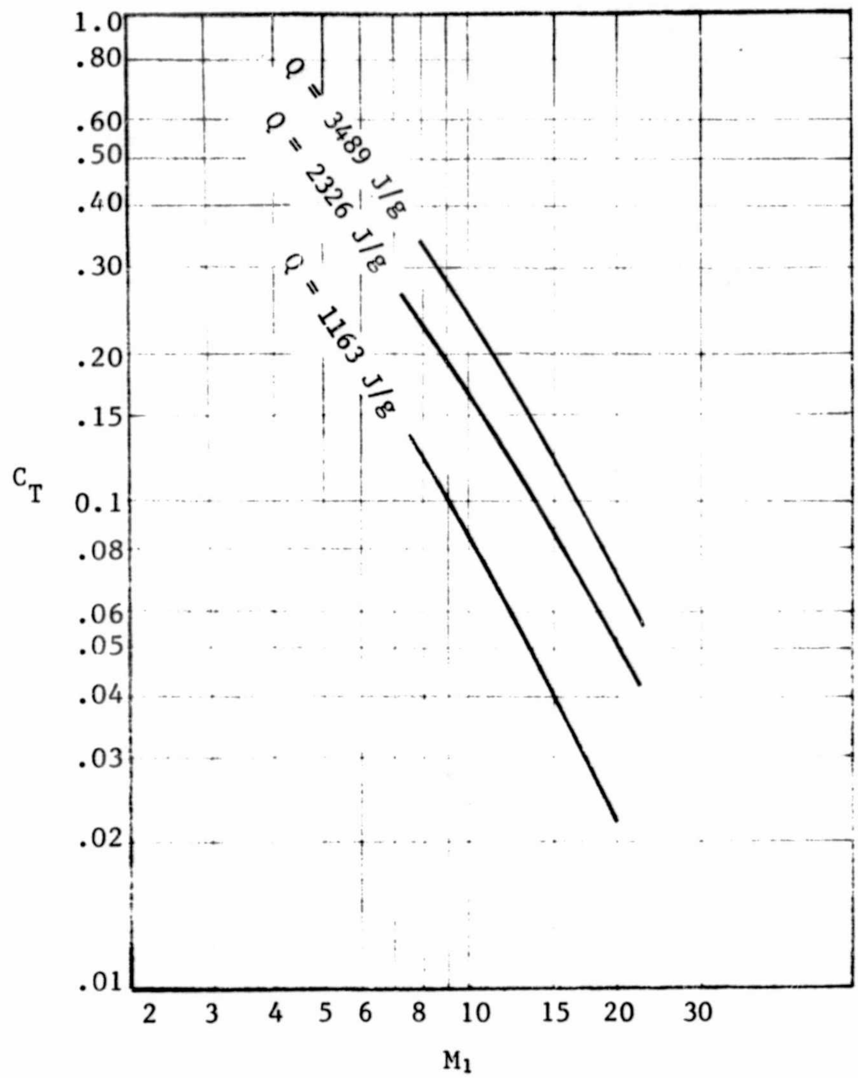
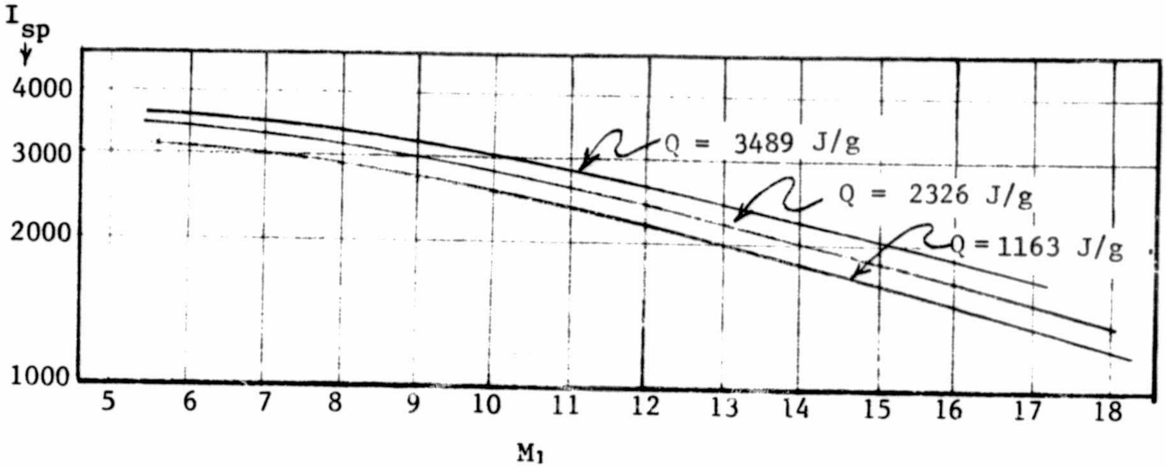


FIGURE 4.3.5 Specific Impulse as a Function of Flight Mach Number for Various Values of Heat Release and for a Stagnation Pressure Loss Ratio of 0.05



5.0 Comparison of Oblique Detonation Wave Ramjet with Diffusive Burning Scramjet

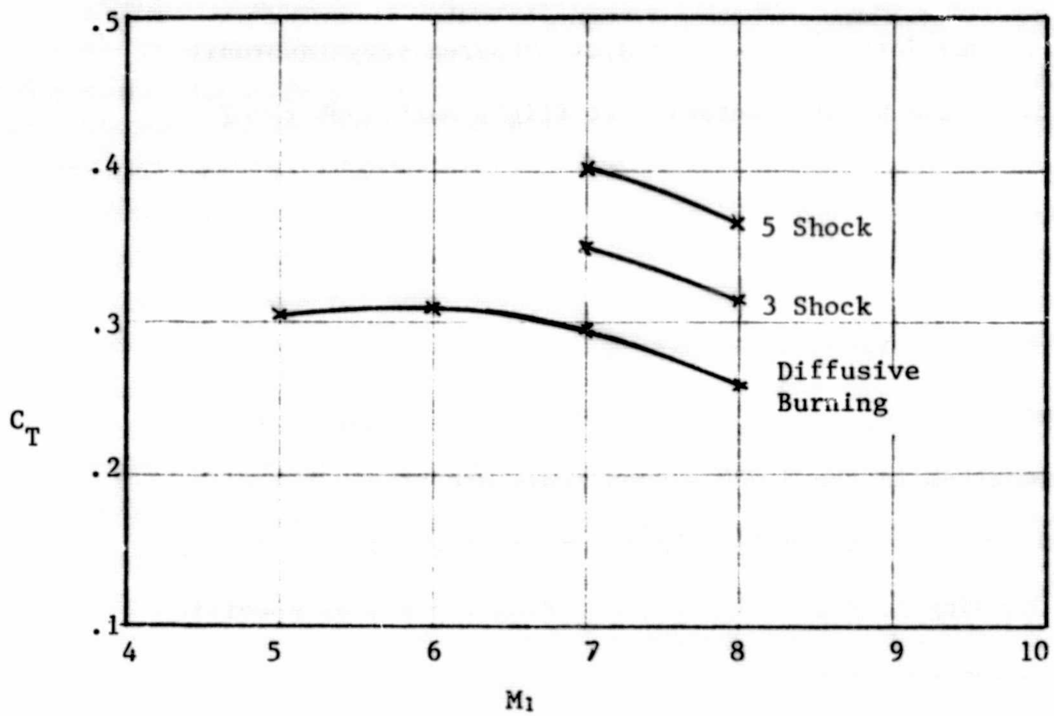
The comparison of the hydrogen fueled detonation ramjet with the hydrogen fueled scramjet is made on the basis of the thrust coefficient and specific impulse. The scramjet information includes losses encountered from fuel injection, diffusive burning and nozzle expansion. This feasibility study for the detonation ramjet includes the losses resulting from diffusion and from the detonation but does not include fuel injection losses or nozzle expansion losses. The detonation ramjet comparison curves do not incorporate the correction factor for fuel-air ratio discussed in section 4.1. The detonative combustion losses, however, include the real gas effects obtained from experimental hydrogen-air detonation properties.

A stoichiometric hydrogen-air mixture is selected as the point of comparison for a range of flight Mach numbers from $M_1 = 5$ to $M_1 = 8$. A flight Mach number of 8 was selected as the point for the detailed analysis of the oblique detonation wave ramjet. This Mach number exceeds the takeover speed of $M_1 = 6$ to 7 sufficiently to allow comparison. The performance at lower Mach numbers are extrapolated results for a constant overall pressure ratio loss.

5.1 Thrust Coefficient Evaluation

The thrust coefficient comparison is shown in Figure 5.1.1. Two configurations of the detonation ramjet are shown i.e., 1) five

FIGURE 5.1.1 Thrust Coefficient Performance Comparison of the Oblique Detonation Wave Ramjet with the Diffusive Burning Scramjet



NOTE: This comparison of the oblique detonation wave ramjet with the diffusive burning scramjet is based upon installed performance for the scramjet (Ref 7) and Analytical results for the detonation jet that are uncorrected for fuel-air ratio and the losses resulting from fuel injection and under expanded nozzles.

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7 Pinckney, S. Z.: NASA TM X-74038, 1978.

oblique shocks of $M_{in} = 1.5$, and 2) 3 oblique shocks of $M_{in} = 1.5$. No nozzle exhaust losses or fuel injection losses are included for the detonation ramjet. Addition of these losses would lower the thrust coefficient a few per cent.

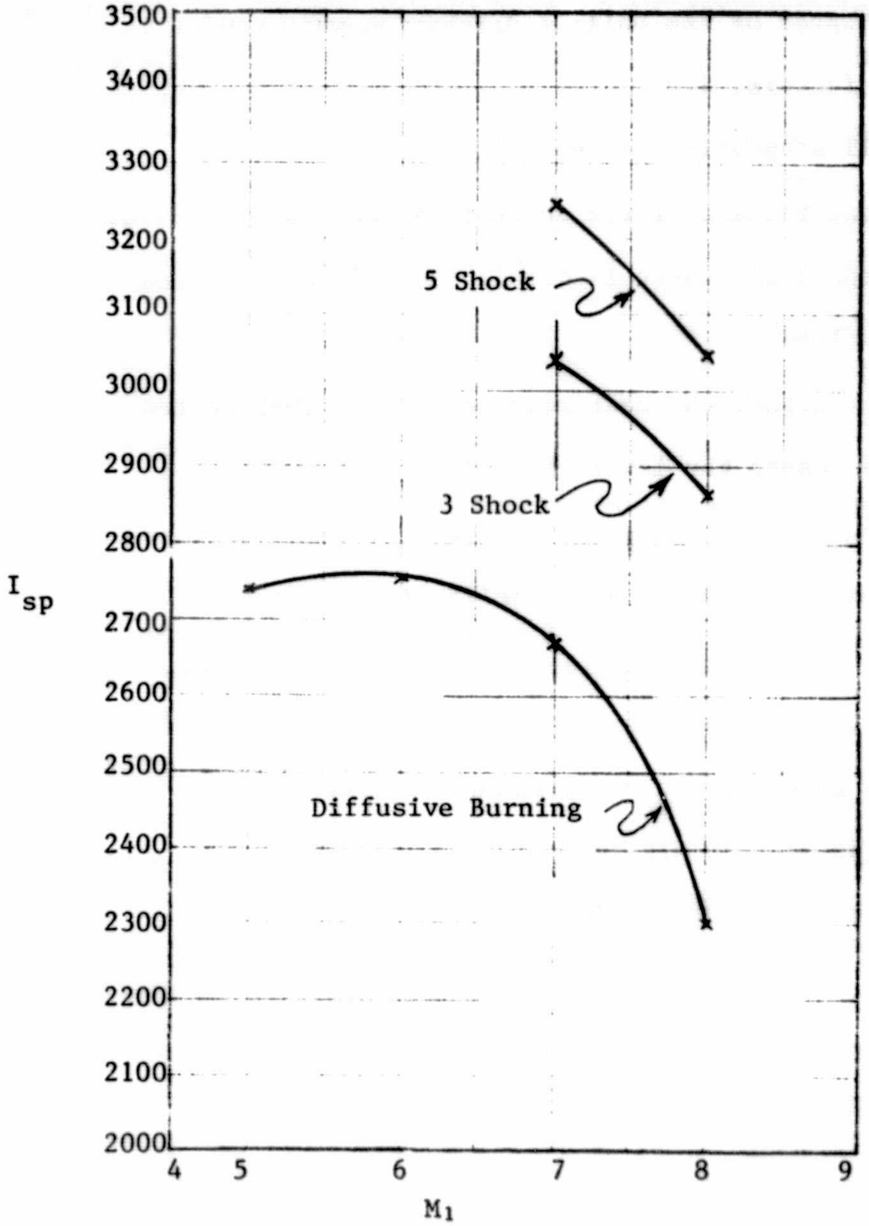
The oblique detonation wave ramjet compares very favorably with the diffusive burning scramjet at flight Mach numbers of 7 to 8.

5.2 Specific Impulse Evaluation

The specific impulse comparison is shown in Figure 5.2.1. Specific impulses in the 3,000 second range are attainable with the detonation ramjet in the flight Mach number range of 7 to 8. The diffusive burning scramjet, in the same range, produces specific impulses in the 2500 seconds range.

If exhaust nozzle and fuel injection losses were included it appears that the detonation ramjet would produce 200 to 300 seconds less specific impulse.

FIGURE 5.1.2 Specific Impulse Performance Comparison of the Oblique Detonation Wave Ramjet with the Burning Scramjet



NOTE: This comparison of the oblique detonation wave ramjet with the diffusive burning scramjet is based upon installed performance for the scramjet and analytical results for the detonation (Ref 7) jet that are uncorrected for fuel-air ratio and the losses resulting from fuel injection and under expanded nozzles.

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7 Ibid.

6.0 Takeover Speed of Oblique Detonation Wave Ramjet

The takeover speed of the oblique detonation wave ramjet is dependent upon two factors, i.e.;

- Overall stagnation pressure losses
- The heat release of the detonative process

Section 5 of the report discusses in detail the influence of the various factors involved.

In general the higher the heat release and the smaller the stagnation pressure losses the lower the flight Mach number for takeover. Referring to the graphs of Section 5 it is noted that peak values of thrust coefficient and specific impulse occur over flat optima of flight Mach numbers (for a given overall stagnation pressure loss). Defining the takeover point to be that of optima values for a given stagnation pressure loss would infer that this is a first point wherein stable operation can occur.

Takeover points for a Q = 3489 K/g air are shown in Table 6.1

TABLE 6.1 TAKEOVER MACH NUMBER FOR Q = 3489 K/g AIR

Overall Stagnation Pressure Ratio Loss	0.5	0.1	0.05	0.02	0.01
Approximate takeover Flight Mach number	4	5	6	7	8

Takeover points for a Q = 2326 J/g air are shown in Table 6.2

TABLE 6.2 TAKEOVER MACH NUMBER FOR Q = 2326 J/g AIR

Overall Stagnation Pressure Loss	0.5	0.1	0.05	0.02	0.01
Approximate Takeover Flight Mach Number	4	5	6	7.5	9

The takeover points for a Q = 1163 J/g air are shown in Table 6.3

TABLE 6.3 TAKEOVER MACH NUMBER FOR Q = 1163 J/g AIR

Overall Stagnation Pressure Loss	0.5	0.1	0.05	0.02	0.01
Approximate Takeover Flight Mach Number	4	5.5	7	9	13

Inasmuch as overall stagnation pressure ratio losses can be held to $Po_5/Po_1 \cong 0.05$ leaving a considerable margin for stability, the takeover speed of the oblique detonation wave ramjet is conservatively a flight Mach number of 6.

7.0 Area Ratio and Variable Geometry Requirements

The area ratio requirements for a multi-shock diffuser are closely approximated by Equation 7.1

$$\frac{A_1}{A_2} \cong \left(\frac{\sin \beta}{\sin \Theta}\right)^N \qquad 7.1$$

Where:

A_1 = Capture area

A_2 = Deflected flow area

N = Number of equal strength oblique shocks

Θ = Deflection angle

β = Shock wave angle

For the M_1 = 8, five shock configuration, ramjet, the area ratios as calculated from Equation 7.1 are given in Table 7.1.

TABLE 7.1 AREA RATIOS OF THE FIVE SHOCK CONFIGURATION DETONATION RAMJET.

N = Number of Equal Strength Oblique Shocks of Mn - 1.5	1	2	3	4	5
Cumulative Area Ratios	2.17	4.71	10.2	22.2	48.1

These area ratios are substantially the same as would be obtained from isentropic compression. Isentropic compression to a Mach number of 3.5, which is the 5 shock case, requires an area ratio 54.3 as compared to the Table 7.1 value of 48.1.

Variable geometry requirements for the detonation ramjet would be confined to the detonation region. To accomodate different fuel-air mixtures with differing oblique detonation wave properties only the deflection angle need be changed. These deflection angles for the case of the $M_1 = 8$ ramjet varied from approximately 10^0 to 14^0 throughout the range of heat releases of 1163 to 3489 J/g air.

Minimal variable geometry should be required for the oblique detonation wave ramjet.

8.0 Compatibility with Lower Speed Airbreathers

The compatibility of the oblique detonation wave ramjet with lower speed airbreathers is limited to flight Mach numbers in the vicinity of 6. In any case the absolute lower limit is the condition of flight Mach number equal to the detonation Mach number. For stoichiometric hydrogen-air mixtures this detonation Mach number is approximately 5 at 289 K. Hydrocarbon-air detonations possess about the same detonation Mach numbers.

The oblique detonation wave ramjet is therefore judged to be incompatible with the subsonic combustion ramjet. It would be compatible, however, with the diffusive burning scramjet as discussed in Section 6.0.

It would also be compatible with composite multimode engine configurations that operate up to flight Mach numbers of 6.

9.0 Problem Areas

The problem areas associated with the oblique detonation wave ramjet are:

- Detonation Limits
- Stability
- Diffuser Performance
- Variable Geometry

These problems are discussed in the following subsections

9.1 Detonation Limits

There is very limited data available for fuel-air detonation limits except for very particular requirements.

Fuel-air explosives (FAE) have been developed to a high degree for producing substantial overpressures to simulate nuclear blast waves and to explode land mines. Many tests of FAE devices have been made by the Navy at the Naval Weapons Center, China Lake, California, with ethylene oxide, propylene oxide, propane, and other specific fuels. These tests were all made at ambient air temperatures.

Needed is detonation limit information for fuel-air mixtures at elevated temperatures. University of Michigan personnel carried out a detonation limit study of ethylene-oxygen mixtures at elevated temperatures for the Ethyl Corporation which substantiated that elevated temperatures narrowed these limits to a point, wherein, at sufficiently high temperatures, the mixture could not be detonated.

The successful operation of the detonation ramjet depends heavily upon such data, and, therefore constitutes a most critical problem for the detonation ramjet.

9.2 Stability

Stability problems of the detonation ramjet are confined to the low speed range of operation as has been discussed in Section 6.0.

There appears to be no detonation stability problems in the high speed range of operation as long as the airflow is not over diffused to a point wherein the detonation Mach number approaches or exceeds the local flow Mach number.

9.3 Diffuser Problems

Diffuser problems fall into two categories associated with the high speed range of operation, i.e.:

- Shock boundary layer interaction
- Sensitivity of very small deflection angles to the diffusion process.

The above sensitivity would likewise reflect upon performance for changes in the angle of the vehicle.

9.4 Variable Geometry

The variable geometry requirements for design point operation at one flight Mach number is covered in Section 7.0.

The requirements of variable geometry for variable flight speeds center upon the diffuser and simple techniques of focusing the oblique shocks. It appears that a number of equal strength oblique shocks could be compromised into a number of unequal strength oblique shocks to provide a variable speed flight speed range of interest.

10.0 Real Gas Effects

Real gas effects upon the performance of the oblique detonation wave ramjet are largely confined to the detonative process. One technique to assess real gas effects for detonative processes is to use the experimental detonation Mach numbers and back calculate the conditions required to produce these experimental Mach numbers. (Ref 8)

The above technique was used in this analysis i.e.; the direct use of experimental detonation Mach numbers together with the semi-empirical equation for detonation Mach number. (Ref 9)

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11.0 Conclusions

The oblique detonation wave ramjet offers great potential as an airbreathing propulsor to extend the useful range of ramjet flight Mach numbers from 6 to 16 and above.

Specific impulses and thrust coefficients that would be attainable in the above flight range would exceed 70% of ideal.

Multi-shock diffusers offer much promise as the means of tailoring a simple configuration to meet requirements of variable speed and variable fuel-air ratios. Three oblique shock configurations appear to represent a best compromise between simplicity and performance.

Stable operation of the detonation wave ramjet exists at flight Mach numbers in excess of 6.

The shock portion of the detonation wave constitutes a most important part of the compression process by alleviating demands upon the diffuser and providing a high temperature for ignition of the mixture. Unlike diffusive supersonic combustion chemical reaction is promoted in very short distances.

Uncertainty of detonation limits at elevated temperatures constitutes the largest unknown parameter influencing performance.

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16. Abstract

The potential performance of oblique detonation wave ramjets is analyzed in terms of multi-shock diffusion, oblique detonation waves, and heat release. Results are presented in terms of thrust coefficients and specific impulses for a range of flight Mach numbers of 6 to 16.

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