von KARMAN INSTITUTE
FOR FLUID DYNAMICS

TECHNICAL NOTE 20

REFLECTED SHOCK INLET DIFFUSER
IN HYPERSONIC FLOW

by

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The work described herein was done in partial fulfilment of the requirements for receiving the Diploma of the von Karman Institute for Fluid Dynamics. Mr Lipfert, an American student, obtained a Distinction Grade, and was winner of the von Karman Prize, awarded annually to the student ranked the first in the graduating class, for the year 1963-64.
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ABSTRACT

A study has been made of hypersonic flow deceleration by means of a plane oblique shock and its reflection from a flat surface.

A theoretical analysis was made of an inlet diffuser using this arrangement followed by a terminal normal shock. The optimum deflection angles were found, and it is shown that the total pressure recovery through such a system of shocks is nearly the optimum for a 3-shock configuration. The optimum is given by the Oswatitsch analysis, which was extended to a Mach number of 10.

The performance of this system was checked experimentally at a Mach number of 5.4, with the terminal normal shock simulated by an impact probe. The results agreed well with the theory; the maximum kinetic energy efficiency was 0.91.
NOMENCLATURE

A  flow area
C_p  specific heat at constant pressure
D  spacing between parallel portion of models
h  enthalpy
L  ramp length
M  Mach number
n  number of shocks in an inlet configuration
P  pressure
S  entropy
T  temperature
V  velocity

\[ \alpha \]  angle of attack
\[ \gamma \]  specific heat ratio
\[ \delta \]  flow deflection angle
\[ \eta \]  diffuser efficiency
\[ \eta \]  kinetic energy efficiency
\[ \theta \]  shock wave angle

Subscript

F  final value
i  oblique shock index
st  stagnation
1  value before entering diffuser
2  value after leaving diffuser
\[ \infty \]  free stream value
p  pitot
4.

**INTRODUCTION**

It has recently been pointed out (1) that chemical rocket engines are reaching a plateau in performance, with improvements in specific impulse becoming more and more difficult. Alternative systems being considered include hypersonic airbreathing vehicles, which, it is shown, could be designed to operate from low supersonic to orbital speeds with significantly higher specific impulse and payload. To achieve the high performance value desired, careful attention must be given to all components to realize the required efficiencies. The inlet diffuser is particularly important, in that it can influence not only the engine thrust and impulse but also the structural and cooling loads by virtue of its mass flow regulation and pressure recovery.

Research on hypersonic inlets in the United States was begun as early as 1953 by the N.A.C.A. Certainly, much additional work has been done in this area both by industrial and government laboratories, but little has been published in the open literature. Some of the available results are summarized below.

<table>
<thead>
<tr>
<th>Ref</th>
<th>Configuration</th>
<th>$M_\infty$</th>
<th>$P_{st,F}/P_{st,\infty}$</th>
<th>$\eta_{KE}$</th>
<th>Remarks</th>
</tr>
</thead>
<tbody>
<tr>
<td>2.</td>
<td>single 27° half-angle cone</td>
<td>5.4</td>
<td>0.137</td>
<td>0.864</td>
<td>required artificial transition for stable operation</td>
</tr>
<tr>
<td>3.</td>
<td>separation spike</td>
<td>5.5</td>
<td>0.138</td>
<td>0.874</td>
<td>extremely sensitive to angle of attack</td>
</tr>
<tr>
<td>4.</td>
<td>isentropic spike</td>
<td>5.6</td>
<td>0.208</td>
<td>0.91</td>
<td>did not capture full stream tube; artificial transition required</td>
</tr>
<tr>
<td>5.</td>
<td>isentropic spike fixed and var. geom</td>
<td>6.0</td>
<td>0.21</td>
<td>0.92</td>
<td>no improvement from variable geometry.</td>
</tr>
</tbody>
</table>

(*) See Appendix A for a discussion of definitions of diffuser efficiencies.
These are all optimal compression designs, which are recommended for hypersonic flight since radiation cooling somewhat alleviates the aerodynamic heating load.

Analysis of the effect of inlet performance on specific impulse at $M = 2.0$ shows that the gains in impulse become unimportant above $\eta_{KE} = 0.92$, and conversely the losses become severe below $\eta_{KE} = 0.90$ (6). Therefore a suitable design target for hypersonic inlets is $\eta_{KE} = 0.90 - 0.92$. The table above shows that this has indeed been realized, but not without difficulties. In one case, a flow spillage problem existed, and in the other, a 3% efficiency improvement expected from the inclusion of variable geometry did not materialize.

So far the discussion has been limited to devices utilizing subsonic combustion. Recently, Dugger and Avery (1) have shown some overall system advantages resulting from the use of supersonic combustion. Since the air supply must be only partially decelerated, the diffuser design problem is alleviated, and structural and cooling loads may be reduced somewhat because of lower temperatures and pressures. It is difficult to specify the diffuser performance requirement for this case, since the combustor Mach number and hence the amount of diffusion may also vary. Values of $\eta_{KE} = 0.97$ have been assumed for combustion Mach numbers in the range 2.5 - 3 for studies comparing subsonic and supersonic combustion systems.

This report is intended to show that the required efficiencies are achievable for the subsonic-burning case from a simple configuration and to present some experimental results from a two-dimensional diffuser that may be suitable for the supersonic-combustion case. The particular diffuser under consideration is a modification of what Hermann (7) calls a "three shock system with partly external, partly
internal compression". It is shown schematically in Figure 1 and provides the parallel flow needed for supersonic combustion. For the subsonic case, a normal shock would be positioned in the parallel channel. Experimentally this was accomplished with an impact probe. The problems of stabilizing and positioning the normal shock were not considered. The total pressure losses resulting from boundary layer growth and from subsonic diffusion after the terminal normal shock are also neglected in this study. The scope of this research was necessarily limited because of the confinement of a student project schedule.
It is well known that a given compressor may be achieved more efficiently by a number of oblique shocks than by a single normal shock or strong oblique shock; this is the principle of the "isentropic diffuser". In 1944 Oswatitsch showed that, for a given number \((n-1)\) of oblique shocks followed by a normal shock, the maximum total pressure recovery results when the oblique shocks are all of the same strength. This is discussed thoroughly by Hermann (7) who also pointed out from the numerical results that the terminal normal shock should be slightly weaker.

That is, for the maximum total pressure recovery

\[
M_i \sin \theta_i = \text{const} \quad (0 < i < n-2)
\]

and

\[
M_{n-1} = 0.94 \quad M_\infty \sin \theta_\infty
\]

Henderson (8) showed recently that a slightly higher recovery results if all the shocks are oblique and of the same strength.

Hermann (7) presents numerical results from the Oswatitsch analysis for Mach numbers up to 6. Since the curves for the required deflection angles reach a peak near this Mach number, it was of interest to extend the data. This was done up to \(M=10\) for \(n = 3\), for a perfect gas with \(\gamma = 1.4\), using an IBM 1620 computer. The details of the calculation program are given in Appendix B, the results are shown in Figure 2. It was also found that the maximum recovery occurs at a slightly higher terminal shock strength at the higher Mach numbers, i.e.,
\[ M_{\text{n-1}} = 0.95 M_o \sin \theta_o \]

However, the differences in efficiency are so small that the point is only an academic one.

Calculations were also carried out for the case of a reflected oblique shock, as in Fig. 1 for which the two flow deflections are necessarily equal. Predictably, the optimum deflection angle lay between the angles resulting from the Oswatitsch analysis, as shown in Figure 2. Moreover, the total pressure recovery for the reflected shock system is only slightly below the optimum, as seen from Figure 3, and is within the \[ \eta_{\text{KE}} = 0.90 - 0.92 \] bond specified above as an inlet diffuser design goal.
EXPERIMENTS

FACILITIES AND APPARATA

This diffuser configuration was tested in the VKIFD Hypersonic Blowdown Tunnel H-1 at a Mach number of 5.4. Stagnation temperatures and pressures were 250 - 150°C and 30-32 atm., respectively. The tunnel is described in detail in Reference 9. It is equipped with an off-axis optical system for flow visualization. A spark light source was used for shadowgraphs; a continuous source was used for Schlieren photographs.

Pressures were measured with a system of scanning valves and variable reluctance transducers. Readout was accomplished with a digital voltmeter. The static pressure orifices were of 1 mm internal diameter. The pitot pressure was measured with an impact prove constructed of 1 mm internal diameter, 1.5 mm external diameter tubing, with a square end. The higher values of pitot pressure were read on a 0-10 atm. bourdon-tube dial-gage. The surface temperature of each model was also measured, in order to get an idea of thermal stabilization times. Equilibrium was reached in about 30 seconds for each model, tested separately with no impinging shocks.

The models were constructed from stainless steel and had average leading edge thicknesses of about .060 mm, varying slowly between a maximum of 0.130 mm and a minimum of 0.030 mm. This variation was the result of thermal distortion from soldering the pressure connections. The
models had a span of 100 mm; the test section span was 144 mm. The ramp length was 96 mm. The upper reflecting plate was mounted on the vertical traverse mechanism in a cantilever fashion. The wedge, or ramp, was mounted on the tunnel carriage, which could be translated along the axis and pitched to positive or negative angles of attack. A movable flap was attached to the rear of the wedge in order to provide the parallel passage of the inlet. The seal between flap and wedge consisted of two close-fitting circular arcs with a rubber "O" ring between. No leakage was observed, although a what appeared to be shock wave was observed emanating from this expansion corner. A pitot pressure traverse was made through the wave to determine its strength, a jump in pitot pressure of 1.5% was recorded, indicating a weak disturbance. Attempts to smooth out the joint had no discernible effect on this wave.

Side plates and spanwise pressure orifices were provided to test for the presence of three-dimensional effects. With the side plates installed, the model was in effect a finite-span rectangular inlet.
RESULTS AND DISCUSSION

Experiments were conducted with flow deflection angles of 5°, 10°, 15° and 18°, with various plate spacings near those given by inviscid theory and shown in Figure 4.

The results are presented in Figures 5 - 8.

The flow patterns seen in the shadowgraphs are similar for all deflection angles. A weak system of waves is seen downstream of the expansion corner, indicating that cancellation of the reflected shock is not complete. The incomplete cancellation undoubtedly results from the fact that expansion around a sharp corner requires a finite distance for completion. Superimposition of an oblique shock then results in a slight overpressure near the corner.

The pressure distributions also reveal this trend. For example, the pressures on the upper plate for 5° show the presence of a wave, followed by an expansion, for all values of the plate spacing. The wave is sufficiently weak that it is felt over only one pressure orifice, since the boundary layer is turbulent at that point. The pattern is similar for the other deflection angles. This lack of complete cancellation seems to have the general effect of raising the pressure level somewhat over the theoretical, although viscous interaction effects at the initial shock impingement point may also play a role. The fact that in most cases the upper plate bow shock wave has coalesced with the reflected shock also has the effect of creating extra compression. The bow shock was found to have finite strength from tests at zero angle of attack.
When the plate spacing is decreased beyond that for the best cancellation, shown in Fig. 4, the reflected shock pressure rise is felt upstream of the corner, and more extraneous waves appear downstream. These waves make it possible to decrease the spacing for beyond that shown in Figure 4, since the additional deceleration outweighs the total pressure losses. An attempt to reach a limit was made at 5°. The spacing was decreased to 3 mm, at which point the flow became unstable, judging by the violent instrumentation oscillations. The tunnel then unstarted after a few seconds, probably because the inlet unstarted first and the subsequent blockage was too great. The delay may have resulted from thermal effects on boundary layer buildup. This data point is included in Table I, where it is seen that this small spacing is physically possible because of the additional compression. This property of the configuration has the effect that the mass flow may be varied during operation simply by decreasing the spacing.

Side plates were installed at 10° to test for three-dimensional effects. Although some differences were recorded, there were not systematic enough to be attributed to cross flow effects, especially in view of the fact that extraneous waves may only affect one pressure orifice. It was therefore concluded that three-dimensional effects were not important.

The results of the pitot pressure measurements are presented in Table I and in Figure 9. Reasonable agreement with theory is seen. At the low angles, the pitot pressures also reveal higher compression than predicted by theory, as discussed above. Also, at the lower angles, most of the compression takes place through the normal shock. If, due to viscous or leading edge effects, the oblique shocks are slightly stronger than predicted, the pressure recovery will be higher, and the effective deflection angle is larger than the physical geometry indicates.
Included in the table are calculations of the stagnation pressure loss through the oblique shock system only, $P_{stF}/P_{st\infty}$ and the corresponding kinetic energy efficiencies. These values indicate the performance of the configuration as a supersonic combustion inlet. It is seen that $\eta_{KE} = 0.97$ is a reasonable figure for combustor Mach numbers above 2.7.

An attempt was made to take data at 18°, but considerable difficulty was encountered in keeping the tunnel started when the plate spacing was decreased to the proper values for good performance. The data shown in Figure 9 were taken transiently, and therefore are open to question.

It is not clear whether this problem resulted initially from flow breakdown inside the model configuration due to the large pressure gradients, or from some adverse condition in flow around the model. The problem was not investigated further because of time limitations.

However, even at 15°, which is less than the optimum angle, a kinetic energy efficiency of 0.91 was realized, which is a respectable figure for such a simple approach.

It should perhaps be mentioned that these tests were confined to the case where the first oblique shock impinges very close to the leading edge of the reflecting plate. This was done for the sake of simplicity, since as the impingement point is moved back, boundary layer separation occurs on the reflecting plate. This results in a more complicated shock reflection pattern consisting of a separation shock, followed by some weak compression waves and finally the reattachment shock. In addition, the reflecting plate bow wave would impinge on the
wedge upstream of the reflected shock system. It is possible that this result could be used to advantage in the process of trying to cancel the reflected shock. Since some axial length is required for the full expansion to be effected around the corner, a more uniform flow may result if the reflected shock compression is also spread over a larger distance at superposition. Again, time did not permit the investigation of this possibility.
CONCLUSIONS

The following main conclusions were drawn from the results of this study:

1. A 3-shock (2 oblique, 1 normal) inlet configuration is theoretically capable of providing satisfactory efficiencies for hypersonic airbreathing flight. The efficiencies resulting from a reflected shock arrangement are sufficiently near the optimum for a 3-shock diffuser.

2. The theoretical predictions were confirmed experimentally at a Mach number of 5.4, in so far as the oblique shock losses are concerned.
REFERENCES


<table>
<thead>
<tr>
<th>α °</th>
<th>D</th>
<th>( \frac{P_{stF}}{P_{st}} )</th>
<th>( \frac{P_{F}}{P_{stF}} )</th>
<th>( \frac{P_{F}}{P_{p}} )</th>
<th>M</th>
<th>M</th>
<th>M</th>
<th>M</th>
<th>( \frac{P_{stF}}{P_{st}} )</th>
<th>( \frac{P_{Ke}}{P_{st}} )</th>
<th>( \frac{A_{F}}{A} )</th>
<th>( \frac{A_{p}}{A} )</th>
<th>( \frac{A_{e}}{A} )</th>
<th>Remarks</th>
</tr>
</thead>
<tbody>
<tr>
<td>5</td>
<td>7.5</td>
<td>.956</td>
<td>.123</td>
<td>.0048</td>
<td>4.35</td>
<td>4.15</td>
<td>4.24</td>
<td>4.41</td>
<td>1.19</td>
<td>.444</td>
<td>.465</td>
<td>.391</td>
<td>Pitot pressure reads high</td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>3</td>
<td>.956</td>
<td>.289</td>
<td>.0263</td>
<td>4.35</td>
<td>3.15</td>
<td>2.85</td>
<td>3.48</td>
<td>.743</td>
<td>.985</td>
<td>.444</td>
<td>.261</td>
<td>.145</td>
<td>Pitot probe failed before accurate reading was taken</td>
</tr>
<tr>
<td>10</td>
<td>6</td>
<td>.774</td>
<td>.238</td>
<td>.0155</td>
<td>3.48</td>
<td>3.37</td>
<td>3.38</td>
<td>3.39</td>
<td>.785</td>
<td>.988</td>
<td>.252</td>
<td>.250</td>
<td>.228</td>
<td>Pitot probe may be in B.L.</td>
</tr>
<tr>
<td>10</td>
<td>6*</td>
<td>.774</td>
<td>.217</td>
<td>.0166</td>
<td>3.48</td>
<td>3.48</td>
<td>3.34</td>
<td>3.13</td>
<td>.573</td>
<td>.97</td>
<td>.252</td>
<td>.250</td>
<td>.243</td>
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</tr>
<tr>
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<td>.774</td>
<td>.253</td>
<td>.0181</td>
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<td>3.30</td>
<td>3.28</td>
<td>3.24</td>
<td>.736</td>
<td>.985</td>
<td>.252</td>
<td>.210</td>
<td>.212</td>
<td>Pitot probe may be in B.L.</td>
</tr>
<tr>
<td>15</td>
<td>6.2</td>
<td>.539</td>
<td>.386</td>
<td>.037</td>
<td>2.74</td>
<td>2.81</td>
<td>2.80</td>
<td>2.78</td>
<td>.554</td>
<td>.969</td>
<td>.180</td>
<td>.205</td>
<td>.192</td>
<td>Pitot probe may be in B.L.</td>
</tr>
<tr>
<td>15</td>
<td>5.5</td>
<td>.539</td>
<td>.392</td>
<td>.0447</td>
<td>2.74</td>
<td>2.52</td>
<td>2.67</td>
<td>2.70</td>
<td>.535</td>
<td>.966</td>
<td>.180</td>
<td>.182</td>
<td>.174</td>
<td>Pitot probe may be in B.L.</td>
</tr>
<tr>
<td>15</td>
<td>4.0</td>
<td>.539</td>
<td>.492</td>
<td>.052</td>
<td>2.74</td>
<td>2.52</td>
<td>2.58</td>
<td>2.49</td>
<td>.451</td>
<td>.956</td>
<td>.180</td>
<td>.139</td>
<td>.169</td>
<td>Pitot probe may be in B.L.</td>
</tr>
</tbody>
</table>

(1) Calculated for incident and reflected shock only - no other losses included

* With side plates installed
DEFINITIONS OF DIFFUSER EFFICIENCY

At least two common definitions of air inlet or diffuser efficiencies are in use. The "kinetic energy efficiency" is defined in NACA (2) publications as the ratio of the kinetic energy of air expanded isentropically from diffuser exit to free stream pressure to the kinetic energy of the free stream.

for a perfect gas

since

Thus

This is seen from the h-S diagram to be

Thus

for a perfect gas

since
now \( \frac{T_1'}{T_1} \) may be evaluated from the expression for a change in entropy of a perfect gas

\[
S_y - S_x = C_p \ln \frac{T_x / T_y}{(P_x / P_y)^{\frac{y-1}{2}}}\]

thus

\[
S_1' - S_1 = C_p \ln \left( \frac{T_1'}{T_1} \right)
\]

and

\[
S_2 - S_4 = C_p \ln \left( \frac{P_{st1}}{P_{st2}} \right)^{\frac{y-1}{2}}
\]

thus

\[
\frac{T_1'}{T_1} = \left( \frac{P_{st1}}{P_{st2}} \right)^{\frac{y-1}{2}}
\]

and

\[
\eta_{KE} = \frac{(P_{st1}/P_1)^{\frac{y-1}{2}} - (P_{st1}/P_{st2})^{\frac{y-1}{2}}}{\frac{y-1}{2} M_1^2}
\]

Now

\[
(P_{st1}/P_1)^{\frac{y-1}{2}} = 1 + \frac{y-1}{2} M_1^2
\]

thus

\[
\eta_{KE} = 1 + \frac{1}{\frac{y-1}{2} M_1^2} - \frac{(P_{st1}/P_{st2})^{\frac{y-1}{2}}}{\frac{y-1}{2} M_1^2}
\]

\[
\eta_{KE} = 1 - \frac{(P_{st1}/P_{st2})^{\frac{y-1}{2}}}{\frac{y-1}{2} M_1^2}
\]

as stated in ref. 2.
Shapiro (3) defines a diffuser efficiency on a basis similar to that for a turbomachine compression process, the ratio of the energy of expansion from diffuser exit to free stream at the entering entropy to the free stream kinetic energy. Referring again to the h-S diagram, it is seen that

\[ \eta_D = \frac{h_3 - h_1}{h_2 - h_1} \]

A similar development leads to the result

\[ \eta_D = \frac{(P_{st2}/P_1)^{\frac{\gamma-1}{\gamma}} - 1}{\frac{\gamma-1}{2} M_1^2 \left( 1 + \frac{\gamma-1}{2} M_1^2 \right) \left( (P_{st2}/P_{st1})^{\frac{\gamma-1}{\gamma}} - 1 \right) \left( \frac{\gamma-1}{2} M_1^2 \right)} \]
This program computes one-dimensional flow properties for supersonic flow in the presence of plane oblique shock compression waves or isentropic plane expansion waves.

**Inputs are**

1. flow deflections in degrees \( i \)
   - 0 for free stream
   - + for compr.
   - - for exp.
2. specific heat ratio for each defl., or only 1 value
3. initial Mach number
4. stag. temperature and pressure \(^\circ\)K, kg/cm\(^2\)
5. number of deflections and number of specific heat ratios

The program first tests the deflection for \((-0+)\)

1. if - or 0,
   
an iteration is made using Prandtl-Meyer expansion equations to find the new Mach number. The tolerance is \(.001 \Delta M\)

2. if +,
   
   the cubic equation in \(\sin^2 \theta\) is solved to find \(\theta\), the shock wave angle. Only the weak shock solution is retained. Then the Mach number behind the shock and the total pressure loss are calculated. The flow deflection is recalculated to find the accuracy of solution.
At this point for all deflections, the new flow properties are computed; static pressure and temperature, total pressure behind a normal shock, stream tube area ratio, and Reynolds number per unit length. For temperature above 100°K the Sutherland law is used. Below 100°K a linear relation is used for viscosity.

The program then returns for a new deflection. If the n° of C's input is less than the n° of D's, the last value is used.
24.

SYMBOL LIST

Rey  Reynolds number per cm.  AA = area ratio
D(I)  deflection in degrees-input-first value must be zero
G(J)  \gamma_1  specific heat ratio
K  no of D's
L  no of G's
XM  free stream Mach no - unchanged in iteration
PO  free stream stag. pressure kg/cm^2
TO  free stream stag. temperature °K
XZ  (Mach number)^2 - not necessarily free stream - changes during iteration
D  D(I) radian
G  G(I)
S2D  sin^2 D
B  constant in cubic equation solution
C  "  "  "  "
D  "  "  "  "
DA  "  "  "  "
P  "  "  "  "
Q  "  "  "  "
U  "  "  "  "
XI  "  "  "  "
S2T  sin^2 \theta
THR  \theta, radians
THD  \theta, degrees
SMX  (sin^2 \theta) max for sonic flow
CTD  cot D - computed
DL  computed flow deflection, degrees
PR  static pressure ratio across flow defl.
PS  static pressure (mm Hg)
TS  static temperature °K
POR stag. pressure ratio across flow defl.
POP stag. pressure behind normal shock
XM2 new Mach number = $\sqrt{XZ}$
C PROGRAM - OBSHOCKEXP.

PROGRAM OBSHOCKEXP.

DIMENSION G(25)

ACCEPT TAPE 1, K, L, XM, PO, TO
1 FORMAT(13, 13, F7.3, F6.2, F5.1)
3 FORMAT(F6:2)

DO 4 J = 1, L
4 ACCEPT TAPE 5, G(J)

5 FORMAT(F5:3)

PPO = PO
X2 = XM**2

DO 200 I = 1, K

ACCEPT TAPE 3, D
D = 57.29578
IF(L = 1) 9, 8, 8

8 J = 1
S = G(J)
E = -1.

9 PPO = 1.
THD = 0.

IF(D) 10, 10, 12

10 WH = N.

20 XH0 = ATAN( (E*(X2-1.)/F)**.5)
XH1 = (F/E)**.5*XM - ATAN((X2-1.)**.5)
IF(WH) 30, 31, 31

30 XN2 = XH1 - D
31 D = XH2 - XN1
IF(D)**.5 = 0.0012) 22, 22, 21

21 D = DM*X2**.5*(1. + E*X2/2.)/(X2-1.)**.5
X2 = X2 + 2.*DM*XM**.5*DM**2

While 1.

IF(WH = 20.) 20, 33, 33

33 PRINT 80
GO TO 200

22 DL = D + DN)*57.29578
GO TO 190

12 S2D = SIN(D)**2
B = (X2 + 1.)/X2 - S2D
C = (2.*X2 + 1.)/X2**2 + (F**2/4.+E/X2)*S2D
DA = (S2D - 1.)/X2**2


IF(D**2 - P**3) 101, 102, 102

102 PRINT 50
GO TO 200

50 FORMAT(2X5HSCF/)
GO TO 200

101 U = ATAN( (P**3 - Q**2)**.5)/Q
IF(Q) 11, 14, 15

14 U = U + 3.141593
15 X1 = 2.*X2**.5*COS(U/3.+4.1888)
S2T = X1 - B/3.

THD = THD*57.29578
SMX1 = (F*X2 - 3.+S)/{(4.*S*X2)
SMX2 = (F*(F*X2**2-2.*(3.-S)*X2+S+9.))***.5
SMX = SMX1 + SMX2/(4.*S*X2)
SM = SMX103, 104, 104

104 PRINT 70
GO TO 200

70 FORMAT(2X6HM2 = 1.0)
GO TO 200

103 CTD = SIN(THD)/COS(THD)*((F*X2/(2.*(X2*S2T-1.)))-1.)
DL = (ATAN(1./CTD))*57.29578
PR = (2.*S*X2*S2T-E)/F
POR = (F*PR+E)/(E*PR+F)**(S/E)
PO = PO*POR
GO TO 80

80 FORMAT(2X6HM=20)
X2 = 2.*X2*(PR+F)+E-2.*(PR**2-1.)

190 Y2 = 2.*X2*(PR+F)+E

PS = (PO/I.+E*X2/2.)**.5*735.59
TS = TO/(1.+E*Y2/2.)
PP = PO*(F/Y2*(E+2.))**(S/E)*F/(2.*S*Y-E)**(1./E)

Y2 = Y2 + 1.

AA = (XM*PP*(Y2*PP*1.)*(1.+E*X2**2/2.)/(1.+E*X2/2.))*(-F/(2.*E))

IF(TS = 100.) 105, 106, 106

105 REL = 1.1176*10.**6*Y2**2*PS/TS*(S/TS)**.5
GO TO 107

106 REL = 5.514*10.**6*Y2**2*PS/TS*(S/TS)**.5

107 PRINT 108, Y2, S, DL, THD, PS, TS, PO, POP, REL, AA
GO TO 108


200 CONTINUE
PAUSE
GO TO 85
END
REFLECTED SHOCK INLET CONFIGURATION

FIGURE 1
DEFLECTION ANGLES FOR OPTIMUM 3-SHOCK INLET
(MAXIMUM TOTAL PRESSURE RECOVERY) $\gamma = 1.4$

DEFLECTION ANGLES, $\delta_1, \delta_2$ DEGREES

FREE STREAM MACH NUMBER, $M_\infty$

FIGURE 2
TOTAL PRESSURE RECOVERY FOR THREE-SHOCK INLETS $\gamma = 1.4$

- Present data ~ 15°
- Isentropic inlet, Ref. 1

Free Stream Mach Number

Figure 3
THEORETICAL SPACING FOR WAVE CANCELLATION

\[ \frac{D}{L} \]

\( \text{INCIDENCE ANGLE, } \alpha \text{, DEGREES} \)

\( M_0 = 5.3 \)
\( \gamma = 1.4 \)

INVISCID, 2-SHOCKS

FIGURE 4
FIG. 5 - SHADOWGRAPHS OF REFLECTED SHOCK INLET CONFIGURATION
PRESSURE DISTRIBUTION -10°

FIGURE 7
PRESSURE DISTRIBUTION - 15°

DISTANCE FROM UPPER PLATE LEADING EDGE

DISTANCE FROM LOWER PLATE LEADING EDGE

FIGURE 8
TOTAL PRESSURE RECOVERY FOR
REFLECTED SHOCK INLET

FIGURE 9