ANALYSIS OF A TWO-DIMENSIONAL
TYPE VI SHOCK-INTERFERENCE PATTERN
USING A PERFECT-GAS CODE AND A REAL-GAS CODE *

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INTRODUCTION

To determine the complete convective heat-transfer distribution for configurations flying at hypersonic speeds, one must consider the viscous:inviscid interactions associated with the complex three-dimensional flow fields. Because of the complexity of the viscous:inviscid interaction phenomena, many investigators have studied the locally perturbed flow fields using models consisting of basic elemental combinations. By varying the relative position of shock-generating elements, Edney observed that there are six basic shock-interference patterns. Hains and Keyes have categorized shock-interference patterns obtained for a variety of space-shuttle configurations in terms of the models by Edney.

Bertin, et al., examined surface-pressure and heat-transfer-rate data for a variety of shuttle-orbiter configurations over an angle-of-attack range from $0^\circ$ to $60^\circ$. It was found that the "type" of shock-interference pattern was dominated by the leading-edge effective sweep angle. For the relatively low sweep angles of the straight-wing orbiters, the interaction between the bow-generated shock-wave and the wing-generated shock-wave was a "Type V" shock-interference pattern. For the delta-wing orbiters, the shock:shock interaction exhibited the characteristics of a "Type VI" pattern for all alphas.

Theoretical calculations of the inviscid shock-interaction flow-field, coupled with empirically derived surface-pressure:heat-transfer correlations of the viscous interaction phenomena, indicate that the local increases in heat transfer and in surface pressure associated with the Type IV and with the Type V interactions become markedly more severe as $\gamma$ decreases. Since the current shuttle entry-configurations have highly swept wings, the present investigation was undertaken to determine the effect of the gas-property model on the Type VI shock-interference pattern.
Numerical codes were developed to calculate the two-dimensional flow field which results when supersonic flow encounters double wedge configurations whose angles are such that a Type VI pattern occurs. The flow-field model included the shock-interaction phenomena observed in ref. 3 for a delta-wing orbiter. Two numerical codes were developed: one which used the perfect gas relations and a second which incorporated a Mollier table to define equilibrium air properties. The two codes were used to generate theoretical surface-pressure and heat-transfer distributions for velocities from 1167 m/sec (3821 ft/sec), i.e., a wind-tunnel condition, to 7610 m/sec (25,000 ft/sec), i.e., an entry condition.
NOMENCLATURE

- $a$ - speed of sound
- $C_p$ - pressure coefficient
- $\frac{dP}{dn}$ - pressure gradient normal to surface
- $f''$ - velocity gradient function
- $g'$ - stagnation enthalpy gradient function
- $h$ - static enthalpy
- $H$ - stagnation enthalpy
- $M$ - Mach number
- $n$ - transformed $\eta$-coordinate
- $P$ - pressure
- $Pr$ - Prandtl number
- $\dot{q}$ - heat-transfer rate
- $\dot{q}_{t,\text{ref}}$ - heat-transfer rate to reference sphere
- $r$ - radius of cross-section for a body of revolution
- $R$ - radius of reference sphere
- $Re_x$ - Reynolds number based on local properties and wetted length
- $R_{\text{gas}}$ - gas constant
- $S$ - wetted distance along wing leading-edge, also entropy
- $T$ - static temperature
- $T_r$ - recovery temperature
- $U$ - streamwise velocity (capital letter denotes free-stream velocity)
- $X$ - axial coordinate
- $\alpha$ - transformation parameter
- $\gamma$ - ratio of specific heats
- $\delta$ - initial turning angle
- $\eta$ - transformed $y$-coordinate
\( \theta \)  - shock angle \\
\( \Lambda \)  - sweep angle (see Fig. 1) \\
\( \Lambda_s \)  - complement of sweep angle (see Fig. 1) \\
\( \mu \)  - viscosity \\
\( \nu \)  - Prandtl-Meyer angle \\
\( \varepsilon \)  - pressure ratio across a shock wave \\
\( \rho \)  - density \\

**Subscript**

\( e \)  - parameter evaluated at the boundary layer edge \\
\( \text{ex} \)  - parameter evaluated in the expansion fan \\
\( \infty \)  - parameter evaluated in the free-stream region \\
\( n \)  - parameter normal to shock wave \\
\( t \)  - indicates stagnation condition \\
\( t_2 \)  - indicates stagnation condition in region 2 \\
\( w \)  - parameter evaluated at the wall
THEORETICAL ANALYSIS

The complex flow field, which is established when a high-speed flow encounters a double-wedge configuration, is dominated by a shock-interaction region which imposes a highly non-uniform flow field adjacent to the wedge boundary layer. If the two wedge angles are not too large, the shock waves associated with the flow deflection are attached. The shock-interference pattern which results for this case has been designated by Edney as a Type VI pattern. For the computer code developed in the present study, it was assumed that the Type VI shock-interference pattern for a double wedge configuration (as shown in Fig. 1) includes:

1) the undisturbed free-stream flow,
2) the flow turned through the angle $\delta$ by a single shock wave,
3) the flow turned through the angle $\Lambda_s$ by two shock waves,
4) the flow processed by the right-running waves of the expansion fan which are centered at the intersection of the two shock waves,
5) the flow which passes through the left-running waves produced by the reflection of the waves of the expansion fan, and
6) the flow turned through the angle $\Lambda_s$ by a single shock wave.

The flow near the juncture of the two wedges, i.e., in region 3, has passed through two shock waves. Further outboard on the downstream wedge, i.e., in region 6, the flow has passed through a single shock wave and is, therefore, at a lower pressure than that in region 3. Thus, although the flow directions are the same, the gas must
undergo an expansion from the root region to equalize the pressure. The flow accelerates isentropically through the expansion regions so that the pressure and the flow direction in region 5E are identical to the values for region 6. The current study is concerned with the shock-interaction flow-field phenomena, which directly influence the wedge surface flow properties (i.e., pressure, shear, and heat transfer). Therefore, the flow-field properties along the wall were calculated including the interactions between the right-running and left-running waves in the numerical code. However, no attempt was made to model the shear layer which develops between the shock wave and the expanding flow, since this shear layer does not interact with the wedge surface for the two-dimensional Type VI interaction.

**Perfect-gas code.** - The first steps were the calculation of the flow conditions downstream of an oblique shock for a given flow-deflection angle and for a given gamma, i.e., in regions 2, 3, and 6. The flow conditions in region 5E could then be calculated, since the static pressure in region 5E is equal to the static pressure in region 6 and, under the isentropic-expansion assumption, the stagnation pressure in region 5E is equal to the stagnation pressure in region 3. The expansion process by which the gas accelerates from region 3 to region 5E was divided into ten equal steps. To satisfy the physical boundary condition that the flow in region 5E be parallel to the wall, the total change in the Prandtl-Meyer angle was divided into ten equal parts: the five right-running waves of region 4 and the five reflected waves constituting region 5. The governing equations for these calculations are summarized in ref. 6. Having defined the
inviscid flow-field and, hence, the conditions at the edge of the boundary layer, the heat-transfer distribution along the downstream wedge was calculated using the Eckert-reference-temperature technique. For the wind-tunnel flow-condition, where the perfect gas relations accurately describe the gas behavior, the Eckert-reference-temperature heating rates compared favorably with the values obtained using the nonsimilar boundary-layer code described below. Due to the simplicity of the Eckert method, it was used with the perfect-gas calculations. The boundary-layer of the downstream wedge was assumed to originate at the intersection of the two wedges (point 0 in Fig. 1).

One way of approximating the high-temperature, or real-gas, properties of air is to use lower values of gamma in the perfect-gas relations. Therefore, the equations for the perfect-gas code have been written so that one can input one value of gamma for regions 1 and 2, another gamma for regions 3 through 5E, and a third value for region 6. Thus, one can "account for" the varying shock strengths. The perfect-gas code was used to generate three "different" types of solution. For the present paper, these types of solution are referred to as:

(a) "perfect-gas" solution for which $\gamma = 1.400$ throughout the flow field,
(b) "constant-gamma" solution for which $\gamma = 1.200$ throughout the flow field,
(c) "variable-gamma" solution for which different values of gamma were assigned to the three input gamma parameters. (The required values were obtained from the real-gas solutions.)

To be consistent in the perfect-gas assumption, the specific heat of air was held constant both for the constant-gamma and the variable-gamma solution. Sutherland's relation was used to calculate the viscosity.
Assuming the Prandtl number to be 0.7 uniquely determined the thermal conductivity.

Real-gas code. - Philosophically, the calculation procedure for the shock-interaction pattern using the real-gas code was similar to that described for the perfect-gas code. However, to account for the high-temperature, or "real-gas", effects all thermodynamic properties are evaluated using numerical charts for air in chemical equilibrium. Reference 9 was used to define the temperature dependence of the transport properties of equilibrium air, i.e., viscosity, thermal conductivity, and specific heat. Once the oblique shock relations have been used to define the static pressure and the entropy for the two end regions, the expansion from region 3 to region 5E is divided into ten equal steps. The flow conditions in the intermediate regions of the isentropic expansion are calculated using the relations of ref. 10. The relations require the static enthalpy and the local speed of sound for the intermediate regions, which are evaluated using the tables for the equilibrium air.

The heat-transfer distribution along the downstream wedge is calculated using a nonsimilar boundary-layer code, which was modified so that the thermodynamic properties for the viscous flow would also be calculated using ref. 8. In addition to accounting for the "real-gas" effects, the effect of the acceleration of the inviscid flow is included in the nonsimilar code. As before, the boundary layer is assumed to originate at the junction of the two wedges.
DISCUSSION OF RESULTS

The objective of the current analytical investigation was to determine how to apply the shock-interference data obtained in the wind tunnel\(^3\) to the reentry of a shuttle orbiter. For simplicity, the numerical code was restricted to a two-dimensional flow model with the second, or downstream, wedge representing the wing leading edge. The initial deflection angle \(\delta\) was 5\(^\circ\) for all solutions. This value was chosen, because the shock standoff distance in the vicinity of the wing leading-edge correlated reasonably well with the bow-wave trace observed in wind tunnel tests. The deflection angle \(\Lambda_s\) of the downstream wedge was varied from 25\(^\circ\) to 63\(^\circ\), which corresponds to leading-edge sweep angles from 65\(^\circ\) to 27\(^\circ\). The dimensions of the 0.009 scale orbiter tested in Tunnel B of AEDC\(^3\) were used to define the characteristic lengths of the wedges.

For this study, numerical solutions were generated to determine the effect of gas properties on the flow field and of the wall temperature on the heat transfer in the region where the Type VI shock-interaction influenced the "wing leading-edge". Flow-field solutions were generated for three free-stream conditions.

1) a wind tunnel condition where
\[
U_\infty = 1167 \text{ m/sec}, \quad P_\infty = 2.98 \text{ mmHg}, \quad T_\infty = 53^\circ \text{K}
\]
\((U_\infty = 3821 \text{ ft/sec}, \quad P_\infty = 0.057 \text{ psia}, \quad T_\infty = 95^\circ \text{R})\)

2) an orbiter entry condition where
\[
U_\infty = 4330 \text{ m/sec}, \quad P_\infty = 0.333 \text{ mmHg}, \quad T_\infty = 273^\circ \text{K}
\]
\((U_\infty = 14,200 \text{ ft/sec}, \quad P_\infty = 0.0064 \text{ psia}, \quad T_\infty = 491^\circ \text{R})\)
3) an orbiter entry condition where
\[ U_\infty = 7610 \text{ m/sec}, \ P_\infty = 0.0268 \text{ mmHg}, \ T_\infty = 1950^\circ K \]
\[ (U_\infty = 25,000 \text{ ft/sec}, \ P_\infty = 0.00052 \text{ psia}, \ T_\infty = 352^\circ R) . \]

Solutions were obtained using both the perfect-gas code and the real-gas code at all flow conditions. In addition, the variable-gamma option was used to generate solutions for flow conditions 2 and 3. Heat-transfer distributions along the downstream wedge were obtained for all three free-stream conditions for a wall temperature \( T_w \) of 3940^\circ K (710^\circ R) and for the two entry conditions for a \( T_w \) of 16400^\circ K (2960^\circ R).

The effect of the gas properties on the calculated geometry of the Type VI shock-interference pattern is illustrated in Fig. 2. The output from the real-gas solutions was used to define the gamma distribution for input for the variable-gamma solution. Thus, \( \gamma_1 = \gamma_2 = 1.400, \gamma_3 = \gamma_4 = \gamma_5 = 1.214, \) while \( \gamma_6 = 1.163. \) The geometry for the variable-gamma solution compares favorably with the real-gas geometry. The region where the leading edge is influenced by the shock interaction is essentially the same for these two solutions. The shock layer is thicker for the real-gas solution. For these deflection angles, the location and the extent of the surface affected by the expansion fan differs little between the perfect-gas solution and the real-gas solution. This similarity exists even though, in region 3, \( \gamma \) is 1.214 for the real-gas solution and is, of course, 1.4000 for the perfect-gas solution. However, because the density ratio across a shock wave is higher when the real-gas properties are accounted for, the shock layer along the wing leading-edge is markedly thinner for the real-gas solution.

The pressure distributions along the leading edge of a 60^\circ sweep "wing" are presented in Fig. 3 for the entry velocity of 4330 m/sec. So that the pressure variations could be seen more clearly, the scale
has been greatly expanded for the region where the expansion fan impinges on the "wing leading-edge". The vertical marks in Fig. 3 indicate the various regions of the field, using the same legend as the theoretical distributions.

As noted in Fig. 2, the interaction-perturbed region for the perfect-gas solution is inboard relative to the real-gas solution, although the locations differ only slightly. However, for a given region, the perfect-gas surface-pressure correlates quite well with the real-gas value (Fig. 3a), with the perfect-gas solution yielding a slightly higher pressure. Over the range of the free-stream conditions of the present study, the perfect-gas surface-pressure in a given region of the flow field was within 10% of the real-gas value. As noted in Fig. 2, the locations of the interaction-perturbed regions for the real-gas solution and for the variable-gamma solution are in close agreement. However, the pressures for the variable-gamma solution are significantly lower than the real-gas solution. The discrepancy between the real-gas and the variable-gamma values of surface pressure in a given region typically varied from 15% to 25%.

The heat-transfer distributions along the wing leading-edge are presented in Fig. 4 for the entry velocity of 4330 m/sec. The local heat-transfer rate has been divided by the stagnation-point heating rate for a reference sphere, whose radius was chosen to be 0.0027m and which is at the same temperature as the wedge surface. The dimensionless heat-transfer parameter \( \dot{q}/\dot{q}_{t,\text{ref}} \) (or, since the wall temperatures are equal, the equivalent ratio of heat-transfer coefficients) is commonly used in shuttle application. Heat-transfer distributions were calculated for wall temperatures of 394°C and of 1640°C.
The dimensionless heat-transfer distributions computed using the real-gas code are compared with the perfect-gas solutions and the variable-gamma solutions in Figs. 4a and 4b, respectively. The wall-temperature variation had little effect on the heat transfer. The heat-transfer distributions for the variable-gamma solution correlate closely with the real-gas heat-transfer distributions. The difference in the location of the interaction-perturbed region contributes to the only significant difference between the perfect-gas heat-transfer and the real-gas heating (and the difference is magnified by the expanded scale). However, for engineering applications, both the perfect-gas distribution and the variable gamma distribution are in satisfactory agreement with the real-gas distribution. Similar correlations were found for the highest velocity solution. However, near the wing root, the difference between the perfect-gas heat-transfer and the real-gas heating was slightly greater at the highest velocity (i.e., condition 3 in Fig. 9).

The local increases in the real-gas heat-transfer distributions which are evident at the beginning of each flow region are due to the local acceleration of the inviscid flow. The waves of the expansion fan produce a step-function decrease in pressure and a corresponding step-increase in the local velocity at the edge of the boundary layer. Thus, the nonsimilar boundary-layer solutions yield local increases in heating due to the local velocity gradient. There are no locally severe heating rates, which could cause design problems, indicated either in the real-gas solutions or in the perfect-gas solutions. However, the flow model for the calculations does not include imbedded shock waves or other three-dimensional flow phenomena, which might occur near the wing-root fairing. Such flow phenomena caused local increases in heating to delta-wing orbiter configurations.
The location and the extent of the interaction-perturbed region of the "wing leading-edge" for the real-gas solutions are presented in Fig. 5 as a function of leading-edge sweep. The limits of the band represent the intersections of the limits of the centered expansion fan, i.e., region 4 of Fig. 1, with the leading edge. The locations are presented as the distance from the junction of the two wedges (S) divided by the radius of a reference sphere (R, which is equal to 0.0027m). Calculated locations from the real-gas solutions are presented for sweep angles from 60° down to the minimum sweep angle for which a Type VI pattern exists. The minimum sweep angle decreases as the free-stream velocity increases. For the higher velocity entry-condition, the Type VI pattern is possible for sweep angles as low as 27°. The perturbed region moves inboard toward the wedge junction as the velocity increases.

The effect of the free-stream velocity on the nondimensionalized heat-transfer rate is indicated in Fig. 6. Calculations are presented for region 3 (upstream of the expansion fan) and for region 5E (downstream of the expansion fan). Since the interaction-perturbed region moves inboard as the velocity increases, the point in region 3 is near the inboard edge of the interaction region for the highest velocity. Correspondingly, the point of region 5E is near the outboard edge for the lowest velocity. For both the perfect-gas solutions and the real-gas solutions and for both wall temperatures, the nondimensionalized heat-transfer rate increases significantly with velocity. The dimensionless heating for the higher entry velocity is roughly twice that for the wind-tunnel condition for region 3, somewhat less for region 5E. This implies that one should not extrapolate wind-tunnel data directly to flight conditions. Instead the wind-tunnel data should be used to construct a viable model of the flow field. The flow-field model can then be used to generate the required aerothermodynamic environment at the conditions of interest.
The shock-interaction geometry, the surface-pressure distributions, and the heat-transfer distributions are presented in Figs. 7, 8, and 9, respectively, for the highest velocity condition (i.e., condition 3) with a sweep angle of 60°. The correlations between the perfect-gas solution, the variable-gamma solution, and the real-gas solution are similar to those observed in the previous figures for the middle velocity condition.

As can be seen in Fig. 7, the interaction-perturbed regions are virtually the same for the variable-gamma solution and for the real-gas solution. The location for the perfect-gas solution is only slightly inboard. The shock layer for the real-gas solution is slightly thinner than that for the variable-gamma solution and markedly thinner than that for the perfect-gas solution.

The relations between the various solutions for the leading-edge pressure-distributions obtained at the highest velocity correspond to those noted for the middle velocity solutions. For a given region of the expansion fan, the perfect-gas pressure is in reasonable agreement with the real-gas value. The difference between the perfect-gas distribution and the real-gas distribution is accentuated by the expanded scale of Fig. 8. Even though the real-gas solution was used to specify the input values of gamma for the variable-gamma solution, the variable-gamma pressures do not match the real-gas values as well as the perfect-gas pressures.

As can be seen in Fig. 9, the wall-temperature variation has no significant effect on the theoretical heat-transfer distributions. The variable-gamma distribution closely follows the real-gas distribution. The differences between the perfect-gas heat-transfer and the real-gas heating are greatest in regions 3 and 4AW, i.e., near the junction of the two wedges. Downstream, the correlation between the real-gas solution and the other two solutions is similar.

Theoretical solutions have been obtained for a variety of sweep angles.
The solutions of the Type VI shock-interaction for flow condition 2 are presented in Figs. 10 through 12 with a sweep angle of 40°. These solutions exhibit several interesting characteristics. For this flow condition, the shock-interference pattern undergoes critical changes as the sweep angle varies. No perfect-gas solution is presented. This is because, for this wedge geometry, the perfect-gas relations require a Type V pattern. Even at hypersonic speeds, perfect air can not turn through a single, linear shock wave parallel to a 50° wedge. Also note that the curved shock region near the reflected waves is not presented in Fig. 10a. This is because the numerical scheme used in the present real-gas code does not yield the continuously curved shock required to turn the flow from the free-stream direction parallel to the flow in region 4E. However, a solution still exists for region 6 and, therefore, region 5E. Therefore, the shock-interference pattern is still basically a Type VI pattern. The curved shock would, of course, be produced in the actual flow. But since the velocity gradients and the pressure gradients of curved streamlines are not modeled numerically in the current code, the code outputs for this situation: "curved shock not modeled". It has been noted that the geometry for the variable-gamma solution is only slightly different than the real-gas geometry. However, because of the difference, the complete variable-gamma geometry (within the assumptions of the numerical code) is generated (Fig. 10b).

As was noted at the larger sweep angles, the variable-gamma pressure do not correlate exceptionally well with the real-gas values (Fig. 11), but the heat-transfer rates do correlate well (Fig. 12).

To simulate the large density changes which exist across a shock wave in hypersonic flight, experimental investigators often make use of wind tunnels for which the test gas has a relatively low value of gamma, e.g., ref. 13. Using tetrafluoromethane (CF₄) as the test gas, the free-stream
specific-heat ratio for the Langley facility varies from 1.17 to 1.31 at Mach 6 (ref. 14). Therefore, theoretical solutions of the Type VI shock-interference pattern for the double-wedge configuration with $\Lambda = 60^\circ$ have been computed for a perfect gas with $\gamma = 1.2$ throughout the flow-field. These solutions are designated "constant-gamma" solutions.

The location of the interaction-perturbed region is presented in Fig. 13 as a function of the free-stream velocity. Reviewing the legend:

(a) "perfect-gas" uses the perfect-gas relations with $\gamma = 1.400$ throughout the flow field,
(b) "constant-gamma" uses the perfect-gas relations with $\gamma = 1.200$ throughout the flow field,
(c) "variable-gamma" uses the perfect-gas relations with $\gamma_1$, $\gamma_3$, and $\gamma_6$ specified from the real-gas solution ($\gamma_1 = \gamma_2$ and $\gamma_3 = \gamma_4 = \gamma_5$), and
(d) "real-gas" uses the equilibrium air properties to describe the gas behavior.

Since $\gamma$ for air is, in fact, essentially 1.4 throughout the flow field for the lowest velocity considered, no variable-gamma solution was obtained for this condition (i.e., condition 1). With the exception of the constant-gamma solution, the various solutions provide similar locations of the interaction-perturbed region over the velocity range considered. At the higher velocities, the interaction-perturbed region is relatively inboard for the constant-gamma solution. For a given geometry of the double-wedge configuration, the location of the interaction-perturbed region is a function of the bow shock-wave angle (generated by the initial deflection), of the leading-edge shock-wave angle (generated by the second deflection), and of the Mach numbers in regions 3 and 4 (which determine the expansion waves). The shock angle for $\gamma = 1.2$ is nearer the surface than is the shock angle for
Thus, since the actual $\gamma$ is essentially unchanged for the first deflection angle, the bow shock-wave for the constant-gamma solution intersects the leading-edge shock nearer the root than is the case for the real-gas solution or for the perfect-gas solution (which yield essentially identical results). The low-velocity, constant-gamma solution for the interaction region correlates with the other solutions because both the bow shock angle and the leading-edge shock angle are relatively small causing the shock-shock intersection to be relatively outboard.

The pressure distributions along the wing leading-edge are compared in Fig. 14. For the wind-tunnel condition, i.e., condition 1, the perfect-gas solution ($\gamma = 1.400$ everywhere) yields flow conditions in regions 3 and 6 which are identical to the corresponding flow conditions computed using the real-gas code. Differences occurred in the perfect-gas solution and the real-gas solution for the local pressures and the interaction locations of the expansion fan. Because regions 3 and 6 are identical and because the differences between the two solutions are attributed to the difficulty in using the Mollier charts at these low temperatures, the perfect-gas solution with $\gamma = 1.400$ (Fig. 14a) represents the actual flow. The constant-gamma solution yields pressures which are significantly lower than the other two solutions. As noted previously, the angles both for the bow shock and the leading-edge shock are relatively small for $\gamma = 1.200$ and, therefore, the interaction region matches that for the other two solutions.

For the middle velocity condition (Fig. 14b), the shock-interaction region for the constant-gamma solution is relatively inboard. It has been established that the difference is due to the fact that the initial shock wave is weak and, therefore, does not significantly alter gamma from its actual free-stream value of 1.400. When comparing the pressure from a
given region, the constant-gamma value is lower than the real-gas value by approximately the same amount that the perfect-gas value is higher than the real-gas value.

For condition 3 (Fig. 14c), the constant-gamma pressure in a given region is in very good agreement with the perfect-gas pressure for that region. For a given region, the real-gas solution yields pressures somewhat lower than the two other solutions. Again, however, because the bow shock-wave generated by the initial turning of the flow is much closer to the body when $\gamma = 1.200$ than it is for the real-gas solution or for the perfect-gas solution, the constant-gamma interaction region is markedly inboard.

The heat-transfer distributions for the leading-edge of a wing with 60° sweep are presented in Fig. 15. The constant-gamma solution is compared with the perfect-gas and the real-gas solutions. Because the free-stream values of temperature and of pressure were used to define the flow condition, negative heat transfer, or cooling, existed when $\gamma = 1.200$ for the wind-tunnel flow condition. Thus, heat-transfer distributions are not presented for the wind-tunnel condition. Heat-transfer distributions are presented for conditions 2 and 3 in Figs. 15a and 15b, respectively. The comparisons between the various theoretical solutions are similar for both flow conditions. The differences in the heat-transfer distributions are due principally to the differences in the locations of the interaction-perturbed regions. These differences are greatest at the higher velocity. The perfect-gas solutions ($\gamma = 1.400$ throughout) compare more favorably with the real-gas heat-transfer distributions than do the constant-gamma solutions ($\gamma = 1.200$ throughout).
CONCLUDING REMARKS

Using a two-dimensional flow model of the Type VI shock-interaction pattern, the aerothermodynamic environment has been calculated for a "simulated" wing leading-edge of a delta-wing orbiter. Calculations have been made for velocities from 1167 m/sec to 7610 m/sec for perfect-gas properties, for constant-gamma gas properties, for variable-gamma gas properties, and for real-gas properties. Based on the calculations of the present study, the following conclusions are made.

1. Free-stream flight conditions were found to produce Type VI interaction patterns for effective wing leading-edge sweep angles as low as $27^\circ$, when the real-gas effects were considered.

2. Perfect-gas solutions for the flow geometry and the pressure distribution were in good agreement with the real-gas solutions. The use of effective gammas did not adequately represent real-gas effects in the surface-pressure distribution.

3. The correlation between the perfect-gas solution and the real-gas solution for the heat-transfer distribution was essentially independent of the wall temperature, but depended on the free-stream velocity. The heat-transfer distributions for the variable-gamma solution correlated closely with the real-gas heat-transfer distributions. No locally severe heating rates, which would cause design problems, were found.

4. When the local heat-transfer rates were nondimensionalized using a current shuttle design parameter, the dimensionless heat transfer increased significantly with velocity. The increase occurred both for the perfect gas solutions and for the real-gas solutions and for both wall temperatures. Thus, one
should not extrapolate wind-tunnel data directly to flight conditions. Instead the wind-tunnel data should be used to construct a realistic model for the flow-field, which can be used to generate the required aerothermodynamic environment.

5. The shock-intersection geometry and its effect on the local flow-field was found to be a complex function of gamma. The interaction-perturbed region depends on the "bow" shock-wave angle, the "leading-edge" shock-wave angle, and the expansion wave angles (or, equivalently, the local Mach numbers). Although gamma often had a significant effect on the locations of the interaction-perturbed region, for a given flow condition, the location differences were never severe. This consistent correlation occurred because the shock angle discrepancies tended to be compensating in many cases and because the interaction occurred relatively near the wing root so that significant differences in angle resulted in relatively minor differences in length.
REFERENCES


Figure 1. - Flow model of the Type VI shock-interference pattern for a double wedge.
Figure 2. - Calculated geometry of the Type VI shock-interference pattern; $U_\infty = 4330 \text{ m/sec}$, $P_\infty = 0.333 \text{ mmHg}$, $T_\infty = 273^\circ \text{K}$, $\Lambda = 60^\circ$.
Figure 2. - Continued.

(b) Real-gas solution
Figure 2. - Concluded.

(c) Variable-gamma solution ($\gamma_3 = 1.214$, $\gamma_6 = 1.163$)
Figure 3. The pressure distributions for the wing leading-edge; $U_\infty = 4330$ m/sec, $P_\infty = 0.333$ mmHg, $T_\infty = 273^\circ$K, $\Lambda = 60^\circ$. 

(a) Real-gas compared with perfect-gas
FREE-STREAM CONDITION 2

--- REAL-GAS SOLUTION
--- VARIABLE-GAMMA SOLUTION

\((\gamma_3 = 1.214, \gamma_6 = 1.163)\)

(b) Real-gas compared with variable-gamma

Figure 3. - Concluded.
FREE-STREAM CONDITION 2

--- PERFECT-GAS SOLUTION, $T_w = 394^\circ K$
--- PERFECT-GAS SOLUTION, $T_w = 1640^\circ K$
--- REAL-GAS SOLUTION, $T_w = 394^\circ K$
--- REAL-GAS SOLUTION, $T_w = 1640^\circ K$

(a) Real-gas compared with perfect gas

Figure 4. - The heat-transfer distributions for the wing leading edge;
$U_\infty = 4330$ m/sec, $P_\infty = 0.333$ mmHg, $T_\infty = 273^\circ K$, $\Lambda = 60^\circ$
--- REAL-GAS SOLUTION, $T_w = 394^\circ K$
--- REAL-GAS SOLUTION, $T_w = 1640^\circ K$
--- VARIABLE-GAMMA SOLUTION, $T_w = 394^\circ K$ \( \gamma_3 = 1.214, \gamma_6 = 1.163 \)
--- VARIABLE-GAMMA SOLUTION, $T_w = 1640^\circ K$

(b) Real-gas compared with variable-gamma

Figure 4. - Concluded.
Figure 5. - The location and the extent of the interaction-perturbed region of the wing leading-edge as a function of leading-edge sweep (real-gas solutions).
Figure 6. - The effect of free-stream velocity and of wall temperature on the leading-edge heat-transfer.

(a) Point in Region 3, \( \Lambda = 60^\circ \).
Figure 6. - Continued.

(b) Point in region 5E, \( \Lambda = 60^\circ \) \( (T_w = 394^\circ K\ only) \)

\[ \frac{\dot{q}_s}{\dot{q}_{t,\text{ref}}} = 26.5 R \]
FILLED SYMBOLS: PERFECT-GAS SOLUTIONS

OPEN SYMBOLS: REAL-GAS SOLUTIONS

(c) Point in region 3, $\Lambda = 48^\circ$ ($T_w = 394^\circ K$ only)

Figure 6. - Continued.
FILLED SYMBOLS: PERFECT-GAS SOLUTIONS

OPEN SYMBOLS: REAL-GAS SOLUTIONS

\[ \frac{q_s}{q_{t,ref}} = 16.4 \]

(d) Point in region 5E, \( \Lambda = 48^\circ \) \( (T_w = 394^\circ K \text{ only}) \)

Figure 6. - Concluded.
Figure 7. - Calculated geometry of the Type VI shock-interference pattern, \( U_\infty = 7610 \text{ m/sec}, \ P_\infty = 0.0268 \text{ mmHg}, \ T_\infty = 195^\circ \text{K}, \ \Lambda = 60^\circ. \)
Figure 7. - Continued.

(b) Real-gas solution
Figure 7. - Concluded.

(c) Variable-gammas solution ($\gamma_3 = 1.169$, $\gamma_6 = 1.216$)
FREE-STREAM CONDITION 3

PERFECT-GAS SOLUTION

REAL-GAS SOLUTION

Figure 8. - The pressure distributions for the wing leading-edge; \( U_\infty = 7610 \) m/sec, \( P_\infty = 0.0268 \) mmHg, \( T_\infty = 195^\circ \)K, \( \Lambda = 60^\circ \).
FREE-STREAM CONDITION 3

--- REAL-GAS SOLUTION

----- VARIABLE-GAMMA SOLUTION

($\gamma_3 = 1.169, \gamma_6 = 1.216$)

(b) Real-gas compared with variable-gamma

Figure 8. - Concluded.
FREE-STREAM CONDITION 3

- PERFECT-GAS SOLUTION, \( T_w = 394^\circ K \)
- PERFECT-GAS SOLUTION, \( T_w = 1640^\circ K \)
- REAL-GAS SOLUTION, \( T_w = 394^\circ K \)
- REAL-GAS SOLUTION, \( T_w = 1640^\circ K \)

Figure 9. - The heat-transfer distributions for the wing leading-edge; \( U_\infty = 7610 \text{ m/sec}, P_\infty = 0.0268, \text{mmHg}, T_\infty = 195^\circ K, \Lambda = 60^\circ \).
(b) Real-gas compared with variable-gamma

Figure 9. - Concluded.
Figure 10. - Calculated geometry of the Type VI shock-interference pattern; \( U_\infty = 4330 \) m/sec, \( P_\infty = 0.0333 \text{mmHg}, T_\infty = 273^\circ \text{K}, \Lambda = 40^\circ \).

Curved shock not modeled for solution (at this sweep)

(a) Real-gas solution
Figure 10. - Concluded.

(b) Variable-gamma Solution

\( \gamma_3 = 1.159, \quad \gamma_6 = 1.158 \)
FREE-STREAM CONDITION 2

--- REAL-GAS SOLUTION

--- VARIABLE-GAMMA SOLUTION

($\gamma_3 = 1.159$, $\gamma_6 = 1.158$)

Figure 11. - A comparison of the real-gas and the variable-gamma pressure distribution for the wing leading-edge; $U_\infty = 4330$ m/sec, $P_\infty = 0.333$ mmHg, $T_\infty = 273^\circ K$, $\Lambda = 40^\circ$. 
Figure 12. A comparison of the real-gas and the variable-gamma heat-transfer distributions for the wing leading-edge; $U_\infty = 4330$ m/sec, $P_\infty = 0.333$ mmHg, $T_\infty = 273$°K, $\Lambda = 40$°.
Figure 13. - The location of the interaction-perturbed region on the wing leading-edge as a function of velocity ($\Lambda = 60^\circ$).
PERFECT-GAS SOLUTION (γ = 1.4 THROUGHOUT)

- CONSTANT-GAMMA SOLUTION (γ = 1.2 THROUGHOUT)

(a) Condition 1: $U_\infty = 1167$ m/sec, $P_\infty = 2.98$ mmHg, $T_\infty = 53^\circ K$

Figure 14. - The pressure-distribution along the wing leading-edge, $\Lambda = 60^\circ$. 
Figure 14. - Continued.

(b) Condition 2: $U_\infty = 4330$ m/sec, $P_\infty = 0.333$ mmHg, $T_\infty = 273^\circ$K

Figure 14. - Continued.
(c) Condition 3: $U_\infty = 7610$ m/sec, $P_\infty = 0.0268$ mmHg, $T_\infty = 195^\circ$K

Figure 14. - Concluded.
(a) Condition 2: \( U_\infty = 4330 \) m/sec, \( P_\infty = 0.333 \) mmHg, \( T_\infty = 273^\circ\)K

Figure 15. - The heat-transfer distribution along the wing leading-edge for \( \Lambda = 60^\circ\).
PERFECT-GAS SOLUTION
---
REAL-GAS SOLUTION
---
CONSTANT-GAMMA SOLUTION

(b) Condition 3: \( U_\infty = 7610 \text{ m/sec}, P_\infty = 0.0268 \text{ mmHg}, T_\infty = 195^\circ\text{K} \)

Figure 15. - Concluded.
APPENDIX A. - GENERAL DESCRIPTION OF PERFECT GAS CODE

The general solution procedure for the perfect-gas code has been discussed previously. This section will provide a more detailed description of the governing equations and their place in the numerical routine. Input data for the perfect-gas code consists of the free-stream flow-field conditions of region 1 and the model geometry. The procedure used to calculate flow conditions in regions 2, 3, and 6 is to first call subroutine DELTAK, which solves for the shock wave angle, and second to call subroutine PTHETA, which calculates the flow conditions of regions 2, 3, and 6 using the computed shock wave angle.

Flow-Field Conditions in Regions 2, 3, and 6. The subroutine DELTAK uses the known turning angle $\delta$ to calculate the shock wave angle $\theta$ in the following equation:

$$
\delta = \tan^{-1} \left\{ \frac{1}{\tan \theta \left[ \frac{(\gamma+1)M_1^2}{2(M_1^2 \sin^2 \theta - 1)} - 1 \right]} \right\}
$$

(1)

where the subscript 1 denotes conditions upstream of the shock. A half-interval, iteration procedure is used to determine $\theta$. The half-interval method is started by assuming $\theta$ to be an average between a lower limit, which is equal to $\delta$, and an upper limit, equal to 90°. This average shock-wave angle is used to calculate the corresponding $\delta$ from equation (1). A comparison is then made between the calculated $\delta$ and the actual $\delta$. If this comparison is not within a prespecified tolerance (i.e., 0.0001 radians) either the lower, or upper limit of $\theta$, depending upon the comparison, is set equal to the previous iteration's average value of $\theta$. A new average value of $\theta$ is computed using the new limit, and the procedure is repeated until the
calculated $\delta$ equals the actual $\delta$ (within the prespecified tolerance). Once the shock-wave angle has been determined the subroutine PTHETA uses this $\theta$ to calculate the pressure ratio by the following:

$$\frac{P_2}{P_1} = \xi = \frac{2\gamma M_{1}^{2} \sin^{2} \theta - (\gamma-1)}{\gamma + 1} \quad (2)$$

The subscript 2 denotes the conditions downstream of the shock. Thus, to solve for the flow-field conditions in region 3, conditions in region 2 are actually the conditions upstream of the shock, and conditions in region 3 are the conditions downstream of the shock. The density ratio and temperature ratio are then calculated as function of $\xi$ as follows:

$$\frac{\rho_2}{\rho_1} = \frac{(\gamma+1)\xi + (\gamma-1)}{(\gamma-1)\xi + (\gamma+1)} \quad (3)$$

$$\frac{T_2}{T_1} = \frac{\xi (\gamma-1)\xi + (\gamma+1)}{(\gamma+1)\xi + (\gamma-1)} \quad (4)$$

The Mach numbers of regions downstream of the shock wave are calculated from the equation:

$$M_2 = \left[ \frac{M_1^2 \left[(\gamma+1)\xi + (\gamma-1)\right] - 2(\xi^2-1)}{\xi[(\gamma-1)\xi + (\gamma+1)]} \right]^{1/2} \quad (5)$$

The pressure coefficient of each region is found from the equation:

$$C_p = \frac{P-P_1}{0.5\gamma P_1 M_1^2} \quad (6a)$$

which, when combined with equation 2, yields the following relation for regions 2 and 6:

$$C_p = \frac{4(M_1^2 \sin^2 \theta - 1)}{(\gamma+1) M_1^2} \quad (6b)$$

The procedure to compute the stagnation conditions for regions 1, 2, and
and 3 is to assume the flow decelerates isentropically to zero velocity. The equation for calculating the stagnation pressure is:

\[
P_{ti} = P_i \left[1 + \frac{\gamma_i \cdot 2}{2 \cdot M_i^2} \right] \frac{\gamma_i}{\gamma_i - 1}
\]

where these calculations are carried out for \(i = 1\) (the free-stream), 2, and 3 (which serves as the stagnation pressure for the isentropic expansion).

**Expansion Fan.** The flow in region 3 is assumed to accelerate isentropically to subregion 5E. Region 3 is uniquely determined, as described above. Since the streamlines in the subregions from subregion 5E through region 6 are straight and parallel to the surface, i.e., not curved,

\[
\frac{dP}{dn} = 0
\]

Thus, the static pressure in region 5E is equal to the static pressure in region 6, which is known. The isentropic assumption requires that the stagnation pressure in subregion 5E is equal to the stagnation pressure in region 3, which is known. Thus, the flow in subregion 5E is uniquely defined.

The flow field in the expansion fan is calculated in the subroutine EXPAN and the locations of the intersection points of the right running and left running waves of the fan are calculated by the subroutine INTRST. Since the waves are assumed to be linear, the subroutine INTRST requires the knowledge of two initial points and the angle between these points to the point of intersection. The intersection point can then be calculated using linear relations. The required angles are the shear layer angles and the expansion wave angles (i.e., Mach waves).
For the isentropic expansion from region 3 to region 5E, the Prandtl-
Meyer expansion equations are used. First, the Prandtl-Meyer angle \( \phi \) for
region 3 and for subregion 5E is calculated as follows:

\[
\phi = \sqrt{\frac{\gamma + 1}{\gamma - 1}} \left( \tan^{-1} \sqrt{\frac{\gamma - 1}{\gamma + 1} (M^2 - 1)} \right) - \tan^{-1} \sqrt{M^2 - 1}
\]  

(8)

(When using the variable-gamma option of the perfect-gas code, \( \phi \) both for
region 3 and for subregion 5E is calculated using the gamma for region 3).
The difference between these two Prandtl-Meyer angles is divided into ten
equal parts to give the five waves in region 4 and the five waves in region 5.
When crossing the right-running waves of the centered expansion fan,
the change in the Prandtl-Meyer function is related to the change in the
flow direction by

\[
d\phi = d\theta
\]  

(9-a)

When crossing the left-running waves reflected from the wall,
\[
d\phi = -d\theta
\]  

(9-b)

Thus, there is no net change in flow direction, satisfying the condition
that the flow in subregion 5E is parallel to the surface. The subroutine
EXPAN calculates the local Mach numbers of the expansion fan from which the
local pressures can then be calculated. The local Prandtl-Meyer angle is
calculated by adding one tenth of \( \Delta \phi \) to the \( \phi \) for the previous region. Then
using equation (8), the local Mach number can be calculated using the half-
interval iteration method. After all the local Mach numbers have been cal-
culated, the local pressures are calculated using the following equation:

\[
P_{ex} = \frac{P_{te}}{\left[ 1 + \frac{\gamma_3 - 1}{2} M_{ex}^2 \right]^{\frac{\gamma_3}{\gamma_3 - 1}}}  
\]  

(10)
The surface pressures have been calculated for the interaction between the right-running waves of the expansion fan (region 4) and the reflected left-running waves (region 5). See Figure A-1. The flow in subregion 4A is directed away from the surface by the angle $\theta$. The expansion of the flow from 4A so that it is parallel to the surface in subregion 4AW is accomplished along right-running characteristics. Thus

$$v_{4AW} = v_{4A} - (\theta_{4AW} - \theta_{4A})$$

(11)

A similar procedure is used for subregions 4BW, 4CW, and 4DW. Once the Prandtl-Meyer functions for these subregions are known, the remaining properties are calculated using EXPAN.

**Wing Leading-Edge Heat-Transfer-Rate Calculations.** After the wall pressures in regions 3, 4, and 5 have been found, the final step is to calculate the heat-transfer rate on the "wing leading-edge" in these regions. The subroutine ERTQDOT calculates the desired heat-transfer rate. The technique used to calculate the heat-transfer rate employs the Eckert's reference temperature equation for a laminar boundary-layer, as follows:

$$q = 0.332(Re_x^*)^{0.5} \ (Pr^*)^{0.333} \ k^* \ (T_r - T_w) (x)^{-0.5}$$

(12)

This heat-transfer rate is divided by the reference stagnation-point heat-transfer rate to give a non-dimensionalized heat-transfer for use in correlation.

**Calculation of the Stagnation Conditions Behind a Normal Shock and of the Reference Heating.** In addition, the stagnation pressure across a normal
shock wave (which of course does not exist in the flow-field) is calculated for region 3 and for region 6 using the equation:

\[ \frac{P_{t2}}{P_1} = \left\{ \frac{(\gamma+1)M_1^2}{2} \right\}^{\frac{\gamma}{\gamma-1}} \left\{ \frac{\gamma+1}{2\gamma M_1^2 - (\gamma-1)} \right\}^{\frac{1}{\gamma-1}} \]  \hspace{1cm} (13)

with the proper subscription of pressures and Mach numbers. The stagnation temperature is constant throughout the regions and is calculated by the adiabatic perfect-gas equation:

\[ T_t = T_1 \left[ 1 + \frac{\gamma-1}{2} \frac{M_1^2}{1} \right] \]  \hspace{1cm} (14)

These normal shock values of stagnation temperature and of stagnation pressure, in region 6 are used to calculate the stagnation point heat-transfer rate to a reference sphere.

The following equation is used:

\[ q_{t,\text{ref}} = 0.64\left(\rho_t u_t\right)^{0.4}\left(\rho_w u_w\right)^{0.1} \left(C_p T_t - C_p T_w\right) \left[ \frac{du}{dx} \right]^{0.5} (Pr)^{-1} \]  \hspace{1cm} (15)

where

\[ \left[ \frac{du}{dx} \right] = \sqrt{\frac{2 R \text{gas}}{R}} \frac{T_t}{T_w} \]  \hspace{1cm} (16)

and the subscript t refers to stagnation temperature, the subscript w refers to wall temperature, and R is the radius of the reference sphere.

**Calculation of the Shock-Wave Angles and the Intersection Points of the Shock Wave with the Expansion Fan.** Once the flow in subregion 5E is defined, the isentropic expansion flows inside of the shear layer in regions 4 and 5 is defined. It is not possible to match both the pressure and the flow direction across the "shear layer" which divides subregion 4ES, and subregion 4E (and, sequentially, 5AS, 5A, 5BS, 5B, 5CS, 5C, and 5DS, 5D). The actual flow
in subregions 4ES through 5DS is believed to be more complex than the flow-field model allows. The required pressure adjustment is assumed to be accomplished by the pressure gradient across curved streamlines:

\[
\frac{dP}{dn} = \frac{pu^2}{r}
\]

For the flow conditions studied, the radius of curvature is very large, i.e., only slight streamline curvature is needed. The subroutines DELTAK and PTHETA are used with the flow angle to generate approximate values of the shock-wave angle and the flow conditions in the subregions between the shock wave and the shear layer, i.e., subregions 4ES, 5AS, 5BS, 5CS, and 5DS. The intersection of the shock wave and the left-running expansion wave is computed using the subroutine INTRST. The expansion waves are assumed to be linear from their intersection with the shock wave. Because the pressure decreases in the expansion fan subregions, the shock-wave inclination decreases for each subsequent calculation. This results in the "curved" shock wave characteristic of the Type VI pattern, when the reflected waves interact with the wing leading-edge shock (see Figure 1).
Streamlines between the wall and the shock wave downstream of this reflected wave are parallel and straight

\(\frac{dP}{dn} = 0\)

Flow between this shear layer and the wall is isentropic

Appendix A: Figure 1. - Flow model for Type VI showing regions for which flow conditions were determined using numerical codes
Input Cards

Card # 1 NC - number of cases

Card # 2 FSMACH - free-stream Mach number

PINF - free-stream pressure (lbf/ft^2)

TINF - free-stream temperature (^O_R)

LAMDAS - the deflection angle, i.e., complement of sweep angle (^O)

Card # 3 RGAS - gas constant (1716 ft^2/sec^2^O_R)

GAMMA - free-stream gamma (usually 1.4)

Card # 4 P2 - pressure in region 2 (lbf/ft^2)

 aşağı (optional)

THETA - first shock wave angle (^O)

(f Thought)

DELTA - angle of first wedge (^O)

KTHETA - does not equal zero if DELTA, δ, is known, equals zero if P_2 or THETA, δ, are known.

KNOWN - does not equal zero if DELTA, δ, is known, equals zero if DELTA, δ, is unknown.

Card # 5 TW - wall temperature (^O_R)

RNØSE - radius of reference sphere (ft.)

PRNØ - Prandtl number

Card # 6 CPO, CP1, CP2, CP3, CP4 - specific heat coefficients in the equation

\[ C_p = C_{p0} + C_{p1}T + C_{p2}T^2 + C_{p3}T^3 + C_{p4}T^4 \] (BTU/sec)

Card # 7 XO - x-coordinate of "nose" (ft.)

Y0 - y-coordinate of "nose" (ft.)

X30 - x-coordinate of wedge intersection point (ft.)

Y30 - y-coordinate of wedge intersection point (ft.)
Card # 8  GAMMA3 - gamma in region 3 (can be used to "simulate" real gas effects)

GAMMA6 - gamma in region 6 (can be used to "simulate" real gas effects)

Card # 9  VISS0, VISS1, VISS2, VISS3, VISS4 - viscosity coefficients in the equation

\[ \mu = \mu_0 + \mu_1 T + \mu_2 T^2 + \mu_3 T^3 + \mu_4 T^4 \left( \frac{\text{lbf-sec}}{\text{ft}^2} \right) \]

(Note: Since the Prandtl number for air is approximately 0.7 for most temperatures and pressures,

\[ K = \mu \frac{C_p}{Pr} \]

\( C_p \) and \( Pr \) are assumed constant for perfect air. Since \( \mu \) is accurately calculated with the above polynomial, a reliable value for \( K \) is obtained).
PROGRAM SHOINT (INPUT, OUTPUT)

*****THIS PROGRAM WILL CALCULATE SHOCKWAVE INTERSECTIONS AND THE FLOW FIELD CONDITIONS BEHIND THE SHOCKWAVES GIVEN FREE-STREET CONDITIONS*****

REAL M1, M2, M3, M6, LAMDA, M2S
HEAD 111, NC
111 FORMAT(15)
   UO 99 IC=1, NC
HEAD 101, FSMACH, PINF, TINF, LAMDA
101 FORMAT(4E12.5)
HEAD 102, RGAS, GAMMA
102 FORMAT(2E12.5)
HEAD 103, P2, THETA, UELTA, KTHETA, KNOWN
103 FORMAT(3E12.5,2I5)
HEAD 104, TW, MIN, PHNO
104 FORMAT(3E12.5)
HEAD 105, CP0, CP1, CP2, CP3, CP4
105 FORMAT(5E12.5)
HEAD 106, Y0, Y30, Y30
106 FORMAT(4E12.5)
HEAD 107, GAMMA3, GAMMA6
107 FORMAT(2E12.5)

THETA=THETA/57.296
UELTA=UELTA/57.296
LAMDA=LAMDA/57.296
IF (KNOWN.EQ.0) GO TO 10

*****CALCULATING FLOW CONDITIONS IN REGION 2*****

CALL UELTAK (FSMACH, GAMMA, DELTA, THETA)
10 CALL PTHETA (FSMACH, M2, PINF, P2, PHAT21, DRAT21, TRAT21, THETA, UELTA,
   KTHETA, KNOWN)

RH0INF=PINF/(RGAS*TINF)
P2=PHAT21*PINF
T2=TRAT21*TINF
RH02=DRAT21*RH0INF

*****CALCULATING FLOW CONDITIONS IN REGION 3*****

UELTA3=LAMDA-DELTA
CALL UELTAK (M2, GAMMA3, UELTA3, THETA3)
CALL PTHETA (M2, M3, P2, P3, PHAT32, DRAT32, TRAT32, THETA3, UELTA3,
   KTHETA3, KNOWN)

RH03=DRAT31*RH0INF
KCOEF3=2.0*(PRAT31-1.0)/(GAMMA3*FSMACH*FSMACH)

G3P1=GAMMA3+1.0
G3M1=GAMMA3-1.0
PMF3=SQRT(G3P1/G3M1)*ATAN(SQRT((G3M1/G3P1)*(M3*M3-1.0)))-ATAN(SQRT
1((M3*M3-1.0))
PTE=P3*((1.0*(G3M1/2.0)*M3*M3)*(GAMMA3/G3M1))
****CALCULATING FLOW CONDITIONS IN REGION 6****

DELTA6=LAMDA6
CALL DELTAK(FSMACH, GAMMA6, DELTA6, THETA6)
CALL PTHETA(FSMACH, M6, PINF, P6, PHAT61, DRAT61, TRAT61, THETA6, UELTA6, IP16, GAMMA6, KTHETA, KNOWN)

P6=PHAT61*PINF
I6=THAT61*TINF
H06=DRAT61*H01INF

****CALCULATING FLOW CONDITIONS IN REGION 5E****

P5E=P6
PRAT5E=PRAT/P5E
EXM5E=SQRT(2.0*(((PRAT5E)**(GAMMA1/GAMMA3)-1.0)/GAMMA3)
PMF5E=SQRT((GAMMA1/GAMMA3)*ATAN(SQRT(((GAMMA1/GAMMA3)**(EXM5E*EXM5E-1.0)))-1)*ATAN(SQRT((GAMMA1/GAMMA3)**(EXM5E*EXM5E-1.0))))
UEXP=P5E*(PMF5E-PMF3)

****COMPUTING STAGNATION CONDITIONS****

M1S=FSMACH*FSMACH
M2S=M1S*M2
GM1=GAMMA-1.0
GP1=GAMMA+1.0
EXPON=GAMMA/GM1
PT1=PINF*(((1.0+GM1*M1S/2.0)**EXPON)
PT2=PINF*(((1.0+GM1*M2S/2.0)**EXPON)
PT3=P2*(((GP1*M2S/2.0)**EXPON)*((GP1/(2.0*GAMMA*M2S-GM1))**(1.0/GM1)
PT6=PINF*(((GP1*M1S/2.0)**EXPON)*((GP1/(2.0*GAMMA*M1S-GM1))**(1.0/GM1)
TT=TINF*1.0+GM1*M1S/2.0
TET=TT

CALL QDOT(P1b, NGAS, TWT, ITR, RNOSE, CP0, CP1, CP2, CP3, CP4, QTSPR, PRNO)

****FIND BOW AND WING SHOCKWAVE INTERSECTION****

AGL3=THETA3+DELTA
CALL INTRST(X0, Y0, THETA, AGL3, X30, Y30, XI, YI)

****FIND INTERSECTION FOR THE FIRST EXPANSION WAVE****

****EQUATION FOR THE ANGLE OF LINE 4A****

WAVE3=ASIN(1.0/M3)
AGL4A=LAMDA4A=WAVE3
CALL INTRST(X1, Y1, AGL4A, LAMDA4A, X30, Y30, X4A, Y4A)
THETA4A=LAMDA4A+DEXP

****LINE 4B****

CALL EXPAN(PMF3, DEXP, GAMMA3, PMF4A, EXM4A, M3)
WAVE4A=ASIN(1.0/EXM4A)
AGL4B=LAMDA4A+DEXP=WAVE4A
CALL INTRST(X1, Y1, AGL4B, LAMDA4B, X30, Y30, X4B, Y4B)
THETA4B=THETA4A+DEXP
CALL EXPAN(PMF4A, DEXP, GAMMA3, PMF4B, EXM4B, EXM4A)
WAVE4A = ASIN(1.0/EXM4B)
AGL4C = LAMDA + 2.0*DEXP - WAVE4B
CALL INTRST(XI, Y1, AGL4C, LAMDA, X30, Y30, X4C, Y4C)
THTA4C = THTA4B + DEXP

CALL EXPAN(PMF4B, DEXP, GAMMA3, PMF4C, EXM4C, EXM4B)
WAVE4B = ASIN(1.0/EXM4C)
AGL4D = LAMDA + 3.0*DEXP - WAVE4C
CALL INTRST(XI, Y1, AGL4D, LAMDA, X30, Y30, X4D, Y4D)
THTA4D = THTA4C + DEXP

CALL EXPAN(PMF4C, DEXP, GAMMA3, PMF4D, EXM4D, EXM4C)
WAVE4C = ASIN(1.0/EXM4D)
AGL4E = LAMDA + 4.0*DEXP - WAVE4D
CALL INTRST(XI, Y1, THTA4E, AGL5A, X4A, Y4A, X5A, Y5A)
THTA5A = THTA4E - DEXP

CALL EXPAN(PMF4D, DEXP, GAMMA3, PMF4E, EXM4E, EXM4D)
WAVE4D = ASIN(1.0/EXM4E)
AGL5B = THTA4E + WAVE4E
CALL INTRST(X5A, Y5A, THTA5A, AGL5B, X4B, Y4B, X5B, Y5B)
THTA5B = THTA5A - DEXP

CALL EXPAN(PMF4E, DEXP, GAMMA3, PMF5A, EXM5A, EXM4E)
WAVE5A = ASIN(1.0/EXM5A)
AGL5B = THTA5A + WAVE5A
CALL INTRST(X5A, Y5A, THTA5B, AGL5C, X4C, Y4C, X5C, Y5C)
THTA5C = THTA5B - DEXP

CALL EXPAN(PMF5A, DEXP, GAMMA3, PMF5B, EXM5B, EXM5A)
WAVE5B = ASIN(1.0/EXM5B)
AGL5C = THTA5C + WAVE5C
CALL INTRST(X5C, Y5C, THTA5C, AGL5D, X4D, Y4D, X5D, Y5D)
THTA5D = THTA5C - DEXP

CALL EXPAN(PMF5B, DEXP, GAMMA3, PMF5C, EXM5C, EXM5B)
WAVE5C = ASIN(1.0/EXM5C)
AGL5D = THTA5C + WAVE5C
CALL INTRST(X5A, Y5A, THTA5D, AGL5E, X4E, Y4E, X5E, Y5E)
WAVE5E = ASIN(1.0/EXM5E)
THTA5E = THTA5D + DEEXP

***** CALCULATING Pressures IN THE Expansion Fan *****

P4A = PTE / (((1.0 + (G3M1/2.0)*EXM4A*EXM4A)**((GAMMA3/G3M1)))
P4H = PTE / (((1.0 + (G3M1/2.0)*EXM4B*EXM4B)**((GAMMA3/G3M1)))
P4C = PTE / (((1.0 + (G3M1/2.0)*EXM4C*EXM4C)**((GAMMA3/G3M1)))
P4D = PTE / (((1.0 + (G3M1/2.0)*EXM4D*EXM4D)**((GAMMA3/G3M1)))
P4E = PTE / (((1.0 + (G3M1/2.0)*EXM4E*EXM4E)**((GAMMA3/G3M1)))
P5A = PTE / (((1.0 + (G3M1/2.0)*EXM5A*EXM5A)**((GAMMA3/G3M1)))
P5B = PTE / (((1.0 + (G3M1/2.0)*EXM5B*EXM5B)**((GAMMA3/G3M1)))
P5C = PTE / (((1.0 + (G3M1/2.0)*EXM5C*EXM5C)**((GAMMA3/G3M1)))
P5D = PTE / (((1.0 + (G3M1/2.0)*EXM5D*EXM5D)**((GAMMA3/G3M1)))

***** CALCULATING FLOW ALONG THE WING IN THE EXPANSION REGION *****

EXP4AW = ABS(LAMDA4 - THTA4A)
CALL EXPAN(PMF4A, EXP4AW, GAMMA3, PMF4AW, EXM4AW, EXM4A)
EXP4BW = ABS(LAMDA4 - THTA4B)
CALL EXPAN(PMF4B, EXP4BW, GAMMA3, PMF4BW, EXM4BW, EXM4B)
EXP4CW = ABS(LAMDA4 - THTA4C)
CALL EXPAN(PMF4C, EXP4CW, GAMMA3, PMF4CW, EXM4CW, EXM4C)
EXP4DW = ABS(LAMDA4 - THTA4D)
CALL EXPAN(PMF4D, EXP4DW, GAMMA3, PMF4DW, EXM4DW, EXM4D)

EXP4AW = PTE / (((1.0 + (G3M1/2.0)*EXM4AW*EXM4AW)**((GAMMA3/G3M1)))
EXP4BW = PTE / (((1.0 + (G3M1/2.0)*EXM4BW*EXM4BW)**((GAMMA3/G3M1)))
EXP4CW = PTE / (((1.0 + (G3M1/2.0)*EXM4CW*EXM4CW)**((GAMMA3/G3M1)))
EXP4DW = PTE / (((1.0 + (G3M1/2.0)*EXM4DW*EXM4DW)**((GAMMA3/G3M1)))

***** COMPUTING SHOCK WAVE ANGLES AND INTERSECTING POINTS OF REGIONS 4 E AND 5 *****

CALL DELTAK(FSMACH, GAMMA6, THTA4E, THT4ES)
CALL PTHTETA(FSMACH, EXM4ES, PINF, P4E, PR4E1, DR4E1, TR4E1, THT4ES,
1DELTAX, PCOFF, GAMMA6, KTHETA, KNOWN)
CALL INTRST(X1, Y1, THT4ES, AGL5A, X4A, Y4A, X5AS, Y5AS)
CALL UELTAK(FSMACH, GAMMA6, THTA5A, THT5AS)
CALL PTHTETA(FSMACH, EXM5AS, PINF, P5A, PR5A1, DR5A1, TR5A1, THT5AS,
1DELTAX, PCOFF, GAMMA6, KTHETA, KNOWN)
CALL INTRST(X5AS, Y5AS, THT5AS, AGL5R, X4B, Y4B, X5BS, Y5BS)
CALL UELTAK(FSMACH, GAMMA6, THTA5B, THT5RS)
CALL PTHTETA(FSMACH, EXM5BS, PINF, P5R, PR5B1, DR5B1, TR5B1, THT5BS,
1DELTAX, PCOFF, GAMMA6, KTHETA, KNOWN)
CALL INTRST(X5BS, Y5BS, THT5BS, AGL5C, X4C, Y4C, X5CS, Y5CS)
CALL DELTAK(FSMACH, GAMMA6, THTA5C, THT5CS)
CALL PTHTETA(FSMACH, EXM5CS, PINF, P5C, PR5C1, DR5C1, TR5C1, THT5CS,
1DELTAX, PCOFF, GAMMA6, KTHETA, KNOWN)
CALL INTRST(X5CS, Y5CS, THT5CS, AGL5D, X4D, Y4D, X5DS, Y5DS)
CALL DELTAK(FSMACH, GAMMA6, THTA5D, THT5DS)
CALL PTHTETA(FSMACH, EXM5DS, PINF, P5D, PR5D1, DR5D1, TR5D1, THT5DS,
1DELTAX, PCOFF, GAMMA6, KTHETA, KNOWN)
CALL INTRST(X5DS, Y5DS, THT5DS, AGL5E, X4E, Y4E, X5ES, Y5ES)

THT4ES = THTA4ES + 57.296
THT5AS = THTA5AS + 57.296
THT5BS = THTA5BS + 57.296
THT5CS = THTA5CS + 57.296
THT5DS = THTA5DS + 57.296
UELTA=DELTA*57.296
THETA=THETA*57.296
UELTA3=DELTA3*57.296
ITETA3=THETA3*57.296
UELTA6=UELTA6*57.296
ITETA6=THETA6*57.296

*****PRINTING THE SOLUTIONS*****

PRINT 200
PRINT 299, 1C
PRINT 201
PRINT 202
PRINT 203, FSMACH, PINF, TINF, RHOINF, PT1, TI
PRINT 204
PRINT 205, M2, PHAT2, U2, T2
PRINT 206, M3, PHAT3, U3, T3
PRINT 207, M6, PHAT6, U6, T6

PRINT DELTA12, THETA12
PRINT DELTA23, THETA23, GAMMA3
PRINT DELTA6, THETA6

*****END OF THE SOLUTIONS*****
PRINT 219, UELTA6, THETA6, GAMMA6
219 FORMAT(11X,*UELTA16 =*,E12.5,5X,*THETA16 =*,E12.5,5X,*GAMMA6 =*,E12.5)
PRINT 220
220 FORMAT(/,13X,*POINTS OF INTEREST IN THE FLOW FIELD AND ON THE BOUNDARY (IN INCHES)*:///)
PRINT 221
221 FORMAT(16X,*PTET =*,E12.5,5X,*TTE =*,E12.5,/)!
PRINT 222
222 FORMAT(11X,*INITIAL POINTS ON BOW SHOCK-WAVE*)
PRINT 223
223 FORM.T(/,13X,*WING INTERSECTION POINT*)
PRINT 224
224 FORMAT(/,13X,*BOW SHOCK-WING SHOCK INTERSECTION POINT*)
PRINT 225
225 FORMAT(/,13X,*FIRST EXPANSION WAVE WING INTERSECTION POINT*)
PRINT 226
226 FORMAT(/,13X,*LINE 4B WING INTERSECTION POINT*)
PRINT 227
227 FORMAT(/,13X,*LINE 4C WING INTERSECTION POINT*)
PRINT 228
228 FORMAT(/,13X,*LINE 5A SHEAR INTERSECTION POINT*)
PRINT 229
229 FORMAT(/,13X,*LINE 5B SHEAR INTERSECTION POINT*)
PRINT 230
230 FORMAT(/,13X,*LINE 5C SHEAR INTERSECTION POINT*)
PRINT 231
231 FORMAT(/,13X,*LINE 5D SHEAR INTERSECTION POINT*)
PRINT 232
232 FORMAT(/,13X,*FLOW IN REGION 5E FROM EXPANSION*)
PRINT 233
233 FORMAT(/,13X,*MACH NUMBERS IN EXPANSION FAN*)
PRINT 234
234 FORMAT(/,13X,*M4A =*,E12.5,5X,*M4B =*,E12.5,5X,*M4C =*,E12.5)
PRINT 235
235 FORMAT(/,13X,*M5A =*,E12.5,5X,*M5B =*,E12.5,5X,*M5C =*,E12.5)
PRINT 236
236 FORMAT(/,13X,*M5D =*,E12.5)
**PRINT 236**

**FORMAT(13X,*PRESSURES IN THE EXPANSION FAN*)**

**PRINT 237**

**FORMAT(11X,*P4A =**E12.5,5X,*P4B =**E12.5,5X,*P4C =**E12.5)**

**PRINT 238**

**FORMAT(11X,*P5A =**E12.5,5X,*P5B =**E12.5,5X,*P5C =**E12.5)**

**PRINT 239**

**PRINT 240**

**FORMAT(13X,*FLOW ALONG THE WING IN THE EXPANSION REGION*)**

**PRINT 241**

**FORMAT(13X,*MACH NUMBERS*)**

**PRINT 242**

**FORMAT(11X,*P4A =**E12.5,5X,*P4B =**E12.5,5X,*P4C =**E12.5)**

**PRINT 243**

**PRINT 244**

**FORMAT(13X,*PRESSURES*)**

**PRINT 245**

**FORMAT(11X,*P4A =**E12.5,5X,*P4B =**E12.5,5X,*P4C =**E12.5)**

**PRINT 246**

**FORMAT(13X,*FLOW IN THE SHOCKED REGION OF EXPANSION FAN*)**

**PRINT 247**

**FORMAT(13X,*FLOW IN THE WING LEADING EDGE*)**

**C ****CALCULATING HEAT-TRANSFER ALONG WING-LEADING EDGE******

**CALL ERTQDOI (PTE, TTE, P3, GAMMA3, HGA3, X30, X4A, Y4A, Y4B, Y4C, Y4D, T4E, P4AW, P4BW, P4CW, P4DW, P5E, QTSPI, PINF)**

**99 CONTINUE**

**END**

**SUBROUTINE UELTAI (FSMACH, GAMMA, DELTA, THETA)**

**C ****SOLVING FOR THETA GIVEN DELTA******

**MEAL M1S**

**CONV=0.0001**

**M1S=FSMACH*FSMACH**

**GP1=GAMMA+1.0**
IHETAL=DELTA
IHETAR=1.571
5 IHETAL=(IHETAL+IHETAR)/2.0
DELTA=ATAN(1.0/(TAN(IHETAL)*((GP1*M1S)/(2.0*(M1S*((SIN(IHETAL))**2)
1)-1.0)))-1.0))
DIFF=DELTA-DELTA
IF (ABS(DIFF) .LE. CONV) GO TO 2
IF (DIFF) 3 TO 4
3 IHETAL=THETA
GO TO 5
4 IHETAR=THETA
GO TO 5
2 CONTINUE
RETURN
END

SUBROUTINE PTHETA(FSMACH,M2,PINF,P2,PRAT21,DRAT21,TRAT21,THETA,
1 UELTA,PCOEF,GAMMA,KTETA,KNOWN)
******SOLVING FOR FLOW CONDITIONS BEHIND A WEDGE SHOCK GIVEN EITHER
P2 OR THETA*****

HEAL M1S, M2
M1S=FSMACH*FSMACH
UP1=GAMMA+1.0
GM1=GAMMA-1.0
IF (KTETA.EQ.0) GO TO 2
PRAT21=(2.0*GAMMA*M1S*((SIN(THETA))**2)-GM1)/(GP1)
IF (KNOWN.NE.0) GO TO 4
GO TO 3
2 PRAT21=P2/PINF
THETA=ASIN(SURT(((GP1*PRAT21+GM1)/(2.0*GAMMA*M1S))))
3 DELTA=ATAN(1.0/(TAN(IHETAL)*((GP1*M1S)/(2.0*(M1S*((SIN(IHETAL))**2)
1)-1.0)))-1.0))
4 M2=SURT(((M1S*(GP1*PRAT21+GM1))-2.0*(PRAT21**2-1.0))/(PRAT21*(GM1*
1PRAT21+GP1)))
DRAT21=(GP1*PRAT21+GM1)/(GM1*PRAT21+GP1)
TRAT21=(PRAT21*(GM1*PRAT21+GM1))/(GP1*PRAT21+GM1)
PCOEF=(4.0*(M1S*((SIN(IHETAL))**2)-1.0))/(GP1*M1S)
RETURN
END

SUBROUTINE WDOT(PTREF,RGAS,TW,TT,RNOSE,CP0,CP1,CP2,CP3,CP4,UTSPR,
1 PRNO)
******SUBROUTINE TO CALCULATE THE REFERENCE HEAT TRANSFER*****

RHOW=PTREF/(RGAS*TW)
R_HOT=PTREF/(RGAS*TT)
VIST=2.27*E-08*(TT**1.5)/(TT+198.6)
VISW=2.27*E-08*(TW**1.5)/(TW+198.6)
UEUX=(12.0*RGAS*TT)**0.5)/RNOSE
CPW=CP0+CP1*TW+CP2*TW**2+CP3*(TW**3)+CP4*(TW**4)
CPT=CP0+CP1*TT+CP2*TT**2+CP3*(TT**3)+CP4*(TT**4)
UEL=CP0*TT+CP1*TW+CP2*TW**2+CP3*(TW**3)+CP4*(TW**4)
UTSPK=0.6*((R_HOT*VIST)**0.4)*((RHOW*VISW)**0.1)*DELH*(UEUX**0.5)
1/PRNO
RETURN
END

SUBROUTINE INTS!(X1,Y1,A1,A2,X2,Y2,X1,Y1)
******SUBROUTINE TO FIND THE INTERSECTION POINT OF THE SHOCK WAVES
AND THE INTERSECTION POINTS IN THE EXPANSION FAN*****

C
C
**SUBROUTINE EXPAN(PMF1,PMF0,GAMMA,PMFO,EXMACH,EM1)***

\[ x1 = (y2 - y1 + x1 \tan(a1) - x2 \tan(a2)) / (\tan(a1) - \tan(a2)) \]
\[ y1 = y1 + (xt - x1) \tan(a1) \]
RETURN
END

**SUBROUTINE TO CALCULATE MACH NUMBERS IN THE EXPANSION FAN**

\[ GM1 = \text{GAMMA} + 1 \]
\[ DP1 = \text{GAMMA} + 1 \]
\[ PMFO = \text{PMF1} \cdot \text{EXP} \]
\[ A = \text{SQRT}(DP1 / GM1) \]
\[ PCONV = 0.0001 \]
\[ STM = EM1 \]
\[ HM = 1.1 * STM \]

5 EXMACH = (STM + HM) / 2.0
EXMS = EXMACH * EXMACH
PMFOS = A * TAN(SQRT((EXMS + 1.0) * GM1 / DP1)) - TAN(SQRT((EXMS + 1.0)))
PIFF = PMFOS - PMFO
IF (AHS(PU1FF) .LE. PCONV) GO TO 2
IF (PI1FF) 3, 2, 4
3 STM = EXMACH
GO TO 5
4 HM = EXMACH
GO TO 5
2 CONTINUE
RETURN
END


**DIMENSION S(150), OUT(150), PE(150), TE(150), HM(150), STAR(150), VIS2(150), ICONF(150), RNS(150), AF(150), HE(150), KNS(150), TR(2150), PRR(150), VISE(150), HOF(150), RFS(150), DMTR(150), PRAT(150)**

**FORMAT (3E12.5)**

1 FORMAT (3E12.5)
12 FORMAT (3E12.5)

A1 = X1 - X4A
A2 = X1 - X4B
A3 = X1 - X4C
A4 = X1 - X4D
A5 = X1 - X4E
Y1 = Y1 - Y4A
Y2 = Y1 - Y4B
Y3 = Y1 - Y4C
Y4 = Y1 - Y4D
Y5 = Y1 - Y4E
S(1) = 0 * 0
SNEWT = DELS
S4A = SQRT((X1 * X1 + Y1 * Y1)) / 12.0
S4B = SQRT((X2 * X2 + Y2 * Y2)) / 12.0
S4C = SQRT((X3 * X3 + Y3 * Y3)) / 12.0
S4D = SQRT((X4 * X4 + Y4 * Y4)) / 12.0
S4E = SQRT((X5 * X5 + Y5 * Y5)) / 12.0
DELTS = 0.02 * (S4E - S4A)
HEC = SQRT(PR)
D0 6 = 2.0
S(1) = S(1-1) + SNELT
IF (S(I) <= 4A) GO TO 41
IF (LL <= 2) GO TO 47
IF (LL <= 1) GO TO 47
DEL = DEL + ST
S(I) = 4A - N; C * DEL
PF(I) = PF
IF (LL <= 2) GO TO 44
DEL = DEL
LL = 1
GO TO 44
47 CONTINUE
IF (S(I) <= 4A) GO TO 42
IF (S(I) <= 4C) GO TO 43
IF (S(I) <= 4D) GO TO 45
IF (S(I) <= 4F) GO TO 46
PF(I) = PF
IF (LL <= 2) GO TO 44
DEL = DEL
LL = 2
GO TO 44
41 PF(I) = PF
GO TO 44
42 PF(I) = PF
GO TO 44
43 PF(I) = PF
GO TO 44
45 PF(I) = PF
GO TO 44
46 PF(I) = PF
GO TO 44
44 CONTINUE
IF (I) = TF *((PF(I) / PF)**((5 - 1) / 5))
PF(I) = TF / PF(I)**((5 - 1) / 5)
EM(I) = SQRT(PF(I) - 1) * 2 / (G - 1)
LSTAR(I) = 0.5 * TE(I) + Tw(I) + 7/2 * HEC((FM(I)**2.)*1E(I)*(G - 1) / 2)
VISF(I) = (VIS20*VISI*TE(I) + VISS2*TE(I)**1E(I) + VISS3*TF(I)**3) +
VIS4***(TF(I)**4)
VIS2(I) = (VIS20 + VISI*TSTAR(I) + VISS2*TSTAR(I) * TSTAR(I) +
VIS3*ISTAR(I) ** ISTAR(I) + VIS4*TSTAR(I) * TSTAR(I) * TSTAR(I) +
ISTAR(I))
ICON(I) = + 2432.1763 * VIS2(I) / PH
HROS(I) = 0 (I) / (H**TSTAR(I))
HHEI(I) = PE(I) / (H**TSTAR(I))
AF(I) = SQRT(G**N**T(I))
UF(I) = EM(I) * AF(I)
HKNOS(I) = HROS(I) / UE(I) / VIS2(I)
HES(I) = HHEI(I) / UE(I) / VIS2(I)
TF(I) = TE(I) * (1.0 + HEC((G - 1) / 2) + (FM(I)**2) / 2)
WDT(I) = 3.32 * SQRT(HKNO(I)) * (PR**3.3) * ICON(I) * (TR(I) - TW(I)) / SORT(I)
WDT(I) = WDT(I) / QTSPR
PRAT(I) = PE(I) / F14F
6 CONTINUE
PRINT 60
60 FORMAT (1H1)
PRINT 77
77 FORMAT (24X,*FLOW PARAMETERS ALONG WING-LEADING EDGE*//)
PRINT 7
7 FORMAT (14X,*S(I) ,*7X, *PE(I) ,*7X,*RES(T) ,*9X,*WDT(I) ,*7X,*WGT(I) ,*15X,*PRAT(I) ,*7X)
UN 1 J = 2,
PRINT 8S(I) * PE(I) * RES(I) * WDT(I) * CONT(I) * PHAI(I)
10 CONTINUE
H. T. H. N.
End
DESCRIPTION OF OUTPUT

The output for the perfect-gas code includes the flow conditions from each region of the flow-field model, the geometry of the shock waves and the expansion waves, and the heat-transfer distribution along the second wedge, i.e., the wing leading-edge. The units for a particular parameter in any region will be the same as the free-stream parameter, unless otherwise noted. The output for the free-stream flow includes:

- FSMACH - free-stream Mach number
- P1 - free-stream static pressure (lbf/ft^2)
- T1 - free-stream temperature (°R)
- RHOO1 - free-stream density (slugs/ft^3)
- PT1 - free-stream stagnation pressure (lbf/ft^2)
- TT - stagnation temperature of entire flow model (°R)
- RGAS - gas constant (ft^2/sec^2-°R)
- GAMMA - γ for the free-stream and region 2
- QTSPR - reference heating rate (Btu/ft^2-sec)

The following flow conditions are output for each region "I", where I = 2, 3, or 6:

- MI - the Mach number of the region
- PI/P1 - the static pressure ratio
- TI/T1 - the temperature ratio
- RHOI/RHOO1 - the density ratio
- PCOEFI - the pressure coefficient
- PI - the static pressure
- TI - the temperature
- RHOOI - the density
- PTI - the stagnation pressure
DELTAJI - the change in flow-direction between two consecutive regions, in degrees.

THETAJI - the shock wave angle between two consecutive regions, in degrees

GAMMAI - the γ of the region

The output for the expansion fan region includes the following stagnation condition and intersection points:

PTE - stagnation pressure at the edge of the boundary layer

TTE - stagnation temperature at the edge of the boundary layer

Intersection points include:

INITIAL POINTS ON BOW SHOCK WAVE, i.e., the origin of the coordinate system, or the nose.

WING INTERSECTION POINT, i.e., intersection of the two wedges

BOW-SHOCK:WING-SHOCK INTERSECTION POINT, i.e., the intersection of the shock of the first wedge with the shock of the second wedge

Next Five Points - the intersections of the centered expansion fan with the wing leading edge

Last Five Points - the intersection of the reflected waves with the inboard shear layer

Output for the flow in region 5E include:

THETA - the flow direction, in degrees

M5E - the Mach number

P5E - the static pressure

The Mach numbers and pressures of each of the other nine subregions of regions 4 and 5 are the output under the next two headings.

Output listed for the interaction region between the left running and right
running expansion fan waves includes:

EXM4IW - the Mach number

P4IW - the static pressure

where "I" = A, B, C, and D.

Output for the subregions between the inboard shear-layer and the "curved" shock wave include:

THETA - shock wave angle

X - x-coordinate point of the intersection of the shock wave and the left-running expansion wave

Y - y-coordinate point of the intersection of the shock wave and the left-running expansion wave

PR - pressure ratio with respect to region 1

DR - density ratio with respect to region 1

TR - temperature ratio with respect to region 1

M - Mach number

The output for the flow parameters and the heat-transfer distribution along with wing leading edge include:

S(I) - distance along wing leading edge

PE(I) - pressure at the edge of the boundary layer

RES(I) - Reynolds number

QDOT(I) - heat transfer rate

QDOTR(I) - heat transfer rate ratio with reference heat transfer rate

PRAT(I) - pressure ratio with respect to region 1

where the (I) index refers to the station for which the calculations are made.
CASE = 1

FREE-STREAM FLOW CONDITIONS

<table>
<thead>
<tr>
<th>$F_{SWACH}$</th>
<th>$P_1$</th>
<th>$T_1$</th>
<th>$\rho_1$ (SLUGS/CUFT)</th>
<th>$P_T$</th>
<th>$T_T$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.31000E+01</td>
<td>9.29874E-01</td>
<td>4.90569E+02</td>
<td>1.10460E-06</td>
<td>2.43547E+05</td>
<td>1.73279E+04</td>
</tr>
</tbody>
</table>

GAS CONSTANTS

$R_{GAS} = 1.71600E+03$  \[\text{GAMMA} = 1.40000E+00\]  \[\text{QTSPR} = 5.89275E+02\]

FLOW CONDITIONS IN REGION 2

<table>
<thead>
<tr>
<th>$M_2$</th>
<th>$P_2/P_1$</th>
<th>$T_2/T_1$</th>
<th>$\rho_2/\rho_1$</th>
<th>$P_{COEF2}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.02440E+01</td>
<td>4.03662E+00</td>
<td>1.60644E+00</td>
<td>2.51277E+00</td>
<td>2.52784E-02</td>
</tr>
</tbody>
</table>

$P_2 = 3.75355E+00$  \[T_2 = 7.98071E+02\]  \[\rho_2 = 2.77562E+06\]  \[P_T = 1.87100E+05\]  \[\text{DELTA12} = 5.00000E+00\]  \[\text{THETA12} = 8.33116E+00\]

FLOW CONDITIONS IN REGION 3

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$P_3 = 1.27959E+02$  \[T_3 = 5.24002E+03\]  \[\rho_3 = 1.42305E+06\]  \[P_T = 5.08890E+02\]  \[\text{DELTA23} = 2.50000E+01\]  \[\text{THETA23} = 3.19361E+01\]  \[\text{GAMMA3} = 1.40000E+00\]

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$P_6 = 7.03174E+01$  \[T_6 = 6.65973E+03\]  \[\rho_6 = 6.15396E+06\]  \[P_T = 2.05891E+02\]
DELTA16 = 3.00000E+01  THETA16 = 3.79703E+01  GAMMA6 = 1.40000E+00

POINTS OF INTEREST IN THE FLOW FIELD AND ON THE BODY (IN INCHES)

PTE = 8.41422E+03  TTE = 1.73279E+04

INITIAL POINTS ON BOW SHOCKWAVE
x = 0.  
WING INTERSECTION POINT
x = 9.00000E+00  
BOW SHOCK-WING SHOCK INTERSECTION POINT
x = 9.83909E+00  y = 1.48083E+00
FIRST EXPANSION WAVE WING INTERSECTION POINT
x = 1.02588E+01  y = 1.53677E+00
LINE 4B WING INTERSECTION POINT
x = 1.02787E+01  y = 1.54827E+00
LINE 4C WING INTERSECTION POINT
x = 1.03007E+01  y = 1.56097E+00
LINE 4D WING INTERSECTION POINT
x = 1.03282E+01  y = 1.57508E+00
LINE 4E WING INTERSECTION POINT
x = 1.03526E+01  y = 1.59093E+00
LINE 5A SHEAR INTERSECTION POINT
x = 1.06104E+01  y = 1.97692E+00
LINE 5B SHEAR INTERSECTION POINT
x = 1.06524E+01  y = 1.97892E+00
LINE 5C SHEAR INTERSECTION POINT
x = 1.07626E+01  y = 1.98792E+00
LINE 5D SHEAR INTERSECTION POINT
x = 1.08216E+01  y = 1.99792E+00
FLOW IN REGION SE FROM EXPANSION
THETA = 3.00000E+01  M5E = 3.82344E+00  PM5E = 7.03174E+01

MACH NUMBERS IN EXPANSION FAN
M4A = 3.43600E+00  M4B = 3.47627E+00  M4C = 3.51700E+00
M4D = 3.43899E+00  M4E = 3.60131E+00  M5C = 3.73228E+00
M5A = 3.64211E+00  M5B = 3.68763E+00
M5D = 3.77748E+00

PRESSURES IN THE EXPANSION FAN
P4A = 1.20858E+02  P4B = 1.14106E+02  P4C = 1.07689E+02
P4D = 1.01494E+02  P4E = 9.56164E+01
P5A = 9.00470E+01  P5B = 8.47671E+01  P5C = 7.96868E+01
P5D = 7.48814E+01
FLOW ALONG THE WING IN THE EXPANSION REGION

MACH NUMBERS

\[ M_{4AW} = 3.47627 \times 10^0 \quad M_{4BW} = 3.55910 \times 10^0 \quad M_{4CW} = 3.64408 \times 10^0 \quad M_{4DW} = 3.73268 \times 10^0 \]

PRESSURES

\[ P_{4AW} = 1.14106 \times 10^2 \quad P_{4BW} = 1.01467 \times 10^2 \quad P_{4CW} = 9.06333 \times 10^1 \quad P_{4DW} = 7.96434 \times 10^1 \]

FLOW IN THE SHOCKED REGION OF EXPANSION FAN

REGION 4ES

\[ \Theta = 4.24924 \times 10^1 \quad x = 1.14125 \times 10^1 \quad y = 2.88224 \times 10^0 \quad PR = 9.11872 \times 10^1 \quad DR = 5.63987 \times 10^0 \quad TR = 1.61683 \times 10^1 \quad M = 2.43377 \times 10^0 \]

REGION 5AS

\[ \Theta = 4.15671 \times 10^1 \quad x = 1.16193 \times 10^1 \quad y = 3.05599 \times 10^0 \quad PR = 8.79715 \times 10^1 \quad DR = 5.62755 \times 10^0 \quad TR = 1.56323 \times 10^1 \quad M = 2.50954 \times 10^0 \]

REGION 5BS

\[ \Theta = 4.06615 \times 10^1 \quad x = 1.18523 \times 10^1 \quad y = 3.26571 \times 10^0 \quad PR = 8.48364 \times 10^1 \quad DR = 5.61469 \times 10^0 \quad TR = 1.51097 \times 10^1 \quad M = 2.58621 \times 10^0 \]

REGION 5CS

\[ \Theta = 3.97480 \times 10^1 \quad x = 1.21149 \times 10^1 \quad y = 3.48414 \times 10^0 \quad PR = 8.16890 \times 10^1 \quad DR = 5.60086 \times 10^0 \quad TR = 1.45851 \times 10^1 \quad M = 2.66626 \times 10^0 \]

REGION 5DS

\[ \Theta = 3.88555 \times 10^1 \quad x = 1.24107 \times 10^1 \quad y = 3.72243 \times 10^0 \quad PR = 7.86319 \times 10^1 \quad DR = 5.58644 \times 10^0 \quad TR = 1.40755 \times 10^1 \quad M = 2.74724 \times 10^0 \]
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APPENDIX B. - GENERAL DESCRIPTION OF REAL-GAS CODE

As has been discussed previously, philosophically the real-gas code is approached in the same way as the perfect-gas code. As with the perfect-gas code, the free-stream conditions and the model geometry serve as input. The real-gas code uses a Mollier gas table to find the flow-field conditions in place of the perfect gas relations of the perfect-gas code.

Calculation of Flow-Field Conditions. The flow conditions of regions 2, 3, and 6 are calculated in subroutine RELGAS. The procedure of this subroutine is to assume an initial value for the shock wave angle $\theta$, and solve for the density as follows.

\[
\rho_2 = \frac{\rho_1 \tan \theta}{\tan(\theta - \delta)} \quad (18)
\]

The velocity behind the shock wave can then be determined by using the conservation of mass equation:

\[
U_{2n} = \frac{\rho_1 U_{1n}}{\rho_2} \quad (19)
\]

The pressure is then calculated from the conservation of normal momentum equation:

\[
P_2 = P_1 + \rho_1 U_{1n}^2 - \rho_2 U_{2n}^2 \quad (20)
\]

Then the enthalpy behind the shock wave is calculated using the conservation of energy equation:

\[
h_2 = h_1 + 0.5U_{1n}^2 - 0.5U_{2n}^2 \quad (21)
\]

Once the static pressure and the static enthalpy are known, the remaining flow properties behind the shock wave can be found by using the subroutine MOLIER. MOLIER, which incorporates the gas table, needs only two properties as
input to find the remaining flow field properties. The density found from MØLIER is compared to the density behind the shock wave calculated from equation (18). If the comparison is not acceptable (to within a prespecified tolerance), a half-interval iteration method is used to redetermine the shock wave angle. The procedure to calculate the flow condition behind the shock wave is then repeated until successive values of the densities converge.

**Calculation of Stagnation Conditions.** The stagnation conditions are calculated after the flow conditions in regions 2, 3, and 6 are calculated, by first calculating the stagnation enthalpy by the following equation:

\[ H = h_1 + 0.5 U_1^2 \]  

(22)

Using the stagnation enthalpy and the entropy of regions 1, 2, 3, and 6, the stagnation conditions for an isentropic deceleration of the local flow are found from MØLIER.

**Calculation of Expansion Fan Flow.** To calculate the expansion fan flow, the flow is assumed to expand isentropically from region 3 to region 5E. Thus, the entropy of region 3 defines the entropy of the expansion fan. Therefore, the two properties serving as input for the MØLIER subroutine, are the local static enthalpy and the entropy.

As before, the streamlines for subregion 5E through region 6 are parallel and straight for the assumed flow model. Therefore,

\[ \frac{dP}{dn} = 0 \]  

(23)

and the static pressure of subregion 5E is equal to the static pressure of region 6. The enthalpy of region 5E is found by using the static pressure of region 6 and the entropy of region 3. To calculate the enthalpy of each of the other nine regions in the expansion fan, the difference between the
enthalpies of region 3 and 5E is divided by ten to get ten equal increments. The enthalpy of region 4A is, therefore, the enthalpy of region 3 plus the above increment. The enthalpies of the remaining subregions are gotten by appropriately incrementing the previous enthalpy.

As a check to the above procedure the Prandtl-Meyer angle for the sub-region of interest was calculated using the equation:

\[ v - v_3 = \frac{1}{2} \left( \frac{2(H-h)}{a^2} - 1 \right)^{1/2} \frac{d(H-h)}{H-h} \]  

When crossing a right-running wave, the change in the Prandtl-Meyer angle is equal to the change in the flow direction, i.e.,

\[ dv = d\theta \]  

When crossing the reflected left-running waves,

\[ dv = -d\theta \]  

Thus, since the flow in region 3 is parallel to the wall, the sum of the change in \( v \) through region 4 should equal to the sum of the change in \( v \) through region 5. For all cases computed thus far, the difference between the \( v \)-sums has been small. Thus, the net change in flow angle has been essentially zero, as it should. The intersection points of the shock waves and expansion fan waves are calculated the same way as in the perfect gas code using the subroutine INTRST.

The flow properties at the wall account for the waves of the centered expansion fan, i.e., region 4, and of the reflected left-running waves, i.e., region 5. As noted before,

\[ v_{4AW} = v_4A - (\theta_{4AW} - \theta_{4A}) \]  

so that

\[ v_{4AW} = v_{4B} \]
Similarly,

\[ v_{4BW} = v_{4D} \quad (27b) \]
\[ v_{4CW} = v_{5A} \quad (27c) \]

and

\[ v_{4DW} = v_{5C} \quad (27d) \]

Once the Prandtl-Meyer functions are known, the remaining properties are readily calculated, since flow in this region is isentropic.

Wing Leading-Edge Heat-Transfer Rate Calculations. After the inviscid flow-field conditions have all been calculated, the heat-transfer rate along the "wing leading-edge" is calculated. The procedure for calculating the heat-transfer is to first use the numerical routine as described in ref. 15 to set up the initial boundary-layer profile.

This initial profile serves as an input into the NONSIMBL code (ref. 11). Several modifications were made to this numerical routine for a laminar boundary-layer. One modification was to eliminate the need for user "experience" in establishing the initial, input guesses for the wall values of the shear function, \( f''(0) \), and of the heat-transfer function, \( g'(0) \). Instead, since the temperature of the edge of the boundary layer and at the wall in region 3 were known from the previous subroutines, the initial value of \( g'(0) \) was assumed to be

\[ g'(0) = \left[ 1 - \frac{T_w}{T_{t3}} \right] 0.6 \]

The initial guess for the shear function was assumed to be: \( f''(0) = 0.47 \), which seemed to provide reasonable results over the entire velocity range (in the absence of gas injection of the wall).

Another modification was to incorporate into the routine a procedure for calculating alpha (i.e., the coordinate transformation parameter).
The above procedure is designated subroutine PIGYBAK and transforms the distribution of the dimensionless velocity function $F$ and the temperature ratio $\theta$ (THETA) into the new coordinate using alpha ($\alpha$). The transformation is

$$n = 1 - e^{-\alpha \eta}$$

where $n$ is the newly transformed $y$-coordinate (ref. 11) and $\eta$ is the transformed $y$-coordinate using the standard Lees-Dorodnitsyn transformation. To calculate the appropriate value of $\alpha$, the value of $\eta$ where $u = 0.99u_e$ in the similar solution for the first station was identified. Then, the value of $\alpha$ was calculated to be the value which makes the produce $\alpha \eta$ equal to three at this point, i.e., $n = 0.95$.

This viscous-layer profile for a similar boundary-layer which is subject to the flow conditions at the initial point in region 3 is used as an initial condition for the subroutine EJ0YCE, which solves the nonsimilar boundary layer. This subroutine is actually the numerical code described in ref. 11 with modifications. Modifications were made to this code so that one could obtain real gas values of density, thermal conductivity, viscosity, and specific heat of air. The MOLIER subroutine was used to calculate the density in the boundary layer. The tabulated values of Hansen (ref. 9) for thermal conductivity, viscosity, and specific heat were input as a function of temperature. Then, the subroutine SPLNTRP was used to calculate the transport properties at the desired temperatures. SPLNTRP is a curve fitting subroutine. The final output of EJ0YCE gives boundary layer profiles, heat-transfer rates, and other data for five points in each of the six regions and subregions on the "wing leading-edge".

**Calculation of the Shock-Wave Angles and the Intersection Points of the Shock Wave with the Expansion Fan.** As was the case for the perfect-gas code the flow direction in subregions 4E, 5A, 5B, 5C, and 5D of the expansion
fan are used as boundary conditions for computing the shock waves in this region. The subroutine RELGAS is used to generate the shock wave angle and the flow conditions in the subregions between the shock wave and the shear layer, i.e., subregion 4ES, 5AS, 5BS, 5CS, and 5DS. The intersection of the shock wave and the left-running expansion wave is computed using the subroutine INTRST using the same procedure as discussed in the perfect-gas section.
Input Cards

Card # 1  NC - number of cases

Card # 2  XO - x-location of "nose" (ft)
          YO - y-location of "nose" (ft)
          X30 - x-location of wedges intersection point (ft)
          Y30 - y-location of wedges intersection point (ft)

Card # 3  U1 - free-stream velocity (ft/sec)
          P1 - free-stream pressure (lbf/ft²)
          H1 - free-stream enthalpy (ft²/sec²)
          DELTA - angle of the first wedge (°)
          LAMDAS - complement of sweep angle (°)
          NØPT - option in MØLIER for the two input properties enthalpy and pressure (°)

Card # 4  TWALL - wall temperature (°R)
          RADIUS - radius of reference sphere (ft)
          PR - Prandtl number

Card # 5  E - convergence criterion on boundary condition (typically .0005)
          DELT - step size in y-direction (typically .05)
          EPS - convergence criterion when variable step size is used.

Card # 6  KEY - equals 1 if using variable step size (equals 0 for the fixed size routine)

Card # 7  IN - number of y-points
          ALF - coordinate transformation parameter

Card # 8  PIGYBK - if equals 1.0 edge properties are not read in as input

Card # 9  MM - number of x-points for calculations
          N - number of y-points in boundary layer
          KK - equals 0 for two-dimensional flow, equals 1 for axi-symmetrical flow
          NØPRINT - if equals 0 output will be printed.
Card # 10  WMI - molecular weight of injectant
          WMS - molecular weight of stream (WMI = WMS for this routine)
Card # 11-13  TEMP - temperature (°R), fifteen temperatures on three cards
Card # 14-16  VISC - viscosity (\( \frac{\text{lbf} - \text{sec}}{\text{ft}^2} \)) fifteen viscosities corresponding to above temperatures.
Card # 17-19  TC - thermal conductivity (\( \frac{\text{BTU}}{\text{ft-sec-°R}} \)), fifteen thermal conductivities corresponding to above temperatures
Card # 20-22  CP - specific heat (\( \frac{\text{BTU}}{\text{slug-°R}^2} \)), fifteen specific heats corresponding to above temperatures

Card # 23 equals Card # 3 if NC > 1

Card # 24 equals Card # 4 if NC > 1
PROGRAM REALGAS (INPUT, OUTPUT)

DIMENSION H4(50), P4(50), T4(50), M4(50), G4(50), A4SU(50), Y(50),
  1*NU(50), Z4(50)

DIMENSION WAVE4(50), AGL4(50), X4(50), Y4(50), THETA4(50), U4(50),
  1*NC4(-1), SIGNU(50)

DIMENSION HTS4(11), A5(11), YS4(11), US4(11), PS4(11), TS4(11),
  1*HS4(11), *S4(11), ZS4(11), SS4(11)

COMMON/KONSI/TWAI, RADIUS, PR
COMMON/WING/PT3, TT3, U3, T3, X3, Y3, P3, T4, T4, T5, T6
COMMON/CASE/NCAS
COMMON/SHOCK/NFLUB, ISHUCK, JSHUCK
REAL LAMDA5
HEAD 222, NC

222 FORMAT (15)
HEAD 101, X0, Y0, X30, Y30
101 FORMAT (*E12.5)
U0 99 II = 1, NC
HEAD 200, U1, P1, H1, DELTA1, LAMDA1, NOPT1
200 FORMAT (*E12.5, I5)
HEAD 102, TWALL, RADIUS, PR
102 FORMAT (*E12.5)

C  H1 = H1/(32.0176*7788)
C  CALL MOLIER(H1, P1, NOPT1, T1, Z1, S1, RH01, GAMMA1)
  H1 = H1*32.176*77880
C

C *****CALCULATING FLOW CONDITIONS IN REGION 2*****
C  CALL RELGAS(U1, P1, RH01, H1, DELTA1, NOPT1, U2, P2, RH02, H2, T2, Z2, GAMMA2, 1
C
C *****CALCULATING FLOW CONDITIONS IN REGION 3*****
C  DELTA3 = LAMDA3 - DELTA1
C  CALL RELGAS(U2, P2, RH02, H2, DELTA3, NOPT2, U3, P3, RH03, H3, T3, Z3, GAMMA3, 1
C
C *****CALCULATING FLOW CONDITIONS IN REGION 6*****
C  DELTA6 = LAMDA6
C  NFLUB = 1
C  CALL RELGAS(U1, P1, RH01, H1, DELTA6, NOPT6, U6, P6, RH06, H6, T6, Z6, GAMMA6, 1
C
C *****COMPUTING STAGNATION CONDITIONS*****
C
C  HSTAG = H1 + 0.5*U1*U1
C  HSTAG = HSTAG/(32.176*7788)
C  CALL MOLIER(HSTAG, PT1, TT1, ZT1, S1, RT1, GT1)
C  CALL MOLIER(HSTAG, PT2, TT2, ZT2, S2, RT2, GT2)
C  CALL MOLIER(HSTAG, PT3, TT3, ZT3, S3, RT3, GT3)
C  CALL MOLIER(HSTAG, PT6, TT6, ZT6, S6, RT6, GT6)
C
C  PRAT21 = P2/P1
C  PRAT31 = P3/P1
C  PRAT61 = P6/P1
C  TRAT21 = T2/T1
C  TRAT31 = T3/T1
C  TRAT61 = T6/T1
C  DRAT21 = RH02/RH01
C  DRAT31 = RH03/RH01
C  DRAT61 = RH06/RH01
C
C  H1 = H1/(32.176*7788)
****COMPUTING EXPANSION FAN FLOW****

IF(ISSHOCK.EQ.1) GO TO 703
CALL MOLIER(H5E,P6,1,T5E,Z5E,S3,RH05E,G5E)
A3SQ=P3*GAMMA3/RH03
A5ESQ=P6*GAMMA5/RH05E
A3SQ=A3SQ/(32.176*778.0)
A5ESQ=A5ESQ/(32.176*778.0)
ETA1=HSTAG-M3
ETAN=HSTAG-M5
N=11
Y(1)=SQRT(2.0*ETA1/A3SQ-1.0)/(2.0*ETA1)
Y(N)=SQRT(2.0*ETAN/A5ESQ-1.0)/(2.0*ETAN)
M=N-1
MSTEP=(ETAN-ETA1)/M
ETA=ETA1

U0 98 I=2*M
ETA=ETA+HSTAG
M4(I)=HSTAG-ETA
CALL MOLIER(H4(I),P4(I),3,T4(I),Z4(I),S3,R4(I),G4(I))
A4SQ(I)=P4(I)*G4(I)/R4(I)
A4SQ(I)=A4SQ(I)/(32.176*778.0)
Y(I)=SQRT(2.0*ETA/A4SO(I)-1.0)/(2.0*ETA)
98 CONTINUE

SIGDNU(1)=0.0
SIGDNU(2)=SIGDNU(1)+(Y(1)+Y(2))*HSTEP/2.0
U0 97 J=3*N+2
SIGDNU(J)=SIGDNU(J-2)+(Y(J-2)+4.0*Y(J-1)+Y(J))*HSTEP/3.0
IF(J+1.GT.M) GO TO 97
SIGDNU(J+1)=SIGDNU(J-1)+(Y(J-1)+4.0*Y(J)+Y(J+1))*HSTEP/3.0
97 CONTINUE

U0 90 JP=2,N
UNU(JP)=SIGDNU(JP)-SIGDNU(JP-1)
90 CONTINUE

DN4=DN4(2)+DN4(3)+DN4(4)+DN4(5)+DN4(6)
DN5=DN5(7)+DN5(8)+DN5(9)+DN5(10)+DN5(11)
DNUDIF=DN4-DN5

****FINDING VELOCITIES AND MACH NUMBERS IN THE EXPANSION FAN****

HSTAG=HSTAG*32.176*778.0
U0 96 K=2*M
H4(K)=H4(K)*32.176*778.0
U4(K)=SQRT(2.0*(HSTAG-H4(K))
A4SQ(K)=A4SW(K)*32.176*778.0
EXM4(K)=U4(K)/SQRT(A4SQ(K))
H4(K)=H4(K)/(32.176*778.0)
96 CONTINUE

H5E=H5E*32.176*778.0
A5ESQ=A5ESQ*32.176*778.0
U4(N)=SQRT(2.0*(HSTAG-H5E))
EXM4(N)=U4(N)/SQRT(A5ESQ)
H5E=H5E/(32.176*778.0)

****FINDING INTERSECTION POINTS OF SHOCK WAVES AND EXPANSION FAN****
**C**

IHETA = THETA / 57.296
IHETA3 = THETA3 / 57.296
LAM = LAM / 57.296
DELTA = DELTA / 57.296
DELTA0 = DELTA0 / 57.296

AGL = IHETA3 * DELTA
CALL INTPST (X0, Y0, THETA, AGL3, A3, Y30, XI, Y1)

A3Q = A3Q / 3.176 * 778.0
SAVE = AS1 (1 + / 13 / SQRT(A3Q))
AGL4 = LAM / WAVE3
CALL INTPST (X1, Y1, AGL4, LAM, X30, Y30, A4, Y4)

C

EXM4 (1) = U3 / SQRT (A3Q)
A4 (2) = X4
Y4 = Y4
AGL4 (2) = AGL4
LL = 1
NL = (NL + 1) / 2
00 95 L = 2 * M
WAVE4 (L) = AS1N (1 / EXM4 (L))
IF (NN = L) Y4, Y3, Y2, Y1

92 AGL4 (L+1) = LAM / WAVE4 (L)
CALL INTPST (X1, Y1, AGL4 (L+1), LAM, X30, Y30, X4 (L+1), Y4 (L+1))
00 TO 95

94 LL = LL + 1
THETA4 (L) = THETA4 (L-1) + WAVE4 (L)
AGL4 (L+1) = THETA4 (L) + WAVE4 (L)
CALL INTPST (X1, Y1, X4 (L), Y4 (L), AGL4 (L+1), X4 (L), Y4 (L), A4 (L+1),

Y4 (L+1))
00 TO 95

93 THETA4 (L) = LAM / WAVE4 (L)
LL = LL + 1
AGL4 (L+1) = THETA4 (L) + WAVE4 (L)
CALL INTPST (X1, Y1, THETA4 (L), AGL4 (L+1), X4 (L), Y4 (L), A4 (L+1), Y4 (L+1))
}

95 CONTINUE

C

C***Calculating Flow Conditions in Shocked Regions 4E and 5***

C

L = 1
FLU = 2
H1 = H1 / 32.176 * 778.0
00 77 I = 1 * M
THETA4 (I) = THETA4 (I) + WAVE4 (I)
CALL RELAS3 (I), PSI, H1, THETA4 (I), NOP1, US4 (I), PS4 (I), RS4 (I),

TS4 (I), SJ (I), ZS6 (I), GS4 (I), THS4 (I), SS4 (I))
THS4 (I) = THS4 (I) / 57.296
L = L + 1
IF (I.GT.6) 00 TO 75
CALL INTPST (X1, Y1, THS4 (I), AGL4 (I+1), X4 (L), Y4 (L), XS4 (I+1), YS4 (I+1))

00 TO 76

75 CALL INTPST (X4 (I), Y4 (I), THS4 (I), AGL4 (I+1), X4 (L), Y4 (L), XS4 (I+1),

YS4 (I+1))
76 THS4 (I) = THS4 (I) + WAVE4 (I)
77 CONTINUE

H1 = H1 / 32.176 * 778.0
THETA3=THETA3*57.296
LAMDA3=LAMDA3*57.296
VELTA=VELTA*57.296
VELTA=DELTA*57.296
THETA=THETA*57.296

703 CONTINUE
PRINT 100
100 FORMAT(1H1)
PRINT 300, 11
300 FORMAT(33X,"CASE =",D12.7)
PRINT 201
201 FORMAT(16X,"FREE-STREAM FLOW CONDITIONS\\\\")
PRINT 202
PRINT 217
PRINT 203, U1, P1, T1, KIN1, PT1, TT1
PRINT 204, GAMMA, H1, Z1
204 FORMAT(11X, D12.7, 5X, "H1 =", D12.7, 5X, "Z1 =", D12.7)
PRINT 206
206 FORMAT(16X,"FLOW CONDITIONS IN REGION 2\\\\")
PRINT 207
PRINT 208, U2, PT2, TT2
208 FORMAT(10X, D12.7, 3X, D12.7, 4X, D12.7, 9X, D12.7)
PRINT 212, GAMMA, H2, Z2
PRINT 214, T2 = D12.7, Z2 = D12.7)
PRINT 215, T2 = D12.7, Z2 = D12.7)
PRINT 209, UELTA, THETA
PRINT 216
216 FORMAT(16X,"FLOW CONDITIONS IN REGION 3\\\\")
PRINT 213
PRINT 214, U3, PT3, THETAT3, GAMMA3
PRINT 215, T3 = D12.7, Z3 = D12.7)
PRINT 218, GAMMA3, THETA3 = D12.7)
PRINT 221, T3 = D12.7, Z3 = D12.7)
PRINT 220, T3 = D12.7, Z3 = D12.7)
PRINT 211
211 FORMAT(16X,"FLOW CONDITIONS IN REGION 4\\\\")
IF(ISHOCK.EQ.1) GO TO 701
PRINT 214
PRINT 216, U6, PT6, THETAT6, GAMMA6
PRINT 217, T6 = D12.7, Z6 = D12.7)
PRINT 219, GAMMA6, THETA6 = D12.7)
PRINT 218, GAMMA6, THETA6 = D12.7)
PRINT 212, GAMMA, H2, Z2
PRINT 211
211 FORMAT(16X,"FLOW CONDITIONS IN REGION 6\\\\")
PRINT 301
301 FORMAT(10X,"FLOW CONDITIONS IN THE EXPANSION FAN\\\\")
Y(1) = Y(2)
Y(2) = Y(3)
Y(3) = (1 / A1) * ((Y(1) + x2) * Y(3))
Y(4) = Y(5)
Y(5) = (1 / (A3 * x1)) * ((Y(1) + A3 * x2) * Y(5) + ((X1 * Y(3) ** 2) - Y(1) * Y(2) * Y(3) ** 2) * A4)
RETURN
SUBROUTINE NUMIN(N, DELT, UP, TE, E0, EPS, DEP, EPS, DEP, EPS, EPS, EPS, KEY, DERIV, 10, N)

C------GENERAL NUMERICAL INTEGRATION WITH SINGLE-STEP ERROR ANALYSIS
DOUBLE PRECISION UPT, SIX, TWO, UPSAVE
DIMENSION A1(4), B(5), U(N), P(N), E(N), E0(N)
DIMENSION YPR(N, A), UEP(N), UPSAVE(N)

C------
EXTERNAL UEP

C-------- DATA
UPT TO (10*10+7) INUX
10 SIX = 24.000
ZERO = 0.000
A(1) = -4.000 / SIX
A(2) = 37.000 / SIX
A(3) = -57.000 / SIX
A(4) = 55.000 / SIX
B(1) = 1.000 / SIX
B(2) = -5.000 / SIX
B(3) = 1.000 / SIX
B(4) = -A(1)
MATU = 19.000 / 270.0
SIX = 6.000
IWO = 2.000
M1 = 4
M2 = 1
M3 = 2
M4 = 3
RETURN

C-------- SET UP FOR INTEGRATION

C-------- ENTRY RESET
15 ASSIGN 2000 TO IPS5
KOP = 0
KOUNT = N
DELBY6 = DELT / SIX
DELBY2 = DELT / TWO
U0 16 J=1*N
UPSAVE(J) = UEP(J)
16 U(J) = UEP(J)
CALL UEP(J, U0, P(J), 6, N)
17 CONTINUE
U0 18 J=1*N
YPR(J, M1) = P(J)
18 EJR(J) = ZERO
19 CONTINUE
RETURN

C-------- ENTRY-POINT FOR ONE NUMERICAL INTEGRATION STEP
20 V = T
I = V + DELT
KOP = KOP + 1
U0 TO IPS5
C-------- RUNGE-KUTTA PROCEDURE
2000 VV = V + DELT * Y2
U0 250 J=1*N
250 UV(J) = (DEP(J) + YPH(J,M1))*DELHY2
    CALL DERIV(UV,UV,P,3,N)
251 CONTINUE
250 UV(J) = (DEP(J) + P(J))*DELHY2
    CALL DERIV(UV,UV,TE,4,N)
260 UV(J) = DEP(J) + TE(J)*DELT
270 IF (J = 2*0* (TE(J) + P(J))
    CALL DERIV(1,UV,K5,N)
271 CONTINUE
270 UV(J) = DEP(J)
    CALL DERIV(1,UV,P5,N)
281 CONTINUE
290 YPR(J,M4) = P(J)
291 CONTINUE
COUNT = KOUNT + 1
C------ CHECK THE NUMBER OF INTEGRATION STEPS MADE BY P-K PROCEDURE
IF (KOUNT.LT.3) GO TO 5000
292 ASSIGN 1000 TO ILS
    TO 5000
5000 MU = M4
    M4 = M3
    M3 = M2
    M2 = M1
    M1 = M0
    IF (KEY*EG+0) HFTURN
    IF (KUP*LE+3) HFTURN
    KINC=0
    500 J1 = 1, N
    IF (EM(J)*GT*EPS) GO TO 515
    IF (EM(J)*10*00*0.LT.EPS) KINC=KINC+1
500 CONTINUE
510 CONTINUE
    IF (KINC*EQ+N) GO TO 550
    510 J1 = 1, N
    UPSAVE(J) = UEP(I)
510 CONTINUE
    RETURN
515 CONTINUE
    IF (KUP*EQ+4) GO TO 540
    1 = T - DELT
520 VELT = 0.5*0*DELT
    00 530 J1 = 1, N
    UEP(J) = DPSAVE(I)
530 CONTINUE
    TO 15
540 I = T - 4*0*DELT
    TO 520
550 VELT = 2*0*DELT
    TO 15
C------- ADAMS PREDICTOR-Corrector PROEDURE
1000 00 710 J1 = 1, N
    IE(J) = H(3)*YPHR(J,M1) + B(2)*YPHR(J,M2) + A(1)*YPHR(J,M3)
    DV(J) = DEP(J) + DELT* (A(4)*YPHR(J,M1) + A(3)*YPHR(J,M2))
    IF (J = 4*0* (TE(J) + P(J))
    CALL DERIV(1,DV,P2,N)
C------- SAVE PREDICTED VALUE
710 IE(J) = DV(J)
    CALL DERIV(I,DV,P1,N)
PRINT 302
PRINT 314
PRINT 315
PRINT 303
PRINT 304
PRINT 305
PRINT 306
PRINT 307
PRINT 308
PRINT 311
PRINT 310
PRINT 313
PRINT 314
PRINT 303
PRINT 305
PRINT 307
PRINT 309
PRINT 311
PRINT 401
401 FORMAT (11A,*UNUSUF ==,E12.5,///)
PRINT 316
PRINT 317
PRINT 318
PRINT 319
PRINT 320
321 FORMAT (/13A*,#EXPANSION FAN INTERSECTION POINTS*)
   IL2=2
88 PRINT 318,X4(IL),Y4(IL)
   PRINT 400
400 FORMAT (4H1,10X,*FLOW ALONG THE WING IN THE EXPANSION REGION*)
   PRINT 405
405 FORMAT (4H1,13X*,#VELOCITIES*)
   PRINT 402,44(3),45(5),47(7),49(9)
402 FORMAT (11X,9H4Aw =**E12.5,3X,*4Aw =**E12.5,3X, 19X,4Dw =**E12.5,25/)
   PRINT 403
403 FORMAT (113X*,#PRESSURES*)
   PRINT 404,44(3),45(5),47(7),49(9)
404 FORMAT (11X,9H4Aw =**E12.5,3X,*4Aw =**E12.5,3X, 19X,4Dw =**E12.5,25/)
   PRINT 500
500 FORMAT (/16X*,#FLOW CONDITIONS IN SHOCKED REGION OF EXPANSION FAN*)
   IF (JSHOCK=EW.1) GO TO 702
   PRINT 501
501 FORMAT (26X**,REGION 4FS*)
   PRINT 502,INT4(-1),X5(7),Y5(7)
502 FORMAT (13X*,THETA =**E12.5,5X,*A =**E12.5,5X,*Y =**E12.5,25/)
   PRINT 503, US4(6),PS4(8),TS4(6),RS4(7),MS4(4),GS4(6)
503 FORMAT (13X*,J =**E12.5,3X,*P =**E12.5,3X,*T =**E12.5,3X,*H =**E12.5,25/)
   PRINT 504
504 FORMAT (26X**,REGION 5AS*)
   PRINT 502,INT5(7),X5(8),Y5(8)
   PRINT 505
505 FORMAT (26X**,REGION 5HS*)
   PRINT 502,INT5(-1),X5(9),Y5(9)
   PRINT 503, US4(8),PS4(8),TS4(8),RS4(8),MS4(8),GS4(8)
   PRINT 506
506 FORMAT (26X**,REGION 5CS*)
   PRINT 502,INT5(10),X5(11),Y5(10)
   PRINT 503, US4(9),PS4(9),TS4(9),RS4(9),MS4(9),GS4(9)
   PRINT 507
507 FORMAT (26X**,REGION 5SS*)
   PRINT 502,INT5(10),X5(11),Y5(11)
   PRINT 503, US4(10),PS4(10),TS4(10),RS4(10),MS4(10),GS4(10)
   IF (JSHOCK=NE.1) GO TO 704
702 PRINT 602
602 FORMAT (26X**,CURVE1 SHOCK NOT MODELED*)
   CONTINUE
C
   NCASE=11
   CALL PI9YBAN(HSTAG)
GO TO 99
701 PRINT 601
601 FORMAT (26X**,NOT A TYPE VI PATTERN*)
99 CONTINUE
END

SUBROUTINE HELGAS(U1,P1,RH01,M1,DELTA,NOPT,J2,P2,RH02,M2,J2,22,
   GAMMA2,THETA,S2)
   COMMON/SHOCK/NFL,HSHOCK,JSW
   DELTA=DELTA/57.2
   THETAL=DELTA
   THETAR=1.571
5 THETA=(THETAL+THETAR)/2.0
IF(THETA.LT.1.57, 79) GO TO 7
IF(NFLUID.GT.1) GO TO 8
JSHOCK=1
RETURN
7 CONTINUE
TANT=TAN(THETA)
TANTMUT=TAN(THETA-DELTA)
SINT=SIN(THETA)
SINTMUT=SIN(THETA-DELTA)
COST=COS(THETA)

C EPS=TANTMUT/TANT

C RHO2=G/RHO1/CPS
U1N=U1*SINT
U1T=U1*COST
U2N=EPS*U1N
U2T=U
P2=P1+RHO1*U1N*U1N=RHO2*U2N*U2N
P2=H1*0.5*U1N*U1N=U2N*U2N
U2=U2N/SINTMUT

C H2=H2/(32+16*77*10)
CALL MULTI(RH01, RHO2, RHO3, T2, Z2, S2, RHCP, GAMMA2)
RHOFF=RHO2*RHCP
CONV=ABS(RHUFF/RHO2)
IF(CONV.0.01) 2, 2
6 IF(RHUFF) 3*2+4
3 THEATA=THETA
GO TO 5
4 THEATA=THEATA
GO TO 5
C CONTINUE

C H2=H2/32+17+777
DELTA=DELTA*77+2
THETA=THEATA*77+2

C RETURN
END

SUBROUTINE INTHST (X1, Y1, A1, A2, X2, Y2, XI, Y1)

****SUBROUTINE TO FIND THE INTERSECTION POINT OF THE SHOCKWAVES
AND THE INTERSECTION POINTS IN THE EXPANSION FAN****

XI=(Y2-Y1+X1*TAN(A1)-X2*TAN(A2))/(TAN(A1)-TAN(A2))
Y1=Y1+(XI-X1)*TAN(A1)
RETURN
END

SUBROUTINE PIGYH-K(HS)
DIMENSION F(200), G(200), FN(200), GN(200), YN(200), ETA(200), ETA(200)
DIMENSION Y(5), FUA(3), GA(3)
DOUBLE PRECISION Y, AA, BB, CC, DD, EE
DIMENSION DV(5), W(5), IE(5), YPR(5+4)*DEF(5)
DIMENSION EM(5), DPSAVE(5)
DIMENSION X4(50), Y4(50), P4(50), T4(50), U4(50)
DOUBLE PRECISION DPSAVE, DEP
EXTERNAL UHIV
COMMON/CONST/A1, A2, A3, A4, E, EPS
COMMON/KONS/TWAI/L+RAUS+PR
COMMON/WING/PT3+TT3+U3+T3+P3+X30+Y30+X4+Y4+P4+P0+T4+U4+T5E
COMMON/CASE/NCASE
IF(NCASE.GT.1) GO TO 1100
READ 1, E*DELT, EPS
1 FORMAT(3E12.5)
1100 CONTINUE
A1=U3*U3/(2.*MSPH)
A2=1.*A1
A3=1.*PH
A4=U3*U3*(PH-1.01)*(MS*PH)
IF(A3)999,774,20
20 CONTINUE
A4=0.,
B4=0.,
CC=0.47
DD=TWALL/TT3
EE=(1.-DU)*PH
IF(NCASE.GT.1) GO TO 1200
READ 13*KEY
13 FORMAT (115)
READ 14*IN*ALF
14 FORMAT(412.*L12.5)
1200 CONTINUE
Y(1)=AA
Y(2)=BB
Y(3)=CC
Y(4)=DD
Y(5)=EE
3 UPSAVE = Y(3)
FPPSAV = Y(3)
FPPU1 = Y(3)
UP01 = Y(5)
FPPPOS = Y(3)
UPPOS = Y(3)
II = 1
30 CALL NUMIN((4,DELT,T0,DV,P,T,EE,YPR,YEPS,UPS SAVE,KEY,DERIV)1)
I=0.
II=5
I=1
CALL NUMIN((4,DELT,T0,DV,P,T,EE,YPR,YEPS,UPS SAVE,KEY,DERIV)2)
5 CALL NUMIN((4,DELT,T0,DV,P,T,EE,YPR,YEPS,UPS SAVE,KEY,DERIV)3)
IF (ABS(Y(3)) .LT. E) GO TO 33
1 = 1 + 1
GO TO 5
33 IF (ABS(1.0000-Y(2)) .GT. E) GO TO 34
IF (ABS(1.0000-Y(4)) .LT. E) GO TO 99
34 IF (II.EQ.2) GO TO 35
IF (II.NE.3) GO TO 36
Y(1) = AA
Y(2) = BB
Y(3) = FPPSAV
Y(4) = DD
Y(5) = 1.0*S*GPPSAV
FPP03 = Y(3)
UP03 = Y(5)
FPPPOS = Y(3)
UPPOS = Y(3)
GO TO 30
35  
Y(1) = AA
Y(2) = DB
Y(3) = 1.05*PPS
Y(4) = UD
Y(5) = GPSAVE
PPO2 = Y(3)
WPO2 = Y(5)
PPOS = Y(3)
MPOS = Y(7)

DO TO 30

36  
UFPA = FPA(1)-FPA(2)
UGA = GA(1)-GA(2)
UFPA1 = FPA(1)-FPA(3)
UGA1 = GA(1)-GA(3)
UFPA0 = PPP1-PPP02
UGPO = GPO1-GPO3
AB = DEFPA/UPPO
CI = UFPA1/UGPO
C1 = 1.0000*FPA(1)
U1 = UGA/UFPA1
L1 = 1.0000*GA(1)
UEPP = (C1*E1*F1*B1)/(AB*E1*U1*B1)
UEGP = (L1*E1*E1*C1)/(AB*E1*U1*B1)
FPP = PPP1+UEPP
GP = GPC1+UGC

99  
CONTINUE

Y(1)=AA
Y(2)=DB
Y(3)=PPPOS
Y(4)=UD
Y(5)=GPSUS
CALL NUMIN(N,DELT,T,DU,P,T,E,ER,YPR,YEPS,DPSAVE,KEY,DERIV,1)
I = 0.
N = 5.
I = 1
ULST1 = (Y(4)-A1*(Y(2)**2))/A2 = Y(2)
IHTA1 = (Y(2))**(1.0-Y(2))
IHTE1 = (Y(2))**(1.0-(Y(2)**2))
I11 = 1
U(111) = Y(2)
U(I11) = Y(4)
E(TA11) = T
CALL NUMIN(N,DELT,T,DU,P,T,E,ER,YPR,YEPS,DPSAVE,KEY,DERIV,2)
ULST2 = (Y(4)-A1*(Y(2)**2))/A2 = Y(2)
IHTA2 = (Y(2))**(1.0-Y(2))
IHTE2 = (Y(2))**(1.0-(Y(2)**2))
ADELS = (0.5*ULST1 + ULST2)*DELT
ATMETA = (0.5*IHTA1 + IHTA2)*DELT
ATMETE = (0.5*IHTE1 + IHTE2)*DELT
55  
CALL NUMIN(N,DELT,T,DU,P,T,E,ER,YPR,YEPS,DPSAVE,KEY,DERIV,3)
UDELST = ((Y(4)-A1*(Y(2)**2))/A2 = Y(2))#DELT
UTMETA = ((Y(2))**(1.0-Y(2)))*DELT
UTMETE = ((Y(2))**(1.0-(Y(2)**2)))*DELT
ADELS = ADELS + UDELST
ATMETA = ATMETA + UTMETA
ATMETA = ATMETA + UTHETA

DECLARE THE = AT/ETE, TH/ETE
1 = 11 + 1
ETAD(1) = T
Y(1) = Y(2)
Y(1) = Y(4)
IF (ABS(Y(3)) .LT. E) GO TO 990
I = 1 + 1
GO TO 55
990 CONTINUE
N1 = 1
AAL = A1
SUM = 0.0
CROSS STREAM STATION TRANSFORM WITH =IN= NO. OF STATIONS
U0 791 I = 1,N1
Y(I) = SUM
IF(ALF) Y2 = Y1
92 IF(F(I) .LT. 9) Y1,Y1,Y1,90
90 ALF = 3.0/ETAD(I)
91 CONTINUE
SUM = SUM + .05
751 CONTINUE
J = 1
U0 752 J = 1,N1
ETAD(I) = ALF(1.0/(1.0-Y(I)))/ALF
U0 753 JJ = 1,N1
IF(ETAD(I) = ETAD(J)) 760,761,762
762 CONTINUE
J = J+1
GO TO 752
761 CONTINUE
Y(I) = 0(J)
N(I) = F(J)
GO TO 752
760 J = J-1
IF(ETAD(I) = ETAD(J)) 770,761,772
770 PRINT 788
780 FORMAT (2A* PUNNY *)
GO TO 794
772 CONTINUE
AF = (ETAD(I) = ETAD(J))/(ETAD(J+1) = ETAD(J))
FN(I) = AF*(F(J+1) = F(J) + F(J)
UN(I) = AF*(G(J+1) = G(J) + G(J)
753 CONTINUE
752 CONTINUE
U0 781 I = 1,N1
HTTHA(I) = UN(I) = AAL*FN(I) = FN(I)
791 CONTINUE
FN(IN + 1) = 1.0
UN(IN + 1) = 1.0
HTTHA(IN + 1) = 1.0 = AAL
799 CONTINUE
CALL TJOYCE(FN,THTTHA,ALF)
999 CONTINUE
RETURN
END

SUBROUTINE UENUV(Y,YUY,YULOC,N)
DIMENSION Y(5),UY(5)
DOUBLE PRECISION X,X1,A2
COMMON/CONS1/A1,A2,A3,A4,E
A2 = ((Y(4)-(A1*Y(2)*Y(3)))/A2)
A1 = (X**(*Y(2)**2))
A2 = (-0.25)*(*X***(-1.25))*(Y(5)/A2)-2.0*(A1/A2)*Y(2)*Y(3))
C--------- DOUBLE PRECISION ADAMS MOULTON PECE

C--------- 110 0 LONT

730   V(J) = DEP(J) * DELT*(H(4) * P(J) * TE(J))
    CALL DETV(T,NV,*,R,A)

731   CONTINUE

740   J=1,N

TPR(J,M4) = P(J)
DEP(J) = UER(J) * DELT*(H(4) * P(J) * TE(J))

741   CONTINUE

750   IF (NCA - GT. 10) GOTO 9999

51 )= Y30-Y4*(T+1)
71(I) = Y30-Y4*(T+1)
11 CONTINUE
  AINT = .1 * S1(11)
  A(1) = AINT
  DEL = 0.2 * S1(11)
  P(1) = P3
  R(1) = R6
  T(1) = T3
  V(1) = V3
  U(1) = U3
  i = 1
  L = 1
  IF (P21.GEQ.1.0) GO TO 13
  READ (12,PE(M),TE(M),UE(M),TW(M),RAX(M)
12 FORMAT (5E12.5)
  GO TO 13
63 IF (KK.EQ.0) GO TO 64
  M(M) = M5
  K(M) = K5
  3 FORMAT (6E12.5)
  GO TO 13
64 K(M) = RADIUS
13 CONTINUE
  P(E(M)) = PE(L)
  T(E(M)) = T(L)
  U(E(M)) = U(L)
  CALL MOLER (M,PE(M),Z,TE(M),Z,ENT,RHO,G)
  KHOE(M) = RHO
  VISE(M) = SQUALL (14,TEMPE,VIS,VI,VP,VS,TE(M))
  T(M) = T(M)
  THE(M) = TE(M)/TIE
  THE(M) = TW(M)/TIE
  U(M) = 4H(M) * CE(M)/VISE(M)
  DEL(M) = DEL
  A(M+1) = X(M+1) * DELX(M)
  IF (X(M+1) * DEL > S1(11)) GO TO 61
  GO TO 14
61 L = L + 1
  i = 1
  IF (DEL.GE.0.2 * (S1(11) + S1(I-1)))
  A(M+1) = S1(I-1) + 0.5 * DEL
  GO TO 14
62 L = L + 1
  i = 1
  S(1) = S1(2) + X(M+1)
  DEL = DELX(I)
14 CONTINUE
  PRINT 9
9 FORMAT (5H1,2X,*OUTPUT DATA**/**)
  DELT = 1.0 / (N-1)
  HH = N-1
  MM = M-1
  REAXINT(1) = URENCE(1) * XINT
  DU 1 = M = 2 * MM
  REAXINT(M) = REAXINT(M-1) + (URENCE(M-1) * URENCE(M)) * DELX(M)/2 * U
16 CONTINUE
  PRINT 17
17 FORMAT (2X,*PE(M),*THETA(M),*UE(M),*U,14X,*URENE(M),*3X,*REAXINT(M))
  DU 19 M = 1 * MM
  PRINT 18-*PE(M),*REXINT(M)
EVALUATION OF S

\[ S(1) = H\mu E(1) \cdot VIS(1) \cdot U(1) \cdot ((\text{RAD}(1) \cdot \mu K) \cdot 2) \cdot \text{INT} / (3 \cdot \mu K) \]

\[ U(M) = S(M-1) + (H\mu E(M) \cdot VIS(M) \cdot U(M) \cdot ((\text{RAD}(M) \cdot \mu K) \cdot 2) \]

\[ I \cdot K\mu E(M+1) \cdot VIS(1) \cdot U(M+1) \cdot ((\text{RAD}(M+1) \cdot \mu K) \cdot 2) \]

\[ \text{DELAX} / (2 \cdot U) \]

\[ S(M) = S(\text{MMAX}) + (H\mu E(M) \cdot VIS(M) \cdot U(M) \cdot ((\text{RAD}(M) \cdot \mu K) \cdot 2) \]

\[ \text{DETA}(1) = (C - 0.8 \cdot S(1) + U(1)) \cdot (U(2) - U(1)) \cdot (U(2) - U(1)) / (S(2) - S(1)) \]

\[ U(2) = 2 \cdot \text{MMAX} \]

\[ U(S(M)) = (S(F+1) - C(M-1) + 2 \cdot 0 \]

\[ \text{DETA}(M) = (C - 0.8 \cdot S(M) - U(E(M)+1) - U(E(M)-1)) / (U(M) \cdot (S(M+1) - S(M-1)) \]

CONTINUE

PRINT 43

\[ \text{FORMAT} / / \text{AX} \cdot 4,6, \text{AX} \cdot M \cdot \cdot B X, \cdot S(M) \cdot 7 X, \cdot U M(M) \cdot 0, \cdot \text{HETA}(M) \cdot 0, \cdot B X, \cdot \text{FLOW}(M) / \]

\[ U(M) = 45 \cdot \text{MMAX} \]

PRINT 44 \cdot M(M) \cdot C(M) \cdot U(M) \cdot HETA(M) \cdot FLW(M)

\[ \text{FORMAT} / / 45,5,5,6,5 ; \]

CONTINUE

PRINT 51 \cdot P"L"T, F"L", A"L"F

\[ \text{FORMAT} / / 45,5,5,6,5 ; \]

CONTINUE

\[ \text{PRINT} / / 48, F(1), \text{THETA}(1), C S(1) \]

\[ \text{FORMAT} / / 3 E 12, 5 \]

CONTINUE

\[ \text{CONTINUE} \]

\[ \text{CONTINUE} \]

\[ \text{CONTINUE} \]

\[ \text{CONTINUE} \]

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\[ \text{CONTINUE} \]

\[ \text{CONTINUE} \]

EVALUATION OF THE STREAM FUNCTION, FLC

\[ \text{FLC}(1) = \text{FLC}(M) \]

\[ \text{FLC}(2) = \text{FLC}(1) + (F(1) \cdot \text{ALF} + F(2) / (\text{ALF} \cdot 1,0 - \text{DELT})) \cdot \text{DELT} / (2 \cdot 0 \]

\[ U(0.5) = 1 \cdot \text{NN} \]

\[ F(1) = F(1) / (\text{ALF} \cdot (1,0 - (1,0 - \text{DELT})) \]

CONTINUE
$F_L(C(I+1) = F_L(C(I-1) + (F_F(I-1) + 4.0*F_F(I) + F_F(I+1)))*\Delta I/J.0$

CONTINUE

51 CONTINUE

52 CONTINUE

PRIN 53

53 FORMAT (11H1*2X*1*,5X*F(I)*5X*THETA(I)*5X*CSS(I)*7/

CONTINUE

54 CONTINUE

55 CONTINUE

56 CONTINUE

57 CONTINUE

58 CONTINUE

59 CONTINUE

C

EVALUATION OF THE THERMO PROPERTIES OF THE COMPONENTS
C EVALUATION OF THE THERMU PROPERTIES OF THE MIXTURE
N1 = (wmi/cb) * (wms/cb) * (wms/wmi)
N2 = (wms/cb) * (wms/wmi)
VIFC(I) = C.649/(1.1e8) * (T(I) ** 1.5) / (PE(M) * F1 * FS)
57 CONTINUE
DO 58 I = 1, MN
IF (CS(I) .EQ. 0) GO TO 71
71 CONTINUE
CPM(I) = CPS(I)
VISM(I) = VIS(I)
ICONM(I) = ICONS(I)
GO TO 73
72 CONTINUE
CPM(I) = CPS(I) * CS(I) * (CPM(I) - CPS(I))
N2 = (wms/cb) * (wms/wmi) * (wms/wmi)
VSM(I) = (CS(I) * (1.0 + G12(I) * X1(I) / AS(I)))
V1 = (1.0 + (VIS(I) / V1(S(I))) * (VIS(I) / V1(S(I)))
ICONM(I) = ICONS(I) / (1.0 + 2.65 * G12(I) * X1(I) / AS(I))
GO TO 73
58 CONTINUE
CALL MOLIER(M, P, E, M) * Z(I) * Z(I) * ENT + PH0 + G
KHOM(I) = HMO
C==C COMPUTATION OF FLOW PROPERTIES AT THE B - L - EISE
1 = N
ICONM(I) = ICONS(I)
VISM(I) = VIS(I)
CPM(I) = CPS(I)
CALL MOLIER(M, P, E, M) * Z(I) * Z(I) * ENT + PH0 + G
KHOM(I) = HMO
KHOM = KHOM(M) * VISE(K)
DO 60 I = 1, N
CKF(I) = KHOM(I) * VISM(I) / KHOM(I)
SCNO(I) = VISM(I) / (KHOM(I) * DIFFC(I))
FRNO(I) = VISM(I) * CPM(I) / TCONS(I)
60 CONTINUE
C COMPUTATION OF THE MATRIX ELEMENTS
DO 94 I = 1, MN
ALFT = ALF * (I + 1) * (I - 1) * DELT
ALF(I) = (CKF(I) * (ALFT ** 2) / (2.0 * (DELFT ** 2)) + ALFT / (4.0 * DELT))
ALF(I) = (ALF * CDF(I) - F1(I)) / (DELFT ** 2) - 2.0 * S(M) * 2 * DFLCUS(I)
94 CONTINUE
i(i) = 0

104 J = 1,3
104 k = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

111 CONTINUE
104 J = 1,3
104 k = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

112 CONTINUE
104 J = 1,3
104 k = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

113 CONTINUE
104 J = 1,3
104 k = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

114 CONTINUE
104 J = 1,3
104 k = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

121 CONTINUE
104 J = 1,3
104 k = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

122 CONTINUE
104 J = 1,3
104 k = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

123 CONTINUE
104 J = 1,3
104 k = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

124 CONTINUE

C SOLUTION FOR THE w VECTOR
w(1) = w(0)
w(N+1) = w(N)
N = N + 1
w(N) = w(N)
1 = N - 1
104 J = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)

w(1) = w(0)
w(N+1) = w(N)
1 = N - 1
104 J = 1,3
104 n(J.K) = 0
104 L = 1,3
104 i = n(J.K) + h(i, j, k) + h(i-1, l, k)
 CONTINUE
 DO 133 J=1,N
 \*(I,J,1) = \*(I,J-1) - \*(I,J,1)
133 CONTINUE
 F(I) = W(I,J-1)
 THETA(I) = THETA(I+1)
 CS(I) = W(I,J-1)
134 CONTINUE
 CONV = \*(F(2)-F(1)+F(2)+F(1))/(F(2)-F(1))
 IF (NOPRINT.GT.0) GO TO 98
 PRINT 999*CONV
998 FORMAT (10X*, CONV = *E12.5)
98 CONTINUE
 IF (ABS(F(2)-F(1)+F(2)+F(1))/(F(2)-F(1)) LE 0.0005) GO TO 150
 DO 136 I = 1,N
 F(I) = F(I)
 THETA(I) = THETA(I+1)
 CS(I) = CS(I)
136 CONTINUE
 DO 137 I = 1,N
 F(I) = 0.5*F2(I)*F(I)
 THETA(I) = 0.5*(THETA(I+1) + THETA(I))
 I(I) = I*I*THETA(I)
 CS(I) = 0.5*(CS2(I) + CS1(I))
137 CONTINUE
C EVALUATION OF THE STREAM FUNCTION, FCL?
C
 DO 140 I = 1,N
 FF(I) = F(I)/(ALF*(I-1)-I*VEL(I))
140 CONTINUE
 FCL(1) = 0.5*(FCL(1)+FCL(M+1))
 FCL(2) = FCL(1) + (F(I)/ALF + F(I)/ALF*(I-1)-VEL(I))**4*DEL/2.0
 DO 141 I = 2,N
 FCL(I) = FCL(I-2) + (FF(I-2) + FF(I-1) + FF(I))**4**DEL/2.0
 FCL(I+1) = FCL(I-1) + (FF(I-1) + FF(I))**4**DEL/2.0
141 CONTINUE
 DO 142 I = 1,N
 FLC2(I) = FLC(I)
 UFLC(I) = FLC2(I)-FLC(I)
142 CONTINUE
 IF (NOPRINT.GT.0) GO TO 90
 PRINT 999
999 FORMAT (5A1, HE JOYCE, SALVATION IS NEAR*)
90 CONTINUE
 DO 150 M = 1,N
 ETA(I) = ALUG((1.0-YN(I))/ALF
 I(I) = I*I*THETA(I)
 CALL MOLER(*, PE(M), 2*1(I), ZENT*, RHO*, G)
 RHOM(I) = RHO
150 CONTINUE
 M = M+1
 DO 160 I=1,N
 ETA(I) = ALUG((1.0-YN(I))/ALF
 I(I) = I*I*THETA(I)
 CALL MOLER(*, PE(M), 2*1(I), ZENT*, RHO*, G)
 RHO(M) = RHOM(I)
160 CONTINUE
 FACTOR = \*(Z.5*S(M))**5/(RHUE(M)*UE(M)**((RAU(M))**KK))
 Y(I) = 0.0
 DO 161 I = 2,N
 ETA(I) = ETA(I) - ETA(I-1)
 Y(I) = FACTOR*RHUE(M)**(1.0/RHUM(I-1) + 1.0/RHUM(I))**ETA(I)/2.0 +
 IY(I-1)
161 CONTINUE
ICUNM(1) = ICO(b_1) 
3n2 CONTINUE
TAU = RHM(M) * V1N(M) * UE(M) * UE(M) * (RAD(M) ** KB) * ALF * (F(2) - F(1)) / 
1 * DELT * ((DEL + KB) ** 0.5)) 
CF = TAU / (DEL ** KB) 
HEAT = -TQMUMM(M) * TOE(M) * UE(M) 

3n4 CONTINUE
MU = -HEAT 
CPE=SPNVAL(14 + MM, MP, CP, CP, CP, CP, TF(M)) 
ICUNM=SPNVAL(14 + TEM, IC + IC, IC + IC, IC + IC, TE(M)) 
PRNM(N) = V1SE(M) * CPE / ICUNM 
REC = QUE(M) * PRNM(N) 
IREC = REC + (TE - TE(M)) * TE(M) 
STUN = QUE(M) * TOE(M) * CPE(M) * (THFC - THETW(M) * T1P)) 
IUNE(M) = IREC 
WATOMT(M) = WUOT 
STNOM(M) = STNU 
SPECH(M) = SPE 
IF (NOPRINT .GT. 0) GO TO 309 
PRINT 306, TAU, CF 
3n5 CONTINUE
PRINT 307, TAU, WUOT, STUN 
3n6 CONTINUE
PRINT 308, TAU, WUOT, STUN 
3n7 CONTINUE
PRINT 309, TAU, WUOT, STUN 
3n8 CONTINUE
PF(1) = F(1) / (ALF * (1.0 - (I-1) * DELT)) 
2a0 CONTINUE
FLC(I) = 0.0 * (FL(I) / ALF * (1.0 - (I-1) * DELT)) 
FLC(2) = FLC(I) + (F(I) / ALF * (1.0 - (I-1) * DELT)) 
IF (I .EQ. 1) GO TO 282 
280 CONTINUE
IF (I > 1) GO TO 281 
FLC(1) = FLC(I-2) + (FF(1-2) + 4.0 * FF(I-1) + FF(I)) / DELT/3.0 
281 CONTINUE
FLC(1) = FLC(I-1) + (FF(I-1) + 4.0 * FF(I) + FF(I+1)) / DELT/3.0 
290 CONTINUE
HFLCOS(I) = FLC(I) - FLC(1) / VS(M) 
292 CONTINUE
HFLCOS(I) = FLC(I) - FLC(1) / VS(M) 
F1(I) = F(I) 
IHEA(I) = IHEA(I) 
CLS(I) = CLS(I) 
FLC(I) = FLC(I) 
I(I) = IHEA(I) 
290 CONTINUE
F2(1) = F1(I) 
F2(2) = F2(I) 
IF (M-MM) 579, 0, 900, 900 
900 CONTINUE
HEUHN 
END

SUBROUTINE SPLNTHP(K, X, F, Z, T) 
C *******************************
C ACCEPTS AND LEAVES UNALTERED...
C* N = NUMBER OF POINTS = 1 (CURRENTLY DIMENSIONED FOR 30 PTS)
C* X = INDEPENDENT VARIABLE ARRAY
C* F = DEPENDENT VARIABLE ARRAY
C* BOTH X AND F MUST BE DIMENSIONED AS THEY ARE IN THIS ROUTINE
C* PRODUCES AND RETURNS THE PARAMETERS R, Z, T
C* PARAMETERS OF SPLNTHP CHARACTERIZE THE INTERPOLATING NATURAL SPLINE
C* FUNCTION S(X)
C------------
C DIMENSION X(30), F(30), K(31), Z(30), T(30), HH(29), AA(24), GG(27), YU(29)

C MON=1
DO 1 J=1,N
1 K(I)=(F(I+1)-F(I))/(X(I+1)-X(I))
DO 2 J=1,NU
DO (J)=6+X*(F(J)-F(J+2))/(X(J)-X(J+2))-K(J)/(X(J+2)-X(J+1))
DO (M)=X(J+2)-X(J)
AA(J)=(X(J+1)-X(J))/UM
BH(J)=2*N
DO (G(J))=(X(J+1)-X(J+1))/UM
CALL TRISULV(1, N, B, G, U) Z)
NP=N+1
4 Z(NP)=0.0
Z(I)=0.0
DO 3 K=1,N
3 T(K)=Z(K+1)-Z(K))/((X(K+1)-X(K))/6.0)
K=(J+2)=T(I)+((X(K)-X(I))//2)
C K(N+c)=K(1)
K(NP)=R(N)+1(X(NP)-X(N))/Z(N)/Z(N)*Z(N)*Z(N)*Z(N)*Z(N)*Z(N)*Z(N)*Z(N)
RETURN
END

SUBROUTINE TRISULV(1, N, A, B, G, U) Z)
C DIMENSION ALPHA (29), BETA (29), DELTA (29), GAMMA (29), Z (31)
C MON=1
1 IF (2 GT NU) GO TO 3
DO 1 I=2,NU
UM=ALPHA(I))/BETA(I-1))
BETA(I)=BETA(I)/UM*GAMMA(I-1)
DELTA(I)=DELTA(I)/UM*DELTA(I-1)
3 Z(N)=DELTA(N)/BETA(N)
1 IF (2 GT NU) GO TO 4
DO 2 J=2,NU
JI = N-J
J1=J1+1
2 Z(J1)= (DELTA(J1)-GAMMA(J1)*Z(J1+1))/BETA(J1)
CONTINUE
K=K
RETURN
END

FUNCTION SPLNVAL(N, X, H, Z, T, ARG)
C **********************
C MUST FIRST CHECK TO SEE IN WHAT INTERVAL ARG LIES
C IN=1
1 IF (X(1), LE, ARG) GO TO 4
SPLNVAL= F(1)*H(N+2)* (ARG-X(1))
GO TO 99
4 IF (X(INN), GT, ARG) GO TO 5
SPLNVAL= F(INN)*HR(INN)* (ARG-X(INN))
GO TO 99
5 DO 1 I=2, INN
1CNT=1
1 IF (X(1), LE, ARG) GO TO 2
CONTINUE
IVAL = ICNT
IVAL+1 = ICNT - 1
IVAL = Z(IVAL1)/2 + T(IVAL) * (ARG+X(IVAL) - 2.0*X(IVAL1))
IVAL = VAL* (ARG+X(IVAL)) + K(IVAL1)
IVAL = VAL* (ARG+X(IVAL1)) + F(IVAL1)
SPLNV = VAL

CONTINUE
M = TOWN
END

SUBROUTINE MULIEV(H, P, NOPT, T, Z, S, RHO, GAMMA)

C NOPT = 0 LOOK UP DROPS BASED ON P AND H
C NOPT = 1 LOOK UP DROPS BASED ON H AND S
C NOPT = 2 LOOK UP DROPS BASED ON P AND T
C NOPT = 3 LOOK UP DROPS BASED ON H AND S

DIMENSION FLP(33, 20), HZ(33, 20), TT(33, 20), ZT(33, 20), GAME(33, 20),
1. CTRU(33, 20), FLP(1660), HZ(1660), TT(1660), ZT(1660), GAME(1660),
2. NTHUOD(1660), FLP(1660), HZ(1660), TT(1660), ZT(1660), GAME(1660).

E QI VI AL E N C E ( FLP, HZ, TT, ZT, GAME,
1. NTHUOD, ENTRU)

UNTA P0 = CP0, OH, Q, HD, ALE, RTO, CP/2110, 3.48158, 17.345, 23.6,
3. 2, 35, 35, 3.32485, 33, 105, 23867

COMMON/FLAG/ IC = L, IFP
COMMON/CHECK/ JL

IDEAL = 0
J = 1
GAMMA = 1.4
IF (I2, EQ, J3) GO TO 6
2 K1 = 120
LL = 33*(K - 1)
2 L = 133
FLPO(=[LL + L1) = FLP(1)
IF (K = 0) GO TO 12
FLPO(=[LL + L2) = HCU(L)

CONTINUE

1 = 33
J = 2
1 I = 1 + IZ
MTML(I) = MT(I + 1)
1 J = 1 + JZ
ENTROV(J, I) = ENTRV(I, JZ - J + 1)
1 FLPV(J, I) = FLP(1, IZ - J + 1)

5 IF (NOPT = 0) GO TO 40
FLP = ALU210(P/2110.) + 10.
1 IF (NOPT = 1) IF = 20, 30
10 IF (H = 100) GO TO 100
CALL UNT((H, HZ, TT, ZT, FLP, FLVZ, IC, JL, T, Tt, S, HTHUOD, GAMMA, GAME)
1 IF (L = 22222 + 30) I = 12, 12
11 H = P/(32, 2, 53, 75, 27, 71)
GO TO 50
100 IF (H = 100) GO TO 12
1 IF (CP0 = CP0, OH, ALU210(P/PU) * AL, * SUH
GO TO 50
1 IF (PL = FLVZ(I)) GO TO 22
21 IF (PL = FLVZ(J)) 23, 20221
CONTINUE
DOUBLE INTERPOLATION SUBROUTINE IF IL=1, THE SECOND INDEPENDENT VARIABLE IS NOT CONSTANT WITH THE FIRST, (THE RANGE OF THE SECOND IS NOT THE SAME AT EACH VALUE OF THE FIRST)

**DIMENSION ZI(NL,AT(ML,NL),Y1I(ML,NL),Y2(ML,NL),Y3(ML,NL)**

**DIMENSION Y4(ML,NL)**

**IF(ZZ-ZT(I))=0.001 THEN 801**

**IF(IL=I) GO TO 804**

**IF(XX+XT(I))=60 TO 802**

**DO 800 I=1,NL**

**L=I**

**IL=I-1**

**IF(ZZ-ZT(I))=0.004 THEN 800**

**CONTINUE**

**RETURN**

**802 HAT1(ZZ-ZT(1))/ZT(1)-ZT(1)**

**IF(IL=I) GO TO 804**

**803 IF ZI=ZT(I) 802=I=Z02**

**DO 800 I=1,NL**

**L=I**

**IL=I-1**

**IF(ZZ-ZT(I))=0.004 THEN 800**

**CONTINUE**

**RETURN**

**902 HAT1(ZZ-ZT(1))/ZT(1)-ZT(1)**

**IF(IL=I) GO TO 804**

**800 CONTINUE**

**RETURN**

**902 HAT1(ZZ-ZT(1))/ZT(1)-ZT(1)**

**IF(IL=I) GO TO 804**

**CONTINUE**

**GO TO 802**

**905 HAT1(ZZ-ZT(1))/ZT(1)-ZT(1)**
111 HRETURN

112 IF(XX-XT(J,L))911,912,913

913 CONTINUE

113 GO TO 802

917 Y5=YT1(LM+L)
Y7=YT2(LM+L)
Y9=YT3(LM+L)
Y0=YT4(LM+L)
GO TO 1114

116 HRETURN

914 GO TO 921

921 CONTINUE

920 Y4=YT1(LM+L)
Y6=YT2(LM+L)
Y8=YT3(LM+L)
YA=YT4(LM+L)
DOUBLE INTERPOLATION SUBROUTINE

DIMENSION ZI(NL), XT(KL+NL), YT1(ML+NL)
IF (ZL-ZT(Il)) 902, 901, 901
801 IF (XX-XT(Il,J)) 803, 803, 803
802 JL=2222222222
KFTURN
803 UO 30 1=1, NL
L=1
LL=I-1
IF (ZL-ZT(Il)) 902, 904, 900
800 CONTINUE
UO TO 842
902 KATI0=(ZL-ZT(Il))/ZT(L)-ZT(LL)
GO TO 1113
804 UO 904 J=1, ML
LM=J
LLM=J-1
IF (XX-XT(Il,J)) 905, 906, 904
904 CONTINUE
GO TO 842
905 KATI0=(AX-XI(LLM,L))/XT(LM,L)-XT(LLL,L)
Y1=YT1(LLM,L)+KATI0*YT1(LLM,L)-YT1(LLL,L)
KFTURN
906 Y1=YT1(LM,L)
KFTURN
1113 UO 918 J=1, ML
LM=J
LLM=J-1
IF (XX-XT(Il,J,L)) 915, 917, 918
918 CONTINUE
UO TO 842
917 Y5=YT1(LML,L)
GO TO 1114
916 KATI0=(AX-XI(LLM,L))/XT(LM,L)-XT(LLL,L)
Y5=YT1(LLM,L)+KATI0*YT1(LLM,L)-YT1(LLL,L)
1114 UO 921 J=1, ML
LM=J
LLM=J-1
IF (XX-XT(Il,J,L)) 919, 920, 921
921 CONTINUE
UO TO 802
920 Y4=YT1(LML,L)
GO TO 925
919 KATI0=(AX-XI(LLM,L))/XT(LM,L)-XT(LLL,L)
Y4=YT1(LLM,L)+KATI0*YT1(LLM,L)-YT1(LLL,L)
925 Y4=KATI0*(Y4-Y5)
KFTURN
END
DESCRIPTION OF OUTPUT

The output for the real-gas code includes the flow condition from each region of the flow-field model, the geometry of the shock waves and the expansion waves, the heat-transfer distribution along the second wedge (i.e., the wing leading-edge), the boundary-layer velocity profile and temperature distribution profile, and boundary-layer parameters. The units for a particular parameter in any region will be the same as the free-stream parameter, unless otherwise noted. The output for the free-stream flow includes:

\begin{itemize}
  \item \textbf{U}_1 \quad \text{free-stream velocity (ft/sec)}
  \item \textbf{P}_1 \quad \text{free-stream static pressure (lbf/ft}^2\text{)}
  \item \textbf{T}_1 \quad \text{free-stream temperature (°R)}
  \item \textbf{RHO}_1 \quad \text{free-stream density (slugs/ft}^3\text{)}
  \item \textbf{PT}_1 \quad \text{free-stream stagnation pressure (lbf/ft}^2\text{)}
  \item \textbf{TT}_1 \quad \text{free-stream stagnation temperature (°R)}
  \item \textbf{GAMMA}_1 \quad \text{free-stream } \gamma
  \item \textbf{H}_1 \quad \text{free-stream enthalpy (Btu/lbm)}
  \item \textbf{Z}_1 \quad \text{free-stream molecular weight ratio}
\end{itemize}

Output for the flow condition in region "I", where I = 2,3, or 6 includes:

\begin{itemize}
  \item \textbf{U}_I \quad \text{the velocity}
  \item \textbf{PI}/\textbf{P}_1 \quad \text{the static pressure ratio}
  \item \textbf{TI}/\textbf{T}_1 \quad \text{the temperature ratio}
  \item \textbf{RHO}_I/\textbf{RHO}_1 \quad \text{the density ratio}
  \item \textbf{PI} \quad \text{the static pressure}
  \item \textbf{TI} \quad \text{the temperature}
  \item \textbf{RHO}_I \quad \text{the density}
  \item \textbf{PT}_I \quad \text{the stagnation pressure}
  \item \textbf{TT}_I \quad \text{the stagnation temperature}
\end{itemize}
HI - the enthalpy
ZI - the molecular weight ratio
GAMMAI - the $\gamma$ for the particular region
DELTA - the change in flow direction from one region to the next
THETA - the shock wave angle

The output for the flow conditions in the expansion from regions 4 and 5, where I = A, B, C, D, or E (i.e., the five subregions) includes:

UI - the velocity
PI - the static pressure
TI - the temperature
RHOI - the density
HI - the enthalpy
GI - the effective $\gamma$ for the particular region
NUDIF - the difference between the total change of the Prandtl-Meyer angle in region 4 and the total change of the Prandtl-Meyer angle in region 5 (should be zero so that flow in region 5e is parallel to the wing)

Output of intersection points include:

INITIAL POINTS ON BOW SHOCK WAVE, i.e., the origin of the coordinate system, or the nose.

WING INTERSECTION POINT, i.e., intersection of the two wedges.

BOW SHOCK: WING SHOCK INTERSECTION POINT, i.e., the intersection of the shock of the first wedge with the shock of the second wedge.

Next Five Points - the intersection of the centered expansion fan with the wing leading edge.

Last Five Points - the intersections of the reflected waves with the inboard shear layer.
Output listed for the interaction region between the left running and right running expansion fan waves includes:

- \( U_{4W} \) - the velocity
- \( P_{4W} \) - the static pressure

where "I" = A, B, C, and D.

The output for the subregions between the inboard shear layer and the shock waves between the shock layer and the free stream include:

- \( \Theta \) - shock wave angle
- \( x \) - x-coordinate point of the intersection of the shock waves of two adjacent regions (in.)
- \( y \) - y-coordinate point of the shock waves intersection (in.)
- \( v \) - the velocity
- \( P \) - the static pressure
- \( T \) - the temperature
- \( R \) - the density
- \( H \) - the enthalpy (\( \text{ft}^2/\text{sec}^2 \))
- \( G \) - the effective \( \gamma \) for the region

The output for the viscous-region subroutines is tabulated under the title "OUTPUT DATA" includes:

- \( M \) - station number in x-direction
- \( P_{E}(M) \) - Static pressure at the edge of the boundary layer
- \( \Theta_{E}(M) \) - temperature ratio (\( T_{e}/T_{te} \)) at the edge of the boundary layer
- \( U_{E}(M) \) - velocity at the edge of the boundary layer
- \( U_{RENOE}(M) \) - unit Reynolds number at the edge of the boundary layer
- \( R_{EXINT}(M) \) - Reynolds number at the edge of the boundary layer integrated over the distance from the origin
- \( X(M) \) - distance along wing leading edge (ft)
- \( S(M) \) - transformed x-coordinate
DS(M) - step-size of S(M)
BETA(M) - velocity gradient at the edge of the boundary layer
FLOW(M) - stream function at the wall used to indicate mass injection
PTE - stagnation pressure at the edge of the boundary layer
TTE - stagnation temperature at the edge of the boundary layer
ALF - coordinate transformation parameter

The next page of output is the initial boundary-layer profile for the first x-station. The output that follows is at each of the next (M-1) x-stations and includes:

M - station number in x-direction
X(M) - distance along wing leading edge (ft)
I - station number in y-direction
YN(I) - transformed n-coordinate (see Ref. 11), n
Y(I) - physical y-coordinate (ft)
ETA(I) - transformed y-coordinate, n
F(I) - velocity profile in the boundary layer
THETA(I) - temperature distribution profile in the boundary layer
CS(I) - mass fraction of the stream species
TAU - skin friction (lbf/ft²)
CF - skin friction coefficient
TREC - recovery temperature, °R
QDOT - heat transfer rate (Btu / ft²·sec)
STNO - Stanton number
DELST - displacement thickness (ft)
THMOM - momentum thickness (ft)
DSTAR - displacement thickness with mass injection (see Ref. 11) (ft)
### CASL = 1

**FREE-STREAM FLOW CONDITIONS**

<table>
<thead>
<tr>
<th>UI</th>
<th>P1 (Lbf/ft²)</th>
<th>T1 (°RANKINE)</th>
<th>RH01 (SLUGS/ft³)</th>
<th>PT1</th>
<th>TT1</th>
</tr>
</thead>
<tbody>
<tr>
<td>1,4200E+04</td>
<td>9,24E+02</td>
<td>4,97E+02</td>
<td>1,1052E+04</td>
<td>n/a</td>
<td>n/a</td>
</tr>
</tbody>
</table>

\[ \text{\textit{Gamma}}_1 = 1,3930E+00 \]
\[ \text{\textit{M}}_1 = 1,1764E+02 \]
\[ \text{\textit{Z}}_1 = 1,1460E+00 \]

**FLOW CONDITIONS IN REGION 2**

<table>
<thead>
<tr>
<th>UI</th>
<th>P2/P1</th>
<th>T2/T1</th>
<th>RH02/RH01</th>
</tr>
</thead>
<tbody>
<tr>
<td>1,4075E+04</td>
<td>4,0317E+00</td>
<td>1,6033E+00</td>
<td>2,5100E+00</td>
</tr>
</tbody>
</table>

\[ \text{\textit{P}}_2 = 3,474E+02 \]
\[ \text{\textit{T}}_2 = 7,352E+02 \]
\[ \text{\textit{H}}_2 = 2,779E+02 \]
\[ \text{\textit{Z}}_2 = 1,1080E+00 \]

\[ \text{\textit{Gamma}}_2 = 1,3911E+00 \]
\[ \text{\textit{Theta}} = 3,3111E+00 \]

**FLOW CONDITIONS IN REGION 3**

<table>
<thead>
<tr>
<th>UI</th>
<th>P3/P1</th>
<th>T3/T1</th>
<th>RH03/RH01</th>
</tr>
</thead>
<tbody>
<tr>
<td>1,2165E+04</td>
<td>4,3113E+02</td>
<td>1,3194E+02</td>
<td>1,5655E+01</td>
</tr>
</tbody>
</table>

\[ \text{\textit{P}}_3 = 1,5193E+02 \]
\[ \text{\textit{T}}_3 = 4,734E+02 \]
\[ \text{\textit{H}}_3 = 1,739E+03 \]
\[ \text{\textit{Z}}_3 = 1,1060E+00 \]

\[ \text{\textit{Theta}} = 3,5080E+01 \]
\[ \text{\textit{Gamma}}_3 = 1,2142E+01 \]

**FLOW CONDITIONS IN REGION 4**

<table>
<thead>
<tr>
<th>UI</th>
<th>P4/P1</th>
<th>T4/T1</th>
<th>RH04/RH01</th>
</tr>
</thead>
<tbody>
<tr>
<td>1,1825E+04</td>
<td>7,0379E+01</td>
<td>9,23E+00</td>
<td>7,5943E+00</td>
</tr>
</tbody>
</table>

\[ \text{\textit{P}}_4 = 6,777E+01 \]
\[ \text{\textit{T}}_4 = 4,5219E+03 \]
\[ \text{\textit{H}}_4 = 8,3941E+06 \]
\[ \text{\textit{Z}}_4 = 1,7841E+04 \]
FLOW CONDITIONS IN THE EXPANSION FAN

HELIUM 4

VELOCITIES

UA = 1.6e2824E+04 UN = 1.22613E+04 UC = 1.22943E+04 UU = 1.23271E+04 UE = 1.23974E+04

PRESSURES

PA = 1.14594E+04 PM = 1.04536E+02 PC = 1.02334E+02 PD = 0.44458E+01 PE = 4.09521E+01

TEMPERATURES

TA = 4.60581E+03 TH = 3.44715E+03 TC = 3.94470E+03 TU = 1.90224E+03 TF = 7.14501E+03

DENSITIES

W@OA = 1.51239E+03 RMH = 1.47553E-05 WMOC = 1.30256E-05 RM@0 = 1.43270E-05 WOF = 1.34641E-05

ENTHALPIES

NA = 1.1577E+03 NH = 1.14179E+03 NC = 1.12559E+03 ND = 1.10937E+03 NE = 1.94909E+03

EFFECTIVE GAMS

VA = 1.4377E+00 VM = 1.22149E+00 VC = 1.22950E+00 GD = 1.22067E+00 GE = 1.24932E+00

HELIUM 6

VELOCITIES

UA = 1.6e3961E+04 UN = 1.24253E+04 UC = 1.24578E+04 UU = 1.24902E+04 UE = 1.25521E+04

PRESSURES

PA = 4.63760E+01 PM = 8.09807E+01 PC = 7.13019E+01 PD = 7.14849E+01 PE = 6.57792E+01

TEMPERATURES

TA = 3.17864E+03 TH = 3.17864E+03 TC = 3.73569E+03 TD = 3.49487E+03 TF = 7.63324E+03
DENSITIES

\( \rho_{HA} = 1.304 \times 10^4 \quad \rho_{MB} = 1.243 \times 10^5 \quad \rho_{MC} = 1.1327 \times 10^5 \quad \rho_{PHON} = 1.0511 \times 10^5 \quad \rho_{OF} = 1.0621 \times 10^5 \)

FATALITIES

\( \text{HA} = 1.0447 \times 10^6 \quad \text{MB} = 1.0473 \times 10^6 \quad \text{MC} = 1.0647 \times 10^6 \quad \text{HD} = 1.0365 \times 10^6 \quad \text{WE} = 1.0173 \times 10^6 \)

EFFECTIVE GAMMAS

\( \text{HA} = 1.2419 \times 10^5 \quad \text{MB} = 1.2419 \times 10^5 \quad \text{MC} = 1.2419 \times 10^5 \quad \text{HD} = 1.2419 \times 10^5 \quad \text{AF} = 1.2419 \times 10^5 \)

\( \text{UNJIF} = 4.089 \times 10^{-2} \)

POINTS OF INTEREST IN THE FLOW FIELD AND ON THE BODY (IN INCHES)

INITIAL POINTS ON HOW SHOCK WAVE
\( x = 0 \quad y = 0 \)

\( \text{WIND INTERSECTION POINT} \)
\( x = 1.0000 \times 10^1 \quad y = 2.197 \times 10^{-1} \)

\( \text{HOW SHOCK-WIND SHOCK INTERSECTION POINT} \)
\( x = 1.0000 \times 10^1 \quad y = 2.197 \times 10^{-1} \)

EXPANDING FAN INTERSECTION POINTS
\( x = 1.0000 \times 10^1 \quad y = 2.197 \times 10^{-1} \)
\( x = 2.197 \times 10^1 \quad y = 2.197 \times 10^{-1} \)
\( x = 2.197 \times 10^1 \quad y = 2.197 \times 10^{-1} \)
\( x = 2.197 \times 10^1 \quad y = 2.197 \times 10^{-1} \)
\( x = 2.197 \times 10^1 \quad y = 2.197 \times 10^{-1} \)
\( x = 2.197 \times 10^1 \quad y = 2.197 \times 10^{-1} \)
\( x = 2.197 \times 10^1 \quad y = 2.197 \times 10^{-1} \)
\( x = 2.197 \times 10^1 \quad y = 2.197 \times 10^{-1} \)
FLUID ALONG THE WING IN THE EXPANSION REGION

VELOCITY

$u_{\text{in}} = 1.2043\times 10^4$  $u_{\text{in}} = 1.2327\times 10^4$  $u_{\text{in}} = 1.2926\times 10^4$  $u_{\text{in}} = 1.2457\times 10^4$

PRESSURES

$p_{\text{in}} = 1.3023\times 10^2$  $p_{\text{in}} = 4.4646\times 10^4$  $p_{\text{in}} = 8.5708\times 10^4$  $p_{\text{in}} = 7.6391\times 10^4$

FLUID CONDITIONS IN SHOCKED REGION OF EXPANSION FAN

REGION 45

$T_{\text{in}} = 3.0527\times 10^4$  $x = 1.1725\times 10^4$  $y = 2.6471\times 10^4$
$u = 1.1107\times 10^4$  $v = 7.6194\times 10^4$  $T = 0.7584\times 10^4$  $R = 0.3722\times 10^4$  $H = 1.2047\times 10^4$  $\theta = 1.1545\times 10^4$

REGION 45-

$T_{\text{in}} = 3.1745\times 10^4$  $x = 1.1402\times 10^4$  $y = 2.5402\times 10^4$
$u = 1.1107\times 10^4$  $v = 7.4763\times 10^4$  $T = 0.7407\times 10^4$  $R = 0.2749\times 10^4$  $H = 1.1946\times 10^4$  $\theta = 1.1510\times 10^4$

REGION 50-

$T_{\text{in}} = 3.4105\times 10^4$  $x = 1.1384\times 10^4$  $y = 2.6363\times 10^4$
$u = 1.1030\times 10^4$  $v = 7.3271\times 10^4$  $T = 0.7363\times 10^4$  $R = 0.2749\times 10^4$  $H = 1.1946\times 10^4$  $\theta = 1.1510\times 10^4$

REGION 55-

$T_{\text{in}} = 3.9480\times 10^4$  $x = 1.1534\times 10^4$  $y = 2.7364\times 10^4$
$u = 1.1467\times 10^4$  $v = 5.9327\times 10^4$  $T = 0.4912\times 10^4$  $R = 0.4333\times 10^4$  $H = 1.3043\times 10^4$  $\theta = 1.1819\times 10^4$

REGION 60-

$T_{\text{in}} = 3.2774\times 10^4$  $x = 1.1757\times 10^4$  $y = 2.8365\times 10^4$
$u = 1.1547\times 10^4$  $v = 7.0472\times 10^4$  $T = 0.3984\times 10^4$  $R = 0.4177\times 10^4$  $H = 1.3942\times 10^4$  $\theta = 1.1747\times 10^4$
### OUTPUT DATA

<table>
<thead>
<tr>
<th>PE(M)</th>
<th>METAL(M)</th>
<th>UE(M)</th>
<th>URENE(M)</th>
<th>HEXINT(M)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.42137E+02</td>
<td>4.2744E-01</td>
<td>1.2195E+04</td>
<td>1.5251E+04</td>
<td>1.9315E+04</td>
</tr>
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THEC = 7.1314E+00
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THMOM = 7.35693E+00

STAN = 1.13467E+03
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CONV = 3.43942E+00

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REJOICE, SALVATION IS NEAR
CONV = 3.43942E+00
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APPENDIX C. - SOLUTIONS FOR A SINGLE WEDGE CONFIGURATION

It has been noted that the local heat-transfer rates along the "wing leading-edge" (which were nondimensionalized using a current shuttle design parameter) increased significantly with velocity. The velocity dependence which has been found in the present study of the Type VI shock-interference pattern had been found in previous studies of cones (work done at the Johnson Space Center) and of wedges (work done at the University). Using procedures similar to those described previously, theoretical solutions have been obtained for a single wedge configuration with a 30° deflection angle. The solutions for the three flow conditions of the main text:

Condition 1. - $U_\infty = 1167$ m/sec, $P_\infty = 2.98$ mmHg, $T_\infty = 53^0\text{K}$
Condition 2. - $U_\infty = 4330$ m/sec, $P_\infty = 0.333$ mmHg, $T_\infty = 273^0\text{K}$
Condition 3. - $U_\infty = 7610$ m/sec, $P_\infty = 0.0268$ mmHg, $T_\infty = 195^0\text{K}$

are reviewed briefly in this appendix.

The heat-transfer rate at a point 0.30m (1.0 ft) from the apex is presented in Fig. C.1 as a function of the free-stream velocity. The local heating has been divided by the theoretical heat-transfer to the stagnation point of a sphere whose radius is 0.30m. The nondimensionalized heat-transfer rates increased by approximately 40% over the velocity range considered. The increase in heating with velocity for the single wedge is not as great as that observed for the double-wedge configuration which produces the Type VI shock-interference pattern. For a given velocity, the difference between the perfect-gas and the real-gas solutions is relatively small.

Other flow parameters are presented in Table C1. The pressure ratio across the oblique shock wave, the temperature ratio, the density ratio, the entropy increase, the shock-wave angle, and the effective gamma of the
shock-layer flow. The relatively high density ratio which occurs when the real-gas effects are accounted for results in a thinner shock-layer for the real-gas solution. Thus, the effective gamma and the shock-wave angle are smaller for the real-gas solution. However, for a given flow condition, the pressure ratio for the perfect-gas solution is approximately equal to that for the real-gas solution.

Also presented are the theoretical value of the heat-transfer rate to the stagnation point of a sphere whose radius is 0.30m and the entropy increase across the normal shock for the assumed stagnation point. Note that the ratio

$$\left( \frac{S_2 - S_1}{S_1} \right) / \left( \frac{S_{t,ns} - S_1}{S_1} \right)$$

i.e., the increase of entropy across the oblique shock-wave divided by the increase of entropy across a normal shock-wave, is approximately 0.5 for all three velocities. The constancy of this ratio suggests that the mechanism which produces the velocity dependence of the nondimensionalized heat-transfer rate is not related to the shock-induced entropy change (which relates to the shock-wave strength). This question is relevant because the heat transfer for the wedge flow (which passes through an oblique shock) is divided by the stagnation-point heat-transfer for the sphere flow-field (which passes through a normal shock wave).
Table C.1. - Flow parameters for a 30° single-wedge configuration

<table>
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<tr>
<th>Flow Condition</th>
<th>Gas</th>
<th>$P_2/P_1$</th>
<th>$T_2/T_1$</th>
<th>$\rho_2/\rho_1$</th>
<th>$(S_2 - S_1)/S_1$</th>
<th>$\theta_s$</th>
<th>$\gamma$</th>
<th>$q_{t,R=0.3m}^*$ (watts/m$^2$)</th>
<th>$(S_{t,ns} - S_1)/S_1$</th>
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<td>Perf</td>
<td>29.75</td>
<td>5.925</td>
<td>5.021</td>
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<td>39.27</td>
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<td>$3.47 \times 10^4$</td>
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<td>5.915</td>
<td>5.000</td>
<td>0.1111</td>
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<td>0.1666</td>
<td>35.41</td>
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<td>Real</td>
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<td>20.269</td>
<td>11.530</td>
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<td>1.216</td>
<td>$1.12 \times 10^6$</td>
<td>0.696</td>
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Appendix C: Figure 1. - Effect of free-stream velocity on the dimensionless heat-transfer for a wedge whose deflection angle is 30°. $T_w = 394^\circ K$