CONFIDENTIAL

Copy
RM L57D22c

c2

NACA

RESEARCH MEMORANDUM

AERODYNAMIC HEAT TRANSFER TO WING SURFACES AND WING LEADING EDGES

By Aleck C. Bond, William V. Feller,
and William M. Bland, Jr.

Langley Aeronautical Laboratory
Langley Field, Va.

CLASSIFICATION CHANGED

To UNCLASSIFIED

By authority of NASA ltr Nov. 28, 1962 By HGR
dtd Nov. 14, 1962
s/ Boyd C. Myers II
Effective date: Apr. 23, 1962.

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
June 5, 1957

CONFIDENTIAL

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

AERODYNAMIC HEAT TRANSFER TO WING SURFACES AND WING LEADING EDGES

By Aleck C. Bond, William V. Feller,
and William M. Bland, Jr.

SUMMARY

A compilation is presented of some recent results obtained at various facilities of the Langley Aeronautical Laboratory on the aerodynamic heat transfer to wing surfaces and wing leading edges at high speeds. Data obtained from hypersonic tunnel tests, rocket-powered-model flight tests, and high-stagnation-temperature jet tests are included and compared with applicable theories. Measured heat transfer to wing surfaces exposed to high-heat fluxes, in general, showed good agreement with theory. Theory was also shown to be adequate in predicting the heat transfer to wing surfaces over a range of local Reynolds numbers from 1×10^6 to 20×10^6. Heat-transfer measurements on a blunted slab wing at a Mach number of 6.86 showed that surface heating at zero angle of attack was essentially the same for sweep angles of 0°, 40°, and 60°; also, increasing the angle of attack of the slab wing increased the heating of the windward surface and decreased the heating of the leeward wing surface. Measurements at high Reynolds numbers showed that rates of heat transfer that are much higher than laminar rates can be experienced on leading edges in the region of the wing-body juncture. Transpiration-cooling tests on a wedge surface showed that theoretical predictions on the effectiveness of nitrogen as a transpiration coolant apply equally as well for conditions of either high or low heating potentials. Further cooling tests showed that helium is about five times more effective as a transpiration coolant than nitrogen.

INTRODUCTION

Considerable emphasis has recently been placed on the experimental study of the aerodynamic heat transfer to wing surfaces and wing leading edges. The wing leading edge has received a large amount of consideration and has been treated both theoretically and experimentally in references 1 to 4. A qualitative type of investigation on wing leading edges at conditions of high stagnation temperature has been reported in reference 5. Investigations are reported in references 6 to 8 of the aerodynamic heating of wing surfaces at angles of attack at low supersonic speeds; however, data on the effects of leading-edge bluntness, sweep, and angle of attack, particularly at hypersonic speeds, as well as the effects of high heating potentials and high Reynolds numbers on wing-surface heating are indeed lacking in the current literature.

Results of experiments conducted at various facilities of the Langley Aeronautical Laboratory have recently become available which provide some extension of the state of the art regarding the heat transfer to wing leading edges and wing surfaces. Rocket-powered-model flight tests have provided some large-scale measurements of the heat transfer to wing surfaces as well as of the heating of blunt leading edges in the vicinity of the wing-body juncture. Hypersonic wind-tunnel tests have provided information on both the effects of sweep and angle of attack on the heat transfer to a wing having large leading-edge bluntness. Tests in a jet having a high stagnation temperature have also provided results on the heating of wings under conditions of high rates of heat transfer. Furthermore, data on the effectiveness of transpiration cooling of flat surfaces, under conditions of high stagnation temperature, have also recently become available from tests in a hot-jet facility. It is, therefore, the purpose of this paper to present a compilation of these recent results which are applicable to the high-speed wing-heating problem.

SYMBOLS

F	injection parameter, $\frac{\rho_c V_c}{\rho_l V_l}$
M	Mach number
N_{St}	Stanton number based on local conditions, $\frac{h}{c_{p_l} \rho_l V_l}$
$N_{St,\infty}$	Stanton number based on free-stream conditions, $\frac{h}{c_{p_\infty} \rho_\infty V_\infty}$
R	Reynolds number, $\frac{\rho V s}{\mu}$
R_d	Reynolds number based on diameter, $\frac{\rho_\infty V_\infty d}{\mu_\infty}$
T	temperature, $^{\circ}F$ or $^{\circ}R$

V velocity, ft/sec

c_p specific heat of air at constant pressure, $\frac{\text{Btu/slug}}{^{\circ}\text{F}}$

d diameter of cylindrical leading edge, ft

h local aerodynamic heat-transfer coefficient, $\frac{\text{Btu}}{(\text{sq ft})(\text{sec})(^{\circ}\text{F})}$

q dynamic pressure, lb/sq ft

s length from stagnation point to measurement station, as indicated when used, ft

Λ sweep angle, deg

α angle of attack, deg

μ viscosity of air, lb-sec/sq ft

ρ density, slugs/cu ft

Subscripts:

AW adiabatic wall

c coolant

EXP experimental

l conditions outside boundary layer

0 zero coolant flow rate

TH theoretical

t total conditions

W conditions pertaining to skin of model

∞ free-stream conditions

RESULTS AND DISCUSSION

Heat Transfer at High Stagnation Temperature

In order to study heating problems associated with high-heat fluxes, one of the free jets of the Langley preflight jet of the Pilotless Aircraft Research Station at Wallops Island, Va., has been modified to allow testing of relatively large test specimens at stagnation temperatures up to about 3,500° R which corresponds to a Mach number of about 6.5 at altitude. The test Mach number is maintained the same ($M = 2.0$); however, the stream temperature is increased by burning ethylene gas upstream of the exit nozzle. This mode of testing provides heating rates corresponding to Mach numbers much higher than the stream Mach number in a test medium not much different from air. Recent tests with wings subjected to high heating rates have yielded heat-transfer data which show good agreement with theory. Results for a wing tested at a stagnation temperature of 3,476° R, a stream Mach number of 2.0, and a stream Reynolds number of 2.4×10^6 per foot are shown in figures 1 and 2. The wing was a tapered unswept wing of hexagonal cross section, was constructed of magnesium, and had a span of 11 inches. The leading edge which was blunted to a 1/16-inch radius was covered with a 1/32-inch Inconel cap to increase its endurance at the test temperature. Sweep of the leading edge was 17°. At the 63-percent spanwise station (coinciding with the center line of the jet), temperature measurements were made at four chordwise locations: the stagnation-point and the 1-, 2-, and 3.77-inch stations. Temperature time histories measured on the wing are shown in figure 1. The temperature at the stagnation point rises rapidly and in approximately 2.1 seconds (time of failure of the leading-edge cap) has reached 2,731° R, just about 229° R below the melting temperature of Inconel. The temperatures at the three rearward stations show a maximum rise of about one-half the stagnation-point value. It should be noted, however, that the temperature of the 2-inch station, which is immediately behind the end of the leading-edge cap, is generally about 200° higher than the temperatures of the other two stations during most of the test.

In figure 2 these wing heating data are reduced to the nondimensional Stanton number evaluated at local conditions and are also plotted as a function of time. In the plot at the upper left, the stagnation-point data are compared with the average laminar stagnation-point theory of Goodwin, Creager, and Winkler (ref. 4). Comparison with the average theoretical stagnation-point values rather than with local values is made since it was felt that, because of the small physical size of the leading edge, the measurements at this point more nearly represented average values rather than local point values. The first data point shows good agreement with the theory; however, with increasing time, the data show increasing deviation from the theory. This disagreement is believed to be due mainly to large lateral conduction losses from the stagnation point as the leading edge increases in temperature, since it is known that the temperature gradient across the jet is not uniform but tends to peak in the vicinity of the jet center line. The data for the other three stations are compared with the flat-plate theories of Van Driest (refs. 9, 10, and 11) and, in general, show good agreement. The theoretical curves were determined by using calculated local conditions and a Reynolds number length equal to the streamwise distance from the stagnation point to the measurement station. These data show that the flow was initially laminar at the 1-inch station and that transition took place between the 1- and 2-inch stations and then gradually moved forward with time. Initially, transition could have been caused by the abrupt discontinuity at the end of the leading-edge cap, and the forward movement with time may have been due to the increased temperature (i.e., temperature ratio) of the surfaces with time. It is obvious that, had the transition point not moved forward, a serious hot spot would have developed at or near the 2-inch measurement station. The comparison of the data for the three downstream stations with those of the stagnation point show that the turbulent level of heating of the downstream stations was of the order of one-half that of the stagnation point.

It might be mentioned at this time that, in the reduction of the heating data for the 1-inch station to local Stanton number, the local conditions were determined by considering the losses through the normal shock caused by the blunted leading edge. This procedure gave much better agreement with the theory than was obtained when the losses were neglected. For the two stations downstream, the agreement between theory and experiment was not enhanced when taking these losses into account; thus, the effects of this small amount of bluntness did not propagate very far downstream in this case.

Large-Scale Heat-Transfer Measurements

In recent rocket-powered-model flight tests, some relatively large-scale heat-transfer measurements have been obtained on two different wing configurations at free-stream Reynolds numbers up to approximately 27×10^6 based on the wing chord. This Reynolds number is equivalent to that of a wing of 120-inch chord flying at an altitude of 50,000 feet and a Mach number of 2.5. In order to show the general agreement which was obtained with theory over this large Reynolds number range, representative data from these tests are shown in figure 3 as the ratio of experimental to theoretical Stanton number as a function of the local Reynolds number. It might be mentioned at this time that there was no indication of laminar heating on either of the wings throughout the usable portion of the flight tests; however, at the beginning of the tests there was some indication of transitional flow at the most forward measurement stations. The theoretical Stanton numbers were, therefore, evaluated from the turbulent flat-plate theory of Van Driest (refs. 10

and 11) by using calculated local conditions and a value of 0.6 for the ratio of Stanton number to skin-friction coefficient.

One of the test wings was unswept, untapered, of approximately 20-inch chord and span, employed a 5-percent circular-arc airfoil section, and had for all practical purposes a sharp leading edge. The other wing, the heating data of which are reported in reference 12, had a leading-edge sweep of 30°, a span of 25 inches, and employed a hexagonal airfoil section with a leading-edge radius of approximately 1/8 inch. The data for the unswept wing are shown by the circular symbols and represent measurements at seven chordwise stations (35.7-percent span) and a free-stream Mach number range from 1.75 to 2.66. The data for the swept wing, indicated by the square symbols, were obtained at five chordwise stations (42-percent span) and are for a free-stream Mach number range from 2.0 to 3.64. Local Reynolds numbers up to about 20×10^{6} were obtained on the unswept wing; however, on the swept wing the maximum local Reynolds number is only of the order of 10×10^{6}. Even though the stream Reynolds numbers of the two wings were of the same order of magnitude, the local Reynolds numbers for the swept wing were lower as a result of its blunted leading edge.

In the lower Reynolds number range the agreement with theory is generally within ±15 percent, with the majority of the points showing agreement within ±10 percent. At the higher Reynolds numbers the data show agreement within about 20 percent of the theory. The decreasing trend with increasing Reynolds number which is exhibited by the data may be due to the reduction in the ratio of Stanton number to skin-friction coefficient with increasing Reynolds number which was observed by Seiff in reference 13. Since the present data were obtained in the presence of temperature, pressure, and Mach number variations, it is felt that no definite conclusions can be drawn on this point. The important feature of these data is that reasonable agreement with theory is shown for two totally different wing configurations over a rather wide variation of Reynolds number.

Heat Transfer at Angle of Attack and Sweep

Theory and experiment have shown that both leading-edge sweep and bluntness have the general effect of reducing the local rates of heat transfer to the leading edge. In order to study the effects of sweep as well as the effects of angle of attack on the heating of the surfaces of a wing with a blunt leading edge, tests have been conducted in the Langley 11-inch hypersonic tunnel at a Mach number of 6.86 on a slab wing with a semicircular leading edge. In figure 4 are presented some of the results of these tests which show the effect of sweep on the wing-surface heat transfer at zero angle of attack. The data are for a stream Reynolds number of 0.151×10^6 based on the 3/4-inch diameter of the leading edge and correspond to an actual flight altitude of about 76,000 feet at the test Mach number. The data are plotted as Stanton number based on free-stream conditions as a function of s/d, where s is the surface length, measured in a streamwise plane, from the stagnation point to the measurement station, and d is the leading-edge diameter. Data are presented for sweep angles of 0°, 40°, and 60°. In order to show the relative magnitude between the leading edge and surface heating, theory curves for the cylinder are presented for each of the sweep angles; however, since heat transfer to swept cylinders has been presented in the past (refs. 3 and 4), attention is called to the flat-plate portion of the wing. The measured data for all sweep angles lie essentially on one line and show no effect of sweep angle on the surface heating. The theory curve included in figure 4 was computed by using the Van Driest theory for laminar boundary layer (ref. 9) with local conditions determined by taking into account the losses through the normal shock and by assuming the local static pressure on the surface equal to stream static pressure. In the vicinity of the leading edge ($s/d = 1$ to 2) the data are considerably higher than this theoretical calculation, mainly as a result of the higher-than-stream pressures which are known to exist there; but as the pressure falls off with distance downstream, the data show better agreement with the theory. Measured pressure distributions on the wing showed that, just behind the leading edge, the surface pressures were several times higher than stream static pressure and dropped off to about twice the stream static pressure at values of s/d of about 7.5. The theory curve shown in the figure was actually computed for zero sweep, but it is interesting to note that calculations for the 40° and 60° sweep angles showed only small variation from the 0° sweep condition.

Measured surface heat transfer at angle of attack for the 40° swept slab wing is presented in figure 5 as being generally typical of the effect of angle of attack on the wing surface heating for all the sweep angles. Here, again the data are plotted as Stanton number based on free-stream conditions as a function of streamwise s/d. Measurements are presented of surface heating of the lower or windward surface with the wing at angles of attack of 5° and 7.5° and of the upper or leeward surface with the wing at angles of attack of 5° and 10°. The curve of zero-angle-of-attack data, of course, applies to both upper and lower surfaces of the wing. Increasing the angle of attack increases the heating of the lower surface and decreases the heating of the upper surface. The variation of heating with angle of attack is essentially linear for both the upper and lower surfaces; however, the rate of increase of heating on the lower surface is about twice the rate of decrease of heating on the upper surface.

Heating of Wing Leading Edge in Vicinity of Wing-Body Juncture

The heat transfer to swept cylinders in an undisturbed flow field has been studied over a wide range of conditions and, in general, the experimental data agree well with existing theories. A rocket-powered-model flight test was recently conducted for the purpose of measuring the heat transfer to leading edges in the vicinity of the wing-body juncture at relatively high Reynolds numbers. A sketch is shown in figure 6 of the rocket-powered model which carried small stubs representing only the leading-edge portions of 0° and 75° swept wings. The stubs had cylindrical leading edges of 3/4-inch diameter which became tangent to flat surfaces inclined at 4.3° to the chord plane. Thermocouple measurements were made in a direction perpendicular to the leading edge of each stub at locations shown in the cross-section sketch. Four measurement points were obtained on the 0° swept stub; however, on the 75° swept stub, the most rearward thermocouple failed to operate properly and, hence, measurements at only the three forward locations were obtained.

Measured heating data for the two leading edges at a stream Mach number of 3.12 and a stream Reynolds number of 18.7×10^{6} per foot are presented in figure 7 along with appropriate theoretical curves for comparison. The data are presented as the nondimensional Stanton number evaluated at free-stream conditions as a function of s/d where s is the surface length, measured in a plane normal to the leading edge, from the stagnation point to the measurement station and d is the leading-edge diameter. The measured data on the cylindrical portion of the leading edge for both the 0° and 75° swept stubs are considerably higher than that predicted by the laminar leading-edge theory in reference 4. On the 0° swept leading edge, the heating at the stagnation point is of the order of 2 times that which is predicted by laminar theory, and on the 75° swept leading edge, the heating is of the order of 4.5 times as great. Since there is such a large difference between the measured values and the laminar theory for the cylindrical portion of the leading edges, it was felt that the flow over this portion must be turbulent. This was verified for the 0° swept leading edge by integrating the measured values over the cylindrical portion to obtain the average heat transfer. This average value $\left(N_{St,\infty} = 34.96 \times 10^{-4}\right)$ was found to agree very well with the theoretical average $\left(N_{St,\infty} = 34.85 \times 10^{-4}\right)$ computed from the turbulent theory in reference 3. Comparison of the measured data on the flat portion following the cylinder with the Van Driest turbulent flat-plate theory (refs. 10 and 11) also shows good agreement and tends to establish further that the flow on the 0° swept leading edge was turbulent. The analysis for turbulent heating on the 75° swept leading edge was not carried out because of the absence of information on the local flow conditions at this sweep angle and Mach number.

The reason for turbulent flow over the 0° swept leading edge cannot be deduced from the measurements made on the model. Since the measurements were made at a distance of only $1\frac{1}{4}$ inches from the body, the heating to the leading edge could have been influenced by conditions existing in the body boundary layer. The Reynolds number for the body at the leading-edge body juncture was approximately 66×10^{6} and the body boundary-layer thickness was estimated to be of the order of 1/2 inch. With these conditions prevailing, it is possible that interaction between the bow shock ahead of the leading edge and the thick turbulent boundary layer of the body could have increased the heating of the leading edge to the turbulent level. Although exact simulation of the flow field at the wing-body juncture may not have been provided by the short leading-edge stubs, these data indicate that rates of heat transfer much higher than laminar rates can be experienced on leading edges in the region of the wing-body juncture, and that, further, more complete investigations are needed to understand this phenomenon.

Transpiration Cooling at High Stagnation Temperatures

At lower flight speeds the problem of convective heating of an aircraft surface can generally be handled by designing the surface as a heat sink and allowing the skin to attain some allowable equilibrium temperature. As flight speeds increase, the idea of some type of cooling for the hotter surfaces appears more attractive. One type of cooling which shows promise is transpiration cooling, in which the coolant passes from the interior of the aircraft through a porous skin into the hot boundary layer. Experimental data on transpiration cooling at relatively low heating potentials have been reported in references 14, 15, and 16, and have shown appreciable reduction in the convective heating with transpiration cooling. In order to investigate the effectiveness of transpiration cooling at higher heating potentials, exploratory tests have been conducted at high stagnation temperatures in the Langley preflight jet of the Pilotless Aircraft Research Station at Wallops Island, Va. The model employed for the tests was a blunted wedge of 20° half-angle which had a porous stainless-steel segment inserted in one surface through which coolant was ejected. Tests were conducted at a nominal free-stream dynamic pressure of 5,000 pounds per square foot and a free-stream Mach number of 2.0; however, the local-surface Mach number on the wedge was of the order of 1.2. Results of these tests showing the effect of transpiration cooling on the wedge-surface temperature for both nitrogen and helium coolants are presented in figure 8. The ordinate is the nondimensional wall-temperature parameter $\frac{T_W - T_c}{T_{AW,O} - T_c}$ where T_W is the porous-wall temperature, T_c is the coolant temperature, and $T_{AW,O}$ is the boundary-layer recovery temperature for zero coolant flow rate. The abscissa is the coolant flow rate in pounds per square foot per minute. Such a presentation of cooling data shows directly the reduction in wall temperature which can be achieved for a given coolant flow rate. Data are presented for stagnation temperatures in the range from 2,355° R to 3,370° R for nitrogen coolant and from 1,755° R to 3,195° R for helium coolant. For the various tests, ratios of wall temperature to local temperature ranged from 0.2 to 0.5 and the local Reynolds numbers ranged from 0.6×10^6 to 8.2×10^6. It might also be mentioned that the present data were obtained for average operating temperatures of the porous wall in the range from about 200° F to 1,300° F. Comparison of the nitrogen and helium data in figure 8 shows that the helium performs as a much more effective coolant than nitrogen, as would be expected because of the higher specific heat of the helium. For example, in order to maintain the skin at a temperature of about 0.3 of the uncooled value, approximately 10 pounds of nitrogen per square foot per minute would be required as compared with 2 pounds per square foot per minute for helium. It might be added that this ratio of 5 to 1 in required coolant flow rates is roughly the same as the ratio of the specific heat of helium to that of nitrogen.

In order to show how these high-temperature data compare with theory and other low-temperature data, as well as to show the effect of cooling on the heat transfer, the data were reduced to heat-transfer coefficients and are presented in dimensionless form in figure 9. The ordinate is the ratio of the Stanton number with cooling to the Stanton number for zero cooling, and the abscissa is the injection parameter F (the ratio of coolant weight flow to local stream weight flow) divided by the Stanton number for zero cooling. In the reduction of the measured data to Stanton number, the recovery temperature with coolant flow was evaluated by using recovery-factor values computed from the theory of reference 17, which gives the variation of recovery factor with Mach number, coolant flow rate, and Reynolds number. Included for comparison are the theoretical curve applicable to nitrogen from the theory in reference 18 for $M = 1.0$ and a ratio of wall temperature to local temperature of 0.2, and the experimental data in reference 14 for the transpiration cooling of an 8° cone with both nitrogen and helium coolants at a stagnation temperature of 1,060° R. The experimental wedge data for nitrogen coolant form a band which has the same trend as the theory and shows slightly greater cooling effectiveness. The present data for the nitrogen also show good agreement with the trend established by the 8° cone data. The present helium data do not show as good agreement with the prior cone data with helium coolant as is observed in the case with nitrogen. Since both sets of data for the helium coolant are for small flow rates, the discrepancy between the two sets of data may be due to the lesser accuracy in determining the flow rates in the lower range of flow rates.

With this type of correlation, the comparison of the effectiveness of helium with nitrogen as transpiration coolants is about in the same relation as noted in figure 8. The theory which is strictly derived for transpiration of air to air but is applicable also to nitrogen because of the similarity of the physical characteristics may be said to apply equally as well for either high or low heating potentials.

CONCLUSIONS

The pertinent conclusions which may be drawn from the foregoing compilation of wing-heating data may be summarized briefly as follows:

1. Measured heat transfer to wing surfaces exposed to high-heat fluxes have, in general, shown good agreement with theory. Theory has also been shown to be adequate in predicting the heat transfer to wing surfaces over a wide range of Reynolds number.

2. Heat-transfer measurements on a blunted slab wing at a Mach number of 6.86 showed that the surface heating at zero angle of attack was essentially the same for sweep angles of 0°, 40°, and 60°. Increasing the angle of attack of the slab wing increased the heating of the windward surface and decreased the heating of the leeward surface.

3. Measurements at high stream Reynolds numbers have shown that rates of heat transfer much higher than laminar rates can be experienced on leading edges in the region of the wing-body juncture.

4. Theoretical predictions of the effectiveness of nitrogen as a transpiration coolant have been shown to apply equally as well for conditions of either high or low heating potentials. Measurements have shown that helium is about five times more effective as a transpiration coolant than nitrogen.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., March 6, 1957.

REFERENCES

1. Reshotko, Eli, and Cohen, Clarence B.: Heat Transfer at the Forward Stagnation Point of Blunt Bodies. NACA TN 3513, 1955.

2. Beckwith, Ivan E.: Theoretical Investigation of Laminar Heat Transfer on Yawed Infinite Cylinders in Supersonic Flow and a Comparison With Experimental Data. NACA RM L55F09, 1955.

3. Beckwith, Ivan E., and Gallagher, James J.: Experimental Investigation of the Effect of Boundary-Layer Transition on the Average Heat Transfer to a Yawed Cylinder in Supersonic Flow. NACA RM L56E09, 1956.

4. Goodwin, Glen, Creager, Marcus O., and Winkler, Ernest L.: Investigation of Local Heat-Transfer and Pressure Drag Characteristics of a Yawed Circular Cylinder at Supersonic Speeds. NACA RM A55H31, 1956.

5. Bland, William M., Jr., and Bressette, Walter E.: Some Effects of Heat Transfer at Mach Number 2.0 at Stagnation Temperatures Between 2,310° and 3,500° R on a Magnesium Fin With Several Leading-Edge Modifications. NACA RM L57C14, 1957.

6. Carter, Howard S.: Heat Transfer on the Lifting Surfaces of a 60° Delta Wing at Angle of Attack for Mach Number 1.98. NACA RM L56C23, 1956.

7. Bartlett, G. E., Hilton, J. H., Vidal, R. J., and Woolard H. W.: Experimental Investigations of Heat Transfer on a 10-Percent Double-Wedge Airfoil at Mach No. 2.0. Rep. No. CAL/CM-832 (Contract NOrd-14523), Cornell Aero. Lab., Inc., Jan. 1955.

8. Sterbutzel, Gerald A., and Kajencki, Stephen S.: Experimental Investigation of Heat Transfer From Aerodynamic Bodies in Supersonic Flow. Rep. No. AF-473-A-9 (Contract W33-038-ac-16701), Cornell Aero. Lab., Inc., Apr. 1950.

9. Van Driest, E. R.: Investigation of Laminar Boundary Layer in Compressible Fluids Using the Crocco Method. NACA TN 2597, 1952.

10. Van Driest, E. R.: The Turbulent Boundary Layer for Compressible Fluids on a Flat Plate With Heat Transfer. Rep. No. AL-997, North American Aviation, Inc., Jan. 27, 1950.

11. Van Driest, E. R.: The Turbulent Boundary Layer With Variable Prandtl Number. Rep. No. AL-1914, North American Aviation, Inc., Apr. 2, 1954.

12. Swanson, Andrew G., and Rumsey, Charles B.: Aerodynamic Heating of a Wing As Determined From a Free-Flight Rocket-Model Test to Mach Number 3.64. NACA RM L56F11a, 1956.

13. Seiff, Alvin: Examination of the Existing Data on the Heat Transfer of Turbulent Boundary Layers at Supersonic Speeds From the Point of View of Reynolds Analogy. NACA TN 3284, 1954.

14. Chauvin, Leo T., and Carter, Howard S.: Exploratory Tests of Transpiration Cooling on a Porous 8° Cone at $M = 2.05$ Using Nitrogen Gas, Helium Gas, and Water as the Coolants. NACA RM L55C29, 1955.

15. Leadon, B. M., and Scott, C. J.: Measurement of Recovery Factors and Heat Transfer Coefficients With Transpiration Cooling in a Turbulent Boundary Layer at $M = 3$ Using Air and Helium as Coolants. Res. Rep. No. 126, Univ. of Minnesota Inst. Tech., Dept. Aero. Eng. (Contract AF 18(600)-1226), Feb. 1956.

16. Rubesin, Morris W., Pappas, Constantine C., and Okuno, Arthur F.: The Effect of Fluid Injection on the Compressible Turbulent Boundary Layer - Preliminary Tests on Transpiration Cooling of a Flat Plate at $M = 2.7$ With Air as the Injected Gas. NACA RM A55I19, 1955.

17. Rubesin, Morris W.: An Analytical Estimation of the Effect of Transpiration Cooling on the Heat-Transfer and Skin-Friction Characteristics of a Compressible, Turbulent Boundary Layer. NACA TN 3341, 1954.

18. Dorrance, William H., and Dore, Frank J.: The Effect of Mass Transfer on the Compressible Turbulent Boundary Layer Skin Friction and Heat Transfer. Rep. ZA-7-013, Convair, Aug. 5, 1954.

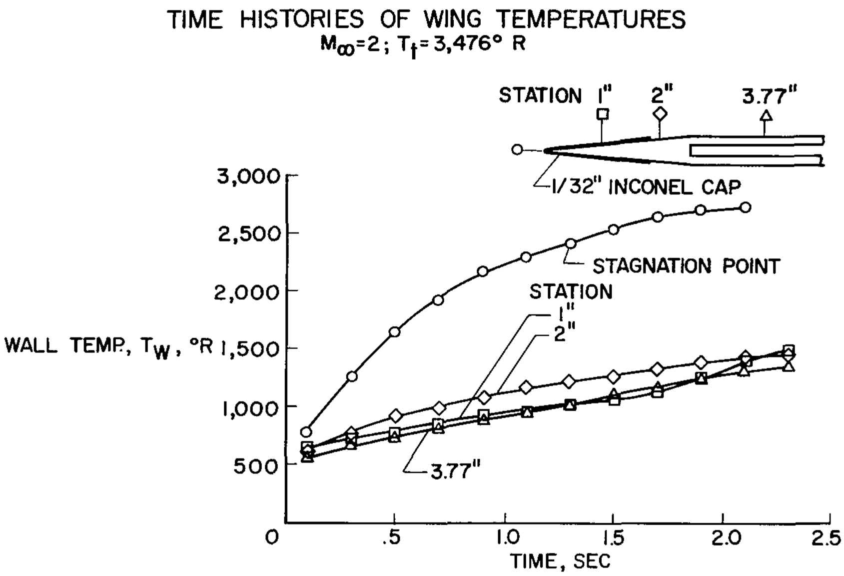

Figure 1

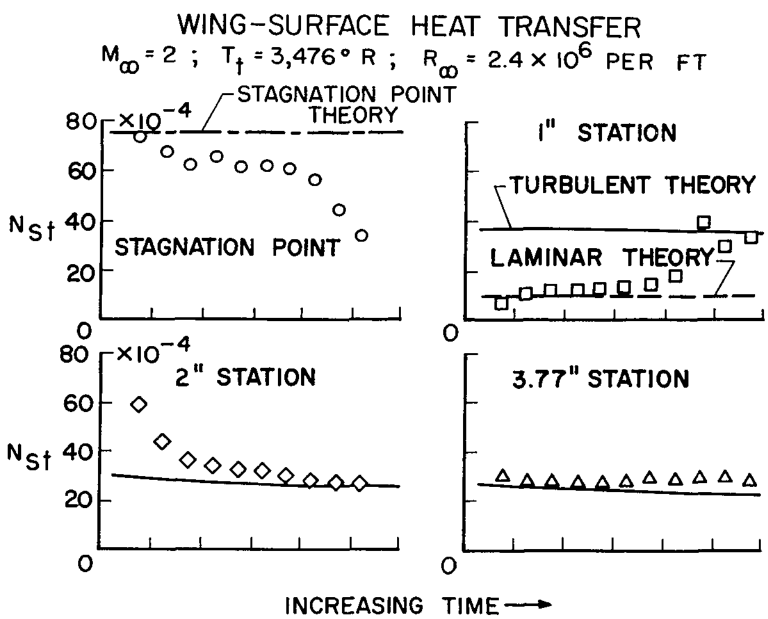

Figure 2

COMPARISON OF EXPERIMENT AND THEORY FOR
LARGE-SCALE HEAT-TRANSFER TESTS

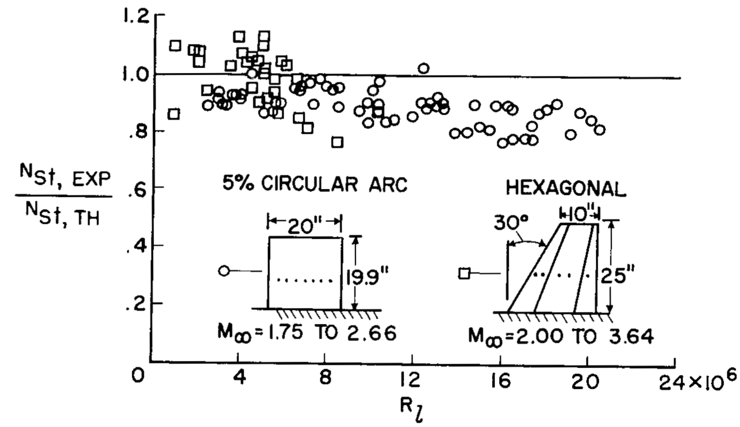

Figure 3

EFFECT OF SWEEP ON HEAT TRANSFER TO A BLUNT L.E. WING
$\alpha = 0°$; $M = 6.86$; $R_d = 0.151 \times 10^6$

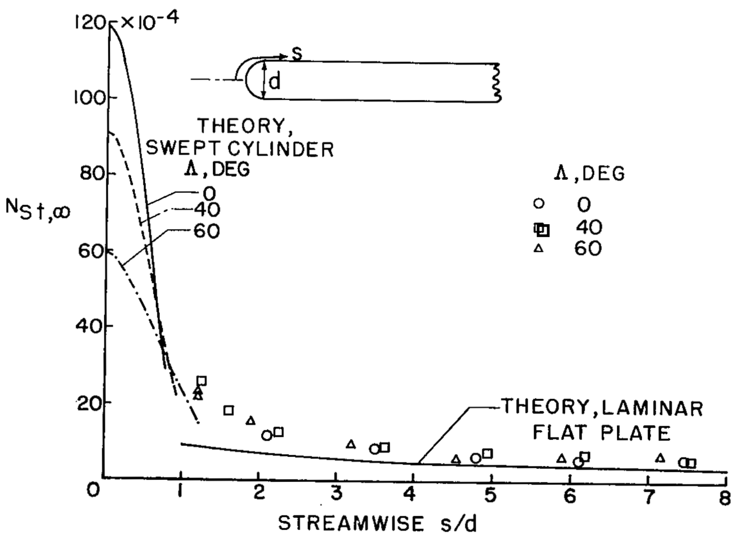

Figure 4

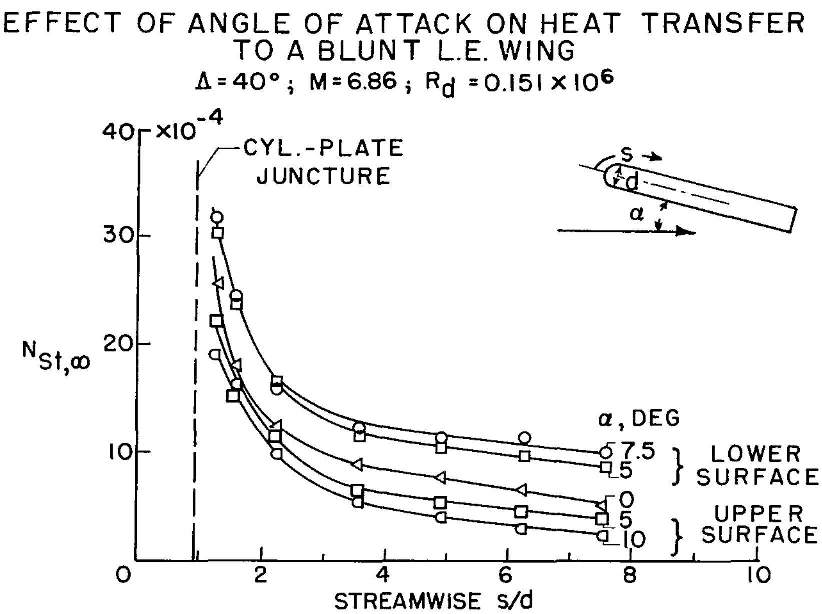

Figure 5

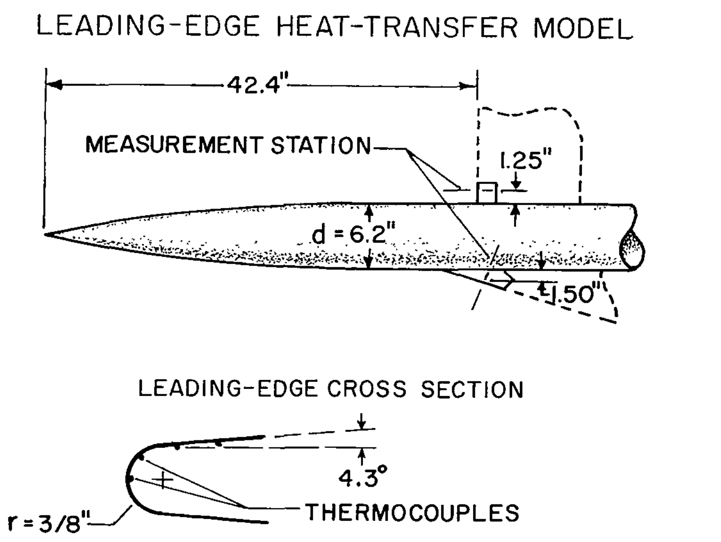

Figure 6

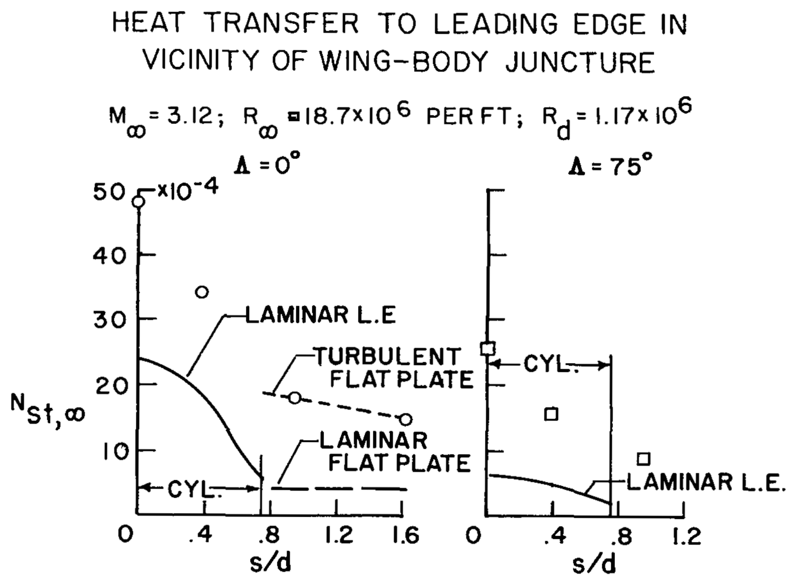

Figure 7

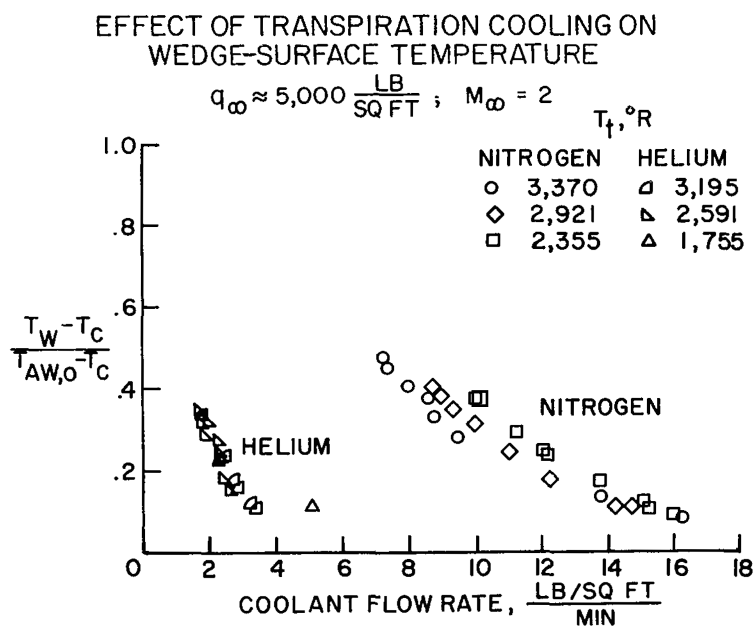

Figure 8

EFFECT OF TRANSPIRATION COOLING ON HEAT TRANSFER

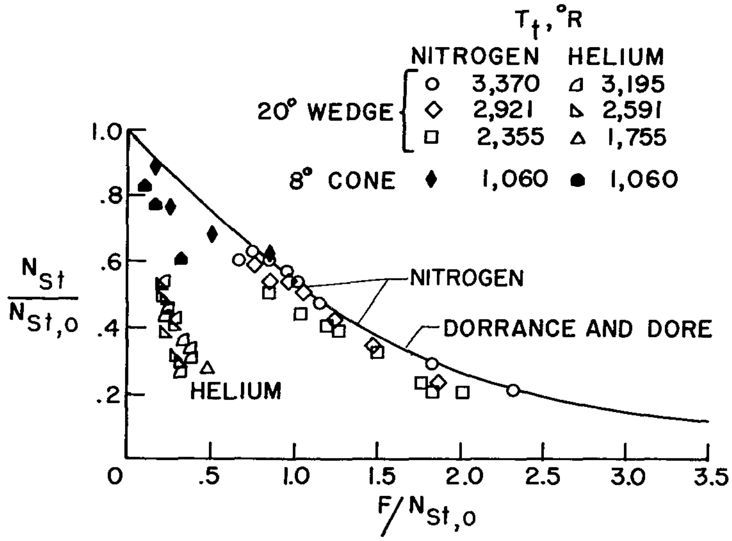

Figure 9