Surface Temperature Effect on Subsonic Stall

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This paper presents the results of an analytical and experimental study of boundary-layer flow over an aerodynamic surface rejecting heat to a cool environment. Analytical studies involving both laminar and turbulent boundary layers revealed that a surface to freestream temperature ratio, T_w/T_∞, greater than unity tended to destabilize the boundary layer, hastening transition and separation. The boundary-layer displacement thickness was found to increase in direct proportion to T_w/T_∞ while the momentum thickness remained unaffected. The effect of heat transfer was to accentuate the effect of an adverse pressure gradient. Wind-tunnel tests of 0012-64 NACA airfoil showed that the stall angle was significantly reduced while drag tended to increase for T_w/T_∞ up to 2.2.

Nomenclature

a = speed of sound
b_0 = half width of the wake
C = chord length, Chapman viscosity coefficient
c_f = skin-friction coefficient
C_p = specific heat at constant pressure
H = shape parameter, enthalpy
M = Mach number
P = pressure
Pr = Prandtl number
Re = Reynolds number
T = absolute temperature
U = velocity
u, v = streamwise and normal boundary-layer velocity components, respectively
X, Y = streamwise and normal spatial coordinates, stretched coordinates
γ = ratio of specific heats
δ, δ^*, θ = velocity, displacement and momentum thicknesses, respectively
μ, ν = coefficients of absolute and kinematic viscosity, respectively
ρ = density
τ = shearing stress
ψ = stream function

Superscripts

$'$ = differentiation with respect to x

Subscripts

t = total or stagnated conditions
w = conditions at wall
1 = conditions at outer edge of boundary layer
∞ = freestream conditions

Introduction

WITH the advent of maneuverable re-entry vehicles such as the proposed space shuttle, a new problem involving heat transfer from aerodynamic surfaces has evolved. During entry into the atmosphere the craft is to be subjected to large heat loads which depend upon the geometric configuration as well as the attitude of the vehicle during re-entry. Some of this heat load will inevitably soak into the structure and aerodynamic surfaces. After the re-entry phase of the flight, the vehicle will slow to subsonic speeds and maneuver for a conventional aircraft landing. During the maneuvering and landing, the heated aerodynamic surfaces will be injecting heat into the boundary layer.

The inverse problem, that of heat transfer from the flow to the wall, has received considerable attention in the past, particularly from high Mach number flows.[1–3] These studies have investigated either the case of adiabatic flow in which the wall assumes an equilibrium temperature, or the case of constant temperature wall. In each case the flow adjacent to the surface is at a higher temperature than the surface. The resultant cooling of the boundary layer tends to have a stabilizing effect on the flow and the point of separation is moved downstream when compared to the zero heat-transfer case. For the problem considered here, heat is transferred from the surface to a subsonic boundary layer. Hence, the flow is at a lower temperature than the surface and the effect is to destabilize the boundary layer. The objective of this investigation was to determine the effects of heat transfer on the boundary layer and aerodynamic characteristics of an aerodynamic surface rejecting heat to the atmosphere. This problem was originally suggested by a high angle of attack re-entry of an aerodynamic space vehicle. The study was conducted initially on an analytical basis to estimate the magnitude of the effects of heat addition to a subsonic boundary layer. The analytical study suggested several adverse effects, chiefly in the area of destabilization of the boundary layer so that transition and separation were hastened. After examination of both laminar and turbulent flows in adverse pressure gradients, flows over cylinders and airfoils were studied. Experiments were then performed to confirm the analytical results.

Analytical Studies

Laminar Flow

There exists a considerable body of literature concerning the behavior of laminar flow over a heated body. The solutions for this problem are limited primarily to the case of linear retarded flow, although others have been studied. The results of a literature survey indicated the following general conclusions: heating the surface increases the boundary-layer

Presented as Paper 72-960 at the AIAA 2nd Atmospheric Flight Mechanics Conference, Palo Alto, Calif., September 11–13, 1972; submitted September 26, 1972; revision received April 23, 1973.

Index categories: Boundary Layers and Convective Heat Transfer—Turbulent; Entry Deceleration Systems and Flight Mechanics (e.g., Parachutes).

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thickness, increases the direct effect of an adverse pressure gradient, decreases skin friction in an adverse pressure gradient, and moves the point of laminar flow separation upstream. These conclusions have been reached on the basis of a Karman-Pohlhausen type of analysis by Morduchow and Grape,[4] on the basis of similarity solutions by Cohen and Reshotko[1] and Li and Nagamatsu,[5] and by the use of the Stewartson-Illingworth transformation.[6]

In considering the effects of temperature on boundary-layer properties, it is instructive to examine the definitions of the various boundary-layer thicknesses. Momentum thickness is defined as

$$\theta = \int_0^\infty \frac{\rho u}{\rho_1 u_1}\left(1 - \frac{u}{u_1}\right) dy \tag{1}$$

Applying the Dorodnitsyn type of transformation

$$\partial\eta/\partial y = \rho/\rho_1 \tag{2}$$

the momentum thickness becomes

$$\theta = \int_0^\infty \frac{u}{u_1}\left(1 - \frac{u}{u_1}\right) d\eta \tag{3}$$

Thus, the momentum thickness is independent of the temperature in the boundary layer. This is illustrated in Fig. 1 which shows the momentum thickness as a function of wall temperature for both laminar and turbulent flows. The numerical results are from calculations based on the boundary-layer solution technique of Cebeci, Smith, and Wang.[7]

Now consider the definition of displacement thickness

$$\delta^* = \int_0^\infty \left(1 - \frac{\rho u}{\rho_1 u_1}\right) dy \tag{4}$$

Again applying the Dorodnitsyn transformation, and assuming constant pressure through the boundary layer, the displacement thickness can be written as

$$\delta^* = \int_0^\infty \left(\frac{T}{T_1} - \frac{u}{u_1}\right) d\eta \tag{5}$$

Now assume the temperature profile through the boundary layer can be defined by a form of Croccos' relation between velocities and enthalpies in the boundary layer

$$T/T_1 = (T_w/T_1) + (u/u_1)[1 - (T_w/T_1)] \tag{6}$$

which has been simplified by assuming $Pr = 1.0$ and letting $M_1 \to 0$. Substituting this temperature profile into the integral, Eq. (5) reduces to

$$\delta^* = \frac{T_w}{T_1}\int_0^\infty \left(1 - \frac{u}{u_1}\right) d\eta \tag{7}$$

Thus displacement thickness is directly proportional to the temperature of the wall. Numerical results agree with this conclusion, and that the momentum thickness remains unaffected by the temperature ratio. Results for displacement thickness are presented in Fig. 1. Boundary-layer growth of this magnitude could alter the pressure distribution over an airfoil and thus affect the lift and drag characteristics.

The effect of heat transfer to the boundary layer on the skin friction may now be examined by considering the integral form of the momentum boundary-layer equation

$$c_f/2 = \frac{\tau_w}{\rho_1 u_1^2} = \frac{d\theta}{dx} + \frac{\theta}{u_1}\frac{du_1}{dx}\left[2 + \frac{\delta}{\theta} - M_1^2\right] \tag{8}$$

The momentum thickness remains essentially unaltered by heat addition while the displacement thickness grows approximately as

$$\delta^* = \delta_0^*(T_w/T_1) \tag{9}$$

where δ_0^* = the displacement thickness of an unheated flow. Thus, at a given position and edge condition, we can form the change of C_f with temperature ratio T_w/T_1

$$\frac{\partial c_f/2}{\partial(T_w/T_1)} = \frac{du_1}{dx}\frac{\delta_0^*}{u_1} \tag{10}$$

Now in adverse pressure gradients du_1/dx is negative, therefore the skin friction is reduced by increasing the temperature ration. Figure 2 presents the effect of T_w/T_∞ on skin friction for a linearly retarded flow.

It now remains only to show that for laminar flow the effect of the pressure gradient is enhanced by heat addition to the boundary layer. This can be shown conveniently for the case where $Pr = 1.0$ and $M \to 0.0$ by following the work of Stewartson.[6] It is accomplished by employing the stream function, the Howarth-Dorodnitsyn transformation, as well as other stretching coordinates. This derivation is presented in some detail in Ref. 8. The result is listed below.

$$\frac{\partial\psi}{\partial Y}\frac{\partial^2\psi}{\partial X\partial Y} - \frac{\partial\psi}{\partial X}\frac{\partial\psi^2}{\partial Y^2} = V_1\frac{dV_1}{dX}\cdot S + C(X)\frac{T_\infty}{\rho_\infty}\frac{\partial^3\psi}{\partial Y^3} \tag{11}$$

where

$$V_1(X) = a_\infty V_1/a_1 = a_\infty M_1(X) \tag{12}$$

The quantity S, which may be a function of X, can be shown to be of the following form

$$S = \frac{T_w(X)}{T_\infty[1 + \frac{1}{2}(\gamma - 1)M_\infty^2]} \quad \text{at } Y = 0 \tag{13}$$

$$S \to 1 \quad \text{as} \quad Y \to \infty$$

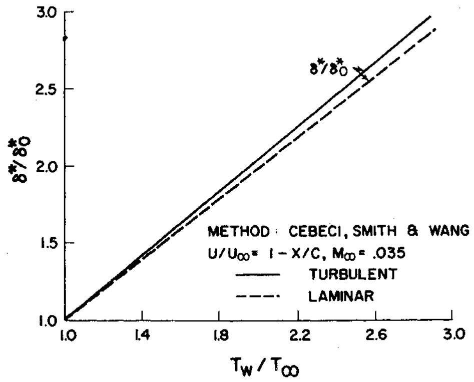

Fig. 1 Effect of surface temperature on boundary-layer thickness.

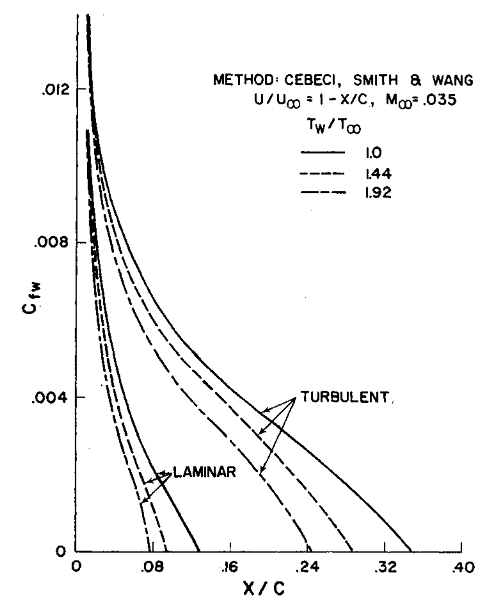

Fig. 2 Effect of surface temperature on skin friction.

Thus, S varies from one at the outer edge of the boundary layer to a value of T_w/T_∞ at the wall for the case $M \to 0$ and $Pr = 1.0$. From Eq. (11) it can be seen that a heated wall will tend to increase the magnitude of the pressure gradient term in the momentum equation. Since adverse pressure gradient directly affects how long the flow is able to remain attached to the surface, it can be expected that heating the surface will cause the separation point to move upstream.

Previous investigations of the effect of wall temperature on the separation point have been limited to laminar flows of the simple retarded or wedge flow types. Li and Nagamatsu[5] extended the work of Hartree[9] on boundary-layer equations which reduce to the similar solutions of Faulkner-Skan[10] for flow past a wedge. They generalized the equations to include the case of constant temperature wall which is different from the freestream. The results of their studies indicated that separation could be expected to occur for a relatively weaker pressure gradient for flow over a heated body.

Curle[11] has developed an approximate method for computing the compressible laminar boundary layer with pressure gradient and uniform wall temperature based on the incompressible work of Thwaites.[12] Thwaites had examined all of the known solutions of laminar boundary-layer flows and recognized that they could be represented in general by two parameters. Curle extended this work to cover compressible flows by introducing an integral of the energy equation based on an approximate temperature profile. A transformation of the normal coordinate partially reduces the momentum equation to an incompressible form. With the approximate temperature profile, the integration of the momentum equation is reduced to a form requiring only two quadratures. Curle's method has been applied to the problem of linearly retarded flow and the results indicate a forward movement of separation for the case of the heated wall.

Morduchow and Grape[4] present a theoretical analysis of the effect of pressure gradient and wall temperature on laminar boundary characteristics and, in particular, on the separation point in an adverse pressure gradient. A simple method of locating the separation point in a compressible flow with a heat transfer is developed. Assumptions made are that the Prandtl number is unity and the viscosity is proportional to temperature by a factor derived from the Sutherland relation. The usual transformation of the normal coordinate is performed and the velocity and stagnation enthalpy profiles are represented, respectively, by sixth and seventh degree polynomials. Concerning the point of flow separation, Morduchow and Grape show, by differentiating the momentum partial differential equation, that at the point of separation the additional boundary condition at the wall $(\partial^4 u/\partial y^4)_w = 0$ must be satisfied. A seventh-degree velocity profile is chosen to satisfy this new boundary condition in addition to those satisfied by the sixth-degree profiles. From this, Morduchow and Grape developed an expression for the point of separation which requires a single quadrature.

Illingworth[13] has developed an approximate method to treat laminary boundary layers with pressure gradient and large wall to freestream temperature differences. Applying von Mises's transformation, he writes the momentum and energy boundary-layer equations in terms of the coordinate x and the stream function ψ. Using several approximations, Illingworth finds similar solutions of the boundary-layer equations for flows defined by a power series $U \alpha X^n$ and flows of the linearly retarded type. For a linearly retarded flow, at zero Mach number Illingworth concluded that the point of laminar flow separation is given by

$$(x/c)_{sep} = 0.12[T_\infty/T_w]^{1/2} \qquad (14$$

Two other approaches to the calculations of the laminar boundary layer with a heated wall are the integral method of Walz[14] and the finite-difference, eddy viscosity method of Cebeci, Smith, and Wang.[7] Since these methods are also

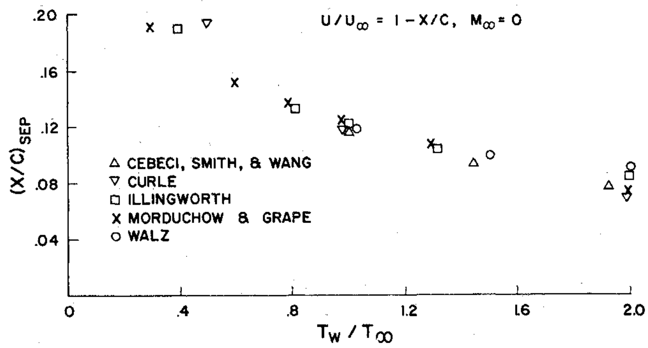

Fig. 3 Effect of surface temperature on laminar separation.

applicable to turbulent flow, they are of particular interest and will be considered in greater detail later. It is sufficient to mention here that both of these methods predict the same trend as the previously cited methods on the forward movement of the separation point over a heated surface.

For a comparison of the various procedures reviewed, Fig. 3 presents results for a linearly retarded flow at zero Mach number and for a range of surface temperatures. It can be seen from the results that all of the methods investigated predict a forward movement of the point of laminar separation for the case of the heated wall.

Turbulent Flow

Consideration of the separation point of turbulent flow over a heated surface has received little attention in the literature. Some knowledge of this phenomenon, however, can be obtained by investigating methods that solve the turbulent boundary-layer equations with heat transfer. Two methods, completely different in approach were used in this study. They are the integral relation method of Walz[14] and the finite-difference, eddy conductivity, eddy viscosity method of Cebeci, Smith, and Wang.[7] Both of these techniques were rated in the upper one-third of the 1968 Stanford Conference[15] on turbulent boundary layers.

The method of Walz is based on the integral relations for momentum and energy.

$$\frac{d\theta}{dx} + \frac{\theta}{u_1}\frac{du_1}{dx}\left[2 + \frac{\delta^*}{\theta} - M_1^2\right] - \frac{\tau_w}{\rho_1 u_1^2} = 0 \qquad (15)$$

$$\frac{d\delta_3}{dx} + \frac{\delta_3}{u_1}\frac{du_1}{dx}\left[3 + 2\frac{\delta_4}{\delta_3} - M_1^2\right] - \frac{2}{\rho_1 u_1^3}\int_0^{u_1} \tau du = 0 \qquad (16)$$

where

$$\delta_3 = \int_0^\infty \frac{\rho u}{\rho_1 u_1}\left[1 - \left(\frac{u}{u_1}\right)^2\right] dy \qquad (17)$$

$$\delta_4 = \int_0^\infty \frac{\rho u}{\rho_1 u_1}\left[\frac{\rho_1}{\rho} - 1\right] dy \qquad (18)$$

The last terms in Eqs. (15) and (16) are, respectively, the wall shearing stress and the dissipation integral. These quantities are determined empirically. Walz then reduces Eqs. (15) and (16) to two first order, ordinary differential equations

$$(dZ/dx) + Z(1/u_1)(du_1/dx)F_1 - F_2 = 0 \qquad (19)$$

and

$$(dH_{32}/dx) + H_{32}(1/u_1)(du_1/dx)F_3 - (F_4/Z) = 0 \qquad (20)$$

where

$$Z = \theta R_\theta^n; \qquad H_{32} = \delta_3/\theta; \qquad F_i = F_i(H_{32}, M_1, \theta) \quad (21)$$

These equations are then amenable to numerical integration by a finite-difference technique.

The approach of Cebeci, Smith, and Wang[7] is to solve directly the turbulent boundary-layer equations for continuity, momentum, and energy

$$(\partial/\partial x)(\rho u + \langle\rho' u'\rangle) + (\partial/\partial y)(\rho v + \langle\rho' v'\rangle) = 0 \quad (22)$$

$$\rho u \frac{\partial u}{\partial x}(\rho v + \langle\rho' v'\rangle)\frac{\partial u}{\partial y} = -\frac{dp}{dx} + \frac{\partial}{\partial y}\left[\mu\frac{\partial u}{\partial y} - \langle\rho v' u'\rangle\right] \quad (23)$$

$$\rho u \frac{\partial H}{\partial x} + (\rho c + \langle\rho' v'\rangle)\frac{\partial H}{\partial y} = \frac{\partial}{\partial y}\left[\frac{\mu}{Pr}\frac{\partial H}{\partial y} - \langle\rho v' H'\rangle + \mu\left(1 - \frac{1}{Pr}\right)u\frac{\partial u}{\partial y}\right] \quad (24)$$

This is possible after eddy viscosity and eddy conductivity terms are introduced. These terms are defined as

$$\varepsilon \equiv -\frac{\langle u'v'\rangle}{\dfrac{\partial u}{\partial y}} = \text{eddy viscosity} \quad (25)$$

$$\lambda_t \equiv -\frac{Cp\langle v'H'\rangle}{\dfrac{\partial H}{\partial y}} = \text{eddy conductivity} \quad (26)$$

In the region near the wall, the eddy viscosity is given by

$$\varepsilon_1 = l^2\ \partial u/\partial y \quad (27)$$

where l is Prandtl's mixing length modified to account for the viscous sublayer. In the outer region, a constant eddy viscosity is used

$$\varepsilon_0 = k_2 U_1 \delta^* \quad (28)$$

Figure 4 compares results for the shape parameter $H_{12} = \delta^*/\theta$ from the two methods for a particular Reynolds number. Agreement appears good to a temperature ratio of $T_w/T_\infty = 2.0$. Figure 5 illustrates the effect of wall temperature on the forward movement of separation of a linearly retarded flow. The turbulent separation criterion used in the Walz method for these calculations is actually a lower bound on separation and therefore, for many cases, predicts separation too far down stream($H_{32} = 1.515$). For the method due to Cebeci, Smith, and Wang[7] $C_{fw} = 0$ was employed as the separation criteria. This explains the disagreement in the figure between the two methods at $T_w/T_\infty = 1.0$. However, the magnitude of the forward movement of separation as the temperature is increased is approximately the same for both techniques. The effect of increasing Reynolds number does delay separation, however, the shape of the curves remain similar to Fig. 5.

Flow over a Two-Dimensional Airfoil

Now consider the boundary-layer flow over a two-dimensional airfoil shape. For this purpose, wind-tunnel surface pressure data for a NACA 0012-64 airfoil section is employed

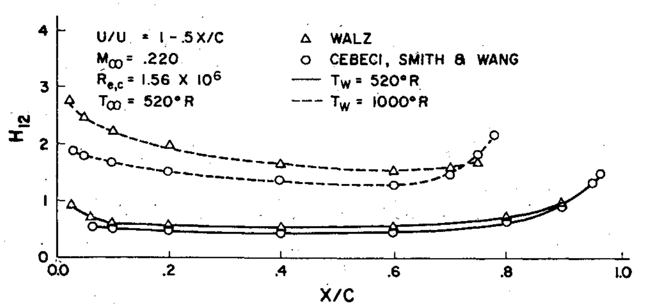

Fig. 4 Shape parameter for a retarded flow over a heated surface.

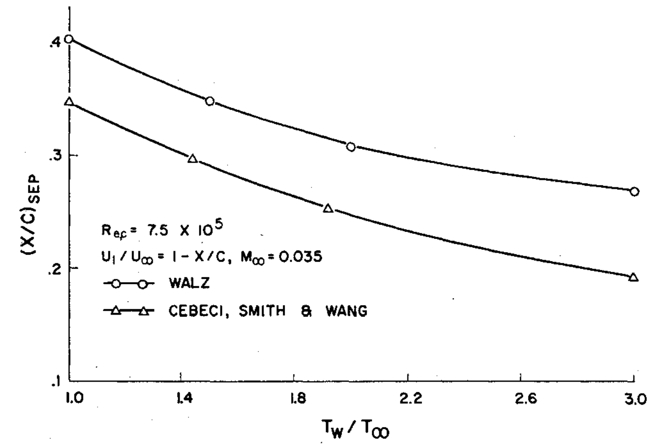

Fig. 5 Effect of surface temperature on turbulent separation.

as input into the numerical methods of Walz and Cebeci, Smith and Wang. Attention is given to the effect of wing temperature on the point of flow separation. Figure 6 shows the velocity distribution over the upper surface of the section at an angle of attack of 12°. As the temperature ratio T_w/T_∞ is increased from one, the separation point gradually moves forward in the region of large velocity gradient at the trailing edge. If the temperature is increased further, the separation point suddenly moves to the region of large gradient just behind the leading edge. Thus, an airfoil which normally stalls at an angle of attack greater than 16° has been forced to stall at 12°. Here, stall is considered to be flow separation just behind the minimum pressure point. Figure 12 presents the stall angle as a function of wing temperature according to the two numerical methods.[7,14]

Experimental Studies

Wind-Tunnel Tests

A model was built and tested in the Texas A & M Univ. 7 ft × 10 ft wind tunnel to obtain experimental data on the effects of wing temperature on the aerodynamic characteristics of an airfoil. Figure 7 is schematic of the experimental setup. The model is an NACA 0012-64 airfoil section with a span of 26 in. and a chord of 1 ft. It was shaped from a single piece of 7075-T6 aluminum. The model was strut mounted through the floor of the tunnel to the six-component

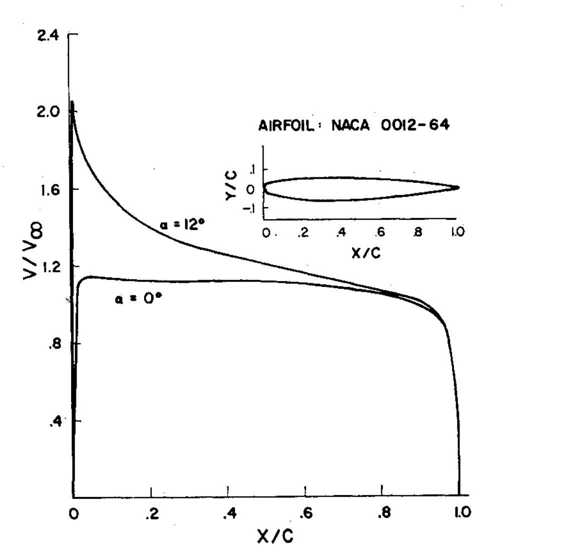

Fig. 6 Velocity distribution over a symmetrical airfoil.

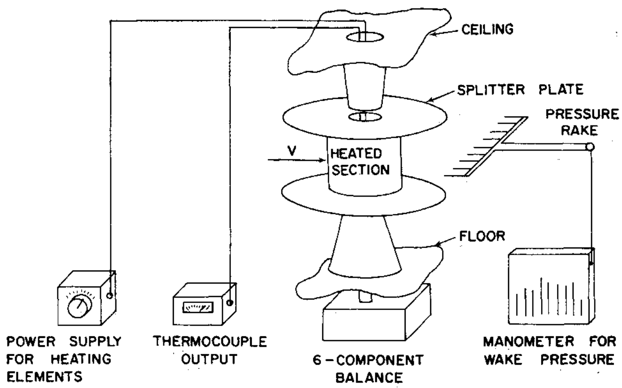

Fig. 7 Schematic of wind-tunnel test.

force/moment measuring balance. Wind forces on the strut were eliminated by a fairing which was geared to the turntable so as to remain at a zero angle of attack when the model was rotated. An image fairing was placed above the model which enclosed the electrical and instrumentation connections. Total pressure and temperatures in the wake of the airfoil were measured by a rake probe. The probe was mounted 0.75 of the chord downstream from the model.

Circular plates two chord lengths in diameter were attached to the ends of the airfoil to improve its two-dimensional characteristics. Tuft studies were conducted on an unheated model to visualize the flow pattern over the wing and to detect the need for air injection to eliminate possible early separation at the roots induced by the end plates. While there was some indication of early separation at the roots, the flow appeared generally to possess good two-dimensional characteristics without blowing.

In fact, overblowing induced recirculation cells over the aft portion of the wing. Since stall was virtually simultaneous over the span without blowing, it was decided that air injection was unnecessary. The end plates increased the apparent aspect ratio of the model from 2.17 to 5.7 while maintaining reasonably two-dimensional flow. Figure 8 shows the model mounted in the wind-tunnel test section during tuft test.

Lift, drag, and stall data were taken at tunnel velocities of 135 fps and 160 fps (Reynolds numbers of 7.5 and 9.25×10^5). These velocities insured that sufficient electrical power was available to heat the model to the desired temperatures. The model was heated by sixteen resistance type cartridge heaters connected in parallel and placed in reamed $\frac{3}{8}$ in. diam holes drilled spanwise through the wing. The heaters maintained a constant resistance of 30 ohms through the temperature range so that heating was regulated by varying the potential across the heaters from 0 to 240 v. Wing temperatures were monitored through measurements by iron-constantan thermocouples embedded near the model surface. The thermocouples were located 1 in. from the leading and trailing edges and at the midchord. Surface temperatures were maintained within 50°F of a nominal temperature setting over the chord length. Uniform wing temperatures were chosen to minimize thermal distortion problems, and to permit comparisons with the theory. Maximum temperatures achieved during the tests were 700°F (1160°R) or approximately 2.2 times the freestream temperature. At these temperatures, the structural strength of the aluminum is reduced to approximately 10% of the room temperature value. The aluminum was chosen for its excellent conductive properties to aid in maintaining a uniform temperature.

Lift force measurements made by the six component balance on which the model was mounted were fed into an analog to digital converter and stored on magnetic tape. Reduction of the data indicated resolution on the lift coefficient C_L was approximately ± 0.025. Figure 9 presents the lift data for Reynolds numbers of 7.5 and 9.25×10^5 for wing-to-freestream temperature ratios T_w/T_∞ of 1.0 and 2.0.

The deviation of the lift slope from the two-dimensional lift slope for this airfoil section is attributed to the finite aspect ratio of the model. Applying a correction factor for the finite aspect ratio results in agreement between the published two-dimensional and the corrected results within experimental error. The apparent effect of wing temperature is to reduce the maximum lift coefficient of the airfoil from 1.0 with no heating to 0.7 for a temperature ratio of 2.0. For the equivalent two-dimensional wing, this corresponds to a variation in C_L from 1.6 to 1.2. Changes in lift due to heating at angles of attack less than 10° were below the resolution of the measuring system. It appears, however, that there may be a slight downward shift of the lift curve with heating. A slight dependence of C_L on Reynolds number is also observed.

Measurements of the drag were made by making total pressure measurements in the wake of the airfoil. The assumption is made that the measurements are taken far enough downstream from the model so that the static pressure has returned to the value of the freestream pressure ahead of the model. Consideration must be given to the dependence of the velocity in the wake on the density, and thus, the temperature. Consequently, measurements with a thermocouple were made to determine the temperature distribution through the wake. Figure 10 presents the variation of temperature in the wake of the heated airfoil. The measured values are compared to those of Reichart[16] for the wake of a heated cylinder. Figure 11 presents the results of the drag measurements and the estimated uncertainty in the measurements.

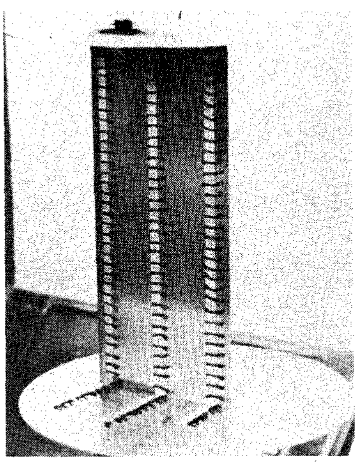

Fig. 8 Hot wing model mounted in wind tunnel.

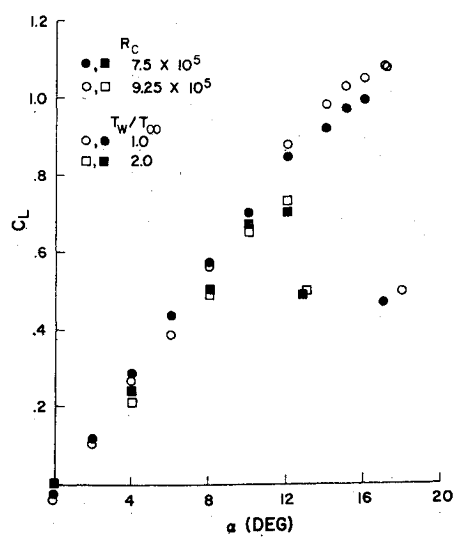

Fig. 9 Experimental lift slope curve for NACA 0012-64 airfoil.

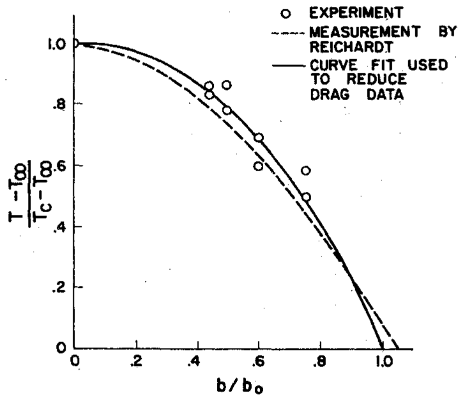

Fig. 10 Temperature distribution in the wake of heated airfoil.

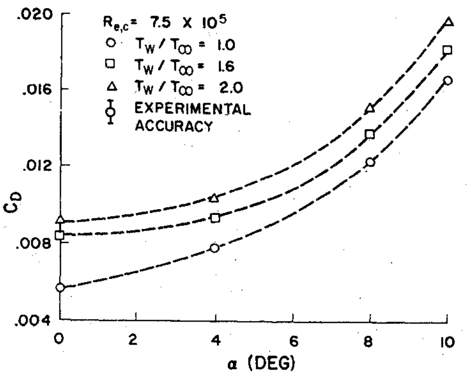

Fig. 11 Drag vs angle of attack for heated airfoil.

An increase in drag was noted; however, due to the uncertainties in the measurements the results should be regarded as trends only.

These trends are of the type obtained when an airfoil with a blunt trailing edge is tested.[18] Since the effect of the heating was to thicken the boundary layer and hasten separation it is not surprising that the two effects are similar.

Tests were made to determine the effect of wing temperature on the stall characteristics of the airfoil. Tufts could not be used on the heated model to visually observe the occurrence of stall. Alternately, stall was defined by the sudden increase in drag indicated by the pressure rake connected to a monometer board. Thus, stall is defined here as completely separated flow, as opposed to the maximum lift point which occurs at a slightly lower angle of attack.

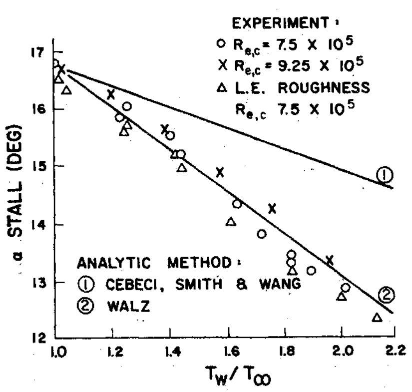

Fig. 12 Comparison of predicted and experimental effect of wing temperature on stall.

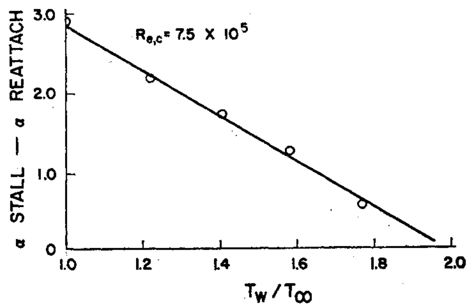

Fig. 13 Reduction in angle of attack to reattach stalled airfoil.

The procedure followed in taking stall data was to heat the wing to the desired temperature at an angle of attack of 10° and then increase the angle of attack until the airfoil stalled. A sudden, well-defined increase in drag at stall was observed at all temperatures. Stall data was taken at Reynolds numbers of 7.5 and 9.25×10^5 for temperature ratios ranging from 1.0 to 2.2. In addition, data was taken with a boundary-layer strip of #60 grit 0.25 in. wide at 1 in. from the leading edge to insure turbulent flow over the wing. The results are presented in Fig. 12. The figure shows a virtually linear decrease in the stall angle from approximately 17° at a temperature ratio of 1.0 to 12.5° at a temperature ratio of 2.2.

Measurements were also taken on the angle of attack at which the separated flow reattached to the airfoil. At the freestream temperature, the reattachment was sudden and well-defined. At high temperatures, however, reattachment appeared to be less definite. Figure 13 presents the angle through which the angle of attack must be decreased to reattach the flow.

Conclusions

The following conclusions are based on the analytical and experimental considerations of this investigation as applied to a two-dimensional symmetrical airfoil.

1) The stall angle of an airfoil is reduced as the wing is heated. Experimental results indicate an approximately linear decrease in the angle of attack at which the model stalls from approximately 17.0° at $T_w/T_\infty = 1.0$ to 12.5° at $T_w/T_\infty = 2.2$. Experimental results compare qualitatively with the predictions of the numerical methods. The method of Walz agrees best with the test results. The method of Cebeci, Smith, and Wang[7] appears to be somewhat conservative in estimating the effect of wing temperature in reducing the stall angle of attack. This could be due to the models for eddy viscosity and conductivity breaking down under the high-heat transfer rates considered here.[17] More accurate analytical models will require coupling of the heat transfer, boundary layer, and flowfield calculations.

2) The maximum lift coefficient C_L is reduced as the wing is heated. The results of the experiments show that the maximum C_L is reduced from 1.0 to 0.7 when the wing-to-freestream temperature ratio T_w/T_∞ is increased from 1 to 2. A slight reduction in lift at smaller angles of attack (less than 10°) appears to be indicated by experiment.

3) The drag of an airfoil appears to increase as the wing is heated. The wind-tunnel tests conducted during this investigation indicated that the drag of a symmetrical airfoil will increase over a range of angle of attack from 0° to 10° as the temperature ratio is increased from 1 to 2. It was shown analytically that wing temperature does influence the flow by increasing the boundary-layer thickness and by shifting the point of separation forward. As a temperature ratio of 2.0, the boundary-layer thickness has doubled. Over the aft portion of the airfoil, this increase is significant compared to

the wing thickness. Therefore, it can be postulated that a slightly earlier separation (~5% of the chord) and a wider wake increases the net drag force by creating a larger region of suction pressure acting on the trailing edge of the airfoil.

4) Experimental results indicate that reattachment of separated flow over a heated airfoil is less well-defined than reattachment over an unheated wing.

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