Engineering Notes

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Effect of Wall Temperature Spatial Variations on Aerodynamic Heating to an Actively Cooled Aircraft

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Nomenclature

Q_w = heat transfer rate for a constant wall temperature of 250°F
ΔQ_w = maximum deviation of heat transfer rate from constant wall temperature value
$R(x_w)$ = local Reynolds number based on x
T_w = wall temperature
ΔT_w = maximum deviation of wall temperature from constant value of 250°F
w = tube space or wall temperature cycle length
x = surface distance from nose
x_w = surface distance from nose to point where cyclic wall temperature begins

PRIOR studies have indicated the potential advantages of an active cooling system approach over alternate cooling concepts and uncooled structure for liquid hydrogen-fueled hypersonic aircraft.[1,2] One concept shown to be feasible for Mach numbers up to 8 employs unprotected aluminum alloy structure cooled by a liquid convective system to normal working temperatures of 250°F to 300°F. This concept involves the incorporation of tubular coolant passages in the load carrying skin. Each loop of the coolant distribution system contains a network of lines that connect the airframe skin panels, where heat is absorbed by a secondary water-glycol coolant, to a central heat-exchanger where heat is rejected to the liquid hydrogen fuel on its way to the engine. The structural skin represents a very large surface area from which heat must be removed. Thus the design of the cooled skin panels, which requires consideration of many parameters such as heat load, skin thickness, panel size, tube spacing, tube size, temperature limits, coolant flow, etc., must minimize weight and thermal stresses.

A design procedure for such panels has been developed by Bell Aerospace Company under Langley contract.[2] This design procedure assumes a uniform heating rate to the panel. In other words, the variation of aerodynamic heating rate due to the cyclic wall temperature distribution, which results from the discrete coolant tube locations, in such a panel has been neglected. The present study was made to determine if these variations were sufficiently large to cause variations in aerodynamic heating rate which would require coupling of boundary layer and conductive thermal analysis solutions.

An evaluation of the magnitude of the variations in aerodynamic heating rate resulting from such cyclic wall temperature distributions for an actively cooled aluminum Mach 8 aircraft was made in the following manner. First the range of aerodynamic heating rates to a Mach 8 aircraft having a constant wall temperature of 250°F were determined. Then, using the method of Ref. 2, the maximum and minimum wall temperatures for a range of suitable coolant tube spacing were calculated. Figure 1 illustrates a typical cooled skin arrangement of the type of wall temperature distribution that results. Of course, the variation in wall temperature, ΔT_w, depends on the tube spacing, the heating rate, and other parameters. The last step was to calculate a new aerodynamic heating rate assuming a cyclic wall temperature distribution.

To simplify these calculations, the wall temperature was held constant at 290°F for a distance, x_w, at which point a sine wave wall temperature distribution was assumed where the cycle length was taken to be w and the temperature oscillated by the amount ΔT_w about the mean value of 290°F. Calculations were made for distances representing several temperature cycles; three different values of ΔT_w; three different Reynolds numbers; lengths, x_w, of 1, 2, 4, 7, 10, 15, 25, and 50 ft; and cycle lengths, w, of 0.05, 0.1, 0.2, 0.3, 0.4, and 0.5 ft. Other assumptions were: fully turbulent flat plate flow from the nose, freestream Mach number of 8, total temperature of 5958°F, and a flow deflection angle of 8° giving a local Mach number of 6.18. Results for many cases were obtained using the computation procedure of Ref. 3. Figure 2 shows the heat transfer results for one case. The maximum deviation of heat transfer rate from the constant wall temperature solution, ΔQ_w, was measured as indicated in the figure. In addition to the magnitude of the deviation in heat transfer for the variable wall temperature case, two other factors illustrated in this typical result are of interest when considering active cooling for high speed aircraft. First the total or integrated heat load for the variable wall temperature case is slightly higher than that for the constant wall temperature calculation and second there appears to be no cumulative effect on heat transfer of multiple wall temperature cycles. A check of these two results was made by making similar calculations using an entirely different method; i.e., that of Ref. 4. Using this method indicated that the integrated heating rates for constant and cyclic wall temperatures were the same, that there was no cumulative effect of multiple wall temperature cycles for many cycles and approximate agreement with the method

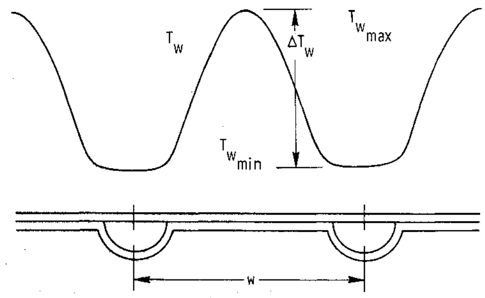

Fig. 1 Possible coolant tube arrangement and wall temperature distribution.

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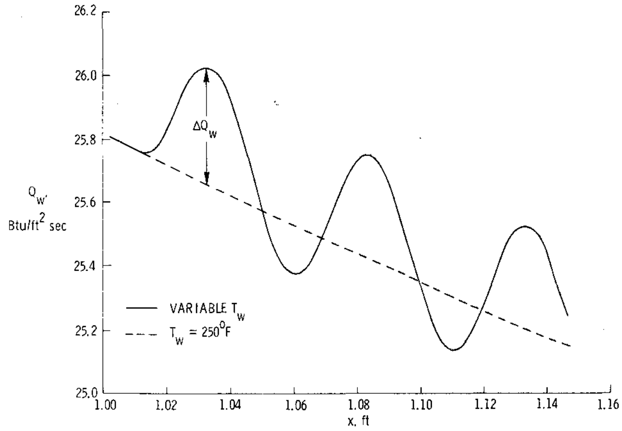

Fig. 2 Heat transfer results for typical case $R(x_w) = 1.34 \times 10^6\ \mathrm{ft}^{-1}$, $\Delta T_w = 25°F$, $w = 0.05$ ft, $x_w = 1.00$ ft.

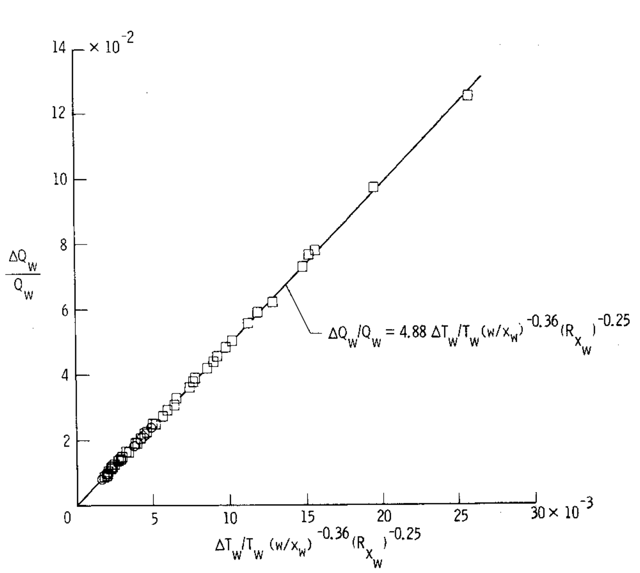

Fig. 3 Correlation of cyclic wall temperature results.

of Ref. 2 as to the magnitude of ΔQ_w. These results, of course, lack experimental verification.

An attempt was made to correlate the results of the cyclic wall temperature cases in terms of the maximum deviation of heating rate, ΔQ_w, from the constant wall temperature data such as $\Delta T_w/T_w$, w, x_w, R_w, and $R(x_w)$. In order to assist in correlating these data, several cases were run in which the values of $\Delta T_w/T_w$ and w are outside the range of practicality for actively cooled transports. In all a total of 50 cases were analyzed. The result for one apparently successful correlation is shown in Fig. 3. The straight line faired through the 50 data points has the equation

$$\frac{\Delta Q_w}{Q_w} = 4.88 \frac{\Delta T_w}{T_w}\left(\frac{w}{x_w}\right)^{-0.36}\left(R_{x_w}\right)^{-0.25}$$

Using this correlation one sees that it is possible to have rather large variations in the heat transfer rate due to a cyclic wall temperature distribution. However, when one examines the result for values of $(\Delta T_w/T_w)$, w, x_w, and $R(x_w)$ that are practical for a Mach 8 aircraft, the variation in heat transfer rate is less than about six percent which is believed sufficiently small so that there is no need to couple the conductive thermal analysis used for panel design with a boundary-layer program.

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Calculation of Nonlinear Lift and Pitching Moment Coefficients for Slender Wing-Body Combinations

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Nomenclature

U_∞ = freestream velocity
α = angle of incidence
b = wing span
$\bar{c}$ = mean wing chord = S/b
$\bar{\bar{c}}$ = mean aerodynamic chord = $\frac{1}{S}\int_{-b/2}^{+b/2} c^2(y)dy$
S = wing area
A = aspect ratio
λ = taper ratio
ν = kinematic viscosity
Re = Reynolds number = $U_\infty \bar{c}/\nu$
C_L = lift coefficient = $L/(\rho(U_\infty^2)S/2)]$
C_m = pitching moment coefficient = $M/(\rho(U_\infty^2)S\bar{\bar{c}}/2)$, nose-up positive, referred to the quarter-chord point of the mean aerodynamic chord projected on the plane of symmetry, N_{25}

Introduction

IT is well known that lift and pitching moment coefficients of slender wings and fuselages depend nonlinearly on the angle of incidence.[1,2] This behavior is related to flow separation which leads to the formation of trailing vortices above the upper surface of such wings and bodies. The nonlinearities arise from the fact that the position of the trailing vortices with respect to the wing and the body changes with the angle of incidence.

For combinations of slender wings and fuselages very strong interference effects have to be expected and the occurrence of nonlinear lift and pitching moment coefficients may result from the following effects: 1) effect of the wing vortices on the wing, 2) interference effect of the wing vortices on the fuselage, 3) effect of the fuselage vortices on the fuselage, and 4) interference effect of the fuselage vortices on the wing. The effect 1) has been dealt

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