# Leeward Heat Transfer Experiments on the Shuttle Orbiter Fuselage

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## Nomenclature

$Re_{ns}$ = Reynolds number based on conditions behind bow shock and reference length of 0.01 ft (0.305 cm)
$St$ = Stanton number, $q[(\rho u c_p)_{ns}(T_{o_e} - T_w)]^{-1}$
$\alpha$ = angle of attack measured between approach velocity vector and longitudinal axis of model
$\Delta$ = viscous layer thickness, $\Delta/\delta = \int_0^l \frac{\rho u}{\rho_e u_e} \frac{T_c - T_o}{T_{o_e}} d(y/\delta)$
$\phi$ = angular location on leeward surface (Fig. 1)

*Subscripts*

$c$ = Crocco stagnation temperature profile
$ns$ = conditions aft of bow shock
$0$ = isentropic stagnation conditions
$\infty$ = approach flow conditions

## Introduction

A RECENT study[1] of convective heat transfer on the leeward surface of a space shuttle orbiter fuselage configuration has verified that, by cooling the windward fuselage surface, proportional reductions in leeward heating also occur due to decreased convection from the windward boundary layer into the near wake region. The investigation produced experimental data on two configurations: a fuselage-wing geometry and a bare fuselage (without wings). Data for the winged configuration have been discussed elsewhere.[2]

The present Note will summarize that portion of the investigation related to the bare fuselage tests in which data were obtained for an angle of attack of 90° so as to produce a nominal planar flow. In addition some supplementary data were obtained at angle of attack of 30°. It is clear that the bare fuselage geometry, when emersed in a nearly normal approaching flow, is merely a cylinder of noncircular cross section and moderate aspect ratio. Among previous investigations of cylinders normal to supersonic or hypersonic streams are those reported by Penland,[3] Dewey,[4] Beckwith and Gallagher,[5] and Bertin et al.[6]

## Experimental Program

Data were obtained on a 0.01-scale model of the shuttle orbiter fuselage (Fig. 1). The model was constructed of stainless steel (17-4PH) and incorporated an asbestos insulation plate and associated steel radiation shield which divided the geometry longitudinally into windward and leeward sections. The interior of the windward section was designed to accept either an Incaloy heating element or a copper cooling coil through which freon was circulated. Thus, the windward surface could be both heated and cooled while the leeward section was operated with a cooling coil only. Heat transfer rates were determined from transient surface temperatures through use of thin-film resistance thermometers having surface dimensions of 1×5 mm. Locations of the 31 heat transfer and 3 pressure gages used in the cylinder tests are shown in Fig. 1; it is seen that the angular coordinate $\phi$ varies between 0° at the leeward pitch plane to 90° at a point on the side of the model. The nominal cylindrical cross section began at $x/L = 0.279$.

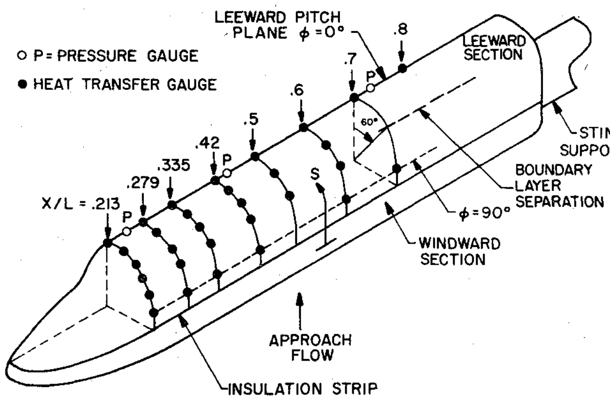


**Fig. 1 Isometric sketch of model illustrating locations of heat transfer and pressure gages. $L = 1.075$ ft. (32.8 cm).**

Present data were obtained with air as the test gas in the Calspan 96-in. hypersonic shock tunnel over a range of test conditions given in Table 1. The test program included three values of windward surface temperature ($T_{wwd}$) and two values of leeward section temperature ($T_{lee}$) so as to produce nominal conditions which are characteristics of both atmospheric reentry and continuous flow, supersonic wind tunnels. Further details of the test program are presented elsewhere.[1]

## Discussion of Results

Experimental circumferential distributions of local heat transfer rate indicated, as expected, a steep decrease in $q$ from an attached flow value at $\phi = 90°$ to a nearly constant level within the separated region. Although a precise location of boundary-layer separation was not indicated due to the spacing of the gages, the extent of the recirculating region was reasonably well defined by $\phi \leq 60°$ (see Fig. 1). As expected, distorted distributions were observed at each end of the model due to threee-dimensional effects associated with the nose section and the sting support. However, it is believed that flow near the midspan was substantially planar for the $\alpha = 90°$ runs.

It was expected initially that data for the two $\alpha = 30°$ runs would would display significant longitudinal variations in heat transfer similar to those found in the corresponding fuselage-wing tests.[2] However, all distributions were qualitatively equivalent, thus suggesting that the perturbations of leeward heat transfer for the fuselage-wing geometry were produced primarily by the interaction between vortices shed from the nose and from the wing leading edge as well as by shear layer transition which is promoted by the strong vortices.

Two values of Stanton number may be used to characterize leeward convection, viz., $St_{af}$ for the attached flow just prior to separation ($\phi = 90°$) and $St_{lee}$ for the recirculating zone ($\phi \leq 60°$). The characteristic Reynolds number which correlates these Stanton numbers was shown previously[1,2] to be $Re_{ns}$.

Values of the two characteristic Stanton numbers for the complete set of 10 test runs are tabulated in Table 1 and presented as a function of $Re_{ns}$ in Fig. 2. The indicated variations in heat transfer rate display three major influences as follows:

1) Runs 2, 26, and 28 and runs 3, 31, and 27 indicate the effect of increases in $T_{wwd}/T_{lee}$ from 1.0 to 1.62 to 2.72 whereas

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**Table 1 Test conditions and selected results**

| Run | $M_\infty$ | $Re_\infty/m$ ($\times 10^{-6}$) | $Re_{ns}$ | $T_0$ (K) | $T_{wwd}$ (K) | $T_{wwd}/T_{lee}$ | $St_{lee}$ $\times 10^4$ | $St_{af}$ $\times 10^3$ | $\Delta/\delta \times 10^2$ (See note "a" below) |
|---|---|---|---|---|---|---|---|---|---|
| Runs with $\alpha = 90°$ | | | | | | | | | |
| 2 | 12.8 | 1.85 | 570 | 2570 | 233 | 1.0 | 5.05 | 1.95 | 0.93 |
| 3 | 11.9 | 5.34 | 1710 | 2580 | 233 | 1.0 | 3.28 | 1.23 | 0.96 |
| 26 | 12.3 | 1.77 | 510 | 2410 | 478 | 1.62 | 5.61 | 2.54 | 1.00 |
| 27 | 11.8 | 5.21 | 1720 | 2640 | 794 | 2.72 | 3.79 | 1.59 | 1.32 |
| 28 | 12.2 | 1.67 | 510 | 2560 | 800 | 2.72 | 7.20 | 3.04 | 1.28 |
| 29 | 11.7 | 0.36 | 120 | 2540 | 800 | 2.72 | 13.4 | 6.02 | 1.27 |
| 30 | 15.7 | 1.90 | 360 | 2580 | 800 | 2.72 | 6.76 | 3.25 | 1.18 |
| 31 | 11.8 | 5.29 | 1750 | 2660 | 478 | 1.62 | 3.34 | 1.24 | 1.19 |
| Runs with $\alpha = 30°$ | | | | | | | | | |
| 1 | 12.3 | 1.85 | 570 | 2570 | 233 | 1.0 | 2.51 | 1.04 | 0.82 |
| 32 | 12.3 | 1.88 | 580 | 2540 | 811 | 2.72 | 3.87 | 1.56 | 1.20 |

[a] Boundary-layer parameters shown are for $\phi = 90°$ (Fig. 1).

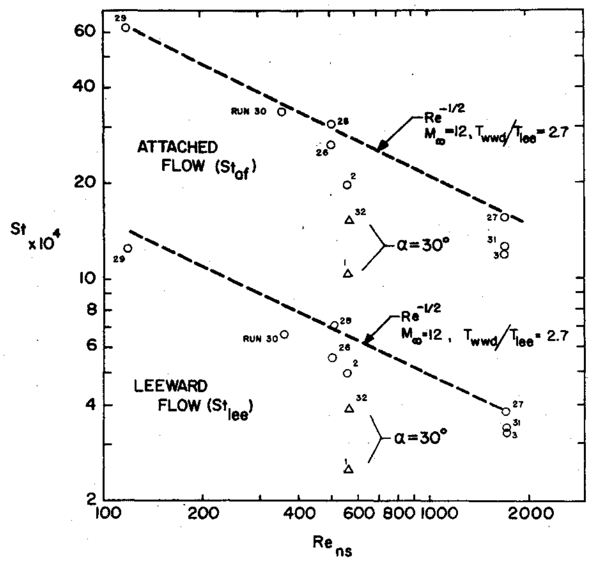


**Fig. 2 Variation of Stanton numbers for attached flow at $\phi = 90°$ and for separated region ($0 \le \phi \le 60°$) with Reynolds number $Re_{ns}$.**

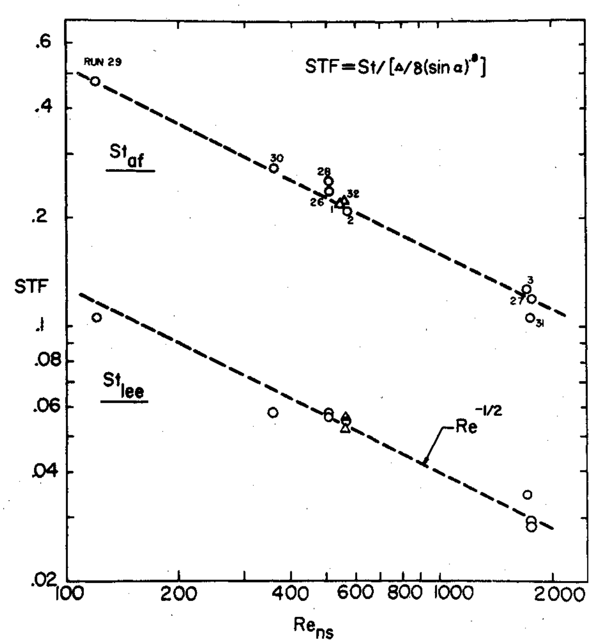


**Fig. 3 Correlation of characteristic Stanton numbers with Reynolds number $Re_{ns}$. Data for $\alpha = 30°$ denoted by triangles.**

runs 1 and 32 display, for a 30° angle of attack, the effect of an increase in $T_{wwd}/T_{lee}$ from 1.0 to 2.72. In all these runs the approach Mach number was nominally 12. It is observed that heat transfer levels are lowered significantly as $T_{wwd}/T_{lee}$ is decreased to unity; furthermore the effect of a smaller angle of attack is also to decrease heat transfer rates.

2) Runs 27, 28, and 29 display, for $T_{wwd}/T_{lee} = 2.72$, the influence of a 15-fold increase in $Re_{ns}$. The nominal approach Mach number was 12 for these runs. The Stanton number variation with Reynolds number appears to confirm the existence of both a laminar boundary layer and a laminar near wake for these approach flow conditions.

3) Runs 28 and 30 indicate, for $T_{wwd}/T_{lee} = 2.72$, the effect of an increase in approach Mach number from 12 to 16. This effect is seen to be small, which is not unexpected for a blunt body in hypersonic flow.

The influence of $\alpha$ on windward heat transfer has been correlated previously[5] with a power-law relationship, viz., $St \sim (\sin\alpha)^n$ where $1 \le n \ge 1.5$. Although limited, the present data for $\alpha = 30°$ suggest an exponent of $n \simeq 0.8$. Correlation of the effect of $T_{wwd}/T_{lee}$ is facilitated through the recognition that any variation in surface temperature affects the shape of the stagnation temperature profile within the viscous wall layer prior to separation. One can therefore devise an integral thickness parameter which reflects the difference between the actual $T_o$ profile and some baseline profile, such as the Crocco profile in which stagnation temperature is proportional to velocity. Current boundary-layer computations, which are summarized in Table 1, suggest that the effect of the windward to leeward temperature ratio can be correlated by $St \sim \Delta/\delta$ (see Nomenclature).

The foregoing correlation proposals are incorporated in the parameter $STF = St/[\Delta/\delta(\sin\alpha)^{.8}]$ which is shown as a function of $Re_{ns}$ in Fig. 3. The correlation lines are given by $STF_{lee}(Re)^{1/2} = 1.24$ and $STF_{af}(Re)^{1/2} = 5.16$. Dispersion of the data is characterized by mean deviations of 6% and standard deviations of 8%. Relative values of two levels of heat transfer in Fig. 3 are of interest although the actual ratios are known to be configuration dependent. The correlation equations show that $St_{lee} \simeq 0.24\, St_{af}$ which compares with a value of 0.6 from Chapman's early theoretical model[7] for negligible boundary layer prior to separation.

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[3]Penland, J.A., "Aerodynamic Characteristics of a Circular Cylinder at Mach Number 6.86 and Angles-of-Attack up to 90°," NACA, RM L54A14, Washington, D.C., March 1954.

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# Asymmetric Shock-Wave Oscillations on Spiked Bodies of Revolution

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## Nomenclature

$D$ = overall model diameter
$F$ = frequency
$p_1, p_2$ = pressure sensed by transducers 1 and 2
$p_{ns}$ = pressure downstream of normal shock
$p_\infty$ = stream static pressure
$S$ = Strouhal number
$T_o$ = wind-tunnel supply temperature
$u_\infty$ = freestream flow speed
$x$ = coordinate in stream direction
$y$ = coordinate normal to $x$
$(\bar{\ })$ = time-averaged quantities
$\langle\ \rangle$ = root-mean-square quantities

IT is well known that the shock-wave configuration in front of "spiked" axisymmetric bodies in supersonic flow can be unstable.[1,2] This instability arises primarily in cases where the forebody geometry has a forward-facing step such as those found in supersonic inlet diffusers with conical centerbodies, or on blunt bodies with drag-reducing spikes. For certain ranges of forebody geometry and flow parameters[3-5] these configurations do not allow a steady-state shock shape capable of satisfying the momentum and continuity equations. As a result, the shock structure oscillates harmonically with a Strouhal number based on body diameter and stream speed which, characteristically, is of order 0.2. This oscillation has been termed a "catastrophic" or "E-oscillation." Until recently, the shock motion was thought limited to one degree of freedom in the radial direction; that is, the shock has been observed to move in an expanding-collapsing fashion symmetrically about the body axis. In this Note, we report observations which imply a more complex motion such as would result from two or more degrees of freedom. Time-averaged and instantaneous surface pressure data presented should be of some interest also since they control the time-averaged drag, and since their prediction by theory is difficult at this juncture.

Experiments were performed in the Aeronutronic Ford Supersonic Wind Tunnel with a model, sketched in Fig. 1, which consisted of a sphere-cone-cone cylinder. In units of the cylinder afterbody diameter $D$, the hemisphere at the model tip had a 0.05 $D$ diam. The cone following it had a half-angle of 20° and the second cone a half-angle of 80°. The model was positioned at zero incidence in continuous flow at Mach $3.02 \pm 0.02$ and supply temperature $T_0$ of 100°F. The Reynolds number based on $D$ and freestream properties varied from 40,000 to 150,000 and the model wall temperature was nearly equal to $T_0$.

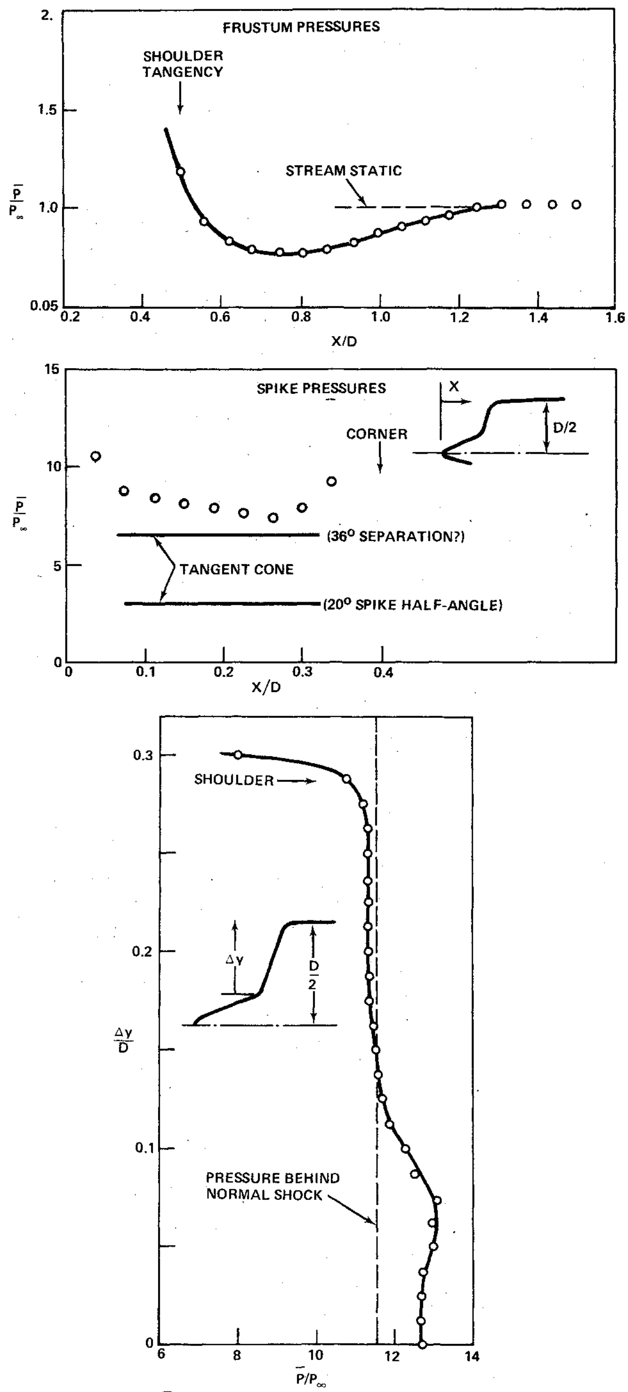


**Fig. 1 Time-averaged surface pressure distribution over the spike, front "face" and frustum of the model.**

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