Table 1 Stagnation convective-heating calculations

Freestream velocity, km/s U_∞	Wall temperature, K T_w	Convective heating, W/cm^2 q_{VLE}	q_{CLE}	q_{LE4}	Diffusion heating ratio $(q_d/q_c)_{\mathrm{VLE}}$	$(q_d/q_c)_{\mathrm{CLE}}$	$(q_d/q_c)_{\mathrm{LE4}}$	Conduction heating ratio $(q_k/q_c)_{\mathrm{VLE}}$	$(q_k/q_c)_{\mathrm{CLE}}$	$(q_k/q_c)_{\mathrm{LE4}}$
6	1400	21.2	23.6	24.4	0.01	0.02	0.01	0.99	0.98	0.99
	1000	21.7	24.2	24.9	~0	~0	~0	1.0	1.0	1.0
12	2280	154	212	222	0.32	0.28	0.30	0.68	0.72	0.70
	1000	158	216	226	~0	~0	~0	1.0	1.0	1.0
14	2580	226	321	342	0.53	0.48	0.50	0.47	0.52	0.50
	1000	232	329	347	~0	~0	~0	1.0	1.0	1.0

Subscripts:
c = convective (diffusion and conduction)
d = diffusion
k = conduction
VLE = variable Lewis number
CLE = constant Lewis number
LE4 = Lewis number = 1.4

Table 2 Wall Lewis numbers

Freestream velocity, km/s U_∞	Wall temperature, K T_w	Lewis number Le
6	1400	1.294
	1000	1.304
12	2280	1.271
	1000	1.311
14	2580	1.252
	1000	1.309

wall temperature condition for the variable Lewis number case, is held constant through the layer for the second case. The third case is for flowfield calculations using a constant value of 1.4 for the Lewis number. Shock-layer Lewis number (Le) distributions are shown in Fig. 1 for different freestream velocities and wall temperatures. At a freestream velocity of 14 km/s, different wall temperature values are shown to have an insignificant influence on the Le distribution through the shock layer as might be expected. However, the very large variations in the Le distribution at 14 km/s were not expected. A more likely anticipated Le distribution is shown by the results for the 6-km/s case. The minimum and maximum Le values for the 14-km/s case are computed at shock locations which also represent conditions of full oxygen and nitrogen dissociation. The Le values shown for the two velocity cases at a normalized shock-layer distance η of 0.15 are essentially constant to the shock location ($\eta = 1.0$). The wall Le for the conditions considered in this Note are presented in Table 2.

Several observations can be noted from these tabulated data. At higher velocities, the discrepancy in the predicted heating rate for different assumptions of the Lewis number through the shock layer is as large as 50%. The differences illustrate the need for a more detailed evaluation, i.e., a multicomponent calculation of the diffusion contribution to the heating. For the high and low wall temperature values for each velocity condition and each assumption for implementing the Lewis number, there is only a 2–3% difference in the corresponding heating rates. This result is very interesting since for the higher wall temperature values, diffusion comprises about 30 and 50% of the convective flux for the 12- and 14-km/s cases, respectively. The heating values that are presented for the lower velocity condition indicate that the present methods for implementing the Lewis number in the flowfield calculations have little impact on the heating rates. A VSL analysis of Moss[5] showed that the influence of Lewis number, even for multicomponent diffusion calculations, was also small for calculations at similar low values (~6 km/s) of velocities. Another study[6] using the boundary-layer equations also included multicomponent diffusion effects in calculations at about 6 km/s. The analysis of Ref. 5 was based on a 5-species air mixture whereas the study of Ref. 6 employed a 7-species model. The conclusions of both investigations were the same including the influences of multicomponent diffusion on the surface heating. However, at hypervelocity conditions, the insensitivity of the Lewis number calculation on the heat-transfer calculation is obviously not a correct conclusion even for low wall temperatures for which the surface diffusion contribution to the convective flux is very small in comparison to the conduction term. Thus, the need for multicomponent diffusion calculations is clearly demonstrated at hypervelocity conditions even for nonablating surfaces.

Note that an effective binary diffusion coefficient[7] was used in the flowfield study of Ref. 8. This coefficient is based on a weighted-average calculation of the individual binary coefficients and provides a significantly faster computational method than the corresponding multicomponent calculation. However, the accuracy with which this effective coefficient approximates the multicomponent diffusion characteristics in a flowfield is not presently known.

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Leeside Shock-Layer Transition and the Space Shuttle Orbiter

David A. Throckmorton*
NASA Langley Research Center, Hampton, Virginia 23681

Nomenclature

R_{NS} = Reynolds number evaluated behind a normal shock, based on Orbiter length (32.77 m)

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*Assistant Head, Aerothermodynamics Branch. Associate Fellow AIAA.

St = Stanton number, based on freestream total enthalpy
x/L = nondimensional vehicle length coordinate

Introduction

IN 1981–1982, a heavily instrumented Shuttle Orbiter Columbia was flown on a series of orbital flight test missions (STS-1 thru STS-5) which preceded the operational phase of the Space Shuttle program. Analysis of data obtained during atmospheric entry on the STS-3 mission revealed that a sudden and unexpected increase in heat transfer occurred, at about Mach 16 and 62-km altitude, simultaneously at multiple measurement locations on the Orbiter's leeside fuselage and wing.[1] The hypothesis of Ref. 1 suggested that this heating increase resulted from transition of the flow in the leeside shock layer from a laminar state to a transitional or turbulent state. Since the heating increase was observed to occur almost simultaneously at multiple measurement locations on both the fuselage and wing, the flow transition was presumed not to be a localized phenomenon, but rather a "global" phenomenon which, once initiated, rapidly affected the entire leeside shock layer.

The original analysis[1] which indicated the occurrence of this apparent leeside flowfield transition was based primarily upon data from a single Orbiter entry. Since that time, the Orbiter Columbia has flown additional missions with instrumentation which have enabled further observations of the phenomenon. This Note provides additional documentation of the occurrence of this leeside flowfield transition event, by compiling results from a total of seven Orbiter entries which occurred during the time period of 1981–1991.

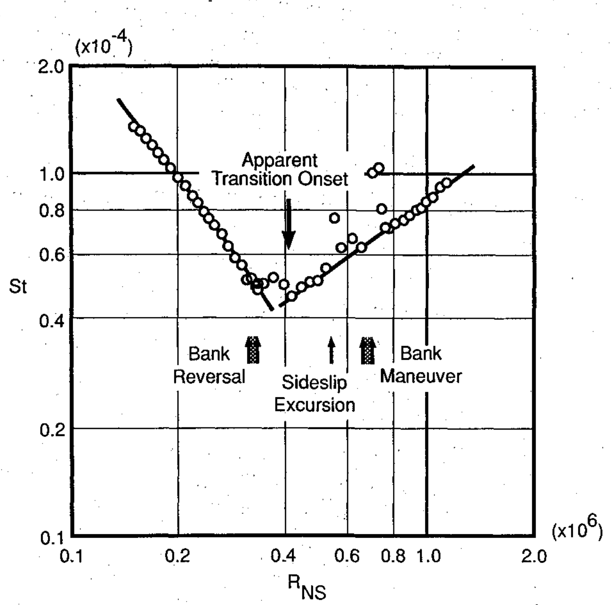

Fig. 1 STS-28 heat transfer to orbiter leeside centerline at $x/L = 0.7$.

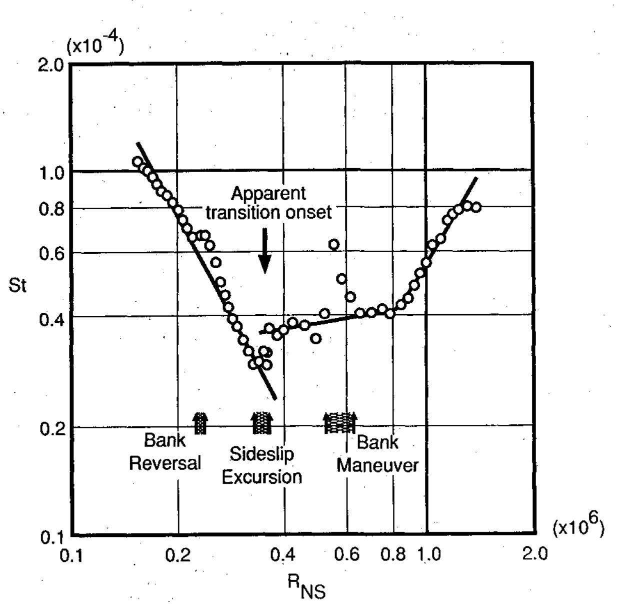

Fig. 2 STS-40 heat transfer to orbiter leeside centerline at $x/L = 0.7$.

Flight Heat Transfer Data

During the orbital flight test missions of the Orbiter Columbia, the vehicle was equipped with an instrumentation system referred to as the development flight instrumentation (DFI). Included among the DFI were measurements of aerodynamic surface temperature at multiple locations on the Orbiter's upper fuselage and wing. These measurements were obtained using thermocouples embedded at the aerodynamic surface of the thermal protection system. Prior to STS-28, leeside temperature sensors which had been a part of the DFI system were reactivated. Consequently, leeside fuselage temperature data are available from missions STS-2, -3, -5, -28, -32, -35, and -40. A one-dimensional, transient analysis[2] of heat conduction within, and reradiation from, the thermal protection system materials was used to determine the time-varying heat transfer rate at each measurement location, using the measured temperature data as a surface boundary condition.

Discussion

Background

The flowfield over the leeward side of the Shuttle Orbiter during the high angle-of-attack, hypersonic portion of entry is a complex, three-dimensional, and separated flow regime. The nature of this flowfield, and the resultant aerodynamic heat transfer to leeside surfaces, may be acutely sensitive to changes in the freestream flight environment, i. e., Mach and Reynolds numbers; and vehicle attitude, i.e., angles-of-attack and sideslip. During the portion of entry on which this study is focused, the Orbiter is flown at a constant attitude of 40-deg angle-of-attack and zero sideslip, with minimal variation about these nominal values. (Trajectory control is accomplished by vehicle banking about the velocity vector.) Thus, changes in surface heat transfer rates which occur during this portion of entry are indicative of changes in the structure of the vehicle's leeside flowfield which result from the continually changing freestream flight environment, not from steady-state changes in the vehicle's nominal flight attitude.

The analysis of STS-3 heat transfer data documented in Ref. 1 revealed a sudden increase in heat transfer to Orbiter leeside surfaces, which occurred at a flight condition of Mach 16.3. Because this abrupt heating increase was observed to occur simultaneously at multiple locations on both the upper fuselage and wing, its cause was attributed to a transition in the leeside shock-layer flow. It is postulated that such a flow transition originates in the viscous shear layer downstream of lines of flow separation which exist along the vehicle's forward fuselage and wing leading edges. Thus, this transition is presumed to be a "global" phenomenon which rapidly affects the entire leeside flowfield. Such a shear-layer transition phenomenon was discussed extensively by Bertin and Goodrich[3] in their analysis of Shuttle Orbiter leeside heat transfer data obtained in a hypersonic shock tunnel.

It must be noted that this apparent leeside shock-layer transition occurred at a flight condition at which the windward-surface boundary-layer state was laminar. Significant movement of the lower-surface boundary-layer transition front, from the aft portion of the vehicle toward the nose, occurred more than 250 s later, after

Table 1 Transition data summary

Mission	R_{NS} ($\times 10^{-6}$)	Mach number	Altitude, km
STS-2	0.39	16.4	61.5
STS-3	0.42	16.3	61.7
STS-5	0.38	17.3	61.9
STS-28	0.41	17.5	62.8
STS-32	0.50	16.7	59.5
STS-35	0.56	15.9	57.8
STS-40	0.35	18.3	64.3

the Orbiter had decelerated to below Mach 9, at significantly lower altitude and angle-of-attack.[4–5]

Current Results and Observations

As indicated in the Introduction, heat transfer data obtained on multiple Orbiter entries have provided evidence of the occurrence of an apparent flow transition in the leeside shock layer. Representative flight data are presented herein, for a measurement location on the Orbiter's leeside centerline, at a longitudinal position of 70% of vehicle length. This location was selected because of the availability of data from a maximum number of Orbiter flights. Data are presented only for flight conditions at which the vehicle's nominal angle-of-attack was 40 deg. These data were obtained over a Mach number range from 24 to 10, with a corresponding altitude range from 72 to 51 km.

For the purpose of the current analysis, the flight-derived heat transfer data are cast in the form of the nondimensional Stanton number St, based on stream total enthalpy. Stanton numbers are presented as a function of Reynolds number evaluated downstream of a normal shock, based on vehicle length. This "normal-shock Reynolds number" R_{NS} was proposed by Bertin and Goodrich[3] for correlation of heat transfer data in the leeside, separated flow region. They demonstrated that, for Orbiter ground-test data, better data correlation was achieved by using this parameter than by using the independent freestream parameters of Mach and Reynolds number. The normal-shock Reynolds number is a function of both freestream Mach number and Reynolds number. Figures 1 and 2 present heat transfer data from missions STS-28 and STS-40, respectively, for the leeside centerline measurement at 70% of vehicle length. The time resolution (i.e., the delta time between points, along the entry trajectory) of the data presented is 10 s.

The STS-28 Stanton number data (Fig. 1) display a generally decreasing trend with increasing normal-shock Reynolds number for flight conditions of R_{NS} less than approximately 0.42×10^{-6}, and a generally increasing trend for flight conditions of R_{NS} greater than that value. The decreasing Stanton number trend for R_{NS} less than 0.42×10^{-6} suggests that the leeside, viscous flowfield was laminar during this early portion of the atmospheric entry. The sudden reversal in trend to an increasing Stanton number for R_{NS} greater than 0.42×10^{-6} suggests the occurrence of a transition in the leeside flowfield at that flight condition. Short-period departures of the heating data from the general trends are evident (Fig. 1) at several different flight conditions. In each instance, however, the heating-data excursion is observed to correlate precisely with a transient perturbation of the vehicle's flight attitude, indicating the sensitivity of the leeside flowfield structure to vehicle attitude. (Although bank maneuvers are, ideally, rotations purely about the velocity vector, they are invariably accompanied by small perturbations to the vehicle's wind-relative attitude, both angles-of-attack and sideslip.) The STS-40 data (Fig. 2) also display a decreasing, apparently laminar, trend during the early portion of entry. At the flight condition of R_{NS} approximately equal to 0.35×10^{-6}, the data slope abruptly becomes positive indicating apparent transition onset. As with the STS-28 results, heating-rate data excursions from the general trend lines correlate with vehicle attitude perturbations.

Any number of fluid dynamic phenomena may affect the level and distribution of heat transfer observed on the leeside of the Orbiter for a given vehicle attitude over a range of flight conditions. These include standing shock waves and embedded vortices, as well as the state (i.e., laminar, transitional, or turbulent) of the local fluid flow. It is this author's conjecture that the abrupt change observed in the slope of the Stanton number data with normal-shock Reynolds number is indicative of a transition in the leeside shock-layer flow.

Figures 1 and 2 illustrate the manner in which the apparent occurrence of leeside shock-layer transition has been inferred from Orbiter surface heat transfer data. The flight condition of apparent leeside-transition onset has been determined in this manner for each of the Orbiter flights for which data are available. Table 1 contains a summary of the seven-mission leeside-transition-onset events. The data contained in Table 1 indicate that the phenomenon of transition in the leeside shock layer of the Shuttle Orbiter, during atmospheric entry, occurs within a relatively narrow range of flight conditions.

Review of the relationship between the apparent transition-onset events and vehicle attitude perturbations suggests that the transition process may be "tripped" as a result of flowfield disturbances which accompany vehicle-attitude perturbations. Apparently, leeside flowfield transition may be described as "incipient" over a well-defined segment of the entry corridor, during which transient disturbances of the leeside flowfield structure may serve as effective transition "trips."

It is recognized that the quantification of apparent leeside shock-layer transition onset in flight is unique to the geometry of the Shuttle Orbiter. However, leeside flowfield transition was not observed during aerothermodynamic-design wind-tunnel testing of the Shuttle Orbiter. The Orbiter aerothermodynamic subsystem managers, Lee and Harthun, have stated[6] that during the development of the Orbiter aerothermodynamic design data base, "Wind tunnel-tests were run where boundary-layer trips were placed on the nose of the Orbiter in an attempt to induce turbulent flow to the leeward side. The resultant heating measured on the leeward surfaces showed no effect of the trips. It was concluded that either the turbulent flow on the windward surface relaminarized when it expanded to the leeward side, or the flow on the leeward side was turbulent without the trips." The differing observations of leeside shock-layer state in flight vs the wind tunnel suggest that tests conducted in hypersonic wind tunnels may not provide a proper simulation of the Orbiter's leeside flowfield for high-altitude, low-Reynolds-number flight conditions.

Concluding Remarks

Heat transfer data obtained on multiple flights of the Shuttle Orbiter Columbia have been used to establish the flight conditions at which transition apparently occurs in the Orbiter's leeside shock layer during entry. This transition is postulated to originate in the shear layer downstream of lines of flow separation which exist along the vehicle's forward fuselage and wing leading edges. This apparent transition phenomenon may be described as "incipient" over a well-defined segment of the entry corridor, during which small disturbances to the leeside flowfield structure, resulting from transient vehicle-attitude perturbations, may serve as effective transition "trips." The flight conditions at which this phenomenon occurs are significantly different (higher altitude and Mach number) from those at which windward-surface boundary-layer transition occurs.

Understanding the probable state of the leeside shock layer in flight is important when flight heat transfer data are used as benchmarks for comparison with wind-tunnel data or the results of state-of-the-art computational fluid dynamic simulations. Valid comparisons between the various data sets demand understanding of the similarities, or possible differences, in the leeside shock-layer flows observed or simulated in those data sets.

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E. Vincent Zoby
Associate Editor