Orbiter Entry Leeside Heat-Transfer Data Analysis

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Heat-transfer data measured along the Space Shuttle Orbiter's leeward centerline and over the wing leeside surface during the STS-2 and STS-3 mission entries are presented. The flight data are compared with available wind-tunnel results. Flight heating levels are, in general, lower than those which are inferred from the wind tunnel results. This result is apparently due to the flight leeside flowfield remaining laminar over a larger Reynolds number range than that of corresponding ground test results. The flight/wind tunnel data comparisons confirm the adequacy of, and conservatism embodied in, the direct application of wind tunnel data at flight conditions for the design of Orbiter leeside thermal protection.

Nomenclature

c	= local wing chord length
DFI	= Development Flight Instrumentation
L	= scaled Orbiter characteristic length, $L = 32.77$ m (107.5 ft) full scale
M	= freestream Mach number
q	= convective heat-transfer rate
q_{ref}	= heat-transfer rate to the stagnation point of a scaled 0.305 m (1 ft) radius sphere
R_L	= freestream Reynolds number based on Orbiter characteristic length
$R_{NS,L}$	= Reynolds number evaluated behind a normal shock based on Orbiter characteristic length
St	= Stanton number
STS	= Space Transportation System
X	= longitudinal distance measured from Orbiter nose
x	= longitudinal distance measured from wing leading edge
$Y/b/2$	= nondimensional spanwise distance measured from plane of symmetry
α	= angle of attack

Introduction

THE flowfield over the leeward side of the Space Shuttle Orbiter during the high angle-of-attack portion of entry when aerodynamic heating is significant, may be characterized as a complex, three-dimensional, viscous, and separated flow regime. The nature of the flowfield, and the resultant surface heat transfer, are acutely sensitive to changes in the freestream flight environment (i.e., Mach and Reynolds numbers), vehicle angle of attack, and leeside geometry. This complex flowfield defies analysis by available computational methods. Consequently, definition of the Orbiter's leeside aerothermodynamic environment for thermal protection system design was based solely on wind tunnel data and empirical correlations.[1]

Past analyses of Shuttle leeside flows, which are reported in the literature,[2-7] have been restricted primarily to the leeward symmetry plane. This paper will present typical convective heat-transfer data for the Orbiter leeward centerline and also the wing leeside from the STS-3 mission entry. Limited STS-2 results are included where available. The flight data are compared with appropriate wind tunnel results for a wide range of flight conditions in order to assess possible conservatism in the design results and determine any dissimilarities in trends of the flight and ground-test data.

Flight Data

Source

During the orbital flight test missions, the Orbiter was equipped with an instrumentation system referred to as the Development Flight Instrumentation (DFI). The DFI was comprised of over 4500 sensors, associated data-handling electronics, and recorder, which provided data to enable postflight certification of Orbiter subsystems design. Included among the DFI were measurements of aerodynamic surface temperature at approximately 50 locations on the upper fuselage and wing surfaces (Fig. 1). These measurements were obtained from thermocouples mounted within the thermal protection system, in thermal contact with the surface coating.[8] Temperature measurements were made at all locations indicated on Fig. 1 for STS-3 and subsequent missions. On STS-1 and STS-2, temperature measurements were made only at those locations denoted by the closed symbols; calorimeters were installed at locations denoted by open symbols. Calorimeter data are not considered in this paper.

DFI temperature data were recorded once each second throughout the time period of entry from Earth orbit. The measured surface temperature-time histories provided for determination of vehicle surface heat-transfer rates.

DFI tape-recorder malfunctions on missions STS-1 and STS-4 resulted in the loss of all thermal data, during that portion of entry when the vehicle was not in communications contact with the ground. No data were obtained at flight Mach numbers above approximately 12. Consequently, only data from missions STS-2 and STS-3 are considered herein.

Data Reduction

A one-dimensional, transient conduction analysis[9] was used to determine the total heating rate to each measurement location. The flight-measured surface temperature data provided a time-dependent boundary condition for the analysis, which assumes an initially uniform temperature throughout the thermal protection materials. The analysis is a mathematically rigorous simulation of the heat conduction within the thermal protection system, and reradiation from its surface, so as to provide a "benchmark" determination of the flight heat-transfer rates.

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On the vehicle leeside, where convective heating rates are low, it has been shown[10] that a significant amount of the total surface heat transfer may result from solar radiation, and additionally for the wing surface, cross radiation from the hot (relatively) Orbiter fuselage. In order to account for the impact of these sources on the thermal response of leeside surfaces in flight, values of solar- and cross-radiation heat-transfer rates to these surfaces were computed. The solar- and cross-radiation heating-rate values were then applied as "correction factors" to the total heating-rate data in order to isolate the convective component of heat transfer. The technique used for the determination of the solar- and cross-radiation heating rates has been fully described in Ref. 10.

Flight Environment

Determination of the vehicle attitude and freestream flight environment data used herein was accomplished through reconstruction of the Orbiter entry trajectory, reconstruction of the atmosphere on the day of entry, and correlation of these two data sets to provide an analytically consistent definition of the entry flight environment. The trajectory reconstruction process[11] utilizes ground tracking data and onboard measurements of Orbiter inertial attitude, linear accelerations, and angular rates to determine the vehicle's inertial position, velocity, and attitude throughout the entry. The atmospheric reconstruction process[12] combines atmospheric modeling with direct measurement of atmospheric profiles on the day of entry in order to define the freestream atmospheric properties (pressure, temperature, density, and winds). The results of the trajectory and atmospheric reconstruction processes are melded together to provide an accurate, analytically and physically consistent definition of the freestream flight environment.

The angle-of-attack and reference-heating-rate histories for STS-3 are shown in Fig. 2. The reference heating rate is that to the stagnation point of a 0.305 m (1 ft) radius sphere in radiation equilibrium at the flight condition. The real-gas heat-transfer rate computation was made by the method of Ref. 13 using the Fay and Riddell[14] expression for stagnation-point heat transfer.

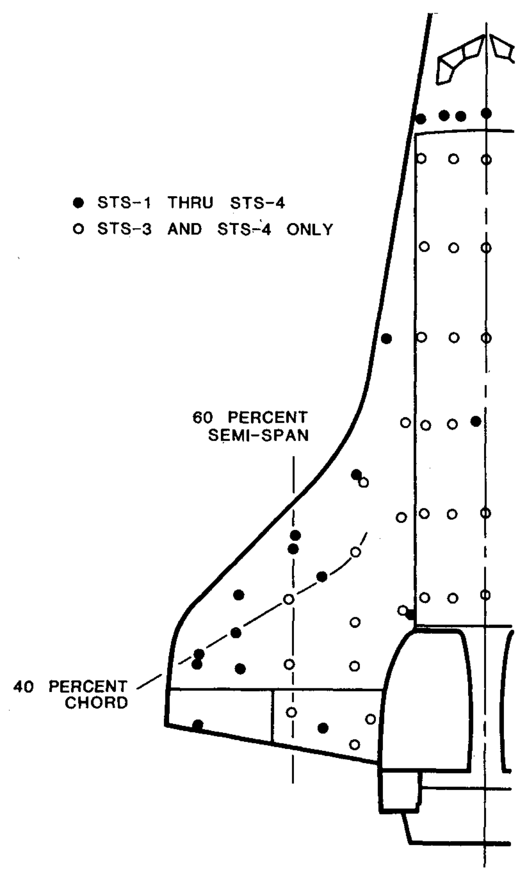

Fig. 1 DFI leeside surface temperature measurement locations.

Data Analysis

Previous investigations of leeside heating to the Shuttle Orbiter or orbiter-like bodies (Refs. 2-6 are typical), confirm that leeside heating levels are acutely sensitive to freestream conditions, vehicle attitude, and vehicle geometry. As a consequence of these sensitivities and the current inability to computationally simulate these complex leeside flowfields, wind tunnel results were applied directly at flight conditions for the purpose of leeside thermal protection system design.[1] This was thought to be a conservative approach based upon prior experience with the Apollo capsule.[1]

While there exists a substantial database of leeside heating information for the Shuttle Orbiter configuration, the data were primarily obtained for thermal protection system design purposes. Analyses of these data are limited in number (i.e., Refs. 4-6), and additionally the analysis discussions are limited to the plane of symmetry. However, these investigators have reached one consistent conclusion; i.e., at high angle of attack, in the wind tunnel, the flow in the leeside separated region is apparently turbulent. Zakkay et al.[5] noted that turbulent boundary-layer theory, predicted the heat-transfer rates satisfactorily. Bertin and Goodrich[6] correlated leeside heating data in the form of Stanton number with a normal shock Reynolds number which resulted in trends indicative of turbulent flow. Lee and Harthun[1] have noted no difference between leeside heating data obtained with or without boundary-layer trips located on the nose of Orbiter models in order to induce turbulent flow on the leeward side. They conclude that "either the turbulent flow relaminarized when it expanded to the leeward side, or the flow on the leeward side was turbulent without the trips."

A more recent investigation[7] resulted in the development of an empirical technique for predicting heating to the leeside centerline based upon knowledge of local leeside surface flow directions as determined from oil-flow tests. The analysis used in the development of the technique assumes turbulent flow. The technique provides remarkably accurate predictions of the heat-transfer levels for the test conditions upon which its development was based.

Flight and wind tunnel results will be presented not only for the leeward plane of symmetry, but also for the wing. It will

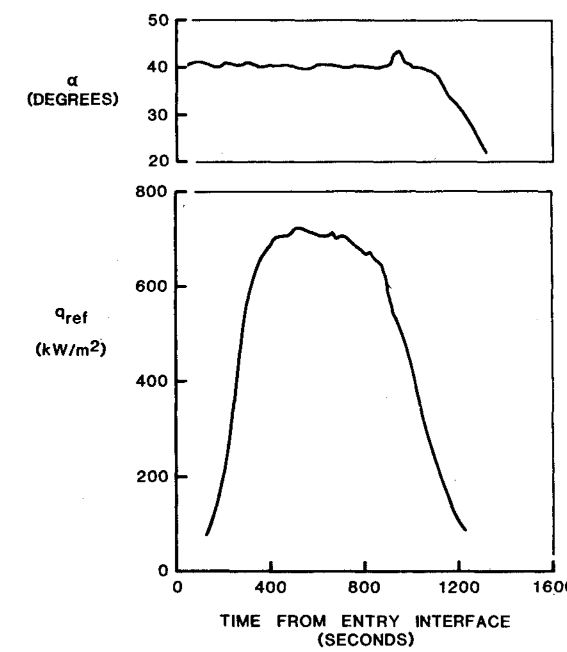

Fig. 2 STS-3 angle of attack and reference heating-rate histories.

be shown that for the fuselage, the flow in the separated region in flight is apparently laminar during a substantial portion of entry, unlike the wind tunnel results which appear universally turbulent. For the wing, the results are somewhat different than for the plane of symmetry. However, the general conclusion is the same. The state of the flow in the separated region in flight was apparently different from that obtained in the wind tunnel at similar test conditions.

Data Presentation

The presentation of data and accompanying discussions contained in the remaining sections of this paper are organized with respect to geometric location of measurements (i.e., fuselage centerline or wing) and are limited to conditions where the vehicle angle of attack is nominally 40 deg. This limitation results from the nature of the Orbiter entry angle-of-attack history. During the STS-2 and STS-3 atmospheric entries, the Orbiter angle of attack was initially set at 40 deg and maintained constant at that nominal value until the Orbiter had passed through the maximum aerodynamic heating portion of the trajectory. (Trajectory control was accomplished by banking the vehicle about the velocity vector.) Thereafter the angle of attack continually decreased as the Orbiter transitioned to aircraft-like flight. While flying at the constant 40-deg angle of attack, the freestream Mach and Reynolds numbers varied over a wide range. During the angle-of-attack rampdown, however, only a single Mach/Reynolds number combination was experienced for any given angle of attack. Consequently, one can observe from the flight data, the effects of Mach and Reynolds number variations on heat transfer to the Orbiter only at 40 deg angle of attack. At other angles of attack, one cannot observe trends with freestream parameters, as only one flight condition is traversed at each angle of attack. The conference paper upon which this manuscript is based contains the available data for angles of attack of 35 and 30 deg. Space limitations have dictated their omission from this article, however.

Fuselage Centerline

Heat-transfer distributions along the leeside plane of symmetry are presented in Fig. 3a for several trajectory points from STS-3. These data were obtained over a Mach number range from 23 to 11 and a freestream Reynolds number range R_L from 0.5 to 5×10^6. The data are nondimensionalized by the heat-transfer rate to the stagnation point of a 0.305 m (1 ft) radius sphere at the flight condition. The q/q_{ref} parameter was used for application of wind tunnel results to the flight environment for the purpose of thermal protection system design.[1] Several important observations are made from the data of Fig. 3a.

1) The measured heat-transfer rates for the leeward centerline at no time exceed 1% of the reference heating rate over this entire portion of the entry flight regime, which includes the time period normally considered to be that of "peak" aerodynamic heating. (Peak windward side heating rates were on the order of 10-15% of the reference rate.)

2) The "roller-coaster" nature of the longitudinal distributions attests to the complexity of the vortex-dominated leeside flowfield.

3) A perturbation in the nature of the flowfield apparently occured between the trajectory points at Mach 17.3 and 13.5 which resulted in a substantial increase in the general levels of the nondimensional heating rate values. This phenomenon will be shown to be the result of apparent laminar to turbulent transition of the flow in the separated region.

Typical wind tunnel results are presented in Fig. 3b. These data[7] were obtained at Mach 6 and 10 over a Reynolds number range consistent with that of the flight data. Additional data[15] from the Orbiter aerothermodynamic design data base, obtained at Mach 8 but not included in this figure (in order to preserve some semblance of clarity), would fall within the bounds of that data shown. The wind tunnel data are strongly Reynolds-number dependent.

Comparison of the flight and wind tunnel data indicates that the nondimensional heat transfer experienced in flight was substantially less than that observed in the wind tunnel for freestream Reynolds numbers of 2×10^6 or less. At higher Reynolds numbers, the flight results were roughly equivalent to those observed in the wind tunnel.

As previously mentioned, Bertin and Goodrich[6] proposed the use of a Reynolds number evaluated downstream of a normal shock for correlation of leeward surface heat-transfer rates. This "normal shock Reynolds number" is a function of both freestream Mach number and freestream Reynolds number, and thus provides a single, representative parameter with which to characterize the flowfield. The parameter, as will be used herein, differs from its use in Ref. 6 only by selection of the characteristic length. For data correlations as a function of the normal shock Reynolds number, the heat-transfer data will be presented in the form of Stanton number based upon stream total enthalpy.

Figure 4 presents both flight and ground test[6,7,15] heat-transfer data presented as Stanton number as a function of normal shock Reynolds number for several leeward centerline locations. In Fig. 4a it is seen that, as reported in Ref. 6, a turbulent correlation fits the ground test results well. There appears to be one correlation line which represents the level of the data of Ref. 6, and another which represents the level of the data of Refs. 7 and 15. Reference 6 data were obtained in a hypersonic shock tunnel while the data from Refs. 7 and 15 were obtained in conventional hypersonic wind tunnels. It is

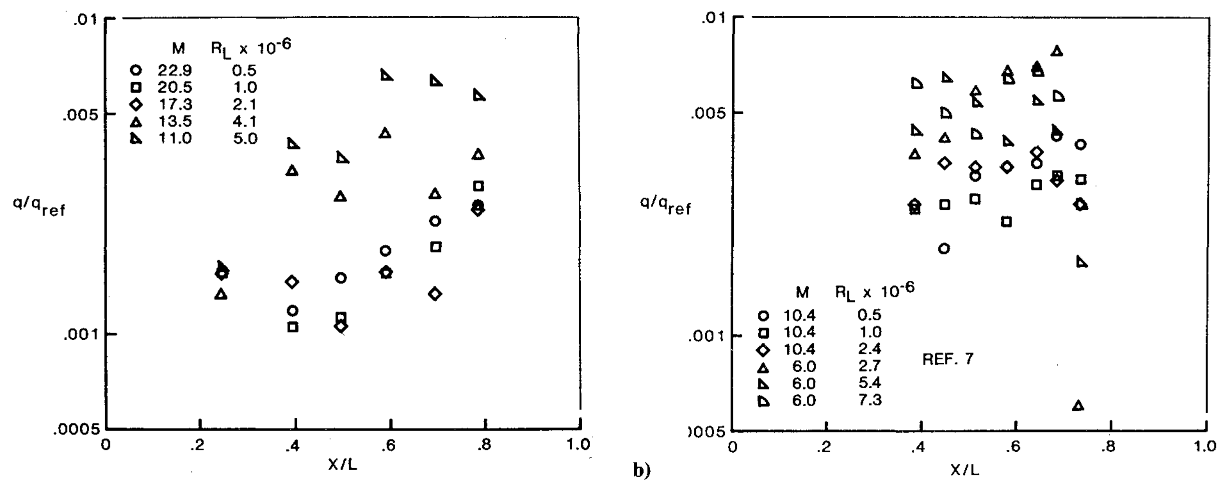

Fig. 3 Leeside centerline heat-transfer distributions, $\alpha \simeq 40$ deg. a) Flight, STS-3. b) Wind tunnel.

postulated that the differing Stanton number levels result from differences in the location of the virtual origin of the turbulent flow in the viscous shear layer in the two types of facilities. Comparison of the flight and ground test results suggests that, in flight, the viscous layer apparently remained laminar for normal shock Reynolds numbers below approximately 0.4×10^6, and then experienced a transition to turbulent flow. Thus the flight heating rate levels were less than those inferred by the ground test results for much of the entry. Similar comparisons between flight and ground test results are observed at other leeside centerline locations (Figs. 4b-d). On STS-3, the laminar-to-turbulent transition was apparently "tripped" as a result of vehicle maneuvering, for the transition correlates precisely with the completion of a vehicle bank reversal. Similarly, on STS-2, transition was apparently "tripped" by vehicle maneuvering (Fig. 4c), as the transition correlates precisely with the occurrence of a flight test maneuver which consisted of a body-flap pulse, with resultant small angle-of-attack fluctuations. There is some evidence of apparent attempts by the flow to "relaminarize" after the transition (Fig. 4a), but such an observation is merely conjecture.

The flight/wind tunnel data comparisons indicate two significant observations:

1) Transition of the separated flow occurred at much higher Reynolds numbers in flight than in the wind tunnel.

2) When the flow in the separated region is turbulent, the flight and wind-tunnel Stanton numbers are comparable.

These results are consistent with those observed for the leeside of the Apollo capsule.[6]

Wing

The nondimensional heat-transfer distributions along the wing chord at the 60% semispan location are presented in Fig. 5a for several STS-3 trajectory points. The following observations are made from Fig. 5a:

1) As was observed on the fuselage centerline, measured heat-transfer rates at this spanwise location are very low, generally less than 1% of the reference heat-transfer rate.

2) The trends of the data with Mach number and Reynolds number are quite consistent. In general, the nondimensional heat-transfer levels increase as a result of the combined effects of decreasing Mach number and increasing Reynolds number.

Typical wind tunnel results[15] obtained at Mach 8 for this spanwise location are shown in Fig. 5b. The trends in the wind tunnel results are not unlike those observed in the flight data. However, at equivalent values of freestream Reynolds numbers, the nondimensional heat transfer experienced in flight was substantially less than that observed in the wind tunnel.

A more revealing view of the heat transfer to the wing leeside may be obtained by considering the spanwise distribution of heat transfer along a line of constant chord, x/c. Figure 6a presents such spanwise distributions along the 40% wing chord for STS-3. Heating-rate levels are once again observed to be very low. Except at the highest Reynolds

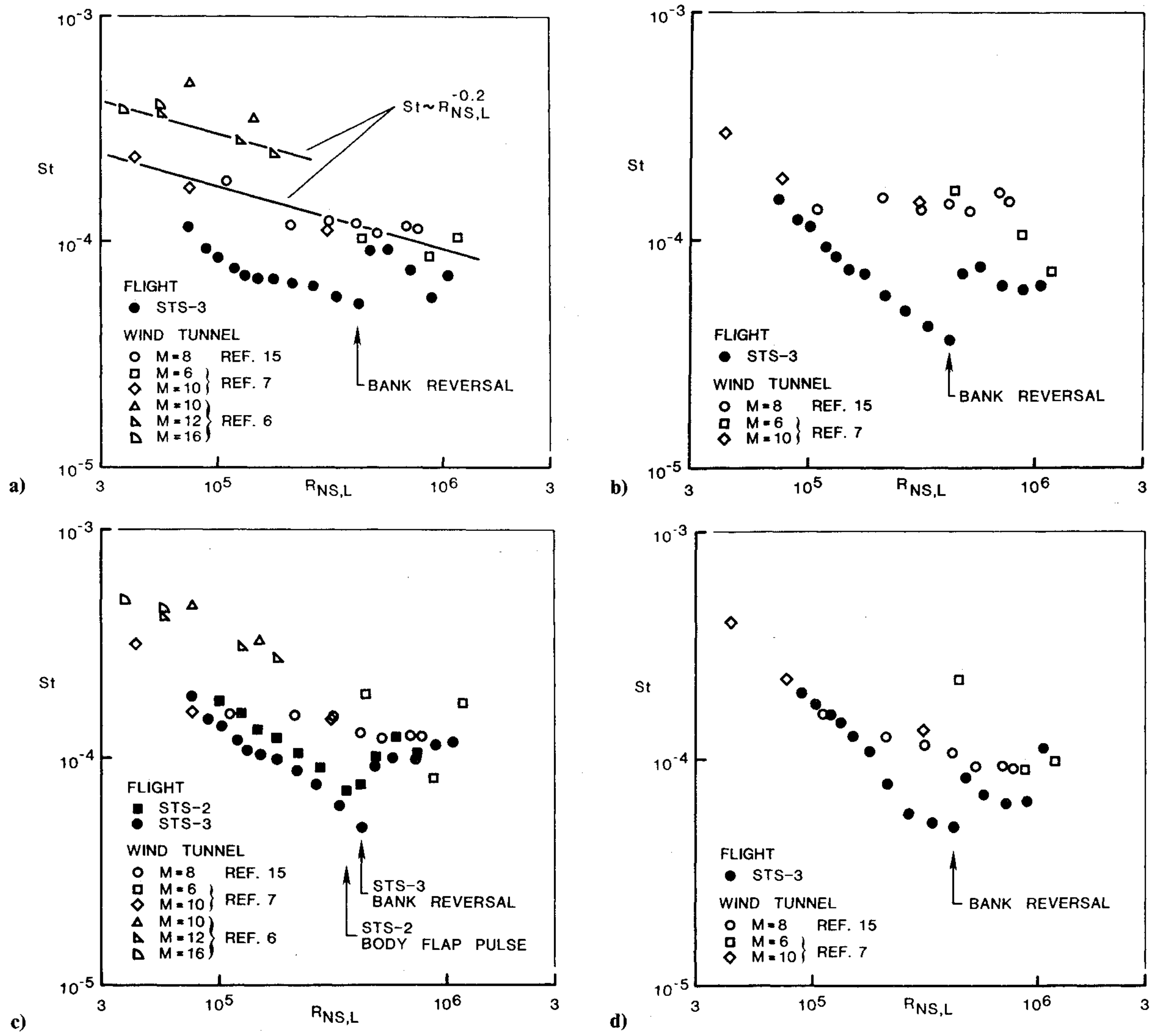

Fig. 4 Stanton number vs normal shock Reynolds number for points on the leeside centerline, $\alpha \simeq 40$ deg. a) $X/L \simeq 0.40$. b) $X/L = 0.50$. c) $X/L \simeq 0.60$. d) $X/L \simeq 0.70$.

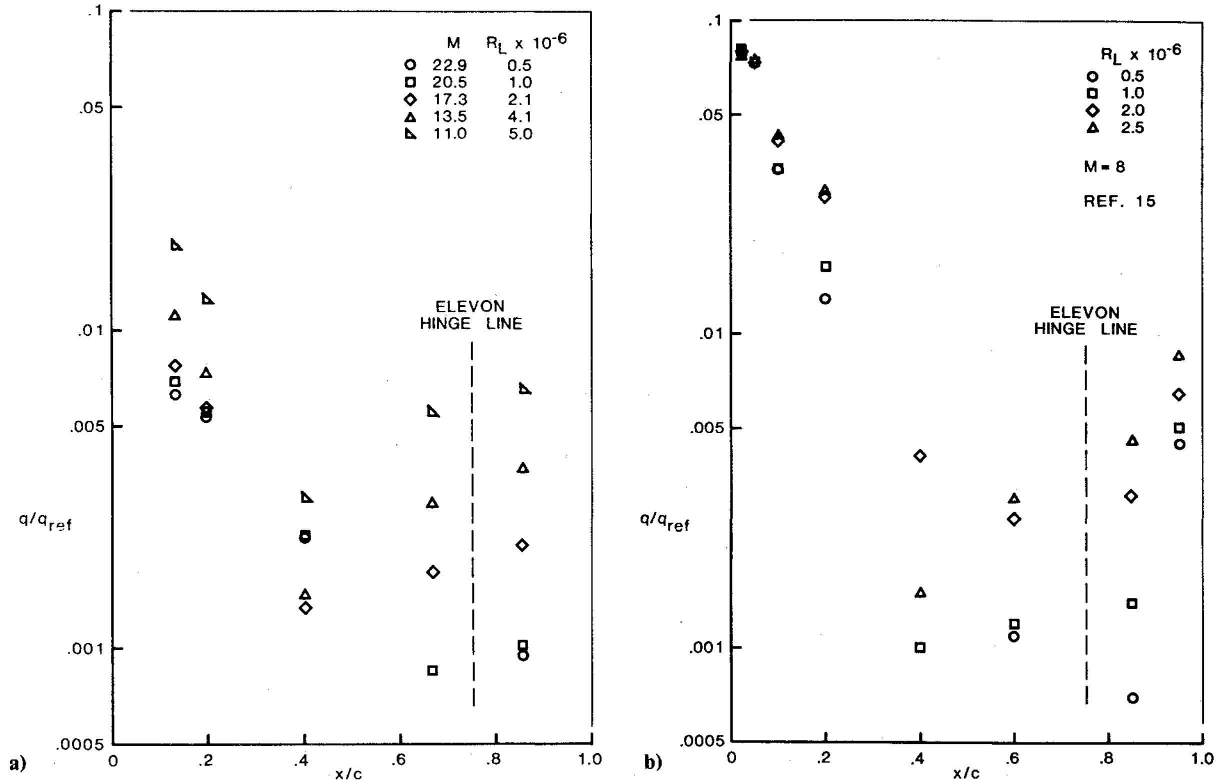

Fig. 5 Chordwise heat-transfer distributions on the wing leeside at the 60% semispan, $\alpha \simeq 40$ deg. a) Flight, STS-3. b) Wind tunnel.

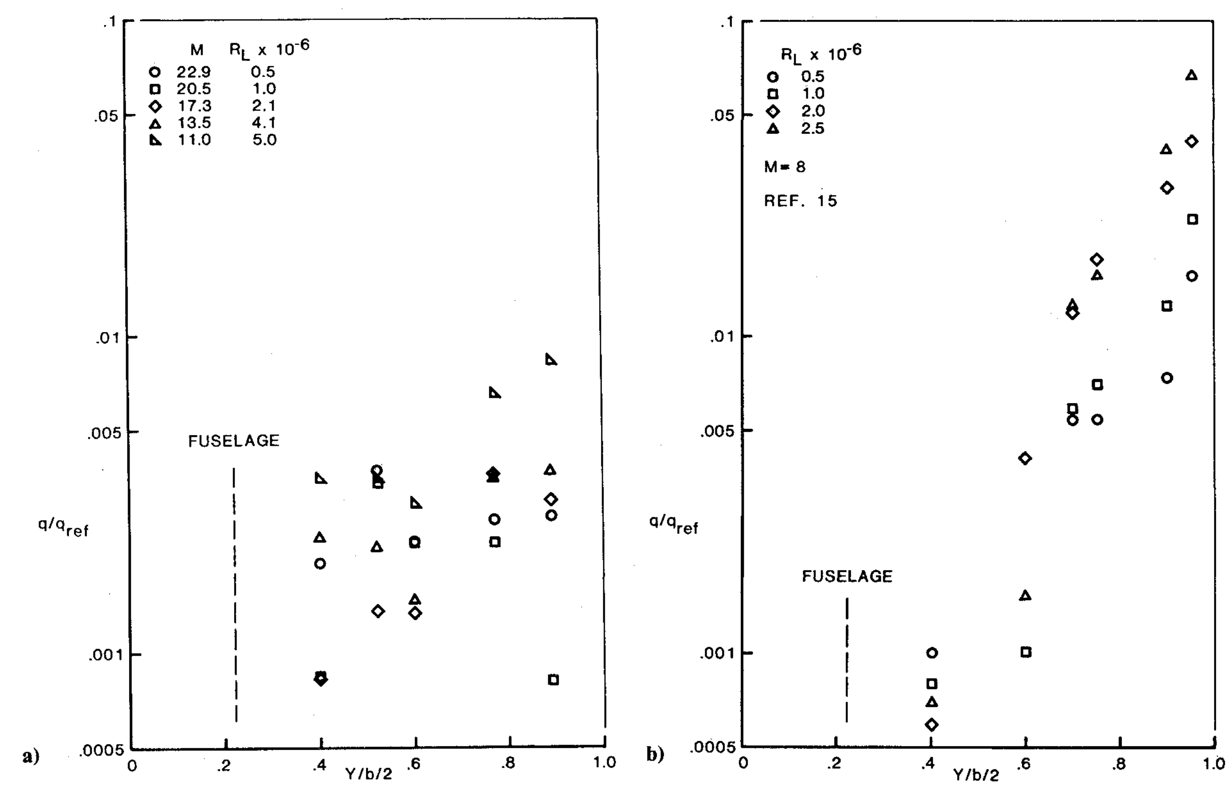

Fig. 6 Spanwise heat-transfer distributions on the wing leeside at the 40% chord, $\alpha \simeq 40$ deg. a) Flight, STS-3. b) Wind tunnel.

number/lowest Mach number, the nondimensional heating-rate levels do not vary dramatically with spanwise location. There is evidence of heating peaks and valleys which move spanwise as the Mach/Reynolds numbers change. These are most probably related to flow disturbances emanating from the region of the leading-edge fillet of the Orbiter's double-delta wing, and from the region of interaction between the Orbiter bow shock and wing leading-edge shock.

Wind tunnel data for the 40% chord line are shown in Fig. 6b. The wind tunnel results indicate a strong dependence on spanwise location which is not evident in the flight data. Outboard of approximately the 60% semispan location, the nondimensional heat transfer observed at equivalent freestream Reynolds numbers in the wind tunnel is greater than that experienced in flight by factors as great as an order of magnitude.

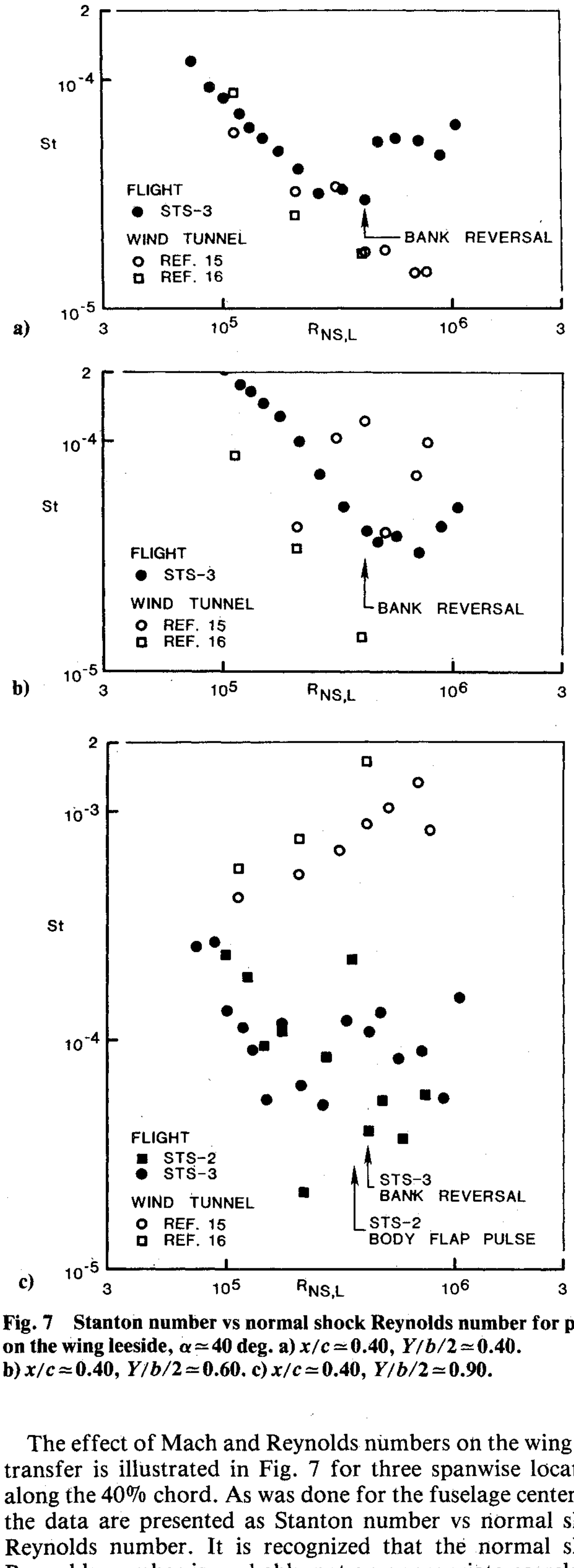

Fig. 7 Stanton number vs normal shock Reynolds number for points on the wing leeside, $\alpha \approx 40$ deg. a) $x/c \approx 0.40$, $Y/b/2 \approx 0.40$. b) $x/c \approx 0.40$, $Y/b/2 \approx 0.60$. c) $x/c \approx 0.40$, $Y/b/2 \approx 0.90$.

The effect of Mach and Reynolds numbers on the wing heat transfer is illustrated in Fig. 7 for three spanwise locations along the 40% chord. As was done for the fuselage centerline, the data are presented as Stanton number vs normal shock Reynolds number. It is recognized that the normal shock Reynolds number is probably not an appropriate correlation parameter for wing data because of the complex shock structure ahead of the wing. However, because it is a parameter which, as has been previously discussed, is a function of both freestream Mach and Reynolds numbers, it is thought to be appropriately useful as a parameter to indicate trends in the wing data under conditions of varying freestream Mach and Reynolds numbers. At the most inboard location (Fig. 7a), the trends of the flight and wind tunnel[15,16] results are in good agreement at values of the normal shock Reynolds number below approximately 0.4×10^6. At that stream condition, the flight data trend indicates the onset of a flow transition, which was probably triggered by the bank reversal, and was not observed in the wind tunnel. It appears that for this inboard wing location, the flow in the separated region was laminar both in flight and the wind tunnel. Note that the wind tunnel test conditions which appear to produce laminar flow at this wing point are the same as those which were previously shown to yield turbulent flow at the symmetry plane. Such dissimilarities in the test results attest to the complexity of the leeside separated flowfield.

The trends of the flight data obtained at the 60% semispan location (Fig. 7b) are similar to those observed at the inboard location. The corresponding wind tunnel results, however, do not provide readily observable trends, but rather indicate large scatter. (The two wind tunnel data sets were each obtained on 0.0175-scale models in the AEDC Tunnel B.) Flow transition in the wind tunnel is possibly incipient at these stream conditions and thus the extreme data scatter observed. For those data which exhibit a laminar trend, the wind tunnel results are lower than the corresponding flight results.

At the most outboard wing location (Fig. 7c), the flight/wind tunnel data comparison is substantially different. For this location, the wind tunnel results imply the presence of transitional flow in the separated region over this range of normal shock Reynolds number, while the flight results are apparently influenced by laminar flow. The dynamic character of the flight data observed at this location is probably the result of vehicle sideslip excursions. It has been previously observed[10] that the heat transfer to some Orbiter leeside surfaces is extremely sensitive to vehicle sideslip. The flight data presented in Fig. 7c can be shown to correlate with vehicle sideslip angle.

Summary and Conclusions

Flight-derived aerodynamic heat-transfer data for the Orbiter leeside centerline and wing surfaces have been presented and compared with appropriate ground test results. Heat-transfer rates to these surfaces in flight are very low, in general less than 1% of the 0.305 m (1 ft) radius sphere reference value (5-10% of windward side heating rates). Flight heating levels are, in general, less than those which are inferred from the ground test results. This result is apparently due to laminar-to-turbulent transition of the flow in the separated region having occurred at a much larger Reynolds number in flight than in the wind tunnel. When the state (laminar/turbulent) of the separated flow is the same both in flight and in the wind tunnel, the levels of heat transfer expressed in the form of Stanton number are comparable. Although the Orbiter configuration is in no way similar to the Apollo entry capsule, it is interesting to note that these results are consistent with those previously observed for the Apollo capsule. The flight/wind tunnel data comparisons confirm the appropriateness of the Orbiter thermal protection system design approach which utilized wind tunnel results applied directly at flight conditions to define the flight leeside aerothermodynamic environment.

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