# EXPERIMENTAL AND NUMERICAL STUDY ON TRANSITION CHARACTERISTICS AT SUPERSONIC SPEED

**Hiroki Sugiura, Kenji Yoshida, Masayoshi Noguchi, Yoshine Ueda, Naoko Tokugawa, Shohei Takagi**
**(National Aerospace Laboratory of Japan)**

## Abstract

*In order to increase the lift-to-drag ratio of next-generation supersonic transport (SST), the consistent prediction of boundary-layer transition on the wings and fuselage is critical to the engineering design and analysis of next-generation SST. Transition on a Sears-Haack (S-H) body and a supersonic natural laminar flow (NLF) wing is studied numerically and experimentally at supersonic speed. Several tunnels with different freestream turbulence levels were used in order to quantify the influence of freestream turbulence on transition. As a result, transition characteristics of the S-H body at Mach 1.2 and the NLF wing at Mach 2 were obtained and, the influence of freestream disturbance on the transition was investigated. Results obtained by a practical transition prediction method based on a compressible $e^N$ method agreed well with the experimental results of the natural laminar flow wing where the NLF effect exists. However, the calculated results did not agree with the measured results in the case of the S-H body. Since this seems to be caused by the large freestream disturbance present at Mach 1.2, further transition measurement in a low-disturbance facility at Mach 2 is necessary.*

## 1 Introduction

Civil aircraft manufacturers have made significant efforts to reduce aircraft drag over the last three decades to reduce the specific fuel consumption of aircraft. Friction drag makes up more than one-third of the drag of next generation supersonic transport (SST) and its reduction by delaying the boundary-layer transition is mandatory.

Friction drag has two important generators, wings and fuselage; these together account for over 80% of the total friction drag [1]. Achievement of laminar flow over up to 60% chordwise station of a natural laminar flow (NLF) wing of an SST configuration has been estimated to result in a total-aircraft drag reduction of 7% [2]; achievement of laminar flow over portions of the forward fuselage of transonic transport aircraft has been estimated to result in a total-aircraft drag reduction of 4% [3].

In view of the large potential reductions in drag, the consistent prediction of boundary-layer transition on the wings and fuselage is critical to the engineering design and analysis of next-generation SST.

A transition prediction technique, the compressible $e^N$ method [4], is currently drawing attention as a strong candidate for an effective design tool of SST. Successful application of the $e^N$ method for the prediction of the transition location in design effort relies on the correlation of experimental transition data with the calculated disturbance amplification rates and amplitude factors, called N-factors.

However, to date, published correlations of the N-factor are very limited and clearly insufficient for supersonic flows. The purpose of the present research is to acquire correlations of experimental transition data with the calculated disturbance amplification rates and amplitude factors under supersonic conditions.

The reason for the dearth of correlations under supersonic conditions is that these correlations are strongly affected by freestream disturbances in supersonic wind tunnels, which can be entrained in the laminar boundary layer. Supersonic wind tunnels have historically exibited orders of magnitude greater disturbances compared with flight at high altitudes.

Hence, most of the efforts in the past experimental research can be classified into two main groups. The first group includes significant effort to thoroughly investigate three-dimensional (3-D) boundary-layer transition under incompressible flow conditions, because compressibility does not change the fundamental physics of the 3-D boundary layer. However, the transition prediction requires the consideration of the role of compressibility. The second group includes prodigious efforts to create a tunnel environment at high Reynolds numbers which approximates that of flight at high altitudes. They are termed *quiet* tunnels. Approximately only five quiet high-Reynolds-number supersonic wind tunnels exist in the world today (1997) [5].

A pessimistic conclusion could be that correlations of the N-factor for supersonic flows are impossible without using these tunnels. In spite of this negative situation, the predictions of transition on SST configurations must be made.

In the present research, we used continuous types of supersonic wind tunnels with low amplitude of pressure fluctuation in the range of the locally maximum amplifying instabilitiy, because a forced disturbance of a frequency will excite a boundary-layer eigenmode having the same frequency but a different wavelength [4].

However, the overall staitic pressure fluctuation (normalized by dynamic pressure) of the facilities lies in the range from 0.2 to 0.4%. According to several studies, such as Chen [6], the fluctuation of that order has a possibility of changing the transition characteristics entirely.

Thus, we planned to check the measured transition characteristics with three different approaches. Firstly, we compared the measured results with calculated results using two $e^N$-method codes based on the compressible linear stability theory. Secondly, we also used several supersonic wind tunnels with different freestream turbulence levels with a view to quantifying the influence of freestream turbulence on transition. The strength and spectral content of freestream disturbances were measured in these tunnels in detail. And finally, we have conducted some transition measurements in a low-Reynolds-number tunnel of in-draft type whose freestream turbulence level approximates that of flight at high altitudes.

## 2 Experimental Procedure

### 2.1 Freestream Turbulence Levels of the Test Facilities

As pointed out by Morkovin [7], freestream disturbances are velocity fluctuations, pressure fluctuations and temperature fluctuations. Although little information on velocity fluctuations has been published for supersonic wind tunnels, it is now clearly established that pressure fluctuations play a major role [4].

The pressure fluctuations in ground facilities are usually characterized by a single parameter, the static pressure fluctuation normalized by dynamic pressure ($C_{Prms}$). Although, pressure fluctuations were measured with bandwidth from 125Hz to 100kHz, all the $C_{Prms}$ in the present paper is calculated with bandwidth from 125Hz to 20kHz for the sake of comparison.

For Mach 2 tests, we used the FHI tunnel and a continuous supersonic tunnel from ONERA (S2MA).

The $C_{Prms}$ of 0.05% has been reported at Mach 2 in the FHI tunnel [8], which lies within the range of the static pressure fluctuation at Mach 2 in flight, which is from 0.03 to 0.09% [9]. In-draft tunnels realize a very low turbulence environment at low Reynolds numbers, for they differ from other supersonic tunnels in not having disturbance sources, such as pressure control valves and blowers, upstream of the test section. Its test section is 0.61m in both length and width.

$C_{Prms}$ of 0.20% in the S2MA has been reported at Mach 2 [10] which is one order of magnitude lower than the value of 2.0% in a typical blowdown tunnel from ONERA called

ONERA-S3 [9]; the test section of S2MA is 1.93m wide and 1.75m high.

For Mach 1.2 tests, we used four test facilities: the FHI tunnel, the S2MA, and a continuous transonic wind tunnel from NAL (NAL-TWT), which has a porous- and a slotted-wall test section cart. At transonic speeds, wind tunnels have large pressure fluctuations due to hole/slot resonance in porous- or slotted-wall test sections. According to our measurement of static pressure fluctuation in the laminar boundary layer on an axisymmetric model, $C_{Prms}$ was 1.50% at Mach 1.2 in the FHI tunnel; $C_{Prms}$ was 2.25% in the S2MA tunnel at Mach 1.2. The $C_{Prms}$ of 0.34% and 1.01% have been reported at Mach 1.2 in the porous- and slotted-wall test section carts of NAL-TWT, respectively [11]; both test sections are 2m in both length and width.

## 2.2 Experimental Definition of Transition Location

Since there is considerable scatter in transition data due to the different methods used for transition measurement, consistent definitions of the transition point for supersonic flows are required for the correlation [4].

For supersonic speeds, an analysis of a large body of heat-transfer data by Bertram and Neal [12] has shown that the choice of the virtual origin at the point of maximum surface temperature gives the most consistent results. Accordingly, the location of the maximum surface temperature is defined as the transition point in the present research.

It was established by Owen [13] that the location of peak voltage fluctuation determined using hot-film sensors on a surface coincides with the maximum burst frequency location and the maximum surface temperature location. Hence, the location of peak voltage fluctuation is also defined as the transition point in the present research. Examples of the hot-film rms voltage and transition point are given in Fig. 1; an arrow indicates the transition point.

We used hot-film sensors and an infrared camera for the present transition measurement. Figure 2 shows the effect of total pressure on the transition Reynolds numbers measured using hot-film sensors and the infrared camera. Transition locations identified using the infrared camera agree well with those obtained using hot-film sensors. This confirms Owen's report and proves the consistency of the present definition of the transition points.

## 2.3 Pressure Distribution on the Testing Configurations

Aiming for the consistent prediction of boundary-layer transition on the wings and fuselage of next-generation SST, we chose a Sears-Haack (S-H) body and an NLF wing as testing configurations.

With a view to diminishing the influence of surface imperfections on the transition, the arithmetic average roughness amplitudes of all following models are set below 0.2μm which is lower than the 0.25μm of a well-known transition model, the Arnold Engineering Development Center (AEDC) 10-degree cone [9].

### *2.3.1 Sears-Haack Body*

When the volume is given, the Sears-Haack (S-H) body has the smallest drag coefficient in the supersonic flow; it serves as a guideline for the forward fuselage configuration for supersonic transports.

We used the same S-H body model in all the test facilites. The S-H model has an unsteady pressure transducer and four hot-film sensors mounted flush with the surface.

Prior to the transition measurements, pressure distributions on the S-H model were compared with the CFD results calculated by an Euler code which is validated by experiments and another numerical code [14]. Configurations and pressure distributions of the two S-H bodies with different diameters, S-H (I) and S-H (II), and 10-degree cone are shown in Figs. 3 and 4, respectively.

Figure 4 shows that a large favorable pressure gradient exists on the S-H bodies compared to the sharp cone. Hence, streamwise instability (Tollmien-Schlichting wave) amplification will be the dominant transition cause of laminar flow because crossflow is negligible in the boundary layer on axisymmetric shapes.

### *2.3.2 Natural Laminar Flow Wing*

The NLF wing was designed at the National Aerospace Laboratory, Japan, for an SST configuration [15]. The principle behind the NLF wing is to keep the growth of these disturbances within acceptable limits so that 3-D and nonlinear effects do not cause breakdown to turbulence. With this philosophy, one only deals with linear disturbances and thus, the difficulties with transition prediction do not directly arise [16]. These features make the NLF wing suitable for application to compressible linear stability calculation.

We used large-scale models to achieve as high a Reynolds number condition as possible.

A 23.3% scale model of an experimental airplane, which has 1.87m length and 1.1m span width, was used in S2MA. The model has 37 ports on the port wing; it has 28- and 40-element hot-film sensors mounted flush with the starboard wing surface at 30 and 70% semi-spanwise stations, respectively.

We used the 15.7% scale half-model with a configuration of 1.139m length and 0.37m span width in the FHI tunnel; the model (Fig. 3) has 28- and 32-element hot-film sensors at 30 and 70% semi-spanwise stations, respectively. With a view to avoiding any influence of static pressure ports on the transition, a pressure model of the identical configuration with 40 static pressure ports was designed. (See details of the NLF wing models in ref.15.)

Prior to the transition measurements, pressure distributions for the NLF wing models were compared with the CFD results obtained using the 3-D Navier-Stokes code. This code uses the Baldwin-Lomax model and was previously validated by experiments and other numerical codes [17].

Pressure distributions at the 70% semi-spanwise station are shown in Fig. 5. The pressure decreases very rapidly in the leading edge region of the wing and maintains nearly zero gradients downstream at angles of attack of 2 to 5 degrees, such that the development of crossflow velocity components which amplify crossflow instability is minimized.

## 3 Linear Stability Analysis Code Based on the Compressible $e^N$ Method

We analyzed the transition characteristics of the S-H body and the NLF wing using a linear stability analysis code based on the 3-D compressible linear stability theory and the $e^N$ method with an envelope method strategy.

Compressible laminar boundary layer profiles were calculated using methods developed by Kaups & Cebeci [18]. N is defined as an integration of amplification rates of small disturbances with several frequencies. As for the wing, N was integrated along a polar arc normal to the leading edge in the code, because the Kaups & Cebeci method was formulated in a polar coordinate system [18]. Nonparallel and curvature effects are not included in the present code.

The code is validated experimentally and using another linear stability code developed by Arnal et al. based on the envelope method [4]. The chordwise distribution of the N values and propagation directions of disturbances on a 10-degree cone, the S-H body and the NLF wing obtained using the two codes agreed very well [19] .

## 4 Transition Characteristics of the Sears-Haack Body at Mach 1.2

### 4.1 Pressure Fluctuations in the Test Facilities at Mach 1.2

We measured static pressure fluctuation on the S-H body model at Mach 1.2 to determine the freestream turbulence levels of the facilities. Power spectrum density curves of pitot pressure fluctuations in the four facilities at Mach 1.2 are shown in Figs. 6 and 7. All porous facilities, that is, all the facilities except the slotted cart of NAL-TWT, have peaks in the curve associated with hole resonance.

In the slotted cart of NAL-TWT, most of the energy of the fluctuation is concentrated below 1kHz; the amplitude of the fluctuation is very low above 5 kHz, which is well below the range of the frequency of the locally maximum amplifying disturbances at supersonic speeds.

The shape of the curve in S2MA differs little with the change of the freestream Reynolds number, unless the boundary layer is laminar (40kPa<$P_0$<70kPa). This implies that the influence of freestream disturbance remains almost the same in the range of test Reynolds number. Hence, by varying the total pressure in S2MA, the unit Reynolds number effect on transition can be studied.

### 4.2 Transition Locations on the Sears-Haack Body

The effect of freestream turbulence level on transition locations on the two S-H bodies and on the 10-degree cone is shown in Fig. 8. The dot-dash-line shows the correlation drawn from transition data of the AEDC 10-degree cone in 23 major tran/supersonic wind tunnels by Dougherty and Fisher [9]; the two dashed lines correspond to its ±20% band.

The figure shows that the transition locations move downstream as the magnitude of favorable pressure gradient increases. It also shows that the local slopes of the curves for each configuration coincide. A general conclusion might be drawn that the effect of the freestream turbulence level on transition locations works in a similar manner on these axisymmetric bodies; however, much more data is necessary to validate this.

The slope of the curve seems to suddenly change at $C_{Prms}$ =1%; this is because the data are measured in the slotted tunnel for $C_{Prms}$ <1% and the data in the perforated tunnels, which have large sound disturbances associated with hole resonance, for $C_{Prms}$ >1%.

### 4.3 Comparisons with Compressible Linear Stability Analysis

The transition locations of S-H (II) in the three different test facilities are shown in Fig. 9. Three kinds of transition points are shown in the figure, namely, onset and end of transition, and peak voltage fluctuation. The transition locations move downstream as $C_{Prms}$ increases. The transition Reynolds number for the slotted cart of NAL-TWT is much larger than those for other facilities; only the onset of transition was detected in the slotted cart.

The figure shows that in each facility, all of the data fall above straight lines with the same slope. However, the slope for each facility seems to differ. There are many possible contributing factors for the transition represented in the figure; however, the general conclusions can be drawn that the transition location on the Sears-Haack body is proportional to the inverse of the freestream Reynolds number and that its proportionality constant is determined by the freestream turbulence level; these trends seem to prevail over the range of the present investigation.

Figure 10 shows comparison of the distributions of the power spectrum density of hot-film voltage and the calculated amplification factor. The frequency of an unstable wave showed good agreement between experiment and analysis.

Figure 11 shows the comparison of the calculated and measured transition locations on the S-H body. Despite the good agreement in the frequency of the unstable wave, the transition locations and N values showed poor agreement.

It is possible that large freestream disturbance present in the porous/slotted test sections greatly affected the transition measurement results. This is supported by the fact that the data for the slotted cart of NAL-TWT, that is, the facility with the least freestream turbulence level, shows good agreement with the N-factor curve.

To clarify this point, further transition measurement in the slotted cart and is necessary. We are also planning to conduct transition measurement at Mach 2 in a low-disturbance facility, that is, the FHI tunnel, because transonic facilities have large freestream disturbances.

## 5 Transition Characteristics of the Natural Laminar Flow Wing at Mach 2

### 5.1 Pressure Fluctuations in the Test Facilities at Mach 2

We measured pitot pressure fluctuation in the tunnels at Mach 2 to determine their freestream turbulence levels of the tunnels.

Figure 12 shows the pitot pressure fluctuation plotted against total pressure at M=2. Figure 13 shows the power spectrum density curves of pitot pressure fluctuations in the S2MA and FHI tunnel at Mach 2. It is clear from the figures that the total pressure has little effect on pitot pressure fluctuation, i.e., the freestream turbulence level in S2MA, so that by varying the total pressure in S2MA, the unit Reynolds number effect on transition can be studied.

The average pitot pressure fluctuation at Mach 2 was 0.29% in S2MA and a very low value of 0.05% in the FHI tunnel. This value lies within the range of the pitot pressure fluctuation at M=2 in flight, which is from 0.02 to 0.06% [9].

### 5.2 Transition Locations on the NLF Wing

Figure 2 shows the effect of angle of attack on the transition Reynolds numbers at the 70% semi-spanwise station measured using hot-film sensors and the infrared camera. All the data fall above a single curve. This is noteworthy because it shows that there is no unit Reynolds number effect on the transition Reynolds number of the NLF wing; this is quite similar to results for flat plates and sharp cones. The reason for this similarity can be found in the very flat pressure distributions of the NLF wing, as in the cases of flat plates and sharp cones.

Figure 2 shows that the longest laminar region at the 70% semi-spanwise station is achieved at an angle of attack of 5 degrees, which is far from the design angle of attack of 2 degrees. The transition location moves rearwards with increasing angle of attack from 0 degrees. This indicates that the NLF effect of the pressure distribution exists in a wide range of angles of attack from 1.5 to 5 degrees on the outer part of the wing, where a narrow accelerating region and flat-top pressure distributions are realized.

### 5.3 Comparisons with Compressible Linear Stability Analysis

Figure 14 shows a typical stability analysis result at the 70% semi-spanwise station at M=2; the coordinates correspond to the envelope of N values. The crossflow and streamwise instabilities are not distinguished here; the growth of disturbances of different frequencies is computed for whichever propagation angle of disturbance that gives the maximum growth rate. According to a transition measurement on a 10-degree cone in S2MA by Fontaine et al. [10], N=7 corresponds to the end of the transition at M=2 in the tunnel. The figure indicates that the end of the transition is located at the 36% chordwise station when N=7. Figure 15 summarizes the measured transition location and that calculated by the same approach in the range of angles of attack from 1.5 to 5° where the existence of the NLF effect is inferred from the shape of the pressure distribution in Fig. 11. It is clear from Fig. 15 that the calculated locations agree quantitatively with the measured locations.

This implies that the present analytical results reproduce the real transition on the NLF wing satisfactorily when the natural laminar flow effect exists. In other words, when the transition is dominated by streamwise instabilities, the present N value corresponds to the transition location, but when the dominance of crossflow instability grow stronger, the corresponding N value seems to be larger.

## 6 Conclusions

Transition on a Sears-Haack (S-H) body and a supersonic natural laminar flow (NLF) wing is studied numerically and experimentally at supersonic speed. Several tunnels with different freestream turbulence levels were used in order to quantify the influence of freestream turbulence on transition. As a result, transition characteristics of the S-H body at Mach 1.2 and the NLF wing at Mach 2 were obtained and the influence of freestream disturbance on the transition was investigated. Results obtained by a practical transition prediction method based on a compressible $e^N$ method agreed well with the experimental results of the natural laminar flow wing when the NLF effect existed. However, the calculated results did not agree with the measured results in the case of the S-H body. Since this seems to be caused by the large freestream disturbance present at Mach 1.2, further transition measurement in a low-disturbance facility at Mach 2 is necessary.

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