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Effects of High-Speed Tunnel Noise on Laminar-Turbulent Transition

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It is well known that the high levels of noise present in conventional hypersonic ground-test facilities cause transition to occur earlier than in flight. Flight measurements of incoming noise are reviewed and compared with measurements in ground-test facilities, of both conventional and quiet design, at hypersonic and high supersonic speeds. The low noise present in flight is apparently the reason for the very large transition Reynolds numbers sometimes measured in flight, when roughness, crossflow, and other factors are controlled. Design will usually involve consideration of the trend in transition when a parameter is varied. The effect of facility noise on these trends is reviewed. In some cases, the trend of conventional-tunnel data is opposite to the trend in quiet-tunnel data. Thus, transition measurements in conventional ground-test facilities are not reliable predictors of flight performance, except perhaps in special cases.

Nomenclature

k	=	roughness height
M_e	=	Mach number at the boundary-layer edge
M_∞	=	freestream Mach number
$p'_{s,\mathrm{rms}}$	=	rms surface pressure fluctuations
$p_{t,\mathrm{mean}}$	=	mean pitot pressure
$p'_{t,\mathrm{rms}}$	=	rms pitot pressure fluctuations
q	=	dynamic head
Re_b	=	Reynolds number based on freestream conditions and average leading-edge thickness
Re_k	=	Reynolds number based on conditions in the undisturbed boundary layer at the roughness height, $Re_k = U_k \rho_k k/\mu_k$
Re_R	=	Reynolds number based on nose radius
Re_T	=	Reynolds number at transition onset, based on arc length from the leading edge and local conditions at the boundary-layer edge
$Re_{T,s}$	=	Re_T for the smooth-wall case
Re/m	=	unit Reynolds number per meter, usually in the freestream
Re_∞	=	freestream unit Reynolds number
T_e	=	temperature at the boundary-layer edge
X_{end}	=	arc length to end of transition
X_{onset} or x_T	=	arc length to transition onset
x_k	=	streamwise distance to roughness element
x_s	=	streamwise location of the tip of the cone, cm
α	=	angle of attack, deg
δ_k^*	=	displacement thickness at roughness element
θ_c	=	half angle of cone, deg

Introduction

LAMINAR-TURBULENT transition in high-speed boundary layers is important for prediction and control of heat transfer, skin friction, and other boundary-layer properties. However, the mechanisms leading to transition are still poorly understood, even in low-noise environments.[1] Applications hindered by this lack of understanding include reusable launch vehicles, high-speed interceptor missiles, hypersonic cruise vehicles, and reentry vehicles.[2]

The transition process is initiated through the growth and development of disturbances originating on the body or in the freestream.[3] The receptivity mechanisms by which the disturbances enter a boundary layer are influenced by roughness, waviness, bluntness, curvature, Mach number, and so on. The growth of the disturbances is determined by the instabilities of the boundary layer. These instabilities are in turn affected by all the factors determining the mean boundary-layer flow, including Mach number, transverse and streamwise curvature, pressure gradient, and temperature.[1] Relevant instabilities include the concave-wall Görtler instability,[4] the first- and second-mode TS-like instability waves described by Mack,[5] and the three-dimensional crossflow instability.[6] The first appearance of turbulence is associated with the breakdown of the instability waves, which is determined by various secondary instabilities.[7] Local spots of turbulence grow downstream through an intermittently turbulent region whose length is dependent on the local flow conditions and on the rate at which spots are generated.[8]

In view of the dozens of parameters influencing transition, classical attempts to correlate the general transition point with one or two parameters such as Reynolds number and Mach number can only work for cases that are similar to those previously tested. For general flight data, the scatter is large.[9] Transition estimation methods that are reliable for a broad range of conditions will need to be based on an understanding of the physical mechanisms involved.[10]

The simplest and best developed of the mechanism-based methods are the e^N methods, which attempt to correlate transition with the integrated growth of the linear instability waves. Although these methods neglect receptivity and all nonlinear effects, such as wave interactions, nonlinear breakdown effects, roughness, and so on, they have shown promising agreement with experiment.[1] Agreement is fairly good for a variety of conditions where the environmental noise is generally low.[11,12] For incompressible and transonic flow, the e^N method appears to be the state of the art for industrial use, for transition caused by a single instability with primarily linear amplification.[13] Where wave interactions or nonlinear wave amplification dominate transition, nonlinear methods must be used.[14] For supersonic flows, the situation is more uncertain, in part because of the difficulty of accurately computing the mean flows,[15] and in part due to the difficulty of making validation-quality measurements. Even for a sharp round cone at zero angle of attack, reliable agreement of measured and computed wave growth remains to be achieved.[16] The accuracy of many existing high-speed e^N computations remains uncertain, for accurate instability computations require highly accurate mean-flow computations, complete with accurate second derivatives, and it is often not clear whether bluntness effects, wall temperature distributions, and so on were handled with sufficient accuracy.[17] Although wave interaction effects and three-dimensional effects can sometimes be handled with correlations,[18] e^N methods are only an intermediate step along the way to reliable mechanism-based methods.

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Direct simulations of transition[19] and the recently developed parabolized stability equations[20] have in some ways advanced theoretical-numerical work far ahead of the experimental database. The numerical work is not yet able to include complex effects such as roughness, waviness, internal shocks, and most bypasses. Three-dimensional mean flows and their instabilities are only beginning to be treated correctly from first principles.[21] The simulation of bluntness effects[22] and chemistry effects[23,24] from first principles is only beginning. When the numerics are based on the correct physical mechanisms, however, they can provide much more detail regarding the transition process. Experimental work that describes not only the location of transition but also the mechanisms involved is needed to improve these modern theories. The key mechanisms need to be identified, in part through experimental work, and the key numerical results need to be validated experimentally.

Unfortunately, most of the ground-test data are ambiguous, due to operation in high-noise conventional wind tunnels and shock tunnels, with disturbance levels much higher than in flight.[25] The mechanisms of transition operational in small-disturbance environments can be changed or bypassed altogether in high-noise environments.[26]

This paper updates the surveys presented earlier in Refs. 26 and 27. The flight data surveyed here are a small subset of that discussed in Ref. 9.

Noise Levels in Flight and Ground Testing

Transition occurs much earlier in wind tunnels than it does in flight, presumably because of the higher noise levels measured there. Flight data taken from Fig. 1 of Ref. 9 are here combined with ground test data and shown in Fig. 1. References for the first four legend items are detailed in Ref. 9. The reentry-F data are from Ref. 28, and the Sherman data are from Ref. 29. The ground-test data labeled "DiCristina" were digitized from Fig. 15 of Ref. 30, and are from nine sources, for sharp cones at zero angle of attack, near a unit Reynolds number of 2×10^6 per ft, for various wall temperatures. Full citations are given in Ref. 30. The NASA Langley quiet-tunnel data were taken from the tabulations in Ref. 31, for a unit Reynolds number near 2×10^6 per ft. The noisy-facility ground-test data are significantly below the quiet-tunnel data point and below the trend in the flight data.

The sources of disturbances in flight and in ground testing include temperature spottiness, particulates, vorticity fluctuations, and acoustic disturbances.[3] Particulates are a common and rarely discussed problem. In many facilities, model nosetips that begin tests as smooth are sandblasted rough by tunnel particulate.[32] At low speeds, the effect of particles is well known.[33] At high speed, little is known, although Holden has studied controlled particle impacts in the shock tunnel.[34]

Acoustic disturbances are particularly difficult to remove from ground-test facilities because high levels of acoustic disturbances are radiated from the turbulent boundary layers normally present on the test-section walls.[35] The magnitude of this noise increases with the fourth power of the Mach number, and so the effect is much worse in hypersonic facilities compared with supersonic ones. In the JPL 20-in. tunnel at Mach 4.5, Kendall had fully laminar flow on a flat plate at $Re_T = 3.3 \times 10^6$ when the tunnel sidewall was laminar and the noise level was low. In the same tunnel at the same Mach number, Coles as reported by Reshotko observed transition on a plate at $Re_T \simeq 1 \times 10^6$, under noisy flow, with turbulent sidewall boundary layers.[36]

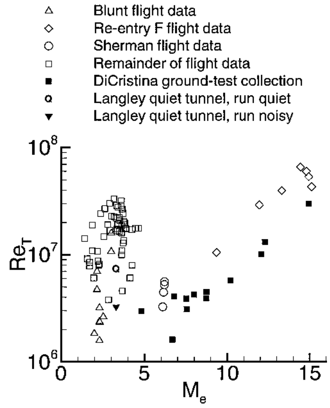

Fig. 1 Transition on sharp cones near zero angle of attack in flight and ground test. (Various sources.)

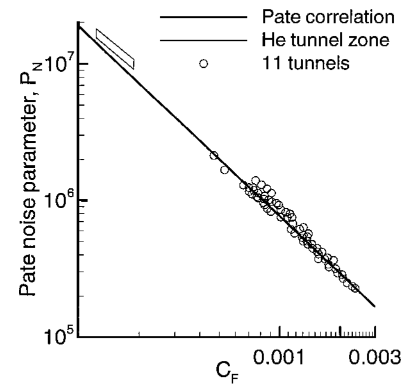

Fig. 2 Correlation of transition on sharp cones using tunnel-noise parameter. Adapted from Ref. 40. Representative data.

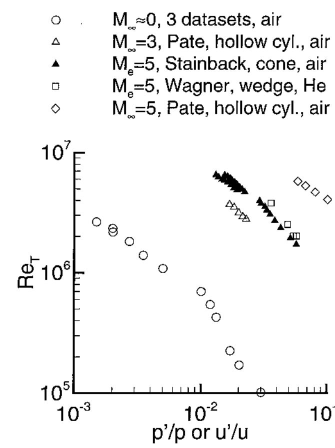

Fig. 3 Transition Reynolds number data plotted as a function of the measured disturbance levels. Adapted from Ref. 41.

As shown in Fig. 2, Pate correlated transition measurements in conventional tunnels using tunnel-wall boundary-layer noise parameters, independent of Mach number.[37–40] Here, the Pate noise parameter $P_N = (Re_t)_\delta \sqrt{(\delta^\star/C)}/[0.8 + 0.2(C_1/C)]$, where $(Re_t)_\delta$ is the Reynolds number for the end of transition based on local conditions, $\delta^\star$ is the turbulent boundary-layer displacement thickness in the tunnel test section, C is the tunnel test-section circumference, and C_1 is the circumference of a 30.5×30.5 cm tunnel (122 cm). The horizontal axis plots C_F, which is the mean turbulent skin friction coefficient in the tunnel test section. The data from the NASA Langley helium tunnel lie within the "He tunnel zone," and the "Pate correlation" is given by $(Re_t)_\delta = \{48.5 C_F^{-1.40} [0.8 + 0.2(C_1/C)]\}/\sqrt{(\delta^\star/C)}$. The high transition Reynolds numbers observed in the Langley helium tunnels are explained by the reduced influence of radiated noise.

Stainback shows sharp-cone transition Reynolds numbers that vary by a factor of 4, at the same edge Mach number and unit Reynolds number, for measurements in six different tunnels at NASA Langley.[41] Figure 3, redrawn from Fig. 3 of Ref. 41, shows the variation of transition Reynolds number with measured tunnel noise for several datasets. Here, Re_T for the Pate data is taken at the end of transition, although for the other datasets, it is for transition onset. For the $M \simeq 0$ data, the horizontal axis is the rms velocity fluctuations divided by the mean, u'/u, and for the rest of the data, the horizontal axis shows the rms pressure fluctuations divided by the mean, p'/p. Transition Reynolds number decreases with increasing noise, in a manner dependent on model geometry and tunnel conditions. Dougherty correlated pressure fluctuations measured on the surface of a cone with transition, and related noise directly to Pate's correlation.[42] Stainback et al. related freestream fluctuation measurements to cone-surface fluctuation measurements.[43,44] Clearly, noise has a dramatic effect on transition.

Mateer and Larson made transition measurements in the Ames 3.5 ft hypersonic tunnel, which was unusual in that cold gas was injected into the nozzle wall boundary layer using a slot placed in the subsonic region of the tunnel.[45] When the gas injected at the wall was changed from air to helium, transition moved downstream. As

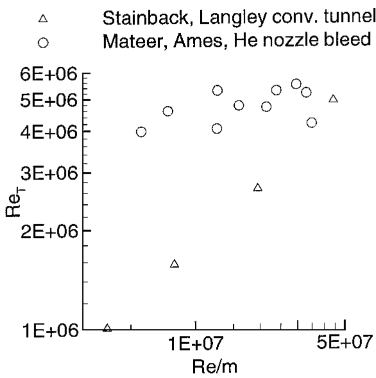

Fig. 4 Effect of unit Reynolds number on transition Reynolds number in two wind tunnels. Adapted from Ref. 45.

Papamoschou has shown for a jet, this is almost certainly due to a reduction in the noise radiated from the wall boundary layer.[46] Figure 4, redrawn from Mateer and Larson's Fig. 3a, shows a more important result. The variation of transition Reynolds number is plotted against unit Reynolds number. Both datasets were obtained on 5-deg half-angle cones, at edge Mach numbers of about 6.8. For the Stainback et al. data, obtained in a conventional tunnel at Langley, transition Reynolds number increased with unit Reynolds number, as is commonly observed. However, the Ames tunnel shows no variation of transition Reynolds number with unit Reynolds number, as has also been observed in the Ludwieg tubes at Göttingen.[47]

Pate[48] reviews the data in detail, arguing that much of the unit Reynolds number effect is due to changes in radiated noise. He concludes that "if a true Mach number effect exists, it is doubtful it [a Mach number effect] can be determined from data obtained in conventional supersonic/hypersonic wind tunnels because of the adverse effect of radiated noise." What are the noise levels in flight and in ground-test facilities?

Measurements of Free-Air Noise in Flight

There are few measurements of noise levels in flight that are carried out at sufficiently high frequencies to be useful for transition studies. The available literature was reviewed by Bushnell,[3] who cites 61 references. The primary source is the work of Fisher and Dougherty,[49,50] who performed measurements with a 5-deg half-angle cone and a pitot tube in wind tunnels and in flight. Microphones were used for static-pressure measurements at the cone surface, and a fast-response pressure transducer was used for the impact-pressure measurements. The bandwidth for both instruments was roughly 0–25 kHz. Both cone and pitot tube were placed ahead of the F-15 aircraft. As might be expected, the rms pressure fluctuations on the pitot tube decrease by a factor of about 5 as the Mach number increases from 0.4 to 1.6. This is presumably caused by the inability of aircraft noise to propagate forward in a supersonic flow. At the highest Mach number flown, Mach 2, Table 1 in Ref. 49 shows that $p'_{t,\mathrm{rms}}/p_{t,\mathrm{mean}}$ was 0.020% for flight 346 and 0.0057% for flight 340. Both flights were at about 12 km altitude.

Fisher (private communication, August 1996) believes that both of these values were obtained with good signal-to-noise ratio and that the difference reflects real variations in small-scale turbulence at altitude. The corresponding values of $p'_{t,\mathrm{rms}}/q$ are 0.058 and 0.016%. Considerable care was required to achieve these measurements; for example, Fisher commented that it was necessary to require that the pilot not use the radio during periods of data acquisition. Static-pressure measurements were obtained on the cone surface for these same two flights; the values for the forward microphone are $p'_{s,\mathrm{rms}}/q = 0.059$ and 0.033%, respectively. Fisher and Dougherty compare these flight results for noise and transition on the cone to values measured in various supersonic wind tunnels around the world. They show that the flight noise level is at least three times smaller than in the best wind tunnels, even at Mach 2.

Haigh et al.[51] performed the only known hypersonic noise measurements. Details are sparse because much of the work is classified. A 10–200 kHz microphone was placed flush with the surface of a cone that reentered the atmosphere at a speed of about 7 km/s. Haigh et al. report that $p'_{s,\mathrm{rms}}/q$ was 0.003–0.0055% at transition onset, and 0.055–0.09% for "fully turbulent" conditions.

Measurements of Freestream Noise in Conventional Wind Tunnels

Donaldson and Coulter[52] report measurements of freestream fluctuations carried out at Mach numbers ranging from 4 to 8 in AEDC tunnels A and B. These measurements are less conservative than Fisher's flight data, for the wind-off noise was directly subtracted from the measurements during flow. A direct subtraction assumes complete correlation of the two random signals; a more conservative approach would be to difference the squares of the signals and take the square root.[53] The mass-flow fluctuations at Mach 8 in tunnel B are about 1–2% of the mean; the total temperature fluctuations are about 0.1% of the mean. The levels measured at Mach 6 are similar. Although the scaling among pitot-pressure fluctuations, mass-flow fluctuations, and static-pressure fluctuations remains unresolved, available data indicate these quantities normalized by their mean values are generally within a factor of 2 (Ref. 54). The high level of mass-flow fluctuations reflects the high levels of acoustic noise radiated from the nozzle walls. These noise levels are 10–100 times larger than those measured in flight by Fisher.

Wagner measured fluctuations in the 1.5-m helium wind tunnel at NASA Langley, using a constant-current hot wire.[55] Mass-flow fluctuations were about 2%, and total temperature fluctuations were about 0.03%. Stainback et al.[56] correlated transition location and noise measurements for sharp cones in several wind tunnels at NASA Langley. They also correlated pressure fluctuations at the cone surface with freestream fluctuations. Figure 6a in Ref. 39 shows that an insufficiently low dewpoint can also affect transition.

Measurements in Quiet Wind Tunnels

Quiet wind tunnels have been constructed to reach much lower levels of freestream noise.[25,57] In these tunnels, the fluctuation levels normalized by their averages are reduced to less than 0.1%. However, the clearest indication of quiet flow is the absence of noise radiated from the turbulent spots that form on the wind-tunnel walls when the flow is not quite quiet.[58,59] Hypersonic tunnels may be considered quiet when such turbulent spots pass by only a small percentage of the time and the rms fluctuation levels are less than 0.1%. Rare intervals of noise contamination will have to be dealt with using filtering or repeated sampling. Although the fluctuation level in these tunnels is an order of magnitude smaller than that measured in conventional tunnels, it still appears to be larger than that measured by Fisher in flight. Figure 5 shows that transition Reynolds numbers measured in the Mach 3.5 quiet tunnel are for the first time in the range of flight data.[60] The "conv. tunnel" and "flight" regions enclose areas in which the conventional-tunnel and flight data fall, as reported in Fig. 4 of Ref. 60. Both the conventional-tunnel and the flight data are for $M_e = 1.4$–4.6. The "quiet tunnel run noisy" data were obtained with flow through the throat-region suction slot turned off. Both the noisy and quiet quiet-tunnel data are for $x_s = 12.7$ cm, where x_s is the streamwise location of the tip of the cone.

References 61 and 62 show that the noise levels in the Mach 3.5 quiet tunnel are in the range of flight noise levels, and that noise incident on the forward portion of the model is the most important for transition. In many of the Langley experiments, the aft end of the model was placed outside the quiet region, and high transition Reynolds numbers nevertheless resulted. The residual effect of the aft-impinging acoustic radiation was rarely determined.

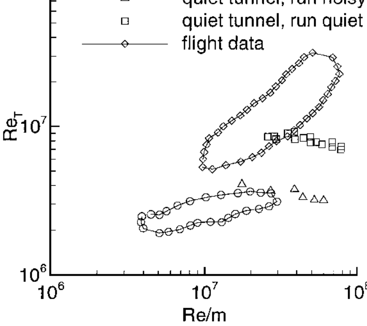

Fig. 5 Comparison of transition-onset Reynolds numbers for sharp cones at zero angle of attack. Adapted from Ref. 60.

Measurements of Freestream Noise in Shock Tunnels

It is sometimes said that the noise levels in shock tunnels may be lower than in conventional hypersonic wind tunnels.[63] This argument is usually made by comparing transition measurements carried out in conventional wind tunnels and in flight, and by assuming a particular scaling. Unfortunately, direct measurements of the fluctuation levels in shock tunnels are sparse. Ross et al.[64] measured heat-transfer fluctuations to a thin-film gauge positioned on a small wedge, at Mach 14, with a bandwidth reported at 800 Hz–1 MHz. The rms heat transfer divided by the mean ranged from 1.5 to 3%. These measurements are in the same range as those in conventional wind tunnels. These high noise levels might be expected, because turbulent boundary layers should still form on the nozzle walls, and, in addition, there are likely to be substantial fluctuations in entropy or stagnation temperature. Holden et al. have argued that the Pate correlation does not apply to shock tunnels, because transition occurs at Reynolds numbers well below those computed from the correlation.[63,65] However, it could also be argued that transition occurs earlier because large fluctuations in stagnation conditions increase fluctuation levels well above those due to tunnel-wall acoustic radiation alone.

Bergstrom[66] reports hot-wire measurements in a Mach 7 gun tunnel, with static-pressure fluctuations of 1–3%. Bergstrom's hot-wire mode diagrams indicated that the primary source of disturbances was the turbulent boundary layer on the nozzle walls.[67]

He and Morgan measured pitot-pressure fluctuations in the T4 free-piston shock tunnel using fast pressure transducers, with response frequencies up to about 250 kHz.[68] The rms pitot pressure was roughly 3% of the mean, although varying over a large range between roughly 2 and 8%. The power spectrum showed a large, dominant peak of unknown origin, whose frequency varied with the square root of the temperature.

Comment on the Ballistic-Range Data

A number of measurements have also been made of transition in free flight in ballistics ranges.[69–72] The range complements the wind tunnel because it uses independent techniques and can be made free of radiated noise.[69] However, the range suffers from its own set of difficulties. The unit Reynolds number effect observed there remains to be conclusively explained, and the transition Reynolds numbers that are observed do not seem to be characteristic of flight values.[9] Because the model is usually destroyed at the end of the shot, measurements are generally limited to transition location and angle of attack is difficult to control. Roughness, tip radius, angle of attack, vibration, and suspended dust are possible causes of early transition.[69] Figure 12 in Ref. 73 shows transition Reynolds number varying by a factor of 3 during a single shot, due apparently to angle-of-attack oscillations with a 1-deg amplitude, for a 5-deg half-angle cone. Pate[74] speculates that Potter's relatively large nose radius may have caused early transition, although the smaller radius used by Sheetz[75] may explain the higher transition Reynolds numbers that he has sometimes observed. It must be remembered that nearly all of these data were obtained when our understanding of transition was much less sophisticated. In particular, nonlocal effects due to convection from upstream were often neglected. For example, Potter's examination of roughness effects mostly neglects roughness present near the tip,[69] although some discussion is given on pages 106–112 of Ref. 76. Although range data can be acquired free of radiated noise, they suffer from another set of experimental difficulties, which remain to be resolved.

Summary of Noise-Level Data

For predictions of hypersonic boundary-layer transition in flight, one would like to have measurements of the noise levels at appropriate altitudes. Detailed measurements of the three-dimensional spectra of the entropy, vorticity, and acoustic fluctuations are desirable, along with measurements of the particle content and so on. A consistent set of measurements is also needed for ground-test facilities, using a consistent method for determining instrument bandwidth and signal-to-noise ratio. The difficulty of making low-noise, high-frequency measurements is second only to the difficulty of fabricating a quiet tunnel.

Some preliminary conclusions can nevertheless be drawn. Although there are few flight data above Mach 2, the fluctuation level in the atmosphere should not be affected by the (supersonic) speed of the aircraft carrying the measurement device. The key limitation to the Fisher data is rather the limited and poorly understood bandwidth and the relatively low altitudes. The Fisher data do clearly show that the fluctuation levels present in the atmosphere at low spatial frequencies are very small, and that they correspond to pressure-fluctuation levels that are 10–100 times smaller than in conventional wind tunnels. Even the "quiet" wind tunnels appear to suffer from fluctuation levels that are larger than in flight.

It is often argued (e.g., Stetson et al.[77,78]) that most of the freestream noise is not important for instability and transition studies; the only part of significance is said to be that which is at frequencies similar to those of the dominant instabilities on the model. A conventional hypersonic wind tunnel is thus sometimes said to be effectively quiet because the noise at the high frequencies of the second-mode instability may be small. However, there is at present no direct evidence to substantiate this interesting conjecture. True, the noise measurements in tunnel B reach a signal-to-noise ratio of 1 at about 70–200 kHz (Ref. 52); however, AEDC has not yet reported a quantitative upper bound for noise in the high-frequency second-mode band, nor has it been shown that this noise does not affect the measurements. In fact, second-mode transition in tunnel B occurs at a computational N-factor of about 5 (Ref. 79). Because second-mode transition occured at a computational N-factor of about 8 in the NASA Langley Mach 6 quiet tunnel (cf. Figs. 2 and 9 in Ref. 80), it seems probable that the low N-factor is caused by the large noise level.

The work of Stetson et al. and also of Kosinov et al. (e.g., Ref. 81) is based on the idea that the local growth of the instability waves is unaffected by the broadband noise environment in which the waves grow. The problem is clearly very difficult, and measurements based on this assumption clearly are more useful than measurements of transition onset only. However, the assumption neglects all nonlinear and three-dimensional interactions, whereas three-dimensional interactions in particular have been shown to be significant in low-speed flow (e.g., Ref. 82). It would be interesting to see the assumption tested by repeating some of the existing measurements in a different facility with a substantially different noise level, although in this case it might be difficult to rule out the effects of unit Reynolds number.

As Stetson et al. have noted, it is possible that some unit Reynolds number effects may be traced to changes in the local boundary-layer profiles. Such changes might make the integrated growth of the instability waves scale with something other than the overall length Reynolds number of the model (e.g., temperature effects or transverse curvature). For example, Mack shows that changes in stagnation temperature can affect stability and transition on sharp adiabatic cones at identical Mach and Reynolds numbers, through temperature-dependent changes in thermal conductivity and viscosity.[83,84] Experimental verification has been provided by Kimmel and Poggie.[85] It would seem that quiet-tunnel measurements will be needed to determine the effect of tunnel noise on instability measurements in conventional tunnels.

Measurements of Transition at High Reynolds Numbers

The flight data were recently reviewed in Ref. 9. The following selects some cases where transition occurred at very high Reynolds number, as sometimes can occur under quiet conditions when roughness, three-dimensionality, and other early transition factors are controlled.

Reentry F

This flight test was of a 4-m-long beryllium cone with a half angle of 5 deg and an initial nose radius of 0.25 cm (Ref. 28). Surface heat-transfer and pressure data were obtained during reentry at altitudes ranging from 30.5 to 18.3 km. The freestream Mach number was about 20, and the ratio of wall temperature to total temperature was about 0.1. The angle of attack was controlled within a degree of zero, and the total enthalpy ranged from 18.3 to 16.9 MJ/kg.

Length Reynolds numbers at transition, based on edge conditions, ranged from 40×10^6 to 60×10^6 at the higher altitudes. Limited e^N computations indicate the transition was dominated by second-mode disturbances.[86]

Sternberg V-2 Flight Test

Sternberg[87] conducted a cold-wall flight test of a 10-deg half-angle cone at nearly zero angle of attack and about Mach 3. The nose radius was small, and the cone was more than 2.4 m long. The length Reynolds number at transition was more than 40×10^6.

Flight Tests of Reentry Vehicles

Although much of this literature is classified, some of it is openly available. Haigh et al.[51] summarize various flight tests, most of which are classified. Their Fig. 48 shows that some zero angle-of-attack flight data exhibited local length Reynolds numbers at transition that were about 20×10^6 or more. Figure 27a in Ref. 88 shows that local arc-length transition Reynolds numbers above 10×10^6 have been measured numerous times.

Rumsey and Lee Flight Test

This cold-wall 1956 flight test of a 7.5-deg half-angle cone with a 0.25-mm nose diameter exhibited local length transition Reynolds numbers as high as 30×10^6, at a local Mach number of 2.91 (Ref. 89). The ratio of wall to static temperature was 1.2.

Ground Tests in NASA Langley Mach 3.5 Quiet Tunnel

The only comparable high-speed ground-test results were obtained in the Mach 3.5 quiet tunnel at NASA Langley.[60] These results were obtained under conditions that were only partially quiet: only the forward portion of the model was in the quiet-flow region. Transition onset on a flat plate with a nose bluntness of 0.02 mm occurred at length Reynolds numbers as high as 17.9×10^6 (see Ref. 31). Transition onset for a 5-deg half-angle cone occurred as late as 9.1×10^6; this cone was at zero angle of attack and had a sharp tip. Both these models were studied at adiabatic wall temperatures. A small bluntness would be expected to delay transition further; however, quiet flow can only be maintained in this facility to a length Reynolds number of about 10×10^6. Thus, the Rumsey and Lee flight test still cannot be duplicated in this facility.

Other Hypersonic Ground Tests

Softley et al. report transition at length Reynolds numbers of 10×10^6 or more, on sharp cones at an edge Mach number of 12 in a shock tunnel, in a residual favorable pressure gradient.[90] Maddalon and Henderson reported values as high as about 50×10^6 for measurements at an edge Mach number of 16 in the NASA Langley helium tunnel.[91] These length Reynolds numbers are again based on edge conditions. Although it is sometimes argued that these (and other) high transition Reynolds numbers are an indication of relatively low levels of facility noise, the shock-tunnel noise measurements discussed here suggest that a receptivity or instability mechanism must instead be sought. This is directly shown by Fischer and Wagner,[92] who measured high freestream noise levels above a sharp cone in the helium tunnel at NASA Langley, while at the same time measuring transition-length Reynolds numbers in the range of 30×10^6–40×10^6. Recall that Pate's correlation included these helium-tunnel data (Fig. 2).

The use of local-length Reynolds number as an appropriate scaling for hypersonic transition is, of course, open to question. For example, Adam and Hornung report measurements at high-enthalpy conditions and Mach 5, on a sharp cone, in the free-piston shock tunnel T5.[93] Their transition-length Reynolds numbers, scaled on edge conditions, were 2×10^6–4×10^6, an order of magnitude below the flight data; but both the T5 measurements and the reentry F measurements show transition at length Reynolds numbers of about 1×10^6 when based on the reference temperature. In view of the differing Mach number, the difference in wall temperature, and the high noise levels in T5, this agreement is probably fortuitous. The comparison points out clearly how flight and ground tests carried out under different conditions can appear to agree or disagree depending on the scaling chosen.

Summary of High Reynolds Number Transition Data

It is clearly possible to delay transition in flight to very high Reynolds numbers, when roughness, crossflow, ablation, and other effects are controlled appropriately. It is also possible to achieve high transition Reynolds numbers in ground tests (especially in the helium tunnel). Noise levels are clearly a significant factor in ground tests, so transition Reynolds numbers reported without measurements of freestream noise levels are of limited value. The 50-year search for a single correlating parameter that will reconcile the various flight and ground-test measurements appears futile, in view of the many factors influencing transition.

Effect of Tunnel Noise on Parametric Trends

Ratio of Transition Reynolds Numbers for Cones and Flat Plates

Although an e^N estimation suggests that transition should occur on flat plates at length Reynolds numbers higher than those on round cones, measurements in conventional wind tunnels consistently showed the opposite behavior.[60] Figure 6, redrawn from Ref. 60, shows a region containing Pate's collection of conventional-tunnel data, along with measurements in the Langley quiet tunnel (at $x_s = 12.7$ cm). When the bleed valve is closed, the Langley tunnel becomes noisy, yet even in this case, the results are substantially different from those measured in conventional tunnels. Figure 7, again redrawn from Ref. 60, shows the comparison to e^N theory, along with a few correlations to conventional-tunnel measurements. The conventional-tunnel, flat-plate transition Reynolds numbers increase with tunnel size and show the usual unit Reynolds number dependence. The AEDC data are for the end of transition, and the rest of the data are for transition onset. The quiet-tunnel measurements are independent of unit Reynolds number and in general agreement with an $N = 10$ prediction. These quiet-tunnel results did much to validate the e^N approach.

Note that a designer relying on the conventional-tunnel data in Fig. 6 would seek to create a conical forebody to delay transition; only with quiet-tunnel data does the designer now see that the flat-plate geometry is to be preferred. The cause of the anomaly in the conventional-tunnel data is addressed by Stetson and Kimmel in their Sec. 5.4 (Ref. 10). Low-frequency disturbances below the first-mode band are amplified on the flat plate in the Mach 8 conventional tunnel, although they are not amplified in the cone case.

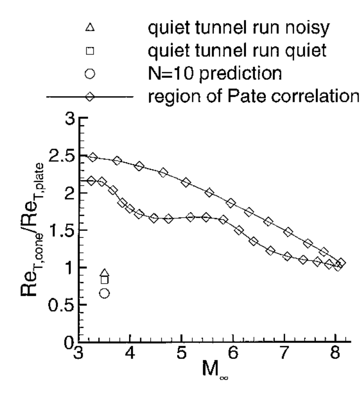

Fig. 6 Ratios of cone-to-flat-plate transition Reynolds numbers from Pate, compared with quiet-tunnel data and linear stability theory predictions. Adapted from Ref. 60.

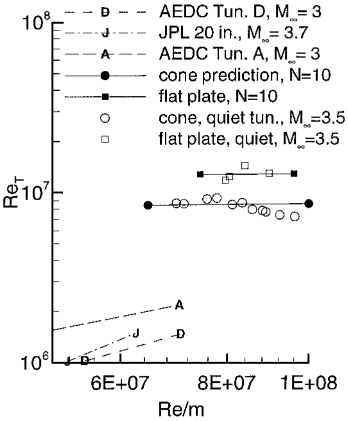

Fig. 7 Comparison of transition-onset Reynolds numbers on cone and flat plate. Adapted from Ref. 60.

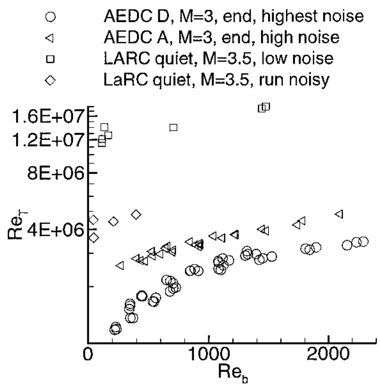

Fig. 8 Effects of noise levels and leading-edge bluntness on flat-plate transition.

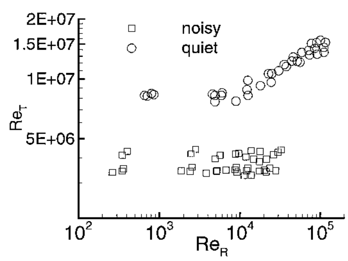

Fig. 9 Nose bluntness effects on cone boundary-layer transition, under quiet and noisy conditions, zero angle of attack. Adapted from Ref. 95. Representative data.

These low-frequency disturbances are presumably the result of the high level of tunnel noise. This example emphasizes the main point: although stability measurements in conventional facilities can be applicable to flight, transition measurements in such facilities suffer from uncertain and uncontrollable ambiguities.

Bluntness Effects Are Influenced by Noise Levels

Figure 8 shows transition Reynolds numbers measured on flat plates in various wind tunnels with various bluntnesses. The figure was inspired by Fig. 4 in Ref. 94 but was redrawn from tabulated data in Refs. 31 and 48. The data for the AEDC 16 × 16 ft tunnel were omitted, because the data in Ref. 48 indicate they are contaminated by high dewpoint problems. For the Langley data from Ref. 31, only the data for $x_s = 12.7$ cm are used, because this is the lowest noise position for the model. The Langley data are for transition onset, and the AEDC data are for the end of transition. The highest-noise data, from the conventional tunnel D at AEDC, show not only the lowest transition Reynolds number, but also the strongest effect of bluntness. Noise affects not only the levels of the curves but the slope. The quantitative significance of the effect remains unclear, as Stetson has commented, because the slopes are fairly similar except for the tunnel D data. See Pate[48] for a detailed discussion of the AEDC data. The Langley data were obtained with four bluntnesses (all less than 0.025 mm), and so the accuracy with which these bluntnesses could be controlled appears uncertain.

Figure 9, digitized from Fig. 13 of Ref. 95, shows measurements of transition onset made using a thin-skin model with thermocouples on a round cone at zero angle of attack. The data taken with the bleed valve open were obtained under quiet conditions, and the data obtained with the bleed valve closed were under noisy conditions. The Reynolds number based on nose radius is plotted on the horizontal axis. Under noisy conditions, small increases in nose bluntness do not delay transition, but under quiet conditions, with sufficient nose bluntness, they do. Figure 7 in Ref. 60 also shows a larger effect of bluntness for the low-noise flow. These trends appear to be opposite from those suggested by the flat-plate data shown in Fig. 8. However, this may be because Fig. 8 compares high and very high noise levels, and Fig. 9 compares low and moderately high noise levels. The geometry also differs. Further investigation is clearly needed.

Angle of Attack and Crossflow Effects Are Influenced by Noise Levels

Transition on a sharp, slender cone is known to be strongly sensitive to angle of attack. Transition measurements can be made on a smooth, sharp cone at zero angle of attack and then compared to measurements at angle of attack. Figure 10 shows data from

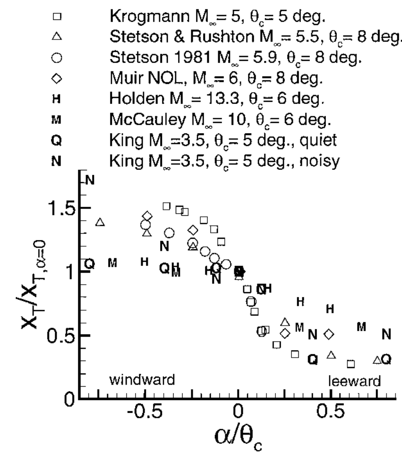

Fig. 10 Comparison of the movement of transition on a sharp cone at angle of attack.

seven sources, in different facilities, with different cone angles and freestream Mach numbers. The transition onset locations in the first six datasets were taken from heat-transfer measurements. The data labeled “McCauley” were digitized from Fig. 6 in Ref. 96, and they were apparently taken at $Re_\infty = 5.2 \times 10^6$/m. The “Holden” data were digitized from Fig. 12 in Ref. 65 and were taken at $Re_\infty = 10 \times 10^6$/m. The “Muir” data are from Fig. 18 in Ref. 97; the leeward data are at $Re_\infty \simeq 9.2 \times 10^6$/m, and the windward, at $Re_\infty \simeq 3.2 \times 10^7$/m. The two different unit Reynolds numbers were used to keep transition on the instrumented section of the model; it is not certain if no other datasets also varied unit Reynolds number. Such a variation does, of course, also vary tunnel noise at the same time. The data labeled “Stetson 1981” were digitized from Fig. 3 of Ref. 98 and were taken at $Re_\infty = 3.2 \times 10^7$/m. The data labeled “Stetson and Rushton” were digitized from Fig. 7 of Ref. 99 and were taken at $Re_\infty = 4.9$–15.4×10^6/m. There is a 4% error in the anchor point at zero angle of attack, which is due to the limited accuracy of the original figure. The data labeled “Krogmann” were digitized from Fig. 7 of Ref. 47 and were taken at $Re_\infty = 1.78 \times 10^7$/m.

In all cases, transition moves aft on the windward ray and forward on the leeward ray. However, the large scatter in the figure shows the limitations of a simple nondimensionalization, which cannot account for all of the variables that differ between the datasets. To the author’s knowledge, these data have never been explained, although various attempts have been made to correlate subsets of the data. Unit Reynolds number, noise, freestream Mach number, and cone angle are all parameters for this complex problem; bluntness can also be added. Although the crossflow instability is obviously critical away from the windward and leeward rays, flow on those rays is affected only indirectly by crossflow, through changes in the boundary layer profiles. One would thus expect that transition on the windward and leeward rays is caused by the first-mode or second-mode instabilities, at least for small bluntness.

Available quiet-tunnel data show that noise has a major impact on this flow, as is also shown in Fig. 10, using King’s data from the Langley Mach 3.5 quiet tunnel.[100] Tabulated data were supplied by King (private communication, Dec. 1996); the onset of transition was obtained from Preston tube measurements. On the leeward ray, transition moves forward for both quiet and noisy flow; it moves forward by a greater amount in the quiet flow. However, on the windward ray, transition moves aft only in the noisy flow; for the quiet flow, the windward ray transition location stays approximately at the zero angle-of-attack value. Quiet-tunnel measurements of transition location for various unit Reynolds numbers and nose bluntnesses are not yet available in the open literature.

King’s data are sometimes used to argue that the crossflow instability is not affected by tunnel noise. However, Fig. 9 of Ref. 100 shows only that the delay in transition due to quiet flow decreases from about 100% at low angle of attack to about 25% at higher angle of attack. The results are not conclusive because the aft end of the cone was outside the quiet region in a way that varied with angle of

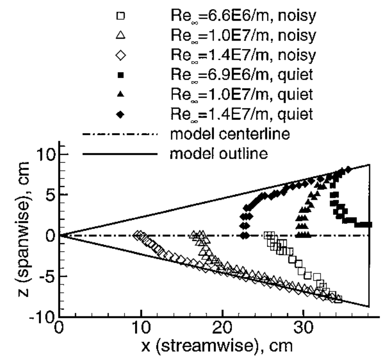

Fig. 11 Effects of noise level on transition on a swept-wing, leading-edge model for various free-stream unit Reynolds numbers.

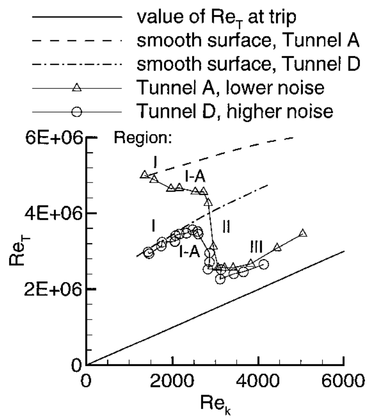

Fig. 12 Variation of transition Reynolds number with trip Reynolds number for M_e = 2.89, tunnels D and A. Adapted from Ref. 105.

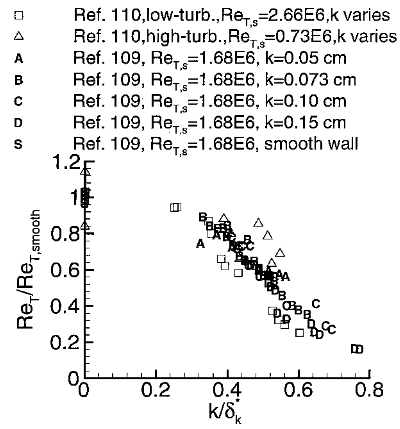

Fig. 13 Ratio of transition Reynolds numbers for a plate with single roughness element to that for a smooth plate in airstreams of different turbulence.

attack (Figs. 1–3 of Ref. 100), and King measured only transition location.

Cattafesta et al. made measurements on a swept-wing model at Mach 3.5, under both quiet and noisy conditions.[101,102] Although the comparison of the effect of noise has not previously been published, it is presented here as Fig. 11, drawn from data provided by Cattafesta (private communication, Oct. 2000). For these measurements, $\alpha = -2$ deg and $M_\infty = 3.5$. The noisy-tunnel transition-onset location is drawn below the model centerline only to clarify the figure; transition onset was determined from thermocouple measurements that were all obtained on the same side of the model centerline. Transition moves forward in the noisy flow, compared with the quiet flow at the same unit Reynolds number. It also appears that the pattern of transition onset may vary between quiet and noisy flow for $Re_\infty \simeq 6.6 \times 10^6$/m. Cattafesta et al. present evidence to suggest that transition on their smooth-wall model was caused by the traveling form of the crossflow instability, which is likely to be strongly affected by tunnel noise. With the addition of small roughness dots, stationary crossflow vortices seemed to appear; however, the effect of tunnel noise on these vortices remains to be studied. Reference 102 also correlated transition onset with an envelope-method e^N computation with $N = 13$. At low speeds, the evidence indicates that freestream turbulence is critical to traveling crossflow instabilities but not to stationary ones.[103] However, the effect of tunnel noise on the high-speed crossflow instability remains to be established clearly.

Effect of Noise on Roughness and Waviness Effects

The limited data of Morrisette et al.[104] show little effect of surface waviness for transition on round cones at zero angle of attack. In measurements in the Mach 3.5 quiet tunnel at Langley, transition moved forward about 20% when a surface waviness with height-to-length ratio of 0.01 was introduced. Morrisette et al. also show that the trends appeared similar with high and low levels of tunnel noise.

Figure 10a from Pate,[105] redrawn here as Fig. 12, shows the effect of tunnel noise on roughness effects for a round cone at zero angle of attack with hemispherical roughness elements. Although tunnels A and D are both conventional, the noise level in tunnel A is smaller because it is larger.[48] For both sets of data, the roughness height is 0.25 mm and the roughness was located approximately 125 mm downstream of the nosetip. The figure shows that for smooth models, transition indeed occurs later in tunnel A. However, it also shows that the parametric effect of roughness also enters differently. As the roughness Reynolds number Re_k is increased, there is a gradual forward movement of transition in tunnel A, but transition moves aft in tunnel D (region I-A). At an Re_k of about 2500, transition moves suddenly forward in tunnel D, with the same sudden movement occuring in tunnel A at a slightly higher Reynolds number. The 10% forward movement of transition that occurs in tunnel A for Re_k between 1000 and 2500 is opposite to the 20% rearward movement that occurs in tunnel D. What variation would occur under the much lower noise conditions expected in flight?

For roughnesses large enough to cause transition directly at the roughness element ("effective" roughnesses), it appears that noise has little effect. However, in the case of design for flight, what is needed is a specification of the roughness that will not move transition forward too far; this roughness will not be an "effective" roughness, and the level of noise in a ground-test facility apparently has a major influence on the determination of these kinds of roughness effects. It is the ambient noise levels interacting with the roughness that cause transition; both are important. This has been clearly shown in various low-speed receptivity experiments (e.g., Wlezian[106]).

Dryden collected low-speed data to show the combined effect of freestream noise and roughness effects.[107,108] Figure A,5h of Ref. 107 was used as a model to generate Fig. 13. The vertical axis is the transition Reynolds number normalized by the average smooth-wall value, $Re_{T,s}$, and the horizontal axis is the roughness height normalized by the displacement thickness at the roughness element. The data were regenerated from the tables of Ref. 109 and Ref. 110; see also Ref. 111. These data are for low-speed, flat-plate transition, tripped by a single two-dimensional cylindrical-wire roughness element. Data where transition occurs less than 100 element heights downstream of the roughness element location are not shown. Where $Re_{T,s}$ is larger, the turbulence level is expected to be larger, although it was apparently not measured directly. Dryden's Fig. A,5h contains additional data that are not shown here, along with curve fits apparently developed from the semiempirical theory of Ref. 108. Dryden's figure also presents turbulence levels for the Tani source data; these were apparently obtained by using a monotonic correlation between tunnel turbulence and smooth-wall transition Reynolds number.[108,111] Dryden states that these data "disprove" his earlier and often-quoted hypothesis that transition Reynolds number depends on roughness Reynolds number alone.[112] Although there is considerable scatter, Fig. 13 shows that transition depends on both freestream turbulence and on the roughness height. Again, this is expected from receptivity research (e.g., Ref. 113). Further research is still needed.

Additional measurements were recently obtained by Ito et al. on a blunt axisymmetric shape with two compression corners.[114] Measurements were obtained under quiet and noisy conditions, both with and without roughness. The measurements were carried out with arrays of hot-film sensors at the wall; the time traces were used to compute intermittency. Noise affected the onset of transition on

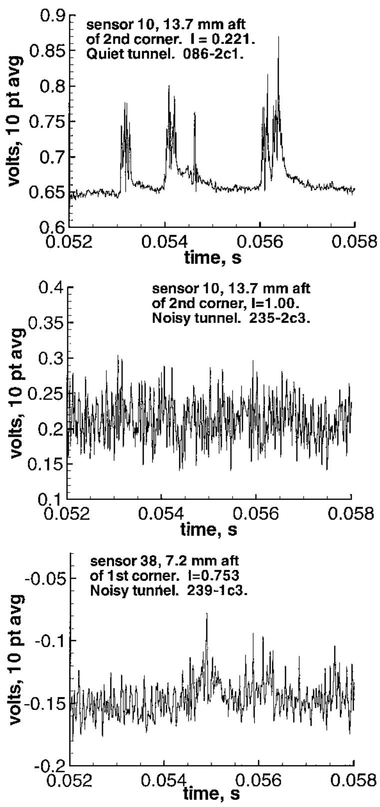

Fig. 14 Hot-film time-traces for smooth scramjet inlet model under quiet and noisy conditions. From Ito et al.[114]

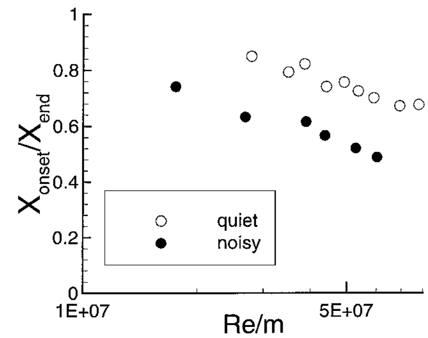

Fig. 15 Ratios of transition locations measured on a sharp cone. From Chen.[95]

both the smooth and the rough models. The transition process was dramatically affected, as shown in Fig. 14. Here, time traces are shown from the hot films. Under quiet conditions, turbulent spots develop as shown in the upper frame, where the intermittency is 22%. Under noisy conditions, the flow is fully turbulent at the same mean-flow condition, as shown in the middle frame. Even at lower Reynolds numbers, as shown in the lower frame, turbulent spots do not appear in the noisy-freestream flow; the hot-film fluctuation level just increases smoothly. In the general case, both noise and roughness must be taken into account, and noise can change the whole transition process.

Preliminary measurements of roughness effects were also carried out by Morissette and Creel in the NASA Langley Mach 3.5 quiet tunnel. These transition measurements showed that noise changed the critical roughness height of hemispherical elements by only 10%, so the sparse available data are not clearcut.[115]

Effect of Noise on Attachment-Line Transition

Creel et al. measured transition on swept cylinders at Mach 3.5 in the Langley quiet tunnel.[116] When no end plates or trips were present, transition appeared to be independent of tunnel noise. Creel et al. speculate that this noise independence is a result of the spanwise-independent attachment-line boundary layer containing no streamwise variation at which receptivity to acoustic disturbances can occur. However, this was also the case for the limited measurements taken off the attachment line, perhaps due to a dominance by the stationary crossflow instability. When trips were used, the forward movement of transition was larger in noisy flow compared with quiet flow, so both roughness and tunnel noise were then important.

Measurements of the transition mechanisms on the attachment line of a swept cylinder were obtained by Coleman et al. under low-noise conditions at Mach 1.6.[117,118] Unfortunately, the wall-temperature distributions differed between the noisy and quiet-flow cases. Figures 4-25 and 4-27 of Ref. 118 show that the instability-wave spectra on the attachment line differ significantly between quiet and noisy flow, although Fig. 4-28 therein shows no net effect on transition onset. Coleman et al. believe that the effect of the increased freestream disturbance level was counteracted by the effect of a lower wall temperature. In addition, they found that tunnel noise has a substantial effect on roughness-induced transition.

It thus appears that tunnel noise has an important effect on roughness-induced transition on attachment lines. This is again in the case where transition occurs downstream of the roughness elements. For a nominally smooth-wall model, the effect of tunnel noise is not yet established clearly.

Effect of Noise on Free Shear-Layer Transition

King et al. measured transition in the free shear layer above a cavity at Mach 3.5 under both quiet and noisy conditions.[119] They found that tunnel noise had a minimal effect on transition and speculated that this was due to disturbances fed back upstream within the cavity, from the oscillations impinging on the downstream lip. Because their cavity was nominally two-dimensional, end effects are also a concern.

Effect of Noise on Transition Extent

A last example, taken from Chen,[95] shows the effect of noise on measurements of transition extent (the length of the region of intermittent flow between the onset of transition and its end). Figure 15 shows measurements for a round cone at zero angle of attack; the aft end of the cone is outside the quiet-flow region. At the highest unit Reynolds number, at noisy conditions with the bleed valve closed, transition onset occurs at about 50% of the distance to the end of transition. This ratio of onset to end is typical of measurements in conventional wind tunnels.[120] Under low noise conditions, with the bleed valve open, at the same unit Reynolds number, transition onset occurs at about 70% of the distance to the end of transition. Both datasets show a unit Reynolds number dependence, the cause of which is unknown. This trend, of transition onset occuring closer to the point where transition is complete as noise decreases, appears to agree with the flight measurements of Fisher and Dougherty,[50] who show transition onset occurring at about 85% of the length to transition end.

Conclusions

Laminar-turbulent transition is affected by a substantial number of parameters, including freestream fluctuations (or noise). In high-speed, conventional-tunnel experiments, high levels of acoustic noise radiate from the turbulent boundary layer on the nozzle wall. Any variation in tunnel conditions varies both the model flowfield and the impinging noise field. Even for a fixed noise field, variations in the model flowfield may change the spectral sensitivity of the instabilities, thus yielding a tunnel-dependent transition process. Conventional-tunnel measurements of transition location thus suffer from the contaminating ambiguity of unknown variable noise. This ambiguity is seldom resolvable, and it can contaminate parametric trends as well as absolute magnitudes.

Shock tunnels have high levels of fluctuations in the stagnation conditions, in addition to the nozzle-wall acoustic radiation. Thus, transition measurements in shock tunnels also suffer from the noise contamination. Even in a quiet wind tunnel, with noise levels comparable to flight, the trends in transition location must depend on both the model characteristics and on any variations in residual tunnel fluctuations. However, low-speed transition measurements have been generally useful, even when incompletely controlled, as long as the tunnel fluctuations are representative of flight.

Although the ballistic range can be made free of radiated noise, ballistic-range experiments have not been able to reproduce flight measurements of transition, probably because of other difficulties. Reliable estimates of transition must be based on detailed instability experiments that yield an understanding of the mechanisms underlying transition. As Stetson pointed out some two decades ago, detailed study of the instability mechanisms is at least as important as development of quiet tunnels.

Even in conventional tunnels, however, transition measurements do have residual value. In all cases current known, the conventional-tunnel transition location represents a worst-case Reynolds number, as tunnel noise appears able only to move transition forward. Furthermore, when transition occurs at or very near a roughness element, so that it is completely roughness dominated, transition appears to be insensitive to tunnel noise. A similar dominance of nonfreestream disturbances was apparently observed in the cavity-flow transition case and in one example of the attachment-line case.

When transition is caused by stationary streamwise vortices, such as Görtler or stationary crossflow vortices, it may be that receptivity to surface roughness dominates any effects of freestream noise. However, this conjecture remains to be supported by detailed measurements at high speed, and, in any case, can only be true when roughness-derived mechanisms are so strong that they completely dominate mechanisms dependent on freestream noise. When transition involves general geometrical changes, angle of attack, bluntness, smaller roughness or waviness, an attachment line, or other factors, any other use of conventional-tunnel transition measurements is suspect and may be misleading.

However, most quiet-tunnel experiments to date have only measured transition locations. Just as in the low-speed case, detailed measurements of the mechanisms of transition under controlled conditions are needed. Linear amplification may be independent of tunnel noise, as has been assumed in several conventional-tunnel studies. Quiet facilities are needed to determine the noise dependence of conventional-tunnel measurements; they will probably be required for receptivity and nonlinear amplification work, at least for traveling instabilities. Reliable transition prediction methods will have to be developed and validated based on detailed measurements of the actual transition mechanisms. Such mechanism-based methods will need to incorporate knowledge of the freestream disturbance environment. Predictions of transition in flight will thus require estimates of the disturbance environment.

Acknowledgments

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