RECENT DEVELOPMENTS IN BOUNDARY LAYER TRANSITION RESERACH

Eli Reshotko*
Case Western Reserve University
Cleveland, Ohio, U.S.A.

Abstract

The NASA Transition Study Group was founded in late 1970 to develop and implement a program that would do something constructive toward resolving the many observed anomalies in boundary layer transition data and that might provide some basis for future estimation of transition Reynolds numbers. The group formulated specific experimental programs emphasizing careful and redundant measurements, documentation of the disturbance environment and eliminating, wherever possible, facility induced transition. It recommended continued study of stability characteristics as well as theoretical studies of the coupling of various types of disturbances to boundary layers. This paper describes the nature of the program and some of the results obtained to date.

I. Introduction

Reasonably dependable transition information is required in the design of high speed aerodynamic configurations because of the following factors that are affected by transition:

1. Aerodynamic heating and its influence on thermal protection systems
2. Observables
3. Vehicle dynamics

The interest in transition as affecting thermal protection system design for entry vehicles and shuttle systems is obvious. But even for hypersonic cruise vehicle designs with length Reynolds numbers as large as 200×10^6 and predominantly turbulent flow the designer will ask: What parts of the configuration are laminar? Can we count on them being so? What is an effective means of boundary layer tripping so that such configurations can be tested at smaller length Reynolds numbers[2,3]?

Regarding observables; simply put, a vehicle can be observed not only by "seeing" its physical size but also by its flow field and particularly its wake. Transition is known to affect wake behavior.

Vehicle stability, both static and dynamic, is affected when transition is observed on the back of an entry cone, particularly at angle of attack. The effect is greatly enhanced when ablation is involved[4,5].

It was early hypothesized by Reynolds that transition is a consequence of the instability of the laminar boundary layer. This hypothesis was further developed by Rayleigh and to this day remains most highly regarded by workers in the field. It has certainly stimulated much theoretical and experimental work in boundary layer stability. The excellent agreement between the boundary layer stability experiments of Schubauer and Skramstad[6], Liepmann[7], Laufer and Vrebalovich[8], and Kendall[9], with appropriate theories has provided a basis for proceeding in developing the consequences of the Reynolds-Rayleigh hypothesis.

Nevertheless, over the years transition data have been accumulated and correlated by traditional aerodynamic testing procedures and quite independently of stability considerations. These tests have yielded lots of information on the effects of Mach number, surface temperature level, the mysterious "unit Reynolds number", surface roughness, bluntness, pressure gradient, suction and blowing, angle of attack, sweep, etc. These efforts, however, have yielded neither a transition theory nor any even moderately reliable means of predicting transition Reynolds numbers.

Attention has again focused in recent years[10,11,12,13] on the importance in the transition process of the response of the boundary layer to the available disturbance environment. For example, it has been shown that the transition behavior in supersonic wind tunnels above Mach number 2.5 can be clearly ascribed to the noise radiated from the turbulent boundary layers on tunnel walls[14,15,16,17]. In the JPL 20" Supersonic Tunnel at $M = 4.5$, Kendall observed no transition on a flat plate of length Reynolds number 3.3×10^6 when the tunnel wall boundary layer was laminar ("quiet" operation) whereas in the same tunnel at the same Mach number but with turbulent side wall boundary layers, Coles[18] observed transition on a plate at Reynolds numbers of the order of 1×10^6. Thus the vast body of transition data obtained in supersonic wind tunnels is suspect.

Nor are ballistic ranges free of difficulties. In a series of experiments in an enclosed range where the model precedes any disturbances resulting from sabot impact, Potter[19] nevertheless obtains a variation of transition Reynolds number with unit Reynolds number that has yet to be explained.

*Associate Fellow, AIAA; Professor of Engineering and Head, Department of Fluid, Thermal and Aerospace Sciences; Chairman, NASA Transition Study Group

These severe effects of facility on transition Reynolds number are some of the salient difficulties among the many catalogued and discussed in Morkovin's[13] comprehensive report. It is clear that the interpretation and utilization of wind tunnel and ballistic range transition data will require resolution of these various difficulties.

II. Formation of NASA Transition Study Group

The state of the transition problem was taken up by the NASA Research and Technology Advisory Subcommittee for Fluid Dynamics about five years ago*. Recognizing the importance of the problem, the Subcommittee endorsed the need for doing something constructive toward resolving the observed anomalies and providing some basis for future estimation of transition Reynolds numbers.

Following a recommendation by the Subcommittee, NASA Headquarters requested the participation of a number of recognized transition investigators encompassing the directly interested federal agencies and government laboratories plus one member from the Subcommittee**.

The NASA Transition Study Group had its first meeting in November 1970. After presentations by the membership and subsequent deliberation, the Group arrived at the position that there was clearly a need for further research of a fundamental nature toward resolving inconsistencies and anomalies in the transition picture. It was also made clear that the transition information available to that time was dominated by factors such as wind tunnel boundary layer noise and other facility and model disturbances whose specific influence on transition had not been delineated quantitatively. The group agreed that an effective, practicable research program, limited in scope, would be undertaken with attention directed to identification and evaluation of the effects of the disturbance environment in quiet wind tunnels and in ballistic ranges. The program would be primarily experimental but with close theoretical support and should be directed toward understanding of fundamental processes. Program coordination should be accomplished by a committee of people who are active in transition research; it was concluded that the Transition Study Group could satisfactorily perform this function. The facilities of the represented laboratories were deemed adequate to initiate such a program, and the group members expressed a willingness to participate in a cooperative effort. It is to be stressed that the program is not to be considered an exclusive effort of the represented laboratories and it is hoped that the work of other groups will complement the program.

*More particularly, the matter was dealt with by an ad hoc committee consisting of Drs. C. D. Donaldson, A. Goldburg, and M. V. Morkovin.

III. Proposed Programs

The Group met again in January 1971 at which time preliminary plans for its program were formulated. At a subsequent meeting in July 1971, the program was subjected to further scrutiny and refinement.

The deliberations of the group yielded a number of guidelines to be used in the formulation of specific programs:

1. Any effects specifically and only associated with test facility characteristics must be identified and, if possible, avoided. This points to emphasizing studies in ballistic ranges and "quiet" tunnels.

2. Attention must be given to disturbances introduced by model surface, model material and internal structure. This includes effects of tip material and integrity, model ringing and the effect of non-uniform temperature distribution due to tip heating. Experimental studies should include documentation of these various factors.

3. Details of coupling of disturbances of various kinds to the boundary layer must be understood theoretically and experimentally, so that the sensitivity of the transition process to the flight environment might be determined.

4. The models should be of simple geometry. A slender cone ($5^{\circ} - 10^{\circ}$ half angle) is favored with tip bluntness (needed to avoid melting in range tests) held to less than 3%.

5. Tests should have ranges of overlapping parameters, and where possible, experiments should have redundancy in transition measurements.

6. When coupled with information on atmospheric disturbances, it should be shown how the work can be related to transition in flight.

**The original membership of the NASA Transition Study Group was:

- Eli Reshotko, Chairman - Case Western Reserve University
- Richard D. Wagner, Secretary - NASA-Langley
- Mitchel H. Bertram - NASA-Langley
- Paul F. Brinich - NASA Lewis
- Alfred Gessow - NASA Headquarters
- James M. Kendall, Jr. - Jet Propulsion Laboratory
- Philip S. Klebanoff - National Bureau of Standards
- W. Carson Lyons, Jr. - Naval Ordance Laboratory
- Joseph G. Marvin - NASA-Ames
- E. D. McElderry, Jr. - AFFDL-Wright Patterson AFB
- Jack D. Whitfield - von Karman Gas Dynamics Facility-ARO

The only changes have been the succession to the group in late 1972 of

Ivan E. Beckwith - NASA-Langley

following Mitchel Bertram's untimely death and the much regretted resignation of Richard Wagner because of a new work assignment.

The specific programs proposed were:

A. Program for Resolution of Wind Tunnel Data

B. Program at M = 4.5

C. Program at M = 2

D. Low Speed Experimental Program

E. Development of Quiet Wind Tunnels

F. Theoretical Program

The level of effort available for the program was estimated as 8 man-years per year which was about 30% of the total existing effort in the transition area for the combined laboratories. The Group agreed that the program should be incorporated into the current research at a high level of priority.

The Transition Study Group or subgroups thereof have continued meeting on the average of twice a year for program planning, direction and review.

IV. Program Details and Results

The elements of the overall program will now be described in greater detail together with the results obtained to date.

A. Program for Resolution of Wind Tunnel Transition Data

The Group recognized a responsibility to assist in the resolution of differences in transition data in noisy tunnels to gain better understanding of the limitations of wind tunnel transition data. These differences could be ascribed to different disturbance environments in the test section, different measurement techniques, different definitions of transition, or a combination of these and other factors.

Specifically an investigation was undertaken to resolve previously reported differences between boundary layer transition Reynolds number data obtained on two similar 5° half-angle cones in the Ames 3.5-Foot Hypersonic Wind Tunnel using thermocouples and in the Langley 18" Variable Density Wind Tunnel using thermal paint and thermocouples. These data (Figure 1) measured at similar test conditions display different levels of transition Reynolds number as well as different trends with unit Reynolds number.

To investigate these differences, new measurements were made in slightly modified versions of both facilities. Two models, one instrumented with thermocouples and the other with thin film gauges were used in both facilities. This was done in an attempt to alleviate any inconsistencies due to unknown model influences and differences between the detection techniques. Turbulence levels and pressure fluctuation levels were also measured in both facilities.

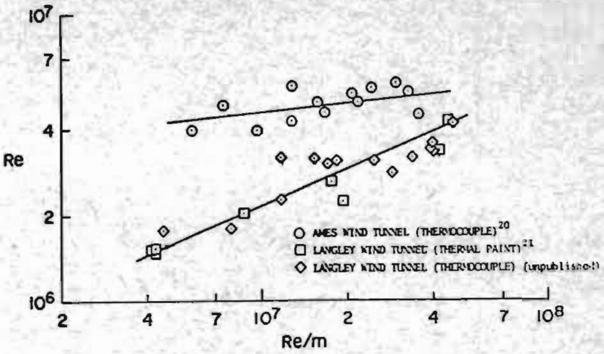
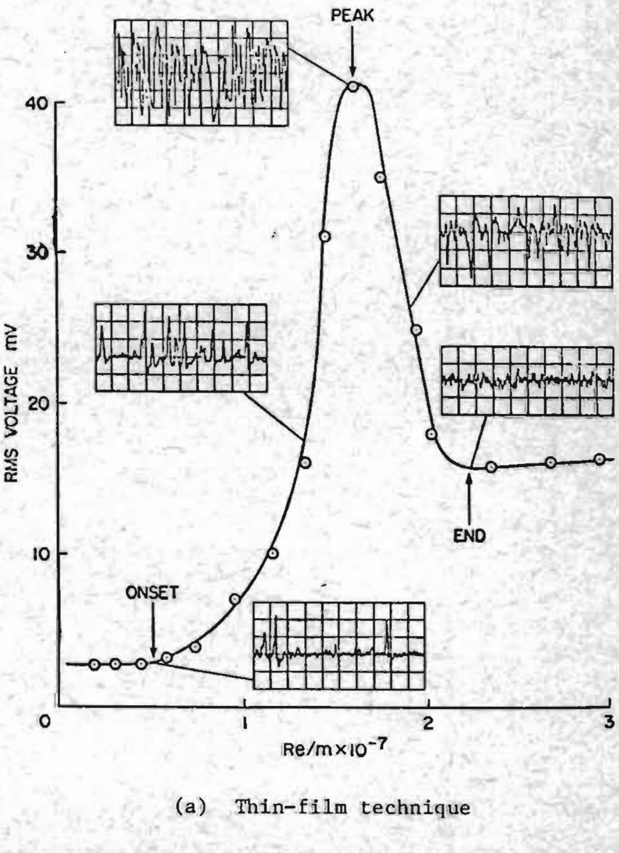


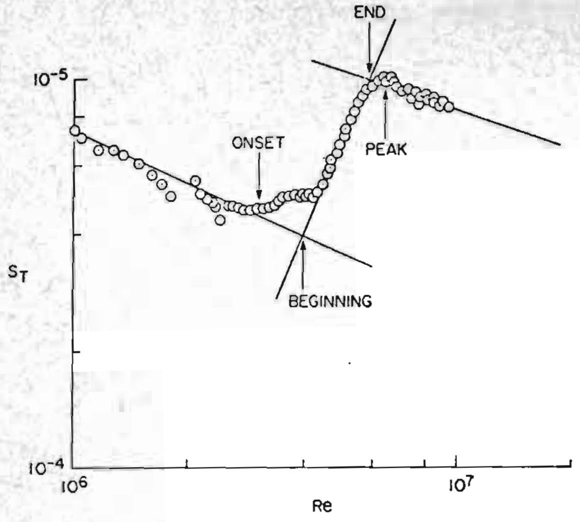
Figure 1. Comparison of previous transition Reynolds number data (beginning) from two wind tunnels, $M_\infty \approx 7.5$, $\theta_c = 5^\circ$, $T_w/T_o \approx 0.4$ (from Ref. 22).

The results as described in the recent paper by Owen et al[22] show a difference in the transition signatures displayed by thin film gauges and by thermocouples (Figure 2). The variation of the rms voltage fluctuations of a single thin film gauge over a range of unit Reynolds numbers (Figure 2a) clearly shows a rise from the laminar to turbulent level with an intermediate peak. The onset of intermittency, the peak signal and the end of transition can be clearly identified. From the variations in Stanton number through transition as measured using the thermocouple technique (Figure 2b), four points in the transition region can be identified: the onset or first



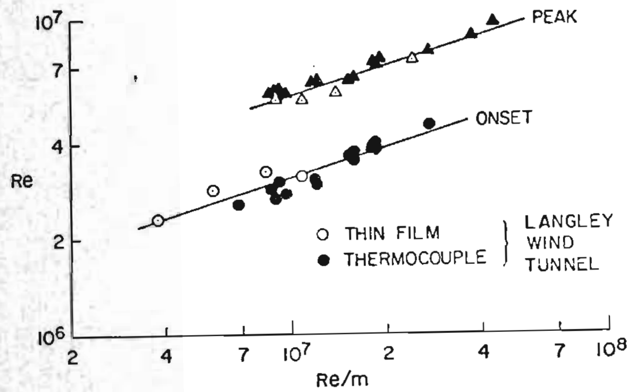
(a) Thin-film technique

Figure 2. Determination of transition location (from Ref. 22).

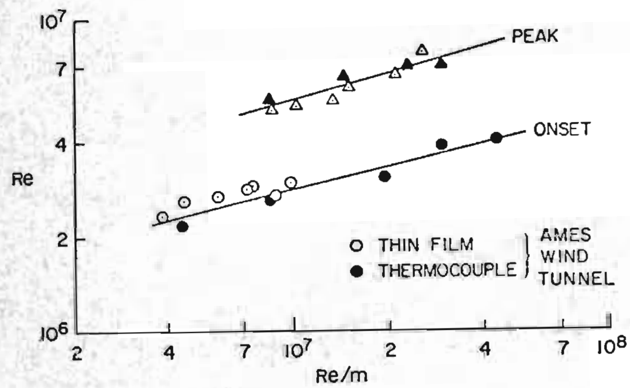


(b) Thermocouple technique

Figure 2. Determination of transition location (from Ref. 22).



(a) Langley wind tunnel



(b) Ames wind tunnel

Figure 3. Comparison of transition detection techniques, $M_\infty \approx 7.5$, $\theta_c = 5^\circ$, $T_w/T_o \approx 0.4$ (from Ref. 22).

consistent departure from a laminar heat transfer rate, the peak Stanton number and then the conventional "beginning" and "end" of transition obtained from the intersection of a line through the transitional data with the laminar and turbulent curves respectively.

Interestingly enough, the onset and peak locations were the same by both techniques in both facilities (Figure 3) but there was a difference in the "beginning" of transition between the two facilities (Figure 4) as obtained by the thermocouple technique. Note however that by its definition, the "beginning" of transition depends on the transitional behavior of heat transfer which in turn may depend on the slightly different disturbance levels and spectra in the two facilities. These factors are discussed in more detail in the paper by Owen et al[22]. In both facilities the dimensionless pressure fluctuation level decreases with unit Reynolds number and so increased transition Reynolds numbers are associated with decreasing dimensionless free stream pressure fluctuation levels.

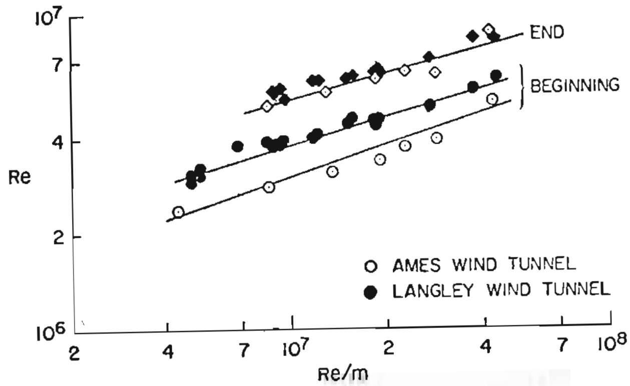


Figure 4. Comparison of thermocouple transition data in the two wind tunnels $M_\infty \approx 7.5$, $\theta_c = 5^\circ$, $T_w/T_o \approx 0.4$ (from Ref. 22).

In summary, the previously reported differences in transition data between the Ames and Langley tunnels were not apparent in the results of Owen et al. However, no inferences as to universality should be drawn from these results. Measurements made on the centerline of the Ames tunnel indicate disturbances whose length scales are several times their width, as would be expected from sound radiated by the turbulent boundary layers on the tunnel walls.

B. Program at M = 4.5

C. Program at M = 2

These two programs are related in conception and were both designed to be performed in the same facilities at three of the represented laboratories - AEDC, NOL and JPL. Hence they will be described together. The following are the reasons for choosing these two particular Mach numbers:

Mach number 4.5 was chosen because, (a) the JPL "quiet" tunnel has a range of quiet operation at this Mach number; (b) in the AEDC ballistic range, the model precedes disturbances resulting from sabot impact; (c) NOL ranges are capable of overlapping test parameters of both of the above facilities; (d) cone tip ablation can be avoided at $M \approx 4.5$; (e) higher modes of boundary layer instability are active and therefore data at this Mach number are somewhat indicative of what may occur at higher Mach numbers; and (f) this Mach number is very much within the range where transition results in conventional wind tunnels are dominated by the noise radiated by the turbulent boundary layers on the tunnel walls.

At Mach number 2, on the other hand, only the lowest mode of boundary layer instability is generally present. Further, it is generally felt that at this Mach number, the noise radiated from the tunnel walls is not a significant factor. Hence, if the tunnel boundary layer were either all laminar or all turbulent, it would have negligible effect on transition on a model at Mach 2. If transition occurs somewhere on the tunnel wall, then waves result which can affect transition on a model.

AEDC Ballistic Ranges - The studies recommended for AEDC were:

(i) Effects of Cone Vibration Characteristics - Boundary layer transition should be measured on models constructed of different materials. Bench tests can be used to determine the vibration characteristics of the models (ringing frequency for example).

(ii) Effects of Small Surface Roughness - For the small models and high unit Reynolds numbers of the range experiments, model boundary layers are quite thin and inadvertent model surface roughness may influence transition.

NOL Thousand Foot Hyperballistics Range - Because of its unique temperature control section, the NOL range offered considerable promise for increasing the understanding of the transition process at Mach number 4.5 where the stability theory predicts diverse T_w/T_{aw} sensitivity for the various instability modes. Unfortunately, this portion of the program could not as of this date be implemented.

JPL 20" Supersonic Tunnel Including "Quiet" Operation - The JPL Tunnel was to be used to investigate those factors that could not adequately or systematically be studied in the ballistic range, but which might influence boundary layer transition in the range experiments. These factors include:

(i) Small Angle of Attack - It is difficult to hold angle of attack variations in range tests to less than about 3 degrees. An assessment of the effects of these small angles of attack on transition is critical to the interpretation of ballistic range data.

(ii) Model Surface Temperature Variations - Aerodynamic heating of the test model as it proceeds through the range can raise the model tip surface temperature above $2{,}000^{o}R$ at $M = 4.5$ while the remaining parts of the model have a surface temperature close to $500^{o}R$. The effects of the hot tip and the large surface temperature gradients near the tip should be documented.

(iii) Model Vibration - Launch accelerations in the ballistic range can be 100 to 200 thousand g's, and ringing of the model in flight introduces a source of disturbance not normally encountered in full scale vehicle flight or in wind tunnel tests.

Unfortunately, at the test conditions for "quiet" operation in the JPL 20 Inch Supersonic Tunnel, boundary layer transition cannot be obtained on a model. Conclusions, as to the behavior of transition, will have to be inferred from boundary layer stability measurements.

The principal facility used in these programs was the AEDC-VKF Aeroballistic Range K. This is a 100 ft. long 6 ft. diameter range. The range air was quiescent prior to launch and there was no significant sound pressure level measurable near the range centerline at the photographic stations prior to arrival of the cones at either of the test Mach numbers. Further information on the range, the launcher and range instrumentation are given by Potter[23]. The technique of determining the transition location from focused shadowgrams is described in some detail by Potter[24]. Although when measured this way transition is determined by observation of the outer edge of the boundary layer, Potter argues that these locations are the same as would be obtained by a measurement taken on the cone surface if one considers the inherent scatter in transition data from shadowgrams[24].

A free-flight model rarely maintains zero angle of attack throughout its flight in the range. Using reported results of wind tunnel experiments by Ward[25] and Mateer[26] as well as corroborative data from Kendall at JPL, Potter[23] devised a technique for utilizing observations obtained at small angles of attack and adjusting them to zero angle of attack. The results presented here are for zero angle of attack after adjustment.

To determine the influence of surface roughness under the cold wall conditions of range testing, the results for models with machined (screw-thread) roughness were compared with those for nominally smooth models. The data shown in Figures 5 and 6 for Mach 5 and Mach 2 respec-

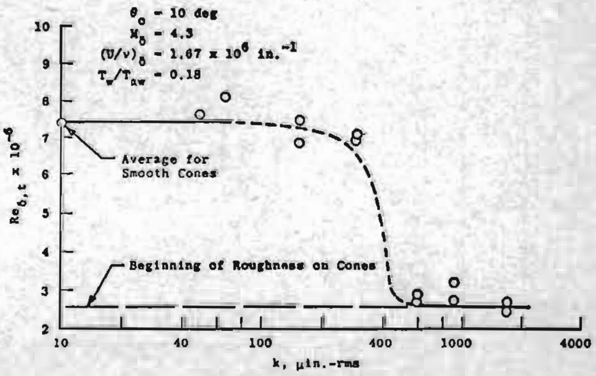


Figure 5. Effect of distributed surface roughness on Cone at Mach 5.1 (from Ref. 23).

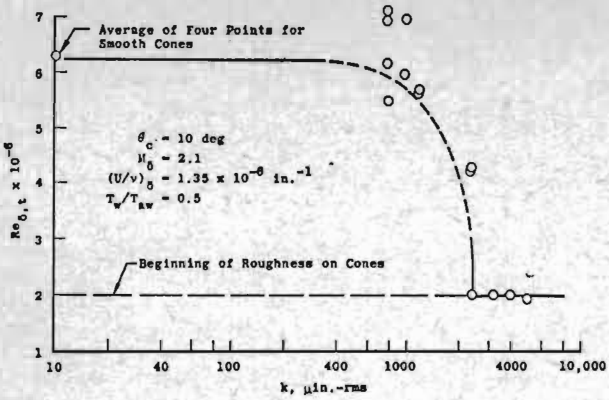


Figure 6. Effect of localized roughness at Mach 2.2 (from Ref. 23).

tively show no discernable effect of roughness until the roughness height is well above that for nominally smooth cones. Based on these data and other presented arguments regarding effects of the sabot on model finish, Potter [23] concludes that surface roughness has not significantly affected any of the data represented as smooth.

Potter investigated effects of model vibration by using cones of three different materials and/or construction: an aluminum cone whose natural frequency was 6880 Hz, a lexan cone of natural frequency 2060 Hz and a ballasted lexan cone that vibrated at 1250 Hz. Sufficient launches were carried out to report that no significant differences in transition Reynolds number were detected between the models. In a corroborative study by Kendall in the JPL 20" tunnel, the vibration of a 10 degree cone model at 3100 Hz at an amplitude that could be sensed by touch, failed to move transition at $M_\infty = 2.2$ and 4.76[23].

Another potential influence on transition under range conditions is the non-uniform surface temperature of the model due to aerodynamic heating. Potter concludes that this hot-tip

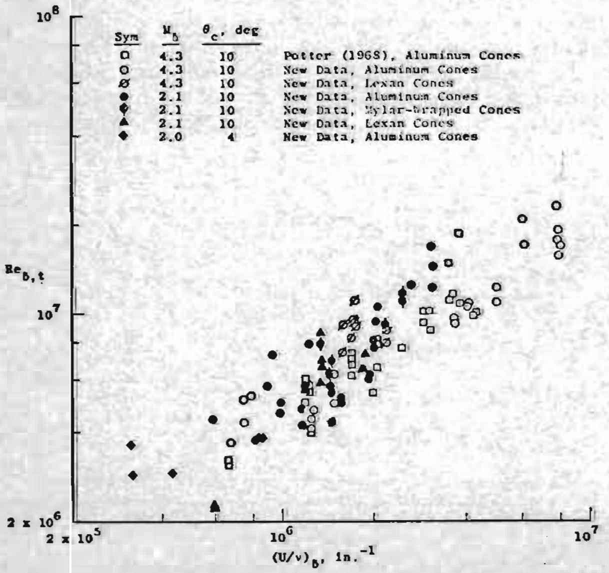


Figure 7. All smooth-cone, quiet-range transition results (from Ref. 23).

effect is not an obvious factor in his results since the tips of his cones show no discernable ablation and the transition which invariably takes place well downstream of the tip seems to be independent of cone or tip material[23].

Potter's transition data for smooth cones at zero angle of attack are shown in Figure 7. The results fully support Potter's earlier range results[19] in showing a variation of transition Reynolds number with $(\frac{u}{\nu})^{0.6}$. The dominating length (or frequency) suggested by Reshotko[11] has yet to be identified for these range tests. The surprising equality of transition Reynolds numbers at the two Mach numbers deserves further investigation. Whether or not the different ratios of T_w/T_{aw} compensate for a Mach number effect remains a question.

This study has more intensely focused on the original question rather than resolve it. It has shown that a unit Reynolds number dependence is obtained in ballistic range tests that cannot be explained by any of the range-peculiar factors investigated. The resolution of this question remains a challenge to the Transition Study Group and to the transition community-at-large.

D. Low Speed Experimental Program

Measurements with hot wire anemometers at low speeds enable a microscopic examination of the transition process to a degree of detail that is not available in the compressible high speed studies. Evidence exists that some of the phenomena observed at low speeds still are present at high Mach number. For example, NBS measurements of the amplification of free stream disturbances by low speed boundary layers reveals amplification of low frequency disturbances that should be stable according to stability theory[27]; the same phenomenon seems to be present in measurements at JPL at $M_\infty = 1.6$.

Because of the apparent promise for uncovering mechanisms that may be fundamental to the transition process at high Mach number, the NASA Transition Study Group recommended continued study of amplification of free stream disturbances in low speed, laminar boundary layers. Such work is in fact continuing in a number of laboratories.

E. Development of Quiet Wind Tunnels

The "quiet" tunnel - having laminar rather than turbulent boundary layers on the nozzle walls - has shown itself to be attractive for transition related studies because it averts a prime cause of facility induced transition at supersonic Mach numbers above 2.5. This does not mean that the test section of a "quiet" tunnel is disturbance-free and one must proceed cautiously in assessing the usefulness of such facilities.

The JPL 20" Supersonic Tunnel achieves quiet operation when operated at low stagnation pressure. However, the unit Reynolds number is such that transition is not generally observed under the limited conditions of quiet operation.

It was felt desirable to develop a high unit Reynolds number quiet tunnel for transition research and also for the study of turbulent boundary layer development in the absence of noise contamination due to turbulent nozzle wall boundary layers. As with other facilities, the proper utilization of a quiet tunnel depends on identification and documentation of its disturbance environment.

Measurements at JPL and at Langley have shown that when tunnel sidewall boundary layers are laminar, stream disturbance levels are greatly reduced. Hence, one of the principal design objectives for a quiet tunnel is to maintain laminar boundary layers on the walls of the nozzle and test section up to sufficiently large Reynolds numbers so that model boundary layers are not affected by noise radiated from the tunnel walls.

Klebanoff and Spangenberg at the National Bureau of Standards have undertaken a study of nozzle laminarization by suction through a longitudinally slotted wall in the NBS Mach number 2 tunnel. In a cooperative effort, NASA-Langley is providing to NBS a porous wall of weave construction to fit the NBS nozzle. These should enable a critical test of the validity of an area suction approach.

NASA-Langley is seeking to build a quiet tunnel with test section unit Reynolds numbers up to 10×10^6 per foot (30×10^6 per meter) at Mach = 5. A conception of such a facility as described by Beckwith[28] is shown in Figure 8.

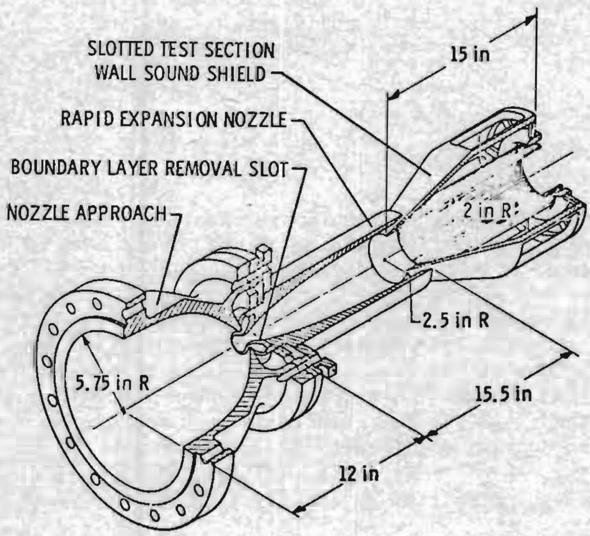


Figure 8. Mach 5 Pilot Quiet Tunnel - Slotted nozzle with rodded wall sound shield installed in test section (from Ref. 28).

The nozzle consists of a convergent approach section followed by an annular slot just ahead of the throat and a rapid expansion just beyond the slot. The function of this slot (Figure 9) is to remove the turbulent boundary layer that forms in the settling chamber and nozzle ap-

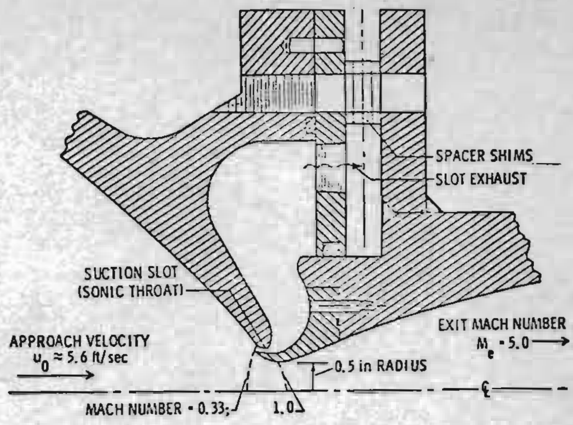


Figure 9. Mach 5 Pilot Quiet Tunnel - Transonic region of nozzle (from Ref. 28).

proach including the Taylor-Görtler vortices that would be present due to the concave curvature in the approach. The slot lip is placed just downstream of the inflection point in the local streamlines. The new boundary layer that forms is therefore not subjected to concave curvature until far downstream of the throat. The rapid expansion section expands the flow to the desired test Mach number and a rodded sound shield is installed at the nozzle exit.

This rodded shield encloses the test section. The nozzle and shield are mounted in a vacuum chamber so that the nozzle wall boundary layer can be removed before the flow enters the shielded region. The new boundary layer that forms on the rods is also partially removed through the gaps between the rods. If sonic flow is maintained in these gaps then noise generated outside the shield will not be transmitted into the test section. If laminar flow is maintained on the shield itself then it will not itself radiate sound into the test section.

Preliminary tests of the pilot slotted nozzle indicated that transition occured at test section $Re/_{ft} \simeq 1.5$ to 2.5×10^6 ($Re/_m \simeq 5$ to 8×10^6). With some initial modification of the slot lip and slot spacing this was increased to 4 to 5×10^6 per ft (12 to 15×10^6 per meter) and there is a reasonable hope of achieving the design objective of 10×10^6 per ft (30×10^6 per meter). To be noted however is that at $Re/_{ft} = 13 \times 10^6$ (40×40^6 per meter) where the boundary layer was fully turbulent, the measured disturbance levels were considerably smaller than in conventional nozzles[28].

Arrays of longitudinal rods were tested in Langley wind tunnels at $M_\infty = 6$ and $M_\infty = 8$. To simulate suction on the rods, the arrays were mounted at an angle of attack of 10° (Figure 10). Pressure fluctuation levels in the shielded region were reduced to half the value in the open tunnel as long as the boundary layer on the rods remained laminar. The laminar nature of the rod boundary layers ahead of the indicated transition (at $Re/_{ft} = 5.5 \times 10^6$ for $M_\infty = 8$ and at $Re/_{ft} = 8 \times 10^6$ for $M_\infty = 6$) was confirmed by the good agreement between measured heat transfer rates on the windward side of the rods

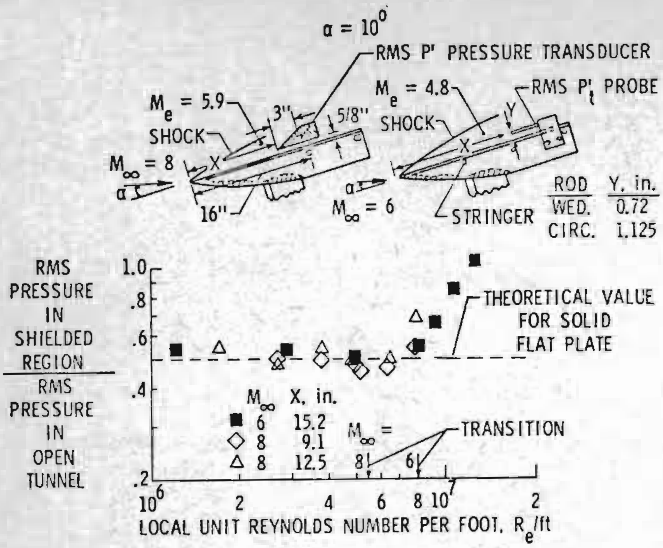


Figure 10. Noise reduction in rod model flow field (from Ref. 28).

and the theoretical predictions for swept infinite cylinders[28]. Beckwith argues that the observed transition on the shield arrays was possibly due to crossflow instability of the flow about the swept rods and could be delayed to higher unit Reynolds numbers by cooling the rods.

Hence with further development of the nozzle throat slot and the sound shield for the test section, it seems entirely possible to realize a facility that will give "quiet" flow to unit Reynolds numbers of 10 x 10^6 per ft or 30 x 10^6 per meter. Such a facility should be valuable in the laboratory study of factors affecting transition and in the study of the nature of turbulent shear flows at supersonic speeds. A 20-inch quiet tunnel has been proposed[28] predicated on the successful operation of the 5-inch pilot facility.

F. Theoretical Program

While the Transition Study Group did not formulate a specific theoretical program, it nevertheless recommended continuation of the study of boundary layer stability characteristics and further theoretical studies of the coupling of various types of disturbances to the boundary layer to complement the experimental treatment of these effects. These would include: external moving pressure fields, free stream turbulence, model vibrations, surface roughness, etc.

A most successful blending of theory with experiment has been in the work of Mack and Kendall at JPL. Over the years Mack has done extensive calculations of the stability of compressible boundary layers by direct numerical solution of the complete parallel flow disturbance equations. He obtained the very important result that when the relative flow between the wall and the propagating Tollmien-Schlichting wave is supersonic, there are multiple modes of instability with the higher modes having somewhat different properties than the traditional first Tollmien-Schlichting mode. These findings are presented in great detail by Mack in Reference 29. A particular result that underlies subsequent work is shown in Figure 11 where it is to be noted that maximum spatial amplification rates for first mode disturbances at supersonic speeds occur for oblique waves propagating at 45° - 65° to the flow direction. The most unstable second (and higher) mode disturbances propogate in the flow direction ($\psi = 0^\circ$). The Mach number zero point is of course for $\psi = 0^\circ$ by virtue of Squire's theorem.

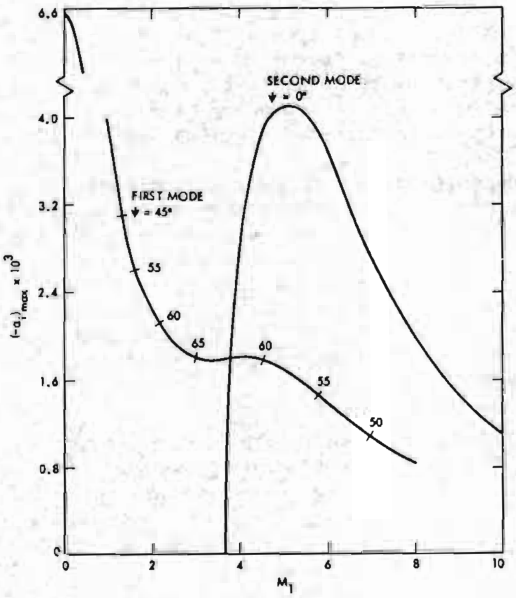


Figure 11. Effect of Mach number on maximum spatial amplification rate at $R = \sqrt{Re_x} = 1500$. Insulated wall, wind-tunnel temperatures (from Ref. 32).

The findings of supersonic stability theory were put on a firm footing by the experiments of Kendall[9] at $M_\infty = 4.5$ in the JPL 20-inch tunnel operated in the "quiet" mode, that is with laminar wall boundary layers. By introducing artificial oblique periodic disturbances into a flat plate boundary layer, Kendall verified the predicted unstable frequencies, phase speeds and amplification rates of the first mode. Further, upon introducing two-dimensional disturbances, the existence of amplified second mode disturbances with the predicted characteristics was confirmed.

Nevertheless, it was early determined that a mechanism other than simple instability was operative at $M_\infty = 4.5$ when the tunnel wall boundary layers were turbulent. Kendall[30] observed that fluctuations of all frequencies grew monotonically from the flat plate leading edge, that is, ahead of the region of expected instability. In an attempt to account for these observations Mack extended his theory to include the response of a layer to incoming sound waves[31,32]. He, in fact, identified the radiated sound from the tunnel walls with incoming supersonic disturbances to the test boundary layer and proceeded to calculate the response of the boundary layer to this incoming forced disturbance. The results gave realistic trends; fluctuations of all frequencies were predicted to grow rapidly larger with increasing distance from the leading edge reaching a broad peak in the vicinity of the region at which amplification due to instability begins. The magnitude of the peak was found to be inverse to the frequency.

Mack then calculated the growth of disturbances at selected frequencies by using the forcing theory up to the neutral stability point and stability theory beyond the neutral point. The calculated results compare very well with Kendall's measurements of disturbance amplitudes at three different frequencies (Figure 12)[33] lending some credence to the hypothesis. However, the forced disturbance is different from the free disturbance and the mechanism by which the forced wave turns into the free wave is unknown and has been ignored.

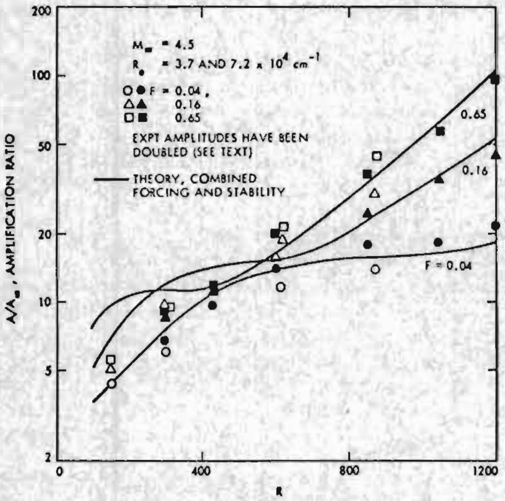


Figure 12. Comparison with combined theory, $M_\infty = 4.5$ (from Ref. 33).

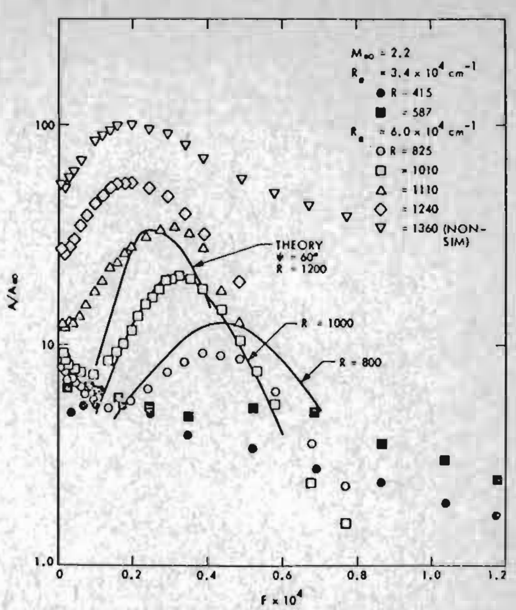


Figure 13. Boundary layer frequency response, $M_\infty = 2.2$ (from ref. 33).

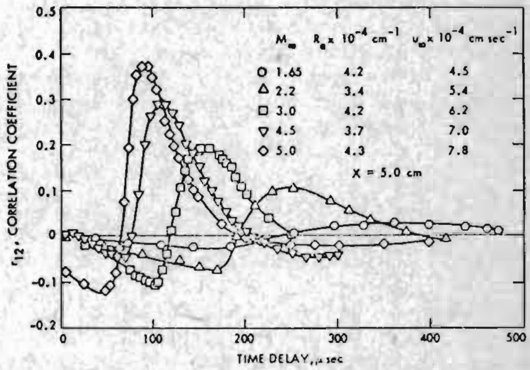


Figure 14. Cross correlation of free-stream and boundary layer fluctuations (from Ref. 33).

The measurements shown in Figure 12 are representative of those by Kendall at Mach numbers 3, 4.5 and 5.6. At $M_\infty = 2.2$ (Figure 13), the frequency response in the boundary layer shows definite peaks at the most unstable frequencies and seems to be what would result from instability amplification of the free stream disturbance spectra by $60°$ waves. The data for $M_\infty = 1.6$ also show the same frequency selective amplification patterns as for $M_\infty = 2.2$.

Kendall further measured the correlation coefficient between the free stream and boundary layer fluctuations (Figure 14). The fluctuations have poor correlation at $M_\infty = 1.6$ but the degree of correlation improves with increase in Mach number[33]. Kendall concludes from these sets of measurements that while wind tunnel sound exists in sufficient intensity to be a potentially important source of boundary layer disturbances for all Mach numbers considered, it seems less effective than other unsteadinesses in disturbing the boundary layer at $M_\infty = 1.6$ and $M_\infty = 2.2$[33].

In a hypersonic test at $M_\infty = 8.5$, the boundary layer fluctuation spectra show decided peaks at the frequencies of second mode amplification and the spatial amplification rates measured in the boundary layer of a $4°$ half angle cone show acceptable agreement with stability theory (Figure 15) , and little if any effect of tunnel sound. Taking all of these results together, the most pronounced contamination of the wind tunnel environment by radiated tunnel sound seems to be in the middle Mach number range that stretches from 2.5 to 7 or so.

The impact of tunnel disturbance environment on transition was of course realized long before these detailed experiments by Kendall. Mack has over the years tried to reproduce observed

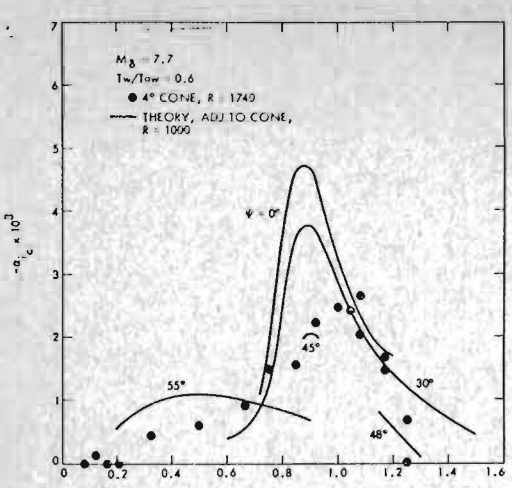


Figure 15. Comparison with stability theory, $M_\infty = 8.5$ (from Ref. 33).

transition features by calculations based on stability theory, and with some success. A recent set of calculations is particularly interesting.

On the premise that transition can be correlated with the attainment of a certain disturbance amplitude level, Mack set out to calculate the effect of cooling at $M_\infty = 3$ on the length required to obtain this prescribed disturbance amplitude level. In less precise language he sought to obtain the effect of T_w/T_{aw} on $Re_{tr}/(Re_{tr})_{aw}$ for a noisy wind tunnel environment. This was done by calculating the growth of the most unstable disturbance from its initial amplitude in the tunnel. The results of his various calculations are shown in Figure 16[32]. The ordinate is the ratio of the length to attain amplitude A with cooling to that without cooling.

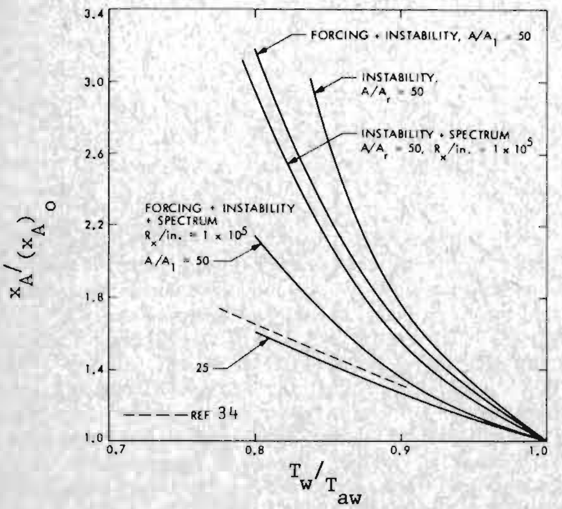


Figure 16. Five theoretical calculations of the effect of cooling on $x_A/(x_A)_o$ at $M_1 = 3.0$ (from Ref. 32).

The abscissa is of course the wall temperature ratio T_w/T_{aw}. The dashed line in the figure is the experimental result of Van Driest and Blumer in the JPL 20-inch tunnel at $M_\infty = 2.7$[34]. The first calculation is that based on stability alone to give $A/A_r = 50$ where A_r is fixed reference amplitude. The first improvement that can be made in the calculation is to account for the fact that as the boundary layer is cooled, the most unstable frequency at a fixed unit Reynolds number shifts to a lower value and hence to a region of the spectrum with increased energy. The spectra as measured by Kendall at $M_\infty = 3$ are used by Mack in this calculation. This result (instability + spectrum) is a slight improvement over the initial calculation. The third calculation uses the forcing theory in the sense illustrated by Figure 12. The resulting curve (forcing + instability, $A/A_1 = 50$) is no improvement over the earlier result. Here A_1 is the amplitude of the mass flow fluctuations in the incoming sound wave.

The final calculations combine all the elements of the previous ones: the forcing theory, the stability theory and the spectrum of free stream disturbances. As seen in Figure 16 a much reduced effect of cooling is obtained and for $A/A_1 = 25$, the trend is in close agreement with the measurements of Van Driest and Blumer*.

While this calculation is very speculative, it gives a hint of what is possible and also shows the importance of knowing the disturbance environment in the test facility. Further, it is an initial example of the theoretical consideration of forced disturbances in a transition context.

Another recent study of forced disturbances is that of Rogler and Reshotko[37]. To acquire insight into the role of free stream turbulence on transition, they studied the interaction between an incompressible boundary layer and an array of single wave number vortices convected at the mean free stream velocity. They found the amplitudes in the boundary layer to grow in the downstream direction with the maximum amplitude arising near the leading edge for small vortex diameters and further downstream for larger diameters. This is closely related to the trend obtained by Mack[32] for tunnel sound. Rogler and Reshotko further speculate that if initial turbulence levels are sufficiently large the disturbances can grow by forcing mechanisms to nonlinear levels and lead to turbulent flows without resort to Tollmien-Schlichting amplification.

There are thus two identified mechanisms, associated with tunnel sound and free stream turbulence respectively, that can lead to growth of boundary layer disturbances apart from considerations of instability.

The consideration of forced disturbances is an important new direction in theoretical studies related to instability and transition.

*Note that all these amplitude ratios are well below the e^9 or e^{10} levels suggested by Smith and Gamberoni[35] and Jaffe, Okamura and Smith[36].

V. Future Directions

While the Transition Study Group has not formally developed specific continuation programs, the thinking of the Group is to continue with programs emphasizing cooperation between different groups of investigators, ranges of overlapping parameters in different facilities, redundancy in transition measurement and above all complete documentation of all factors contributing to the test disturbance environment. Three proposed areas of concentration are:

1. Study of disturbance generating mechanisms- continuation of experimental and theoretical study of facility-connected and model-connected disturbances on the transition process.

2. Configurations other than nearly sharp slender cones - primarily introducing the additional factor of bluntness. It is not within the nature of the program to introduce too many new factors at one time.

3. Flight program - if the work of the Transition Study Group is to have relevance to the prediction of transition in atmospheric flight, then some carefully planned flight tests within Transition Study Group guidelines are in order.

VI. Concluding Remark

The program of the NASA Transition Study Group is a fundamental one seeking to develop procedures yielding information relevant to transition in flight. This is not a modest objective; nor is the adopted cooperative approach entirely conventional. Success is felt to depend on just this kind of concerted effort, experimental and theoretical, encompassing the talents and facilities of the many participating laboratories.

The author wishes to thank the participants in the Transition Study Group program for the help given in preparing this paper and for the permission to freely use some of their yet unpublished material.

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DISCUSSION

D.J. Peake (National Research Council, Ottawa, Canada): With respect to the design of the Mach Number 5 pilot wind tunnel at Langley; as well as installing boundary layer control on the tunnel walls, is the tunnel mainstream treated upstream of the test section to remove acoustic and vorticity fluctuation disturbances?

E. Reshotko: Mr. Beckwith of NASA-Langley assures me that the settling chamber of the M = 5 pilot Quiet Tunnel is expressly designed for the removal of acoustic and vorticity disturbances. The settling chamber is about 100 inches long. Within the first 50 inches there is an entrance baffle followed by a muffler 3.5" wide consisting of steel wool between a perforated plate and rigimesh plate, followed in turn by seven screens of progressively smaller mesh starting at 20 x 20 mesh with the last three being 50 x 50 mesh. This leaves more than 50 inches of length for remaining disturbances to decay before the flow reaches the nozzle entrance.