# Boundary-Layer Transition on Supersonic Cones in an Aeroballistic Range

J. LEITH POTTER*

*ARO, Inc., Tullahoma, Tenn.*

**Research was undertaken with the purpose of determining the effect of the unit Reynolds number on boundary-layer transition under conditions where disturbances associated with wind tunnel flows would not be present. The location of boundary-layer transition was determined from shadowgrams of nominally sharp, 4° and 10° semi-angle cones in an aeroballistic range at freestream Mach numbers of 2.3 and 5.0 and unit Reynolds numbers of $0.3 \times 10^6$ to $8 \times 10^6$ per in. Owing to constant and equal freestream and cone skirt temperatures, the average ratio of cone wall-to-adiabatic recovery temperature was 0.52 at Mach 2.3 and 0.19 at Mach 5.0. Features of free-flight experimentation that may be suspected of influencing boundary-layer transition were investigated. These included 1) oscillatory motion and finite angles of attack, 2) surface roughness, 3) vibration of the model, and 4) non-uniform (hot-tip) surface temperature. There was no evidence that any of these conditions influenced the major results. The data show local Reynolds number of transition increasing with unit Reynolds number for both Mach numbers. A siren was used to elevate the fluctuating sound pressure ratio by a factor of 200, but that produced no measurable effect on transition locations.**

## Nomenclature

$c_r$ = phase velocity of disturbance = velocity of wave propagation in freestream direction
$f$ = frequency of vibration of cone; also "function of"
$k$ = roughness depth profilometer reading
$M$ = Mach number
$N$ = number of cycles of vibration
$p$ = pressure
$\tilde{p}$ = fluctuating sound pressure amplitude
$Re$ = Reynolds number
$s$ = distance measured along surface from stagnation point
$T$ = temperature
$U$ = velocity
$\alpha$ = total angle of attack
$\alpha_p$ = angle of attack in plane of photograph
$\beta$ = characteristic angular frequency of disturbance spectrum
$\gamma$ = ratio of specific heats
$\delta$ = boundary-layer thickness
$\theta$ = characteristic orientation of disturbance spectrum
$\theta_c$ = cone semi-apex angle
$\lambda$ = characteristic wavelength of disturbance spectrum
$\mu$ = absolute viscosity
$\nu$ = kinematic viscosity = $\mu/\rho$
$\rho$ = mass density
$\phi$ = orientation of a cone meridian relative to the windward stagnation line where $\phi = 0$
$\omega$ = exponent in the approximation $\mu \propto T^{\omega}$

*Subscripts*

$aw$ = adiabatic wall
$j$ = at station of temperature discontinuity
$o$ = total, e.g., total temperature; also designates $\alpha = 0$
$t$ = transition
$w$ = cone wall
$\delta$ = local flow parameter at outer edge of boundary layer
$\infty$ = freestream
$\alpha$ = denotes $\alpha \neq 0$

Presented as Paper 74-132 at the AIAA 12th Aerospace Sciences Meeting, Washington D.C., January 30–February 1, 1974; submitted April 3, 1974; revision received September 26, 1974. The research reported herein was conducted partly at the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC) and partly at Vanderbilt University, Nashville, Tenn. Further reproduction is authorized to satisfy needs of the U.S. Government. This work forms part of a dissertation submitted to the Graduate School of Vanderbilt University in partial fulfillment of the requirements for the Ph.D. degree in Mechanical Engineering.

Index category: Boundary-Layer Stability and Transition.

* Deputy Director, Technology. Associate Fellow AIAA.

## I. Introduction

THE state of knowledge of boundary-layer transition has been summarized in notable reviews by Morkovin,[1,2] Mack,[3] and Mack and Morkovin.[4] It is a justifiable conclusion, after studying these reviews, that significant new contributions by experimentalists are necessary but that all possible care must be taken to control and define factors influencing the transition process in the experimental environment. By far, most previous experimental investigations of transition have been conducted in wind tunnels where it has been recognized that coupling between "tunnel" disturbances and flow in the boundary layer under observation almost always made such absolute measurements as local transition Reynolds number, $Re_{\delta,t}$, inapplicable in other environments, even when all of the more obvious dynamic similarity conditions were matched.

An investigation reported by Potter[5] was conducted to answer the previously unresolved question whether the local Reynolds number of transition, $Re_{\delta,t}$, would vary as local unit Reynolds number, $(U/\nu)_\delta$, varied under aeroballistic range conditions. Such a variation, commonly called a unit Reynolds number effect, was indeed found. Under the conditions existing, it was seen that $Re_{t,\delta} \propto (U/\nu)_\delta{}^{0.6}$. This discussion takes Ref. 5 as a point of departure and extends the scope to include a second Mach number, a broader range of unit Reynolds numbers, and extensive study of major "range-peculiar" factors at both Mach numbers.

The two Mach numbers were selected on the basis of stability theory[3] which suggests that different modes of boundary-layer instability are dominant at the two local Mach Numbers, $M_\delta$, which were approximately 2.1 and 4.3, respectively. The location of transition was measured on shadowgrams of 4° and 10°, semi-angle, nominally sharp cones at freestream Mach numbers, $M_\infty$, near 2.3 and 5.0. Transition, as the term is used here, refers to the station where the photographed boundary layer appeared to finally depart from a laminar condition.

The free-flight range is an experimental facility not widely exploited for boundary-layer transition studies, though some rather important information may be obtained from range experiments. The quiet atmosphere of the aeroballistic range offers an opportunity for study of boundary-layer transition free of the complex influences of stream vorticity, temperature spottiness, and noise which are known to be present in varying degrees in wind tunnels. However, there are some special features of aeroballistic experimentation which raise questions, and it is appropriate that they be reviewed in the context of their influence on boundary-layer transition. The ones discussed in this paper are: 1) finite angles of attack and oscillatory motion; 2) surface roughness under conditions of cold walls and large unit Reynolds number, i.e., thin boundary layer; 3) vibration of the model resulting from launch acceleration; and 4) nonuniform surface temperature owing to aerodynamic heating.

## II. Background Investigation

The background of the unit Reynolds number effect found in experiments was traced briefly by Potter.[5] Morkovin has since presented a general discussion in his review report[1] where he asserts that the "unit Reynolds number effect" does not represent one effect but a complex superposition of many functional relationships. Thus, he properly cautions against expecting any regular dependence of $Re_{\delta,t}$ on $(U/v)_\delta$ under all circumstances. His complete discussion may be read with profit by anyone interested in this subject.

Reshotko[6] has discussed the roles of $(U^2/v)_\delta$ and $(U/v)_\delta$. He shows that, depending on the circumstances, one could find either $Re_{\delta,t} = f_1(\beta v_\delta/U_\delta^2)$ or $Re_{\delta,t} = f_2(U_\delta \lambda/v_\delta c_r)$. It is possible that both $U_\delta/v_\delta$ and $U_\delta^2/v_\delta$ enter into the process of transition and that the relations would be clearer if laboratory facilities made it easier to vary $U_\delta$ for otherwise fixed conditions. Almost all experiments in which $U_\delta/v_\delta$ or $U_\delta^2/v_\delta$ are varied at constant supersonic Mach number actually involve the variation of $v_\delta$ alone. The term, unit Reynolds number, is used herein because that is the commonly accepted practice.

Recently, in somewhat limited wind-tunnel experiments, Ross[7] investigated this question and showed that $(U/v)_\delta$ seemed to be more significant than $(U^2/v)_\delta$. He produced a variation of $(U/v)_\delta$ by changing total temperature at $M_\infty = 4$ and the result was found to be $Re_{\delta,t} \propto (U/v)_\delta^{0.65}$. In the light of the knowledge of tunnel noise influences,[8,9] nearness to the exponent of 0.6 found in Ref. 5 may be coincidental. Generally, though by no means always, the exponent has been nearer 0.4 in wind tunnels where $(U/v)_\delta$ was varied by changing pressure at constant Mach number.

The correlation of transition Reynolds number with unit Reynolds number in supersonic and hypersonic wind tunnels has been shown to be predictable on grounds of noise pressure fluctuations radiated onto models from turbulent tunnel nozzle boundary layers. Laufer[8] demonstrated the influence of unit Reynolds number on sound pressure measurements and achieved an order of ten reduction in $\tilde{p}/p_\infty$ by dropping unit Reynolds number to a level which resulted in laminar nozzle wall boundary layers. Subsequently, Pate and Scheuler[9] constructed an empirical correlation for $Re_{\delta,t}$ at $3 \leq M_\infty \leq 8$ using tunnel wall boundary-layer parameters to represent the noise production and the resulting influence on transition on a model exposed to that noise. However, there would seem to be no relationship between those results and the present free-flight range experiments which are believed free of wall influence. At least, none has been suggested yet and the proposition credited to Morkovin at the beginning of this section probably is as close to an explanation as now exists.

There is a persistent tendency on the part of some investigators to make quantitative comparisons between sets of transition data from different experimental facilities, e.g., between wind tunnels and aeroballistic ranges. Aside from differences in recognized but incompletely evaluated parameters such as $T_w/T_{aw}$ and $T_\infty$, which may vary widely between tunnel and range at a given Mach number, there are other good reasons to reserve judgment when comparing these types of data. Unless the dominant causes of transition are clear and well controlled, there can be no confidence that the disturbance spectra acting on different boundary layers are equivalent. Whether disturbances originate in a wind tunnel stream or within the flowfield of a body in free flight through quiet air would seem to allow large differences in the amplitudes and frequencies which are imposed on boundary layers. Therefore, it is not surprising that the levels of transition Reynolds numbers for given shapes sometimes differ by factors of two or more between different wind tunnels and an aeroballistic range.

## III. Models and Range Systems

Data on models are given in Fig. 1. The 1.75-in.-diam aluminum cones were used for the earlier launches, but the 2.5-in.-diam aluminum cone later became the principal model. All Lexan® cones were of 1.75-in. diam. Only a few 4° semiangle cones were used, all at Mach 2.

The aluminum models were fabricated from 7075-T6 alloy and were given a surface finish of 10 $\mu$ in., rms, or better. (The subject of surface roughness is discussed later.) A nose radius of 0.005 in. was standard on both aluminum and Lexan cones. The latter is a polycarbonate resin which was selected because it would give a cone of appreciably different vibrational characteristics for comparison to the aluminum cone. This was desired for a study of the possible influence of vibration on boundary-layer transition, which is discussed in Sec. V-B. At first, it was supposed that the Lexan cones would require aluminum tips to prevent ablation, but trial flights in the range proved that an all-Lexan cone surface was feasible under these conditions.

Sabots used with these cones are discussed in the context of their relation to roughening of cone surfaces in Sec. V-B. Portions of the sabots in contact with the cones were given finishes of approximately 30 $\mu$ in., rms.

The aeroballistic range used for this work was AEDC-VKF Range K (Fig. 2). This is a 100-ft-long, 6-ft-diam range equipped with 6 dual-axis shadowgraph systems and a single high-quality Schlieren or focused shadowgraph system at the time these studies began. The latter, with an effective exposure duration of 0.15 $\mu$sec, was used to obtain the principal photographic data in this investigation. During the latter stages of the experiments, a second such photographic station was installed. A laser-front-lighted photographic system, with an effective exposure duration of 20 nsec was used to obtain information on cone surface conditions after launch.

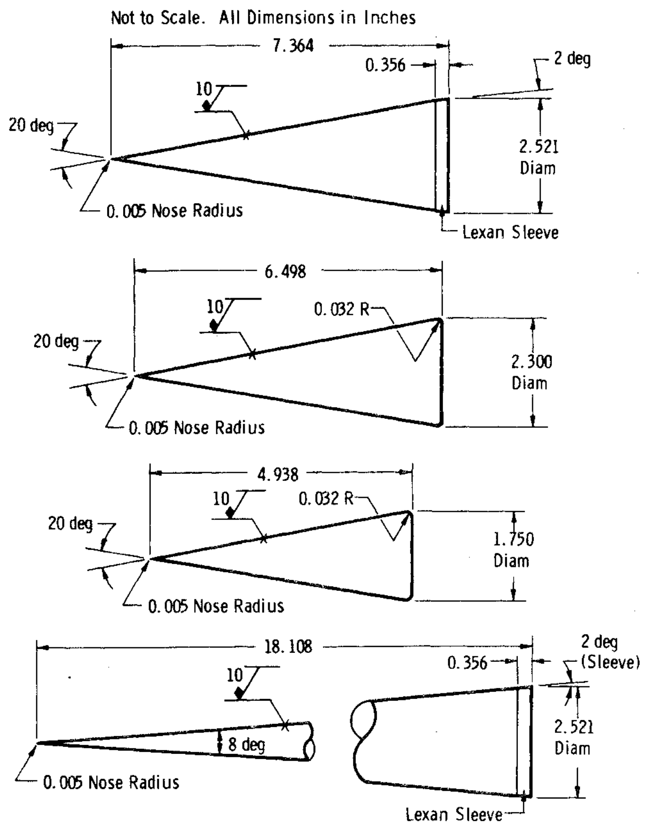


**Fig. 1 Cones used in experiments.**

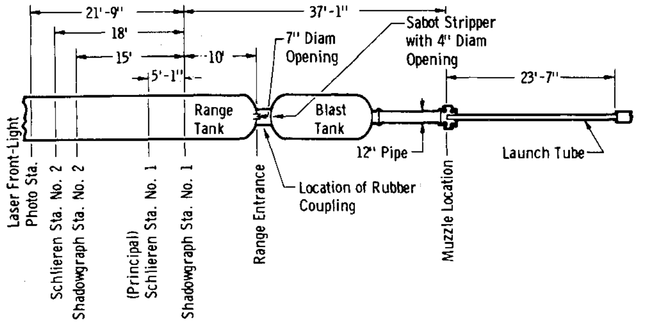


**Fig. 2 Schematic drawing of Range K showing uprange instrumentation and launch tube location.**

A single-stage, 2.5 in. caliber launcher was used. The muzzle of this gun was located approximately 42 ft from the principal focused shadowgraph station. The cones were launched without spin, and sabot separation was caused by aerodynamic force on the sabot components.

Noise in the range was monitored by small microphones, much as reported in Ref. 5. Except during some brief trial launches with a siren in the range, 0.25 in. Bruel and Kjaer® microphones were used. For the trial with a siren, described in Sec. V-B, a B&K 0.5 in. model was installed. Some pertinent characteristics of the complete system for sound pressure measurement are: dynamic range, 70 to 174 db rms, referenced to 0.0002 $\mu$ bar; limiting sound pressure: 185 db; and response: $\pm 3$ db from 40 Hz to 40 kHz at pressures and temperatures of this investigation.

## IV. Range Air Conditions

During the work reported in Ref. 5, the only measurable disturbances in the range air arose from vibration of the range structure. The dominant cause of this was traced to the impact of the sabot sections on the stripper located in the blast tank between the launcher muzzle and the opening into the range proper. Figure 2 illustrates the structural arrangement during the present launchings. If the sabot impact on the stripper were allowed to transmit disturbances through the steel tank structure at the speed of sound in that material, these disturbances would arrive in the wall near the photographic station well ahead of the cones. Vibrations of the structure are transmitted to the adjacent air, so it is apparent that disturbances would be introduced into the air ahead of the cones unless their speed is sufficient to outrun the structure-to-air noise. This was illustrated in Ref. 5.

To suppress this noise near its dominant source, a rubber coupling was installed between the blast tank and the range entrance as shown in Fig. 2. This simple step proved highly effective in that no significant sound pressure level was measurable near the centerline of the range at the photographic station prior to arrival of the cones at either Mach 2 or 5. Based on the characteristics of the microphone system described earlier, it is concluded that sound pressure level in the 40 Hz to 40 kHz regime was therefore no more than $\tilde{p} \approx 4.7 \times 10^{-4}$ mm Hg, rms. Using the lowest value of $p_\infty$, this gives $(\tilde{p}/p_\infty)_{\max} \approx 2 \times 10^{-6}$, rms. This is several orders of ten lower than the levels measured in supersonic wind tunnels at Mach 3.[8,9] However, the earlier discussion of disturbance spectra and the uncertainty about the relative importance of different amplitude-frequency bands should discourage quantitative speculation on the effect of this low-noise environment on transition Reynolds numbers.

Considering that the range air had long settling times between pumping or venting operations and was always near room temperature, there was little likelihood of measurable turbulence or temperature spottiness existing immediately prior to launches. The question of suspended dust was investigated by conducting tests using techniques developed for clean-room monitoring. Repeated tests always showed the number of dirt particles to be less than the level expected in an office area. The data scattered about the curve defining a class 10,000 clean room.

## V. Some Features of Aeroballistic Experimentation

This section is devoted to discussion of some aeroballistic conditions that were regarded as potential influencers of transition. If freestream disturbances are negligible in a range and if external disturbances are necessary for instability and transition in a boundary layer, then candidates may be found here. (Readers who are only interested in the transition Reynolds numbers may proceed directly to Sec. VI.)

### A. Influence of Angle of Attack

It is rare that a free-flight model maintains zero angle of attack throughout its flight. Therefore, it is necessary to establish a method for adjusting or correcting measured $s_t$ to its zero angle-of-attack value. Note that one must distinguish between the total angle and the angle in the plane of a photograph; they usually differ.

For the experiments discussed herein, the angles measured in two orthogonal shadowgraph planes were plotted as a function of length along the range, measured from the first to the last of six dual-axis shadowgraph stations. A typical Mach 5 case was characterized by an average velocity of approximately 5700 fps, a quarter-cycle of motion in roughly 17 ft, and a maximum amplitude somewhat under 2°. The wetted length of the conical model upstream of transition in this case was slightly under 5 in. Thus, there was a change in angle-of-attack of 2° in 17 ft of flight or 0.0029 sec, giving a rate of change of 690 deg/sec. In terms of wetted length to transition, the velocity was 13,700 lengths/sec. This enables expressing the oscillatory motion as 690/13,700 = 0.05°/wetted length. If we assume that the change in angle of attack during a time corresponding to flow from stagnation point to transition location is crucial, then this information seems to warrant the tentative assumption that the oscillations of the models, per se, in these experiments were of low enough frequency for that to be ignored as a factor in boundary-layer transition. However, there remains a need for a steady-state angle-of-attack correction.

Ward's wind tunnel data[10] on the effect of angle-of-attack are of interest because most of the experimental conditions, $M_\delta$, $\theta_c$, and $(U/\nu)_\delta$, were close to the present Mach 5 case. We have made a modification of Ward's result which consists of minor refairing of his curves between 0 and 1° leeward. The result is not in conflict with Ward's actual data points, and it seems to agree better with the present range data (if such a fine point is justified here). The final curve is shown later.

Under the auspices of a NASA Transition Study Group chaired by E. Reshotko, the writer had a series of contacts with J. M. Kendall, of the Jet Propulsion Laboratory in Pasadena, Calif. Kendall agreed to conduct experiments in the NASA/CIT Jet Propulsion Laboratory 20 in. supersonic wind tunnel with the objective of determining the angle-of-attack influence on transition under conditions as close as practicable to those of the present free flights. He also carried out other experiments to be mentioned later.

The data provided by Kendall[11] show that the circumferential variation of transition location on his 4° cone at zero angle of attack was as great (~13%) as the variation measured on the windward meridian when the cone was rolled at $\alpha/\theta_c = 0.25$. This spatial nonuniformity in transition locations is well known, both for two-dimensional and axisymmetric cases. It is important to remember that the wind tunnel data represent time-averaged transition locations. If the temporal fluctuations of transition location on a given meridian were superimposed upon the demonstrated circumferential nonuniformity, the percentage scatter of $s_t$ values at any instant of time could be twice as great. Temporal fluctuations of the type mentioned have been measured by several investigators.[12] When we add to the foregoing considerations the fact that total angle of attack in the aeroballistic experiments has an uncertainty of 0.2 to perhaps as much as 0.5°, while $\alpha_p$ has an uncertainty of the order of 0.1°, the requirements on the precision of the angle-of-attack correction are seen to be rather loose. A large number of data points is the best safeguard against biased results in this situation.

When the data of Kendall, Ward, and Mateer[13] are compared, relatively close agreement results for the dependence of $s_t/s_{to}$ on $\alpha/\theta_c$ for $\phi = 0°$ and 180°. Conditions of the latter two investigators were reasonably close to the present Mach 5 case; Kendall's experiment was performed with Mach number near 2. In view of the level of accuracy justified and the general agreement of the several sets of data, it was convenient to adopt Ward's results for $\phi$ of 0° and 180° as standard. To establish corrections to $s_t$ for intermediate values of $\phi$, the two sets of data given by Mateer and Di Cristina[14] were averaged and small shifts were made to force agreement with Ward's curves at $\phi = 0°$ and 180°. The result is shown in Fig. 3, which was used to adjust all measured $s_t$ to $s_{to}$. With all angles small, $\cos \phi \approx \alpha_p/\alpha$.

In order to eliminate data points for which very large adjustments would be necessary, the following procedures were applied: 1) Use data for $\alpha/\theta_c \leqq 0.6$ if $\phi \leqq 120°$; 2) Use data for $\alpha/\theta_c \leqq 0.25$ if $\phi > 120°$. The effects of small angles of attack on local Mach and Reynolds numbers were neglected.

### B. Influence of Surface Roughness

Because of the typically higher local unit Reynolds number, cold walls ($T_w < T_{aw}$) and consequently thin boundary layers, it has been suggested that transition data from aeroballistic ranges may be affected by surface roughness. Although a method exists for predicting the influence of standard, raised roughness on transition,[12] it was thought best to conduct some experiments under actual aeroballistic range conditions. This was done at Mach numbers of 5 and 2.

In the Mach 5 phase, which was done first, cones otherwise identical to the 1.75 in. models in Fig. 1 were deliberately roughened, beginning at a station 1.5 in. from the tip. Considering that circumferential machine tool marks, or grooves, seemed to be the most general form of roughness encountered on nominally smooth conical bodies, the desired roughness was created simply by changing cutting tool speed. That produced varying degrees of roughness which was measured in the usual manner with a profilometer. The shortcomings of such devices for surface roughness measurements are well known.[15] Mainly, the objections are the possible scratching of the surface by the stylus and the inordinately large radius of the profilometer stylus (500 $\mu$ in.) compared to the smaller dimensions of the roughness. A typical, well-finished cone surface registered less than 10 $\mu$ in. rms, but one must assume that the profilometer stylus could not penetrate to the bottom of surface defects with

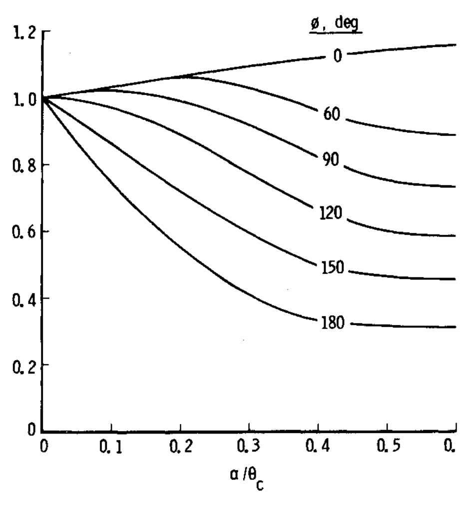


**Fig. 3 Combined data on angle of attack and meridian-angle influence.**

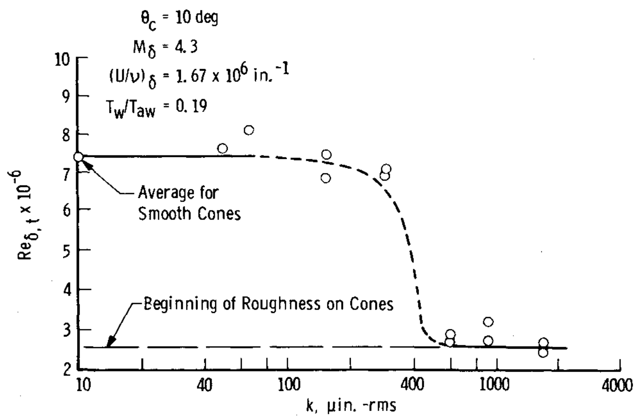


**Fig. 4 Effect of distributed surface roughness on cone at Mach 5.0.**

transverse widths less than stylus diameter. Whether such types of roughness are of any importance in a given case is another subject to consider. In view of the data to be shown later, it is probable that defects of this scale did not matter in the experiments discussed herein.

When the time arrived for the experiments on roughness effect threshold at Mach 2, it was decided to use a narrow band of grooved or screw-thread roughness rather than widely distributed roughness. In addition to yielding data on a different type of roughness, the changed procedure made "reading" the shadowgrams a little easier. Only 2.5 in. diam cones were used for the Mach 2 studies.

Six circumferential v-grooves (screw-threads) were machined in the aluminum 2.5-in.-diam cones beginning 1.5 in. from the apex. Thus, the band extended only about 1/10 in. along the cone. Design of the screw-thread was selected so that profilometer readings on the distributed roughness described previously and on the six rows of threads were consistent. In both cases, the actual dimension from the bottom of a groove to the top was approximately four times the quoted profilometer rms roughness reading. The tops of the grooves lay on the finished surface of the smooth cone.

Figure 4 gives the results for the Mach 5 distributed roughness, while Fig. 5 presents similar results for the Mach 2 localized roughness. The significant point to be made here is that no effect on transition was discerned until levels of roughness were extended well above those measured by the same technique on nominally smooth cones ($\approx 10$ $\mu$ in. rms) which were used for the main body of this research. Even though the critical roughness seems safely higher than any to be expected on the "smooth" cones used in this study, further experiments were conducted because of some markings seen on laser-front-lighted photographs of free-flight cones.

It must be remembered that it is surface condition in flight

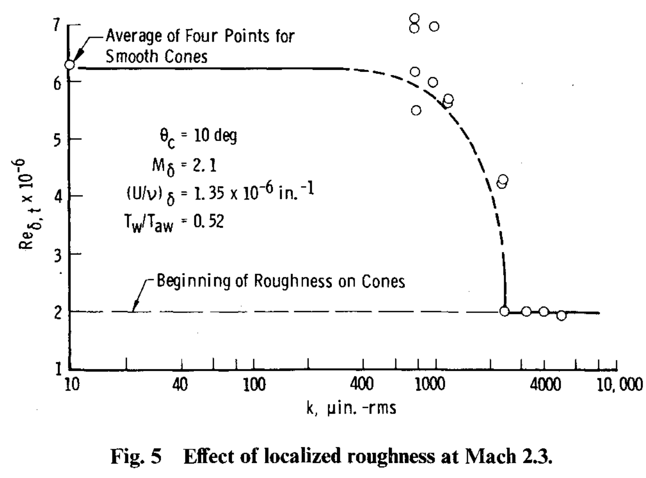


**Fig. 5 Effect of localized roughness at Mach 2.3.**

at the observation station which really matters. Thus, laser-front-lighted photography of the cones in flight was used as a means for identifying any cones with visible defects such as roughened, bent, or ablated surfaces. The technique, as applied to high-speed, free-flight ablation research, has been described by Dugger, Enis and Hill.[16]

Two basic types of sabots were used in this work. In the prior work[5] and part of the present investigation, closed-base sabots were used. For all of the 2.3- and 2.5-in. aluminum cones used later, open-based sabots were used. Cones in these experiments were made from either 7075-T6 aluminum alloy or Lexan. The latter was used for the cones in the study of vibration discussed in the following section. The open-based sabots for the aluminum cones and the closed-base sabots for the Lexan cones were of Lexan, while the major portion of the closed-base sabots interacting with the surfaces of the other aluminum cones was Dylite®. During the experiments of Ref. 5, closed-base sabots made of Lexan were used.

Aside from a few cases of cone damage in launch, e.g., bent noses on the 4° cones, the most important finding in the laser front-lighted photographs was discolored areas on the cone surfaces where cone and sabot were in contact. An example is shown in Fig. 6. Such markings were seen at various times for all types of sabots used.

Laser-front-lighted photography of static aluminum cones subjected to simulated launch loadings in open-based sabots revealed that the discolored area of Fig. 6 could be approximated with roughnesses measuring only 15 $\mu$ in., rms. Early in this investigation, it was also found that the inner surfaces of the Lexan sabots contacting the cones were characterized by profilometer readings of the order of 100 $\mu$ in., rms. Thereafter, it was decided to require inner Lexan sabot surfaces to be finished to the order of 30 $\mu$ in., rms, which is deemed about the best to be expected on a routine basis from ordinary matching on Lexan. Then, it seems a safe assumption that any scuffing or embossing due to the cone pressing against the sabot would be less than 30 $\mu$ in., rms, particularly since the aluminum is the stronger of the two materials. The Dylite foam is so soft, in comparison to aluminum, that any interaction under launch loading should not appreciably roughen the aluminum cones. In addition, the closed-base sabot causes the base of the cone to carry the load, rather than the cone wetted surface, and this lessens the likelihood of surface embossing. More importantly, no difference in transition data was detected in comparisons between "early" and "later" sabot usage. That is, of course, consistent with the results in Figs. 4 and 5 if the roughness due to the sabot is no more than 0(100 $\mu$ in.).

The zigzag or sawtooth pattern of the discoloration on the free-flight cone in Fig. 6 deserves explanation. The sabot quarters have serrated surfaces where they fit together. This is to form a seal against leakage of gun gases and to prevent slippage of the sabot quarters relative to one another. It was clear that the pattern seen in Fig. 6 fitted the sabot serrations, but it was not known if the roughness represented by this pattern was high enough to be important. Presumably the "printing" was caused by hot powder-chamber gas leaking through the seals from the base of the sabot, possibly enhanced by contaminants produced as the sabot was heated by the propelling gas and by friction along the gun barrel.

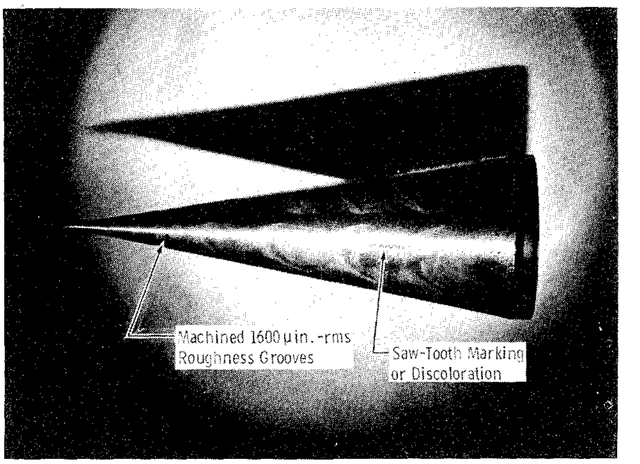


**Fig. 6 Laser-front-lighted photograph of cone in flight.**

To resolve the question raised by the leakage-related, printed pattern described above, a technique of wrapping the cones with a thin sheet of Mylar® was adopted. The Mylar was wrapped around the cones so that all four longitudinal sabot joints were fully covered by the protective sheet. During launch, after exit from the gun barrel, the Mylar sheet was quickly swept aside and fell to the floor. This left the cones unmarked by the sabot serrations. The technique was successful on every launch of 2.5-in.-diam, 10° cones in open-base sabots. The latter was the only type of sabot used during this phase of the investigation. It was not successful when 4° cones were used. Evidence picked up off the blast tank floor showed that the Mylar sheet was cut or burned through in a sawtooth pattern where it fitted against the serrated sabot joints.

Considering that no difference in transition data was found in comparing data for Mylar-wrapped and unprotected 10° cones, it seems safe to conclude that the printed, longitudinal, sawtooth patterns did not systematically influence the results. The Mylar-wrapped cones will be identified when transition data are discussed later in the section on results.

In view of Figs. 4 and 5, and the other results described, it is concluded that surface roughness has not significantly affected the data presented in Ref. 5 or any of the later data reported herein.

#### Influence of Model Vibration

The possible influence of model vibration was examined by comparing transition results obtained from launching cones of two materials under otherwise similar conditions. The materials were chosen on the basis of their being compatible with the rigors of range operations while having appreciably different vibrational characteristics. If a significant change in transition location were made by this variation in cone vibrational behavior, it at least would indicate the need for more careful study. Seeing no change does not prove that cone vibration is not a factor because of the limited frequency range, but the experiment seemed worthwhile.

All the previous work, as well as the current extension, involved aluminum cones. The only readily usable material offering significantly different vibrational characteristics appeared to be Lexan. Table 1 shows relevant data on the two materials. Laboratory experiments have essentially confirmed the computed frequency ratio in Table 1.

**Table 1 Frequency and amplitude data**

| Material | $W$ | $Y$ |
|---|---|---|
| Aluminum 7075-T6 | 1 | 1 |
| Lexan | 0.28 | 3.3 |

$W = (E/\rho)^{1/2}/(E/\rho)_{\text{aluminum}}{}^{1/2}$ = frequency ratio
$Y = (\sigma_y/E)/(\sigma_y/E)_{\text{aluminum}}$ = amplitude ratio
$E$ = Young's modulus of elasticity
$\rho$ = material density
$\sigma_y$ = yield strength in tension

The assumption was made that the cone tip in free flight would tend to vibrate as if the cone were fixed at its center of gravity. Strain gages were attached to the base of a model 1.75-in. cone and wired to measure bending stress. The collar was suspended from wires and struck with a hammer to induce vibration in the cone. Bending strain at the base of the cone simulating the range models was measured with two strain gages which formed adjacent arms of a four-equal-arm bridge circuit. Strain, represented by the bridge output, was amplified using a differential-type d.c. amplifier and was recorded using a preamp and an oscilloscope.

Oscilloscope traces as shown in Fig. 7 were obtained when the cones were struck with a large ball bearing while fixed at their centers of gravity. The lower frequency and greater amplitude of the Lexan cone is obvious. It should be noted that the so-called Lexan cone is not truly representative of a cone made only from Lexan; for aerodynamic stability it was necessary to insert an internal metal ballast in the forepart of the cone. This affected the vibrational characteristics, and it is the ballasted cone which is represented in Fig. 7. Apparently, the ballast acted as a damper because it reduced the frequency.

The oscilloscope traces were read on a film reader to determine frequency and the logarithmic decrement, $\Delta = (1/N)l_n(y_o/y_n)$. In this equation $N$ is the number of cycles, $y_o$ is the original amplitude, and $y_n$ is the amplitude after the $N$ cycles. We obtained $\Delta = 0.047$ for the aluminum cone. The ballasted Lexan cone did not give a constant $\Delta$. The value was lower at later times but was approximately constant at 0.3 for about 8 msec after being struck.

The measured natural frequency of the aluminum cone was 6880 Hz. The ballasted Lexan cone experimentally yielded 1250 Hz and a wholly Lexan cone gave 2060 Hz. Therefore, rather than the computed ratio of Lexan-to-aluminum frequencies, $W = 0.28$, which appears in Table 1, we have $W = 0.18$ for the ballasted Lexan cones actually used. While one cannot say if this 5.6:1 variation in cone vibrational frequency is significant in regard to boundary-layer transition under the circumstances studied, it is at least large enough to be interesting.

The time required for a given amplitude change is given by $t = [1/(\Delta f)]l_n(y_o/y_n)$ where $f$ is the frequency in Hz. Because it would seem very likely that any cone vibration is induced early within the launcher, the time elapsing between, say, the start of motion within the launcher and arrival at the viewing station is relevant insofar as the cone vibration amplitude is concerned. In the case of the Mach 5 experiments, this (average) time was 14.4 msec and for the Mach 2 launches it was 27.9 msec. Therefore, if the initial maximum amplitude, $y_o$, of the cone tip were known, these input data would permit a calculation of the amplitude, $y_n$, at the focused shadowgraph station. This calculation cannot be made because $y_o$ is not known, but an upper limit may be placed on it. For the 10° aluminum cone, it is calculated that a deflection of the tip of 0.065 in. would have caused the metal to yield. No such bending was ever observed in flight on 10° cones, so it is safe to assume

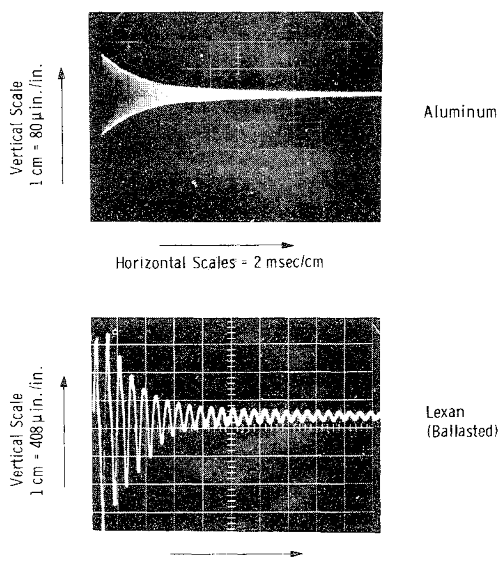


Fig. 7 Vibrational response of aluminum and Lexan cones.

that $y_o$ did not attain that magnitude. Assuming, as an example, $y_o \lesseqgtr 0.06$ in. and substituting into the last equation the quantities $t = 14.4$msec, $\Delta = 0.3$, $f = 1250$ Hz, and $y_o \lesseqgtr 3.3(0.06) = 0.20$ in. leads to $y_n \lesseqgtr 0.00091$ in. Although the Lexan cone may start with more than three times the tip deflection of the aluminum cone, it would be expected to have roughly 1.6 times as much tip deflection at the viewing station.

Because of interest in possible higher vibrational modes, a second type of experiment was conducted with one of the 1.75-in. aluminum cones of the type actually launched. It was suspended by a string at its center of gravity and struck with a hammer. A microphone and recording system of the type used to monitor noise in the range recorded the result. Frequencies of 7680 and 7180 Hz could be identified. Higher modes could not be found by this means, and it was concluded that any higher modes were associated with very much lower amplitudes.

By this comparison of transition on aluminum and ballasted Lexan cones, one is seeing the effect of a reduction of vibrational frequency from 6880 to 1250 Hz, coupled with a corresponding increase in possible tip vibrational amplitude by a factor of 1.6. Sufficient Mach 5 and Mach 2 launches have been carried out so that it may be reported that no significant difference in transition Reynolds numbers has been found. The Lexan cones will be identified when the transition data are discussed in the section on results.

It may be noted that a recent wind tunnel experiment by Oslon et al.[17] revealed no influence of model vibration on transition Reynolds number. In that situation, frequencies of 2900 to 82,000 Hz and peak-to-peak amplitudes of 40 to 1500 $\mu$ in. were explored. The authors believed that artificial roughness heights equal to the vibration amplitudes would have tripped their boundary layer, and they concluded that vibration in their case was less effective as a trip than fixed surface roughness. Perhaps this should not be surprising. The surface deflections owing to vibration would take the form of gentle waviness rather than abrupt discontinuities in the nature of boundary-layer trips.

Again, Kendall has provided the writer with data from the JPL 20-in. supersonic wind tunnel which are of interest. His vibrating a 10° cone model, producing a 3.1 kHz dominant frequency and tip vibrations that could be sensed by touch, failed to move transition at $M_\infty = 2.2$ and 4.76. He has pointed out that according to other experiments he has conducted, such a low frequency would not be amplified, at least at the higher Mach number; thus, the lack of effect is consistent with his earlier findings.

**Non-Uniform Wall Temperature**

A source of potential influence on transition under range conditions is the nonuniform surface temperature arising from aerodynamic heating. A hot nose will be combined with a relatively unheated afterbody, and the boundary-layer profile near the nose will reflect this. Rhudy[18] has made an illustrative calculation of the influence of a hot leading edge section with $T_w/T_o = 0.8$ followed by a cooled plate with $T_w/T_o = 0.2$. He calculated that, for $M_\delta = 6$ and $(U/\nu)_\delta = 1.1 \times 10^6$ in.$^{-1}$, it takes a distance of approximately 600 $\delta_j$ for the product $\rho u$ in the boundary layer at the critical height $y/\delta = 0.9$ to attain the profile that is calculated for a plate with $W_w/T_o = 0.2$ over its entire length. The symbol $\delta_j$ represents boundary-layer thickness at the discontinuous change of wall temperature.

In the range investigations conducted by the author, calculations of stagnation point and afterbody temperature have been made. By an approximate method it was calculated that Lexan cone tips would melt under the Mach 5 conditions and yet no evidence of tip blunting was seen. Thus, it is inferred that the calculated temperature increases either were conservative or that the amount of localized ablation was too small to be seen. Inspection of the laser-lighted photographs of Lexan cones in flight revealed small decreases in length and faintly cusped tip shapes. However, the laser station was 32% further downrange than the principal focused shadowgraph, and ablation would have been negligible at the shadowgraph where transition was determined. If we take a published melting temperature of roughly 990°R for Lexan, it follows that the nose tips of the Mach 5 Lexan cone noses were no more than 1.8 times the temperature of the skirts which heated negligibly in the brief flight. The calculation method yields a maximum tip-to-skirt temperature ratio of roughly 2:1 for the Mach 5 aluminum cones as well. At Mach 2, these ratios are much nearer unity. In keeping with Rhudy's results, it may be inferred that the cone boundary layers at the transition location should be essentially free of "hot-tip" influence when distance $(s_t - s_j)$ is greater than $600\delta_j$. To apply Rhudy's criterion for distance downstream of the temperature discontinuity, it is necessary to designate a station on the cones where the wall surface temperature may be assumed to have become constant ($T_w = T_\infty$). For this purpose it seems conservative to take the order of 10 nose radii or $s_j \approx 0.05$ in. For a sharp cone in the Mach 5 case of higher aerodynamic heating, calculations for a laminar boundary layer give $s/\delta \geqq 400$. Thus, if $s_t \geqq 30 s_j$, as it always was, it follows that

$$(s_t - s_j)/\delta_j \approx s_t/\delta_j = (s_t/\delta_t)(\delta_t/\delta_j) \geqq 400\delta_t/\delta_j$$

But

$$\delta_t/\delta_j = (s_t/s_j)^{1/2} \geqq 30^{1/2}$$

so

$$(s_t - s_j)/\delta_j \geqq 2000$$

Coupled with the lesser tip-to-afterbody temperature ratios and the conservative nature of this comparison with the results in Ref. 18, it is concluded that the hot-tip effect is not an obvious factor in the present work.

## VI. Discussion of Results

The prior section has been used to present at least partial answers to some peripheral questions that concern aeroballistic data on boundary-layer transition. The fact that the models did oscillate, were not perfectly smooth, and did vibrate in small amplitudes may be significant in regard to sources of disturbances imposed upon and possibly amplified within the boundary layer. If so, the absolute levels of $Re_{\delta,t}$ may have been affected. Even this is not certain. All real boundary layers are disturbed in different degrees by different sources. At this time, however, evidence of any explanation of the unit Reynolds number effect arising from these factors is lacking. The main body of data from this investigation is next presented.

### Transition on Smooth Cones in Quiet Range

Figure 8 presents the data on nominally smooth, 4° and 10° semi-angle cones at both Mach numbers. Within the scatter of the data no clear difference appears between the data for 4° or 10° cones or between the data for Mach numbers of 2.3 or 5.0. Lest one be tempted to say that no Mach number effect exists, it must be pointed out that $T_w \approx T_\infty \approx 540$°R and that

$$T_w/T_\delta = \left(1 + \frac{\gamma - 1}{2} M_\delta^2\right) \Big/ \left(1 + \frac{\gamma - 1}{2} M_\infty^2\right)$$

Thus, on 10°, cones

$$T_w/T_\delta = 0.79 \quad \text{at} \quad M_\infty = 5.0$$

and

$$T_w/T_\delta = 0.91 \quad \text{at} \quad M_\infty = 2.3$$

On 4° cones at $M_\infty = 2.2$, $T_w/T_\delta \approx 1$. As mentioned earlier, for 10° cones,

$$T_w/T_{aw} = 0.19 \quad \text{at} \quad M_\infty = 5.0$$

and

$$T_w/T_{aw} = 0.52 \quad \text{at} \quad M_\infty = 2.3$$

For the 4-deg cones at $M_\infty = 2.2$, $T_w/T_{aw} = 0.54$.

The influences of temperature levels and wall heat transfer rates on boundary-layer transition are uncertain because of either or both conflicting data and lack of data for extreme ranges of these parameters. The general impression is that cooling the wall of a model delays transition at $T_w/T_{aw} \gtrsim 0.5$, but there are data[13] showing that cooling can cause earlier transition at $T_w/T_{aw} \lesssim 0.5$. Actually, of course, the question is much too complicated to allow an easy answer. The stability theory of Ref. 3 suggests that there is a complex interplay between different modes of instability and the heat transfer situation existing for a particular type of disturbance at a particular Mach number.

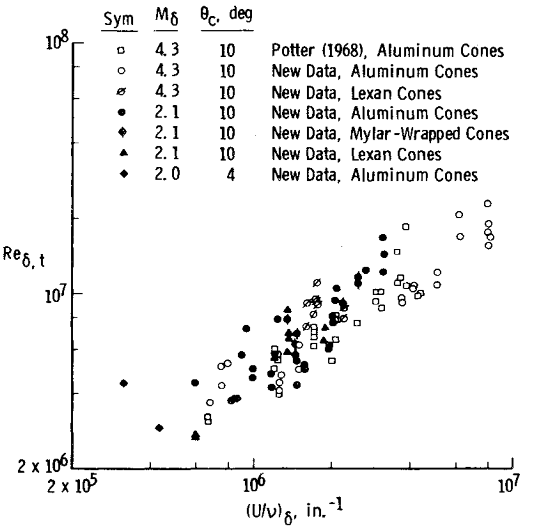


**Fig. 8 All smooth-cone, quiet-range transition results.**

It is relevant to call attention to the cold-wall status ($T_w \ll T_{aw}$) of the aeroballistic range data and the possibly important fact that $T_\delta$ in the range is generally much greater than $T_\delta$ in supersonic wind tunnel cases that might be thought comparable. The point to this latter remark is that, to a good approximation, $\mu_1/\mu_2 = T_1^{\omega_1}/T_2^{\omega_2}$, where $\omega$ is a function of $T$. Under typical aeroballistic range conditions where $T \geqq 530$°R, one finds $\omega \leqq 3/4$. In supersonic wind tunnels, the upper limit on $\omega$ will approach unity. Therefore, equal local values of viscosity-dependent parameters within the boundary layers are not necessarily to be expected even when $(U/v)_\delta$ and $M_\delta$ are equal in the cases considered.

Returning to Fig. 8, it can be seen that a large number of data points and broad range of the independent variable are desirable when working with transition data obtained from a single-spark photograph of each of a series of models at random attitudes. Repetitious spark photographs of a single model at fixed, zero angle of attack have shown a bell-shaped or Gaussian distribution of transition locations with time.[12] The near-instantaneous, single-shot photography represented in Fig. 8 certainly reflects a temporal fluctuation in transition location on a given meridian of a cone and the variation spatially around the circumference. Other factors, including the repeatability of the author's visual process in determining transition location in the photographs, undoubtedly added to the scatter.

To conclude earlier discussion about Lexan cones and the experiment to evaluate possible cone vibration effects, the comparison of Lexan and aluminum cones is included in Fig. 8. No significant difference is apparent for either Mach number. Similarly, data from the Mylar-wrapped cones are indicated in Fig. 8, and no significant difference in results can be attributed to that refinement.

A major message contained in Fig. 8 is that the data of Ref. 5 are supported. After extension, both in regard to $(U/v)_\delta$ and $M_\delta$, the so-called unit Reynolds number effect remains strongly evident in these aeroballistic data where none of the normal wind tunnel disturbances were present. Several analytic curves have been tested by the least-squares method for fitting the data in Fig. 8. The two expressions giving the best results were found to be

$$Re_{\delta,t} = 880(U/v)_\delta^{0.63}$$

and

$$Re_{\delta,t} = 3.3\times10^6 + 2.1(U/\nu)_\delta$$

where $(U/\nu)_\delta$ is given in units of in.$^{-1}$. There is no justification for extending either of these curves beyond the limits of the data, and little space is devoted to discussion of the curve-fitting results. It is only noted that the second of the above expressions implies a secondary Reynolds number equal to 2.1 $(U/\nu)_\delta$ or a virtual origin of transition an average of 2.1 in. upstream of the $s_t$ station. This provides grounds for some plausible speculations about the unit Reynolds number effect and suggests an area for future research, but it does not merit further discussion here.

The mildly surprising equality of Reynolds numbers of transition at $M_\delta \approx 2.1$ and 4.3, with $T_w = T_\infty = 540°R$ in both cases, deserves further investigation. Whether or not the different wall temperature ratios, $T_w/T_{aw}$, compensated for Mach number effect remains a question.

#### Experiment with Noise Introduced into Range

In an effort to perturb the quiet range condition, an available Federal Sign and Signal Corporation Model A siren was installed as near as practical to the position of free-flight cones as they passed the photographic station. Experiments with various orientations of siren and microphones showed that a 130-db field was introduced in the area of the model by this means. The peak amplitude of the fluctuating sound pressure level, normalized by the range static pressure most used for this phase of the study was $\tilde{p}/p_\infty \approx 4.7\times10^{-4}$, rms, for 40 Hz to 40 kHz frequencies, and the dominant frequency of the near-sinusoidal waveform was 800 Hz. A Bruel and Kjaer $\frac{1}{2}$-in. Model 4133 microphone was used for these measurements.

For comparison, it may be noted that sound pressure ratios mentioned in Ref. 9 for a wind tunnel at Mach 3 conditions are $0.014 < \tilde{p}/p_\infty < 0.035$ rms. Thus, while the siren produced an rms level of $\tilde{p}/p_\infty$ more than 200 times greater than the measured maximum in the "quiet" range, the ratio remained an order of ten less than would be expected in a typical supersonic wind tunnel. (Note that these ratios are different if $\tilde{p}$ rather than $\tilde{p}/p_\infty$ is the basis of comparison.)

Figure 9 presents results for several cones launched with the siren operating. Static pressure, $p_\infty$, was 1040 mm Hg for the 10° cones at Mach 2 and 239 mm Hg for the 4° cone at Mach 2. The 10° cones at Mach 5 were launched with 450 mm Hg. It is clear that the particular noise spectrum imposed by the siren did not produce a significant effect on boundary-layer transition. The low frequency of the siren output could be the cause of this, but it is noteworthy that sound pressure ratio, even with the siren, was still very low in comparison to supersonic tunnel test sections.

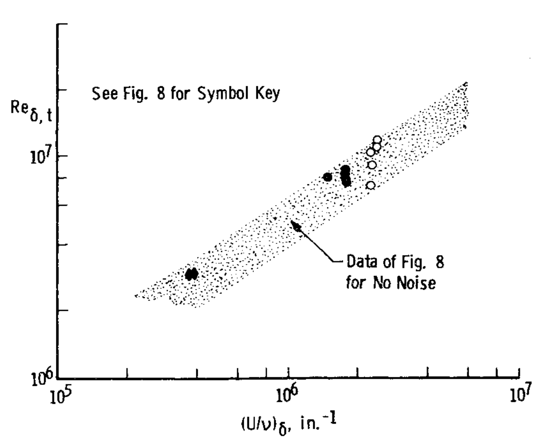


**Fig. 9 Influence of elevated noise level on transition Reynolds number.**

## VII. Conclusion

The principal conclusion is that a unit Reynolds number effect was found in the free-flight range environment. None of the range-peculiar conditions investigated thus far appears to offer an explanation for this result. It must be remembered that the unit Reynolds number effect can only be one of several competing and possibly interacting factors capable of influencing transition. Thus, it can be hidden or modified under many circumstances, but its potential influence in full-scale atmospheric flight should not be neglected.

## References

[1] Morkovin, M. V., "Critical Evaluation of Transition from Laminar to Turbulent Shear Layers with Emphasis on Hypersonically Traveling Bodies," AFFDL-TR-68-149, 1969, Air Force Flight Dynamics Lab., Wright-Patterson Air Force Base, Ohio.

[2] Morkovin, M. V., "On the Many Faces of Transition," in *Viscous Drag Reduction,* Plenum Press, New York, 1969, pp. 1–31.

[3] Mack, L. M., "Boundary Layer Stability Theory," JPL 900-277 Rev. A, Nov. 1969, Jet Propulsion Lab., Pasadena, Calif.

[4] Mack, L. M. and Morkovin, M. V., AIAA short course notes on boundary-layer stability and transition, available from AIAA, New York, 1969.

[5] Potter, J. L., "Observations on the Influence of Ambient Pressure on Boundary-Layer Transition," *AIAA Journal,* Vol. 6, No. 10, pp. 1907–1911, Oct. 1968.

[6] Reshotko, E., "Stability Theory as a Guide to the Evaluation of Transition Data," *AIAA Journal,* Vol. 7, No. 6, pp. 1086–1091, June 1969.

[7] Ross, R., "Influence of Total Temperature on Transition in Supersonic Flow," *AIAA Journal,* Vol. 11, No. 4, pp. 563–565, April 1973.

[8] Laufer, J., "Aerodynamic Noise in Supersonic Wind Tunnels," *Journal of the Aerospace Sciences,* Vol. 28, No. 9, pp. 685–692, Sept. 1961.

[9] Pate, S. R. and Scheuler, C. J., "Radiated Aerodynamic Noise Effects on Boundary-Layer Transition in Supersonic and Hypersonic Wind Tunnels," *AIAA Journal,* Vol. 7, No. 3, pp. 450–457, March 1969.

[10] Ward, L. K., "Influence of Boundary-Layer Transition on Dynamic Stability at Hypersonic Speeds," *Transactions of the Second Technical Workshop on Dynamic Stability Testing,* Vol. II, April 1965, Arnold Engineering Development Center, Tullahoma, Tenn.

[11] Kendall, J. M., private communication, NASA/CIT Jet Propulsion Lab., Pasadena, Calif., 1972–1973; also *AIAA Journal,* Vol. 13, March 1975, pp. 290–299.

[12] Potter, J. L. and Whitfield, J. D., "Effects of Slight Nose Bluntness and Roughness on Boundary-Layer Transition in Supersonic Flows," *Journal of Fluid Mechanics,* Vol. 12, Pt. 4, pp. 501–535, 1962.

[13] Mateer, G. C., "Effects of Wall Cooling and Angle of Attack on Boundary-Layer Transition on Sharp Cones at $M_\infty = 7.4$," TN D-6908, Aug. 1972, NASA.

[14] Di Cristina, V., "Three-Dimensional Laminar Boundary-Layer Transition on a Sharp 8-deg Cone at Mach 10," *AIAA Journal,* Vol. 8, No. 5, pp. 852–856, May 1970.

[15] Wilkins, M. E. and Darsow, J. F., "Finishing and Inspection of Model Surfaces and Boundary-Layer-Transition Tests," NASA Memo 1-19-59A, Feb. 1959.

[16] Dugger, P. H., Enis, C. P., and Hill, J. W., "Laser High-Speed Photography for Accurate Measurements of the Contours of Models in Hypervelocity Flight within an Aeroballistic Range," *Proceedings of the Technical Program,* Electro-Optical Systems Design Conference, New York, Sept. 1970.

[17] Olson, L. E., Gregorek, G. M., and Lee, J. D., "The Influence of Artificially Induced Turbulence upon Boundary-Layer Transition in Supersonic Flows," ARL 71-0022, Jan. 1971, Aerospace Research Labs., Wright-Patterson Air Force Base, Ohio.

[18] Rhudy, J. P., "Effect of Uncooled Leading Edge on Cooled-Wall Hypersonic Flat-Plate Boundary-Layer Transition," *AIAA Journal,* Vol. 8, No. 3, pp. 576–577, March 1970.