are needed to check out the derived inviscid scaling laws. Examination of the deviations between viscous experimental data presently available and the theoretical predictions gives one reason to believe that the derived scaling laws will prove to be correct. The scaling laws, when combined with existing pointed body or Newtonian theories, provide the capability to predict the hypersonic unsteady aerodynamic characteristics of most re-entry bodies of current interest.

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Acoustic Technique for Detection of Flow Transition on Hypersonic Re-Entry Vehicles

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This paper describes the development of an acoustic flow-transition detector for use on hypersonic ablating re-entry vehicles. The acoustic sensor of this system is recessed from the heat-shield surface and thus is protected from the externally developed heat; the sensor communicates with the external pressure field via a small-diameter vent. Results of supersonic wind-tunnel tests are presented which compare the characteristics of signals of an experimental transition detector with those of reference sensors (hot-wire probe, flush-mounted microphone) in transitional and fully developed turbulent flow.

I. Introduction and Summary

A. Statement of the Problem

ACCURATE determination of the time and location of flow transition on hypersonic, ablating, re-entry vehicles is very important for the successful planning and operation of a re-entry mission. Because of the excessive heat development on the surface of the re-entering vehicle, a direct measurement of flow transition by means of surface-mounted sensors is not feasible. Any such sensor would be quickly destroyed. The occurrence of flow transition is therefore usually inferred from data information not directly related to the flow phenomenon, e.g., from the increase of interior vibration levels.

This paper describes the development of an acoustic flow-transition detector system that senses fluctuating pressures directly on the surface of the ablating heat shield. This system makes use of a $\frac{1}{4}$-in.-diam piezo-electric microphone recessed from the surface and communicating with the fluctuating pressure field at the surface via a "probe tube" hole (Fig. 1).

It had been expected originally that a recessed microphone in contact with the surface pressure field would require elaborate thermal protection to assure no damage during the re-entry phase. Accordingly, an earlier developed acoustic system[1] for measuring surface pressure fluctuations involved a rather complicated arrangement of tubes and cavities preceding the microphone. Later arc tests,[2] however,

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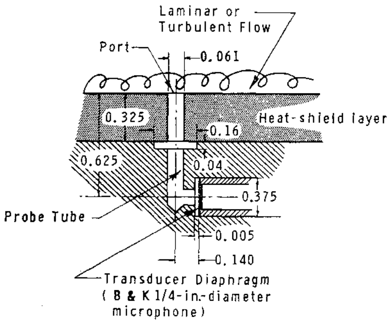

Fig. 1 Transition detector configuration (dimensions in in.).

revealed that a simple bend in the probe tube, arranged so as to avoid a line-of-sight communication between the hot surface gas flow and the microphone provides adequate thermal protection for the microphone. The temperature, measured on the microphone diaphragm, did not unduly rise during the time span typical for a re-entry phase.

The microphone port arrangement of Fig. 1 was chosen as the result of a variety of considerations. It avoids use of a large port which could cause severe heat-shield aggravation around the port region during re-entry, with unpredictable effects on the system acoustics. The system also avoids use of a small-diameter port with a conical transition cavity from the probe tube diameter to the microphone diameter. Such a configuration would be the acoustic analog of a mass-spring system (Helmholtz resonator) that could distort the acoustic response of the system in the frequency range of interest.

In developing the transition detector system, the following questions had to be answered:

1) What characteristics of a fluctuating pressure signal indicate the occurrence of flow transition in supersonic or hypersonic flow?

2) Does the presence of a pressure port in the heat-shield layer prematurely trigger flow transition?

3) How do the internal acoustic resonances in the system affect its frequency response under hypersonic flow?

The study described in this paper was aimed at answering these questions.

B. Experimental Study

A series of measurements were made in the M.I.T. Naval Supersonic Wind Tunnel at a Mach number of 2.9 utilizing a flat plate with a sharply beveled leading edge. This test plate incorporated 1) an experimental transition detector configuration, 2) reference microphones, and 3) a hot-wire probe. By changing the stagnation pressure in the tunnel test section, the Reynolds number of the flow on the test plate was adjusted so as to obtain laminar and transitional flow at the location of the sensing elements. Thus, the signal characteristics for the various flow regimes as seen by the transition detector configuration, by the hot-wire probe and by the flush-mounted reference microphones could be investigated.

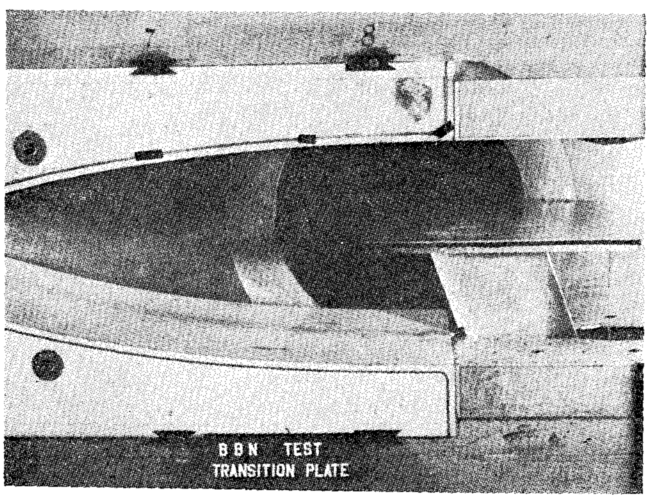

Fig. 2 Test plate in tunnel test section.

C. Summary of Test Results

The results of this study, as described in the body of this paper indicate the following: 1) flow transition in supersonic flow is characterized by a broadband, rapid increase of the fluctuating pressure levels, with a subsequent decrease in levels as more fully developed turbulent flow is approached; 2) transition can thus be detected by acoustic transducers sensing directly the fluctuating pressure field in the surface flow; 3) presence of a probing port does not trigger flow transition prematurely as long as the port diameter is smaller than the boundary-layer displacement thickness; 4) the acoustic response of the transition detector configuration experiences a strong damping in the presence of external supersonic or hypersonic flow. Any resonant peaks as observed under acoustic excitation are suppressed; the response drops off at frequencies beyond a particular break frequency, which is a function of detector internal geometry and external flow parameters.

II. Experimental

A. Test Setup

Figure 2 shows the experimental test plate (20 in. $\times$ 18 in. $\times$ 1 in.) in the test section of the M.I.T. Supersonic Wind Tunnel. This wind tunnel is a continuous flow unit with an 18 in. $\times$ 18 in. test section in which the stagnation pressure can be varied between 3 and 25 psi, at a stagnation temperature of approximately 100°F. The flat test plate was lapped and ground. The leading edge had a radius of about 0.002 in. One center pylon with a sharp leading edge supported the plate. A flush-mounted $\frac{1}{2}$-in.-diam microphone (B&K type 4133), a miniature hot-wire probe (DISA type 55A52), and the experimental transition detector configura-

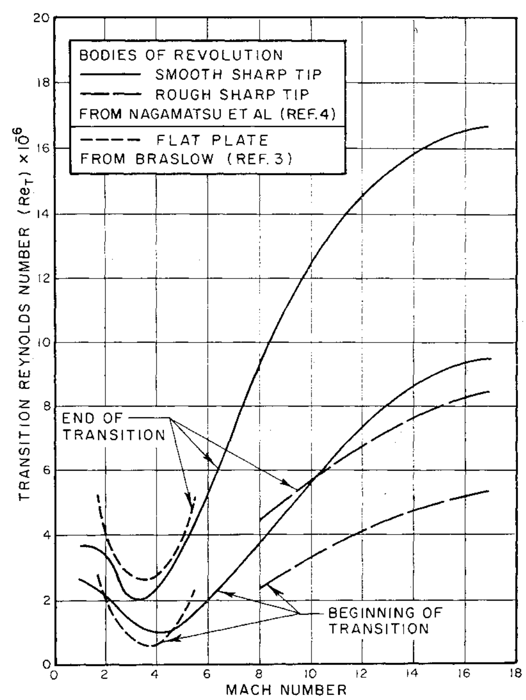

Fig. 3 Effect of Mach number on transition.

tion were mounted centrally in close lateral proximity to each other at 8 in. downstream of the leading edge.

Pertinent details and dimensions of the experimental transition detector are shown in Fig. 1. For the experimental study a $\frac{1}{4}$-in.-diam (B&K type 4135) condensor microphone was used.†

B. Flow Transition Characterization

Boundary-layer transition from laminar to fully turbulent flow for constant flow conditions (Mach number, freestream Reynolds number) on a flat plate occurs over a distance which is many boundary-layer thicknesses long. Flow transition is characterized, for example, by an increase in 1) boundary-layer thickness, 2) surface temperature, 3) hot-wire signal output, and 4) total pressure, measured near the test plate surface. Since this increase is rather gradual and, thus, makes the exact location of transition-onset somewhat ambiguous, many investigators take the easier definable peak value of the surface temperature or pressure as a clear indication of flow transition.

Within our study, however, we were concerned about the first occurrence of transitional flow. We used the very sensitive method of hot-wire probing for the definition of the onset of flow transition. Having established the Reynolds number range within which, for our particular test conditions, transition occurs, we could then associate the microphone signal with the various types of flow (laminar and transitional). Thus knowing the acoustic signal characteristics typical for flow transition, we could compare the signal output of a microphone flush-mounted to the surface with that of a microphone recessed from the surface (i.e., incorporated in the transition detector configuration). This procedure should readily exhibit the triggering effect, if any, that the presence of the detector probing port in the test surface would have on flow transition.

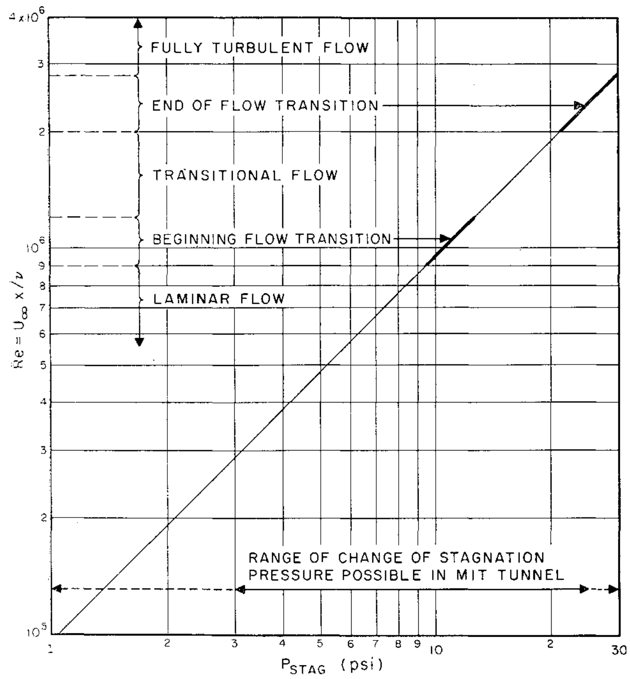

Fig. 4 Reynolds number 8 in. downstream of plate leading edge as function of tunnel stagnation pressure for test conditions: $M = 2.9$, $T_{stag} = 100°F$.

† The transition detector system, now commercially available, uses a piezo-electric microphone (BBN model 370), which is more rugged and almost insensitive to ambient static pressure changes.

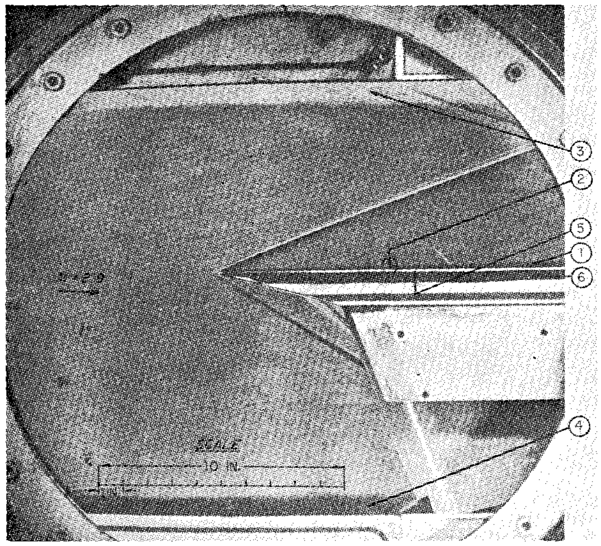

Fig. 5 Schlieren photograph of flow on test plate for $M = 2.9$ and $P_{stag} = 12$ psi: 1) test plate boundary layer; 2) predicted transition onset; 3) tunnel-ceiling boundary layer; 4) tunnel-floor boundary layer; 5) location of microphones, hot-wire probes, and transistion detector; 6) rubber seal.

III. Flow Transition in Supersonic Flow

Various investigators have established transition Reynolds numbers $Re_x = U_\infty \cdot x/\nu$ (where U_∞ is the freestream velocity, x is the distance from the leading edge or the body tip, and ν the kinematic viscosity) in supersonic flow on both flat plates and conical bodies. Braslow,[3] for one, gives the transition range in terms of Reynolds number as function of Mach number (Fig. 3). Accordingly, one should expect flow transition to begin at a Mach number of 2.9 at $Re_x = 0.9 \times 10^6$, and fully developed turbulent flow at $Re_x = 2.8 \times 10^6$. Results reported by Nagamatsu et al.[4] (shown also in Fig. 3) that were obtained from slender cone tests, indicate the flow transition range to correspond to $1.2 \times 10^6 < Re_x < 2.8 \times 10^6$. Experimental results reported by Potter and Whitfield[5] at a Mach number of 3.5 suggest the transition range to be given by $10^6 < Re_x < 1.97\ 10^6$, where the smaller Reynolds number gives the onset of transition, the larger one the range of fully developed turbulent flow.

From these data one should expect onset of transitional flow at Reynolds numbers from 0.9 to 1.2×10^6 and fully developed turbulent flow to occur at Reynolds numbers from about 2.0 to 2.8×10^6.

As a re-entry vehicle re-enters the earth's atmosphere, it encounters increasingly dense air; the freestream Reynolds number increases, causing the flow transition range to move upstream on the cone surface. A fixed point on the forward cone of the vehicle (say the probing port of a transition detector) would thus initially be exposed to laminar flow, then to transitional and finally to fully developed turbulent flow. This time sequence can be simulated in the wind-tunnel experiment by increasing the tunnel stagnation pressure, thus increasing the Reynolds number on the experimental test plate. We used this approach to adjust the flow regimes at the various sensors on the test plate.

The per foot Reynolds number in the M.I.T. wind tunnel obeys

$$Re/\text{ft} = 1.42 \times 10^5 \cdot P_{stag}$$

where P_{stag} is the stagnation pressure in psi. Figure 4 shows the Reynolds number at the location of the sensors (8 in. downstream of the plate leading edge) as a function of the tunnel stagnation pressure, for the test conditions of $M = 2.9$

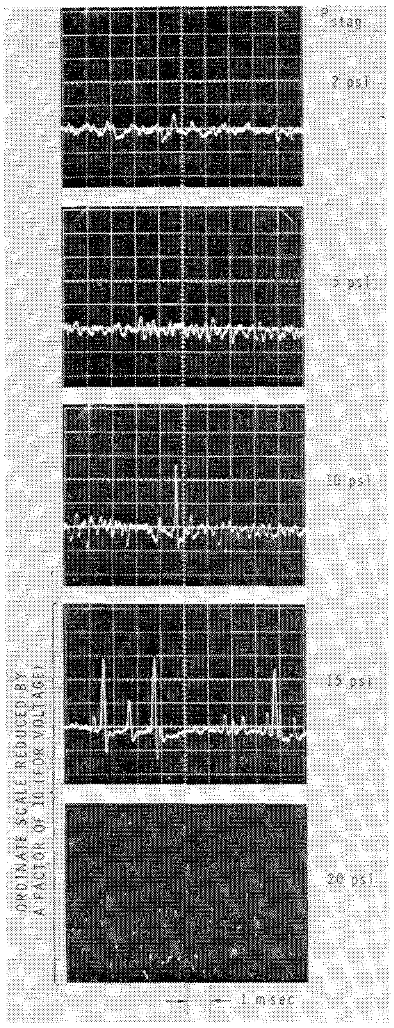

Fig. 6a Single sweep scope pictures of hot-wire signals.

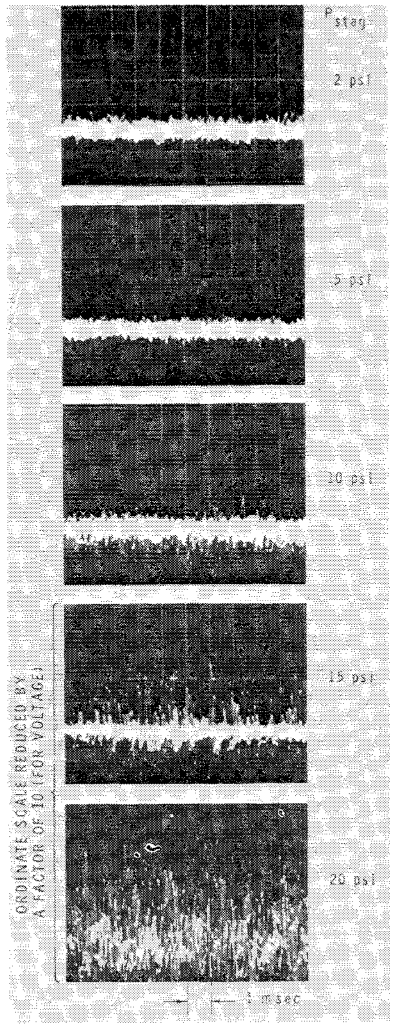

Fig. 6b Continuous sweep scope pictures of hot-wire signals.

and a stagnation temperature of $T_0 = 100°F$. Accordingly, one expects natural onset of flow transition to occur at stagnation pressures between 9.5 and 12.5 psi.

IV. Results and Discussion

The first objective of the study was to obtain a clear indication of the occurrence of transitional flow, and to compare the characteristics of the signals of the various sensors to identical flow patterns.

A. Schlieren Visualization

Figure 5 shows a Schlieren picture of the flow in the test section, corresponding to a tunnel stagnation pressure of 12 psi (which is in the range where flow transition is expected to begin at the sensor location). The tunnel-floor boundary layer (dark area) and the tunnel-ceiling boundary layer (light area) both having a thickness of about 1 in., are clearly discernible. On the test plate one sees the boundary layer (dark area above the test plate) as it develops and increases in thickness in the downstream direction. At $P_{stag} = 12$ psi, the transition range should extend from approximately 7 to 24 in. downstream of the leading edge. The expected sudden increase in boundary-layer thickness somewhere near the location of the sensors is very difficult to discern; however, the dark area on the test plate does seem to thicken past a distance of 8 to 10 in. downstream of the leading edge. It is, of course, well known (see for example, Ref. 5) that Schlieren flow visualization is not very appropriate to define flow regimes.

B. Hot-Wire Measurements

A more precise definition of flow regimes is possible by hot-wire probing. Figure 6 presents the time histories of the hot-wire signal for various stagnation pressures, as obtained with hot-wire sensors about 0.04 in. above the test plate surface. (The hot-wire signal was filtered with the "B-weighting curve," commonly in use on sound-level meters. This filter curve is flat between 400 and 3200 Hz, and then rolls off, with 3 db-down points at 150 Hz and 8000 Hz.) Figure 6a shows single-sweep signals, Fig. 6b shows a time exposure of approximately 1 sec.

At 10 psi (single sweep) we find only one turbulence "spike" during the period of display, whereas at 15 psi and, more so, at 20 psi there are many more spikes—most certainly due to the approach of the fully turbulent flow regime. Corresponding results may be observed in the time exposures of Fig. 6b. It is interesting to note that the spikes are not of a discrete frequency nature, as evident from Fig. 7, which shows $\frac{1}{3}$-octave band spectra of the hot wire signal for various P_{stag}. The most striking feature of these data is the clustering of the spectra for stagnation pressures from 2 to 10 psi, and the dramatic increase in level for higher pressures. This level increase, as may be observed, is of broadband nature. This observation agrees with the well-known[7] fact that transition is caused directly by random disturbances, and is not preceded by selective amplification of sinusoidal oscillations unless the degree of turbulence in the oncoming flow is extremely low. Quite obviously, flow corresponding to stagnation pressures below 10 psi is laminar, or, more precisely, has an extremely low degree of turbulence, since otherwise no signal at all should be observable. The fluctuations that are found in the freestream, and, also on the plate

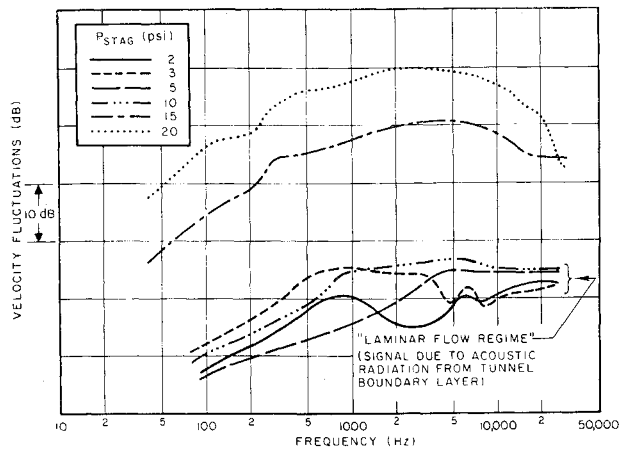

Fig. 7 Spectra of velocity fluctuations for various tunnel stagnation pressures.

surface are due to sound radiated from the thick turbulent boundary layers on the tunnel walls. Transitional flow sets in above $P_0 = 10$ psi, in agreement with the predictions from Fig. 4.

C. Fluctuating Pressure Measurements

The signals obtained from the flush-mounted microphone for the various flow regimes are presented in Fig. 8. This figure shows fluctuating pressure p (in third-octave bands) normalized with respect to the freestream dynamic head q_∞ as a function of a dimensionless Strouhal frequency $S = f\delta^*/U_\infty$ (where f is the center frequency of the third octave band, δ^* the turbulent boundary-layer displacement thickness, and U_∞ the freestream velocity of the tunnel flow). The displacement thickness δ^* of the turbulent boundary layer was calculated[6] for the flow conditions at the location of the sensors and was found to be 0.09, 0.07, and 0.06 in. for stagnation pressures of 5, 15, and 25 psi, respectively. Again, the fluctuating-pressure spectra for stagnation pressures from 3 to 10 psi coincide; they undoubtedly correspond to the "laminar flow regime." The spectra for higher stagnation pressures "break away," exhibiting increasingly higher levels, and reaching 15 to 18 db above the laminar regime for $P_{\text{stag}} = 20$ psi before dropping in level at $P_{\text{stag}} = 25$ psi. Evidently, at 20 psi the transitional flow regime has been passed and at higher stagnation pressures, fully turbulent flow is approached.

Since it was not possible to increase the tunnel stagnation pressure to the extent, where fully turbulent flow could be

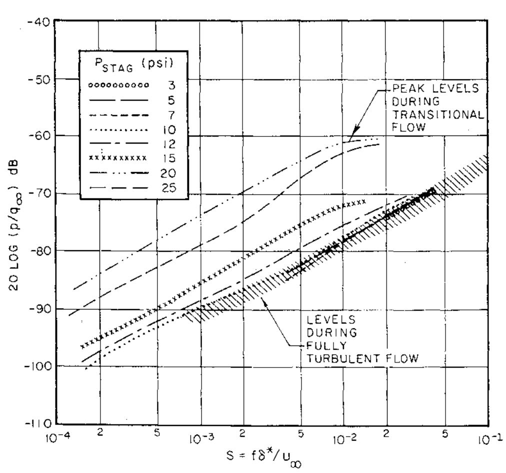

Fig. 8 Normalized fluctuating pressure level spectra measured by a flush-mounted ½-in. diameter microphone.

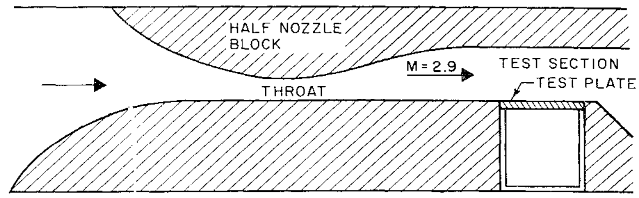

Fig. 9 Test setup for fully-turbulent flow measurements.

established at the sensor location on the test plate a supplementary experiment was performed to determine the amount of level decrease that follows after flow transition. A test plate, incorporating a number of flush-mounted $\frac{1}{2}$-in.-diam microphones was arranged so as to form part of the bottom wall of the wind tunnel test section (see sketch, Fig. 9). Natural boundary-layer development along the tunnel wall for 7 ft following the nozzle throat resulted in fully turbulent flow at the test section for all stagnation pressures possible. The flow Mach number used was again 2.9, and the turbulent boundary-layer displacement thickness at the microphone location was 0.22 in. Third-octave band fluctuating-pressure level spectra were measured at P_{stag} equal to 10 psi and 20 psi; these spectra are incorporated in Fig. 8. They scale with the dynamic pressure and thus appear superposed in the dimensionless representation used in this figure. (It is, of course, purely accidental that these curves coincide with those that were obtained previously corresponding to the laminar flow regime.)

The general conclusion drawn from this supplementary experiment is that highest fluctuating pressure levels occur during transitional flow, and that a significant decrease in (normalized) level occurs as the fully turbulent flow regime is approached. One should, however, realize that the level decrease appears more drastic in this normalized representation than in an absolute one; in approaching the fully turbulent flow regime, the freestream dynamic pressure q_∞ (or, equivalently, the freestream Reynolds number) increases, resulting in inherently higher "absolute" fluctuating pressure levels.

D. Response of the Transition Detector

Prior to exposing the transition detector configuration to transitional flow conditions, the configuration was calibrated 1) under purely acoustic excitation in the absence of flow, and 2) under a fully turbulent supersonic flow (using the test setup shown in Fig. 9). Figure 10 shows the results. The acoustic calibration curve was obtained by exposing the detector configuration to a pure incident sound field. Two pronounced resonances occur. The response in the presence of supersonic flow ($M = 2.9$), on the contrary exhibits no resonances at all, but merely a 6 db/octave drop-off at a characteristic break-frequency (2800 Hz). Supersonic flow over the port evidently causes the system to be damped

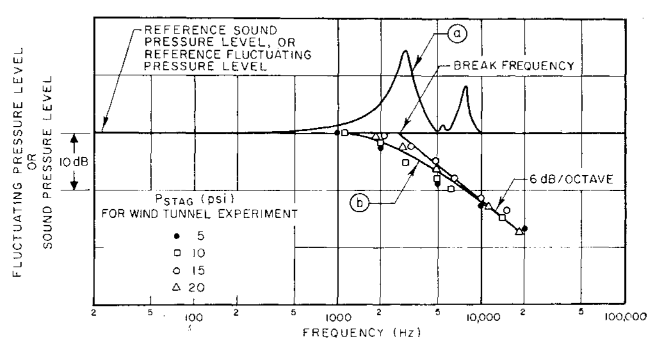

Fig. 10 Frequency response of transition detector for a) acoustic excitation, b) excitation by supersonic flow ($M = 2.9$, $T_{\text{stag}} = 100°F$).

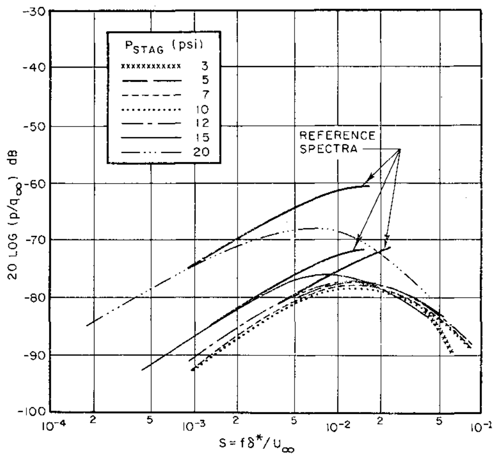

Fig. 11 Normalized response of transition detector configuration for various tunnel stagnation pressures.

so heavily that the resonances, which occur with acoustic excitation are not observed. The reasons for the occurrence and detailed explanations of this damping mechanism were presented in an earlier paper.[1]

After establishing the response characteristics of the transition detector configuration under supersonic flow, the transition detector was exposed to laminar and transitional flow (using the test setup as shown in Fig. 2). The results are presented in Fig. 11 again in normalized form. The response follows that of the flush-mounted reference microphone up to the break frequency. Again, the characteristic cluster for tunnel stagnation pressures below 10 psi is discernible, indicating "laminar" flow conditions. The (normalized) levels are seen to coincide with those measured by the flush-mounted microphone. Similarly, the spectrum for P_{stag} = 15 psi and for 20 psi are found to coincide with the corresponding reference spectra that are also drawn into Fig. 11. This coincidence at all stagnation pressures indicates the feasibility of the transition detector configuration for measuring the onset of flow transition, without a premature triggering effect. If flow transition had been initiated by the probing port, one should have expected the breakaway from the "laminar" flow conditions at some lower stagnation pressures. It is interesting to note that no early transition initiation occurred, although the probing port diameter was of the order of the local displacement thickness in the experiment. Use of the discussed transition detector configuration on full-scale re-entry vehicles, where larger displacement thicknesses occur typically, therefore, should cause no transition triggering problems.

If the transition phenomenon under atmospheric free-flight conditions is also characterized by a broadband level increase as the wind tunnel experiments suggest, then this fact could be utilized to reduce the telemetry bandwidth, necessary to indicate the onset of flow transition.

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