SOME RECENT INFORMATION ON AIRCRAFT VIBRATION DUE TO AERODYNAMIC SOURCES

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The purpose of this paper is to point out some of the aerodynamically induced vibration problems of aircraft. Specifically, the problems to be discussed are listed on figure 1. The problem area is shown on the left, and the bar graph alongside each illustrates the time during the flight that the vibration is most significant. Going down the list, it is shown that, in general, (1) boundary-layer noise is of significance during the cruise or major portion of the flight; (2) buffet for subsonic aircraft is similarly of importance during the high-speed flight, whereas for supersonic aircraft the buffet region is normally during the ascent-descent phases when the aircraft is in the vicinity of Mach No. 1; (3) for gust response, flight in the lower portion of the atmosphere represents the more important portion of flight time, whether for supersonic or subsonic aircraft; (4) engine noise and sonic fatigue are important, of course, during ground operation and take-off; (5) with regard to helicopters, the picture is black throughout the whole flight range.

Boundary-Layer Noise

Noise from the boundary layer which, of course, is in a turbulent condition, is important from two aspects: (1) The noise generated is transmitted through the vehicle skin into the interior, which could damage equipment or cause discomfiture of passengers. (2) The noise generated could damage the exterior skin structure through long-term exposure and resulting fatigue failure. A tremendous amount of literature has been generated in this area, for instance, Alan Powell and T. J. B. Smith prepared a bibliography in 1962 (ref. 1), at which time they noted 2,000 articles, and the first reference in this list was a paper by Michael Faraday in 1818, "On Sound Produced by Flames in Tubes." The basic work of Hans Liepmann as well as the definitive experimental work of W. W. Willmarth are noted. Ribner of the University of Toronto and Maestrello of The Boeing Company have been active in both the experimental and theoretical areas of the problem of boundary-layer noise.

Before discussing some analytical approaches, reference is made to some work done by D. A. Bies of Bolt Beranek and Newman (ref. 2). He examined recent literature concerning the measurement of the pressure fluctuations in the boundary layer, and devised a nondimensionalizing parameter which would collate the data into a logical pattern. After scanning the literature, he settled on about 30 sources of data, and on figure 2 is shown a summary of his results, where he selected for his ordinate, the quantity

$$10 \log \frac{U_\infty F_{(f)}}{2\pi q^2 \ C_f^{\ 2} \ \delta^*}$$

where U_∞ = stream velocity

$F_{(f)}$ = frequency spectra

q = dynamic pressure

C_f = friction coefficient

δ^* = boundary-layer displacement thickness

For the abscissa, he chose the Strouhal number where ω is the frequency, V is the free-stream velocity, and δ^* is the boundary-layer momentum thickness. There is a very large scatter in the data, so that either the correct parameter has not been found or the measurements themselves are not accurate. The line in the middle indicates a region he was able to identify as an area of concentration of results. It is apparent, then, that there is work to be done in the experimental determination of these fluctuating pressures, as the proper nondimensionalizing parameter has not been determined.

Also, a definitive and satisfying theory of boundary-layer turbulence has not been determined. What is the mechanism whereby the flow becomes turbulent? What is the triggering mechanism, and in what form does this oscillation exist? What are the nondimensionalizing parameters? In many places in the literature are found statements that offer no hope for a rational explanation, but this is a rather bleak outlook, and some day there will be a satisfying explanation. For instance, Theodorsen in 1958 proposed a model for turbulence which consisted essentially of the formation of horseshoe vortices in the boundary layer, and the subsequent growth and decay as the cause for the pressure fluctuations.

Black's hypothesis.- Following this line of attack, Thomas J. Black, TRACOR, has developed what may be the beginning of a rationale for boundary-layer noise (ref. 3). At the present time, this is just a physical model and the mathematics still must be developed. On figure 3 is shown the concept of the basic mechanism for turbulence generation, where is plotted a velocity profile, but the velocity is against a relative velocity, $\dot{X} - U_i$, where U_i is the velocity of the disturbance, thus for an observer on the disturbance, the wall appears to be moving upstream, while the free stream is moving downstream. Black postulates that the velocity distribution is initially laminar, having the profile as shown on the top right, but as the flow progresses certain nonlinear effects cause the flow to gradually deviate from this initial flow, as shown in the figure. Below this figure is plotted the difference in the original purely viscous velocity distribution and the newer velocity distribution caused by the nonlinear effects. The supposition is now that a vortex pair is formed due to the shearing action in the laminar sublayer, as shown on the bottom left. The upper vortex will then float upward due to a lifting force similar to the bound vortex on an aircraft wing. A vortex must either be infinite in extent, end on a solid surface, or close on itself. For this case it will form a complete circuit, such as shown on figure 4, and this picture is identical to that depicted for a lifting wing. Eventually, the vortex on the wall will dissipate, and a horseshoe vortex attached to the wall and extending off into the boundary layer will remain. Another interesting facet to this physical model is that it is possible to explain the presence of small eruptions or jetlike flows in the main stream, which have been observed experimentally, and the explanation could be that the induced velocity on the underside of the vortex resulting in a flow which, when it reaches the edge of the boundary layer, would look like small random jets. Of course, it is presumed that the strength of these vortices would be variable and thus provide random pressures. Further, Black points out that the scaling distance for the smaller and higher frequency disturbances may be the laminar sublayer thickness, whereas the lower frequency, larger vortices may scale with the boundary-layer thickness. This attack should be pursued further to see if a mathematical model could be developed. Some of the questions to be determined are: At what point in the flow will the vortices form, and what will be their strength, and what determines their strength?

Houbolt's method.- One more semianalytical method for turbulent boundary-layer flows is research recently done by Dr. John C. Houbolt of Aeronautical Research Associates of Princeton (refs. 4 and 5). The problem was to determine the fluctuating pressures in the boundary layer at hypersonic speeds. On figure 5 is plotted the root-mean-square of the pressure fluctuation divided by the dynamic pressures plotted against Mach number. Here, it can be seen that a rough configuration, such as the Mercury spacecraft, may have pressure fluctuations ranging around 5 percent of the free-stream dynamic pressure, which would be in the buffet range, whereas for smooth shapes the order of magnitude is around 1/2 percent of the dynamic pressure. Some vehicles enter the atmosphere at very high dynamic pressures and high Mach number, and utilizing this constant value for the same response would indicate extremely high values of the pressure fluctuations, enough so that the vehicle would certainly be destroyed or seriously damaged, and experience has shown that this is not the case. So what Houbolt did was to derive a more rational variation of σ with Mach number. He used as a basis the local mean density of the flow in the region of large velocity gradient in the boundary layer. With this assumption, he derived an expression for the rms pressure as shown on the top of figure 6, $\sigma = c\rho_1 V_0^2$, where C is a constant to be determined, ρ_1 is the density at the point of maximum velocity gradient, and V_0 is the free-stream velocity. Utilizing a recovery factor and fitting the expression to the known subsonic and low supersonic results, he obtained

$$\left(\frac{\sigma}{q} = \frac{0.007}{1 + 0.012\, M^2}\right)$$

Note that for $M = 0$, σ/q reduces to 0.007, a value in agreement with the experimental results previously shown.

Also, by assuming a model for convection velocity, he was able to derive an expression for the power spectrum, as follows:

$$\varphi(\omega) = \frac{0.00002\, \frac{1}{V}\, q^2\, \delta^*}{1 + \left(\frac{\omega\delta^*}{V}\right)^2}$$

On the two plots of figure 6 are shown the σ/q against Mach number and the spectrum for various velocities. Note that with the model Houbolt selected, the σ/q does indeed drop off in the high Mach number region. As far as is known, this has not been confirmed experimentally, due principally to the difficulty of measuring fluctuating pressure under high-temperature conditions. On the same figure are shown some spectra for several flight velocities. As the flight speed increases, the spectra are reduced in magnitude, but are very flat and extend to higher frequencies.

Buffet

Buffeting of aircraft is a phenomenon related to boundary-layer noise. The scale lengths are much larger than the boundary-layer thickness and approach the dimensions of the body dimensions, such as the wing chord or body diameter. With regard to the aerodynamic input, there are no theoretical means for estimating the buffeting unsteady pressures, and thus resort is made to experimental methods, principally wind-tunnel tests on scaled models. On figure 7 are shown some types of buffet problems which have arisen on aircraft. On the top is a very common type which involves the vibration of the tail resulting from unsteady flow from the wing. Another type involves the flow around a body with unsteady incidence on a canard, such as happens on the B-70 for some subsonic flight conditions. Another type can occur in cutouts or bays, and this is usually important solely for the design of the payload in the bay, such as rockets and missiles. Another type can be due to protuberance on aircraft, for instance, for camera windows or other necessary bumps.

Tail buffet.- With regard to tail-induced buffet, a dynamically induced aeroelastic model in which both Reynolds number and Mach number are thus scaled can provide adequate prediction for full-scale aircraft as shown by A. G. Rainey (ref. 6).

Cavity buffet.- With regard to bay or cavity buffet, some excellent work was accomplished by Plumblee et al. (ref. 7).

Protuberances.- Protuberances on aircraft can cause a local flow breakdown, and result in rather severe but area-restricted pressure fluctuations which can degrade or damage sensitive instruments. For instance, in an investigation of the flow around a step protuberance on a model tested in the Transonic Dynamics Tunnel at Langley Research Center, the measured rms buffeting pressures appear as shown in figure 8 for two configurations. The aerodynamic shapes are shown on the right of the figure, and the measured pressures plotted against Mach number. An important factor in this figure is the fact that the phenomenon is more severe at subsonic Mach numbers peaking about $M = 0.7$, although the tests were extended to the low supersonic range. The model was then reshaped to remove the step by refairing the nose as shown (the dotted lines show the first shape), and the reduction in buffeting loads is dramatic; however, there is still a slight peak at $M = 0.88$.

Response to canard buffet.- Early in the flight program, it became evident that the XB-70 was experiencing stall buffet of the canard at low values of dynamic pressure for subsonic flight. The following results represent unpublished work by Dr. Eldon Kordes of the NASA Flight Research Center. In order to determine the effect of a strong disturbance applied through the canard structure on the nature of the airplane response, the acceleration at the center of gravity was analyzed for the condition of $M = 0.4$ at 10,000 feet (3,048 meters) altitude. The power spectral density estimates of the normal and lateral accelerations obtained from a 40-second record sample are shown in figure 9. The results for the normal acceleration show the response of several structural modes with a maximum structural response at 13.4 cps which corresponds to the first symmetrical bending mode of the canard. The response for this flight condition contains a large amount of energy from structural modes above 6 cps and with a total rms value of 0.046g. The lateral acceleration response shows a rms level of 0.025g with almost all of the energy between 5 and 11 cps.

Comparison of the power spectral density estimates of center-of-gravity accelerations with the estimates shows that whereas the primary structural response for canard buffet is at 13.4 cps, this frequency does not appear in the gust response. Even the acceleration response at the pilot's station does not contain structural response at 13.4 cps. Unfortunately, the accelerometers at the pilot's station were not operating on the flight when canard buffet was experienced, so that a direct comparison of the pilot's station response cannot be made with the response in turbulence.

Gust Response

A history of the development of the gust criteria over the years follows: On figure 10 are shown six airplane types representing six identifiable time periods of development of the gust criteria. On the upper left is shown a biplane in the period of the 1920's. There is no information as to how, if at all, the response of loads to gust was performed; the likelihood is that no attempt was made to design the airplane for this condition. About 1934, a sharp-edge gust criterion was developed by Rhode et al. (ref. 8) which was used for several years. About 1942 a ramp gust was introduced, and some account was taken of the relieving factor of the vertical acceleration of the airplane as well as the effects of unsteady aerodynamics. A good summary of the status of gust work was made by P. Donely (ref. 9) at this time. In 1955 K. G. Pratt (ref. 10) introduced the effective gust factor which he terms Kg. In this case, a 1 - cos gust having a length of 25 chords and a maximum velocity of 50 ft/sec, Pratt provided tables of Kg with which the correction to be made to the older type of criteria could be calculated. About 1960, when the present fleet of jets were being designed, the same 1 - cos gust was used, with two changes: first, the length of the gust was made variable and calculations were made until the maximum response was obtained, and second, the flexibility of the wings was taken into account.

For the future, concepts of continuous random turbulence will almost certainly become the design standard. At the present time, both the 1 - cos variable gust as well as the random turbulence concepts are being used. The random approach has been pioneered by Etkin (refs. 11 and 12) in Canada, and Houbolt (ref. 14), Press (ref. 12), and Diederich (ref. 13) in the United States.

For the supersonic aircraft, such as the B-70 and SST, it is not the wing which is the main contributing factor to turbulence, but rather the fuselage-wing combination, or more specifically, the complete airplane vibration modes, which for these long slender configurations contain a large degree of flexibility in the fuselage, as opposed to the rather stiff fuselages and flexible wings of the present subsonic jets. On figure 11 are shown the acceleration spectrum for the pilot's station for the XB-70 at $M = 2.4$ and altitude 55,000 ft, and for a typical subsonic jet. The large response at the low frequency portion is due to the rigid body "short-period" response, typical of all aircraft. However, the unusual response is at the higher frequency portion, and it will be noted that the XB-70 has two rather large peaks as compared to the subsonic jet. These two peaks correspond to the third and fourth airplane vibration modes. This results in a rather rough ride for the pilots, even in extremely light turbulence. There have been times during the flight of the B-70 when the pilot reported light to severe turbulence, when the nearby chase airplane pilot reported no turbulence. It is apparent, then, that some method for reducing these large responses is needed and some work is now underway. One method would be to automatically sense the motion of the aircraft and attempt to dampen out the motion, including the flexible mode of the airplane. This has actually been demonstrated on a B-52 airplane, and the results are shown on figure 12, from reference 15. Here is shown damping ratio plotted versus dynamic pressure for two modes: the Dutch roll mode and the fuselage side bending mode. Of course, an increase in damping means a corresponding decrease in response of the aircraft. There is a large increase in damping for both modes for the system on, as compared to the system off. Also shown are the results of flight tests of the actual automatic system and the agreement is excellent. Thus, it appears that the tools necessary to reduce the response of these very flexible airplanes to random turbulence are in hand.

Flutter.- Flutter is a self-induced oscillation of a surface which can result in the destruction of the surface. The first recognized flutter occurred on a World War I bomber, and the solution was obtained by Lancaster and Bairstow who advised an increase in torsional stiffness of the tail surface. Since that time, there have been rapid advances made in the state of the science. The flutter speed of wings throughout the subsonic range as well as the supersonic range can be analytically predicted. The one remaining gap lies in the transonic speed range, where the theories are still not adequate, and wind-tunnel testing is mandatory. For this range, model tests are run and the Transonic Dynamics Tunnel at Langley Research Center has been used to proof-test every military aircraft of recent vintage.

To provide a graphical view of the transonic problem, on figure 13 is shown the true airspeed for flutter plotted against Mach number. Here is noted a very small variation in speed, until approaching $M = 1$, where there is a rather large reduction in flutter speed and, finally, a rapid increase upon entering the supersonic region. (It is interesting to note that this curve is very similar to the reciprocal of the slope of the lift curve, when plotted against Mach number.)

To round out the flutter picture, a plot illustrating one other area that requires additional work, namely, the coplanar case, and some experimental results are illustrated in figure 14 (ref. 15). At the top is shown the configuration when the main wing is pivoted, and flutter speed is plotted against sweep angle. As the angle of sweep increases for the wing alone, the usual increase in speed with increasing sweep angle is noted; however, when a fixed tail is placed on the aircraft, the flutter speed suddenly decreases.

Sonic Fatigue

By sonic fatigue is meant the damaging of a small section of the aircraft by noise generated mainly by the exhaust of jets or due to the boundary layer itself, although similar results on fuselage areas near the plane of the propeller can result. There are essentially three problem areas: namely, what are the noise spectrum and orientation generated by the jet? What is the response of the panel due to this noise? And finally, what is the fatigue life of the jet? Two conferences were held on this subject: one in 1966 at the University of Minnesota and published in WADC TR 59-676 (ref. 16), and a second at Dayton, Ohio, the proceedings of which were published in a book entitled "Acoustical Fatigue in Aerospace Structures" (ref. 17).

Jet noise.- The famous work of Lighthill (ref. 18), set the pattern for theoretical jet noise prediction, wherein he stated that the noise produced by a jet was essentially due to shearing action on the jet boundary, and the noise was proportional to the V^8. This velocity dependence has been rather well substantiated in the past; however, some recent work has shown regions where this may not be entirely the full story. On figure 15 this problem is illustrated qualitatively. Here, noise is plotted against jet exhaust velocity. The central part of the curve seems to follow nicely the 8th power law. However, it has been observed in experimental work of actual configurations that the noise reduction in the low velocity range does not decrease as rapidly as predicted by the Lighthill theory, and is somewhere between the 4-6th power. Similarly, for the higher jet velocities the noise does not seem to be as great as the 8th power indicates. For the lower velocity range, this problem has been experimentally studied by Gordon and Maidanik of Bolt Beranek and Newman (ref. 19). It is their conclusion that noise generated inside the pipe by obstruction as well as rotor noise may cause a noise which is proportional to the 4-6th power, and can be explained by the use of dipole or doublet distributions, that is, a sort of lifting surface in the pipe.

With regard to panel response, Alan Powell (ref. 20) has proposed the more or less classical procedure of calculating the response of a panel utilizing many vibration modes and the complete noise field over the panel with all the attendant correlation of the pressure field. This is quite an imposing job, and B. L. Clarkson has proposed what may be an easier out, wherein he focuses on one vibration mode (ref. 21). In that paper, Clarkson points out that from experiments most of the panel response is in a single vibration mode, and it is usually the lowest mode. With this concept then, he utilizes a result of Miles for the response of a single-degree-of-freedom system to random noise. Specifically, the equation, shown at the top of figure 16, is

$$\sqrt{\sigma^2 \, \delta(t)} = \left[\frac{\pi}{48} f_r \, G_p \, (f_r) \right]^{1/2} \sigma_0$$

σ = vision damping ratio

f_r = frequency of predominant mode

$G_p \, (f_r)$ = spectral density of pressure at fr

σ_0 = stress at point of interest due to a uniform static pressure of unit magnitude

To illustrate the adequacy of the method, results taken from Clarkson's report illustrate the results of a number of experiments versus the analytical estimates, where the measured rms stress is plotted on the ordinate. This is quite remarkable agreement, and it should constitute the beginning of a semirational approach. Of course, the next step is to estimate the fatigue life, and experimental data are lacking, since it would be necessary to have S-M curves from random input having a Rayleigh distribution of stress and having rms stress and the number of reversals as ordinates. A considerable amount of experimental work would be necessary to gather these data.

Helicopter Vibration Problems

The helicopter has by far the most severe vibration problems of any aircraft, resulting from the fact that the main lifting surfaces operate in a completely nonuniform flow field. Some of the aerodynamic sources of the vibration are shown on figure 17 along with a conceptual plot of the vibration level plotted against forward speed. At the lower speeds, there is a rather severe vibration due to interaction of the tip vortex generated by a blade on the following blade. This noise is usually termed blade slap. This phenomenon is surprising, since it has usually been assumed that the tip vortex is normally deflected down and that it could pass under the following blade. This is still a research problem, and the fixes are being worked on. At the higher flight speeds, there are a number of problems such as stall, compressibility effects, stall flutter, and blade-motion instability. To obtain a better idea of how the blade operates, figure 18, taken from a paper by Al Gessow of NASA (ref. 22), illustrates the flow field. Looking down on the blade field, the portion of the rotational field shows certain important factors. The airflow is from top to bottom. Regions of stall and high Mach number operation are shown. For instance, a blade tip will be at $M = 0.9$ on the advancing side, whereas the blade root is at $M = 0.3$. When the blade is on the retreating side, the blade tip is at $M = 0.3$ and the root is practically at $M = 0$, and the whole event occurs once per revolution. On the other hand, the angle-of-attack ranges from -2 at the tip on the advancing blade to 4°-5° at the root, but on the retreating side can go as high as 14°. The hatched area shows the area of importance from the standpoint of stall and stall flutter. Stalling of the blade results in a more or less random input, whereas stall flutter involves a sinusoidal oscillation at the natural torsional frequency of the blade and is more or less proportional to the square root of the torsional frequency. Therefore, a possible fix is to increase the stiffness of the system. Of course, using airfoil shapes that will stall at a higher angle of attack will be beneficial as well as boundary-layer control. The stalling effect is one of the principal effects that limits the flight speed of a helicopter.

Concluding Remarks

This paper has been principally aimed at pointing out some major aerodynamically induced vibration problems of aircraft, and to provide some insight into the progress being made. Specifically, the paper has covered the following areas: (1) boundary-layer noise, (2) buffet, (3) gust response, (4) canard buffet, (5) flutter, (6) sonic fatigue, and (7) helicopter vibration.

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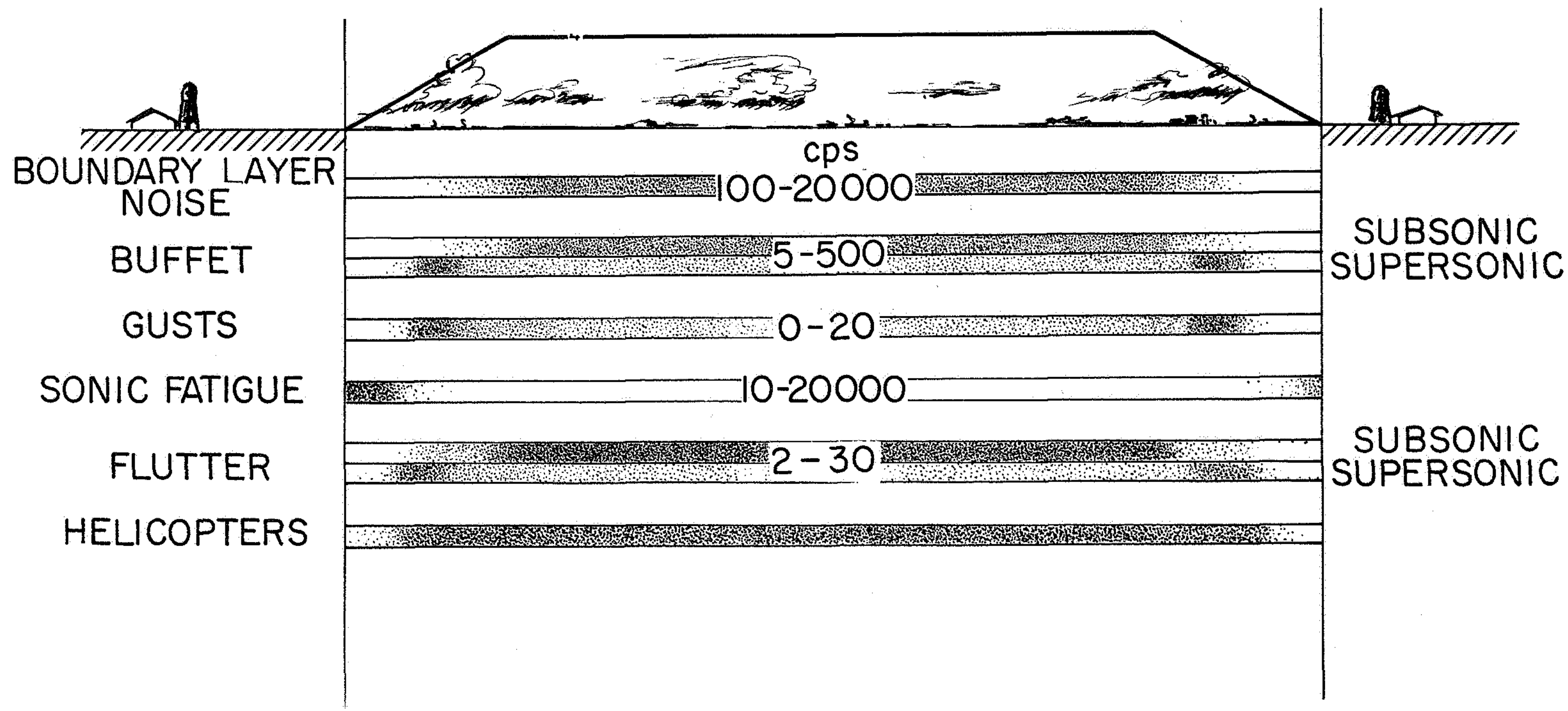

Figure 1.- Aircraft vibration.

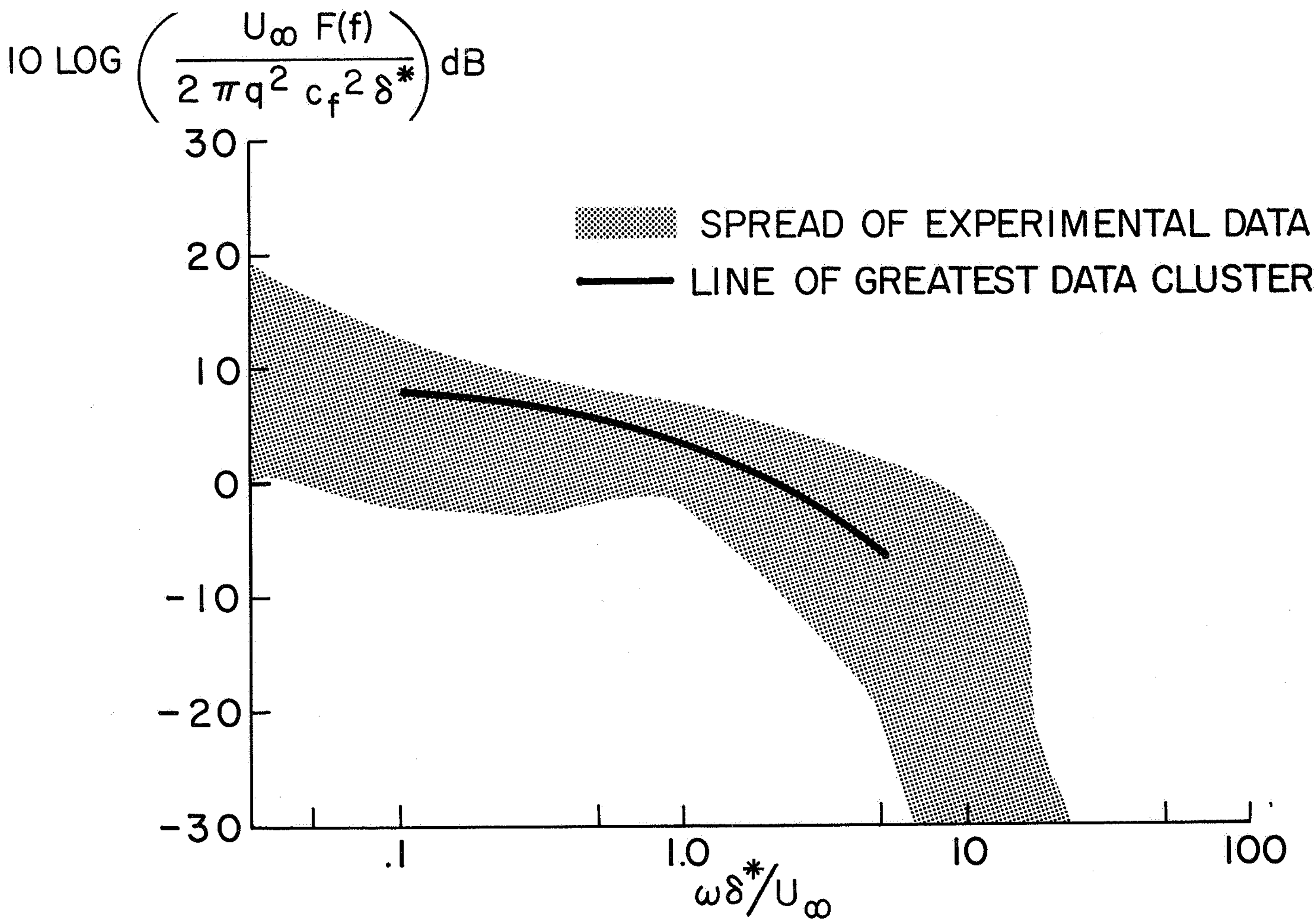

Figure 2.- Tunnel measurements of boundary layer pressure fluctuations.

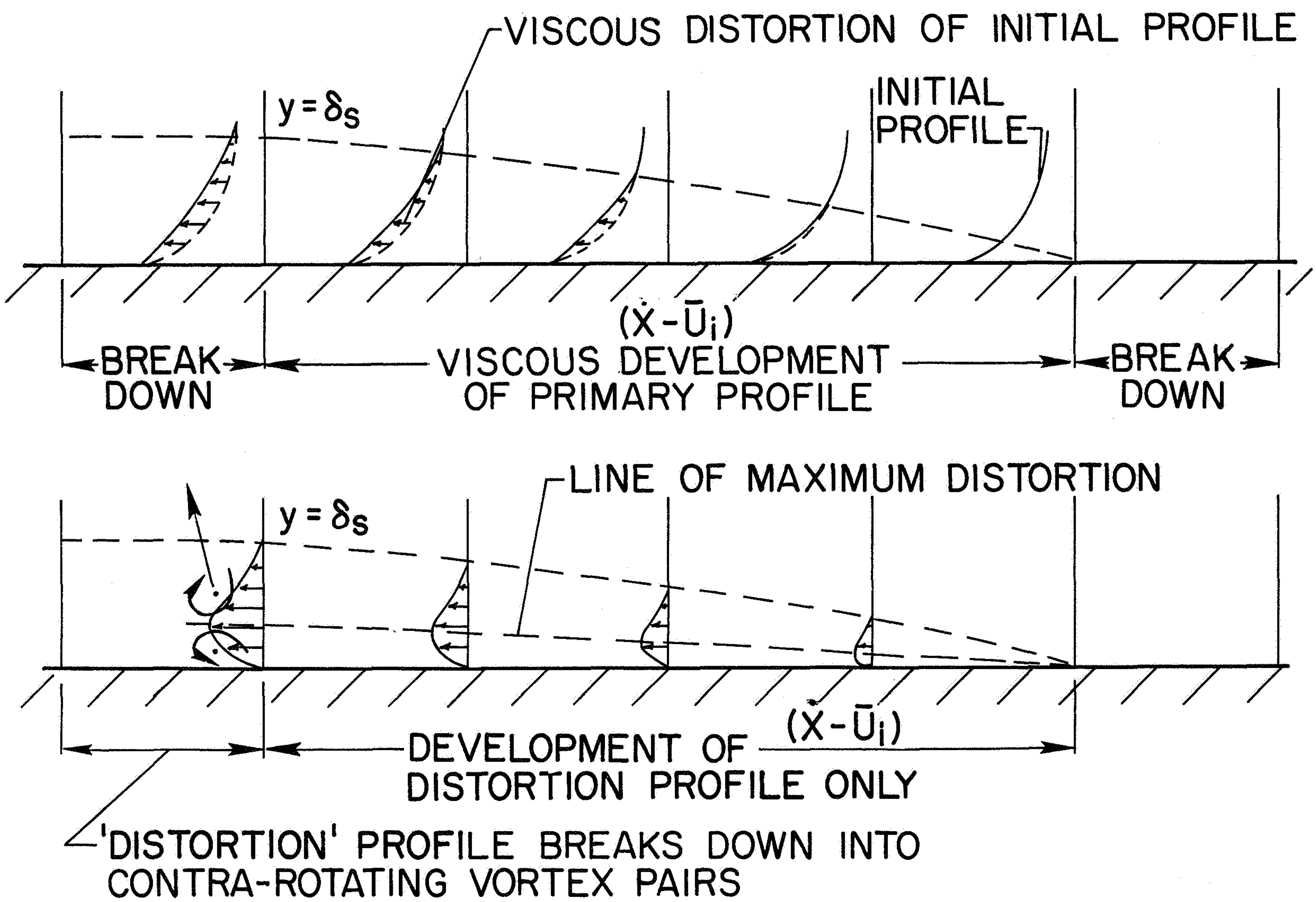

Figure 3.- Vortex-pair generation explained by breakdown of viscous "distortion" profile.

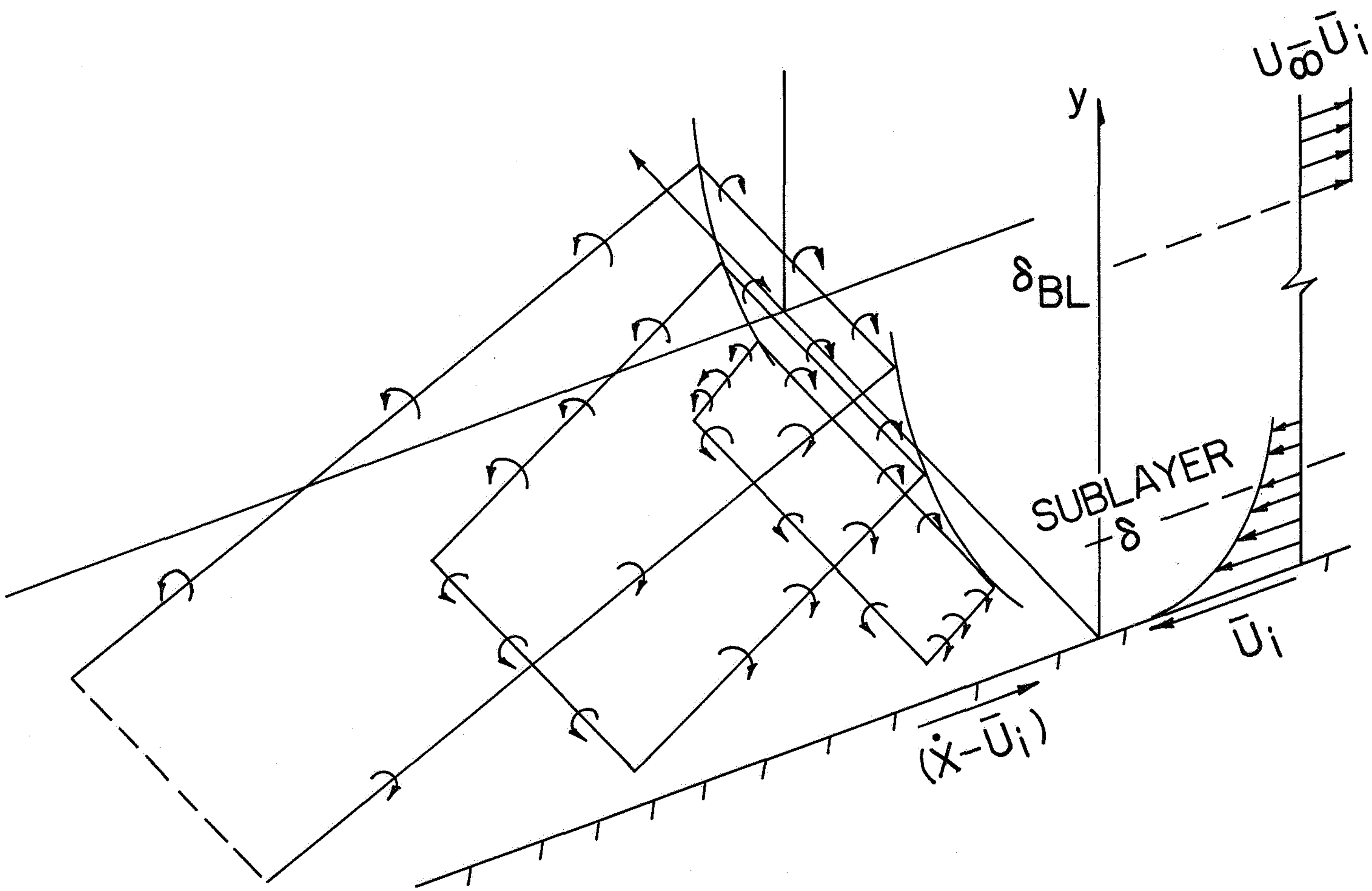

Figure 4.- Schematic diagram of vortex structure generated by instability.

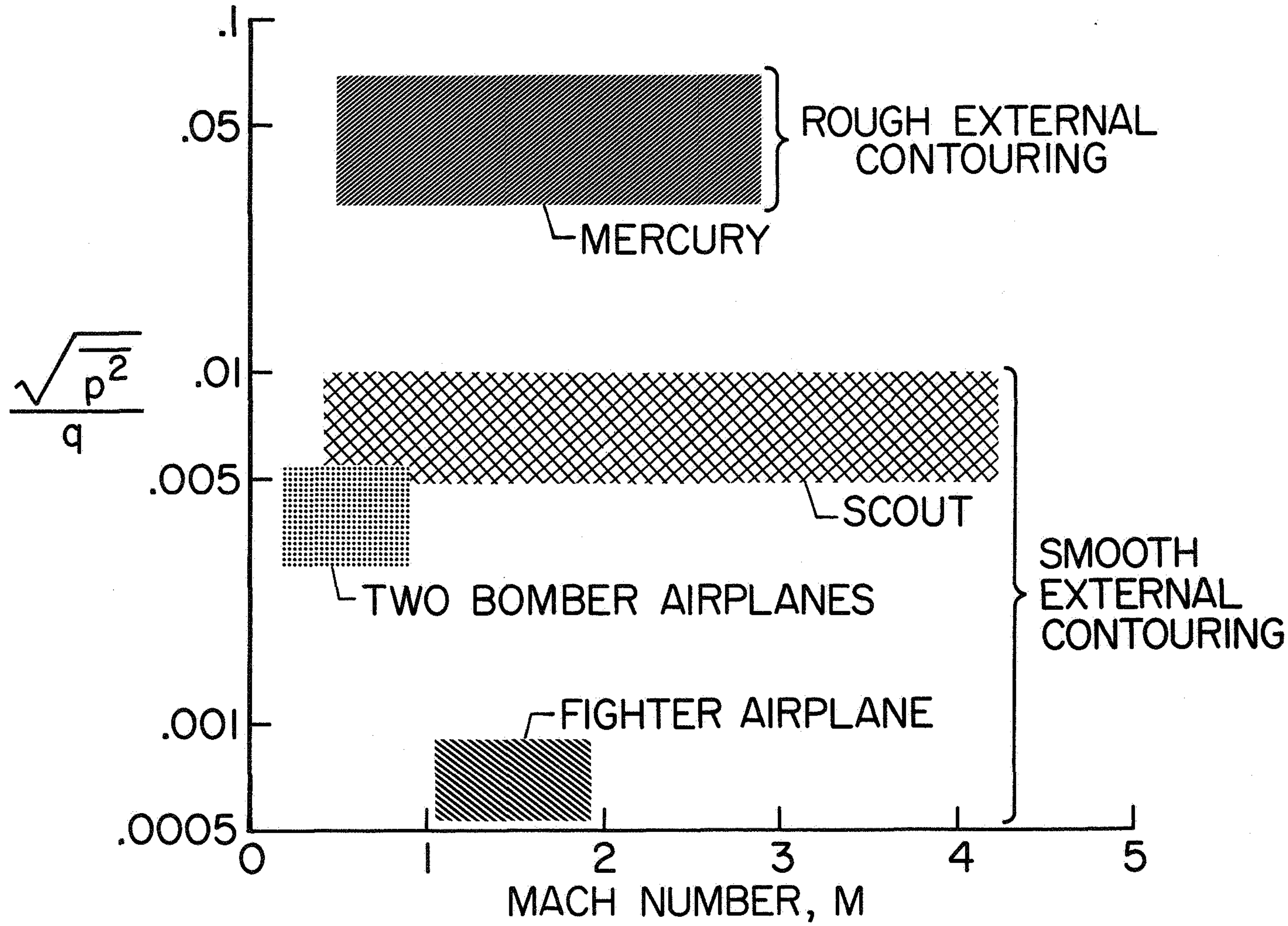

Figure 5.- Surface-pressure coefficients for various aerospace vehicles.

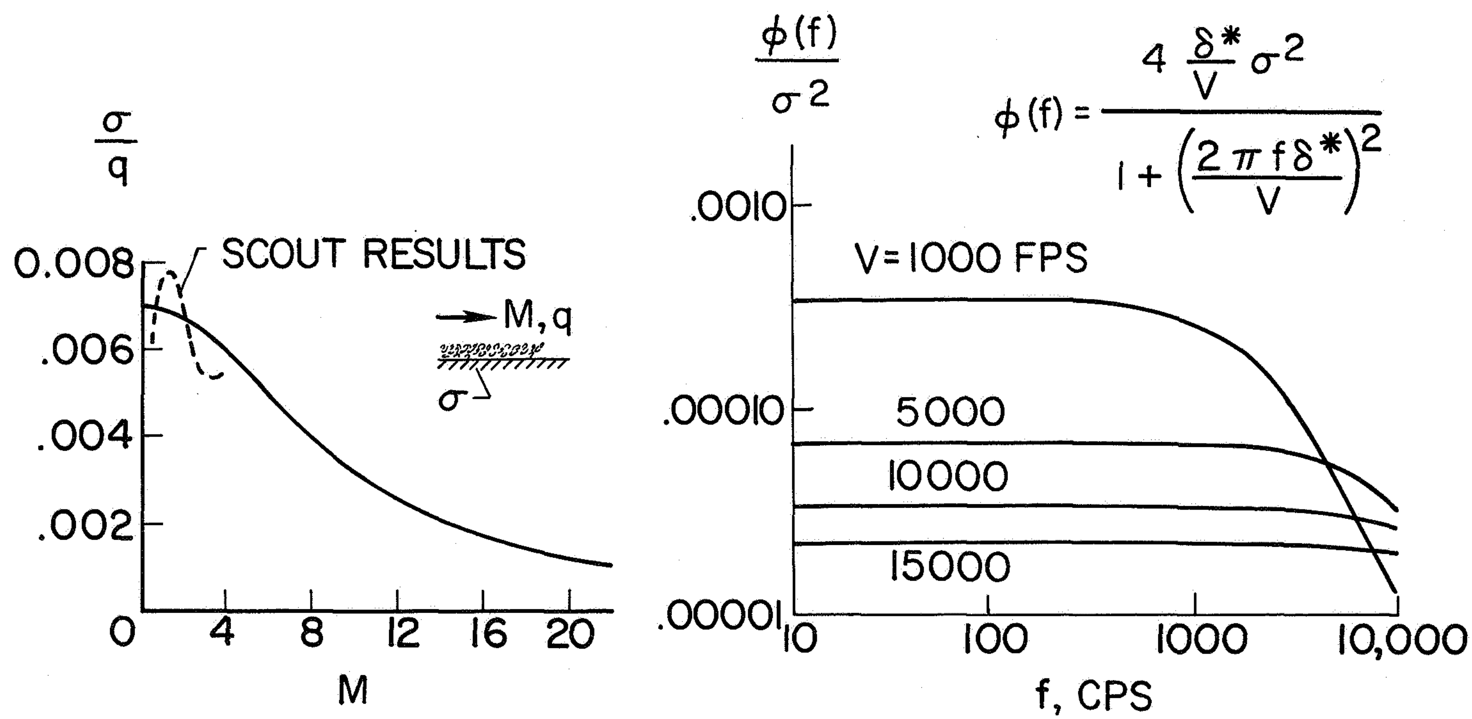

Figure 6.- Boundary-layer noise at high Mach number.

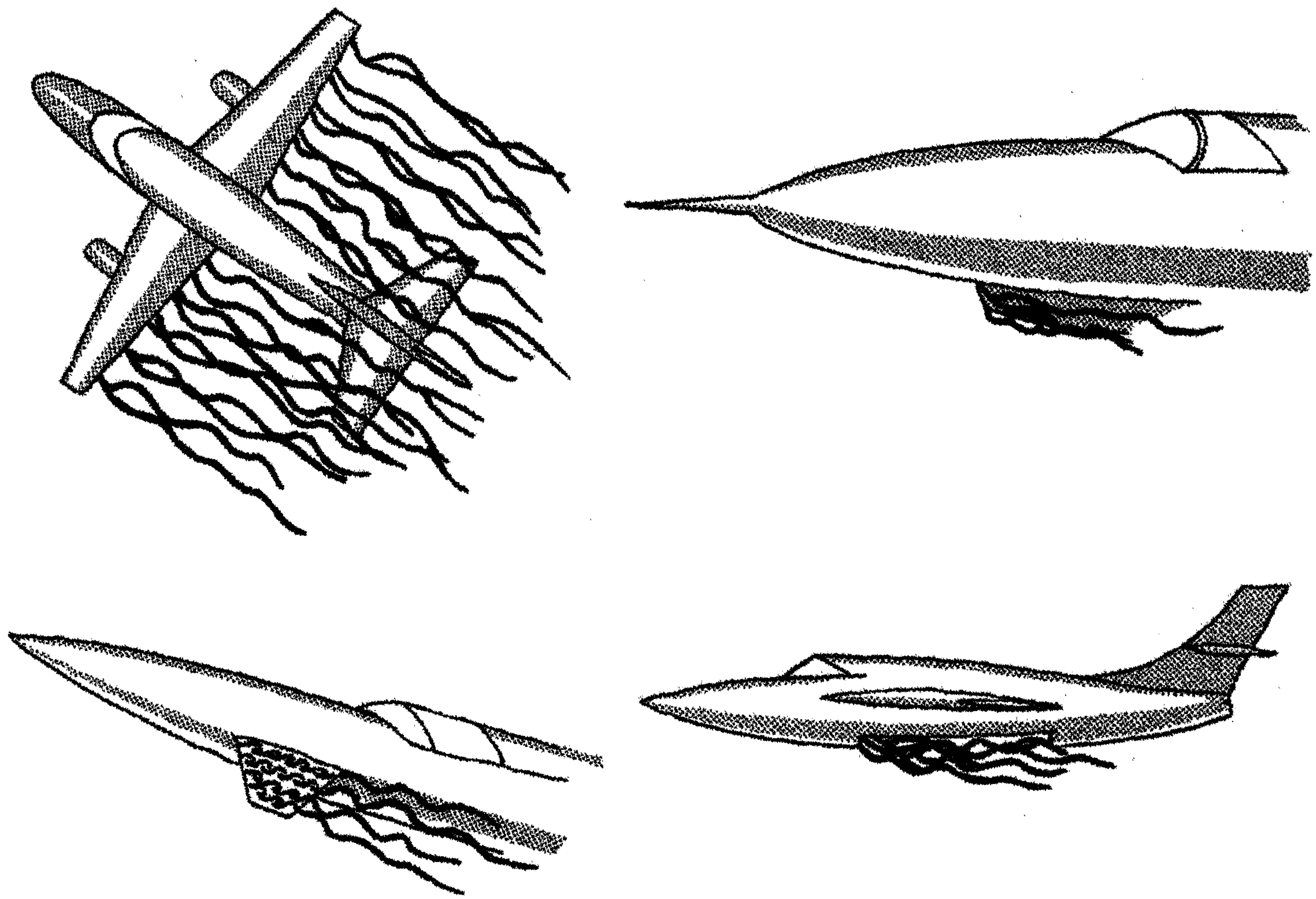

Figure 7.- Type of aircraft buffet.

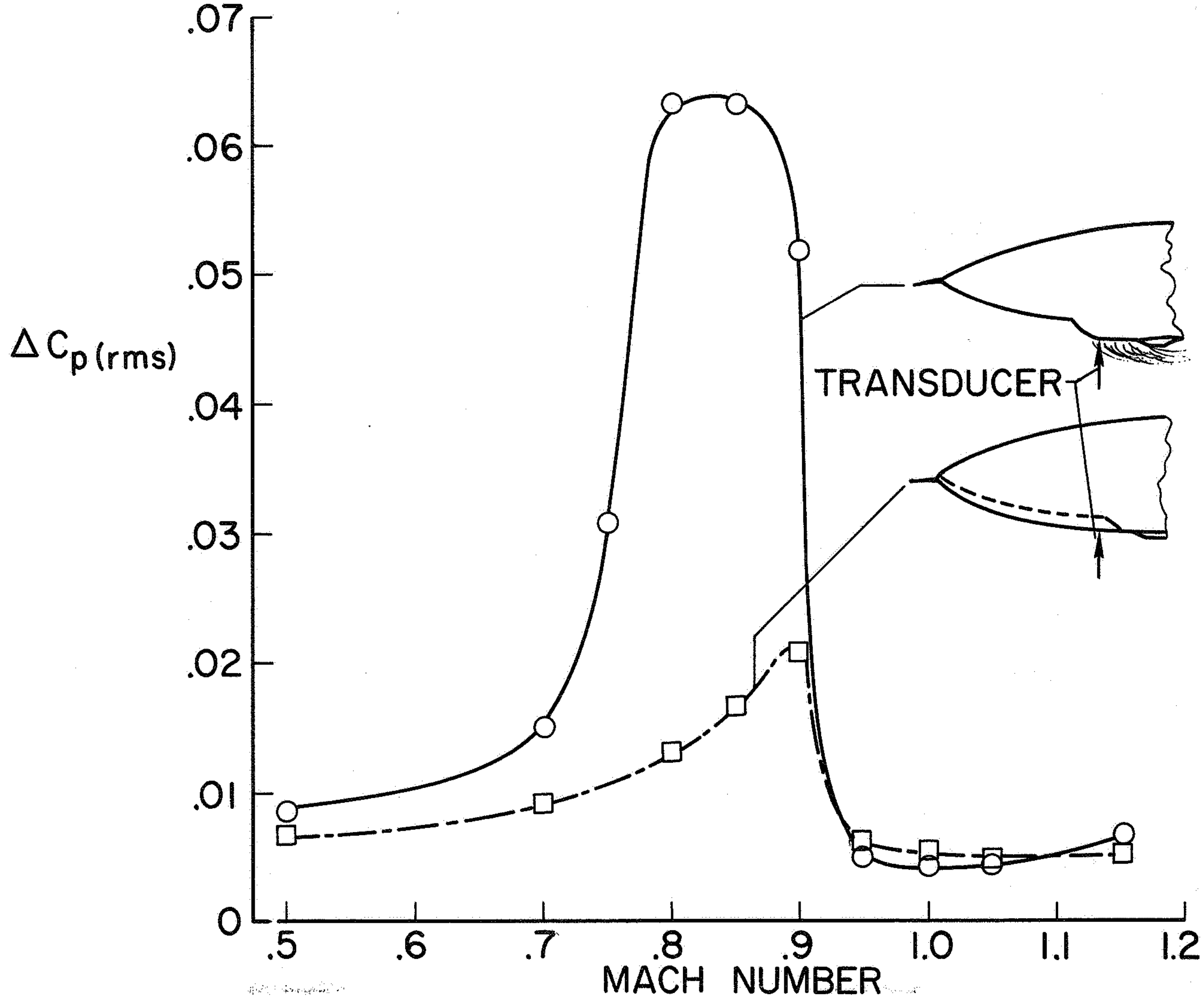

Figure 8.- Variation of fluctuating pressure coefficient with Mach number.

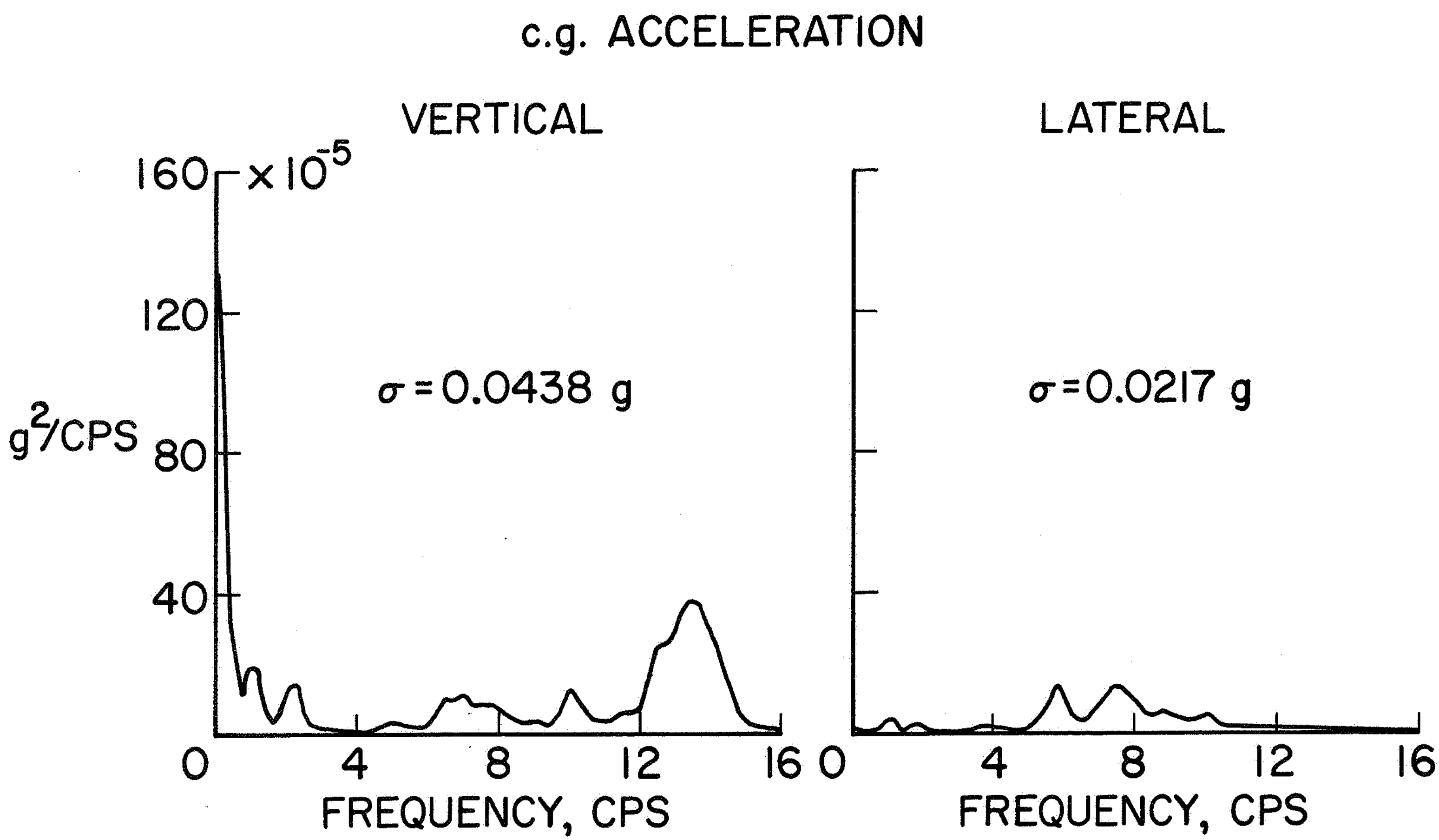

Figure 9.- Power spectra of XB-70 airplane response to canard buffet.

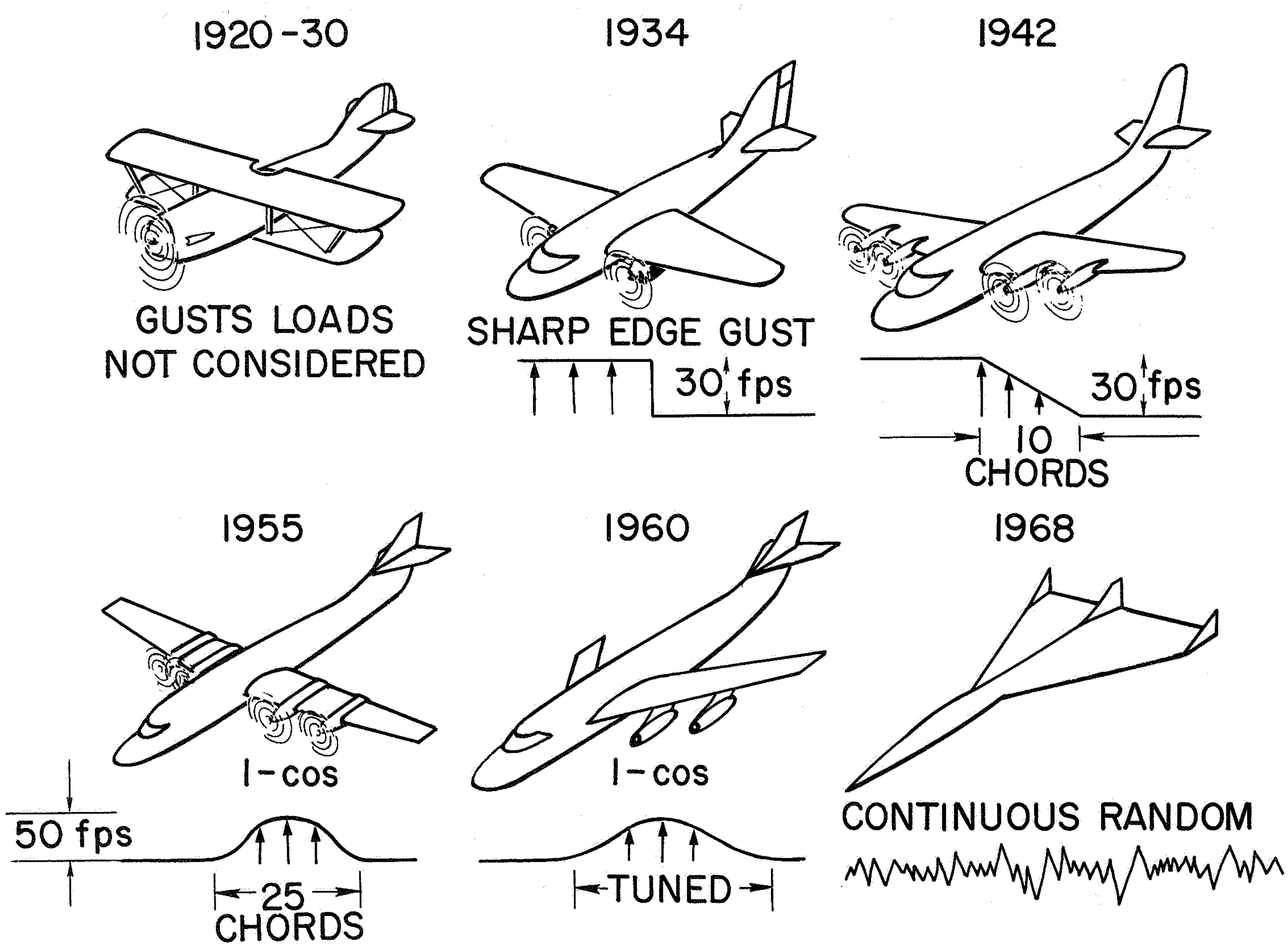

Figure 10.- Progress of gust-load criteria.

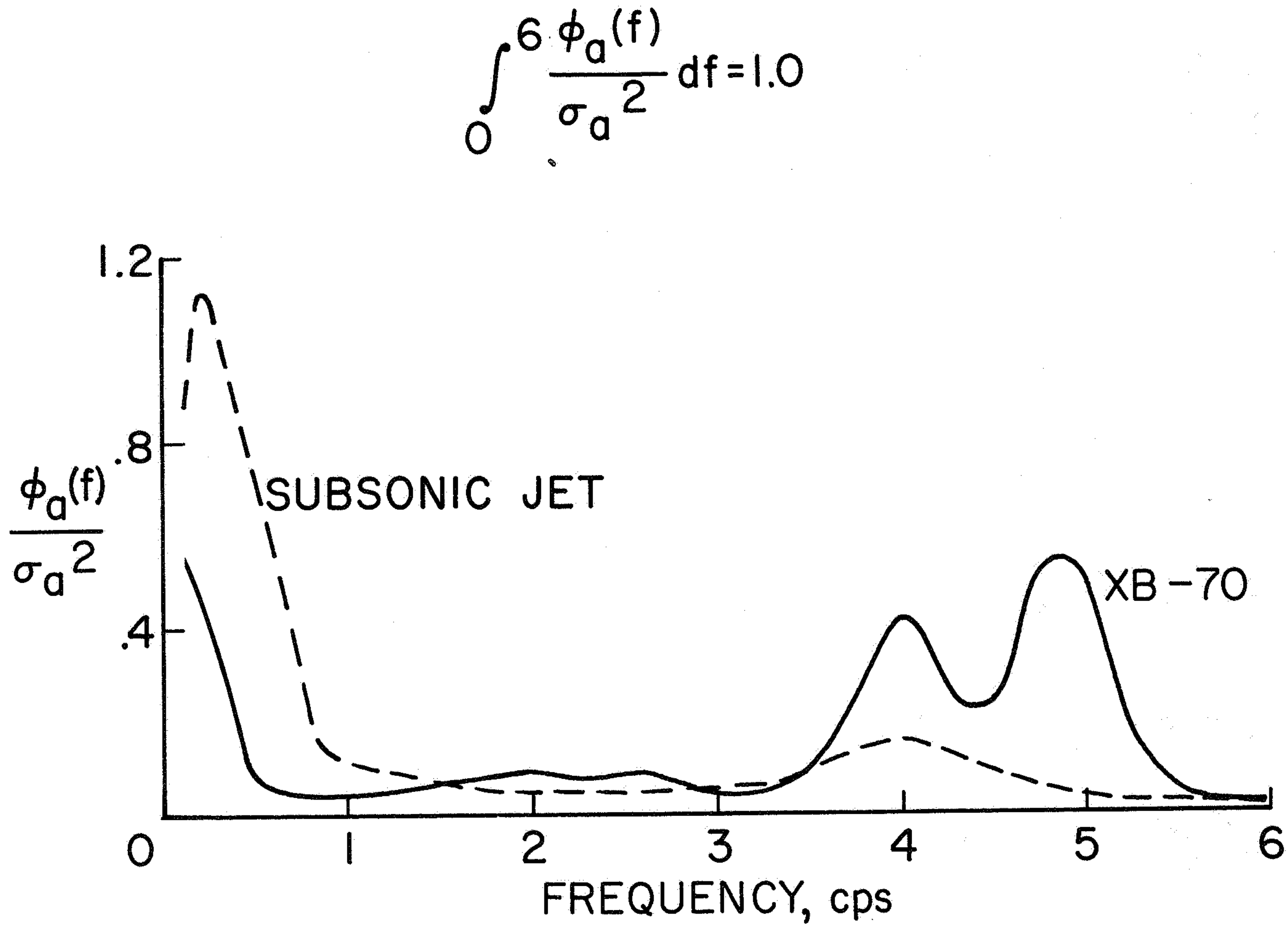

Figure 11.- Pilot-station vertical-acceleration spectra shapes.

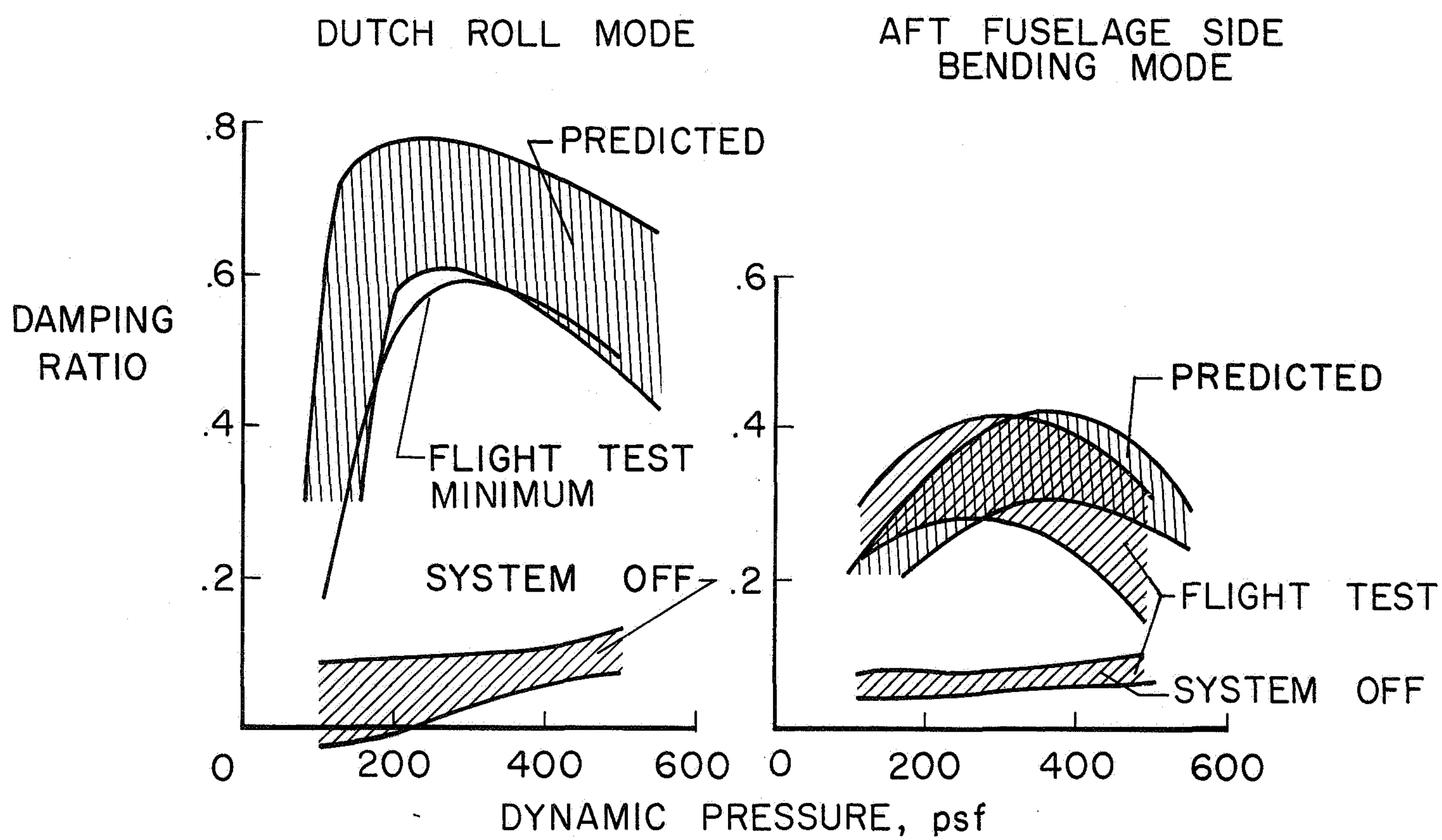

Figure 12.- Effect of modal suppression system on B-52 modal damping.

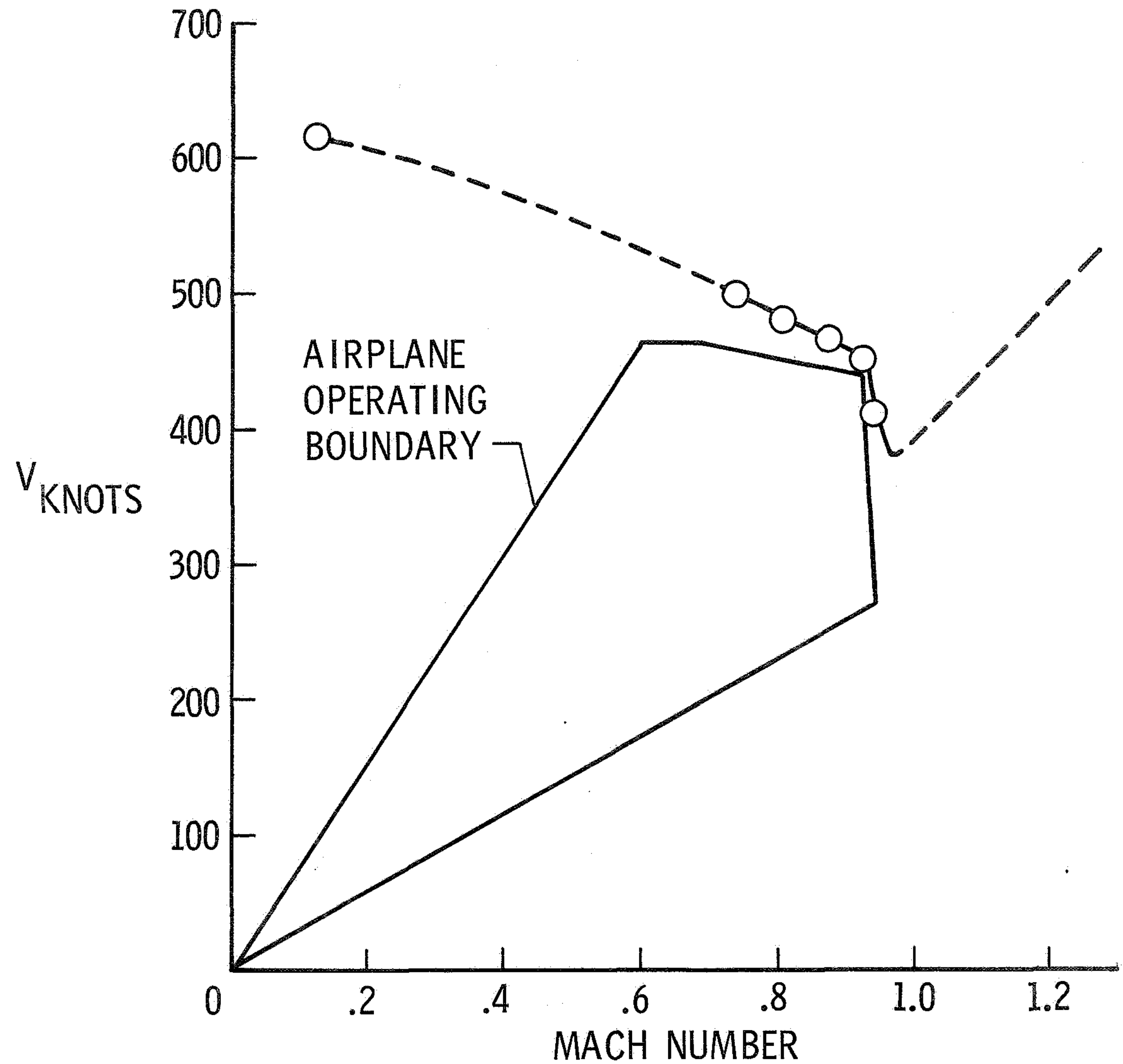

Figure 13.- Typical flutter boundary.

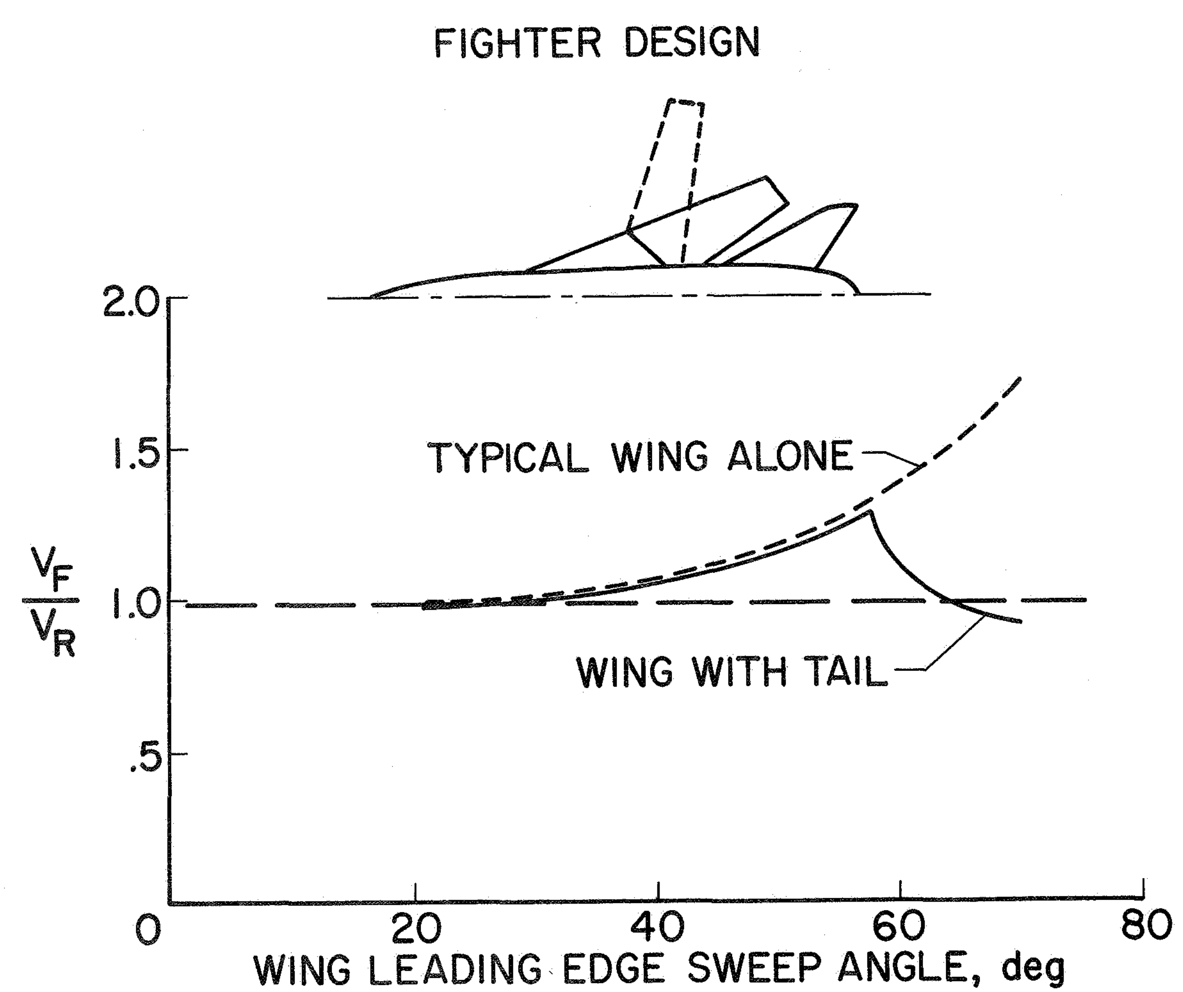

Figure 14.- Effect of wing-tail interference on flutter speed of a fighter design.

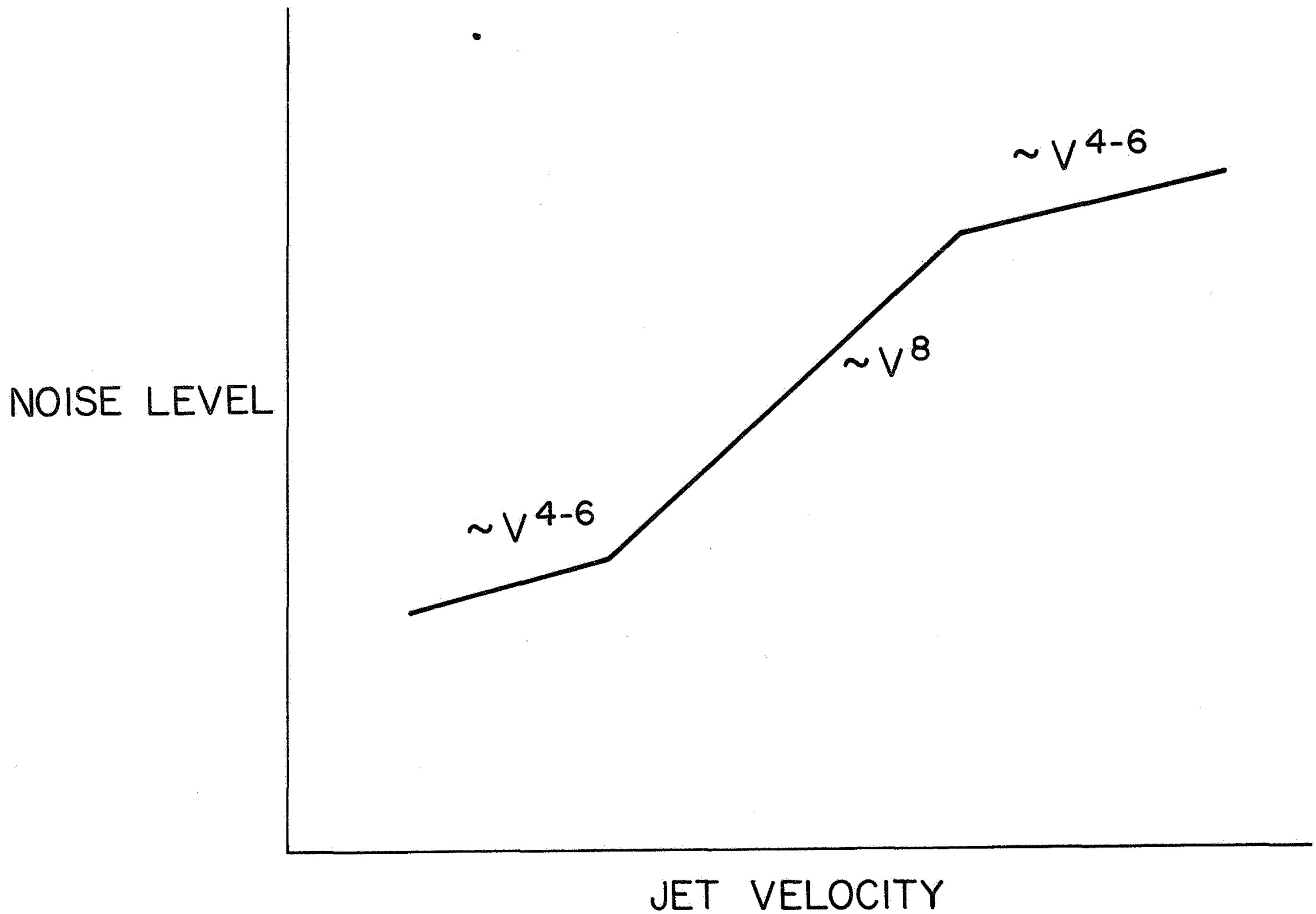

Figure 15.- Jet noise.

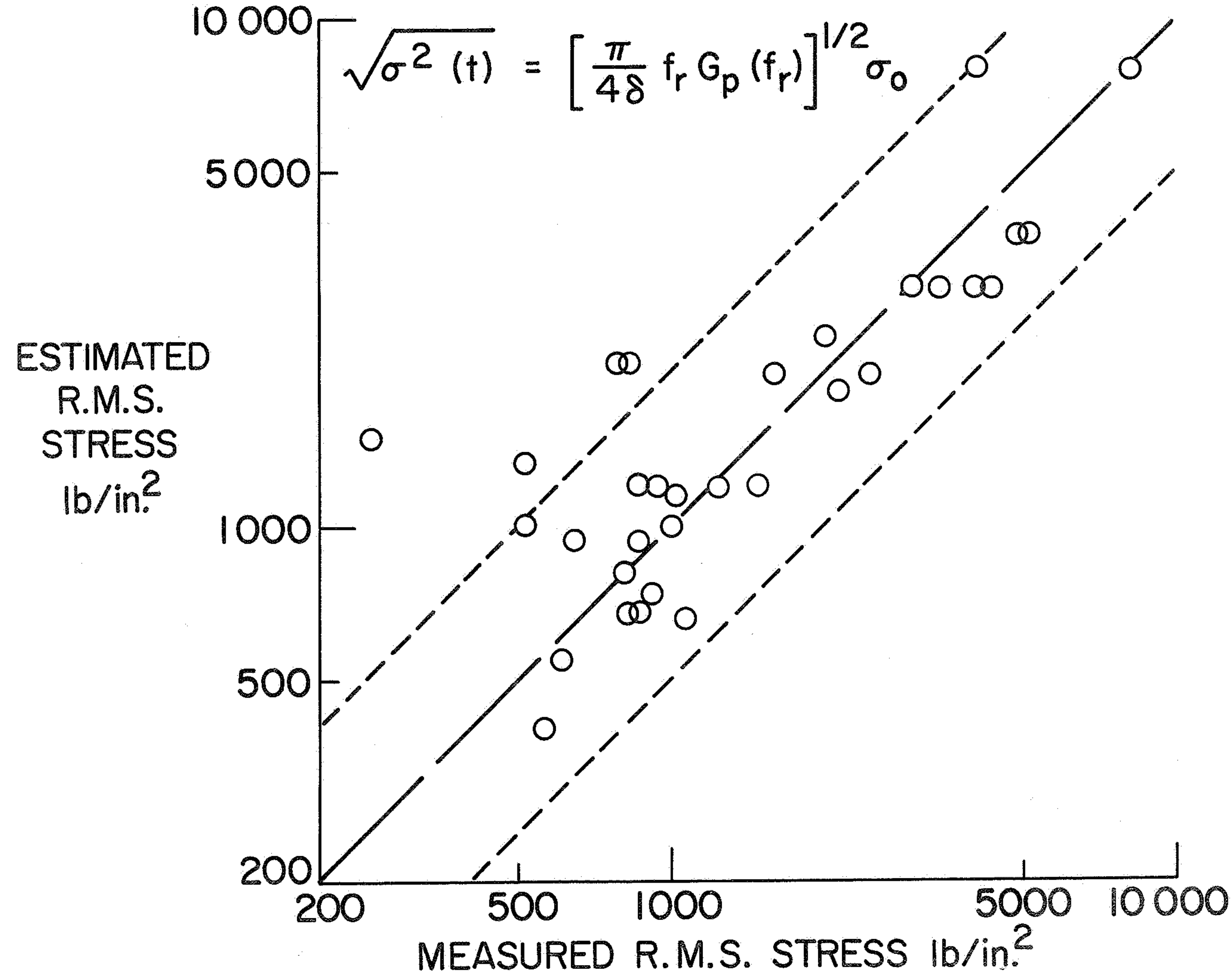

Figure 16.- Comparison of estimated and measured stress levels.

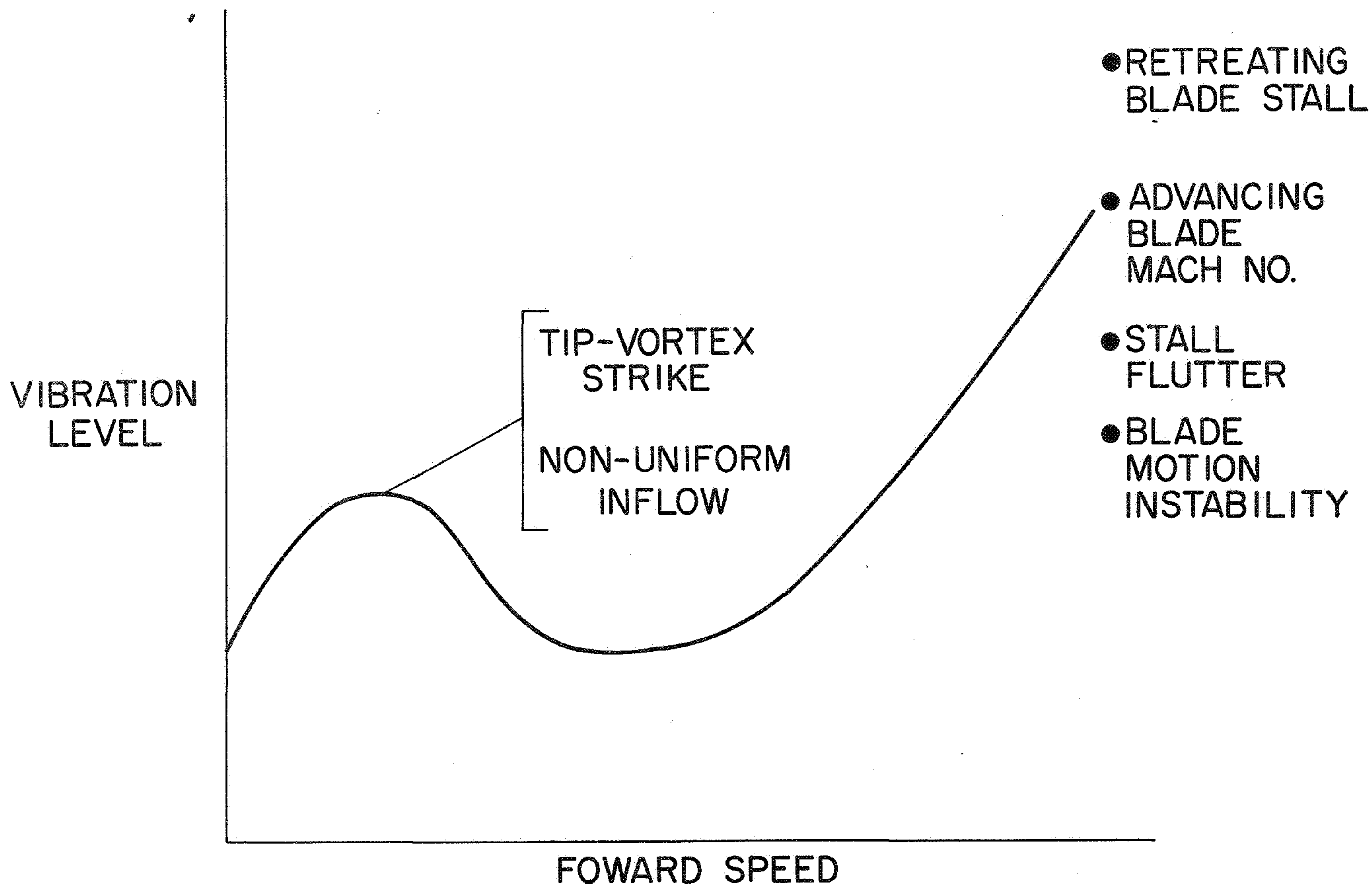

Figure 17.- Helicopter vibration levels vs forward speed.

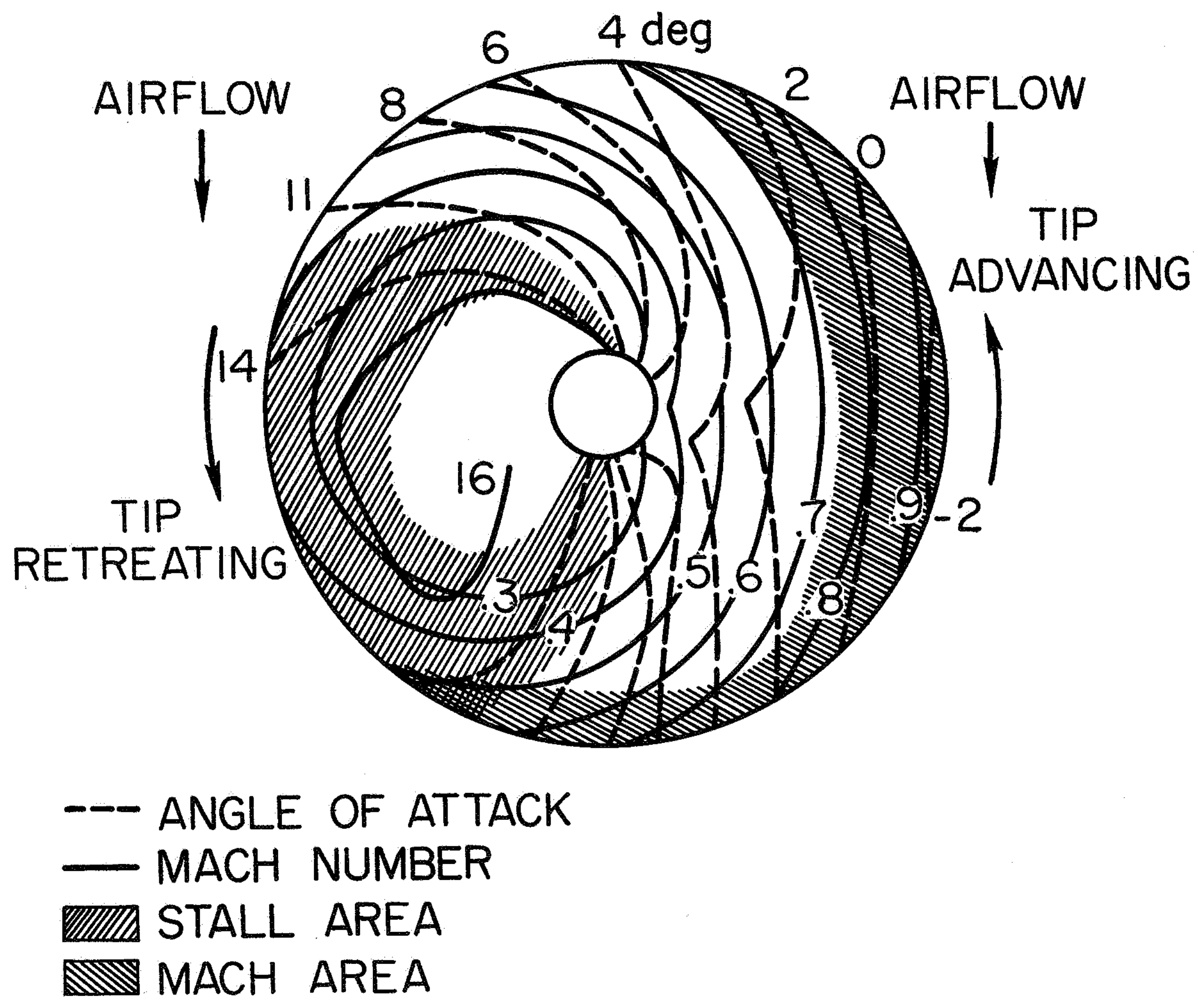

Figure 18.- Typical flowfield for high speed helicopter.