[5] Donnell, L. H., "Stability of thin-walled tubes under torsion," NACA Rept. 479, pp. 95–115 (1934).

[6] Flugge, W., *Stresses in Shells* (Springer-Verlag, Berlin, 1962).

[7] Green, A. and Zerna, W., *Theoretical Elasticity* (Oxford University Press, London, 1954).

[8] Gerard, G. and Becker, H., "Handbook of structural stability, Part III—Buckling of curved plates and shells," NACA TN 3783 (August 1957).

[9] Hess, T. E., "Stability of orthotropic cylindrical shells under combined loading," ARS J. **31**, 237–246 (1961).

[10] Timoshenko, S. and Gere, J., *Theory of Elastic Stability* (McGraw-Hill Book Co. Inc., New York, 1961).

[11] Love, A. E., *A Treatise on the Mathematical Theory of Elasticity* (Dover Publications, New York, 1944).

[12] March, H. W., Norris, C. B., Smith, C. B., and Kuenzi, E. W., "Buckling of thin-walled plywood cylinders in torsion," Forest Products Lab. Rept. 1529 (June 1945).

[13] March, H. W., "Buckling of long, thin plywood cylinders in axial compression," Forest Products Lab. Rept. 1322-A (March 1956).

[14] Novozhilov, V. V., *The Theory of Thin Shells* (P. Noordhoff Ltd., Groningen, The Netherlands, 1959).

[15] Von Kármán, T. and Tsien, H. S., "The buckling of thin cylindrical shells under axial compression," J. Aeronaut. Sci. **8**, 303–312 (1941).

[16] Ho, B. P. C. and Cheng, S., "Some problems in stability of heterogeneous aeolotropic cylindrical shells under combined loading," AIAA J. (to be published).

APRIL 1963 AIAA JOURNAL VOL. 1, NO. 4

Some Recent Contributions to Panel Flutter Research

Y. C. Fung*

California Institute of Technology, Pasadena, Calif.

With the objective of formulating a realistic computing program to analyze panel flutter in aerospace vehicles, plausible simplifying assumptions are examined in the light of experimental results. It is shown that in certain areas very simple analysis yields respectable results, whereas in other areas great elaboration is necessary to obtain an accurate prediction. In particular, the role played by the boundary layer flow is discussed. The attenuation and phase shift in pressure-deflection relationship caused by the boundary layer can become important under certain circumstances. Examples are given which show that the boundary layer greatly stabilizes flat plates in a transonic or low supersonic flow and circular cylindrical shells at higher Mach numbers. Some recent contributions to panel flutter research by the author and his colleagues and students at the California Institute of Technology are summarized. Although details are to be published elsewhere, a brief description of experimental results concerning flat plates and cylindrical shells is given here. The experimental and theoretical investigations taken together provide a fairly clear picture with regard to proper assumptions for an accurate analysis. Recommendations for future research in this field are given.

Nomenclature

A = $\rho U^2L^3/MD$, ratio of dynamic pressure to panel rigidity $= \pi^4$ (Q of Ref. 1)

A_ν = coefficients of Fourier series of $z_0(x,t)$, Eq. (9)

a_m = $m = 1, 2, \ldots$, coefficients of sine series of $z_0(x,t)$, Eqs. (29) and (30)

a, a_δ = velocity of sound, in main flow and boundary layer, respectively

B_ν = coefficients of Fourier series of $z_1(x,t)$, Eq. (10)

C_ν, D_ν, E_ν = coefficients, see Eqs. (12) and (13)

D = $Eh^3/[12(1 - \mu^2)]$, bending rigidity of plate

f = frequency, cps

g = structural damping factor

h = thickness of plate or shell wall

k = $\omega L/U$, reduced frequency in main flow

k_δ = $\omega L/U_\delta$, reduced frequency in boundary layer

L = chord length

M, M_δ = Mach number of main flow and of boundary layer, respectively

n = number of waves along circumference (number of nodes $= 2n$)

Δp = see Eq. (33)

$p(x,t)$ = wall pressure

p, p_δ = static pressure in freestream and in boundary layer, respectively

p_m = excess of model internal pressure above p_δ, psig

p_t = wind tunnel stagnation pressure

$p_0(x,t)$ = wall pressure in potential flow without boundary layer

q = $\frac{1}{2}\rho U^2$, dynamic pressure of main flow

R = radius of middle surface of circular cylinder

r,θ,x = cylindrical polar coordinates

T,T_δ = absolute temperature in freestream and in boundary layer, respectively

---

Presented at the IAS 31st Annual Meeting, New York, January 21–23, 1963; revision received March 4, 1963. This research was supported by the U. S. Air Force through the Office of Scientific Research, Office of Aerospace Research. Howard Wolko and Joseph Long of the Office of Scientific Research provided much help to the author. The experimental work at NASA Ames Research Center was carried out with the help and guidance of Lloyd Jones and Ralph Huntsberger; to them and to the staff of the Unitary Plan wind tunnel, the author is sincerely grateful. The Guggenheim Aeronautical Laboratory, California Institute of Technology team whose work is reported here consists of Hans Krumhaar, William J. Anderson, Ronald O. Stearman, Malcolm Lock, Benjamin Cummings, George Watts, and J. Stuart Keith; to their contributions, individual references are made in this paper. Ernest E. Sechler provided much inspiration and guidance to experimentation. Louis Schmidt and Richard Luntz offered tireless help during the wind tunnel tests. Above all, Marvin Jessey designed and made many of the instrumentation components; to his patience, ability, and accuracy the author owes much for any success of his research. Dorothy Eaton did most of the numerical calculations. Helen Burrus and Jeanette Siefke prepared the manuscript. To all these people and to many others not here mentioned, the author wishes to express his sincere appreciation and thanks.

* Professor of Aeronautics.

t = time
U = velocity of freestream
u,v = velocity components in x,y directions
V_ν = velocity of traveling waves, see Eq. (9)
$w(x,r,t)$, w_0 = radial velocity on wall and amplitude, respectively
w_{rms} = root mean square value of the deflection (radial or vertical) of an oscillation shell or plate
x,y = rectangular Cartesian coordinates; x in flow direction
z_0 = constant
$z_0(x,y)$ = wall displacement
$z_1(x,y)$ = displacement of the edge of boundary layer
α_ν = $\nu\pi/L$, wave number, see Eqs. (9–11)
$[\alpha_1]_{mn}$, $[\alpha_2]_{mn}$ = see Eq. (36)
β_δ = $[1 - M_\delta^2]^{1/2}$
$\bar{\beta}$ = $[M^2 - 1]^{1/2}$
γ_ν = constants, see Eqs. (14) and (23)
δ = idealized boundary layer thickness
$\bar{\delta}$ = apparent boundary layer thickness (wall to 99% freestream-velocity point)
ζ_ν = constants, see Eqs. (15) and (23)
κ_ν = $\zeta_\nu\delta$, boundary layer thickness parameter, see Eq. (18)
μ = Poisson's ratio
ν = $\pm 1, \pm 2, \ldots$, an index
ρ, ρ_δ = density in freestream and in boundary layer, respectively
σ_ν = constants, see Eqs. (19) and (23)
ϕ, ϕ_δ = velocity potentials in main flow and in boundary layer, respectively
ω = circular frequency
$\mathcal{B}_m$, $\mathcal{B}_{-m}$ = pressure coefficients, see Eqs. (31) and (32)
$\mathfrak{M}(\kappa_\nu)$ = ratio of wall pressure with and without boundary layer, see Eqs. (20) and (21)

5. Role of Boundary Layer Flow

The boundary layer flow over a solid body has important influence in many phenomena in aeroelasticity. In buffeting or stall flutter, the boundary layer may detach from the solid wall, causing complete change in flow pattern. In panel flutter, the boundary layer causes changes in amplitude and phase relationship between pressure and wall displacement.

To formulate a simple idealized problem, which sheds some light on the role of boundary layer in panel flutter, consider the following: an infinite flat plate oscillates harmonically in a standing sine wave with straight nodal lines perpendicular to the flow. The unperturbed flow in the half-space above the plate is a uniform supersonic flow of Mach number M; in between the supersonic flow and the plate is a layer of parallel uniform subsonic flow of constant thickness δ and Mach number M_δ. The interface between the supersonic flow and the subsonic layer is a vortex sheet of constant strength. Assume that the amplitude of oscillation of the plate is small compared with the thickness of the subsonic layer δ and that the perturbed flows in both the subsonic layer and the supersonic half-space are isentropic and irrotational. The problem is to relate the pressure distribution on the plate with the surface displacement.

Although the idealization just named is so severe that the conditions are unrealizable, it does preserve two features that are important to the problem. First, the entire flow outside of the boundary layer is influenced by the oscillation of the wall; second, the pressure across the boundary layer does not remain constant but is variable throughout the thickness.

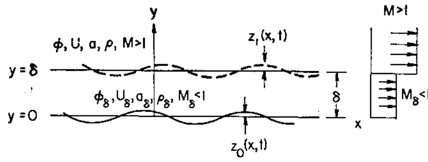

Fig. 13 Notations for an idealized boundary layer

Let (u,v) be the perturbation velocity and ϕ be the perturbation velocity potential, so that $\nabla\phi = (u, v)$ and $(u^2 + v^2)/U^2 \ll 1$. The uniform main flow velocity, speed of sound, fluid density, and Mach number will be denoted by U, a, ρ, M, respectively, and the corresponding quantities in the subsonic layer will be indicated with a subscript δ (see Fig. 13). The linearized equations of motion are (see Ref. 26, pp. 419, 432)

$$\frac{1}{a^2}\frac{\partial^2\phi}{\partial t^2} + \frac{2M}{a}\frac{\partial^2\phi}{\partial x\partial t} + \bar{\beta}^2\frac{\partial^2\phi}{\partial x^2} = \frac{\partial^2\phi}{\partial y^2} \qquad (y \geqslant \delta) \quad (1)$$

$$\frac{1}{a_\delta^2}\frac{\partial^2\phi_\delta}{\partial t^2} + \frac{2M_\delta}{a_\delta}\frac{\partial^2\phi_\delta}{\partial x\partial t} - \beta_\delta^2\frac{\partial^2\phi_\delta}{\partial x^2} = \frac{\partial^2\phi_\delta}{\partial y^2} \quad (2)$$

$$\text{for} \quad (0 \leqslant y \leqslant \delta)$$

$$\bar{\beta}^2 = M^2 - 1 > 0 \qquad \beta_\delta^2 = 1 - M_\delta^2 > 0$$

The boundary conditions are $(-\infty < x < \infty)$, $(-\infty < t < \infty)$

$$y = 0: \quad \frac{\partial\phi_\delta}{\partial y} = \frac{\partial z_0}{\partial t} + U_\delta\frac{\partial z_0}{\partial x} \quad (3)$$

$$y = \delta - 0: \quad \frac{\partial\phi_\delta}{\partial y} = \frac{\partial z_1}{\partial t} + U_\delta\frac{\partial z_1}{\partial x} \quad (4)$$

$$y = \delta + 0: \quad \frac{\partial\phi}{\partial y} = \frac{\partial z_1}{\partial t} + U\frac{\partial z_1}{\partial x} \quad (5)$$

$$y = \delta: \quad p(x,\delta^{-0},t) = p(x,\delta^{+0},t) \quad (6)$$

i.e.,

$$-\rho_\delta\left(\frac{\partial\phi_\delta}{\partial t} + U_\delta\frac{\partial\phi_\delta}{\partial x}\right) = -\rho\left(\frac{\partial\phi}{\partial t} + U\frac{\partial\phi}{\partial x}\right) \text{ at } y = \delta$$

$$y \to \infty: \text{ finiteness and radiation condition} \quad (7)$$

Consider a standing wave on the wall:

$$z_0(x, t) = e^{i\omega t}\sum_{m=1}^{\infty} a_m \sin\frac{m\pi x}{L} \quad (8)$$

In the process it will be shown that the perturbed oscillation of the boundary layer [interface $z_1(x, t)$] consists of not only a standing wave but also a traveling wave. The traveling wave becomes more important to the stability of the panel as the supersonic Mach number is reduced ($M \to 1$). Thus a surprising reconciliation between the "standing wave theory of panel flutter" and the "traveling wave theory of panel flutter" (Miles[27]) is obtained. Write

$$z_0(x, t) = \sum_{\nu=-\infty}^{\infty} A_\nu e^{i(\omega t + \alpha_\nu x)} = \sum_{\nu=-\infty}^{\infty} A_\nu e^{i\alpha_\nu(x - V_\nu t)} \quad (9)$$

$$z_1(x, t) = \sum_{\nu=-\infty}^{\infty} B_\nu e^{i(\omega t + \alpha_\nu x)} = \sum_{\nu=-\infty}^{\infty} B_\nu e^{i\alpha_\nu(x - V_\nu t)} \quad (10)$$

where

$$\alpha_\nu = \nu\pi/L \qquad V_\nu = -\omega/\alpha_\nu \quad (11)$$

It is easy to see that the differential Eqs. (1) and (2) have the solutions

$$\phi = \sum_{\nu=-\infty}^{\infty} E_\nu e^{i(\omega t + \alpha_\nu x - \gamma_\nu y')} \qquad (y' = y - \delta) \quad (12)$$

$$\phi_\delta = \sum_{\nu=-\infty}^{\infty} [C_\nu \sin\zeta_\nu y + D_\nu \cos\zeta_\nu y] e^{i(\omega t + \alpha_\nu x)} \tag{13}$$

where

$$\gamma_\nu^2 = \frac{\omega^2}{a^2} + \frac{2M\omega\alpha_\nu}{a} + (M^2 - 1)\alpha_\nu^2$$

$$\gamma_\nu = \frac{\pi}{L}\left[M^2\left(\frac{k}{\pi} + \nu\right)^2 - \nu^2\right]^{1/2} \quad (\nu = \pm 1, \pm 2, \ldots) \tag{14}$$

$$\zeta_\nu^2 = \frac{\omega^2}{a_\delta^2} + \frac{2M_\delta\omega\alpha_\nu}{a_\delta} + (M_\delta^2 - 1)\alpha_\nu^2$$

$$\zeta_\nu = \frac{\pi}{L}\left[M_\delta^2\left(\frac{k_\delta}{\pi} + \nu\right)^2 - \nu^2\right]^{1/2} \quad (\nu = \pm 1, \pm 2, \ldots) \tag{15}$$

and

$$k = \omega L/U \qquad k_\delta = \omega L/U_\delta \tag{16}$$

are reduced frequencies.

As ν ranges over $-\infty$ to ∞, γ_ν^2, ζ_ν^2 can have both plus and minus signs. For γ_ν, the selection of branches of the multi-valued function must be based on the radiation and finiteness conditions at ∞. For ζ_ν, it is arbitrary. Choose γ_ν, ζ_ν to be real and positive if γ_ν^2, ζ_ν^2 were real and positive, and γ_ν, ζ_ν to be imaginary with complex argument $-\pi/2$ (on the lower half-plane) if γ_ν^2, ζ_ν^2 were real and negative. With this choice, $\phi(x, y; t)$ represents an outgoing wave in the y direction if $\gamma_\nu^2 > 0$ or decreases exponentially as $y \to \infty$ if $\gamma_\nu^2 < 0$. Note that, if ω were allowed to be complex, then a divergent wall oscillation, $Im\ \omega < 0$, will correspond to a complex-valued γ_ν with $Im\ \gamma_\nu < 0$. Then ϕ decreases with increasing y, as the radiation condition would imply.

The constants E_ν, C_ν, D_ν can be determined from the boundary conditions (3–5). Then condition (6) gives the desired result:

$$B_\nu/A_\nu = 1/(\cos\kappa_\nu - \sigma_\nu \sin\kappa_\nu) \tag{17}$$

where

$$\kappa_\nu = \zeta_\nu\delta \tag{18}$$

$$\sigma_\nu = -i\frac{\rho}{\rho_\delta}\frac{(\omega + U\alpha_\nu)^2}{(\omega + U_\delta\alpha_\nu)^2}\frac{\zeta_\nu}{\gamma_\nu} = -i\frac{\rho U^2}{\rho_\delta U_\delta^2}\left(\frac{(k/\pi) + \nu}{(k_\delta/\pi) + \nu}\right)^2\frac{\zeta_\nu}{\gamma_\nu} \tag{19}$$

Thus the problem is solved. The pressure on the wall is obtained:

$$p(x,0;t) = \rho\sum_{\nu=-\infty}^{\infty} i(\omega + U\alpha_\nu)^2\frac{A_\nu}{\gamma_\nu}\mathfrak{M}(\kappa_\nu)e^{i(\omega t + \alpha_\nu x)} \tag{20}$$

where

$$\mathfrak{M}(\kappa_\nu) = \frac{\cos\kappa_\nu + (1/\sigma_\nu)\sin\kappa_\nu}{\cos\kappa_\nu - \sigma_\nu \sin\kappa_\nu} \tag{21}$$

The function $\mathfrak{M}(\kappa_\nu)$ represents the influence of the boundary layer. Since $\mathfrak{M}(\kappa_\nu) \to 1$ as $\delta \to 0$, the expected solution of an oscillating wall in a uniform supersonic main flow is obtained:

$$p_0 = \text{wall pressure in potential flow}$$
$$= \rho\sum_{\nu=-\infty}^{\infty} i(\omega + U\alpha_\nu)^2\frac{A_\nu}{\gamma_\nu}e^{i(\omega t + \alpha_\nu x)} \tag{22}$$

To examine the nature of the solution, it is sufficient to consider $\nu = 1$ and -1, corresponding to traveling waves of wave length $2L$ up- and downstream, respectively; other values of ν merely represent waves of shorter wave length

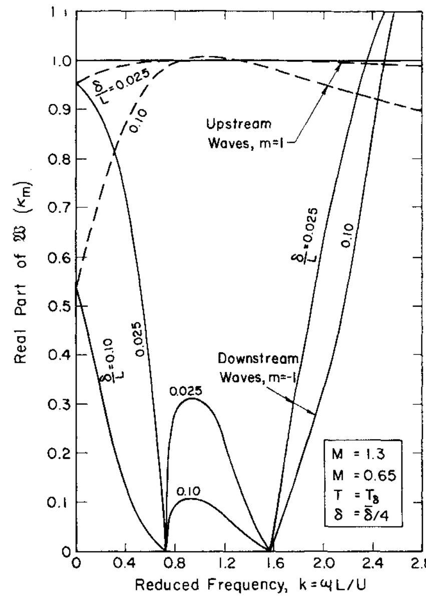

Fig. 14a Real part of the function $\mathfrak{M}(\kappa_\nu)$, ratio of aerodynamic pressure on wall with and without boundary layer; traveling waves; $M = 1.3$, $M_\delta = 0.65$

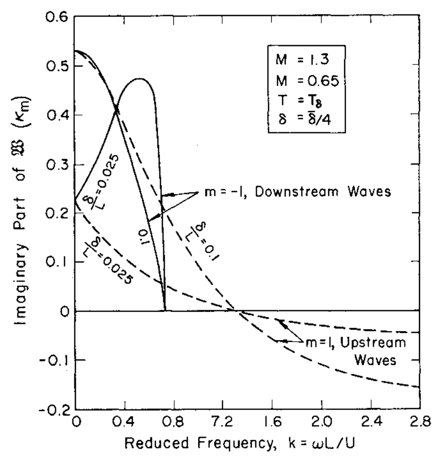

Fig. 14b Imaginary part of the function $\mathfrak{M}(\kappa_\nu)$, ratio of aerodynamic pressure on wall with and without boundary layer; traveling waves; $M = 1.3$, $M_\delta = 0.65$

$2L/|\nu|$. As traveling waves, the individual terms can be understood better in terms of the phase velocity V_ν relative to the panel [see Eqs. (9–11)]. Thus

$$\gamma_\nu = (|\alpha_\nu|/a)[(U - V_\nu)^2 - a^2]^{1/2}$$
$$\zeta_\nu = (|\alpha_\nu|/a_\delta)[(U_\delta - V_\nu)^2 - a_\delta^2]^{1/2} \tag{23}$$
$$\sigma_\nu = -i\frac{\rho U^2}{\rho_\delta U_\delta^2}\left(\frac{U - V_\nu}{U_\delta - V_\nu}\right)^2\frac{\zeta_\nu}{\gamma_\nu}$$
$$p(x,0;t) = \rho\sum_\nu i\alpha_\nu(U - V_\nu)^2\frac{A_\nu}{\gamma_\nu}\mathfrak{M}(\kappa_\nu)e^{i\alpha_\nu(x - V_\nu t)}$$

It is seen that γ_ν, ζ_ν vanish when, respectively,

$$U - V_\nu = \pm a \qquad U_\delta - V_\nu = \pm a_\delta \tag{24}$$

Further, σ_ν becomes either zero or indeterminate if

$$U = V_\nu \qquad U_\delta = V_\nu \tag{25}$$

These correspond to steady transonic flows or static conditions, respectively, with respect to an observer moving with the traveling wave. In the former case, Eq. (24), the approximation is not valid.

The influence of boundary layer is exhibited by the function $\mathfrak{M}(\kappa_\nu)$, which is the ratio of pressures in flows with and without boundary layer. If one sets $T = T_\delta$, so that $p = p_\delta$, $\rho = \rho_\delta$, $a = a_\delta$ for the undisturbed flow, then $\mathfrak{M}(\kappa_\delta)$ depends on the parameters M, M_δ, δ/L, and k.

The behavior of $\mathfrak{M}(\kappa\nu)$ is exhibited numerically in Fig. 14, which is very instructive in showing that a wide variation in $\mathfrak{M}$ is possible. On the other hand, a standing wave of the wall,

$$z_0(x,t) = z_0\, e^{i\omega t} \sin(\pi x/L) \tag{26}$$

corresponds to

$$A_{\pm 1} = \pm z_0/2i \qquad \alpha_{\pm 1} = \pm \pi/L \tag{27}$$

$$z_1(x,t) = \frac{1}{\cos\kappa_1 - \sigma_1 \sin\kappa_1}\, z_0 e^{i\omega t} \sin\frac{\pi x}{L} - \left(\frac{1}{\cos\kappa_{-1} - \sigma_{-1}\sin\kappa_{-1}} - \frac{1}{\cos\kappa_1 - \sigma_1 \sin\kappa_1}\right)\frac{z_0}{2i}\, e^{i[\omega t - (\pi x/L)]} \tag{28}$$

This shows clearly that the edge of the boundary layer appears as an attenuated wall oscillation plus a traveling wave.

The foregoing results cannot be used directly in analyzing panel flutter of finite panels, because the influence of a leading edge is not clarified. It is known that for $1 < M < 1.4$ the potential theory of supersonic flow shows a strong leading edge effect (no disturbance in front of the leading edge). For the idealized boundary layer (uniform subsonic flow in $0 < y < \delta$), the writer has worked out a complete solution for an arbitrary oscillation of a finite wall, but the results are complicated. For a qualitative examination, two simple alternatives are suggested. The first ignores the leading edge effect, treating a finite panel as one period of an infinite wall. Thus, if

$$\begin{aligned} z_0(x,t) &= 0 && \text{for } x < 0 \\ &= e^{i\omega t} \sum_{m=1}^{\infty} a_m \sin\frac{m\pi x}{L} && \text{for } x \geqslant 0 \end{aligned} \tag{29}$$

it is assumed that the wall pressure $p(x,t)$ is the same as that induced by

$$z_0(x,t) = e^{i\omega t} \sum_{m=1}^{\infty} a_m \sin\frac{m\pi x}{L} \quad \text{for } -\infty < x < \infty \tag{30}$$

Hence, from Eq. (20), one has

$$p(x,0;t) = \frac{\rho U^2}{2L^2}\, e^{i\omega t} \sum_{m=1}^{\infty} (\mathfrak{B}_m e^{im\pi x/L} - \mathfrak{B}_{-m} e^{-im\pi x/L}) a_m \tag{31}$$

where

$$\mathfrak{B}_m = \frac{(k + m\pi)^2}{\gamma_m}\, \mathfrak{M}(\kappa_m) \qquad \mathfrak{B}_{-m} = \frac{(k - m\pi)^2}{\gamma_{-m}}\, \mathfrak{M}(\kappa_{-m}) \tag{32}$$

The second alternative assumes that the change of aerodynamic pressure due to boundary layer on a finite panel is the same as that on an infinite wall with the same wave form repeated periodically. Thus, if $z_0(x,t)$ is given by (29), one assumes that

$$\text{pressure on wall} = p_0 + \Delta p \tag{33}$$

where p_0 is the wall pressure corresponding to (29) in a potential flow without a boundary layer, and Δp is the difference of p from Eqs. (20) and (22). The function $p_0(x,t)$ is known (see Miles[29]). Simplifications are discussed by Luke[30] and Lock and Fung.[14] The function Δp is

$$\Delta p = \frac{\rho U^2}{2L^2}\, e^{i\omega t} \sum_{m=1}^{\infty} \left\{ \left[\mathfrak{B}_m - \frac{(k + m\pi)^2}{\gamma_m}\right] e^{im\pi x/L} - \left[\mathfrak{B}_{-m} - \frac{(k - m\pi)^2}{\gamma_{-m}}\right] e^{-im\pi x/L} \right\} a_m \tag{34}$$

The first alternative is used in the flutter calculations to be discussed in the next section, in which the analysis is extended to circular cylinders.

It is hoped that such a simple analysis can be supplemented by a comparison of the final results with those obtained in more exact theories. A theoretical-empirical scheme of fixing δ, M_δ, T_δ might be evolved which could be sufficiently accurate for practical purposes. This, however, has not been done yet. Two improved theories have been published so far. One, due to Miles,[31] considers the boundary layer as an inviscid, parallel shear flow over an infinite, plane panel. The other, due to McClure,[32] treats the full problem in the Heisenberg, Tollmien, Lin, Lees, Lighthill tradition, extending the boundary layer problem to oscillating walls. Both Miles and McClure applied their theories to panel flutter. Miles[31] showed that, for a circular cylinder, the stability boundary of short-wavelength traveling waves does not change much on account of the shear layer, but the rate of divergence in the unstable regime may be reduced by an order of magnitude. McClure's solution of the transonic flutter of a flat plate is truly remarkable. In application to the Lock-Fung experiment, McClure obtained the stability boundary as shown in Fig. 3, which is rather close to the experimental value. However, it must be remembered that McClure ignores the leading edge effect in the manner of Eq. (31). How the leading edge effect would influence McClure's stability boundary is yet unknown.

Miles and McClure's analyses are, of course, much more complicated than what was presented here. There are still weakness and difficulties that make an extension of their solutions to an arbitrary wall oscillation impossible. Fortunately, the flexible wall problem has attracted much attention among aero- and hydrodynamicists recently, owing to a great debate about the possibility of reducing the drag of a body in a flow by elastic walls. The names of Kramer, Benjamin, Landahl, Becker, and Laufer are becoming well known in this newly discovered field. It is expected that the matter will be settled before long and with it the aerodynamic aspects on panel flutter, but the matter will not be discussed further here.