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SEPT.-OCT. J. AIRCRAFT VOL. 3, NO. 5

Some Experimental Contributions on Single Degree-of-Freedom Flutter in Two-Dimensional Low Supersonic Flow

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Two experimental investigations on the low supersonic instabilities were conducted in transonic wind tunnels. Firstly, the results are presented of the measurement of the unsteady pitching moment of a two-dimensional airfoil performing pitching oscillations in low supersonic flow. Secondly, a rotational motion of a flap has been investigated with a two-dimensional airfoil-flap combination model. A simple free oscillation method was adopted in the test in order to obtain the unsteady aerodynamic coefficients for oscillations at small amplitudes. Optical observations were made of the flowfield around the airfoil-flap combination model by using a highspeed cine camera for the case where the flap was hinged freely by ball bearings at both ends. In the first experiment, pure pitching instability of the airfoil was observed with the appropriate axis positions for all the values of Mach number and reduced frequency that were obtained. Rotational instability of the flap was observed also in the second experiment at every Mach number of the test. It is concluded from the results that the inviscid flow over the surfaces of the airfoil would be responsible primarily for the onset of these instabilities.

Nomenclature

I = moment of inertia of the system, kg-m-sec^2
$\bar{H}$ = unsteady aerodynamic hinge moment, $\bar{H} = \rho_\infty v_\infty^2 c_F^2 l(h_\beta \beta + c_F/v_\infty\, h_{\dot\beta}\dot\beta)$
$\bar{M}$ = unsteady aerodynamic pitching moment $\bar{M} = \rho_\infty v_\infty^2 c^2 l(m_\theta \theta + c/v_\infty\, m_{\dot\theta}\dot\theta)$
M_∞ = freestream Mach number
c = airfoil chord
c_F = flap chord
d = thickness of the metal strips used for cross spring pivots mm
f_0 = still air frequency of the system, cps
f = frequency of flap oscillation, cps
g_θ, g_β = structural damping of the system
h_β = aerodynamic stiffness coefficient of the unsteady hinge moment
$-h_{\dot\beta}$ = aerodynamic damping coefficient of the unsteady hinge moment
k = reduced frequency, $k = \omega c/2v_\infty$ or $\omega c_F/2v_\infty$
k_θ = torsional rigidity of the cross spring pivots, kg·m/rad
l = airfoil span
m_θ = aerodynamic stiffness coefficient of the unsteady pitching moment
$-m_{\dot\theta}$ = aerodynamic damping coefficient of the unsteady pitching moment
t = time
v_∞ = air velocity
x_0 = nondimensional distance along the chord from the leading edge
β = flap angle or its amplitude of oscillation
θ = pitching angle or its amplitude of oscillation
ρ_∞ = air density
ω = circular frequency, rad/sec
$\bar\omega$ = $2kM_\infty^2/(M_\infty^2 - 1)$
$\cdot$ = time derivative

Presented at the AIAA Symposium on Structural Dynamics and Aeroelasticity, Cambridge, Mass., August 30–September 1, 1965; submitted October 18, 1965; revision received March 14, 1966. The authors would like to express their cordial thanks to R. Kawamura and K. Washizu, of the University of Tokyo, for their instructive advice during the course of this study. They are also very grateful to N. C. Lambourne and H. C. Garner of the Aerodynamics Division, National Physical Laboratory, United Kingdom, who read the manuscript and offered valuable comments and criticism.

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Introduction

ACCORDING to the linearized unsteady theory for two-dimensional transonic potential flow, the possibility of single degree-of-freedom flutter for a range of small values of reduced frequency is shown with both the pitching motion of an airfoil and with the rotational motion of a flap.[1–3]

However, there is a need for some experimental verification to support the foregoing theoretical statement since, in the

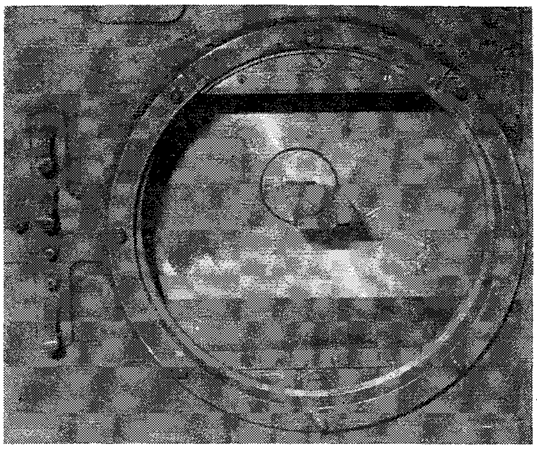

Fig. 1 The model in the working section of 18-cm sq transonic wind tunnel.

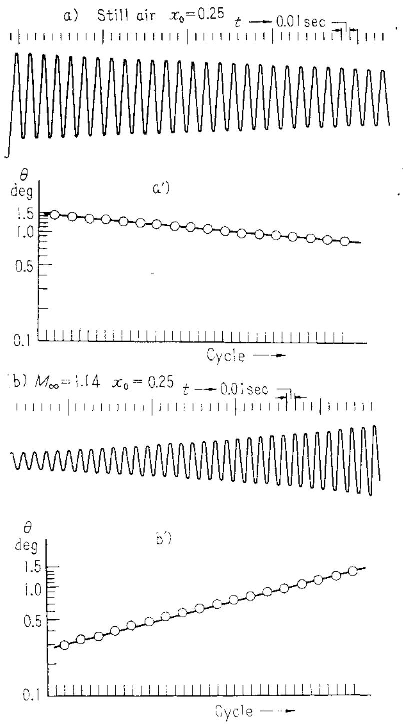

Fig. 3 Original oscillograms and the variations of amplitude of pitching oscillation with cycle.

transonic range, the so-called nonlinear accumulation of the disturbance might invalidate the linearization of the flowfield, and the statement does not seem to be self-evident. There is, on the other hand, very little experimental work on two-dimensional unsteady transonic flow. This is possibly because of the difficulties that are encountered in dynamic measurement in a wind tunnel.[4]

The present report contains the results for the following two experimental investigations in transonic wind tunnels: the measurement of the unsteady pitching moment of a two-dimensional airfoil performing pitching oscillations in low supersonic flow, and the investigation on a self-excited oscillation of a control surface at low supersonic speeds with a two-dimensional airfoil-flap combination model. It is shown experimentally that single degree-of-freedom flutter can occur in two-dimensional low supersonic potential flow.

Apparatus and Method

The test for the pitching measurement was carried out at the National Aerospace Laboratory (NAL) 18-cm square continuous transonic wind tunnel, the working section of which was equipped with the perforated walls having a 20% open area ratio at the top and bottom, with solid walls at the vertical sides. Figure 1 shows both the working section of the tunnel and the model which were used for this test. The model was a two-dimensional steel airfoil having NACA 64A010 section (Table 1). In order to maintain two-dimensional flow, a pair of circular disks 6 cm in diameter were attached to each end of the model. The gaps between these disks and the side walls were about 1 mm. The model was mass-balanced with respect to the axis of rotation, and was supported by the cross spring pivots at both ends, the center of which coincided with the axis position.

A free oscillation method was adopted for measuring the unsteady pitching moment. An amplitude limiter was attached to the cross spring pivots, so that for the divergent oscillations the pitching angle could not exceed 2°. For the damping oscillations, the initial deflection as large as 2° was given by pulling the ends of the cross spring pivots by a thin piano wire of 0.2 mm in diameter from outside of the working section; and the oscillation was started by suddenly cutting the wire. The pitching angle of the model was recorded on the oscillograph as an electric signal that was transmitted by the strain gages bonded on the cross spring pivots. The aerodynamic pitching moment was obtained with usual procedures by reading the frequency and the decrement (or increment) of the amplitude of the oscillatory pitching angle, the details of which are not included in this paper.

The test was made for four values of the axis positions, i.e., 15, 25, 37.5, and 50% chord behind the leading edge. Three different values of still-air frequency were obtained by

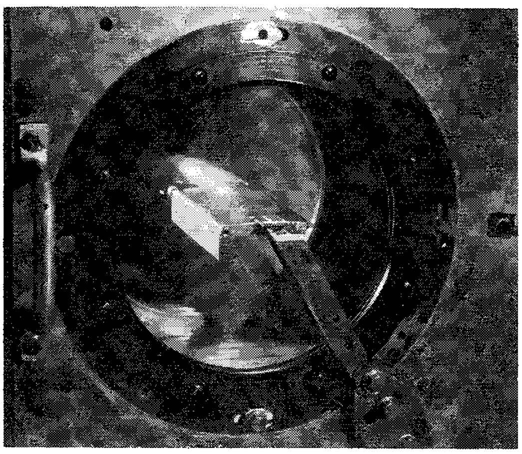

Fig. 2 The model with freely hinged flap in the working section of a 20-cm × 12-cm transonic wind tunnel.

Table 1 Geometrical and dynamical properties of the model for measuring the unsteady hinge moment[a]

Model no.	x_0	f_0 cps	d mm	I kg-m-sec²	k_θ kg-m/rad	g_θ
A 1	0.15	36.5	1.0	5.225×10^{-5}	2.75	0.00383
A 2	0.25	47.4	1.0	3.098	2.75	0.00383
A 3	0.375	53.2	1.0	2.460	2.75	0.00606
B 1	0.15	48.4	1.25	5.436	5.03	0.00419
B 2	0.25	62.6	1.25	3.249	5.03	0.00428
B 3	0.375	70.6	1.25	2.555	5.03	0.00976
B 4	0.50	59.3	1.25	3.621	5.03	0.00745
C 1	0.15	67.9	1.5	5.216	9.50	0.00377
C 2	0.25	87.4	1.5	3.148	9.50	0.00287
C 3	0.375	99.0	1.5	2.454	9.50	0.00375
C 4	0.50	83.6	1.5	3.441	9.50	0.00433
D 4	0.50	103.2	1.75	3.558	14.97	0.00379

[a] Airfoil section is NACA 64A010. Thickness to chord ratio is 0.10. Position of maximum thickness is 0.40 chord. Airfoil chord is 5 cm. Airfoil span is 18 cm.

Table 2 Geometrical and dynamical properties of the model with freely hinged flaps

Airfoil section	NACA 64A010
Thickness to chord ratio	0.10
Position of maximum thickness	0.40 chord
Airfoil chord	8 cm
Flap chord	2 cm
Airfoil span	12 cm
Inner diameter of bearing	2 mm
Moment of inertia of flap about its hinge	
Steel flap	2.565×10^{-7} kg-m-sec^2
Duralmin flap	0.910×10^{-7} kg-m-sec^2

adopting three kinds of metal strips with different thickness for the cross spring pivots (Table 1).

For the investigation of the flap oscillation, the 20-cm × 12-cm rectangular intermittent transonic wind tunnel at the Institute of the Aeronautical and Space Science, University of Tokyo was used. The top and bottom walls of the working section were slotted and with 10% open area ratio, while the side walls were solid. The working section expanded by 0.6° on each slotted wall to allow for the growth of boundary layer. As is shown in Fig. 2, a two-dimensional airfoil-flap combination model was used. The steel airfoil was clamped to the side walls at both ends by use of steel arms and set at zero incidence.

For optical observation, the flap was hinged freely about the axis by small steel ball bearings at both ends, which were fixed in the glass windows. Two flaps of the same section, respectively, made of steel and duralmin, were adopted for this experiment to find the effects of frequency parameter (Table 2). The experiment was concerned with the analysis of the high-speed schlieren films for the transient motion of the flap, which were taken by using a high-speed ciné camera immediately after the quick-acting valve of the tunnel was opened, and included the initiation and the growth of the instability. Although the flap was hinged freely, the oscillation, in most cases, started after the steady flow was established in the working section.

The measurement of the unsteady hinge moment was performed also by the same free oscillation method as adopted in the pitching measurement. Table 3 shows the dimensions of the model for the hinge moment measurement.

Reynolds numbers based on the airfoil chord length were small for both the experiments (0.8×10^6 and 1.2×10^6, respectively), so that the trip wires were cemented on both surfaces of the airfoils in order to fix the boundary-layer transition. In the first experiment they were located at 8% chord behind the leading edge, and at 5% chord in the second experiment. Their diameters were 0.3 mm and 0.2 mm, respectively.

Measurement of the Unsteady Pitching Moment of a Two-Dimensional Airfoil

The test was made at zero mean incidence, and all the data were close to 1° for the amplitude. The tested Mach number range was from 1.03 to 1.20, approximately. The value of reduced frequency was varied from approximately $k = 0.02$ to $k = 0.04$. Typical oscillograms and the corresponding curves showing the logarithmic decay or growth of amplitude with the cycle of oscillation in still-air and in freestream are

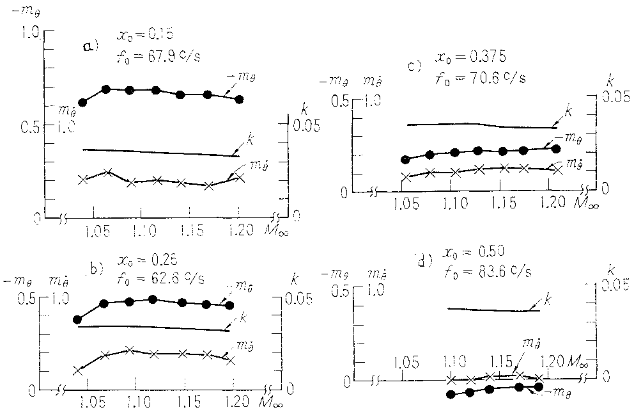

Fig. 4 The aerodynamic pitching-moment coefficients vs Mach number for approximately constant values of reduced frequency at various axis positions.

presented in Figs. 3a, 3á, 3b, and 3b′, respectively. The frequency and the rate of growth of amplitude of oscillation did not change with amplitude at least up to 1° for all the cases, but at the larger amplitudes there were cases in which the changes in rate of growth of amplitude were observed.

In Figs. 4a, 4b, 4c, and 4d, the experimental curves showing the variations of the aerodynamic stiffness and damping coefficients of pitching moment with Mach number for four different axis positions are presented. The variations of the same coefficients with Mach number for three different values of still-air frequency of the system for the axis position at 37.5% chord behind the leading edge are shown in Figs. 5a, 5b, and 5c.

The values of the two coefficients were essentially constant over the range of Mach number, as is seen in Fig. 4. The aerodynamic stiffness coefficients were negative for all the axis positions except for the one at 50% chord behind the leading edge. At the same time, in Figs. 4a, 4b, and 4c, the values of the aerodynamic damping coefficient were found to be negative at all the tested Mach numbers. This indicates that, in most cases where the airload was greater than the structural damping of the system, spontaneous flutter of pure pitching mode occurred. This is seen more directly from the oscillogram in Fig. 3b. Figure 5 also shows that both the co-

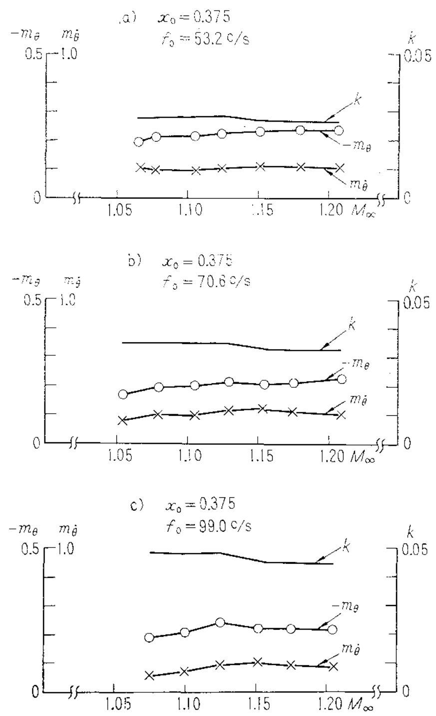

Fig. 5 The aerodynamic pitching-moment coefficients vs Mach number for three values of still-air frequency at 37.5% chord axis.

Table 3 Dynamical properties of the model for measuring the unsteady hinge moment

Model no.	f_0 cps	I kg-m-sec^2	g_β
1	52.5	6.242×10^{-7}	0.0065
2	76.0	6.242×10^{-7}	0.0071
3	109.4	6.242×10^{-7}	0.0097

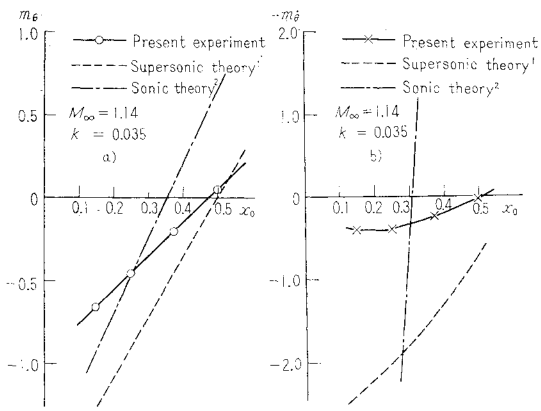

Fig. 6 The variations of the aerodynamic coefficients of pitching moment with axis position at M_∞ = 1.14 for k = 0.035.

efficients were nearly constant for three different values of reduced frequency at the same axis position.

Figs. 6a and 6b show the variations of both the coefficients with the axis position at approximately M_∞ = 1.14 and for k = 0.035. The experimental graph for the aerodynamic stiffness coefficient was straight, whereas that for the aerodynamic damping coefficient was parabolic. The calculated values corresponding to the linearized supersonic theory at M_∞ = 1.14 and the sonic theory for k = 0.035 are given also for comparison. These theories state that pure pitching flutter is possible over the wide ranges of Mach number and reduced frequency in two-dimensional low supersonic flow including M_∞ = 1. As is seen in Fig. 6, the quantitative agreement for both the coefficients between the present experiment and the supersonic theory is not good; the validity of the sonic theory for such small values of reduced frequency is rather doubtful.

However, schlieren films that were taken by use of a high-speed ciné camera showed that in this Mach number range, at least up to the amplitude of 2°, the flow over the surfaces of the airfoil was attached essentially with the shock waves remaining at the trailing edge during whole cycles of oscillation. This leads us to conclude from the present experimental results that spontaneous flutter of pure pitching mode is possible in two-dimensional low supersonic potential flow for a wide range of small values of reduced frequency with appropriate axis positions; and it can be guessed that the basic (steady) flow pattern over the surfaces of the airfoil, i.e., the variations of the local Mach number of the flow, should be responsible mainly for the large discrepancies between experiment and theories.

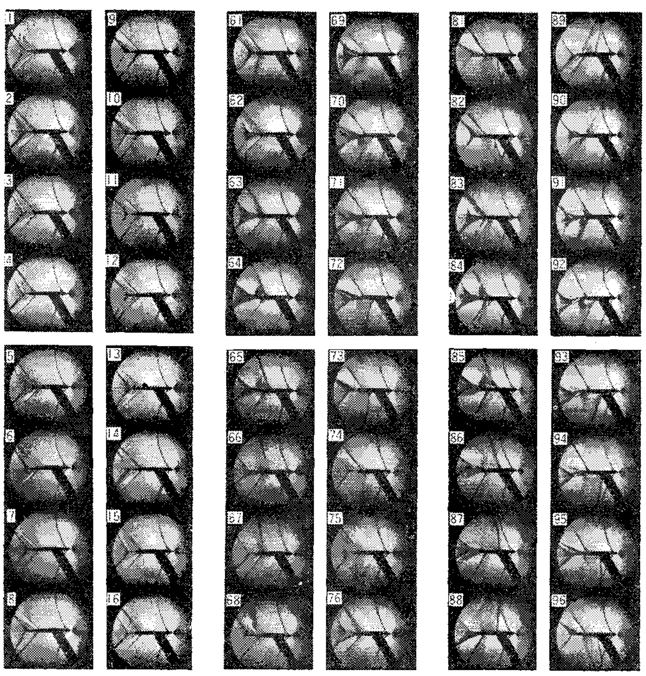

Fig. 7 High-speed schlieren films showing the growth of buzz oscillation: M_∞ = 1.18, incidence 0°, and film speed 3600 fps approximately; frames are numbered by frame number.

Self-Excited Rotation of a Control Surface

The instability treated in this section is a self-excited rotation of a control surface occurring at transonic speeds, which is commonly known as aileron buzz. Much work has been done already on these instabilities since 1945 when aileron vibration of a novel type was encountered in high subsonic flight. Recent investigation by N. C. Lambourne[5] studied the effects of Mach number, airfoil section, and incidence on buzz, and pointed out that more than one variety of buzz can occur in transonic flow.

However, even at the present stage of investigation, our knowledge about the mechanism of these instabilities is far from complete. The experimental evidence accumulated so far indicates that the buzz oscillations are featured by the strong effects of nonlinearity of the flow; that is, the motion of the flap cannot be divergent, but, at the final stage, it has a sustained oscillation of a finite amplitude, namely a limit cycle. Therefore, in order to investigate the initiation of the buzz, particular attention was paid in this experiment to the growth of the oscillation from its initiation up to the limit cycle, whereas all the previous researches were confined to a study of the limit cycle of the oscillation.

Experimental Results and Discussions

In this experiment, spontaneous flutter of the flap was observed at all supersonic Mach numbers that were obtained, i.e., M_∞ = 1.06 ~ 1.18. Figure 7 shows a typical example of the high-speed schlieren films of buzz for the model with a duralmin flap, the Mach number being 1.18 and the film speed approximately 3600 fps; Figs. 7a, 7b, and 7c, respectively, correspond to the three stages of the growth of the instability, the limit cycle being shown in 7c.

At small amplitudes the shock waves remained attached to the trailing edge and the supersonic flow was established over the whole surfaces of the flap. The Mach angle of the wave produced at the hinge was about 50° at zero flap angle, by which the Mach number of the flow immediately ahead of the hinge was calculated to be 1.30. Figure 7a indicates that the type of buzz in this Mach number range is featured by soft oscillator characteristics; that is, the oscillation always grows spontaneously from very small amplitude. It is concluded also that the negative damping of the motion of the flap is caused primarily by the characteristics of the inviscid flow, since the shock waves were attached to the trailing edge during the oscillation at small amplitudes, and the effect of the boundary layer is not considered to exist.

When the amplitude of oscillation reached approximately 8°, the waves at the hinge were strengthened and curved. With further increase of amplitude to about 15°, the waves began to move backwards and forwards over the surfaces of the airfoil. The severe separation of the boundary layer was observed then at the feet of these waves. The amplitude of the limit cycle was 29° (Figs. 7b and 7c). It should be mentioned that the effects of the top and bottom walls are insignificant during the small amplitude oscillations; whereas, at larger amplitudes, the shock waves reflected from the slotted walls and impinged again on the surfaces of the flap, as is seen in Fig. 7c.

The variations of amplitude and frequency of flap angle with cycle of oscillation are shown in Fig. 8. The amplitude increases exponentially with cycle at small amplitudes, and at about 8°, the rate of growth increases rapidly, and finally slows down. The frequency of oscillation, which is constant at small amplitudes, decreases abruptly at approximately 15°. The distortion and the initiation of the movement of the waves in the schlieren films correspond to the nonlinear behavior of the oscillation.

The variations of the aerodynamic stiffness and damping coefficients of the unsteady hinge moment with Mach number and with reduced frequency are shown in Figs. 9a and 9b. Reduced frequency is defined as $k = \omega c_F/2v_\infty$ in this section. White symbols indicate results obtained by the free oscillation method. Black symbols correspond to the results which were obtained from the analysis of the high-speed ciné films for the model with freely hinged flaps; the frictional torque of the bearings which hinged the flap was neglected in the calculation on the assumption that it would be much smaller than the aerodynamic loads. Agreement between these two experimental results is fairly good for the stiffness coefficients, whereas the free oscillation method gave slightly lower damping coefficients than those for the freely hinged flaps. The values of both coefficients were nearly constant over the Mach number range, primarily because of the Mach number freezing that is applied to the flow ahead of the hinge of the flap. Their variations with reduced frequency were also small for the range that was obtained in the present test.

The flow around the flap at low supersonic speeds, as was stated before, remained attached throughout the cycle of small oscillations. Under this condition, and since the section of the flap was wedge-shape, it can be stated that the linearization for the flowfield of the present experiment will be possible with a modified freestream Mach number; the value of reduced frequency may be small. In other words, the motion of the flap of the airfoil in uniform supersonic flow with Mach number M_∞ is identical to the pitching oscillation of a flat plate with zero thickness in supersonic potential flow where M_∞ is equal to the local Mach number of the flow immediately ahead of the hinge of the flap. The value of this local Mach number in the present experiment was approximately 1.30 through all the tested Mach numbers.

According to the unsteady linearized theory for two-dimensional supersonic flow, it is well known that a wide range of negative damping exists with the pitching oscillation of a flat plate for small values of reduced frequency.

In case of

$$\bar{\omega} = 2kM_\infty{}^2/(M_\infty{}^2 - 1) \ll 1 \quad (1)$$

the aerodynamic stiffness and damping coefficients of pitching moment about the leading edge are given by the approximate formula[1]:

$$-m_\theta = 1/(M_\infty{}^2 - 1)^{1/2} \quad (2)$$

$$-m_{\dot\theta} = \tfrac{2}{3}\cdot 1/(M_\infty{}^2 - 1)^{1/2}\cdot[2 - M_\infty{}^2/(M_\infty{}^2 - 1)] \quad (3)$$

It is seen readily that the right-hand term of Eq. (3) is negative for $1 < M_\infty < (2)^{\frac{1}{2}}$, which shows the possibility of the existence of pure pitching flutter in this range. It is interesting to compare the experimental results with the theoretical values that are given by Eqs. (2) and (3) and shown as the dashed lines in Figs. 9a and 9b. However, for reasons indicated, the theoretical values at $M_\infty = 1.30$ should be taken as a reference for the present experimental results. On this basis, the agreement of the aerodynamic damping coefficient between theory and experiment is fairly good, whereas the experimental values of the aerodynamic stiffness coefficient are somewhat lower than the theoretical one.

General Discussions

It has been confirmed now by the present experiment that the negative damping motions for the pitching mode of an airfoil and for the rotational mode of a flap in two-dimensional low supersonic potential flow are possible. Although the numerical agreement between the linear theory and the present experiment is poor for the pitching motion of the airfoil and fairly good for the rotational motion of the flap on the basis of the local Mach number, these instabilities are considered to be caused by the properties of the inviscid low supersonic flow.

Indeed, the negative damping moment for very slow oscillation of an airfoil in low supersonic flow can be deduced from Eq. (3). But an alternative explanation is found from the characteristic delay in the propagation of the pressure signal traveling downstream in the low supersonic flowfield. Although the flow speed is larger than, but as is close to that of the sound, the delay is considerably large. In the linear supersonic theory, the physically simple assumption that this delay is proportional to the distance from the leading edge is sufficient to establish that negative damping occurs. This is shown easily by obtaining the value of the unsteady pressure coefficient in Ref. 1 for the case where $\bar{\omega} \ll 1$ and $k \ll \bar{\omega}$.

In the same way, it also is expected theoretically that the damping moment will be changed from negative to positive as the Mach number becomes sufficiently high, so that k and $\bar{\omega}$ are the same order. However, owing to the limitation of the operation of the tunnels, the validity of the theoretical upper flutter boundary could not be checked in the present experiment. It would be interesting to obtain the experimental flutter boundary through further test and compare it with the theoretical one.

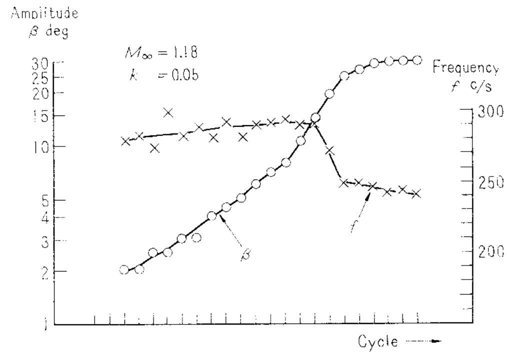

Fig. 8 The variation of the amplitude and frequency with cycle during the growth of buzz oscillation.

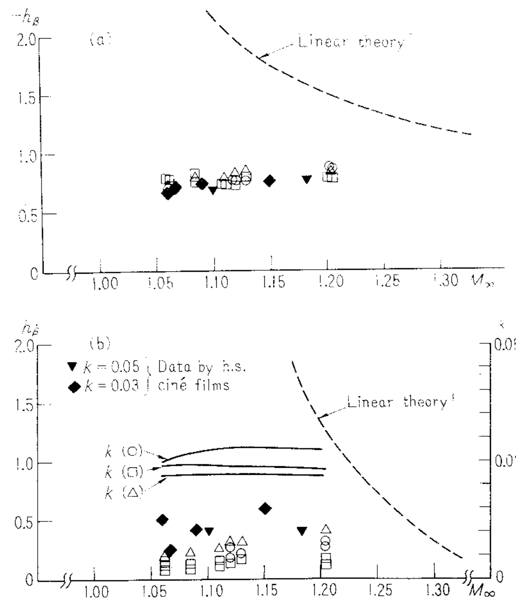

Fig. 9 The results of the measurement of the unsteady hinge moment of the flap at low supersonic speeds.

No corrections for the wall interferences were applied to the present experimental results. However, because of the large sizes of the models as compared with the heights of the working sections of the tunnels, various wall interferences would be considered to exist. The experiment should be repeated in a larger wind tunnel and with different slotted or perforated wall sections, and results should be compared with the present experimental results.

Conclusions

The measurement of the unsteady pitching moment for a two-dimensional airfoil was made at low supersonic speeds by use of a free oscillation method. The test was performed for four axis positions and for three values of reduced frequency at each Mach number. At low supersonic speeds spontaneous flutter of pure pitching mode was observed for axis positions forward of midchord. The values of the aerodynamic stiffness and damping coefficients were essentially independent of the changes of Mach number and reduced frequency ($1.03 \leq M_\infty \leq 1.20$ and $0.02 < k < 0.04$).

The instability of the flap rotation in low supersonic flow was studied with a two-dimensional airfoil-flap combination model. The test consisted of the optical observation of the growth of the self-excited oscillation of the flap by use of a high-speed ciné camera, and of the measurement of the unsteady hinge moment of the flap by a free oscillation method. The rotational instability of the flap also was found to be a spontaneous one, and at least at small amplitudes the flow around the airfoil remained essentially attached and neither the boundary layer nor the shock waves would be responsible for this instability. The measured unsteady hinge moment is compared with that calculated by linear theory; on the basis of the local Mach number, the numerical agreement for the aerodynamic damping coefficient between theory and experiment was fairly good, whereas the experimental values for the aerodynamic stiffness coefficient were somewhat lower than the calculated one.

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