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# TECHNICAL REPORT OF NATIONAL AEROSPACE LABORATORY

## TR-57T

## Measurements of the Aerodynamic Derivatives of an Oscillating Biconvex-Flat Airfoil in Supersonic Flow at Mach Number 2 to 3

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CHOFU, TOKYO, JAPAN

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# Measurements of the Aerodynamic Derivatives of an Oscillating Biconvex-Flat Airfoil in Supersonic Flow at Mach Number 2 to 3*

By Takao ISHII** and Mitsunori YANAGIZAWA**

## SUMMARY

The aerodynamic derivatives of a biconvex-flat airfoil (thickness ratio 10 per cent), performing pitching oscillation in supersonic flow at the range of Mach number 2~3, was measured and compared with the linearized theory, piston theory and Van Dyke's nonlinear theory which includes the effect of the airfoil thickness. The tested reduced frequency range was from 0.01 to 0.05 and the tested Reynolds number range was $2.8\times10^6 \sim 3.6\times10^6$ based on the chord length. In spite of the existence of the detached shock wave in front of the leading edge, the experimental results agreed quite well with the nonlinear theory.

## Introduction

The study of aerodynamic forces and moments produced by the oscillating airfoils and bodies has been of great importance in the field of aeroelasticity especially with reference to flutter and other related problems. Experimental measurement of oscillating air forces has been required because of their importance in aeroelasticity and also because of the need to assess current theoretical works. However, as only limited experimental results have been available up to now, our research was directed towards acquiring information which would be of use in determining the reliability of these theories when they are applied to aeroelastic problems of airfoil.

The theory of unsteady supersonic flow for an oscillating thin wing was originally developed by Possio[1], and later the refined analysis and the extensive numerical results were obtained by Schwarz[2], Temple and Jahn[3], Garrick and Rubinow[4], and others. M.D. Van Dyke developed a second order two dimensional unsteady supersonic flow theory that included the effects of the thickness of the airfoil. Both theories did not take into consideration the existence of leading edge shock wave: the former treated thin airfoil and the latter, by assuming a sharp flexible nose that is kept in the streamwise direction, was able to avoid the shock wave problem.

H. Ashley and G. Zartarian developed a very convenient aerodynamic tool for aeroelasticians -piston theory-based on Hayes' hypersonic similitude. This piston theory can describe local pressure on the airfoil simply by local velocity of the oscillating airfoil surface. Due to this simplicity it has been used in attempting to solve various aeroelastic problems even in the supersonic range, especially when the exact expression of the aerodynamic forces makes the

* Received on Dec. 27, 1963.

** The First Aerodynamics Division.

aeroelastic problems intractably intricate.

The present investigation was undertaken with the aim of comparing these theories with experimental results. Presented in this paper are moment coefficien of a biconvex-flat plate airfoil associated with the pitching oscillation about an axis at 5, 15, 25, 35, and 45 per cent chord length from the leading edge. Moment coefficients were measured over a reduced frequency range from 0.01 to 0.05 and a Mach number range from 2 to 3.

The Reynolds number range based on the chord length varied from $2.8\times10^6$ to $3.6\times10^6$. The measurements were made by using a free oscillation method in a 20×15 cm supersonic tunnel with an asymmetric nozzle on an airfoil suspended between the side-walls by a pair of cross-flexure pivots.

## Apparatus

TUNNEL: Tests were made in a 20×15 cm blow-down supersonic tunnel equipped with an asymmetric nozzle which could provide a wide continous variation of Mach number from 1.8 to 3.5*1. (Fig. 2) The Mach number could be changed continuously during a run simply by sliding the lower nozzle with a motor-driven device. Supplied dry air pressure of 20 ata. was reduced through a control valve to the desired pressure up to 10 ata.

For these tests, the airfoil was suspended over the whole test section width by a pair of cross flexures at both sides of the wing. (Fig. 3 and 4)

MODEL: The tested airfoil had a 80 mm chord length, 8 mm thickness and a biconvex (circular arc)-flat wing section as shown in Fig, 1.

The positions of the rotation axis of the model were located at 5, 15, 25, 35, and 45 per cent of the chord length from the leading edge. Oscillation frequency was changed by using five degrees of cross flexure thickness varing 1~3 mm as shown in Fig. 5.

## Device to Give an Initial Deflection

Initial angular deflection for the free oscillation method was given by attracting, with a pair of electric magnets, a pair of iron arms which extended from the rotation axis of the model. These devices were located outside the tunnel wall and were covered with an air-tight box. When the magnetizing current was abruptly cut, the model was released and went into free oscillation.

## Instrumentation and Test Procedures

One strain gage was glued to each side of a flexure in order to measure the amplitude of oscillation; and this amplitude signal was recorded on oscillograph paper moving at a speed of 300 mm/sec. Before each run of the tunnel test, the decaying free oscillation of the airfoil in still air was measured in order to estimate the structural damping of cross flexures and

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*1 The coordinates for the nozzle contour were taken from Ref. 7 and 8 with courtesy of Wright Air Development Center and Prof. H. Buning, University of Michigan.

moment of inertia of the airfoil with accessaries from its logarithmic decrement and frequency*2.

## Data Reduction

By assuming the virtual moment of inertia of air to be negligiblly small when compared to that of the airfoil, we get the following equation of motion of the airfoil in air flow.

$$I\ddot{\alpha}+C_0\,\dot{\alpha}+K_0\,\alpha=M_\alpha\,\alpha+M_\alpha\,\dot{\alpha} \tag{1}$$

where

- $I$: mass moment of inertia per unit span
- $C_0$: structural damping coefficient of cross flexures per unit span*3
- $K_0$: torsional spring constant of cross flexures per unit span
- $\alpha(t)$: angle of attack (rotation)
- $M_\alpha$: aerodynamic restoring moment coefficient per unit span
- $M_\alpha$: damping coefficient of aerodynamic moment per unit span

$$\cdot\equiv\frac{\partial}{\partial t}$$

By solving Eq. (1), we obtain the circular frequency and logarithmic decrement:

$$\omega=\frac{\sqrt{4\,I\,(K_0-M_\alpha)-(C_0-M_{\dot\alpha})^2}}{2\,I} \tag{2}$$

$$\delta=\frac{C_0-M_{\dot\alpha}}{2\,I}\,\frac{1}{f},\quad f=\frac{\omega}{2\pi} \tag{3}$$

Circular frequency and logarithmic decement for the free oscillation in still air is obtained by putting $M_\alpha=M_{\dot\alpha}=0$ in Eq. (2) and Eq. (3):

$$\omega_2=\frac{\sqrt{4\,I\,K_0-C_0^2}}{2\,I} \tag{4}$$

$$\delta_0=\frac{C_0}{2\,I_0}\,\frac{1}{f_0},\quad f_0=\frac{\omega_0}{2\pi} \tag{5}$$

From Eq. (3) and Eq: (5) we obtain

$$-M_{\dot\alpha}=\frac{K_0}{2\pi^2}\,\frac{\delta_0}{f_0}\left\{\frac{f}{f_0}\,\frac{\delta}{\delta_0}-1\right\} \tag{6}$$

As $(C_0-M_\alpha)^2$ is usually small in comparison with $4\,I\,(K_0-M_\alpha)$ in Eq. (2), we can approximate

$$\omega=\sqrt{\frac{K_0-M_\alpha}{I}} \tag{7}$$

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*2 From the view point of more rigorous attitude, this calibration should have been done in vacuum. However, as the aerodynamic damping of still (non-viscous) air is, as is well known, theoretically zero, the aerodynamic damping comes only from viscosity of the air. This damping coefficient was measured in the separated preliminary test, and was proved to be approximately 3~5 per cent of the structural damping coefficient. The theoretical ratio of the virtual moment of inertia of the surrounding air to that of the wing model was estimated to be 0.2~0.5 per cent corresponding to the axis position at the midchord and at the leading (or the trailing) edge respectively.

*Considering the above facts, the present calibration was substitutionally done in the atmospheric still* air.

*3 Another assumption for structural damping is that the magnitude of the damping moment is proportional to the elastic restoring moment and is in phase with velocity. However, at $\omega\simeq\omega_{naturai}$, as in this free oscillation method for the wing of big mass ratio, both definition on the mechanism of structural damping do not differ from each other.

With the same assumption, from Eq. (4) we get

$$\omega_0=\sqrt{\frac{K_0}{I}} \tag{8}$$

From Eq. (7) and Eq. (8) we obtain the aerodynamic restoring moment coefficient

$$-M_\alpha=K_0\left\{\left(\frac{f}{f_0}\right)^2-1\right\} \tag{9}$$

Eq. (6) and Eq. (9) give the data reducing formulae by which we can obtain $-M_\alpha$ and $-M_{\dot\alpha}$.

Final results are shown in nondimensional form:

$$-m_\alpha=\frac{-M_\alpha}{4\rho U^2b^2} \tag{10}$$

$$-m_{\dot\alpha}=\frac{-M_{\dot\alpha}}{8\rho Ub^3} \tag{11}$$

A typical experimental record is shown in Fig. 6 for an airfoil of $a=0.45$ at $M=2.59$.

A typical example of amplitude decrement is shown in Fig. 7 and from this the logarithmic decrement is determined.

## Results and Discussion

$-m_\alpha$ and $-m_{\dot\alpha}$ vs. Mach number for various positions of rotation axis are shown in Fig. 8 to 12. Van Dyke's nonlinear theory is in close agreement with our experimental results both for $-m_\alpha$ and $-m_{\dot\alpha}$. The linearized theory gives higher $-m_\alpha$, but the deviation from the experimental value is almost constant (between 0.8 and 1.2) for the Mach number range of the present experiment.

The piston theory also gives a slightly higher value for $-m_\alpha$, however its deviation from the experimental value is much smaller (approximately 0.02~0.03) than that of the linearized theory.

As for $-m_{\dot\alpha}$, both the linearized thory and piston theory give higher values than experimental results for $a<0.45$.

In Fig. 13 and 14, experimental results are replotted on $-m_\alpha$ vs. $a$ plane for Mach number 2 and 3. Although the nonlinear theory, as mentioned above, is in close agreement with the experiment, the slope of $-m_\alpha$ with respect to $a$ axis given by the linearized theory and by the piston theory have exactly the same value as that of the experimental results at Mach number 2.

At Mach number 3, however, the linearized theory and nonlinear theory have exactly the same slope as the experimental result.

As for $-m_{\dot\alpha}$, the nonlinear theory agrees quite well with the experimental results, especially at $a$ below 0.25 for both Mach number 2.0 and 3.0.

In Fig. 15, $-m_\alpha$ is replotted vs. Mach number and is compared with the nonlinear theory. It is interesting that the experimental value is higher than the theoretical value above a certain Mach number which decreases as $a$ increases.

In Fig. 16, $-m_\alpha$ and $-m_{\dot\alpha}$ are shown for the various reduced frequencies of the present experiments. These figures imply that, for the small $k$ of the experiment $-m_\alpha$ is almost the value of the steady state flow, and the higher terms than $k^2$ in the expression of $-m_{\dot\alpha}$ is negligiblly small.

Considering the fact that the leading edge of the airfoil model had a biconvex shape of which included angle was 73°44', one could expect the critical Mach number for the attached shock wave to bo about 3.47.

Therefore it was considered that the shock wave was detached at the leading edge for the entire Mach number range of the experiment. However, as the greatest part of the model was a flat plate it was expected that the region on the airfoil, which was markedly subjected to the effects of the strong (nearly normal) part of the shock wave, would be restricted to the rather small part of the airfoil (near the leading edge). Most of the airfoil surface was immersed in the region where the effect of weak shock (nearly Mach wave) played the major part.

As no theoretical work is available at present for the estimation of the effect of a blunt leading edge on aerodynamic derivatives, a qualitative analysis, which will explain the deviation of the experimental results from the theories, can not be made. However, it should be noted that Van Dyke's nonlinear theory, which considered the thickness of an airfoil but avoided, by a mathematical contrivance, the existence of a strong leading edge shock wave, is still applicable for an airfoil with a blunt leading edge (at least for a blunt-flat airfoil).

## Conclusions

1) The nonlinear theory gives quite accurate values both for $-m_\alpha$ and $-m_{\dot{\alpha}}$ in the whole range of the present experimental conditions.

2) In the Mach number range $M>2$, the piston theory gives bette estimation of $-m_\alpha$ than the linearized theory.

3) On the other hand, as for $-m_{\dot{\alpha}}$, the linearized theory gives the closer value than the piston theory for $0<a<0.5$, especially for the bigger $a$.

## Acknowledgement

During the present research the authors have benefited from contact with many colleagues. In particular they wish to express their appreciation to Mr. I. HIRAKI, Head of the First Aerodynamics Division of the NAL, and Mr. H. NAGASU, Acting Chief of Unsteady Aerodynamics Section, for their encouragement, and to Dr. K. WASHIZU and Dr. J. SHIOIRI, Profssors of the University of Tokyo, for their helpful suggestions. Mr. I. UCHIDA and Mr. Y. HARATA have been of invaluable assistance during the past two years and have contributed much to this experiment.

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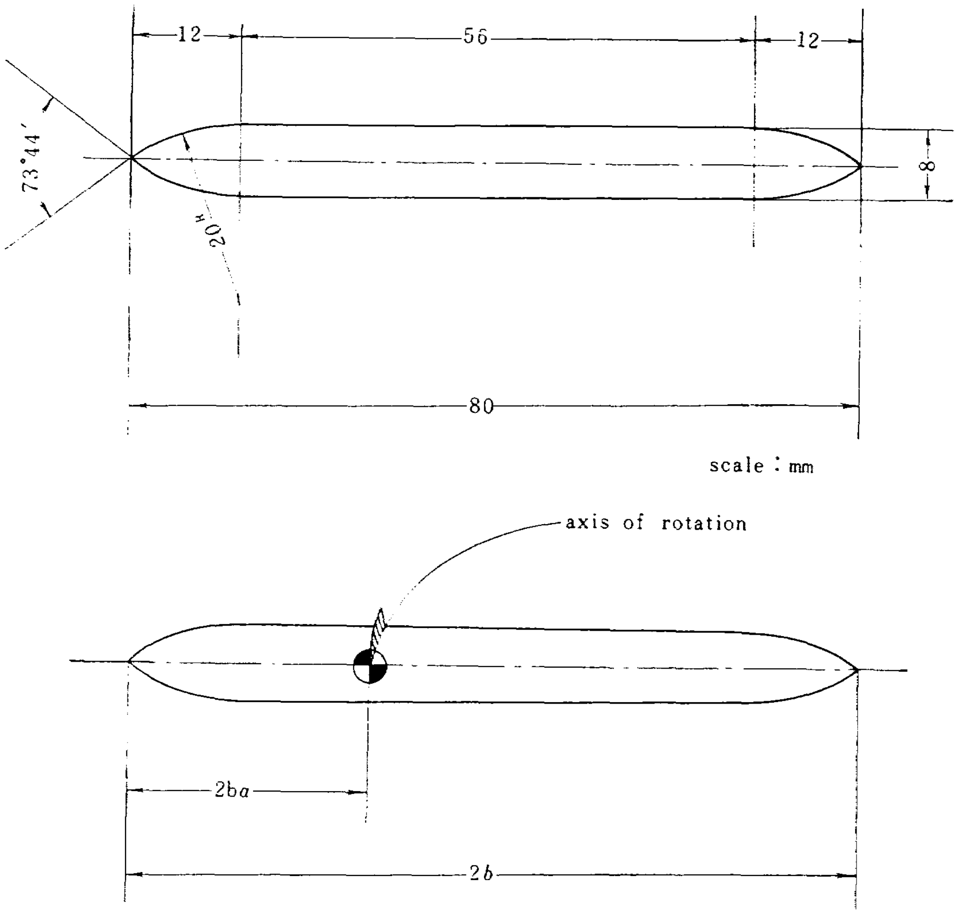



Fig. 1 Dimensions and notations of the tested airfoil

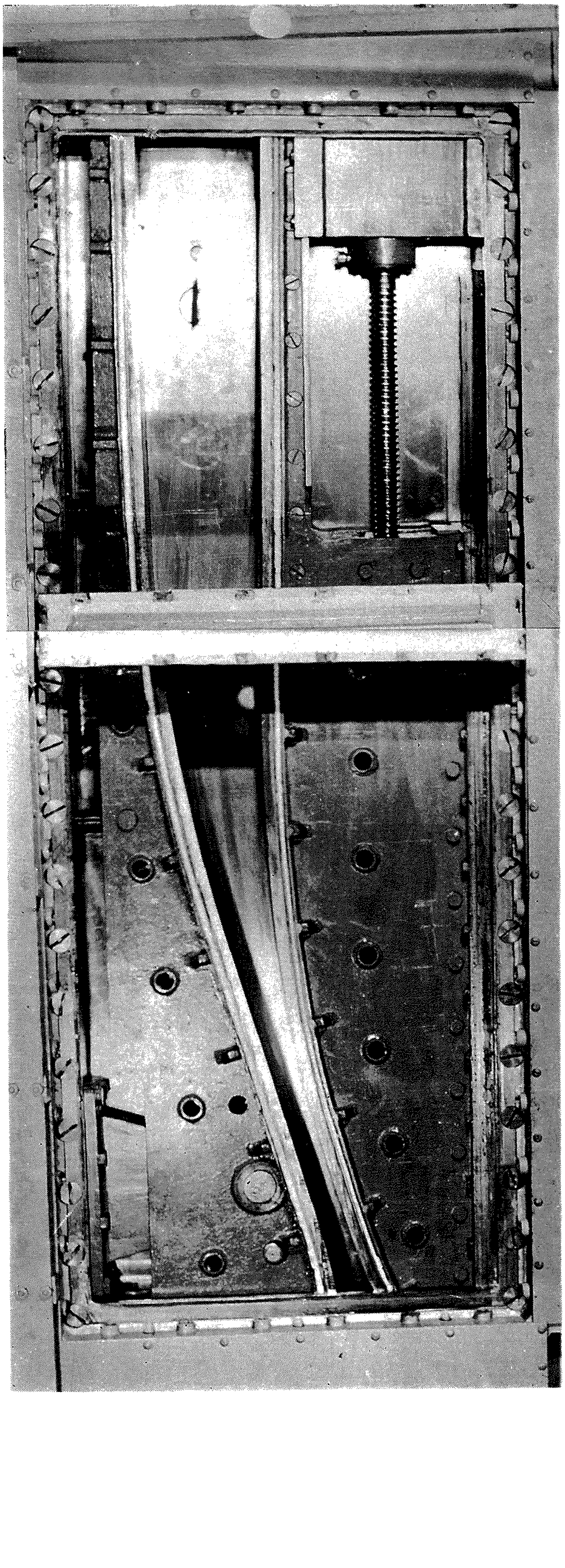


Fig. 2 The variable Mach number tunnel with an asymmetric nozzle

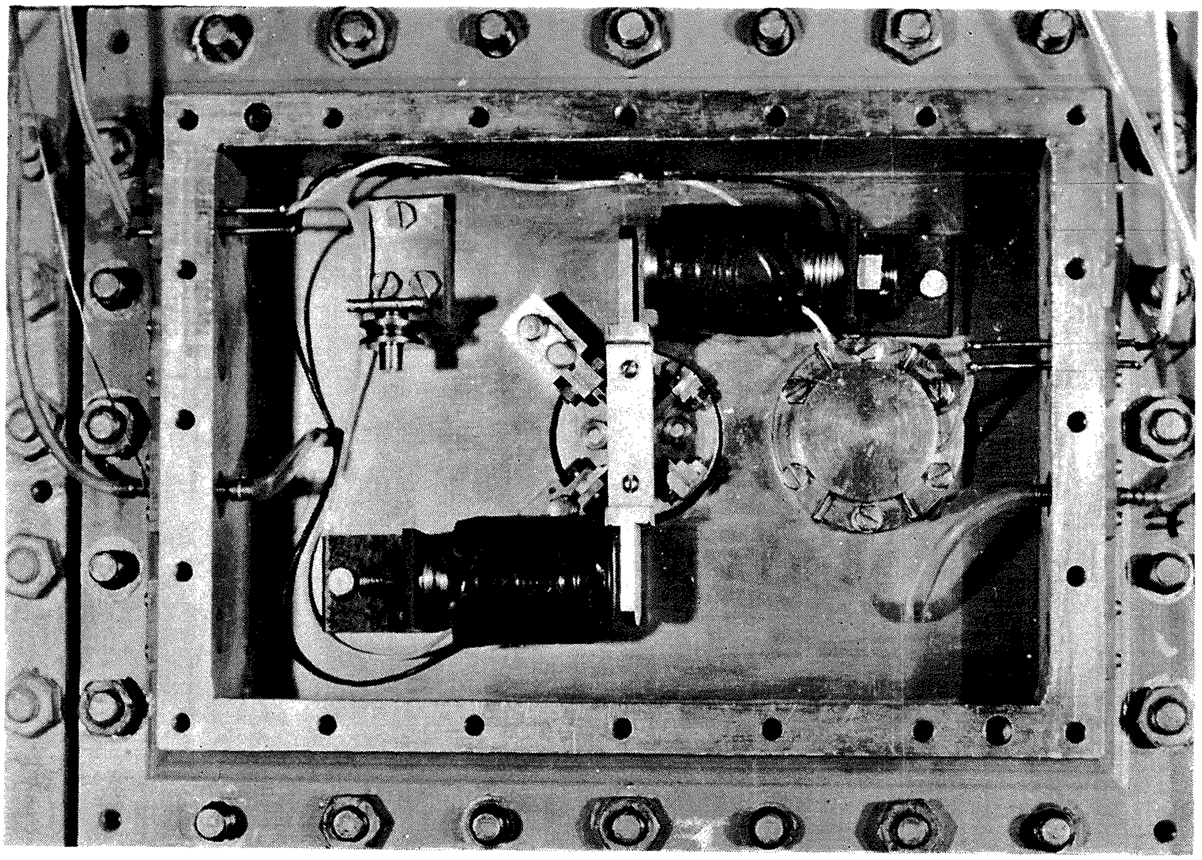


Fig. 3a Devices to give an initial deflection

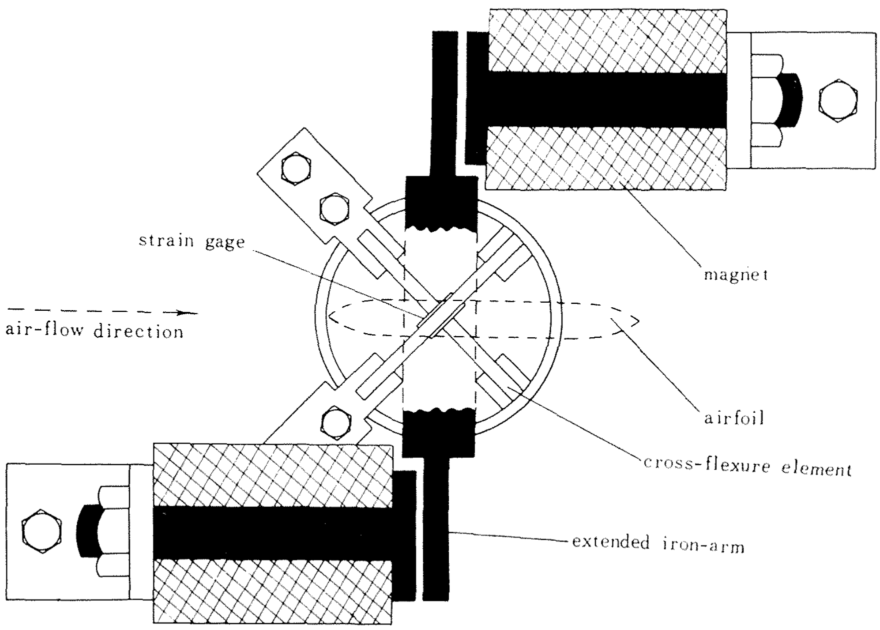



Fig. 3b Schematic illustration of the magnets and the flexures

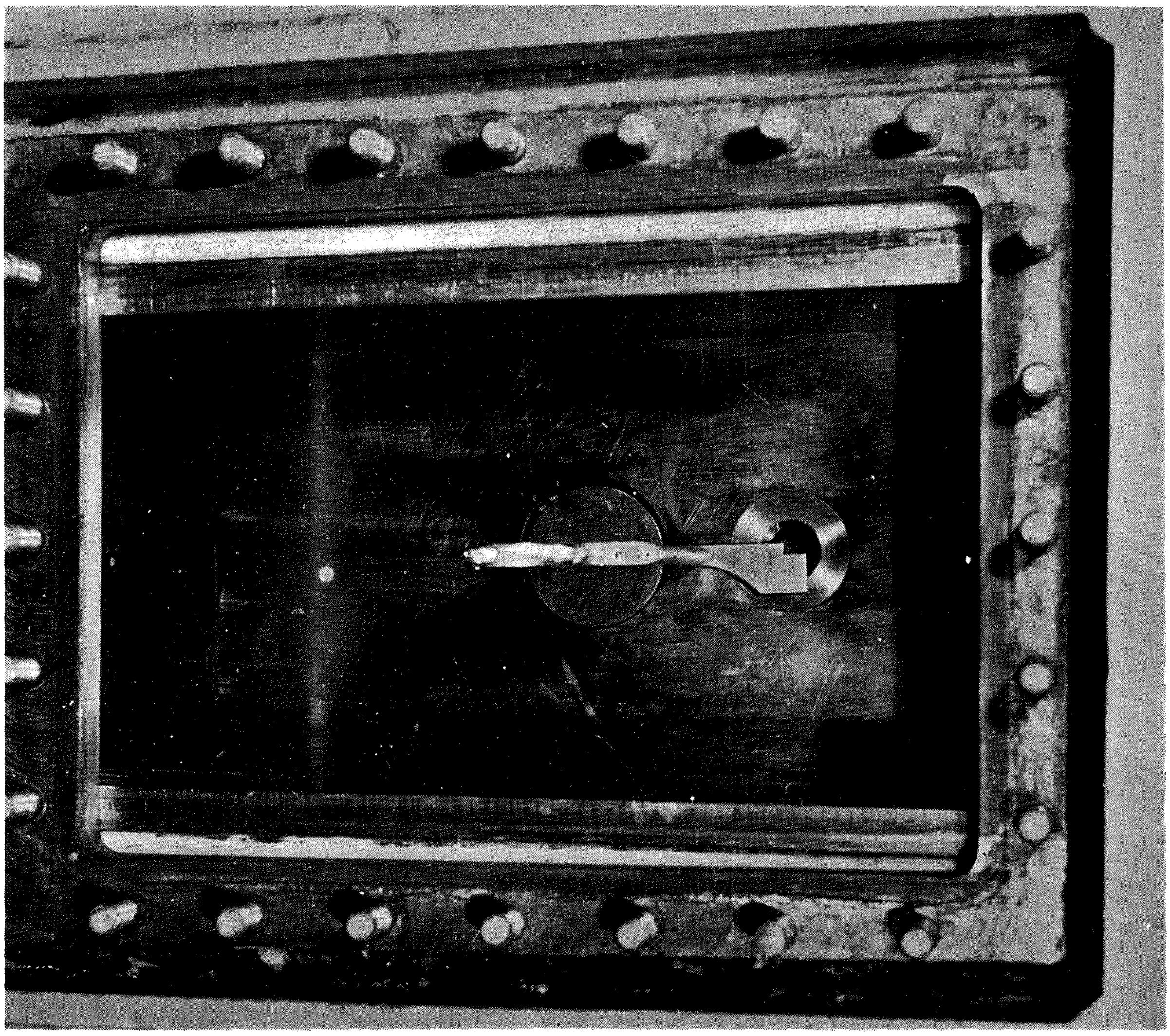


Fig. 4 A model installed in the test section

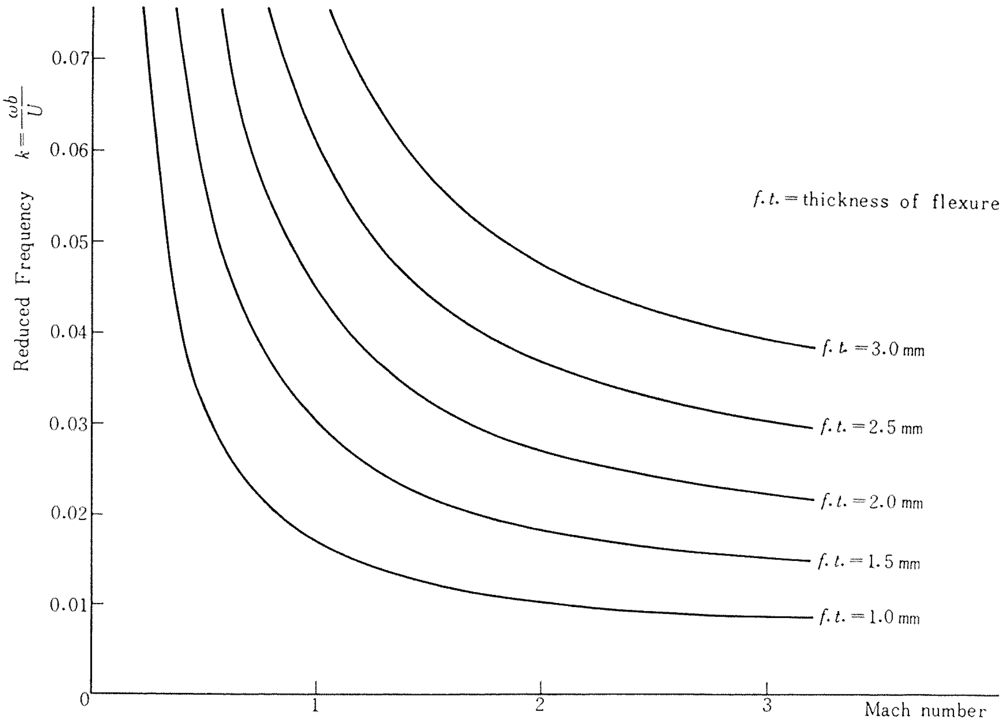



Fig. 5 Estimated reduced frequency for the five degrees of flexure thickness

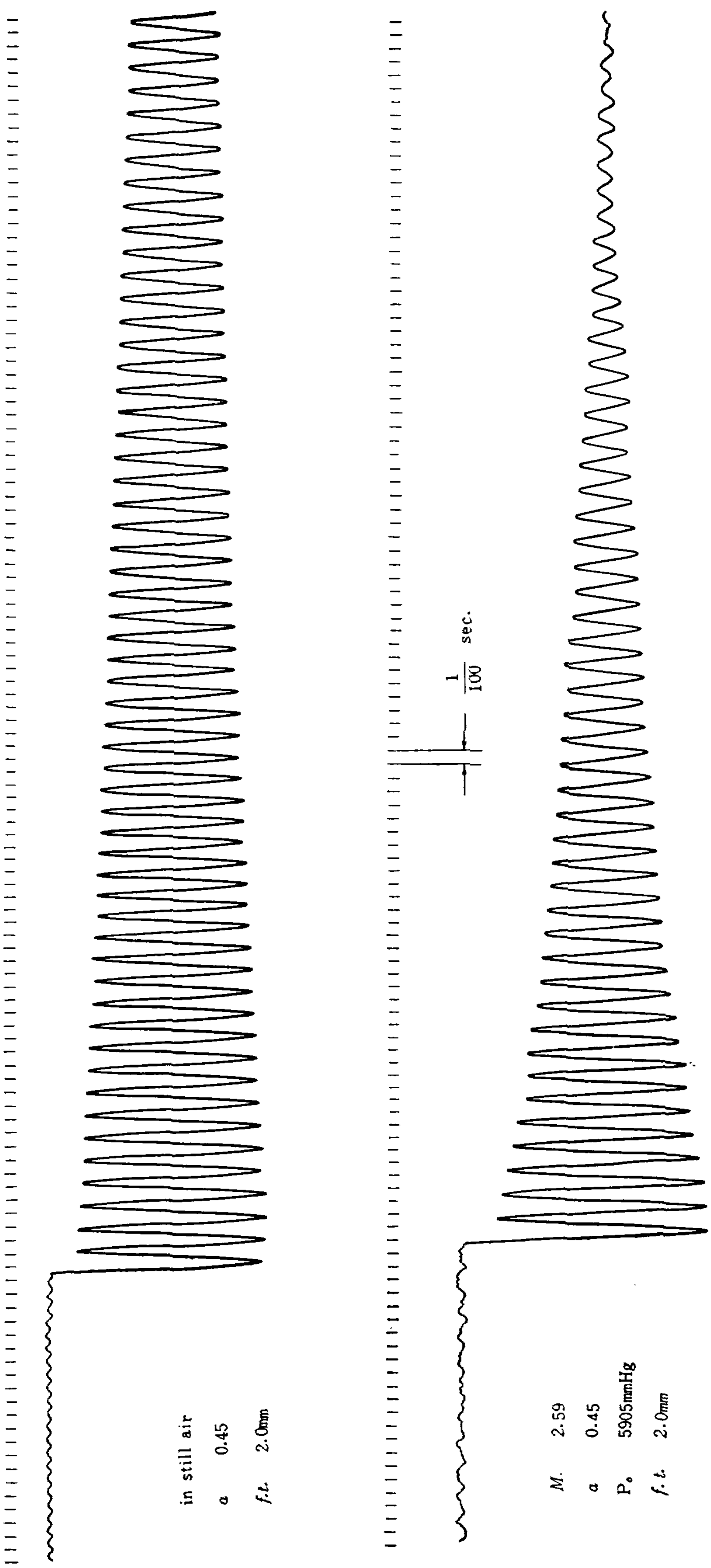



*Fig.* 6 A typical experimental record for an airfoil of $a=0.45$ at $M=2.59$

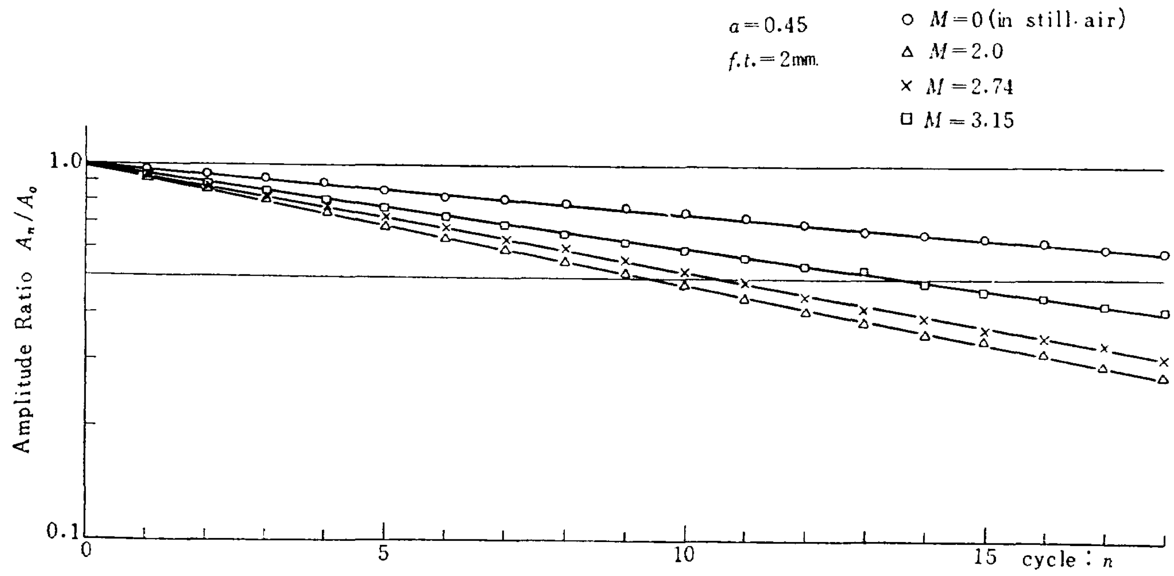



Fig. 7 A typical example of amplitude decrment for $a=0.25$ and the flexure thickness=2 mm

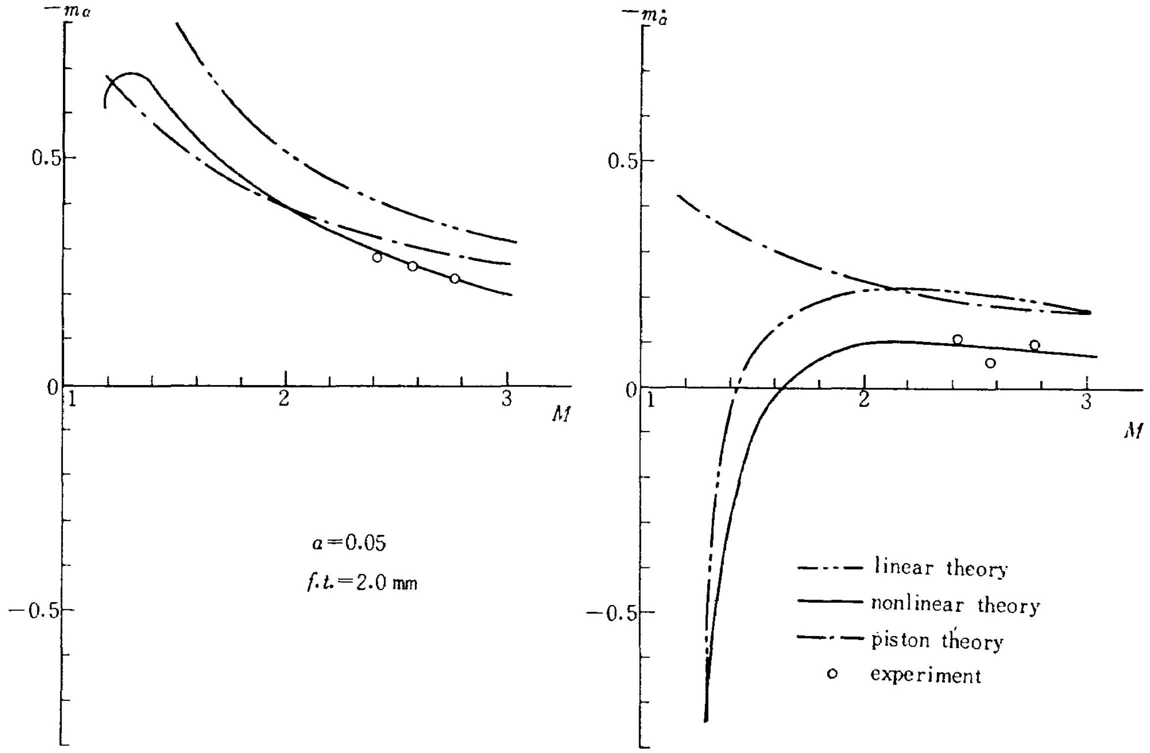
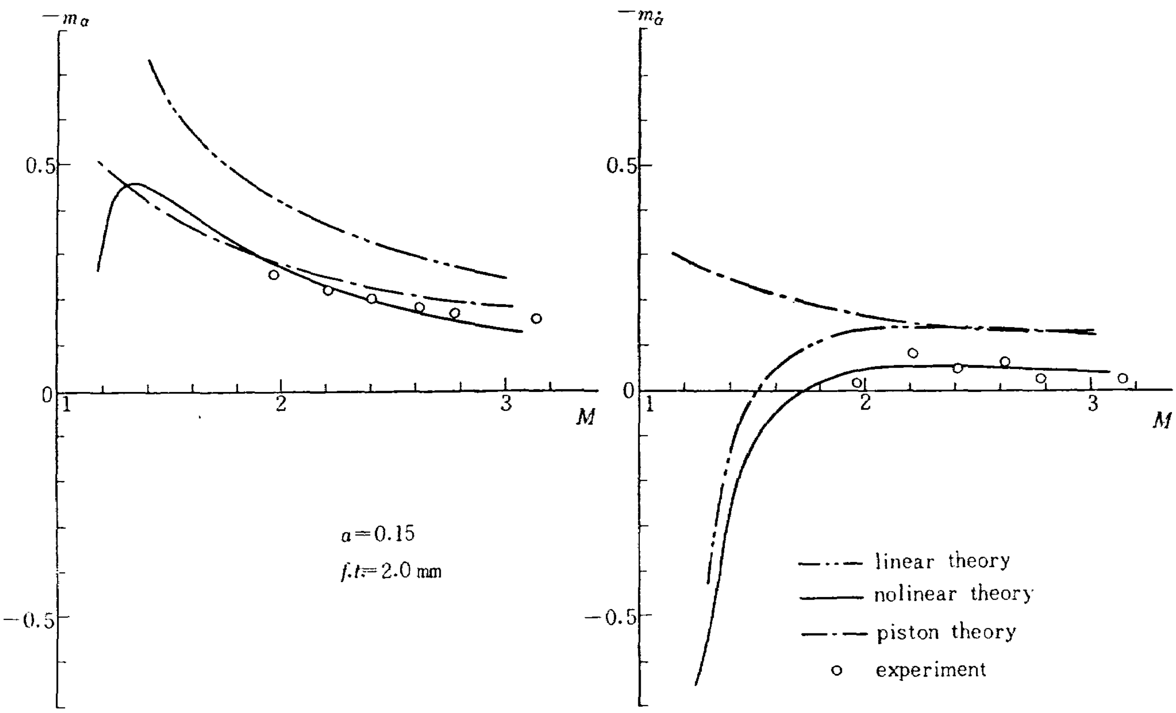
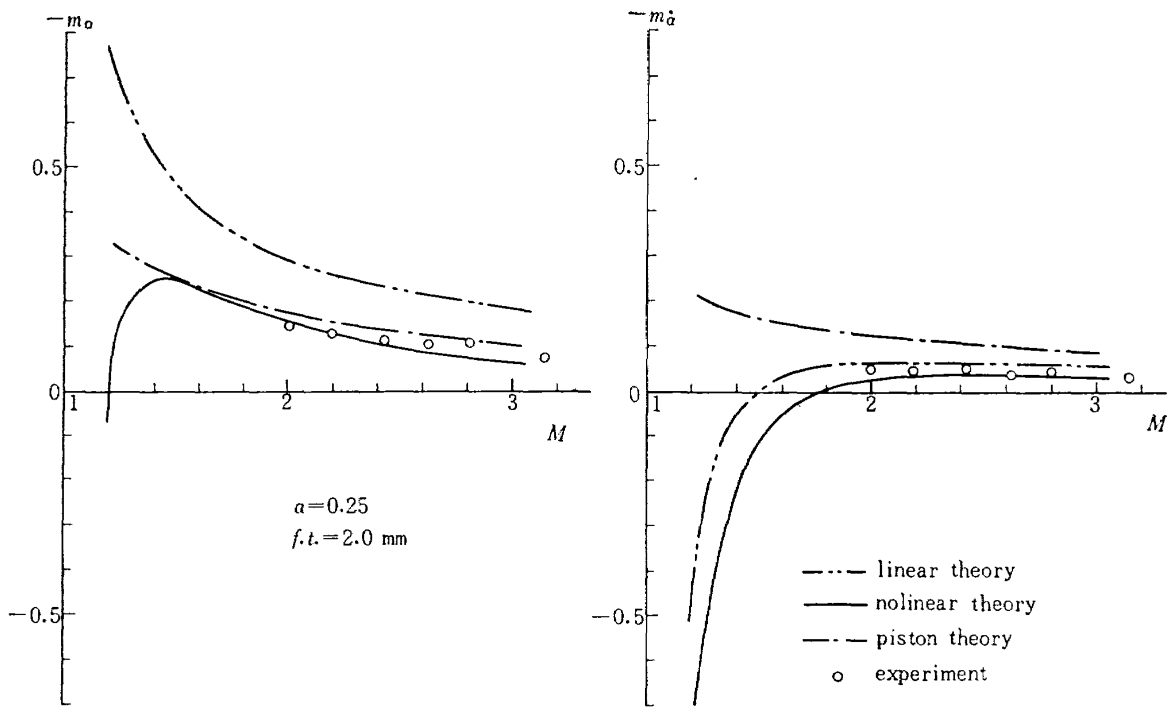



Fig. 8



$-m_\alpha$
0.5
0
−0.5
1
2
3
$M$
$a=0.15$
$f.t.=2.0$ mm
$-m_{\dot\alpha}$
0.5
0
−0.5
1
2
3
$M$
linear theory
nolinear theory
piston theory
experiment


Fig. 9



$-m_\alpha$
0.5
0
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$M$
$a=0.25$
$f.t.=2.0$ mm
$-m_{\dot\alpha}$
0.5
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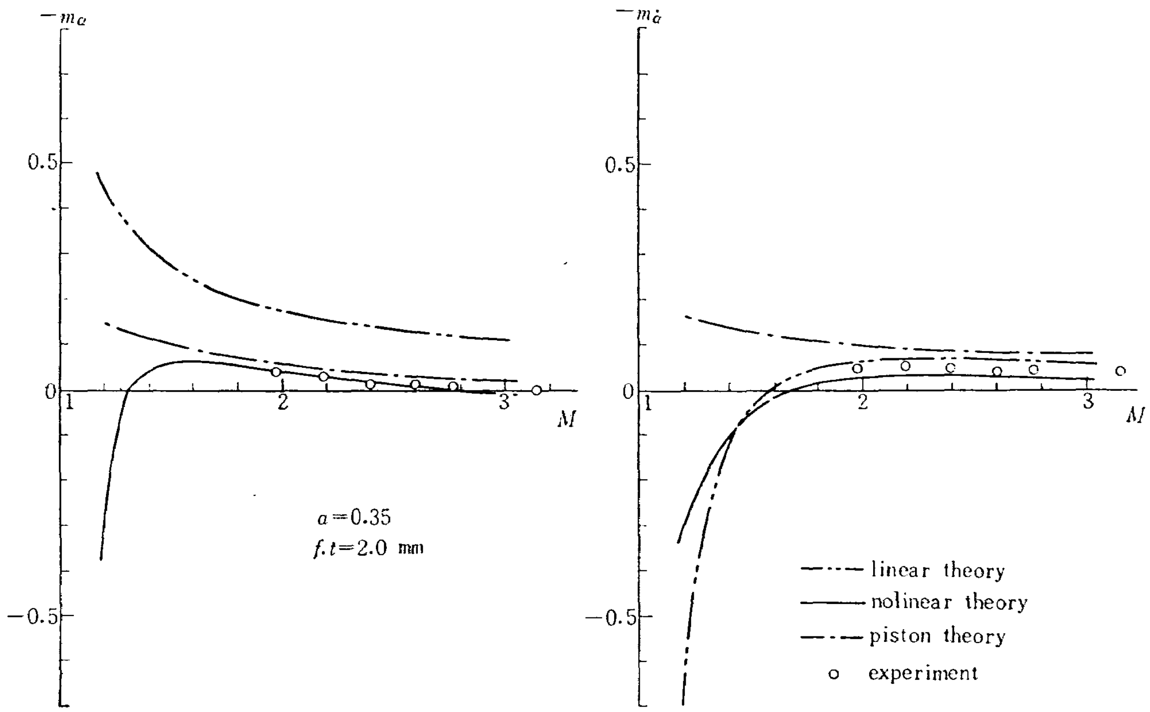
$M$
linear theory
nolinear theory
piston theory
experiment


Fig. 10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linear theory
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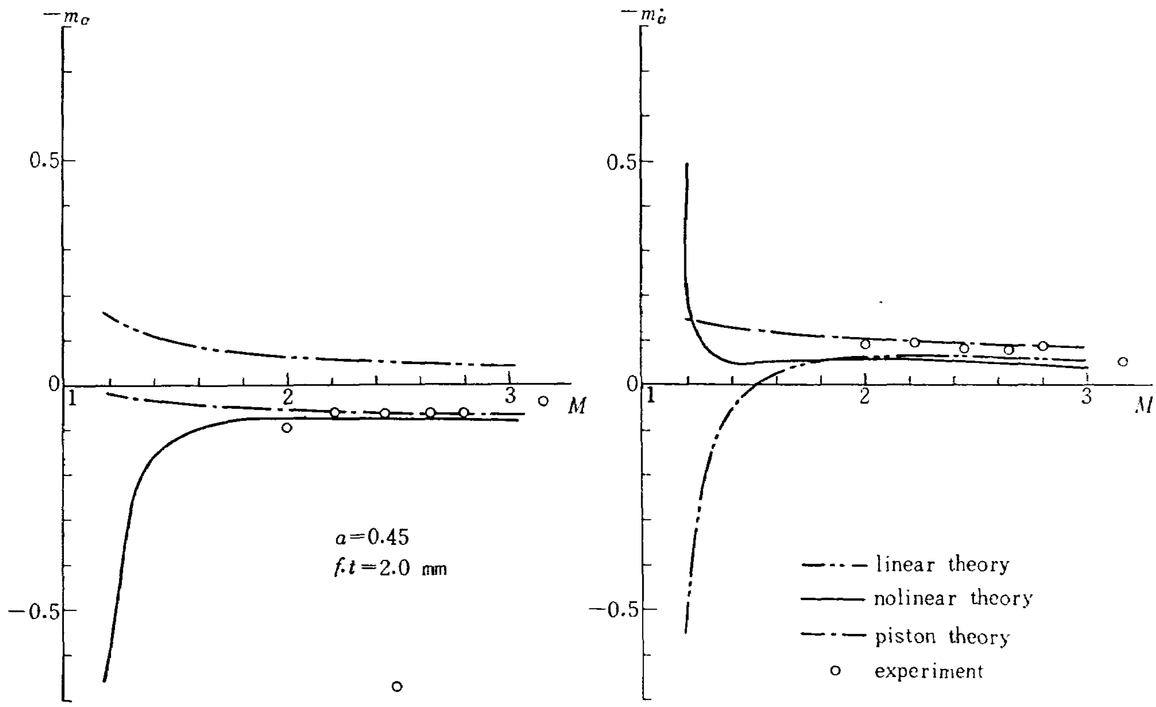
experiment

Fig. 11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$a=0.45$
$f.t=2.0$ mm
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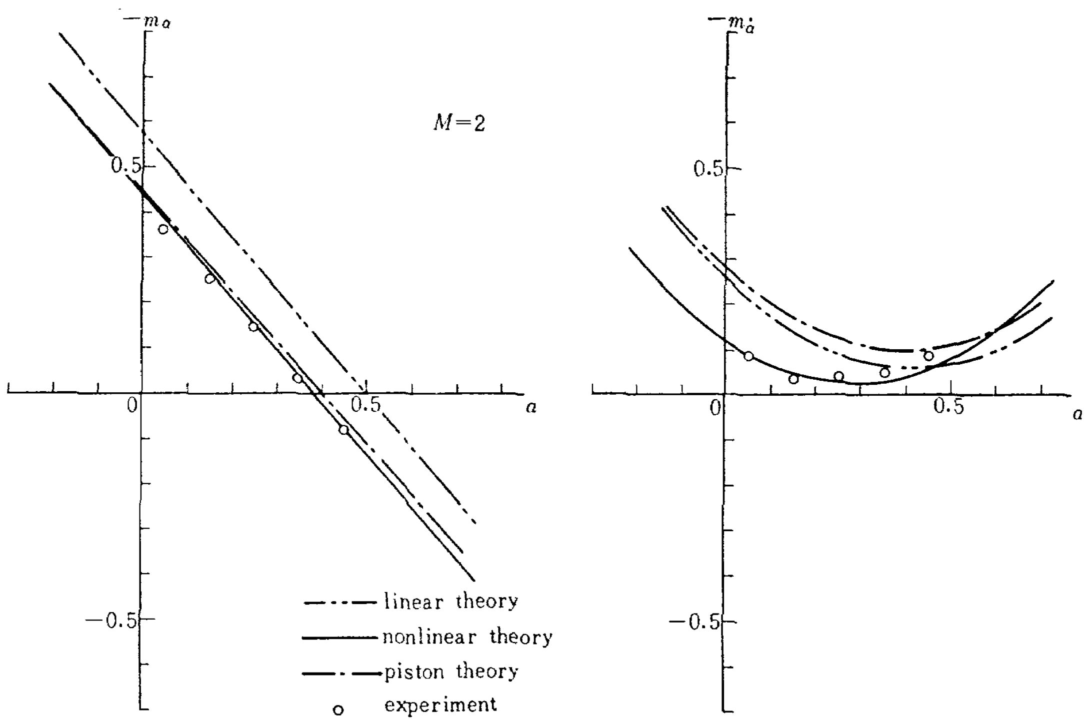
linear theory
nolinear theory
piston theory
experiment

Fig. 12


$-m_\alpha$
$M=2$
0.5
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0.5
$a$
−0.5
$-m_{\dot{\alpha}}$
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$a$
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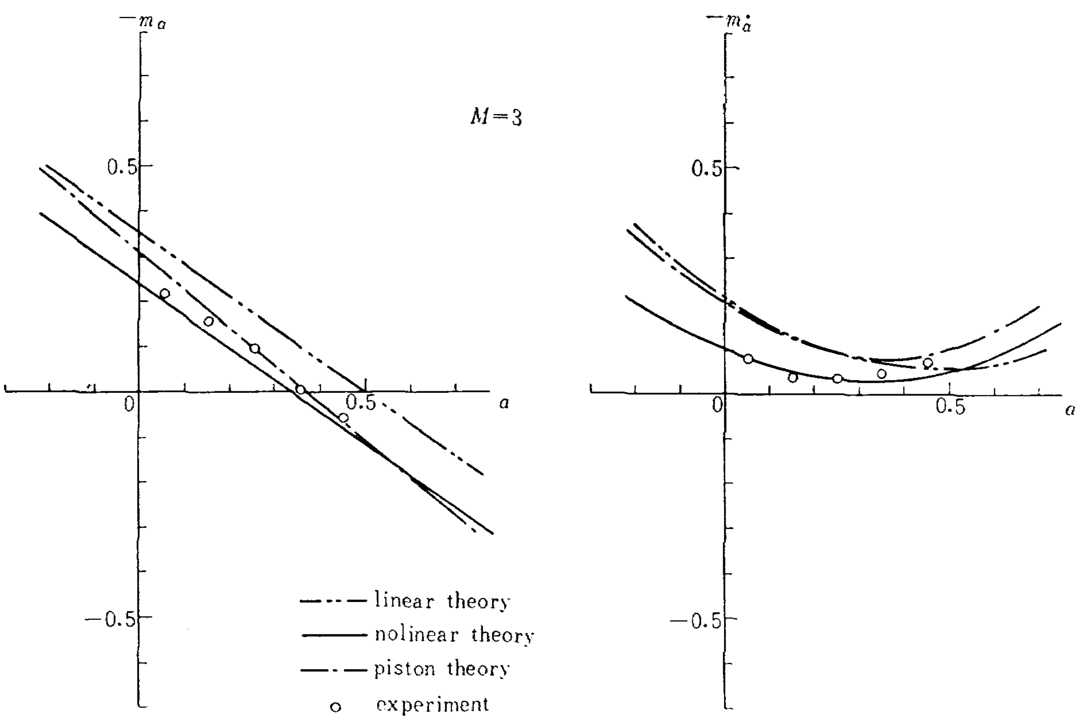
linear theory
nonlinear theory
piston theory
experiment

Fig. 13


$-m_\alpha$
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linear theory
nolinear theory
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experiment

Fig. 14

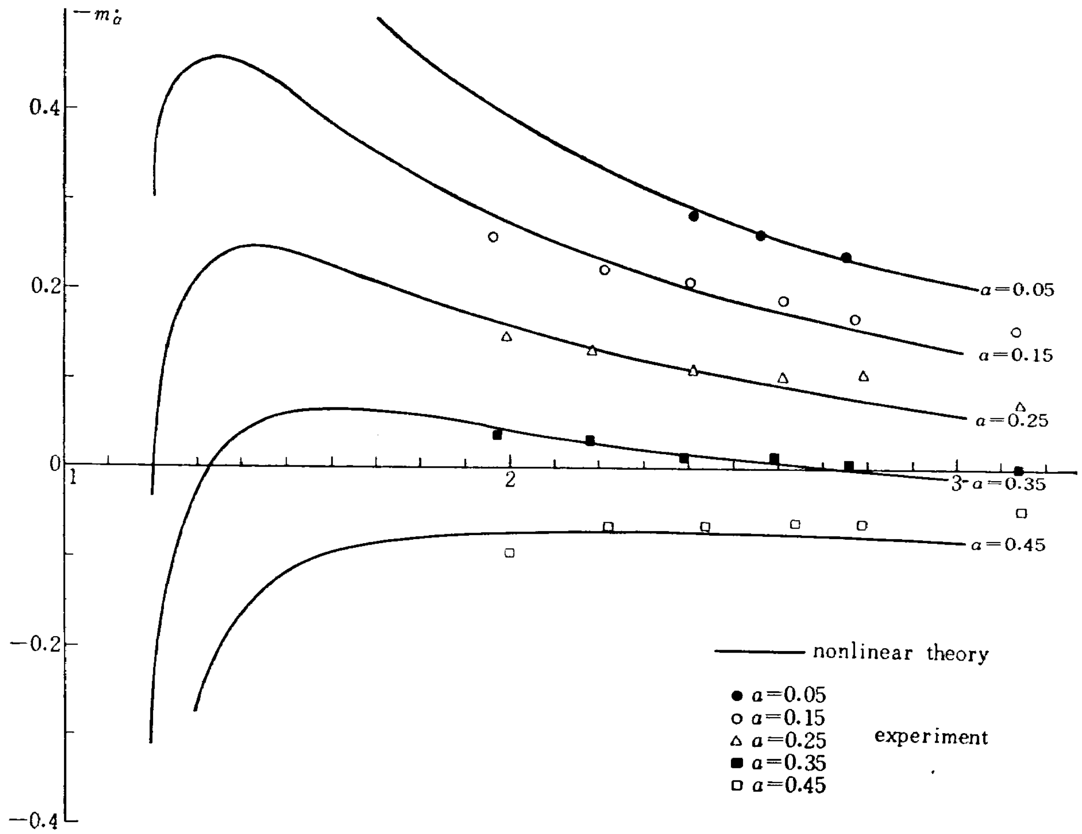



Fig. 15 a

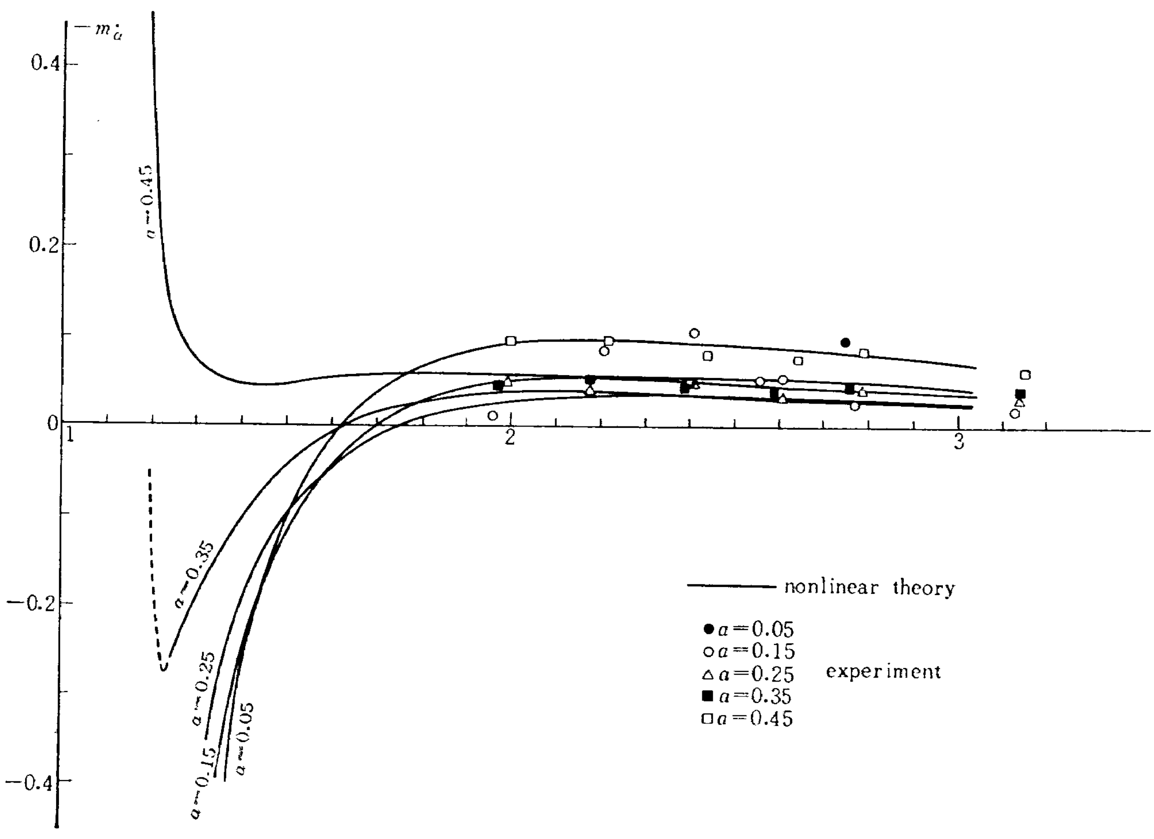
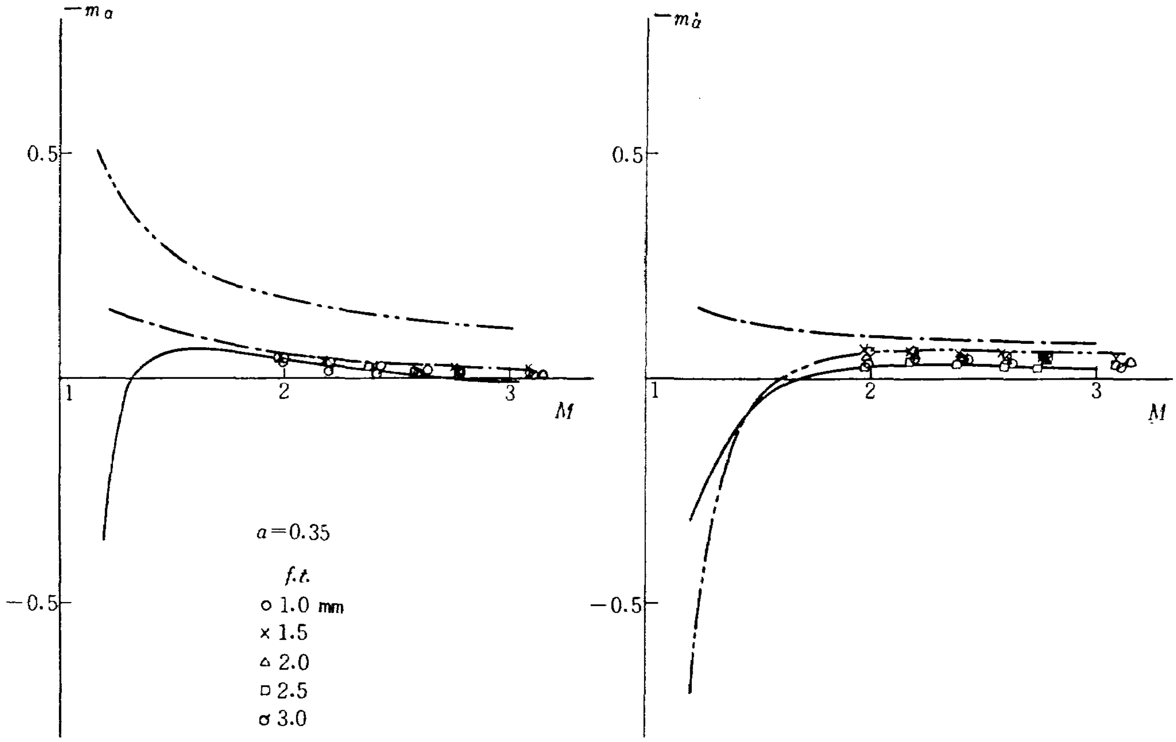



Fig. 15 b



$-m_\alpha$
0.5
−0.5
1
2
3
M
$a=0.35$
f.t.
○ 1.0 mm
× 1.5
△ 2.0
□ 2.5
σ 3.0
$-m_{\dot\alpha}$
0.5
−0.5
1
2
3
M


Fig. 16

NAL TR-57T

航空宇宙技術研究所

円弧–平板結合翼の超音速流（マッハ 2～3）における動的空力微係数の測定

1964 年 1 月　　17 ページ

前縁が頂角 73°44′ の複合円弧型の平板翼について，動的空気の微係数を超音速流（マッハ数 2～3）中で測定し，これを線型理論（薄翼理論）ピストン理論（極超音速相似則より導かれた準定常理論），翼の厚みを考慮した非線型理論と比較し，次の結果を得た。

(1) 動的空力微係数に及ぼす前縁衝撃波の影響は小さい。(2) 非線型理論は reduced frequency の一次の項までで，十分良く一致する。(3) マッハ数 2 以上ではピストン理論の与える $m_\alpha$ は線型理論より良いが，(4) 他方，$m_{\dot{\alpha}}$ については線型理論の方がピストス理論より良い値を与える。

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