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AD-A030 966

Unsteady Pressures Due to Control Surface Rotation at Low Supersonic Speeds Comparison between Theory & Experiment

Advisory Group for Aerospace Research & Development Paris France

Sep 76

AGARD-R-647

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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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AGARD REPORT No. 647

on

Unsteady Pressures Due to Control Surface Rotation at Low Supersonic Speeds

Comparison between Theory and Experiment

by

C.G.Lodge and H.Schmid

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AGARD Report No.647

UNSTEADY PRESSURES DUE TO CONTROL SURFACE ROTATION AT LOW SUPERSONIC SPEEDS –

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A

Paper presented at the 42nd Structures and Materials Panel Meeting, Ottawa, April 1976.

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Published September 1976

ISBN 92-835-1223-5

Printed by Technical Editing and Reproduction Ltd
Harford House, 7-9 Charlotte St, London, W1P 1HD

REPORT DOCUMENTATION PAGE

1. Recipient's Reference	2. Originator's Reference	3. Further Reference	4. Security Classification of Document
	AGARD-R-647	ISBN 92-835-1223-5	UNCLASSIFIED

5. Originator Advisory Group for Aerospace Research and Development
North Atlantic Treaty Organization
7 rue Ancelle, 92200 Neuilly sur Seine, France

6. Title UNSTEADY PRESSURES DUE TO CONTROL SURFACE ROTATION AT LOW SUPERSONIC SPEEDS – Comparison between Theory and Experiment

7. Presented at
the 42nd Structures and Materials Panel Meeting, Ottawa, April 1976.

8. Author(s)	9. Date
C.G.Lodge* and H.Schmid †	September 1976

10. Author's Address		11. Pages
*British Aircraft Corporation Ltd United Kingdom	† Messerschmitt-Bolkow-Blohm GmbH Federal Republic of Germany	22

12. Distribution Statement This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.

13. Keywords/Descriptors

Unsteady flow	Rotation
Supersonic flow	Predictions
Pressure distribution	Pressure measurements
Control surfaces	

14. Abstract

This paper was presented during the 42nd Meeting of the Structures and Materials Panel in Ottawa in April 1976.

It deals with a serious difficulty in unsteady aerodynamics, that is the prediction of the pressure field induced by the rotation of a control surface. Much work has already been done on this subject in subsonic flow, but this is one of the first approaches to the supersonic problem. Predictions have been made by two methods developed separately by BAC and MBB. They have been compared with windtunnel tests made at NLR using more than 80 pressure tubes. Pressure distributions, hinge moments and lift have been measured for different sections of the wing. As the two theories that have been used are linearised, the agreement between theory and experiments is not perfect but appears to be adequate for flutter speed prediction.

PREFACE

This paper by Lodge and Schmid was presented to the Sub-Committee on Aeroelasticity and Unsteady Aerodynamics on 5 April 1976 during the 42nd Meeting of the Structures and Materials Panel in Ottawa.

It deals with one of the most serious difficulties in unsteady aerodynamics, that is the prediction of the pressure field induced by the rotation of a control surface. Much work has already been done on this subject in subsonic flow but this publication presents one of the first approaches to the supersonic problem.

Predictions have been made by two distinct methods developed separately by BAC and MBB. They have been compared with windtunnel tests made at NLR using more than 80 pressure tubes. Pressure distributions, hinge moments and lift have been measured for different sections of the wing.

As the two theories that have been used are linearised, the agreement between theory and experiment is not perfect but appears to be adequate for flutter speed prediction.

This paper is of great interest to the aeroelasticians of the NATO community and will help both flutter prediction and active control design.

G COUPRY

Unsteady Pressures due to Control Surface Rotation at Low Supersonic Speeds - Comparison between Theory and Experiment

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Summary:

Most aircraft flutter problems have featured control surfaces, and it is necessary that unsteady aerodynamic forces generated by their motions should be accurately predicted. Therefore, theoretical and experimental studies on a planform with part-span control surface oscillating in the control surface rotation mode at low supersonic Mach numbers have been effected and the results of these are presented and discussed. It is shown that these studies must be of a high accuracy so that the more critical aerodynamic coefficients, such as hinge moment damping, might be determined with confidence.

1. NOTATION

Symbols

$c(y)$	local chord of aerofoil (including control surface)
$\bar{c}$	mean chord
f	frequency
M	Mach number
p	pressure
s	semi-span
V	free-stream velocity
w	non-dimensional upwash amplitude
x,y,z	non-dimensional rectangular co-ordinates referred to s
ρ	density of air
ϕ	non-dimensional amplitude of velocity potential

Subscripts

h.l.	hinge line
l.e.	leading edge
t.e.	trailing edge
r.	rudder
l.s.	lower surface
u.s.	upper surface

Definitions

k	=	$\omega\bar{c}/V$	reduced frequency parameter
q_∞	=	$\frac{1}{2}\rho V^2$	stagnation pressure
R^2	=	$(x-\xi)^2 - \beta^2(y-\eta)^2 - \beta^2 z^2$	hyperbolic distance
β^2	=	M^2-1	
Δ_p	=	$p_{l.s.} - p_{u.s.}$	
ΔC_p	=	$\Delta_p/q_\infty = \Delta C_p' + i\Delta C_p''$	
ω	=	$2\pi f$	circular frequency

$$l(y) = s \int_{x_{h.l.}}^{x_{t.e.}} \Delta_p \, dx = q_\infty \, c(y).c_l$$ sectional lift (positive upwards)

$$h(y) = s^2 \int_{x_{h.l.}}^{x_{t.e.}} \Delta_p \, (x - x_{h.l.}) \, dx = q_\infty \, c(y)^2.c_h$$ sectional hinge moment (positive nose downwards)

$$L = s^2 \int_0^s \int_{x_{l.e.}}^{x_{t.e.}} \Delta_p \, dx \, dy = q_\infty \, \bar{c}.s.C_L$$ total lift (positive upwards)

$$H = s^3 \int_0^{y_r} \int_{x_{h.l.}}^{x_{t.e.}} \Delta_p \, (x - x_{h.l.}) \, dx \, dy = q_\infty \, \bar{c}^2.s.C_H$$ total hinge moment (positive nose downwards)

2. INTRODUCTION

The accuracy of the control surface unsteady aerodynamics is particularly important on modern combat aircraft at low supersonic Mach number/high frequency parameter combinations.

In order to assess the accuracy of current theories for control surface unsteady aerodynamics, a nominally rigid model has been designed, built and tested at NLR, Amsterdam, at Mach numbers up to 1.3 and reduced frequency parameters, based on semi-span, up to 1.6. Unsteady pressure distributions induced over the main and control surfaces by rotational control surface oscillations have been measured and compared with predictions. Since dynamically scaled control surface models can involve a combination of rigid rotation and torsion, which both generate basically similar unsteady aerodynamic effects, model data for a rotation mode can be used to quantify the accuracy of flutter calculations involving the real aircraft vibration modes.

Predicted and measured results are presented here for M = 1.1, 1.3, and values of k from 0.7 up to 1.4.

3. MODEL DETAILS

The model under consideration is a swept main surface with part-span control surface, of aspect ratio 2.0, as described in Figure 1a. Its profile is symmetric with a maximum thickness to chord ratio of 5.5%. Gaps between the main and control surfaces have been kept down to the order of 0.1 mm.

In the model construction, both main and control surfaces were made as stiff as possible, to minimise any vibration mode effects. The main surface is of steel and the control surface of Dural, to reduce the inertia forces.

Results were obtained from measurements at pressure holes situated along 3 streamwise stations (Figure 1a). These are more closely spaced near the hinge line and the control surface tip.

During the model design stage, research work (References 1 and 2) revealed inadequacies in the calibration factors hitherto applied by NLR to the measured pressures, to account for tube system dynamics with wind-on. Therefore, special measures were taken during these model tests, to establish the correct calibration, by installing some direct measuring pressure transducers (Reference 3).

In addition, accelerometers were installed to determine the true model vibration mode during oscillatory control surface excitation.

4. WIND TUNNEL TESTS

Under contract with B.A.C., Warton, oscillatory pressure measurements were made on the model described above in the High Speed Tunnel at the NLR, Amsterdam. In this closed circuit tunnel, with a rectangular test-section of 2.00 metres wide by 1.60 metres high, the values of the Reynolds number, based on mean chord, varied within the range of 3.5×10^6 and 4.7×10^6 (Figure 1b).

The model was mounted into the tunnel wall, with the main surface rigidly clamped. The control surface excitation equipment was outside the tunnel wall. Further tests exhibited that the repeatability of the results was within 10%. The boundary layer transition was fixed by installing strips of carborundum grains of nominal size 74, on both sides of the model. These strips were placed forward of the region of shock onset (Reference 4).

5. AERODYNAMIC THEORIES

5.1 Outlines of B.A.C. Method

The supersonic lifting surface theory of Sadler and Allen (Reference 5) is a linearised theory solving the doublet based integral equation. This involves the solving of an integral equation which relates the down-wash at any given point to the perturbation velocity potential.

Integration has to be performed over that portion of the wing and the wing wake lying in the Mach fore-cone of that point. As the potential is zero forward of the planform leading edge and outside the wing wake, the integrations require less storage and are speedier than those of the more popular integrated downwash methods.

The basic integral equation is:-

$$w(x,y) = \lim_{z \to 0} \frac{\partial}{\partial z} \iint_{\substack{\text{Mach} \\ \text{fore-cone}}} \phi(\xi,\eta)\, K(x-\xi, y-\eta)\, d\xi\, d\eta \qquad (1)$$

where.

$$K(x,y) = -\frac{1}{\pi} \exp\left(-\frac{iM^2 kx}{\beta^2}\right) \cos\left(\frac{Mk\, R}{\beta^2}\right) \frac{1}{R} \qquad (2)$$

Characteristic co-ordinates ξ and η, with their origin at the pivotal point (the point at which the potential is being calculated) and their axes parallel to the two forward pointing Mach lines, are introduced, the lines of integer ξ and η forming a characteristic mesh (figure 2).

By splitting the above equation into its real and imaginary parts, and assuming certain variations of the real and imaginary parts of ϕ along lines of constant ξ and η, known functions of the upwash w at the pivotal point are found as weighted sums of the potentials at the mesh points, from which the potential at the pivotal point can be determined.

When the trailing edge is subsonic, the potential over the wake region is needed and this may be calculated from the condition that no load can be sustained across the wake. Thus we find.-

$$\phi_{wake} = \phi_{t.e.} \exp(-ik(x_{wake} - x_{t.e.})) \qquad (3)$$

Where a box is cut by the leading edge, special weighting has to be introduced for that particular box, to prevent any leading edge inaccuracies.

The integral equation is solved in a "marching" technique such that every potential in the pivotal points forward pointing Mach cone is known. Thus, the potentials at every mesh point can be calculated ΔC_p and the aerodynamic work matrices are then derived from these potentials.

As programmed this theory cannot deal with the singularity associated with the control surface hinge line incidence discontinuity. This problem has been overcome by treating the whole planform as a single surface and smoothing out the hinge line discontinuity by a cubic interpolation.

5.2 Outlines of M.B.B. Theory

The unsteady program used by M.B.B. is an extension of the characteristic box method first proposed by V J E Stark (Reference 6) The assumed small perturbation of the flow field implies a full linearisation of the potential equation The procedure was amended and programmed by C Bohm (Reference 7) and H Schmid (Reference 8) Primarily, the velocity potential of the harmonically varying flow is evaluated by means of a source distribution The basic equation reads as follows -

$$\phi = \iint_{\substack{\text{Mach} \\ \text{fore-cone}}} w(x',y')\, K(x-x', y-y', M, k)\, dx'dy' \qquad (4)$$

The equation relates the velocity potential to the normal velocity w, K represents the contribution to the potential of a harmonically pulsating unit source and is defined in Eq (2)

The integration region is limited by the Mach fore-cone emanating from the collocation point and by the Mach rays, emanating from the wing or control surface apex Within this region, $w \neq 0$ generally (figure 2)

The surface integral taken over Mach fore-cone is subdivided by the program into a number of surface integrals over rhombic or characteristic boxes, the edges of which are parallel to the Mach lines of the fore - and aft cones, respectively. Within each box, the downwash distribution w is assumed to be constant.

If the planform has a subsonic leading or trailing edge, or streamwise parallel side edges, then the concept of diaphragms according to Evvard is introduced into the M.B.B. procedure.

Whereas the source distribution or normal velocity distribution on the lifting surface is known, the source distribution placed on the diaphragm is unknown and has to be determined first by a step-by-step procedure

The method is applicable for wings with arbitrary subsonic leading or trailing edges and for wings with control surfaces. In the case of a subsonic leading edge, the upwash field of boxes cut by and lying in front of the leading edge are presumed to have a square root singularity.

The formulation of present M.B.B. lifting surface theory primarily supplies the velocity potential values Sectional or total loads are easily derived from these values by applying partial integration techniques

If the calculation of pressure coefficients is wanted, this can be achieved by using the formula.-

$$\Delta C_p = 4(\phi_x + ik\phi) \tag{5}$$

This means that a numerically-given function has to be differentiated For this reason, the function must be smoothed first Caution is demanded where the potential offers a kink, i.e. where the function cannot be differentiated.

6. DISCUSSION OF RESULTS

6.1 Comparison of Pressure Distributions

Theory and experiment were compared for four streamwise stations, namely stations 1, 3, 5 and 6 (figures 3, 4, 5, 6). The two theories agree well both in their real and imaginary parts At M = 1 3, the B A C theory predicts pressures forward of the nearly sonic hinge line, this being due to the smoothing technique used in computing pressures from control surface modes Although in all cases, both supersonic theories predict pressure maxima in the proximity of the control surface hinge line, the maxima obtained from wind tunnel tests are, in general, much greater This is probably due to the fact that neither theory accounts for the pressure singularity in the case of a subsonic hinge line (in all cases, the hinge line is subsonic) Along station 6, there occur the greatest deviations between test and theory, with a shift of the test pressure maximum aft

In some test measurements (e.g stations 3 and 5, figure 6), there appears a secondary maximum of pressure, smaller and aft of the hinge line peak This is perhaps evidence of shocks, which are not considered by either theory.

6 2 Comparison of Local Lift and Moment Distributions

In figure 7, local lift and moment derivatives plotted versus span are shown The parameters are M = 1 1 and f = 145 Hz corresponding to k = 0 80

The supersonic theories agree fairly well, but the comparison between theory and experiment does not show satisfactory correlation in all cases, especially for the out-of-phase derivatives The centre of pressure is further outboard in the tests than predicted by theory and, when torsion effects are present in an aircraft flutter analysis, this could possibly lead to larger deviations

Figure 8 presents sectional lift and moment distributions for M 1 1 and f = 260 Hz corresponding to k = 1 40

In this case, there is good agreement between the two theories But both predictions overestimate the real parts of lift and hinge moment at the inboard stations and underestimate them near the outboard control side edge For the imaginary parts of lift and moment, there are considerable deviations especially outboard

The data for M = 1 3 and f = 148 Hz, corresponding to k = 0 70, are given in figure 9 With this parameter configuration, we have deviating predictions for the local lift distribution whereas M B B and B A C predictions for the local hinge moment distribution agree well

The experimental c_l' distribution is well predicted by B.A.C. theory whereas the distribution c_l'' is well predicted by M.B.B. The in-phase hinge moment is over-estimated in all cases. The out-of-phase moment coefficients are in good agreement with test values

Figure 10 presents data for M = 1.3, f = 260 Hz and k = 1.24. As in the previous graph, the agreement of the experimental spanwise lift with theory depends on the procedure considered. For the imaginary coefficients of c_l and c_h, we can observe minor differences at the inboard stations and greater discrepancies at the outboard station

It seems a feature of the sectional lift that theory is greater than test near the root chord and vice-versa at the tip. If the planform is not mirrored exactly which is probable, due to the presence of a boundary layer near to the tunnel wall, then there will be differences in results between test and theory, becoming less significant as the planform tip is approached, since the theories both assume a perfect mirror image. On station 1, it can therefore be expected that theory will produce results of larger magnitude than test, and this is shown to be so at least for real parts. Imaginary parts of local derivatives are, unfortunately, much harder to predict consistently. This is fundamentally due to their being the difference between two relatively large opposing effects.

6.3 Comparison of Total Lift and Total Hinge Moment Coefficients

Figures 11, 12, 13 present the variation of the total lift and total hinge moment coefficient with reduced frequency. Correlation between the results from both theoretical methods is good. This is not contradictory to the results obtained from comparisons of local loads, since sectional over-estimations and under-estimations will partially cancel each other when integrated to obtain total coefficients.

The theoretical values of lift and hinge moment are greater in magnitude than those of experiment, and for both C_L' and C_H', the trends with k and M are correctly represented Again, it is found that the imaginary parts are much more difficult to predict consistently. C_H'' is particularly critical since it should be kept positive, to avoid one degree of freedom flutter. It can be seen from figures 11-13 that this is so for all cases considered

In this case, increasing k is favourable and increasing M is unfavourable, both these effects being predicted.

7 CONCLUSIONS

Both theories in this report are linearised and purely supersonic, whereas the true flow is mixed, especially at the lower Mach number. The Stark theory tends to produce higher pressure magnitudes than Sauler-Allen, and a better correlation with test results This is presumably due to the different hinge line treatments. In terms of overall forces, however, the two theories, one being an integrated downwash method, the other an integrated potential method, are in good agreement

In the context of aircraft flutter analysis, therefore, the differences between these two theoretical treatments are not very significant In general, trends demonstrated by experiment are followed reasonably by both theories, but it is of great importance that both theoretical and experimental studies are of very high accuracy, so that control surface flutter status can be established with confidence at low supersonic conditions

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Acknowledgement:

The authors wish to express their appreciation to M. Burt, B.A.C., Warton, for his technical contributions to this paper.

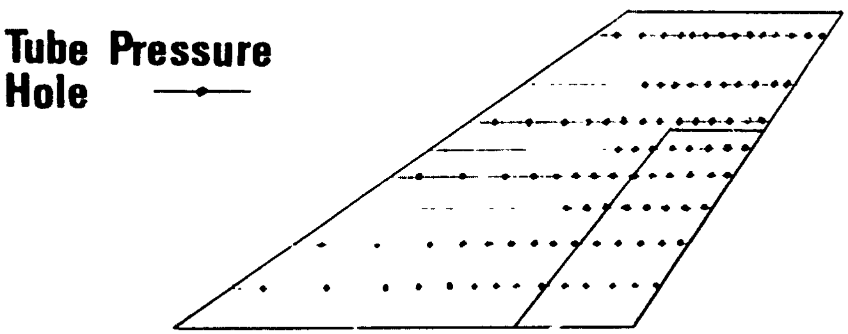


Fig. 1a The location on the model of the tube pressure holes

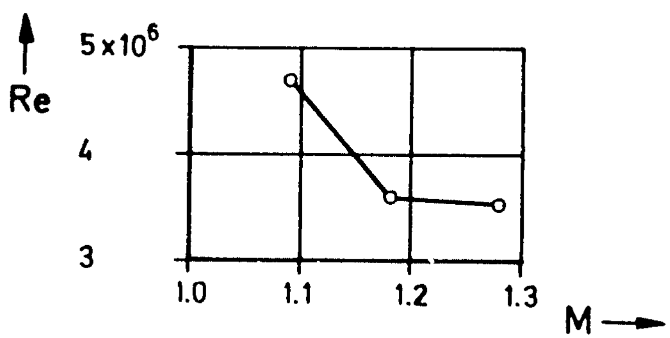
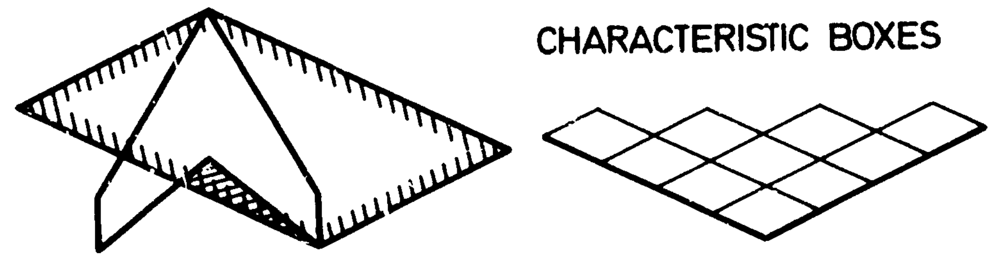


Fig. 1b Reynolds number (based on mean chord) vs. Mach number



INTEGRATED DOWNWASH

$$\phi \langle x,y \rangle = \iint w \langle x',y' \rangle \; K_1 \langle x-x', y-y' \rangle \, dx'dy'$$

$$K_1 = 0 \left\langle \frac{1}{R} \right\rangle$$

INTEGRATED POTENTIAL

$$w \langle x,y \rangle = \iint \phi \langle x',y' \rangle \; K_2 \langle x-x', y-y' \rangle \, dx'dy'$$

$$K_2 = 0 \left\langle \frac{1}{R^5}, \frac{1}{R^3}, \frac{1}{R} \right\rangle$$

FIG. 2 SUPERSONIC LIFTING SURFACE METHODS

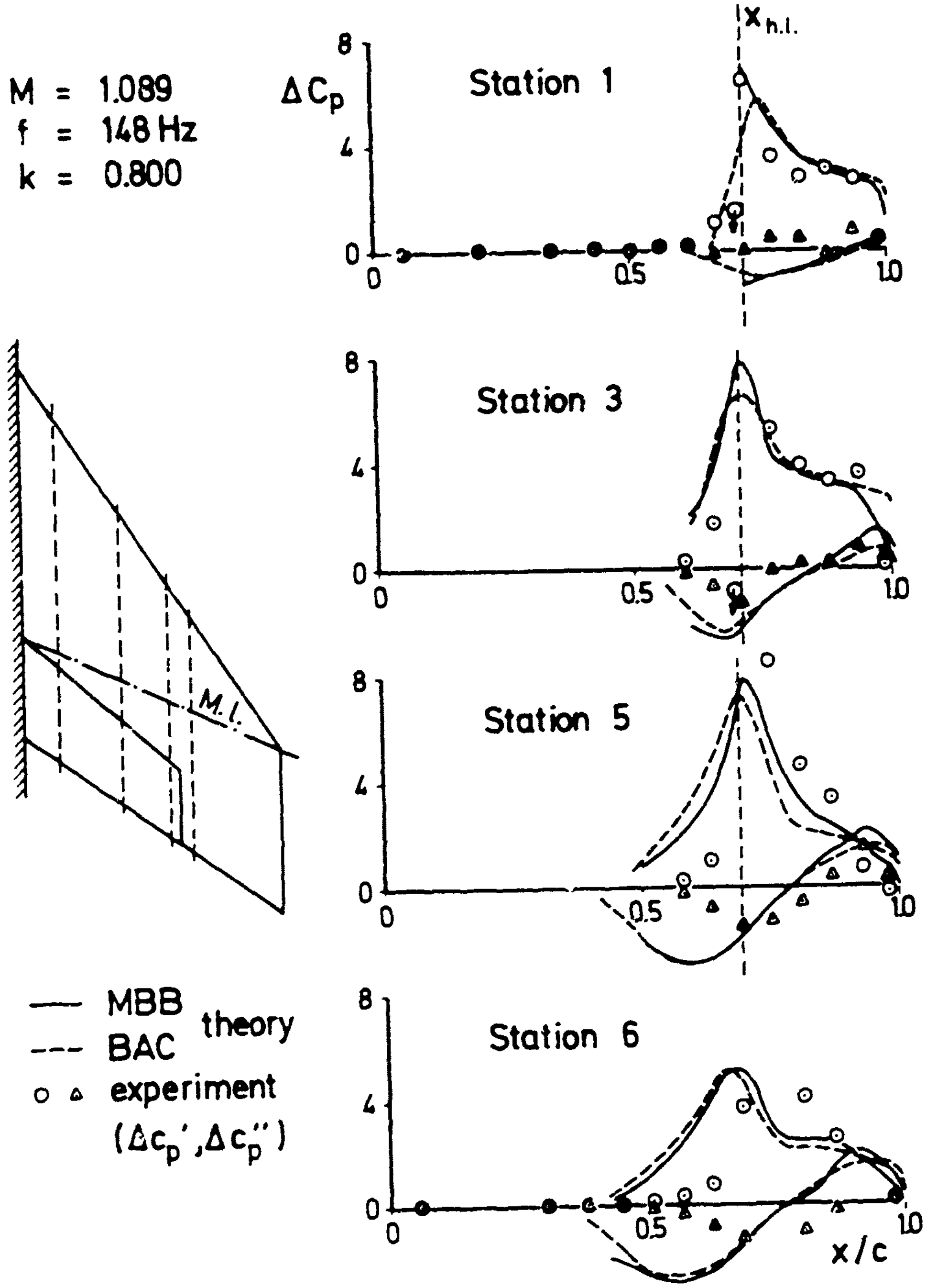


Fig. 3 Experimental and theoretical pressures at test stations

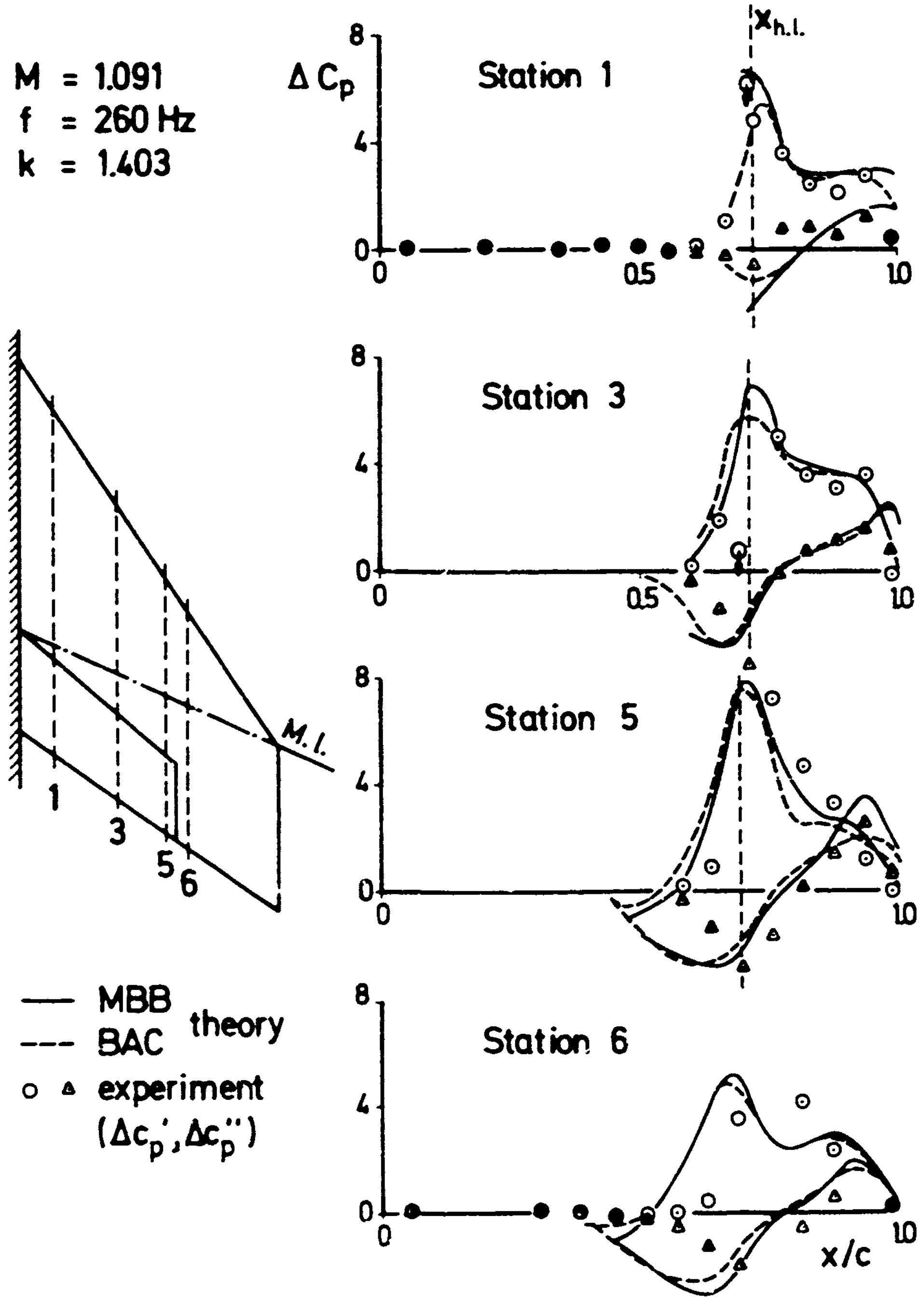


Fig. 4 Experimental and theoretical pressures at test stations

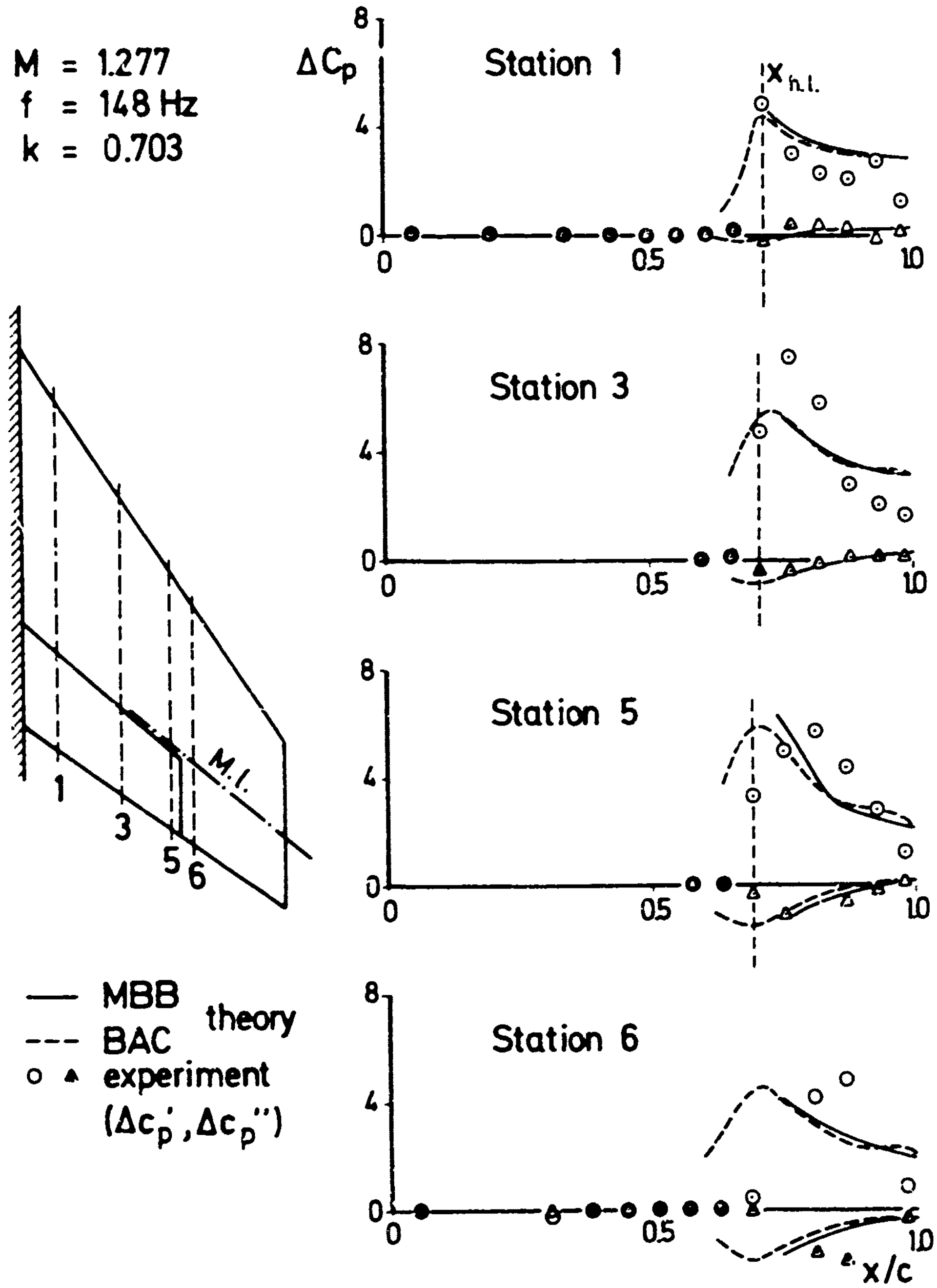


Fig. 5 Experimental and theoretical pressures at test stations

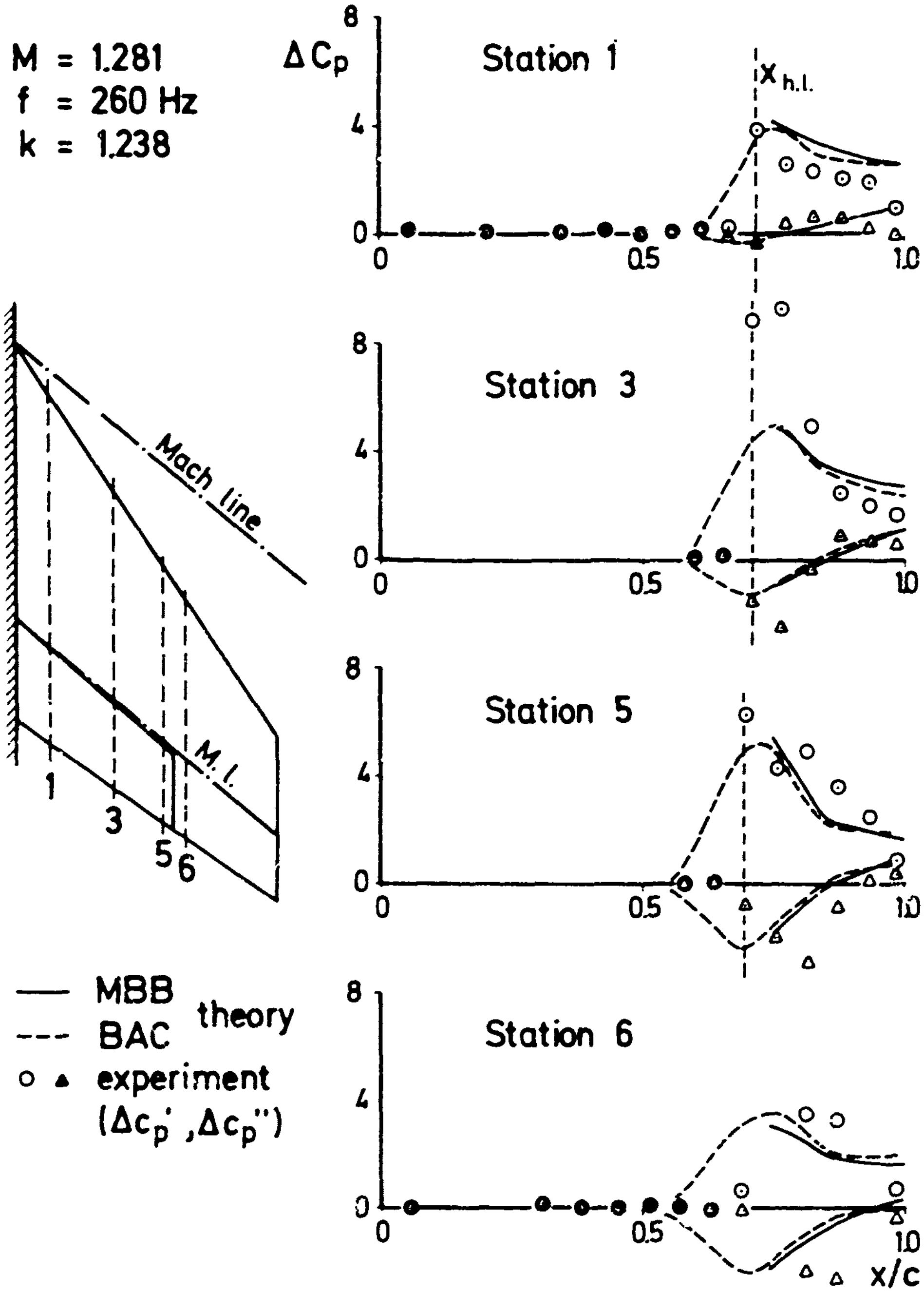


Fig. 6 Experimental and theoretical pressures at test stations

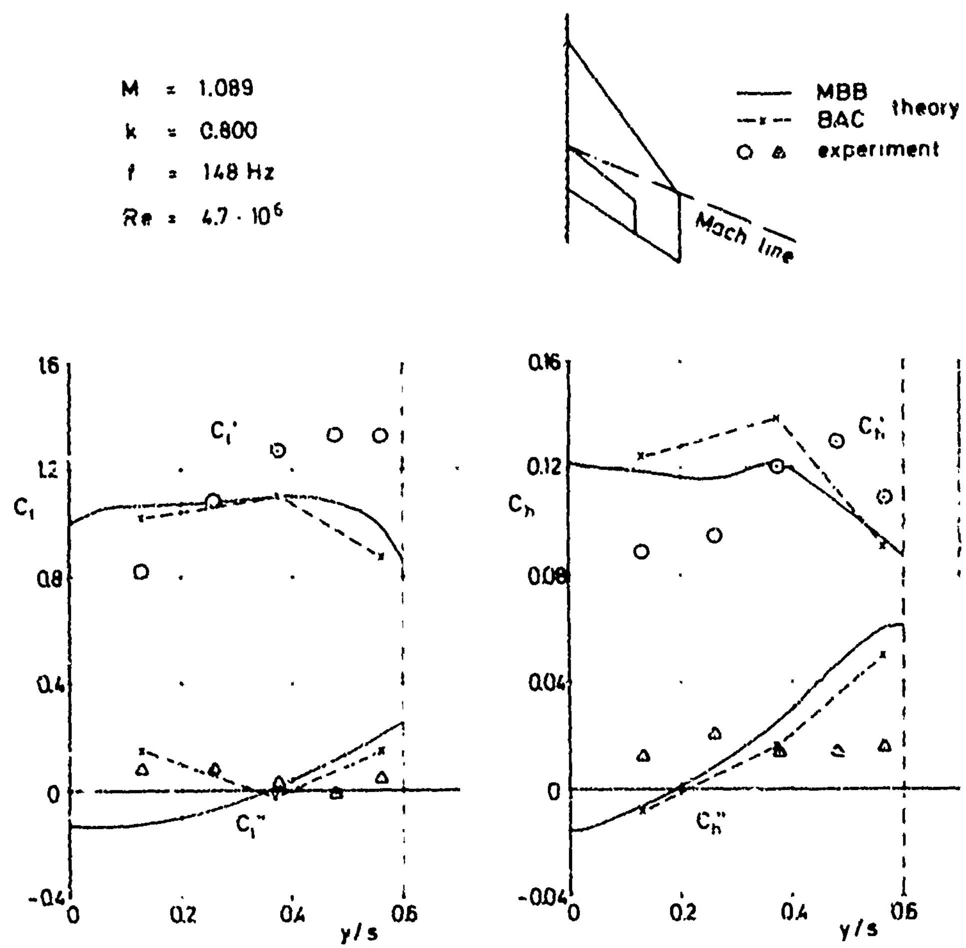


Fig. 7 Sectional control surface lift- and hinge moment coefficient versus span

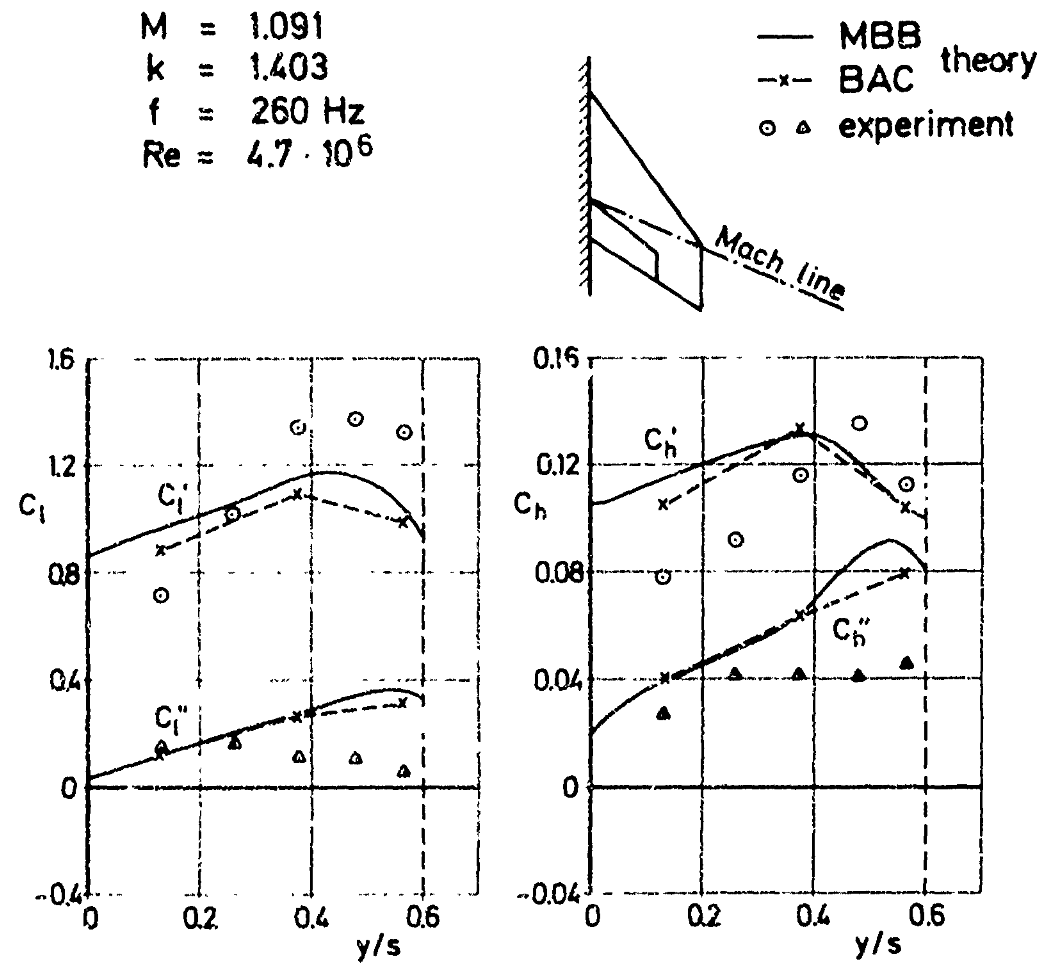


Fig. 8 Sectional control surface lift- and hinge moment coefficient versus span

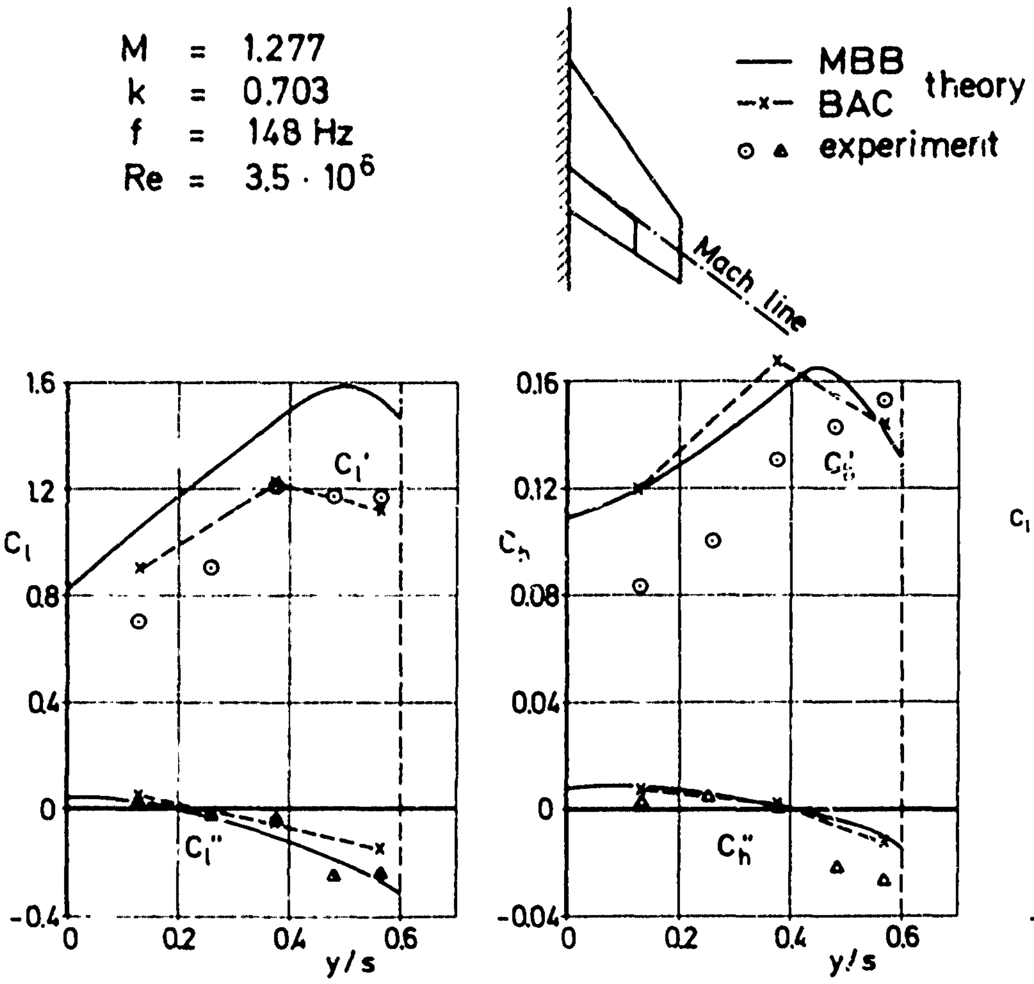


Fig. 9 Sectional control surface lift- and hinge moment coefficient versus span

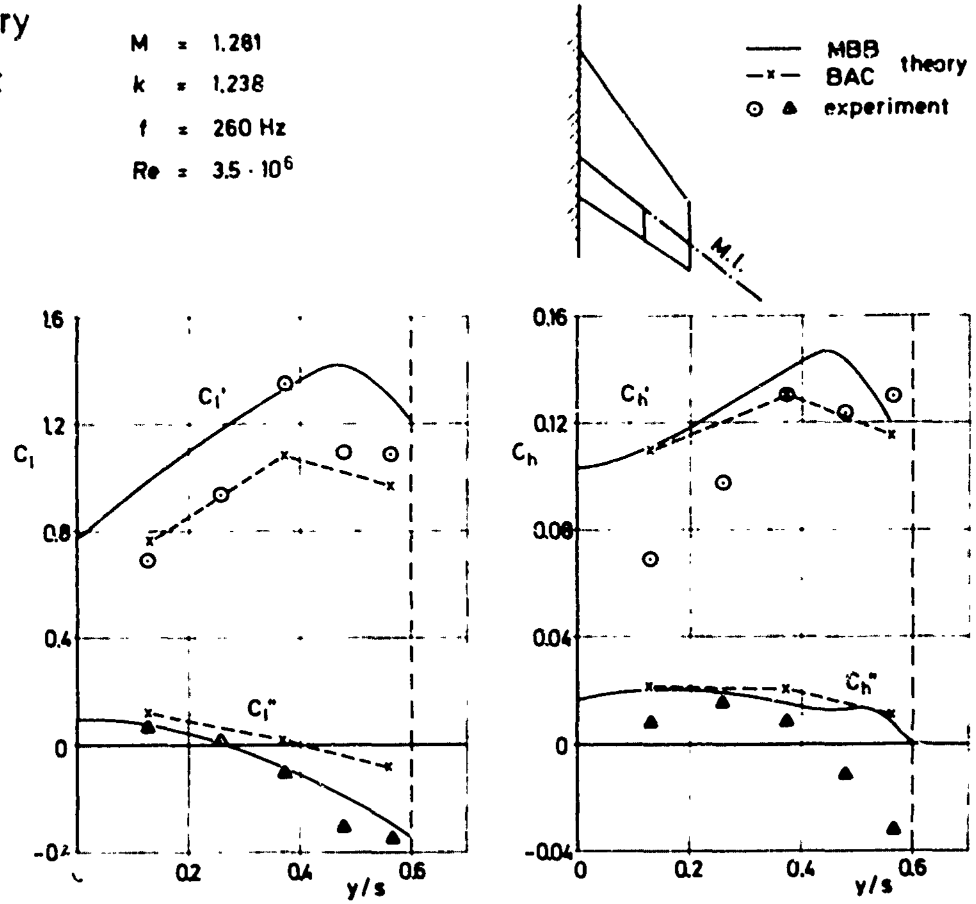


Fig. 10 Sectional control surface lift - and hinge moment coefficient versus span

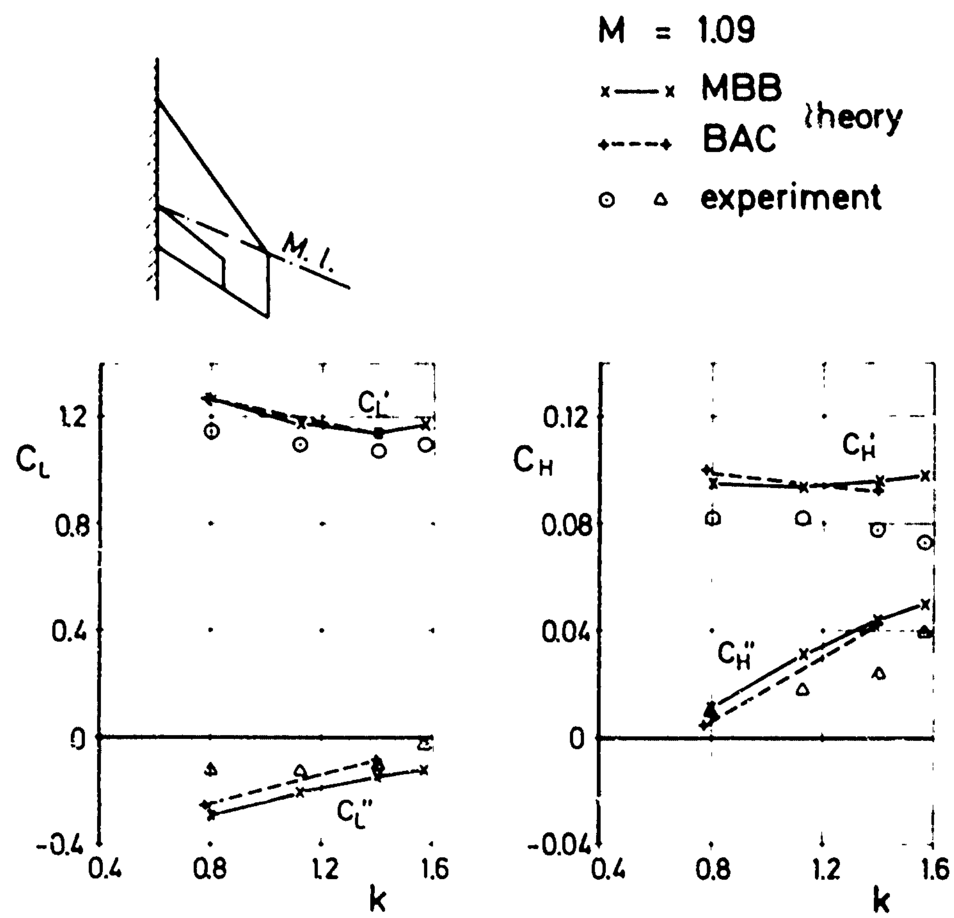


Fig. 11 Total lift- and total hinge moment coefficients versus reduced frequency

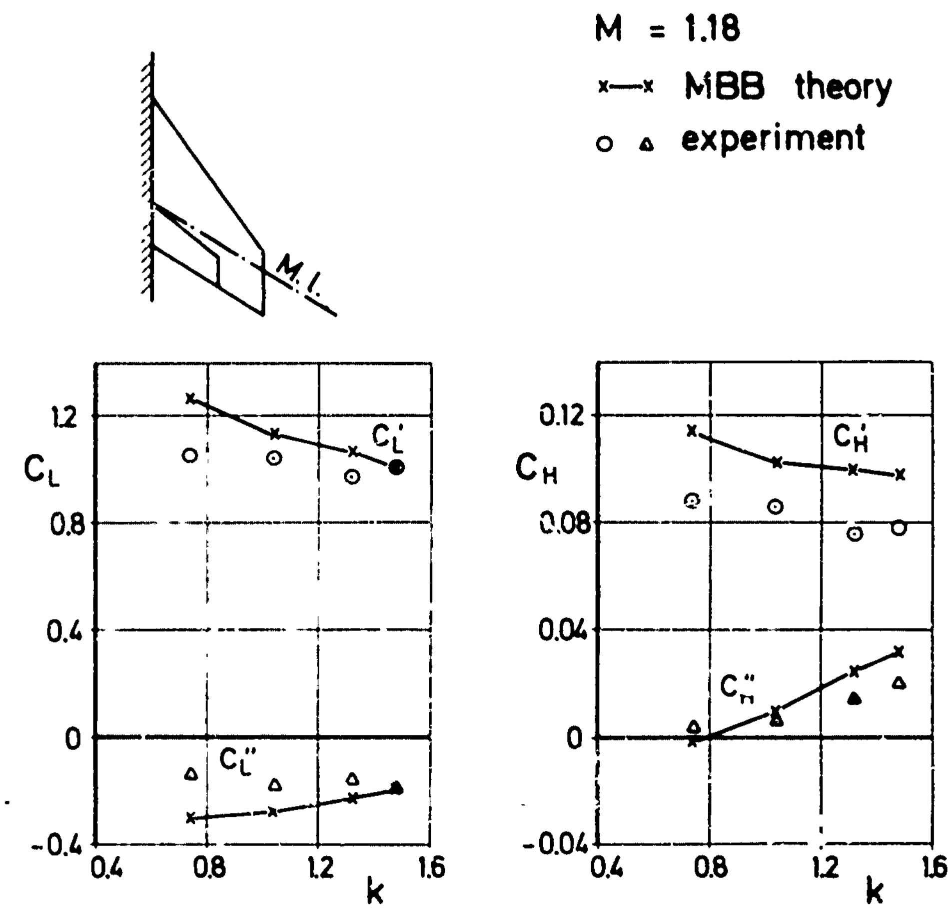


Fig. 12 Total lift- and total hinge moment coefficients versus reduced frequency

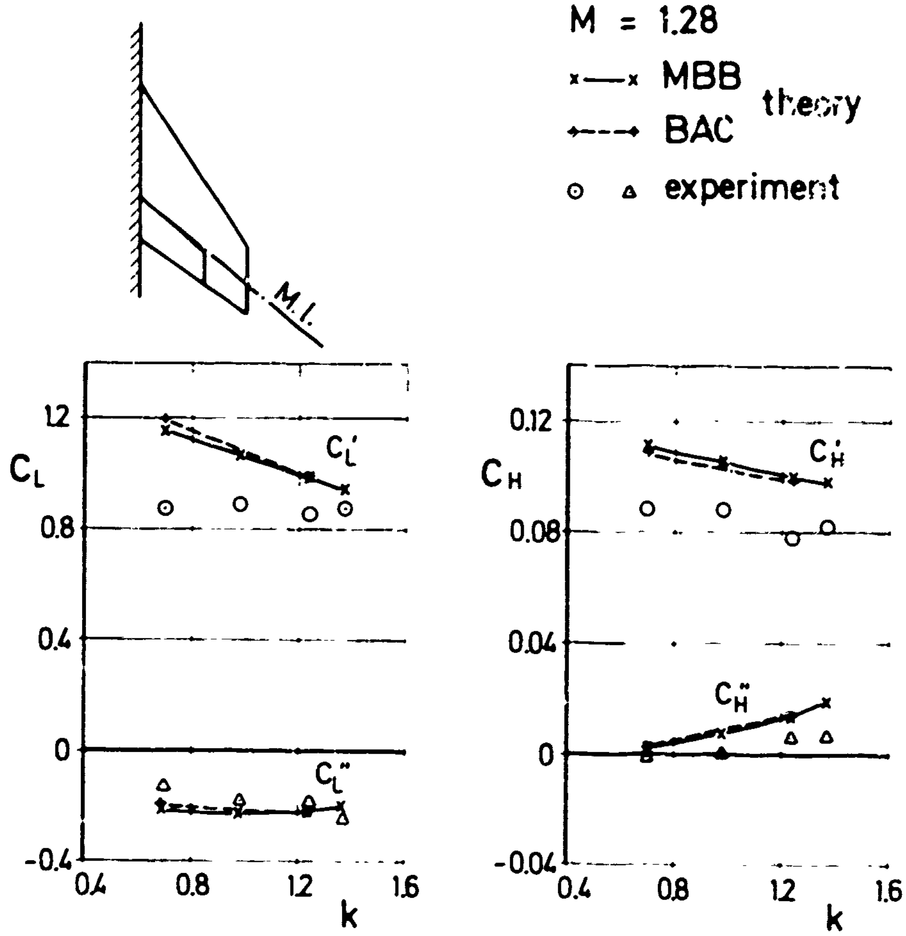


Fig. 13 Total lift- and total hinge moment coefficients versus reduced frequency

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ISBN 92-835-1223-5