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(NASA-CR-146660) [ILIIAC 4 AND LIFTING SURFACE THEORY WITH BOUNDARY LAYER] Semiannual Progress Report, 1 Oct. 1975 - 1 Apr. 1976 (Princeton Univ.) 21 p HC $3.50 CSCL 01A G3/02

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TECHNICAL DISCUSSION

During the present reporting period progress has been made in the following areas:

Illiac IV

The aerodynamic panel flutter program has been re-written in CFD language and is now running on the IBM 360-67 simulator.

Lifting Surface Theory with Boundary Layer

Major progress has been made in this area.

(1) Subsonic and Supersonic, Two-dimensional Steady Flow (M.H. Williams). This work is completed and a report is being prepared. As expected shear layer effects become less important for high supersonic Mach number and are most prominent in the transonic regime. Figure 1 shows the lift curve slope of a flat plate airfoil vs. Mach number for various boundary layer thickness to airfoil chord ratios, $\delta/_C$. Typical pressure distributions are shown in Figure 2 and 3.

(2) Incompressible, Two Dimensional, Unsteady Flow with Control Surfaces (C. S. Ventres, Bolt, Beranek and Newman). Numerical results are showing substantial sensitivity as might be expected from previous potential flow work with control surfaces. Efforts are underway to overcome this problem. This work is described in a recent publication (Ref.1).

(3) Improved Unsteady Theory (M.R. Chi and M.H. Williams). Higher order terms in $\frac{\omega\delta}{U_\infty}$ have been included to assess the accuracy of presently available shear flow models. A typical result is shown in Figure 4 where the real part of the lift curve slope is plotted vs. reduced frequency for three different theoretical

models. $m = 0$, $n = 1$ is the potential flow result; $m = 1$, $n = 1$ the simplest model incorporating shear flow effects and $m = 2$, $n = 1$ is a higher order theory including shear flow effects. For $\delta/_C = .1$ of this example, good convergence is shown for $\frac{\omega c}{U_\infty} < .5$ by comparing results of the two models which include shear flow effects.

(4) Combined transonic airfoil thickness and shear layer thickness effects. (E. H. Dowell).

Work continues as to how these two effects can be combined in a single analysis. For

$$\delta/_C \left(\tau/_C\right)^{1/3} << 1$$

$\tau \equiv$ airfoil maximum thickness

$C \equiv$ airfoil chord

a relatively simple theoretical model appears possible.

(5) A report (Reference 2) has been completed describing the generalization to boundary layer thickness which vary along the airfoil chord. (M. R. Chi).

(6) Bending-Torsion Flutter Calculations (M. R. Chi).

Using the simplest unsteady shear flow model, flutter calculations have been carried out for a typical section airfoil. In convential notation[3] the results are shown in Figure 5, 6 and 7. For small bending/torsion natural frequency ratios, $\omega_h/_{\omega_\alpha} << 1$, the shear flow increases flutter velocity. For larger $\omega_h/_{\omega_\alpha}$ the trend is reversed. It should be noted, however, that for large $\omega_h/_{\omega_\alpha}$, the reduced frequency based upon boundary layer thickness may be sufficiently large that the theory becomes inaccurate.

An examination of the aerodynamic forces (see Figure 8-15) shows that the shear layer affects their magnitude much more than their phase angle. Hence the trends in flutter behavior may be a result of both (destabilizing) aerodynamic stiffness and (stabilizing) aerodynamic damping terms being decreased by the shear flow. The aerodynamic damping terms are relatively more important at high reduced frequencies which occur at larger ω_h/ω_α.

BUDGETARY DISCUSSION

One graduate student and one research staff member have worked on this program at Princeton during the reporting period in addition to the principal investigator. One research staff member at Bolt Beranek and Newman Inc. has also worked under subcontract during this reporting period.

REFERENCES

1. C. S. Ventres, "Non-Steady Shear Flow Lifting Surface Theory", Bolt, Beranek and Newman. Report No. 3235, 30 January 1976.
2. M. R. Chi, "Steady Incompressible Variable Thickness Shear Layer Aerodynamics", Princeton University AMS Report No. 1259, January 1976.
3. R. L. Bisplinghoff and H. Ashley, "Principles of Aeroelasticity", John Wiley and Sons, Inc., 1962.

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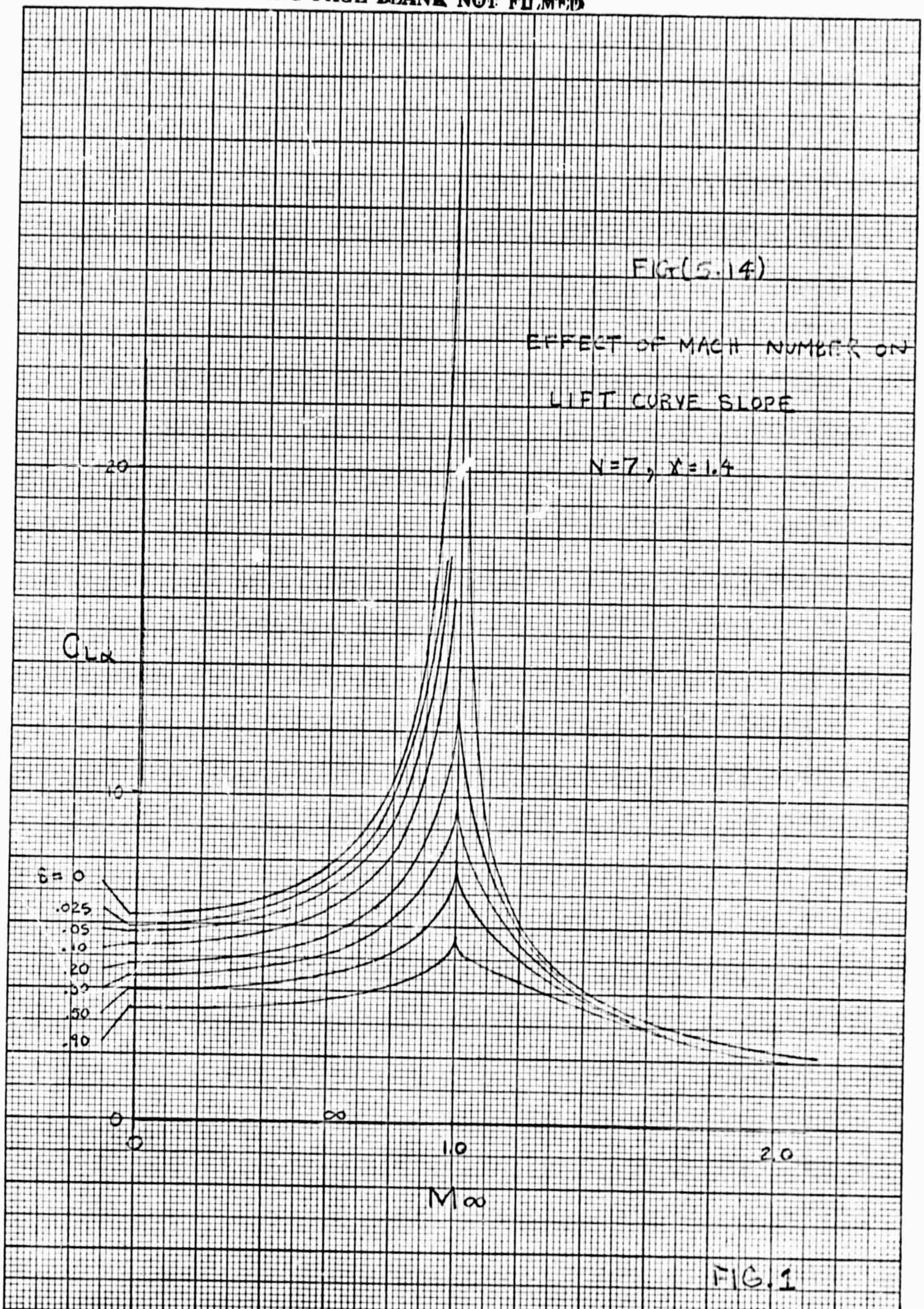


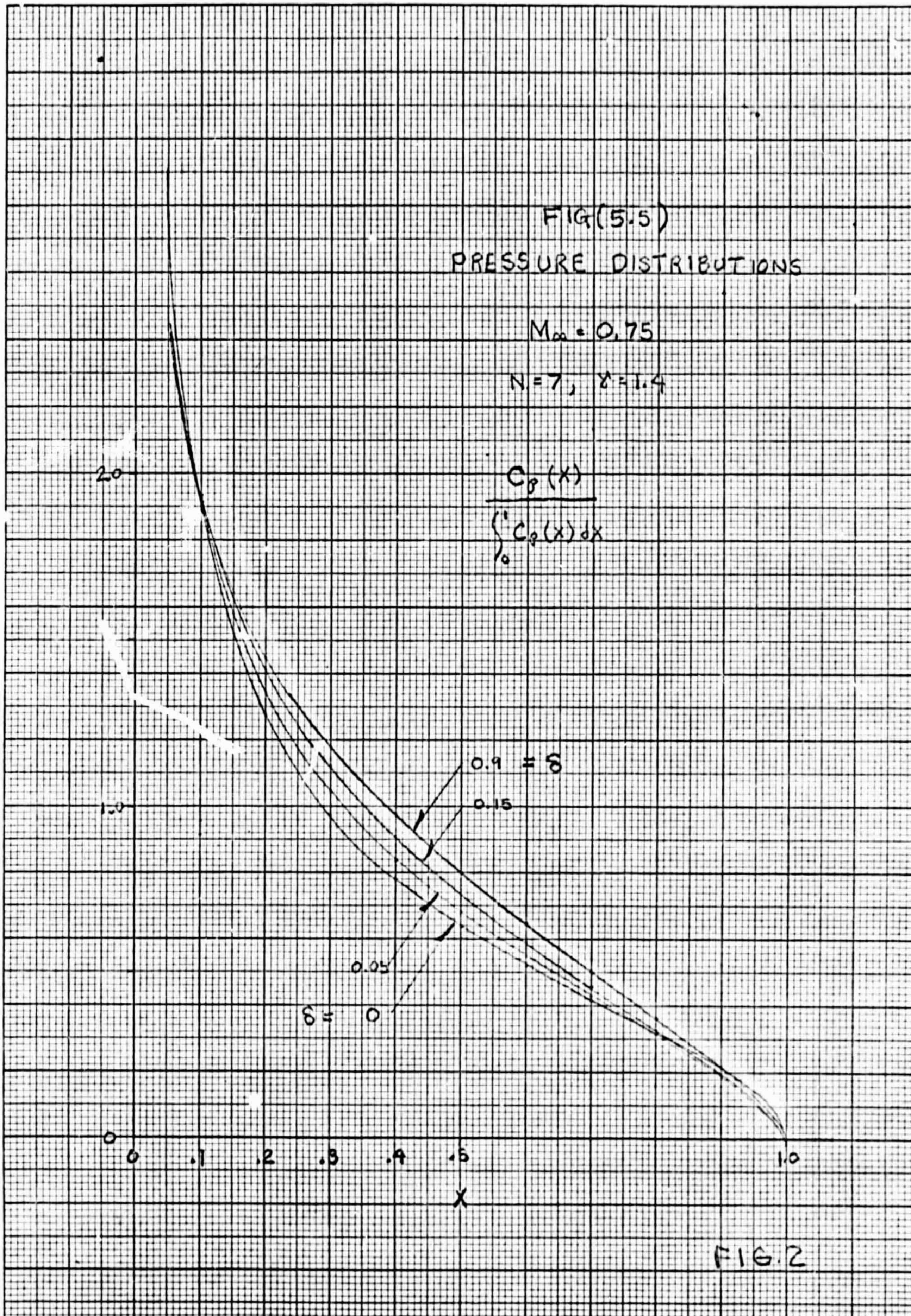
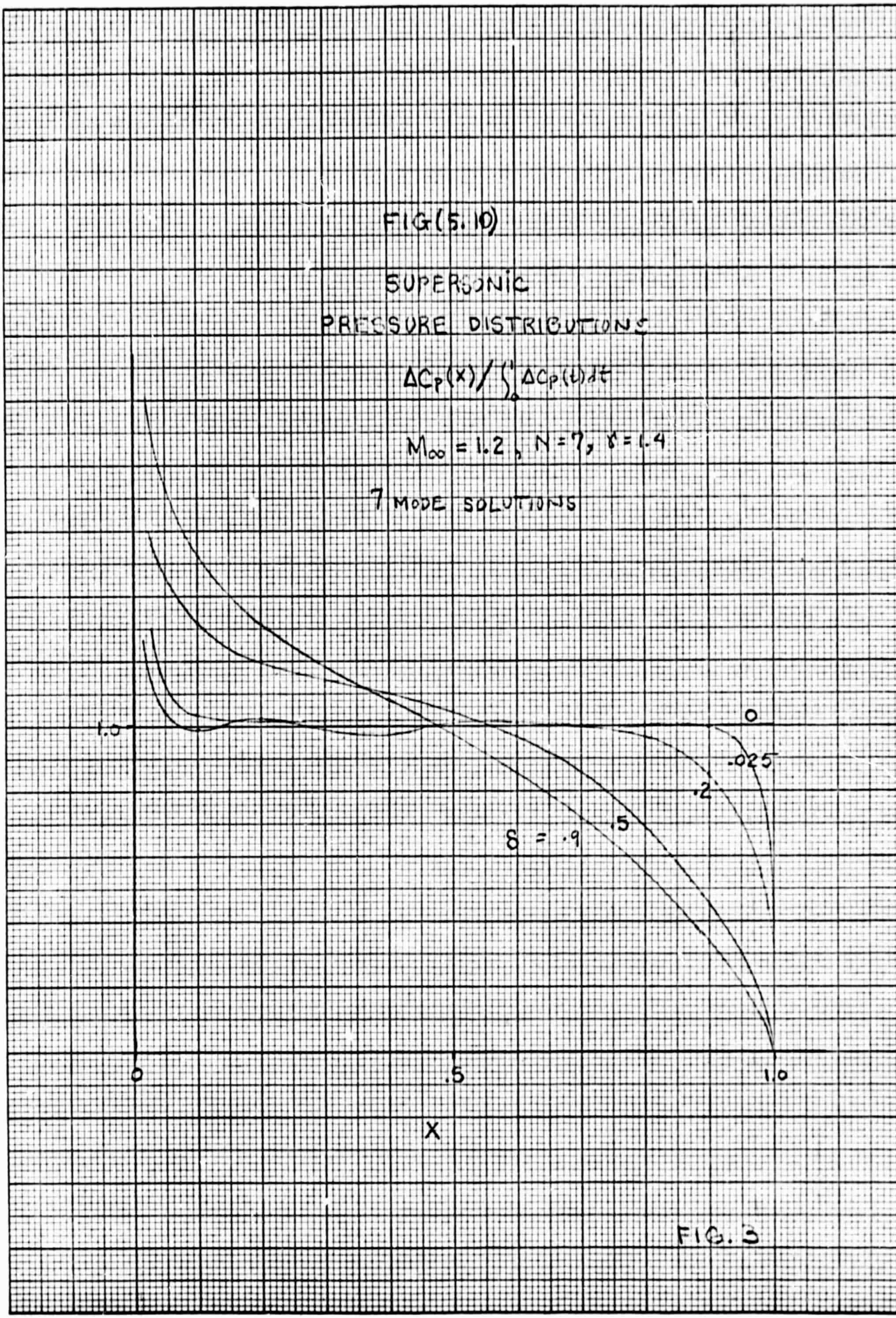


FIG (5.5)
PRESSURE DISTRIBUTIONS
$M_\infty = 0.75$
$N = 7$, $\gamma = 1.4$
$\dfrac{C_p(x)}{\int_0^1 C_p(x)\,dx}$
0.9 = δ
0.15
0.05
δ = 0
2.0
1.0
0
0
.1
.2
.3
.4
.5
1.0
X

FIG. 2



FIG(5.10)
SUPERSONIC
PRESSURE DISTRIBUTIONS
$\Delta C_p(x) / \int_0^1 \Delta C_p(t)\,dt$
$M_\infty = 1.2$, $N = 7$, $\gamma = 1.4$
7 MODE SOLUTIONS
1.0
0
.025
.2
.5
$\delta = .9$
0
.5
1.0
X

FIG. 3

pitching

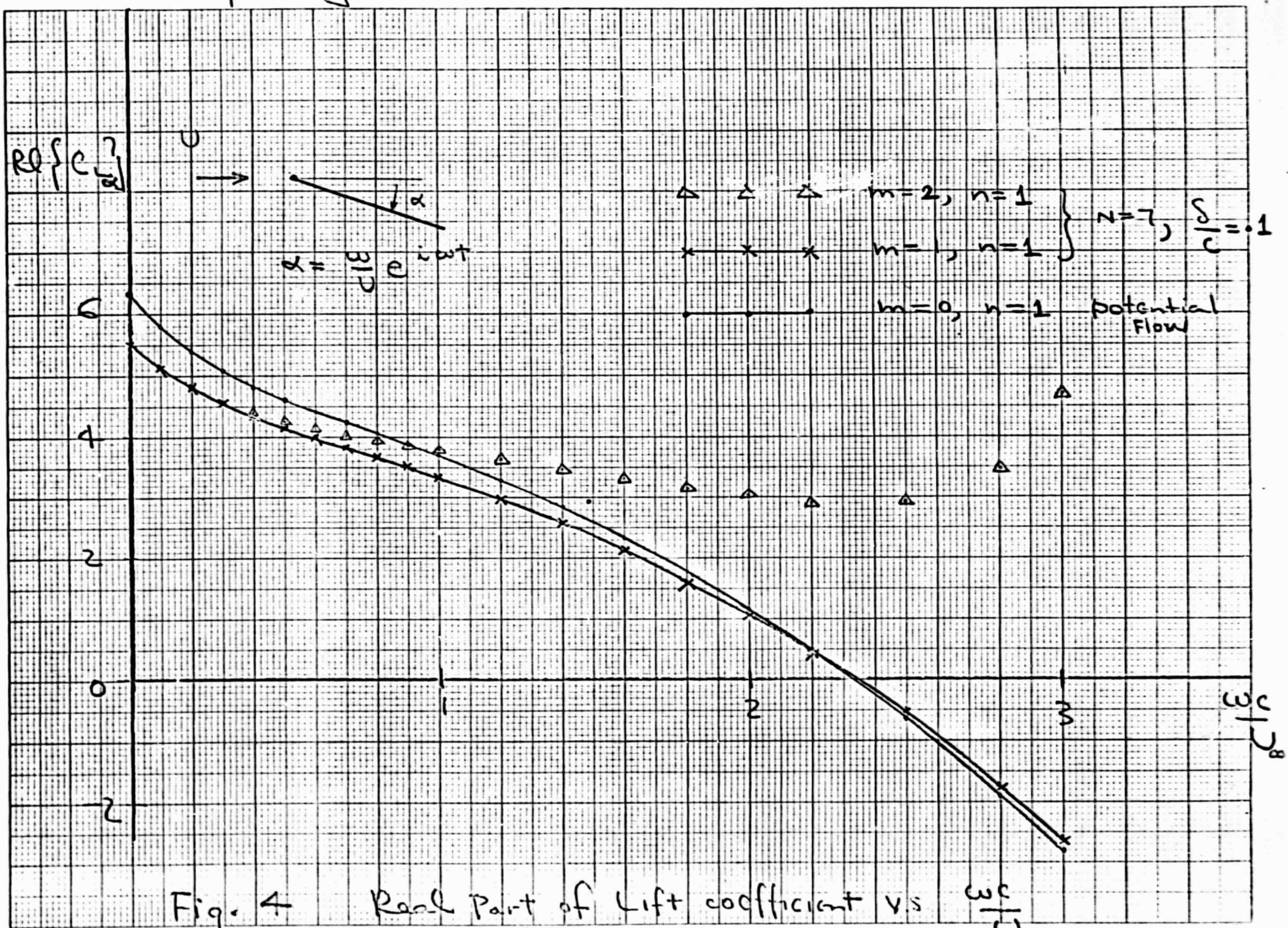


Fig. 4 Real Part of Lift coefficient vs $\frac{\omega c}{U}$

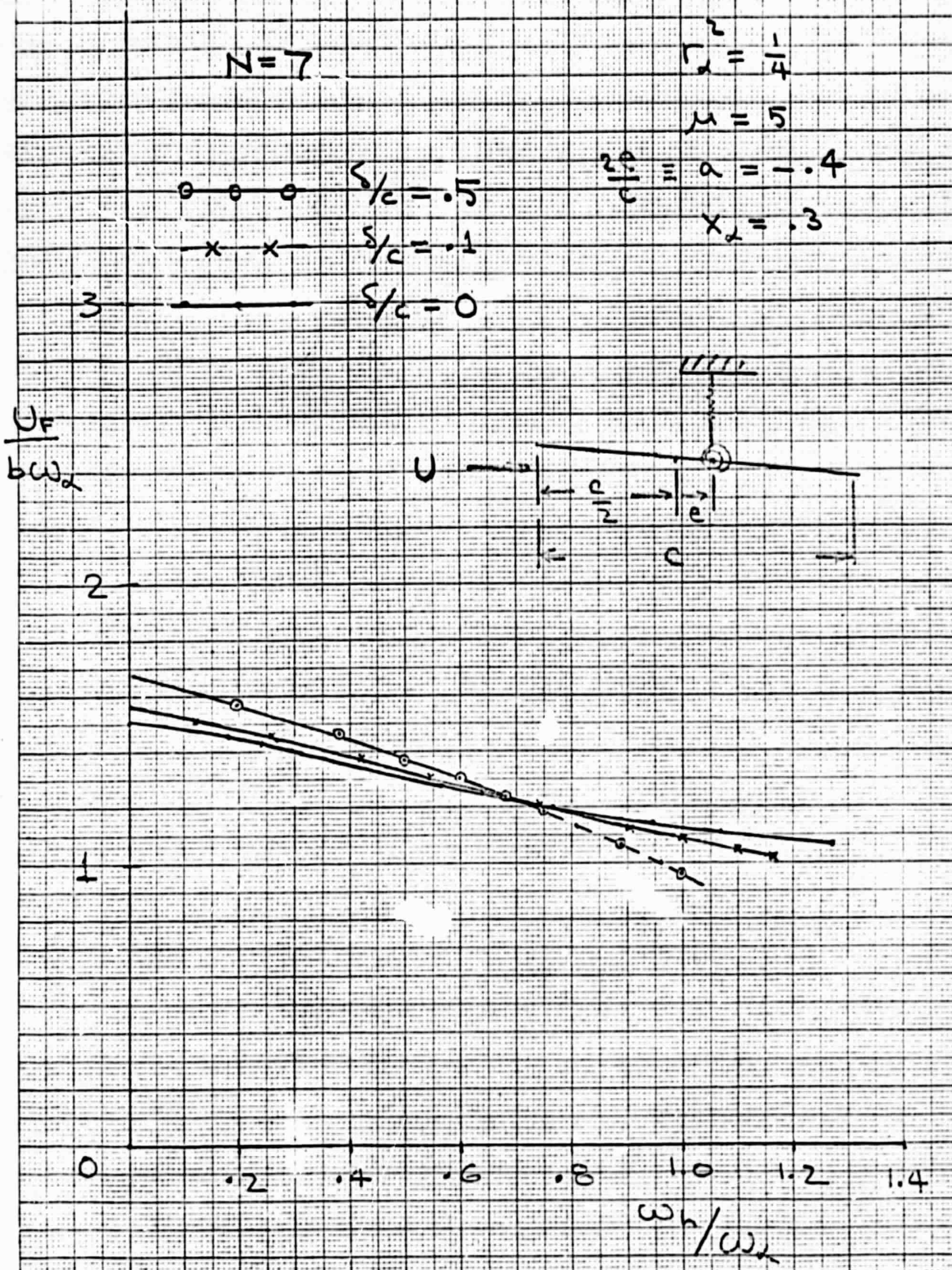


Fig. 5 The Flutter Speed v.s. The Frequency Ratio

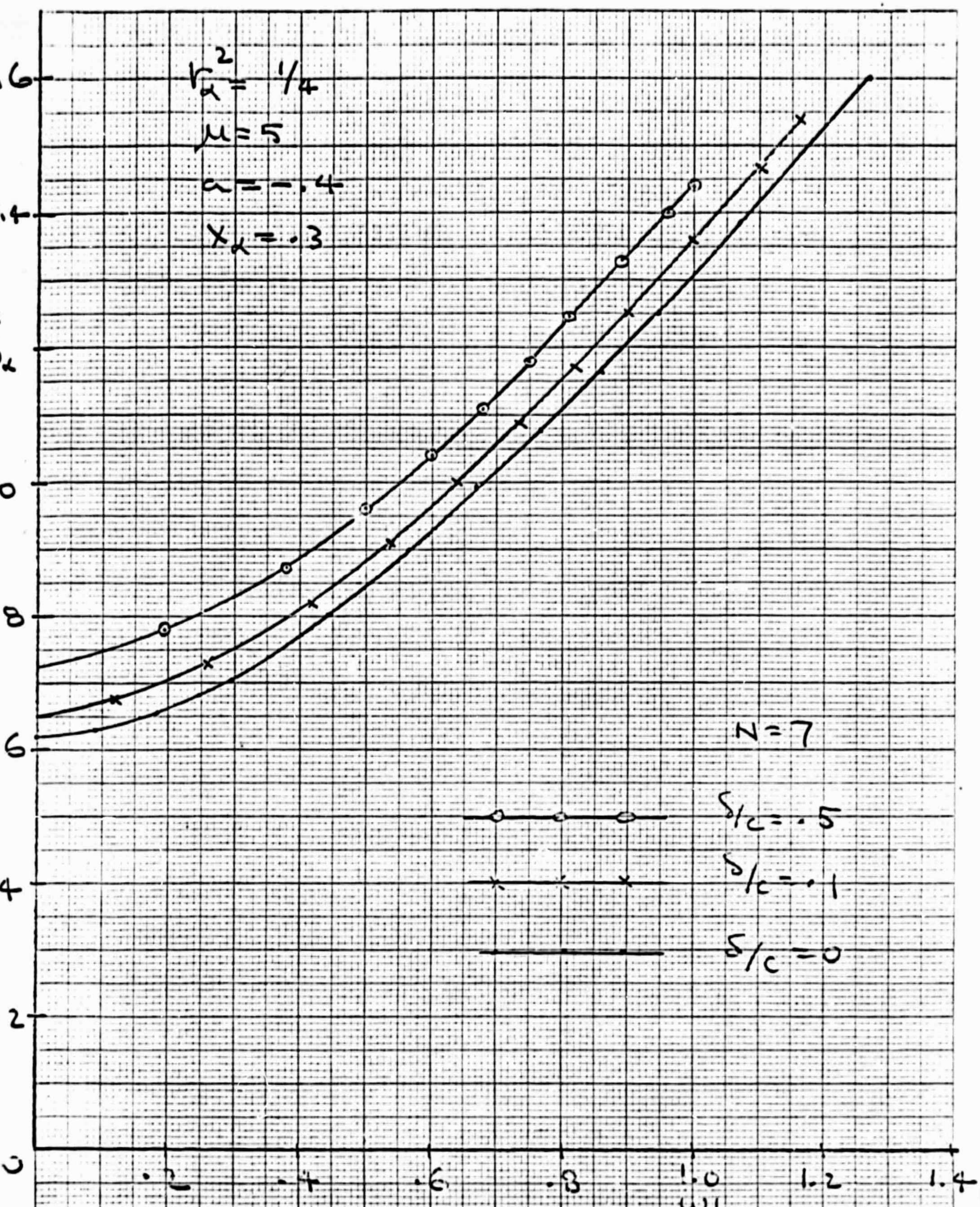
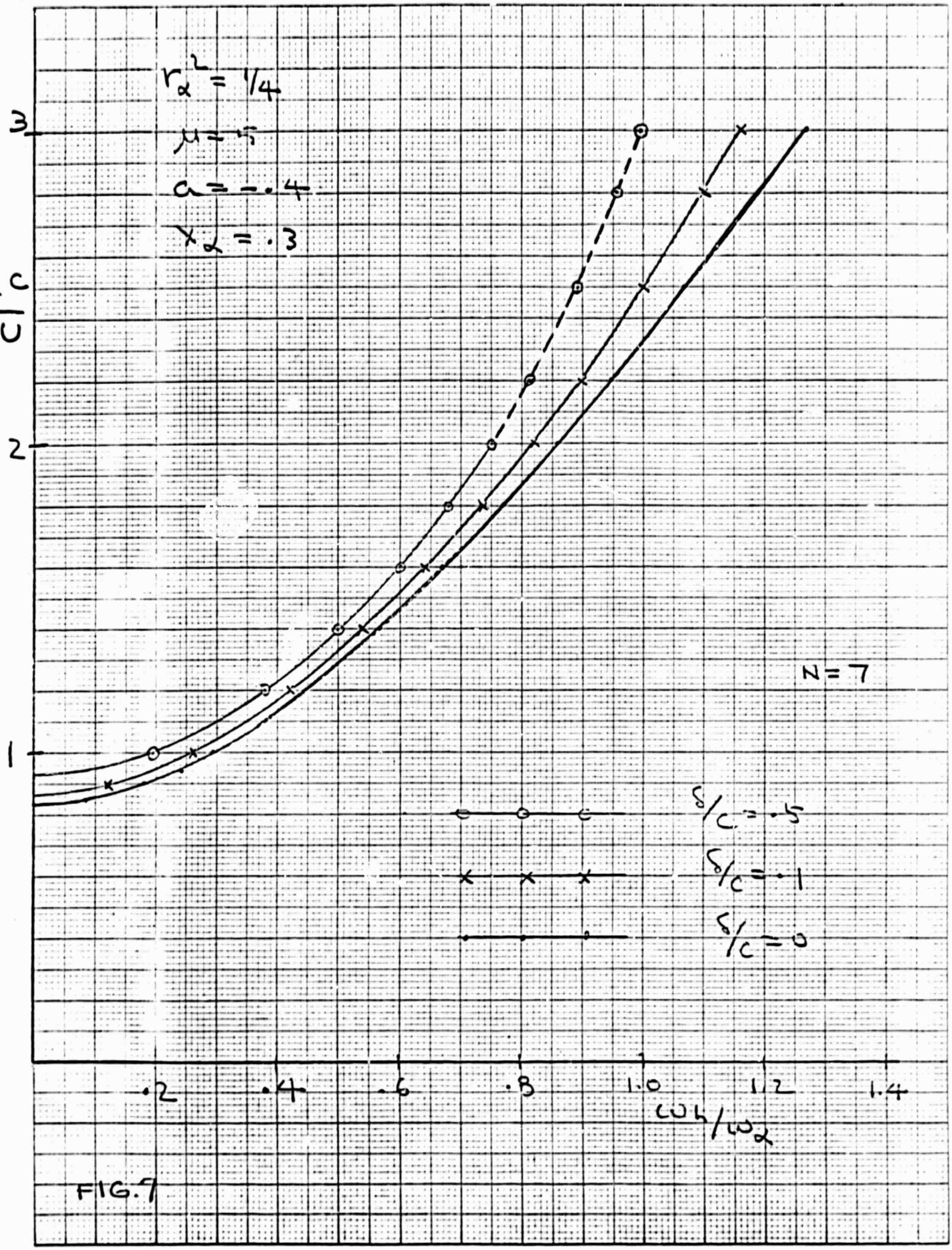
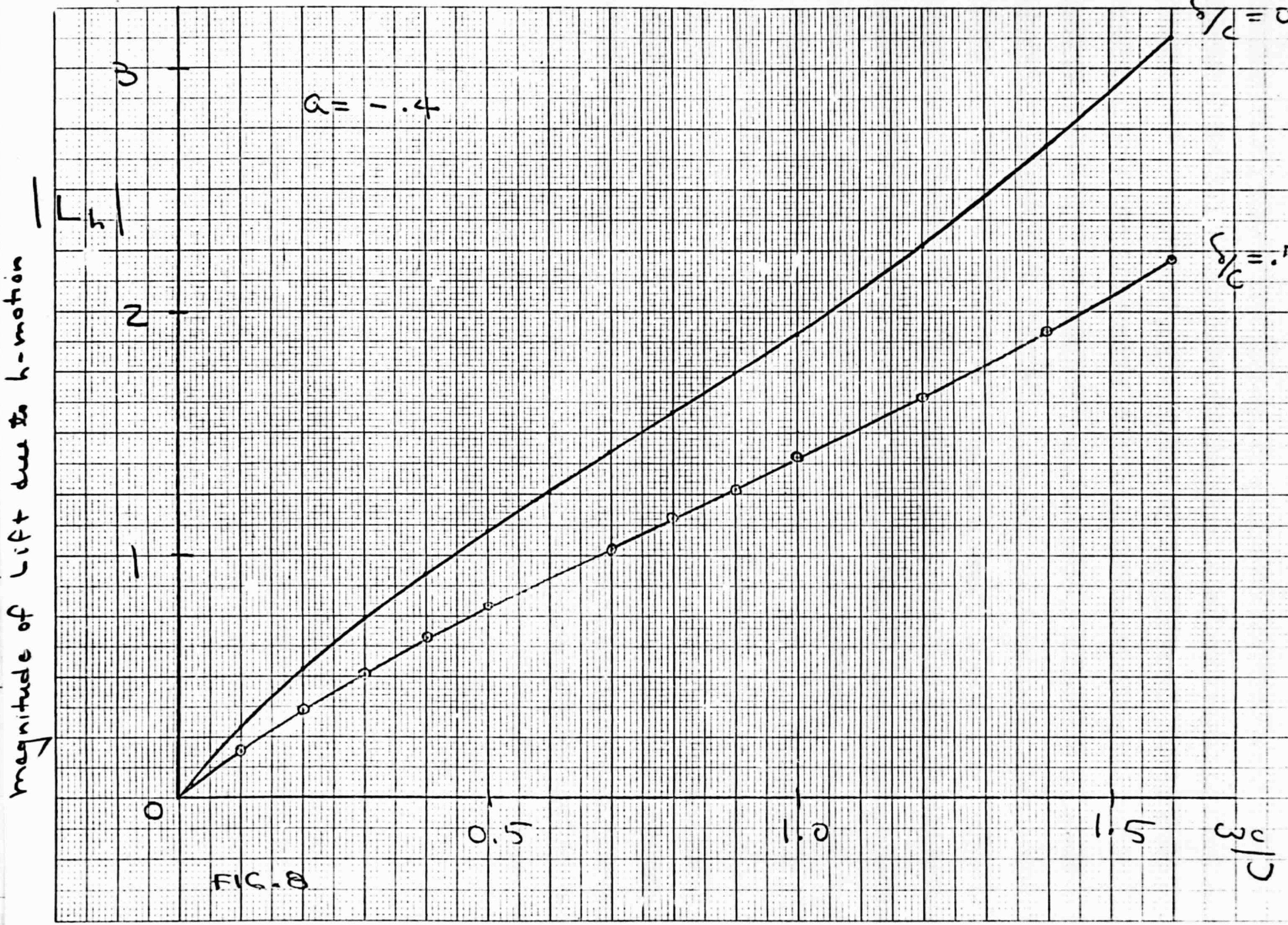


Fig. 6 Flutter Frequency v.s. Frequency Ratio



$r_\alpha^2 = 1/4$
$\mu = 5$
$a = -.4$
$x_\alpha = .3$
$\frac{\omega_F}{C} c$
3
2
1
N = 7
$\delta/c = .5$
$\delta/c = .1$
$\delta/c = 0$
.2
.4
.6
.8
1.0
1.2
1.4
ω_h/ω_α

FIG. 7



a = -.4
δ/c = 0
δ/c = .5
|L_h|
magnitude of Lift due to h-motion
3
2
1
0
0.5
1.0
1.5
ωc/U

FIG. 8

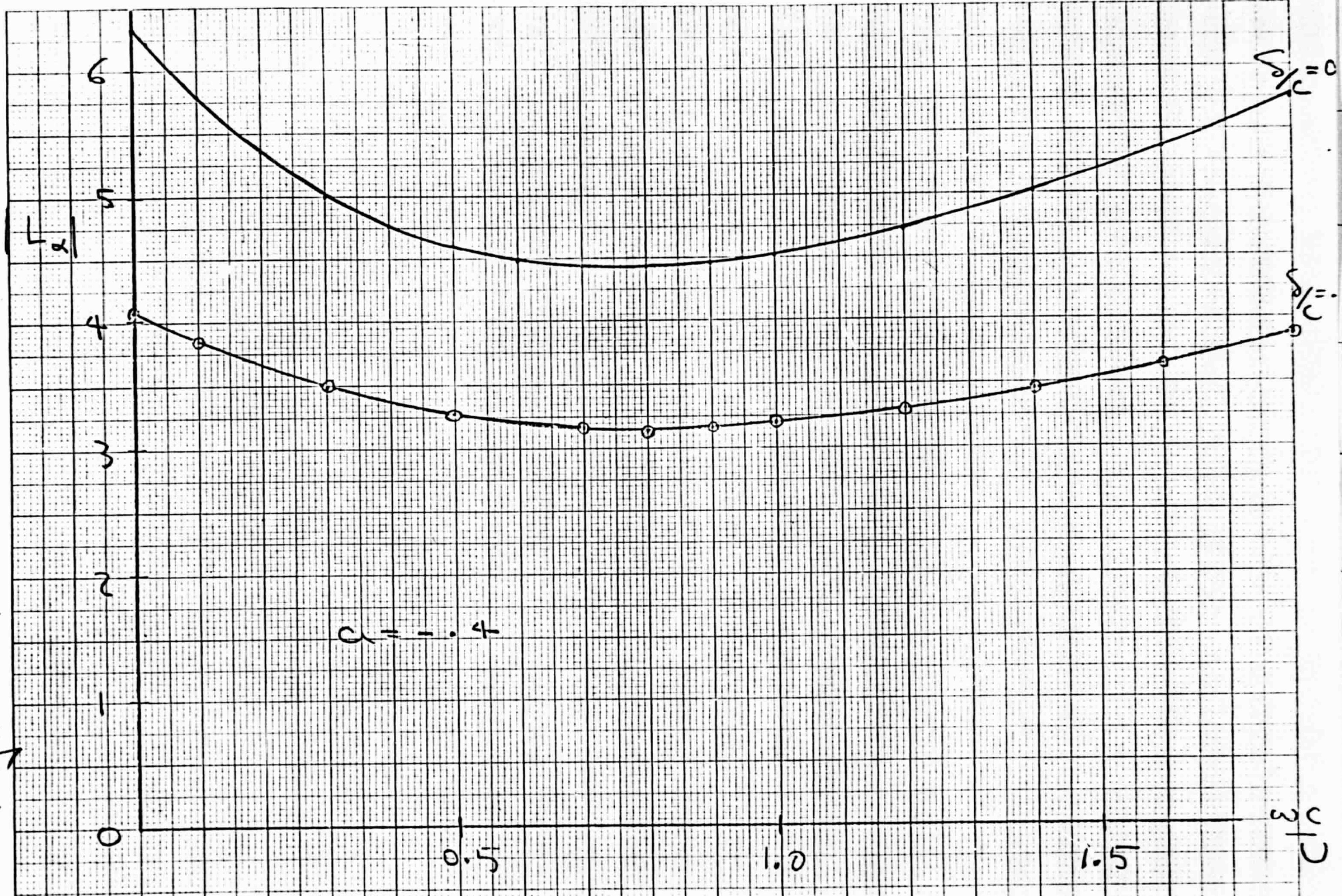

6
5
4
3
2
1
0
|Lα|
Magnitude of Lift due to α-motion
δ/c = 0
δ/c = .
a = -.4
0.5
1.0
1.5
ωc/U

FIG. 9

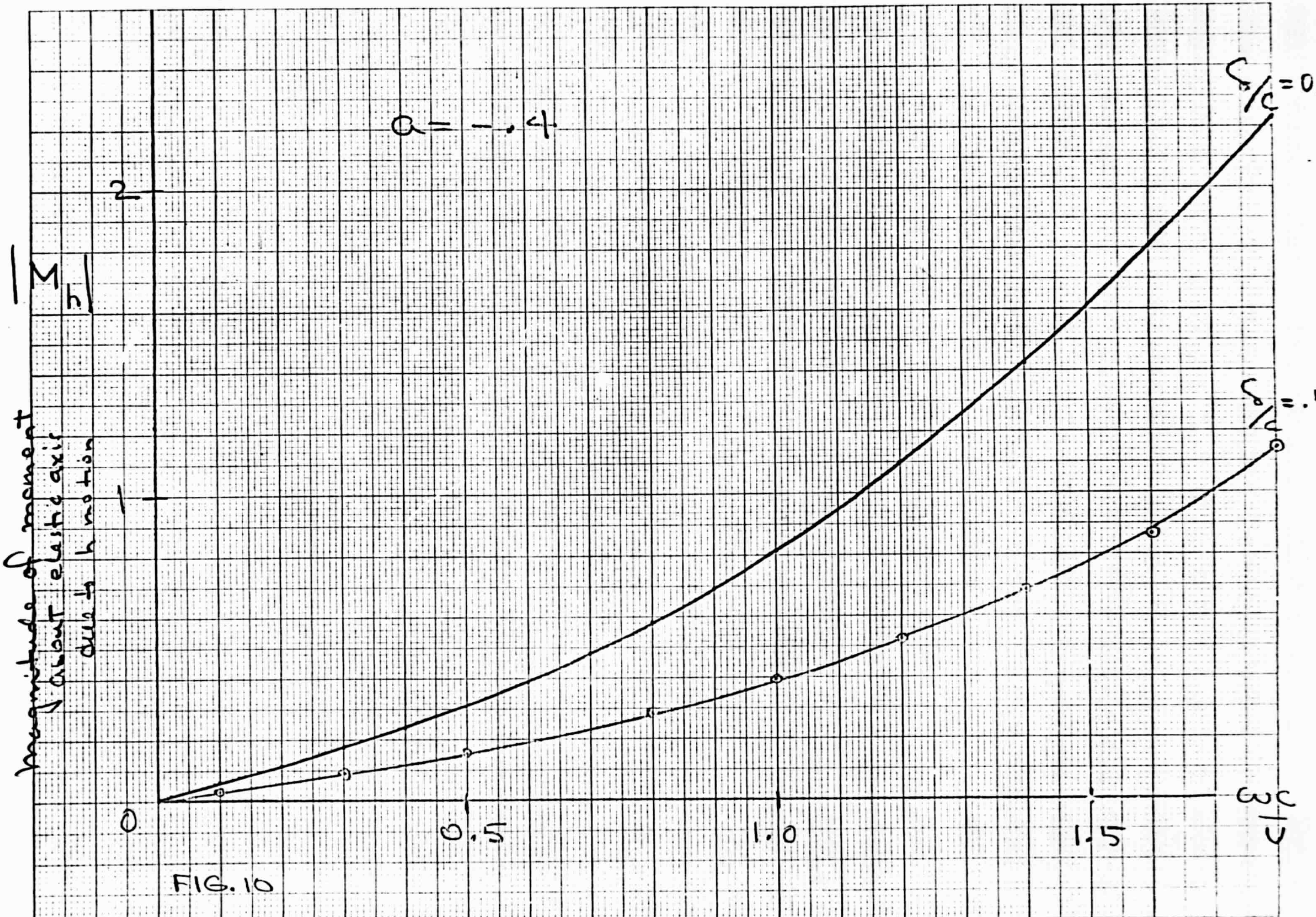

a = -.4
|M_h|
Magnitude of moment about elastic axis due to h motion
2
1
0
0.5
1.0
1.5
ωc/V
δ/c = 0.
δ/c = .5

FIG. 10

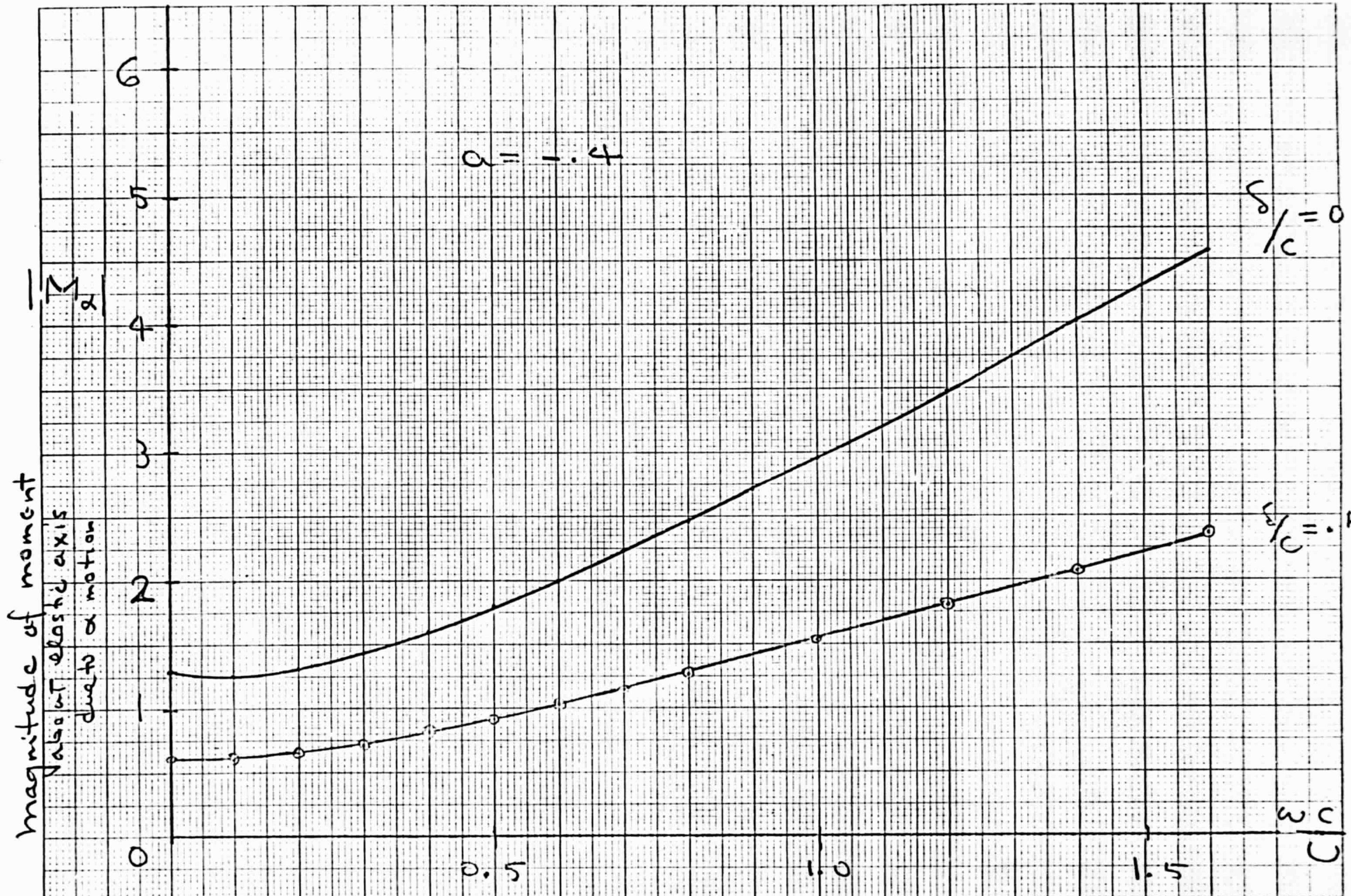

a = -.4
δ/c = 0.
δ/c = .5
|M_α|
magnitude of moment about elastic axis due to α motion
ωc/U
0
0.5
1.0
1.5
1
2
3
4
5
6

FIG. 11

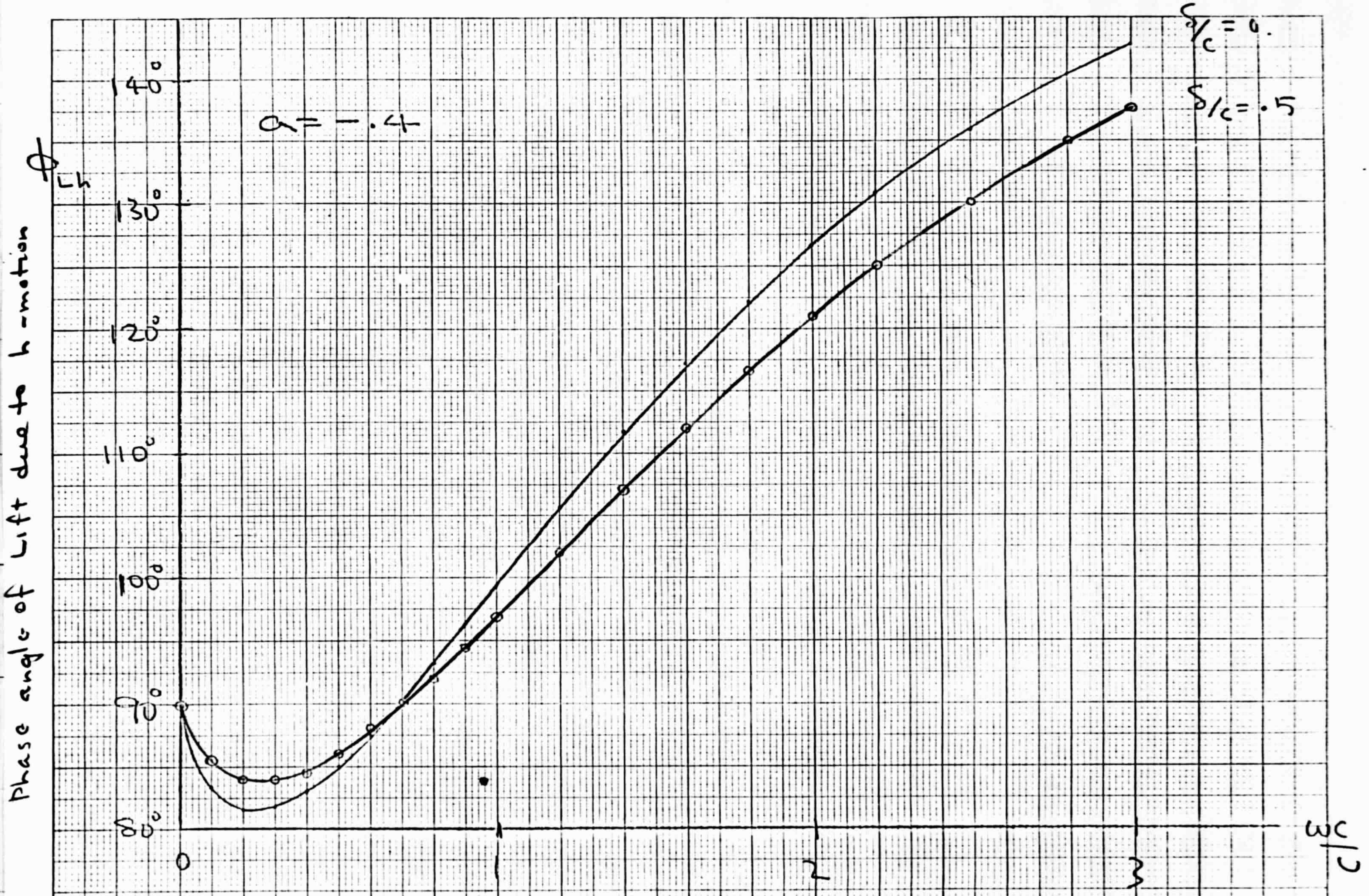
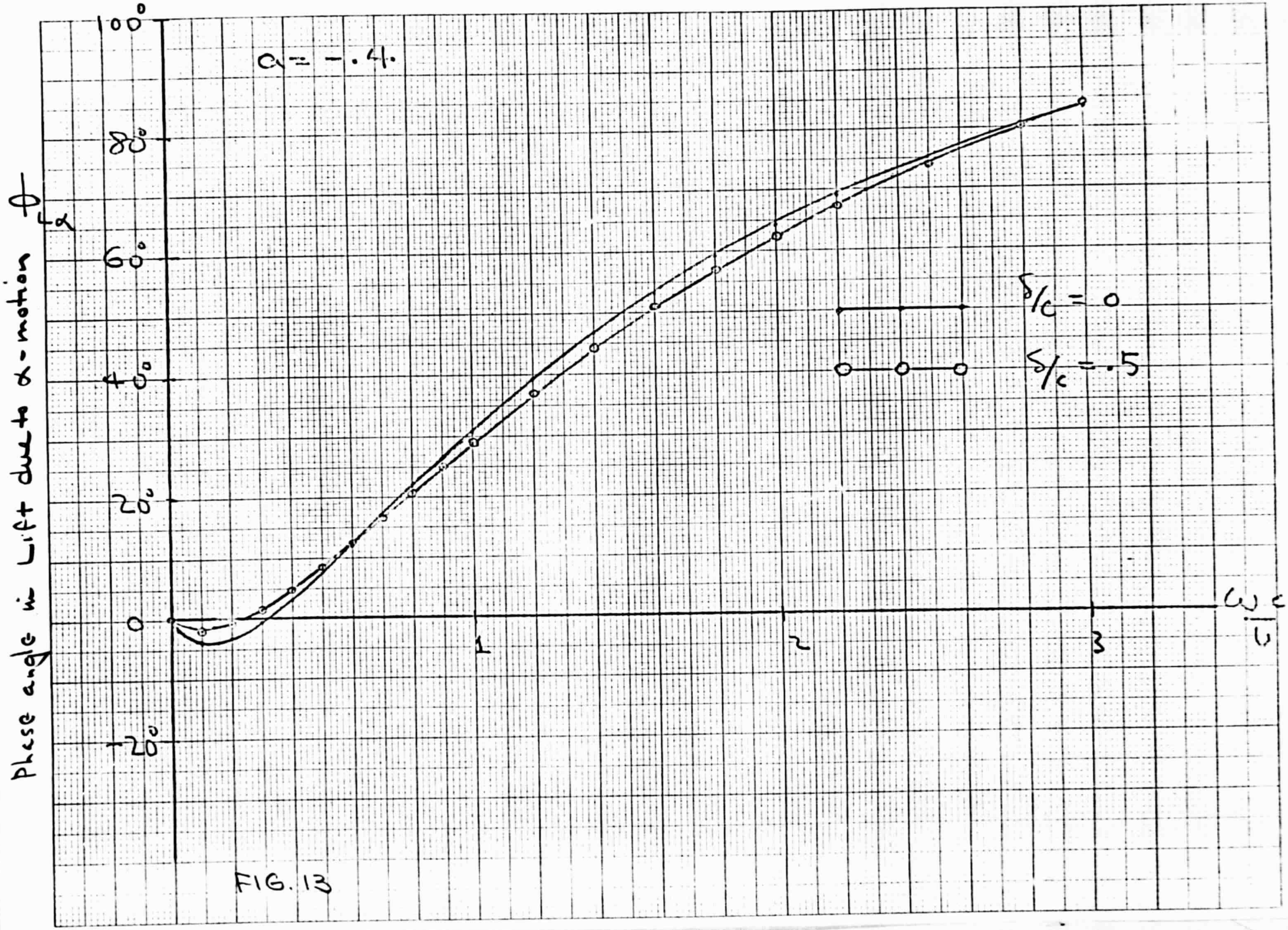

ϕ_{Lh}
Phase angle of Lift due to h-motion
$a = -.4$
140°
130°
120°
110°
100°
90°
80°
0
1
2
3
$\omega c / U$
$\delta / c = 0.$
$\delta / c = .5$

FIG. 12



a = -.4
100°
80°
60°
40°
20°
0
-20°
1
2
3
ωc/U
Phase angle in Lift due to α-motion $\phi_{L\alpha}$
δ/c = 0
δ/c = .5

FIG. 13

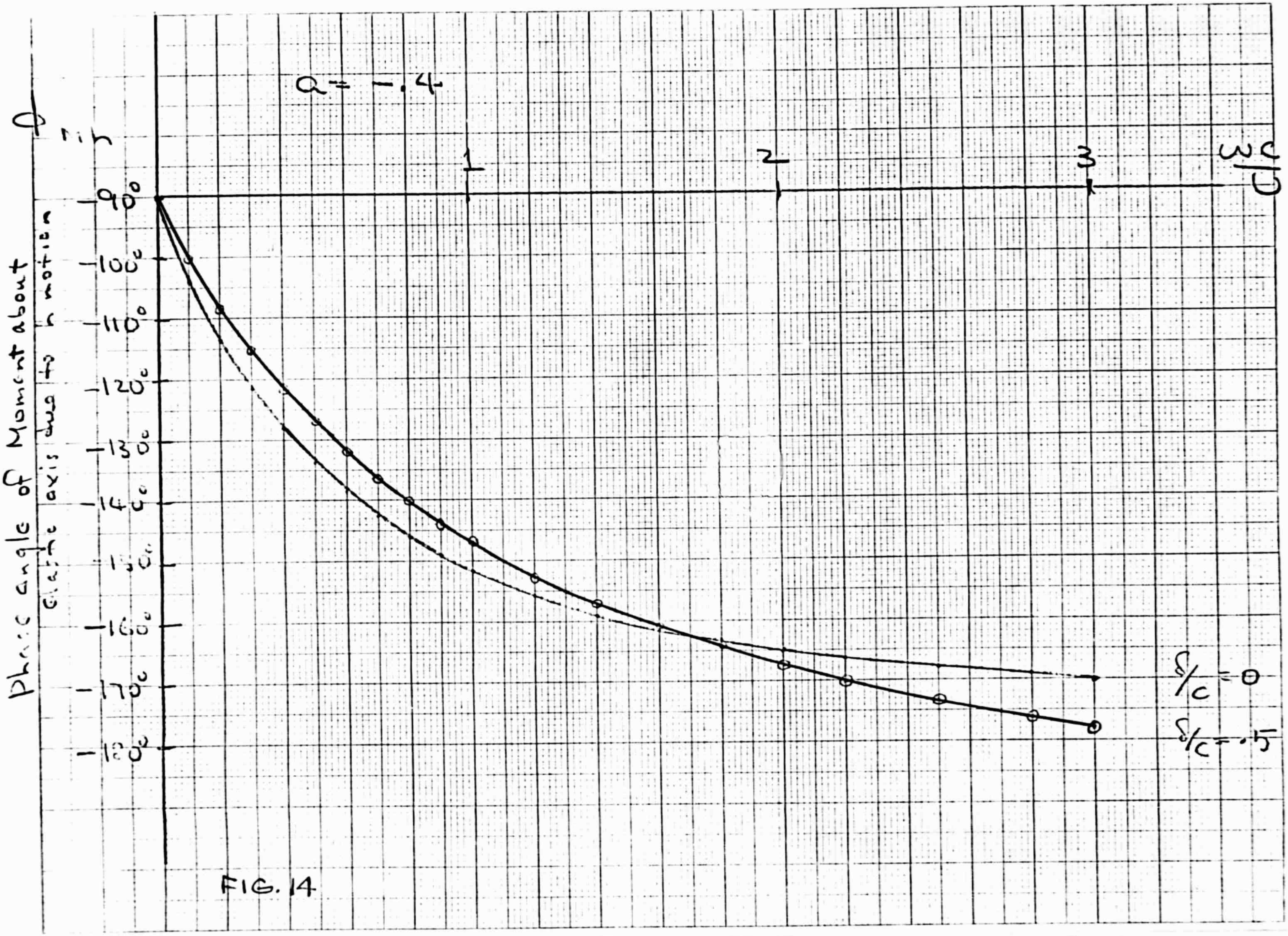

a = -.4
Phase angle of Moment about elastic axis due to h motion
ϕ_{Mh}
1
2
3
ω/b
-90°
-100°
-110°
-120°
-130°
-140°
-150°
-160°
-170°
-180°
S/c = 0
S/c = .5

FIG. 14

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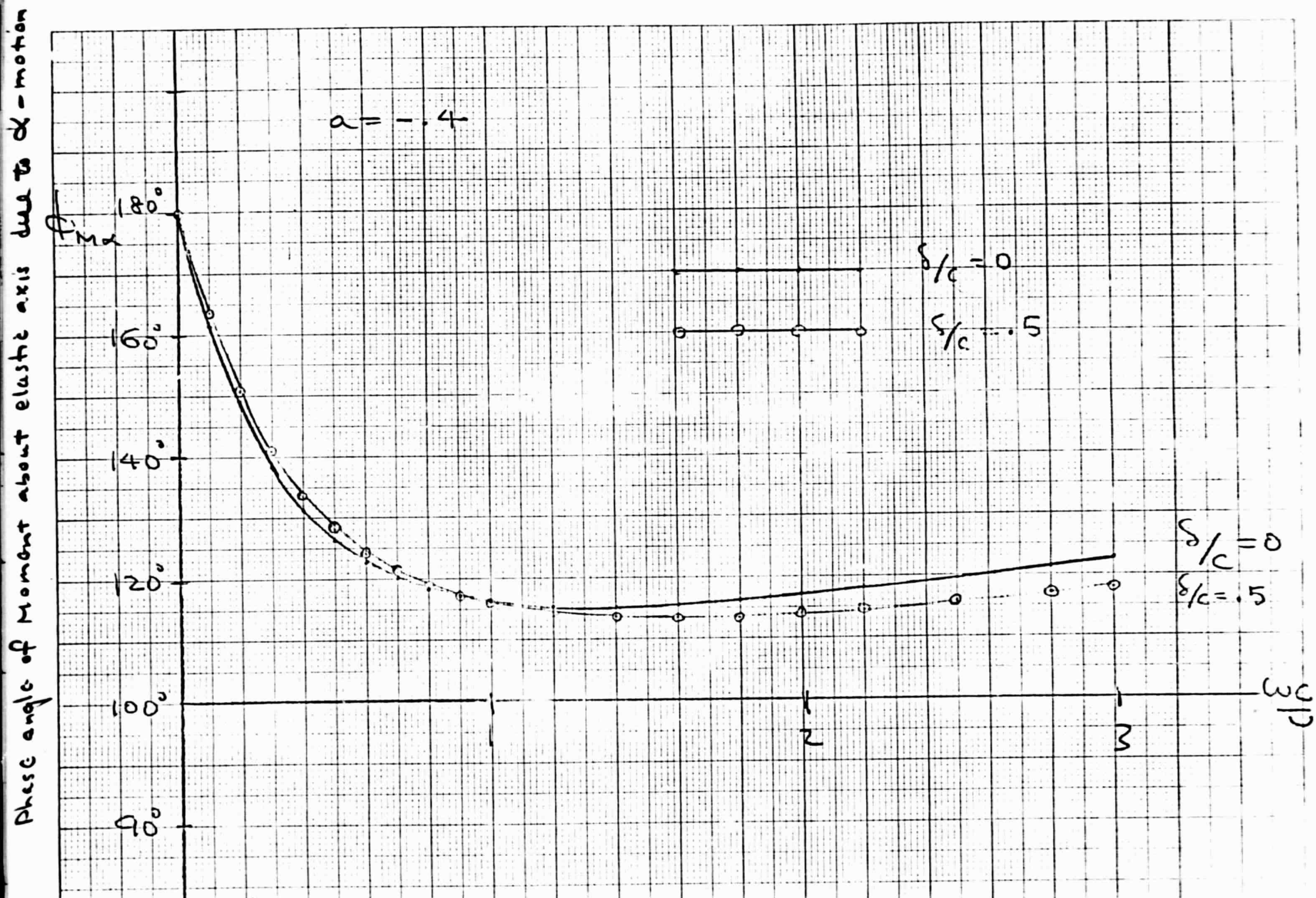

FIG. 15