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PRESSURE-DISTRIBUTION MEASUREMENTS OF A MODEL OF A
DAVIS WING SECTION WITH FOWLER FLAP SUBMITTED BY
CONSOLIDATED AIRCRAFT CORPORATION

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

MEMORANDUM REPORT

for

Materiel Division, Army Air Corps

PRESSURE-DISTRIBUTION MEASUREMENTS OF A MODEL OF A

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INTRODUCTION

Pressure-distribution measurements were made at the request of the Materiel Division, U.S. Army Air Corps, on a 24-inch chord model equipped with a Fowler flap and submitted by the Consolidated Aircraft Corporation. The tests were made in the Langley two-dimensional tunnel at a Reynolds number of about 6,000,000.

The model is of a section of the Davis wing for the XB-32 airplane and is described in reference 1. The tubes for the pressure orifices in the flap were removed for the tests of reference 1 to prevent possible interference with the flow through the slot. These tubes were replaced for the pressure-distribution tests in essentially the same manner as when received except that smaller diameter tubes were used to minimize possible interference effects.

RESULTS AND DISCUSSION

Pressure-distribution diagrams for several angles of attack, and flap deflections of 0° , 20° , and 40° are presented in figures 1 to 16. Pressures are plotted directly as obtained from the manometer in terms of 1/2-inch units of carbon tetrachloride. The abscissa is the measured projection on the chord line of the pressure orifices. The values of the corrected dynamic pressure q and the impact pressure level in terms of the same units are given on each figure. The static pressure level is obtained by adding the value of q to the impact pressure level. The value $\left(\frac{v}{V}\right)^2$, where v is the local velocity and V is the free-stream velocity, is obtained by dividing the local pressure, measured from the impact pressure level, by the value of q .

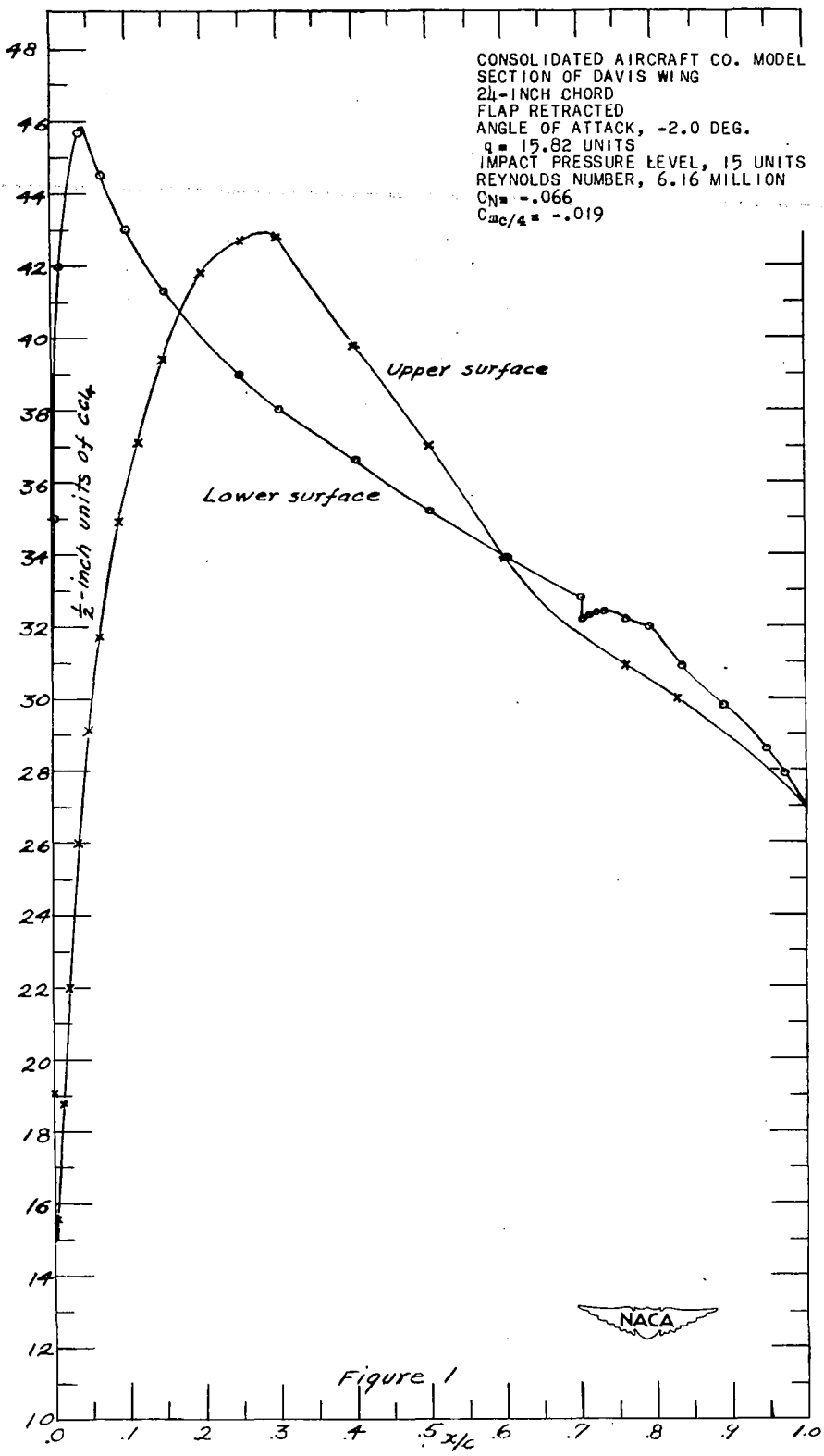
The normal-force coefficient C_N , as obtained by integration of the pressure diagrams, is given on each figure. These normal-force coefficients are in reasonable agreement with the lift coefficients presented in reference 1 except near the maximum lifts with flap deflected. The present tests indicate lower values of the maximum lift coefficient than those presented in reference 1 and this is thought to be caused by the interference of the pressure tubes on the flap. At each flap deflection the pressure distribution presented at the highest angle of attack was taken at or very close to maximum lift.

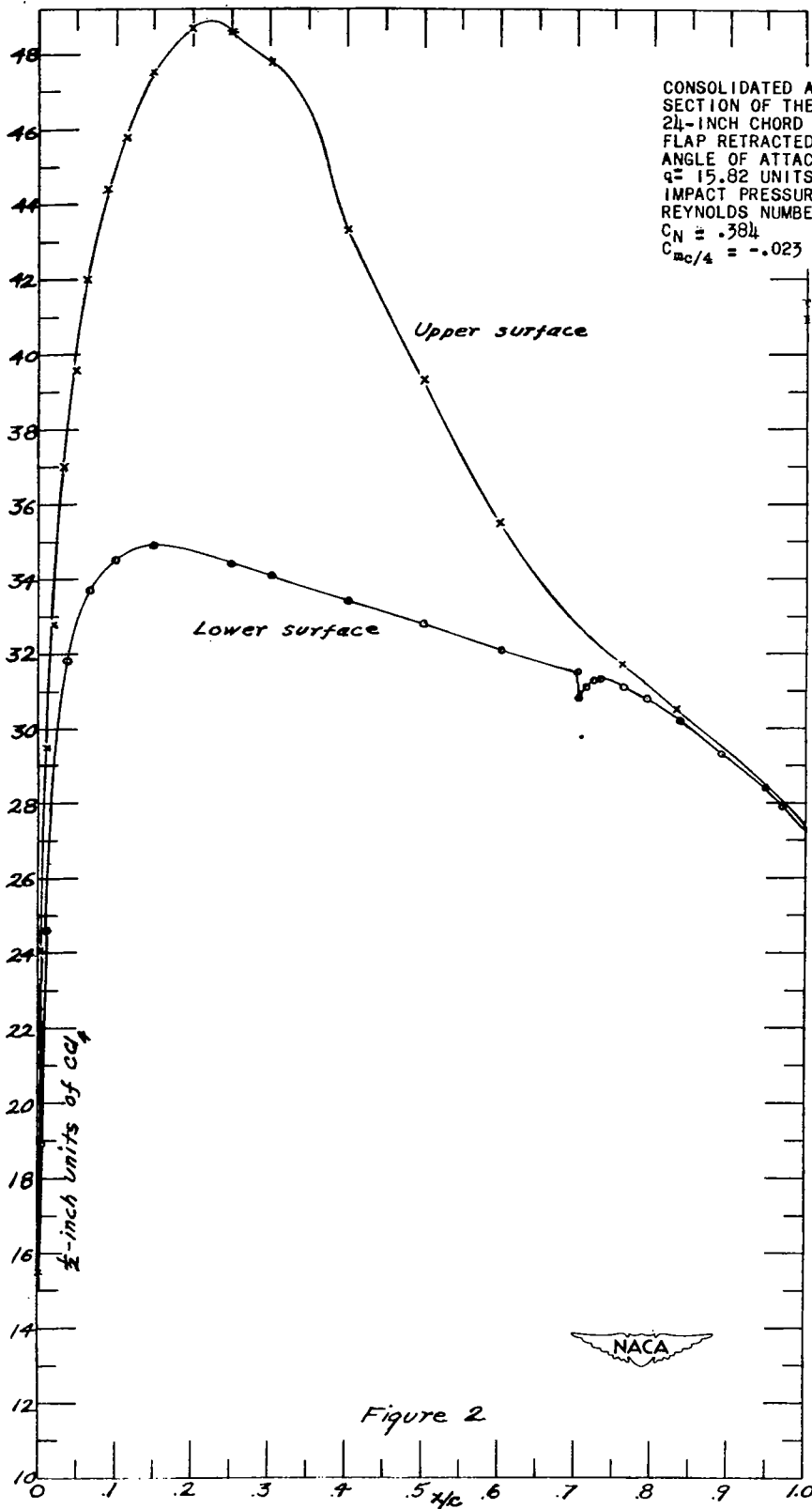
Moment coefficients about the quarter-chord point $c_{m_c/4}$ are also presented as obtained by integration of the diagrams. These moment coefficients do not contain the component of moment due to the chord force which may be appreciable, especially for the flap-deflected conditions. This component of moment may be obtained by replotting the diagrams against displacement of the crifices perpendicular to the chord.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., January 17, 1942.

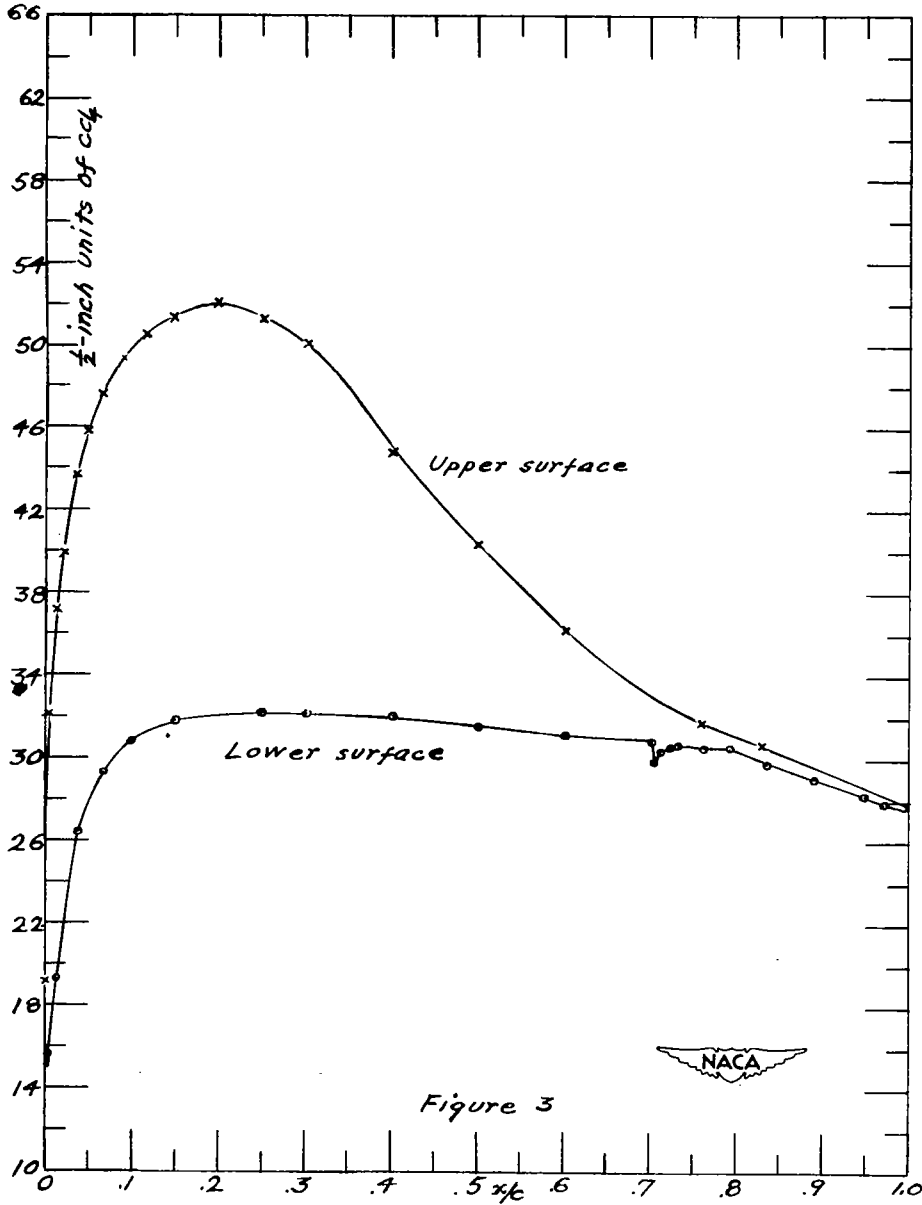
REFERENCE

1. Abbott, Ira H., and Turner, Harold R., Jr.: Lift and Drag Tests of Three Airfoil Models with Fowler Flaps Submitted by Consolidated Aircraft Corporation. NACA MR, Dec. 29, 1941.

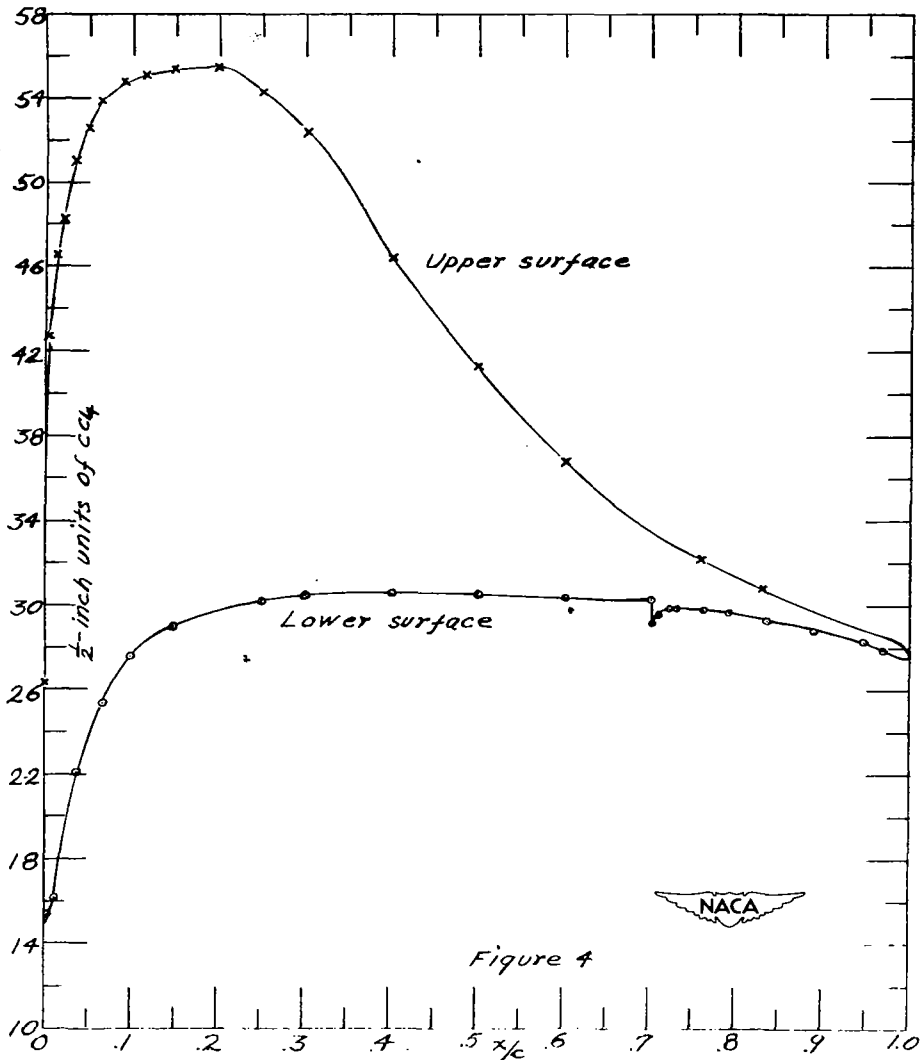




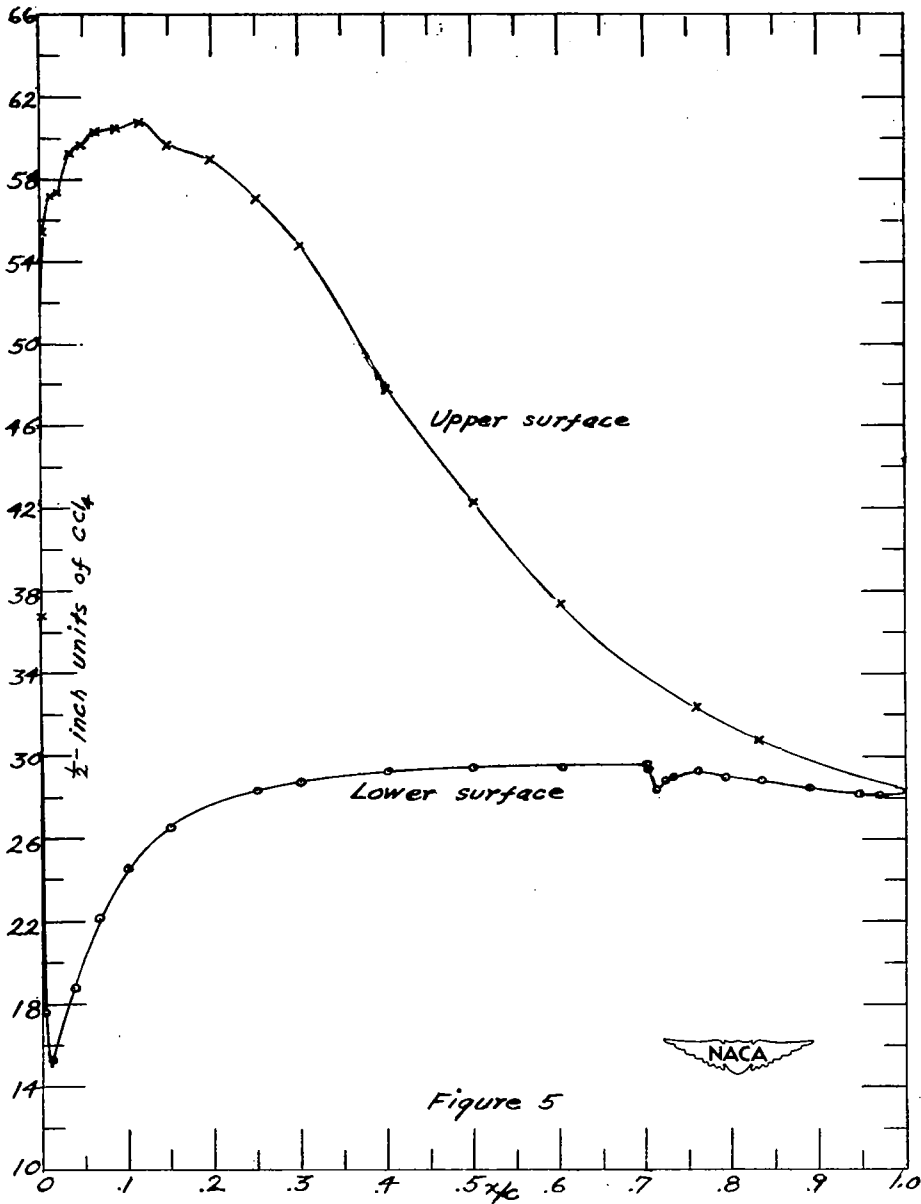
CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP RETRACTED
 ANGLE OF ATTACK, 4.1 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.16 MILLION
 $C_N = .591$
 $C_{mc/4} = -.024$



CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP RETRACTED
 ANGLE OF ATTACK, 6.1 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.16 MILLION
 $C_N = .791$
 $C_{m_{c/4}} = -.025$



CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP RETRACTED
 ANGLE OF ATTACK, 8.1 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.16 MILLION
 $C_N = .991$
 $C_{m_{c/4}} = -.026$



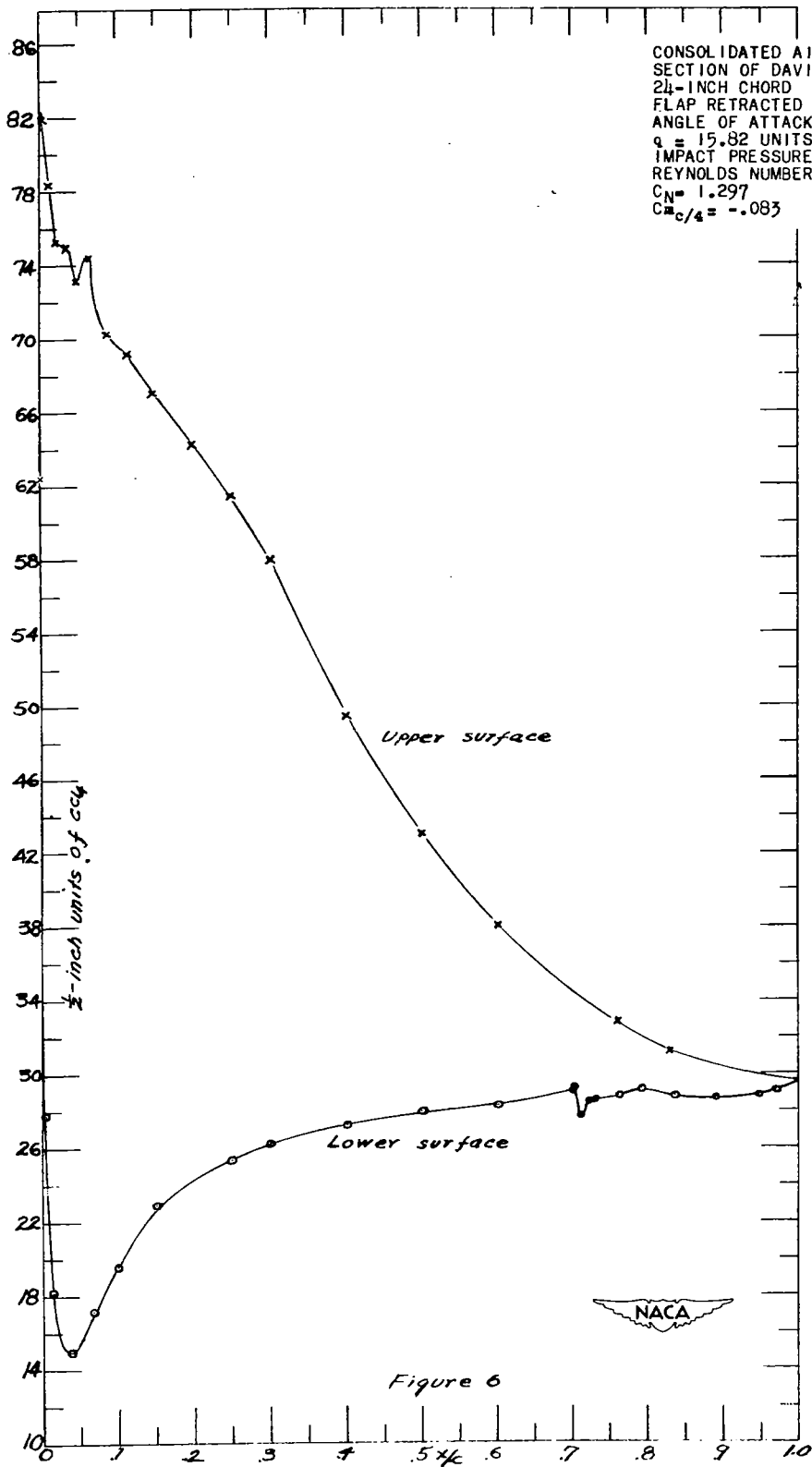


Figure 6



CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP DEFLECTED 20 DEG.,
 ANGLE OF ATTACK, -8.1 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.13 MILLION
 $C_N = -.067$
 $C_{m_c/4} = -.175$

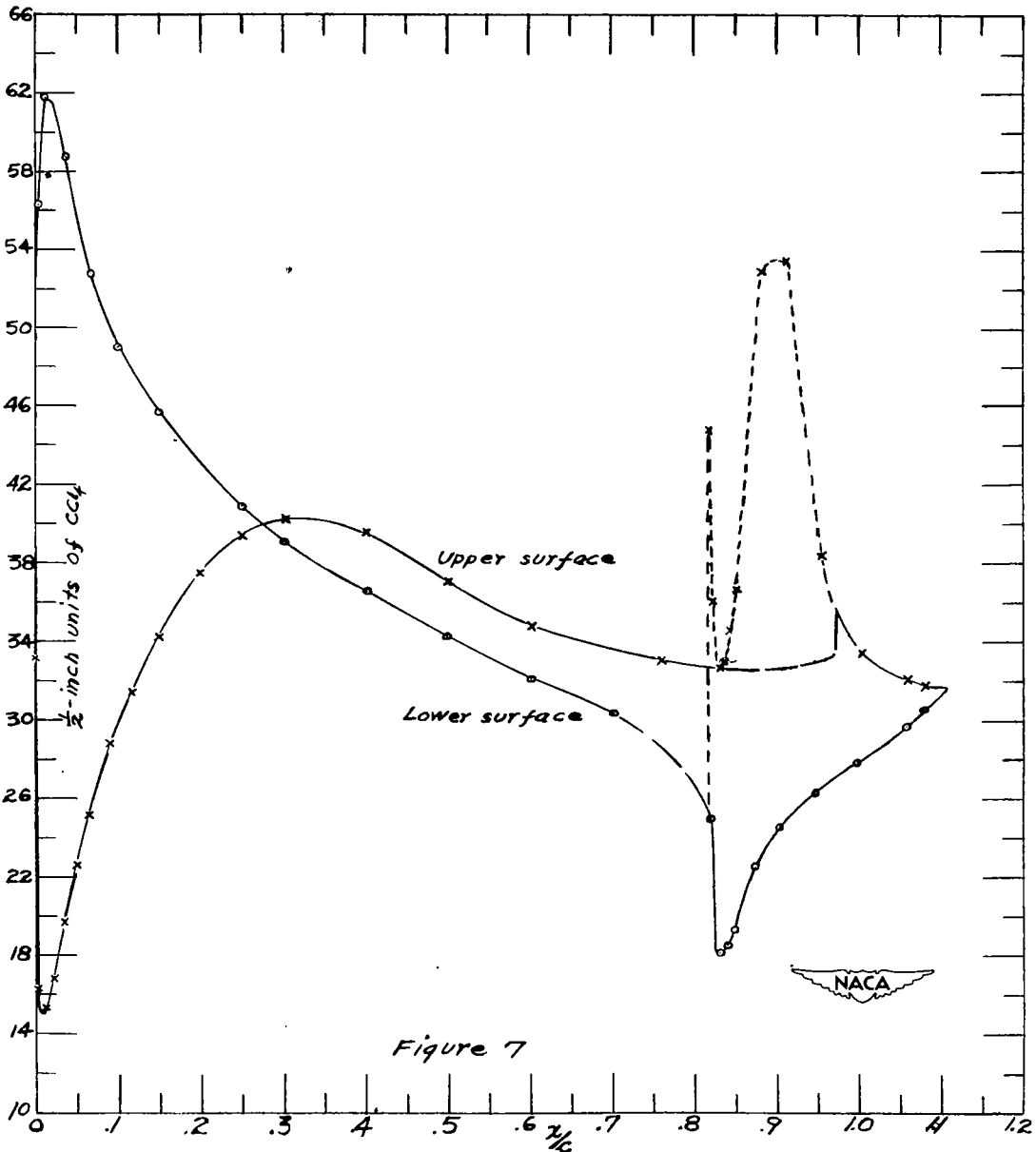
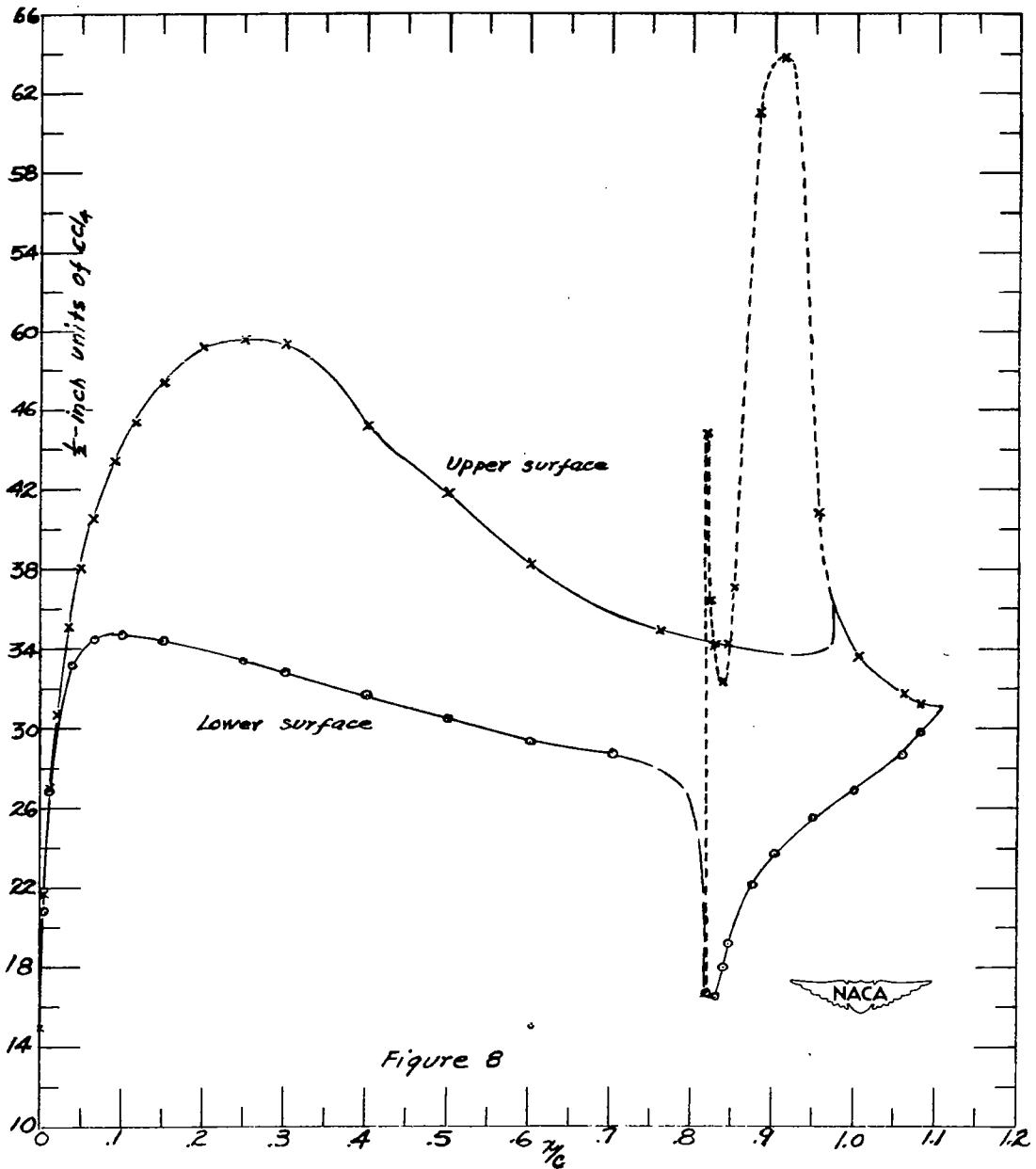


Figure 7

CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP DEFLECTED 20 DEG.
 ANGLE OF ATTACK, -2 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.13 MILLION
 $C_N = .683$
 $C_{m_{c/4}} = -.185$



CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP DEFLECTED 20 DEG.
 ANGLE OF ATTACK, 4.1 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.13 MILLION
 $C_N = 1.413$
 $C_{m/c/4} = -.226$

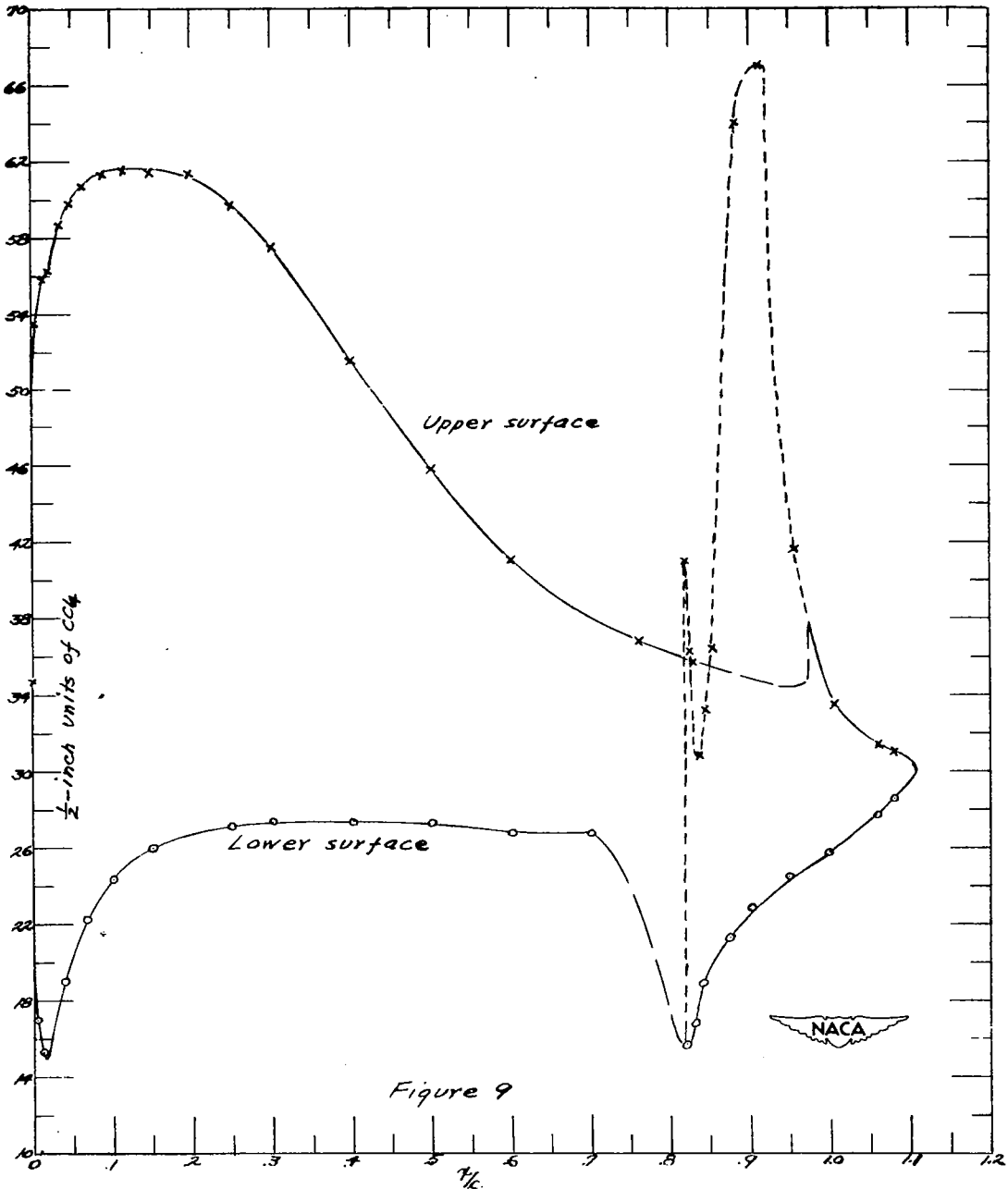


Figure 9

CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP DEFLECTED 20 DEG.
 ANGLE OF ATTACK, 8.1 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.13×10^6
 $C_N = 1.794$
 $C_{M_c/4} = -.221$

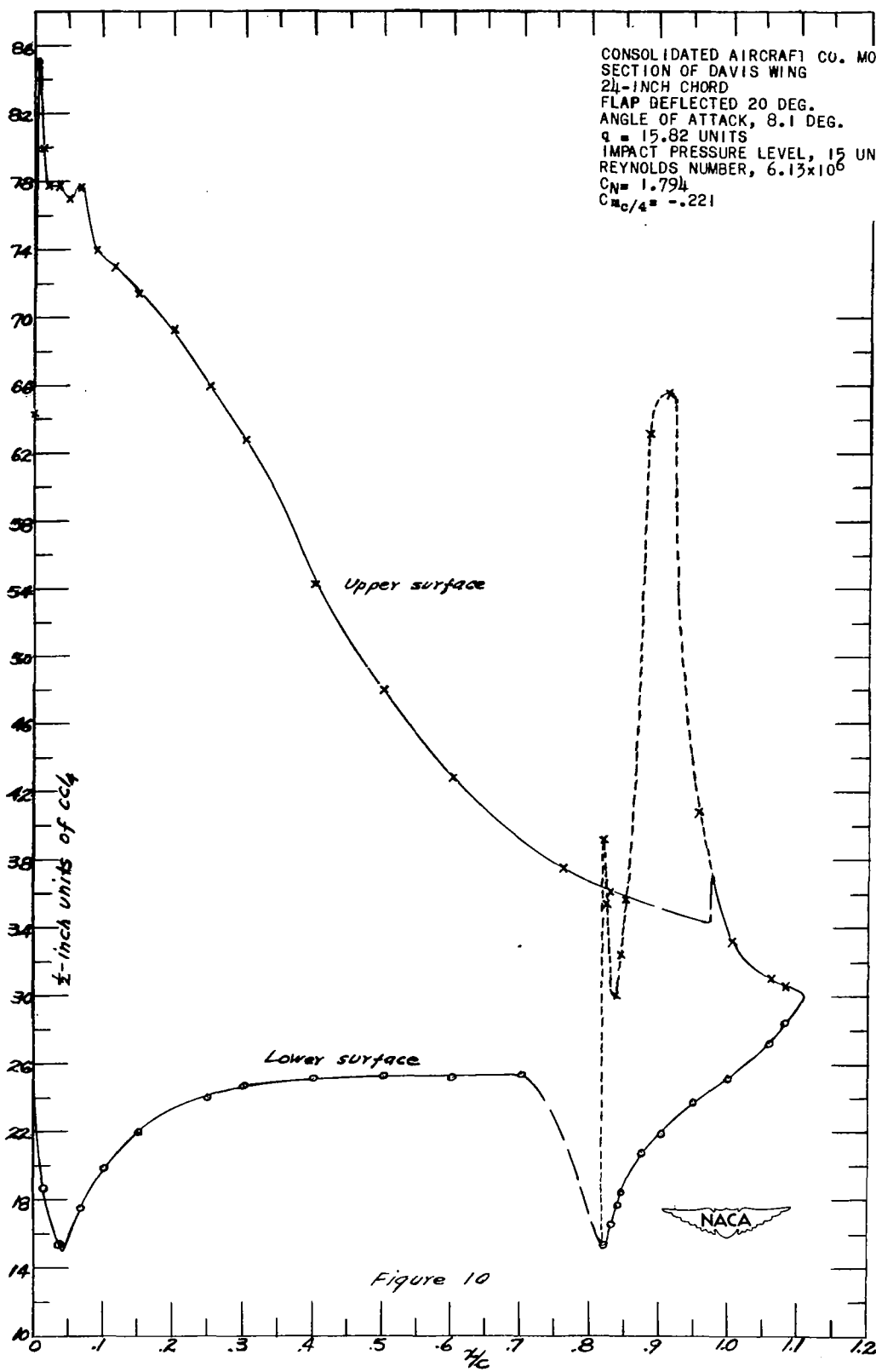
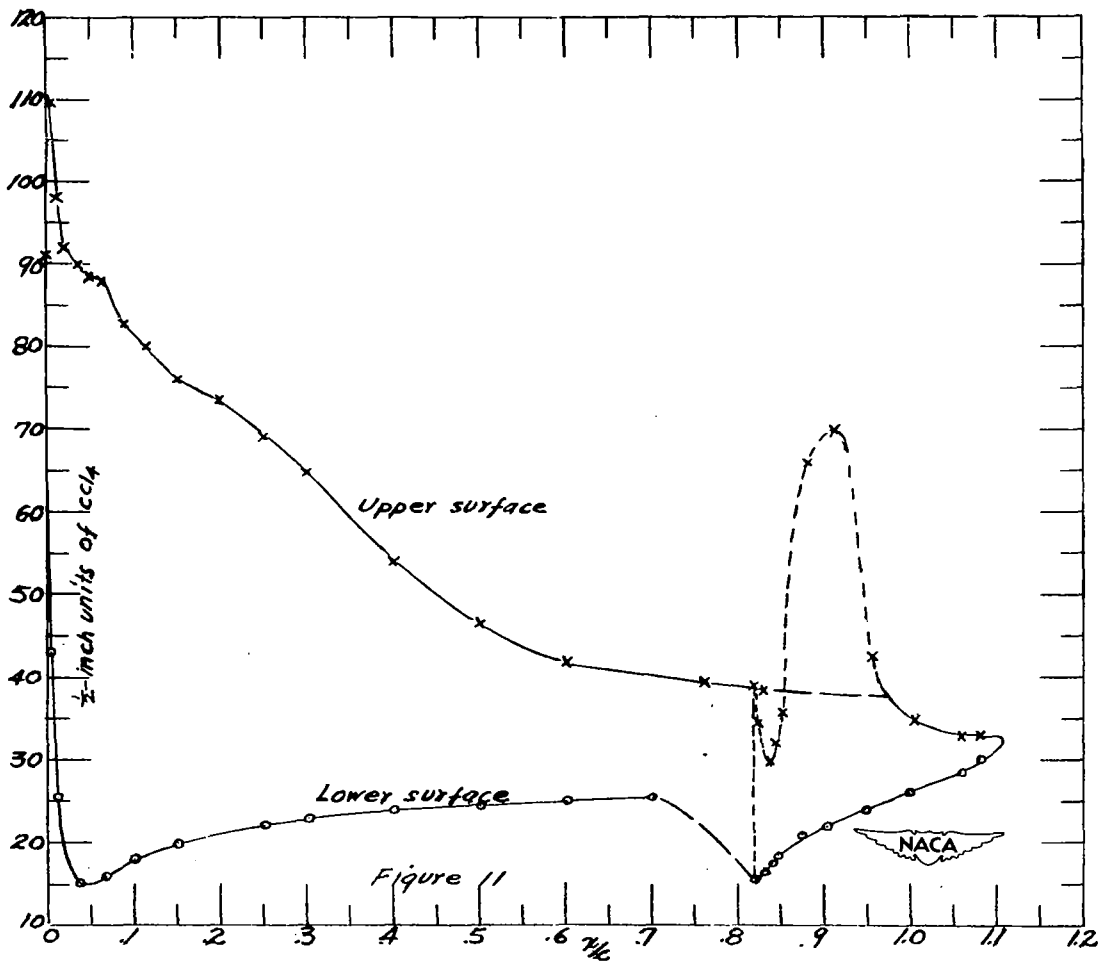
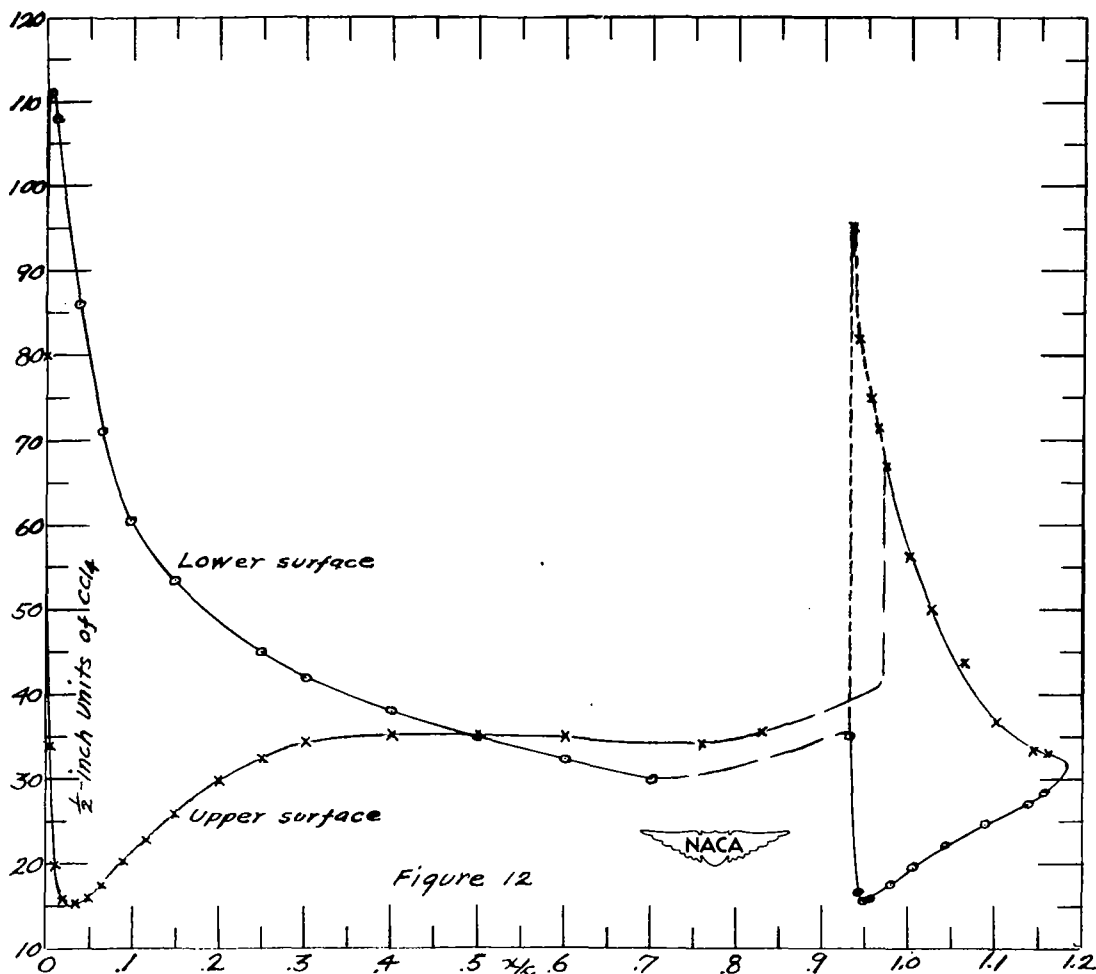


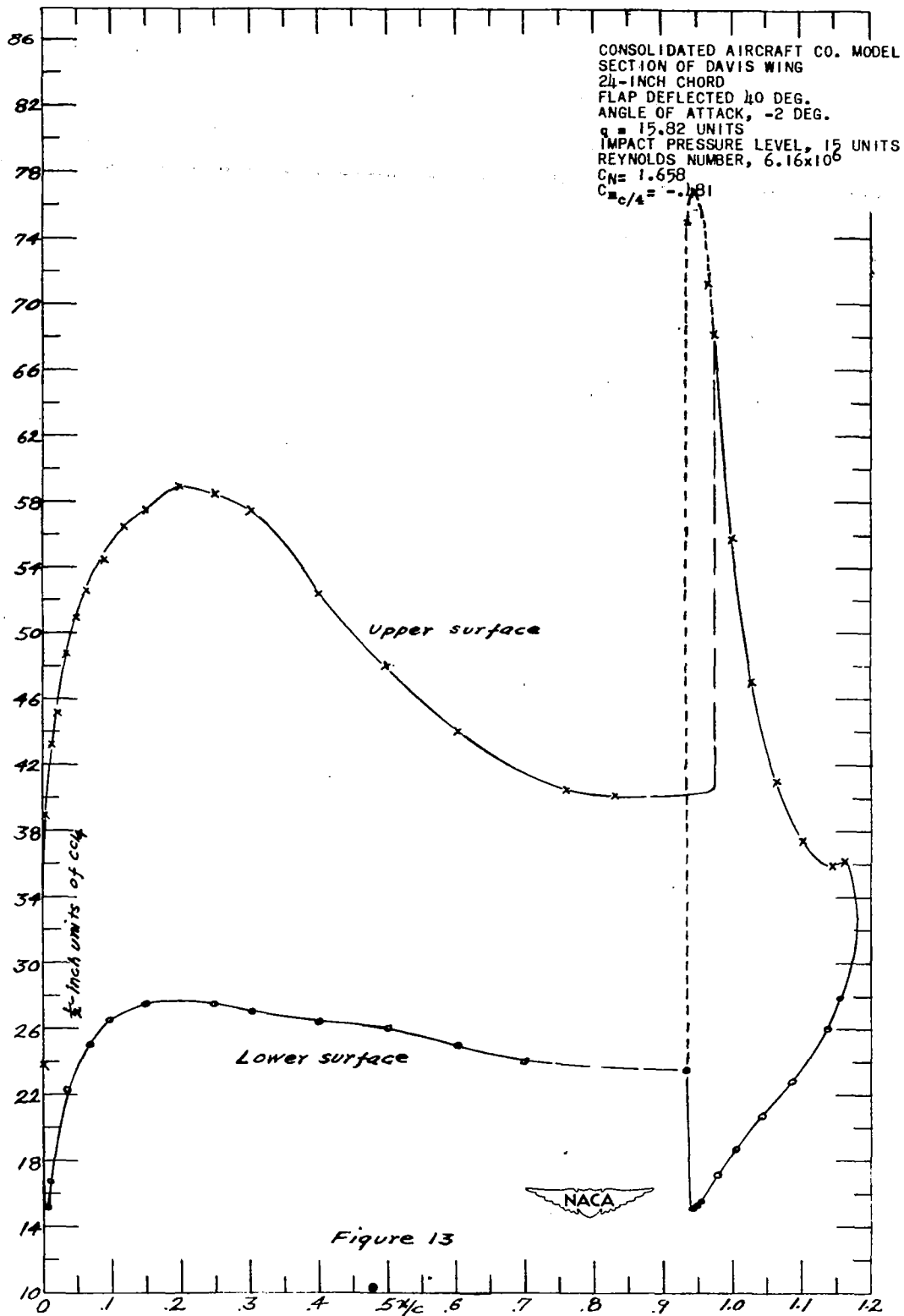
Figure 10

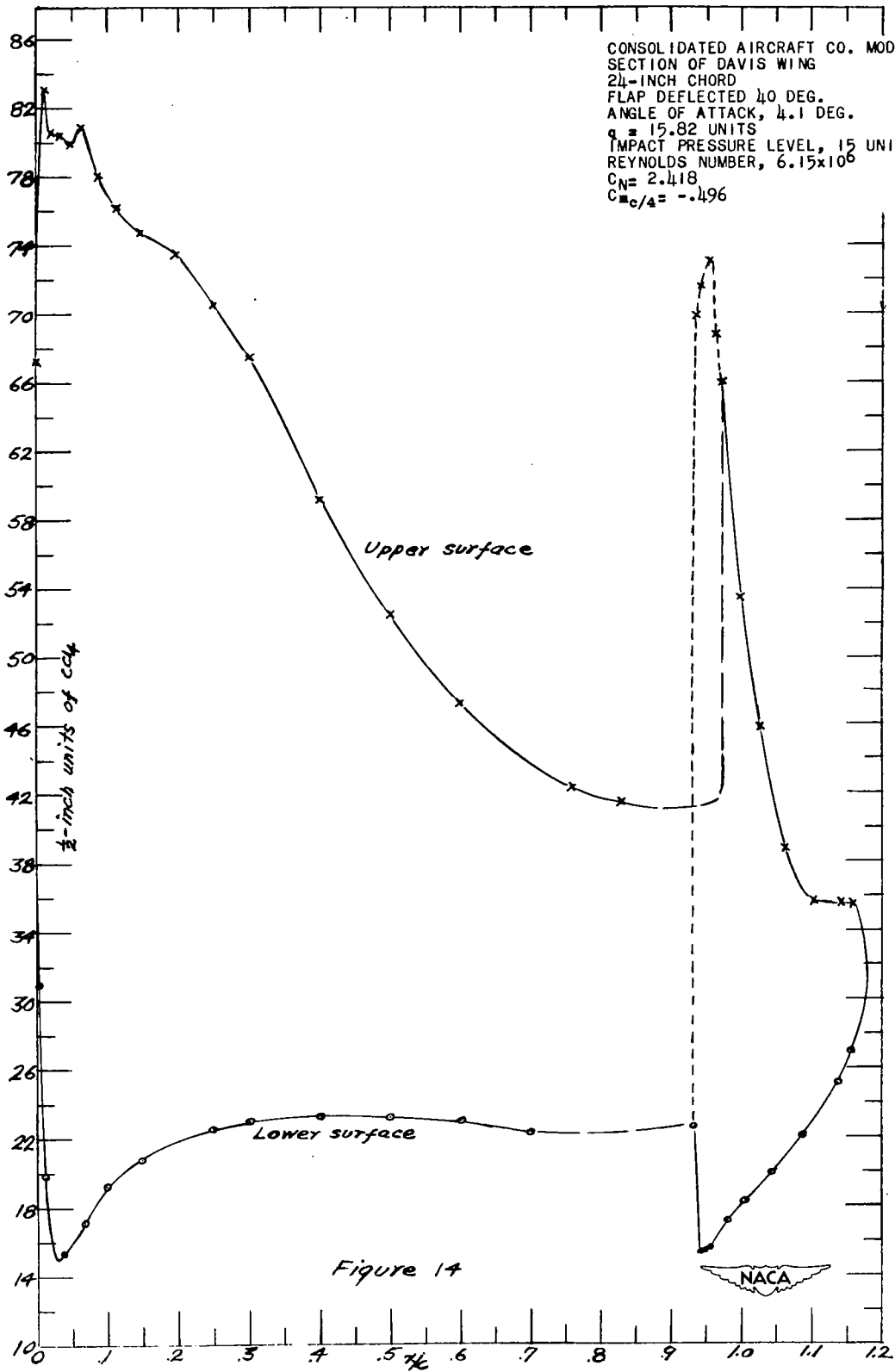
CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP DEFLECTED 20 DEG.
 ANGLE OF ATTACK, 12.3 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.08×10^6
 $C_N = 1.979$
 $C_{m_{c/4}} = -.231$



CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP DEFLECTED 40 DEG.
 ANGLE OF ATTACK, -17.3 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.16×10^6
 $C_N = -.294$
 $C_{m_{c/4}} = -.390$







CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 2 1/2-INCH CHORD
 FLAP DEFLECTED 10 DEG.
 ANGLE OF ATTACK, 8.1 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.15×10^6
 $C_N = 2.725$
 $C_{m/c/4} = -.501$

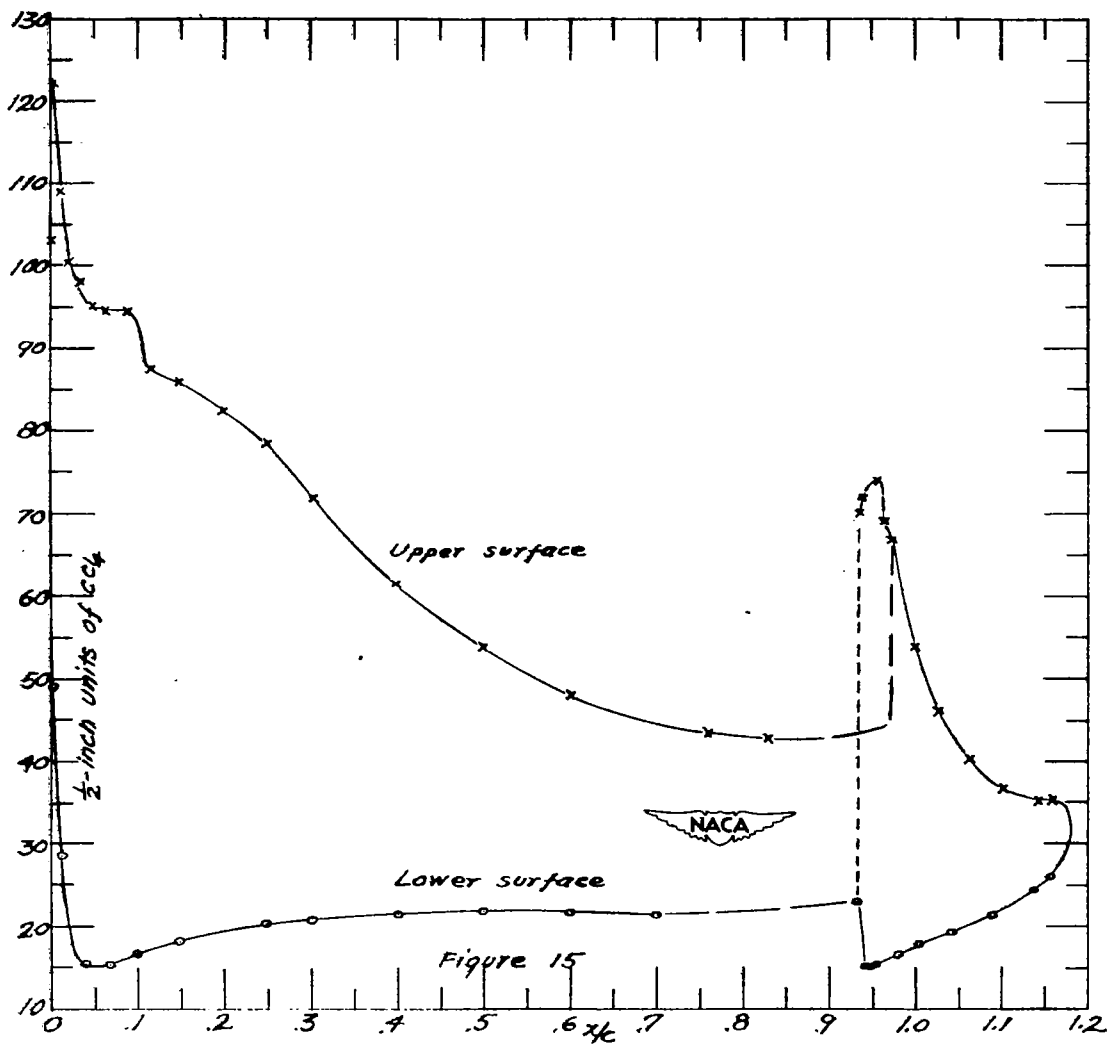


Figure 15

CONSOLIDATED AIRCRAFT CO. MODEL
 SECTION OF DAVIS WING
 24-INCH CHORD
 FLAP DEFLECTED 40 DEG.
 ANGLE OF ATTACK, 11.2 DEG.
 $q = 15.82$ UNITS
 IMPACT PRESSURE LEVEL, 15 UNITS
 REYNOLDS NUMBER, 6.15×10^6
 $C_N = 2.781$
 $C_{m_c/4} = -.552$

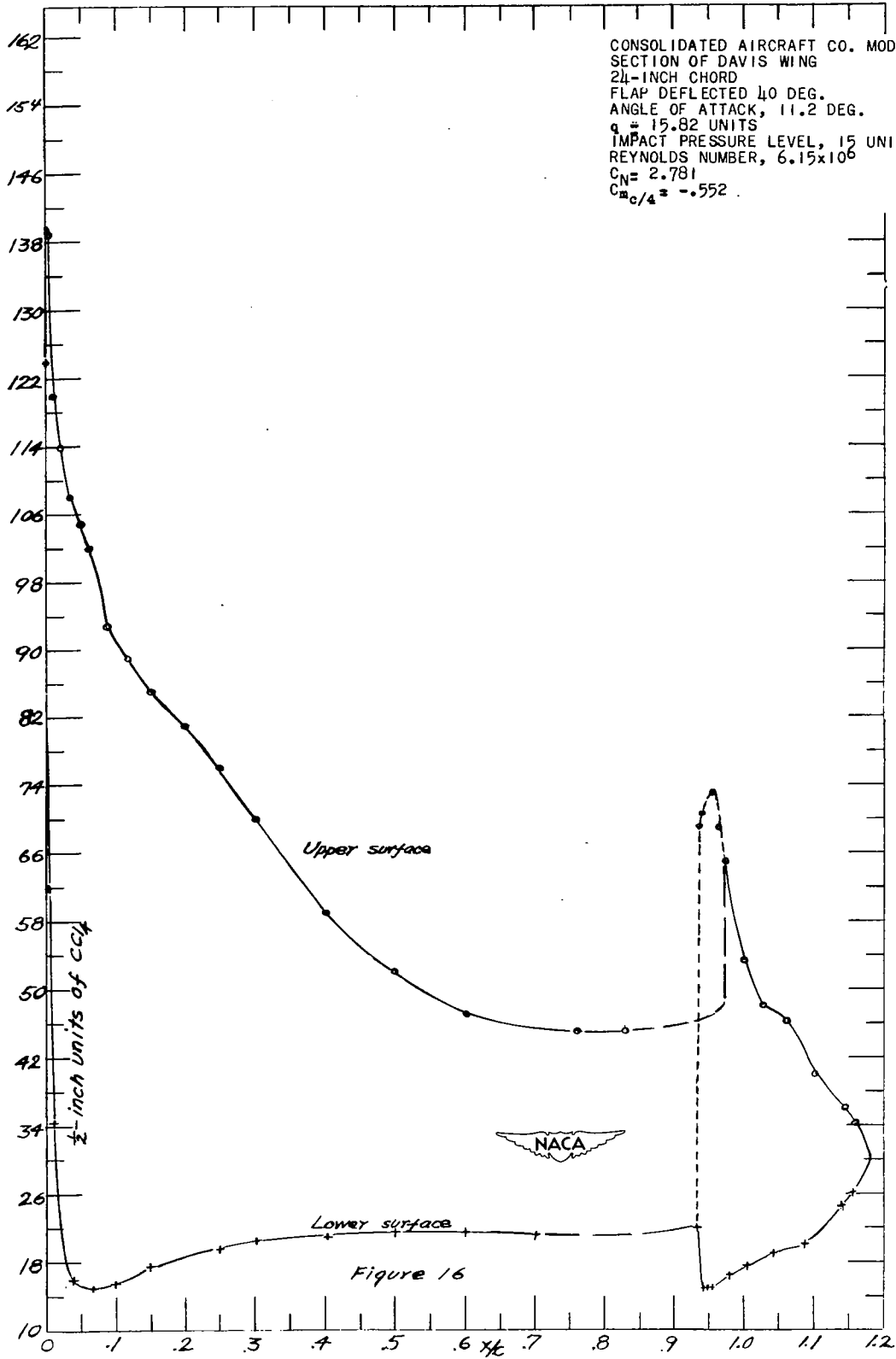


Figure 16



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