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CALIBRATION OF AVERAGING TOTAL
PRESSURE FLIGHT WAKE RAKE AND
NATURAL-LAMINAR-FLOW AIRFOIL
DRAG CERTIFICATION

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ABSTRACT

An averaging total pressure wake rake used by the Cessna Aircraft Company in flight tests of a modified 210 airplane with a laminar flow wing was calibrated in wind tunnel tests against a five-tube pressure probe. The model generating the wake was a full-scale model of the Cessna airplane wing.

Indications of drag trends were the same for both instruments.

INTRODUCTION

A form of the averaging (or integrating) total pressure wake rake designed by A. Silverstein and S. Katzoff, of NACA¹, was built by Cessna Aircraft Company. This rake was used to determine flight drag of a NLF(1)-0414 airfoil.

The tests calibrated the flight wake rake against the Wichita State University five-tube pressure probe. This probe was also used for the airfoil drag estimation.

The investigations were carried out over an angle of attack range of -5 degrees to +7 degrees, and a Reynolds number range between 2.25 million and 5.5 million based on airfoil chord; maximum Mach number was 0.2.

Flow visualization was also done, using fluorescent oil. Tests were conducted in the Wichita State University low speed 7 X 10 foot wind tunnel.

FLIGHT WAKE RAKE

The integrating total pressure flight wake rake (figure 1a,b,c) was developed by A. Silverstein and S. Katzoff of NACA. A rake of this type was used by Cessna Airplane Company in a flight test program of a modified Cessna 210. This wake rake uses 71 total pressure pitot tubes spaced at 2.0 mm intervals. These tubes open to a common chamber, producing a type of average of the total pressure across the wake. This average total pressure and the ambient static pressure permit calculations of mean velocity deficit in the wake, and hence, the mean momentum deficit, which is profile drag.

Data from the rake needs to be corrected for a number of factors. These are explained in the original report by Silverstein and Katzoff; see reference 1.

This flight wake rake is very convenient to use on a flight-test airplane, requiring no complicated or heavy equipment. The mode of attaching the rake on the model trailing edge can be seen in figure 1. The rake is attached to an aircraft wing for flight tests in the same manner.

AIRFOIL DESIGN AND MODEL GEOMETRY

The airfoil section to be tested was designed by Dr. Werner Pfenninger and Mr. Jeffrey Viken under a contract to the NASA Langley Research Center (ref. 2). The airfoil is designated NLF(1)-0414, indicating, natural laminar flow (first airfoil) with a design lift coefficient of 0.4 and a maximum airfoil thickness of 14% of the chord. This airfoil was designed to achieve natural laminar flow over 70% of its upper and lower surface.

The wind tunnel model was constructed by Cessna Aircraft Company as a full-scale duplicate of the mid-span portion of the wing which had been flown on the specially modified BLF Model 210 aircraft. It was built with aluminum ribs and spars covered with a fiberglass skin. This is not a flight-worthy wing section; the ribs are widely spaced and at high tunnel temperature there was some distortion of the surface contour due to expansion of the skin. The model has a span of 83 1/8 inches and a chord of 48 3/8 inches. The pitching pivot point is at 26.5% chord behind the leading edge, on the chord line. The

model also has a simulated flap gap 0.3 inches wide and 0.2 inches deep running the full length of the model, on both sides, positioned at 80% chord. This is a surface gap; there is no leakage from one surface to the other.

The airfoil contours are shown in figure 2.

THE WICHITA STATE UNIVERSITY FIVE-TUBE PRESSURE PROBE

The five-tube pressure probe is based on the "AEROTECH" probe DC 120 #148 (reference 4). This probe has been used for numerous tests and is well calibrated.

The probe was mounted on a streamlined support stem. This stem, in turn, was attached to a manual drive system. The probe had only one degree of freedom and transversed the airstream perpendicular to the flow, in the y-direction. The five-tube pressure probe is shown in figure 3a, and the entire installation can be seen in figure 3b,c,d.

WIND TUNNEL AND INSTRUMENTATION

The Wichita State University Walter Beech 7 feet by 10 feet wind tunnel is a closed throat, single return tunnel. Turbulence factor for the tunnel varies between 1.34 and 1.03 for tunnel dynamic pressures between 10 psf to 70 psf respectively. Wind tunnel test section and model are shown in figures 1b, c and figures 3b, c, d.

All pressures are routed through pressure transducers, and an A/D converter to the tunnel data acquisition system.

TESTS AND METHODS

Two types of tests were conducted using the two different wake survey instruments--tests through a range of angles of attack and tests through a range of Reynolds numbers at the angle of attack for minimum drag. Additional tests were conducted to measure the effect of the use of riblets on the airfoil drag, and an attempt was made to visualize boundary layer flow to locate transition using liquid crystals and fluorescent oil.

The angle of attack sweeps were at a constant tunnel dynamic pressure of 45 psf. At approximately the angle of attack of least drag ($\alpha = -0.5$ degree) runs were made at various dynamic pressures, corresponding to Reynolds numbers from about 2.25 million to 5.5 million, based on the chord.

The flight wake rake was used to calculate the profile drag of the airfoil. The raw data from the rake was fed into a computer program which was a modification of that used during the flight tests of the airfoil (reference 5); this program corrects for tunnel dynamic pressure and static pressure variations and also for other factors as listed in reference 1. An improved version of this program has since been written which includes additional corrections as detailed in reference 6. This program is listed in the Appendix.

Next, the five-tube pressure probe was used to calculate the profile drag of the airfoil. Initially, it was set at 15.6% chord behind the wing trailing edge, the same position as the flight wake rake, in flight and in the wind tunnel tests. For a full pressure recovery in the wake the probe was moved to 51.6% chord behind the

wing trailing edge, and the angle of attack runs were repeated.

Flow visualization tests were carried out using fluorescent oil illuminated by ultraviolet light. Only the upper surface could be monitored.

These tests were followed by the use of riblets applied by a NASA technician to the section of wing immediately ahead of the probe. First, riblets were attached to the aft 30% of the wing; see figure 7a. Then, a section of wing from leading to trailing edge was covered with the riblets; see figure 3c and 7b.

Finally, various formulations of liquid crystals were used in an attempt to visualize the transition point.

RESULTS

Variation of the drag coefficient as a function of angle of attack is graphed in figure 4. These tests were carried out at a Reynolds number of about 5 million, based on model chord. Wake rake results are labelled "Old" (reduced by the Cessna routine) and "New" (reduced using the improved routine, Appendix). It will be noted that the five-tube probe gave consistently lower drag values when mounted at 51.6% chord behind the trailing edge as compared to its 15.6% chord aft position. Although both exhibit the same trend up to about +3 degrees angle of attack, at higher angles of attack the values diverge from each other. The higher values at the closer (15.6% downstream of the trailing edge) position is probably due to the fact that at these angles the airfoil produces a large lifting force ($C_L \sim 0.8$) and hence a strong wake. There is a delay in the pressure recovery in the wake producing higher drag figures.

The flight wake rake (New) gives results having the same general trend as those given by the five-tube probe at 51.6% chord downstream of the trailing-edge, and the results are within 15% of the five-tube data. It is felt that the five-tube probe data obtained with the probe 51.6% of chord downstream of the trailing edge is more accurate than those obtained nearer the trailing edge (where the flight wake rake was located).

Application of the riblets resulted in a decrease in the measured drag coefficient even at the very low drag coefficients resulting from laminar flow past the 70% chord station, as shown in figure 6a and 6b. For the case of riblets attached over the entire airfoil chord, an increase of drag was indicated between angles of attack of -1.0 degrees to $+2.5$ degrees. It is believed that this increase of drag resulted from improper attachment of the riblets. With only 30% of the aft surface covered with riblets consistently lower drag values were obtained when compared to the model with the flap gaps filled (with clay).

Figure 5 shows the variation of drag coefficient as a function of Reynolds number. The trends are as expected, i.e. the drag coefficient does decrease with increasing Reynolds number. The increase of drag at the higher limit for the five-tube pressure probe is probably due to earlier transition resulting in a loss of natural laminar flow over the airfoil.

Use of fluorescent oil confirmed the laminar flow over much of the airfoil at low angles of attack. At an angle of attack of -0.5 degrees and a Reynolds number of 5 million transition from laminar to turbulent flow was observed at 75% chord station; see figures 8a,b.

The use of liquid crystals was not at all successful. During runs at a dynamic pressure of 45 psf, the temperature of the closed return tunnel rose rapidly during the run (typically to above 90 degree F). The liquid crystal formulations lost their color and became clear and invisible as the temperature rose. In one case, the transition at about $x/c = 0.75$ was observed at $\alpha = -0.5$ degrees.

REFERENCES

1. Silverstein, A.; Katzoff, S.: "A Simplified Method for Determining Wing Profile Drag in Flight," Journal of the Aeronautical Sciences, Vol. 7, No. 7, May 1940, pp. 295-301.
2. Viken, Jeffrey K.: Aerodynamic Design Considerations and Theoretical Results for a High Reynolds Number Natural Laminar Airfoil. Masters thesis, George Washington University, January 1983.
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4. Anon, "AEROTECH Calibrated Airflow Probes--Probe Calibration DC 120 #148," AEROTECH Specialties Inc., September 1958.
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6. Irani, Eddie: "Calibration and Modification of a Flight Wake Rake and Drag Tests on a NLF(1)-0414 Airfoil Section." Thesis, Wichita State University, Wichita, KS, May 1988.

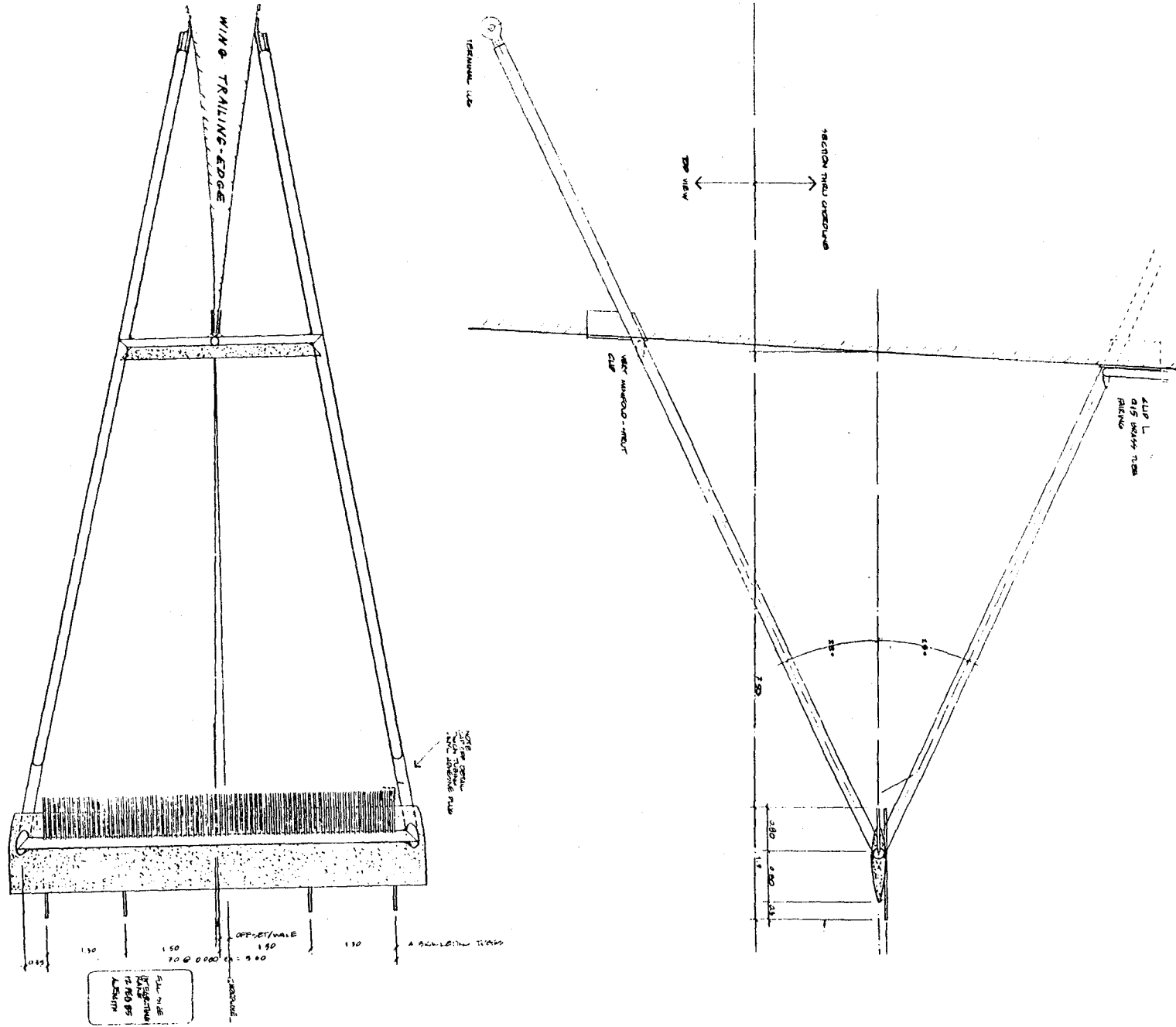


Figure 1a. Averaging total pressure flight wake rake.

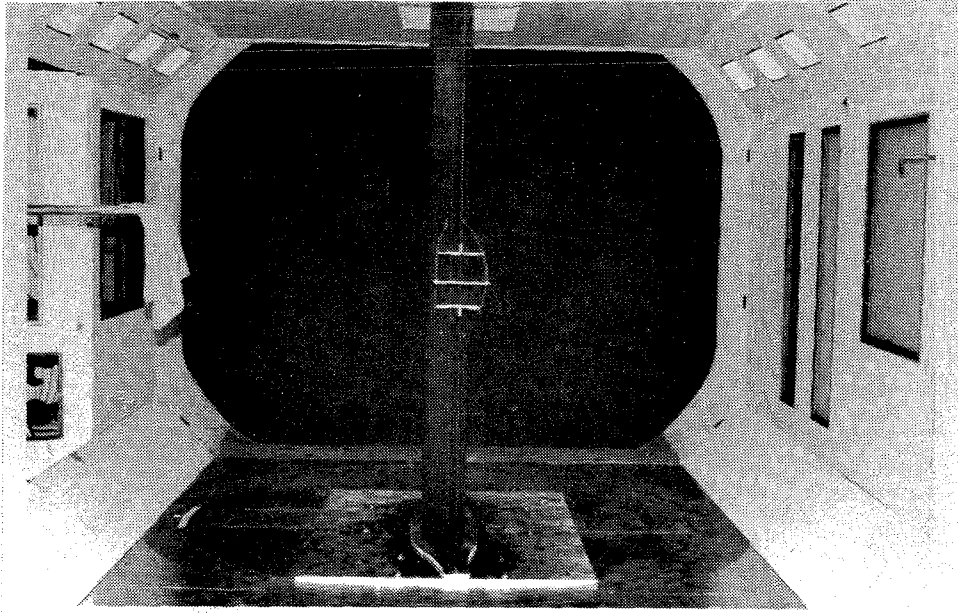


Figure 1b. Model and rake in the WSU 7x10 ft. wind-tunnel.

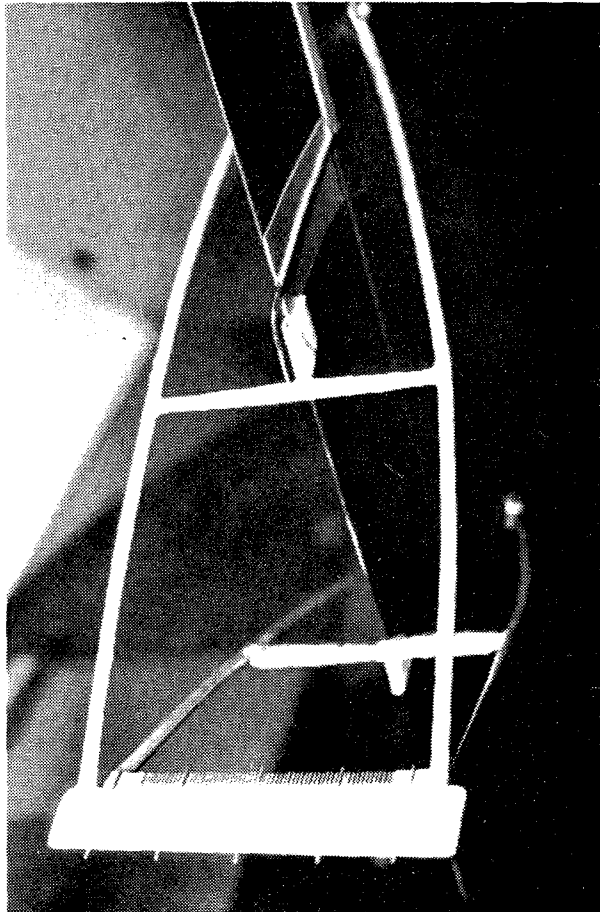


Figure 1c. Flight wake rake attached to model trailing edge.

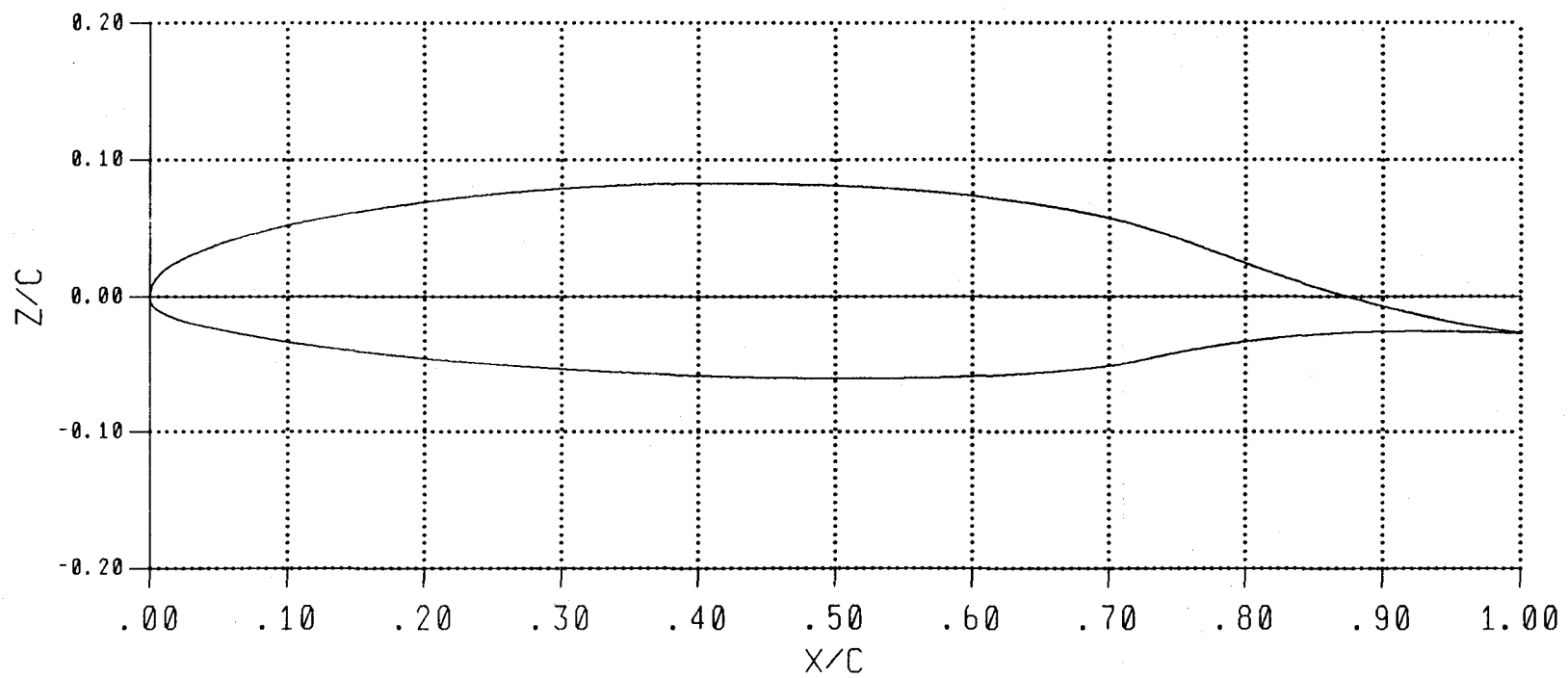


FIGURE 2. PROFILE SHAPE FOR THE NLF(1)-0414 AIRFOIL

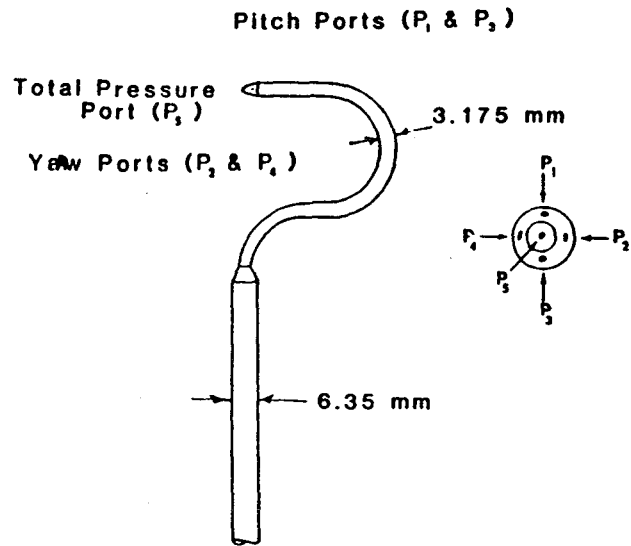


Figure 3a. Five-tube-probe head and tip detail.

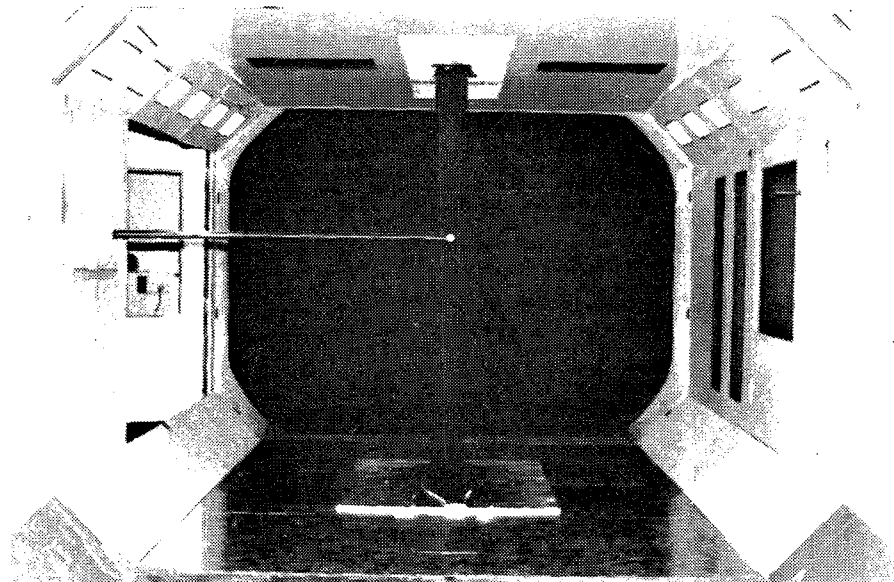


Figure 3b. Model and the five-tube-probe support stem in test section.

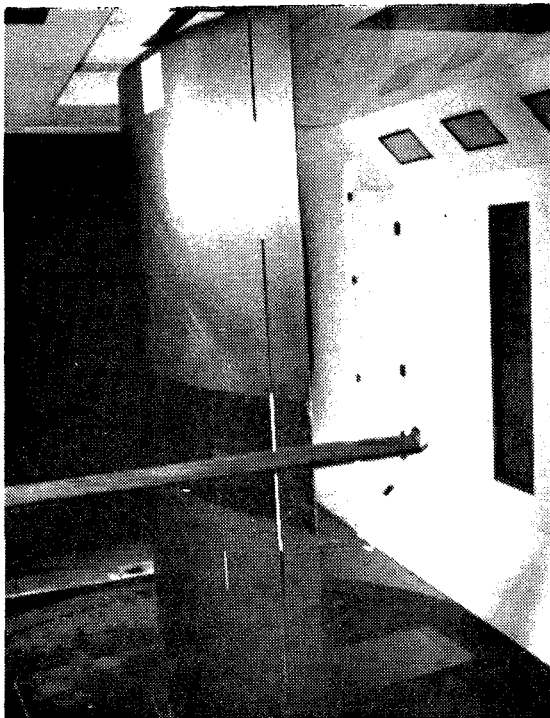


Figure 3c. Five-tube-probe attached on stem behind the model. Riblets attached to model.



Figure 3d. Close up of the five-tube-probe as mounted behind the model.

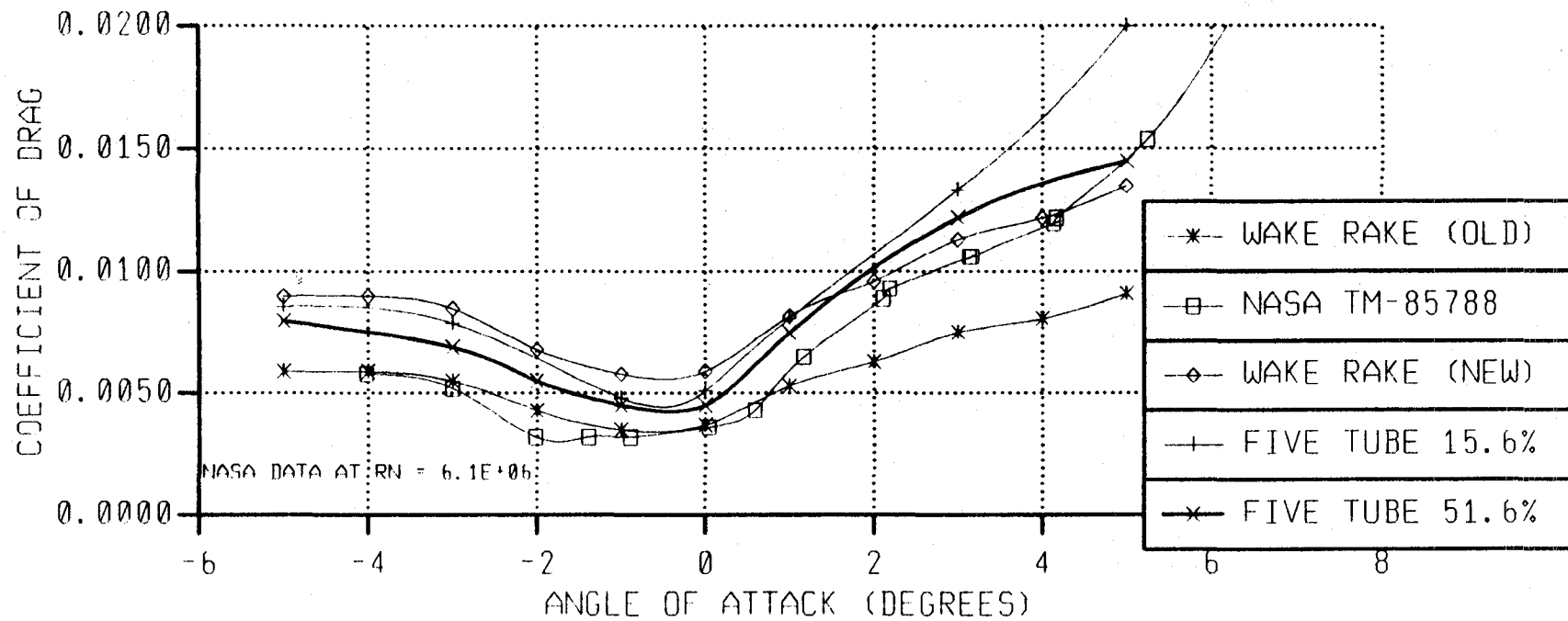


FIGURE 4: CESSNA FLIGHT WAKE RAKE CALIBRATION AND NLF AIRFOIL TEST
 DRAG COEFFICIENT -VS- ANGLE OF ATTACK, RN = 5.0E+06

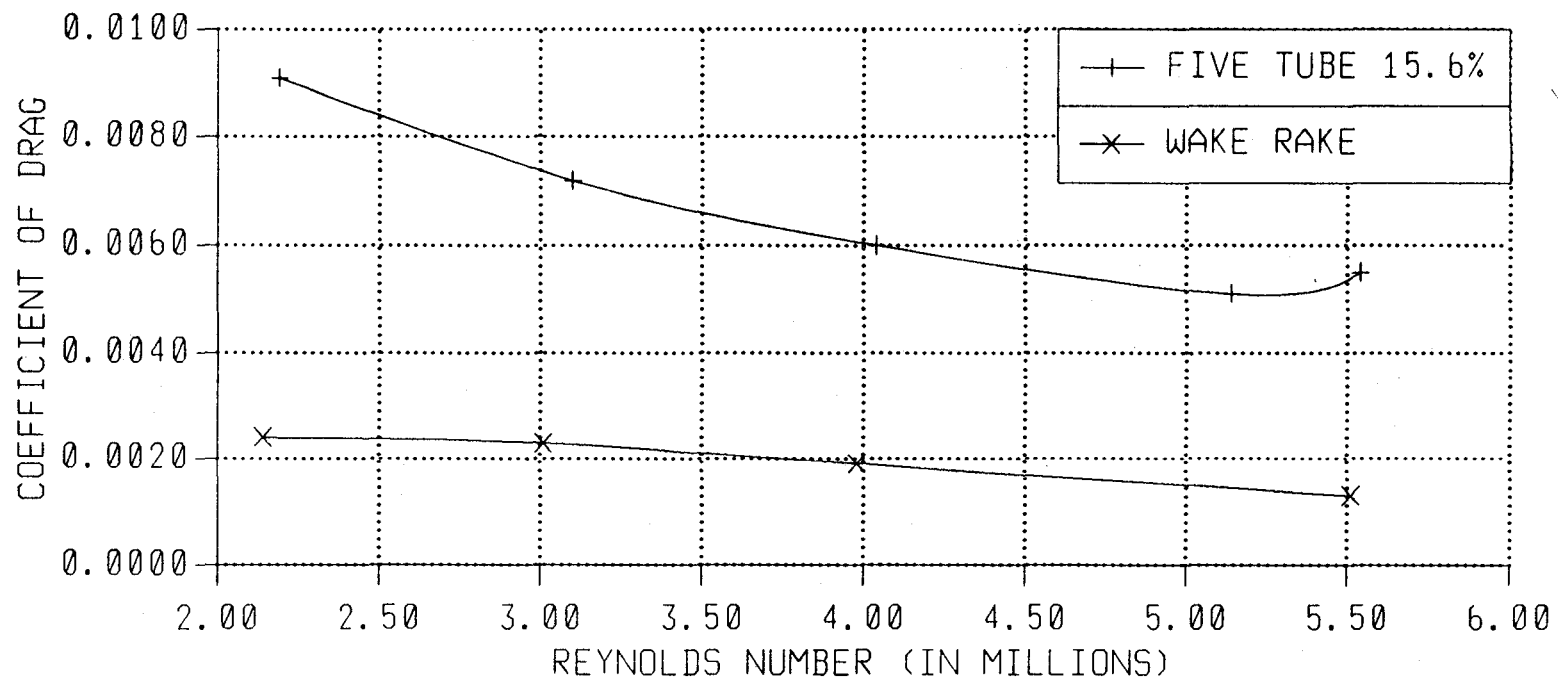


FIGURE 5. CESSNA FLIGHT WAKE RAKE CALIBRATION AND NLF AIRFOIL TEST DRAG COEFFICIENT -VS- REYNOLDS NO., A.O.A. = -0.5 DEG.

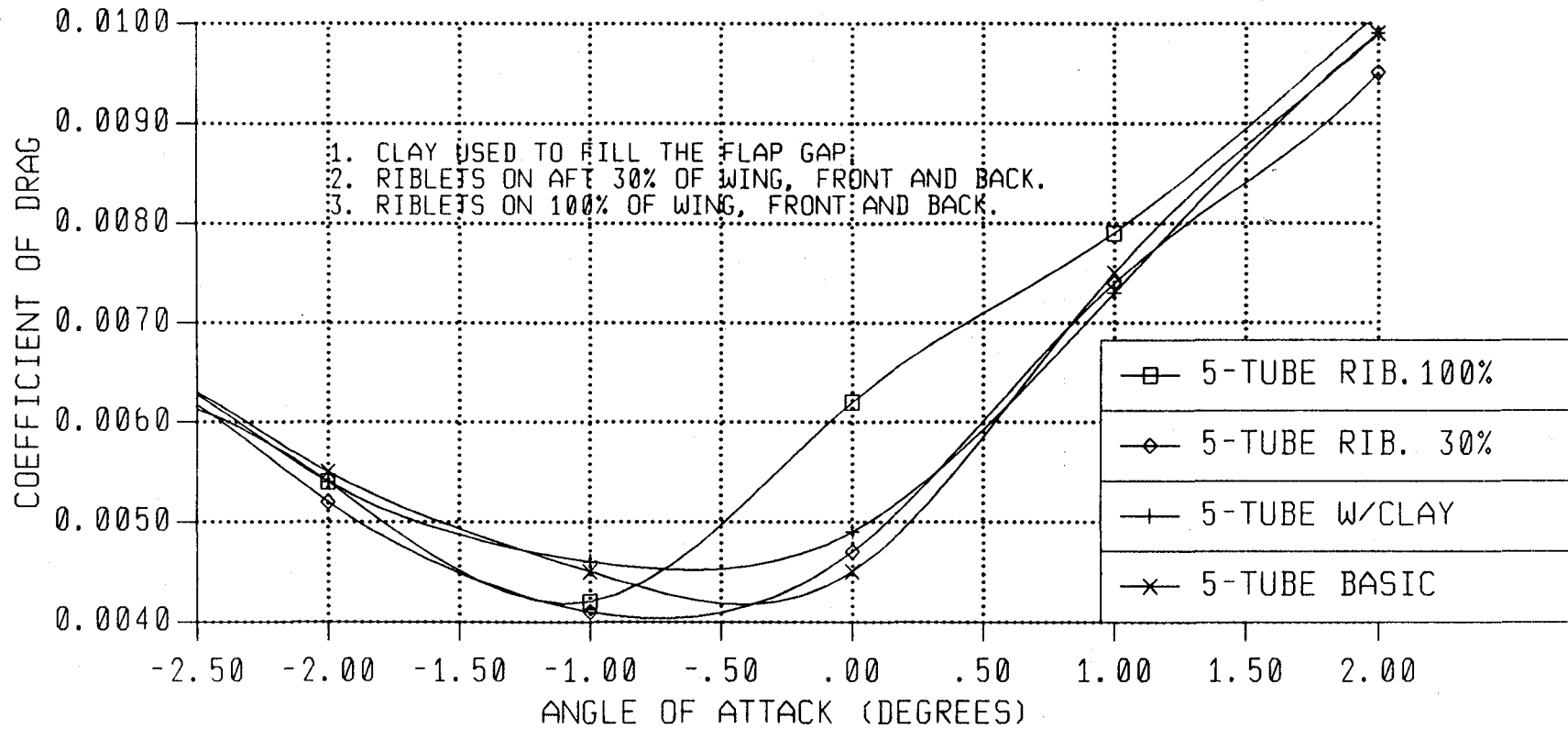


FIGURE 6a. CESSNA FLIGHT WAKE RAKE CALIBRATION AND NLF AIRFOIL TEST
 DRAG COEFFICIENT -VS- ANGLE OF ATTACK ,RN = 5.0E+06.

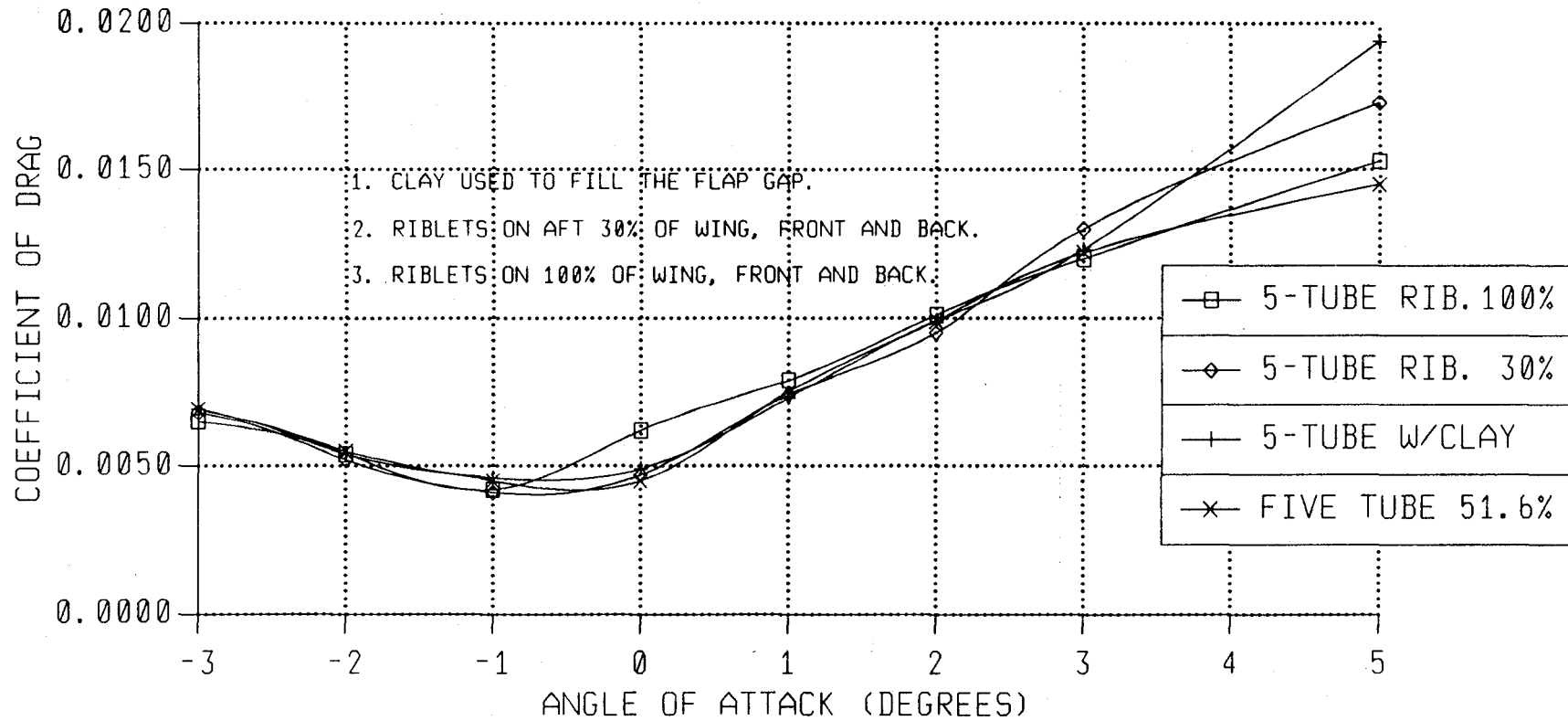


FIGURE 6b. CESSNA FLIGHT WAKE RAKE CALIBRATION AND NLF AIRFOIL TEST DRAG COEFFICIENT -VS- ANGLE OF ATTACK, $RN = 5.0E+06$.

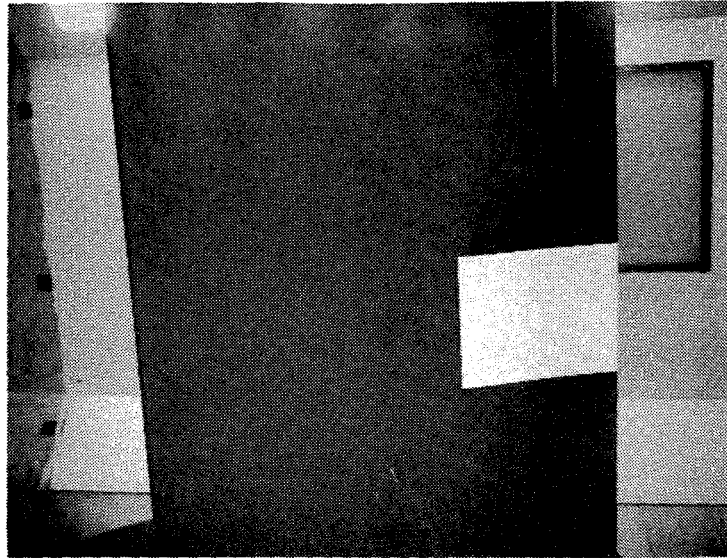


Figure 7a. Riblets attached to aft 30% of the airfoil.

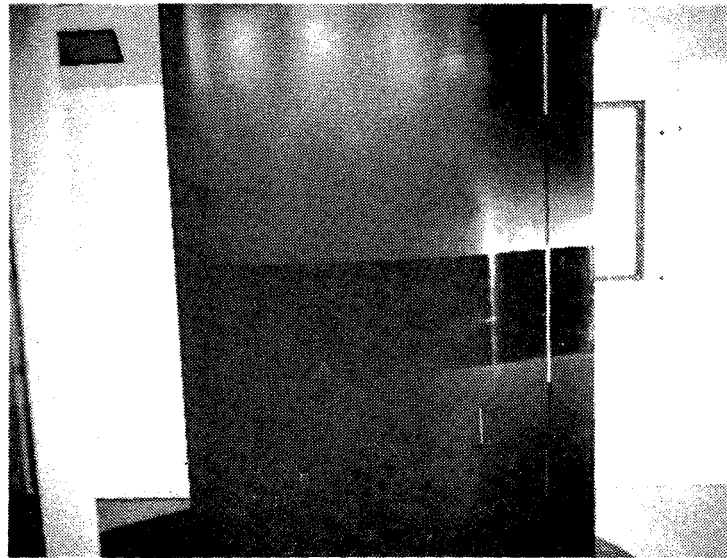


Figure 7b. Riblets as attached on 100% of the airfoil.

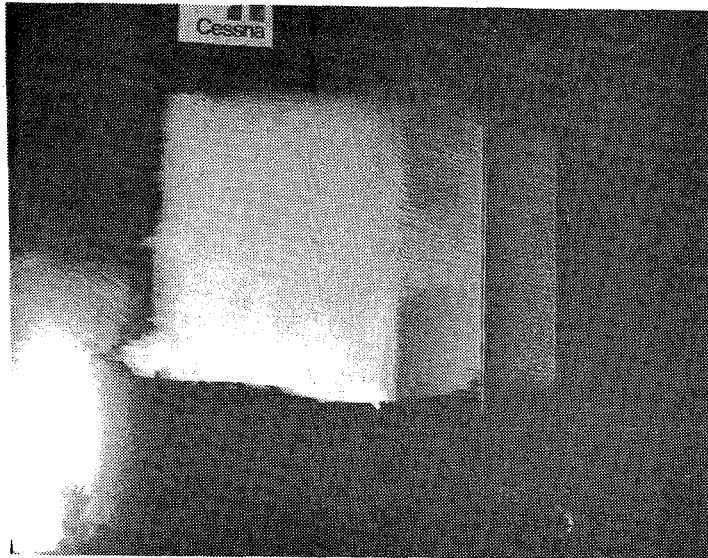


Figure 8a. Fluorescent oil visualization showing transition from laminar to turbulent flow at 75% chord. Angle of attack is -0.5 degrees, $RN = 5$ million.

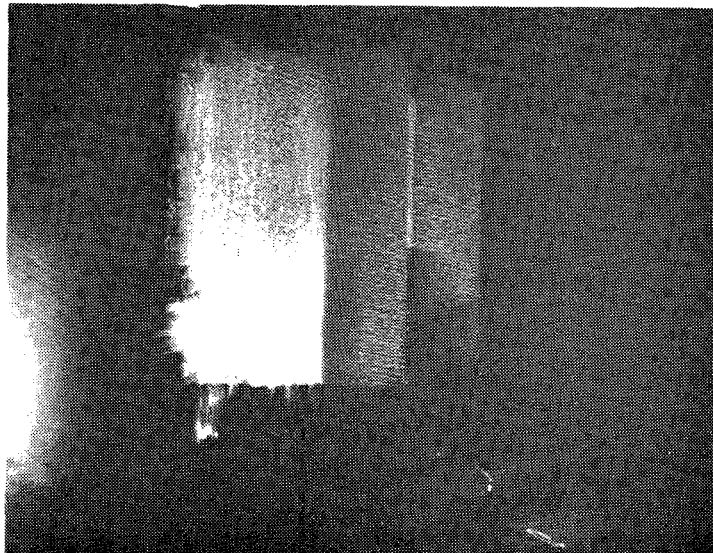


Figure 8b. Transition from laminar to turbulent flow at 75% chord. Note filling of flap gap has caused change of flow pattern, without effecting transition point.

APPENDIX

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$COMPILE
C CESSNA / N.A.S.A. FLIGHT WAKE RAKE PROGRAM
C BY: EDDIE IRANI
C
C THE FOLLOWING ROUTINE WILL CALCULATE THE PROFILE DRAG COEFFICIENT
C OF A TWO DIMENSIONAL WING IN FLIGHT. THIS ROUTINE IS A
C VARIATION OF A PROGRAM WRITTEN BY CESSNA AIRCRAFT CO. FOR A FLIGHT
C WAKE RAKE, ORIGINALLY DESIGNED BY SILVERSTEIN AND KATZOFF OF
C N.A.C.A.
C THE INPUT TO THIS ROUTINE CONSISTS OF TWO PARTS:
C 1. INPUT IS TO BE MADE IN THE BEGINING OF THE PROGRAM AS REQUIRED.
C 2. AT THE END OF THE PROGRAM THE FOLLOWING IS TO BE ADDED;
C AN ESTIMATED DRAG COEFFICIENT FOR THE FIRST RUN; THEN
C AIRPLANE WEIGHT(LBS.), ALTITUDE(FT.), CALIBRATED AIRSPEED(KNOTS)
C AND TOTAL PRESSURE COUNT FROM THE FLIGHT WAKE RAKE,
C FOR EACH RUN.
C
C
C REAL M, RH
C-----
C THE FOLLOWING DATA SHOULD BE ADJUSTED AS REQUIRED
C-----
C????? NUMBER OF RUNS TO BE MADE, ONE RUN PER ANGLE OF ATTACK.
      N = 11
C????? WING CHORD IN INCHES
      WC = 48.375
C????? WING AREA IN SQUARE FEET
      S = 150.0
C????? RAKE HIGHT IN INCHES / WC
      RH = 5.6 / WC
C????? TRAILING EDGE TO RAKE IN INCHES / WC
      TER = 7.55 / WC
C????? FLIGHT WAKE RAKE PITOT TUBE OUTSIDE DIAMETER IN INCHES
      DOUT = 0.035
C????? FLIGHT WAKE RAKE PITOT TUBE INSIDE DIAMETER IN INCHES
      DIN = 0.025
C
C????? FIND VALUE OF  $\Rightarrow (P_1 - P_0) / Q_0 = P_Q$  FOR THE WING. FIND IN
C????? REF. 2, FIGURE 5.
C????? FIND STATIC PRESSURE CORRECTION DUE TO THE EFFECT OF THE FUSELAGE
C????? ON THE RAKE  $\Rightarrow Q_P$  FIND IN REF. 1, FIGURE 6.
C????? FOR ACCURACY  $(P_1 - P_0) / Q_0$  MAY BE DIRECTLY MEASURED IN THE WAKE,
C????? THEN SET  $P_Q$  EQUAL TO THE CORRECT VALUE AND SET  $Q_P$  EQUAL TO ZERO.
C
      P_Q = 0.092
      Q_P = 0.000
C
C????? 'COUNT' IS THE CONVERSION TO CHANGE TOTAL PRESSURE COUNTS FROM
C????? THE FLIGHT WAKE RAKE TO P.S.F.; IF DATA FROM RAKE IS GIVEN IN
C????? P.S.F.. SET COUNT = 1.0
C
      COUNT= 20.0
C-----
C READ AN ESTIMATED DRAG COEFFICIENT THEN,
C READ AIRPLANE WEIGHT, ALTITUDE, CALIBRATED AIRSPEED, AND
C WAKE AVERAGE TOTAL PRESSURE COUNTS
C THESE ARE USED TO CALCULATE THE DRAG COEFFICIENTS.
C-----
C
      WRITE(6,200)
      200 FORMAT('1'.5X,'LIFT COEFFICIENT',.5X,'DRAG COEFFICIENT',//)

```

```

C:::::N ITERATIONS:::::
C
  READ(5,*) ECDO
  DO 99 I = 1, N
C
  READ(5,*) WEIGHT, ALT, CAS, PTWAKE
C
C:::::CALCULATING STATIC PRESSURE, PDIF = P(SL) - PSTAT, MACH NO:::::
C:::::AND INDICATED DYNAMIC PRESSURE:::::
C
  PSTAT = (((ALT) * (-6.8664E-6) + 1.0) ** 5.2561) * 2116.2
  PDIF = (((((CAS ** 2) * 4.5709E-7) + 1.0) ** 3.5) - 1)*2116.2
  M = SQRT(5 * (((PDIF/PSTAT) - 1.0) ** 0.2857) - 1.0))
  QIND = 0.7 * (M ** 2) * PSTAT
C
C:::::ITERATIONS TO DECREASE ERROR DUE TO ESTIMATED CD:::::
C
  DO 88 J = 1, 3
C
C:::::CALCULATING ETA:::::
C
  ETA = 2.23 * SQRT(ECDO) / (TER + 0.3)
  ETA = 2.42 * SQRT(ECDO) / (SQRT(TER) + 0.3)
C
C:::::CORRECTION FOR EQU. (6):::::
C
  DELTA = 0.131*(DOUT) - 0.0821*(DIN)
  COR2 = 2.0 * ETA * DELTA * QIND / RH
C
  DPDQ = PQ - QIND * QP/100.0
  PWT = (PTWAKE/COUNT) + PSTAT
  PW = PWT - COR2
  PT = PSTAT - QIND
  DP = (PT - PW) / QIND
C
C:::::CORRECTION FOR FIG. 11, FOR RAKE/WAKE = 2 ::::::
C
  FRAC = ETA * QIND / PSTAT
  COR1 = -0.0063 - FRAC * (0.3595 - FRAC * (-0.3198 + FRAC *
    - (0.4393)))
C
C:::::CORRECTED DP ==> DELP:::::
C
  DELP = DP - (COR1 * DP)
C
C:::::CALLING SUBROUTINE "FINDF" TO CALCULATE THE VALUE OF F:::::
C
  CALL FINDF(RH, ETA, DPDQ, F)
C
C:::::COMPC= COMPRESSIBILITY CORRECTION TO THE JONES EQUATION:::::
C
  COMPC = 0.9994 + M*(0.0152 - M*(-0.4706 + M*(0.1984
    + M*(-0.0173))))
C-----
C THE DRAG COEFF. IS CALCULATED AND PRINTED CORRESPONDING TO THE ALPHA.
C-----
C
  CDO = ABS(F * COMPC * DELP * RH)
C
  88 ECDO = CDO
C
C:::::CALCULATING THE LIFT COEFFICIENT:::::
C
  CL = WEIGHT / (QIND * S)
C

```



```

C:::::WRITE STATEMENTS:::::
C
      WRITE(6,210) CL. CDO
210  FORMAT('  .10X.F6.4,16X.F6.4)
C
      99 CONTINUE
C
      WRITE(6,1000)
1000 FORMAT('1')
      STOP
      END
C
C
C-----
      SUBROUTINE FINDF(RH, ETA, DPDQ, F)
C-----
C      CALCULATING "F" FROM SILVERSTEIN AND KATZOFF, BY A
C      TEN POINT GAUSSIAN QUADRATURE FOR SINGLE INTEGRATION
C-----
      DOUBLE PRECISION A(10), T(10), Y, DY, EQU, ANS, RES
C:::::LOWER LIMIT OF INTEGRATION
      AY = -(RH / 2.0)
C:::::UPPER LIMIT OF INTEGRATION
      BY = (RH / 2.0)
C
      PZ = 3.1415927 / RH
C
C:::::ROUTINE FOR A SINGLE INTEGRATION:::::
C
      A(1) = 0.066671344308688
      A(2) = 0.149451349150581
      A(3) = 0.219086362515982
      A(4) = 0.269266719309996
      A(5) = 0.295524224714753
      A(6) = 0.295524224714753
      A(7) = 0.269266719309996
      A(8) = 0.219086362515982
      A(9) = 0.149451349150581
      A(10) = 0.066671344308688
C
      T(1) = -0.973906528517172
      T(2) = -0.865063366688985
      T(3) = -0.679409568299024
      T(4) = -0.433395394129247
      T(5) = -0.148874338981631
      T(6) = 0.148874338981631
      T(7) = 0.433395394129247
      T(8) = 0.679409568299024
      T(9) = 0.865063366688985
      T(10) = 0.973906528517172
C
      ANS = 0.0
      DY = (BY-AY) / 2.0
      DO 11 K = 1, 10
      Y = (BY-AY) / 2.0 * T(K) - (AY-BY) / 2.0
C
C:::::THE EQUATION TO BE INTEGRATED:::::
      EQU = DSQRT(1.0 - DPDQ - ETA * (DCOS(PZ * Y)) ** 2) *
      (1.0 - DSQRT(1.0 - ETA * (DCOS(PZ * Y)) ** 2))
C:::::
C
      EQU = EQU * A(K)
      ANS = ANS - EQU
      11 CONTINUE
      RES = DY * ANS
C:::::CALCULATING "F":::::
      F = (4.0 / (ETA * RH)) * RES
C
      RETURN
      END

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