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Wind Tunnel Tests of Rotor Blade Sections With Replications of Ice Formations Accreted in Hover

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John D. Lee
The Ohio State University
Aeronautical and Astronautical Research Laboratory
Columbus, Ohio

and

Jack H. Berger and Timothy J. McDonald
FluiDyne Engineering Corporation
Columbus, Ohio

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SUMMARY

Full scale reproductions of ice accretions molded during the documentation of a hover test program were fabricated by means of epoxy castings and used for a wind-tunnel test program. Surface static pressure distributions were recorded and used to evaluate lift and pitching moment increments while drag was determined by wake surveys. Through the range of the tests, corresponding to those conditions encountered in hover and in flat pitch, integration of the pressure distributions showed negligible changes in lift and in pitching moment, but the drag was significantly increased.

NOMENCLATURE

C	Chord Length (0.827 meter or 1.750 feet)
C_D	Drag Coefficient
C_L	Lift Coefficient
C_M	Pitching Moment Coefficient (about quarter-chord)
C_P	Pressure Coefficient
LWC	Icing Cloud Liquid Water Content (gm./m^3)
M	Mach Number
R	Radius of Rotor (7.315 m., 24 ft.)
r	Radial Distance Along Rotor
X	Chordwise Distance from Leading Edge
	Attack Angle

INTRODUCTION

As part of the joint NASA/Army HIFT program (Helicopter Icing Flight Test), ice was accreted on the main rotor of a UH-1H helicopter during several hover flights at the Canadian hover spray rig at Ottawa and documented upon landing. The program and the documentation procedures are described in references 1,2 and 3. Table I lists the pertinent information for the five flights which were successfully documented. In order to determine the aerodynamic characteristics of the blade with the ice, a wind tunnel test program was planned and executed using cast replicas of the ice formations on full scale models.

One of the methods for documenting the ice shapes, a silicone-rubber molding technique, was so successful in capturing the fine details of the ice formations that it provided the basis for the wind tunnel program. Molds were obtained at up to eight spanwise stations on the blade, depending on the extent of the ice, with each mold being about 10 inches in length (spanwise). The last of the five flights (flight E) was selected as having the greatest potential for correlation between the measured aircraft performance and performance calculations based on section aerodynamics. For that flight, Figure 1 shows the projections from the ends of the silicone molds, from which the accreted ice-shapes could be reproduced in some detail. The spanwise distribution of the ice is presented in Figure 2.

Models were fabricated by repetitively casting replicas of the ice formations from the molds onto sections of UH-1H helicopter blades which were then pressure tapped in the front 35% (of chord) to acquire pressure distributions. The wind tunnel test program was performed by FluiDyne Engineering Corporation (Minneapolis) in their 66-inch transonic tunnel [4].

This report summarizes the wind tunnel program and the results obtained from it.

WIND TUNNEL MODELS

Four wind tunnel models were fabricated using 66 inch sections of UH-1H main rotor blades supplied by the Army; for reference purposes, the section was basically a NACA-0012 profile. The ice formations were reproduced in epoxy castings from the silicone molds directly onto the blade-model; the casting was performed repetitively using a single mold, starting with the first casting spanning the centerline and then alternating on either side to form a 66 inch span quasi-two-dimensional model. This procedure was followed in order that the best reproduction would be at the mid-span in case any mold material was lost in removing it from each epoxy casting; the detail of the reproductions indicated that this precaution was unnecessary.

Pressure taps were installed at the mid-span of the model by drilling through to the D-spar and inserting steel hypodermic tubing through the hole and out the end of the D-spar. This procedure was possible for only the front 35% of the models; however, some preliminary wind tunnel tests in scale indicated that the pressure field distortion due to the ice accretion shape was confined to the front 35% or less, and the full-scale data later confirmed this. Three taps were located on or beside the main ice formation and thereafter at 5% intervals through 35%. Although data from such a set was insufficient to define the details of the flowfield, it was considered adequate for calculating changes in lift and pitching movement.

Models were fabricated for 4 stations (r/R = 0.44, 0.60, 0.77 and 0.93) as being representative of the blade in providing a minimum amount of data for evaluating the rotor performance. A "clean" blade model provided baseline reference data for the 3 inner stations as well as primary data for r/R = 0.93 since no ice was accreted that far out for flight E. The tap layout for the four models is given in Figure 3.

For two stations (r/R = 0.44 and 0.60) an attempt was made to simulate the ice formations with simple strips of wood cut, in cross-section, in a trapezoidal shape approximating the shape of the main ice formation. These were attached, in turn, to the clean model. The wood strips were not tapped and so the pressure distributions from those tests were incomplete in the vicinity of the leading edge.

FACILITY DESCRIPTION

The full-scale rotor-blade models were tested in the 66 in. X 66 in. transonic wind tunnel of Fluidyne Engineering Corporation, Minneapolis. This is a slotted-wall tunnel driven by air ejectors; it has an atmospheric intake which can be heated to avoid moisture condensation (see Figure 4). A multiple tube wake-rake was used for the determination of drag. Figure 5 shows the arrangement for mounting the models in the wind tunnel and the location of the wake rake (at 0.75 chord length downstream from the trailing edge). A photograph of a model mounted in the tunnel is given in Figure 6. In a typical test sequence, the test section Mach number was attained by adjusting the ejector flow rate, the model pressures and the wake rake pressures were acquired using PSI (Pressure Systems Inc.) pressure transducers and the appropriate reference pressures and temperatures were recorded. The model pressures were reduced to coefficient form while the wake distribution of momentum deficit was integrated to drag coefficient. Reference 4 gives a full description of the wind tunnel and the instrumentation used in the program.

Some of the data from the clean model were used to evaluate the wall interference in the test section. The results of that study are contained in the Appendix; the essential feature is a correction to attack-angle was necessary due to lift-induced downwash:

$$\Delta\alpha = \frac{-0.155}{\sqrt{1 - M^2}} \cdot \alpha_{SET}$$

TEST PROGRAM

For the selected blade stations, the operating conditions were furnished by Bell Helicopter for the hover and the "flat-pitch" (no lift) condition, since these corresponded to the conditions at which the ice was accreted and at which the engine performance was recorded. Figure 7 shows the corresponding range of conditions for the stations. It should be noted that a given station is (ideally) operating at a fixed point on the plotted lines for the aircraft in hover or in flat-pitch; actually, due to the finite though low wind speed (9 knots for flight E) there was a small amount of blade oscillation for this "near-hover" case. Wind tunnel test points for each model were selected to be on the indicated operating line for that station and to bracket the flight test conditions.

Table II contains the list of the conditions for the tests on the different models along with the measured drag coefficients; the detailed distributions of the surface pressures and of the wake profiles are contained in reference 4.

RESULTS AND DISCUSSION

In Figures 8 A-D, the drag coefficients from the wind tunnel program have been plotted for the 4 stations respectively (Note that C_D is on a log scale). The attack angle has been corrected for the tunnel downwash. Although the data are sparse, they are sufficiently regular to allow a reasonable evaluation of the drag change caused by the ice accretions over the range of the testing. However, additional points would be useful in forming a better data base. Some typical sets of pressure distributions are presented in Figures 9A and 9B.

The fact that the very simple simulation of the ice formations by wooden strips gave drag levels comparable with the cast replicas is encouraging from the longer-range viewpoint. The use of simple substitutes of two-dimensional shapes plus distributed grain roughness is not new (reference 5) and the results obtained in this test series indicates that, with further research in this area, reliable information may be obtained by this approach using scale models.

Examination of the pressure distributions (i.e. Figures 9) shows that the disturbances caused by the ice shapes were somewhat erratic, but the basic profile still dominated the field. Differences between the data from the castings and the strip simulations might be due to resolution (insufficient number of taps), placement errors on the model, attack angle differences and deviations between models as well as inexact modelling. The similarities, however, are again encouraging for the application of a modelling technique. Inspection of the data lead to a conclusion that the changes in lift and pitching moment would be small. This was verified by integration of the pressure coefficient distributions which showed maximum changes to be within the scatter of the measurements; these were 0.01 for C_L and 0.005 for C_M over the range of the acquired data.

The above observations apply to the range of conditions for the test program, which was representative of the flight range. For example, it would be expected that the ice formations would produce larger effects outside of the indicated range of attack angle and Mach number, that the stall angle would be reduced and that C_L-Max would be less; however the rotor did not operate under those conditions.

Overall, the techniques which were applied for the hover case have been successful and further work of this type will provide not only data useful for evaluating helicopter performance degradations, but a better understanding of the ice accretion phenomena and of the detailed flow field associated with its formations. It is quite possible that some part of the ice formations may have been lost or eroded during the time taken to land and document the shapes in many cases; the extreme detail observed in the molds of flight E in particular would suggest that that was not the case for that flight. On the other hand, the relatively close agreement between the data from the ice replicas and the simple two-dimensional strip simulations would suggest that the fine detail is not important for the determination of the aerodynamic characteristics of the iced sections.

APPENDIX: EVALUATION OF TUNNEL INTERFERENCE

Since the rotor section was basically a NACA-0012 (with a chord of 21 inches), there were considerable reference data available, both from airfoil analysis codes and from other wind tunnel tests; it was one of the primary airfoils used to determine the magnitude of the wall interference in both the OSU/AARL airfoil tunnels [6] and the same methodology was used in this case.

FEC-channel 10 is a slotted wall tunnel with a common plenum, and hence downwash was expected to dominate the interference. The procedure was to determine the interference in subcritical flow and then to separately evaluate the interactions in supercritical flow.

From low speed tests (M =0.3), the local increment (of pressure coefficient) was determined for specific attack-angles, i.e. the difference between the pressure coefficients on the upper and lower surfaces adjusted by subtracting the same difference at zero attack angle. This procedure was necessary to remove the effects of any local surface irregularities in an otherwise symmetrical model.

These data were then compared with the correct distribution to determine the apparent or free-air attack-angle. In practice, the "correct" data is obtained from other "corrected" wind tunnel tests and airfoil analysis codes. The data may be adjusted to a common Mach number basis, e.g. M=0, by a Prandtl-Glauert conversion. Once the mean downwash has been determined, the variations from it may be used to establish the streamline curvature; in this situation, there were not sufficient data to fully evaluated that part beyond the estimate that it was negligible compared with downwash.

In the supercritical flow range, the use of ventilated walls requires a further analysis since the wall openness determines the location of the transonic terminal shock wave. The correct shock location (and hence the correct wall configuration) can only be determined by testing models of different chord and/or by referring to "interference-free" data. In this case, the comparisons showed that the tunnel ventilation was correct in the range tested; Figures A2 and A3 are typical of cases where a close comparison could be made, assuming that the previous subcritical attack-angle adjustment was correct.

Since the downwash was generated by lift, the application of the attack-angle correction is applicable only where the lift coefficient is linear in alpha. If a fully tapped model had been available, a proper form for the correction (i.e. in terms of lift) would have been possible; however, in the application to this program, the correction is sufficient since the ranges of application are not large. Also, the specified correction applies only to a full span airfoil model of chord = 21 inches.

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2. Lee, John D.; Harding, Rorry; and Palko, Richard L.: Documentation of Ice Shapes on the Main Rotor of a UH-1H Helicopter in Hover. NASA CR-168332, Jan. 1984.

3. Abbott, W. Y.; Williams, R. A.; Belte, D.; and Stellar F. W.: Evaluation of UH-1H Hover Performance Degradation Caused by Rotor Icing. USAAEFA Report 82-12, Aug. 1983. (AD-A141252)

4. Berger, Jack H.; and McDonald, Timothy J.: Wind Tunnel Tests of Airfoil Shapes Altered by Icing and Airfoil Shapes with Deicer Boots. FluiDyne Engineering Corporation Report, Jan., 1984.

5. Bragg, M. B. and Gregorek, G. M., "Aerodynamic Characteristics of Airfoils with Ice Accretions," AIAA Paper No. 82-0282, presented at the AIAA 20th Aerospace Sciences Meeting, Orlando, Florida, January 11-14, 1982.

6. Lee, J. D., Gregorek, G. M. and Korkan, K. D. : Testing Techniques and Interference Evaluation in the OSU Transonic Airfoil Facility. AIAA-78-1118, July, 1978.

TABLE 1. - SUMMARY OF HOVER ICING FLIGHTS

Flight Number	Wind		LWC gm/m^3	Test Temp. C	Time In cloud, min.
	Speed, kn	Gustiness			
A	7	Medium	0.4	-12.0	4.5
B	10	Medium	.4	-9.5	4.25
C	6	Medium	.4	-17.5	4.0
D	4	Low	.4	-21.5	6.0
E	9	Low	.7	-19.0	3.0

TABLE II TESTS RESULTS

CONFIGURATION	RUN #	ALPHA SET	ALPHA CORR.	MACH #	CD
Clean Wing	1	0.00	0.00	0.313	0.0079
	2	0.00	0.00	0.601	0.0079
	4	4.00	3.12	0.603	0.0086
	5	4.00	3.25	0.355	0.0087
	6	8.00	6.50	0.355	0.0103
	7	8.00	6.25	0.599	0.0210
	8	16.00	14.00	0.353	0.0879
	9	11.50	8.98	0.603	0.0524
	10	11.50	9.34	0.357	0.0145
R-11	11	0.00	0.00	0.358	0.0176
	12	5.50	4.47	0.355	0.0256
	13	7.25	5.90	0.355	0.0340
	14	10.00	7.80	0.356	0.0802
	15	2.50	2.00	0.355	0.0195
R-15	16	0.00	0.00	0.423	0.0120
	17	2.50	2.00	0.463	0.0120
	18	5.50	4.40	0.515	0.0152
	19	7.25	5.70	0.543	0.0165
R-19	20	0.00	0.00	0.535	0.0098
	21	2.50	1.95	0.593	0.0094
	22	5.50	4.23	0.648	0.0121
	23	7.30	5.54	0.688	0.0274
Clean Wing	95	0.00	0.00	0.751	0.0080
	96	0.00	0.00	0.795	0.0105
	97	0.00	0.00	0.894	0.0410
	98	1.00	0.75	0.701	0.0080
	99	2.00	1.57	0.573	0.0077
	100	2.00	1.47	0.746	0.0098
	101	2.50	1.95	0.603	0.0078
	102	5.00	4.00	0.507	0.0085
	103	5.00	3.90	0.600	0.0091
	104	5.00	3.77	0.700	0.0192
	105	3.50	2.48	0.799	0.0323
	107	6.50	5.17	0.519	0.0088
	108	6.50	5.08	0.600	0.0104
	109	16.00	13.00	0.600	0.1859
F-11	146	0.00	0.00	0.362	0.0157
	147	2.50	2.00	0.351	0.0168
	150	10.00	7.87	0.356	0.1177
	151	7.50	5.90	0.357	0.0561
	152	5.50	4.47	0.355	0.0300
F-15	153	0.00	0.00	0.417	0.0129
	154	2.50	2.00	0.465	0.0116
	155	5.50	4.38	0.512	0.0150
	156	7.25	5.74	0.539	0.0205

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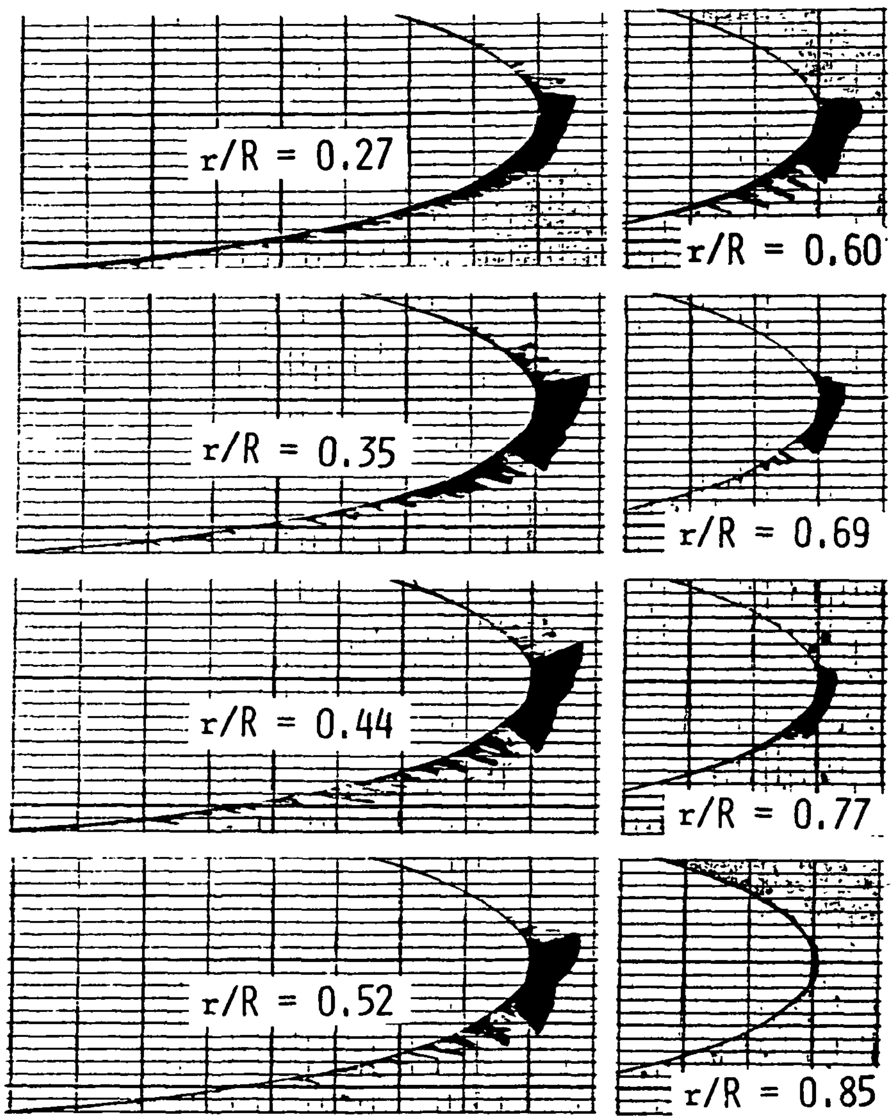


FIGURE 1. PROJECTIONS OF MOLDS OF ICE ACCRETIONS IN HOVER, FLIGHT E.

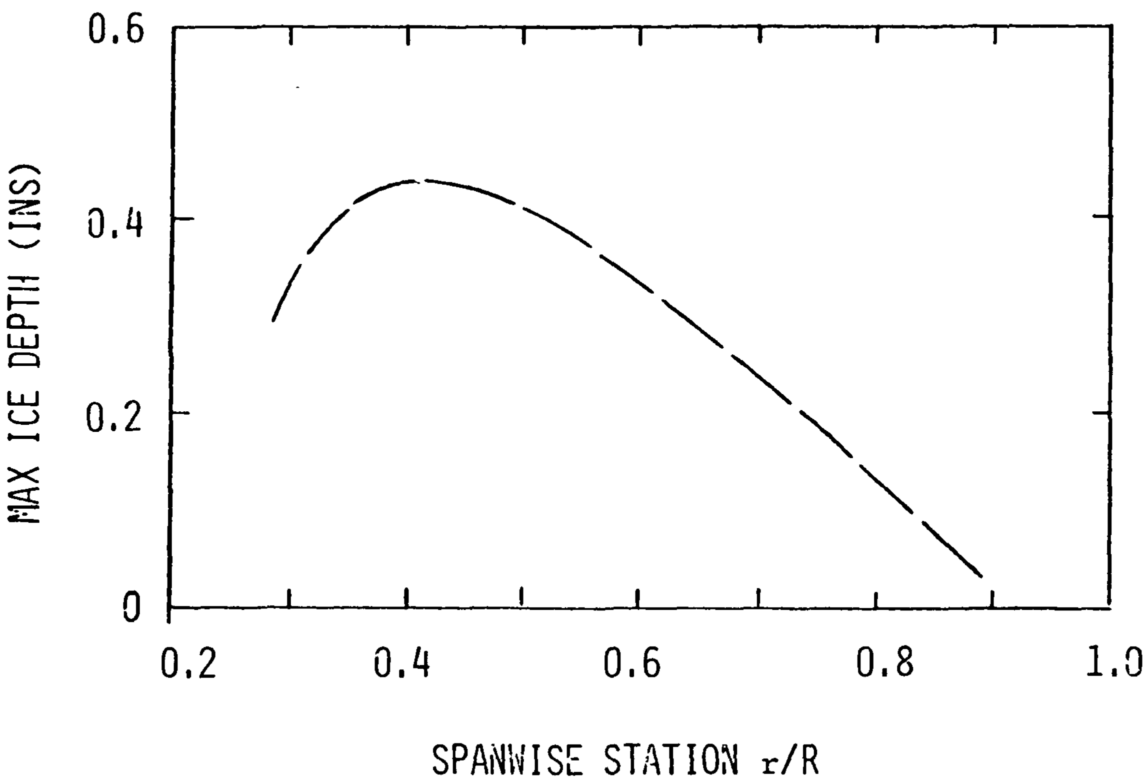
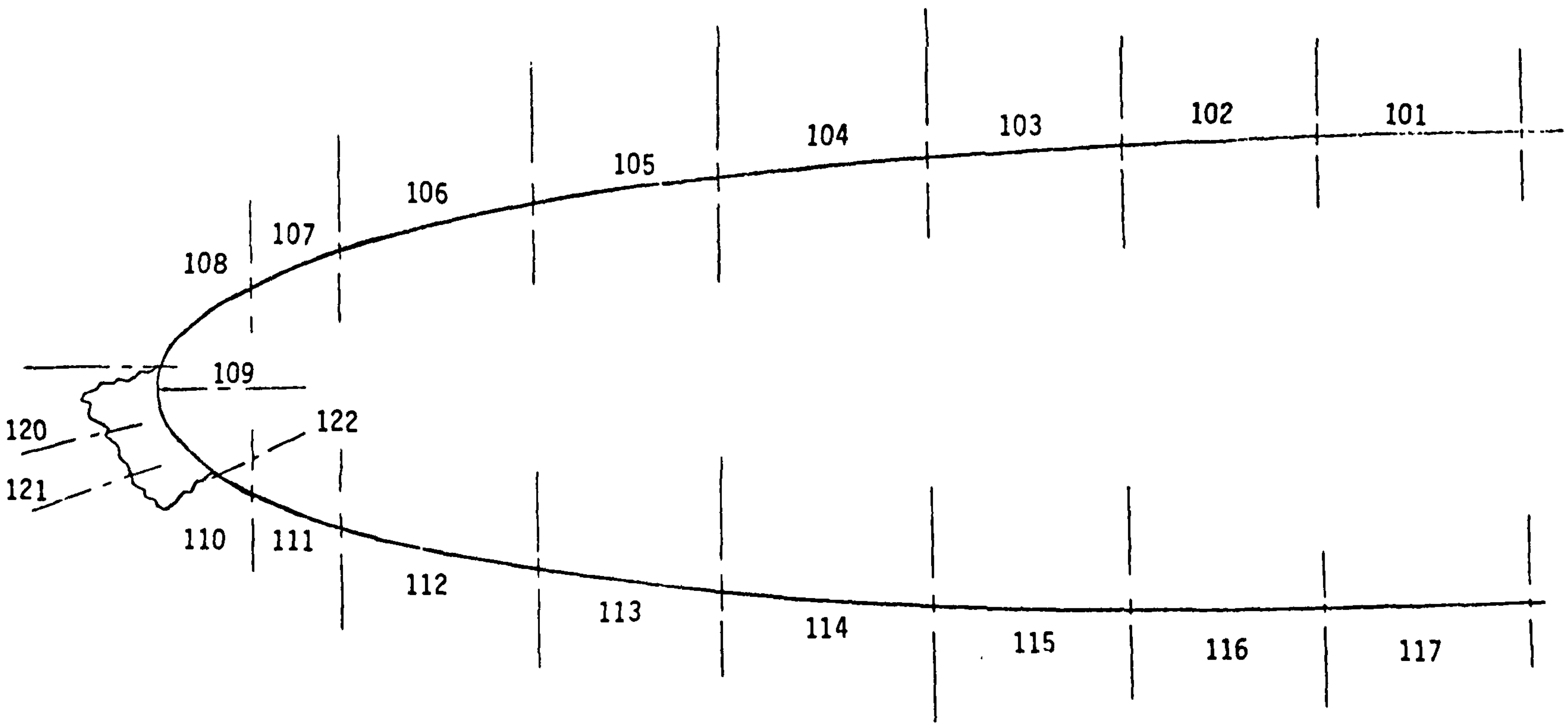


FIG. 2 MAXIMUM DEPTH OF ICE ACCRETED ON THE ROTOR FOR FLIGHT E (HOVER).



Tap No.	x = inches (from L.E.)	X/C
(109-Clean)	0.0	0.0
108,110	0.525	0.025
107,111	1.05	0.05
106,112	2.10	0.10
105,113	3.15	0.15
104,114	4.20	0.20
103,115	5.25	0.25
102,116	6.30	0.30
101,117	7.35	0.35

Tap No.	Location R11	X/C R15	For Models R19
109	-0.005	0.0	0.0
120	-0.015	-0.013	-0.005
121	-0.01	-0.007	0.0
122	0.01	0.01	0.01

FIGURE 3. LOCATIONS OF THE STATIC TAPS ON THE MODELS.

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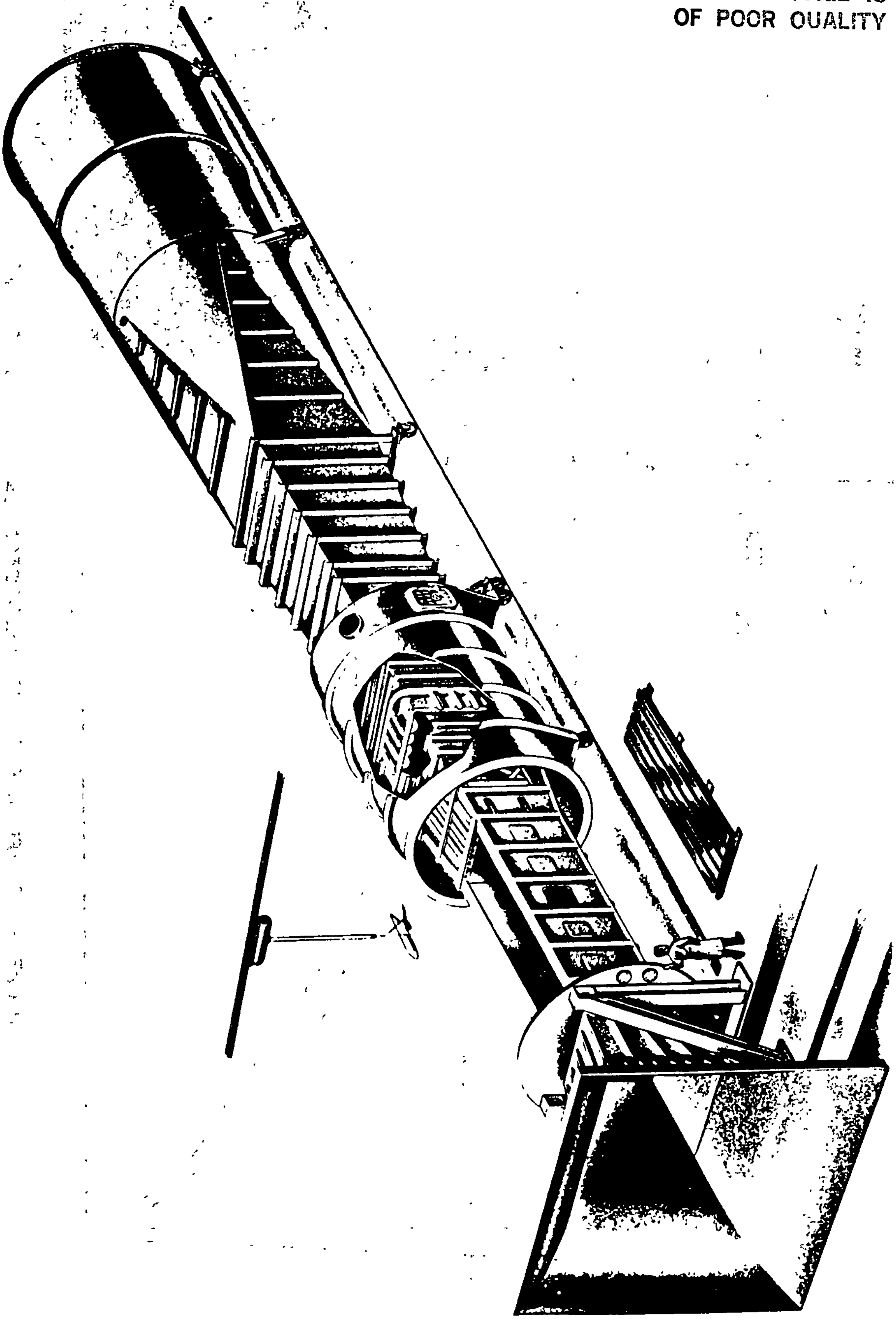

FIGURE 4. FLUIDYNE CHANNEL 10 : 66 INCH X 66 INCH TRANSONIC TUNNEL.

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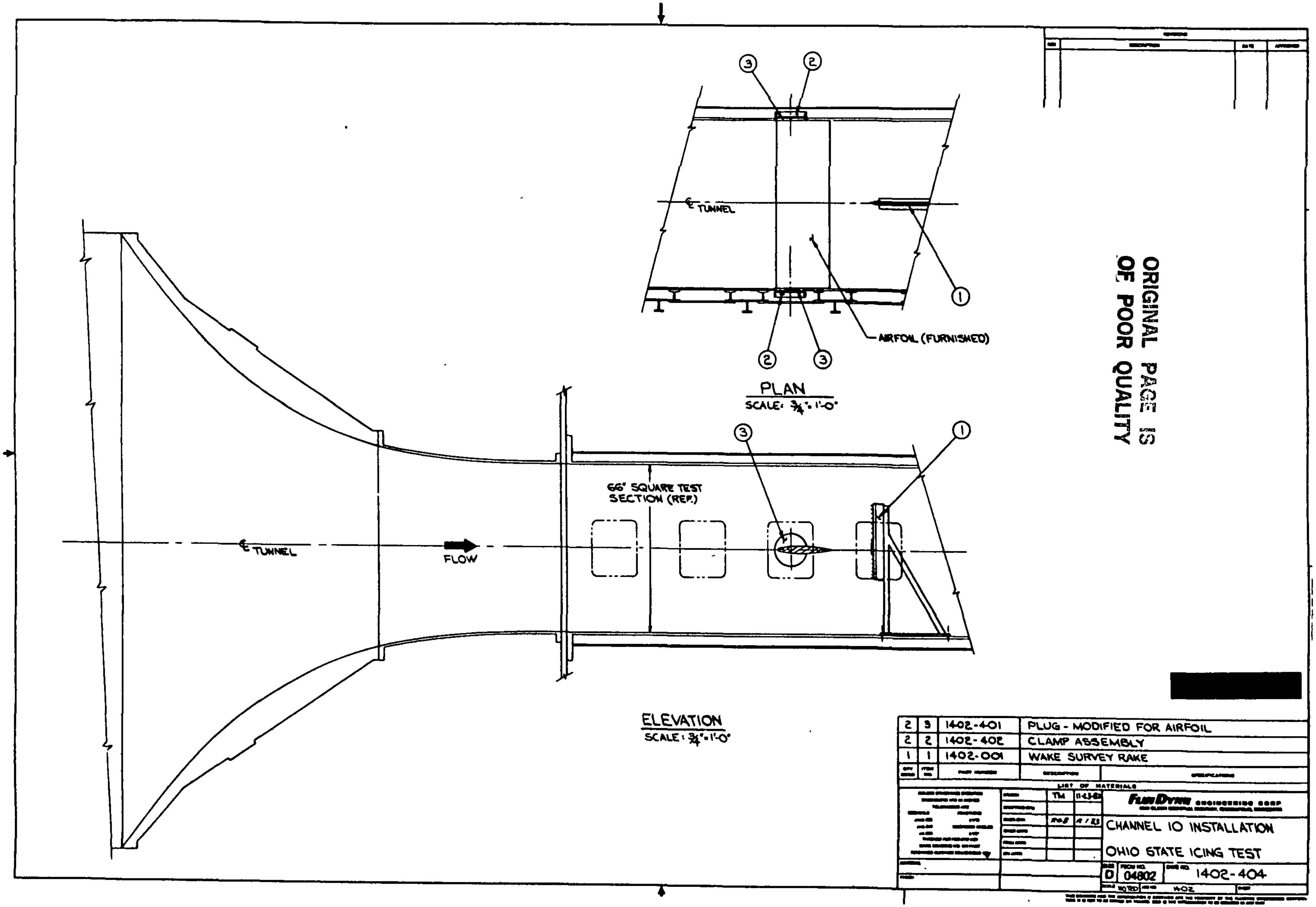

FIGURE 5. AIRFOIL TEST INSTALLATION IN FEC CHANNEL 10.

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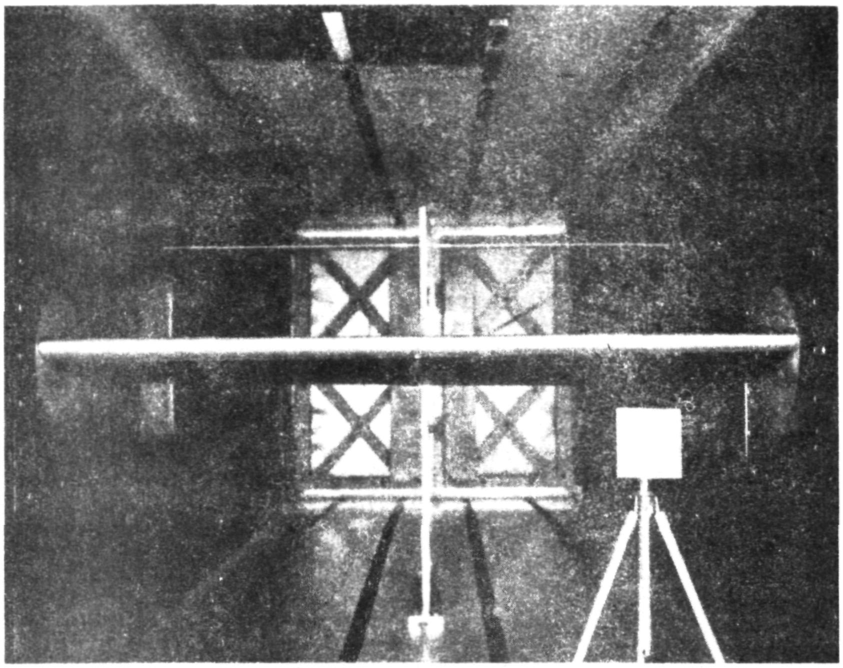
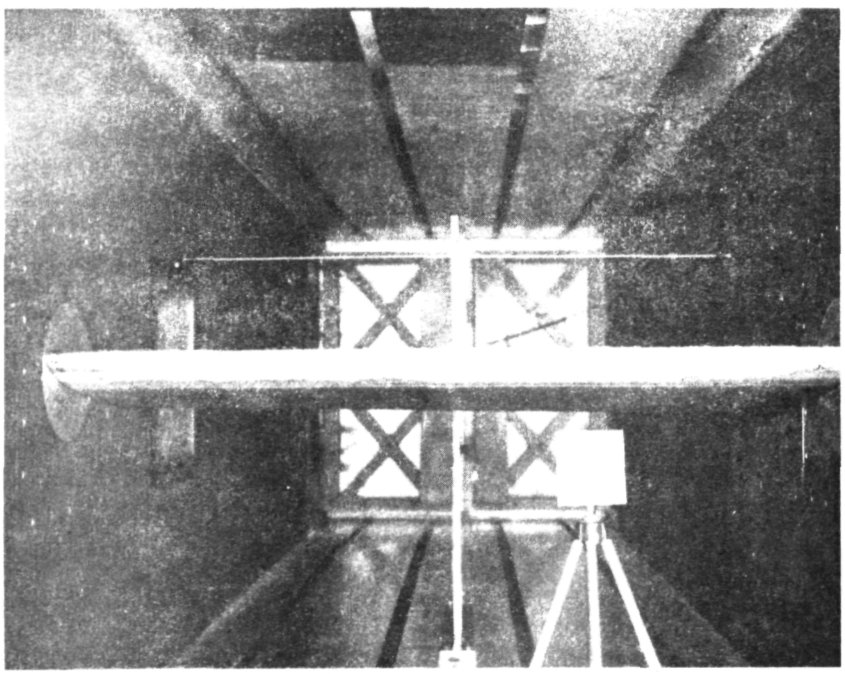


clean wing
installation



wing with
simulated ice
(R11)

FIGURE 6. PHOTOGRAPHS OF MODELS INSTALLED IN FEC CHANNEL 10.

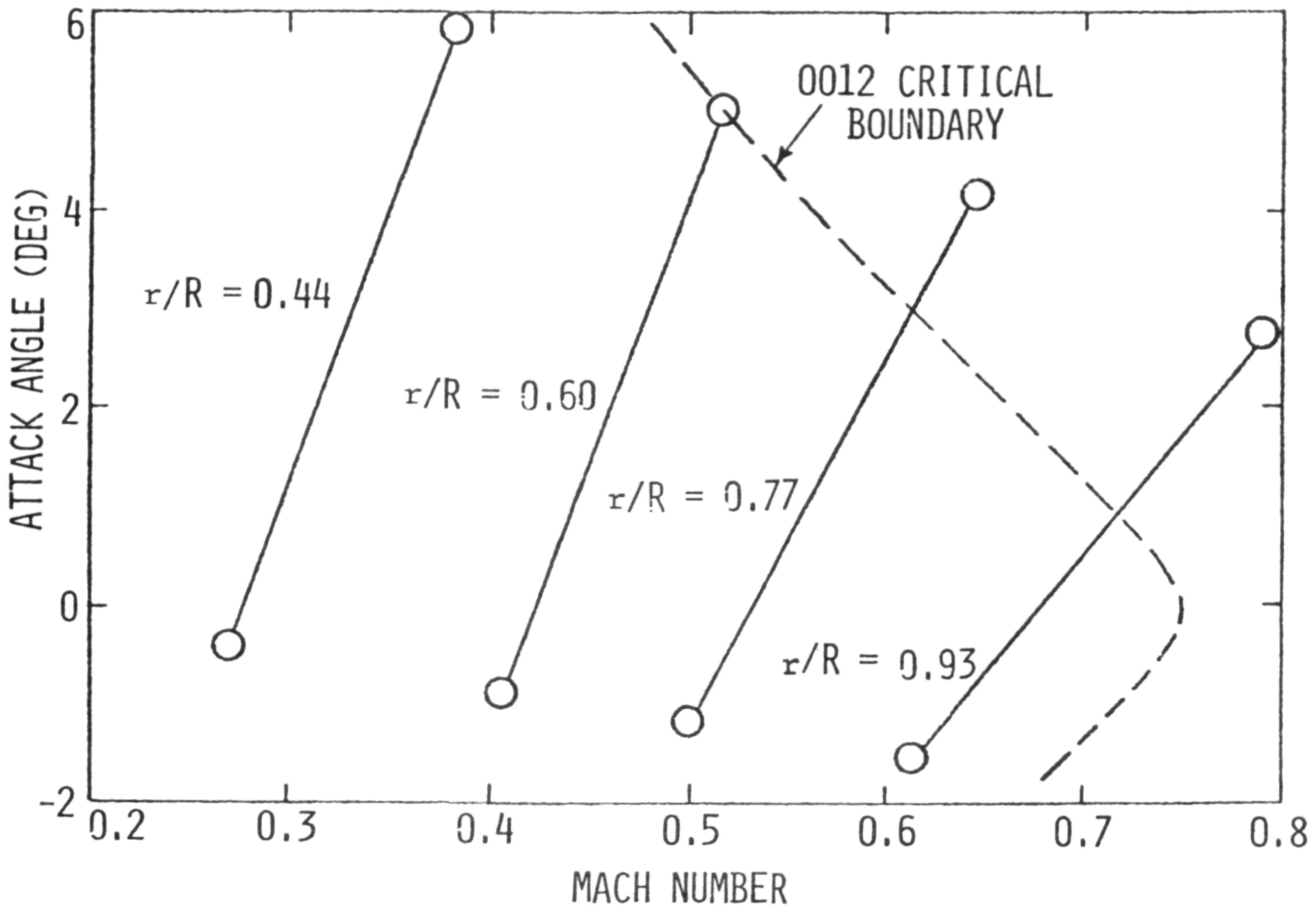


FIGURE 7. RANGE OF TEST CONDITIONS FOR THE FOUR SELECTED STATIONS COVERING HOVER AND FLAT PITCH

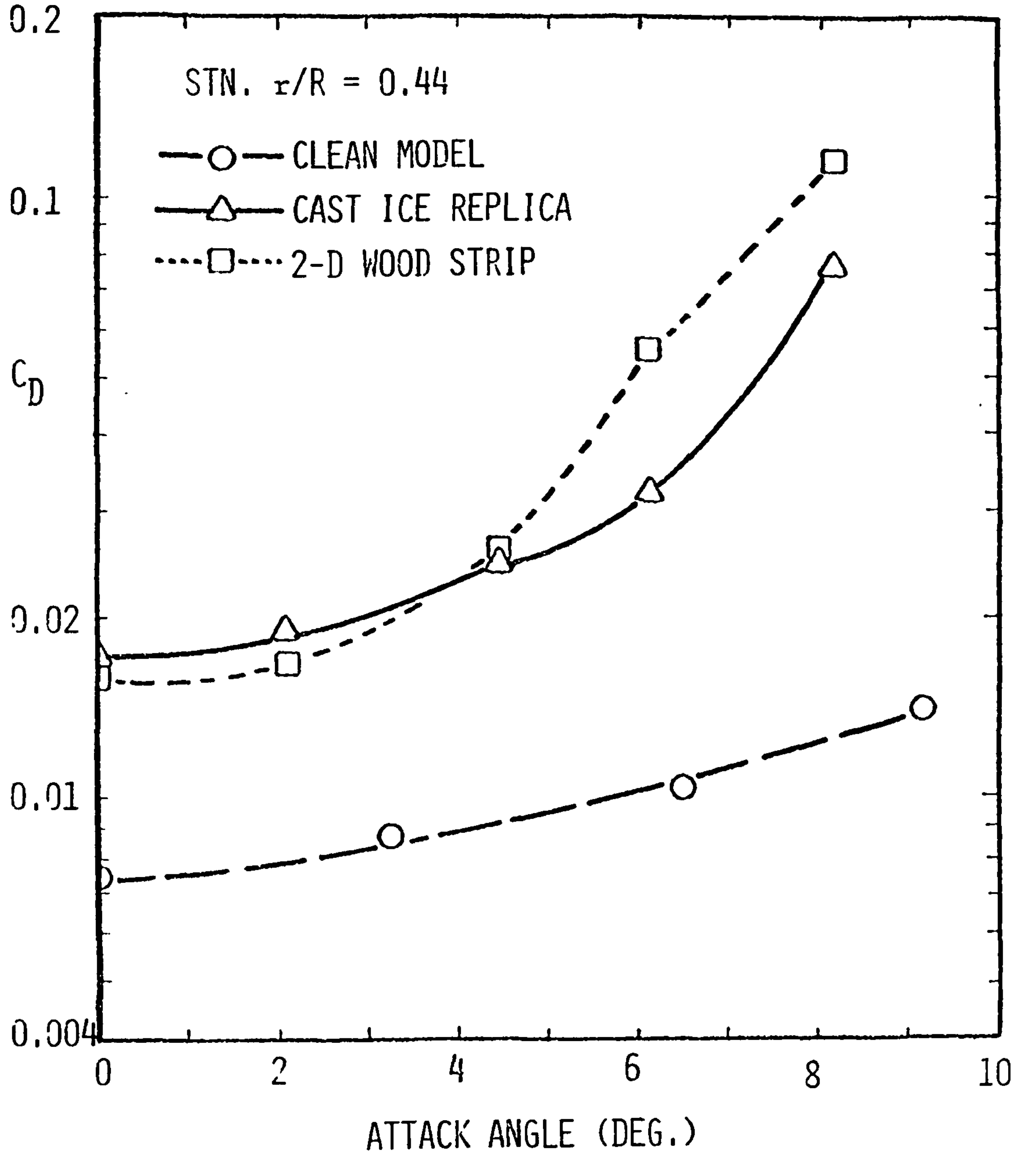


FIGURE 8. DRAG COEFFICIENTS FOR THE FOUR STATIONS THROUGH THE TEST CONDITIONS CORRESPONDING TO FIGURE 7 , CLEAN MODEL , CASTINGS FROM MOLDS AND 2-D STRIP SIMULATIONS

8A RADIAL STATION r/R = 0.44

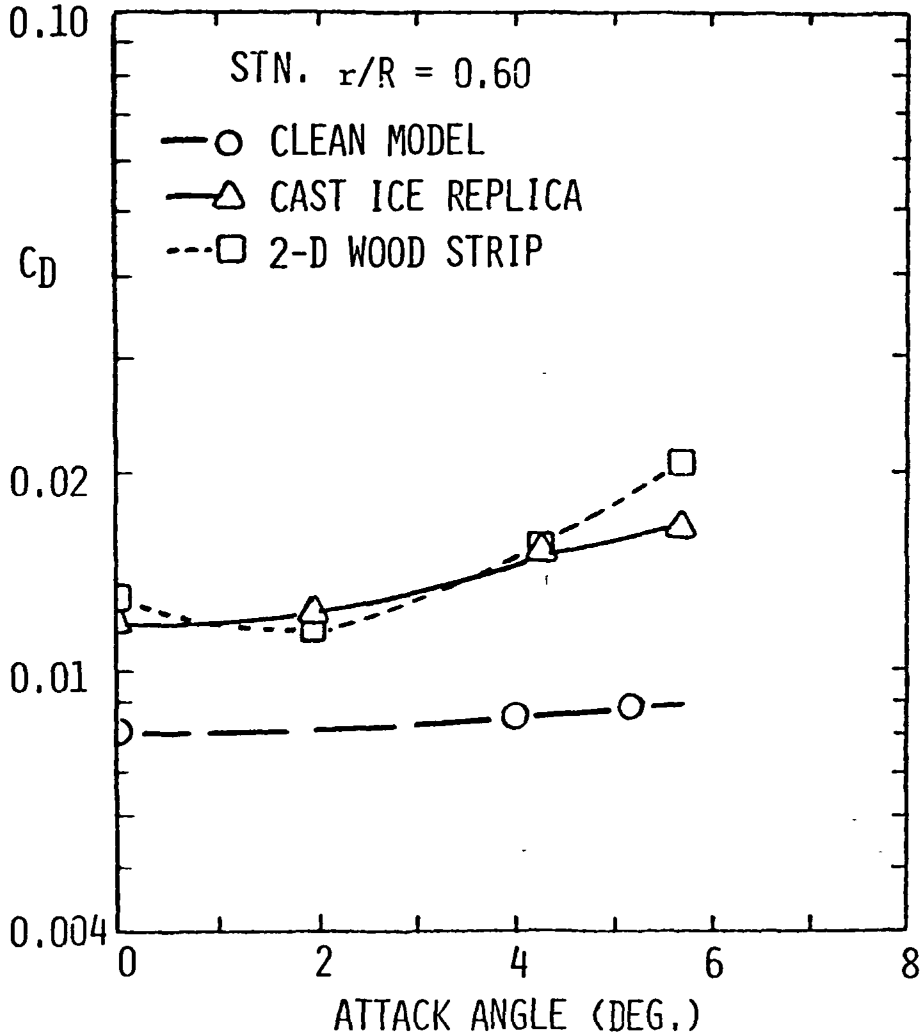


FIGURE 8. DRAG COEFFICIENTS FOR THE FOUR STATIONS THROUGH THE TEST CONDITIONS CORRESPONDING TO FIGURE 7 , CLEAN MODEL , CASTINGS FROM MOLDS AND 2-D STRIP SIMULATIONS

8B RADIAL STATION r/R = 0.60

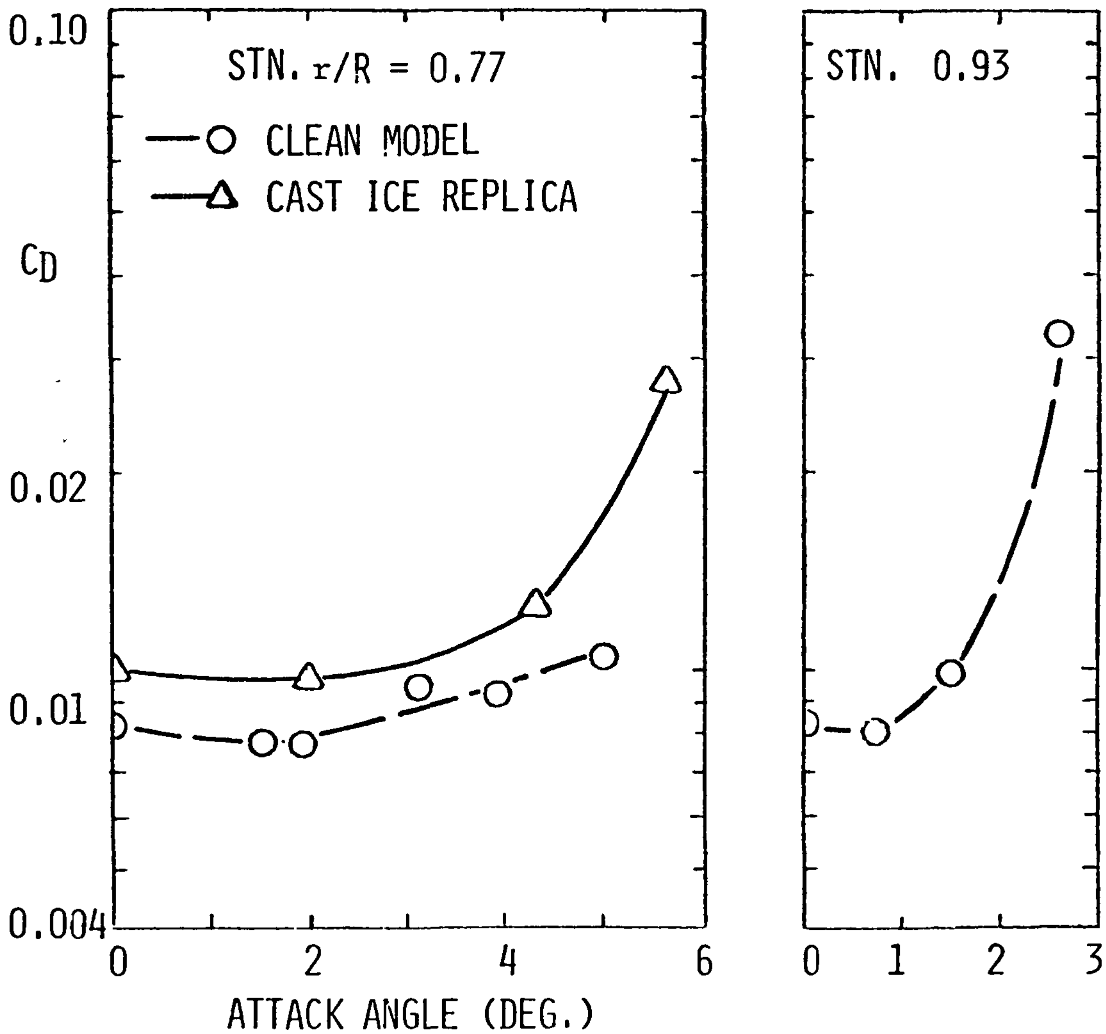


FIGURE 8. DRAG COEFFICIENTS FOR THE FOUR STATIONS THROUGH THE TEST CONDITIONS CORRESPONDING TO FIGURE 7 , CLEAN MODEL , CASTINGS FROM MOLDS AND 2-D STRIP SIMULATIONS

8C RADIAL STATION r/R = 0.77 AND 0.93

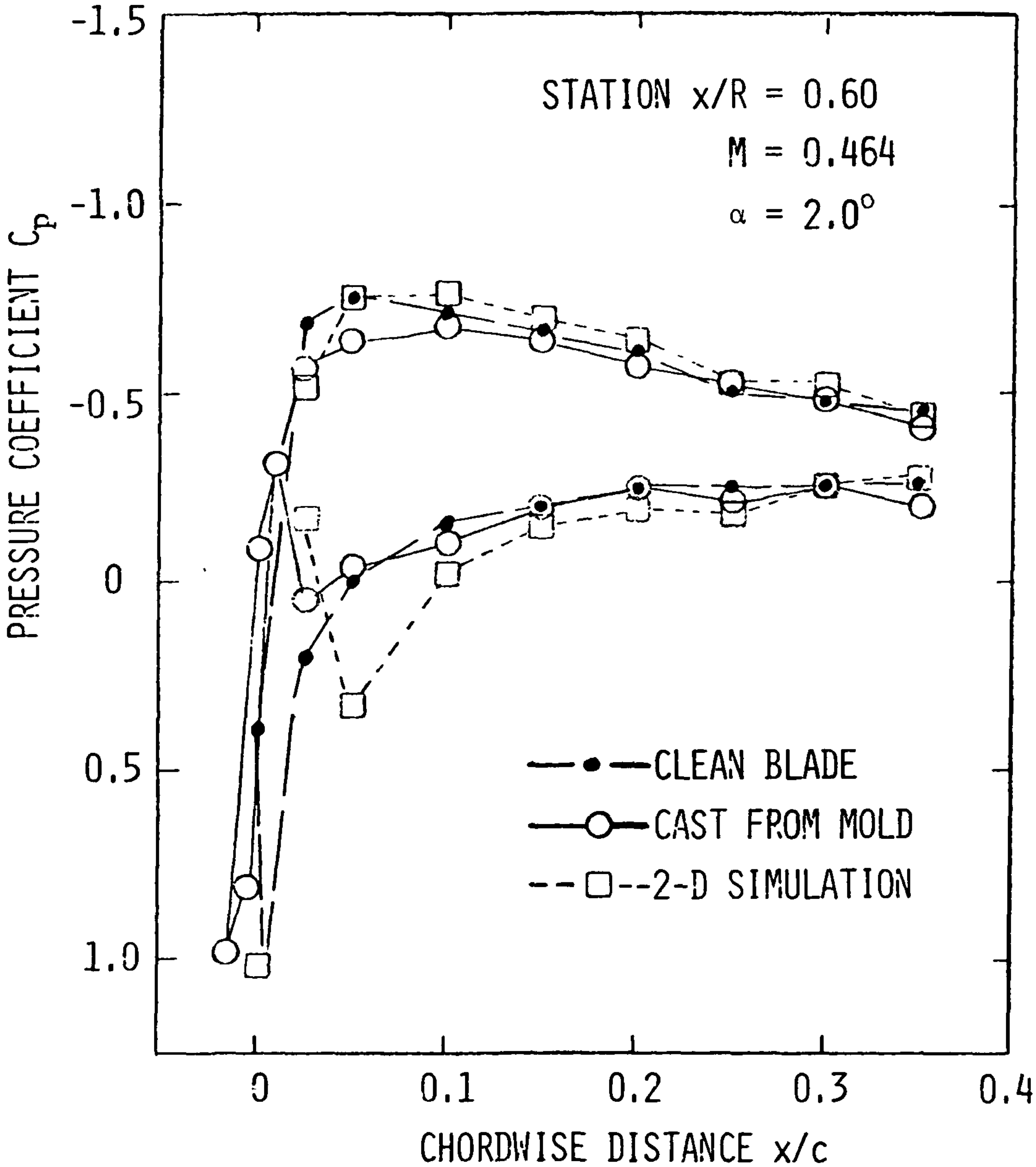


FIGURE 9. PRESSURE DISTRIBUTIONS OVER THE LEADING 0.35 OF THE BLADE SECTION WITH AND WITHOUT ICE SIMULATIONS

9A RADIAL STATION r/R = 0.60

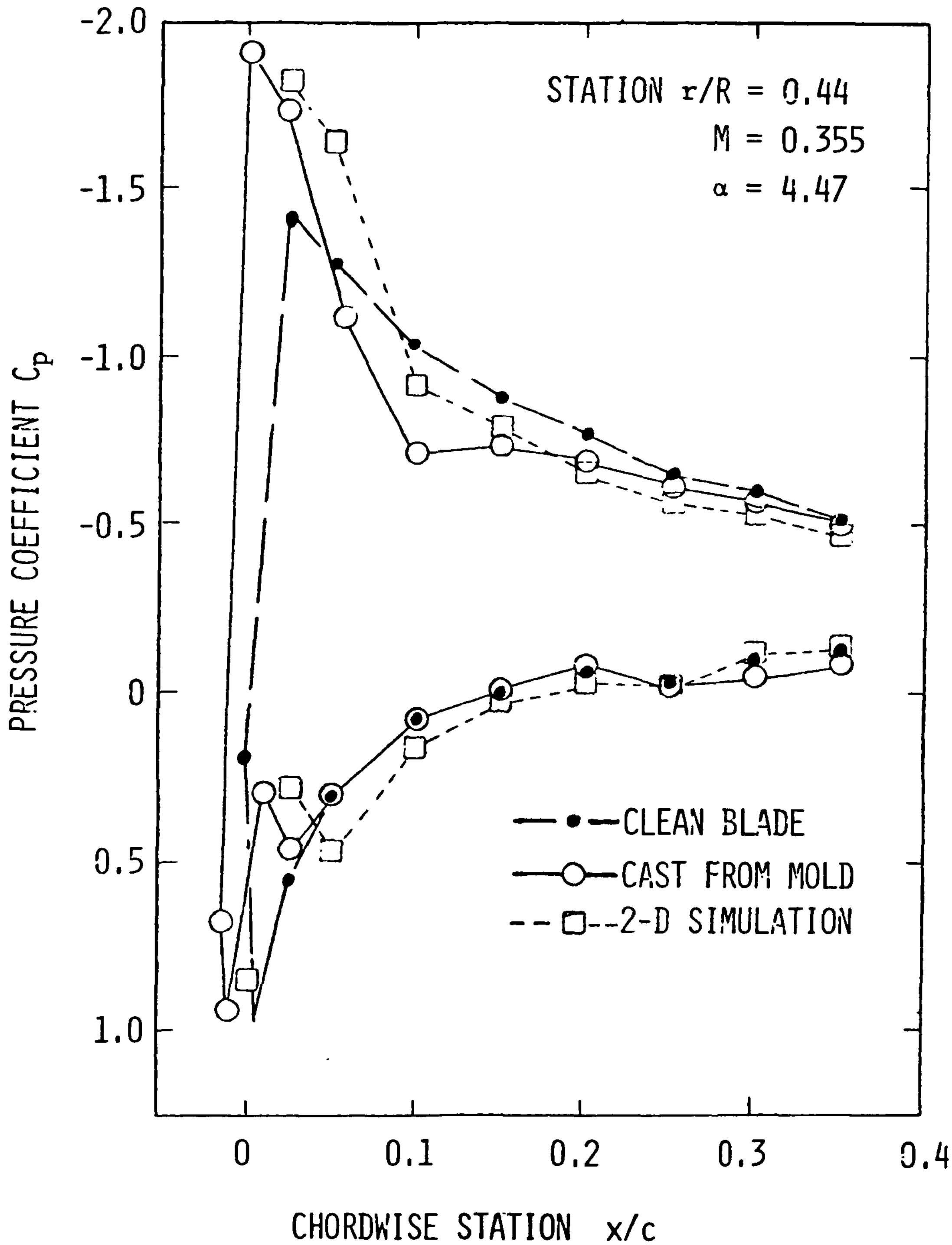


FIGURE 9. PRESSURE DISTRIBUTIONS OVER THE LEADING 0.35 OF THE BLADE SECTION WITH AND WITHOUT ICE SIMULATIONS

9B RADIAL STATION r/R = 0.44

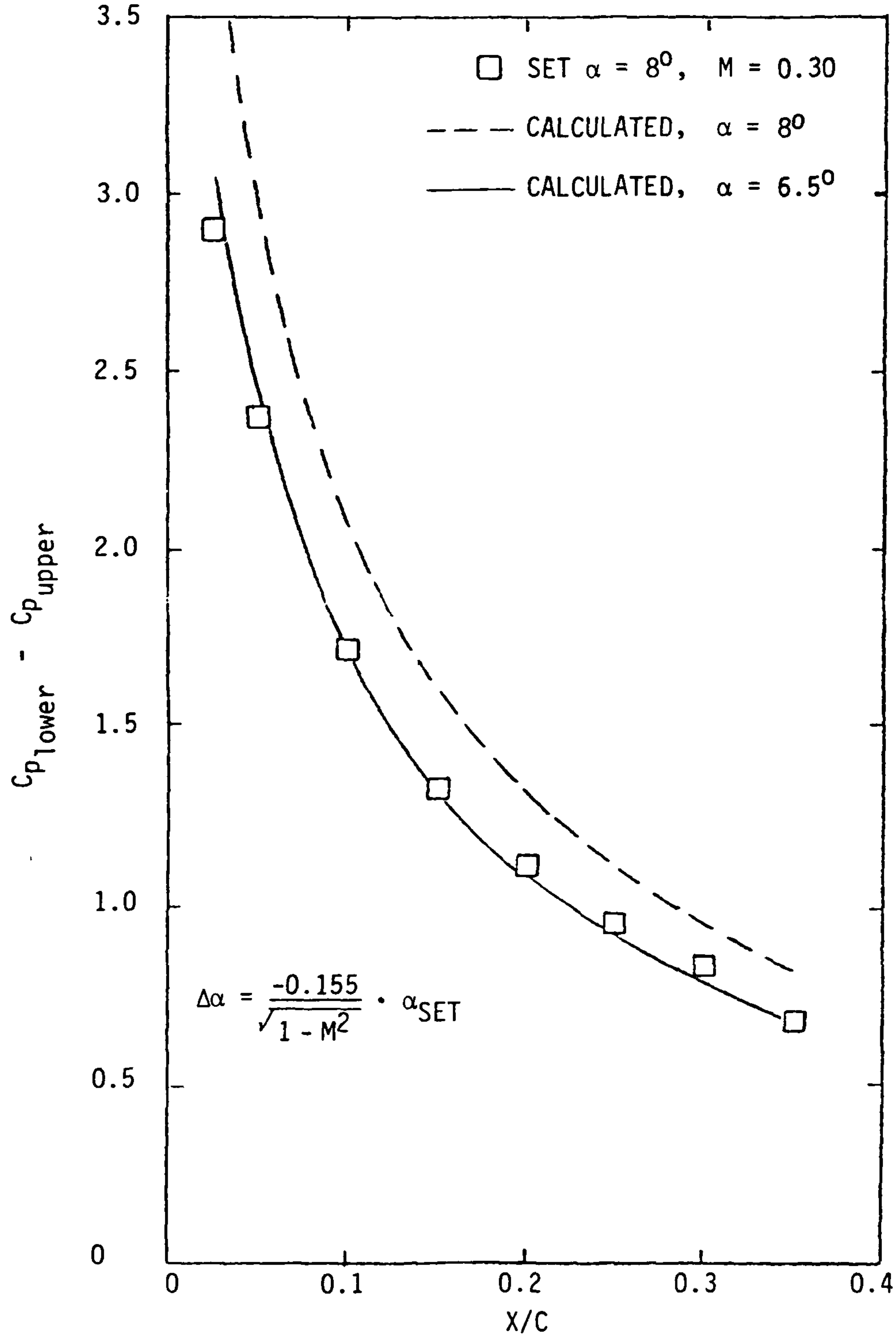


FIGURE A1. DETERMINATION OF THE CORRECTION TO ATTACK ANGLE BY COMPARISONS OF LIFT-INCREMENT DISTRIBUTIONS SUBCRITICAL FLOW

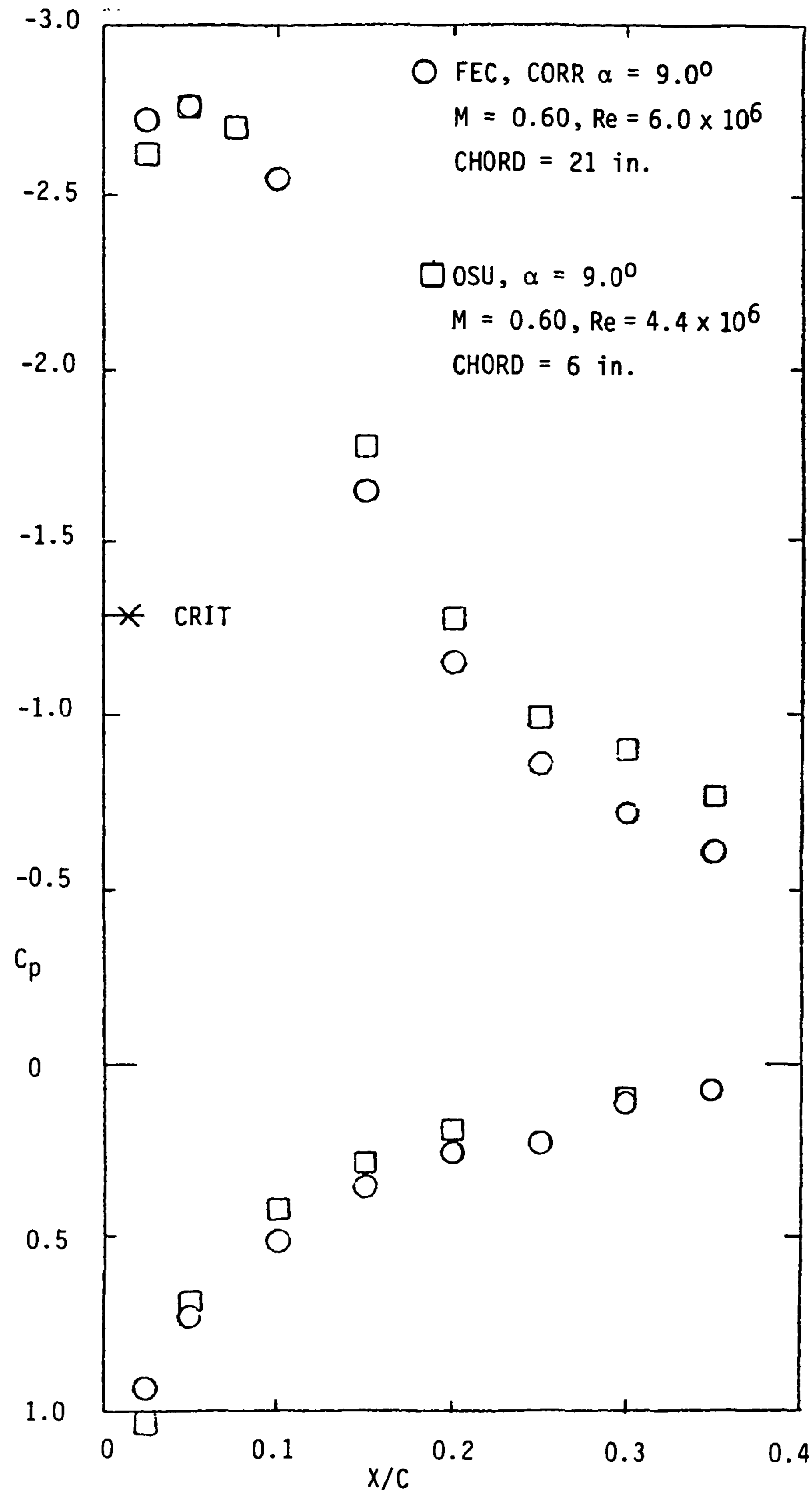


FIGURE A2. COMPARISON OF CP DISTRIBUTIONS IN SUPERCRITICAL FLOW FEC ATTACK ANGLE CORRECTED FROM FIGURE A1. (SET ANGLE = 11.5 DEGREES)

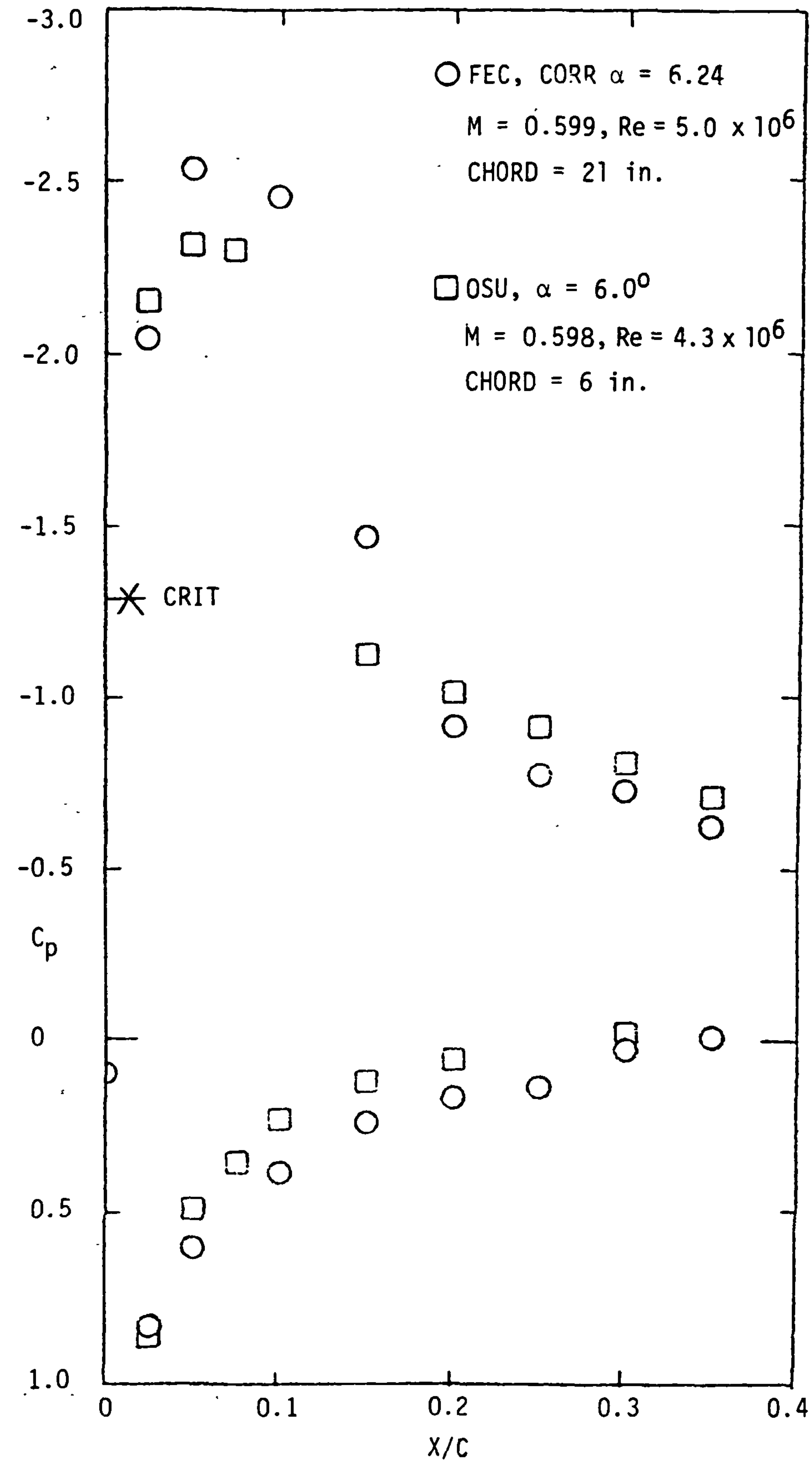


FIGURE A3. COMPARISON OF CP DISTRIBUTIONS IN SUPERCRITICAL FLOW FEC ATTACK ANGLE CORRECTED FROM FIGURE A1. (SET ANGLE = 8.0 DEGREES)

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15 Supplementary Notes

Final report. Project Manager, Robert J. Shaw, Propulsion Systems Division, NASA Lewis Research Center, Cleveland, Ohio 44135. John D. Lee, The Ohio State University; Jack H. Berger and Timothy J. McDonald, FluiDyne Engineering Corporation, Columbus, Ohio.

16 Abstract

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