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## Analysis of Mach Number 0.8 Turboprop Slipstream Wing/Nacelle Interactions

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Contract NAS2-10881  
 August 6, 1981



National Aeronautics and  
 Space Administration

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## I. SUMMARY

An experimental test program of a powered propeller and nacelle mounted on a supercritical wing was conducted by the NASA Ames Research Center in the 14-Foot Tunnel. Analysis of this data by the Douglas Aircraft Company, under contract to NASA with Al Lavin as the program manager, is contained in this report. The design condition for this study was  $M_0 = 0.8$ .

Analysis of the data indicated that the installation of the nacelle significantly affected the wing flow and that the flow on the upper surface of the wing is separated near the leading edge under powered conditions. Comparisons of various theories with the data indicated that the Neumann surface panel solution and the Jameson transonic solution gave results adequate for design purposes. A modified wing design was developed (Mod 3) which reduces the wing upper surface pressure coefficients and section lift coefficients at powered conditions to levels below those of the original wing without nacelle or power. A contoured over-the-wing nacelle is described that can be installed on the original wing without any appreciable interference to the wing upper surface pressures.

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## II. INTRODUCTION

The recent increases in fuel prices for aircraft has resulted in the consideration of alternate propulsion system concepts that would reduce fuel consumption. One of the primary candidates is a propeller-turboshaft (turboprop) powerplant. Several system studies have been conducted that indicate fuel savings from 15 to 30% in fuel burned for a given mission when compared to turbofan engines (References 1 through 6). Flight speeds of  $M_0 = 0.8$  are considered necessary for compatibility with existing airline operation and advanced propeller designs called Prop-Fans have been developed that give efficient performance at these speeds. A wing of the supercritical type is recommended to maximize performance.

One of the aerodynamic concerns about the turboprop installation is the interference drag that will result from the placement of the gas generator/nacelle and propeller on a supercritical type wing. Several years ago, a test was run by Douglas Aircraft under contract to NASA Ames (Reference 7) to experimentally evaluate these interferences. A flow-through ejector powered nacelle located ahead of the wing was used to simulate the onset flow of the propeller. This experimental approach permitted independently varying the various propeller parameters to obtain a basic understanding of the power-wing interactions. Many useful results and observations resulted from this early exploratory program but a more accurate representation of the flow is obtained by using an actual rotating propeller.

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The test results discussed in this report used the same wing design as the previous simulator test. A semispan test installation was used mounted to the floor of the Ames 14-foot tunnel. Test conditions covered the Mach range from 0.6 to 0.85. The Reynolds numbers based on the mean aerodynamic chord varied from  $7.8 \times 10^6$  at  $M_0 = 0.6$  to  $9.1 \times 10^6$  at  $M_0 = 0.85$ . An air-driven motor powered an 8 bladed propfan propeller designated SR-2C (Reference 8). The installation, development of the motor, fabrication and testing were all done by NASA Ames. This report covers the analysis of the data, comparison with theory and configuration modifications performed by the Douglas Aircraft Company (a division of the McDonnell Douglas Corporation) located in Long Beach, California.

### III. NOMENCLATURE

AR	Wing aspect ratio
b	Wing span
c	Local chord
$C_D$	Configuration drag coefficient
$c_l$	Section lift coefficient
$C_{LWB}$	Wing-body lift coefficient based on wing trapezoidal area
$C_{LW}$	Wing lift coefficient based on exposed wing trapezoidal area
$C_p$	Pressure coefficient
$\Delta C_{D_i}$	Incremental induced drag coefficient
$C_{mac}$	Mean aerodynamic chord
$M_o$	Free stream Mach number
$M_L$	Local Mach number
$P_T/P_{T0}$	Propeller total pressure to freestream total pressure ratio
$Re_c$	Reynolds number based on chord
RPM	Revolutions per minute
r/R	Propeller blade local radius divided by maximum radius
$S_{TRAP\_EXP}$	Exposed trapezoidal reference area
t/c	Wing thickness to chord ratio
x,y,z	Coordinate system x streamwise, y spanwise, z vertical
x/c	Fraction of local chord
$\alpha$	Configuration angle of attack measured relative to fuselage reference plane
$\alpha_s$	Swirl angle, degrees
$\beta$	propeller blade angle setting, degrees
$\eta$	Percent semispan of wing



$\lambda$

Taper ratio

$\Lambda c/4$

Wing quarter chord sweep

#### IV. DESCRIPTION OF THE MODEL

A photograph of the model is shown in Figure 1. The wing coordinates were obtained from Douglas Aircraft. The wing had been tested previously as part of the Douglas Aircraft supercritical wing development program and had demonstrated good drag rise characteristics near  $M_o = 0.8$ . The design also had been used previously as part of an earlier NASA sponsored test where the propeller onset flow had been simulated by using an ejector powered flow through nacelle mounted ahead of the wing (Reference 7 and Figure 2). The planform of the NASA model with the nacelle installed is shown in Figure 3 including the location of pressure rows to be discussed later. Coordinates for the wing are given in Table 1 and Figure 4 shows a side view of the nacelle.

The propeller was an SR-2C design (Reference 8) and was powered by an air driven turbine. Air to power the turbine was supplied through the floor and wing and exhausted under the wing through a nozzle at about 50% chord.

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## V. ANALYSIS OF THE DATA

### Force Data

Analysis by NASA and Douglas determined that the force data for this test was not reliable. Subsequent analysis by NASA did result in some usable force data, but for the purposes of the work described in this report the force data will not be referred to.

### Pressure Data

The pressure data on the wing were integrated to obtain the wing lift coefficient ( $C_{L_W}$ ), and these results are shown on Figure 5. The analysis will focus on an angle of attack of 2 degrees or a  $C_{L_W}$  near 0.5.

The pressure distributions on the wing are shown in Figures 6 through 12 for a fixed propeller blade angle of  $57^\circ$ . Figures 6 through 10 show the flow development for a fixed angle of attack of  $2^\circ$  and Figures 11 and 12 show the variation with angle of attack at  $M_0 = 0.8$ . The data indicates that there is a significant effect of the nacelle on the pressures inboard of the nacelle at 36.5 and 41.5 percent semispan, but that there is a negligible effect outboard of the nacelle. The pressure distributions indicate that small separations may occur inboard due to the nacelle installation, and at  $M_0 = 0.8$ , a normal shock is indicated. Available oil flow photographs shown in Figures 13 and 14 indicate that the flow has been significantly disturbed by the nacelle but large regions of flow separation are not apparent. The presence of the normal shock inboard of the nacelle can be seen in Figure 14.

When full propeller power is added, the flow velocities are increased in the propeller wash region (included on the figures) and there is also a change in the local angle of attack. The isolated propeller flow one blade chord downstream of the propeller is shown in Figure 15. (These data were not part of the current test but were measured separately as part of NASA's propeller development program.) The propeller rotation for this test created increased wing section angle of attack inboard of the nacelle (upwash) and decreased angle of attack outboard (downwash). The pressure distributions on the wing due to power show increased upper surface pressure peaks and as the Mach number is increased, the presence of a flow separation becomes more apparent. Available oil flow photographs with power-on, shown in Figures 16 and 17, illustrate clearly that there is a significant flow separation and extreme inboard flow from the outboard region of the nacelle.

The section lift data obtained by integrating the above pressure distributions (and removing obviously bad points) are shown in Figures 18 and 19. The installation of the nacelle reduces the  $c_l$  at the inboard stations with little effect outboard. The windmilling propeller reduces the  $c_l$  further probably due to a loss in dynamic pressure. The application of power significantly increases the  $c_l$  values inboard due to propeller upwash and increases the velocity to levels which caused the flow separations. Outboard the  $c_l$  is reduced because of the propeller downwash.

At  $M_0 = 0.7$ , the  $c_l$  data (solid symbols Figure 20) was used to calculate the induced drag increment due to the nacelle installation by a Trefftz plane method. The increment was found to be 12 counts ( $\Delta C_{D_i} = 0.0012$ ).

To summarize these results for later reference during the wing redesign discussion (Section VII), the region outboard of the nacelle is not seriously affected by the installation of the nacelle or power and wing modifications are not required in this region. However, inboard the nacelle significantly increases the upper surface pressure peaks but not to levels that cause large flow separations. However, a significant 12 count induced drag penalty was calculated. At  $M_0 = 0.8$ , a small normal shock is present near the nacelle on the wing, further contributing to the drag. The increased flow velocity and upwash due to power are sufficient to increase upper surface pressure levels and  $c_l$ 's to conditions where attached flow can not be maintained by the original wing design.

#### Comparison With Simulator Data (Reference 7).

It was found that the flow from the ejector powered simulator most nearly matched the  $59^\circ$  blade angle data (Figure 21), therefore the comparisons are made for this blade angle. Since the simulator test did not have a nacelle mounted on the wing, only incremental effects due to power from each test are compared.

Figures 22 through 27 compare the pressure distributions and span loads between the two tests at  $M_0 = 0.7$  and  $0.8$ . Qualitatively the pressure distributions agree fairly well, with the peak pressure levels and the general shape of the plots being very similar. It is interesting to note that the pressure increase near 60% chord at 50% semispan on the upper surface due to power is indicated in both sets of data (Figure 25).

The incremental  $c_l$  values are not as large for the simulator test as for the propeller test as shown on Figures 26 and 27. However, the  $c_l$  levels with power are comparable.

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## VI. COMPARISON WITH THEORY

### Lifting Line Program

The lifting line program is a method which uses a vortex filament at the quarter chord point at several spanwise locations on the wing and a control point at the three-quarter chord location. The wing zero-lift-line and the propeller onset flow are input. The nacelle geometry cannot be input. The strength of the vortex is then determined to satisfy the condition of tangential flow at the control point. Forces are calculated by repeated application of the Kutta-Joukowski law. This theory is compared to the simulator data in Figures 28 and 29. Agreement with the clean wing is good but the increments due to power are overpredicted. Both positive (up-inboard) and negative (up-outboard) swirl cases are shown.

Comparisons with the propeller data are shown in Figures 30 and 31. Increments due to the nacelle taken from the previous Figures (18 and 19) are indicated. At  $M_0 = 0.7$ , the clean wing data is below predictions inboard in contradiction to the data measured for the simulator case, suggesting a difference in wing geometry. This may have been caused by a difference in wing twist under load caused by the difference in wing fabrication methods or scale. The powered data are well predicted except in the region immediately outboard of the nacelle. At  $M_0 = 0.8$  the same conclusions apply except the clean wing data is underpredicted outboard, again suggesting a difference in wing twist.

In summary, the powered data for the simulator is underpredicted by the theory perhaps because of the presence of the ejector nacelle or nacelle



boundary layer flowing over the wing. Powered predictions for the propeller are generally good except just outboard of the nacelle. Nacelle interferences, not included in the theory, may cause this discrepancy.

### Neumann Theory

The Neumann program is an incompressible surface panel solution using unknown source and dipole singularities to satisfy the zero normal flow condition at control points located in a number of panels describing the body surface (Reference 9). In regions washed by the propeller, the propeller onset flow is input at each affected control point to be included with the free stream flow when the singularity strengths are found. The program can calculate flows about completely arbitrary configurations.

The program contains the option of using the Goethert correction for compressibility. However, when this option is applied, the zero normal flow boundary condition is not exactly satisfied. Because of the complex geometries dealt with and the small included angles between some body surface panels, the compressibility option was not employed in order to avoid potential numerical errors in some cases.

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A description of the paneling used for the simulator case is shown in Figure 32. The results are shown in Figures 33 and 34. (The Neumann program is compared to the data at the same configuration  $C_{L_w}$ . Since the Neumann program did not include viscous effects, the same  $C_{L_w}$  is achieved at an angle of attack  $1^\circ$  less than the data, Figure 5.) The pressure distributions generally agree well but the upper surface peaks near the leading edge are underpredicted. Agreement is good if incremental effects are compared. The section lift agreement is good with the exception of the power and swirl case at 35% semispan.

The paneling for the propeller case is shown in Figure 35. The circular body extending aft of the wing trailing edge is the representation of the exhaust flow. The pressure distributions inboard and outboard of the nacelle for the clean wing, wing plus nacelle and wing plus nacelle plus power are shown in Figures 36 through 41. (Again, note the  $1^\circ$  angle difference. The powered data are for  $\beta = 57^\circ$ , run 69). The agreement of the theory with the data is good. The experimental separation inboard of the nacelle with power can be easily identified in Figure 38.

The section lift distribution data (Figure 42) does not agree as well. Neither the level nor the increments are reasonably predicted inboard of the nacelle but outboard the agreement is better. The disagreement inboard may be caused by local separations and transonic flows not correctly included in the theory.

The Neumann pressure distributions are compared to the data at the same angle of attack as the data ( $2^\circ$ ) in Figures 43 through 48.

### Jameson Theory

The Jameson Theory is a fully transonic solution restricted to wings only. The wing is mapped into a computational plane and the full compressible potential flow equations are solved (Reference 10).

The Jameson theory agreement with the data for the clean wing is excellent as shown on Figures 49 and 50. To approximately account for propeller onset flow effects, the wing was twisted in accordance with the propeller swirl and an incremental free stream Mach number was applied to account for slipstream velocity increases. The nacelle effect is not

accounted for. These results are shown in Figures 51 through 56. As noted, if the incremental Mach effect is not included, the agreement between the theory and data is adequate to be used for design purposes.

The Jameson theory, applied in the manner discussed above, was the primary method used to design the modified wings described in the next section. This computer code was used because it is a transonic method and would give an adequate representation of the transonic flow development. However, nacelle effects are not properly accounted for and the Neumann computer code was used to include these effects. Uncertainty exists by using either program because of a limitation in the geometry capability (lack of a nacelle in Jameson) or transonic flow computational capability (Neumann). These factors will be discussed further during the discussion of the wing design in the next section.

## VII. DESIGN PHILOSOPHY

### Design Criteria

To reduce upper surface separations (Figures 16 and 17) and subsequent degradations in wing performance, it was necessary to reduce the magnitude of the high negative pressures on the wing upper surface inboard of the nacelle (Figures 7 and 9) and reduce the wing  $c_{\ell}$  values (Figures 18 and 19) to acceptable design limits. Since the clean original wing performed well and did not have any flow separations, the clean wing data was used to establish the design criteria. These criteria were minimum upper surface pressure levels, chordwise pressure gradients and spanwise upper surface isobar patterns. The  $c_{\ell}$  levels at any span station and the distribution of  $c_{\ell}$  as it affects induced drag were also used. The objective of the redesign work was to achieve levels on the new wing which were equal to or less than the above criteria.

### Wing Redesign

The airfoil section shape inboard of the nacelle was modified to conform to the above described criteria within the constraints of the existing hardware. No problems were identified outboard of the nacelle so no modifications were made there. The hardware constraints were that the airfoil remain unchanged except for the forward or aft 20 to 25% of the chord.

Modified airfoils that fit within the existing planform could not be found which conformed to the design criteria. Therefore, a leading or trailing edge extension was required. A trailing edge extension was ruled out because of difficulties with the aft pressure gradients and unsweeping

of the isobars. Therefore, a 15% leading edge extension was selected as shown on Figure 57. The closer proximity of the propeller to the leading edge is a concern and may shorten useable propeller test time.

The airfoil section for the modified planform was developed to reduce upper surface peaks and gradients at transonic conditions ( $M_0 = 0.8$ ,  $C_{L_w} = 0.5$ ) and fair smoothly into the existing airfoil shape at about 25% chord. The airfoil shapes were developed using a 2-D transonic analysis method prior to 3-D analysis to insure a pressure distribution over the entire airfoil that conformed to accepted design practice. The 3-D Jameson computer code was used to evaluate the various designs in three dimensions.

Two of the best designs developed are shown in Figures 58 through 62, and are identified as Mod 2 and Mod 3. The  $\eta = 0.12$  section shape is the same for either Mod. The Jameson results with and without power, compared to the original wing, are shown in Figures 63 and 64. Mod 3 has pressure peaks near the leading edge about half of the original wing and the transonic flow over the entire upper surface is free of shock waves even with power. The  $C_p$  levels with power are less than the clean original wing, the gradients are less and constant  $C_p$  levels occur at similar  $x/c$  values indicating swept isobar patterns. Mod 2 has higher nose peaks than Mod 3 but, as will be discussed, Mod 2 may be less sensitive to nacelle interferences. The spanwise distribution of lift is shown on Figure 65 and the  $c_l$  values are less everywhere than for the clean original wing. The desired spanwise distribution of lift to reduce induced drag could not be achieved.

Incompressible Jameson solutions at  $M_0 = 0.1$  were calculated for reference to subsequent discussions of the incompressible Neumann solutions to follow which calculated results for the wing/fuselage and the nacelle.

The Jameson results are shown in Figures 66 and 67. Note that the Mod 3 design has higher nose peaks than Mod 2. However, the transonic development of Mod 3 was much better than Mod 2 indicating that at the design condition of  $M_0 = 0.8$ , Mod 3 will be the better design.

The incompressible Neumann solutions for the 3 wings (original wing, Mod 2, and Mod 3) with and without the nacelle and power at the 37% semispan station are shown in Figures 68, 69 and 70. In all cases, Mod 3 has lower pressure coefficients and gradient levels near the leading edge on the upper surface than the clean original wing, and it is also better than Mod 2.

The same data presentation is shown at the 41% semispan station in Figures 71, 72 and 73. At this station, the Mod 3 design significantly reduces the negative  $C_p$  peaks compared to the original wing, especially with power as shown in Figure 73. However, the negative  $C_p$  peaks are higher than the original wing without nacelle and power and are higher than Mod 2 with nacelle and power. The uncertainty here is whether the results shown at the 41% semispan would also occur at transonic conditions. The Mod 3 results were worse in the Jameson code at freestream conditions similar to the Neumann solution ( $M_0 = 0$ ) but the pressure distribution development into the transonic region ( $M_0 = 0.8$ ) was better. Since a transonic code including the nacelle body was not available for this study, the transonic development in the presence of the nacelle could not be evaluated. Since the Mod 3 design is better at 37% semispan using the Neumann and is better everywhere at transonic conditions as evaluated in the Jameson, it has been selected as the preferred design and is recommended for test. If the adverse effects nearer the nacelle (41%) occur during test, these effects can be treated locally with a small fillet.

To evaluate whether or not the pressure peak calculated for Mod 3 at 41% semispan will cause a flow separation, the results shown on Figure 72 are compared to the power-off flow visualization photos on Figures 13 and 14. The flow photos did not indicate a large region of flow separation although a limited normal shock was indicated at  $M_0 = 0.8$ . The pressure peak calculated by the Neumann at these conditions was -1.4 as shown on Figure 72. Therefore, a -1.4  $C_p$  should be marginally acceptable to avoid significant flow separation.

The peak  $C_p$  for Mod 3 with nacelle and power calculated by the Neumann is -1.4 as shown on Figure 73. This level indicates that flow conditions similar to the power-off case on the original wing would occur (like Figure 13 and 14) and the powered performance of Mod 3 should be acceptable.

Figure 74 shows results for all the configurations and conditions at 56% semispan. In all cases, the pressure distributions are more favorable than the original wing.

The spanwise distribution of lift for all cases is shown in Figure 75. The desired  $c_l$  values (less than the original wing) are achieved inboard of the nacelle for either Mod 2 or 3.

To amplify the conclusions drawn and to indicate sensitivity to angle of attack, the Neumann results are shown at  $1^\circ$  higher angle of attack in Figures 76 through 81. All the conclusions drawn previously apply to these data. Note that in Figure 79, Mod 3 has a higher negative  $C_p$  at the nose without nacelle and power than Mod 2, just like the Jameson results.

### Nacelle Contouring

The streamlines about the isolated wing computed using the Neumann program are shown in Figures 82 and 83.

Underwing Nacelle - The nacelle geometry for this case was constrained to adding fairings to the existing nacelle or contouring aft of the motor so that installation of the air drive motor would be assured.

The aft part of the nacelle and nozzle centerline were contoured to the lower surface streamline as indicated in Figure 84. This contouring did not produce adequate favorable effects to warrant further consideration as shown in Figure 85.

Limited contouring of the nacelle was developed using the upper surface streamlines. As indicated in Figure 83, no contouring is possible inboard of the nacelle because of the relationship of the wing leading edge and rotating hub and propeller. A local fillet can be used outboard to fair between the nacelle and the wing leading edge. In the profile view (Figure 82) a local "bump" on the nacelle upper surface in the region of the wing leading edge will make the nacelle conform more closely with the streamline. The two fairings are shown by photographs of a subscale mock-up in Figure 86.

These limited fairings probably won't have a significant impact on the performance. The geometric limitations precluded contouring to the extent necessary. These fairings can be tested in the tunnel depending on test results of the modified wing and basic nacelle.

Overwing Nacelle - An overwing nacelle was also considered and is shown in Figure 87. For this case, the approach was to aerodynamically contour the nacelle and then determine if the air drive motor can be installed. Since this work was done in parallel with the development of Mod 3, the original



wing was used. Also, because the objective of this work was to evaluate what can be achieved by nacelle contouring with the power effects being treated using wing modifications, the power effects were not included.

The contouring was accomplished by maintaining the longitudinal cross-sectional area distribution of the nacelle to give the best chance of the motor fitting inside. The centers of these areas were shaped to conform to the flow streamlines. The resulting nacelle shape is shown in Figures 88 and 89. The contouring achieved excellent results as shown in Figures 90 through 92. At the 43% semispan location, the upper surface pressure peak and gradient with the nacelle installed is significantly less than for the non-contoured nacelle case and is almost identical to the original wing without the nacelle. Significantly better span loading is also achieved as shown in Figure 93, which should provide significantly less induced drag. It is recommended that this contoured nacelle and the non-contoured nacelle be tested to confirm these results.

## VIII. DESIGN MODIFICATIONS

The wing coordinates for Mod 3 are given at 10 spanwise locations in Table 2. These coordinates conform to the existing wing at approximately 25% of the original wing chord. The geometry of the overwing contoured nacelle has been given to NASA Ames in the form of a computer tape containing a parametric cubic definition of the nacelle. The plan and profile view were shown in Figures 88 and 89.

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## IX. CONCLUSIONS AND RECOMMENDATIONS

The following conclusions are drawn from this study:

- 1) The force data could not be used during this study because of uncertainties in accuracy.
- 2) Both the nacelle and power had significant effects on the wing flow. With power, the upper surface of the wing was separated at a wing  $C_{L_w}$  near 0.5.
- 3) The increments in the pressure data due to power agreed fairly well between the earlier simulator test and the propeller test. The absence of a nacelle in the simulator test makes a direct comparison impossible.
- 4) The Lifting Line computer code agreed better with the propeller power-on data than with the previous simulator data. This program could be used for preliminary design purposes.
- 5) The Neumann Surface Panel computer code predicted the wing surface pressure distributions with sufficient accuracy for design purposes. The agreement with the propeller model  $C_p$  distributions was excellent. The  $c_l$  values were not predicted as well.
- 6) The Jameson Transonic Program gives results adequate for design purposes if the power effects are accounted for by wing twist.

- 7) A new wing planform and new airfoil sections were developed (Mod 3). At  $M_0 = 0.8$ , the upper surface peak  $C_p$  values with power for this configuration are approximately half of the original wing with power. The pressure distribution is shock free and should result in performance with power approaching that of the original clean wing without power. Some uncertainty is present regarding the effects of the nacelle on this wing at transonic speeds. It is recommended that test data on Mod 3 be obtained to evaluate its performance.
  
- 8) A contoured upper surface nacelle was developed for the original wing which can be installed on the wing without producing any appreciable interference to the wing pressures. It is recommended that this nacelle be built and tested to evaluate performance.

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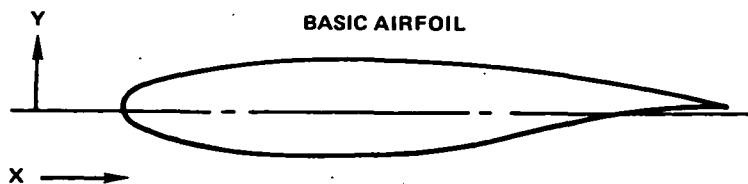
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TABLE 1  
ORIGINAL AIRFOIL COORDINATES

X' IS IN WRP	UPPER SURFACE							
	122 X = 10.5904		352 X = 30.8888		702 X = 61.7775		1002 X = 88.2537	
	Y	Z	Y	Z	Y	Z	Y	Z
100 Z	50.2101	-6.1735	50.2467	-2.0874	63.1242	-1.1690	73.5625	-.9283
97.5	49.1409	-5.9629	50.2149	-1.8628	62.6302	-1.0284	73.2718	-.3584
95.	48.0713	-5.7071	49.4831	-1.6344	62.1359	-.8976	72.9810	-.2963
92.5	47.0017	-5.4580	48.7513	-1.4684	61.6415	-.7803	72.6899	-.2378
90.	45.9325	-5.1671	48.0199	-1.3004	61.1472	-.6756	72.3992	-.1856
85.	43.7933	-4.6320	46.5562	-1.0053	60.1584	-.4940	71.8177	-.0981
80.	41.6545	-4.1119	45.0926	-.7517	59.1701	-.3407	71.2362	-.0272
75.	39.5157	-3.6122	43.6290	-.5358	58.1813	-.2128	70.6447	-.0290
70.	37.3769	-3.1441	42.1654	-.3529	57.1926	-.1077	70.0733	.0720
65.	35.2377	-2.7030	40.7022	-.2055	56.2042	-.0243	69.4918	.1029
58.5	32.4571	-2.1741	38.7993	-.0595	54.9189	.0518	68.7357	.1283
55.	30.8601	-1.9062	37.7749	.0022	54.2268	.0823	68.3285	.1316
50.	28.8809	-1.5502	36.3113	.0639	53.1061	.1088	67.7470	.1312
45.	26.6821	-1.2214	34.8477	.1011	52.2497	.1195	67.1655	.1288
40.	24.5493	-.9193	33.3845	.1150	51.2610	.1150	66.5841	.1059
35.	22.4041	-.6421	31.9209	.1037	50.2726	.0948	66.0022	.0809
30.	20.2653	-.3900	30.4572	.0687	49.2839	.0573	65.4207	.0489
25.	18.1264	-.1658	28.9936	.0004	48.2952	.0004	64.8393	.0
20.	15.9876	.0243	27.5300	-.0967	47.3064	-.0783	64.2578	-.0873
15.	13.8484	.1338	26.0668	-.2319	46.3346	-.1805	63.6763	-.1893
10.	11.7096	.0386	24.6032	-.4320	45.3293	-.3187	63.0948	-.2188
7.5	10.6400	-.1251	23.1396	-.5517	44.8350	-.4142	62.8041	-.2768
5.	9.5708	-.3889	21.1396	-.7164	44.3406	-.5252	62.5130	-.3462
2.5	8.5012	-.8483	22.4077	-.9560	43.8466	-.6874	62.2223	-.4485
1.25	7.9664	-1.2578	22.0420	-1.1490	43.6992	-.8182	62.0391	-.5168
.5	7.6456	-1.6716	21.8222	-1.3322	43.4511	-.9391	61.9900	-.5856
.25	7.5386	-1.8911	21.7491	-1.4953	43.4015	-1.0067	61.9626	-.6237
.05	7.4833	-2.1671	21.6906	-1.5798	43.3622	-1.1019	61.9374	-.6767
0.00	7.4816	-2.3917	21.6759	-1.6625	43.3522	-1.1791	61.9315	-.7189

LOWER SURFACE								
0.00	7.4816	-2.3917	21.6759	-1.6625	43.3522	-1.1791	61.9315	-.7189
.05	7.4533	-2.6185	21.6906	-1.8099	43.3622	-1.2056	61.9374	-.7605
.25	7.5386	-2.9217	21.7491	-1.9605	43.4015	-1.3327	61.7606	-.8141
.5	7.6456	-3.1456	21.8222	-2.0727	43.4511	-1.4269	61.9900	-.8542
1.25	7.9664	-3.6215	22.0420	-2.2822	43.6992	-1.5629	62.0771	-.9284
2.5	8.5012	-4.1920	22.4077	-2.5774	43.8466	-1.7143	62.2223	-1.0089
5.	9.5708	-4.9447	23.1396	-2.8162	44.3406	-1.9050	62.5130	-1.0889
7.5	10.6400	-5.4873	23.8714	-3.0140	44.8350	-2.0285	62.8041	-1.1707
10.	11.7096	-5.8464	24.6032	-3.1621	45.3293	-2.1157	63.0948	-1.2151
15.	13.8484	-6.8087	26.0668	-3.3664	46.3346	-2.2414	63.6763	-1.2667
20.	15.9876	-6.5932	27.5300	-3.5013	47.3064	-2.3163	64.2578	-1.2967
25.	18.1264	-6.7653	28.9936	-3.5936	48.2952	-2.3649	64.8393	-1.3096
30.	20.2653	-6.8535	30.4572	-3.6560	49.2839	-2.3913	65.4207	-1.3100
35.	22.4041	-6.8843	31.9209	-3.6859	50.2726	-2.3979	66.0022	-1.2997
40.	24.5493	-6.8744	33.3845	-3.6873	51.2610	-2.3847	66.5841	-1.2784
45.	26.6821	-6.8166	34.8477	-3.6517	52.2497	-2.3476	67.1655	-1.2445
50.	28.8209	-6.7770	36.3113	-3.5782	53.1061	-2.2855	67.7470	-1.1949
55.	30.9601	-6.7006	37.7749	-3.4828	54.2268	-2.1866	68.3285	-1.1266
58.5	32.4571	-6.6407	38.7993	-3.3813	54.9189	-2.0980	67.7357	-1.0648
65.	35.2377	-6.5234	40.7022	-3.0445	56.2042	-1.8914	69.4918	-.9321
70.	37.3769	-6.4326	42.1654	-2.7864	57.1926	-1.7069	70.0733	-.8134
75.	39.5157	-6.3422	43.6290	-2.5145	58.1813	-1.5129	70.6447	-.6899
80.	41.6545	-6.2869	45.0926	-2.2550	59.1701	-1.3276	71.2362	-.5723
85.	43.7933	-6.1841	46.5562	-2.0352	60.1584	-1.1692	71.8177	-.4705
90.	45.9325	-6.1804	48.0199	-1.8111	61.1472	-1.0619	72.3992	-.3977
92.5	47.0017	-6.1872	48.7513	-1.8727	61.6415	-1.0446	72.6899	-.3820
95.	48.0713	-6.1320	49.4831	-1.9036	62.1359	-1.0600	72.9810	-.3911
97.5	49.1409	-6.1566	50.2149	-1.9440	62.6302	-1.1170	73.2718	-.4046
100.	50.2101	-6.2827	50.2467	-2.1462	63.1242	-1.2218	73.5625	-.4837

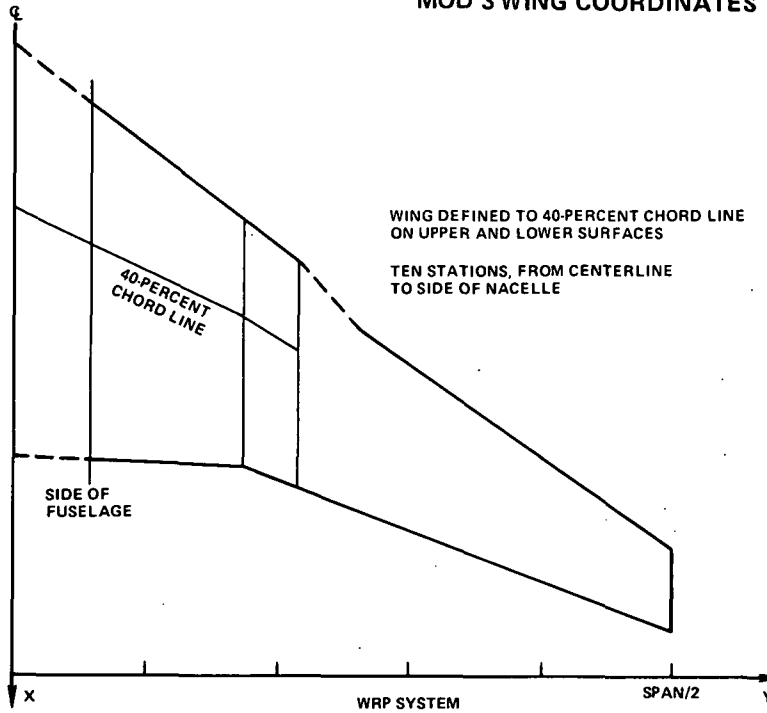


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TABLE 2  
MOD 3 WING COORDINATES



X	Y	Z
15.812060	0.0	-0.740680
14.054580	0.0	-0.460690
12.325220	0.0	-0.207010
10.630840	0.0	0.010970
8.978100	0.0	0.183100
7.373530	0.0	0.316320
5.823460	0.0	0.414810
4.333990	0.0	0.478790
2.911010	0.0	0.508770
1.560150	0.0	0.506260
0.286730	0.0	0.472200
-0.904210	0.0	0.407260
-2.007980	0.0	0.312010
-3.020230	0.0	0.187120
-3.936950	0.0	0.033570
-4.754530	0.0	-0.146090
-5.469750	0.0	-0.350210
-6.079780	0.0	-0.577320
-6.582210	0.0	-0.826880
-6.975060	0.0	-1.098800
-7.256770	0.0	-1.392240
-7.426250	0.0	-1.705150
-7.482820	0.0	-2.041560
-7.426250	0.0	-2.379360
-7.256770	0.0	-2.739880
-6.975060	0.0	-3.103160
-6.582210	0.0	-3.472570
-6.079780	0.0	-3.847270
-5.469750	0.0	-4.226230
-4.754530	0.0	-4.606940
-3.936950	0.0	-4.987230
-3.020230	0.0	-5.364220
-2.007980	0.0	-5.734170
-0.904210	0.0	-6.095380
0.286730	0.0	-6.445230
1.560150	0.0	-6.780510
2.911010	0.0	-7.098000
4.333990	0.0	-7.395130
5.823460	0.0	-7.668320
7.373530	0.0	-7.909590
8.978100	0.0	-8.120780
10.630840	0.0	-8.290130
12.325220	0.0	-8.412710
14.054580	0.0	-8.486420
15.812060	0.0	-8.542960

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TABLE 2 (CONT)  
MOD 3 WING COORDINATES

X	Y	Z
21.010529	10.590440	-0.474080
19.501648	10.590440	-0.303140
18.016922	10.590440	-0.152990
16.562256	10.590440	-0.028640
15.143310	10.590440	0.063920
13.765720	10.590440	0.128370
12.434910	10.590440	0.167240
11.156130	10.590440	0.180550
9.934440	10.590440	0.168860
8.774670	10.590440	0.133300
7.681380	10.590440	0.075040
6.658910	10.590440	-0.005340
5.711270	10.590440	-0.106710
4.842210	10.590440	-0.228500
4.055160	10.590440	-0.370010
3.353230	10.590440	-0.529380
2.739190	10.590440	-0.705650
2.215450	10.590440	-0.897760
1.784100	10.590440	-1.104400
1.446820	10.590440	-1.325060
1.204950	10.590440	-1.558400
1.059440	10.590440	-1.801500
1.010880	10.590440	-2.057810
1.059440	10.590440	-2.314670
1.204950	10.590440	-2.585460
1.446820	10.590440	-2.859330
1.784100	10.590440	-3.137140
2.215450	10.590440	-3.417410
2.739190	10.590440	-3.698900
3.353230	10.590440	-3.979450
4.055160	10.590440	-4.257800
4.842210	10.590440	-4.532260
5.711270	10.590440	-4.800430
6.658910	10.590440	-5.061160
7.681380	10.590440	-5.312740
8.774670	10.590440	-5.553420
9.934440	10.590440	-5.781800
11.156130	10.590440	-5.995530
12.434910	10.590440	-6.192460
13.765720	10.590440	-6.371090
15.143320	10.590440	-6.529240
16.562241	10.590440	-6.661230
18.016922	10.590440	-6.762860
19.501648	10.590440	-6.834500
21.010529	10.590440	-6.877040

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TABLE 2 (CONT)  
MOD 3 WING COORDINATES

V=13.9735			V=17.3585		
X	Y	Z	X	Y	Z
22.670883	13.973500	-0.380840	24.331223	17.356491	-0.287610
21.241489	13.973500	-0.244560	22.981155	17.356491	-0.185980
19.834888	13.973500	-0.127530	21.652879	17.356491	-0.102070
18.456696	13.973500	-0.033660	20.351105	17.356491	-0.038560
17.112411	13.973500	0.033660	19.081487	17.356491	0.002130
15.807320	13.973500	0.076130	17.848982	17.356491	0.021900
14.546550	13.973500	0.094580	16.658142	17.356491	0.021910
13.325060	13.973500	0.091360	15.513960	17.356491	0.002170
12.177660	13.973500	0.065990	14.428840	17.356491	-0.036870
11.078910	13.973500	0.019470	13.383130	17.356491	-0.094350
10.043160	13.973500	-0.046970	12.404910	17.356491	-0.168980
9.074490	13.973500	-0.132730	11.490040	17.356491	-0.260130
8.176720	13.973500	-0.236570	10.621130	17.356491	-0.365440
7.353390	13.973500	-0.357910	9.864540	17.356491	-0.487330
6.607760	13.973500	-0.496880	9.160330	17.356491	-0.622160
5.942770	13.973500	-0.649450	8.532270	17.356491	-0.769520
5.361040	13.973500	-0.817250	7.982950	17.356491	-0.928850
4.864860	13.973500	-0.998650	7.514230	17.356491	-1.099550
4.456210	13.973500	-1.192140	7.128200	17.356491	-1.279870
4.136670	13.973500	-1.397010	6.825500	17.356491	-1.465970
3.907530	13.973500	-1.611900	6.610980	17.356491	-1.665400
3.769680	13.973500	-1.833570	6.478980	17.356491	-1.865650
3.723680	13.973500	-2.064290	6.436440	17.356491	-2.070780
3.769680	13.973500	-2.295470	6.479990	17.356491	-2.276280
3.907530	13.973500	-2.538880	6.510080	17.356491	-2.490700
4.136670	13.973500	-2.783540	6.626500	17.356491	-2.707750
4.456210	13.973500	-3.028290	7.162880	17.356491	-2.927470
4.864860	13.973500	-3.273040	7.421130	17.356491	-3.147920
5.361040	13.973500	-3.533800	7.982950	17.356491	-3.367700
5.942770	13.973500	-3.782090	8.532270	17.356491	-3.584740
6.607760	13.973500	-4.028950	9.160330	17.356491	-3.798300
7.353390	13.973500	-4.269810	9.864540	17.356491	-4.007360
8.176720	13.973500	-4.506470	10.642130	17.356491	-4.210910
9.074490	13.973500	-4.734650	11.490040	17.356491	-4.409950
10.043160	13.973500	-4.954830	12.404910	17.356491	-4.595400
11.078910	13.973500	-5.164330	13.383130	17.356491	-4.775250
12.177660	13.973500	-5.363710	14.428840	17.356491	-4.946620
13.325060	13.973500	-5.550510	15.513960	17.356491	-5.105490
14.546550	13.973500	-5.723140	16.658142	17.356491	-5.253830
15.807320	13.973500	-5.880480	17.848982	17.356491	-5.398980
17.112411	13.973500	-6.028980	19.081487	17.356491	-5.512540
18.456696	13.973500	-6.138830	20.351105	17.356491	-5.618440
19.834888	13.973500	-6.234190	21.652879	17.356491	-5.705510
21.241489	13.973500	-6.301640	22.981155	17.356491	-5.768800
22.670883	13.973500	-6.342700	24.331223	17.356491	-5.808360

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TABLE 2 (CONT)  
MOD 3 WING COORDINATES

V=20.7396			V=24.1227		
X	Y	Z	X	Y	Z
25.991688	20.739578	-0.194370	27.651993	24.122681	-0.101130
24.780947	20.739578	-0.127410	26.460739	24.122681	-0.068830
23.479586	20.739578	-0.076500	25.283529	24.122681	-0.051140
22.245560	20.739578	-0.043520	24.140030	24.122681	-0.048480
21.050620	20.739578	-0.028770	23.019760	24.122681	-0.050660
19.890518	20.739578	-0.031330	21.932144	24.122681	-0.084560
18.769791	20.739578	-0.050750	20.881454	24.122681	-0.123410
17.682982	20.739578	-0.087870	19.874867	24.122681	-0.176220
16.640973	20.739578	-0.139740	18.967318	24.122681	-0.246210
15.687400	20.739578	-0.208180	17.991669	24.122681	-0.322010
14.766710	20.739578	-0.291000	17.128252	24.122681	-0.413010
13.905550	20.739578	-0.387530	16.321259	24.122681	-0.514930
13.107610	20.739578	-0.496310	15.573100	24.122681	-0.626170
12.375750	20.739578	-0.616740	14.886870	24.122681	-0.746150
11.712960	20.739578	-0.748230	14.265590	24.122681	-0.874310
11.121840	20.739578	-0.889590	13.711420	24.122681	-1.009660
10.604740	20.739578	-1.040450	13.226630	24.122681	-1.152050
10.163680	20.739578	-1.200450	12.813130	24.122681	-1.301340
9.800420	20.739578	-1.367610	12.472570	24.122681	-1.455340
9.516390	20.739578	-1.540920	12.206290	24.122681	-1.612880
9.312700	20.739578	-1.718890	12.015320	24.122681	-1.772390
9.190160	20.739578	-1.897730	11.900450	24.122681	-1.929800
9.149270	20.739578	-2.077260	11.862110	24.122681	-2.083750
9.190160	20.739578	-2.257680	11.900450	24.122681	-2.237890
9.312700	20.739578	-2.443320	12.015320	24.122681	-2.395940
9.516390	20.739578	-2.631960	12.206290	24.122681	-2.556170
9.800420	20.739578	-2.822600	12.472570	24.122681	-2.717750
10.163680	20.739578	-3.013170	12.813130	24.122681	-2.878420
10.604730	20.739578	-3.202100	13.226630	24.122681	-3.038490
11.121840	20.739578	-3.387370	13.711420	24.122681	-3.198010
11.712960	20.739578	-3.568550	14.265590	24.122681	-3.338790
12.375750	20.739578	-3.744910	14.886870	24.122681	-3.462460
13.107610	20.739578	-3.916540	15.573100	24.122681	-3.629570
13.905550	20.739578	-4.078480	16.321259	24.122681	-3.752730
14.766710	20.739578	-4.230810	17.128252	24.122681	-3.876170
15.687400	20.739578	-4.365170	17.991669	24.122681	-4.007080
16.640978	20.739578	-4.527520	18.967318	24.122681	-4.140430
17.682982	20.739578	-4.660460	19.874867	24.122681	-4.215430
18.769791	20.739578	-4.784500	20.881454	24.122681	-4.315180
19.890518	20.739578	-4.899290	21.932144	24.122681	-4.440880
21.050620	20.739578	-5.004180	23.019760	24.122681	-4.495820
22.245560	20.739578	-5.097840	24.140030	24.122681	-4.575640
23.479586	20.739578	-5.176830	25.283529	24.122681	-4.648140
24.780947	20.739578	-5.235940	26.460739	24.122681	-4.703080
25.991688	20.739578	-5.274000	27.651993	24.122681	-4.739650

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TABLE 2 (CONT)  
MOD 3 WING COORDINATES

V=27.5057			V=30.8888		
X	Y	Z	X	Y	Z
29.312332	27.505692	-0.007899	30.972717	30.888779	0.085340
28.290485	27.505692	-0.010250	29.940262	30.888779	0.048338
27.106400	27.505692	-0.025689	28.924316	30.888779	-0.000220
26.034454	27.505692	-0.053440	27.928909	30.888779	-0.058400
24.988846	27.505692	-0.099560	26.957977	30.888779	-0.121450
23.973724	27.505692	-0.137799	26.015325	30.888779	-0.191820
22.993057	27.505692	-0.190479	25.104706	30.888779	-0.268730
22.050751	27.505692	-0.265418	24.220691	30.888779	-0.354640
21.150513	27.505692	-0.345489	23.393738	30.888779	-0.448350
20.295883	27.505692	-0.435830	22.600143	30.888779	-0.549650
19.490265	27.505692	-0.535020	21.852051	30.888779	-0.657040
18.736801	27.505692	-0.642320	21.152405	30.888779	-0.769720
18.030513	27.505692	-0.755040	20.503967	30.888779	-0.885910
17.381117	27.505692	-0.875560	19.909202	30.888779	-1.004880
16.818146	27.505692	-1.000330	19.370758	30.888779	-1.126460
16.300919	27.505692	-1.129730	18.890457	30.888779	-1.249810
15.848440	27.505692	-1.263560	18.470291	30.888779	-1.375260
15.462500	27.505692	-1.402240	18.111068	30.888779	-1.503140
15.144650	27.505692	-1.543880	17.810767	30.888779	-1.630810
14.896110	27.505692	-1.684830	17.568968	30.888779	-1.756790
14.717870	27.505692	-1.825890	17.420456	30.888779	-1.879390
14.610650	27.505692	-1.961880	17.320892	30.888779	-1.993560
14.574870	27.505692	-2.090240	17.287674	30.888779	-2.096730
14.610650	27.505692	-2.218890	17.320892	30.888779	-2.199500
14.717870	27.505692	-2.348570	17.420456	30.888779	-2.301190
14.896110	27.505692	-2.480380	17.568968	30.888779	-2.404590
15.144650	27.505692	-2.612910	17.810767	30.888779	-2.508060
15.462500	27.505692	-2.746550	18.111068	30.888779	-2.609330
15.848440	27.505692	-2.879900	18.470291	30.888779	-2.705200
16.300919	27.505692	-2.992550	18.890457	30.888779	-2.795200
16.818146	27.505692	-3.100040	19.370758	30.888779	-2.879290
17.381117	27.505692	-3.220010	19.909202	30.888779	-2.957560
18.030513	27.505692	-3.325610	20.503967	30.888779	-3.030640
18.736801	27.505692	-3.425630	21.152405	30.888779	-3.098530
19.490265	27.505692	-3.519530	21.852051	30.888779	-3.160890
20.295883	27.505692	-3.608090	22.600143	30.888779	-3.218920
21.150513	27.505692	-3.691340	23.393738	30.888779	-3.272540
22.050751	27.505692	-3.770410	24.220691	30.888779	-3.322390
22.993057	27.505692	-3.845870	25.104706	30.888779	-3.370550
23.973724	27.505692	-3.918990	26.015325	30.888779	-3.427500
24.988846	27.505692	-3.987470	26.957977	30.888779	-3.479120
25.034454	27.505692	-4.052550	27.928909	30.888779	-3.522850
27.106400	27.505692	-4.119470	28.924316	30.888779	-3.559000
28.290485	27.505692	-4.178240	29.940262	30.888779	-3.637390
29.312332	27.505692	-4.205310	30.972717	30.888779	-3.670970

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TABLE 2 (CONT)  
MOD 3 WING COORDINATES

V=34.4630			V=38.0373		
X	Y	Z	X	Y	Z
33.245972	34.462992	0.086360	35.519333	38.037338	0.087380
32.258194	34.462992	0.051400	34.576218	38.037338	0.054470
31.286209	34.462992	0.005540	33.648193	38.037338	0.011290
30.333862	34.462992	-0.040490	32.739922	38.037338	-0.040680
29.404938	34.462992	-0.109050	31.852051	38.037338	-0.097840
28.503093	34.462992	-0.195630	30.999552	38.037338	-0.182240
27.631866	34.462992	-0.252210	30.159134	38.037338	-0.275680
26.794768	34.462992	-0.335020	29.350833	38.037338	-0.317230
25.994919	34.462992	-0.427360	28.596207	38.037338	-0.406380
25.235672	34.462992	-0.525990	27.871292	38.037338	-0.502320
24.519943	34.462992	-0.630460	27.187943	38.037338	-0.603880
23.850571	34.462992	-0.740110	26.548943	38.037338	-0.710590
23.230194	34.462992	-0.853460	25.956528	38.037338	-0.821920
22.661255	34.462992	-0.970360	25.413315	38.037338	-0.935750
22.146011	34.462992	-1.090500	24.921371	38.037338	-1.054550
21.686493	34.462992	-1.212340	24.482635	38.037338	-1.174870
21.284500	34.462992	-1.335360	24.098831	38.037338	-1.295460
20.941635	34.462992	-1.459190	23.771469	38.037338	-1.415240
20.659241	34.462992	-1.588840	23.501846	38.037338	-1.530870
20.438446	34.462992	-1.699460	23.291031	38.037338	-1.642130
20.280090	34.462992	-1.814820	23.139847	38.037338	-1.748660
20.184845	34.462992	-1.922480	23.048904	38.037338	-1.851010
20.153076	34.462992	-2.021910	23.018585	38.037338	-1.947090
20.184845	34.462992	-2.121340	23.048904	38.037338	-2.043170
20.280090	34.462992	-2.218250	23.139847	38.037338	-2.135310
20.438446	34.462992	-2.315170	23.291031	38.037338	-2.225760
20.659241	34.462992	-2.412840	23.501846	38.037338	-2.317620
20.941635	34.462992	-2.508740	23.771469	38.037338	-2.408560
21.284500	34.462992	-2.601200	24.098831	38.037338	-2.497290
21.686493	34.462992	-2.688630	24.482635	38.037338	-2.581960
22.146011	34.462992	-2.770520	24.921371	38.037338	-2.661750
22.661255	34.462992	-2.846540	25.413315	38.037338	-2.735510
23.230194	34.462992	-2.917570	25.956528	38.037338	-2.804490
23.850571	34.462992	-2.983670	26.548943	38.037338	-2.868810
24.519943	34.462992	-3.044670	27.187943	38.037338	-2.928460
25.235672	34.462992	-3.101850	27.871292	38.037338	-2.984240
25.994919	34.462992	-3.154900	28.596207	38.037338	-3.036550
26.794768	34.462992	-3.205790	29.350833	38.037338	-3.086180
27.631866	34.462992	-3.255060	30.159134	38.037338	-3.133570
28.503093	34.462992	-3.303280	30.999552	38.037338	-3.179060
29.404938	34.462992	-3.351330	31.852051	38.037338	-3.223530
30.333862	34.462992	-3.400280	32.739922	38.037338	-3.267650
31.286209	34.462992	-3.451680	33.648193	38.037338	-3.312560
32.258194	34.462992	-3.493250	34.576218	38.037338	-3.349110
33.245972	34.462992	-3.529240	35.519333	38.037338	-3.374980

81-GEN-24219

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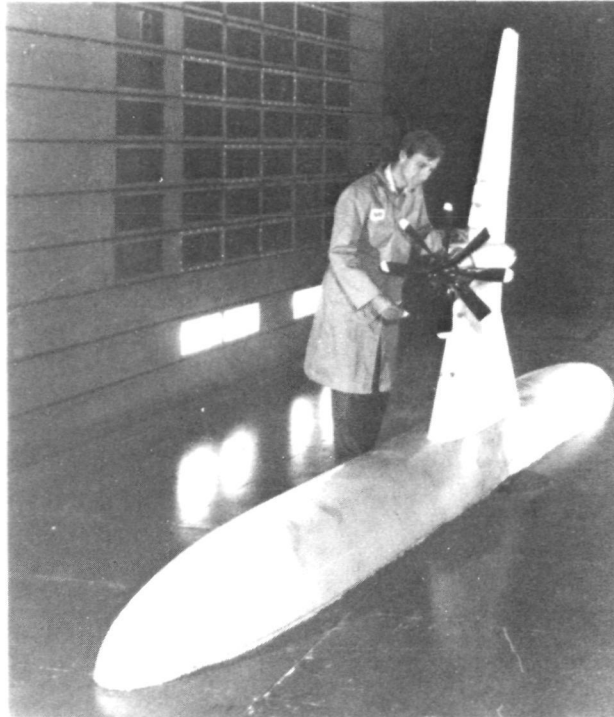
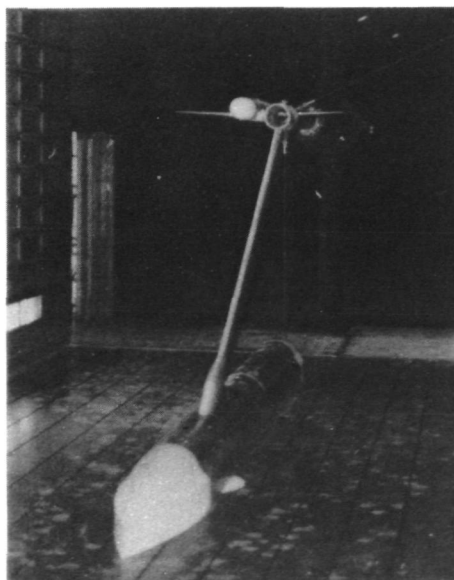
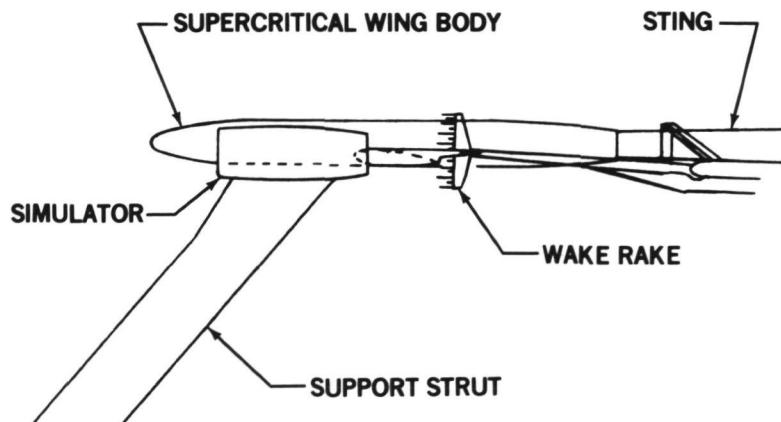


FIGURE 1. NASA MODEL INSTALLED IN AMES 14-FOOT TUNNEL

81-GEN-23245



FROM UPSTREAM OF THE MODEL



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81-GEN-23244

FIGURE 2. SIMULATOR TEST INSTALLATION

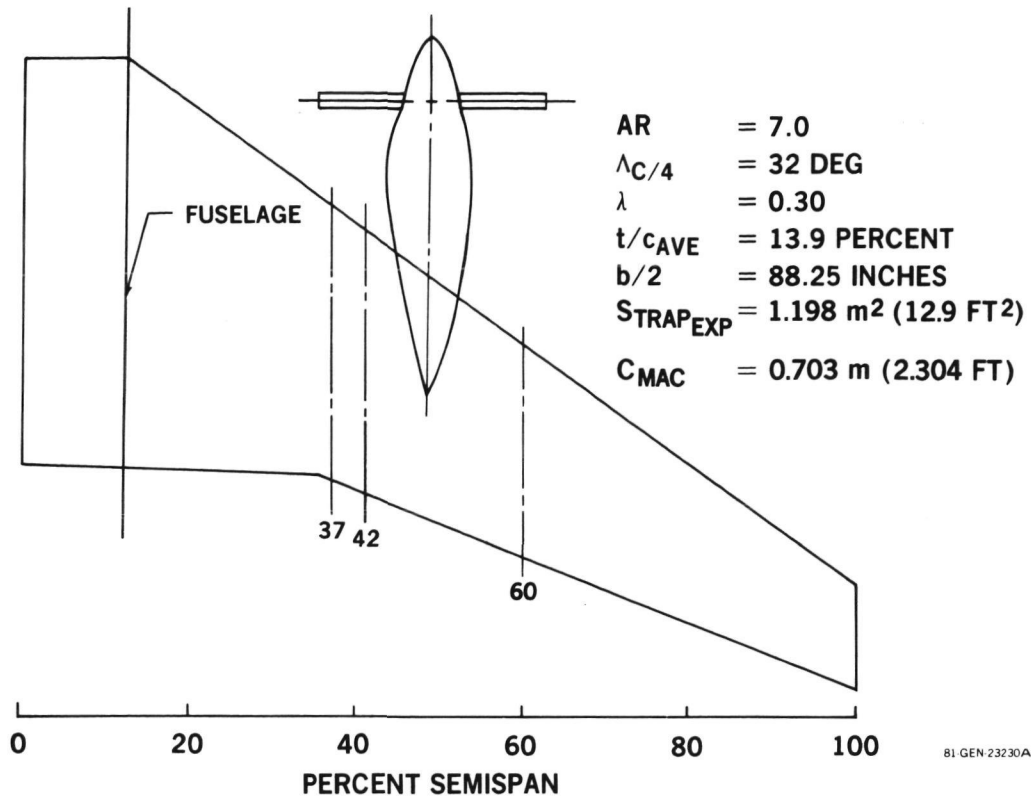


FIGURE 3. NASA MODEL PLANFORM

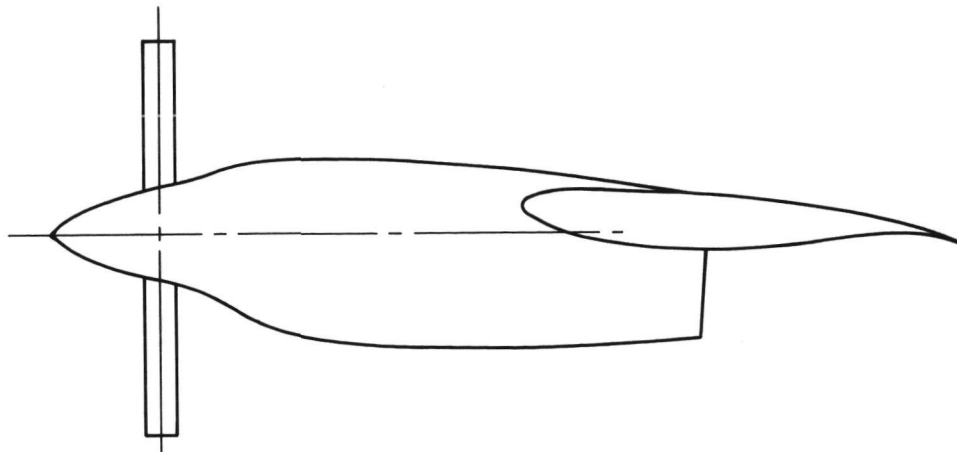
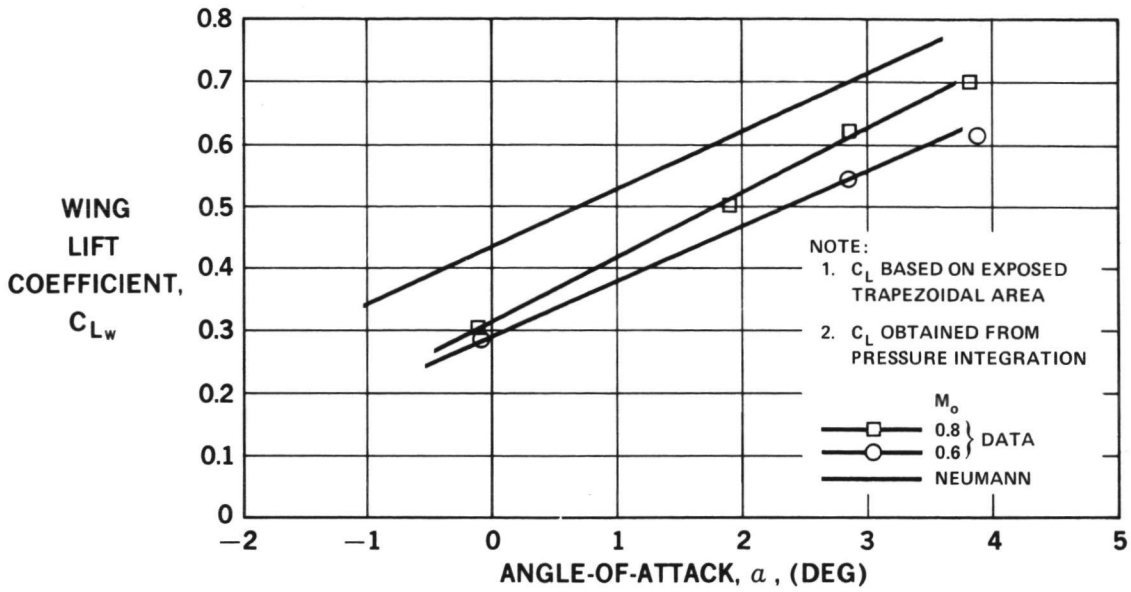


FIGURE 4. SIDE VIEW OF UNDERWING NACELLE

81 GEN 23204



81-GEN-23221

FIGURE 5. WING LIFT CURVES

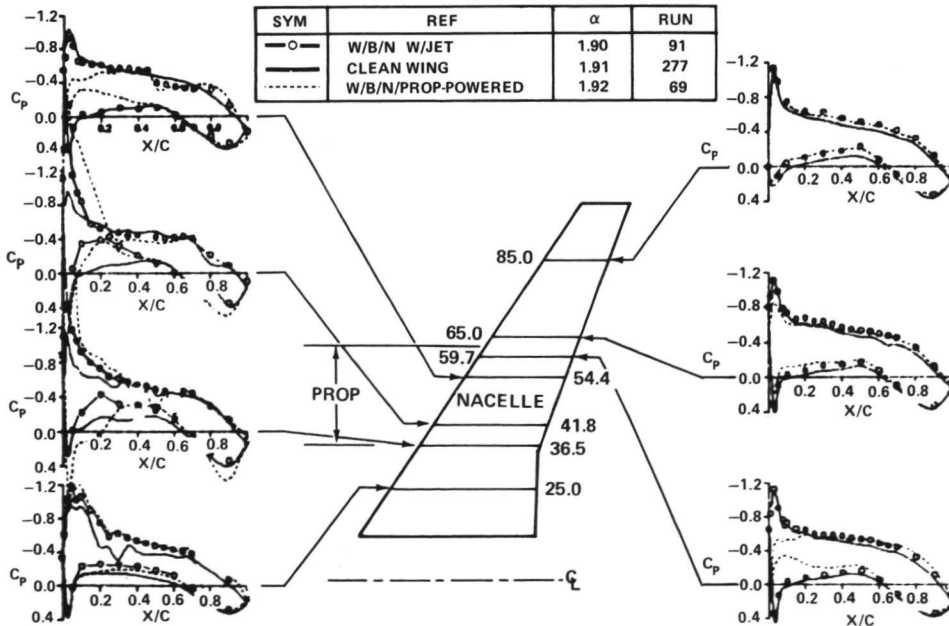


FIGURE 6. CHORDWISE PRESSURE DISTRIBUTION FOR ORIGINAL TURBOPROP CONFIGURATION AT  $M_o = 0.6$

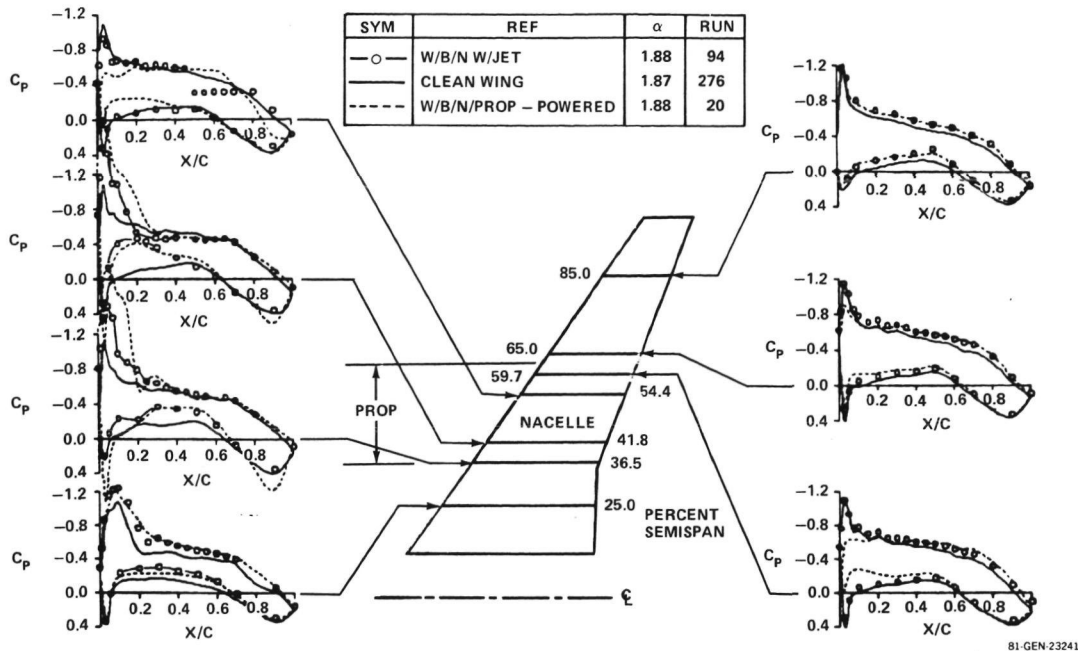


FIGURE 7. CHORDWISE PRESSURE DISTRIBUTIONS FOR ORIGINAL TURBOPROP CONFIGURATIONS AT  $M_0 = 0.7$

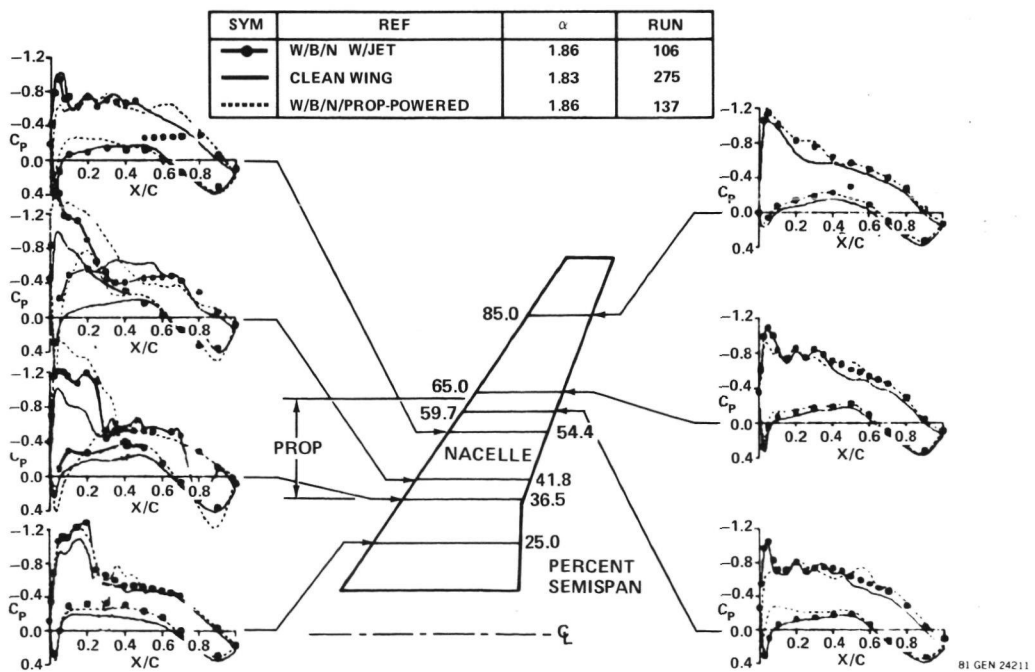


FIGURE 8. CHORDWISE PRESSURE DISTRIBUTION FOR ORIGINAL TURBOPROP CONFIGURATION AT  $M_0 = 0.78$



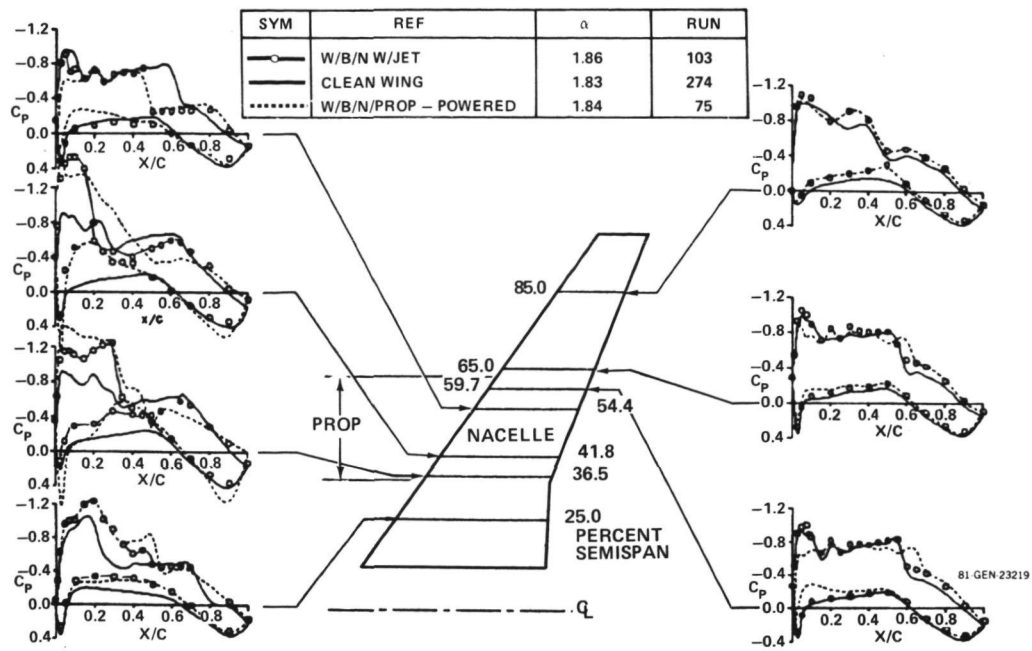


FIGURE 9. CHORDWISE PRESSURE DISTRIBUTIONS FOR ORIGINAL TURBOPROP CONFIGURATION AT  $M_0 = 0.8$

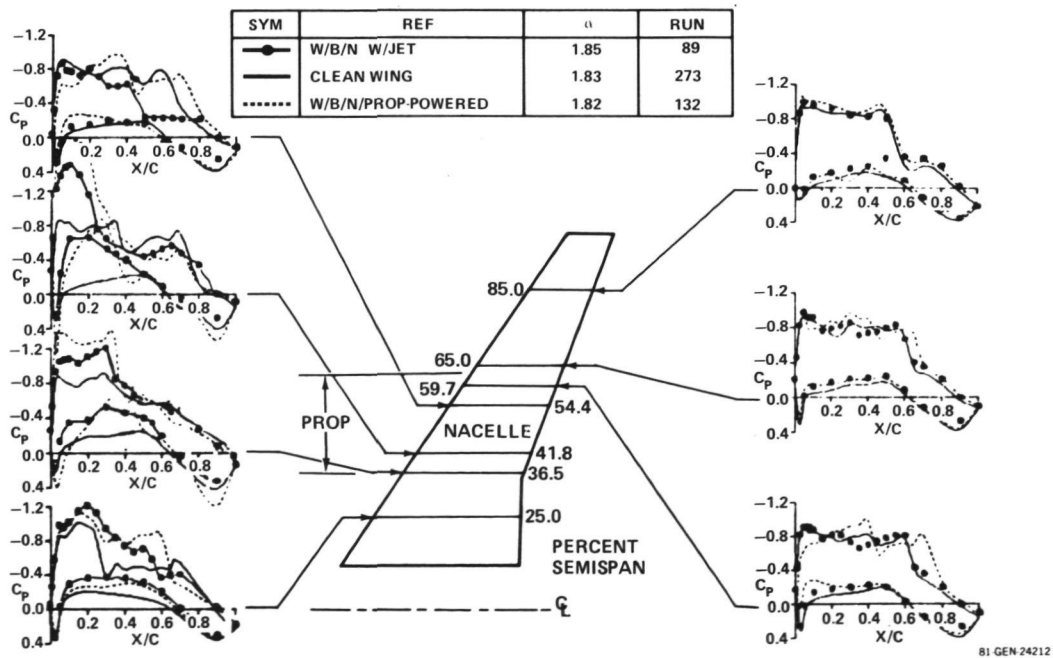


FIGURE 10. CHORDWISE PRESSURE DISTRIBUTION FOR ORIGINAL TURBOPROP CONFIGURATION AT  $M_0 = 0.82$

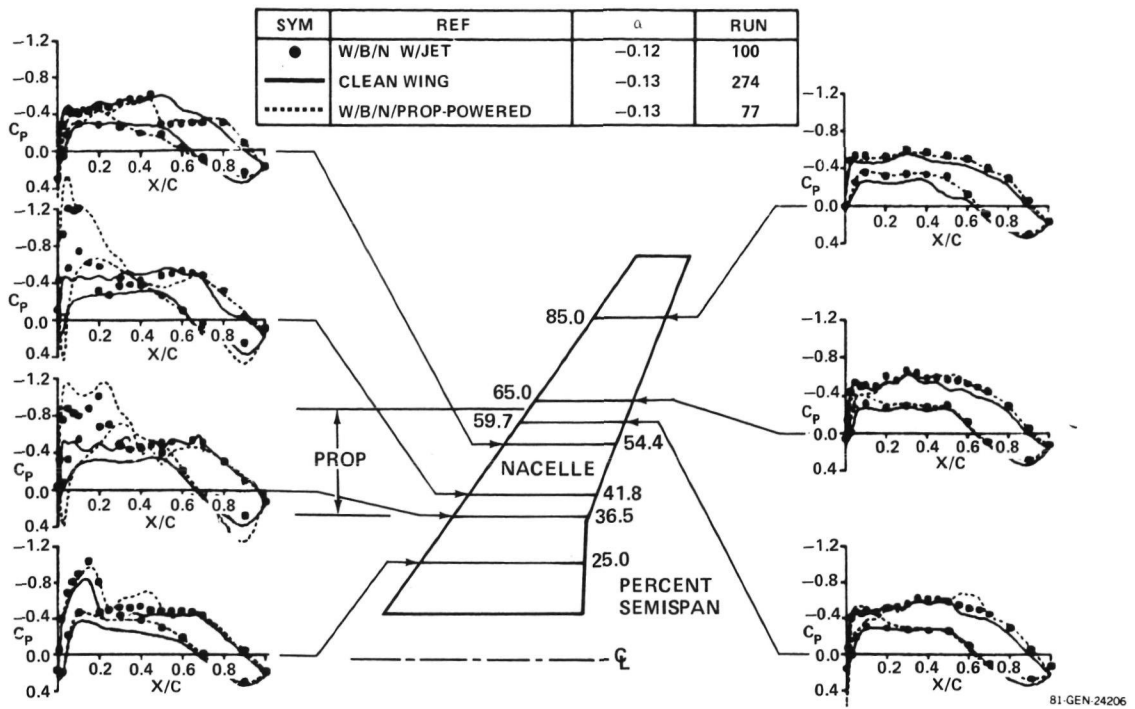


FIGURE 11. PRESSURE DISTRIBUTION AT REDUCED ANGLE OF ATTACK FOR  $M_0 = 0.8$  AND ORIGINAL CONFIGURATION

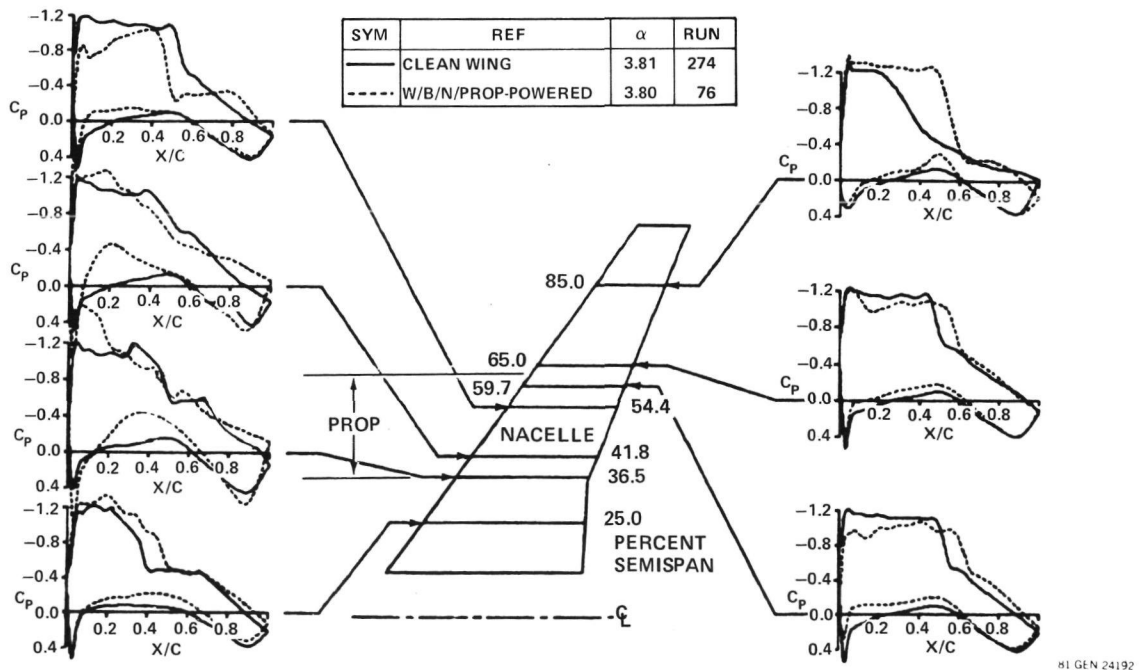
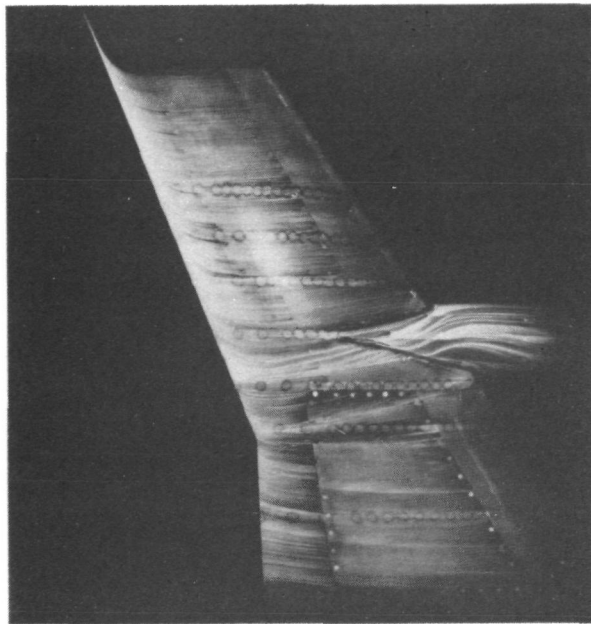


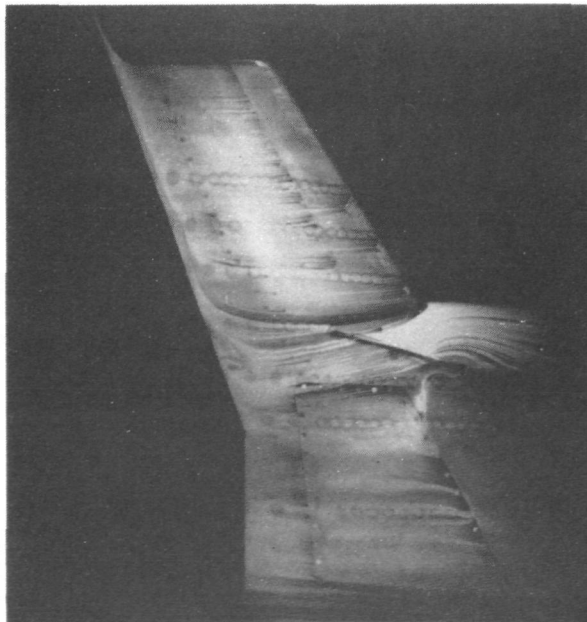
FIGURE 12. PRESSURE DISTRIBUTION AT INCREASED ANGLE OF ATTACK FOR  $M_0 = 0.8$  AND ORIGINAL CONFIGURATION



$M_0 = 0.75$   
 $\alpha = 2 \text{ DEG}$   
WINDMILL  
UPPER SURFACE

81 GEN 23184A

FIGURE 13. OIL FLOW PHOTOGRAPH AT  $M_0 = 0.75$  – WINDMILL CONDITIONS

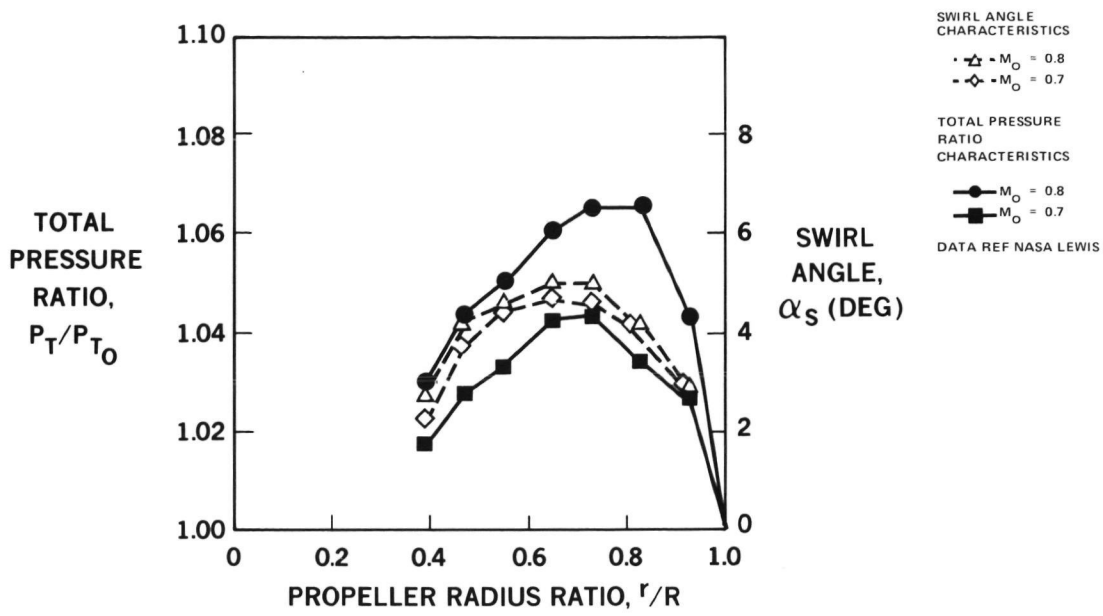


$M_0 = 0.8$   
 $\alpha = 2 \text{ DEG}$   
WINDMILL  
UPPER SURFACE

81 GEN 23181A

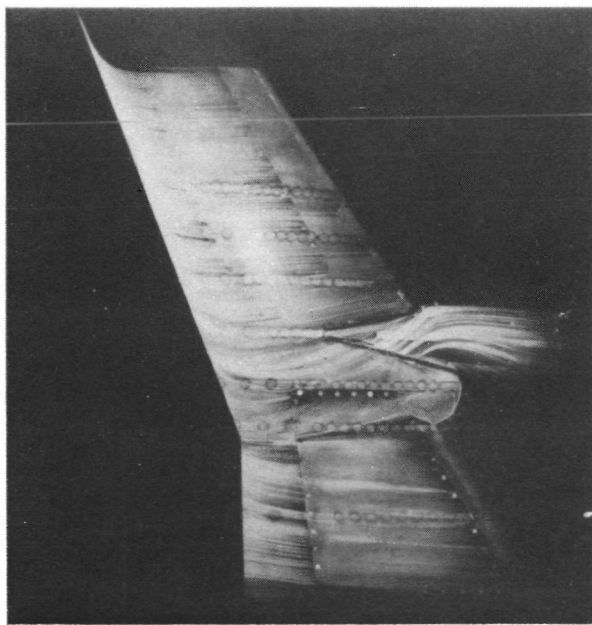
FIGURE 14. OIL FLOW PHOTOGRAPH AT  $M_0 = 0.8$  – WINDMILL CONDITIONS

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81 GEN 24207

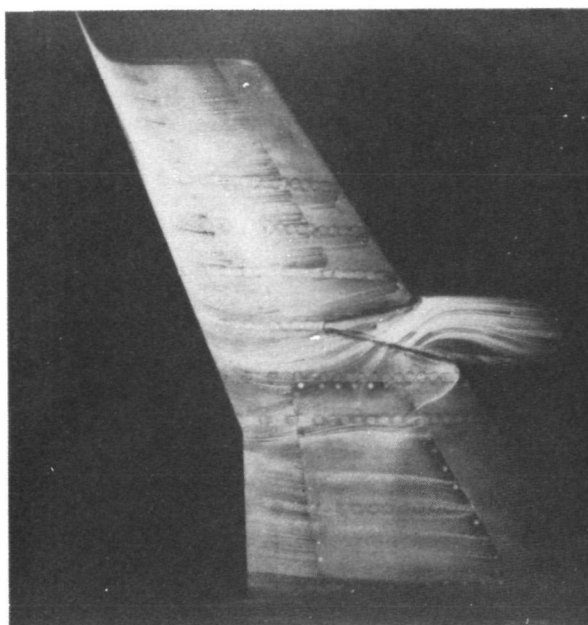
FIGURE 15. PROPELLER SWIRL AND TOTAL PRESSURE RATIO CHARACTERISTICS – ONE BLADE CHORD DOWNSTREAM – BLADE ANGLE,  $\beta = 57$  DEGREES



$M_0 = 0.75$   
 $\alpha = 2$  DEG  
 8100 RPM  
 UPPER SURFACE

81 GEN 23183A

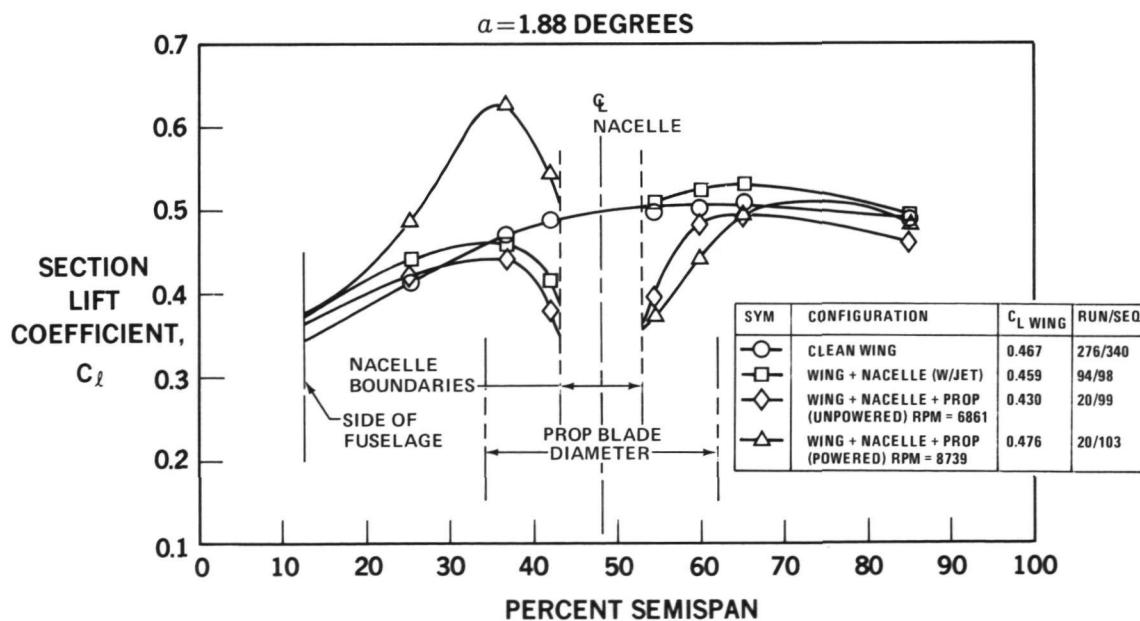
FIGURE 16. OIL FLOW PHOTOGRAPH AT  $M_0 = 0.75$  – MAXIMUM POWER



$M_0 = 0.8$   
 $\alpha = 2 \text{ DEG}$   
 8500 RPM  
 UPPER SURFACE

81 GEN 23182A

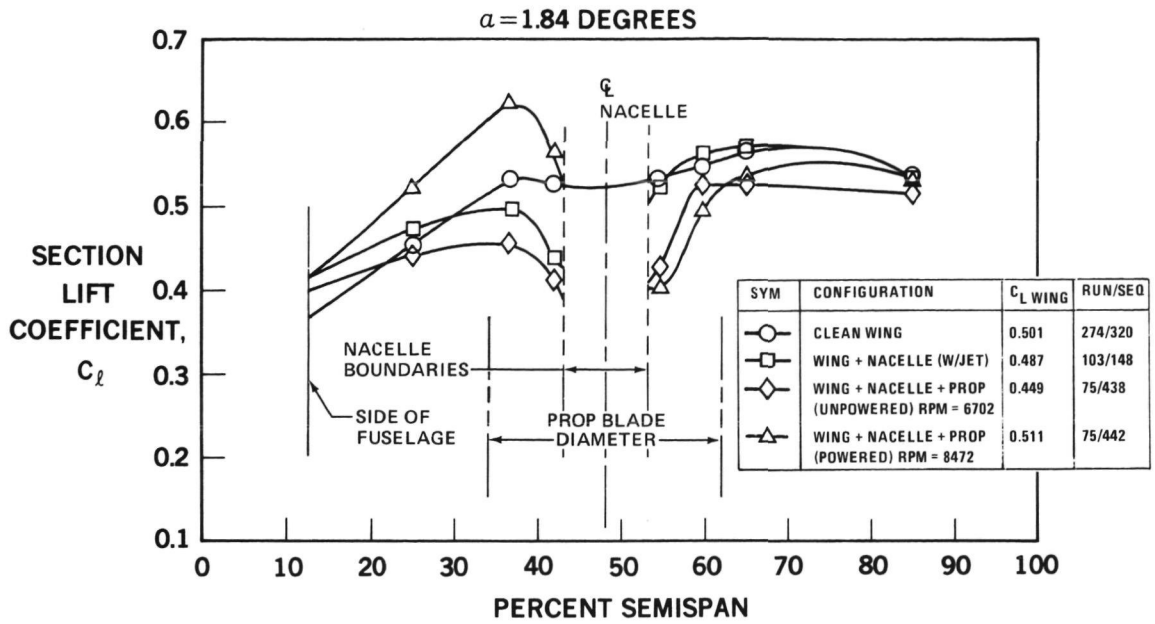
FIGURE 17. OIL FLOW PHOTOGRAPH AT  $M_0 = 0.8$  — MAXIMUM POWER



81 GEN 23189

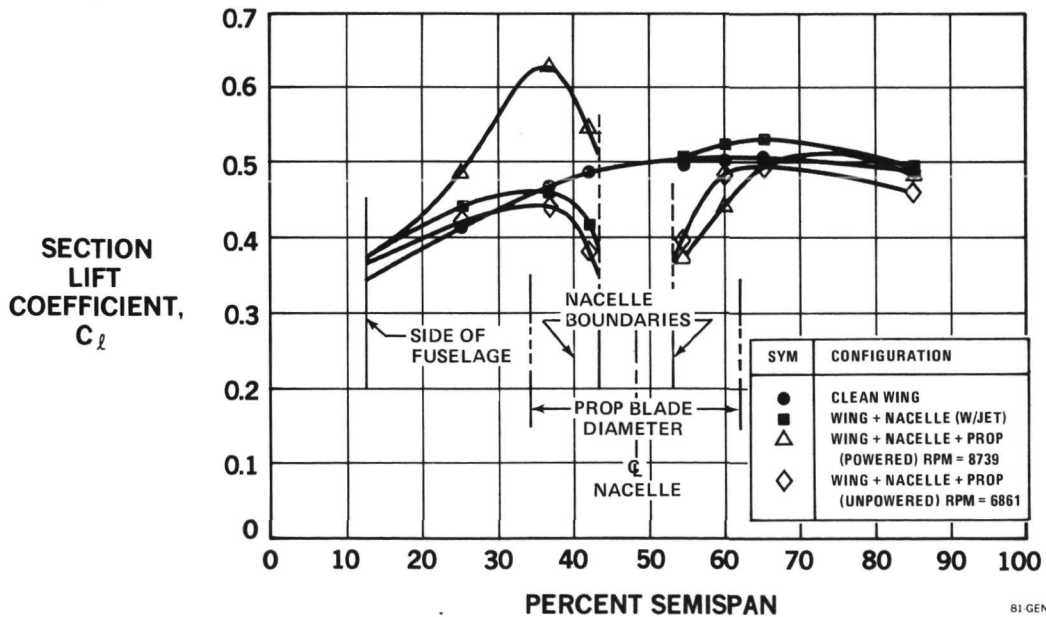
FIGURE 18. COMPARISON OF WING SECTION LIFT DISTRIBUTIONS AT  $M_0 = 0.7$

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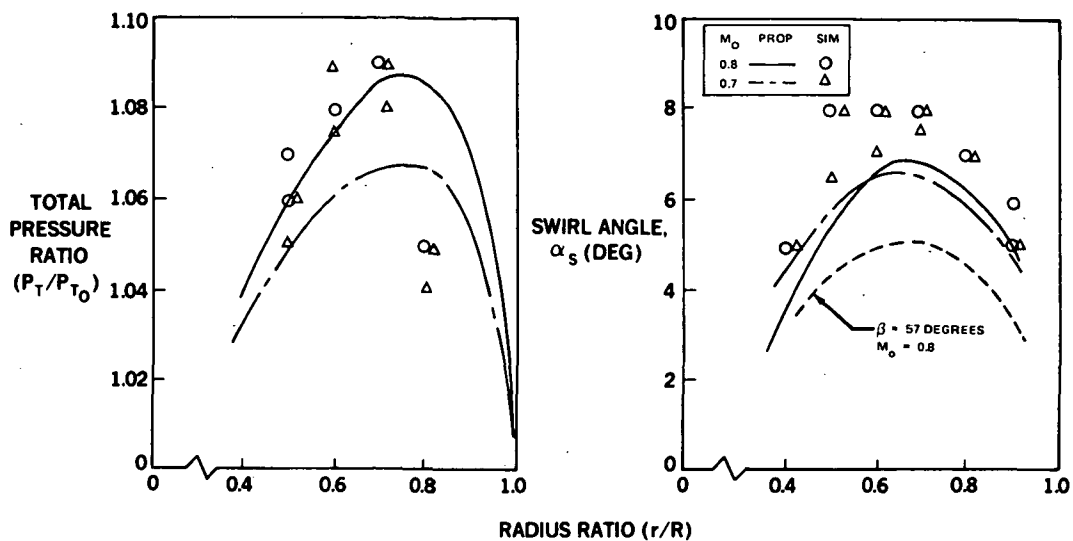
81 GEN 23186

FIGURE 19. COMPARISON OF WING SECTION LIFT DISTRIBUTIONS AT  $M_0 = 0.8$



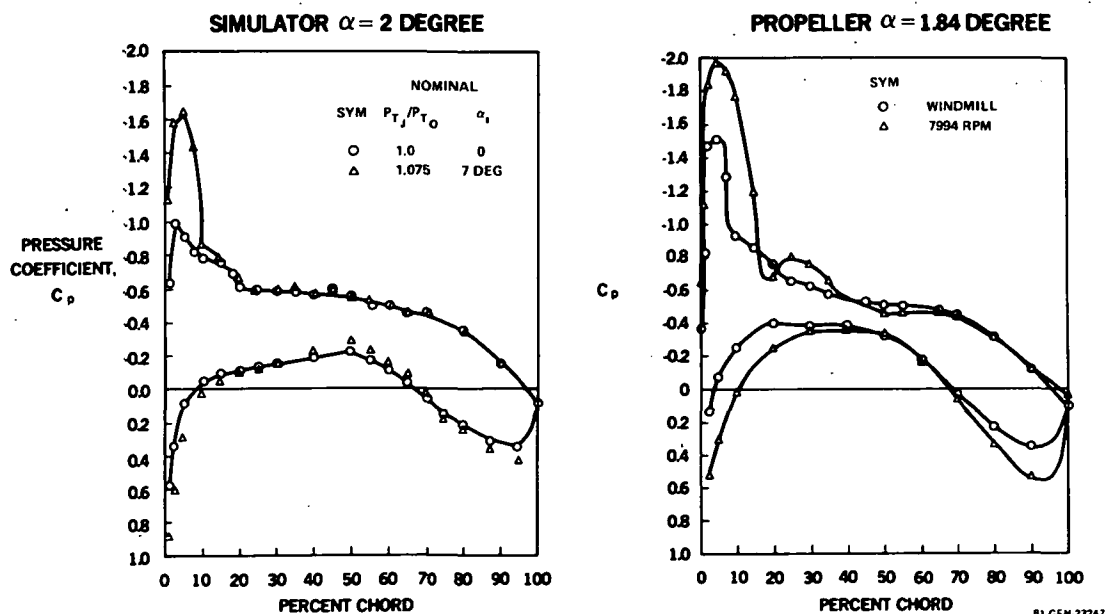
81 GEN 23188

FIGURE 20. LIFT DISTRIBUTION USED FOR INDUCED DRAG CALCULATION



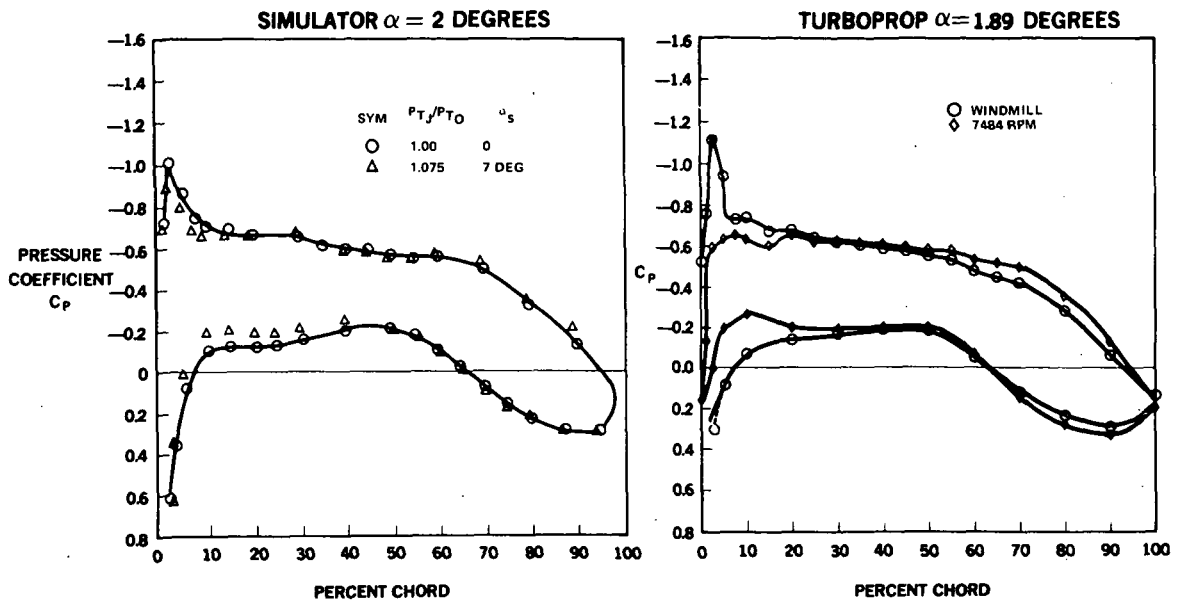
81 GEN 24213

FIGURE 21. COMPARISON OF SIMULATOR EXIT CONDITIONS AND ISOLATED PROPELLER DATA AT  $\beta = 59$  DEGREES



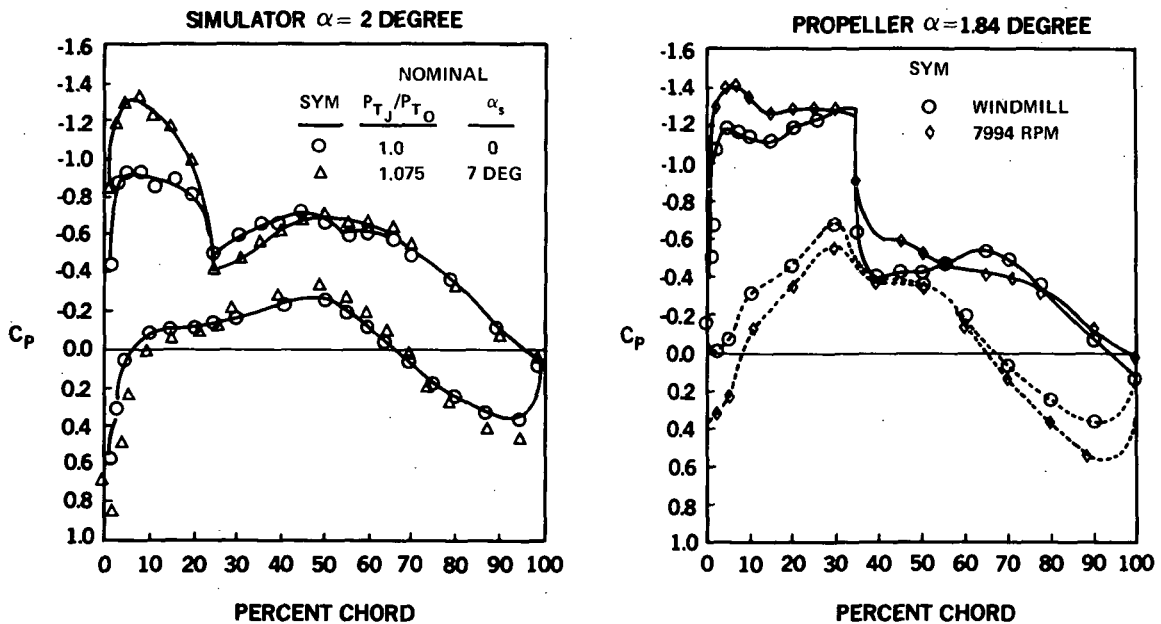
81-GEN-23247

FIGURE 22. COMPARISON OF SIMULATOR AND PROPELLER DATA AT  $M = 0.7$  AND  $\eta = 0.36$



81 GEN 24222

FIGURE 23. COMPARISON OF SIMULATOR AND PROPELLER DATA AT  $M_o = 0.7$  AND  $\eta = 0.5$



81 GEN 23229

FIGURE 24. COMPARISON OF SIMULATOR AND PROPELLER DATA AT  $M_o = 0.8$  AND  $\eta = 0.36$



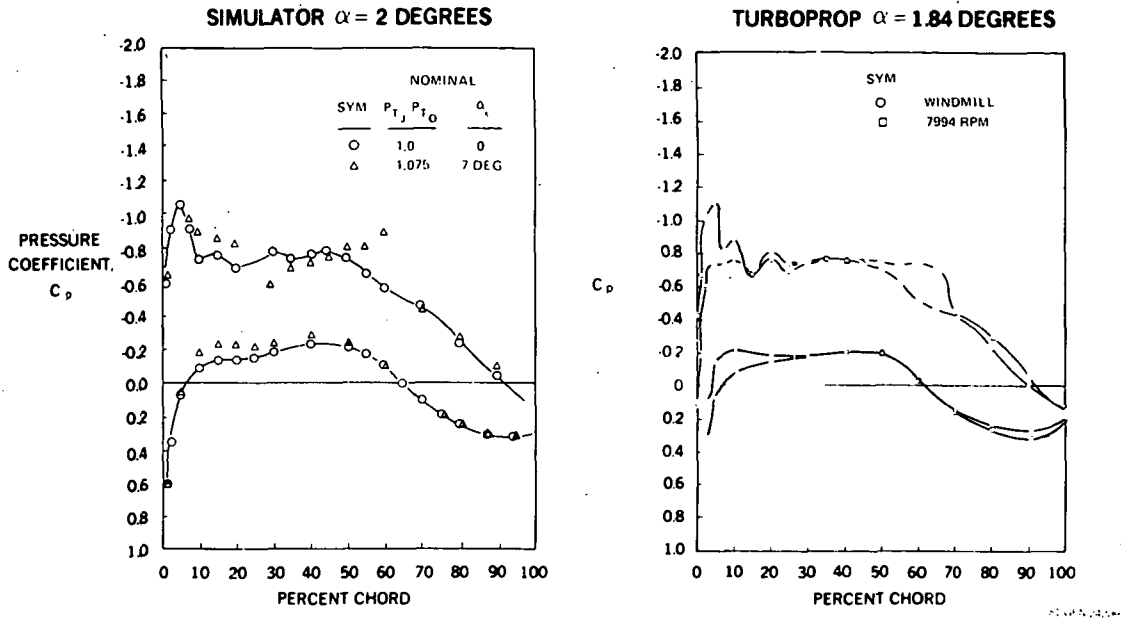


FIGURE 25. COMPARISON OF SIMULATOR AND PROPELLER DATA AT  $M_0 = 0.8$  AND  $\eta = 0.5$

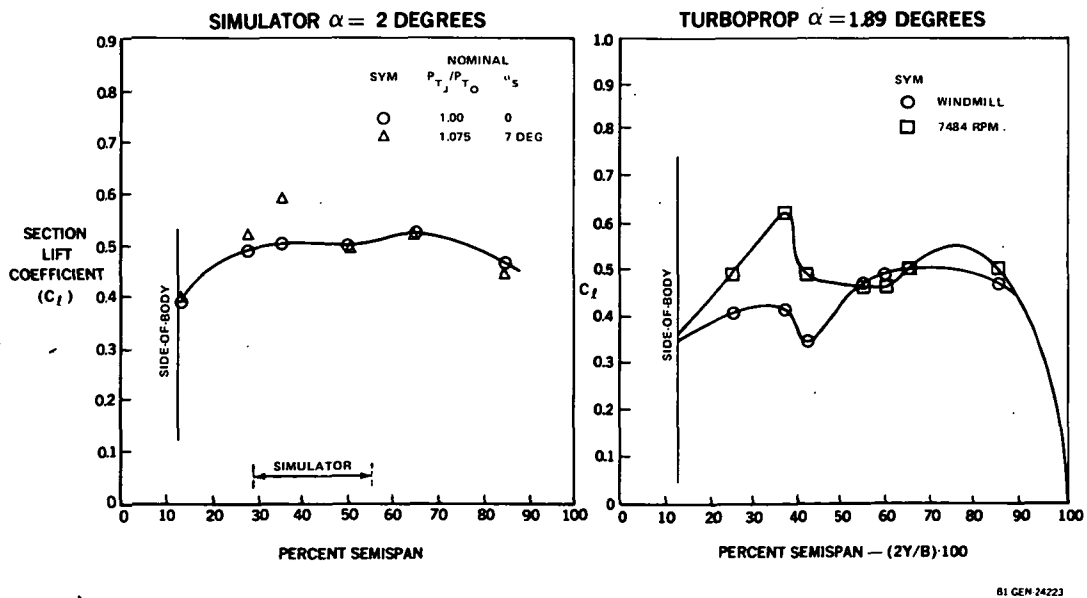
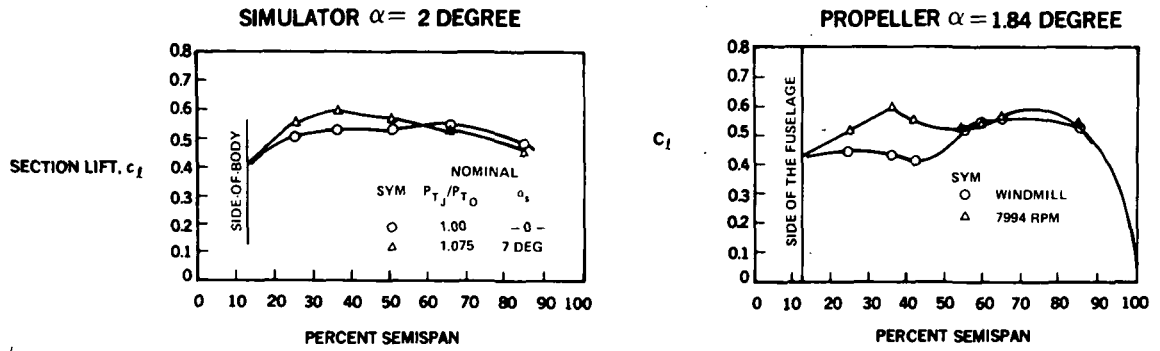
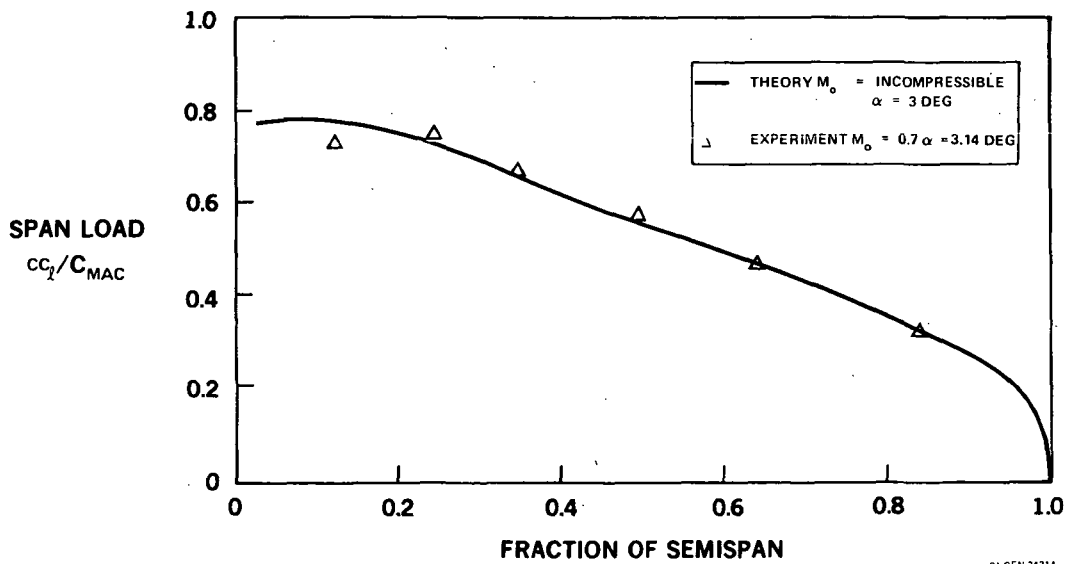


FIGURE 26. COMPARISON OF SIMULATOR AND PROPELLER SECTION LIFT DISTRIBUTION AT  $M_0 = 0.7$



81 GEN 23246

FIGURE 27. COMPARISON OF SIMULATOR AND PROPELLER SECTION LIFT DISTRIBUTION AT  $M = 0.8$



81 GEN 24214

FIGURE 28. COMPARISON OF LIFTING LINE AND EXPERIMENTAL SPAN LOAD FOR CLEAN WING

THEORY	DATA	$M_0$	CONDITION
————		INCOMP	NO SIMULATOR
- - - -	△	0.7	$P_{TJ}/P_{T0} = 1.075$ $\alpha_S = 7$ DEG [UP INBOARD]
- - - -	○	0.7	$P_{TJ}/P_{T0} = 1.075$ $\alpha_S = -7$ DEG [UP OUTBOARD]

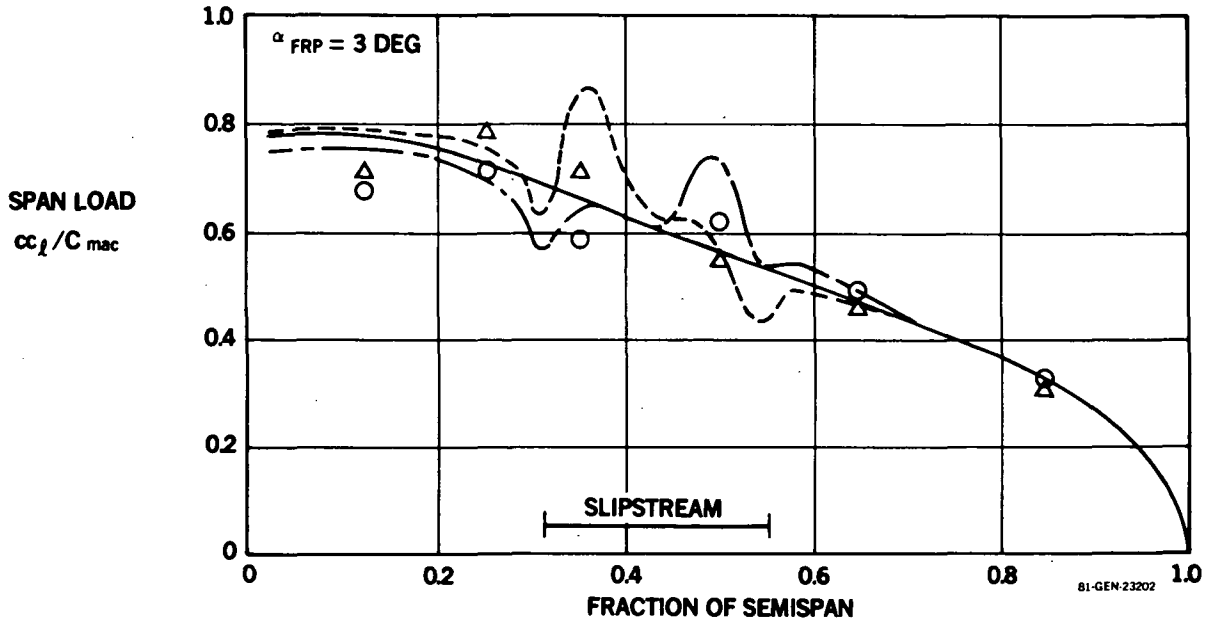


FIGURE 29. COMPARISON OF WING SECTION LIFT DISTRIBUTION WITH LIFTING LINE THEORY - SIMULATOR MODEL

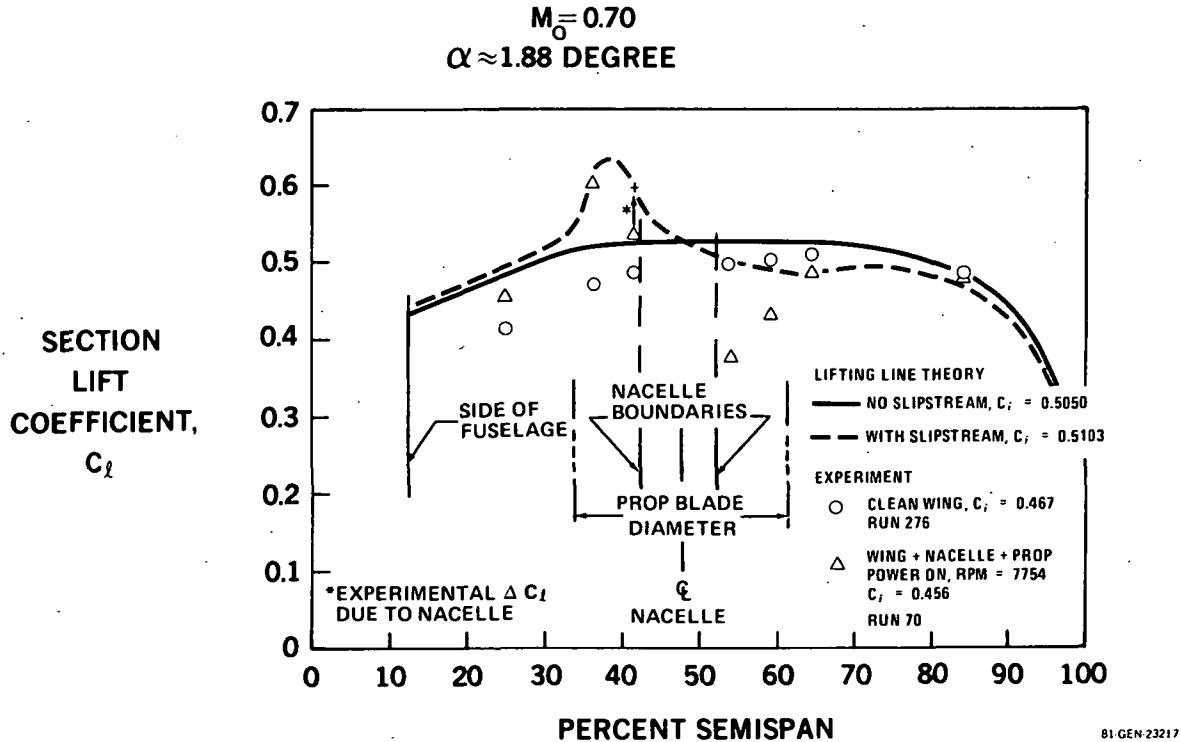


FIGURE 30. COMPARISON OF WING SECTION LIFT DISTRIBUTIONS WITH LIFTING LINE THEORY - PROPELLER MODEL

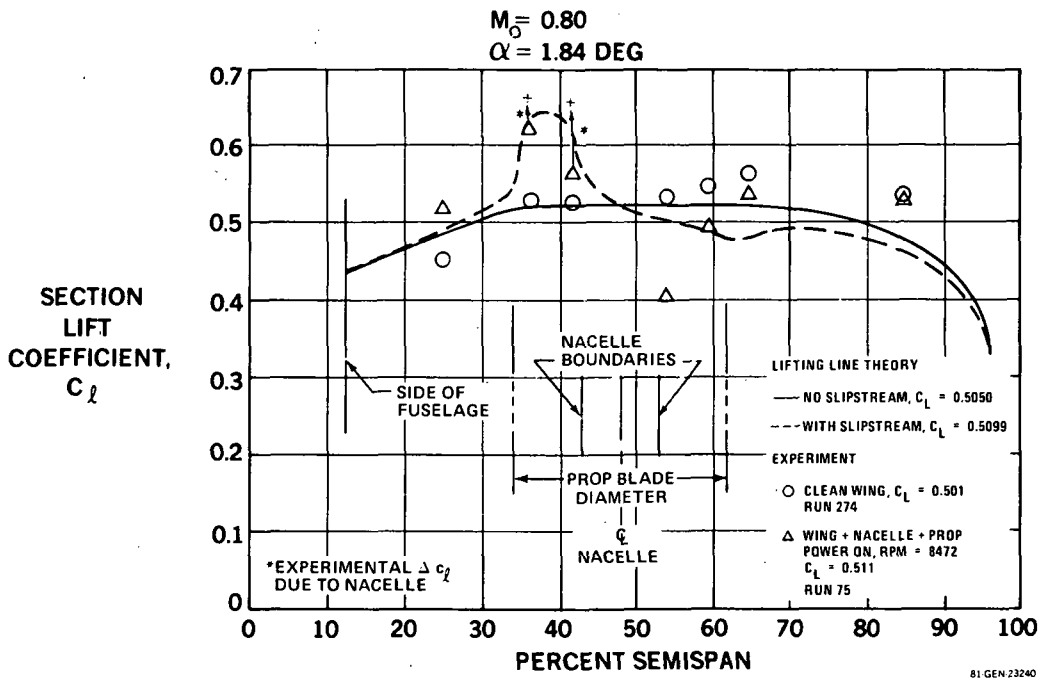
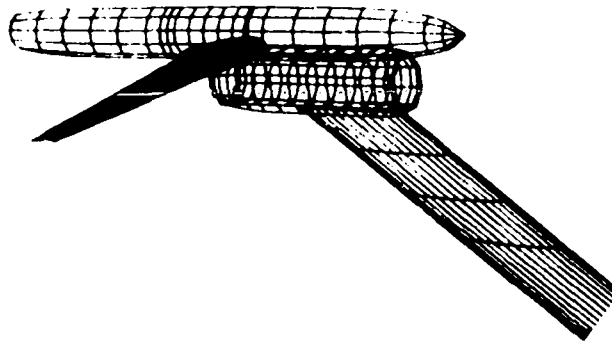


FIGURE 31. COMPARISON OF WING SECTION LIFT DISTRIBUTIONS WITH LIFTING LINE THEORY — PROPELLER MODEL



81-GEN-23191

FIGURE 32. NEUMANN PANELING FOR SIMULATOR MODEL

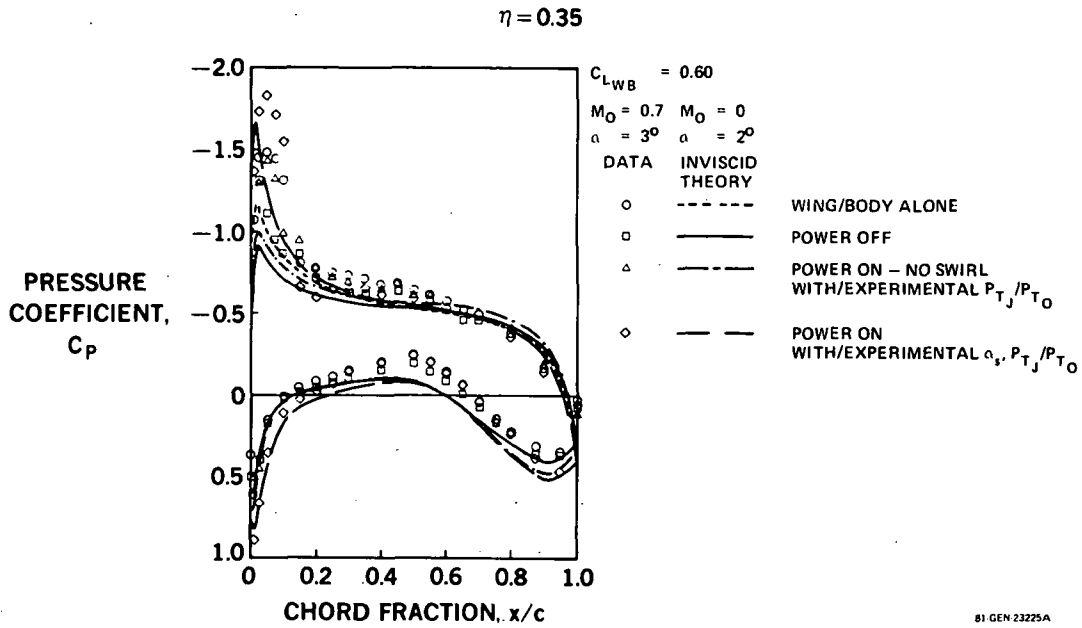


FIGURE 33. SIMULATOR/NEUMANN PRESSURE DISTRIBUTION COMPARISON

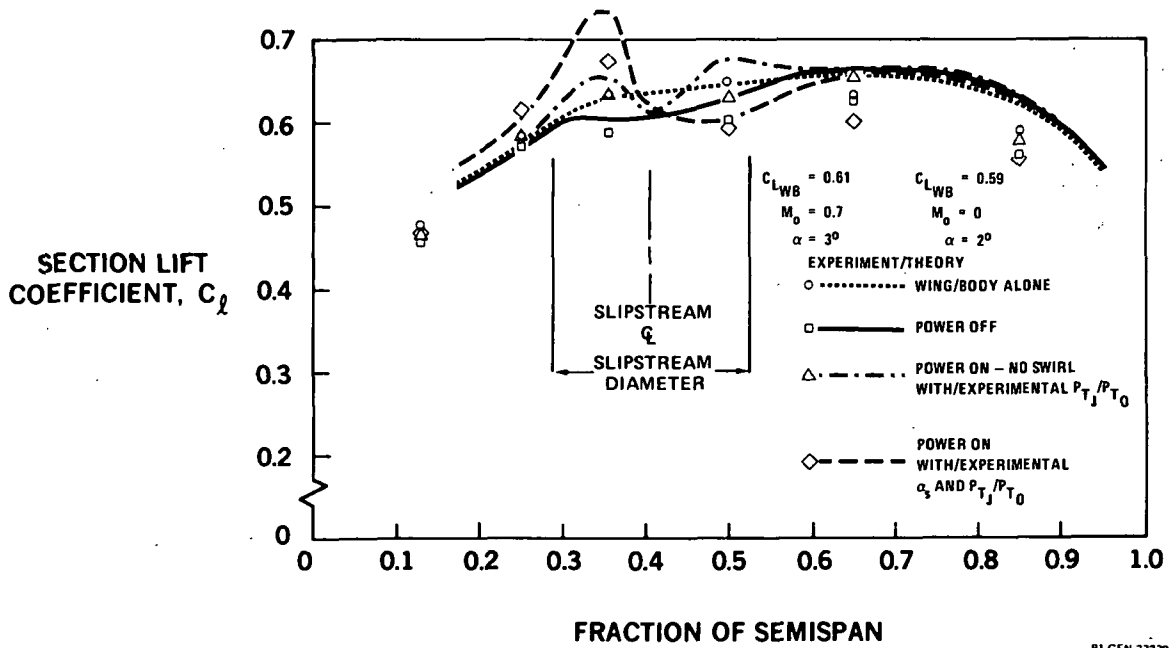
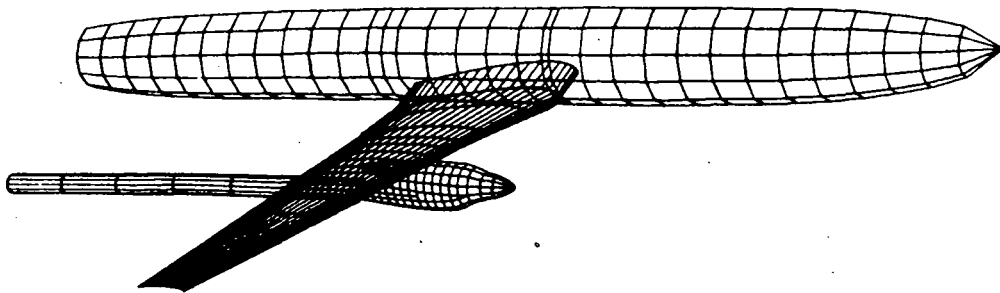
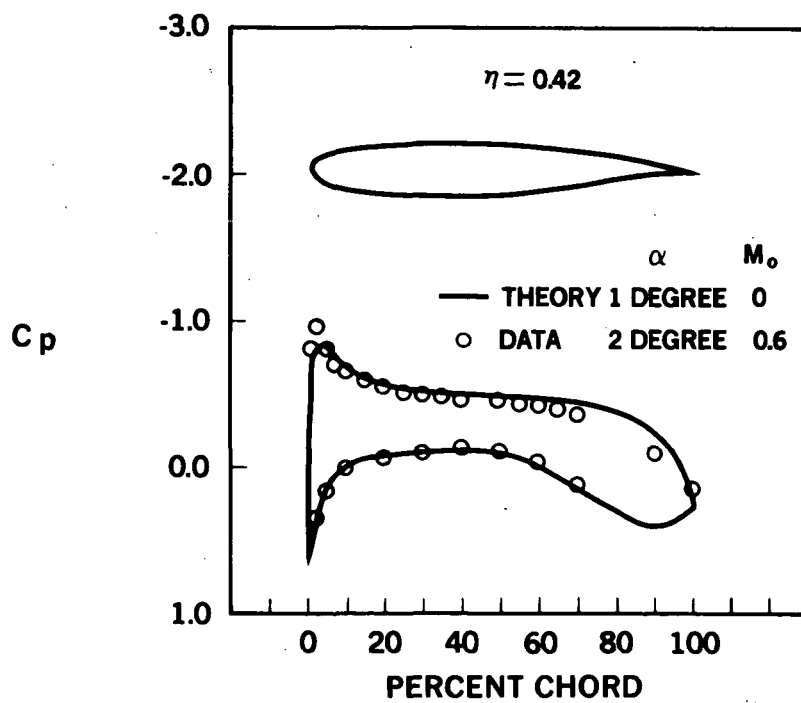


FIGURE 34. SIMULATOR/NEUMANN SECTION LIFT COEFFICIENT COMPARISON



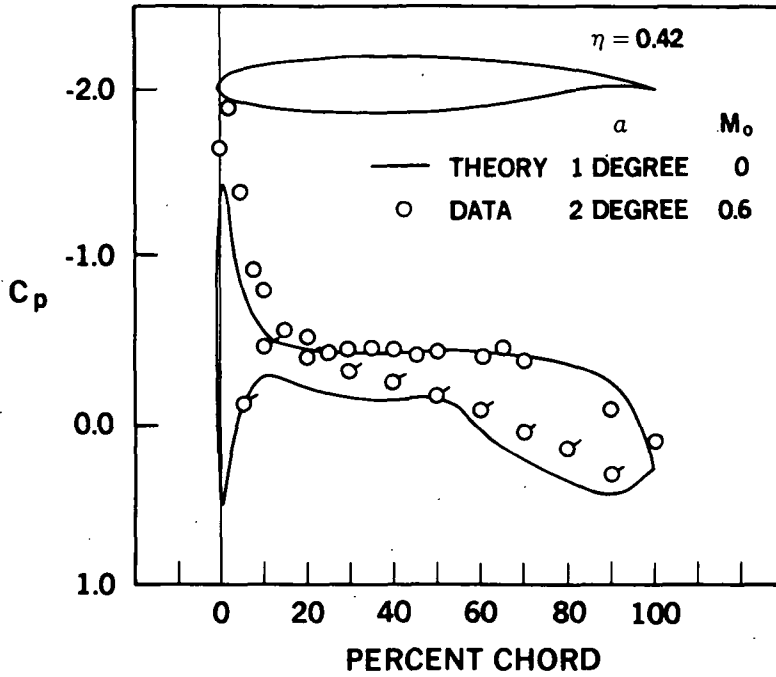
81-GEN-23201

FIGURE 35. NEUMANN PANELING FOR PROPELLER MODEL



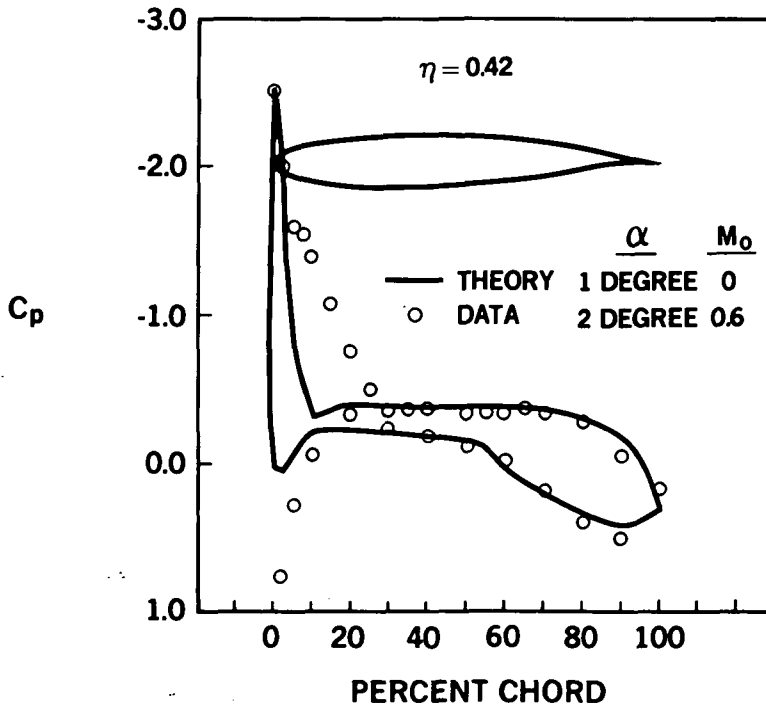
81-GEN-23198

FIGURE 36. COMPARISON OF NEUMANN AND DATA FOR CLEAN WING



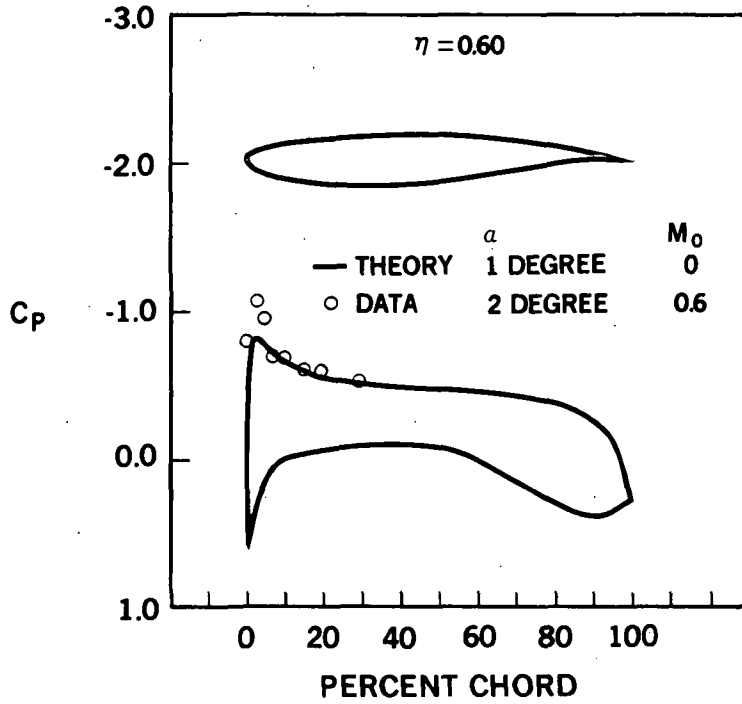
81-GEN-23199

FIGURE 37. COMPARISON OF NEUMANN AND DATA FOR WING AND NACELLE – NO POWER



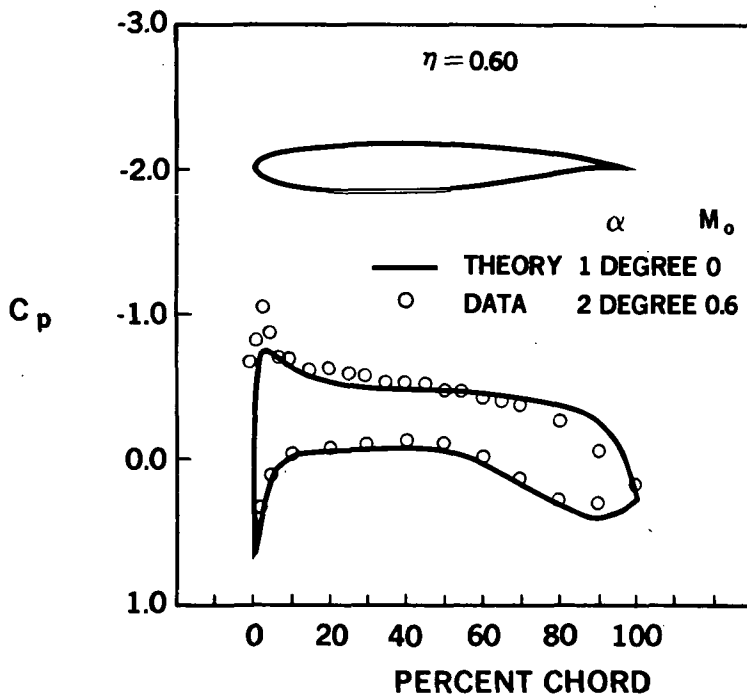
81-GEN-23197

FIGURE 38. COMPARISON OF NEUMANN AND DATA FOR WING AND NACELLE WITH POWER



81-GEN-23196

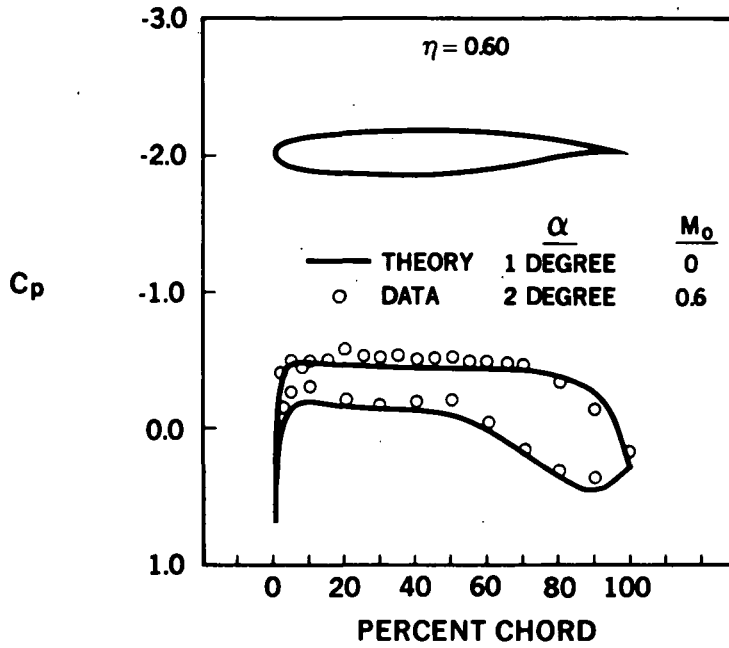
FIGURE 39. COMPARISON OF NEUMANN AND DATA FOR CLEAN WING



81-GEN-23195

FIGURE 40. COMPARISON OF NEUMANN AND DATA FOR WING AND NACELLE - NO POWER





81-GEN-23194

FIGURE 41. COMPARISON OF NEUMANN AND DATA FOR WING AND NACELLE WITH POWER

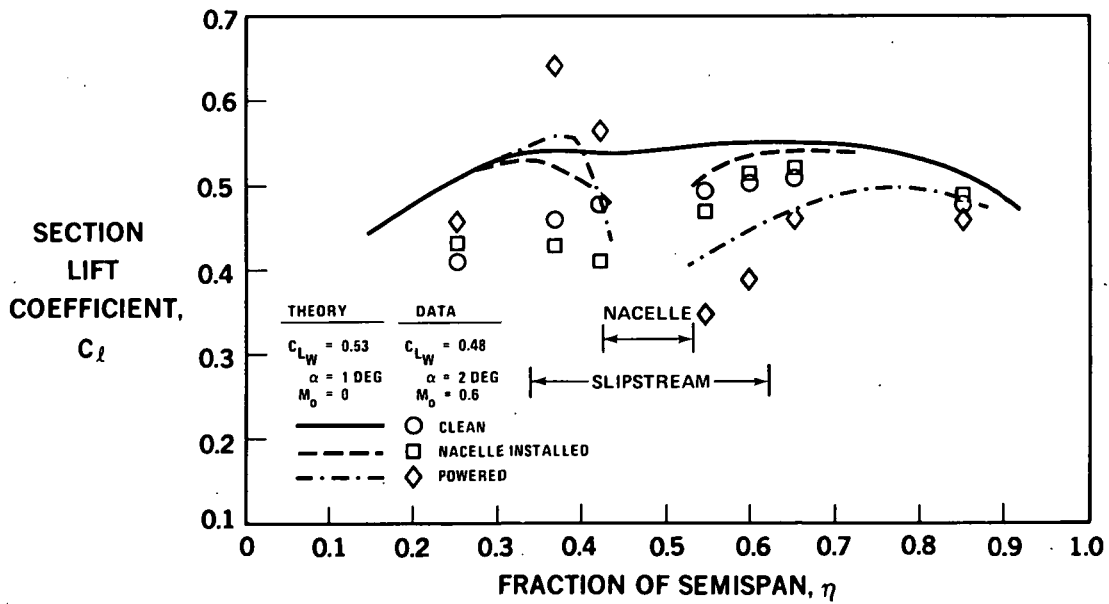


FIGURE 42. DATA COMPARISON OF SECTION LIFT DISTRIBUTION FOR NEUMANN

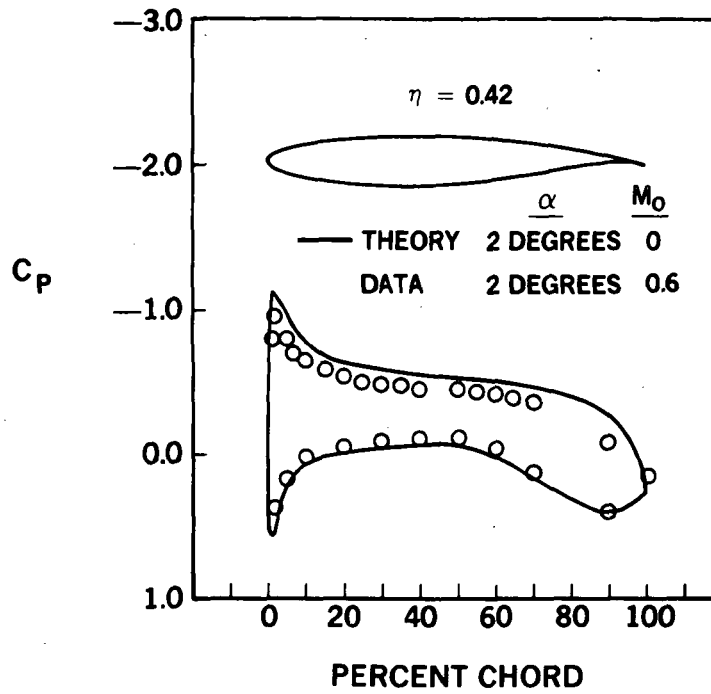


FIGURE 43. COMPARISON OF NEUMANN AND DATA FOR CLEAN WING AT  $\alpha = 2$  DEGREES

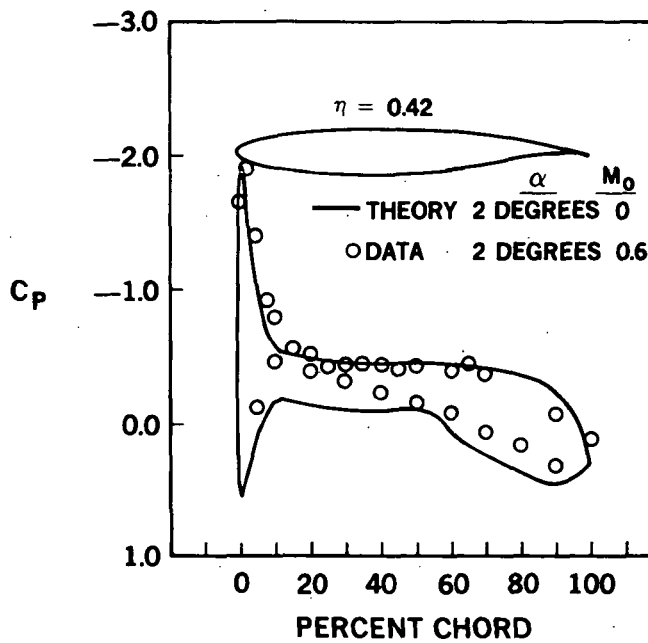
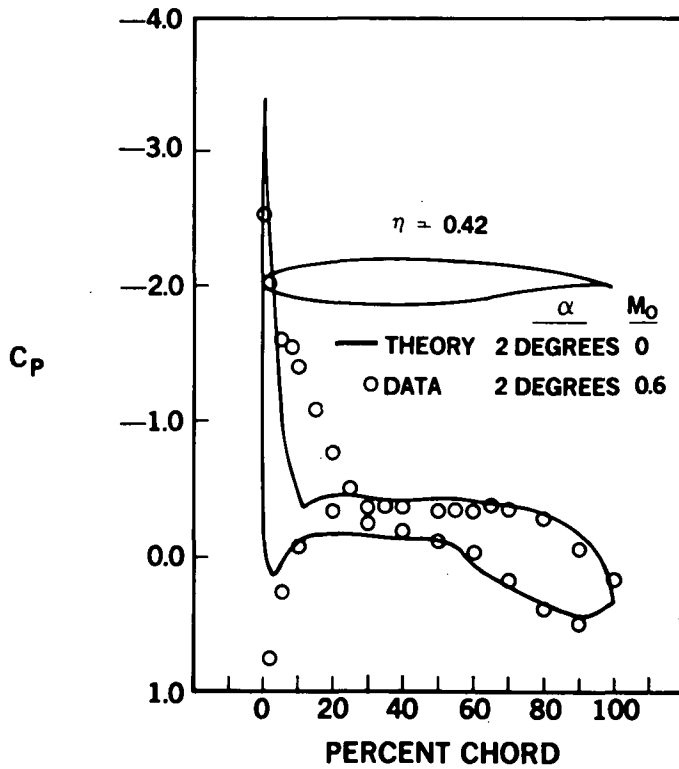


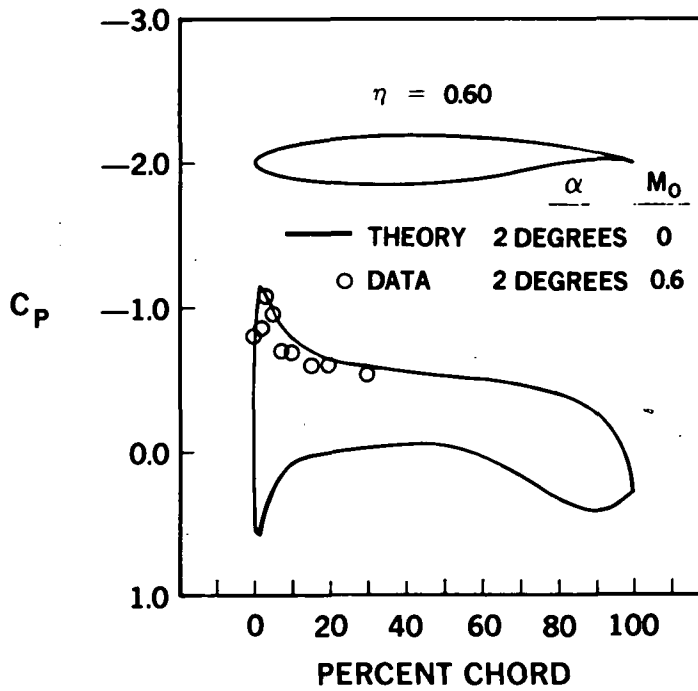
FIGURE 44. COMPARISON OF NEUMANN AND DATA FOR WING/NACELLE NO POWER AT  $\alpha = 2$  DEGREES

81-GEN-24184



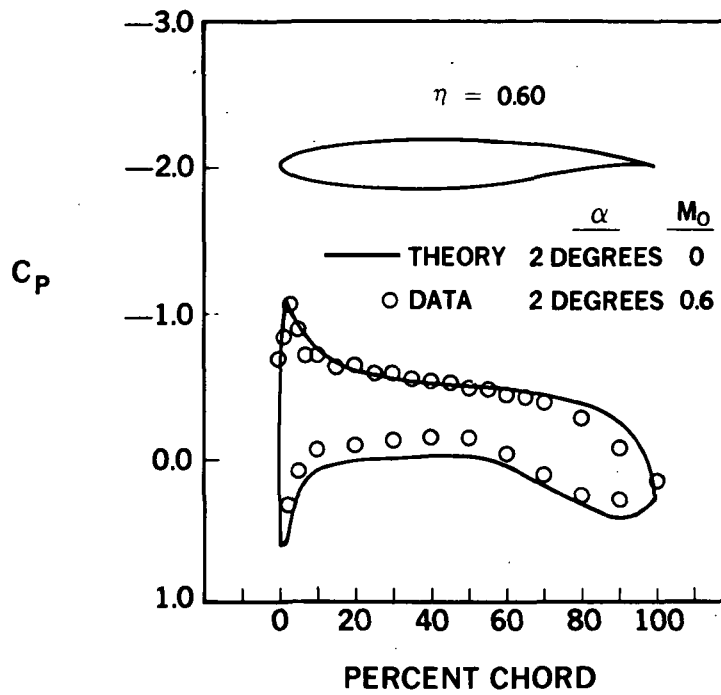
81 GEN 24190

FIGURE 45. COMPARISON OF NEUMANN AND DATA WITH POWER AT  $\alpha = 2$  DEGREES



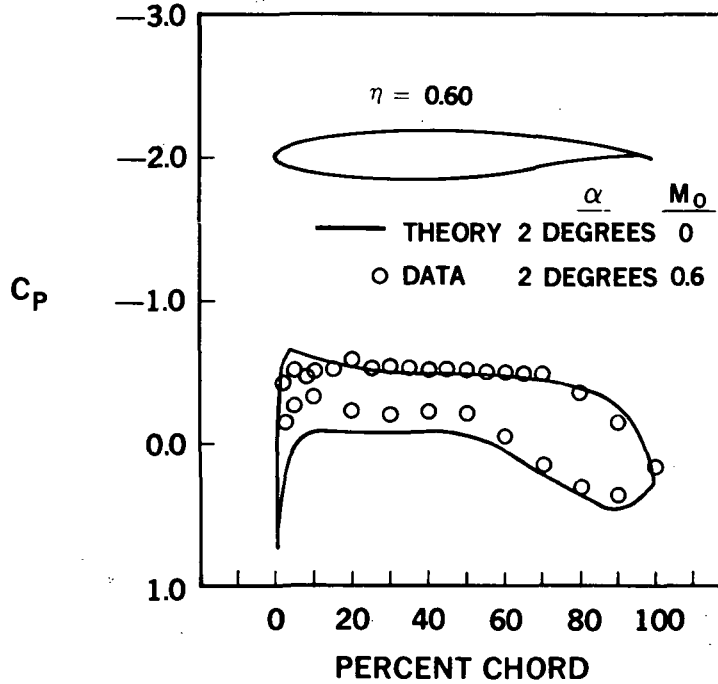
81 GEN 24186

FIGURE 46. COMPARISON OF NEUMANN AND DATA FOR CLEAN WING AT  $\alpha = 2$  DEGREES



81-GEN-24185

FIGURE 47. COMPARISON OF NEUMANN AND DATA FOR WING NACELLE NO POWER AT  $\alpha = 2$  DEGREES



81-GEN 24187

FIGURE 48. COMPARISON OF NEUMANN AND DATA WITH POWER AT  $\alpha = 2$  DEGREES

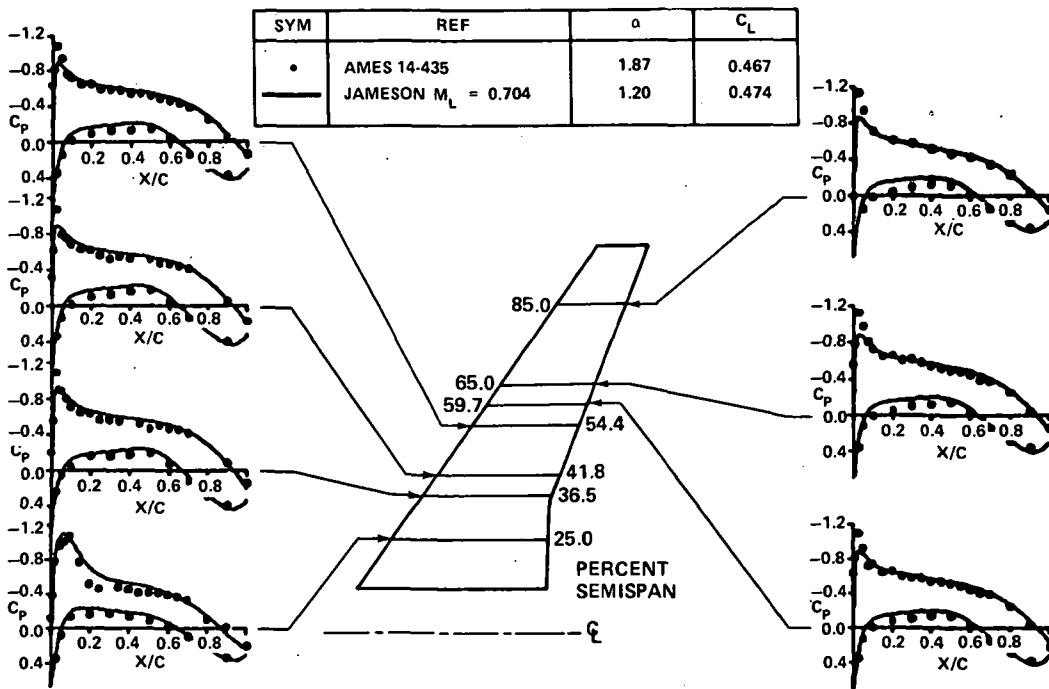
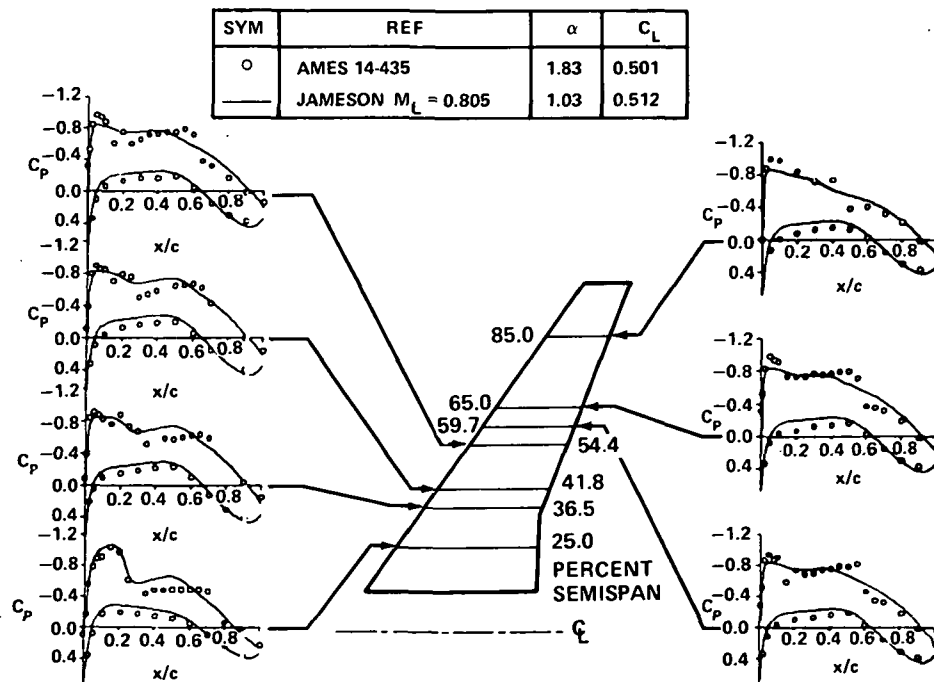


FIGURE 49. COMPARISON OF JAMESON AND EXPERIMENTAL CHORDWISE PRESSURE DISTRIBUTIONS FOR CLEAN WING CONFIGURATION  $M_0 = 0.7$



81-GEN-23243A

FIGURE 50. COMPARISON OF JAMESON AND EXPERIMENTAL CHORDWISE PRESSURE DISTRIBUTIONS FOR CLEAN WING CONFIGURATION  $M_0 = 0.800$

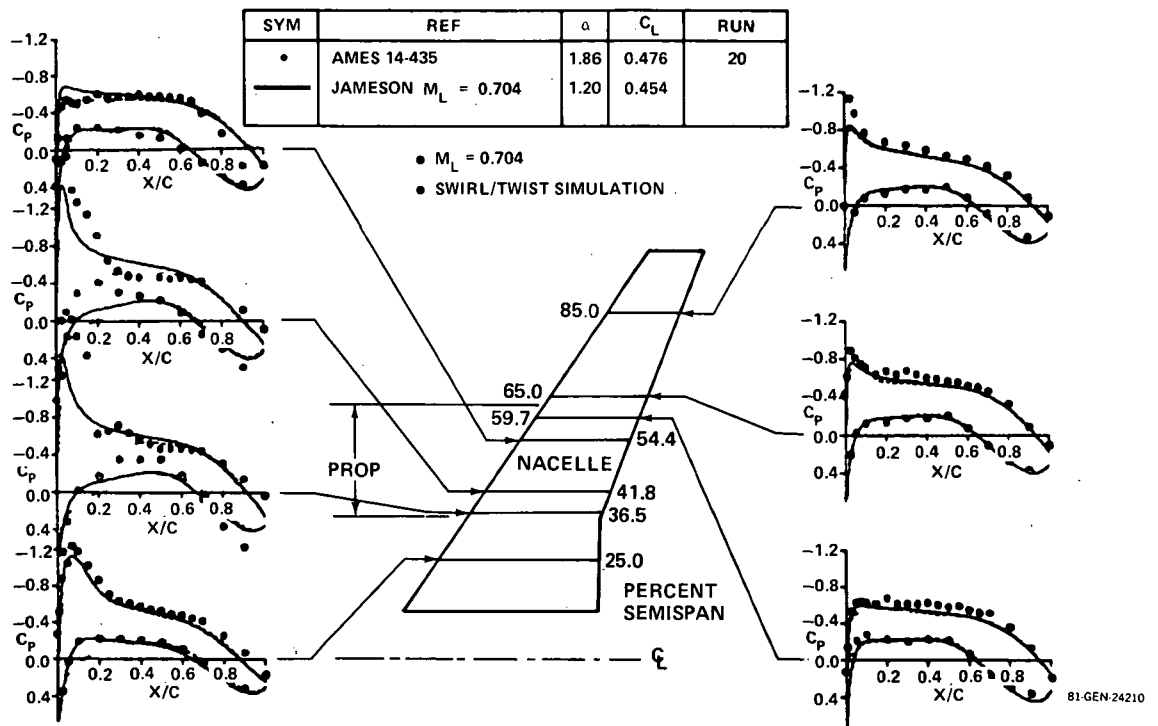


FIGURE 51. COMPARISON OF JAMESON AND EXPERIMENTAL CHORDWISE PRESSURE DISTRIBUTIONS – WITH POWER  $M_0 = 0.7$

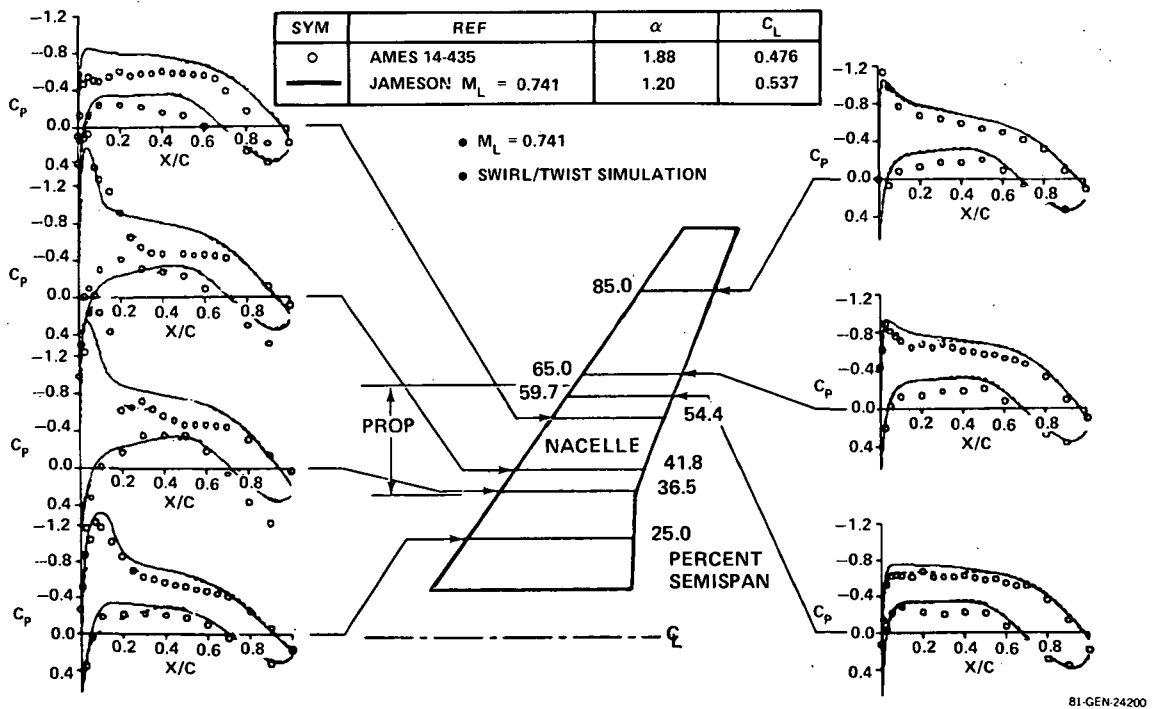


FIGURE 52. COMPARISON OF JAMESON AND EXPERIMENTAL CHORDWISE PRESSURE DISTRIBUTIONS – WITH POWER  $M_0 = 0.7$

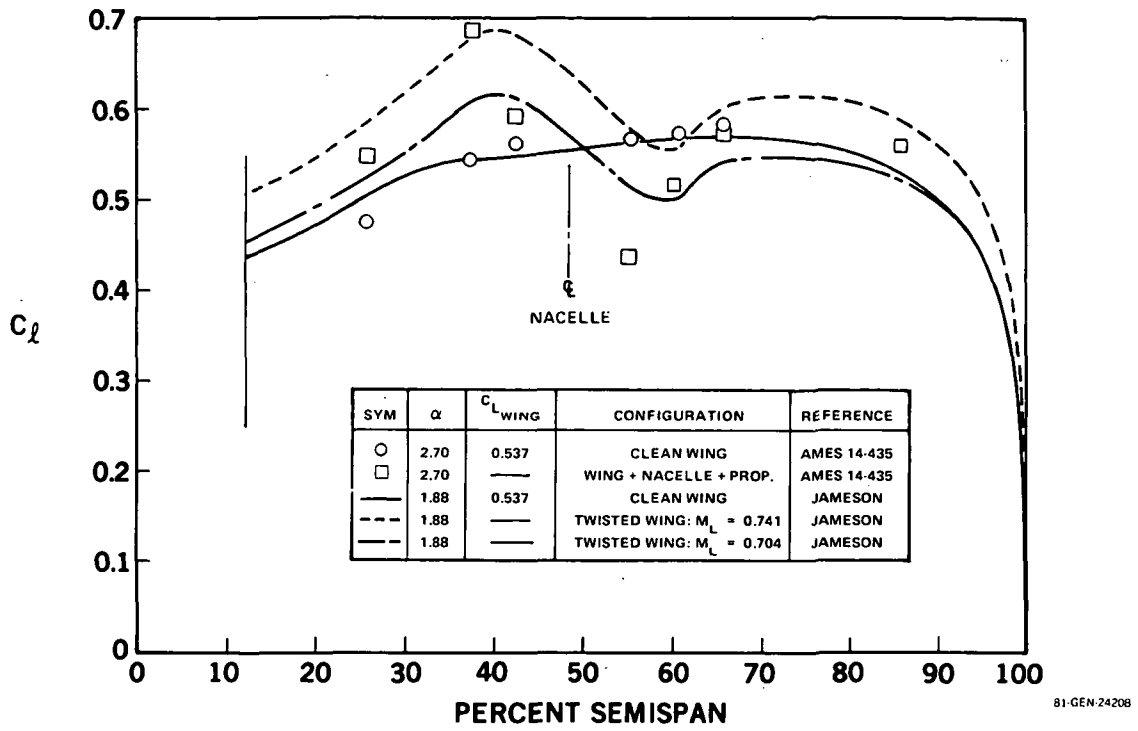


FIGURE 53. EXPERIMENTAL WING SECTION LIFT COMPARISON WITH THEORY AT  $M_0 = 0.7$

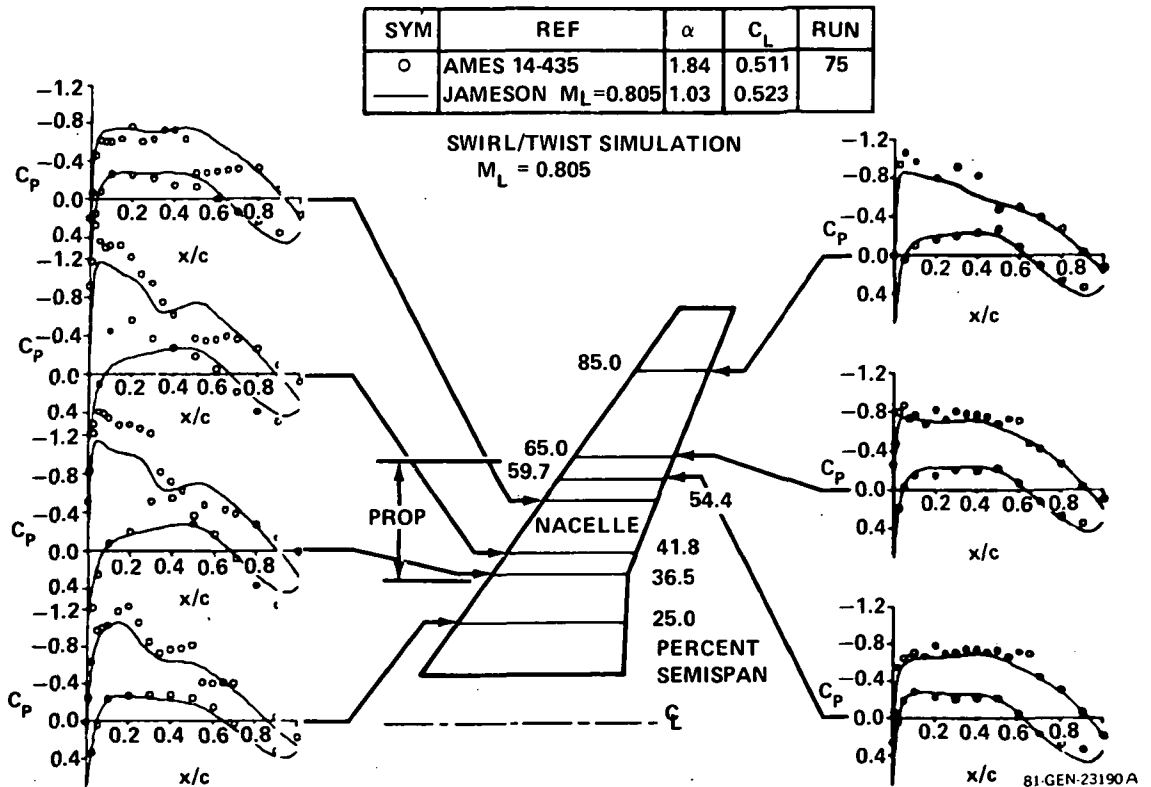
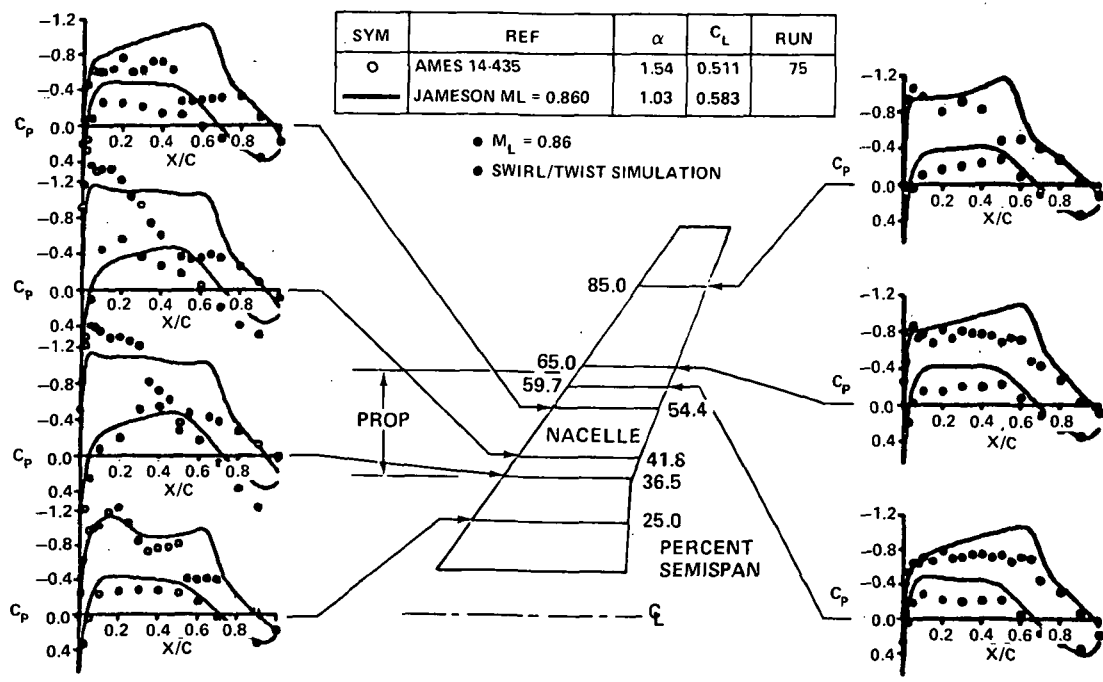
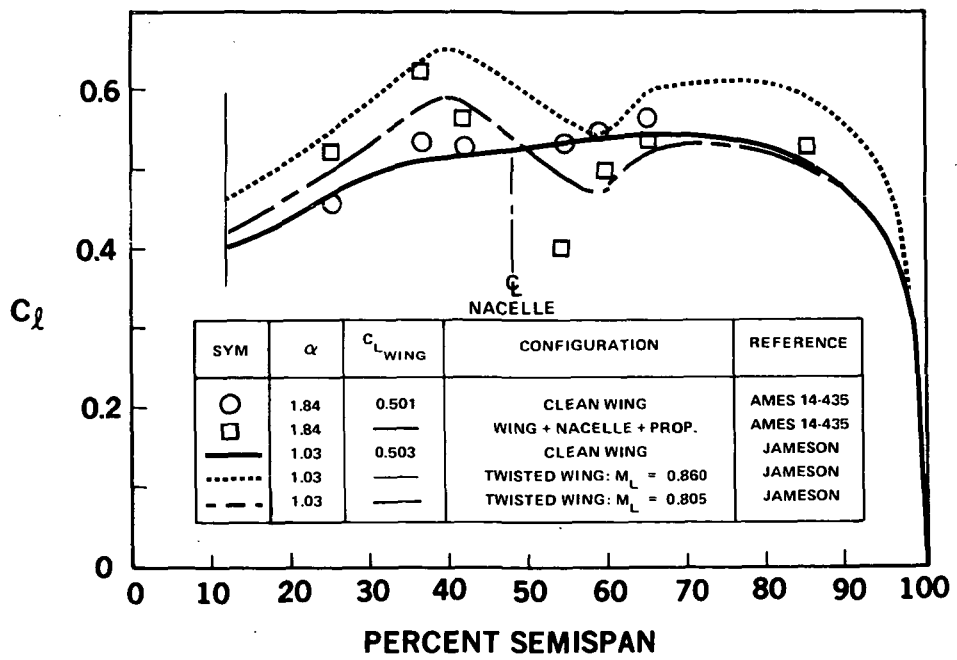


FIGURE 54. COMPARISON OF JAMESON AND EXPERIMENTAL CHORDWISE PRESSURE DISTRIBUTIONS FOR POWERED PROP CONFIGURATION



81 GEN-24201

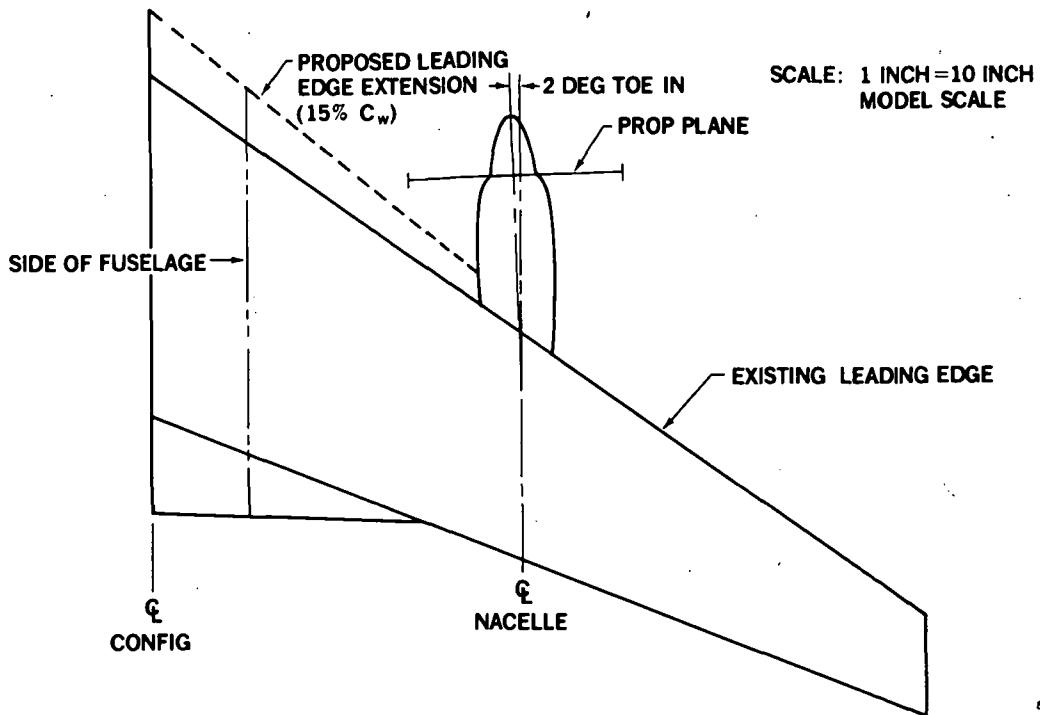
FIGURE 55. COMPARISON OF JAMESON AND EXPERIMENTAL CHORDWISE PRESSURE DISTRIBUTIONS FOR POWERED PROP CONFIGURATION AT  $M_0 = 0.8$



81 GEN-24204

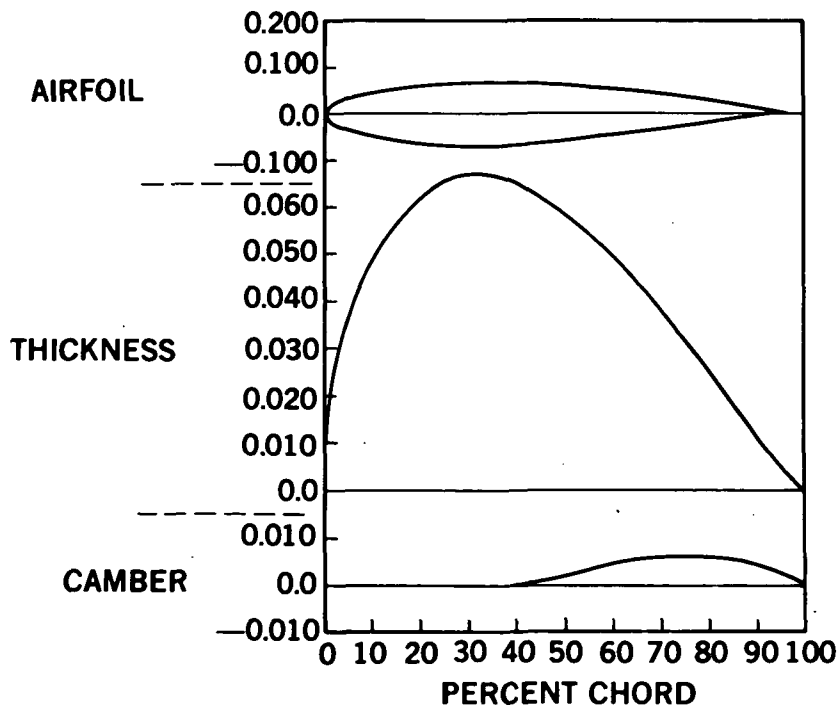
FIGURE 56. WING SECTION LIFT COMPARISON WITH THEORY AT  $M_0 = 0.8$





81-GEN-23213

FIGURE 57. PLANFORM MODIFICATION



81-GEN 24195

FIGURE 58. MOD 2 AND MOD 3 AIRFOILS AT  $\eta = 0.120$

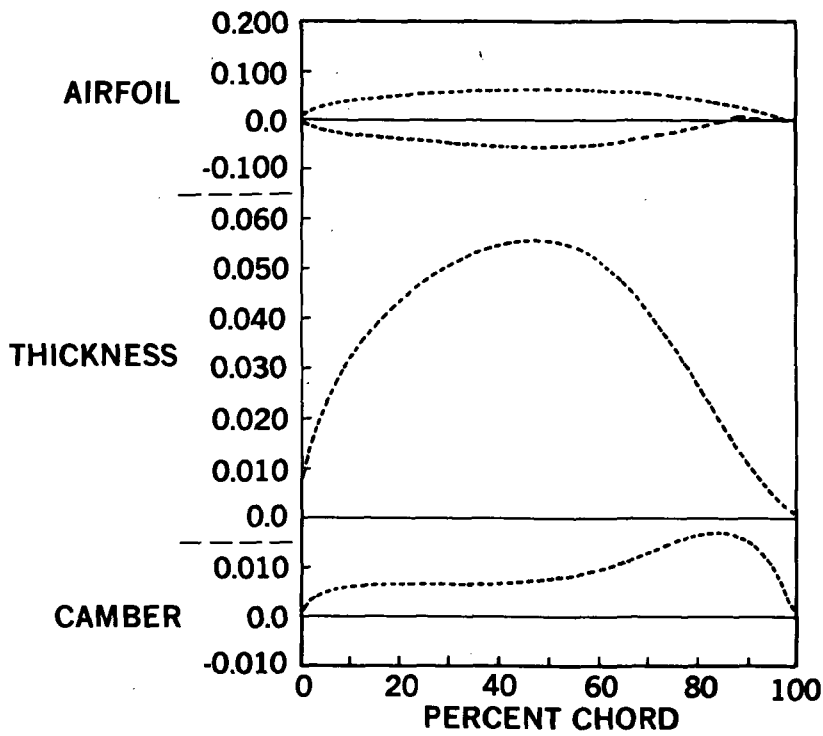


FIGURE 59. MOD 2 AIRFOIL AT  $\eta = 0.350$

81-GEN-23212

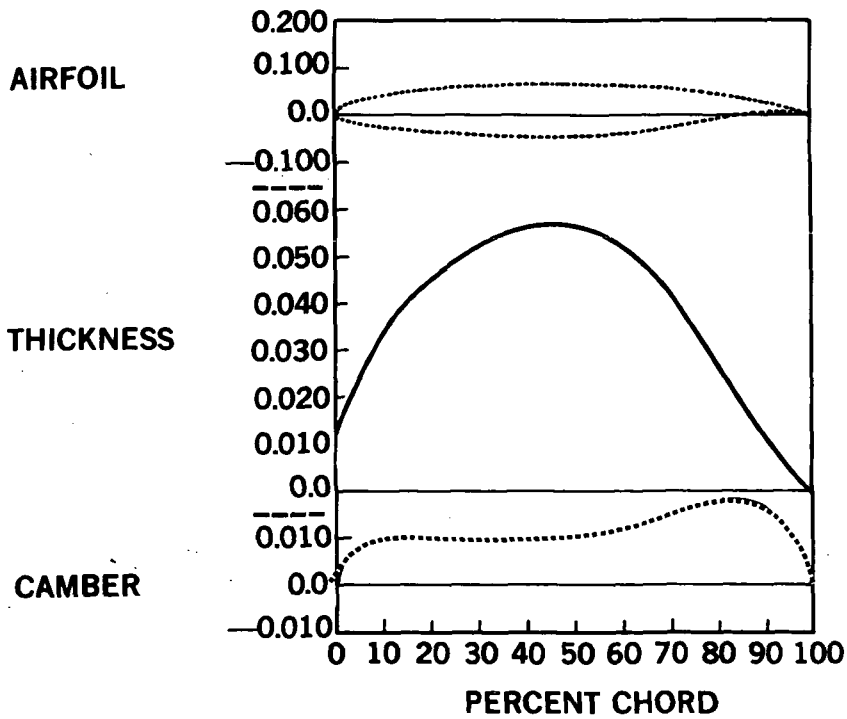


FIGURE 60. MOD 2 AIRFOIL AT  $\eta = 0.43$

81-GEN-24193

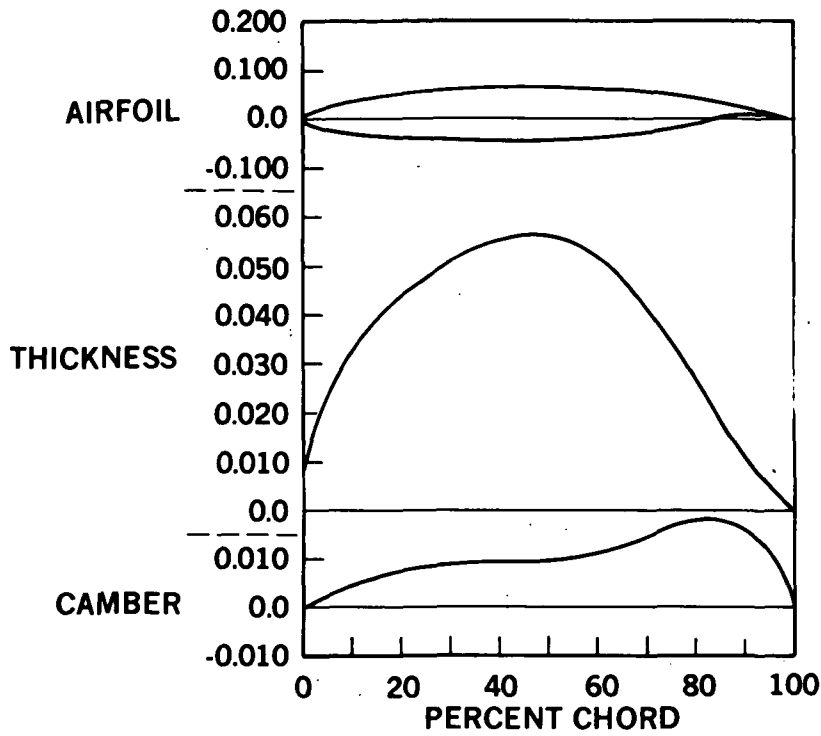


FIGURE 61. MOD 3 AIRFOIL AT  $\eta = 0.350$

81 GEN 23226

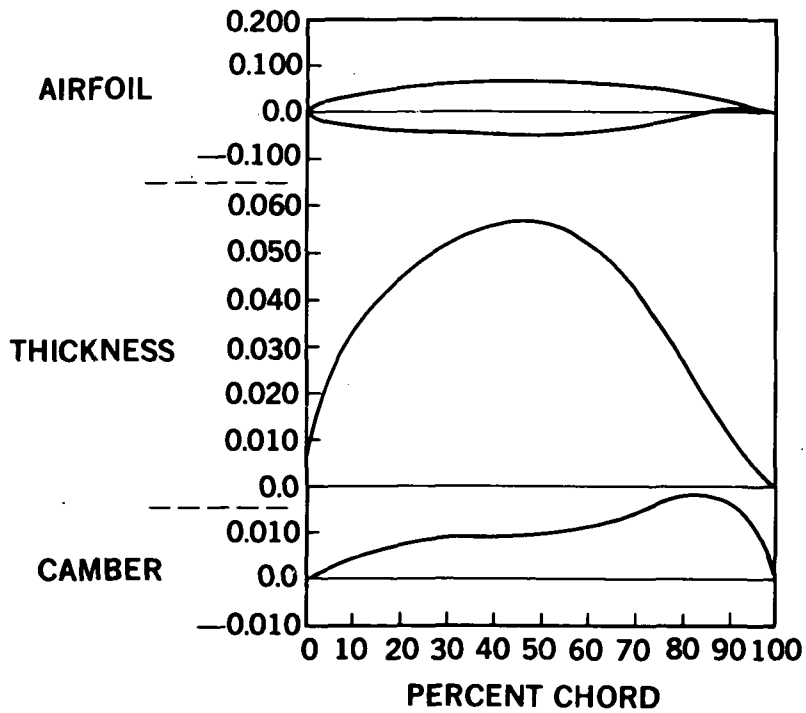


FIGURE 62. MOD 3 AIRFOIL AT  $\eta = 0.43$

81 GEN 24194

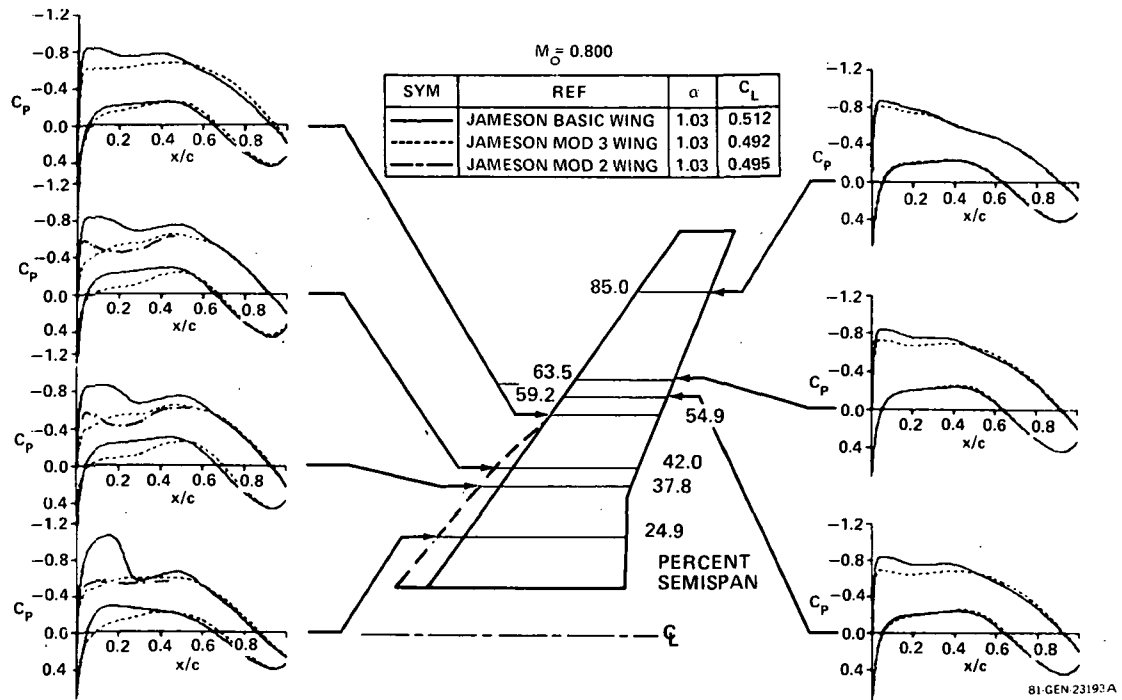


FIGURE 63. COMPARISON OF CALCULATED CHORDWISE PRESSURE DISTRIBUTIONS USING JAMESON — CLEAN WING  $M_o = 0.800$

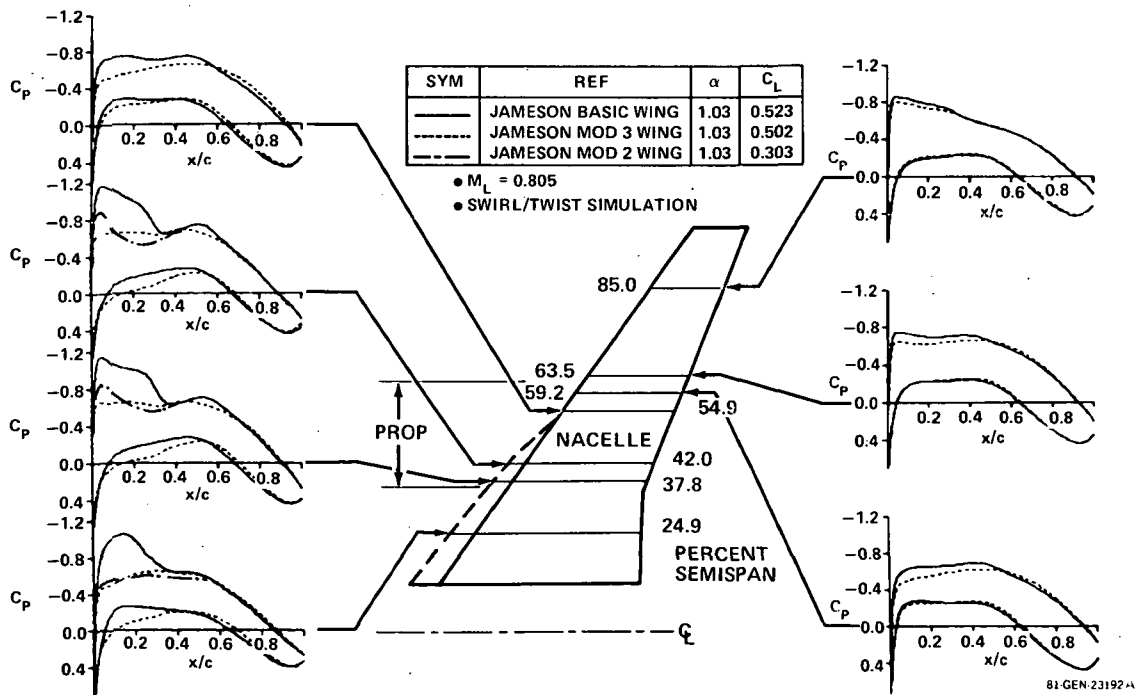
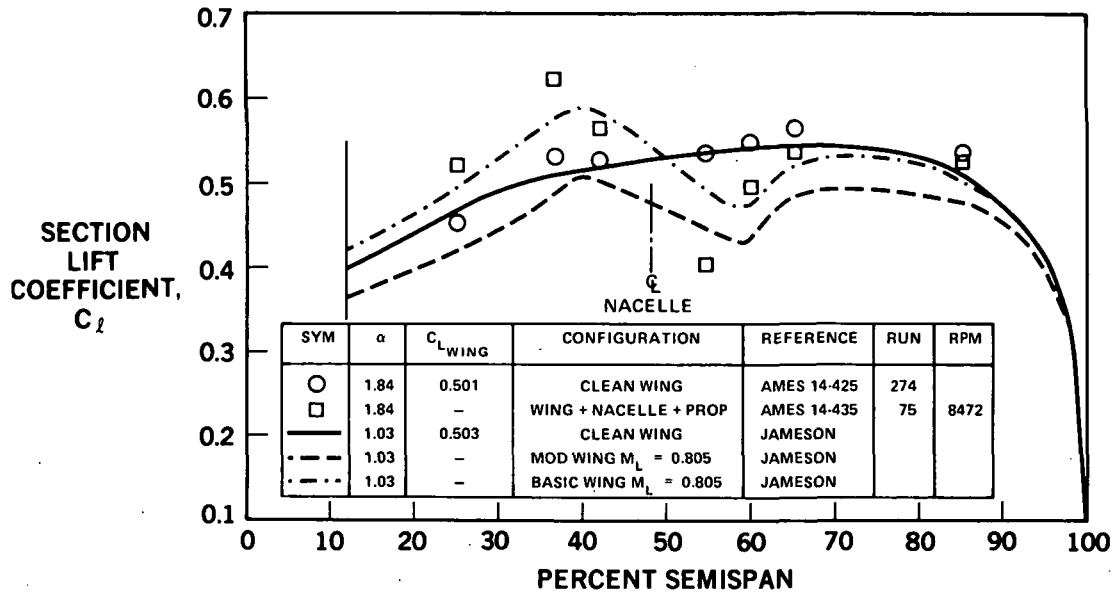
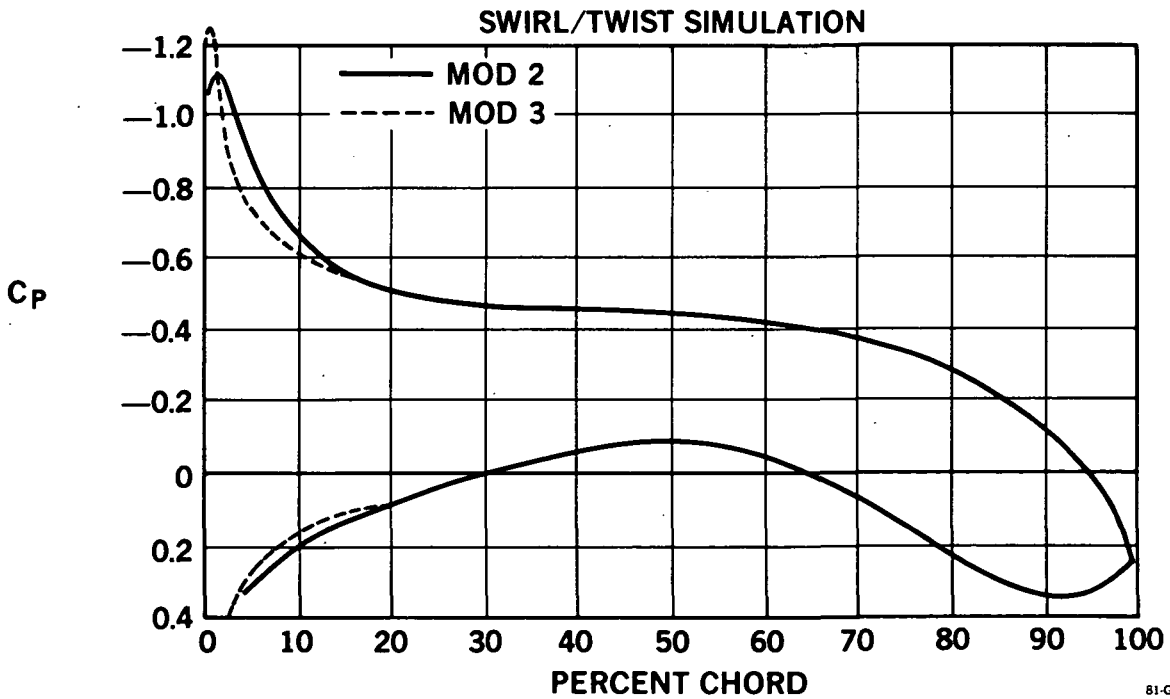


FIGURE 64. COMPARISON OF JAMESON CALCULATED CHORDWISE PRESSURE DISTRIBUTIONS FOR POWERED CONFIGURATIONS  $M_o = 0.800$



81 GEN-23187

FIGURE 65. SECTION LIFT CHANGE DUE TO WING MOD 2 AND MOD 3



81-GEN-23222

FIGURE 66. INCOMPRESSIBLE JAMESON AT  $\eta = 0.38$

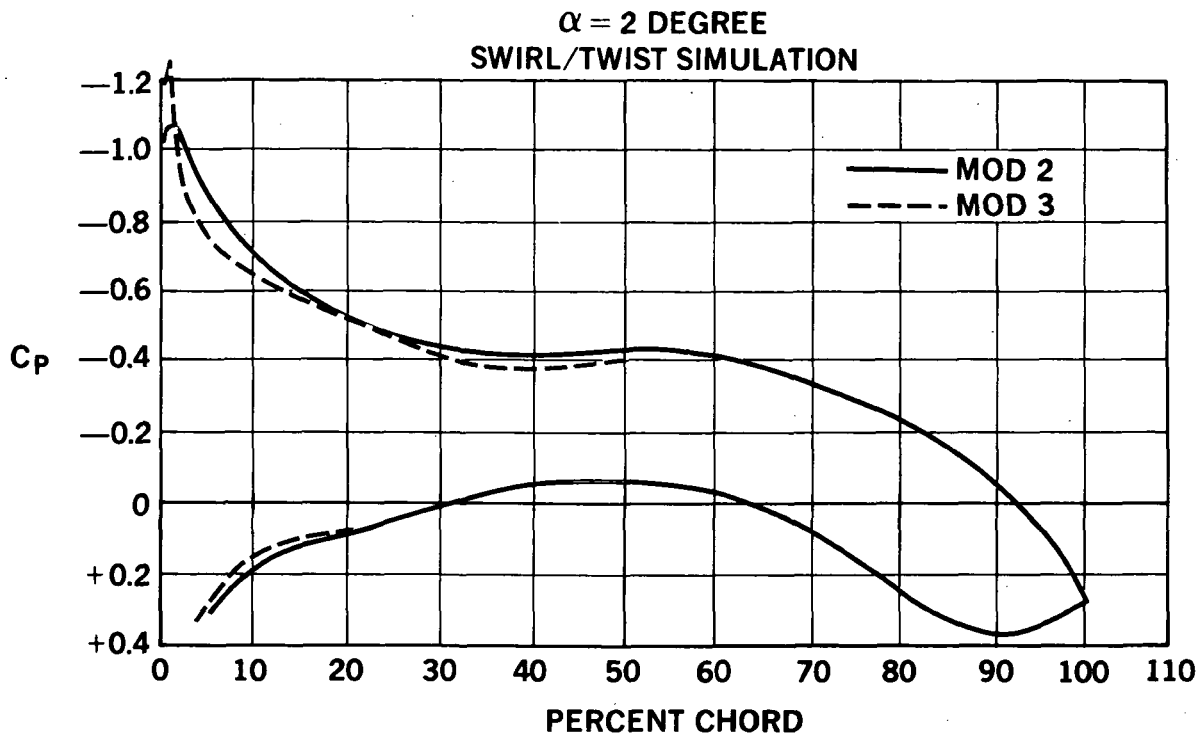


FIGURE 67. INCOMPRESSIBLE JAMESON AT  $\eta = 0.42$

81-GEN-23223

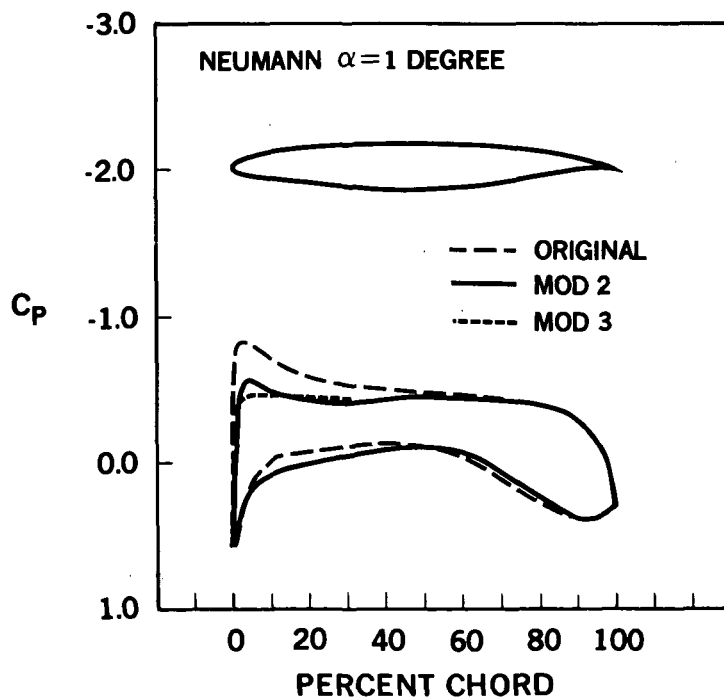
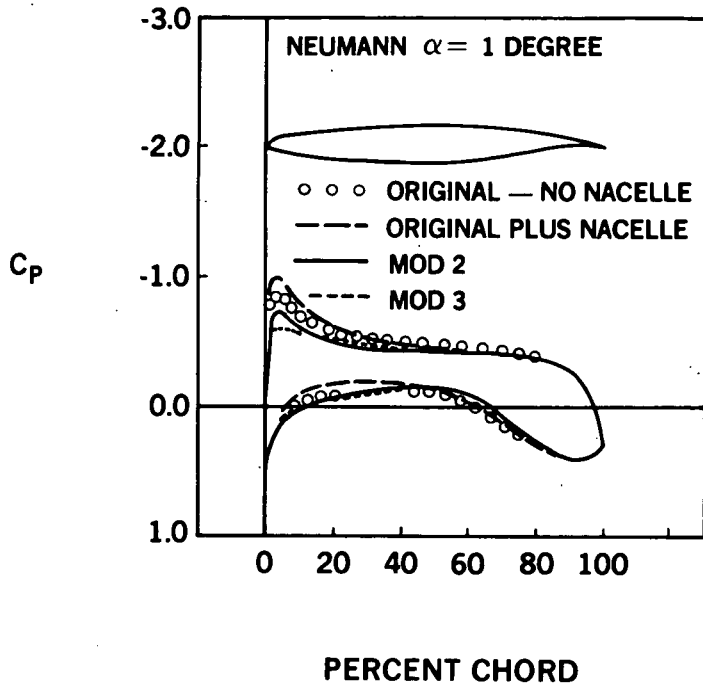


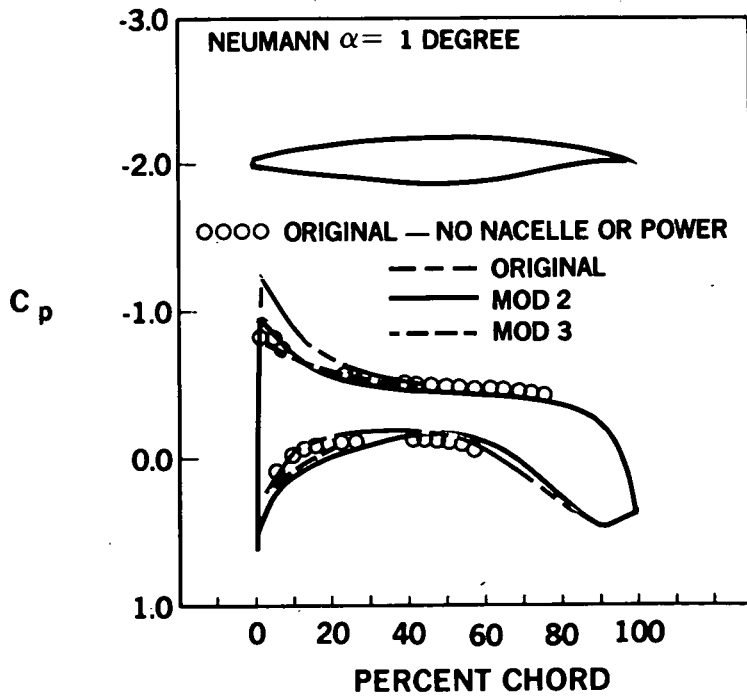
FIGURE 68. PRESSURE DISTRIBUTION FOR CLEAN WINGS AT  $\eta = 0.37$

81-GEN-23237



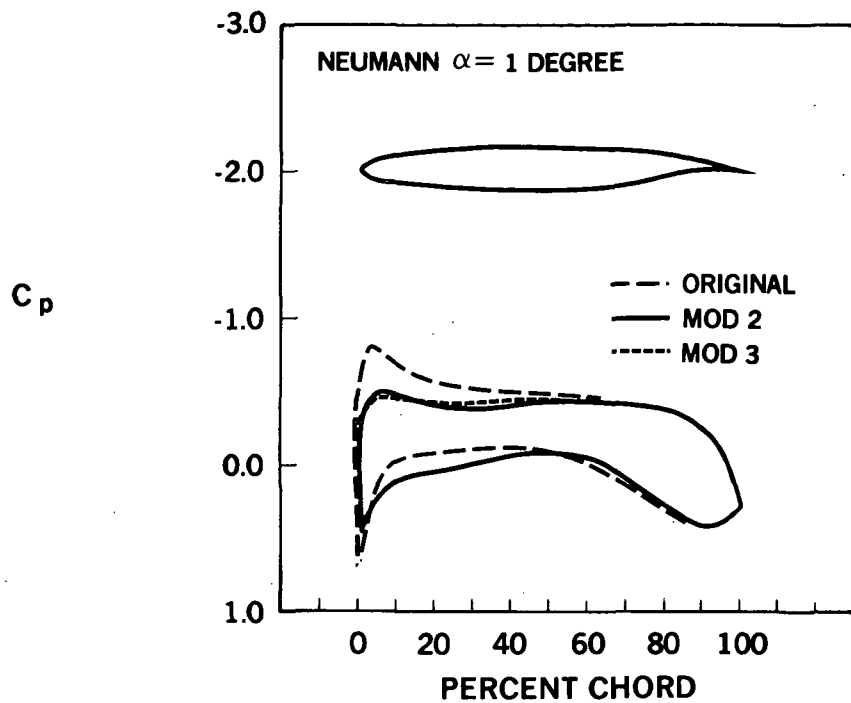
81-GEN-23236

FIGURE 69. PRESSURE DISTRIBUTION FOR WINGS PLUS NACELLE WITHOUT POWER AT  $\eta = 0.37$



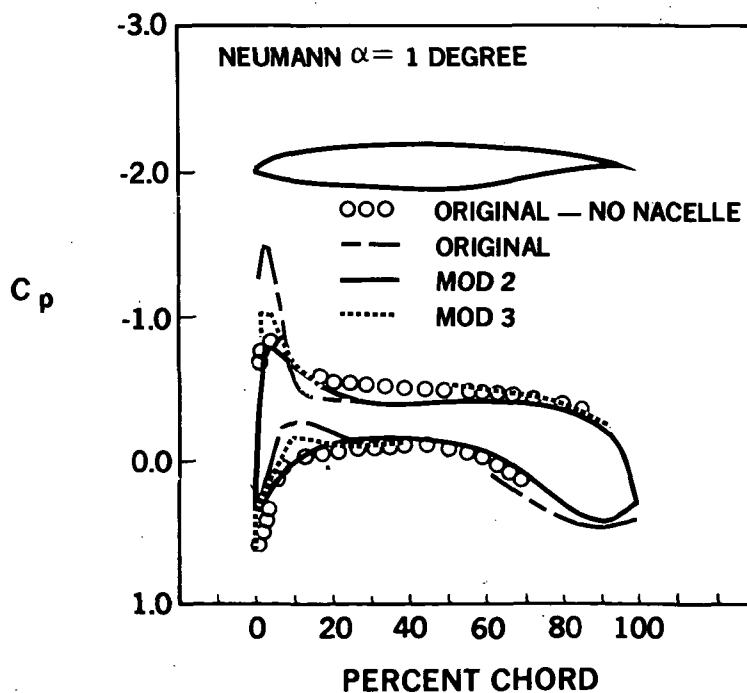
81-GEN-23235

FIGURE 70. PRESSURE DISTRIBUTION FOR WINGS WITH NACELLES AND POWER AT  $\eta = 0.37$



81-GEN-23234

FIGURE 71. PRESSURE DISTRIBUTION FOR CLEAN WINGS AT  $\eta = 0.41$



81-GEN-23233

FIGURE 72. PRESSURE DISTRIBUTION FOR WINGS PLUS NACELLE WITHOUT POWER AT  $\eta = 0.41$



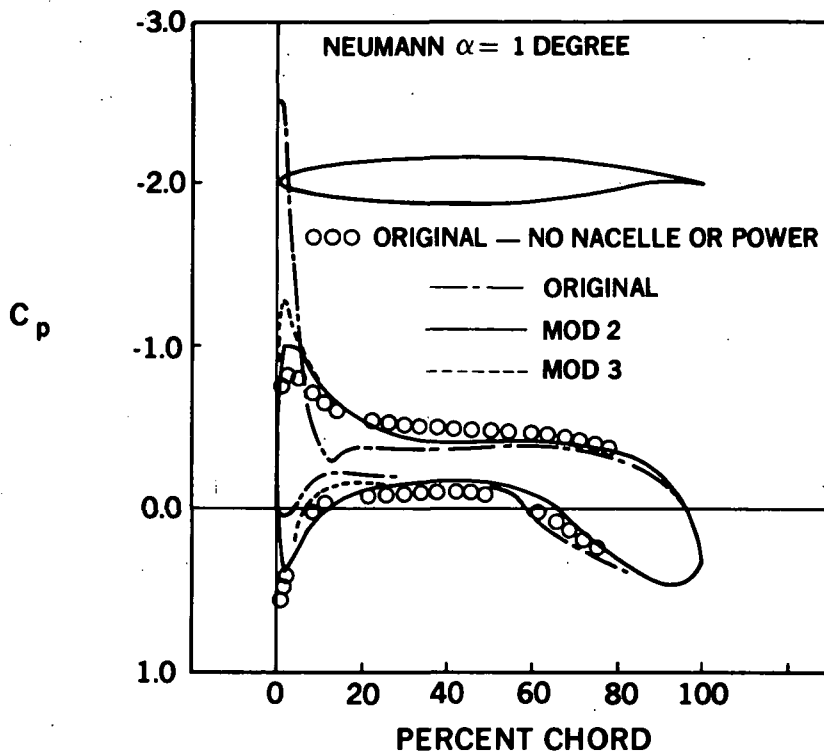


FIGURE 73. PRESSURE DISTRIBUTIONS FOR WINGS WITH NACELLES AND POWER AT  $\eta = 0.41$

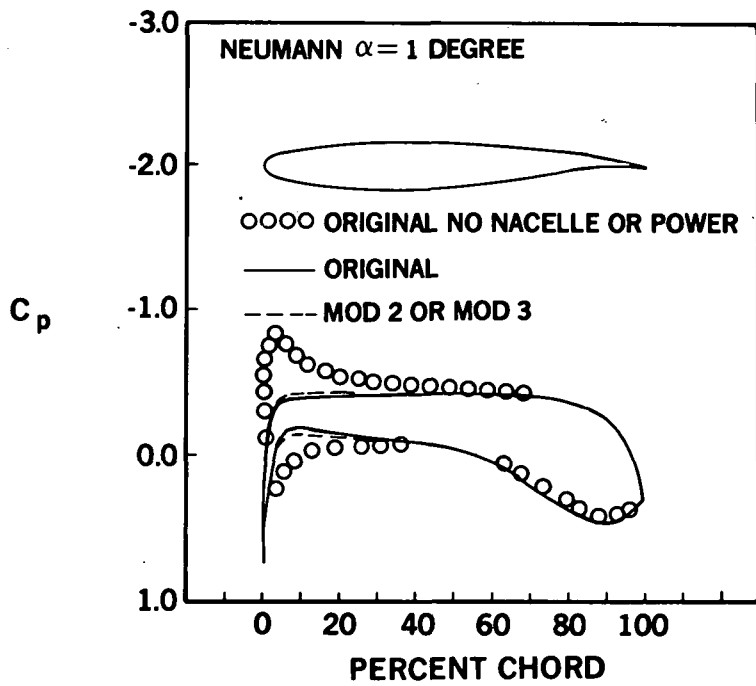
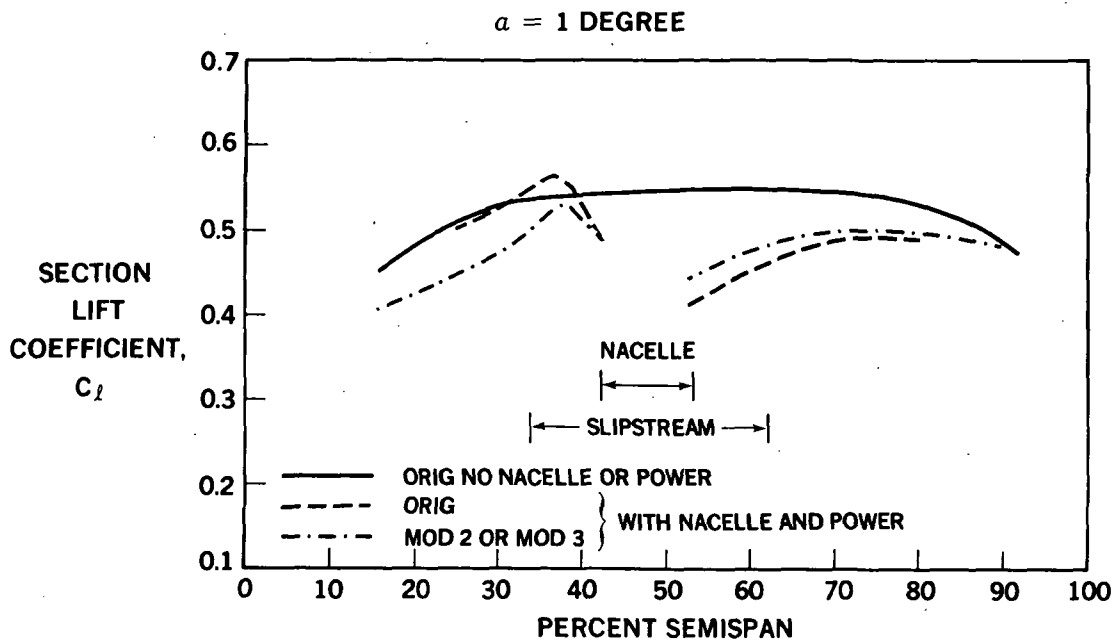
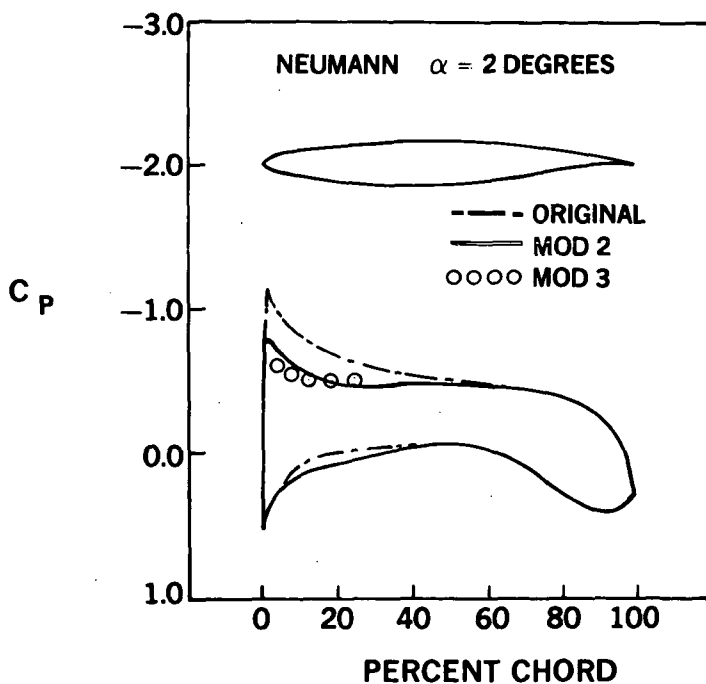


FIGURE 74. PRESSURE DISTRIBUTION OUTBOARD OF NACELLE, WITH NACELLE AND POWER  $\eta = 0.56$



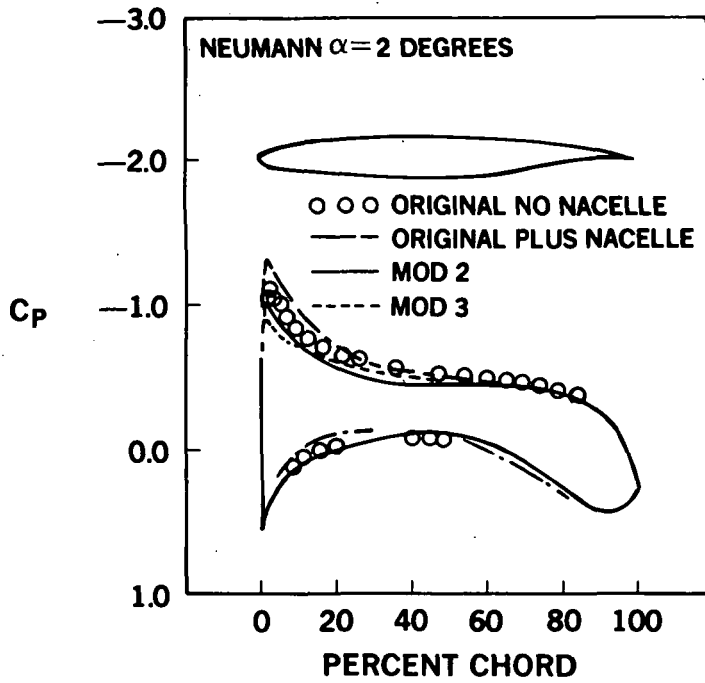
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FIGURE 75. COMPARISON OF SECTION LIFT DISTRIBUTION CALCULATED USING NEUMANN



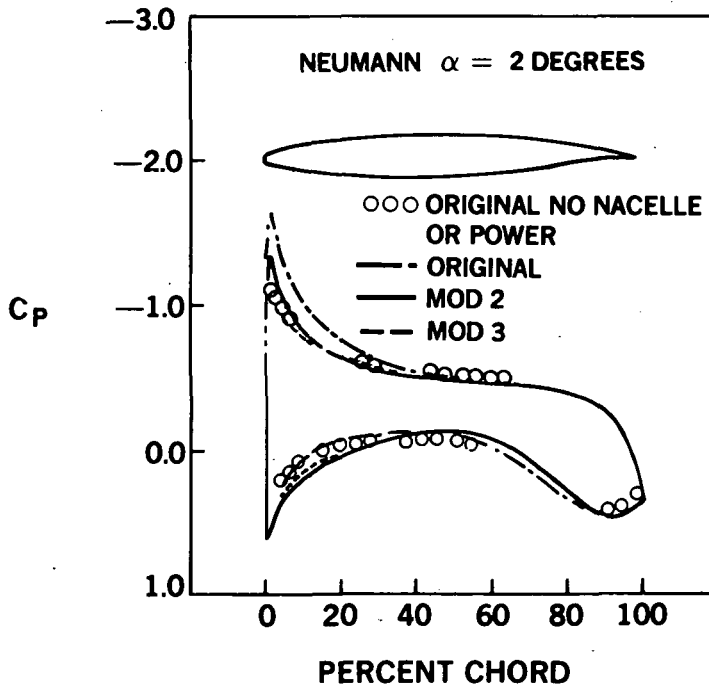
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FIGURE 76. PRESSURE DISTRIBUTION FOR CLEAN WING AT  $\eta = 0.37$  AND  $\alpha = 2$  DEGREES



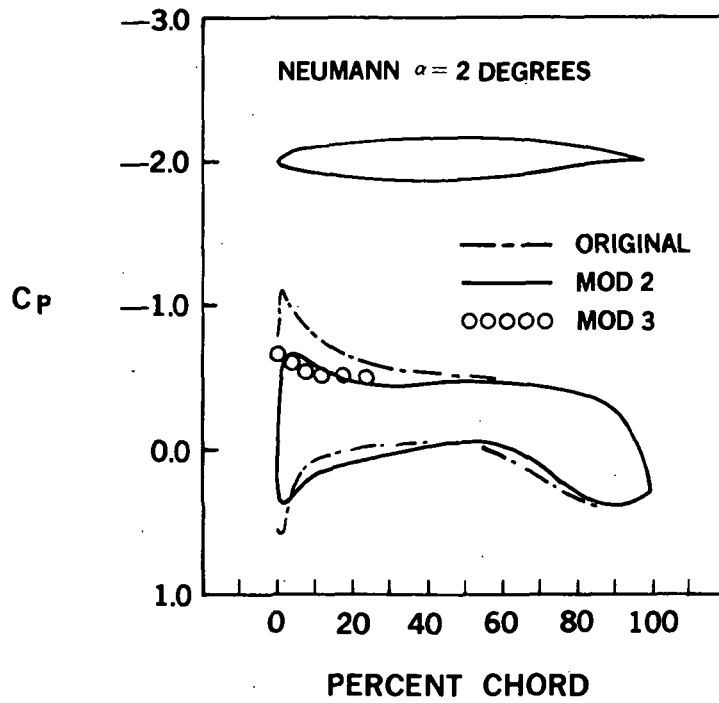
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FIGURE 77. PRESSURE DISTRIBUTION FOR WINGS PLUS NACELLE WITHOUT POWER  $\eta = 0.37$  AND  $\alpha = 2$  DEGREES



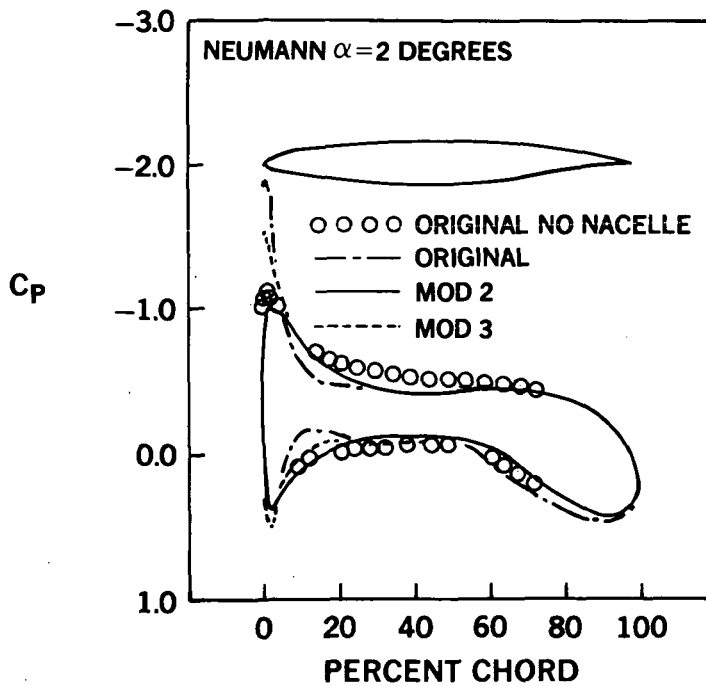
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FIGURE 78. PRESSURE DISTRIBUTION FOR WINGS WITH NACELLE AND POWER AT  $\eta = 0.37$  AND  $\alpha = 2$  DEGREES



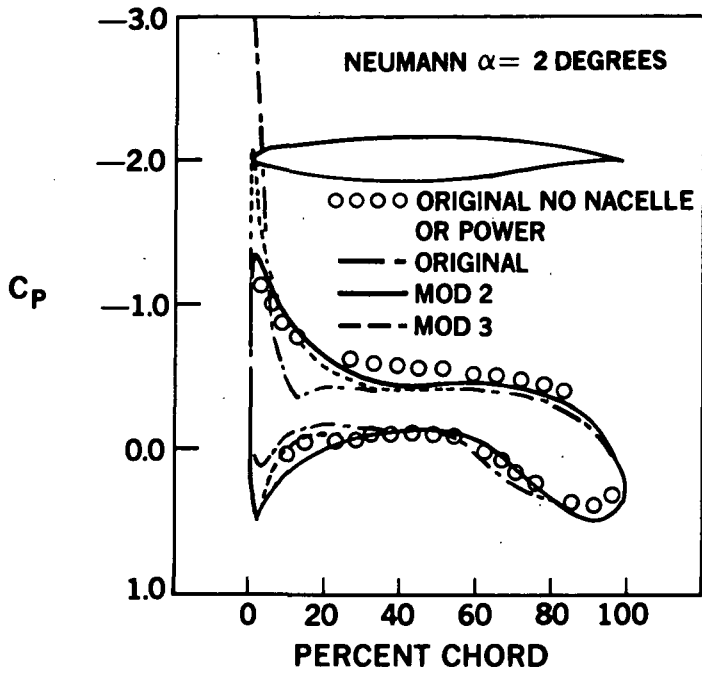
81-GEN-24181

FIGURE 79. PRESSURE DISTRIBUTION FOR CLEAN WINGS AT  $\eta = 0.41$  AND  $\alpha = 2$  DEGREES



81-GEN-24199

FIGURE 80. PRESSURE DISTRIBUTION FOR WINGS PLUS NACELLE WITHOUT POWER AT  $\eta = 0.41$  AND  $\alpha = 2$  DEGREES



81-GEN-24191

FIGURE 81. PRESSURE DISTRIBUTIONS FOR WINGS WITH NACELLES AND POWER AT  $\eta = 0.41$  AND  $\alpha = 2$  DEGREES

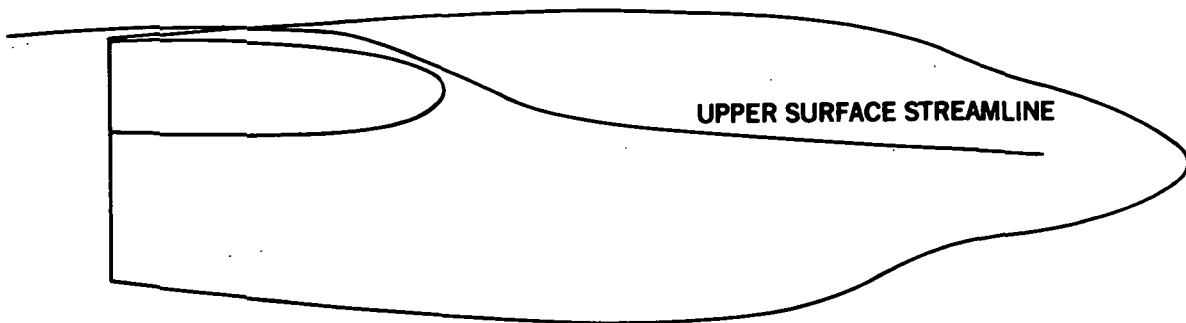
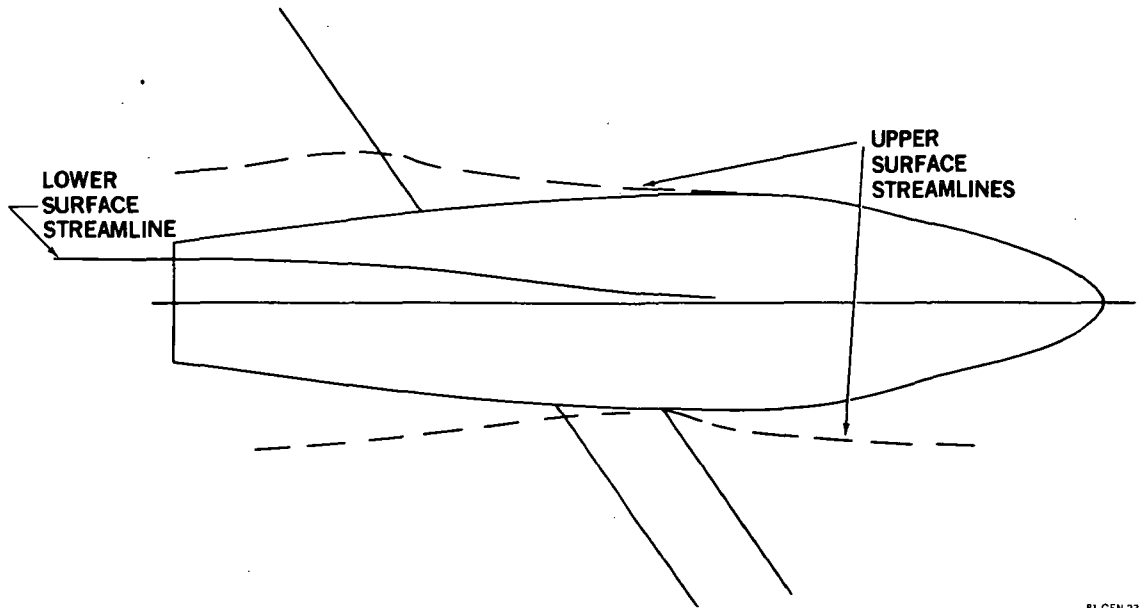


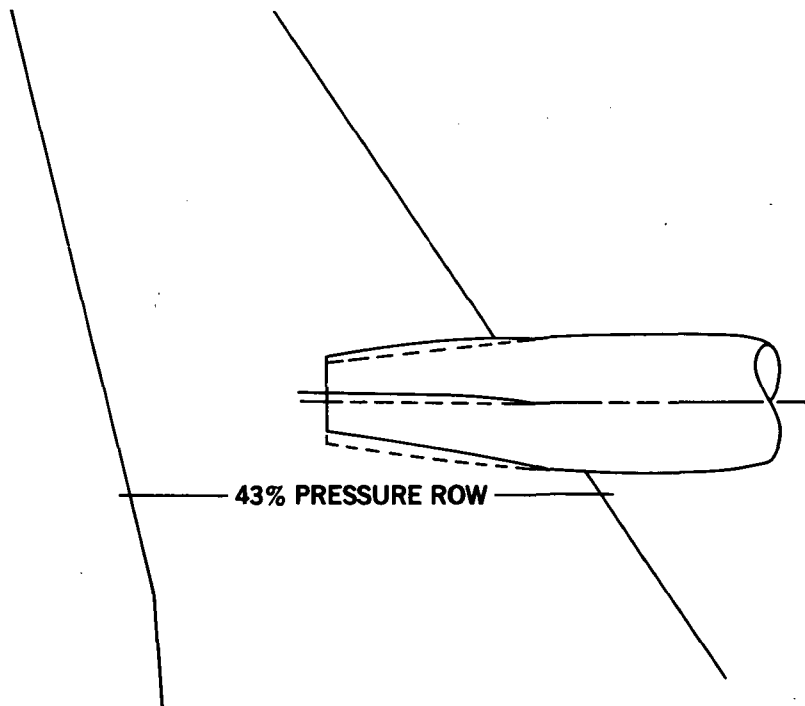
FIGURE 82. CLEAN WING STREAMLINE — PROFILE VIEW

81-GEN-23203



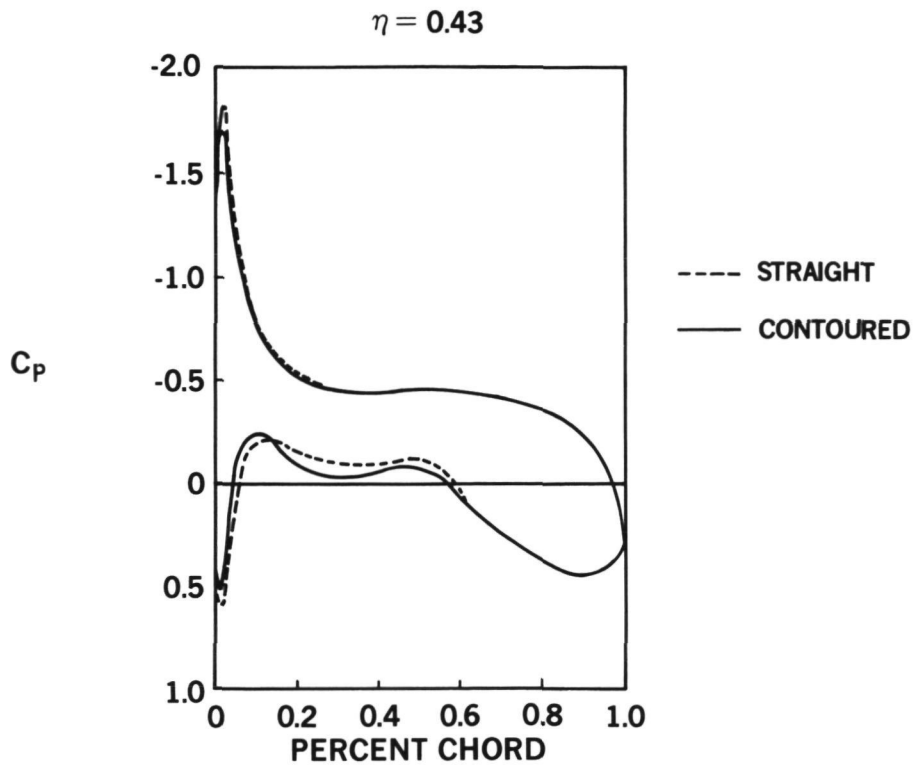
81-GEN-23242

FIGURE 83. CLEAN WING STREAMLINES – PLAN VIEW



81-GEN-23214

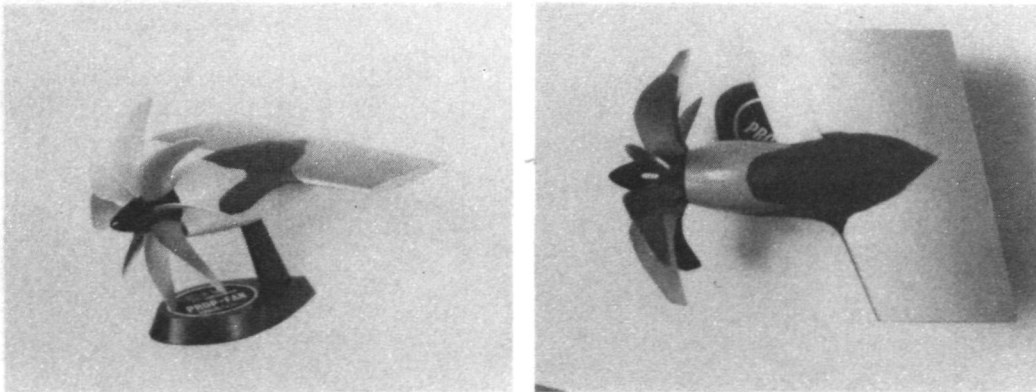
FIGURE 84. STREAMLINE CONTOURED LOWER SURFACE AFT NACELLE



81-GEN-23227

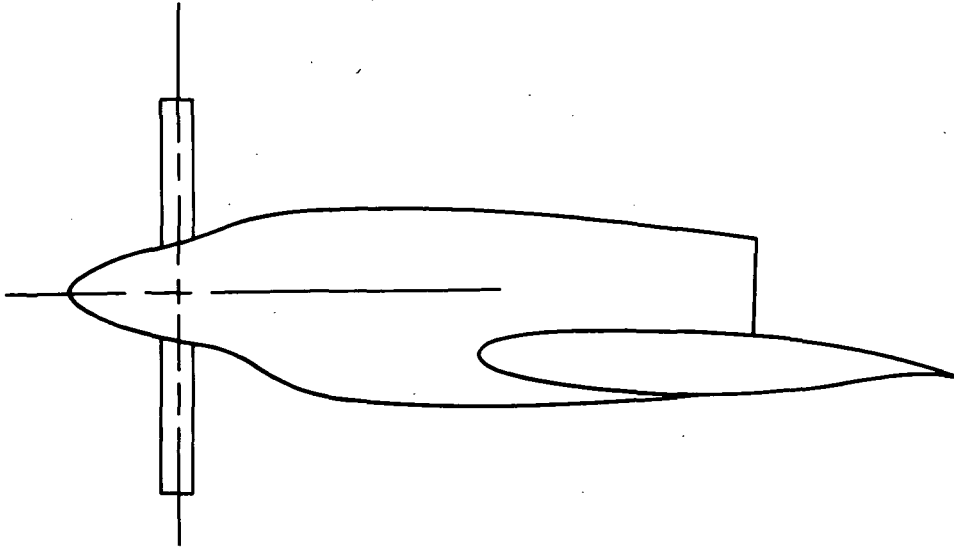
FIGURE 85. EFFECT OF CONTOURING AFT PORTION OF UNDERWING NACELLE

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OF POOR QUALITY



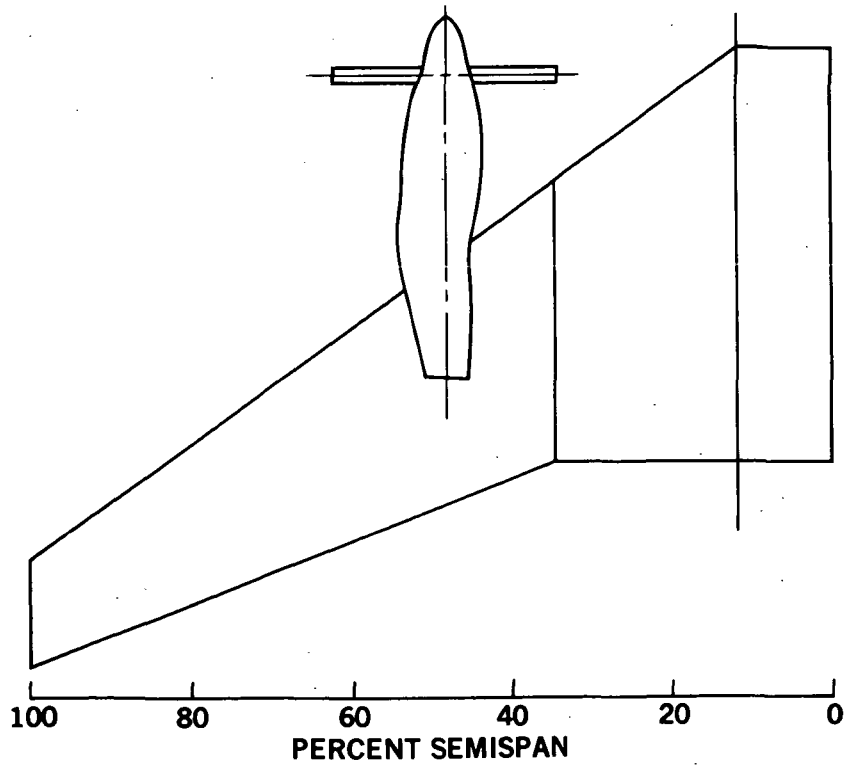
81-GEN-23211

FIGURE 86. SUBSCALE MOCKUP OF POTENTIAL UNDERWING NACELLE CONTOURING



81-GEN-24203

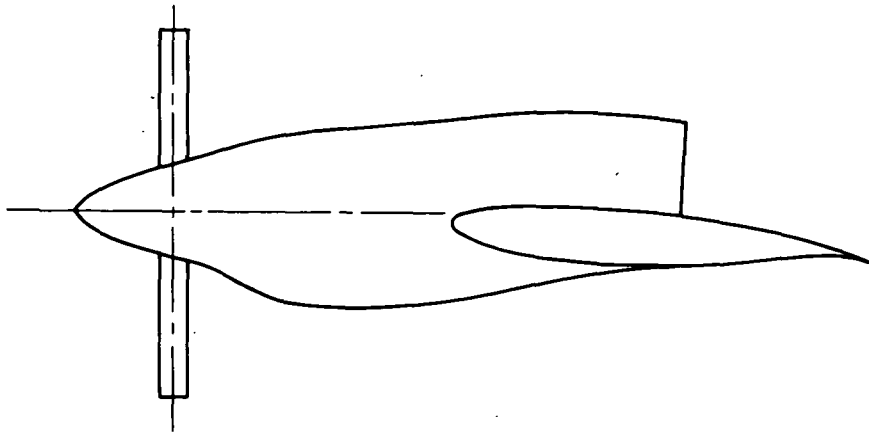
FIGURE 87. SIDE VIEW NONCONTOURED OVERWING NACELLE



81-GEN-23218

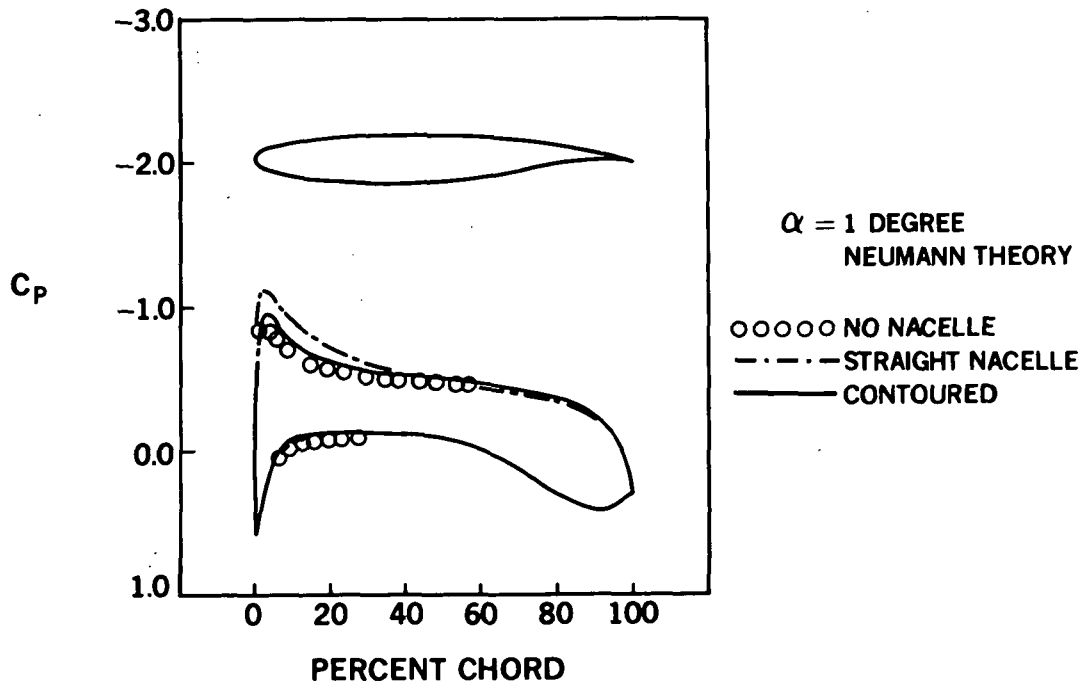
FIGURE 88. PLAN VIEW OF OVERWING CONTOURED NACELLE





81-GEN-23220

FIGURE 89. SIDE VIEW OF OVERWING CONTOURED NACELLE



81-GEN-24196

FIGURE 90. PRESSURE DISTRIBUTION COMPARISON FOR CONTOURED NACELLE AT  $\eta = 0.37$  PERCENT

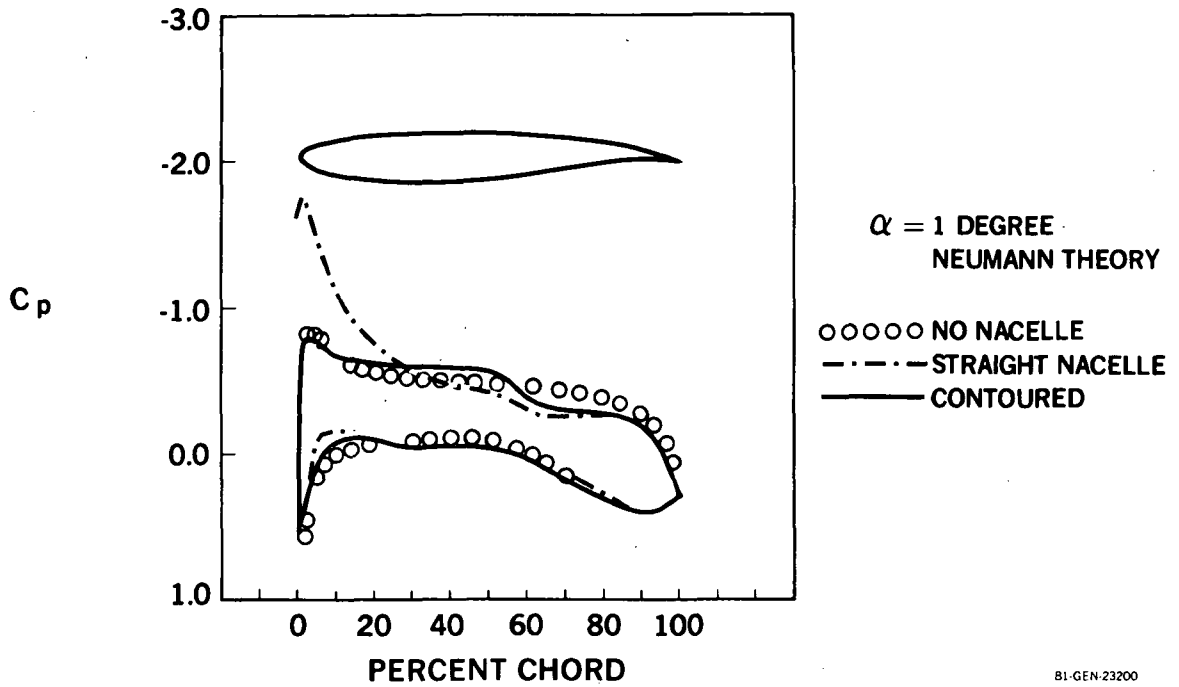


FIGURE 91. PRESSURE DISTRIBUTION COMPARISON FOR CONTOURED NACELLE AT  $\eta = 43$  PERCENT

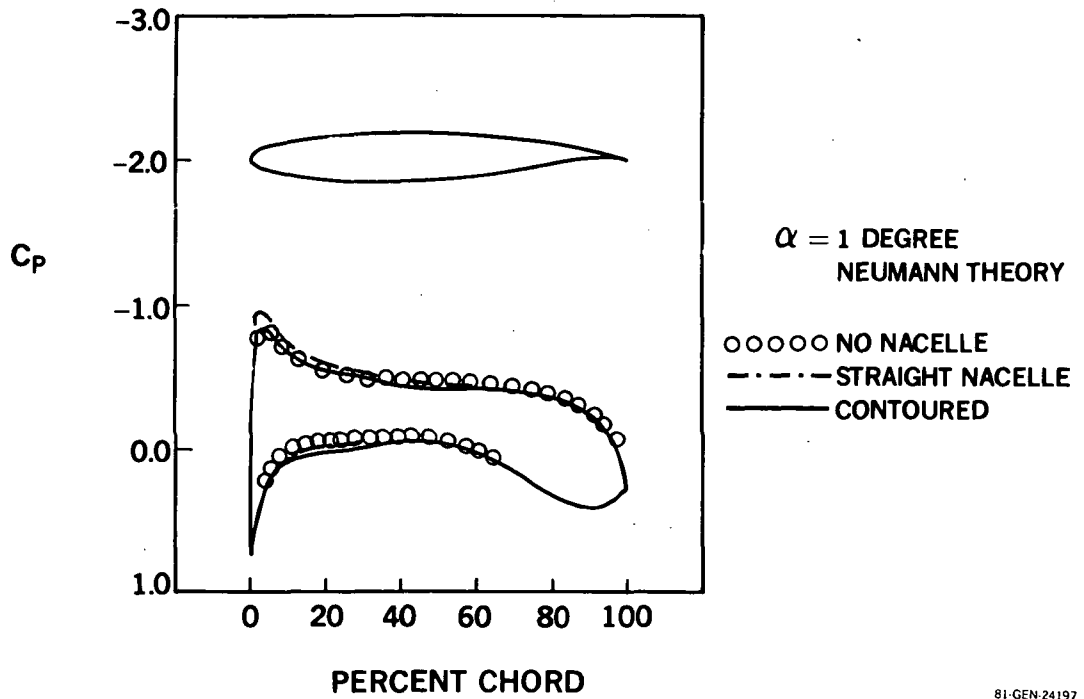


FIGURE 92. PRESSURE DISTRIBUTION FOR CONTOURED NACELLE AT  $\eta = 0.56$  PERCENT

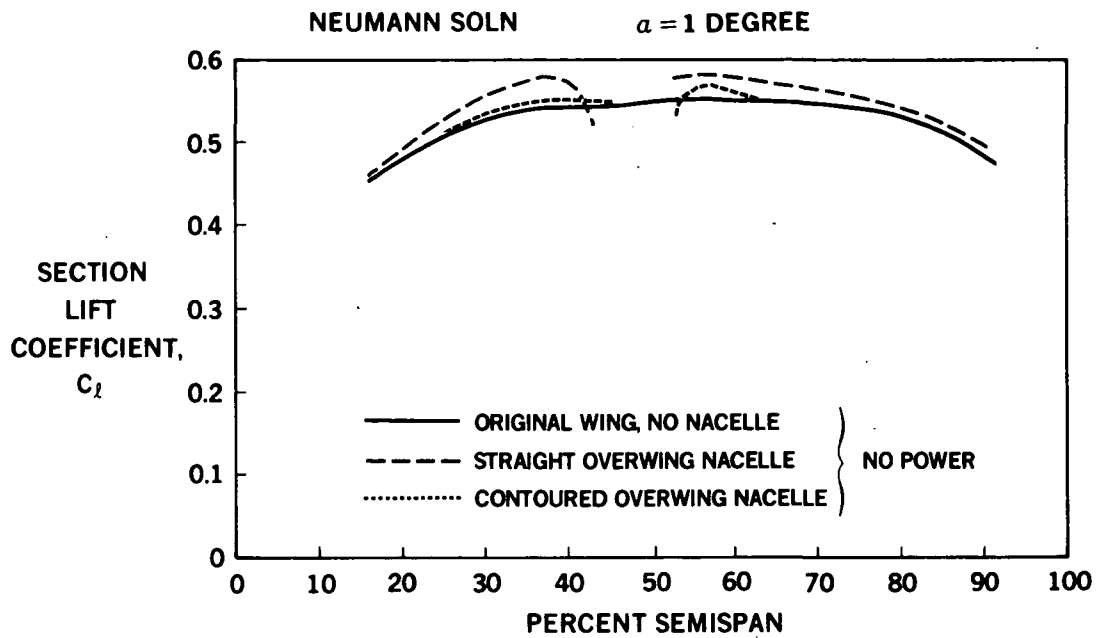


FIGURE 93. EFFECT OF NACELLE CONTOURING ON SECTION LIFT DISTRIBUTION

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15. Supplementary Notes Technical Monitor: Al Levin (227-2) NASA Research Center Moffett Field, CA 94035					
16. Abstract An experimental test program of a powered propeller and nacelle mounted on a supercritical wing was conducted by the NASA Ames Research Center in the 14-foot tunnel. Analysis of this data by the Douglas Aircraft Company, under contract to NASA with Al Lavin as the program manager, is contained in this report. The design condition for this study was $M_0 = 0.8$ .  Analysis of the data indicated that the installation of the nacelle significantly affected the wing flow and that the flow on the upper surface of the wing is separated near the leading edge under powered conditions. Comparisons of various theories with the data indicated that the Neumann surface panel solution and the Jameson transonic solution gave results adequate for design purposes. A modified wing design was developed (Mod 3) which reduces the wing upper surface pressure coefficients and section lift coefficients at powered conditions to levels below those of the original wing without nacelle or power. A contoured over-the-wing nacelle is described that can be installed on the original wing without any appreciable interference to the wing upper surface pressures.					
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