(NASA-CR-169927) COMPUTATIONS AND TURBULENT FLOW MODELING IN SUPPORT OF HELICOPTER ROTOR TECHNOLOGY Progress Report, 1 Jun. - 30 Nov. 1982 (Nevada Univ.) 17 p HC A02/MF A01 CSCL 01A G3/02

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PROGRESS REPORT

JUNE 1, 1982 THROUGH NOVEMBER 30, 1982

NASA GRANT NO. NSG 2291

COMPUTATIONS AND TURBULENT FLOW MODELING IN SUPPORT OF HELICOPTER ROTOR TECHNOLOGY

Prepared for:

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PROGRESS REPORT

JUNE 1, 1982 THROUGH NOVEMBER 30, 1982

BACKGROUND

Continuing efforts on the grant were expended to investigate the feasibility of using Deiwert's time-dependent numerical simulation code to calculate two-dimensional airfoil flows similar to those expected to occur on helicopter rotors. Rather than attempting to calculate the actual helicopter oscillating, unsteady flow, critical components thought to be important to the overall flow are being studied individually. Since the helicopter rotor problem involves airfoil sections at large local angles of attack, both time accuracy and adequate turbulence models are required.

Efforts during the present reporting period continued to investigate the applicability of the Deiwert's time-dependent numerical airfoil calculation code to the simulation of two-dimensional airfoil flows with large amounts of separation present. In the previous Progress Report for the period December 1, 1981 through May 31, 1982, it was shown that at angles of attack of 12.5 to 17.5 deg at a Mach number of 0.4, non-physical behavior of a flow was observed. This was a manifest in a nearly periodic shedding of the entire flow field. In that report, a variation in the wall region turbulence modeling was proposed. The usual wall layer is modeled with the Karman constant as:

$$\ell = ky \tag{1}$$

A model which has a decreasing-wall influence in separations was proposed as equation 2.

$$\ell = ky[10 \log (100 (y_{sep}-y_s)/c)] \qquad (2)$$

The net effect of Equation 2 is to reduce the Karman constant whenever the height of the separation bubble relative to the chord length becomes significant. If, for example, the separation height is approximately 10% of the chord, the Karman constant is increased by a factor of 10. The calculations carried out in the previous progress period had indicated a rather substantial change in the calculated flow properties, particularly over the aft portion of the airfoil. Extensive separation was calculated and velocity profiles similar to those observed experimentally were predicted.

Modification of the Karman constant as proposed during last period's reporting has been one of the uncountable number of modifications to turbulence models which might be proposed. For purposes of discussion here, the boundary layer on an airfoil undergoing a severe adverse pressure gradient leading to separation can be divided into three principle regions. First the outmost layer, or so-called wake region, in which the eddy viscosity or mixing length is held approximately constant. The logarithmic region of the velocity profile, which is described usually by Equation 1, which decreases the mixing length or eddy viscosity to near zero at the wall, and finally an inner-wall region which is usually described by a Van Driest damping term in order to account for the existence of a molecular sub-layer. Clearly the modification discussed in last period's progress report is a modification to the logarithmic region of the boundary layer. It was shown there that minor modifications to this region of the flow produce extensive changes in the calculated flow-field behavior. With respect to the outer region, extensive experiments in the NASA 2'x2' tunnel in making direct measurements of a Reynolds shear stress term with laser velocimetry has indicated that a very good approximation to the outer layer behavior is given by the mixing length term, $.09\delta$. This is true even for flows having just undergone abrupt adverse pressure gradients resulting from shock waves (including separated flow). Thus, the suspected non-equalibrium nature of the outer portion of the flow does not seem to be verified from experiment and that the disagreement between the calculated flow fields and experimental flow fields cannot be blamed on outer layer Reynolds shear stress behavior alone. The modified logarithmic region mixing length term, as noted above, did produce substantial differences in the calculated flow fields for the large separation on the M=0.4, high angle of attack cases.

In this report, that same model has been applied to the previously investigated cases of the 64A010 airfoil section at Mach number 0.8 for angles of attack of 4 deg and 6.2 deg. The interest in these two cases lies in the following:

Although both flow fields contain some separated flow, the flow at the 6.2 deg angle of attack does have a substantial region of backflow, experimentally, and has not been properly calculated in the past using the Escudier model throughout the course of the present study. The 4 deg case contains only a minor amount of separation and it is not expected that the alteration of a logarithmic profile in this particular case should have a substantial influence on the calculated solution. The interest in the 4 deg case at this point lies in the unsteady nature of the solutions obtained using Steger's code by King, Degani and Chyu and not obtained previously with the Deiwert code for the 4 deg case. Effects of these turbulence model alterations and the time dependent behavior from impulsively started conditions are discussed in the present report.

RESULTS AND DISCUSSION

In order to illustrate the behavior of the 64A010 airfoil section at 4 deg angle of attack and Mach number 0.8, the first portion of this study was carried out. Figure 1 shows the upper and lower surface airfoil pressure distributions obtained from the Deiwert code with the Escudier turbulence model for the 4 deg angle of attack condition. Free-air boundary conditions are used throughout this report even though the free air assumption is known to be grossly inadequate in the NASA Ames 2'x2' wind tunnel at M=0.8. These boundary conditions are uniformly duplicable in all of the analytical techniques and are thus chosen here for illustration. The airfoil pressure distributions shown in Figure 1 were obtained after approximately 25 chords of calculation from an impulsive start to the Mach 0.8 condition. Reynolds numbers in this report are 2.0×10^6 based on chord length unless otherwise noted specifically in the discussion or on the figures. The mean average lift coefficient for the 4 deg case calculated from the present theory has a value of approximately 0.72. The time evolution of this lift coefficient is shown in Figure 2 where a monotonic progression from 0 to a final, constant value of approximately 0.72 is seen. Over approximately the last 10 chords of calculation time, the lift coefficient is steady and indicates no trend whatsoever toward the periodic behavior as obtained by others using the Steger code. The effect of changing from the Escudier model to the wall model of Equation 2 is shown in the upper and lower surface pressure distributions at 4 deg and Mach 0.8 in Figure 3. Because only a small amount of separation is present in the calculation for this case, no large differences are indicated between these two turbulence models. Again, the progression from the impulsive start to the condition shown at approximately 25 chords of flow exhibited the exact behavior as that shown for the Escudier model in Figure 2. Unsteady periodic behavior has also been observed in other calculations published recently in the April 1982 AIAA Journal by Sugavanam and Wu. The fact that the Deiwert code does not exhibit this periodic behavior for this type of solution at the lower angles of attack may be due to the use of the MacCormack algorithm as opposed to the Steger algorithm for the solution of the equations. Additional numerical damping inherent in the MacCormack algorithm may be one possible explanation for the lack of calculated periodicity in the flow fields, although this point remains speculative.

In order to investigate the effect of changing turbulence models for a case in which extensive separation is expected to occur, the following series of runs was performed for the 64A010 section at a Mach number of 0.8, again for the free air boundary conditions, at an angle of attack of 6.2 deg. Figure 4 shows a comparison between the final steady upper and lower surface pressure distributions obtained for this flow using the Escudier and modified wall models. The steady lift coefficient for the

Escudier model is approximately 0.78 while that calculated from the modified wall model is approximately 0.94. Unfortunately, the modified wall model predicts a flow that remains attached further than that of the Escudier model. This is opposite to the behavior expected based on the calculations performed during the last reporting period on the 0012 section at M=0.4, for which a much more extensive region of separated flow was calculated with the modified wall turbulence model.

The time history of the lift coefficients for successive changes from Escudier to the wall model are shown in Figure 5. Approximately 25 chords of flow from the impulsive start were calculated with the modified wall model and the turbulence model was changed instantaneously to the Escudier model. A progression of lift coefficient is seen to occur monotonically decreasing from approximately 0.94 value to approximately 0.78 at about 48 chords of flow, at which point the modified wall model was reinstated and a monotonic progression to the final value of approximately 0.96 is shown. No indication of unsteady behavior was seen for this set of calculations.

The inability of the newly proposed wall model to accurately predict the amount of separation or the increased amount of separation for the 6.2 deg case at Mach 0.8 is disheartening. This failure leads us to look even lower into the boundary layer at the near wall model. Runs have not been made changing the value of A+ because of the inappropriately long computation times associated with the Navier-Stokes code. Perhaps the best arena for discussing this type of turbulence wall modification is in the inverse boundary layer code where numerous runs can be achieved in order to investigate parametric variations of varying near wall modeling changes. It should be borne in mind when discussing turbulence modeling changes such as those above that in the thin boundary layers on airfoils, measurements of the Reynolds shear stress terms will be difficult. Measurements using today's known instrumentation technology within the sub-layer portion of the boundary layer will be impossible due to the large size of the instrumentation volumes in comparison with the scales of the motions to be investigated. This leads us to consider experiments involving extremely thick boundary layers undergoing adverse pressure gradients in order to allow state-of-the-art instrumentation to be brought to bear on turbulence modeling in the near wall region. Without proper data in the near wall region, turbulence modeling efforts will progress very slowly. Attention should be given to the possibility of making measurements in the boundary layers of the very large scale facilities including, but not limited to, the NASA Ames 6'x6' Supersonic Wind Tunnel, the 11'x11' Transonic Tunnel, and perhaps the 14' Transonic Tunnel.

One final calculation was made during the present reporting period in order to add credence to the Navier-Stokes solution obtained by King using the Steger code for the case of the

64A010 at M=0.72 and 3.5 deg angle of attack. This is a test case that is being considered by King and Murphy for the coupled TAIR and inverse boundary layer codes. The solution shown in Figure 7 has a time variation in the upper surface pressure distribution as shown at the various computation times. It is also of passing note that most of these solutions represent shock-free upper surface pressure distributions. They are generally found to be in agreement with those solutions given by the Steger code and in substantial disagreement with those given by the coupled code.

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AIRFOIL PRESSURE DISTRIBUTION

SECTION 64A010 ALPHA 4.0 M 0.8 BC FA TURB MODEL Escudier

$C_l \approx 0.72$ time ≈ 25

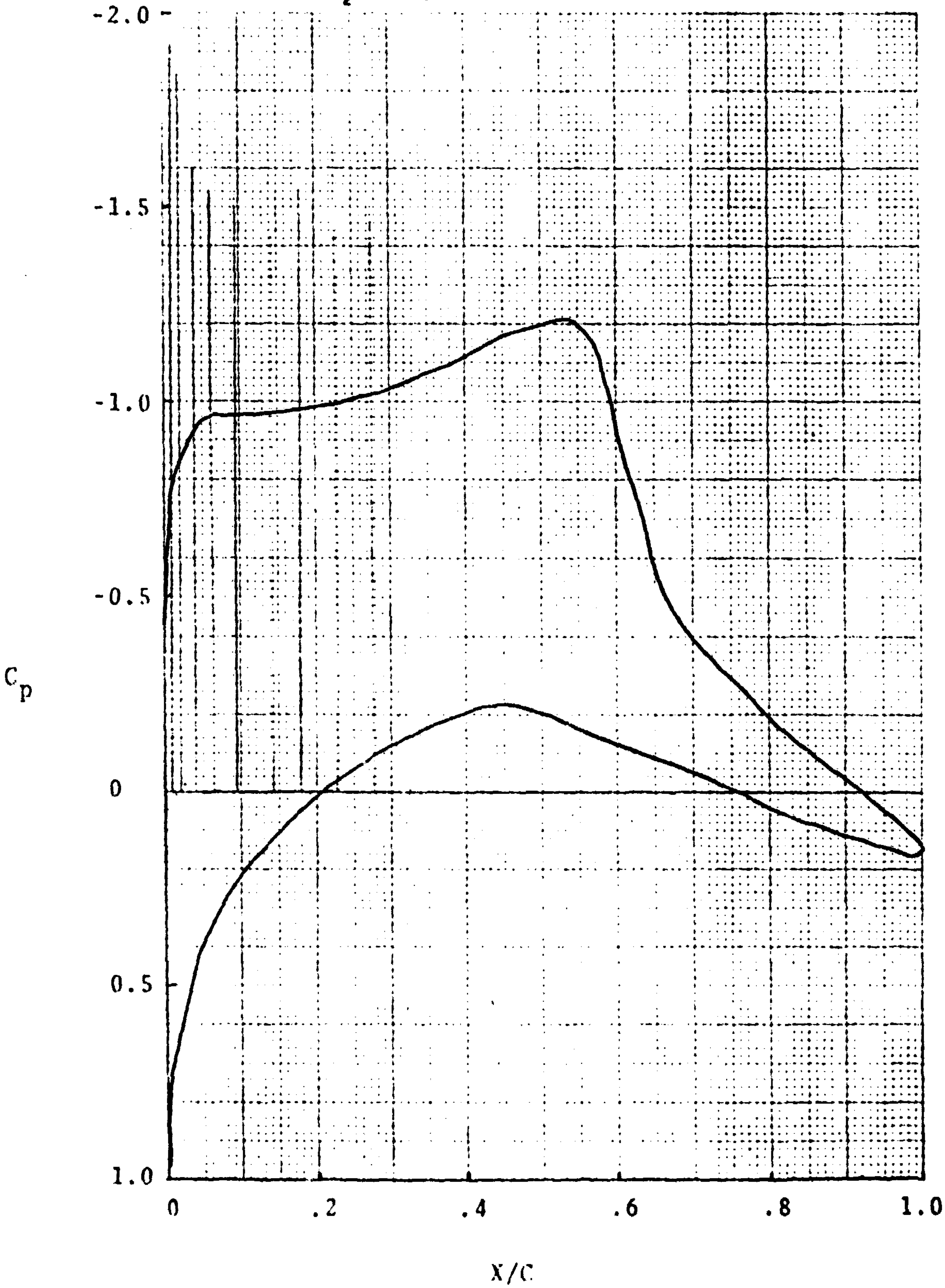

Figure 1. Calculated pressure distribution for the NACA 64A010 at M=0.8 and α 4.0 deg; Escudier model.

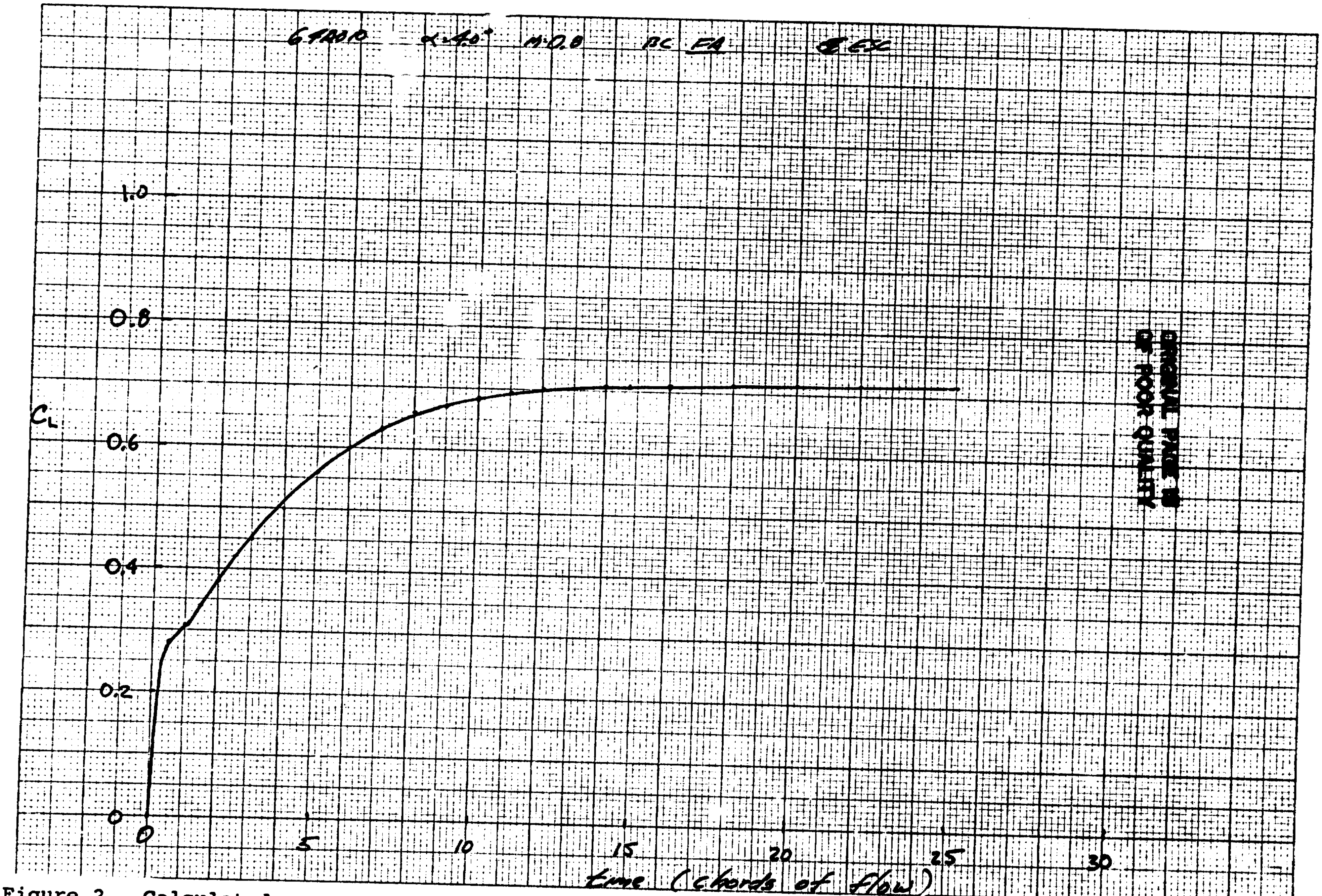

Figure 2. Calculated progression of section lift coefficient from impulsive started flow for the NACA 64A010 at M=0.8 and α=4.0 deg; Escudier model.

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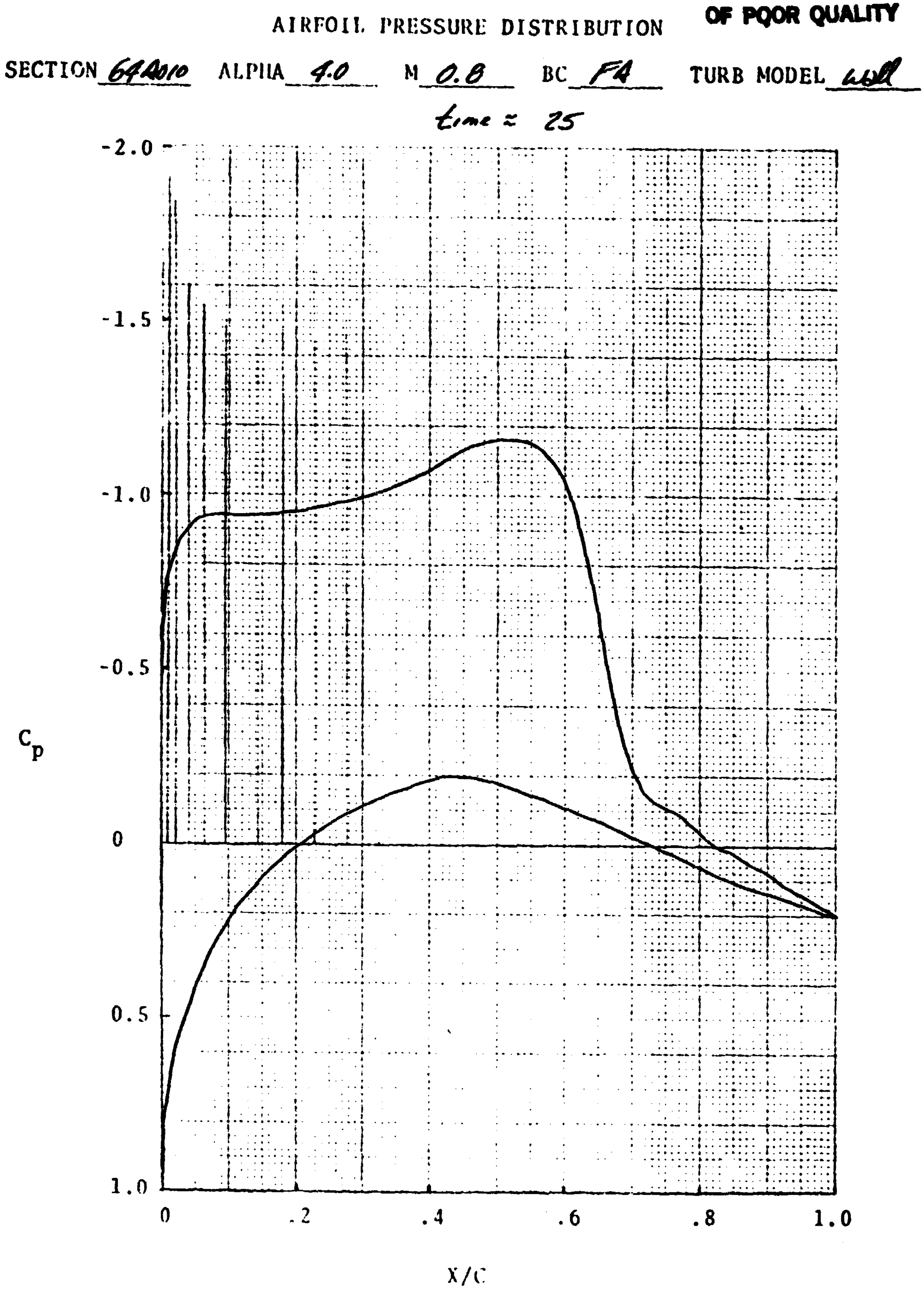

Figure 3. Calculated pressure distribution for the NACA 64A010 at M-0.8 and α=4.0 deg; modified wall model.

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AIRFOIL PRESSURE DISTRIBUTION

SECTION 64A010 ALPHA 6.2 M 0.8 BC FA TURB MODEL 3s noded

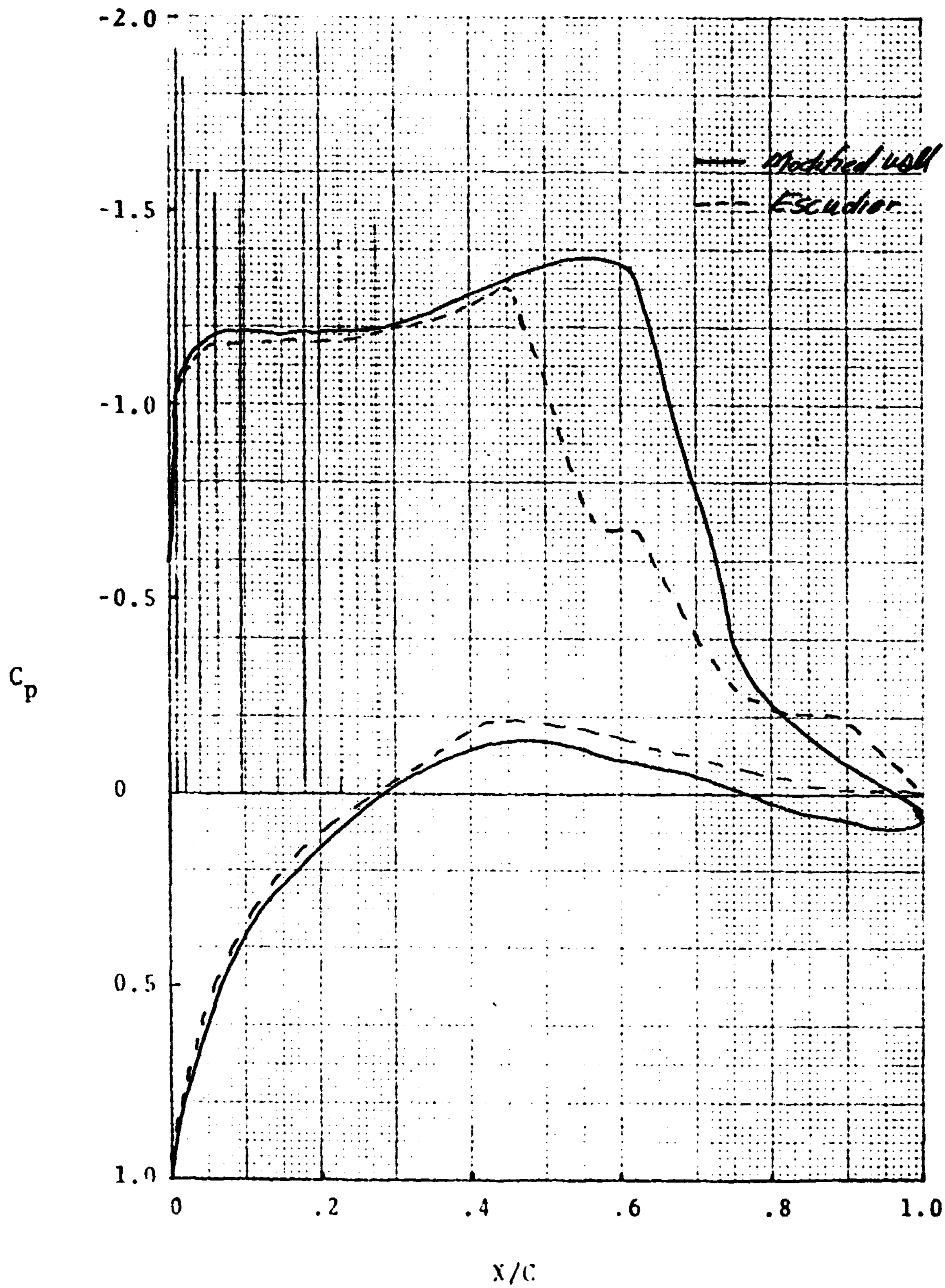

Figure 4. Comparison of calculated pressure distributions using different wall models for the NACA 64A010 at M=0.8 and α=6.2 deg.

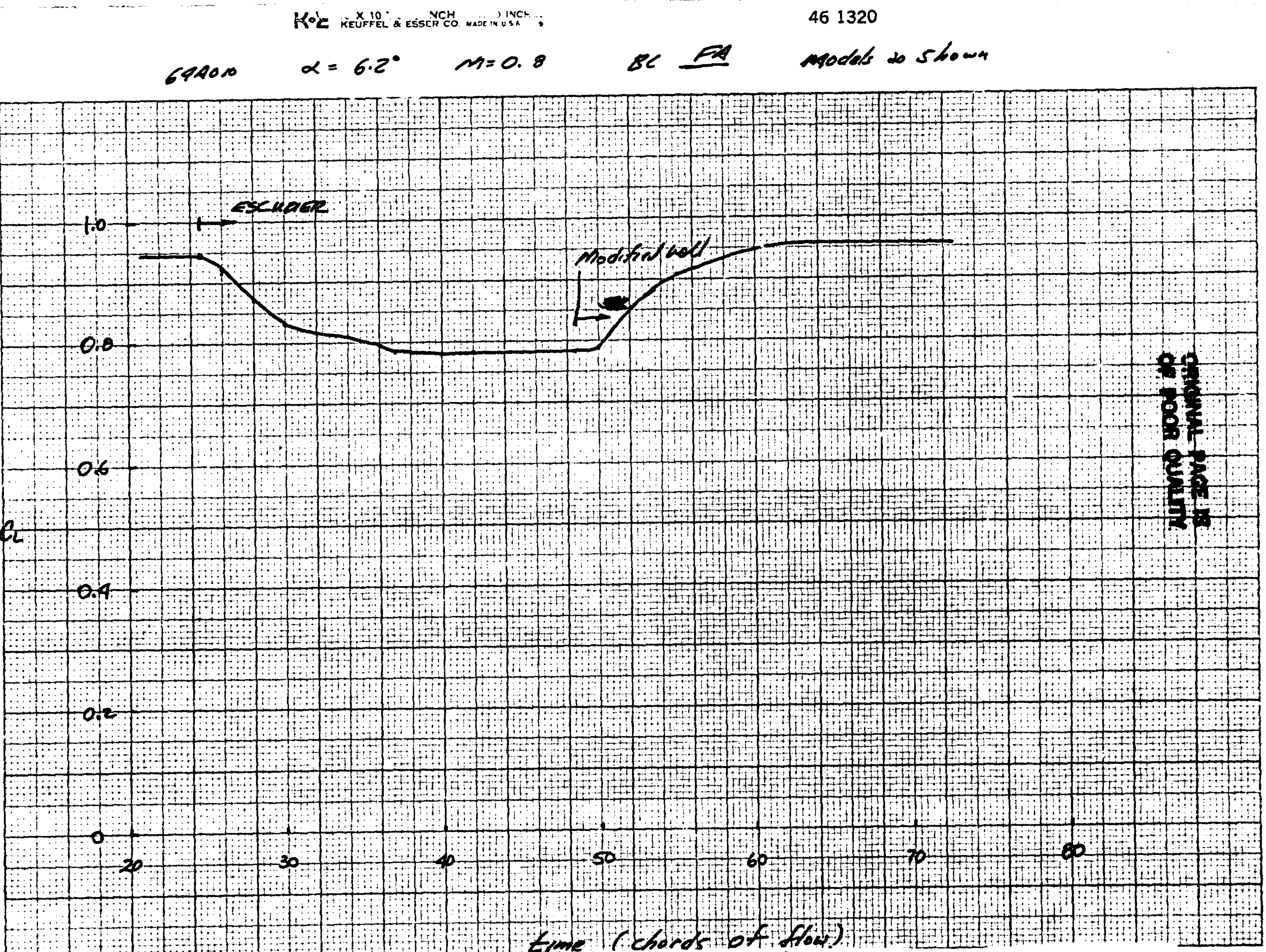

Figure 5. Calculated progression of section lift coefficient with two turbulence models for the NACA 64A010 at M=0.8 and α=6.2 deg.

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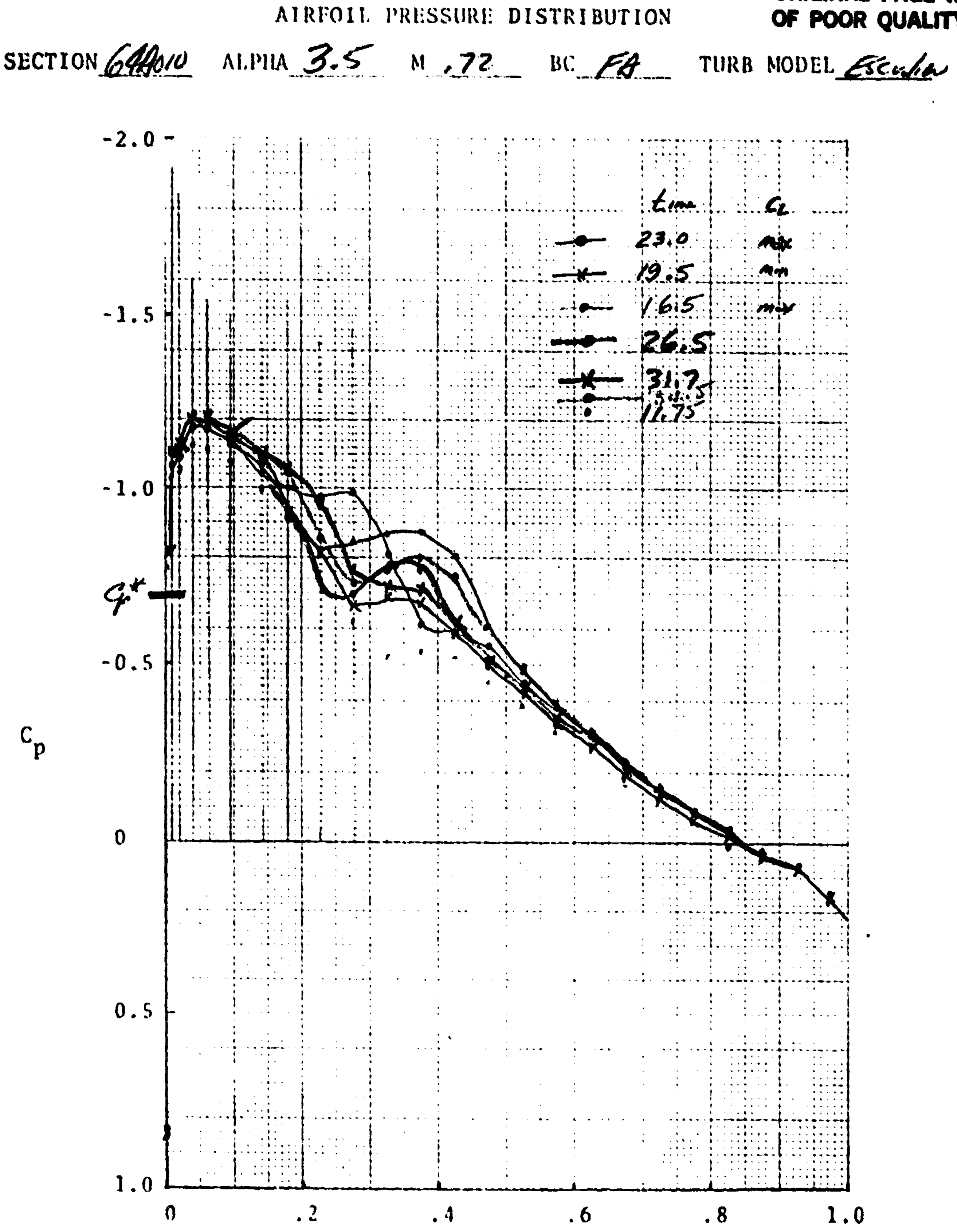

Figure 6. Time variation of calculated upper surface pressure distribution for the NACA 64A010 at M=0.72 and α=3.5 deg, Escudier model.