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Computation of Two-Dimensional Turbulent Flow at Subsonic Mach Numbers Over Thick Trailing Edges

Pius Drescher

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represented in the conventional six degree of freedom, quasistatic
stability derivative model. The amended model was a substantial
improvement over the conventional model, effectively doubling the unsable

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ABS: An implicit time marching finite difference method is used to predict two dimensional turbulent flow at a Reynolds number of 440,000 and a Mach number of 0.574 over a shortened NACA 0012 airfoil with a trailing edge of 4.5% thickness and semicircular shape. The flow is found to be unsteady but periodic in the trailing edge region. Thus, lift and drag fluctuate at small amplitudes around mean values and at distinct frequencies.

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COMPUTATION OF TWO-DIMENSIONAL TURBULENT FLOW AT SUBSONIC MACH NUMBERS OVER THICK TRAILING EDGES

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SUMMARY

An implicit time-marching finite difference method is used to predict two-dimensional turbulent flow at a Reynolds number of 4.4×10^5 and a Mach number of 0.574 over a shortened NACA 0012 airfoil with a trailing edge of 4.5% thickness and semi-circular shape. The flow is found to be unsteady but periodic in the trailing-edge region. Thus, lift and drag fluctuate at small amplitudes around mean values and at distinct frequencies. While the overall features are in qualitative agreement with the few experimental observations reported in the literature, the accuracy is limited because of certain local shortcomings of the computation grid. Several recommendations on how to achieve improved predictions are given for future attempts.

1. INTRODUCTION AND BACKGROUND

Certain airfoils, helicopter blades, and modern turbine bladings often have trailing edges of a finite thickness, usually amounting to a few percent of the chord length. Typically, the flow is compressible and turbulent at high Reynolds numbers. The calculation of the flow field is still an unsolved problem in the vicinity of the trailing edge. This is due to the limited understanding of the quite complex flow phenomena and to the lack of powerful adequate calculation tools previously available. In contrast to many other flow problems where an approximate prediction consistent with experimental observation is achieved on the basis of the Euler equations, the problem of flow over a thick trailing edge cannot be even approximately solved by means of inviscid theory. For subsonic flow, a stagnation point is predicted with an unrealistic pressure peak whereas an infinite number of solutions are found for supersonic conditions. The fact that the Euler equations fail completely in predicting the real flow behavior is probably one of the most striking (and challenging) features of computation of flow over a thick trailing edge.

Therefore, a successful approach has to account for the viscous effects. Several calculation models for base-region flow have been developed in the past. A detailed survey on several of these boundary-layer-like approaches is given in references 1 and 2. Since almost all are derived from the flow over a simple backward-facing step, they are based on the assumptions that

- the separation point is known,
- the flow reattaches to a solid wall of inclination known a priori,

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• the wake flow is steady, and

• the outer flow approaching the step is homogeneous.

A major deficiency is that the interaction between the viscous-dominated boundary-layer and shear-layer flow, and the nearly inviscid outer flow is poorly accounted for. In contrast, flow over the thick trailing edge of an airfoil or turbine blade has the following features:

• There are two separation points which are unknown if the base is rounded.

• Two streams of different properties, e.g., Mach numbers, etc., merge behind the trailing edge; the resulting flow direction is not known a priori.

• At high Reynolds numbers and subsonic Mach numbers, the flow is unsteady; the wake is of the vortex-street-type.

• The outer flow approaching the trailing edge is not necessarily homogeneous.

• The viscous-dominated flow near the body and in the near wake interacts strongly with the nearly inviscid outer flow.

2. MOTIVATION AND DESCRIPTION OF PROBLEM

Several problems with strong viscous-inviscid interaction and unsteady effects have been successfully treated by numerically solving the Reynolds-averaged Navier-Stokes equations or a truncated version thereof (e.g., refs. 3-6). Therefore, the question arises whether such methods can lead to an improved solution of the thick trailing-edge problem. So far, a few attempts have been made, for example:

• Levy, Briley, and McDonald recently used an implicit technique to predict the laminar flow over an ellipse at a Reynolds number of 10^3 and a Mach number of 0.2. Their solution shows steady flow with two standing vortices in the near wake (ref. 7).

• Fanning and Mueller solved the vorticity transport equation and Poisson's equation numerically and were thus able to predict the oscillating incompressible laminar wake behind the square base of a flat plate (ref. 8).

• Waskiewicz, Shang, and Hankey recently published the solution for supersonic turbulent flow over the square base of a wedge-flat plate-model (ref. 9). They used MacCormack's fully explicit method (ref. 10).

To the author's knowledge, the problem perhaps most often encountered in practical application has not been tackled so far, namely subsonic compressible turbulent flow at high Reynolds number over an airfoil with a trailing edge of finite thickness and arbitrary shape. Therefore, the work described herein is dedicated to that specific problem. As in references 7-9, the attempt is restricted to two-dimensional flow. However, in contrast to references 8 and 9, it is not restricted to flat plates and/or square bases.

The basic goal consists of efficiently achieving an improved prediction of the unsteady subsonic flow field near a thick trailing edge by using a modern numerical computation procedure. The specific problems connected with such a prediction should lead to recommendations for future similar attempts and for practical engineering applications. Furthermore, by studying the unsteady flow near the thick base of a slender two-dimensional body by means of computer simulation, the author hopes to provide additional information for a better understanding of the complex-flow phenomena.

3. COMPUTATION METHOD

A large variety of computation methods based on the numerical solution of the Reynolds-averaged Navier-Stokes equations or a truncated version thereof have been developed in recent years. Steger's code (ref. 4) was found to be very powerful and efficient in several applications (e.g., refs. 4-6,11). The use of curvilinear coordinates allows their application to flow fields with arbitrary boundary contours as they are encountered on thick round trailing edges.

Since the details of this computation method are given in several publications (e.g., refs. 4 and 11), only its main features are described here. The governing equations are the continuity equation, the Reynolds-averaged Navier-Stokes equations, and the energy equation written in strong conservation-law form. From the Cartesian coordinate system x,y, they are transformed onto the grid line coordinates ξ,η. As shown in figure 1, the transformation corresponds to a mapping of the physical plane where ξ and η appear as curvilinear, nonorthogonal coordinates, onto the computational plane where they form a Cartesian coordinate system. It is important to note that the wake centerline, including the last point of the airfoil at the trailing edge (see fig. 2), are mapped twice if the flow field in the physical plane is cut along the wake center line. As a consequence, the solution is computed twice for these nodal points. As discussed in references 11 and 12, the grid spacing in the streamwise direction used in the numerical computation of high Reynolds-number flow usually does not allow resolution of the viscous derivatives in the streamwise direction even though the complete Navier-Stokes equations may be programmed. Consequently, the viscous terms in ξ, which is the direction along the body, are neglected and only those in η are retained, as in conventional boundary-layer theory. However, in contrast to the latter, the η-momentum equation is retained. Thus, no assumption is needed on the pressure variation across the viscous layer. This so-called thin-layer approximation is therefore expected to be capable of accounting extensively for the strong viscous-inviscid interaction near a trailing edge of finite thickness. Unlike certain coupling procedures, this approach avoids the difficulties rising from matching an inviscid solution with a viscous-layer solution. Since a correct coupling might be quite difficult in unsteady flow, the application of the thin-layer approach is very attractive here.

The thin layer model imposes two restrictions on the computation grid: (a) a boundary-layer-like coordinate system must be used near the body and in the wake with the η-direction (approximately) normal to the flow direction, and (b) the intersecting trailing-edge contour and wake center line have to be treated as a simple continuous line (see fig. 2) and must be mapped onto the same line of constant η. Therefore, that type of network has to be used which is usually referred to as a c-grid, and which has the property that the ξ-direction coincides with the streamwise direction near the body and in the wake.

The influence of turbulence is modeled using the two-layer algebraic eddy-viscosity model developed by Baldwin and Lomax and described in detail in reference 12. As discussed in section 5, detailed measurements are not available for the unsteady near-wake flow. Therefore, one is forced to adopt a turbulence model which was not developed specifically for that type of flow; however, it is adequate for the region where the flow is attached to the airfoil surface. Since the Strouhal number of vortex shedding from thick-trailing edges is of $O\ (10^{-1})$ and thus much lower than the dimensionless mean frequency of the large-scale turbulent eddies which is of $O\ (10)$, the use of a steady or quasi-steady turbulence model is justified.

If the same notation used in reference 4 is adopted here, the equations to be solved can be written as

$$\frac{\partial \hat{q}}{\partial t} + \frac{\partial \hat{E}}{\partial \xi} + \frac{\partial \hat{F}}{\partial \eta} = \frac{1}{Re} \frac{\partial \hat{S}}{\partial \eta}$$

where the vectors $\hat{q}$, $\hat{E}$, $\hat{F}$, and $\hat{S}$ are given in detail in reference 4. They are efficiently obtained by means of the Beam-Warming delta-form approximate-factorization algorithm (refs. 13 and 14). It is an implicit time-marching finite difference scheme of second-order accuracy in space and of first- or second-order accuracy in time, depending on the type of differencing used to advance the solution in time. The results presented in section 6 are computed using the first-order time-accurate scheme.

4. GRID GENERATION

Thompson, Thames, and Mastin have developed a practical method to generate grids that vary smoothly and fit arbitrary boundary contours precisely (ref. 15). In its simplest form, the grid in the physical plane is found by solving the two Laplace equations

$$\frac{\partial^2 \xi}{\partial x^2} + \frac{\partial^2 \xi}{\partial y^2} = 0$$

$$\frac{\partial^2 \eta}{\partial x^2} + \frac{\partial^2 \eta}{\partial y^2} = 0$$

which, for the actual solution are transformed onto the computational plane and solved there by relaxation on the same grid that is used to solve the flow equations.

This method has the advantage that the user can select the location of the grid points on the boundary contours (Dirichlet boundary conditions). Thus, clustering of the boundary nodes can easily be adjusted to the actual problem which, in the present case, requires a fine resolution of the trailing-edge contour. A major deficiency is that the mesh spacing in η-direction near the body and in the wake is far too coarse for viscous-flow computation. Therefore, Sorenson and Steger proposed to discard the grid point distribution along the lines of constant ξ and to recluster the nodes by a simple monotonic stretching function, which moreover allows the minimum mesh spacing at the surface to be specified by the user (ref. 16). According to reference 16, constant minimum spacing is used along the body. It may be based on a simple estimation of the boundary-layer thickness at the trailing edge by means of the well-known Reynolds-number criterion. In an attempt to account for the variation of the boundary-layer thickness along the surface, the minimum mesh spacing is continuously reduced toward the front part of the airfoil. A similar modification is used in the wake.

A further modification is needed because the lines of constant ξ intersect the surface on the rear portion of the airfoil at an angle well below 90° and thus violate the condition of a boundary-layer-like coordinate system as imposed by the thin-layer approximation. There are several ways to create a grid with lines of constant ξ normal to the wall; one way is to add inhomogeneous terms to the right-hand side of the Laplace equations. Here, however, a much simpler procedure is used. It is required that the nodes on the surface and the lines of constant ξ in the outer flow, which are well distributed, are retained. The basic idea consists of displacing the grid points near the body along the lines of constant η. In the first step, a straight line normal to the surface is drawn. As indicated in figure 3, its last point is connected by another straight line with the first node kept unchanged in the outer part. Smooth curvature is achieved by continuously cutting off the resulting corners. Two quantities need to be specified: (a) the number of nodes which must belong to the strictly normal direction of the new lines of constant ξ, and (b) the number of grid points within which the transition to the outer part has to occur. The new lines of constant ξ look very similar to those that would result from the use of a cubic spline function. In contrast to the latter, the procedure presented here, which should be viewed as a simple engineering approach rather than a very sophisticated solution, avoids any iteration.

5. CHOICE OF A TEST CASE

Experimental data on the unsteady near wake of two-dimensional slender bodies with a trailing edge of finite thickness are very rare. A search in the literature resulted in a very limited collection of Strouhal numbers for flat plates and symmetrical airfoils and is described in reference 17.

A common feature of almost all of the experiments is that no details are available except the Reynolds number, the Mach number, and the body shape. According to reference 17, the frequency of vortex shedding can be strongly affected by the ratio of boundary-layer displacement thickness to trailing-edge thickness. It is therefore astonishing that most of the experimentalists do not give any information on the boundary layer. Often, it is not even known whether the boundary layer is laminar or turbulent. Therefore, we must abandon the attractive idea of confining the computation domain to the trailing-edge region and using measured boundary-layer profiles as upstream boundary conditions.

Numerical computation methods give a detailed prediction of the flow quantities at every grid point. To check these data reliably and to detect the weakness of a method precisely, as well as to improve these data, detailed measurements are needed in the entire region of interest. Due to the lack of such experimental data, it is not possible to establish a test case which allows reliable checks on the numerical prediction by comparing it against measurements. Therefore, the author prefers to design a theoretical test case which allows several internal checks.

For reasons of symmetry, a NACA 0012 airfoil is chosen. Its rear portion is cut off and replaced by a semicircular base with a radius of 2% of the original chord length (see fig. 4). The requirement that the trailing-edge contour fits the airfoil surface and its slope results in a new chord length of 87.3% of the original one. Thus, the actual ratio of trailing-edge thickness to chord length amounts to 4.5%.

From the remarks just given, it follows that the entire flow field around the airfoil must be calculated. The resolution of the trailing-edge region requires a lot of grid points. The total number of grid points available is limited for obvious reasons. Since the resolution of shocks would require a fine grid in these regions and thus would reduce the number of nodes available at the base, the flow should preferably be subsonic everywhere. However, effects of compressibility must not be negligible. This is achieved at the free-stream Mach number of 0.574 used in the computation. The free-stream Reynolds number amounts to 5×10^5 and the boundary layer is assumed to be turbulent over its entire length. Two different angles of attack are selected: 0° and 2°. This choice allows several checks based on physical considerations, as will be seen in the discussion of the results.

Since the governing equations are used in nondimensional form, it should be noted that while the reference quantities are arbitrary, the Reynolds number is defined in terms of these reference values. For convenience, the chord of the original NACA 0012 airfoil is chosen as a reference length. The Reynolds number of 5×10^5 refers to that length. Its value based on the new chord is thus smaller by 12.7%, that is, it amounts to 4.36×10^5.

6. RESULTS AND DISCUSSION

A grid with a total number of 77×34 nodal points is used. The grid section near the body shown in figure 5 and the trailing-edge detail shown in figure 6 illustrate the modification of the grid generation procedure described in section 4. Within the 12 nodes next to the body surface or wake center line, the lines of constant ξ are strictly normal to these contours. As is evident from figure 6, this is not true for the semicircular base. There, the direction is based on a rough estimation of the mean-flow direction expected. The minimum mesh spacing in the η-direction increases from 0.125×10^{-3} at the leading edge up to 0.220×10^{-3} at the trailing edge. A check of the predicted flow field just ahead of the base indicates that 16 grid points are inside the boundary layer.

Figure 6 also shows that the lines of constant η do not curve smoothly as they intersect the two lines of constant ξ emanating from the last nodal point at the trailing edge. This deficiency is likely to cause locally a certain loss of accuracy, a fact which was underestimated when the grid was generated. Indeed, the most crucial detail in creating an adequate network is the intersection of the wake center line with the base contour. To achieve smooth curvature, one would have to accumulate many grid points in that region and probably to introduce a cusp of almost vanishing size. Another way to avoid the deficiency is by using a so-called O-grid where the lines of constant η form closed loops around the airfoil. However, it must yet be examined in that case whether the thin-layer approximation is still valid in the trailing-edge region.

The first prediction was dedicated to symmetrical flow, that is, $\alpha = 0°$. The computation was started using free-stream values at all grid points. Thus any artificial asymmetry was avoided. The length of the time step was doubled after 100 and 200 steps and then kept constant. Already, after 400 time steps corresponding to approximately 3-1/2 actual chord lengths traveled, a symmetrical oscillation of the lift coefficient at a distinct frequency was observed. Since the amplitude and the frequency were still increasing, the computation was continued. After they became constant, several hundred additional time steps were made in order to assure that this was not an intermediate stage but the final solution. This is illustrated in figure 7 where the lift coefficient CL and the pressure drag coefficient CD are shown versus the number of actual chord lengths traveled. The analysis of the flow field shows that the oscillations are due to the periodic motion upwards and downwards of the separated flow at the base. This causes a slight alteration of the pressure distribution up to a certain distance ahead of the trailing edge. The appearance of such fluctuations is in agreement with Summers and Page, who observed it in their experiments with circular-arc airfoil sections with a blunt trailing edge (ref. 18). Unfortunately, no pressure measurements were made which would allow a check of the magnitude of the oscillations.

Since the airfoil is symmetrical and since the angle of attack is 0°, the lift fluctuation must be symmetrical with a mean value equal to zero. Using the definitions

$$\widetilde{CL} = \frac{1}{T}\int_T CL(t)dt$$

$$\overline{CL}_{max} = \frac{1}{n}\sum^{n} CL_{max}$$

$$\overline{CL}_{min} = \frac{1}{n}\sum^{n} CL_{min}$$

where T is the length of a period and CL_{max} and CL_{min} denote positive and negative peak values, we obtain

$$\widetilde{CL} = -0.893\times10^{-5}$$

$$\overline{CL}_{max} - \widetilde{CL} = 0.910\times10^{-2}$$

$$\widetilde{CL} - \overline{CL}_{min} = 0.909\times10^{-2}$$

The Strouhal number based on the trailing-edge thickness, d,

$$Str = \frac{f \times d}{U_\infty}$$

where f is the dimensional frequency, is found to be 0.070. If it is based on the maximum thickness of the body, which is 13.7% due to shortening of the chord, its value amounts to 0.24. Unlike the well-known case of a circular cylinder in cross-flow, the Strouhal number of slender bodies with thick trailing edge is not dependent only on the Reynolds number (ref. 17). There is strong evidence that the Strouhal number is affected by the ratio of boundary-layer thickness to base height. It is also likely to be affected by certain shape parameters, for example, maximum thickness and tail angle. As mentioned previously, boundary-layer transition is assumed at the leading edge and thus it is fairly thick at the trailing edge. Since a reliable well-documented empirical correlation, such as the Strouhal number-Reynolds number relationship of the circular cylinder, is not available for airfoils, the correctness of the predicted Strouhal number cannot be assessed.

According to figure 7, the pressure drag fluctuates at twice the frequency of the lift. This feature is due to the symmetry of the body and the choice of $\alpha = 0°$ and can also be observed on circular cylinders (see ref. 19). It expresses that the flow phenomena occurring in the upper part of the near wake later takes place in the lower half of a CL-period. Essentially, these effects are the washing-in and washing-out process of the separated flow and are connected with the alternating vortex shedding from upper and lower detachment regions. A detailed analysis of the predicted near-wake flow indicates that the time-dependent pressure distribution during the washing-out motion differs from that during the washing-in motion. Since a substantial part of the pressure drag is due to the thick trailing edge, the shape of the drag fluctuation must be asymmetrical. However, two consecutive drag cycles corresponding to two vortices shed from different but symmetrical separation points must be identical. In the author's opinion, there are no physical reasons which contradict this explanation. Moreover, there is another case of self-driven unsteady flow separation with analogous features, namely transitory stall in diffusers. According to Kline (private communication) the time dependent pressure distribution during the washing-out motion of the separation bubble differs from the one during the washing-in process.

Consider the same airfoil at the same Mach and Reynolds number but at a certain angle of attack which, however, is assumed to be small enough to prevent the separation from moving upstream. Beside the fact that the airfoil must have a certain mean lift, at least the following alteration in the flow behavior can be predicted without any calculations: (a) the drag has to fluctuate at the same frequency as the lift, and (b) the Strouhal number changes only slightly. Therefore, a computation was made for $\alpha = 2°$. All other flow parameters and program parameters (e.g., the grid, the time step-length, etc.) were kept unchanged. Furthermore, the computation was started again using free-stream values in all nodal points.

Again, lift and drag fluctuations at a distinct frequency were observed after a few hundred time steps. After constant frequency and amplitudes were achieved, the computation of several hundred additional time steps did not change the prediction. This, as well as the fact that the length of drag and lift cycles are identical, is illustrated in figure 8. The Strouhal number based on the trailing-edge thickness amounts to 0.069 whereas the one based on the maximum thickness is 0.237. The mean lift coefficient $\widetilde{CL}$ is found to be 0.205. The values

$$\overline{CL}_{max} - \widetilde{CL} = 0.7929\times10^{-2}$$

$$\widetilde{CL} - \overline{CL}_{min} = 0.7931\times10^{-2}$$

suggest that the lift still fluctuates nearly symmetrically around its mean value.

Several remarks have to be made with respect to accuracy. In order to give helpful suggestions for future attempts, this is done quite openly. Since the total number of nodes was limited, the grid points used to resolve the base contour had to be saved along the remaining part of the surface. This resulted in streamwise mesh spacings which are somewhat coarse in the mid-chord region. The solution for the last node at the trailing edge is computed twice independently. For example, a check on the numerical values indicates that the static pressures always differ somewhat; in the worst case, the difference is about 5%. This inaccuracy is mainly due to the corner-like curvature of the line of constant η near that grid point.

Another unsatisfactory detail is illustrated in figures 9a, b, c, and d where four large-scale plots of the instantaneous velocity vectors in the near wake are shown for $\alpha = 0°$. They are taken after approximately 1/12, 4/12, 7/12, and 10/12 of a CL-period and imply that the velocity component in the x-direction is not computed correctly along the center line. This is likely at least partially due to the (too) large streamwise spacing in the reverse-flow region. The fact that the shortcoming is the worst at maximum and minimum lift (i.e., when the flow field is the most asymmetrical) and that it disappears at zero lift (fig. 10) may suggest that the boundary condition applied there is questionable. An extrapolation procedure from above and below followed by averaging was used in the computation. Based on these considerations, the accuracy of all the numerical values given above must be considered limited. In the author's opinion, the shortcoming of the grid near the intersection of base contour and wake center line has more negative impact than the boundary condition.

In addition to the points given previously, some further recommendations for future attempts can be derived from the previous discussion. A finer resolution of the reverse-flow region is necessary. As a rule of thumb, its length may be assumed to be roughly equal to the base height. The stretching in the η-direction should preferably be reduced there (see fig. 6). The limited upstream influence of the unsteady flow phenomena at the trailing edge suggests that the number of grid points available in the near wake can be increased by confining the computation domain to the rear part of the airfoil. The present results imply that the inflow plane could be located in the mid-chord region with no adverse effects. Beside the application of an improved procedure for the nodes on the near-wake center line, it may also be useful to scrutinize the terms neglected in the thin-layer model. As mentioned by several authors (e.g., in ref. 20), there is evidence that the normal stress terms in the streamwise direction become significant near separation.

7. SUMMARY AND CONCLUSIONS

An implicit time-marching finite difference scheme was used to solve the thin-layer equations for two-dimensional turbulent flow at a Reynolds number of 4.4×10^5 and a Mach number of 0.574 over a modified NACA 0012 airfoil with a 4.5% thick round trailing edge. The computation grid was generated using an existing automatic procedure. A simple modification was added to achieve strictly normal direction of the radial grid lines near the body. Solutions were computed for $\alpha = 0°$ and $2°$. Due to unsteady but periodic flow in the trailing-edge region, lift and drag fluctuate at distinct frequencies around mean values. The amplitudes are small compared to the mean values. The overall features of the numerical solution are in agreement with the few experimental observations known from literature. The accuracy cannot be assessed precisely due to the lack of detailed measurements. The accuracy is judged to be limited because the streamwise grid lines do not curve smoothly near the intersection point of base contour and wake center line and because the number of computation stations in the streamwise direction is small in the near wake. The recommendations given on how to avoid these shortcomings and on how to improve some further details should be useful in achieving more accurate predictions in future attempts.

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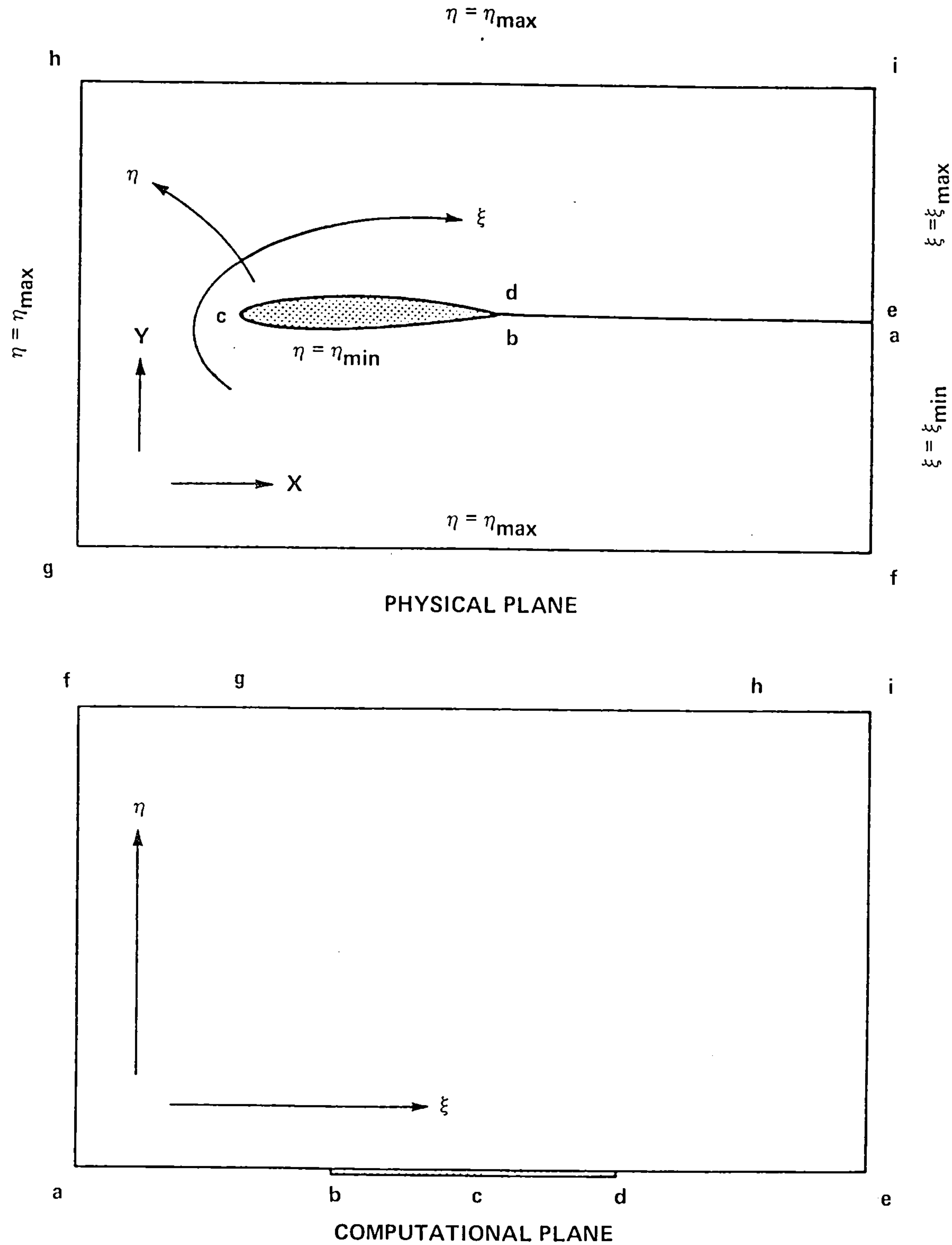


Figure 1.- Physical and transformed computational planes.

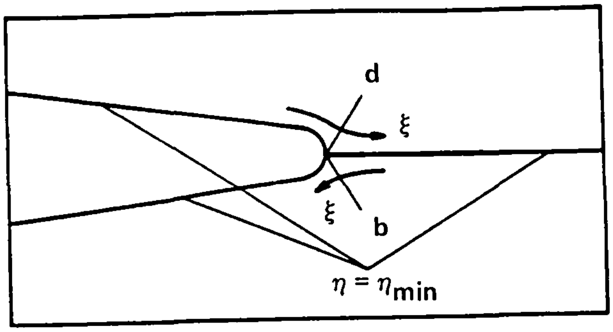


Figure 2.- Mapping of trailing-edge region.

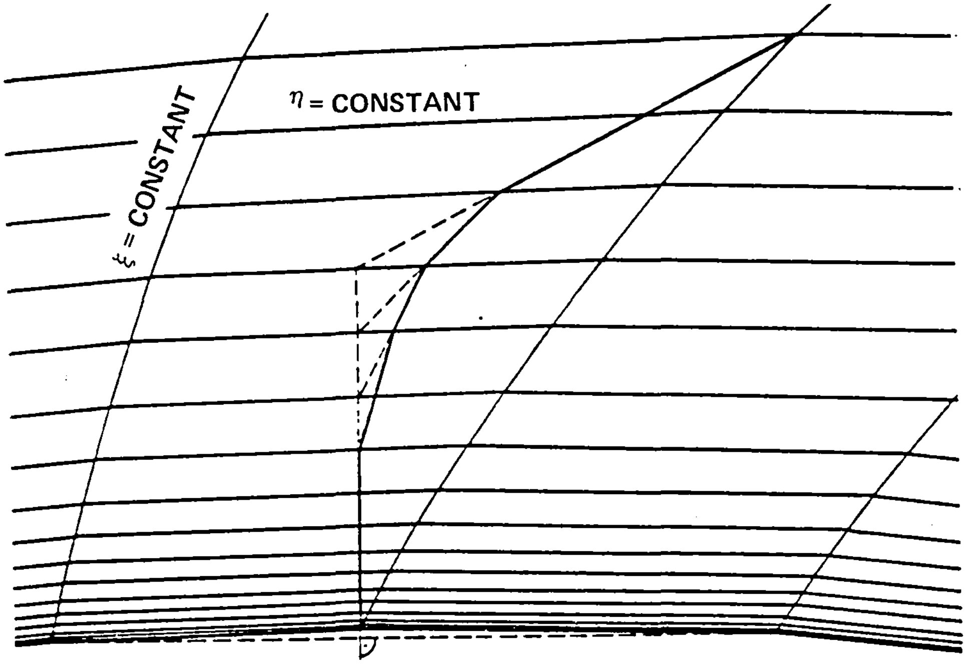


Figure 3.- Grid modification to ensure that the η-direction is normal to the body surface.

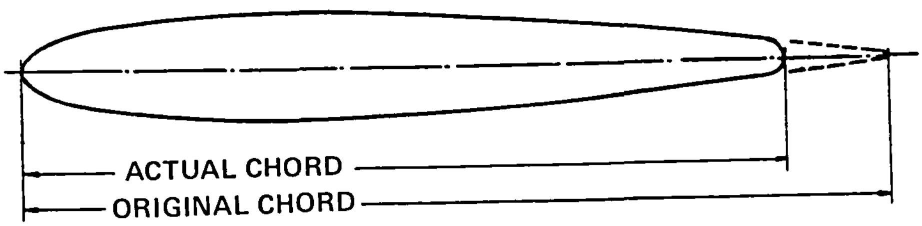


Figure 4.- Geometry of test model.

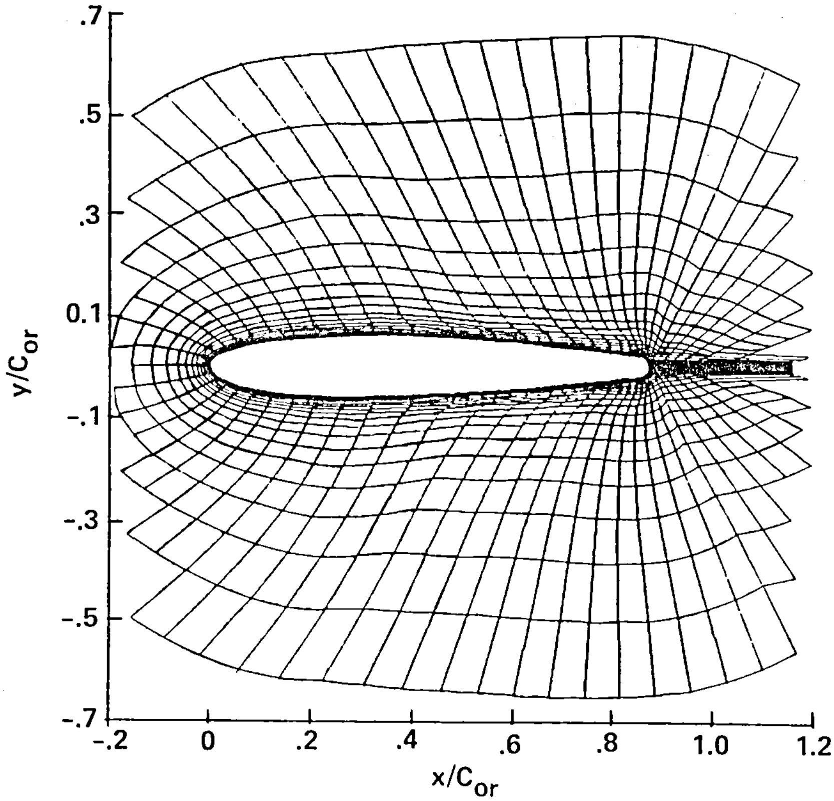


Figure 5.- Grid section near the airfoil.

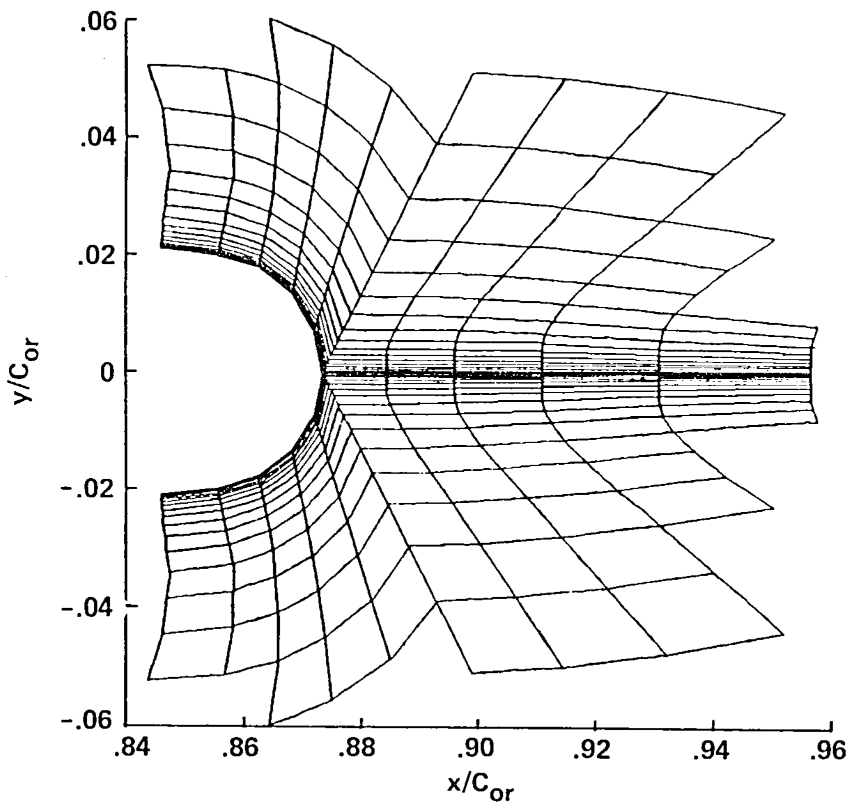


Figure 6.- Grid detail in the base region.

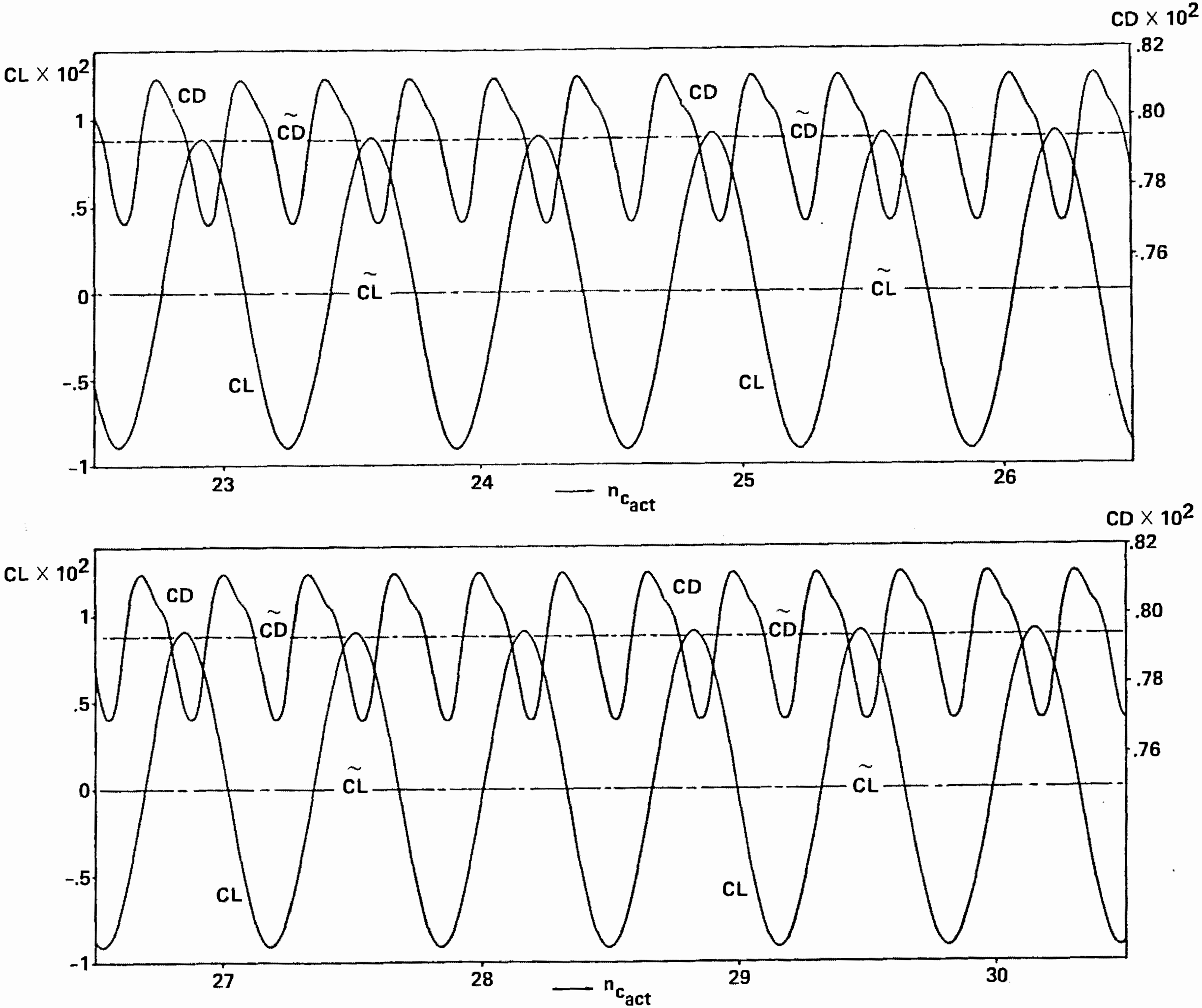


Figure 7.- Lift and drag fluctuations vs number of actual chords traveled; α = 0°.

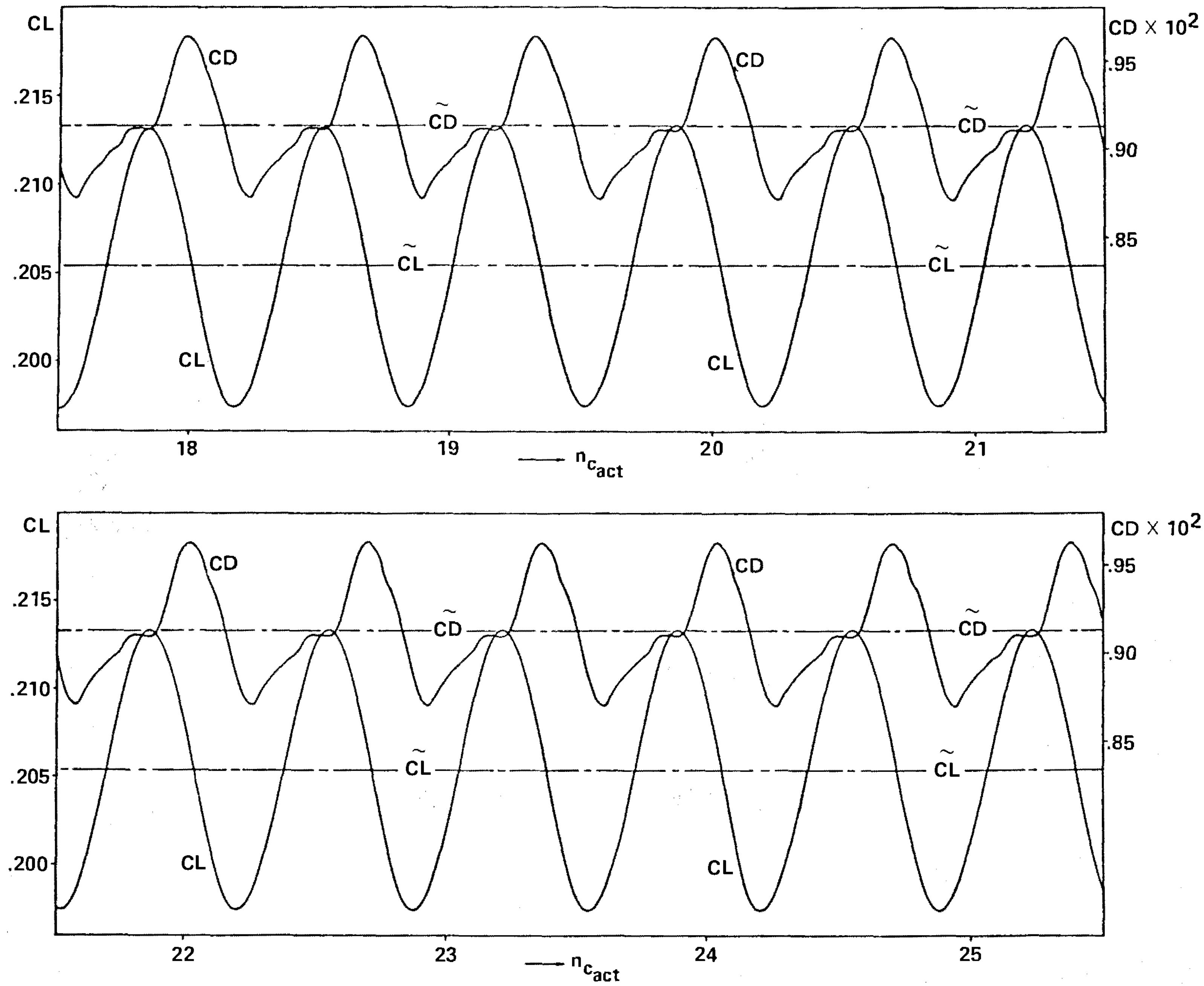
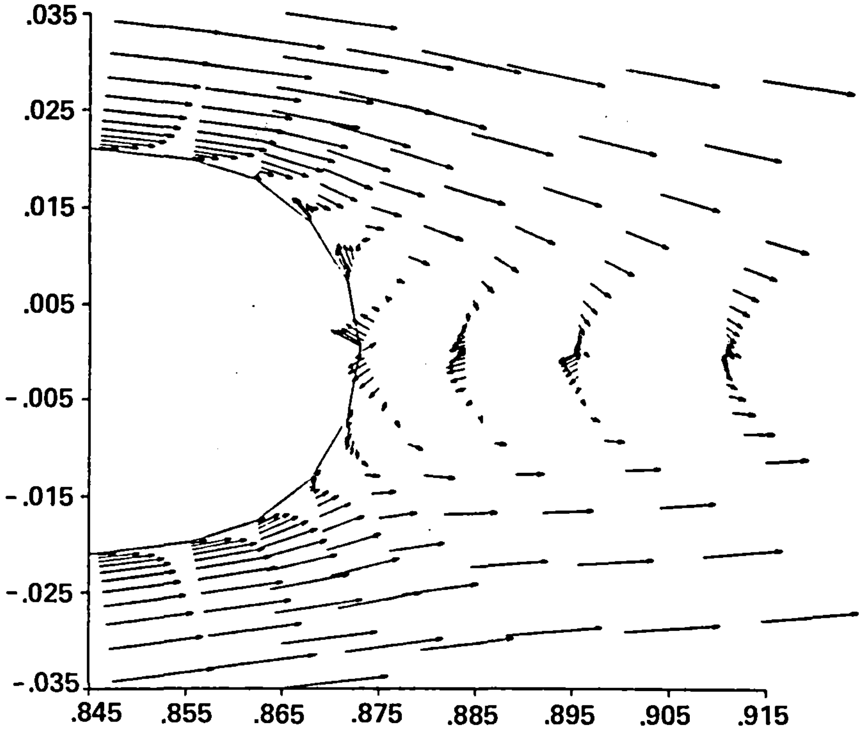
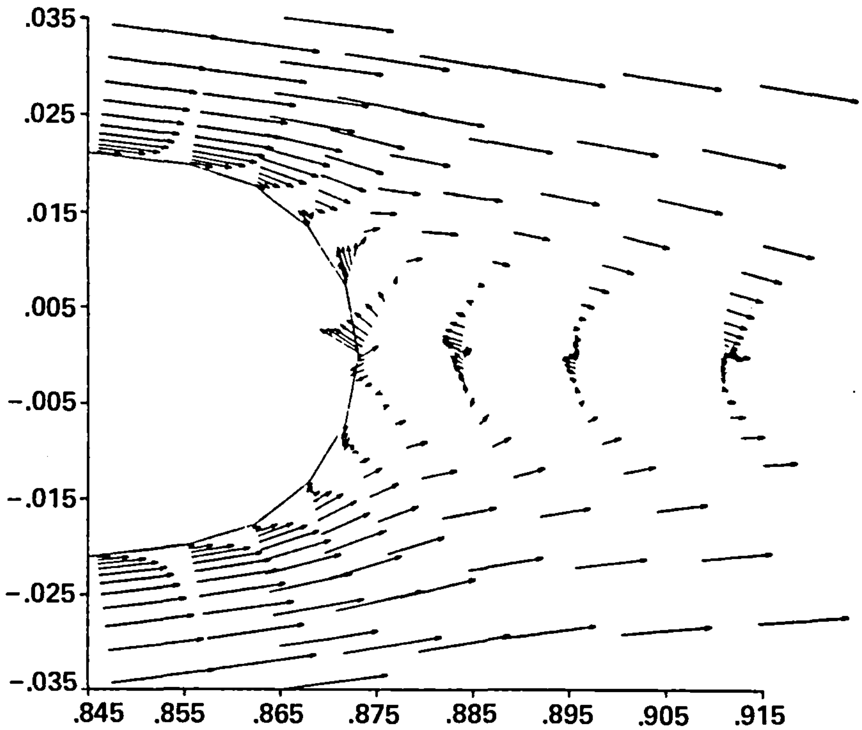


Figure 8.- Lift and drag fluctuations vs number of actual chords traveled; $\alpha = 2°$.

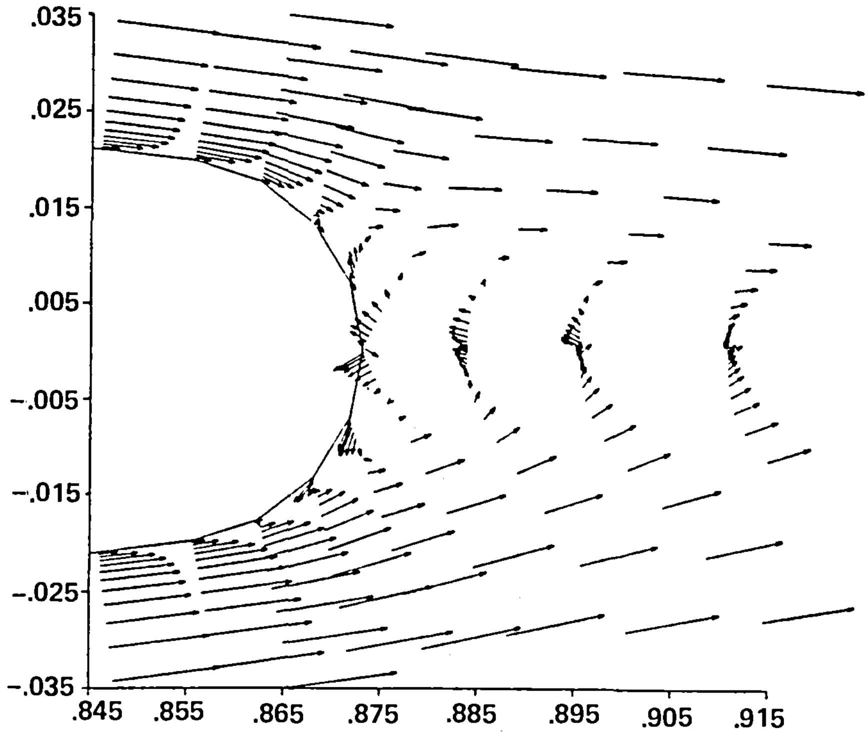


(a) After 4/50 of a CL-period.

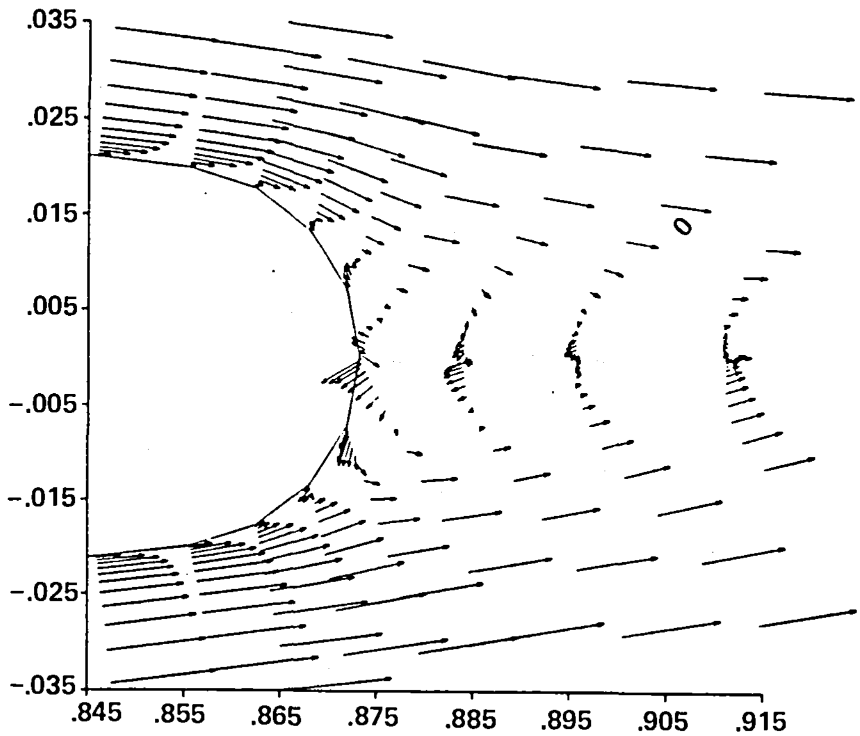


(b) After 16/50 of a CL-period.

Figure 9.- Instantaneous velocity vectors in the base region.

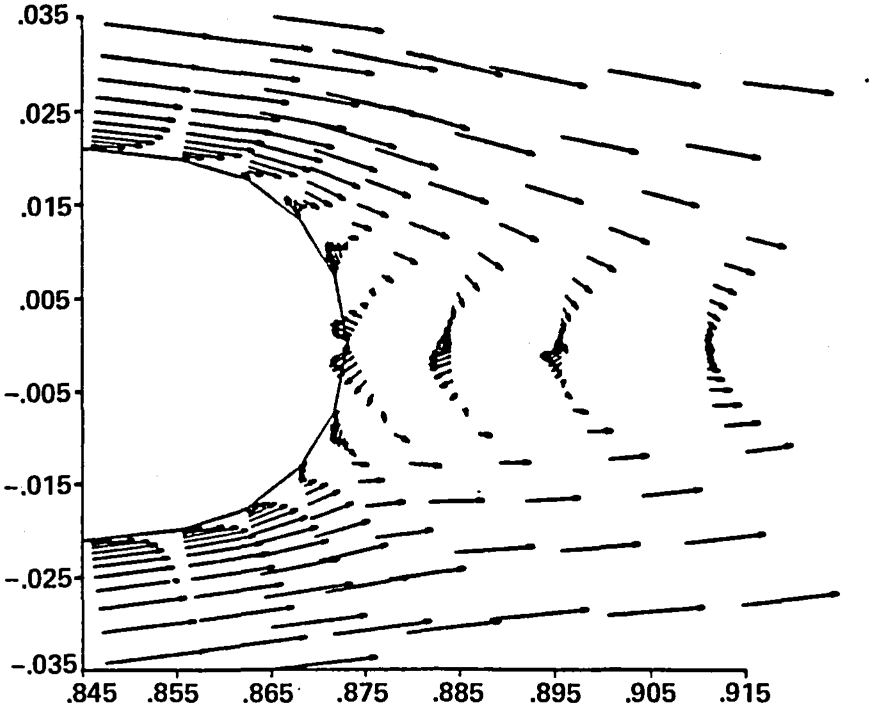


(c) After 29/50 of a CL-period.

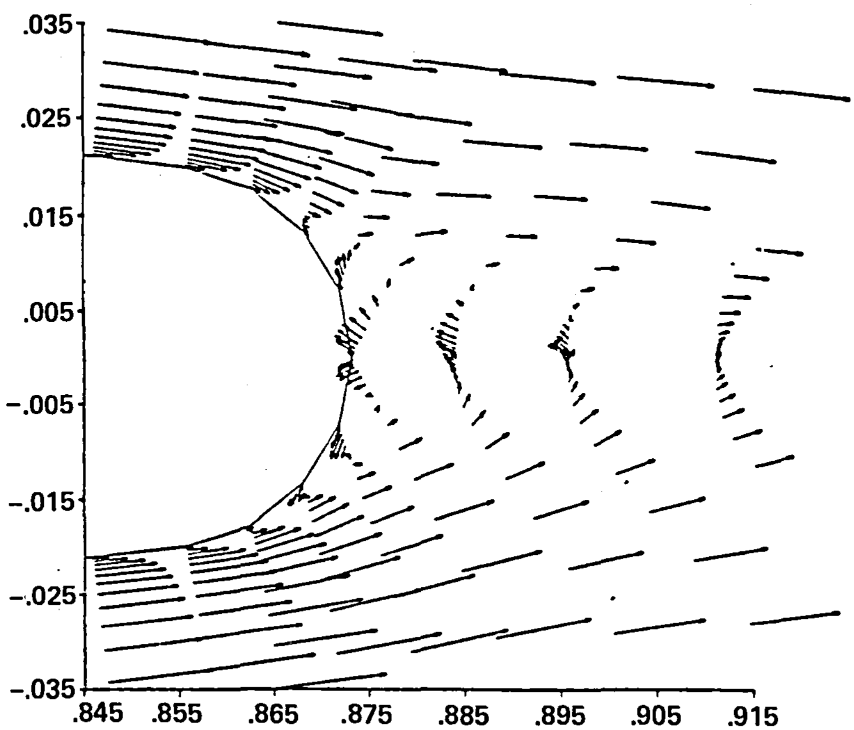


(d) After 41/50 of a CL-period.

Figure 9.- Concluded.



(a) At the beginning of a CL-period.



(b) After half a CL-period.

Figure 10.- Instantaneous velocity vectors in the base region at zero lift.

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16. Abstract

An implicit time-marching finite difference method is used to predict two-dimensional turbulent flow at a Reynolds number of 4.4×10^5 and a Mach number of 0.574 over a shortened NACA 0012 airfoil with a trailing edge of 4.5% thickness and semicircular shape. The flow is found to be unsteady but periodic in the trailing-edge region. Thus, lift and drag fluctuate at small amplitudes around mean values and at distinct frequencies. While the overall features are in qualitative agreement with the few experimental observations reported in the literature, the accuracy is limited because of certain local shortcomings of the computation grid. Several recommendations on how to achieve improved predictions are given for future attempts.

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