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THE USE OF COMPUTER MODELS IN HELICOPTER DRAG PREDICTION

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Abstract

The paper presents a brief review of the developments in airframe drag prediction capability in the last decade. In particular, it details the inability of the first generation panel methods to more than adequately calculate the flow about helicopter fuselages and their failure, despite early promise to be able to predict absolute levels of drag. The development of a second generation of panel methods led the way to an improved flow modeling capability and, in particular, to the ability to model the behavior of viscous flow and to predict and track the development of regions of separate flow. The paper shows, however, that even these models fall short in their ability to accurately predict drag. They do, however, provide the jumping off point for the development of a new generation of panel methods which take full account of the unsteady motion of the large-scale separated flow. These methods are discussed and early results on a series of bluff bodies and typical helicopter tail boom cross sections are presented. In most cases the correlation between predicted and measured drag is excellent.

Introduction

Over a decade ago the paper that introduced the Report of the American Helicopter Society Ad Hoc Committee on Rotorcraft Drag, Ref. 1, outlined a systematic plan for the reduction of helicopter drag. Part of that plan was the development and use of detailed analytical models of the fuselage flow field. While some of the practical suggestions made by the authors have been implemented and production helicopters now are undeniably aerodynamically cleaner than their predecessors, the promise that the authors saw in the theoretical approach to the problem has not been fully realized.

The work in progress at the time and also presented as part of the committee's report, Refs. 1 and 2, used vortex-lattice or source singularities or some combination of the two to represent the flow about simple helicopter shapes. The work of Gillespie (3) is typical of the early approaches to the problem. Based on the work of Hess and Smith (4), Gillespie combined the potential flow method with a streamline/boundary layer analysis to allow calculation of the combined pressure/skin friction drag of bodies. With no way of modeling wakes or separated flow available to him he was forced to compromise his answer by, in fact, assuming some "reasonable" value of base pressure within the separation zone indicated by his viscous analysis. In essence, he had to know the drag in order to calculate it and in his concluding remarks acknowledged this problem, "...reasonable drag prediction could be obtained if an accurate means of predicting the pressure in the separated flow region can be developed." The inability to accurately define separation and hence drag did not, however, prevent the analyses from being used to explore typical helicopter-like shapes and real flow problems, Refs. 1 and 5. Using another approach to the basic potential flow problem (6), Montana used the analysis to explore the viscous flow over a range of fuselage and rotor head shapes developing configurations where the shape was modified in an attempt to prevent separation. The author also used the analysis in an attempt to predict the effect of wind tunnel wall constraints in testing, highlighting in the process one of the weaknesses of the early analyses. That is, their tendency to lose mass flow, to "leak" when used for channel-like and particularly internal flows.

The approach used by the authors of Ref. 2, outlined in more detail in Ref. 7, was a combination of source and surface vortex-lattice elements. This permitted a representation of the separated flow region and modeling of the lift carried by the fuselage. Unfortunately, not all of the empiricisms were removed since it was still necessary to specify some non-zero normal velocity boundary condition inside the separation zone and only a rigid, streamwise wake could be used. Despite this it was possible to calculate the drag on simple shapes (for example, cylinders) with some degree of confidence and the early results on helicopter fuselages were encouraging. However, it was evident that the viscous/potential flow iteration procedure was not stable. This resulted from an inappropriate angle between the (streamwise constrained) wake and the surface at the separation location. The consequence of this was a small stagnation zone ahead of the modeled separation location, an intensification of the unfavorable pressure gradient and an unrepresentative tendency for the separation to move further forward.

Among other attempts to model separation zones using first generation panel methods, perhaps the most notable is Ref. 8. Here, the authors, apparently more interested in the downstream influence of regions of separated flow than absolute levels of drag (9), (10), have developed a model, starting with a potential flow/viscous flow calculation to determine separation locations, which carries the viscous flow from the body into the wake. The wake follows local streamlines and allows for an embedded region of separated flow and wake growth due to entrainment. However, empiricisms remain and it appears that, as with the simpler models noted above, one needs to know the answer before starting the calculation in order to be able to set up the wake model. Further, it is not clear from the published results how the authors handle the situation when portions of the configuration pass through the wake sheath. In the early methods wake/surface impingement generally led to a divergent solution. Truncation of the body in the separation zone, shown in Ref. 8, would avoid the problem and would have little effect on results calculated in downstream regions. It does, however, make a calculation of the drag of the body a more difficult process.

With the advent of a second generation of panel codes in the late seventies and early eighties, it was felt that calculation of absolute levels of drag was a more achievable goal.

Developed specifically with the limitations of the first generation panel methods in mind, program VSAERO (for **V**ortex **S**eparation **AERO**dynamics) (11), is a typical second generation method. Retaining a low-order description of the surface, flat panels, the program uses a compound singularity, combining doublet and source terms for the body panels, using doublet panels for the wake. The source term is defined by the body geometry. The doublet term is the unknown and is solved using a modified Dirichlet boundary condition. Surface velocities are determined by differencing the doublet distribution in the appropriate direction. This approach, together with the program's ability to calculate the wake development free from any constraints, allows calculation of both external and internal flows with no "leakage" problems and permits exploration of complex flows involving wake/wake and wake/surface interactions (12).

Used with considerable success on conventional aircraft configurations, including those with complex high-lift devices deployed, the program has shown that it can calculate aerodynamic forces and moments, including drag, with a high degree of reliability (13). The doublet sheet approach has been extended to separated flow and jet wake modeling where the flow in the zone is assumed steady state (time-averaged). As with earlier approaches, positioning of the separated flow is preceded by a potential flow/viscous flow calculation. Also following earlier practise, the wake strength at the shedding point is carried onto the separated sheet. A constant doublet gradient is assumed and this is set up based on the difference between the resultant velocities inside and outside the wake. This model, although rather crude, has been used with considerable success in cases where the separation line is well defined, as on the leading- and trailing-edge of a wing immersed in rotor downwash (14), or where a jet is being modeled (15). Unfortunately, when separation is less well defined, where the approaching surface is curved or where the surface within the separation zone is non-planar, such as on an automobile aft deck and bumper region (16), or a helicopter transition zone, airloads in the base region are less well predicted and correlation between measured and predicted drag is poor. Attempts to improve this situation and remove the need to assume the wake vorticity gradient by actually determining it from local conditions were only marginally successful.

It would appear, therefore, that based at least on their ability to predict absolute levels of drag on helicopter-like shapes, the panel methods available today are not significantly better than their predecessors of a decade ago. The main reason for this has to be, of course, that although the basic formulations have improved, giving upgraded modeling capability for attached flows, nothing new has been incorporated which would allow the methods to better represent the unsteady flows which predominate in base regions. The assumptions regarding time averaging of unsteady flows which appear justifiable of the basis of good correlation between analytical and experimental results for conventional aircraft, where the flow is lift dominated and the characteristic frequencies of the separations, when present, are high enough to permit temporal averaging, breakdown for the bluff body separations found on most helicopters.

This paper discusses the application of an unsteady, time-stepping panel method program to the problem of drag prediction. Although this approach has the potential to identify and track the dynamic separation and cyclical loading associated with bluff body separation, its principal benefit is that by starting a body impulsively from rest, a time when it has no wakes, and generating the wakes step by step, no assumptions need be made regarding wake strength or characteristics. A pilot version of the method has been exercised on a range of typical helicopter shapes and correlation between measured and calculated drag values was excellent.

Method Development

Potential Flow

Although remarkable advances are being made in flow field calculations using finite-difference and also finite-element methods, the surface integral approach using panel methods coupled with special routines for nonlinear effects still offers distinct advantages for many real flow problems. In particular, panel methods offer greater versatility for practical application to complex configurations and are considerably more efficient in terms of computing effort. However, the concept of zonal modeling--in which a local Navier-Stokes analysis is coupled with a panel method--should not be overlooked. Ultimately, such a coupling should lead to an improved modeling of vortices (e.g., vortex cores, vortex dissipation and breakdown), thick viscous regions, local separation bubbles, and shock wave/boundary-layer interactions.

The present work is based on program VSAERO, (Vortex Separation AEROdynamics) which has been developed at Analytical Methods over the past few years. The program was originally developed for application to high-lift configurations under funding from NASA Ames. The method includes interactive schemes for a wake-relaxation calculation and for viscous effects using coupled integral boundary layer techniques. Under funding from the Office of Naval Research the program was used as a basis for research into modeling extensive separations on three-dimensional configurations (17). This project extended to the three-dimensional case, a separation model that had been successfully developed for the two-dimensional case under funding from the Army Research Office (18) and (19). In this model, which is installed in program CLMAX, the separated zone is enclosed between free vortex sheets in the panel method representation, Figure 1.

The thin vortex sheet model of the upper separated shear layer was demonstrated by Young and Hoad (20) to be a reasonable representation of the flow as far back as the trailing edge. For example, a comparison from (20) of a laser-velocimeter flow survey, and CLMAX program calculation is shown here in Figure 1.

A particular feature of the vortex sheet model enclosing the region of low energy is that surface pressures can be calculated directly in the separated zone (18). This is an additional advantage over the displacement surface approach of Henderson (21) and over the source outflow model of Jacob (22). The CLMAX method generally gives very close agreement with experimental pressure distributions (18), (19).

Program VSAERO is a mature analysis which has been used with considerable success on a wide range of configurations. It has been particularly successful for aircraft application (13). Application of the program to automobile aerodynamics, with shapes and flow features much like the helicopter has met with only qualified success (16), (23). Even with demonstrated ability to predict detailed flow features such as regions of separated flow and wake development, Figures 2 and 3 present results from a recent study (23), and well correlated calculated surface pressures over most of the body, Figure 4, the program falls short with the pressures in the base region. This, of course, seriously impacts the ability to predict drag.

The stumbling block appears to lie with the remaining empirical factors controlling wake length and the assumptions regarding entrainment of flow into the base region. This sensitivity, noted earlier by Stricker and Polz (8), severely limits the production (ideally, hands-off) use of the analysis and a fall-out of the development of an unsteady version of the program was that it became possible, by starting from a wake-free condition (body at rest) to remove any empirical inputs. The unsteady program retains the desirable features of program VSAERO and is described in some detail in Ref. 25.

Preliminary Calculations

The complete procedure coupling the unsteady time-stepping potential flow panel method, the extensive separation wake model and the unsteady boundary layer code has been assembled in the pilot code for a system checkout prior to forming the three-dimensional version. The flow diagram for the procedure is shown in Figure 5. At this time the unsteady boundary layer code is called at each time step and is fed by unsteady derivatives from the potential flow calculation.

Some recent refinements developed in the two-dimensional pilot code have significantly reduced the computing requirement of the time-stepping calculations. The procedure has been applied to a number of basic test cases with very encouraging results (25).

Time-stepping calculations have been performed for cases with known extensive separations. The purpose of these calculations was to check the basic unsteady circulation shedding model in the potential flow code. For the first set of tests, the wake panels were simply transported at the onset flow conditions at separation. Several triangular shapes were considered, each starting impulsively from rest and proceeding forward over 10 time steps for a total time of $\tau = \tau U_\infty/h = 3.0$, where h is the triangle base height. Separation was prescribed at the corners. Figure 6(a) shows the computed history of the drag coefficient from pressure integration for a 60 deg triangle with blunt face forward. A total of 40 panels was used to represent the triangle surface. The calculation was repeated in the presence of wind tunnel walls (also panelled) with a 10% blockage ratio. The indicated blockage correction is somewhat lower than that given by standard techniques. Figure 6(b) compares the computed pressure distributions for this triangle in and out of the tunnel. This "base" pressure has only small variation and is quite close to experimental measurements. Figure 7 shows a summary of computed drag coefficient versus triangle semi-apex angle. The calculated values are slightly high in relation to the experimental data collected from several sources by Hoerner in Fluid Dynamic Drag (26).

One further case was run for the 60 deg, apex-forward triangle in free air with the full wake velocity calculation routine turned on but without the amalgamation and redistribution schemes at this stage. The calculated C_D for this case falls below the experimental value, Figure 7. A series of computed wake shapes is shown in Figure 8. These are samples from a total of 40 time-step calculations. The total computing time for this case was 195 seconds on a PRIME 550 minicomputer--this is equivalent to less than 2 seconds of CRAY time. The solution should benefit from the numerical damping provided by the amalgamation and redistribution schemes described earlier.

Although the two-dimensional pilot program was generated primarily as a tool to examine the behavior of various parts of the dynamic separation calculation, it has shown considerable promise as a general purpose code for two-dimensional calculations and early examples (e.g., Figures 6, 7 and 8) demonstrated a capability to compute base pressures and drag coefficients of blunt sections using an impulsive start.

Discussion of Helicopter Component Analysis Results

The Sections and Flow Conditions

The three tail boom cross sections used as the basis for the analysis were chosen to match the configurations tested in the NASA Langley wind tunnel documented in Ref. 27 and come originally from the UH-1, the UH-60 and the AH-64 helicopters. For this study the cross sections were scaled to match those tested. The analytical modes were constructed by digitising detailed half-scale drawings provided by NASA. The three cross sections are shown in Figure 9, with representative tail rotor drive shaft covers and the strakes as tested. All dimensions in the plotted output data are in feet.

For comparison with the test data a wind tunnel dynamic pressure of 30 lb./ft.^{2} was used throughout the calculation. This, for standard atmospheric conditions, converts to an onset flow velocity of roughly 160 ft./sec. and a Reynolds number which puts the flow around the sections into the supercritical regime. The actual Reynolds number, based on body frontal height for all the cases, was in excess of 1 million.

The Calculations

Program OSCAIR (25) was used for all of the calculations. A typical run begins with the user preparing an input deck with the surface described in enough detail so that the program can reconstruct the shape and then subdivide the true surface into the panel model. For the basic shapes roughly 80 panels were used with this number increasing to roughly 100 for the cases with the drive shaft covers and to 110 for the cases with spoilers.

In the calculation the sections are started impulsively from rest and the user prescribes a schedule of time steps. The size and number of the steps were compromises between run time to a converged solution (convecting the starting vortex out of the range of significant influence) and the local flow detail required. As an example, circular cylinders similar to the basic AH-64 tail boom have a shedding frequency of roughly 25/sec. and to catch the details of this cyclical activity would require time steps of the order of 0.01 sec. With this step size it would require 1000 program cycles to convect the starting vortex only 10 body heights downstream. For the present calculations a step size of 0.1 sec. was used. This relatively crude step was felt to be acceptable in view of the final goal, the "steady" forces and moments on the cross sections.

The separation locations are initially prescribed based on experience and held fixed for the first 10 time steps to allow the starting vortex to convect clear of the body. However, the program progressively moves the (then calculated) separation location every time step thereafter until the position stabilises. If an initial position close to the final location is used, then convergence normally takes only 5 or 6 steps after release. However, even grossly inappropriate assumed values, say 1/4 body circumference away from the final location, can be tolerated at the expense of longer convergence steps.

Results for Basic Sections (Cover Off, Spoiler Off)

A full set of results are presented for the basic cases. These include: summary plots of the vertical and lateral forces on the sections in Figures 10 through 12 (these were determined by resolving local calculated wind axis lift and drag into the body axis system).

Correlation of the calculated body forces, with the experimental data taken from Ref. 27 and presented in Figures 10 through 12, is generally excellent. Trends of variation of the down (Cx) and cross (Cy) loads with onset flow angle are very well followed. Agreement with the UH-1 and UH-60 cases, Figures 11 and 12, is particularly good. For the AH-64 case in Figure 10, although the calculated trends follow the test data, the value of download at zero deg is substantially lower than the test value.

Searching out other sources of data for comparison produced an average value of Cx of 0.34 based on frontal area (Hoerner, Ref. 26) at a Reynolds number of 1 million. This is in the same range as the calculated value of 0.38. A value of download coefficient of 0.6 is very difficult to justify at the test Reynolds number since the flow would be supercritical. The difference is much larger than can be explained with local surface roughness conditions and any nonuniformity in the environment or slight sweep of the test specimen would tend to lower the measured drag. It would be useful to recalculate the section forces by integrating any surface pressure data that may have been taken during the test. In any further tests, surface flow visualization with tufts or oil flow would help confirm the two-dimensionality of the flow around the test article.

These calculated pressure distributions represent snapshots taken of what is in reality a time-varying phenomenon. In the wind tunnel pressure instrumentation with sufficiently high response rates would reflect the fluctuations in loading coming from the dynamic shedding. This is normally lost in the measuring system. Similarly, in the analysis, if short time steps had been chosen, dynamic detail would have been available. The result presented is essentially the time average. The time step shown does, however, show some upper/lower surface asymmetry, a residual effect of the asymmetrical behavior of the developing boundary layer during the start-up sequence. The effect is small, however, and is certainly within the scatter band of data from tests of cylinders at supercritical Reynolds numbers presented in Ref. 26. A comparison between the analysis and the Ref. 26 data is given in Figure 13. Agreement is very good, adding further credence to the analytic results.

Figure 14 shows samples of the progressive development of the pressure distribution on the UH-1 boom section as angle of attack is increased. Note particularly the sharp change in the profile as the separation moves forward between 30 and 40 deg on the UH-1 section.

Similar behavior can be seen in the related wake shapes presented in Figures 15. In these pictures, the wake has been caught at a time when the boundary layer separation has stabilized, but before the starting and adjustment wake has drifted completely downstream. The results at high angle of attack serve to illustrate the robust nature of the iteration procedure where the difference in height between the wake close to the body and that downstream is very marked.

<u>Sections with Covers and Strakes Added</u>

Addition of features such as tail rotor drive shaft covers and spoilers or strakes to the basic cross sections considerably complicates the process of determining local flow development and overall loads. The reason for this is that they introduce local separated flow and potential reattachments beyond the primary base separations. This means that the program must be able to track multiple boundary layers and their behavior following separation as shear layers, including not only reattachment but also possible detachment from the surface and convection downstream. To illustrate the flow processes involved and the features of the model, results from a run with the AH-64 boom with the drive shaft cover on are presented in Figures 16 and 17. The angle of attack is 30 deg. In this case an earlier run had identified a separation from the upper side of the cover and the wake reattaching to the main surface. For the purpose of the demonstration run, a piece of fixed wake was added, starting at the predicted separation point and stopping just short of the predicted impingement point on the main surface. This piece of wake carries the doublet value and gradient from the shedding panel. The upper surface boundary layer calculation was then forced to start just downstream of the "impingement" point. Figure 16 shows the wake configuration at time-step 6. The two rear wakes are being shed from computed separation points. The "bubble" wake is fixed in location and has just a small gap at the downstream end. The velocity distribution, Figure 17(a) shows the rapid deceleration (marked A) approaching the bubble followed by a region of very small but gradually decreasing velocities in the bubble and eventually a flow reversal (marked B) under the end of the wake. Figure 17(b) shows the corresponding pressure distribution. Inside the bubble region the small, gradually decreasing velocities, combined with an increasing $d\phi/dt$ term, result in a fairly constant pressure region (marked C). The Cp spike at the downstream end is partly a result of the peak reversed flow under the end of the fixed sheet and partly an erroneous $d\phi/dt$ value. The upper surface boundary layer calculation was started at the close-to-stagnation point (marked D) just beyond the impingement.

The large region of essentially stagnant flow on the windward side of the cover is very evident in this figure.

Adding the strake further complicates the picture with yet another embedded separation-reattachment zone. Figure 18 shows a typical condition for the full AH-64 configuration, again at 30 deg angle of attack, at a point in time when the drag has stabilized.

Figure 19 is a repeat of the force summary for the AH-64 tail boom with the results for the cover and strake on added at ± 30 deg and 0 deg angle of attack. Correlation of the calculated increments in forces due to the additions is good, but the overall correlation must be questioned until the questions regarding the measured drag of the basic section (relative to the large database available on cylinders) are answered.

<u>Application of the Program to Three-Dimensional Shapes</u>

All of the results presented above relate to the two-dimensional pilot code, program OSCAIR (25). Work is in progress on the full three-dimensional time-stepping analysis which is operational for cases with fully attached flows, for flows with leading-edge separations and for base flows where the separation remains fixed. Correlation to date has been excellent. Figure 20, taken from Ref. 28, shows the behavior of a delta wing at high angle of attack (leading-edge separations) performing a coning motion. Note the close agreement between the calculated and measured behavior, especially the crossover to an autorotative mode as angle of attack is increased.

Conclusions

A panel method has been developed which, when operated in a time-stepping mode, does not require the empirical inputs regarding wake behavior that limited the application of earlier methods for prediction. A pilot version of the program has been used to successfully predict drag/download on helicopter tail boom sections, including drive shaft and spoiler effects and a full three-dimensional, unsteady separated flow model is in an advanced stage of validation.

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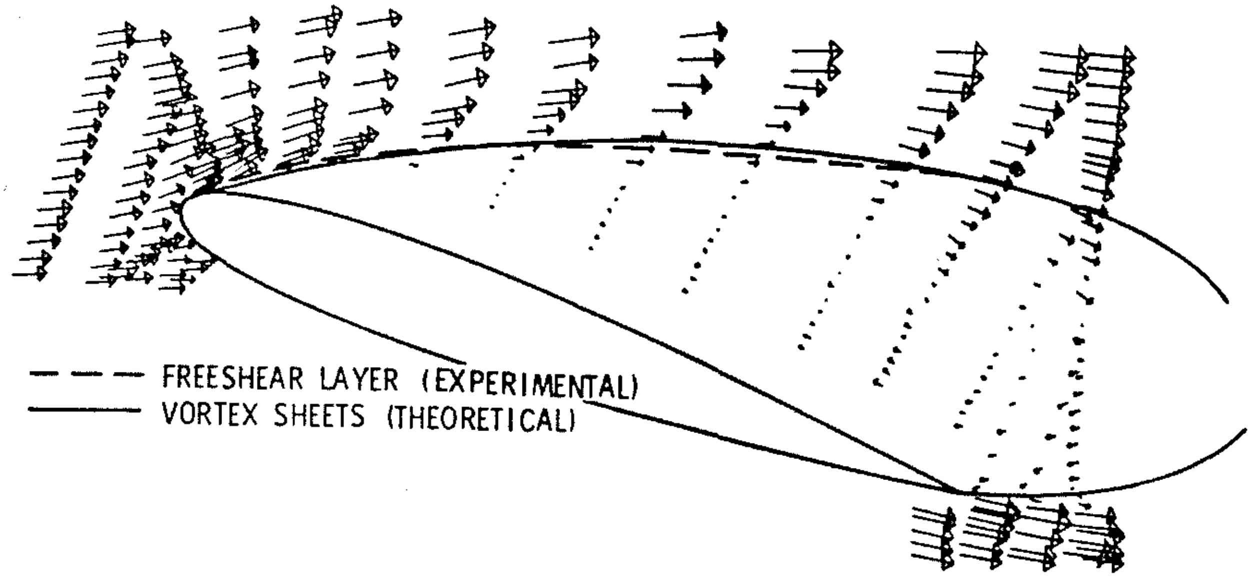


Fig. 1. Comparison of Experimental Free Shear Layer and CLMAX Calculated Vortex Sheet Centerlines (from (20)).

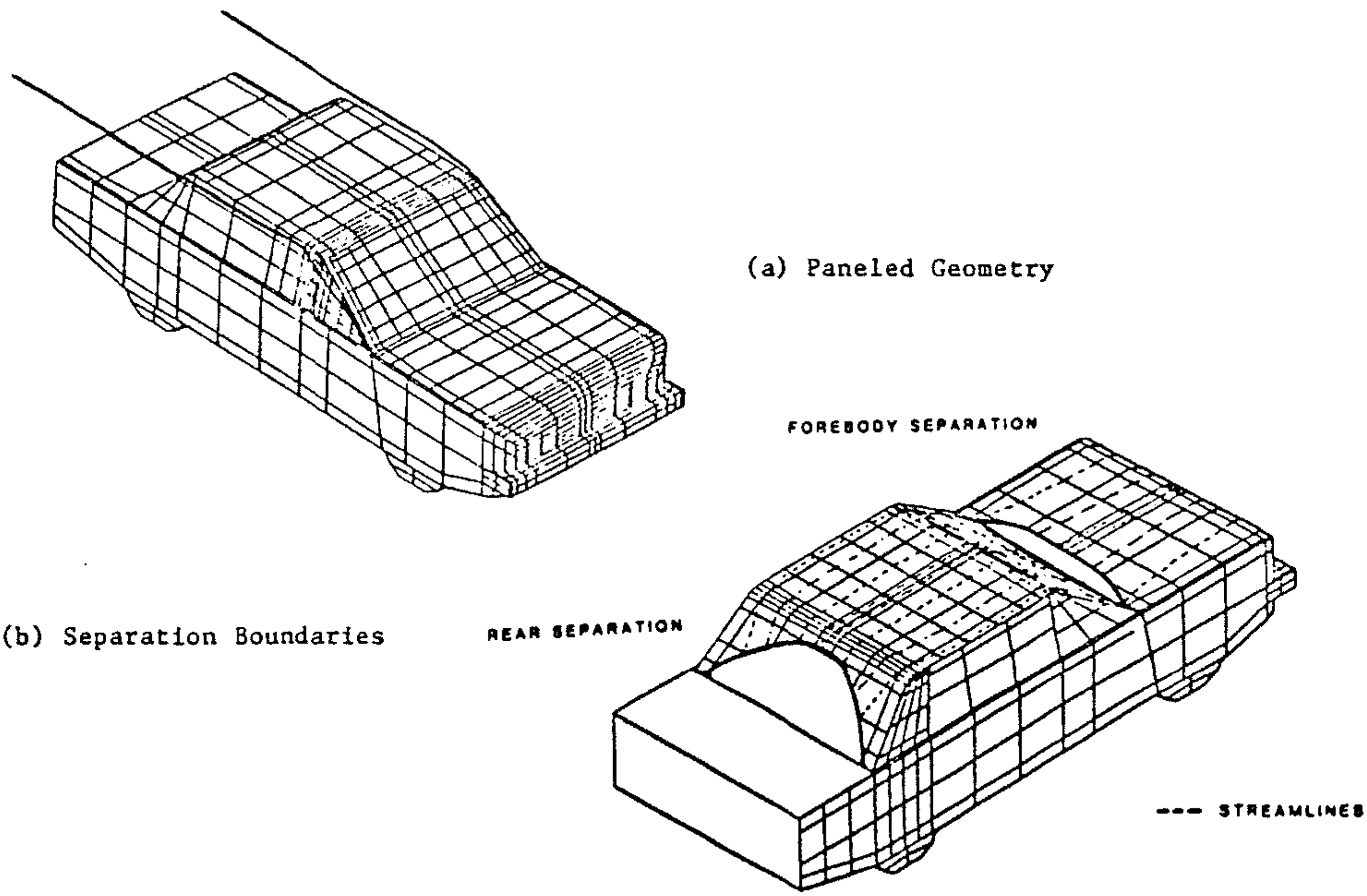


Fig. 2. Panel Model and Predicted Separation Boundaries for a Generic Auto.

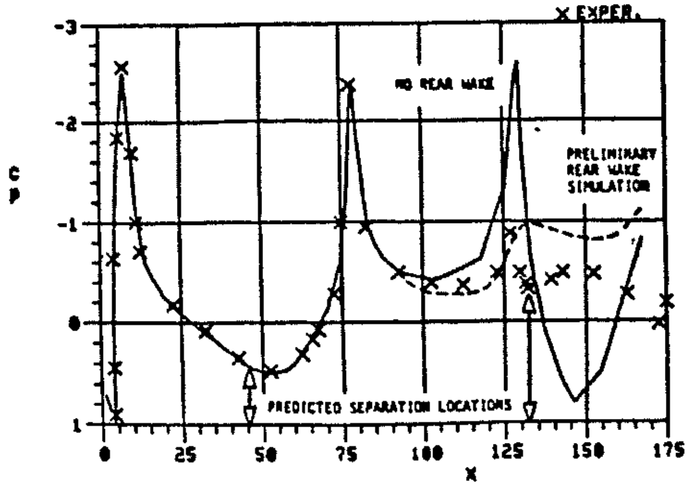


Fig. 3. Comparison of Predicted and Experimental Centerline Upper-Surface Pressures on a Generic Automobile.

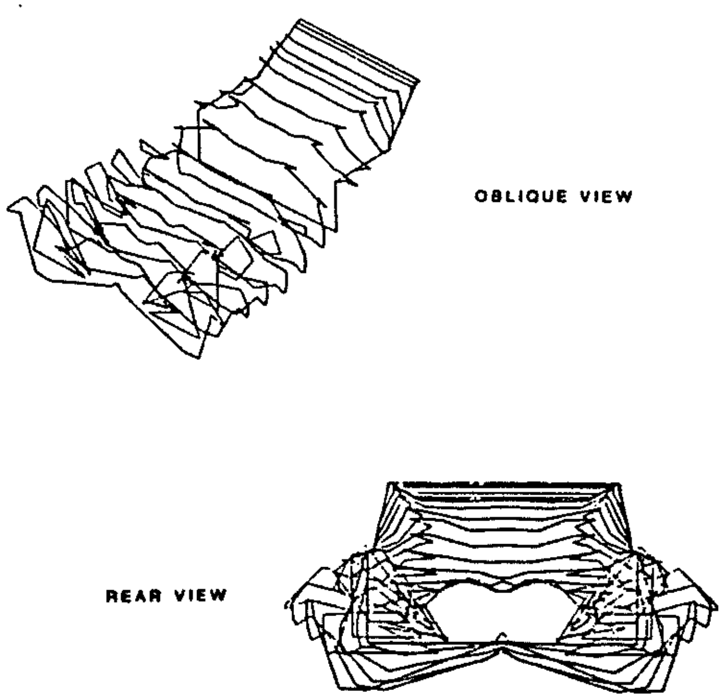


Fig. 4. Preliminary Calculated Rear Wake Shape on Generic Automobile.

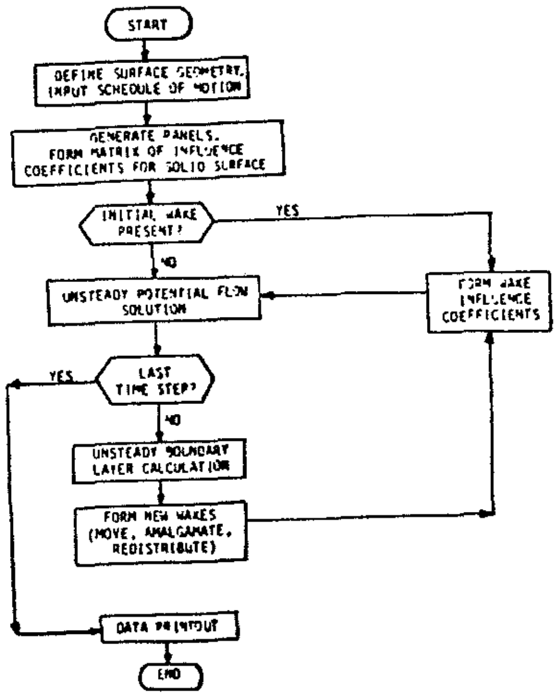
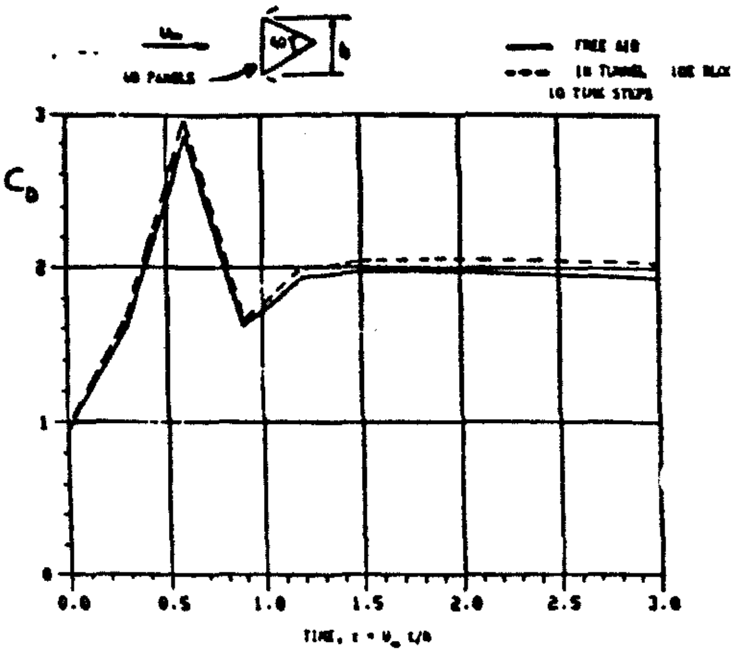
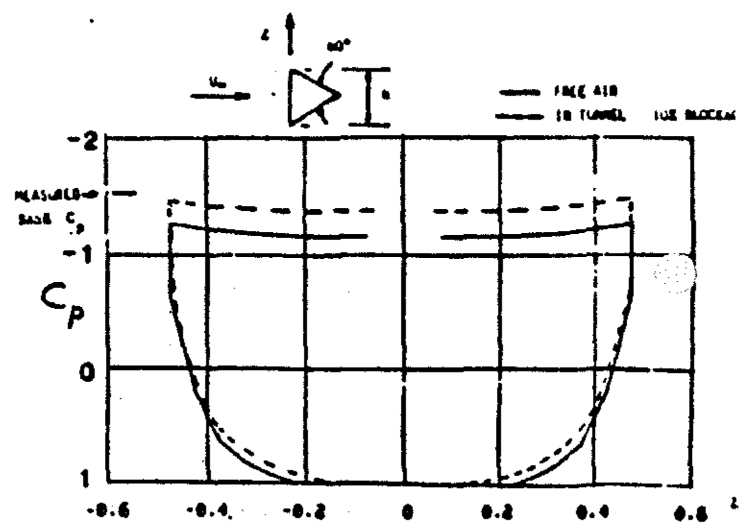


Fig. 5. Flow Diagram for Program OSCAIR.



(a) History of Drag.



(b) Calculated Pressure Distribution at τ = 3.0.

Fig. 6. Calculation on a Triangular Section Started Impulsively from Rest.

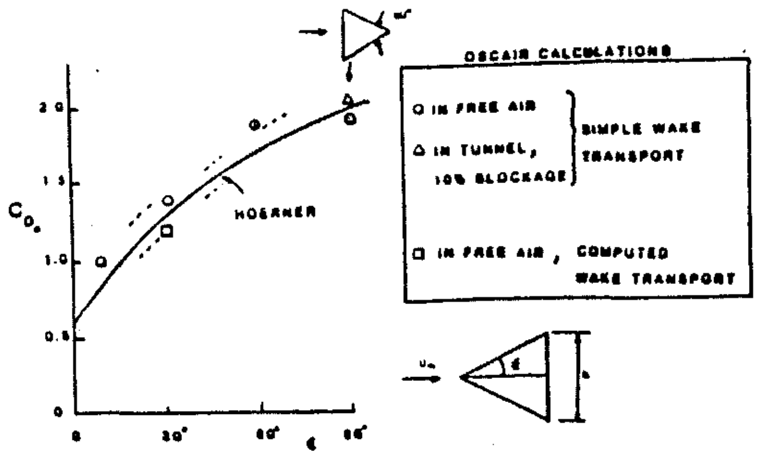


Fig. 7. Calculated Drag Coefficient of Two-Dimensional Wedges as a Function of Apex Angle.

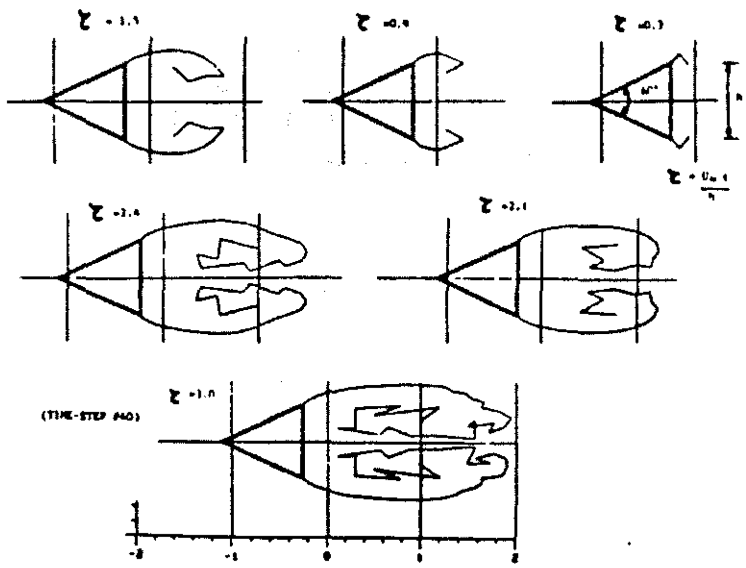
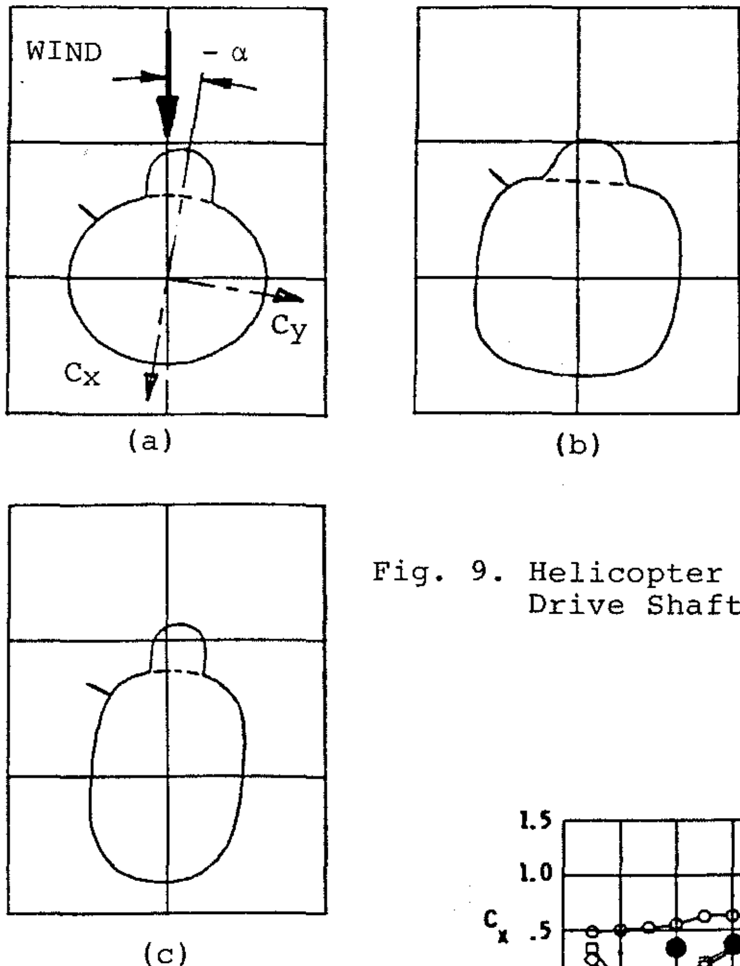


Fig. 8. Computed Wake Shapes for a 60° Wedge Started Impulsively from Rest.



(a) AH-64

(b) UH-1

(c) UH-60

Fig. 9. Helicopter Tail Boom Cross Sections with Drive Shaft Covers and Spoilers Added.

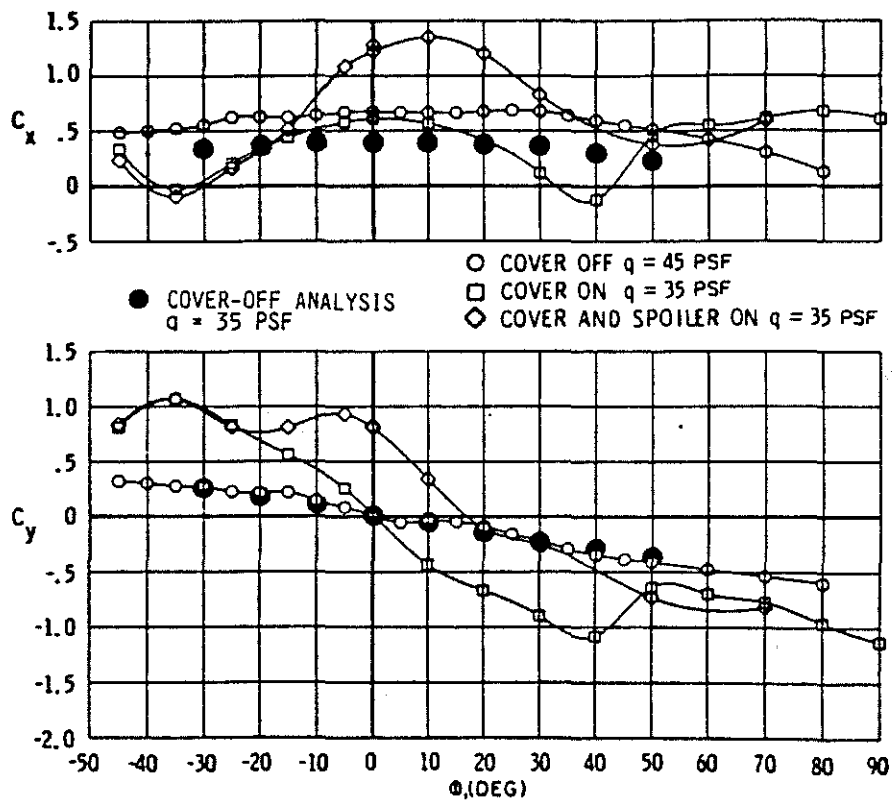


Fig. 10. Comparison of Calculated (Filled Symbols) and Measured (Open Symbols) Loads on AH-64 Tail Boom Sections. Experimental Data from Ref. 22.

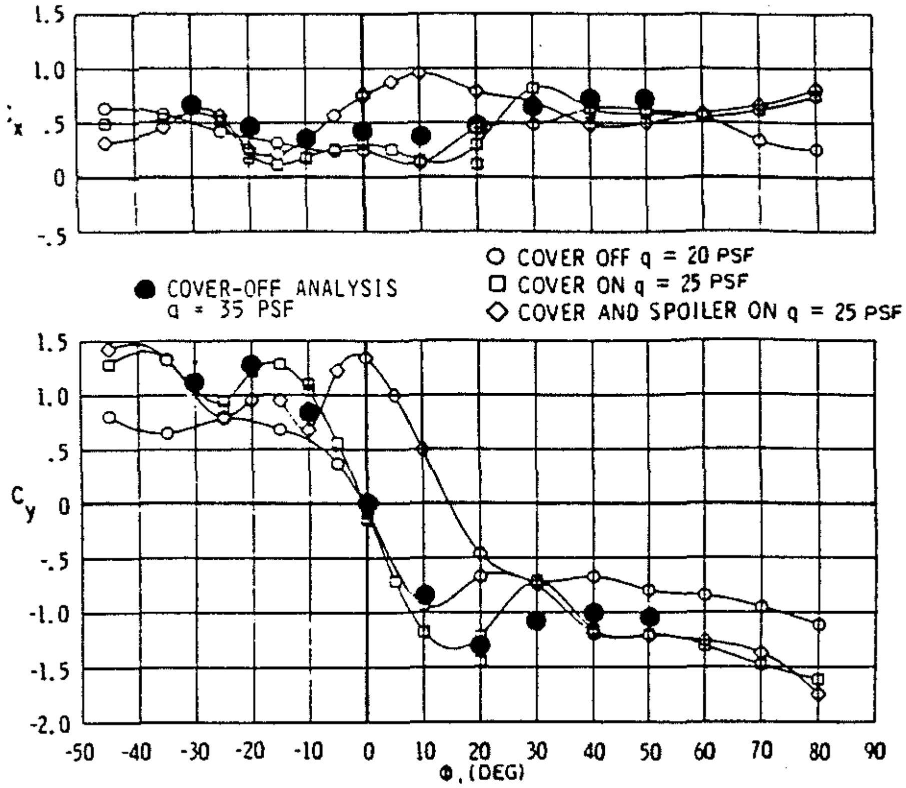


Fig. 11. Comparison of Calculated (Filled Symbols) and Measured (Open Symbols) Loads on UH-60 Tail Boom Sections. Experimental Data from Ref. 22.

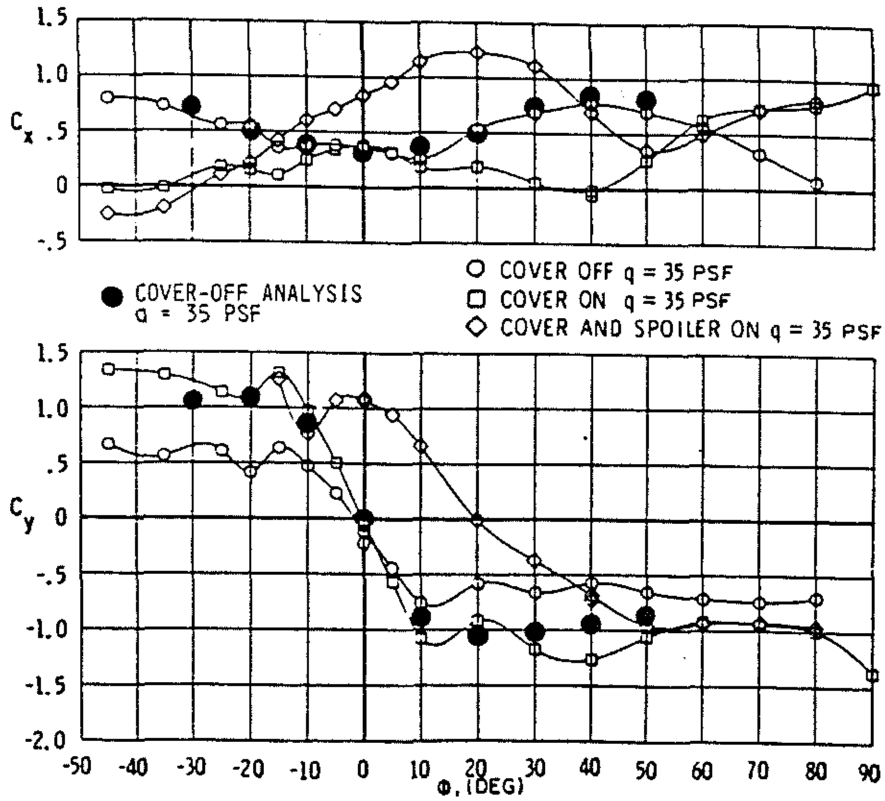


Fig. 12. Comparison of Calculated (Filled Symbols) and Measured (Open Symbols) Loads on UH-1 Tail Boom Sections. Experimental Data from Ref. 22.

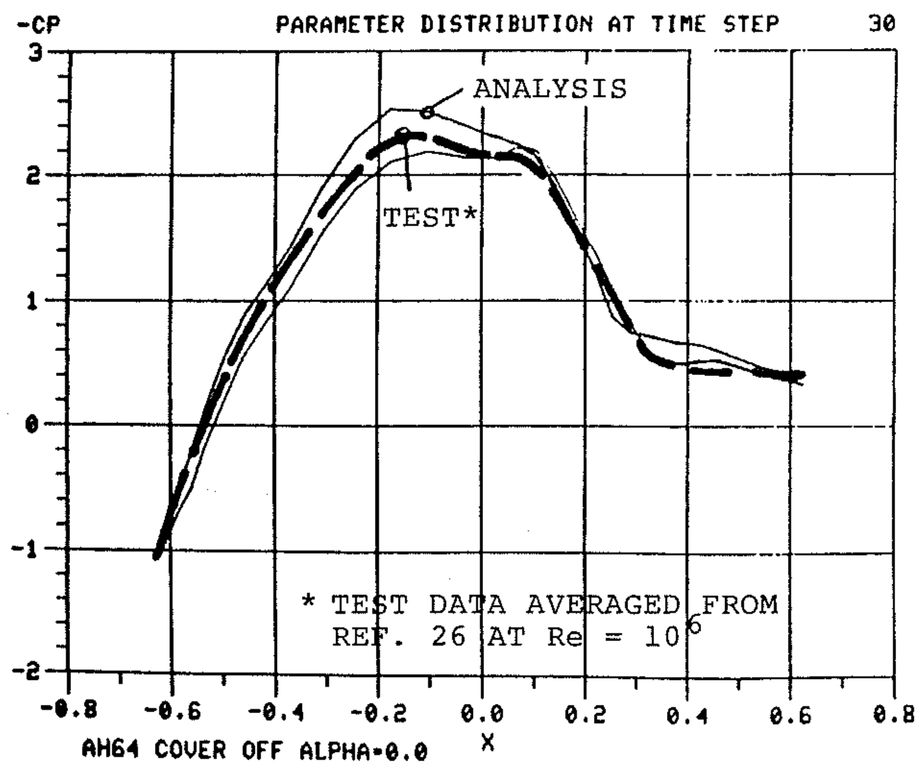


Fig. 13. Comparison of Calculated and Measured Pressure Distributions on Circular Cylinder (AH-64 Tail Boom) at $\alpha = 0°$, $Re = 10^6$.

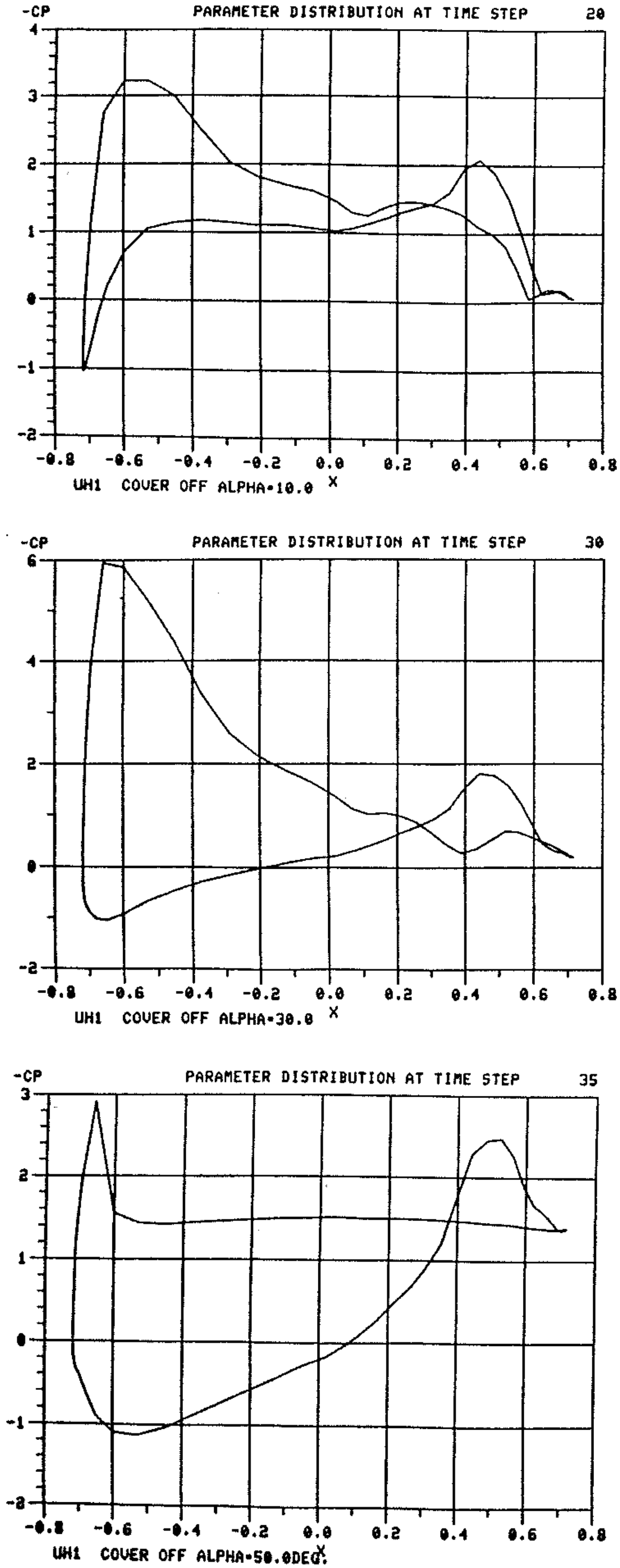


Fig. 14. Calculated Variation of Pressure Distribution with Angle of Attack; UH-1.

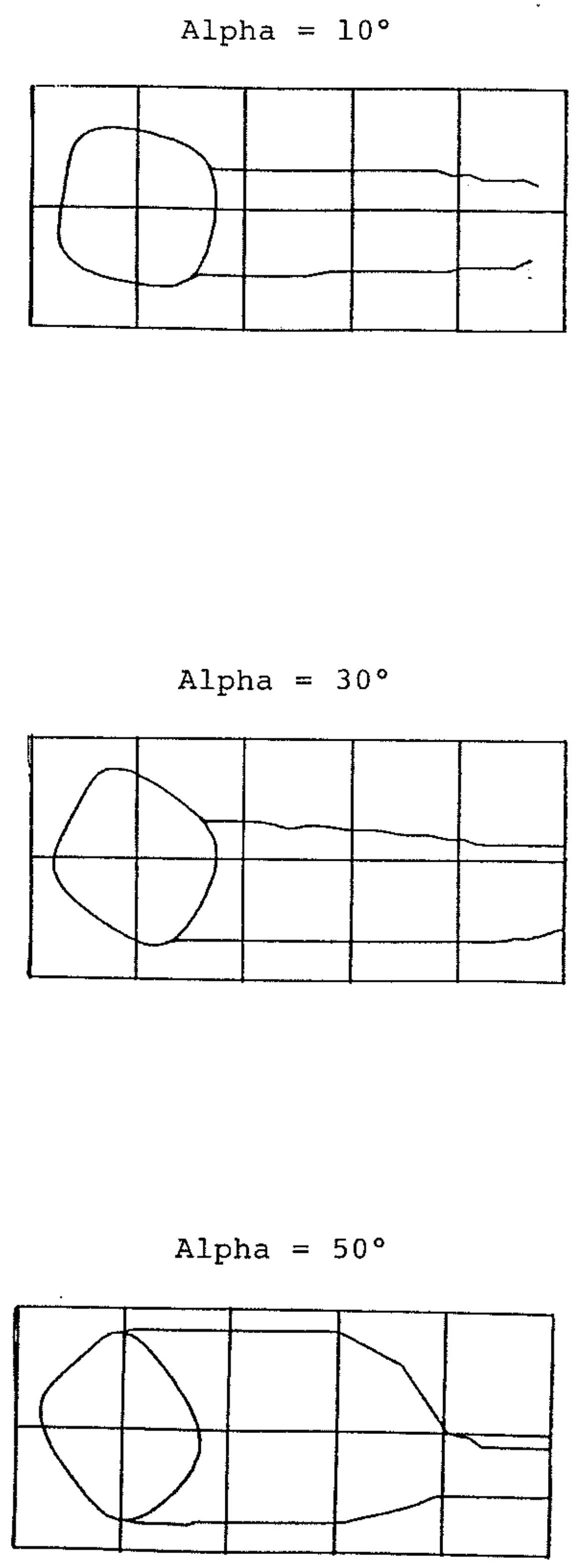


Fig. 15. Calculated Wake Shape Variation with Angle of Attack; UH-1.

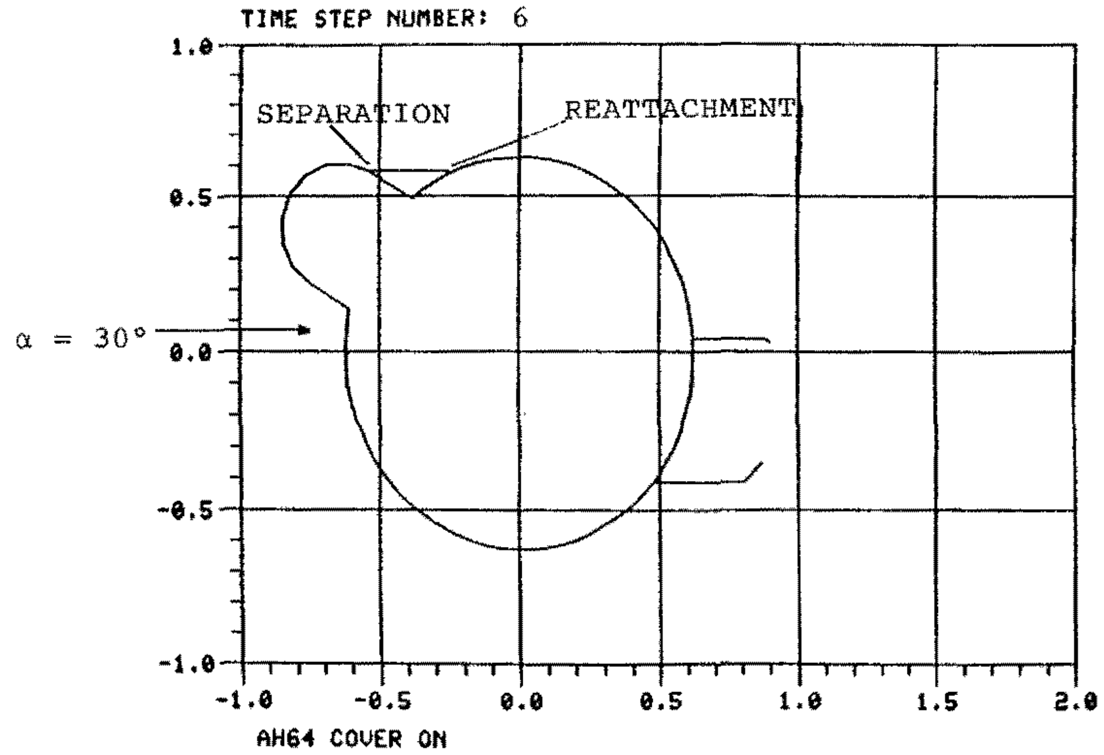


Fig. 16. Test Case to Validate Separation/Reattachment Modeling--
Wake Development at Time Step 6.

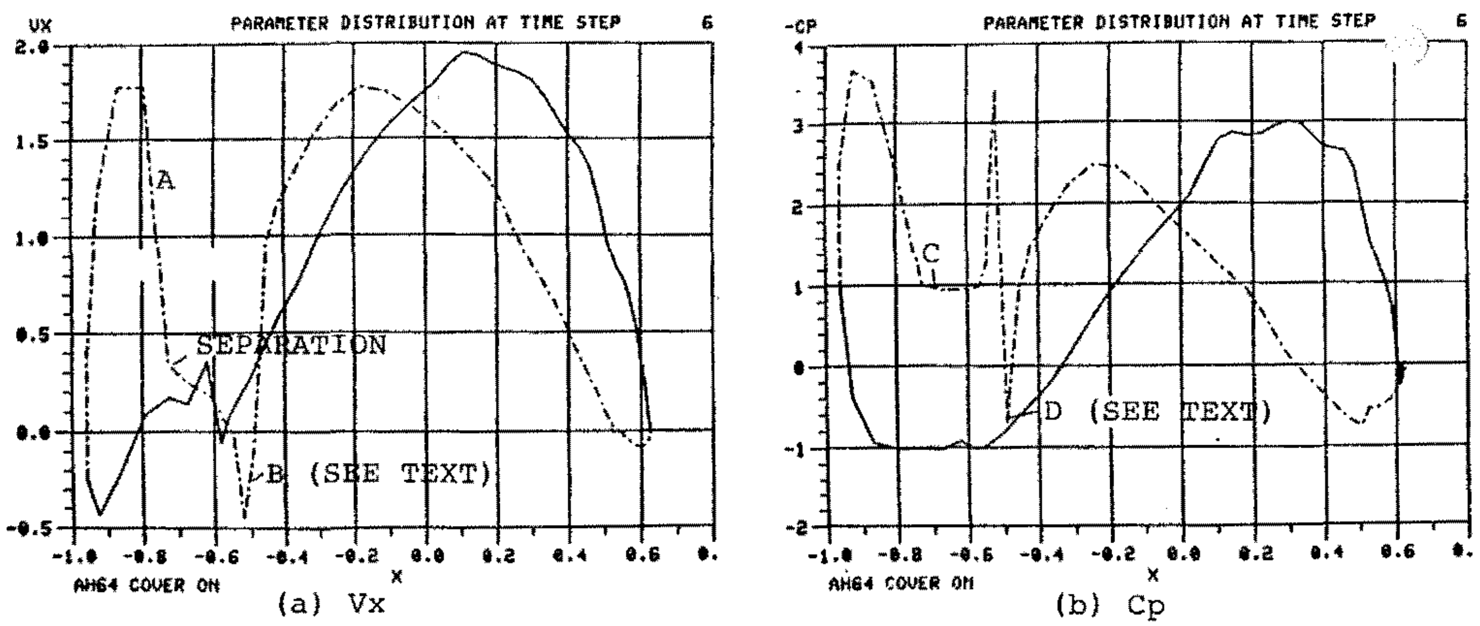


Fig. 17. Calculated Parameter Distribution Including Effect of Separation/Reattachment Zone.

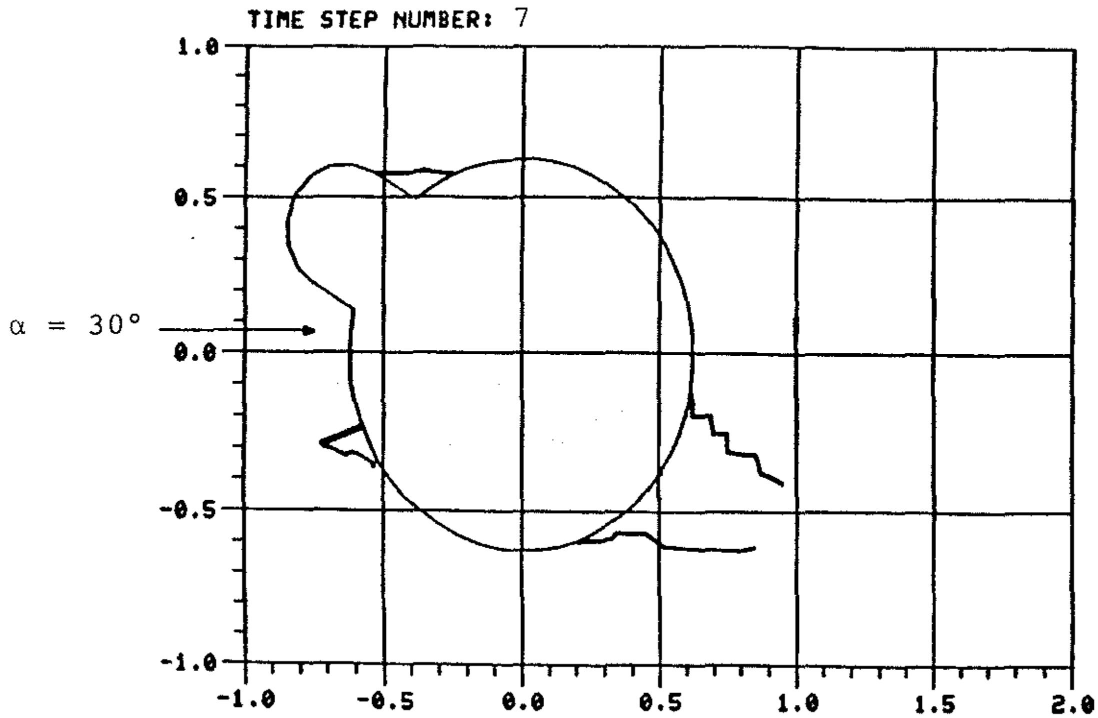


Fig. 18. Typical Wake--AH-64; Cover and Spoiler On.

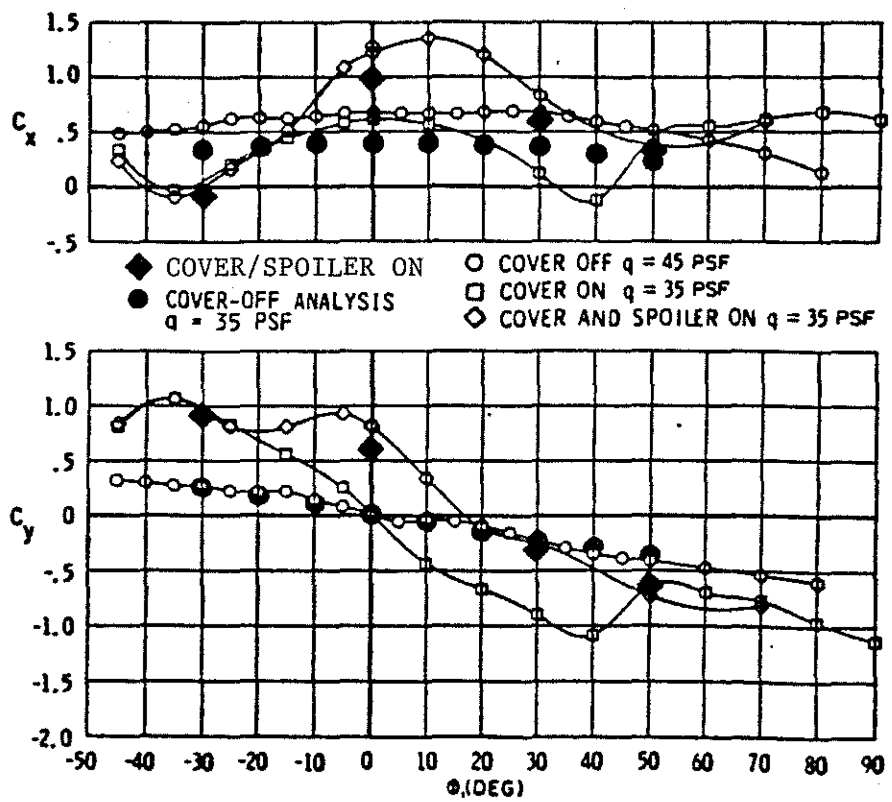


Fig. 19. Comparison of Calculated and Measured Loads--AH-64; Cover and Spoiler On.

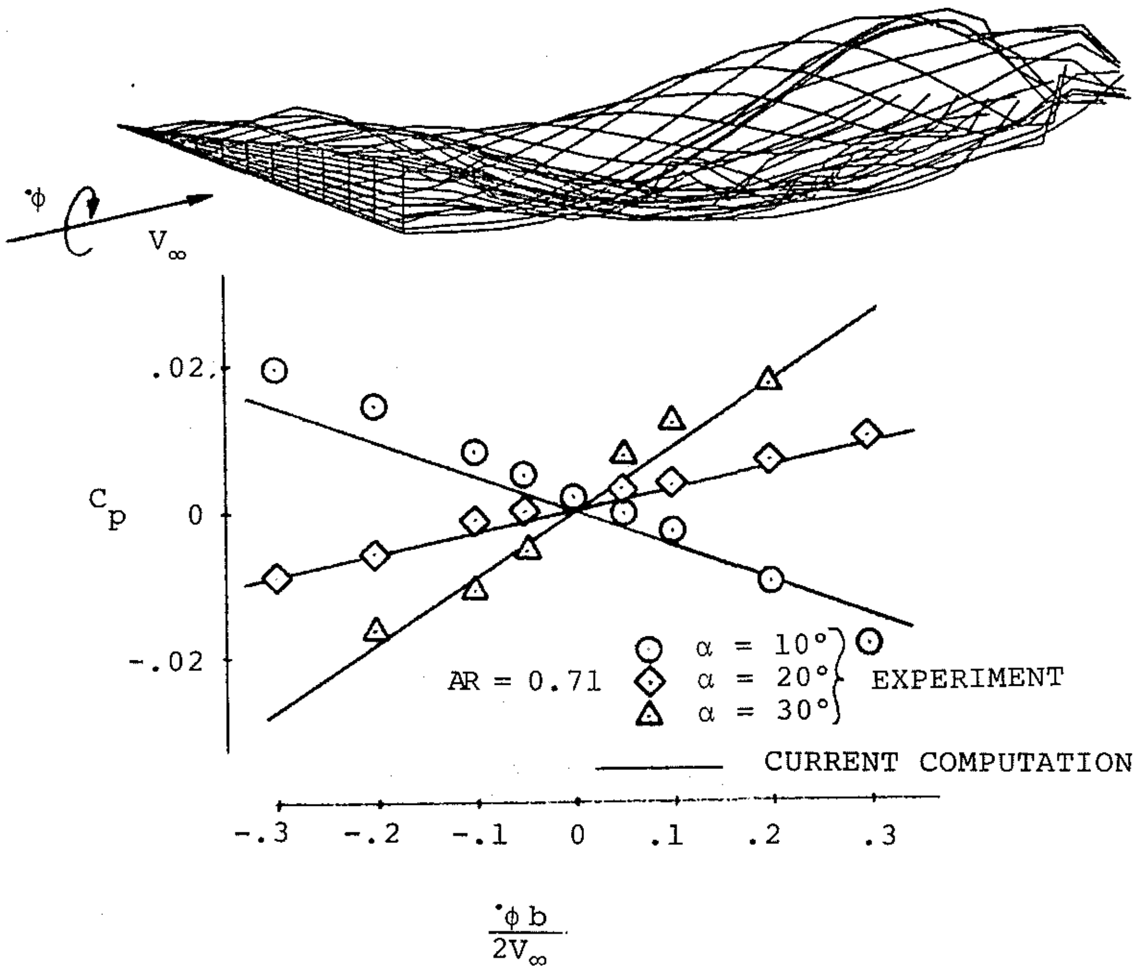


Fig. 20. Delta Wing Performing Coning Motion--Comparison of Analysis and Test.