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Application of a Higher Order Panel Method to Realistic Supersonic Configurations

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A higher-order panel method has been developed for the analysis of linearized subsonic and supersonic flow over configurations of general shape. This method overcomes many of the slender body limitations of present day programs in the analysis of supersonic configurations. The capabilities of this method are demonstrated through its application to the analysis of realistic supersonic cruise configurations. Comparisons are shown with experimental data and with results from other methods in current use. These comparisons demonstrate the unique capabilities of a major new software system called PAN AIR soon to be available as a general boundary value problem solver.

Nomenclature

c	= chord length
C_D	= drag coefficient
C_L	= lift coefficient
C_M	= pitching moment coefficient
C_P	= pressure coefficient
M_∞	= freestream Mach number
$\hat{n}$	= surface unit normal vector
p	= pressure
p_i	= isentropic pressure
p_2	= second-order pressure
$\bar{v}$	= perturbation velocity
$\bar{V}$	= total velocity
V_∞	= freestream velocity magnitude
$\bar{V}_\infty$	= freestream velocity
$\bar{w}$	= perturbation mass flux vector
$\bar{W}$	= total mass flux vector
$\hat{x}$	= unit vector along x-axis
α	= angle of attack
β	$=\sqrt{1-M_\infty^2}$
Δ	= jump in quantity across singularity surface or line
γ	= specific heat ratio
η	= span fraction
μ	= doublet strength
ρ	= fluid density
ρ_∞	= freestream fluid density
σ	= source strength
ϕ	= perturbation potential
ϕ_x, ϕ_y, ϕ_z	= gradient of perturbation potential
∇	= gradient operator

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Index categories: Aerodynamics; Computational Methods; Supersonic and Hypersonic Flow.

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I. Introduction

THE ability to analyze the linearized supersonic flow over configurations of general shape is of great value in the design process. Accurate potential flow analysis, when combined with appropriate pressure level and gradient constraints (to account for real flow characteristics), can greatly enhance the design of efficient supersonic cruise vehicles.[1] Methods in general use today, which have been so successfully applied to slender supersonic transport configurations,[2] are highly restrictive as to the generality of the geometry they are capable of handling. In these methods,[3-6] the geometry is linearized into "thin" wing and "equivalent body of revolution" representations making the simulation of canopies or inlets difficult, if not impossible. Not surprisingly, the resulting designs seem to adhere quite closely to the geometric limitations of the analysis methods as shown in Fig. 1. The application of these methods to not-so-slender type configurations, such as fighters, has not been as successful.

Newer methods developed by Woodward[7] and Morino et al.[8] overcome some of these limitations. Woodward's newer method, USSAERO, has seen extensive application in the last few years. The method features a source paneled fuselage representation to allow more general body simulation but still lacks the ability to simulate inlets or complex multibody configurations. Little is known on the application of Morino's method to any complex configuration, although it seems to work well on simple configurations. Moreover, these methods can show some sensitivity to the choice of paneling and location of control point. This is due to the method's assuring values of source, doublet, or vorticity which are discontinuous between panels. In linearized supersonic flow such discontinuities introduce infinite singularities in velocity.

The evolution of supersonic transport design into blended wing-body configurations[9] and the military interest in supercruiser applications[10] has heightened the need for a reliable analysis method capable of dealing with these geometric and numerical problems. A higher-order panel method overcoming these difficulties has been developed by Ehlers et al.[11] This method, which is valid for both linearized supersonic and subsonic flow, is being incorporated under NASA and Air Force contracts into a general boundary value problem solver called PAN AIR. A developmental version has also been available in pilot code form[12] during the last year. During this time the method has been applied to a variety of

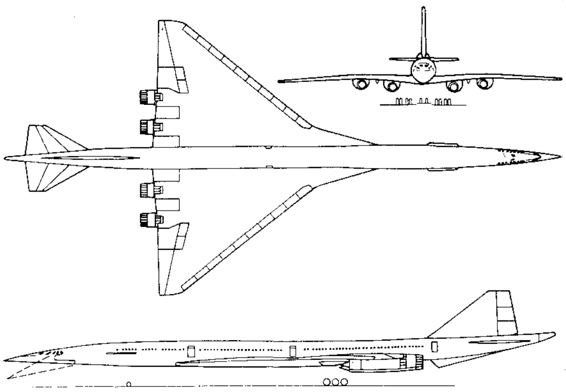

Fig. 1 Boeing 2707-300 Supersonic Transport.

aircraft configurations. Results of these studies, including comparisons with experimental data and with other theoretical methods, will be presented to illustrate the capabilities of this higher order panel method.

The remainder of this paper is organized as follows: In section II, a brief description of the theoretical method is given. Section III discusses the application of the method to a lightweight experimental supercruiser fighter configuration. Comparison of results are presented for a Mach number of 1.2. In section IV, a similar presentation is made for a long-range reconnaissance strike vehicle analyzed at $M=3.0$. In this section a comparison showing the effects of paneling density is also shown. Section V illustrates application of the method to complex flows associated with weapons carriage and delivery. Concluding remarks are presented in section VI.

II. Description of the Method

The solutions presented in this paper are based on the higher-order panel method developed by Ehlers et al.[11]. A complete description of the method is available in Ref. 13. The advanced panel pilot code[12] which was used for the numerical development and validation of the method, was used to obtain the results presented. The advanced panel pilot code is currently available through COSMIC (COSMIC, Barrow Hall, University of Georgia, Athens, Georgia, 30602.) as program ARC-11278. The higher-order panel method is also being incorporated into a user-oriented software system, PAN AIR (Panel Aerodynamics), being developed under contract to NASA and the Air Force Flight Dynamics Lab. Initial deliveries of PAN AIR are scheduled in 1979.

Basic Concepts

The method is intended to solve a variety of boundary value problems in steady subsonic or supersonic inviscid flow. The solutions are governed by the classic Prandtl-Glauert equation for linearized compressible flow

$$\beta^2\phi_{xx}+\phi_{yy}+\phi_{zz}=0 \qquad \beta^2=1-M_\infty^2 \tag{1}$$

The definition of impermeability for the boundary conditions and the pressure formulation appropriate to Eq. (1) are open subjects. Several options are available to the users in the pilot code and PAN AIR. We prefer the mathematically natural choice of zero normal mass flux and the second-order pressure formula. The total mass flux vector $\bar W$ is defined as

$$\bar W=\rho_\infty \bar V_\infty+\bar w \tag{2}$$

where $\bar w$ is the perturbation mass flux vector defined by

$$\bar w=\rho_\infty(\beta^2\phi_x,\phi_y,\phi_z) \tag{3}$$

To first-order in perturbation quantities, $\bar w$ is equal to ρv[14]; hence, the impermeable surface boundary condition can be expressed by

$$(\bar W\cdot\hat n)=0 \tag{4}$$

where $\hat n$ is the surface normal. Equation (1) rewritten as $\nabla\cdot\bar w=0$ expresses conservation of mass, and Eq. (4) then guarantees that even if the configuration is such that the assumptions used to derive Eq. (1) are violated locally, there is still no net production of fluid.

The second-order pressure formula is

$$p_2=p_\infty-[\rho_\infty(\bar V_\infty\cdot\bar V)+(1/2)(\bar V\cdot\bar W)] \tag{5}$$

and the isentropic formula is

$$p_i=p_\infty+\frac{\rho_\infty V_\infty^2}{\gamma M_\infty^2}\left\{\left[1-\frac{\gamma-1}{2}\cdot\frac{M_\infty^2}{V_\infty^2}(V^2-V_\infty^2)\right]^{\gamma/\gamma-1}-1\right\} \tag{6}$$

In the solutions shown, the second-order pressure Eq. (5) is used for the force and moment calculations because it is consistent with conservation of momentum principles. For the pressure distribution comparisons, the isentropic pressure, Eq. (6), is used. Experience has shown that it agrees closely with the second-order equation in the range where linear theory is valid and tends not to degenerate fast as the second-order equation outside the linear theory range.

Panel Solutions

The configuration is represented by a distribution of source and doublet singularities. These singularities, each of which is a solution to Eq. (1), may be placed on the actual configuration surface or may be used to represent components such as wings in linearized "thin wing" fashion by a single surface. A panel method is used for the solution. Here the configuration surface is divided up into networks. A network is defined as a smooth portion of the configuration which has subsequently been divided into panels, each panel representing some source and doublet distribution. A linear source and quadratic doublet distribution over the panels is defined in terms of the values of the singularities at the centers of each panel and neighboring panels by a system of spline type polynomials. Boundary conditions are applied at discrete points associated with each network. Each network is logically independent in that it contributes as many equations as unknowns to the overall boundary value problem. The required integrals are evaluated in closed form, and a set of linear equations results to be solved for the required singularity strength parameters. Once this is accomplished, the potential and velocity fields are known. The pressure field can then be calculated from an appropriate pressure-velocity relationship, and forces and moments calculated by integration.

During the development of the method it became apparent that to insure sufficient accuracy and stability for supersonic flow solutions, continuity of both the doublet strength and geometry must be maintained. This was necessary because such discontinuities introduce infinite singularities in velocity which do not decay with distance. A panel system with all contiguous edges was obtained by dividing the basic four corner non-planar panel into eight triangular subpanels as shown in Fig. 2. A quadratic doublet distribution is applied over each triangular subpanel in such a way that the doublet strength is continuous at panel edges, leading to a nine parameter spline for the complete panel.

Boundary Conditions

In order to eliminate any internal disturbances in closed configurations, combined source and doublet panels are used along with boundary conditions requiring the perturbation

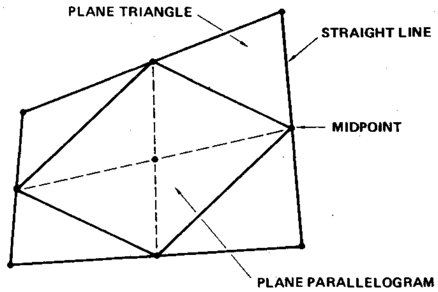

Fig. 2 Piecewise flat panel.

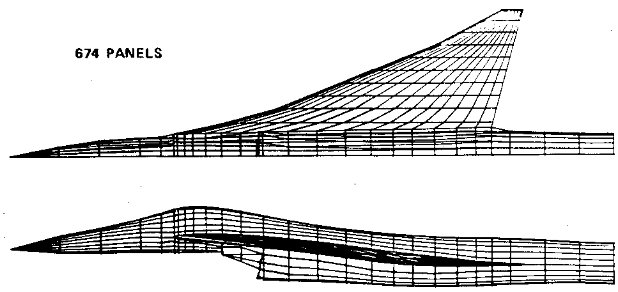

Fig. 3 LES 216 configuration paneling.

potential to be zero inside of the closed configuration.[8] The source distribution strength σ is set equal to the negative of the normal component of the freestream to the surface,

$$\sigma = -\bar{W}_{\infty} \cdot \hat{n} \tag{7}$$

the doublet distribution is determined by setting the perturbation potential ϕ equal to zero on the inside surface.

$$\phi = 0 \tag{8}$$

This eliminates all disturbances in the interior flow and any related numerical interference with the velocity distribution on the exterior surface. These boundary conditions are mathematically equivalent to the impermeability condition given by Eq. (4).

Thin wings simulated with linearized boundary conditions can also be represented by sources and doublets distributed over the planar planform. The thickness distribution is represented by sources while camber shape (and/or angle of attack) is given by doublets.

Solid surfaces represented by sources and doublets must be inclined at angles less than the Mach angle. Singularity surface for panels at angles with respect to the freestream greater than the Mach angle do exist theoretically. These "superinclined" panels, which require initial conditions rather than boundary conditions, are useful for closing inlets or representing boundary conditions on exhausts. On inlets, these superinclined panels simply make the perturbation flow vanish, preventing any internal disturbance.

III. LES 216 Supercruiser Analysis

The higher-order panel method, in pilot code form, was applied to the analysis of a lightweight experimental supercruiser configuration, Boeing model LES 216. Here, a supercruiser is defined as an aircraft designed for efficient supersonic cruise on dry or unaugmented thrust. The configuration representing an extremely small fighter (12,000-15,000 lb gross weight) features a rather large canopy and a chin inlet. These two features are essentially impossible to simulate well using a method which relies on a slender or "equivalent" body of revolution type fuselage representation.

Configuration Modeling

The paneling scheme employed in the pilot code is shown in Figs. 3 and 4. The wake paneling, shed from the wing and fuselage trailing edges, is not shown for clarity. The figures illustrate some of the unique modeling capabilities of this new method. The canopy and the inlet geometries have been faithfully represented. The configuration was modeled without its wing-mounted vertical fins in order to match a wind tunnel configuration for which pressure data were available. The chin inlet was simply modeled by placing a barrier on the face of the inlet. This barrier consisted of a "superinclined" network on the lower part of the inlet face. Here the incoming flow simply disappears. The ramp portion of the inlet barrier is subinclined and, therefore, the mass flow

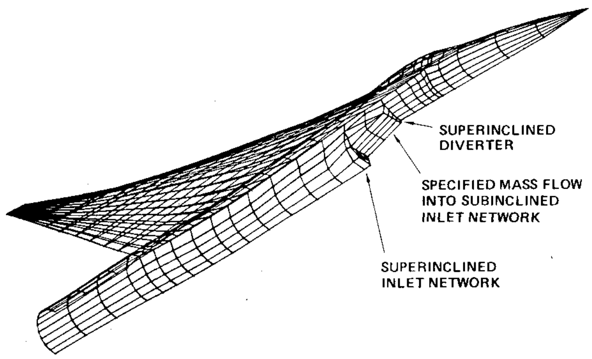

Fig. 4 Inlet and diverter details.

into the barrier must be specified. The amount of mass flow specified controls the amount of inlet spillage. It would have been possible to simulate the finely detailed geometries in the inlet area, but the cost of using additional numbers of panels required for this was not justified in this example.

Another area in which the modeling has been simplified is that of the diverter. The diverter is simply a splitter plate placed between the fuselage and the inlet to protect the inlet from ingesting any fuselage boundary layer air. The flow entering the diverter is split along the plane of symmetry by an ogee-shaped wedge and spilt out the sides of the diverter between the fuselage and the inlet. Due to the large local viscous effects, together with multiple waves that would occur in the narrow channel between the diverter plate and the fuselage, detailed modeling of the internal geometry of the diverter was judged not to be worthwhile. As in the case of the inlet, the modeling of the diverter was greatly simplified by placing a "superinclined" network at the face of the diverter to capture the incoming flow. This flow could then be spilt out the sides of the diverter by specifying a mass flow out of those panels.

It should be emphasized that the simplifications made to the modeling of the inlet and the diverter were made with the desire to adequately simulate their effects on the rest of the configuration. It was not the intention of this analysis to get detailed local flow characteristics in these regions. To do so is probably out of the realm of inviscid linearized supersonic flow theory. The diverter flow is highly viscous dominated, while the internal inlet flow is strongly dependent on the shock wave formation caused by the inlet ramp. In terms of simulating the external effects of these features on the rest of the configuration it is felt that the modeling used was not only adequate, but also very cost effective in terms of panels used.

The total configuration representation consisted of 674 panels and 1195 unknown singularity parameters. Potential boundary conditions discussed in section II and initial conditions for the superinclined panels were used. The CPU time for a solution consisting of one Mach number and one angle of attack was 700 s on a CDC 7600 computer.

It should be pointed out that the computer resources for a solution are usually only a small fraction of the total analysis cost. The engineering cost of initial input data preparation frequently greatly exceeds the computational cost. The geometry input must represent the configuration you are trying to analyze. The biggest problem usually encountered is determining in sufficient detail what the configuration actually looks like. Tables of wing ordinates and fuselage cross-sectional areas by themselves are not always sufficient. Details of not only the fuselage cross sections, but also the actual wing-body intersection must be known. The network geometry for the configuration of this example was extracted from the numerical loft used originally to define the wind tunnel model for numerical machining. The Boeing Geometry Control System (GCS) program was used for most of the geometry definition, extraction, and paneling. Other smaller programs were necessary to manipulate the data, to work out intersections between surfaces, and to plot the paneling for checking purposes. The pilot code itself was used to check on proper abutments between the 32 paneling networks that were necessary for the analysis. But once the model has been assembled and checked out, it can be used throughout the Mach number range from subsonic to supersonic with only changes to the inlet and diverter boundary conditions.

LES 216 Results

Comparison of force data results from the advanced panel pilot code, the Middleton "Mach Box" method,[4] the Woodward constant pressure panel method,[5] and experimental data[15] are presented in Figs. 5 and 6 for a Mach number of 1.2. In the pilot code results, the forces on the inlet and diverter barriers were ignored. Overall, the pilot code appears to best agree with the experimental data. Looking in detail at the lift curve results in Fig. 5, both the Woodward and Middleton solutions yield a higher lift curve slope than the pilot code. The apparent better agreement in slope of these codes with experimental data may be fortuitous. Examination of the surface pressure data as a function of angle of attack indicates the definite onset of a leading edge vortex at around two degrees angle of attack. The resulting additional lift from the vortex cannot be directly predicted from these attached flow solutions. A method for predicting detailed forces and surface pressure distributions in the presence of a leading edge vortex on simplified planforms is presented in Ref. 16.

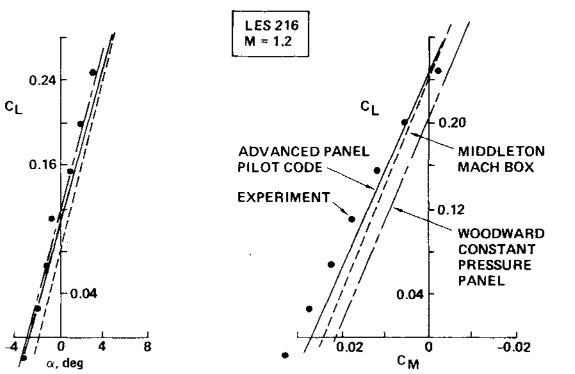

Fig. 5 Longitudinal characteristics.

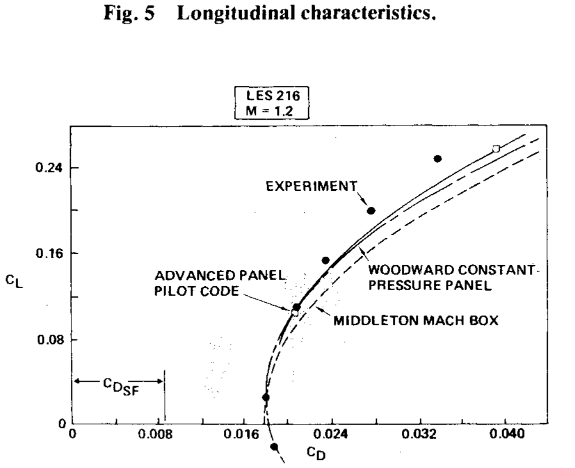

Fig. 6 Drag polar.

Drag polar comparisons are shown in Fig. 6. For these comparisons, an estimate of the skin friction drag for the test conditions was added to the theoretical results in each case. All the methods predict a significantly more closed drag polar than was measured. The main reason for this is the failure of the theoretical methods to adequately capture the portion of leading edge suction developed by the model. The Middleton and Woodward methods employ a thin wing representation which does not directly account for the leading edge thrust. The pilot code, which employs a thick wing representation, fails to capture most of the thrust because of an overly simple pressure integration scheme used to evaluate the forces. The upcoming PAN AIR program, which will embody the higher-order panel method of the pilot code, will have a more accurate singularity integration method to better evaluate the forces. Perfect agreement with experiment on drag will still be elusive because only a portion of the full theoretical leading edge force is ever realized experimentally.[17,18]

Surface pressure distributions, from the Woodward and the pilot code, are compared in Fig. 7. The experimental data is rather scarce, in that only 24 pressure taps were available on the wind tunnel model. The trends between the Woodward constant pressure panel code, the pilot code, and experimental data are, nevertheless, quite clear. The simplified representation necessary in the Woodward case completely fails to predict the correct surface pressure levels, while the exact surface presentation of the pilot code yields results that are in considerably better agreement with the experimental data.

IV. Recce-Strike Model Analysis

Another case in which the advanced panel pilot code was applied was the analysis of a long range reconnaissance strike aircraft. This aircraft, in the 100,000 lb weight class, is designed to cruise at Mach 3.0. The vehicle featured two podded engines and wing-mounted vertical fins, which were not included in this analysis.

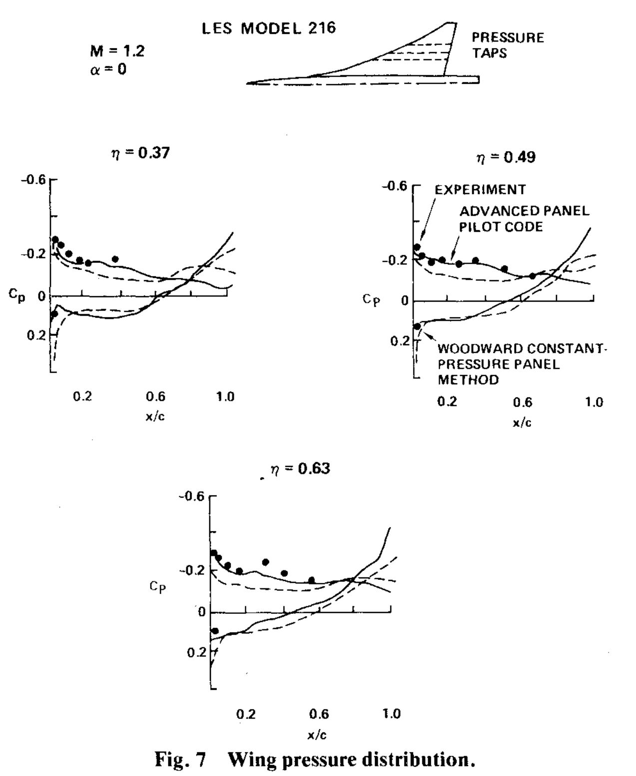

Fig. 7 Wing pressure distribution.

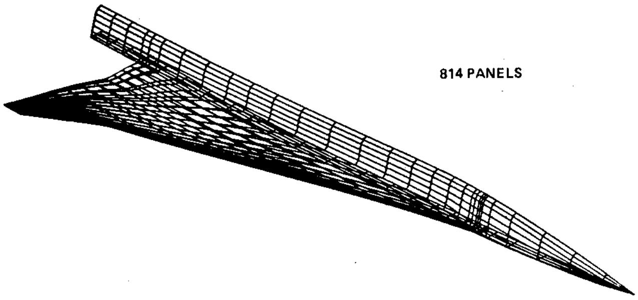

Fig. 8 Recce-Strike configuration paneling.

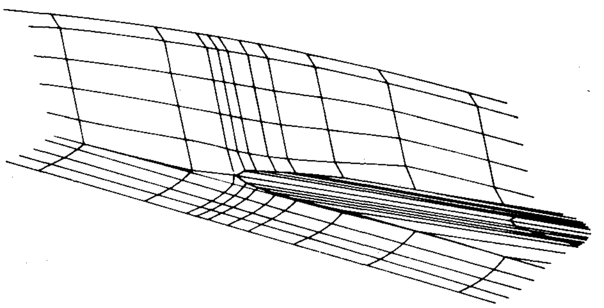

Fig. 9 Wing/body intersection details.

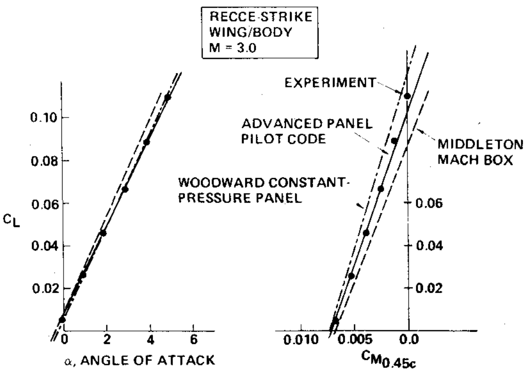

Fig. 10 Longitudinal characteristics.

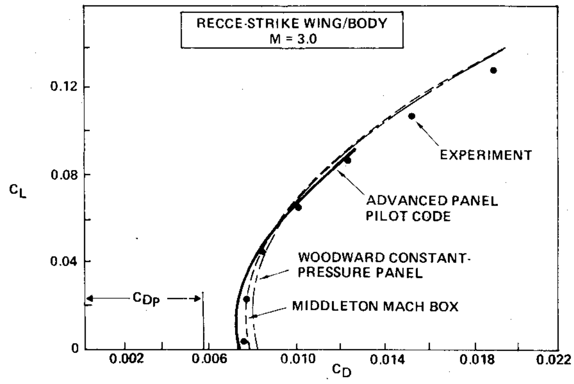

Fig. 11 Drag polar.

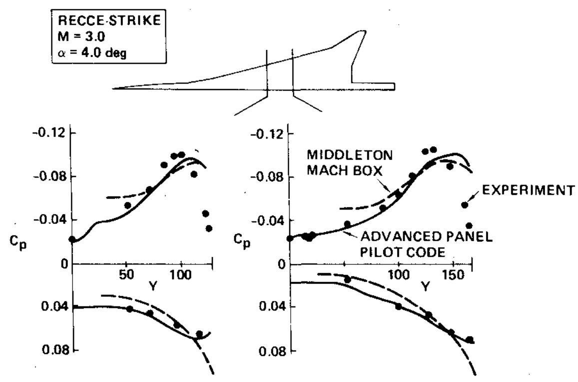

Fig. 12 Wing surface comparisons.

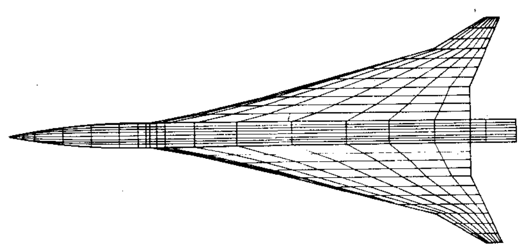

Fig. 13 Sparse paneling for Recce-Strike configuration.

Configuration Modeling

The paneling scheme, used to represent the configuration, is shown in Figs. 8 and 9. The wake paneling again has not been shown for clarity. Note the triangular shape fuselage cross-section and the wing-body intersection details. A total of 810 panels were used to represent the configuration, resulting in 1388 singularity unknowns. Potential boundary conditions were used for the entire configuration. The CPU time for a solution, consisting of one Mach number and one angle of attack, was 914 s on a CDC 7600 computer.

Recce-Strike Results

Comparison of force data results from the advanced panel pilot code, the Middleton "Mach Box" method, the Woodward constant pressure panel method, and experimental data[19,20] are presented in Figs. 10 and 11. For the drag polar comparisons shown in Fig. 11, an estimate of skin friction drag and grit drag for the condition tested was added to each of the theoretical results. The grit drag is the drag of the carborumdum grits placed near the surface leading edges to trip the boundary layer. These comparisons show the pilot code to be in best agreement with the experimental data. Little or no leading edge thrust is developed at these high Mach numbers to complicate the drag comparison.

Surface pressure distribution comparisons between the Middleton "Mach Box" method, the pilot code, and experimental data are shown in Fig. 12. These comparisons show the failure of both theoretical methods to properly predict the upper surface leading edge pressures correctly. Experimental pressure data and surface oil flow visualization data indicated some type of nonlinear disturbance along the wing leading edge. The exact nature of this disturbance could not be determined. Inboard of the leading edge, as the side of the body is approached, the pilot code shows significantly better agreement with the experimental data than does the Middleton method. As was the case on the LES 216 model, the restricted modeling capabilities of the simpler methods do not allow these codes to adequately predict surface pressures for proper surface contour design.

Sparse Paneling Study

The paneling used in modeling the Recce-Strike configuration was rather dense. In order to assess the requirements for the number of panels to adequately represent a configuration, another model of the Recce-Strike configuration was defined by essentially combining adjacent panels from the dense model. This new sparse model, shown in Fig. 13, consisted of 380 panels, resulting in 791 singularity unknowns. The CPU time for a solution was 254 s on a CDC 7600, resulting in a 72% savings over the dense model. The resulting sparse solution was found to be virtually in-

	PANELS	C_L	C_D	C_M	CPU
DENSE	810	0.08953	0.00619	0.00078	914 sec
SPARSE	380	0.08964	0.00625	0.00053	254 sec

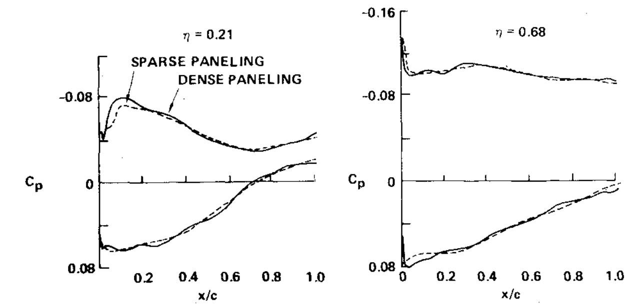

Fig. 14 Effect of paneling density.

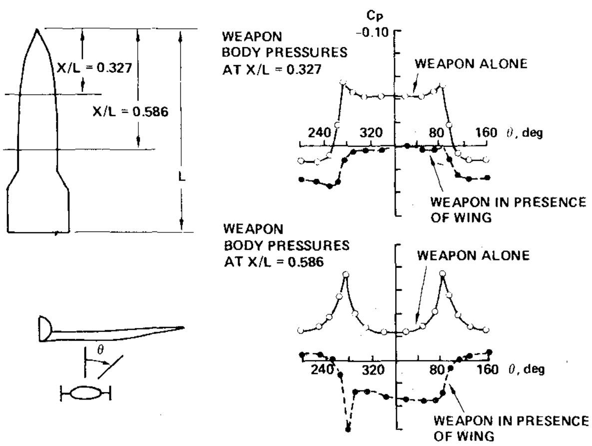

Fig. 15 Weapon-body pressures.

distinguishable from the dense solution. The resulting force and moment data and surface pressure distributions at two wing stations are shown in Fig. 14. Not unexpectedly, the greatest differences (though small) were confined to the wing leading edge, where the surface curvature was changing the most. This comparison illustrates the ability of the code to adequately represent a complex configuration without having to use an excessive number of panels.

V. Weapons Carriage Analysis

Another example in which the advanced panel pilot code has demonstrated unique capabilities is in the field of weapons carriage and separation. In the study of weapons carriage and separation, the capability of simulating detailed and complex geometric shapes is an absolute requirement. The individual aircraft and weapon configurations are generally complex and nonslender in shape. Also, the relative sizes and locations of weapon and aircraft are such that extreme detail is required in describing the geometry in the vicinity of the weapon. This is so that adequate resolution can be achieved for the small-scale (relative to the aircraft) aerodynamic phenomena, forces, and moments associated with the weapon store.

The aerodynamic prediction method must be able to represent the geometries and flow phenomena associated with such configuration details as inlets, canards, conformally carried weapons, recesses, weapon fairings, and gun fairings. At the same time, in the interest of cost, timeliness, and usability, it must have the flexibility to allow the user to represent remote configuration and flow features in a less detailed manner. With the advanced panel method, exact boundary conditions at the surface in critical areas of the configuration or linearized boundary conditions at some idealized representation of the surface for remote areas can be satisfied.

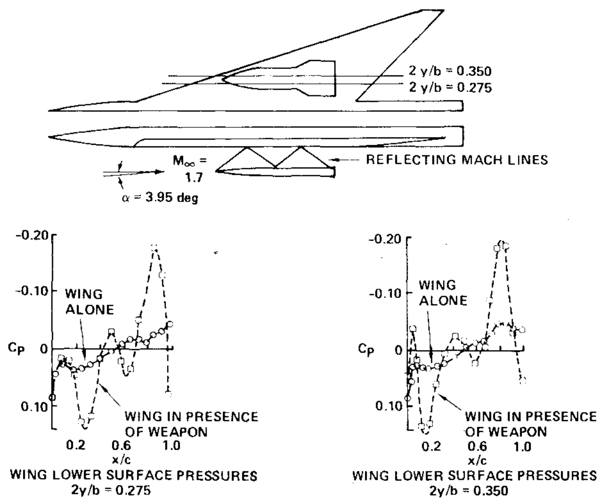

Fig. 16 Lower wing surface pressures.

Configuration Modeling

For the purpose of demonstrating the capability of the advanced panel method in this interest area, a solution was obtained on an arrow wing-body configuration delivering a lenticular shaped store. (At the time these calculations were performed, the arrow wing-body configuration was the only computational wing-body model which had been assembled for the pilot code.) On-the-surface paneling with potential boundary conditions was used for the arrow-wing-body and the body of the lenticular shaped store. Linearized "thin wing" paneling and linearized boundary conditions were used to represent the wings and tip fins of the store.

Computational Results

The results of the analysis of this combination at a Mach number of 1.7 are shown in Figs. 15 and 16. The former shows static pressures at two weapon longitudinal stations: first for the weapon alone, and second in the presence of the wing-body. It can be seen that the presence of the wing-body induces appreciable changes in the weapon surface pressure distributions. The effects of the weapon on the wing-body lower surface pressures at two spanwise locations are shown in Fig. 16. Here the pressure distributions for the wing-body alone are seen to be quite smooth, whereas those in the presence of the weapon are oscillatory in nature. These oscillations are caused by the multiple reflections of the compression wave emanating from the nose of the weapon and are fully consistent with observations of such phenomena in wind tunnel tests of closely located interfering bodies in supersonic flows. More recent weapon-carriage results can be found in Ref. 21.

VI. Conclusions

The application of the advanced panel pilot code to the analysis of realistic supersonic configurations has demonstrated some of the unique capabilities of this method. These studies have shown the code to be a practical analysis method for complex configurations in supersonic flow. Although not shown, this method can also be used for analysis of these configurations in subsonic flow without having to make any paneling changes.

In the detail wind tunnel model design stage, accurate surface pressure data is necessary to use with the appropriate empirical pressure level and gradient constraints in deriving the configuration surface design. With the proper application of these design procedures and use of an accurate potential flow analysis method, such as the advanced panel pilot code, the probability of designing configurations which avoid unnecessary shocks and premature flow separation can be greatly enhanced.

The advanced panel pilot code is currently available through COSMIC as program ARC-11278. As the name "pilot code" implies, this is a no frills software package. The advanced panel method is also being incorporated into the PAN AIR software system, which will be released to NASA, the Air Force, and the Navy during 1979.

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