

APPLICATION OF THE TRANAIR FULL-POTENTIAL CODE TO COMPLETE CONFIGURATIONS

L. L. Erickson*, M. D. Madson† and A. C. Woo ‡

NASA Ames Research Center, Moffett Field, California, USA §

Abstract

The TranAir computer code solves the full-potential equation for transonic flow by combining a rectangular box of flow-field grid points with networks of surface panels. Complex geometry is easily represented since surface conforming flow-field grids are not used. Application of TranAir to the F-16A is presented for free-stream Mach numbers of 0.6 and 0.90, at an angle of attack of four degrees.

Introduction

The major technical obstacle to routinely computing inviscid transonic flow about realistic aircraft "is the difficulty in generating 'suitable grids,'" [1]. Whereas it is difficult to produce flow-field grids that conform to the surface of complicated configurations, it is relatively easy to produce grids that are only on the aircraft surface. Such surface grids are used routinely by linear panel (boundary element) methods.

Recently [2,3,4] there has been progress in solving nonlinear fluid-flow problems by combining flow-field grid methods with surface-panel methods. Such an approach is used in the TranAir computer code. TranAir combines portions of the surface-paneling technology from PanAir [5,6,7,8], with finite difference and optimization techniques to solve the conservative form of the full-potential equation. The basic approach is to embed the surface panels in a rectangular box of grid points. The surface definition generality associated with the panels then enables transonic flow about very complex configurations to be computed without using a surface conforming flow-field grid.

*Aerospace Engineer. Member AIAA.

†Aerospace Engineer. Member AIAA.

‡Research Scientist. Member AIAA.

§This paper is declared a work of the U.S. Government and therefore is in the public domain.

The present paper demonstrates this capability by comparing TranAir solutions with wind-tunnel data for the F-16A at free-stream Mach numbers of $M_\infty = 0.6$ and 0.9, and $\alpha = 4^\circ$. The TranAir model includes fuselage and canopy, engine nacelle (modeled with specified inlet flow velocity), the diverter area between the nacelle inlet and the fuselage, wing and strake, ventral fins, vertical and horizontal stabilizers and the cutouts between these stabilizers, and the fuselage afterbody. These initial results are for the aircraft without wing-tip missiles or under-wing fuel tanks.

TranAir/PanAir Comparison

The input quantities to TranAir are very similar to those of the PanAir linear-potential flow code. The aircraft surface is defined as a collection of abutting networks of surface grid points. Within each network, the surface grid points are used to define surface panels at which boundary conditions are specified. The most apparent input difference between the TranAir and PanAir codes is that a rectangular box of flow-field grid points must be defined for TranAir. If the user does not explicitly define the box, a default box is generated by the program. The paneled model is embedded in this rectangular grid, as illustrated in figure 1 for a two-dimensional airfoil, [9] wherein a portion of the rectangular grid is interior to the paneled airfoil surface.

As in PanAir, each panel is subdivided into eight triangular subpanels so that geometric continuity between all panels within each network is maintained. The program checks all network edges for abutments with adjacent network edges. If unintended gaps are detected by the program they must be closed by the user, or the program can be instructed to change the coordinates of the intended abutting edges so that the gaps are eliminated. After the abutments have been properly established, the code then determines which of the rectangular-grid field

points are interior or exterior to the panels representing the configuration surface. Zero thickness surfaces are also allowed; these are useful for simple models of lifting surfaces, e.g., struts and fins. In addition to defining the aircraft surface, special networks of wake panels are also used to enforce trailing edge Kutta conditions.

User specified boundary conditions can be imposed on either side of a non-wake panel. The program then uses the panel geometry to form relations between the boundary condition data and the flow field properties at the field grid points. The panels are also used to perform pressure integrations over the vehicle surface, and to output surface flow quantities.

TranAir differs from PanAir at the equation solver stage. This is indicated by the absence of certain PanAir options from the TranAir input. For example, the following three features, used to improve upon PanAir's semi-linearized solution (i.e., solving the linear Prandtl-Glauert equation with nonlinear boundary conditions), are not present in TranAir:

1. Various pressure-velocity equations, for example, isentropic, 2nd-order, slender body, linearized.
2. Boundary conditions expressed in terms of either the velocity vector or the linearized mass-flux vector (which is not parallel to the velocity vector).
3. Velocity correction options in regions where the perturbation velocity is not small compared to the free-stream velocity (e.g., near wing leading edge stagnation points).

In TranAir, only the isentropic pressure rule is used, the nonlinear mass-flux vector (which is parallel to the velocity vector) is used, and perturbation velocities do not have to be small compared to the free-stream velocity.

Since TranAir iteratively solves a nonlinear problem, each change in angle of attack or sideslip represents a new computer run through the equation solver. This contrasts to PanAir's ability to obtain approximate solutions for multiple angles of attack and sideslip for nearly the same cost as a single angle. For sideslip cases involving symmetric configurations, TranAir uses both halves of the configuration, whereas PanAir sideslip solutions for symmetric configurations can be obtained using only half of the geometry.

Models of F-16A

A TranAir input model of the F-16A is shown in figure 2. The wing wake has been aligned with the angle of attack (4°). The one-panel wide vertical wake connecting the wing and horizontal stabilizer wakes is a constant-strength doublet network which prevents the doublet strength on the inboard edge of the wing wake from being forced to zero, (a condition which would cause an incorrect loss in wing circulation). For zero sideslip, the

vertical stabilizer wake is omitted. TranAir wakes differ slightly from PanAir wakes. When a TranAir wake emerges from the downstream face of the rectangular box of field-grid points, the program "chops off" the rest of the wake beyond the grid box. It then redefines the section of wake beyond the grid box to be parallel to the body axis. Thus, the wing wake on the TranAir model is parallel to the free-stream until it reaches the aft end of the grid box, where it is modified by the program to run parallel to the body axis. Views of the paneled aircraft (without wakes) embedded in the box of rectangular grid points are shown in figure 3. The orthographic views show the actual grid points used, the isometric view is a schematic.

The right half of the configuration is defined with 3185 surface panels, plus 252 wake panels for the half geometry (Only the right half portion of symmetric configurations are input). The right half of the rectangular grid box contains $129 \times 33 \times 33$ points in the x, y, z directions, respectively. The grid spacing along each coordinate direction is uniform, but $\Delta x, \Delta y,$ and $\Delta z,$ can each be different. Views of the paneled geometry of the F-16A are shown in figure 4. These views show several regions where the panel density is discontinuous, e.g., in the wing-strake/fuselage region where fewer chordwise panels are used on the fuselage than on the wing-strake. The regions of abrupt change in panel density occur at network abutments; within a network the panel spacing is usually non-uniform but continuous. Figure 5 repeats two of the figure 4 views, but with panel edges drawn as dashed lines, and abutments (network edges) drawn as solid lines. The model contains 71 networks, 14 of which are wake networks.

In this initial model, the diverter geometry above the nacelle is included, but the nacelle inlet lip has been omitted. This omission was not due to any limitations within the program. The wind-tunnel model has a flow-through duct connecting the inlet and afterbody regions. The TranAir model of the inlet face is a network of porous panels, on which the boundary conditions are, (1) the velocity on the upstream side of the panels is the normal component of the free-stream velocity, and (2) the perturbation potential on the downstream side of the panels is zero. The wing, strake, vertical and horizontal stabilizers are modeled as thick surfaces (actual-surface geometry), with panels sealing the wing and stabilizer tips. For simplicity, and to show some of TranAir's modeling versatility, the ventral fin is approximated by a zero-thickness flat plate (mean-surface geometry).

Although not shown in figure 4, the afterbody base is closed off with panels in the same manner as the inlet. The boundary conditions used on these base panels are, (1) the total potential is constant on the downstream side and, (2) the perturbation potential is zero on the upstream side. The constant total potential on the downstream side, combined with the wake networks emanating from the afterbody perimeter (see figure 2), cause the flow to separate smoothly from the afterbody

trailing edge, i.e., the flow doesn't turn through a right angle as would occur if the base panels were modeled with solid-surface boundary conditions [5, figure 18].

Results

The wind-tunnel model has pressure taps at six wing stations. These stations are indicated in figure 6. The PanAir/TranAir model's spanwise paneling was arranged so that panel center points coincided with the pressure tap stations on the wind-tunnel model.

Initial results were obtained for $M_\infty = 0.6$ in order to validate TranAir predictions by comparison with PanAir results. The flow at this Mach number is subcritical, so TranAir and PanAir results should be similar except near leading-edge stagnation points, where the Prandtl-Glauert small perturbation assumption employed by PanAir is violated.

TranAir and PanAir results are in very good agreement except near the leading edges and at the most outboard tap station. These initial TranAir results fail to capture the leading edge pressure peaks. Figure 7 shows the TranAir leading edge definition of the F-16 wing, near the outboard tap station, embedded in the rectangular grid. Flow field properties are calculated at the centroid of a flux box. As seen in figure 7, several leading edge panels are contained in one flux box. Presently, surface flow quantities are obtained by a linear interpolation of the flow field quantities, therefore the degree of change that can be predicted for surface quantities is limited by the local density of the flow field grid. This suggests the need for a second-order basis function for potential, and/or local grid refinement. Figure 8 shows Tranair and PanAir results, along with wind tunnel data, for $\alpha = 4^\circ$ and $M_\infty = 0.6$ at the six wing stations. Solid lines represent upper surface pressures, while lower surface pressures are represented by dashed lines.

TranAir results are generally in good agreement with experimental results, except for the following discrepancies:

1. As previously mentioned, TranAir misses the leading edge pressure peaks.
2. At the inboard wing station, wind-tunnel results show a fairly flat pressure distribution over the upper surface, with no leading edge pressure peak, and a slight rise in pressure between about 30% and 60% chord.
3. At the outboard station, the pressure levels between the wind-tunnel and computational models are considerably different. This is probably due to the presence of the tip missile and launcher on the wind-tunnel model, which were not included in the computational model. The wind tunnel data indicates the flow is separated near the trailing edge.

Results for $M_\infty = 0.90$ are shown in figure 9. The wind-tunnel data indicates a shock at approximately 75% chord for the four inboard stations. TranAir also indicates a shock, but it is slightly downstream of the shock predicted by the wind tunnel. This result is generally expected from a conservative full potential solution, due to the absence of a boundary layer correction [10, figure 40] or a correction for entropy conservation at the shock jump. PanAir results, generated to clearly show the difference between a linear and full potential solution for a transonic Mach number, do not predict the shock on the aft portion of the wing. The shock predicted by TranAir is smeared over a range of 3-5 panels. This result suggests that the rectangular grid was too coarse for this solution. As in the $M_\infty = .6$ case, TranAir fails to capture the leading edge pressure peaks. Wind tunnel data indicates that a shock may exist near the leading edge of stations 2-5. At the outboard tap station, TranAir and wind tunnel results differ greatly. Again, this can probably be attributed to the presence of the tip missile and launcher on the wind tunnel model that was absent from the TranAir model.

TranAir and PanAir results were obtained using a CRAY X-MP/48 with 2 million words of available central memory and a solid state disk of 128 million. The PanAir model was run using Boeing's pilot code version of PanAir, since the model contained too many panels to be run with the production code version. For $M_\infty = .6$ and $.9$, PanAir used 3485 CPU seconds for each run. The TranAir solution for $M_\infty = .6$ took 100 iteration cycles and used only 1700 CPU seconds. For $M_\infty = .9$, TranAir took 400 iterations and used 6000 CPU seconds. It should be pointed out, however, that engineering accuracy was obtained after 210 iteration cycles and only 3000 CPU seconds. TranAir run times are expected to be reduced by as much as a factor of two in the near future with the addition of more optimization techniques.

Future Plans

The results presented herein are for the first TranAir application to a relatively complete configuration. Studies are needed to determine the solution sensitivity to the number of panels and field grid points. This will be done on the NAS (Numerical Aerodynamic Simulator) CRAY-2 located at NASA Ames Research Center. With an initial memory size of 64 million words of central memory, TranAir models containing up to 25,000 panels and a box of $516 \times 132 \times 68$ grid points should be possible.

Additional F-16 geometry features will be added to the current TranAir model. These include the wingtip missiles and launcher, the underwing fuel tank and pylon, and the flow-through nacelle of the wind-tunnel model. Sideslip, and deflected leading-edge flap cases are also anticipated.

Refinements currently being made to the TranAir

code include the following:

1. Implementation of a second-order basis function for the potential, which would, in areas of high gradient, increase the range of changes that could be captured when many panels exist within a flux box.
2. Local grid refinement, wherein individual cells of the rectangular grid are subdivided into smaller cells in regions of high panel density. This should allow greater accuracy in regions of high gradients, such as the leading edge of a wing, without having to increase the density of the entire field grid.
3. The Kutta condition model of [11] is being added. In this model, the trailing vorticity vectors in the wake are free to seek their lateral orientation within the user-specified positions of the wake networks. The model used for the results presented herein forces the trailing vorticity to be aligned with the downstream panel direction and ignores the effect of the spanwise velocities.
4. Multigrid is being tested to determine whether it is effective in accelerating the convergence rate.
5. The current version of TranAir requires a subsonic free-stream Mach number. The code will soon accept supersonic free-stream Mach numbers.

Plans for developing TranAir include the addition of wake capturing for wakes separating from sharp edges, and incorporation of routines to solve the Euler equations and the Navier-Stokes equations.

Conclusions

For inviscid flow, the TranAir approach eliminates the need for surface-fitted grids and enables the full geometric generality associated with linear-flow panel methods to be extended to the transonic regime. This has been demonstrated by applying TranAir to the F-16A. The nonlinear full potential equation is solved for both subcritical and supercritical flow cases and compared with wind-tunnel measurements of wing pressures at $\alpha = 4^\circ$.

Acknowledgements

The TranAir code is being developed under NASA contract NAS2-11851. The authors would like to thank F. T. Johnson, J. E. Bussoletti, S. S. Samant, D. P. Young, B. Everson, and R. Burkhart for their work under this contract, and for their invaluable assistance in obtaining the results presented in this paper. The authors also wish to thank K. Li of the University of Illinois for his contributions to the project.

References

- [1] Holst, T. L., Slooff, J. W., Yoshihara, H., and Ballhaus, W. F., Jr., "Applied Computational Transonic Aerodynamics", AGARD AG-266, 1982.
- [2] Piers, W. J., and Slooff, J. W., "Calculation of Transonic Flow by Means of a Shock-Capturing Field Panel Method", AIAA Paper 79-1459, 1979.
- [3] Tseng, K. and Morino, L., "Nonlinear green's function method for unsteady transonic flows", 1982.
- [4] Johnson, F. T., James, R. M., Bussoletti, J. E., Woo, A. C., and Young, D. P., "A Transonic Rectangular Grid Embedded Panel Method", AIAA Paper 82-0953, 1982.
- [5] Carmichael, R. L., and Erickson, L. L., "PAN AIR - A Higher Order Panel Method for Predicting Subsonic or Supersonic Linear Potential Flows about Arbitrary Configurations", AIAA Paper 81-1255, 1981.
- [6] Derbyshire, T., and Sidwell, K. W., "PAN AIR Summary Document (Version 1.0)", NASA CR-3250, 1982.
- [7] Magnus, A. E., and Epton, M. A., "PAN AIR - A Computer Program for Predicting Subsonic or Supersonic Linear Potential Flows about Arbitrary Configurations using a Higher Order Panel Method (Version 1.0), Vol. I, Theory Document", NASA CR-3251, 1981.
- [8] Sidwell, K. W., Baruah, P. K., and Bussoletti, J. E., "PAN AIR - A Computer Program for Predicting Subsonic or Supersonic Linear Potential Flows about Arbitrary Configurations using a Higher Order Panel Method (Version 1.0), Vol. II, User's Manual", NASA CR-3252, 1981.
- [9] Erickson, L. L., and Strande, S. M., "A theoretical basis for extending surface-paneling methods to transonic flow", *AIAA Journal*, 23(12), Dec. 1985.
- [10] Holst, T. L., "Numerical computation of Transonic Flow Governed by the Full-Potential Equation", NASA TM 84310, 1983.

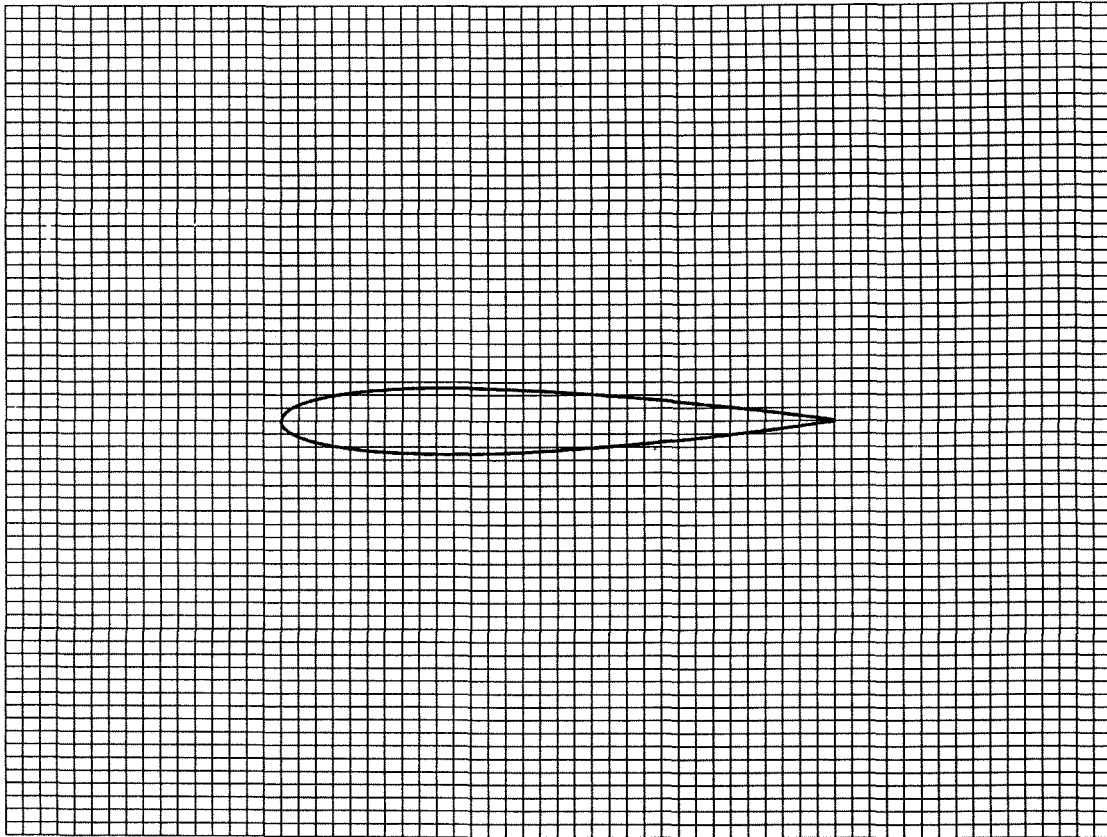


Figure 1. NACA 0012 Airfoil in a 65x65 grid.

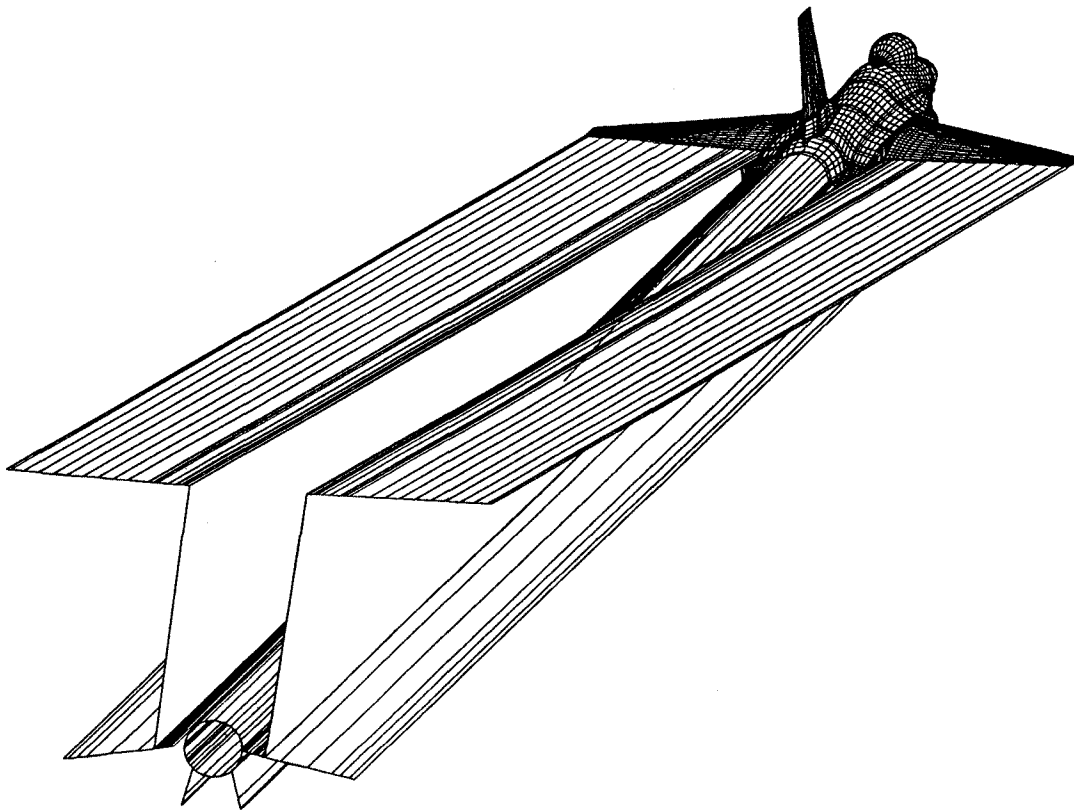
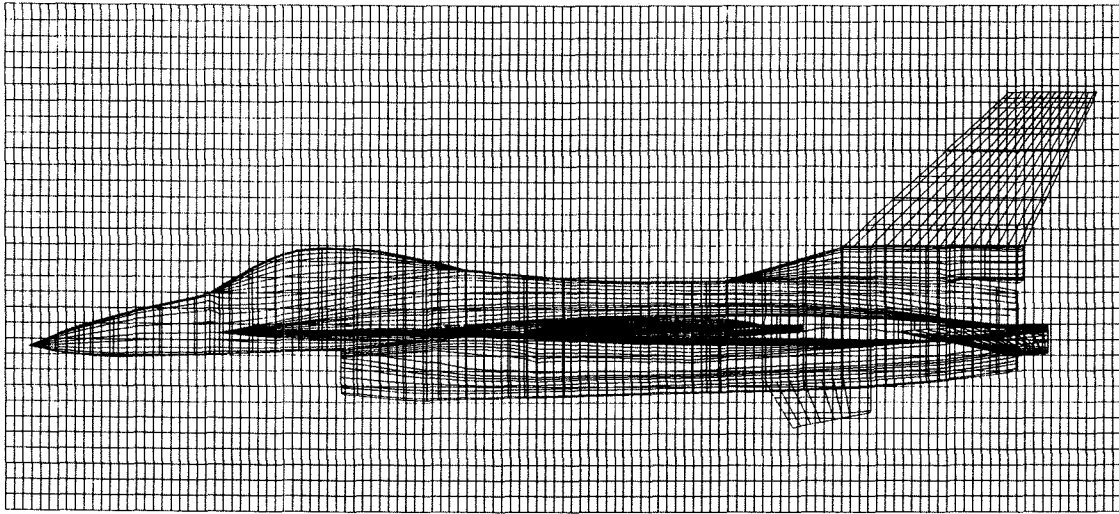
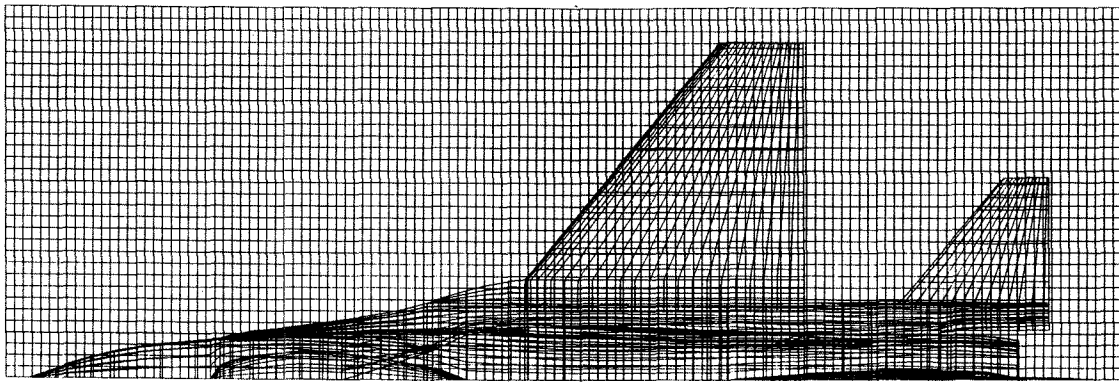


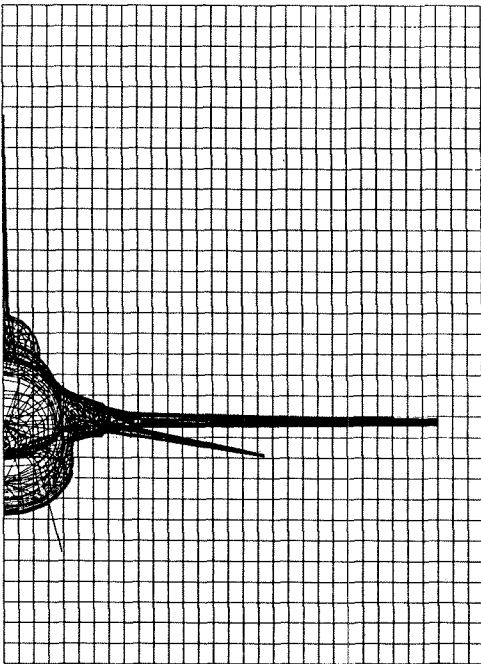
Figure 2. Trailing wake networks.



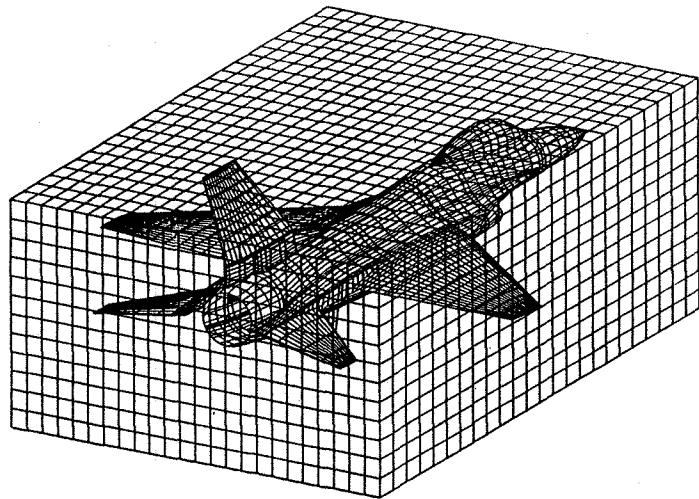
(a) side view.



(b) top view.

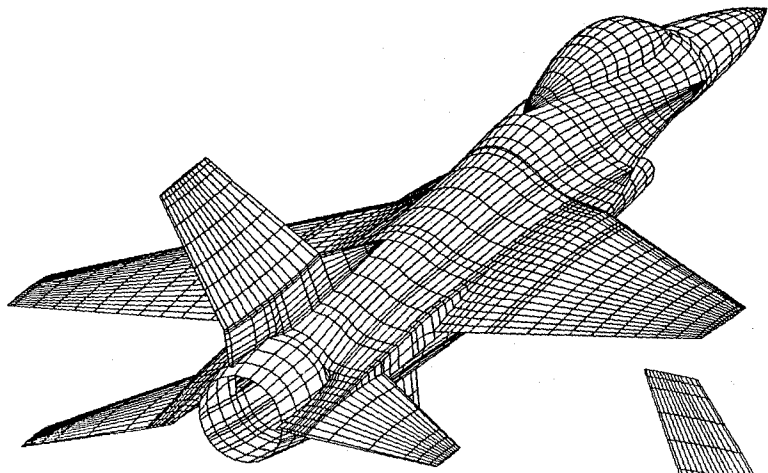


(c) rear view.

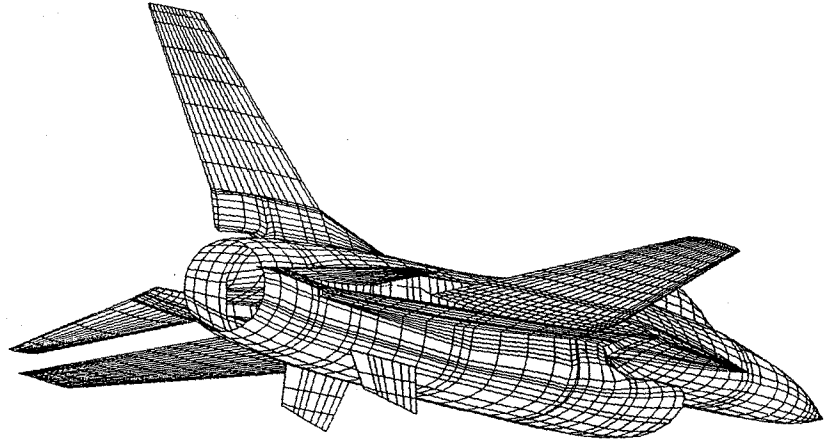


(d) isometric view.

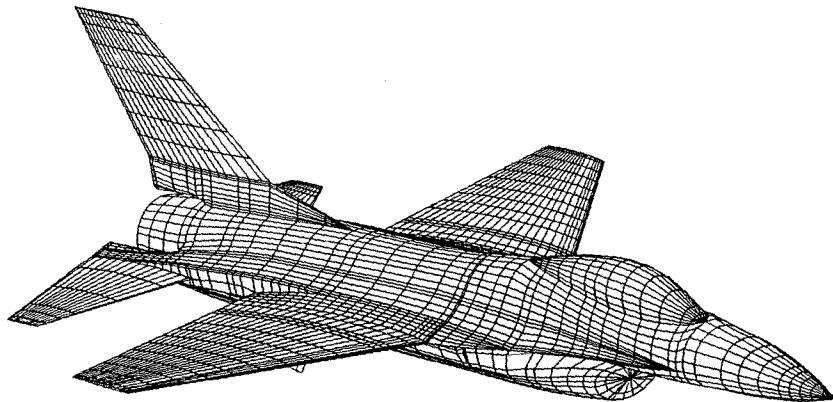
Figure 3. Surface panels embedded in a box of rectangular grid points:



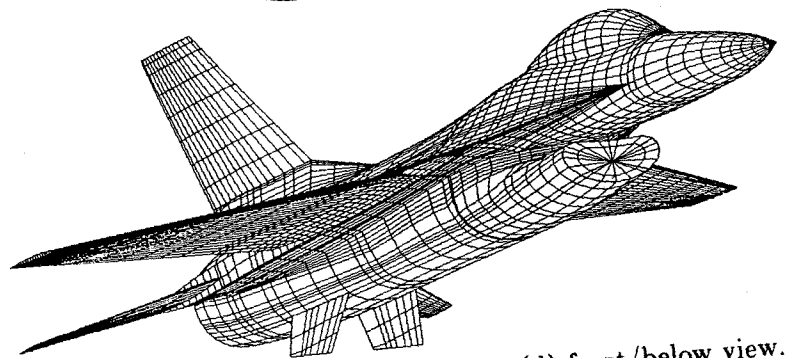
(a) rear/above view,



(b) rear/below view.

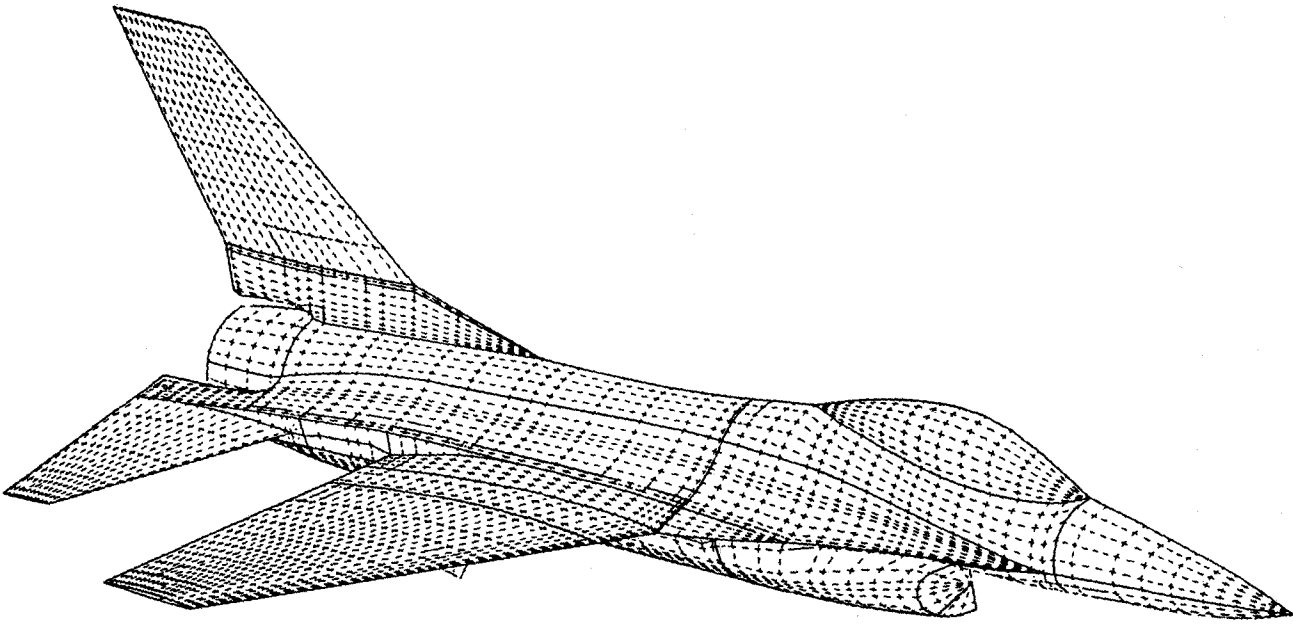


(c) front/above view.

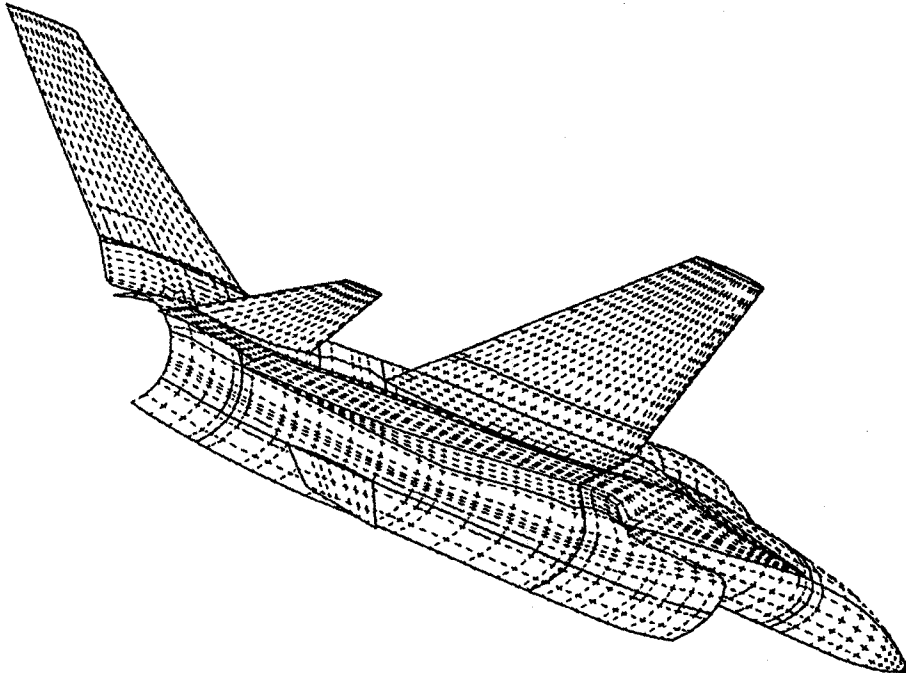


(d) front/below view.

Figure 4. Isometric view of surface paneling :



(a) front/above view.



(b) rear/below view.

Figure 5. Network edges = solid lines, panel edges = dashed lines.

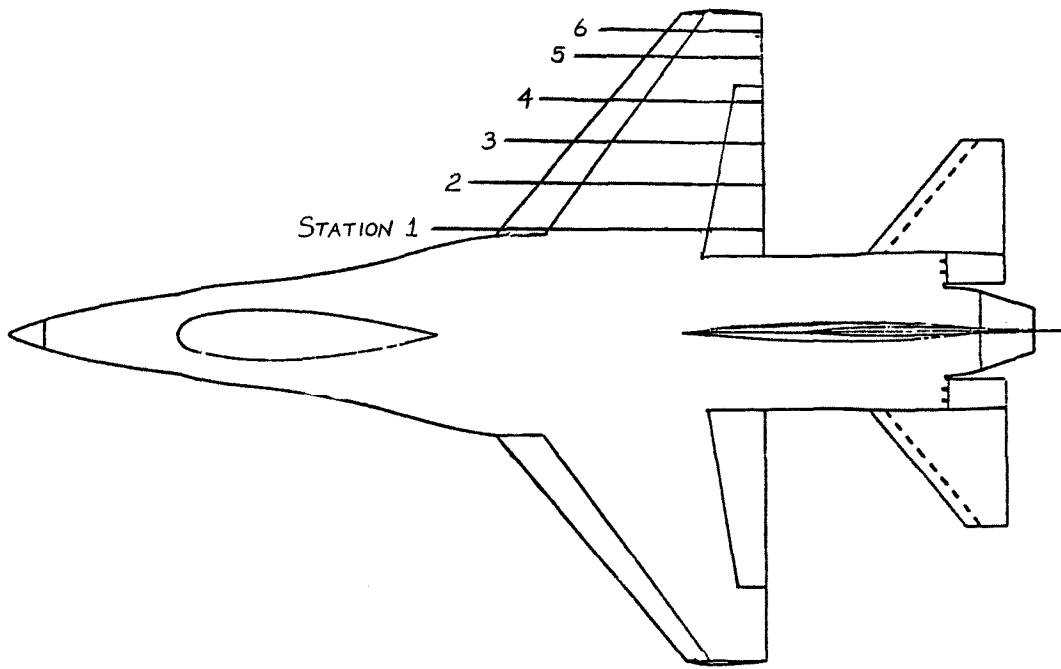


Figure 6. Wing pressure stations.

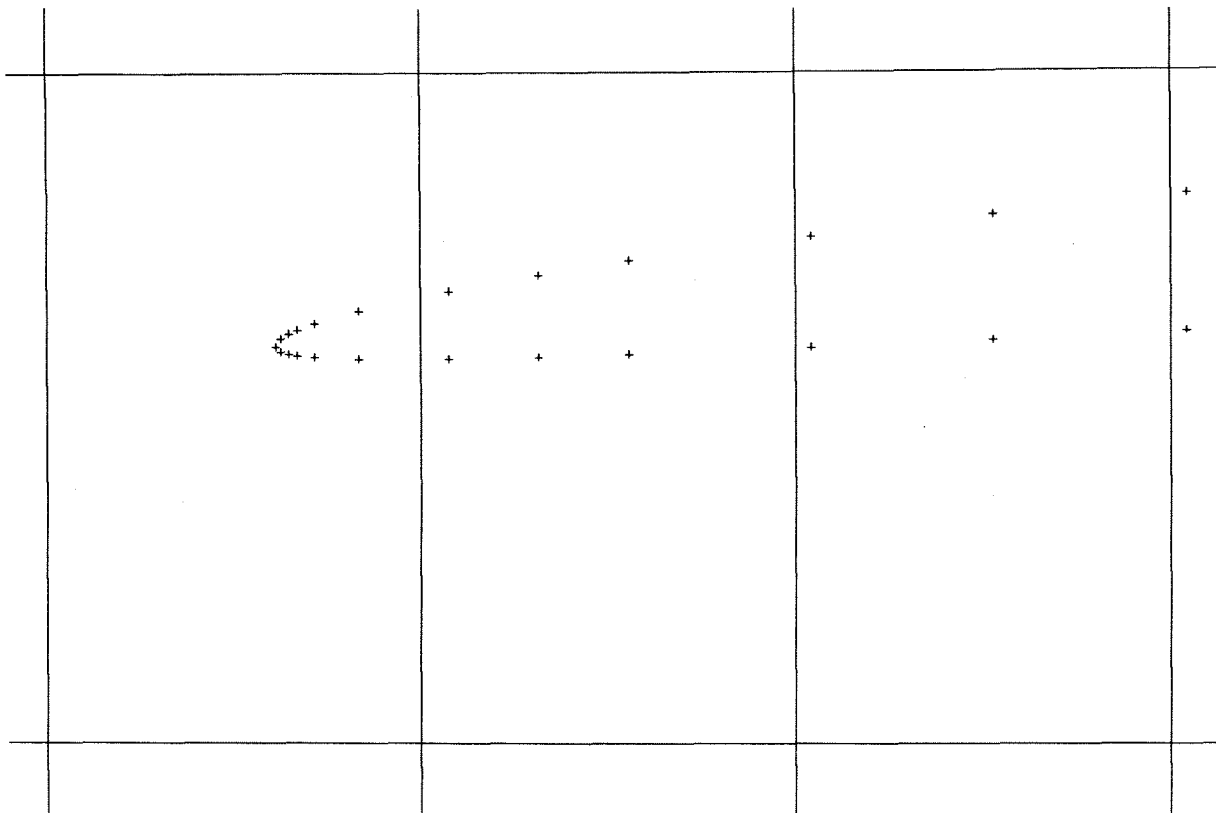
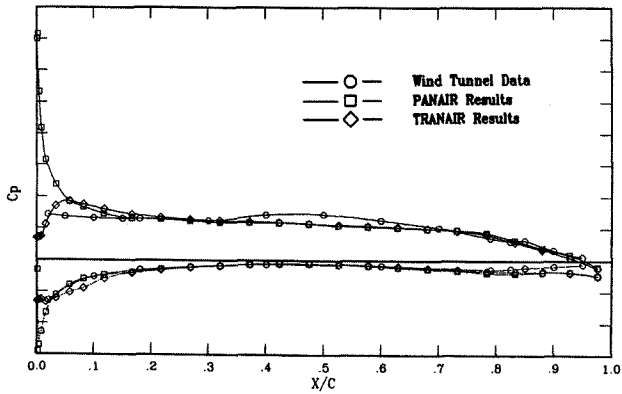
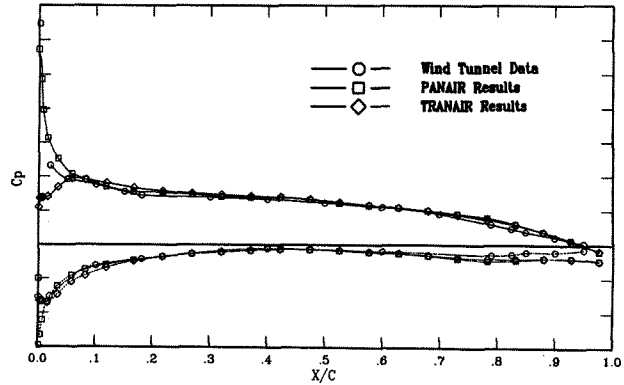


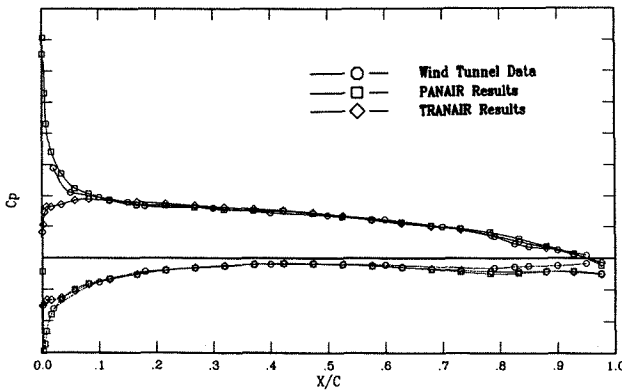
Figure 7. Detail of leading edge paneling within the computational grid which explains the inability to resolve leading edge pressures.



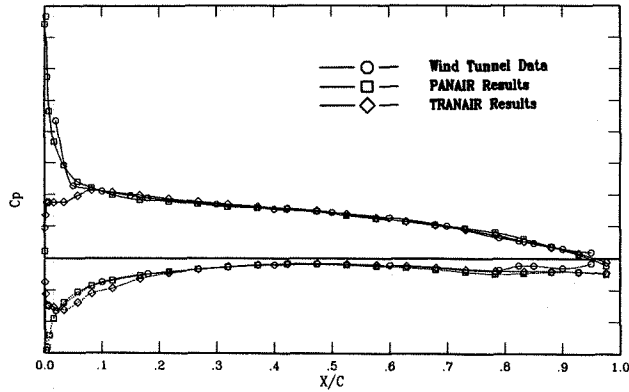
Tap Station 1 (32% semispan).



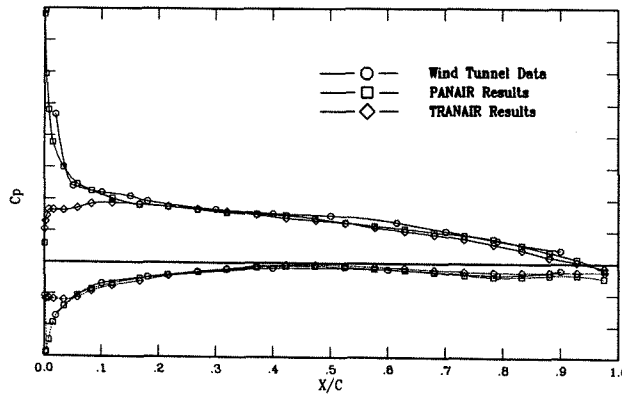
Tap Station 2 (45% semispan).



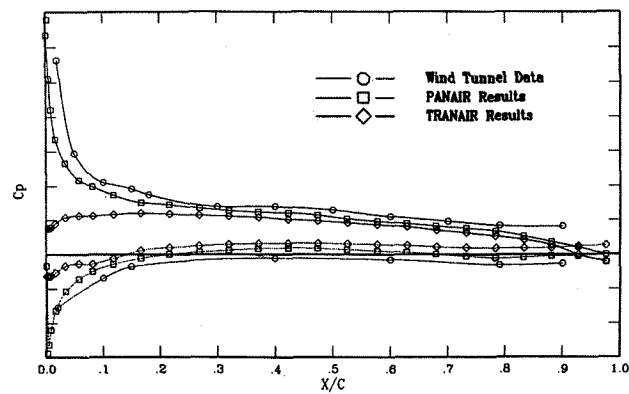
Tap Station 3 (59% semispan).



Tap Station 4 (71% semispan).

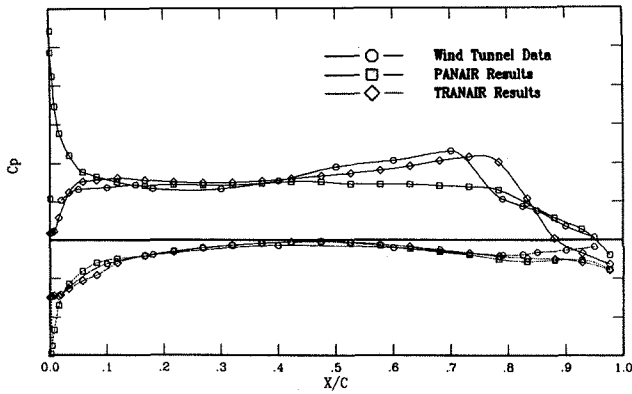


Tap Station 5 (84% semispan).

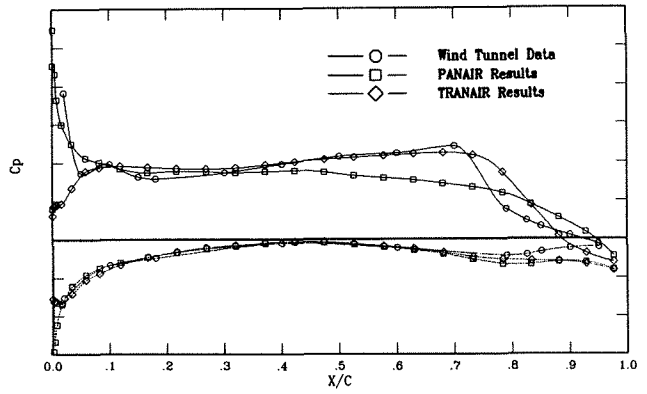


Tap Station 6 (95% semispan).

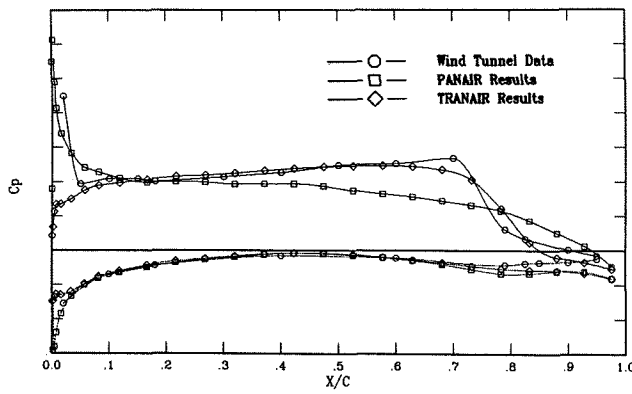
Figure 8. TranAir Results of F-16A Full Configuration: $M = 0.6$ and $\alpha = 4.0^\circ$.



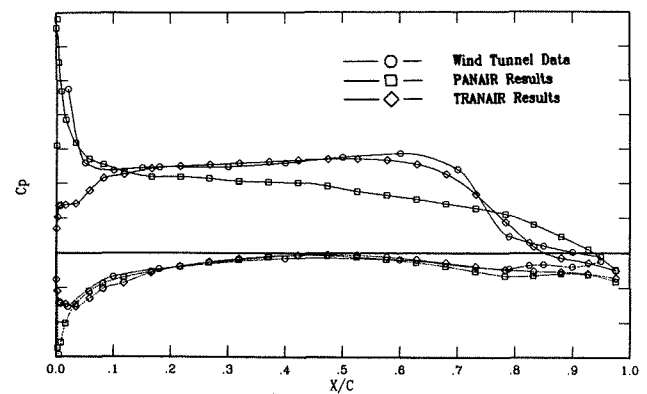
Tap Station 1 (32% semispan).



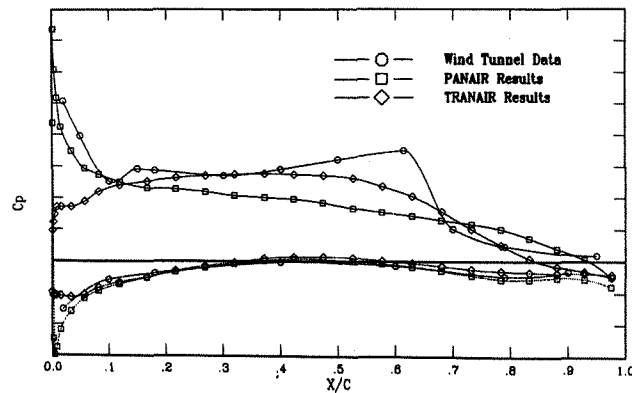
Tap Station 2 (45% semispan).



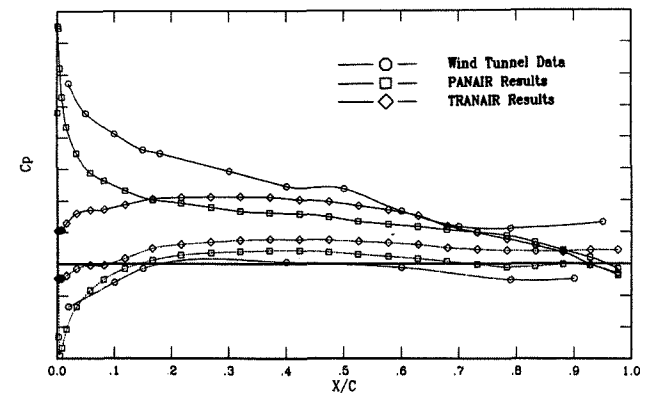
Tap Station 3 (59% semispan).



Tap Station 4 (71% semispan).



Tap Station 5 (84% semispan).



Tap Station 6 (95% semispan).

Figure 9. TranAir Results of F-16A Full Configuration: $M = 0.9$ and $\alpha = 4.0^\circ$.