Transonic Computational Analysis of F-111 TACT

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F-111 TACT tests at $M=0.9$ and $\alpha=6$ deg showed severe wing unloading in the inboard region over the glove and fuselage, and a shock wave unsweeping in the tip region deteriorating the cruise performance. A computational model is developed for use on the Bailey-Ballhaus transonic small-disturbance code which allows a good simulation of the complex fuselage geometry, including inlet effects, of the F-111 TACT. Comparisons with experiments show good agreement. "Fixes" are demonstrated which alleviate some of the aerodynamic deficiencies of the original aircraft.

Nomenclature

c = local wing chord
C_p = pressure coefficient
M = freestream Mach number
n = surface unit normal vector
t = thickness
V = total velocity
V_I/V_∞ = inlet velocity ratio
WBL = wing butt line
x = chordwise body axis coordinate
y = spanwise body axis coordinate
z = body axis coordinate normal to x-y plane
α = angle of attack
η = span fraction

I. Introduction

THE F-111 TACT aircraft (Fig. 1), designed over a decade ago, was intended to demonstrate the performance of an early version of the NASA supercritical airfoil. Tests[1] at high-subsonic Mach numbers showed that adverse three-dimensional (3 D) effects occurred in the wing-fuselage junction region and in the outborad tip region of the wing, as shown in Fig. 2. The inboard deterioration is due to the relatively high pressures existing over the fuselage-glove upper surface. Such high pressures are not only undesirable in themselves, but they further erode the high suctions on the adjacent wing panels. A similar erosion of the upper-surface suctions occurs in the tip region due to pressure leakages over the wing tip. In both regions, the consequent upstream displacement of the primary shock exposes sufficient stretches of aft high-surface convexity to produce a second supersonic region with a terminating shock. Such additional high suctions add lift, but they act on highly rearward inclined slopes to cause an undesirably disproportionate increase in drag.

The primary purpose of the present paper is to explore methods of making configuration improvements to eliminate these shortcomings, using the 3-D transonic small-disturbance relaxation procedure originally developed by Bailey and Ballhaus.[2] Such a procedure has its well-known limitations, due to the small-disturbance assumption, and of course essential viscous interactions must be additionally incorporated. A preliminary calculation was made by Bateley and Yoshihara[3] at $M=0.9$ and $\alpha=6$ deg (a point in the drag rise) using a simplified box-type fuselage with flat sides. The calculation was performed using nonconservative relaxation (NCR) methods. In the absence of viscous corrections, the NCR solutions generally show better agreement with experimental pressure measurements than the more correct fully conservative relaxation (FCR) methods. The resulting

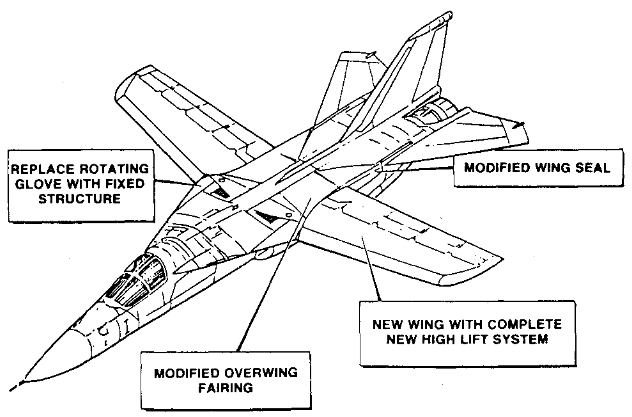

Fig. 1 TACT modifications to F-111.

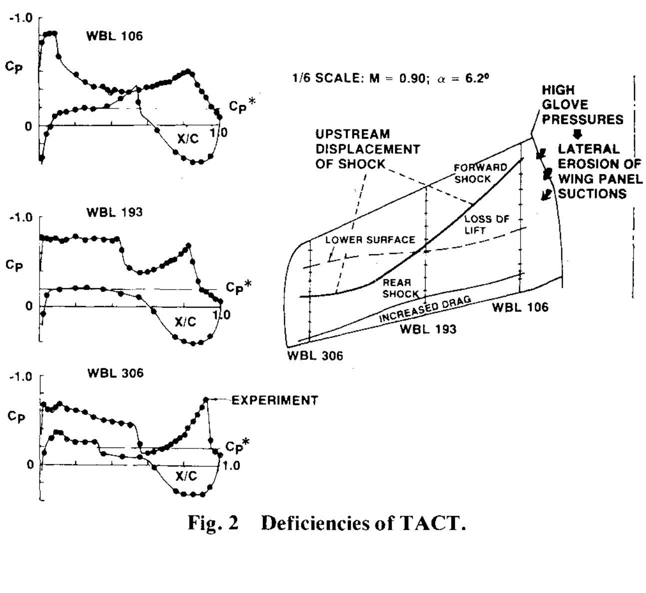

Fig. 2 Deficiencies of TACT.

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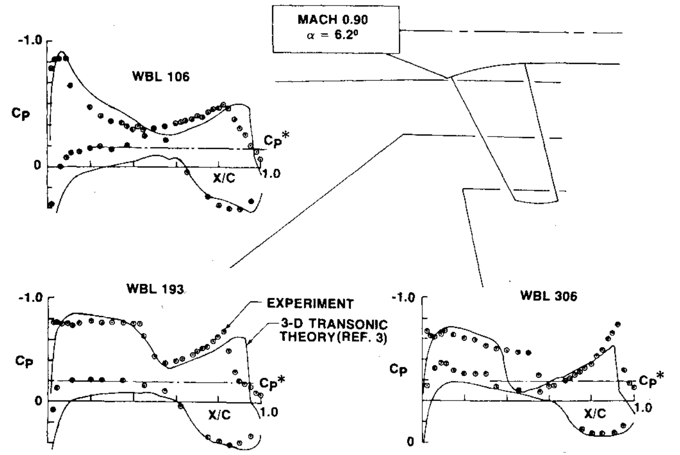

Fig. 3 Early test theory comparison.

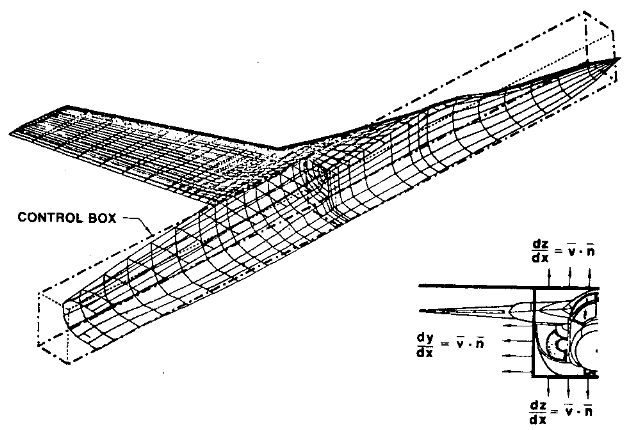

Fig. 4 Complex fuselage modeling scheme.

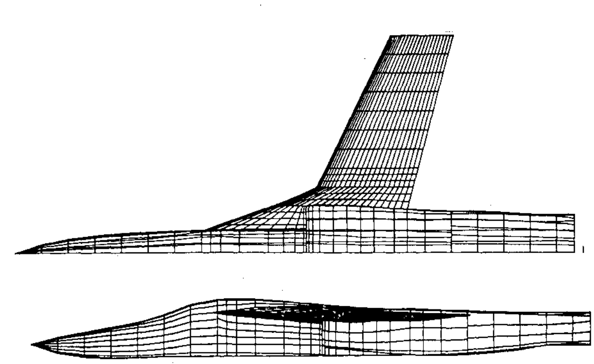

Fig. 5 Subcritical panel modeling.

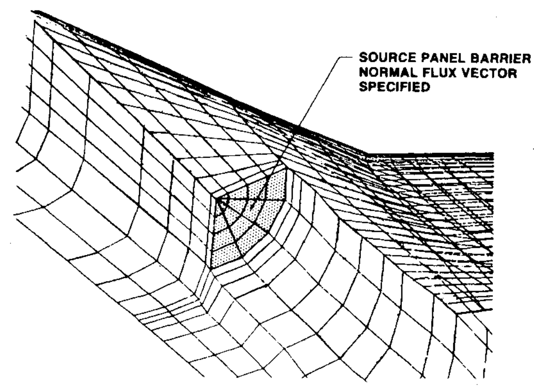

Fig. 6 Inlet modeling.

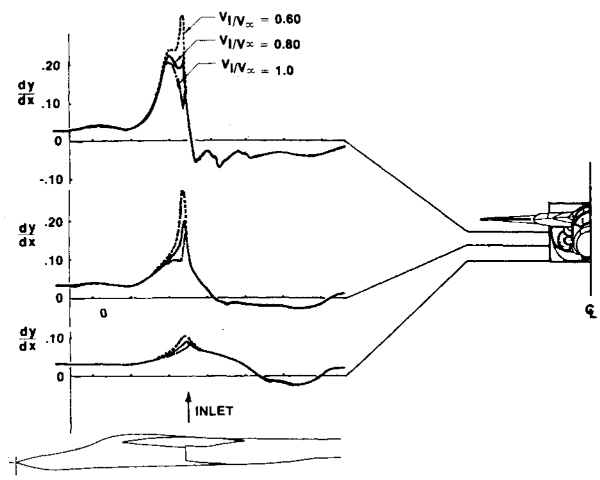

Fig. 7 Body boundary conditions.

pressure distributions are compared in Fig. 3 with the experimental results. The agreement is reasonably good in its qualitative aspects, but significant discrepancies arise, understandably in the shock capture, but also in the lower surface pressures. The latter discrepancy, which most likely can be attributed to the omission of the inlet and the simplification of the side surface of fuselage to one aligned with the freestream, underscores the need for a better method for simulating fuselages.

The paper will first discuss the configuration modeling used in the small-disturbance calculations. A new scheme for modeling complex fuselage representations will be introduced, followed by a brief discussion of the viscous interactions. Once a viable theoretical model has been established, the discussion will shift to design modifications based on theoretical computations on the F-111 TACT configuration. The modifications presented are not necessarily optimal, but serve to demonstrate the use of the fixes as well as the strong spanwise interference effects which arise in transonic flows. A portion of the work presented in this paper was sponsored by the Air Force Flight Dynamics Laboratory. The data presented are based on either the General Dynamics F-111 TACT configuration or a Boeing version of the TACT.

II. Computational Modeling

Current 3-D transonic analysis methods have a very crude body representation capability. Fuselages in these codes have generally been represented by simple bodies of revolution or infinitely long cylindrical bodies with simple box-type cross sections. In transonic flow, body influences can propagate far outboard on the wing to alter significantly the wing surface pressures. It is therefore imperative, if we wish to improve our transonic analysis capabilities, to be able to better simulate complex fuselage configurations.

A scheme has been developed which will allow the simulation of complex geometries within the framework of existing transonic codes. The approach is to specify the boundary conditions on a simple cross section control box enclosing the fuselage. The necessary streamline slope conditions are then calculated by a subsonic panel method for the actual configuration in subcritical flow. Here it is postulated that the flow slopes near the fuselage remain essentially unchanged from subcritical conditions. The streamline slopes, thus determined by the subcritical panel method on a box-like array of off-body points (Fig. 4), would then define a simpler substitute fuselage which would still simulate the influence of the actual fuselage in the transonic calculations. Complexities of the fuselage such as the canopy, the inlets, and the effects of inlet velocity ratio are fully accounted for with the subsonic panel method which does not assume a small-disturbance approximation. As a fallout, the use of the substitute fuselage will bury high-disturbance regions at the inlet lip.

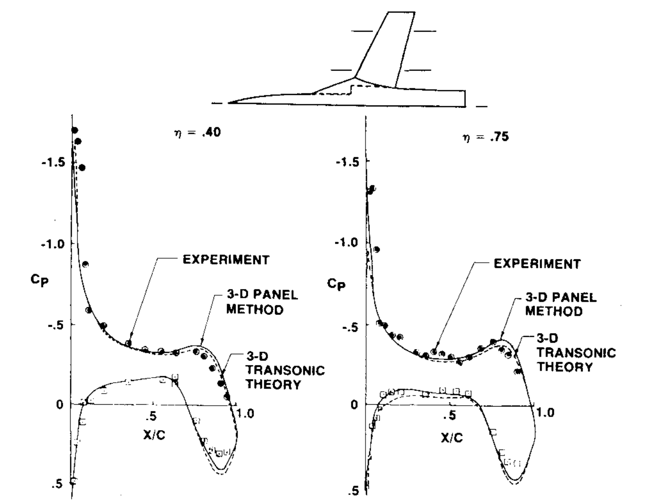

Fig. 8 Wing pressure distribution, $M_\infty = 0.70$, $\alpha = 5.0$ deg.

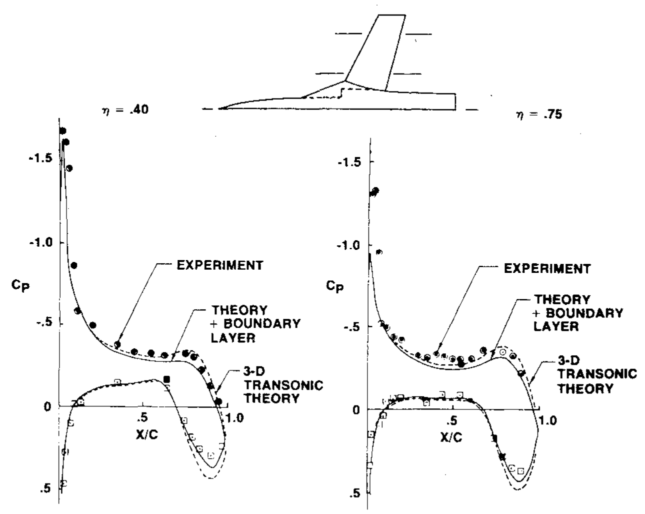

Fig. 9 Effect of adding pseudo-boundary-layer thickness to geometry.

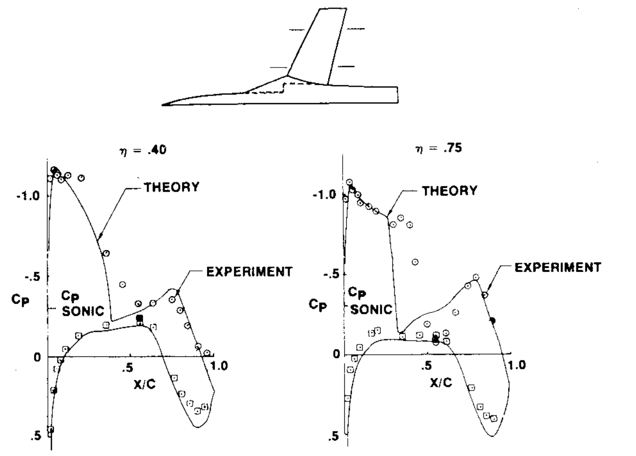

Fig. 10 Wing pressure distribution, $M_\infty = 0.85$, $\alpha = 5.24$ deg.

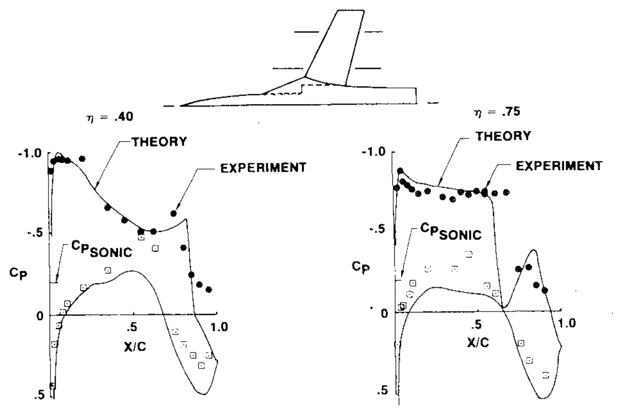

Fig. 11 Wing pressure distribution, $M_\infty = 0.90$, $\alpha = 5.23$ deg.

The subsonic potential flow code used was based on the general method of Rubbert and Saaris[4,5] for the numerical solution of nonplanar three-dimensional boundary-value problems. The method employs a panel solution to the exact incompressible potential flow equation, satisfying boundary conditions on the actual configuration surface. Compressibility effects are approximated by the Goethert rule. The method is incorporated into Boeing computer program A230.

The paneling scheme used to represent the F-111 TACT is shown in Fig. 5. A total of 1684 singularities were used to represent the configuration. The objectives of the analysis required a representation that would yield good wing surface pressures and also provide a good simulation of the inlet effects on the surrounding flowfield. Since the requirement to simulate the inlet did not include the need for detailed surface pressures on the inlet lips, a simple source panel barrier was placed at the inlet face, as shown in Fig. 6. By specifying the mass flux entering the barrier, different inlet velocity ratios could be simulated. A simple jet model of the flow exhausting tangential to a solid surface was used at the aft end of the model. The horizontal and vertical tails were neglected as was the inlet diverter.

The transonic analysis was carried out using essentially the Bailey-Ballhaus 3-D transonic small-disturbance code, featuring the grid embedding scheme of Boppe.[6] A skewed or sheared mesh is used outboard of the glove. Inboard, the mesh is unswept to form a Cartesian grid, nonconformal with the leading edge of the highly swept glove, resulting in a sawtooth-type representation. The inboard Cartesian mesh is well suited for the body representation and the plane of symmetry conditions. The body boundary conditions, derived from the mass flux vectors calculated by the subsonic panel program, were applied on a box-like cross section. Selected distributions are shown in Fig. 7 for the side of the box. Note that the effects of changing inlet velocity ratio can be simulated in this manner.

III. Computed Results

Comparisons of surface pressure distributions at 0.70 Mach number between the A230 subsonic panel and transonic code results and experimental data are shown on Figs. 8 and 9. Considering first the comparison between the two theoretical methods as shown on Fig. 8, note how well the transonic code results agreee with the A230 results. This agreement affirms the consistency between the computational models. Considering now the comparison between the theoretical methods and the experimental data, it is noted that the computed results overpredict the aft loading. This is due to the reduction of the effective aft camber by the viscous displacement effects and the effects of the small-disturbance approximation. In order to approximate the viscous effects, a two-dimensional analysis was carried out using the Bauer-Garabedian transonic viscous analysis program[7] on the supercritical section located at approximately the midspan. The resulting boundary-layer displacement thickness was then applied to the wing, scaled with the local thickness ratio. The resulting improved correlation with experimental data is as shown in Fig. 9. Obviously, such a simple "fix" can in no way account fully for the actual viscous effects, but it is surprising how well it does improve the test theory correlation on the aft portion of the wing. The new method, which accounts for the

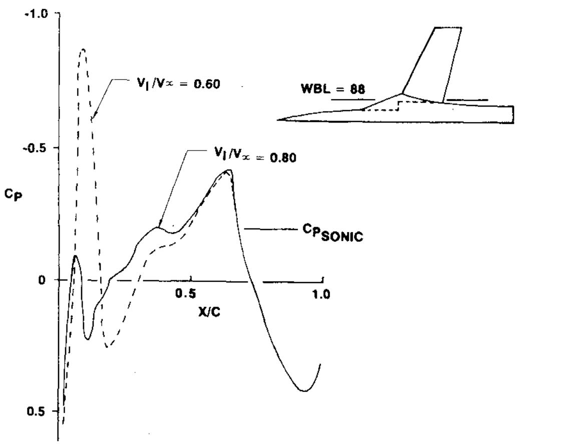

Fig. 12 Effect of inlet velocity ratio, $M_\infty = 0.90$, $\alpha = 6.0$ deg.

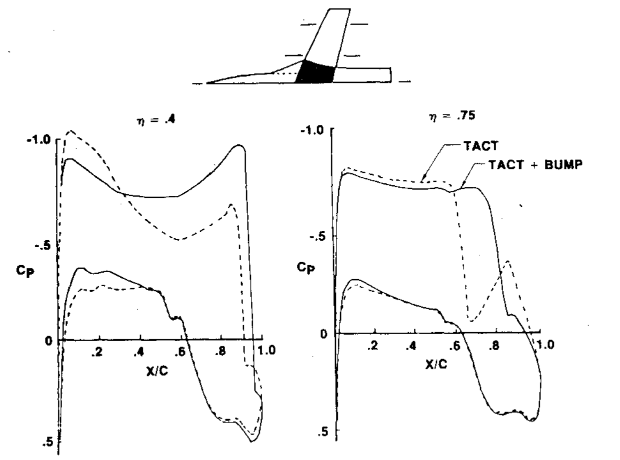

Fig. 14 Effect of bump on wing pressures, $M_\infty = 0.90$, $\alpha = 6.0$ deg.

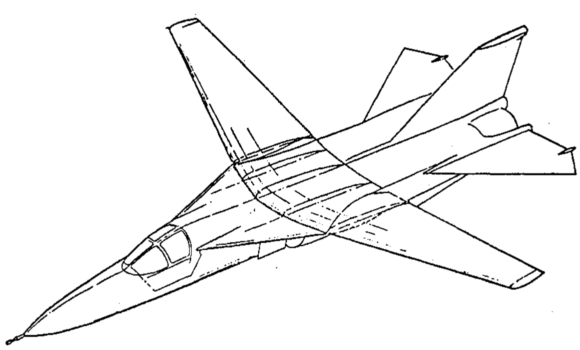

Fig. 13 F-111 TACT bump.

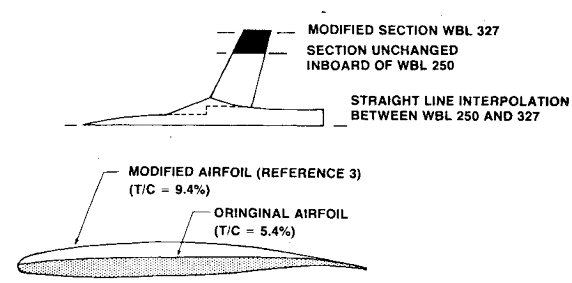

Fig. 15 Wing tip modifications.

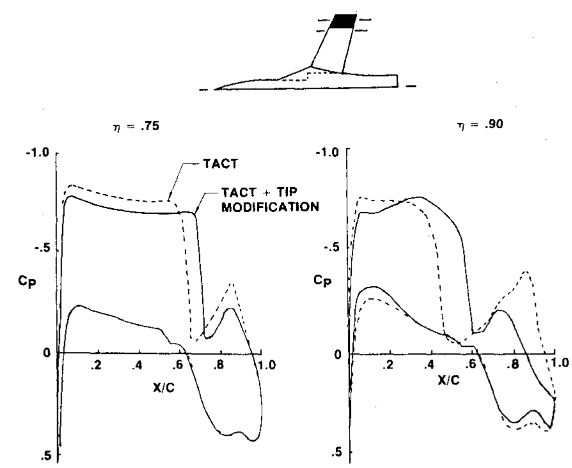

Fig. 16 Effect of tip modification on wing pressures, $M_\infty = 0.90$, $\alpha = 6.0$ deg.

body geometry, shows substantial improvement in correlation with the experimental results on the wing lower surface than was seen for the flat-sidewall results shown in Fig. 3.

Figure 10 shows wing surface pressure distribution comparisons between the transonic code and experimental data for M = 0.85. The correlations are quite good, except for the prediction of the shock location. The pressure distributions on the forward portion of the upper surface and most of the lower surface are well predicted. Although a pseudo-boundary-layer thickness has been added to the wing geometry, the calculations still overpredict the loading at the trailing edge. Data at the inboard wing station indicate the presence of a highly swept forward shock. The transonic code does not sharply define this shock, but rather tends to smear it. At the outboard wing station, the shock is much better defined. The transonic code predicts the shock location 10-15% chord too far forward, and this must be attributed, in part, to the omission of the viscous interactions at and downstream of the base of the shock. If these viscous effects could be better accounted for, then a conservative differencing scheme must be used to properly capture the shock.

Figure 11 shows test-theory comparisons for $M = 0.90$. These comparisons show good correlation on the upper surface ahead of the shock location. Correlation between predicted results and experimental data on the wing lower surface was poorer than at lower Mach numbers. Probable causes of this discrepancy are due to the inadequate modeling of the trailing edge separations. The resulting loss of circulation will allow higher velocity flow on the lower wing surface, as indicated by the test data. More studies are necessary to develop techniques which will provide a means of modeling the separated boundary layer and the wake.

The fuselage modeling used in the transonic calculations shown so far was based on a panel solution for an inlet velocity ratio of 0.8. Calculations were also performed on a model with an inlet velocity ratio of 0.6. These results showed little effect of inlet velocity ratio on the variable sweep portion of the wing. This is because the inlet face is ahead of that portion of the wing leading edge. The inlet does strongly influence the glove portion of the wing, as shown in Fig. 12. The lower inlet velocity ratio results in a strong expansion and resultant shock not seen at the higher inlet values. Unfortunately, experimental data were not available at this wing station.

IV. Numerical Design Modifications

With the empirical fixes and with the effects of the inlet incorporated by the equivalent fuselage, it has been shown that the procedure adequately predicted the primary deficiencies of the basic TACT configuration. Thus, we can use the procedure to determine configuration improvements in the inboard and tip regions of the wing to eliminate the deficiencies.

Consider first the more serious inboard deficiency. Here, a simple cure is to place a suitably shaped half airfoil over the wing-glove upper surface to generate the necessary suctions, as shown schematically in Fig. 13. Results with such a bump are shown in Fig. 14 where again the shock capture is non-conservative. The results indicate that the bump has indeed removed the undesirable saddle-shaped upper surface pressure distribution; and, perhaps not unexpectedly, the effect is felt to the wing tip. Unfortunately, the resulting increases in lift were, in this example, offset by an increase in drag caused, in part, by the large rearward displacement of the shock which would not occur with the proper viscous interaction. Clearly, further effort will be required to obtain the optimal recontouring in the inboard region.

Finally, consider the deficiency in the wing tip region. As noted earlier, the erosion of the upper surface suction plateau caused an upstream displacement of the primary shock leading to a strong second shock wave. To compensate for this plateau erosion, additional upper surface convexity was added at the tip section, as shown in Fig. 15. The wing inboard of wing butt line (WBL) 251 was kept unmodified; the modified section installed at the wing tip section WBL 327, and the intermediate sections then linearly interpolated. The resulting pressure distributions are shown in Fig. 16. The results show that the tip modification has essentially eliminated the double shock configuration in the tip region, resulting in lower section drags. It is interesting to note that this performance improvement was accomplished with a significant increase in wing thickness near the tip. It should also be noted that the inboard bump modification previously shown caused larger pressure distribution changes in the tip region than did the tip modification.

V. Summary and Conclusions

In summary, the preceding results demonstrated that the small-disturbance method, supplemented by the suggested empirical fixes, can play an important role in the configuration development of advanced aircraft. Further experience clearly is needed to establish its dependability. Procedures by which the nonplanar subcritical panel methods can be combined with the nonlinear finite-difference methods can lead to improved solutions with existing transonic codes. The techniques of fuselage modeling shown in this paper can also be applied to other finite-difference methods, such as the full potential method of Jameson.[8] Another example of such combination techniques is given in Ref. 9, in which a finite-difference method is imbedded in a panel solution to substantially reduce the far-field finite-difference calculations. The strong, spanwise propagation of disturbances illustrates the role that the design of the inboard portion of the wing and the fuselage can have on the outboard portion of the wing, and may explain why local "fixes" sometimes do not work. Clearly, the area in greatest need of improvement is that of properly accounting for the viscous effects. The 3-D boundary-layer method of McLean[10] is currently being integrated with several inviscid codes to improve this situation. But more than just a 3-D boundary-layer method will be needed to account for strong shock/boundary-layer interaction and to appropriately model the wing trailing edge wake.

References

[1] Yoshihara, H., Benepe, D., and Whitten, P., "Transonic Performance of Jet Flaps on an Advanced Fighter Configuration," AFFDL-TR-73-97, 1973.

[2] Ballhaus, W., Bailey, R., and Frick, J., "Improved Computational Treatment of Transonic Flow About Swept Wings," NASA CP-2001, 1976.

[3] Bhateley, I. and Yoshihara, H., "A Transonic Calculation Study of Configuration Changes to the F-111 (TACT)," Convair Rept. AM-76-SP-08, Oct. 1976.

[4] Rubbert, P. E. and Saaris, G. R., "3D Potential Flow Method Predicts V/STOL Aerodynamics," *SAE Journal*, Vol. 77, Sept. 1969, pp. 44-51.

[5] Rubbert, P. E. and Saaris, G. R., "Review and Evaluation of a Three-Dimensional Lifting Potential Flow Analysis Method for Arbitrary Configurations," AIAA Paper 72-188, AIAA 10th Aerospace Sciences Meeting, San Diego, Calif., Jan. 17-19, 1972.

[6] Boppe, C. W., "Calculation of Transonic Wing Flows by Grid Embedding," AIAA Paper 77-207, AIAA 15th Aerospace Sciences Meeting, Los Angeles, Calif., Jan. 24-26, 1977.

[7] Bauer, F., Garabedian, P. Korn, D., and Jameson, A., "Supercritical Wing Sections II," *Lecture Notes in Economics and Mathematical Systems*, No. 108, Springer-Verlag, New York, 1975.

[8] Jameson, A. and Caughey, D. A., "A Finite Volume Method for Transonic Potential Flow Calculations," AIAA Paper 77-635, AIAA 10th Fluid and Plasma Dynamics Conference, Albuquerque, N. Mex., June 27-19, 1977.

[9] Chen, A. W., Dickson, L. J., and Rubbert, P. E., "A Far Field Matching Method for Transonic Computations," *AIAA Journal*, Vol. 15, Oct. 1977, pp. 1491-1497.

[10] McLean, J. D., "Three-Dimensional Turbulent Boundary Layer Calculations for Swept Wings," AIAA Paper 77-3, AIAA 15th Aerospace Science Meeting, Los Angeles, Calif., Jan. 24-26, 1977.