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AIRFOIL DESIGN UTILIZING PARALLEL PROCESSORS, PART II: APPLICATIONS

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Abstract

One test case and two airfoil design applications were performed utilizing a parallel optimization scheme coupled to different flow solvers. Parallel processors use computational fluid dynamics to evaluate the aerodynamic performance of multiple geometries simultaneously. The test case designed an airfoil to match the pressure distribution corresponding to an airfoil of a known shape. A transonic design application varied an airfoil's shape to maximize its lift-to-drag ratio. Also, a turbine blade design case used a Navier-Stokes flow solver to minimize viscous losses while maintaining adequate volume in the blade for cooling purposes. These design applications demonstrate the practicality, versatility, and possible design utilizations for aerodynamic design via optimization using parallel processors.

Nomenclature

C_{dw}	coefficient of wave drag
C_l	coefficient of lift
C_{loss}	viscous loss coefficient
C_p	coefficient of pressure
f	objective function
Vol	volume of airfoil

Introduction

This is the second of a two part series which introduces an aerodynamic optimization scheme utilizing parallel processors. Part I presents the implementation of the parallel optimization scheme and compares it with a similar sequential optimization scheme in an airfoil design test case.[1] This paper discusses three applications of aerodynamic design using the parallel optimization scheme.

Airfoil design applications were completed utilizing the parallel quasi-Newton optimization routine PARQNM. PARQNM utilized multiple processors to simultaneously evaluate numerous objective functions based upon the aerodynamic performances of various airfoil shapes. Aerodynamic performance evaluations were necessary to estimate gradients and to search for airfoil shapes which minimized the objective function in the quasi-Newton optimization scheme.

One test case was constructed to design an airfoil to match the pressure distribution corresponding to an airfoil of a known geometry. The design airfoil approached the shape of the airfoil used to calculate the target pressure distribution.

An application involving transonic flow designed an airfoil to increase its lift-to-drag ratio. Another design application used an internal flow Navier-Stokes flow solver to design a symmetric cascade blade to minimize viscous losses

Parallel Optimization Scheme

Airfoil design via optimization methods requires numerous CFD solutions to compute the aerodynamic performances of airfoils with different geometries. In the design process, independent

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variables are varied to determine which geometry optimizes the design performance criteria.

For airfoil design, the designer must first select the desired performance criteria. Next, independent variables are used to describe the geometry of an airfoil's shape. After the initial CFD solution and performance evaluation are calculated, the gradient vector is calculated based upon small perturbations of each independent variable. The quasi-Newton optimization method uses an approximation of the Hessian matrix to calculate a direction of search to vary the independent variables. Multiple CFD solutions are required each optimization cycle to calculate the gradient vector and to minimize an objective function, f, based upon the performance criteria. The flow-field calculations require the vast majority of processing time required for aerodynamic design via optimization.

A quasi-Newton optimization scheme was developed for aerodynamic design utilizing the Intel iPSC/860 hypercube parallel computer. The hypercube is used to simultaneously calculate the flow fields over multiple airfoil geometries for the estimation of the gradient vectors and in directional searches for minimum objective functions. Conducting the gradient calculations and line searches in parallel greatly increases the speed and efficiency of the design procedure.

For a second-order estimation of each component of the gradient, two function evaluations are required. For n independent variables, 2n processors are used in PARQNM to calculate all forward and backward-difference function evaluations simultaneously for the estimation of the gradient vector.

Similar to the gradient calculation, the line search locates a minimum objective function and requires the performance evaluations of numerous airfoil shapes corresponding to different sets of independent variables. The designer must select maximum and minimum variations to the airfoil's geometry. For an optimization cycle, each processor simultaneously evaluates a unique set of independent variables along the direction of search. If the minimum objective function is less than the previously calculated minimum, the new set of independent variables are sent to all processors for the next optimization cycle.

A local search is performed when a line search fails to reduce the objective function. The local search uses random directional searches to check whether a point can be found lower than the estimated minimum. Two line searches in different directions are conducted to find a set of independent variables corresponding to a lower objective function.

Design of a Lifting Subsonic Airfoil

A test case was conducted which utilized the parallel quasi-Newton optimization routine to design an airfoil to match the pressure distribution of a cambered airfoil at a small angle of attack.

Baseline Airfoil and Independent Variables

The baseline airfoil used in this test case is a NACA 1410. The camber and thickness of the airfoil were perturbed each optimization cycle.

The independent variables were selected to be eight collocation points describing the surface of the airfoil. Four independent variables were collocation points on the lower surface of the airfoil, and four independent variables were collocation points on the upper surface of the airfoil. A geometry package developed by the McDonnell-Douglas Corp. was used to compute two fifth-order Chebychev polynomials describing the surfaces.[2] The collocation points were varied slightly each optimization cycle to recompute airfoil shapes needed for the gradient calculation and the directional search.

The GRAPE grid generation program was used to compute a 133 x 34 grid around each airfoil shape described by the independent variables.[3]

Flow Solver

The inviscid pressure distributions around the airfoil shapes evaluated in this test case were calculated using a two-step Runge-Kutta scheme Euler flow solver RK2EULER.[4] RK2EULER updated the flow-field properties around various airfoil shapes at an angle of attack of two degrees and a freestream Mach number of 0.6 to solve for their steady-state pressure distributions. The pressure at the surface of the airfoil shapes were used for the calculation of the objective function in the optimization process. The flow-field properties were initialized to freestream conditions based upon a density of one and pressure equal to the reciprocal of the ratio for specific heat for a perfect gas. Sixteen hundred flow-field iterations were conducted for each performance evaluation.

Performance Criterion

The goal of this test case was to design an airfoil to match or optimize the inviscid pressure distribution around a NACA 2412 airfoil at 2 degrees angle of attack and a Mach number of 0.6. The coefficients of pressure for 73 points around the NACA 2412 for the design flight conditions were used to formulate the objective function,

$$f = \sum_{i=1}^{73}\left[C_{p\text{-calculated}_i} - C_{p\text{-target}_i}\right]^2 \qquad (1),$$

which represents the difference between the desired and actual pressure distributions.

The Mach contours around the baseline airfoil are shown in Figure 1, and the Mach contours around a NACA 2412 in identical flight conditions are shown in Figure 2. Also, the corresponding pressure distributions around both airfoils are shown in Figure 3. Six optimization cycles were performed.

Results

The test case was completed using 16 processors and 8 hours of processing time on the hypercube. The convergence history of this case is shown in Figure 4. The airfoil design application reduced the objective function to less than 10% of its original value in five optimization cycles.

The optimization design airfoil and the NACA 2412 target airfoil are shown in Figure 5. The design airfoil is slightly thinner than the target airfoil after six optimization cycles. The resulting pressure distributions of both airfoils are shown in Figure 6.

Transonic Airfoil Design

A parallel design application was performed to maximize the lift-to-drag ratio of an airfoil in transonic flight conditions. In transonic flow, a small change in the shape of an airfoil results in a large change in its pressure distribution due to changes in shock locations. Unlike the previous test case, a specific airfoil shape was not used as a target for the solution.

Baseline Airfoil and Independent Variables

The baseline airfoil used in this test case was a NACA 0012. Eight independent variables were chosen as the thickness at eight points along the chord of the airfoil. The McDonnell-Douglas geometry package was used to compute a ninth-order Chebychev polynomial describing the thickness distribution of the airfoil. The camber along the airfoil was set to zero and not varied.

GRAPE was again used to compute a 133 x 34 grid around each airfoil shape described by the independent variables.

Flow Solver

The flow field around each evaluated airfoil shape was calculated using the explicit Euler solver RK2EULER. RK2EULER updated the flow-field properties around various airfoil shapes at an angle of attack of one-half degree and a freestream Mach number of 0.8 to solve for their steady-state pressure distributions. The evaluated lift and wave drag coefficients for each geometry were used in the calculation of the objective function.

Flow-field properties were initialized to freestream conditions prior to each evaluation, and 1800 flow-field iterations were performed.

Performance Criterion

The goal of this application was to design a symmetric airfoil to maximize its inviscid lift-to-drag ratio. The objective function was selected as the square of the wave drag-to-lift ratio,

$$f = \left(\frac{C_{dw}}{C_l}\right)^2 \qquad (2).$$

The Mach contours around the baseline airfoil are shown in Figure 7. Three optimization cycles were performed for this design application.

Results

Mach contours around the design airfoil are shown in Figures 8. The thickness of the design airfoil is more evenly distributed than with the NACA 0012. Subsequently, the shock on the design airfoil is weaker and farther aft.

The convergence history for this design application is shown in Figure 9. For this design application, the minimum objective function was not known. The parallel optimization routine reduced the objective function to 30% of its original value after three iterations. Also, the convergence rate of the routine decreased significantly with the final two optimization cycles as the objective function approached a minimum. The test case was completed using 16 processors and 7 hours of processing time on the hypercube.

Cascade Blade Design

The goal of this design application was to design a two-dimensional symmetric cascade blade to minimize viscous losses while maintaining adequate volume for cooling purposes. Most turbomachinery blade designs rely upon subsonic analysis or transonic-potential analysis. This test demonstrates the practicality of utilizing an efficient Navier-Stokes flow solver with the parallel quasi-Newton optimization scheme for aerodynamic design.

This test case has several variations from previous ones. Flows in turbomachinery are highly rotational and can be dominated by shock waves and viscous effects. Since the design criteria is based upon viscous losses, a Navier-Stokes flow solver is used. Also, this application involved internal rather than external flow, and periodic boundary conditions must be applied in the flow solver. These variations demonstrate the versatility of using an optimization routine for aerodynamic design.

Baseline Airfoil and Independent Variables

The baseline cascade blade used for this application was a symmetric NACA 0012 airfoil. Eight independent variables were chosen to represent the thickness at eight points along the chord of the cascade blade. The McDonnell-Douglas geometry package was used to calculate the coefficients of a ninth-order Chebychev polynomial which described the thickness distribution of each cascade blade shape evaluated in the design process.

A modification to the GRAPE grid generation program was used to generate a 250 x 60 C type grid around the airfoils evaluated in the design process. Grids used for Navier-Stokes flow solvers require more grid points than those used with inviscid flow solvers, especially near the surface of the airfoil for calculation of flow-field properties within the boundary layer where viscous effects are significant.

Flow Solver

The selected performance criteria required a Navier-Stokes flow solver to evaluate the viscous losses of the internal cascade flow at a freestream Mach number of 0.6. Chima developed a multi-stage Runge-Kutta scheme algorithm for quasi three-dimensional flows in turbomachinery.[5] This efficient Navier-Stokes code was developed for turbomachinery design and analysis. The thin viscous layer assumption has been invoked to eliminate the streamwise viscous derivatives and reduces the processing time for the computation of separated flows. The algebraic two layer eddy-viscosity model developed by Baldwin and Lomax is used for the evaluation of turbulent flows.[6]

A five-stage Runge-Kutta scheme is applied to this problem. Because of the complexity of the flow, the flow field around each cascade blade geometry was initialized based upon freestream conditions prior to each performance evaluation. The larger grid size, more Runge-Kutta steps and viscous calculations require significantly more processing time per flow-field iteration than was needed in the previous design applications. Therefore, 400 flow-field iterations were performed for each geometry evaluated. Mach contours around the baseline airfoil are shown in Figure 10.

Performance Criteria

The purpose of this design application was to design a cascade blade which minimizes viscous losses while maintaining adequate volume for cooling. An objective function was formulated which accounted for a trade-off of these two factors.

The explicit Navier-Stokes flow solver was used to evaluate a loss coefficient for the viscous losses through the cascade. The loss coefficient was based upon the loss of total pressure from the cascade inlet to the cascade exit due to viscous effects.

For a decrease in viscous losses, the symmetric cascade blade would decrease in thickness and volume. The change in airfoil volume from its original volume was incorporated in the calculation of the objective function

$$f = C_{loss} \frac{Vol_{NACA\ 0012}}{Vol_{DESIGN\ BLADE}} \qquad (3).$$

The stopping criteria was set for 2 optimization cycles because of the increased processing time required for the viscous flow evaluations. Each optimization cycle required approximately 4 hours processing time on 16 processors.

Results

The parallel optimization routine successfully reduced the objective function. Two optimization cycles were completed using 16 processors of the iPSC/i860 parallel computer. Mach contours around the design airfoil are shown in Figure 11. Also, the values of airfoil volume, coefficient of viscous losses and objective function each optimization cycle are shown in Figure 12.

The optimization scheme decreased the thickness of the airfoil near its nose and increased its thickness aft of 60% chord. The flow over the design airfoil accelerated more gradually which resulted in a smaller wake. The overall volume of the airfoil increased 5%, and the viscous loss coefficient decreased 19% in the design process.

Conclusions

The primary purpose of this research was to demonstrate the advantages of using parallel computers for airfoil optimization. This work applies recent advances in parallel supercomputing technology to an intuitively parallel problem.

A major advantage of airfoil design via optimization over inverse airfoil design techniques is the independence of the optimization routine from the flow solver. This allows various flow solvers to be used with the same optimization routine, and the flow solver can be selected based upon the desired performance criteria.

In the past, Navier-Stokes flow solvers have not widely been used in aerodynamic design due to the large amount of required processing time. This restriction in the choice of flow solvers has also restricted the designer's selection of desired performance criteria. This research has proven that successful design applications involving viscous phenomena can be accomplished in a reasonable amount of processing time utilizing an efficient Navier-Stokes flow solver and the parallel quasi-Newton optimization scheme.

Importance of Designer Intervention

The single most important factor in any aerodynamic design application is the supervision and intervention of the designer in a design process. The designer must decide the desired performance criteria to optimize and mathematically formulate this criteria into an objective function. The designer must also select the appropriate flow solver and independent variables for the optimization routine based upon the performance criteria. Furthermore, the designer must carefully examine the results and make necessary changes to the problem in order to ensure a meaningful solution.

Future Design Applications

Many areas of aerodynamics could benefit using parallel optimization design applications, especially areas with little empirical data. Future applications should include the design of hypersonic wings, helicopter rotor blades, and engine compressor and turbine blades. Three-dimensional optimization design can be accomplished using recently developed parallel supercomputers with faster processors and more memory.

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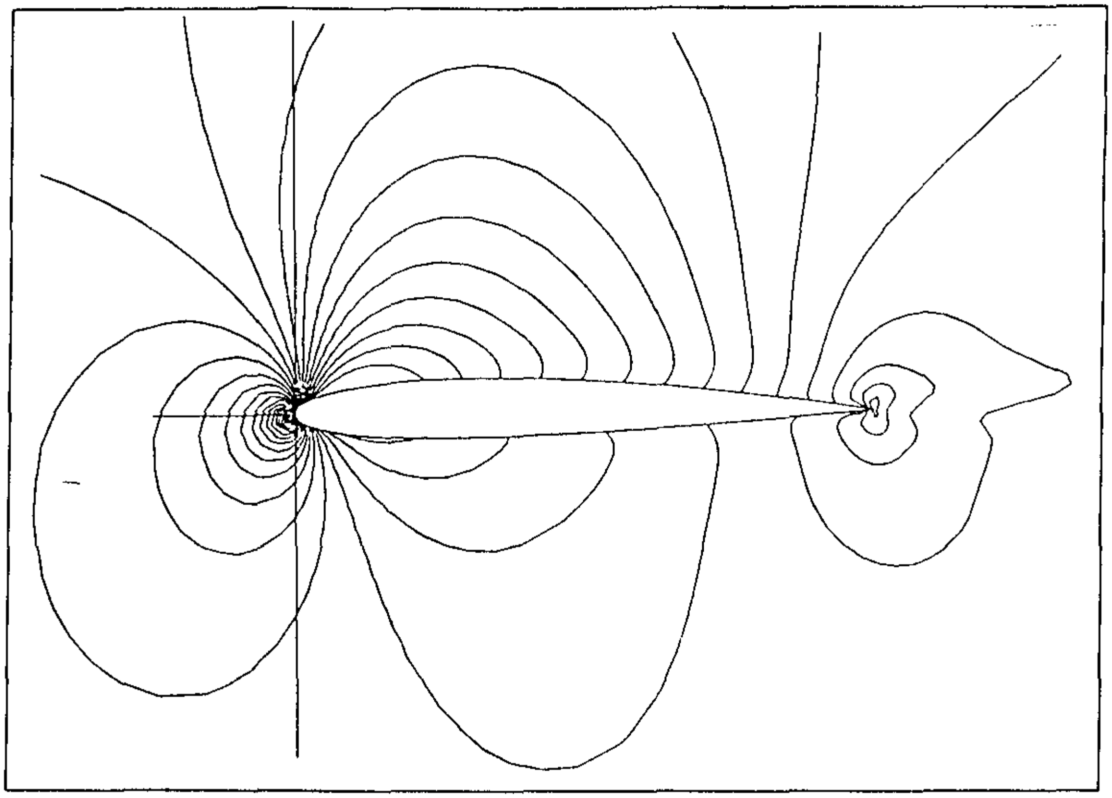

Figure 1: Mach Contours Around NACA 1410 Baseline Airfoil, AOA=2 deg, M=0.6

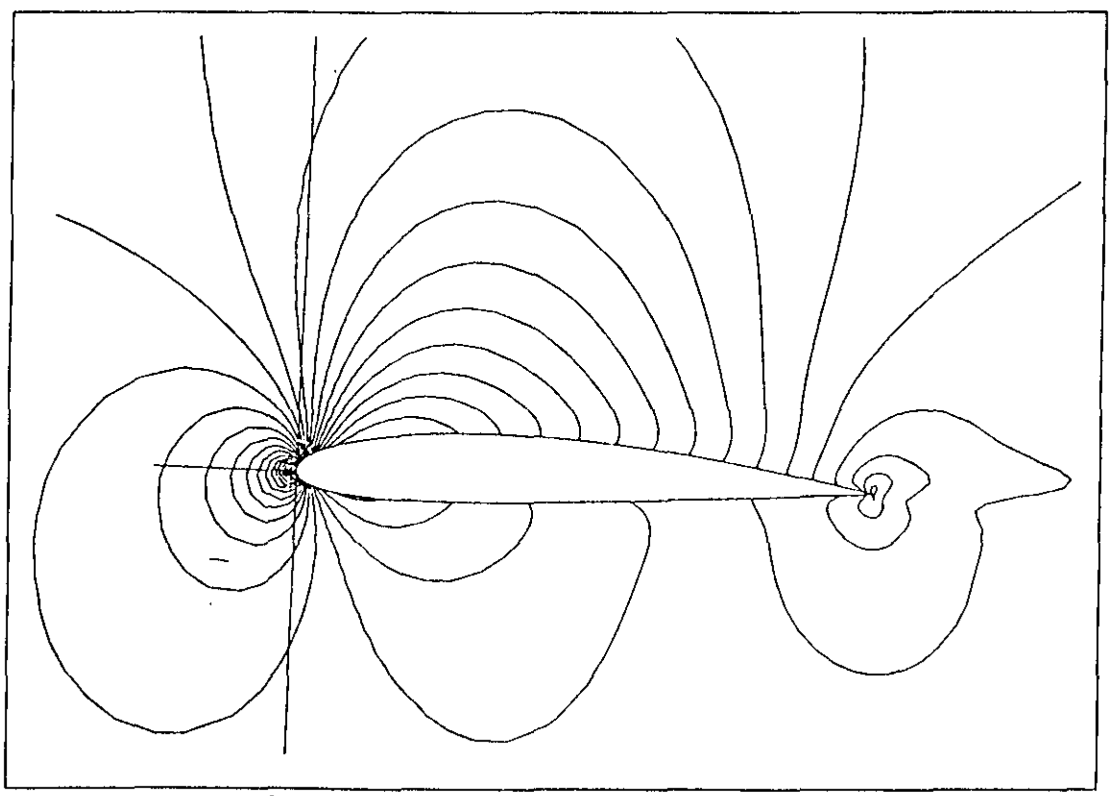

Figure 2: Mach Contours Around NACA 2412 Target Airfoil, AOA=2 deg, M=0.6

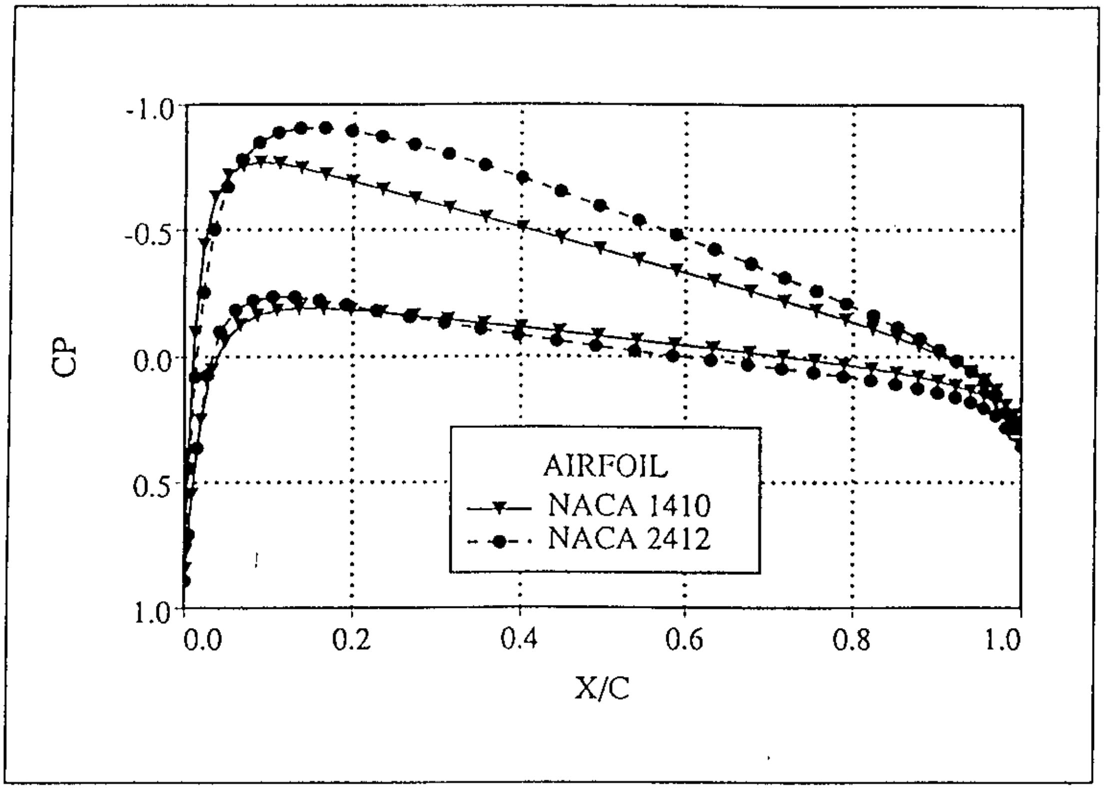

Figure 3: Pressure Distributions of Baseline and Target Airfoils

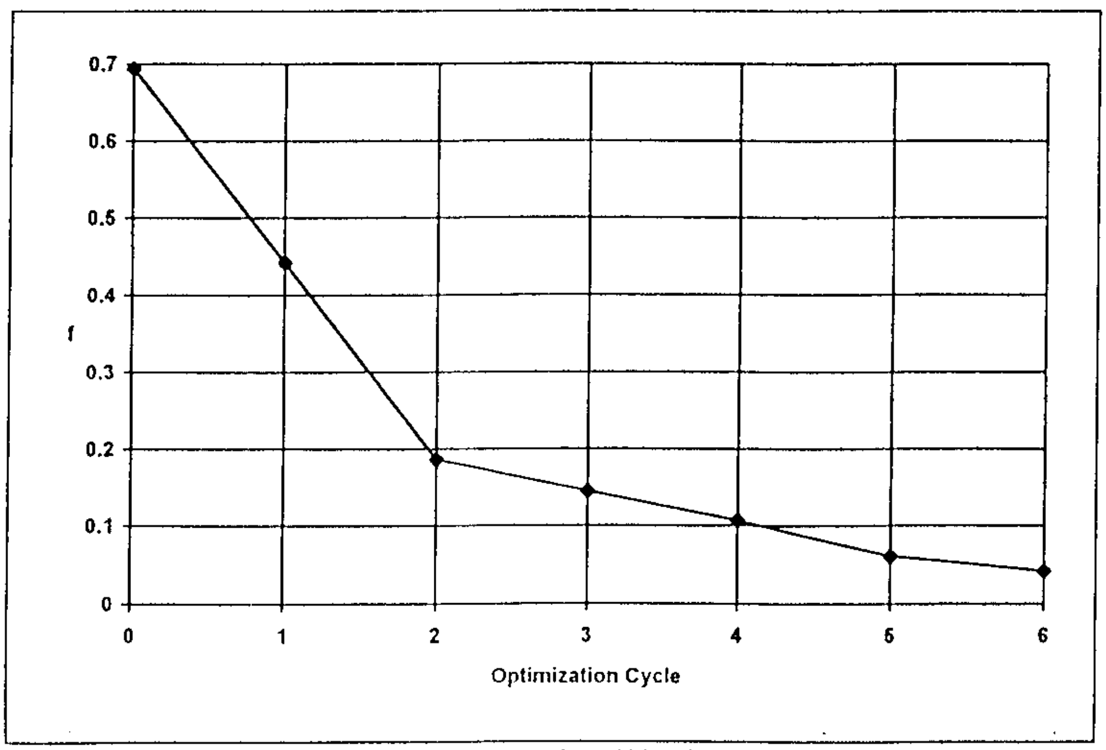

Figure 4: Convergence History for Lifting Subsonic Application

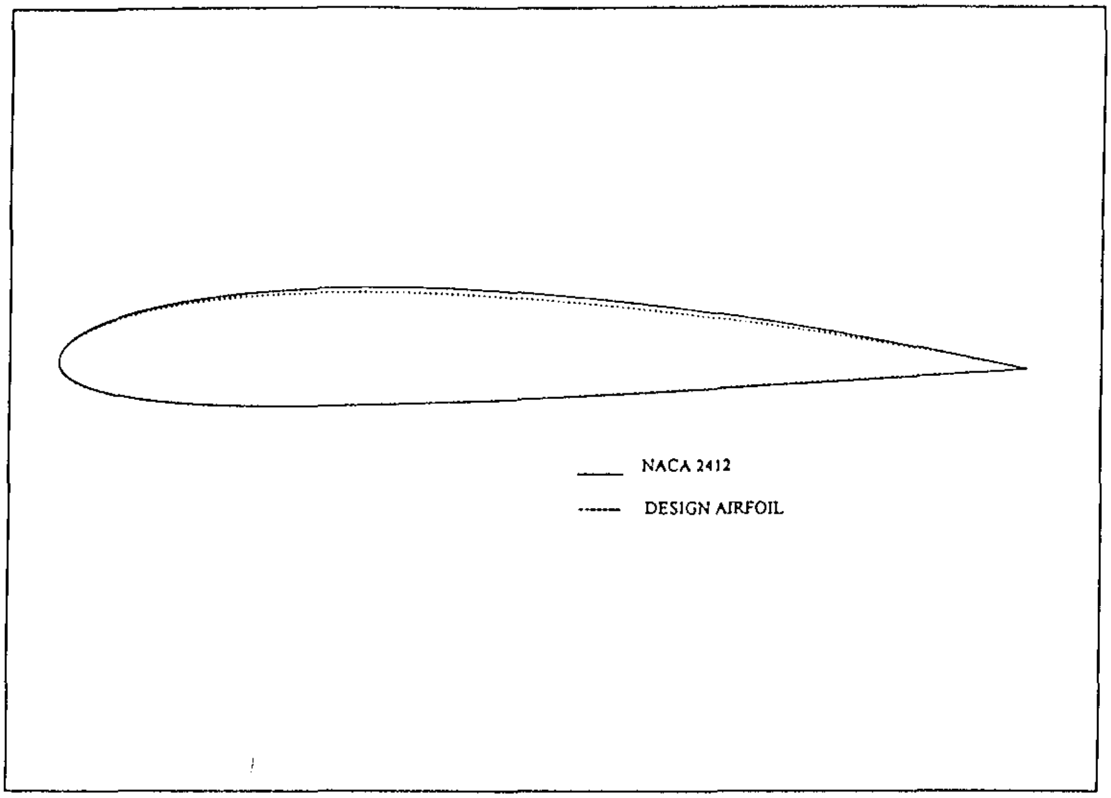

Figure 5: Design and Target Airfoils

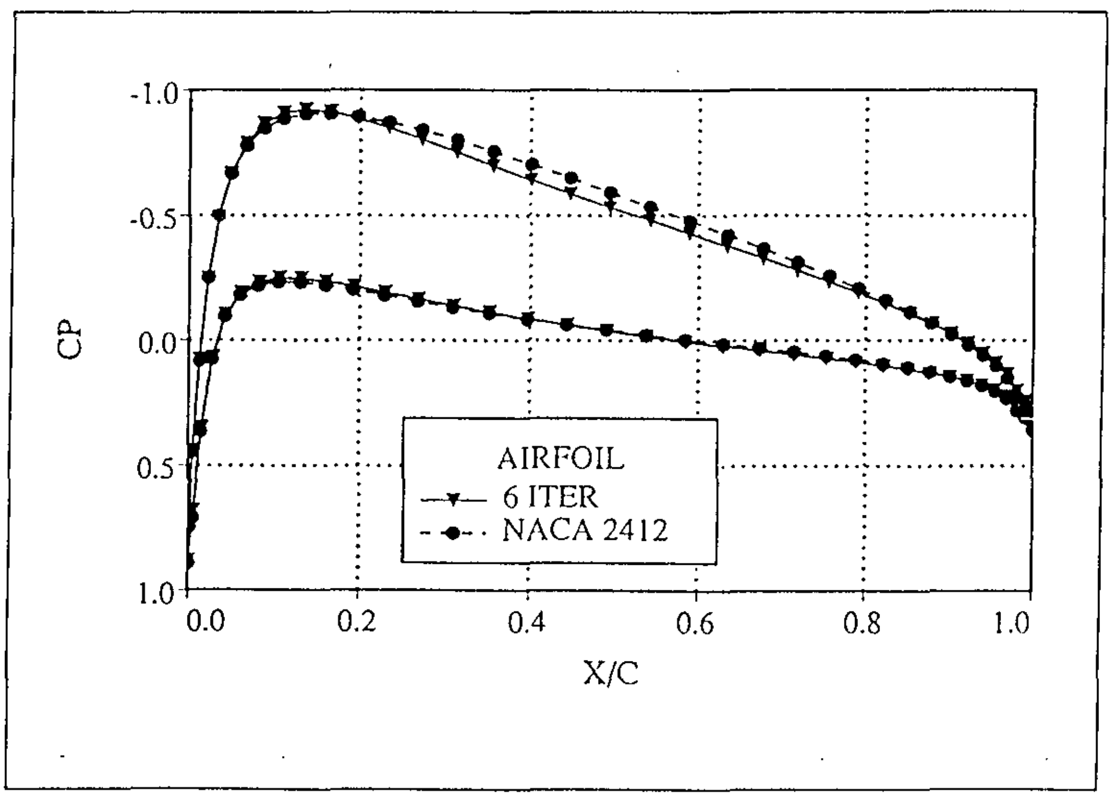

Figure 6: Design and Target Airfoils' Pressure Distributions

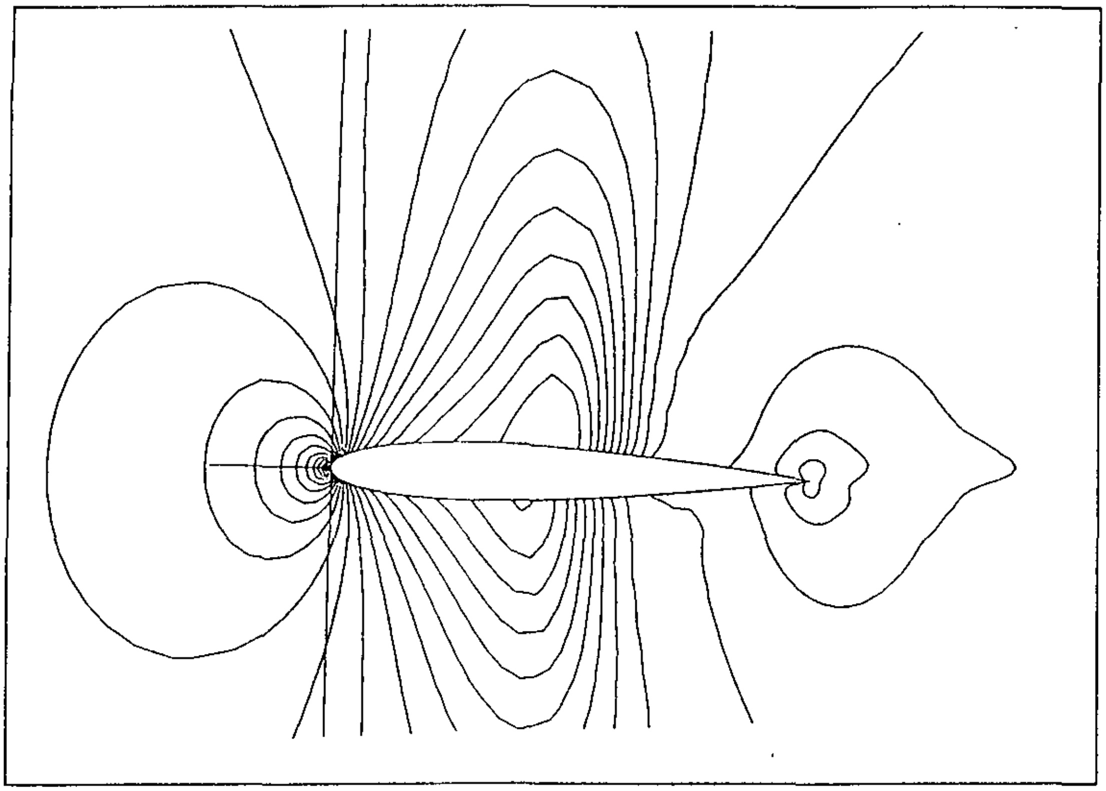

Figure 7: Mach Contours Around NACA 0012 Baseline Airfoil, AOA=0.5 deg, M=0.8

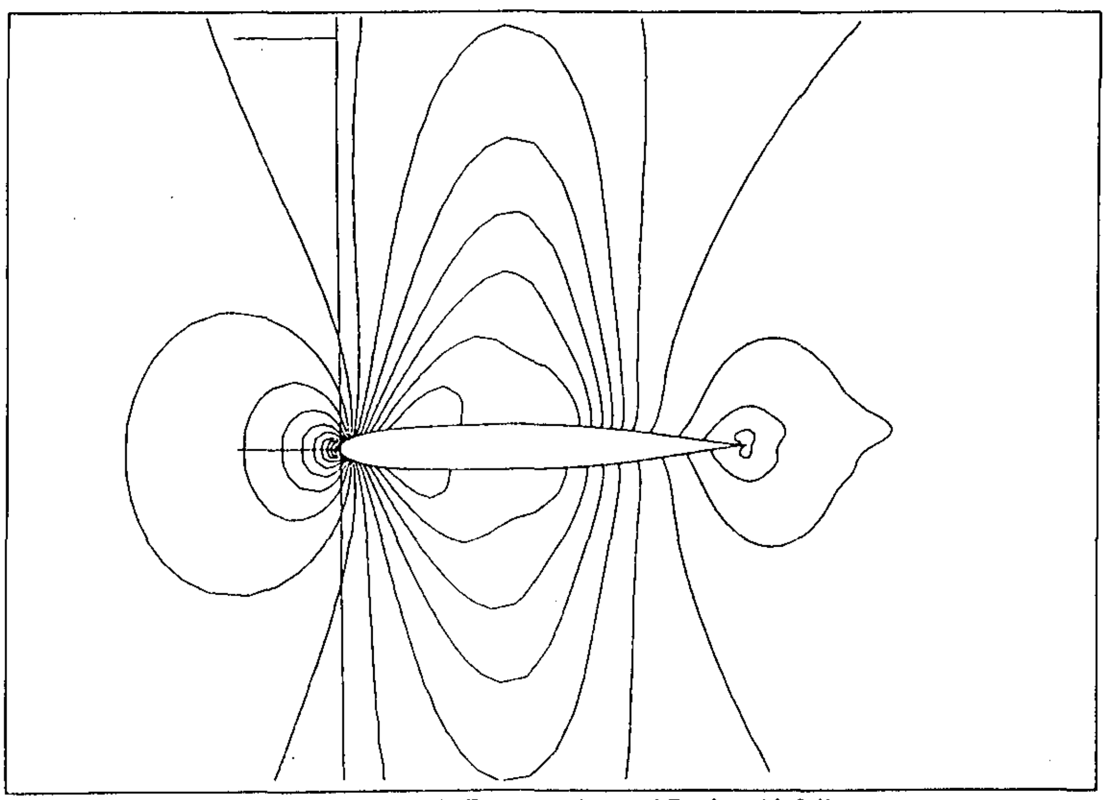

Figure 8: Mach Contours Around Design Airfoil

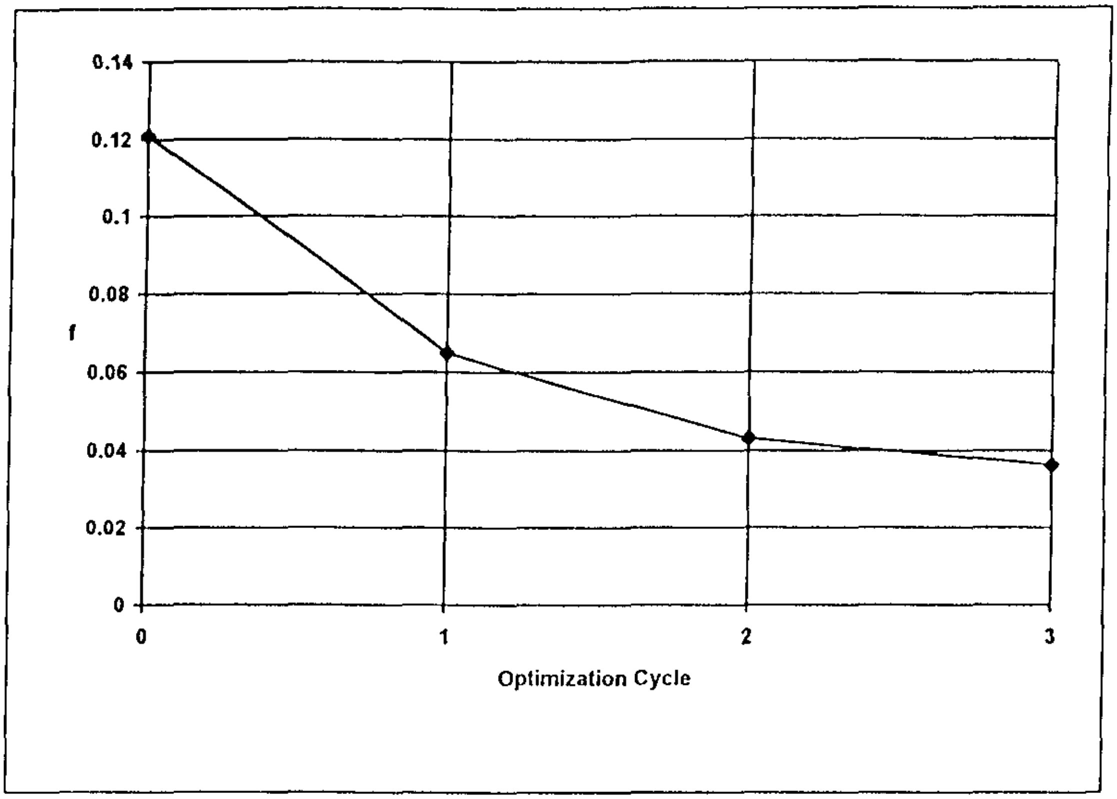

Figure 9: Convergence History for Transonic Design

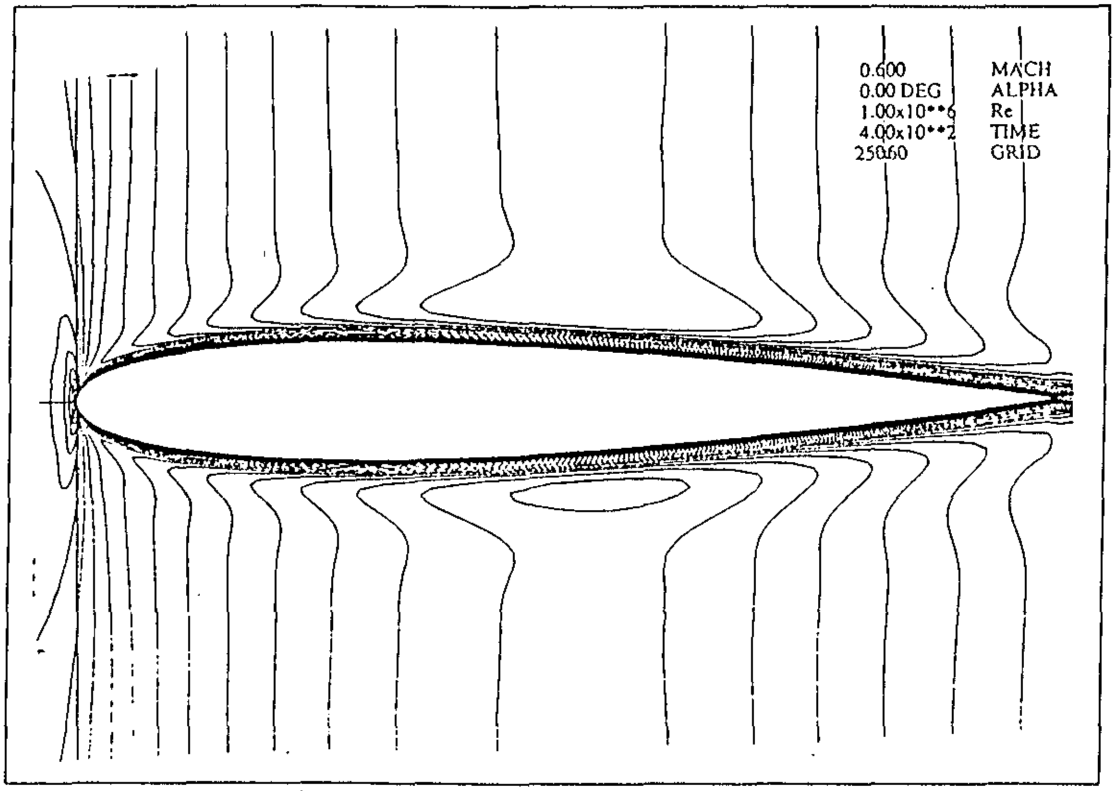

Figure 10: Mach Contours Around Baseline NACA 0012 Cascade Blade, M=0.6

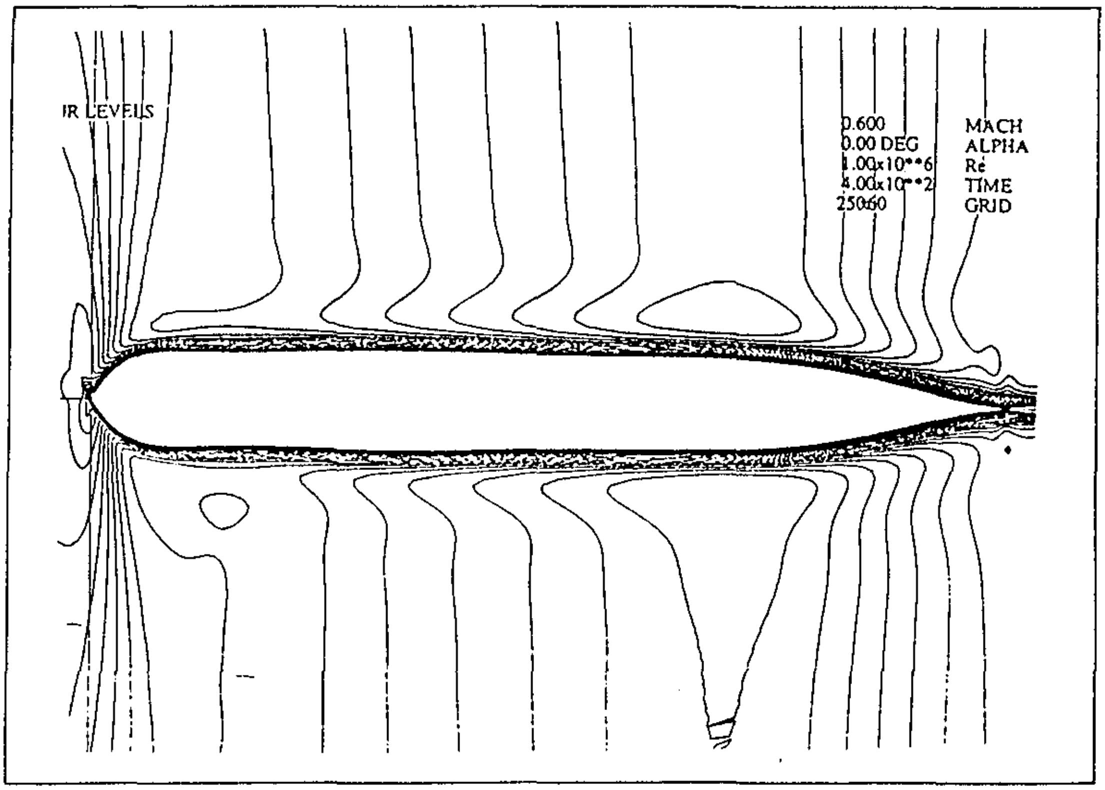

Figure 11: Mach Contours Around Design Cascade, M=0.6

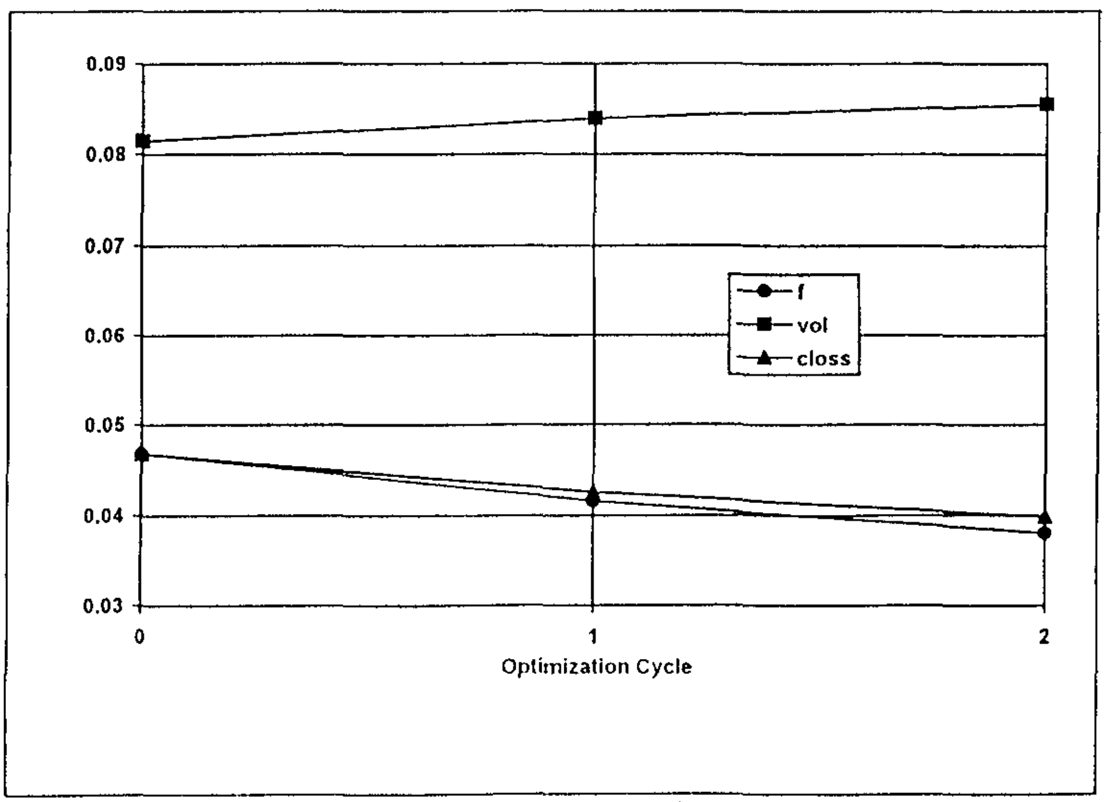

Figure 12: Convergence History for Cascade Design