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The goal of our research under AFOSR sponsorship is to develop mathematical procedures which can be used to arrive at optimum, or near optimum, aerodynamic shapes by merging techniques from computational fluid dynamics and control theory.
With this in mind we have continued to work in two main topics:

1. *Development of high resolution shock capturing schemes with low numerical diffusion*
   Since last year the symmetric limited positive (SLIP) and upstream limited positive (USLIP) schemes have been improved by the introduction of a new flux limiter which guarantees positivity, while maintaining good accuracy in smooth flow regions. A comprehensive theory has also been developed for design of numerical fluxes which guarantee stationary discrete shocks with a single interior point.
2. *Aerodynamic shape optimization by boundary control*
   Progress in aerodynamic shape optimization has been realized on two fronts. First the method has been successfully implemented for two-dimensional lifting potential flows using a general finite volume scheme with numerically generated grids.
   Secondly, the method has been successfully implemented for three-dimensional wing design using the Euler equations.

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Computational Algorithms for Aerodynamic Analysis and Design

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AFOSR 91-0391

September 1994

1 Introduction

The goal of our research under AFOSR sponsorship is to develop mathematical procedures which can be used to arrive at optimum, or near optimum, aerodynamic shapes by merging techniques from computational fluid dynamics and control theory. A prerequisite for the success of such a program is the ability to calculate the aerodynamic flow over a given shape both accurately and efficiently, since optimization will require repeated aerodynamic calculations as the shape is modified. With this in mind we have continued to work in two main topics:

1. Development of high resolution shock capturing schemes with low numerical diffusion.

2. Aerodynamic shape optimization by boundary control.

2 High Resolution Shock Capturing Schemes

Since last year the symmetric limited positive (SLIP) and upstream limited positive (USLIP) schemes have been improved by the introduction of a new flux limiter which guarantees positivity, while maintaining good accuracy in smooth flow regions. In the SLIP scheme artificial diffusion is introduced as

$$d_{j+\frac{1}{2}} = \alpha_{j+\frac{1}{2}} \left\{ \Delta v_{j+\frac{1}{2}} - L\left(\Delta v_{j+\frac{3}{2}}, \Delta v_{j-\frac{1}{2}} \right) \right\}$$

where $L(u, v)$ is a limited average of u and v which is zero when u and v have positive signs. The limiter which has proved most effective has the form

$$L(u,v) = \frac{1}{2} R(u,v)(u+v)$$

where

$$R(u,v) = 1 - \left| \frac{u-v}{\max\left(|u|+|v|\right), \epsilon \Delta x^{\frac{3}{2}}} \right|^{\beta}, \beta \geq 2$$

The threshold in the denominator allows second order accuracy to be preserved at smooth extrema.

A theory has also been developed for design of numerical fluxes which guarantee stationary discrete shocks with a single interior point. While this can be achieved by characteristic based schemes, it can also be achieved by a class of much simpler schemes using scalar diffusion augmented by pressure differences. Both types of schemes can be designed to preserve exactly constant values of stagnation enthalpy, which is not allowed by the usual characteristics based schemes. Figure (1) shows representative a 2D result computed on a 320x64 mesh with just 25 multigrid cycles.

Careful studies have also confirmed that the low numerical diffusion of these schemes results in excellent resolution of viscous boundary layers. These results are reported in Reference [6]

3 Aerodynamic Shape Optimization by Boundary Control

Progress in aerodynamic shape optimization has been realized on two fronts. First the method has been successfully implemented for two-dimensional lifting potential flows using a general finite volume scheme with numerically generated grids. Thus the need to rely on analytic transformations for grid generation has been eliminated. This result is reported in Reference [5].

Secondly, the method has been successfully implemented for three-dimensional wing design using the Euler equations. Since three dimensional calculations are much more expensive than two dimensional calculations, it is extremely important to use fast solution algorithms for both the flow and the adjoint equations. In this case the author's FLO87 computer program has been used as the basis of the design method. FLO87 solves the three dimensional Euler equations with a cell-centered finite volume scheme, and uses residual averaging and multigrid acceleration to obtain very rapid steady state solutions, usually in 25 to 50 multigrid cycles [2, 3]. Upwind biasing is used to produce nonoscillatory solutions, and assure the clean capture of shock waves. This is introduced through the addition of carefully controlled numerical diffusion terms, with a magnitude of order Δx^3 in smooth parts of the flow. The adjoint equations are treated in the same way as the flow equations. The fluxes are first estimated by central differences, and then modified by downwind biasing through numerical diffusive terms which are supplied by the same subroutines that were used for the flow equations.

The method has been tested for the optimization of a swept wing. The planform was fixed while the wing sections were free to be changed arbitrarily by the design method. The wing has a unit-semi-span, with 36 degrees leading edge sweep. It has a compound trapezoidal planform, with straight taper from a root chord of 0.38 to a chord of 0.26 at the 30 percent span station, and straight taper from there to a chord of 0.12 at the tip, with an aspect ratio of 8.7. The initial wing sections were based on the Korn airfoil, which was designed for shock free flow at Mach 0.75 with a lift coefficient of 0.63, and has a thickness to chord ratio of 11.5 percent [1]. The thickness to chord ratio was increased by a factor of 1.2 at the root and decreased by a ratio of 0.8 at the tip, with a linear variation across the span. The inboard sections were rotated upwards to give 3.5 degrees twist across the span.

The two-dimensional pressure distribution of the Korn airfoil at its design point was introduced as a target pressure distribution uniformly across the span. This target is presumably not realizable, since it would correspond to a lifting wing with zero vortex drag. It serves, however, to favor the establishment of a relatively benign pressure distribution. The total inviscid drag coefficient, due to the combination of vortex and shock wave drag, was also included in the cost function. Calculations were performed with the lift coefficient forced to approach a fixed value by adjusting the angle of attack every fifth iteration of the flow solution. A grid with 192x32x48 = 294912 was used, and the wing shape was determined by 133 sections each with 128 mesh points for a total of 4224 design variables. It was found that the computational costs can be reduced by using only 15 multigrid cycles in each flow solution, and in each adjoint solution. Although this is not enough for full convergence, it proves sufficient to provide a shape modification which leads to an improvement. Figures 2,3, and 4 shows the result of a calculation at Mach number of 0.82, with the lift coefficient forced to approach a value of 0.5. The plots show the initial wing geometry and pressure distribution, and the modified geometry and pressure distribution after 8 design cycles. The total inviscid drag was reduced from 0.0185 to 0.0118. The initial design exhibits a very strong shock wave in the inboard region. It can be seen that this is completely eliminated, leaving a very weak shock wave in the outboard region.

To verify the solution, the final geometry, after 8 design cycles, was analyzed with another method, using the computer program FLO67. This program uses a cell-vertex formulation, and has recently been modified to incorporate a local extremum diminishing algorithm with a very low level of numerical diffusion [4]. When run to full convergence it was found that the redesigned wing has a drag coefficient of 0.0107 at Mach 0.82 at a lift coefficient of 0.5, with a corresponding lift to drag ratio of 47. The result is illustrated in Figure 5. A calculation at Mach 0.500 shows a drag coefficient of 0.0100 for a lift coefficient of 0.5. Since in this case the flow is entirely subsonic, this provides an estimate of the vortex drag for this planform and lift distribution. Thus the design method has reduced the shock wave drag coefficient to about 0.0007. For a representative transport aircraft the parasite drag coefficient of the wing due to skin friction is about 0.0050. Also the fuselage drag coefficient is about 0.0050, the nacelle drag coefficient is about 0.0015, the empennage drag coefficient is about 0.0015, and excrescence drag coefficient is about 0.0006. This would give a total drag coefficient $C_D = 0.0243$ for a lift coefficient of 0.5, corresponding to a lift to drag ratio $L/D = 20.5$. This would be a substantial improvement over the values obtained by currently flying transport aircraft.

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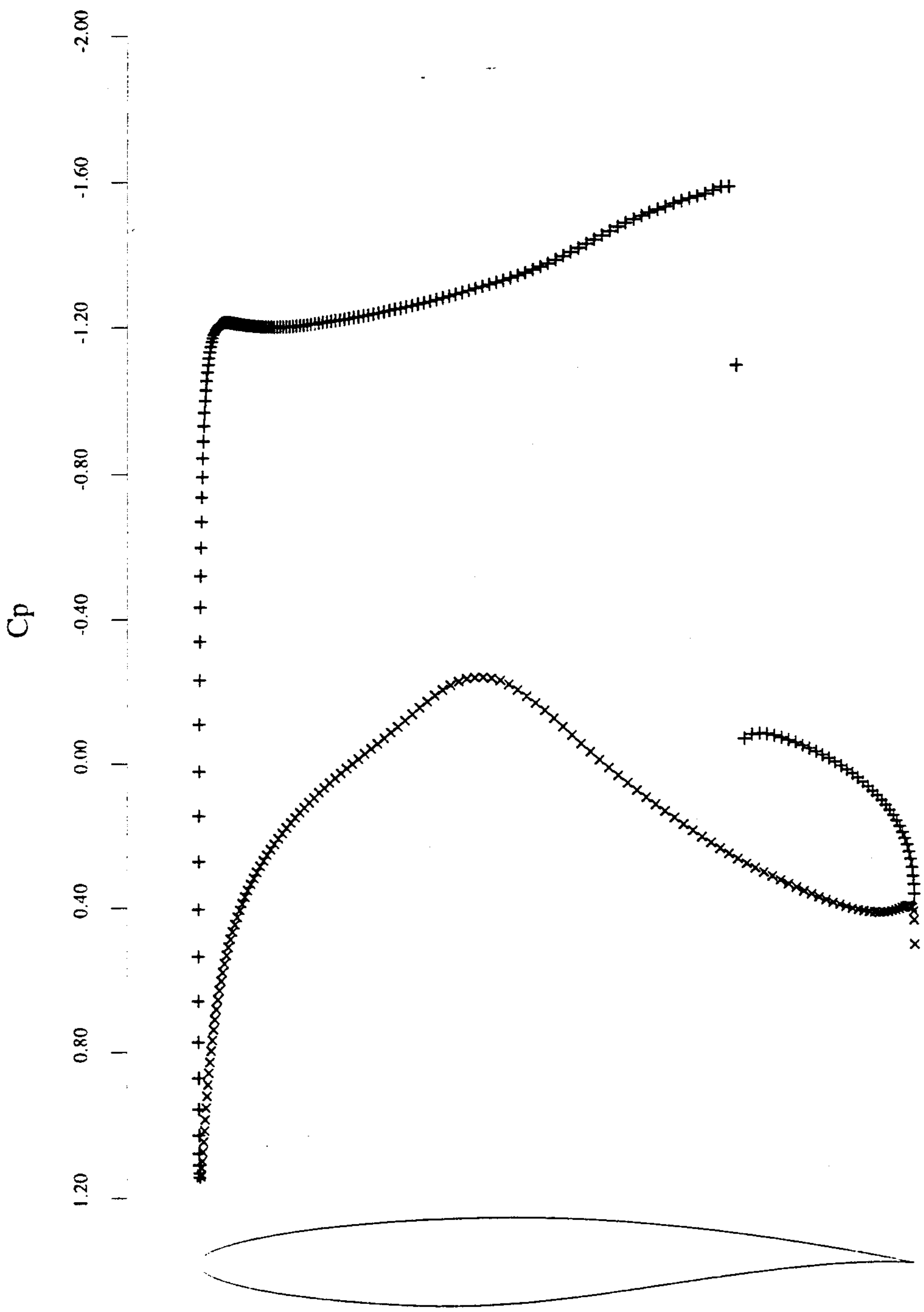
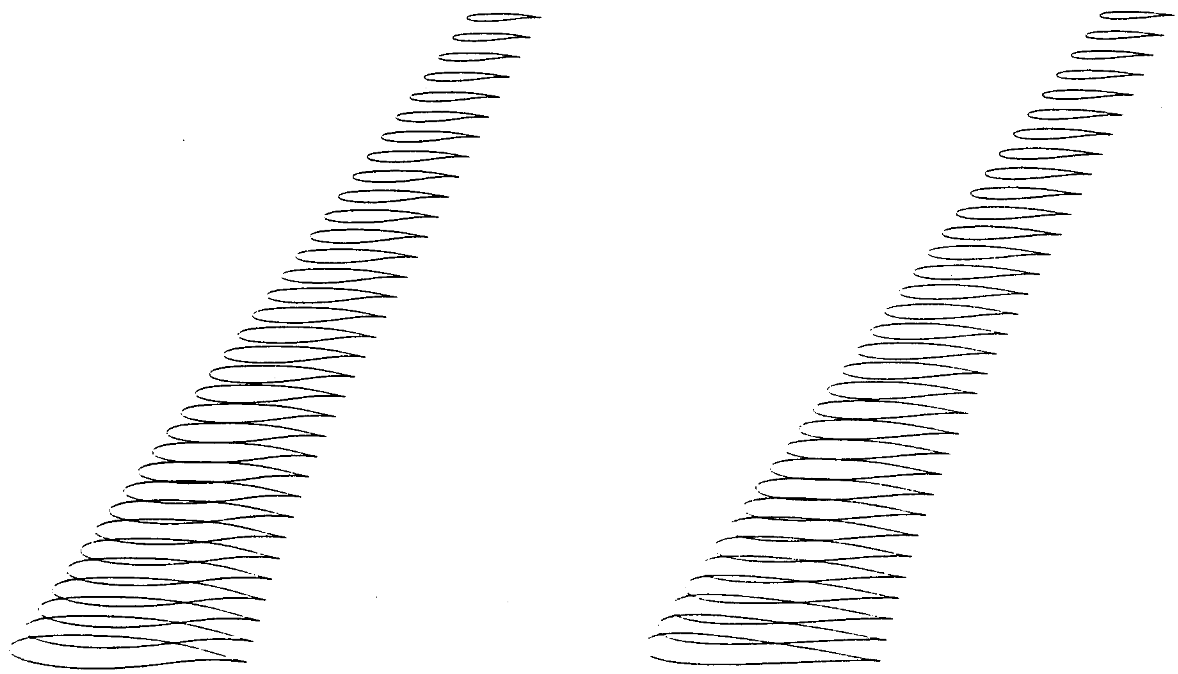


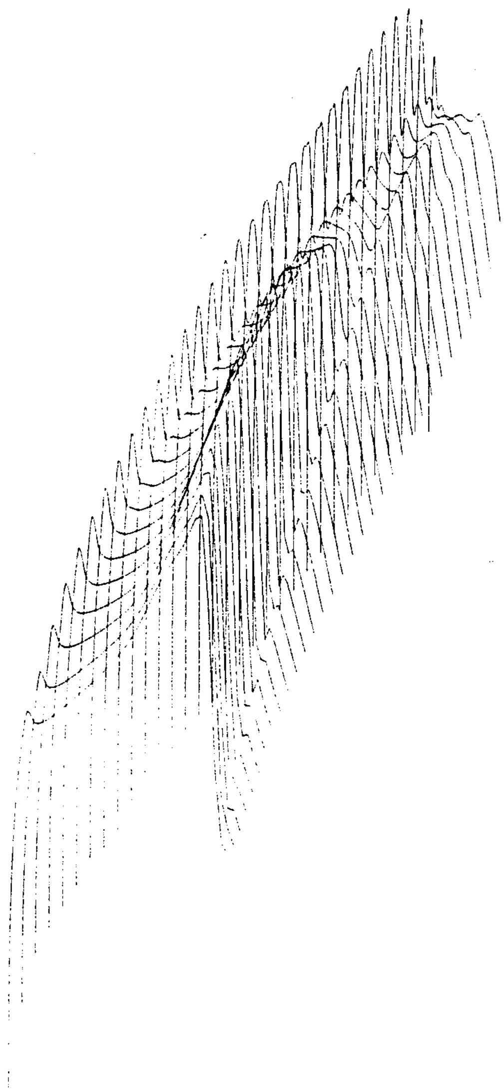
Figure 1: RAE 2822 airfoil
Solution by: SLIP flux limiter with CUSP flux splitting.
$C_l = 1.1313$, $C_D = 0.0469$, $\alpha = 3.000°$
Mach = 0.750, 321x65 grid, number of multigrid cycles = 25
Residual = $0.971 \mathrm{x} 10^{-4}$



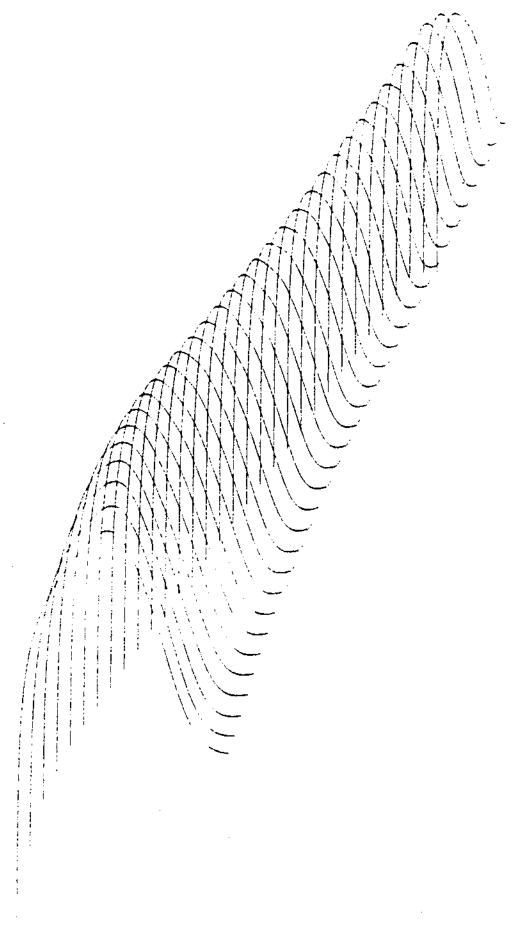
2a: Initial Wing
$C_l = 0.5001$, $C_d = 0.0185$, $\alpha = -0.958°$

2b: 8 Design Iterations
$C_l = 0.4929$, $C_d = 0.0118$, $\alpha = 0.172°$

Figure 2: Lifting Design Case, $M = 0.82$, Fixed Lift Mode.
Drag Reduction



UPPER SURFACE PRESSURE



LOWER SURFACE PRESSURE

Figure 3: Lifting Design Case, $M = 0.82$, Fixed Lift Mode.
Initial Wing: Modfied Korn.
$C_L = 0.5001$, $C_D = 0.0185$, $\alpha = -0.958°$
Drag Reduction

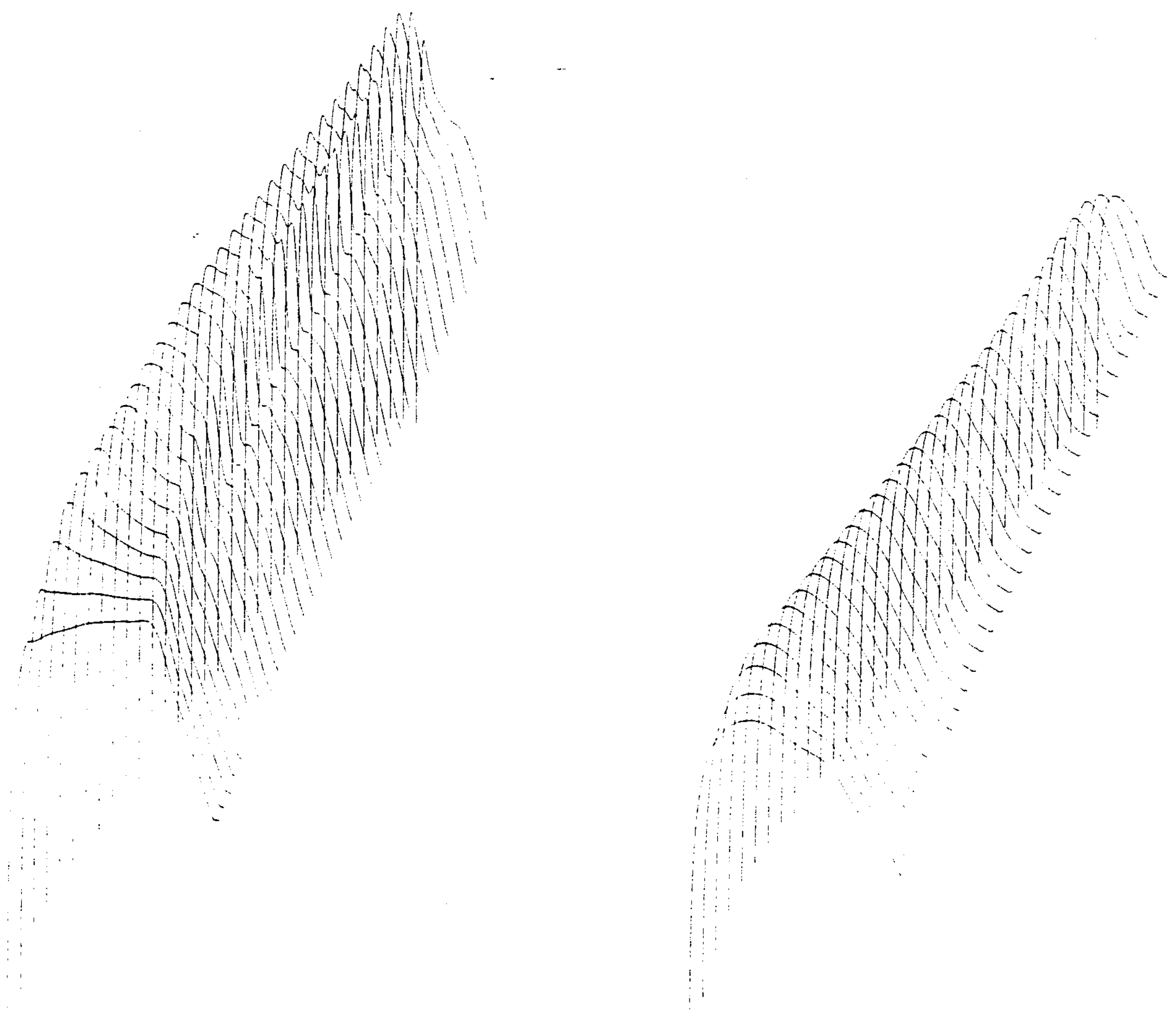
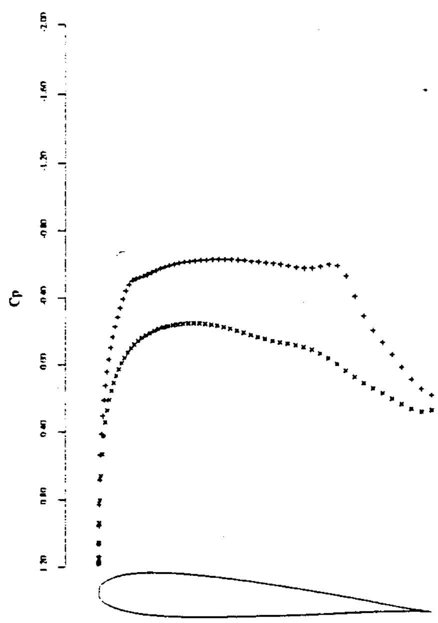
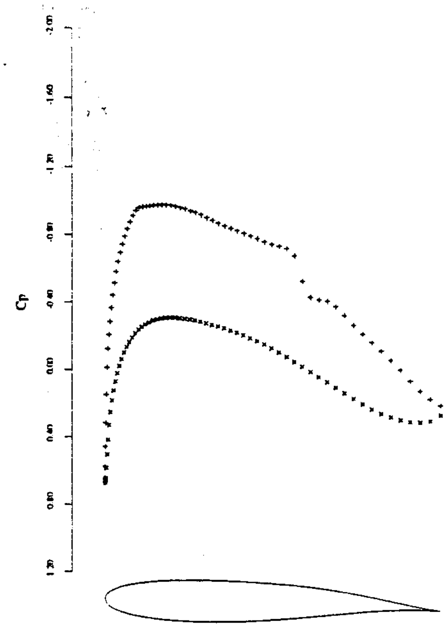


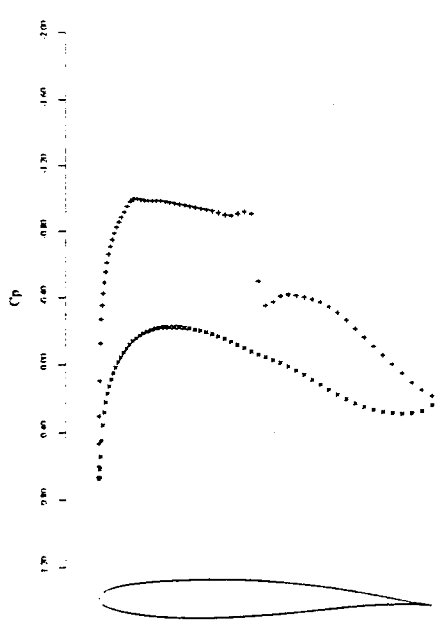
Figure 4: Lifting Design Case, $M = 0.82$, Fixed Lift Mode.
Design after 8 cycles
$C_L = 0.4929$, $C_D = 0.0118$, $\alpha = 0.172°$
Drag Reduction



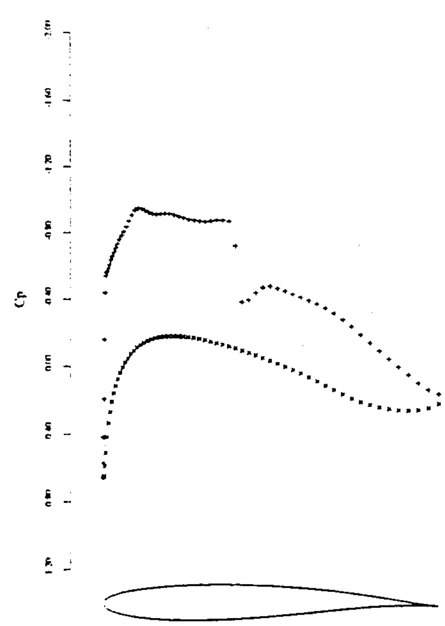
5a: span station $z = 0.00$



5b: span station $z = 0.25$



5c: span station $z = 0.50$



5d: span station $z = 0.75$

Figure 5: FLO67 check on redesigned wing.
$M = 0.82$, $C_L = 0.4975$, $C_D = 0.0107$, $\alpha = 0.200°$