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SOME ASPECTS OF HELICOPTER AIRFOIL DESIGN

BY

Alexander A. Nikolsky

CENTRAL AERODYNAMIC INSTITUTE (TsAGI)
ZHUKOWSKY, MOSCOW REGION, RUSSIA

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TsAGI HELICOPTER DEPARTMENT

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Abstract

Two methods of improving helicopter airfoil performance are presented in this paper. The first one combines the design by optimization method based on the complex method of nonlinear programming with a new approach to the choice of design variables, providing an effective tool to satisfy several aerodynamic and geometric constraints simultaneously. The second one is designing an airfoil with prescribed chord or arc length pressure distribution at a given Mach number by solving the nonlinear transonic inverse problem.

List of symbols

CL	lift coefficient
CDw	wave drag coefficient
Cm_0	pith moment at zero lift coefficient
Cp	pressure coefficient
D	drag
L	lift
M	Mach number
Mdd	drag divergency Mach number
r	polar coordinate in circle plane
s	distance along airfoil surface
t	maximum thickness of airfoil
z	complex variable in airfoil plane (=x+iy)
ξ	complex variable in circle plane (=r exp i0)
θ	polar coordinate in circle plane
ε	trailing edge angle divided by π

Introduction

To design an advanced lifting rotor blade it is important to have airfoils which can effectively operate in a wide range of flow conditions. Usually for the simplicity three specific regions are considered:

M=0.4; CL=CLmax - retreating blade,
M=0.6, (CL/CD)max - hover,
M=Mdd, CL=0 -advancing blade.

Also the curve $Cm_0(M)$ has to be considered at CL=0.

These performance provide an appropriate level of maximum lifting capability, L/D ratio and control system loads of lifting rotor. In general, some geometric conditions (such as maximum thickness and trailing edge angle limitations) also must be satisfied. Usually the nonlinear programming methods are used to optimize one of the aerodynamic characteristics with a set of aerodynamic and geometric constraints.

1. Design by optimization

This approach includes the following basic features: the choice of design variables to define and modify an airfoil contour and generation the objective function and constraints based on the design variables, the choice of flow field model and numerical method to solve a direct problem of an airfoil, the choice of nonlinear programming method to optimize the objective function.

1.1 Direct solver

Aerodynamic parameters are calculated by two-dimensional inviscid transonic analysis code BGKJ [1], which solves the full potential equations (FPE) of fluid flow about an airfoil. A coordinate system for the treating the flow past a profile is generated by mapping the exterior of the profile conformally onto the exterior of a unit circle. The real part of the mapping function derivative $|dz/d\xi|$ can be presented as the power series:

$$\ln(ds/d\theta) - (1-\varepsilon)\ \ln 2\sin\theta/2 = \sum_{n=0}^{n=\infty} An\ \cos n\theta + Bn\ \sin n\theta \qquad (1)$$

1.2 Design variables

A common practice is to define the airfoil coordinates by polinomial spline and different incremental algebraic shape functions. In this case the design variables are the coefficients and parameters of these functions, that is, in the form: of:

$$y = f\,(x;\ a_1, \ldots, a_n),$$

where the coefficients a_n are design variables.

. An alternative choice of design variables, based on equation (1) is suggested in this paper. If $s(\theta)$ is given, then the airfoil coordinates are easy to define:

$$x(\theta) = \int_0^\theta ds/d\theta\ \cos\psi\ d\theta$$

$$y(0) = \int_0^\theta ds/d\theta\ \sin\psi\ d\theta,$$

where

$$\psi + 1/2\ [\theta + \varepsilon\ (\theta - \pi)] = \sum_{n=0}^{n=\infty} An\ \sin n\theta - Bn\ \cos n\theta$$

The design variables are Fourie coefficients An and Bn. The coefficients An define the thickness of a profile and Bn define its meanline. Such a choice has an additional advantage because of two useful relations:

$$\varepsilon = 1 - A_1$$

$$C_{m0} = 4\pi B_2$$

These relations allow to satisfy the subsonic pitch moment condition and the trailing edge angle limitation easily.

1.3 Optimization technique

Two the most used types of optimization technique are known. The constrained function minimization program, CONMIN,[2,3] employs the gradient method of feasible directions to seek the minimum value of the objective function and simultaneously to satisfy a set of constraints. The present program is based on the nongradient complex method [4]. For comparison of these two methods and two types of the design variables, the test example of wave drag minimization of NACA-0012 profile was considered. Fig.1 shows pressure curves and wave drag values with $t < 12\%$ constraint, fig.2 shows the same values with extra $\varepsilon < \varepsilon_{NACA}$ condition. Similar results were presented in Ref.3, where the number of the design variables was equal to six.. In the present paper the number was equal to three and the cost of computations was about two times lower, thus demonstrating the suscessful choice of design variables.

1.4 Application

As has been mentioned earlier, the helicopter airfoils should operate in a wide range of flow conditions. The inviscid flow computations by BGKJ method have showed, when compared with the experimental data, that it reasonably predicts the increments of the Mdd value by CDw(M) curve, of the CLmax value by Cpmin(CL) curve for a similar behaving of Cp(x) curves in trailing edge regions, of the (CL/CD)max by CL(CD) curve and also $C_{m0}(M)$. The increments are added to the correspoding profile prototype values. In the most cases considered only some under/overprediction of absolute values but not signes of these increments was obtained. Nevertheless at the later stages of the airfoil design more accurate prediction codes are desirable.

At the earlier design stages for lower cost consider the following parameters:

Cpmin- suction peak at M=0.4, CL=CLmax

C_{dw6}- CDw at M=0.6, CL=0.7

C_{dw8}- CDw at M=0.8, CL=0

C_{m0} at M=0.4, CL=0.

The design task is to provide the following improvemets of the initial airfoiil parameters: to increase CLmax by 0.05 and to minimize CDw at M=0.8. The values of CDw at M=0.6 and Cm_0 should not change significantly.

Using the appropriate weighting coefficients βi, the objective can be written as:

to minimize G(X),

where $G(X)= \beta_1 (Cpmin- Cpmin_0)+ \beta_2 (CDw_6 -CDw_{60})+ \beta_3 CDw_8$,

X=(A2,...,A4; B2,...,B4) is the design variables vector, index 0 means design value.

It is an uncoditional optimization problem, because the pitch moment and trailing edge constraints are satisfied automatically (Eqs.2). The results are shown in Figs.3,4. In comparison with the initial profile, the optimized profile has a more thin and flat upper surface providing practically shockless flow on it. The lower surface thickness distribution is strongly changed in accordance with the pitching moment constraint. The wave drag is reduced to zero, and the maximum thickness is equal to the initial one. The design constraints are satisfied and computed Mdd value is by 0.01 greater than initial one.

2. Design by inverse problem

An alternative approach to design an airfoil is to solve inverse problem, that is, to define an airfoil contour for prescribed pressure distribution. Known is a method of reference 5, based on Dirichlet problem for full potential equation, but trailing edge closure constraints were not exactly assured. Here briefly described are the method, generalizing the method of Lighthill [6], based on Newmann problem for FPE and the way to accurately satisfy closure conditions.

2.1 Inverse solver

For incompressible fluid flow an exact solution of inverse problem [6] is known by means of conformal mapping of the exterior of the profile onto the exterior of a circle. The modulus of mapping function derivative is written as:

$$ds/d\theta=u_\theta(\theta)/u_s(s) \qquad (3)$$

The magnitude $u_s(s)$ of the velocity along the airfoil surface is prescribed. The magnitude of the circle plane velocity $u_\theta(0)$ is universal for any airfoil for incompressible case but obtained by iterations for FPE, so Eq.3 is solved by successive approximations:

i. Initial profile gives $ds/d\theta$

ii. BGKJ code predicts $u_\theta(\theta)$ for the given $ds/d\theta$

iii. Eq.5 gives a new $ds/d\theta$. Items ii-iii are repeated until convergency is reached.

2.2 Integral constraints

To satisfy trailing edge closure conditions and $|dz/d\xi|=1$ condition it is nesessary to meet three integral constraints:

$$\int_0^{2\pi} \ln(ds/d\theta)\, d\theta = 0$$
$$\int_0^{2\pi} \ln(ds/d\theta) \cos\theta\, d\theta = 0 \qquad (4)$$
$$\int_0^{2\pi} \ln(ds/d\theta) \sin\theta\, d\theta = 0$$

For this reason after item iii an additional item is included:

iiii. Modification of $ds/d\theta$: $ds/d\theta m = ds/d\theta \exp E$,

where a modification interval $[\theta_1,\theta_2]$ ordinary agrees with some fraction of the lower surface in order not to disturb upper surface pressure distribution and E is a finite Fourie series in this interval. The least squares procedure makes it possible to obtain the exact relations for Fourie coefficients, solving a problem. So, the full procedure is to repeat items i-iiii. It is of interest to notice the similarity of inverse problem and viscid-inviscid interaction iteration processes. The same is valid for the cost of computations.

2.3 Examples

Test examples are presented in Figs.5-7. In all cases initial pressure distributions are modified in the supersonic region as follows:

$$Cpmod = Cp^* + (Cpi - Cp^*)\, fct,$$

where Cpi- initial pressure coefficient,
Cp*- critical pressure coefficient,
Cpmod-prescribed pressure coefficient,
fct<1 -reduction factor.

The calculated curves differ from prescribed ones on the lower airfoil surface due to integral constraints (Eqs.4) imposed.

3.Conclusions

The application of nonstandard design variables provided a good statement of airfoil design procedure excluding two constraints from consideration and lowing cost of computations.

A possible way to employ the inverse problem for helicopter airfoil design may consist in proper parametrizing of the pressure distribution at one of design points and using these parameters as design variables. It is the subject of future applications.

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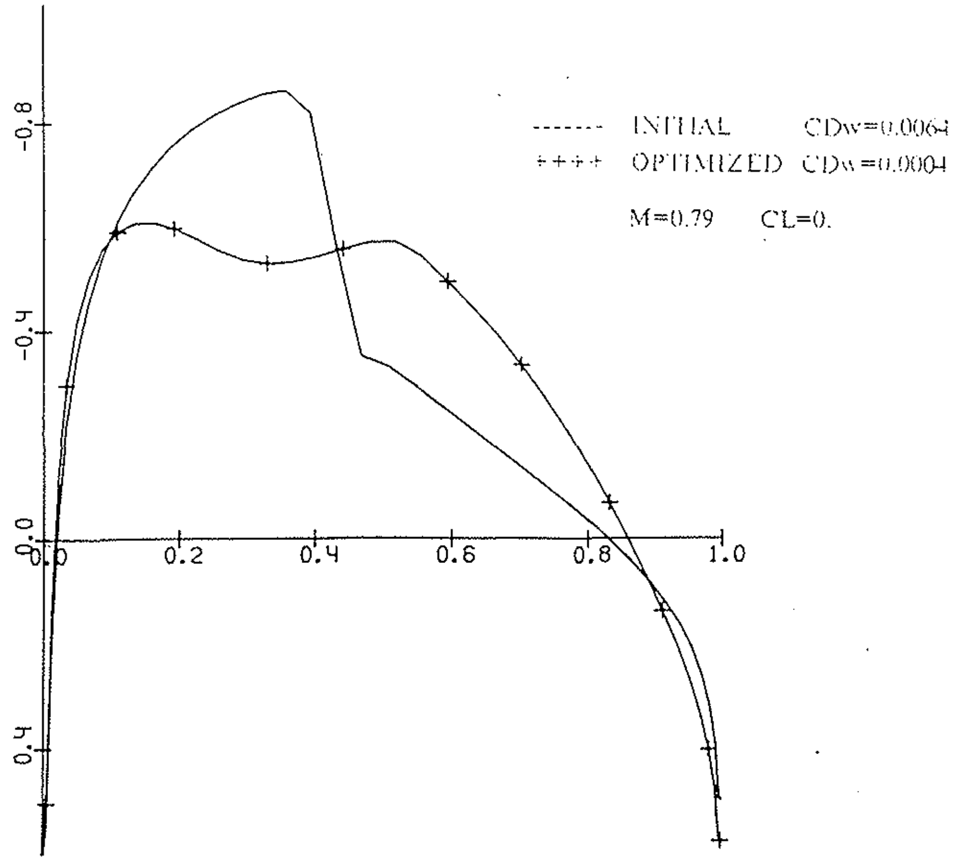


Fig.1. Cp distributions and wave drag values for NACA- 0012 airfoil and modified airfoil, t< 12% constraint.

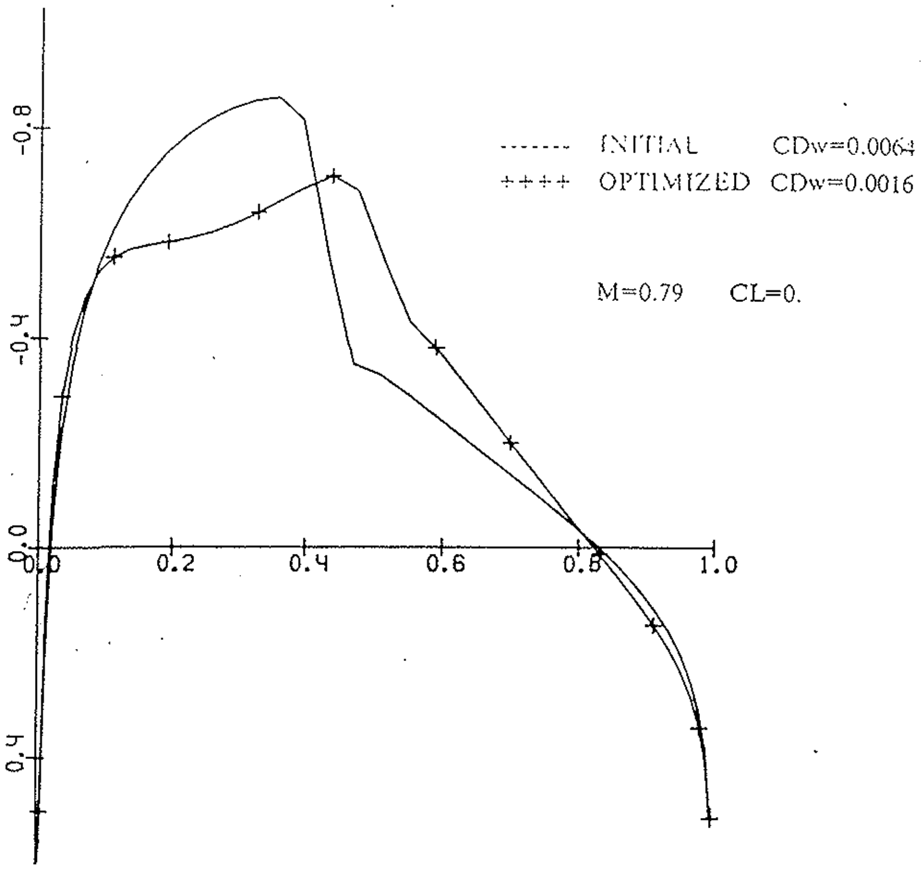
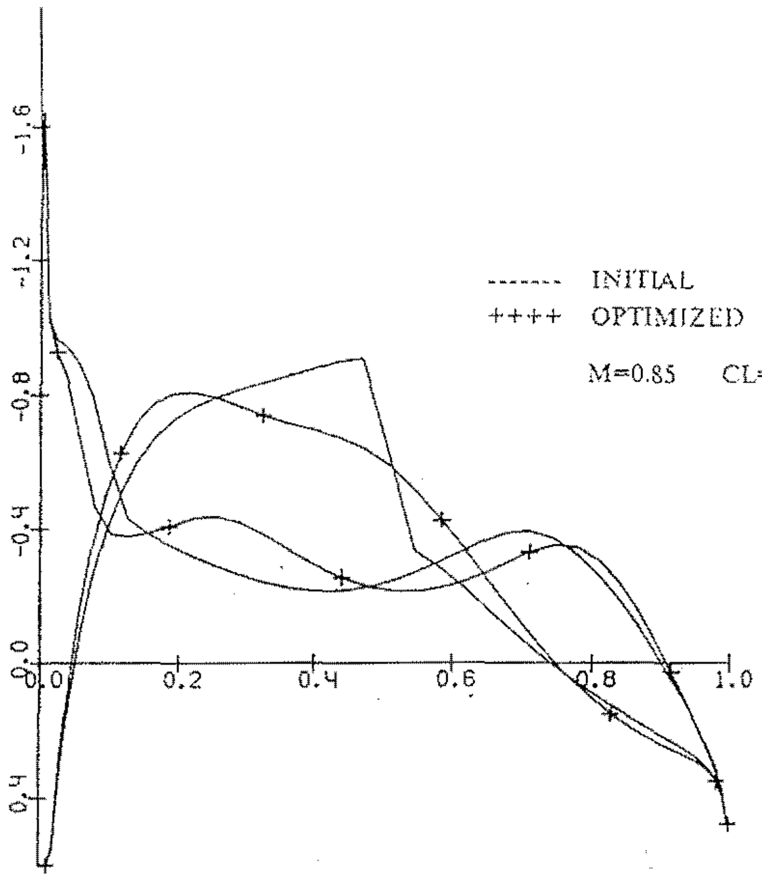


Fig.2. Cp distributions and wave drag values for NACA- 0012 airfoil and modified airfoil, t< 12% , $\varepsilon < \varepsilon_{NACA}$ constraints.



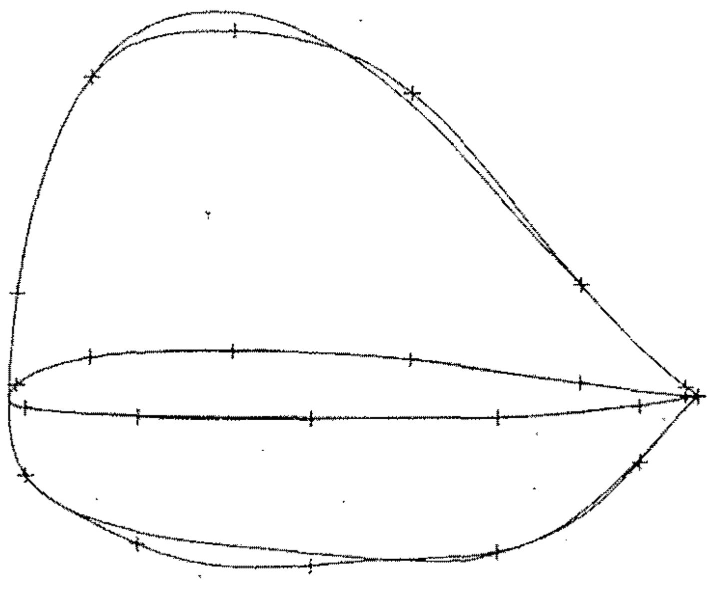


Fig.3. Initial TsAGI helicopter airfoil compared with designed airfoil.
M=0.8 design point.

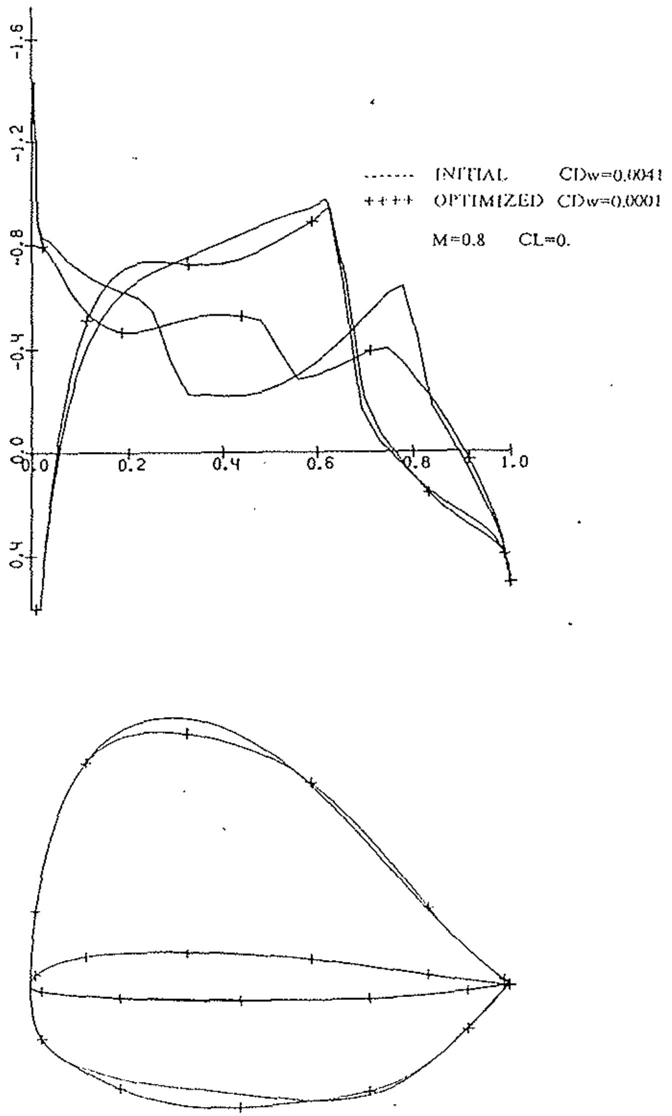


Fig.4.. Initial TsAGI helicopter airfoil compared with designed airfoil M=0.85 design point.

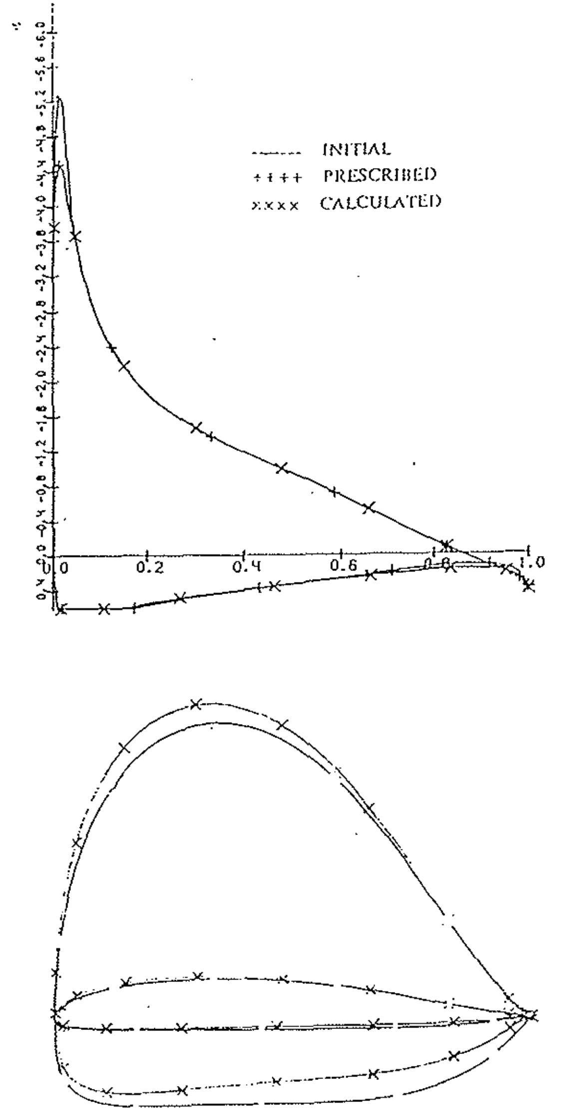


Fig.5. Design by inverse problem. Test example M=0.4.

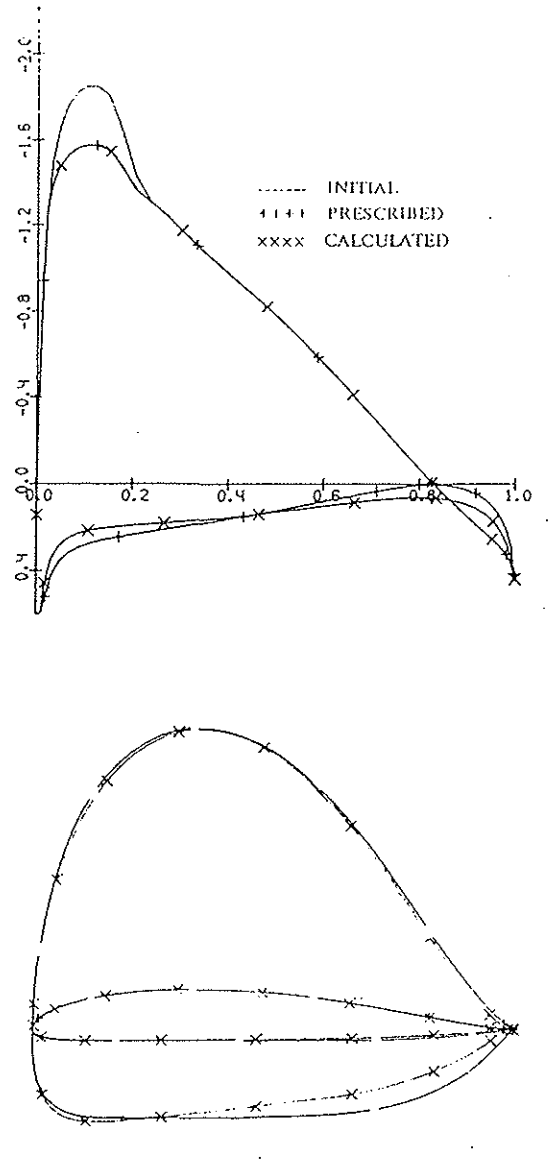


Fig.6. Design by inverse problem. Test example M=0.6.

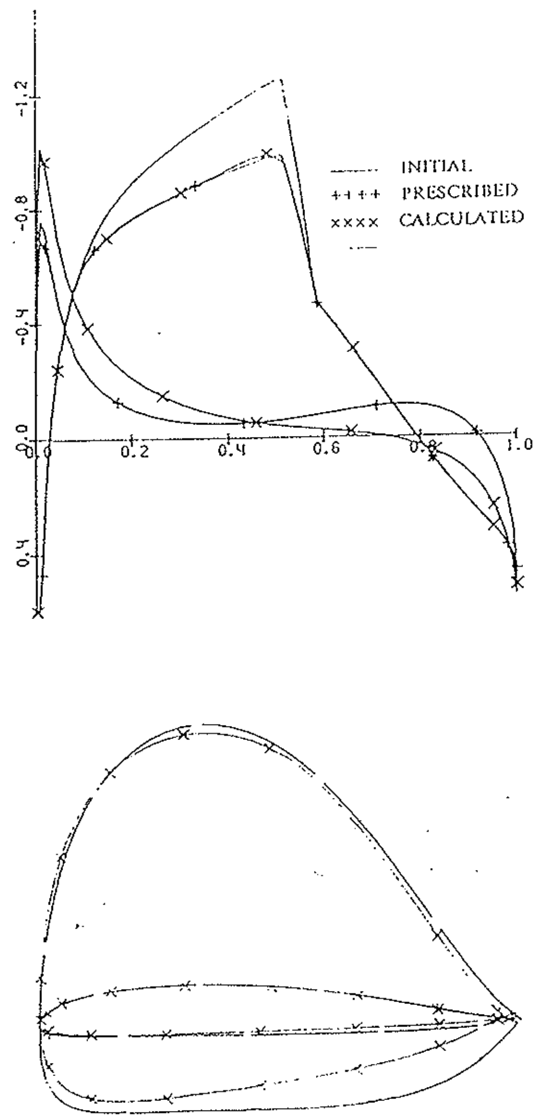


Fig.7. Design by inverse problem. Test example M=0.75.