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TRANSONIC WING DESIGN FOR TRANSPORT AIRCRAFT

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Abstract

The aerodynamic design of two transonic wings is described in the paper. The design method is an improved "iterative residual correction" method with closed form integral formulations to replace numerical integrations, using a new procedure including a weighted smoothing approach. Design criteria, viscous correction method, control of spanwise lift and thickness distributions, and specification of target pressure distributions for transonic wing design are also discussed. Computation results of the supercritical wing designed in this paper show that the optimum cruise speed is improved by 0.05 Mach number while the aerodynamic efficiency increased by 14.5 percent at Cl=0.5, compared with the Boeing 737-300 wing. The designed NPU-NLF1 wing has straight isobars from 0.1 to 0.9 half-span, and reasonable lift and thickness distributions in the span direction. Wind tunnel experiments of the NLF1 wing is under preparation.

1. Introduction

Close attention and great effort are paid in the current aerospace world to the development of new techniques of design aerodynamics, especially for the next generation of transport aircraft [1], [2], [3], [4]. For such purposes, CFD is now playing the most important role, besides wind tunnel experiments and flight tests. It could even reduce the number of test configurations and iterations for a new airplane by approximately 80% [5]. However, compared with the status of direct analysis methods of CFD for given geometries, the progress of inverse design codes is relatively slow.

Based on the extensive use and reviewing of some of the available design methods for transonic airfoils and wings, the following criteria for a practical CFD design method could be put forward [6]:

* The governing equations and numerical approach should be correct , accurate and efficient.

* The input, output, computation grid, boundary conditions and the computer code should be easy to handle, and should be of real-time plotting ability to show the design process and results.

* The geometry of the wing section (eg. trailing edge thickness) should be controlled and the new contour should have smooth curvature distributions.

* The code should be able to design complete airfoils or wings, besides the common ability to modify parts of the contours.

* Viscous effects should be included in the design process.

* The code should also work well for the design case with relatively sharp shocks.

* Empirical coefficients for operating the code should be avoided or limited.

* The design method should be reliable, the result should be satisfactory if checked by other reliable analysis codes or experiments.

* For well-posed target pressure distributions, reasonable design results should usually be obtained.

Few of the existing design methods could fully satisfy the above, then further modifications to the methods or to the designed wing sections have to be carried out [7], [8] ,[9]. This may certainly hamper fast advancement of the research on design aerodynamics. So it becomes necessary that a more practical and efficient transonic design method be developed.

The state of the art of 3-D design methods seems to be the work of Takanashi, first introduced in middle 1980's [10]. The inverse part of this method is independent of the direct analysis code coupled into the design procedure, so the direct part could easily be changed with other advanced methods, as in the case of the world first design method using Navier-Stokes equations [11]. But for engineering applications, still further modifications should be made to this method.

In this paper, closed form integrations are derived to replace the numerical integrals used in the method of Takanashi, a weighted smoothing approach has been introduced, other improvements are supplemented so as to develop an effective design tool ready for practical applications.

The design techniques for supercritical and natural laminar flow (NLF) wings and airfoils have been the current goals of design aerodynamics. It has been estimated that the aerodynamic efficiency (ML/D) of a jet transport could be improved by more than 30% if NLF surface is used [2]. Great economic benefit is also recognized with the application of supercritical wings, as the forth generation of jet airliners, say Boeing 767, Airbus A310 and so on, came into service in the last decade. It becomes then very important how to make effectively use of modern CFD methods to design perfect transonic configurations.

Detailed descriptions concerning design criteria, prescriptions of target pressure distributions, control of lift and thickness distributions along the span, and some analysis results are given in this paper for two transonic wings with supercritical and NLF pressure distributions designed by using the present method. It is also shown that the natural laminar flow could be maintained for extended off-design flight conditions by means of simple variable camber method.

2. Modified Inverse Method and Design Procedure

The inverse method is based on the 3-D full potential equation written in terms of the pertubation velocity potential $\phi(x,y,z)$:

$$(1-M_\infty^2)\bar{\phi}_{\bar{x}\bar{x}}+\bar{\phi}_{\bar{y}\bar{y}}+\bar{\phi}_{\bar{z}\bar{z}}=K\frac{\partial}{\partial\bar{x}}\left(\frac{1}{2}\bar{\phi}_{\bar{x}}^2\right)+\bar{H} \qquad (1)$$

$$\bar{\phi}_{\bar{z}}(\bar{x},\bar{y},\pm 0)=\bar{f}'_{\pm}(\bar{x},\bar{y})+\bar{Q} \qquad (2)$$

$$C_{p\pm}(\bar{x},\bar{y})=-2\bar{\phi}_{\bar{x}}(\bar{x},\bar{y},\pm 0)+\bar{S} \qquad (3)$$

Where $k=(\gamma+1)M_\infty^2$, $\bar{f}(x,y)_\pm$ are the functions of the upper and lower wing surfaces, $\bar{H}$, $\bar{Q}$ and $\bar{S}$ are higher order terms in the transonic small disturbance series expansion.

Transform the above equations into the design coordinate system x,y,z, introduce a new variable $\Delta\phi$, which is considered to be due to the "residual" between the current Cp distribution of the baseline wing sections and the specified target distribution. Using Green's theorem to convert the partial differential equation into an integro-differential equation, introducing an approximate decay function of ϕ in z direction, and then performing some more derivations, the integral formulation of the equation about $\Delta\phi$ could be obtained. Discretizing the equation by making the integrations in each of the small panels dividing the wing surface (Fig. 1), the final discrete formulas could be [10]:

$$\Delta u_s(x_i^j,y_j)=\sum_{l=1}^{I+1}\sum_{m=0}^{J}\mu_{ijlm}^s\,\Delta w_s(x_{l-\frac{1}{2}}^m,y_m)+\chi_s(x_i^j,y_j)+\sum_{l=1}^{I}\sum_{m=0}^{J}[\upsilon_{ijlm}^s\,\chi(x_l^m,y_m,+0)+\hat{\upsilon}_{ijlm}^s\,\chi(x_l^m,y_m,-0)] \qquad (4)$$

$$\Delta w_a(x_i^j,y_j)=\sum_{l=1}^{I}\sum_{m=0}^{J}\mu_{ijlm}^a\,\Delta u_a(x_l^m,y_m)+\sum_{l=1}^{I}\sum_{m=0}^{J}[\upsilon_{ijlm}^a\,\chi(x_l^m,y_m,+0)-\hat{\upsilon}_{ijlm}^a\,\chi(x_l^m,y_m,-0)] \qquad (5)$$

Where μ_{ijlm}^s, υ_{ijlm}^s, $\hat{\upsilon}_{ijlm}^s$, μ_{ijlm}^a, υ_{ijlm}^a and $\hat{\upsilon}_{ijlm}^a$ are the integral representations over each small panel (l,m). These could be further derived and each contains a number of sub-integrations. For example:

$$\mu_{ijlm}^s=-\frac{1}{2\pi}(1-\delta_{l,I+1})\{I_{311}+(1-\delta_{m,0})I_{312}-I_{313}-(1-\delta_{m,0})I_{314}\}-\frac{1}{2\pi}(1-\delta_{l,1})\{I_{315}+(1-\delta_{m,0})I_{316}\} \qquad (6)$$

Where the sub-integrations are:

$$I_{311}=\int_{x_{l-\frac{1}{2}}^m}^{x_{l+\frac{1}{2}}^m}\int_{y_{m-\frac{1}{2}}}^{y_{m+\frac{1}{2}}}\psi_x(x_i^j,y_j,0;\xi,\eta,0)\,d\xi\,d\eta \qquad (7)$$

$$I_{312}=\int_{x_{l-\frac{1}{2}}^m}^{x_{l+\frac{1}{2}}^m}\int_{y_{m-\frac{1}{2}}}^{y_{m+\frac{1}{2}}}\psi_x(x_i^j,y_j,0;\xi,-\eta,0)\,d\xi\,d\eta \qquad (8)$$

and so on.

In this paper, still further derivations lead to the closed forms of these integral representations. For example:

$$I_{311}=\ln\Big(\frac{(y_j-y_{m+\frac{1}{2}})-\sqrt{(x_i^j-x_{l+\frac{1}{2}}^m)^2+(y_j-y_{m+\frac{1}{2}})^2}}{(y_j-y_{m-\frac{1}{2}})-\sqrt{(x_i^j-x_{l+\frac{1}{2}}^m)^2+(y_j-y_{m-\frac{1}{2}})^2}}*\frac{(y_j-y_{m-\frac{1}{2}})-\sqrt{(x_i^j-x_{l-\frac{1}{2}}^m)^2+(y_j-y_{m-\frac{1}{2}})^2}}{(y_j-y_{m+\frac{1}{2}})-\sqrt{(x_i^j-x_{l-\frac{1}{2}}^m)^2+(y_j-y_{m+\frac{1}{2}})^2}}\Big] \qquad (9)$$

in case of m = j. The closed form integral representations have completely removed the numerical integrals used in [10], then the accuracy and stability of the design iterations are improved.

After w_s and w_a are calculated from eqs. (4) and (5), new wing sections could be obtained by adding the correction terms $\Delta f_\pm$ to the baseline sections, where

$$\Delta f_\pm(x,y)=\int_{x_{LE}}^{x}\Delta f'_\pm(x,y)dx=\frac{1}{2}\int_{x_{LE}}^{x}[\Delta w_a(x,y)\pm\Delta w_s(x,y)]\,dx \qquad (10)$$

The iterative design procedure is shown in Fig. 2. In the *present method, some more modifications have been made.* For analysis code with viscous calculations like BGKJ, correction terms are added directly to the wing sections and viscous effect is naturally contained in each of the analysis step. When the inviscid code is coupled with the inverse program, boundary layer displacement thickness is added and subtracted for each two or three design iterations.

After the inverse step, the correction terms are smoothed to keep the new contour smooth in curvature. For viscous design cases with relatively sharp shocks, a special weighted function is included in the smoothing formula to reach a good convergence near the shocks [7]. Fig. 3 shows the effect of weighted smoothing in the case of the re-design of DLR-W1 airfoil with this modified method.

Two inverse codes have been programmed at NPU from the present method for 2-D transonic airfoil design and 3-D wing design, respectively. The 2-D code is coupled with DLR-BGKJm, and the 3-D with DLR-FLO22m. Then a basic transonic design software is constructed which is based on the same design principle and easy to be advanced by simply introducing modern analysis codes into the basic software. The reliability of the 2-D method has been checked by the high speed wind tunnel test of a NPU-NLF airfoil, and more design examples for transonic airfoils and wings have shown that the present method is much practical for engineering applications. [6], [12], [13].

3. Design of NPU-GXX Supercritical Wing

Design Considerations

A common approach to quickly provide commercial aircraft to the airlines is to produce modified versions of existing airplanes, such as Boeing 737-300, -400, 747-400, and so on. But due to the difficulties and expense to modify the whole wing, the economic benefits of those new versions are limited. So it is necessary for the design of a wing that advanced techniques be applied and the potential for further modifications be taken into account.

As a test case, a supercritical wing with the same planform of Boeing 737-300 wing has been designed. The design conditions are taken as: $M = 0.79$, $Cl = 0.55$, and the relative thickness of the outer wing should be increased to about 12.5% for larger fuel tank and reduced structure weight.

Design of the Basic Airfoil

The corresponding design conditions for the basic airfoil could be: $M = 0.72$ to 0.73, $Cl = 0.65$, $T/C = 13.2\%$ and the design Reynolds number is taken to be 10 million.

DLR-R4 airfoil is selected as the initial airfoil and a slightly different target Cp distribution is specified for weaker shock and reduced value of pitching moment. 6 design iterations by the 2-D method of this paper lead to a satisfactory result(Fig. 4). The aerodynamic analysis of this new airfoil NPU8 shows that the design requirement has been fulfilled.

Wing Design

The general requirements for transport wing design are to get straight isobars along percentage chord lines and elliptical lift distribution in span direction. The present design method has provided greater possibility to realize these requirements.

The wing planform is shown in Fig. 5. Six wing sections along the span at $y=0.0$, $y=0.205$, $y=0.297$, $y=0.6$, $y=0.8$ and $y=1.0$ are chosen to be the design sections. An initial wing is first constructed by putting converted NPU8 airfoil to these design sections.

It has been demonstrated that viscous effect must be taken into account for transonic wing design [14]. Because FLO-22 is a inviscid code, adding boundary layer displacement thickness is used for the viscous correction.

Computation is made for the initial wing and the Cp distribution of the middle section at $y=0.5$ is found to be the best coincidence with the design Cp distribution of the basic airfoil. This Cp distribution is then taken to be the target pressure distributions for wing sections from $y=0.205$ to $y=0.8$. The target distributions of the lower wing surface are modified to control the lift and thickness distributions in span direction. Similar target distributions are also given for root and tip sections at $y=0.0$ and $y=1.0$.

With present method, four design iterations lead to a good convergence to the target Cp distributions from the initial wing at all design sections. Fig. 6 shows that straight isobars could have been obtained from span section of $y=0.2$ up to $y=0.9$, so the feature of the basic airfoil is well retained by this new wing. Reasonable thickness distribution in the span direction is shown in Fig. 7. This wing is named NPU-GXX.

Aerodynamic analysis of this wing is made with DLR-FLO22m code for different flight conditions at different altitude with viscous corrections of stripe method. Same computations are also made for Boeing 737-300 wing. Fig. 8 gives the comparison of aerodynamic efficiency of both wings at $Cl = 0.5$. It could be seen that the maximum value of NPU-GXX wing is about 14.5% higher than 737-300 wing and the optimum cruise Mach number at $Cl=0.5$ is also increased from 0.74 of 737-300 wing to 0.79, the design Mach number of NPU-GXX wing.

4. Design of NPU-NLF1 Wing

Design Considerations

Natural laminar flow over wings has been clearly indicated by high Reynolds number flight tests in recent years. It is even estimated that laminar flow techniques will be applied to most transport aircraft by the year of 2000 [15]. But at the present time, reliable 3-D boundary layer instability analysis methods and suitable criteria for NLF wing design are still under investigation. Then a straight isobar criterion is put forward in this paper for transonic NLF wing design due to the following features:

* It could, in a large span area, retain 2-D Cp distributions of the basic airfoil favourable to reduce Tollmien-Schlichting instability.

* Avoiding additional cross flows over the wing surface caused by pressure gradients in the span direction.

* The relative thickness of the outer wing may increase in the span direction which could reduce the induced sweep angle of the wing [1].

Wing Design

Design condition of the wing is $M=0.75$, $Cl=0.50$. The aspect ratio of the wing is 11 and leading edge sweep angle is 19 degrees.

NPU-NLF-No.2 airfoil is chosen to be the basic airfoil which was designed at $M=0.72$, $Cl=0.5$ and has been tested in a high speed wind tunnel of CARDC [16]. Laminar flow up to 72% chord was identified over upper surface at design condition and test Reynolds number of 2 million.

Same design procedure is used as in the case of NPU-GXX wing. Satisfactory convergence to the target Cp distributions is obtained after 4 design iterations from the initial wing by using the present method (Fig. 9). Straight isobars are realized from $y=0.1$ span section to $y=0.9$ section for the upper wing surface. Fig. 10 shows the lift and thickness distributions in the span direction.

High and low speed wind tunnel experiments of this wing are under preparation.

Investigation of variable camber effect

Variable camber technique has been found to be of special advantages for transport aircraft and will be first used on Airbus A330 and A340. For NLF wings, the change of flight conditions could directly influence the extention of laminar flows over the wings. It is then suggested that variable camber technique may have greater effect in maintaining NLF at flight conditions other than design condition, and elementary investigation is made in this paper numerically with NLF1 wing by simply flapping the last 20% chord of the wing up and down.

Fig. 11 shows one of the examples when flight Mach number is lower than design value but with the weight and altitude of the aircraft unchanged. It can be seen that when the trailing edge is flapped down by 1 degree, the NLF pressure distribution is regained and the 2-D analysis of corresponding wing sections shows that the drag is reduced to the NLF level(Fig. 12).

Investigation also shows that at flight Mach number higher than design value, slightly flapping upwards could reduce the strength of the shock and pitching moment. When the wing is flying at design Mach number but at lower lift coefficients, slightly reduce the rear camber could also be favourable in maintaining NLF over lower surface of the wing and then increase the aerodynamic efficiency.

5. Conclusion

Design techniques of supercritical and NLF wings for transport aircraft have been one of the most important topics in design aerodynamics. The transonic design method of this paper has been proven to be an effective and practical design tool. The design procedure of transonic wings described in this paper provides a reasonable application of modern CFD methods and the design results of NPU-GXX wing and NPU-NLF1 wing have been much satisfactory.

6. Acknowledgement

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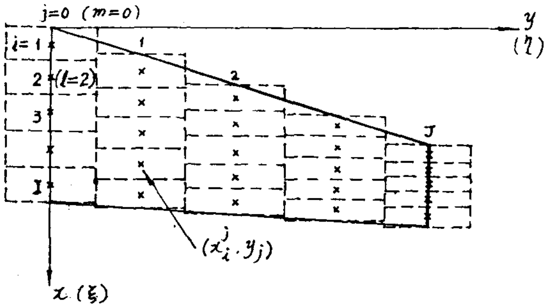

Fig. 1. Small panels dividing the Wing surface

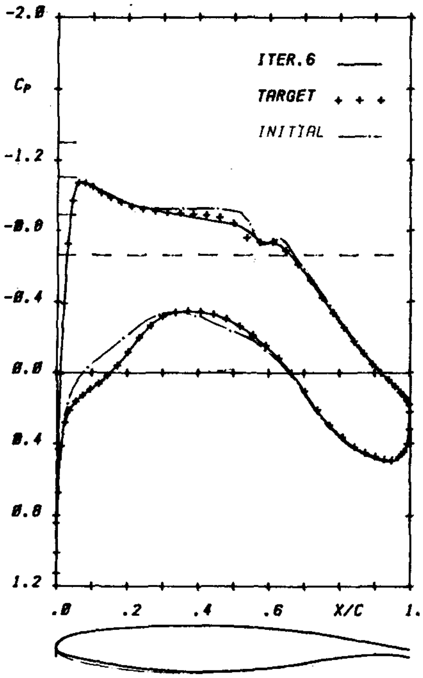

Fig. 4. Design result of NPU8 airfoil

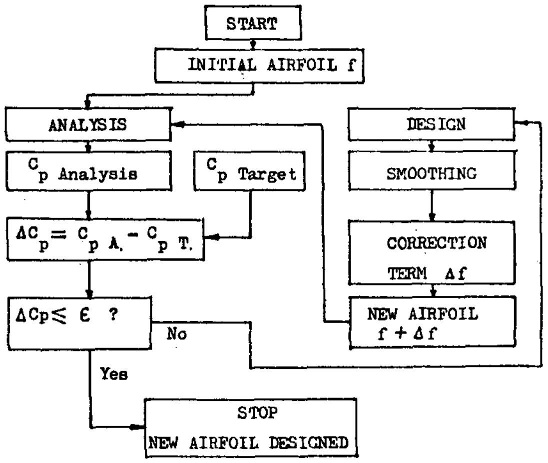

Fig. 2. Flow-chart of the design procedure

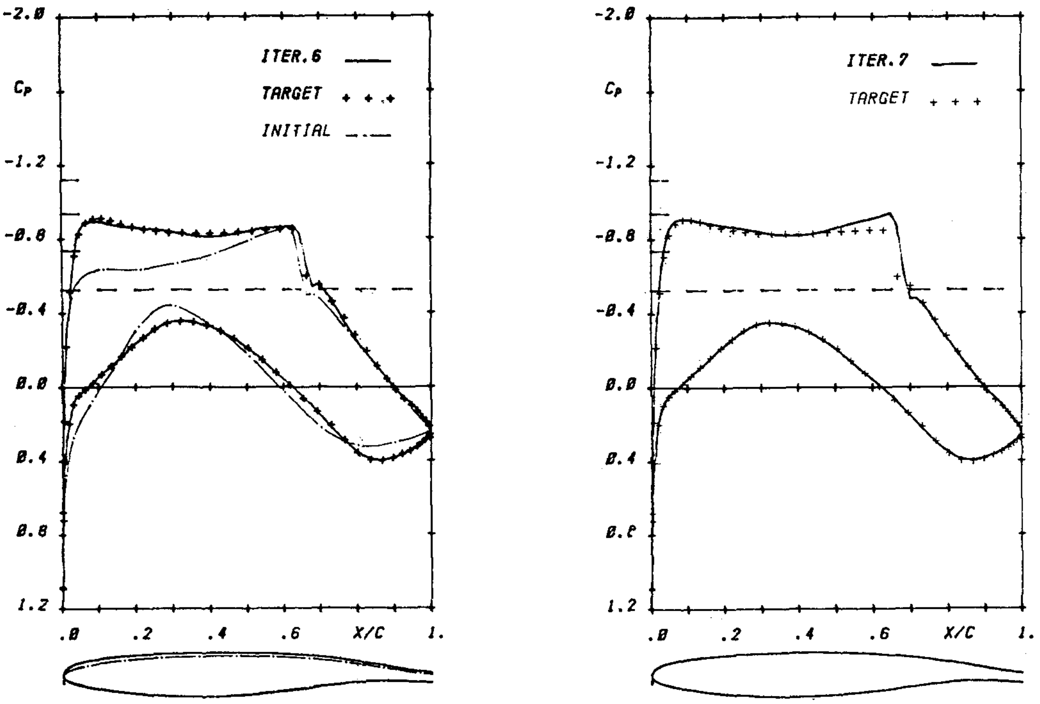

Fig. 3. Re-design of DLR-W1 airfoil with and without weighted smoothing

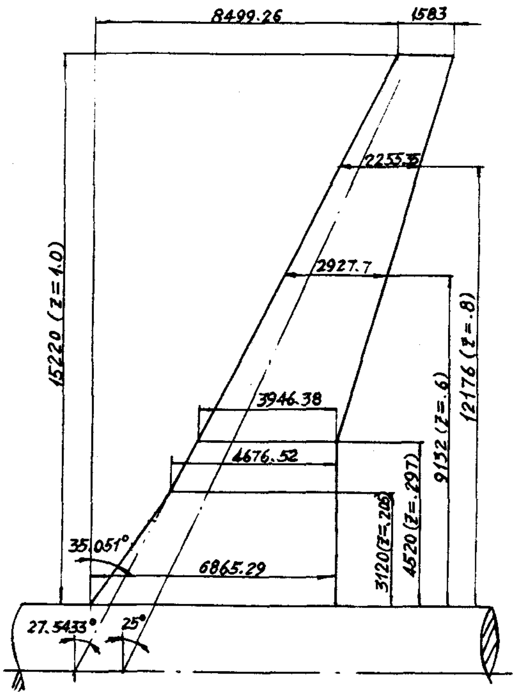

Fig. 5. Planform and design sections of NPU-GXX wing

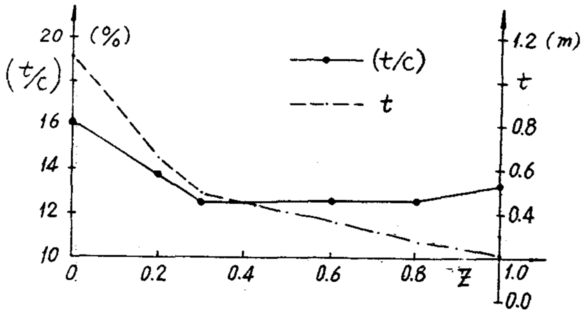

Fig. 7. Thickness distributions of NPU-GXX wing in span direction

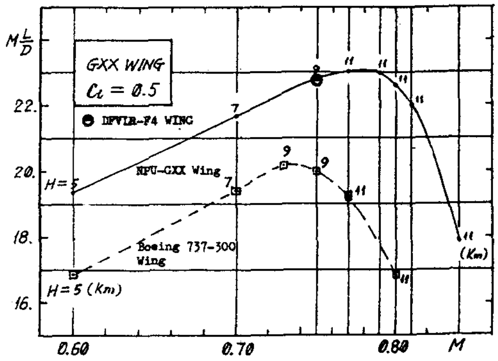

Fig. 8. Comparison of aerodynamic efficiency of NPU-GXX wing with Boeing 737-300 wing

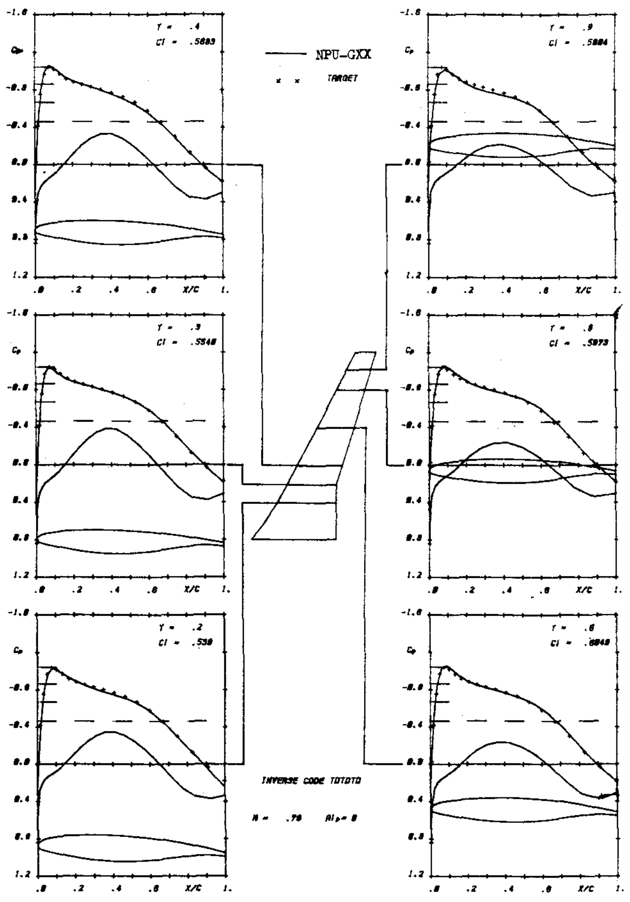

Fig. 6. Comparison of the design result of NPU-GXX wing with a same upper surface target pressure distribution

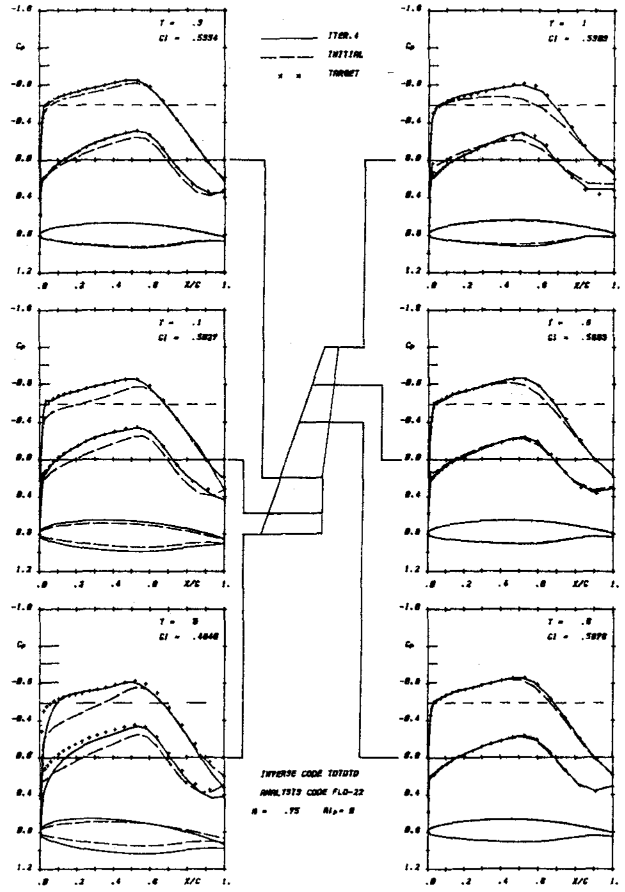

Fig. 9. Design result of NPU-NLF1 wing

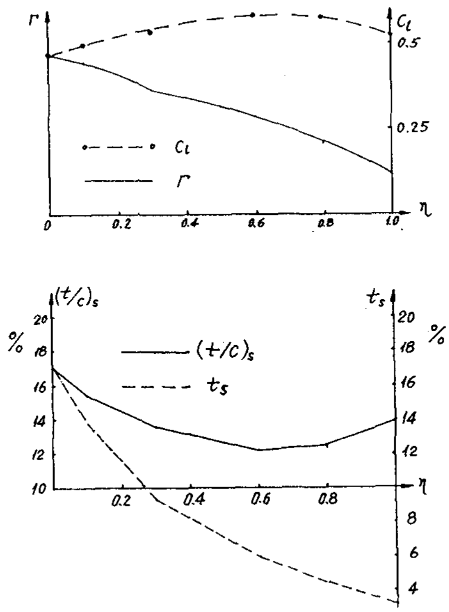

Fig. 10. Lift and thickness distributions of NPU-NLF1 wing in span Direction

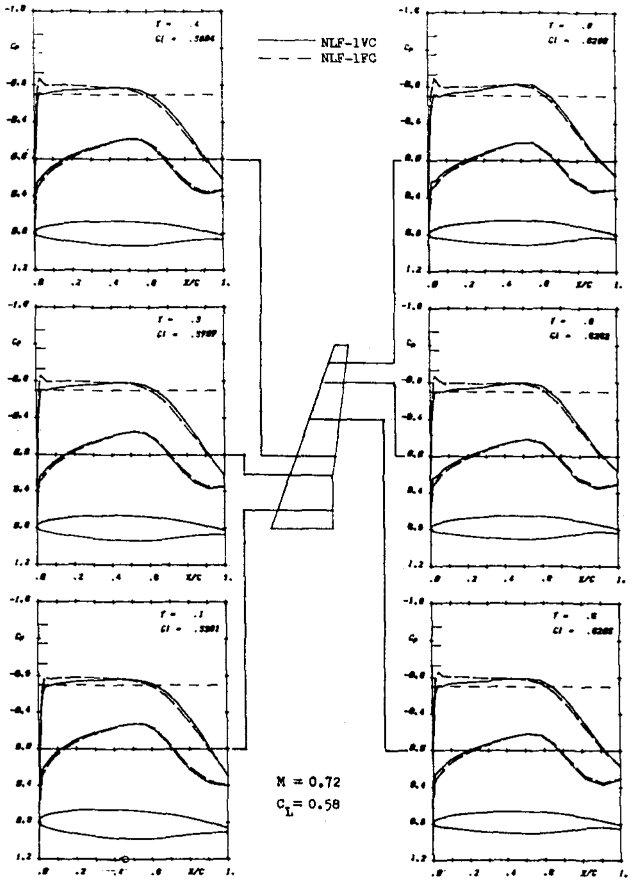

Fig. 11. Effect of variable camber on pressure distributions of NLF1 wing

AIRFOIL	M	Cl	Alp	Cd	Cdw	Re*E-6
NLF-1VC ———	.68	.6691	.3627	.0053	-.0001	6
NLF-1FC – – –	.68	.6669	1.0727	.0064	-.0001	6

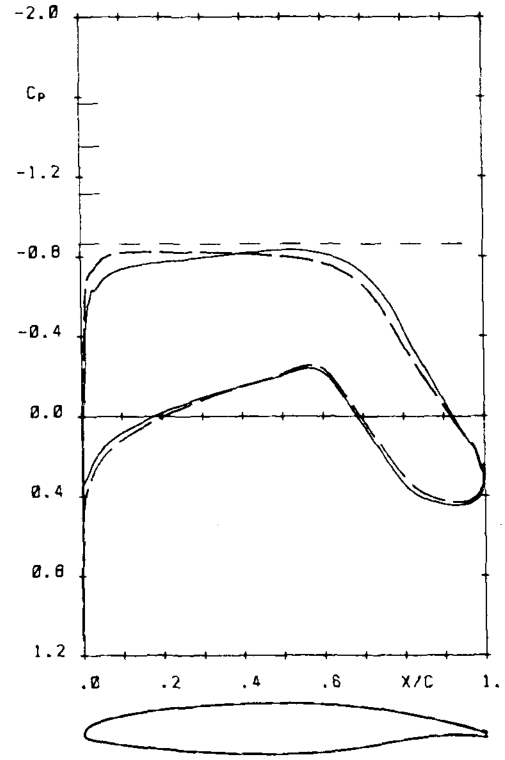

Fig. 12. Effect of variable camber on wing section characteristics