# Aerodynamic Design of Natural Laminar Flow Supersonic Aircraft Wings

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**Abstract**

Aerodynamic design of Natural Laminar Flow (NLF) for supersonic aircraft wings was conducted by using 3-D CFD analysis and boundary layer stability analysis. There are two main subjects for NLF wing design. One is that the wing of the sub-scaled unmanned supersonic experimental aircraft of National Aerospace Laboratory[1] (NAL) was designed to improve the L/D ratio. This wing sections were designed especially to reduce skin friction drag which amounts to nearly half of the total drag. For this purpose, we designed the wing sections to remain laminar flow as far as possible using 3-D CFD solver and 3-D laminar boundary layer stability analysis code and 2-D inverse design method. The other is that the simple sweep-back wing was designed to delay a transition due to cross flow instability in consideration of spanwise pressure distribution. (This design concept of NLF is patented.)

## 1. Introduction

A project led by NAL in Science and Technology Agency is underway to fly sub-scaled, unmanned supersonic aircraft(Fig.1). Kawasaki Heavy Industries participates in the program by designing aircraft with CFD analysis which is important in this project for aerodynamic design and evaluation. For conventional aircraft, the flight efficiency is dependent on L/D (lift to drag) ratio, so the drag should be reduced to improve the efficiency. For this purpose the following three technologies are usually applied:

1) reduction of wave drag due to volume (supersonic area rule, etc.)
2) reduction of induced drag (wing planform and warp-design, etc.)
3) reduction of friction drag (natural laminar flow, laminar flow control, riblets, etc.)

The first two technologies are mainly applied to the sub-scaled supersonic aircraft. Since the aircraft is nearly 12 meters long (about 10% scale of real future Supersonic Transport), friction drag amounts to about half of the total drag. To improve L/D ratio, we tried to apply friction drag reduction technology "Natural Laminar Flow" concept to wing by using "UG3[2]" (KHI's 3-D unstructured grid CFD solver) and "SALLY[3]" ( incompressible 3-D laminar boundary layer stability analysis code ).

## 2. Results of wing design for NLF

### 2.1. Set the target pressure distribution of NAL' experimental aircraft for NLF

We studied a wing without body to understand the relation between airfoil thickness distribution on wing and boundary layer transition characteristics at supersonic cruising condition, M=2.0, CL=0.1. CFD analyses were conducted with five different thickness distributions (NACA0003, 63003, 64003, 65003, 66003)[4] to obtain pressure distributions on wing surface, where the wing planform and the camber line were fixed.

The results of chordwise pressure distribution on the wing were used to perform compressive 3-D laminar boundary layer calculation to get velocity profile in boundary layer. Based on this velocity profile, we conducted 3-D laminar boundary layer stability analysis to get amplification rate of disturbance "N-factor". As an "N-factor" becomes large, the transition occurs earlier. Therefore, we judged a transition characteristics by the "N-factor". From these analyses, a transition point of NACA66003 was found closest to trailing edge. This leads to the conclusion that to optimize chordwise pressure distribution to delay the transition; the pressure distribution should be steep at leading edge and then has a slight acceleration gradient.

Therefore, we studied effects of pressure gradient change at three regions for a transition (Fig.2). After stability analysis of several pressure distributions, we achieved to obtain the optimized chordwise pressure distributions at each wing section for NLF (Fig.3).

To put that into practice, we designed a 3-D wing which realized these target pressure distributions using an inverse design method based on the 2-D Busemann's approximation. We then analyzed the 3-D wing by using UG3 and SALLY (Fig.4). The result was that the designed wing could make the "N-factor" smaller than a conventional wing section (NACA66003), which demonstrated the validity of the optimized chordwise pressure distribution that we proposed. (The increase of the N-factor at $0.05 < x/c < 0.15$ appeared because the pressure distribution of the designed wing was slightly different from the target distribution). These pressure distributions were adopted as the target distributions of NAL's experimental aircraft wing for NLF.

### 2.2. Considerations of Spanwise pressure distribution for NLF

When a wing has a large sweep-back angle, the cross flow instability in the boundary layer plays a dominant role in transition. To reduce cross-flow instability, we observed how the optimized chordwise pressure distribution developed spanwise. The cross flow instability is caused by the presence of cross flow velocity. To reduce cross flow velocity due to spanwise pressure difference, the pressure gradient has to be zero. To achieve such spanwise pressure distribution for NLF, we made two assumptions. One is the flow is supersonic and the other is the lift coefficient is not so large. From these assumptions, the stream line at boundary layer edge of wing upper surface matches with the mean stream, which means cross flow direction is perpendicular to the mean stream. Consequently, the design process was simplified.

Fig.5 shows surface pressure distribution on a simple sweep-back wing. In general, spanwise pressure gradient causes the cross flow velocity in boundary layer, then transition occurs due to cross-flow instability.

On the other hand, Fig.6 shows surface pressure distribution with no pressure gradient along with cross flow direction. That is, when pressure distributions at any section are on one line looking from the side, there is no pressure gradient to the cross flow direction. Consequently, a transition due to cross flow instability doesn't occur. (This wing design method is patented. (No.3005526))

Then, we designed a sweep-back wing by applying the above concept with inverse design method, and observed stream line around boundary layer edge and sublayer of both the initial and the designed configurations at M=2.0

(Fig.7, 8). The tip of the designed wing was twisted up because of applying large acceleration gradient to reduce T-S wave instability (Fig.8). Fig.9 shows that the stream lines on the surface of the designed wing are more straightforward than those of the initial wing, which means the cross-flow velocity in the boundary layer is reduced. (Fig.9, 10)

## 3. Conclusion

From a research on the application of Natural Laminar Flow to supersonic aircraft wings,

- The boundary layer transition is delayed by the optimized chordwise pressure distribution. This chordwise distribution was adopted as the target pressure distribution of NAL's experimental aircraft wing to achieve NLF.
- The optimized spanwise pressure distribution is proposed considering chordwise distribution which reduces cross-flow velocity in the boundary layer.

**Acknowledgment**

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**Reference**

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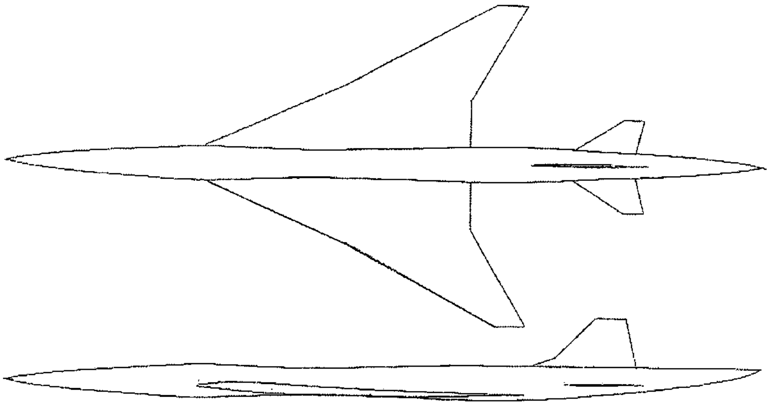

Fig.1 Supersonic sub-scaled experimental aircraft of NAL (1st Baseline)

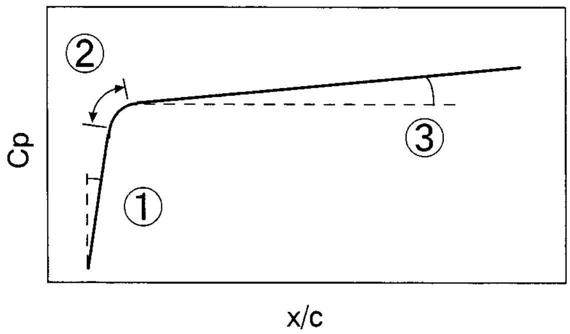


Fig.2 A model of chordwise target pressure distribution for NLF

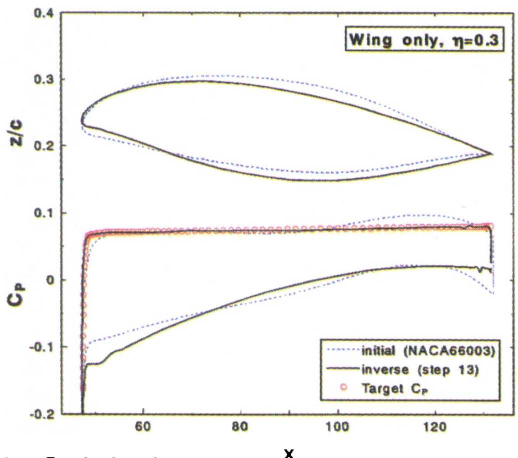


Fig.3 Optimized target pressure distribution for NLF and inverse designed wing of NAL's aircraft ($\eta$ =0.3)

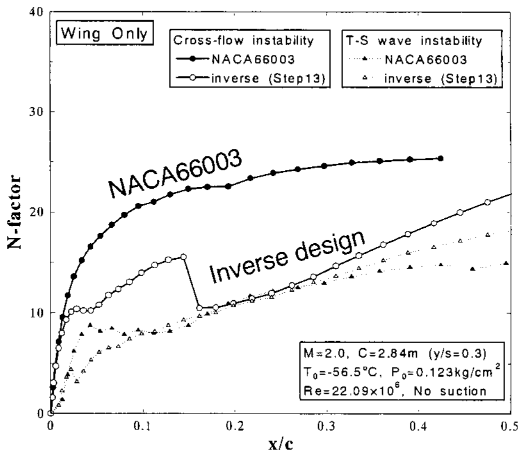


Fig.4 Results of boundary layer stability analysis.

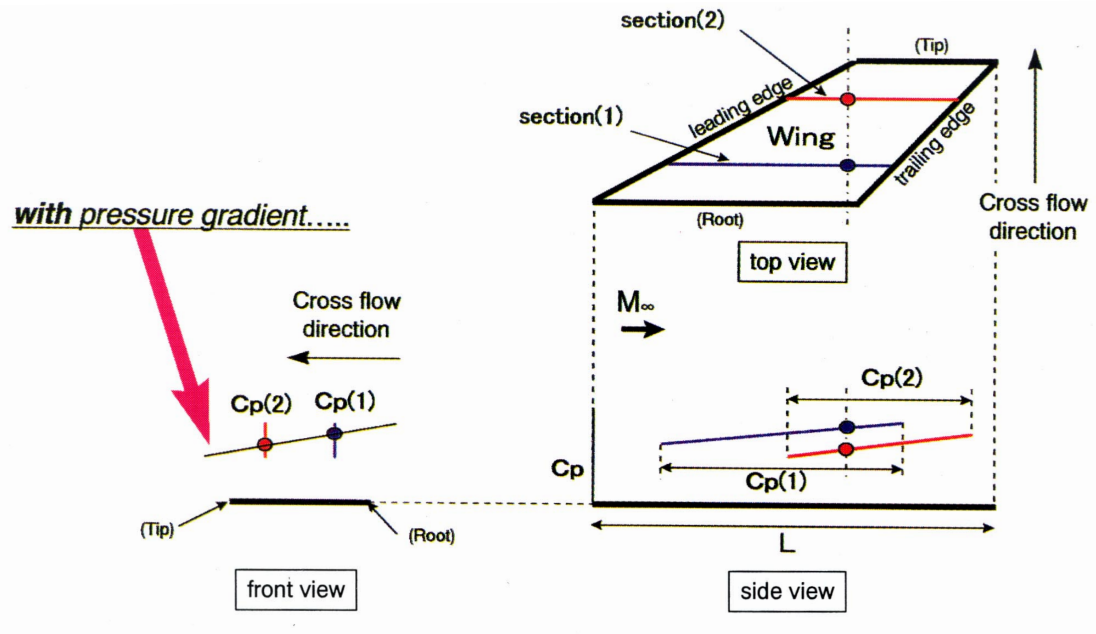


Fig.5 A model of spanwise pressure distribution of a simple sweep-back wing

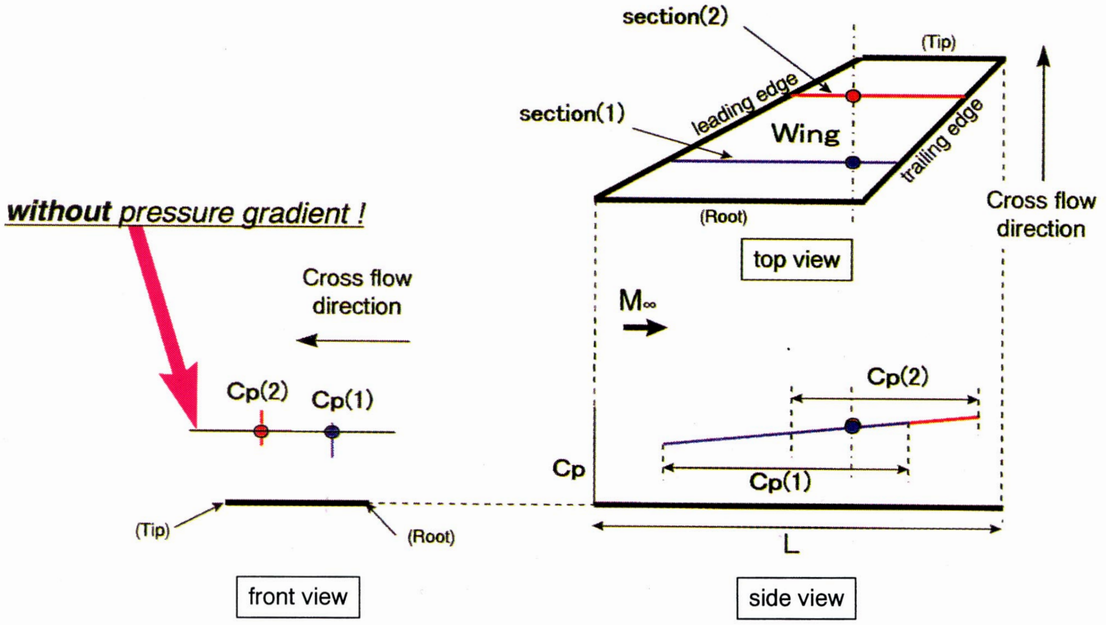


Fig.6 A model of spanwise pressure distribution of new NLF concept wing

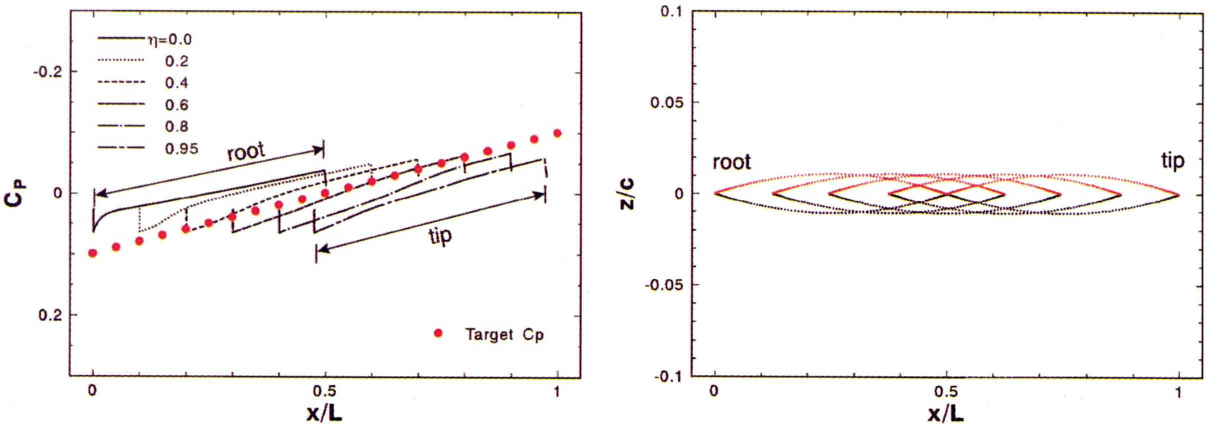


Fig.7 Pressure distributions and wing sections of an initial sweep-back wing

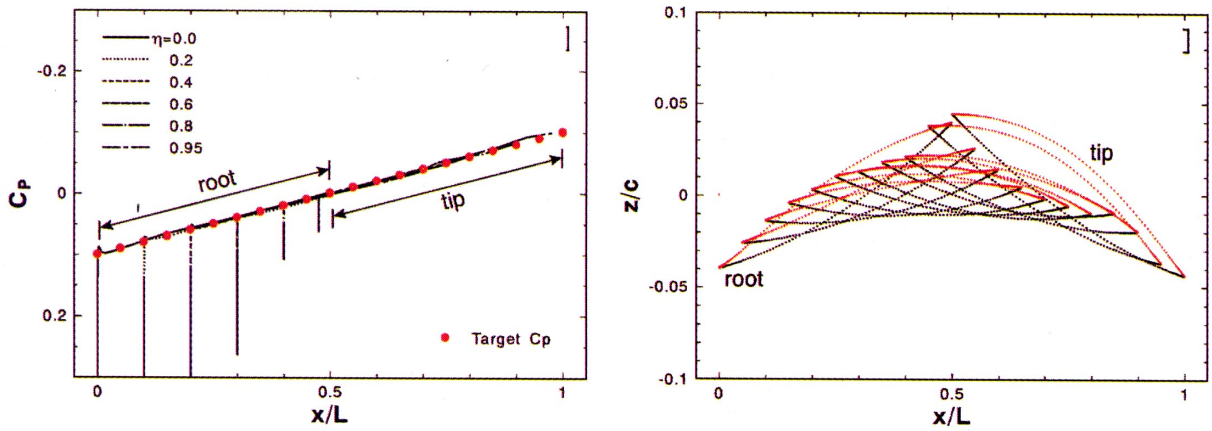


Fig.8 Pressure distributions and wing sections of a designed sweep-back wing by applying the optimized spanwise pressure distribution

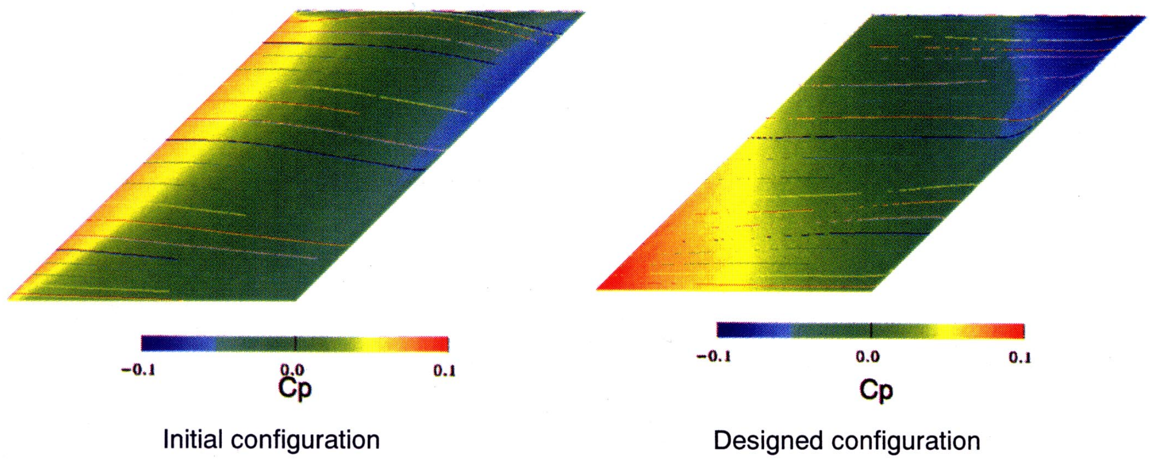


Fig.9 Surface pressure distributions and stream line in sublayer

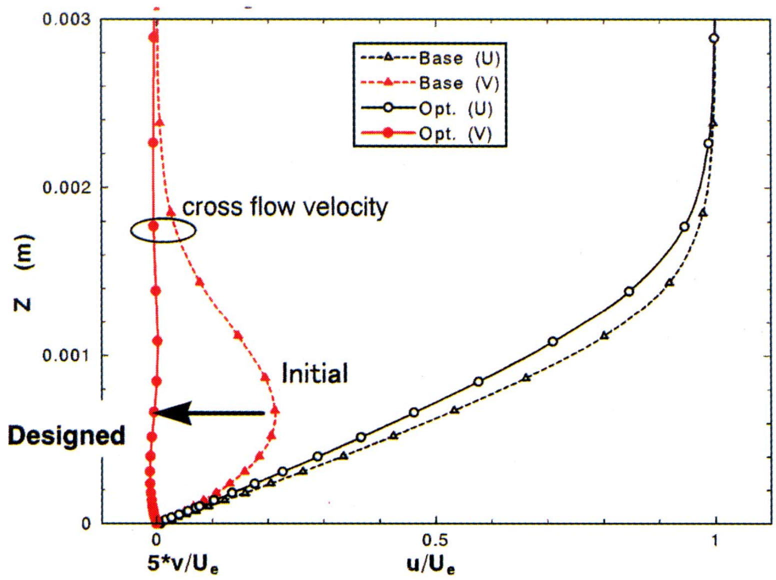


Fig.10 Boundary layer velocity profiles of initial and designed configuration (inverse) at $\eta$ =0.5, x/c=0.8