# Description of the Wind Tunnel Test Data of the NEXST-1

Yuichi SHIMBO
Mitsubishi Heavy Industries, Ltd.
10 Oye-cho, Minato-ku, Nagoya 455-8515, Japan
and
Kenji YOSHIDA
National Aerospace Laboratory
6-13-1, Osawa, Mitaka, Tokyo, 181-0015, Japan

Abstract

The high-speed wind tunnel tests of the NEXST-1 were conducted in 1999 at the 1m x 1m Supersonic Wind Tunnel of NAL. The main objectives of the tests were to confirm the CFD-and-inverse-method-based supersonic natural laminar flow wing design, to collect wind tunnel data to be correlated with CFD and flight test data of the NEXST-1 and to supply data for validation and further development of the CFD tools, other than to collect aerodynamic data to be used in the detailed system design phase of the NEXST-1. At M=2.0 where the data were presented for the workshop, the force test data showed very smooth longitudinal characteristics. The pressure test results proved that the flat-type pressure distribution was achieved on the upper surface as was intended in the wing design, and the data also showed good repeatability and good agreement with ones of a larger model tested in ONERA.

## Introduction

The development of the unpowered National Experimental Supersonic Transport (NEXST-1) was started in 1996 and the basic aerodynamic design was composed of the selection of the wing planform, initial wing warp design and an area-ruled fuselage design based on the linear theory (Ref.1). After a series of basic wind tunnel test campaigns in 1998, the wing was further refined based on the CFD and inverse method, aiming for the supersonic natural laminar flow wing (Ref. 2). Then, the confirmation wind tunnel tests of the NEXST-1 were conducted in 1999 after the aerodynamic configuration was fixed, with the four main objectives below.

(1) To confirm the supersonic natural laminar flow wing design
(2) To collect wind tunnel data to be correlated with CFD and flight test data of the NEXST-1
(3) To supply data for validation and further development of the CFD tools
(4) To collect aerodynamic data to be used in the detailed system design phase

Although there were variety of wind tunnel tests at a wide range of Mach number regime including the launch configuration test (NEXST-1 connected to the solid rocket motor) and the supersonic separation test, the aerodynamic data presented in the workshop were limited to the supersonic test data of the NEXST-1 itself.

## Wind tunnel

The supersonic part of the confirmation wind tunnel test was conducted at the 1m x 1m Supersonic Wind Tunnel in National Aerospace Laboratory (Figure 1), which was the largest supersonic wind tunnel in Japan. It was a blow-down-type wind tunnel and its free stream Mach number ranged from 1.4 to 4.0. The typical flow duration was 40 seconds. Even the wind tunnel tests of the NEXST-1 were conducted at M=1.4 ~ 2.2 and the Reynolds number varied from 22~28million [1/m], only the test data at M=2.0 (Re=25million [1/m]) were presented as the validation data for the workshop. The local Mach number distribution of the typical cross section of the tunnel is shown in Figure 2 (Ref. 3), and

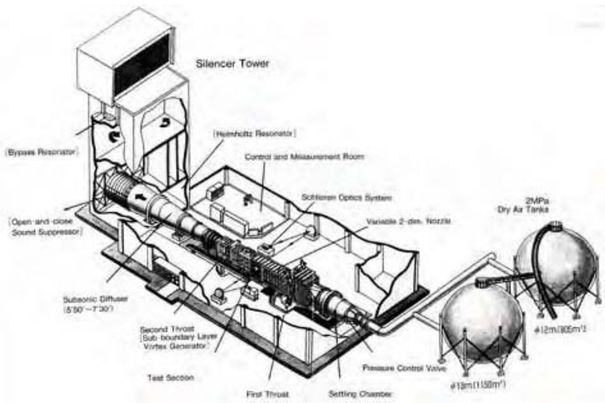


Figure 1 NAL 1m x 1m Supersonic wind tunnel

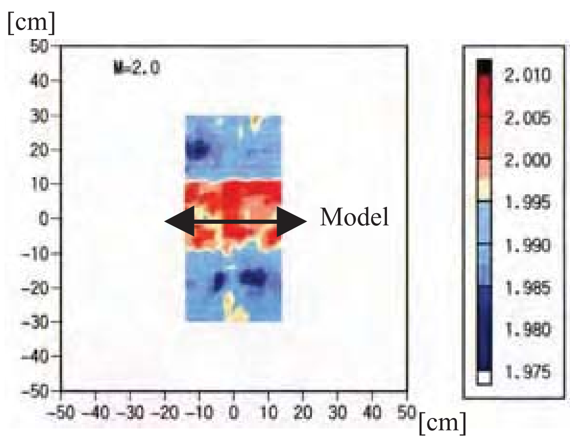


Figure 2 Local Mach number distribution at M=2.0

the Mach number variation at M=2.0 was within about 1% of the indicated Mach number computed from the total pressure of the tunnel measured at the settling chamber, and a typical wall static pressure measured at the test section.

**Force measurement test**

The model used in the experiment was an 8.5% force model of the NEXST-1 (Figure 3). The total length of the model was 790mm, wingspan was 401mm and the mean aerodynamic chord length was 234.1mm. The aft part of the fuselage was cut just behind the trailing edge of the vertical tail to install the sting support system. The airfoil shapes of typical wing sections were checked by the template gauges prior to the experiment. The model had a series of detachable control surfaces (ailerons, horizontal tails and rudder) with strain-gauge-type hinge-moment balances, however, the force data presented in the workshop were limited to the ones with each control surface fixed to its neutral position. In addition, even the wing was intended to the supersonic natural laminar flow wing, tripping discs were put on the 3% local chord location of the wing and tails to prevent unnecessary laminar flow separation at the low Reynolds number wind tunnel test conditions.

The aerodynamic forces and moments of the model were measured by an internal 6-compnent balance located inside the fuselage. The balance load capacity was shown in Table 1 and it was rather large because the model had to stand the starting load of the blowdown wind tunnel up to M=2.2. At M=2.0, the total pressure of the tunnel was 220 kPa absolute and taking 0.2% of the balance load capacity as the typical accuracy of the measurement, it lead to uncertainty in CD by 3 drag counts as is shown in Table 1. The static pressure inside the balance cavity was also measured by a pressure transducer. The balance and pressure transducer data were collected during the continuous pitching up motion at 2deg/sec by the dedicated measurement system of the wind tunnel. The data reduction included the model weight tare correction and the cavity pressure correction. The schrielen flow visualization was also made with the balance load measurement (Figure 4).

The longitudinal force data are shown in Figure 5. The lift and pitching moment characteristics were almost linear and the lift and pitching moment slope were

$CL\alpha = 0.034$ [1/deg]
$CM\alpha = -0.0110$ [1/deg]

Fitting the lift and drag data to a quadratic curve in the form of

$$CD = K(CL - CLo)^2 + CDo$$

it resulted in

Figure 3 Model in the test section

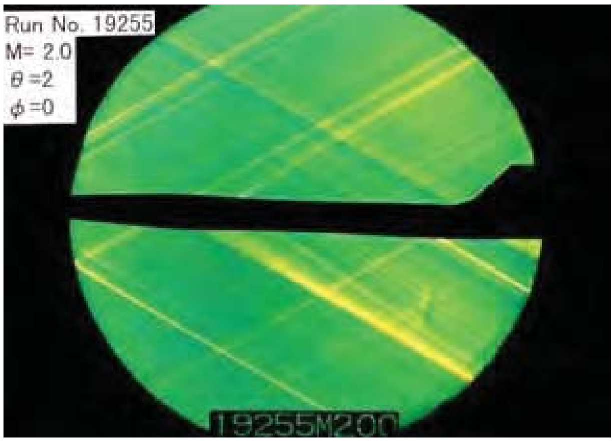


Figure 4 Example of the schlieren flow visualization

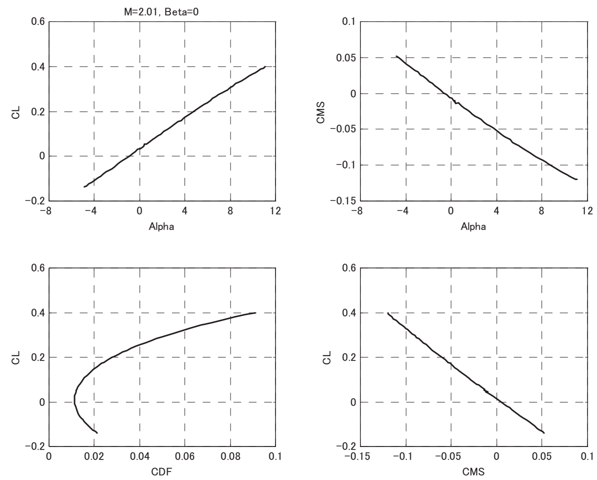


Figure 5 Longitudinal force and moment data at M=2.0

Table 1 Load capacity and expected error of the 6 components internal balance

| | CL | CD | CM |
|---|---|---|---|
| Capacity [N] or [Nm] | 3,923 | 785 | 343 |
| Uncertainty at M=2 | 0.0014 | 0.0003 | 0.0005 |

K=0.5223
CLo＝0.0129
CDo＝0.0104

**Pressure measurement test**

Another 8.5% model with an identical configuration was fabricated for pressure measurement test. This pressure model had no detachable control surfaces and each control surface was fixed to its neutral position. The model had 108 pressure taps, 2 rows arranged on the fuselage (semispan location η=0.0, 0.09), and 5 rows arranged on the wing as is shown in Figure 6. Most of these pressure taps were located on the same place as the NEXST-1 flight vehicle for future comparison among the wind tunnel test, CFD and flight test data. Because the wing was thin and the model was small, upper surface pressure taps were located on the port wing (η=0.15, 0.3, 0.5, 0.7 and 0.9) and lower surface pressure taps were located on the starboard wing (η=0.13, 0.28, 0.48, 0.68 and 0.88). The model was directly attached to the sting support system of the wind tunnel without a balance.

The pressure measurement was conducted with differential-type electrical pressure scanners (EPS). Each sensor unit had 32 ports and 100kPa capacity, and was located inside the sting pod, where pressure tubes from the model were connected. The measurement accuracy of these EPS was 0.08% FS, which corresponded to 0.001 uncertainty in Cp at M=2.0. The total pressure of the tunnel, test section wall pressure and atmospheric pressure used as a reference pressure of the EPS were measured by absolute-type pressure transducers. These transducers were calibrated within 0.08% Reading accuracy, which lead to uncertainty in Cp less than 0.001 at the typical M=2.0 condition.

The measurement was conducted in the pitch-and-pose mode. The typical pressure data at M=2.0 and α=1.5 deg are shown in Figure 7, along with the repeatability data in the same wind tunnel test campaign. The pressure distribution on the wind upper surface at α=1.5 deg was almost flat as was intended in the CFD-and-inverse-method based design for the supersonic natural laminar flow, and the repeatability of the pressure tap measurement was very good.

The effect of the uniform flow quality of the tunnel was also checked by comparing the pressure distribution at the roll angle φ=0 deg 63 deg with the pitch angle θ=0 deg, both of which resulted in α=0 deg. The pressure distribution agreed well again, and the local Mach number distribution was found to have slight effect.

Another pressure model in 23.3% scale of the NEXST-1 was tested in ONERA S2MA wind tunnel in France in 2000 (Figure 9). Although the model was made mainly for the transition point measurement by hot-films, it had 30 pressure taps on the upper surface of

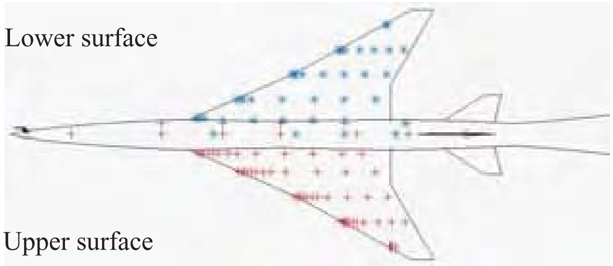


Figure 6 General view of the pressure model

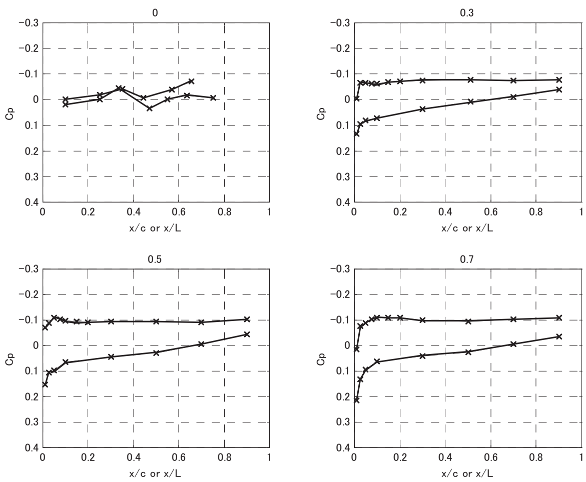


Figure 7 Typical pressure data and repeatability at M=2.0 and α=1.5 deg (Left upper : η=0.0, right upper : η=0.3/0.28, left lower : η=0.5/0.48, right lower : η=0.7/0.68, Line : typical result, x : repeatability data)

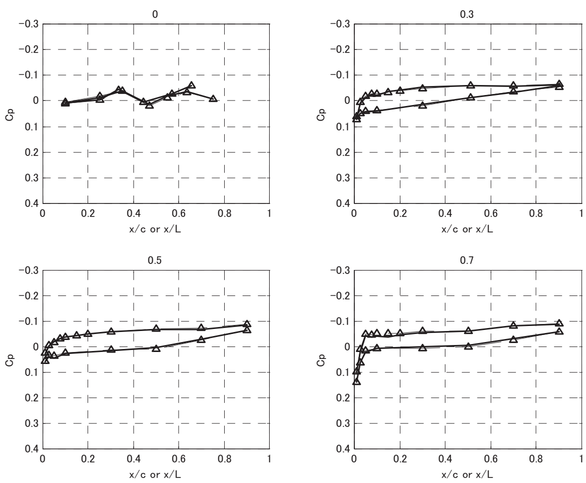


Figure 8 Effect of wind tunnel flow at M=2.0 deg (Left upper : η=0.0, right upper : η=0.3/0.28, left lower : η=0.5/0.48, right lower : η=0.7/0.68, Line : φ=0 deg, triangle : φ=63 deg)

the wing and fuselage. The ONERA test data at Re=5 million [1/m] was compared with NAL data in Figure 10 and it showed slight difference except the leading edge region at η=0.5, even the model, wind tunnel and the test Reynolds number were different.

## Conclusions

A collection of force and pressure distribution data was made through the confirmation wind tunnel test campaign of the NEXST-1. At M=2.0 the force test data showed very smooth longitudinal characteristics and the pressure test results proved that the flat-type pressure distribution was achieved on the upper surface as was intended in the wing design. The data also showed good repeatability and good agreement with ones of a larger model tested in ONERA.

## References


(1) Shimbo, Y., Yoshida, K., Iwamiya, T., Takaki, R. and Matsushima, K., Aerodynamic Design of the Scaled Supersonic Experimental Airplane, International CFD Workshop for Super-Sonic Transport Design, 1998, pp.62-67

(2) Shimbo, Y., Iwamiya, T., Yoshida, K., Suzuki, K. and Matsushima, K., Natural Laminar Flow Wing Design for the NAL's Scaled Supersonic Experimental Airplane, Proceeding of The 30th Anniversary Memorial JSASS Annual Meeting, pp. 159-162

(3) Tate, A., Hamamoto, S., Noda, J., Watanabe, M. and Hara, N., Mach Number Distribution of NAL Supersonic Wind Tunnel, NAL SP-45, pp.29-39


Figure 9 NEXST-1 23.3% model in ONERA S2MA wind tunnel

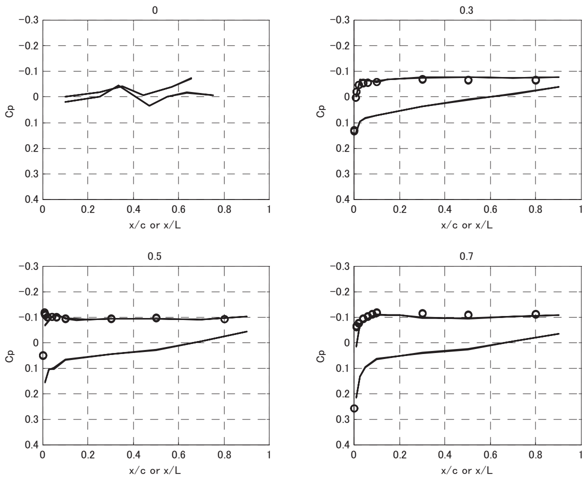


Figure 10 Comparison with 23.3% model data deg (Left upper : η=0.0, right upper : η=0.3/0.28, left lower : η=0.5/0.48, right lower : η=0.7/0.68, Line : 8.5% model at NAL, circle : 23.3% model at ONERA)