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CHŌFU, TOKYO, JAPAN

Evaluation of NACA0012 Airfoil Test Results in the NAL Two-Dimensional Transonic Wind Tunnel*

Norikazu SUDANI*[1], Hiroshi KANDA*[1], Mamoru SATO*[1],
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ABSTRACT

Surface pressure and drag measurements on the NACA0012 airfoil were conducted in the NAL Two-Dimensional Transonic Wind Tunnel. Using a comparison with other wind tunnel data, the wall interference effects are discussed, especially those from the sidewall. The results suggest that the Mach number of the actual flow around the airfoil is lower than the setting Mach number. The Mach number correction for the sidewall boundary-layer effects based on the similarity rule was applied to the present measurements, thereby showing that the shock positions, the pressure distributions, and the minimum drag coefficients are in good agreement with both other wind tunnel results and the Navier-Stokes calculation. It is shown that the evaluation indicates satisfactory transonic airfoil test results in the NAL Two-Dimensional Transonic Wind Tunnel.

Keywords: Airfoil, Transonic Flow, Two-Dimensional Flow, Mach Number Correction, Sidewall Interference

概　　要

航空宇宙技術研究所二次元風洞において NACA0012翼型について表面圧力分布測定，後流測定を行い，他風洞での測定結果との比較を通して本風洞における壁干渉，特に側壁干渉についての検討を行った。その結果，翼のまわりには設定マッハ数よりも低いマッハ数の流れが実現されている可能性のあることが判明した。そこで，今回得られた実験結果に遷音速相似則に基づいた側壁干渉によるマッハ数修正を施し，衝撃波位置，圧力分布，抵抗値等に検討を加えた結果，他風洞と非常に良好な一致が認められた。さらに二次元の CFD 解析結果との対比においても良好な一致が得られ，本検討は二次元風洞における翼型試験に一つの妥当な評価を与えることがわかった。

NOMENCLATURE

AR : aspect ratio ($= b/c$)
b : width of the tunnel, span of the airfoil
c : airfoil chord
C_d : drag coefficient
C_{d_0} : minimum drag coefficient
C_l : lift coefficient
C_n : normal force coefficient
C_p : pressure coefficient
$C_{p,c}$: corrected pressure coefficient
$C_p{}^*$: critical pressure coefficient
H : shape factor of the sidewall boundary-layer
k : constant (see Equation (3))
k_1 : constant ($= \pi \beta b/l$)
l : wavelength of the wavy wall
M : Mach number
M_∞ : free stream Mach number

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*[1]Aircraft Aerodynamics Division

M_c : corrected Mach number
M_s : setting Mach number
Re : Reynolds number based on the airfoil chord
x : streamwise coordinate
x_s : shock location on the airfoil surface
y : normal coordinate
z : spanwise coordinate
α : angle of attack
β : compressibility factor ($= \sqrt{1 - M^2}$)
$\delta*$: sidewall boundary-layer displacement thickness
ϕ : velocity potential

1. Introduction

In the NAL Two-Dimensional Transonic Wind Tunnel (2D-TWT), tests of the 250 mm chord airfoils (aspect ratio of 1.2) are usually performed in order to realize the highest Reynolds number condition (40×10^6)[1]. In such two-dimensional airfoil tests, the effects of the wall interference are generally concerned with the effects of top and bottom walls. Thus, in the NAL 2D-TWT, only the top and bottom walls interference correction by the Sawada method[2] has been made. However, it has recently been recognized that the effects of the sidewall boundary-layers can influence on the measurements on the airfoils at the mid-span to the reasonable extent. In McCrosky's report,[3] the fact is shown that the shock waves on the NACA0012 airfoil at $M = 0.8$ and $\alpha = 0°$ in the NAL's measurements stand at considerably forward positions compared with other wind tunnels. Therefore, in the NAL 2D-TWT, the measured results are largely influenced by the sidewall boundary-layers and the Mach number of the actual flow around the model will be considered lower than the setting Mach number.

In order to remove the influence of the sidewall boundary-layer, boundary-layer suction at the model location has been tried. However, on the contrary, it was found that the sidewall suction caused the nonuniformity of the Mach number along the center-line of the test section in the flow direction. Moreover, the effects of the Mach number nonuniformity on the center-line pressure distributions were remarkable. The purpose of this report is to evaluate the results measured without sidewall boundary-layer removal by means of the sidewall correction, especially the Mach number correction.

As to the sidewall interference correction, Barnwell proposed a simplified analysis in the form of a modified Prandtl-Glauert rule to account for the attached sidewall boundary-layer effects.[4] After that, Sewall extended it to transonic speeds using the von Karman similarity parameter.[5] These corrections have been derived under certain assumptions of simplified boundary-layer treatment and linear variation of the crossflow velocity across the width of the tunnel. These assumptions imply that the change in the streamtube area is gradual so that the sidewall boundary-layer effects can be treated one-dimensionally. Therefore, the corrections are inapplicable to high-aspect-ratio models (in this report, aspect ratio is defined as the ratio of the span to the chord length). Murthy modified the Barnwell-Sewall correction in order to include the effect of the model aspect ratio, and proposed the wavy flow model by considering the compressible flow between a straight wall and a wavy wall.[6]

In this report, by applying the Murthy correction to the results on the NACA0012 airfoils of aspect ratio, 1.2 and 2.0, measured in the NAL 2D-TWT, the sidewall effects on the measured data are evaluated through the comparison with both the results measured in other wind tunnels and the calculation by the Navier-Stokes method.

2. Experimental Apparatus and Numerical Method

The tests were conducted in the NAL Two-Dimensional Transonic Wind Tunnel. The detail description on the wind tunnel facility is given in Reference 1. The tunnel is a blowdown type, and it is capable of operating at Reynolds numbers based on the airfoil chord length (250 mm) from 7×10^6 to 40×10^6 according to the stagnation pressure. The test section is 1 m in height and 0.3 m in width.

The top and bottom walls of the test section are slotted walls with open area ratio of 3%. The tests were made at the setting Mach numbers of nearly 0.8 and Reynolds numbers based on the airfoil chord of 7 x 10^6, 15 x 10^6, 21 x 10^6, 30 x 10^6 and 40 x 10^6. No attempt was made to fix the transition on models.

The model chord length is usually 250 mm (aspect ratio of 1.2), but in the present tests two different chord models were used to investigate the effect of aspect ratio. One is the 250 mm chord and the other is the 150 mm (aspect ratio of 2.0). In this respect, discussion is mainly made on aspect ratio of 1.2 except mentioned.

Normal forces acting on the airfoil were determined from surface static pressure measurements. The surface pressure was measured with Scani-valves placed in the plenum chamber and near the model to keep the small lag time. Pressure signals to be measured were differentially detected to the reference signal of plenum chamber pressure. Drag forces were calculated from vertical variations of the total and static pressure measured across the wake. The wake rake had a static pressure probe and two stagnation pressure probes to confirm the two-dimensionality of the wake. The stagnation pressure probe was positioned in the vertical plane at $z/(b/2) = -0.133$, and the static pressure probes were at $z/(b/2) = 0.2$ and 0.533. They were positioned twice the chord length rearward of the trailing edge of the 250 mm chord airfoil.

The Navier-Stokes method,[7] developed by Takanashi, was applied to simulate the fully turbulent flows past the NACA0012 airfoil at a Reynold number of 21 x 10^6.

3. Mach Number Correction

As to the Mach number correction, the Murthy method is applied to the measurement results. The Murthy correction assumes that the sidewall boundary-layer is a kind of wavy wall and that the ratio of the crossflow velocity at any point in the flow to that at the wall was only a function of the distance from the wavy wall. These assumptions imply that the crossflow velocity variation along the airfoil span with sidewall boundary-layer effects can be represented by the wavy wall flow model. This method of Reference 6 will be briefly summarized.

The development of the sidewall boundary-layer induces the spanwise velocity across the width of the tunnel, and the flow in the tunnel tends to become three-dimensional. In general, the corresponding small perturbation equation for the flow in the wind tunnel is

$$(1 - M_\infty^2)\,\phi_{xx} + \phi_{yy} + \phi_{zz} = 0 \qquad (1)$$

This equation including the three-dimensionality induced by the sidewall boundary-layer effects is two-dimensionally approximated to

$$(1 - M_\infty^2 + k)\,\phi_{xx} + \phi_{yy} = 0 \qquad (2)$$

At the mid-span, the value of k can be estimated as

$$k = \frac{2\delta^*}{b}\left(2 + \frac{1}{H} - M_\infty^2\right)\left(\frac{k_1}{\sinh k_1}\right) \qquad (3)$$

where

$$k_1 = \frac{\pi\beta b}{l} \qquad (4)$$

The value of l represents a wavy length and it is assumed as

$$l = 2c \qquad (5)$$

This assumption is based on the fact that the influence of the sidewall boundary-layer on the airfoil is over a distance of about twice the chord length of the airfoil. Therefore, the value of k includes the effect of the airfoil aspect ratio.

Comparing Equation (1) with (2), the corresponding expressions for the corrected Mach number (M_c) and the corrected pressure coefficient ($C_{p,c}$) are written as

$$\frac{1 - M_\infty^2 + k}{M_\infty^{4/3}} = \frac{1 - M_c^2}{M_c^{4/3}} \qquad (6)$$

$$C_{p,c} = \left(\frac{M_\infty^2}{M_c^2}\right)^{1/3} C_p \qquad (7)$$

Table 1 gives typical examples of the corrected values of the Mach numbers by this method. In the NAL 2D-TWT, the width of the tunnel b is 300 mm, the

Table 1. The values of the corrected Mach numbers in the NAL wind tunnel

Model chord	M_∞	M_c	ΔM
250*mm* (AR=1.2)	0.823	0.800	-0.023
	0.800	0.777	-0.023
150*mm* (AR=2.0)	0.817	0.800	-0.017
	0.800	0.783	-0.017

displacement thickness of the sidewall boundary-layer δ^* is about 4.7 mm and the shape factor H is about 1.5 irrespective of Mach number and Reynolds number.[8] In case of aspect ratio of 1.2, the corrected value of the Mach number is about –0.023 at the setting Mach number of 0.8.

4. Results and Discussion

Shock Wave Position

A series of pressure measurements were conducted in the present study. The present results of the shock position at $M = 0.8$ and $\alpha = 0°$ are shown in Figure 1 in addition to the figure described by McCrosky.[3] The shock position x_s is defined as the approximate midpoint of the pressure rise across the shock wave. As noted in Reference 3, this quantity appears to be particularly sensitive to wall interference effects and to errors in Mach number.

In Figure 1, notwithstanding the majority of the results seem to lie between $x_s = 0.44$ and 0.48, the NAL data in the past (Takashima, et al.; open circles) are largely forward out of the extent. The solid triangles are the NAL present data at the setting Mach number of 0.8. At the same setting Mach number, the present results are in excellent agreement with those of the past. There exists no problem on the repeatability. The solid circles represent the results with the Murthy correction. As shown in Table 1, the setting Mach number is 0.823. In this case, over a wide range of Reynolds numbers the shock waves are positioned within the extent where most of the results in other wind tunnels lie.

The Murthy correction includes the effect of aspect ratio, so that the tests on the higher-aspect-ratio (= 2.0) model were made to investigate this effect. The open hexagons denote the corrected data, and the hexagons with diagonals are the uncorrected. In the uncorrected case (i.e. the setting Mach number of 0.8), the shock waves are positioned rearward than those of the lower-aspect-ratio (= 1.2) model because the sidewall interference is weaker. Although the corrected data are also positioned rearward, all the NAL corrected data are within the good extent. Applying the Murthy sidewall correction gives very satisfactory results.

In addition to these data, the Navier-Stokes calculation data (a double diamond symbol) are also plotted. This is in good agreement with the NAL experimental data on the higher-aspect-ratio model in the condition where the sidewall interference is considered to be smaller.

Figure 2 shows the relation between the shock wave position and Mach number at zero angle of attack. The open symbols represent the uncorrected data, and the solid are the corrected. Using the Murthy correction the curve moves left, and at $M = 0.8$, the shock position is within the good extent which is estimated by McCrosky. Also at high Reynolds number (40×10^6; square symbols), applying the correction gives the good results.

The pressure distributions at $M = 0.8$ and $\alpha = 0°$ compared with the results measured in other wind tunnels and the calculation by the Navier-Stokes method are shown in Figures 3 and 4. In Figure 3, the solid circles represent the results measured in the NAL 2D-TWT, the open squares are Harris' results[9] in the Langley 8-foot Transonic Pressure Wind Tunnel, and the open diamonds are McDevitt's[10] in the Ames High Reynolds Number Facility. Solid line denotes the Navier-Stokes calculation. The experiemental data denote only the upper surface pressure distributions on the airfoils. The NAL data were measured at the corrected Mach number of 0.8 (the setting Mach number of 0.823). All the data are almost in accordance except the slight differences of the shock wave position and the suction peak level upstream of the shock. The disagreements are explained by the following facts.

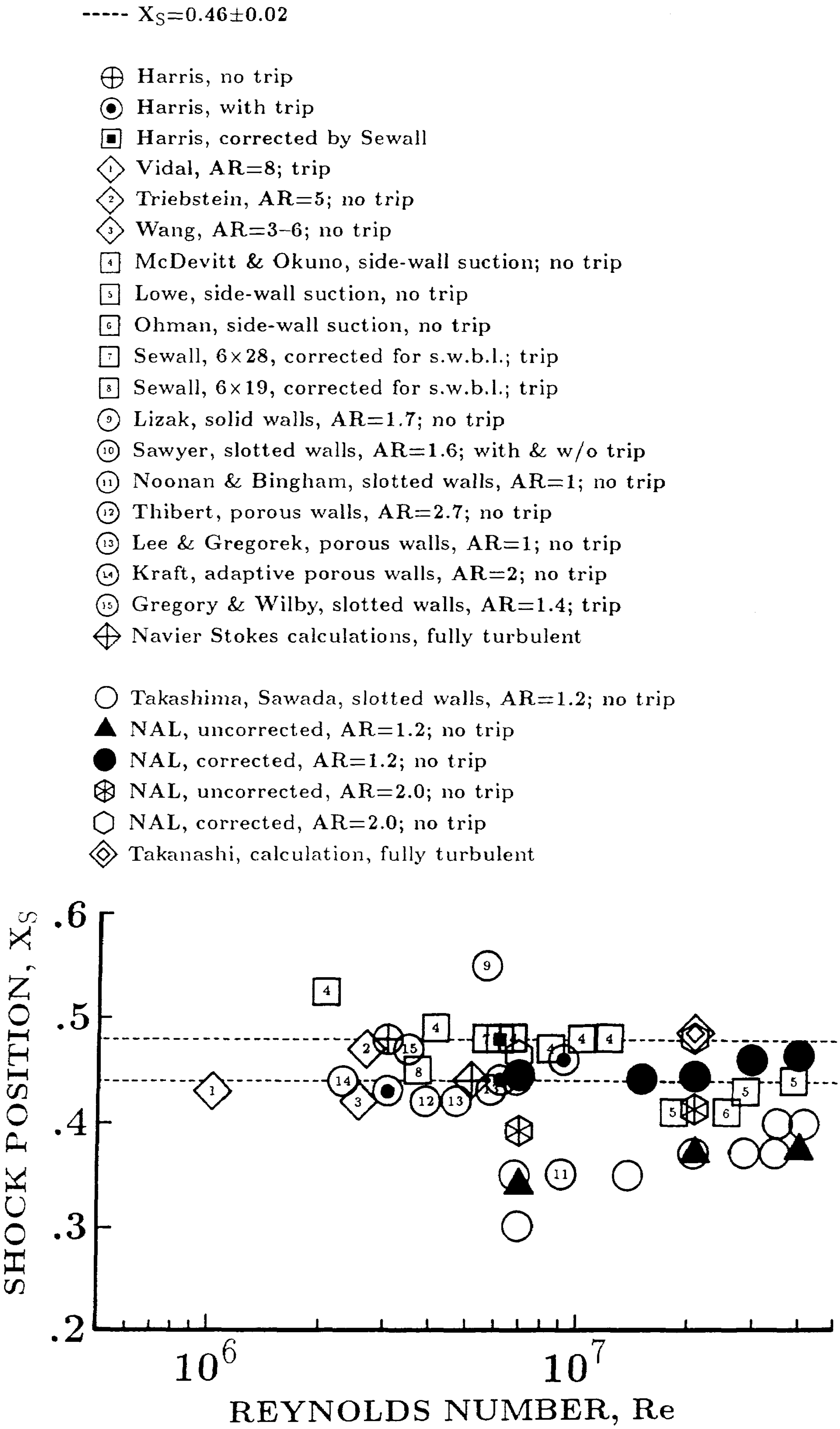

Figure 1. Shock wave position vs. Reynolds number at $M = 0.8$ and $\alpha = 0°$

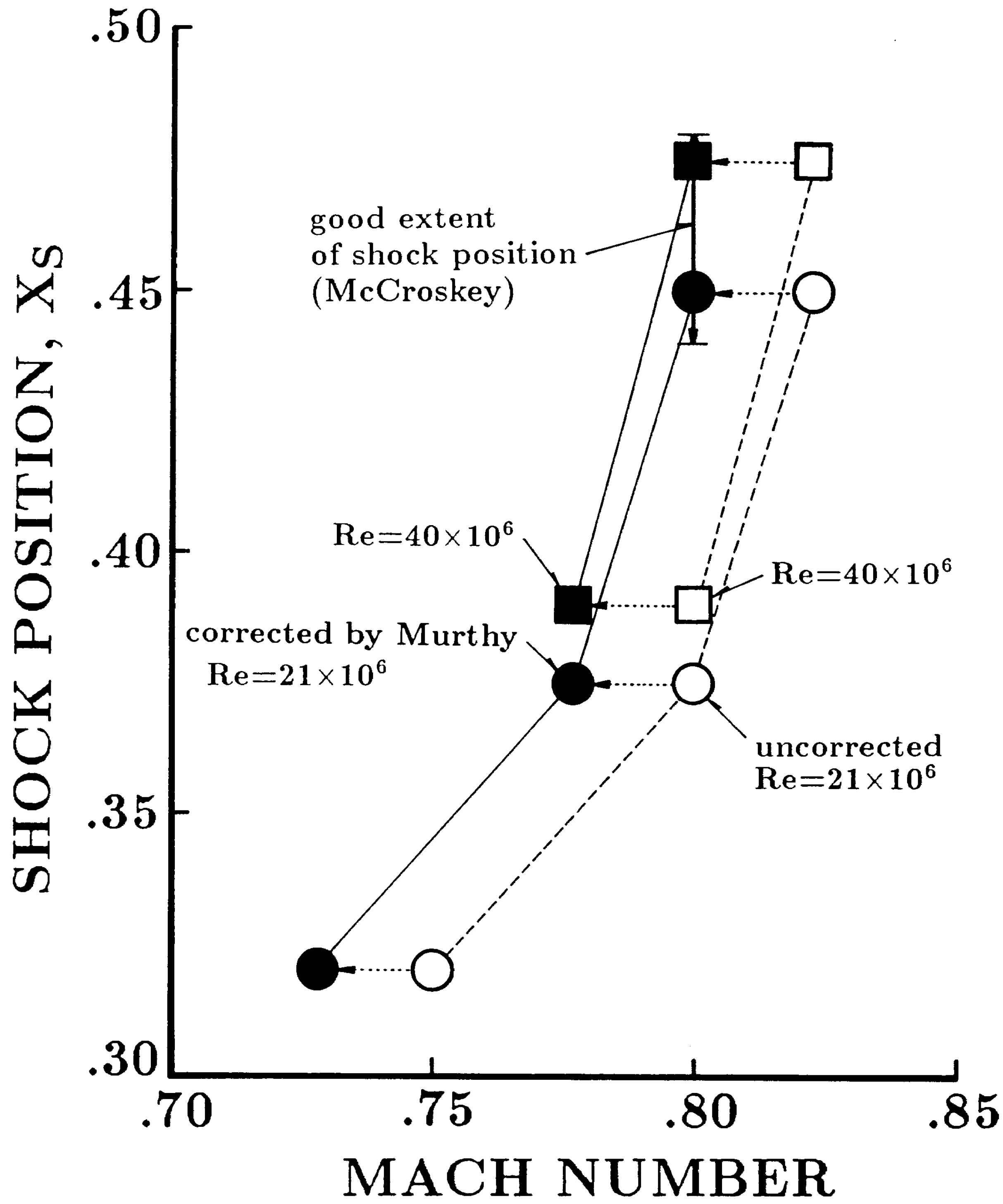

Figure 2. Shock wave position vs. Mach number

One is that, while the calculation assumes the fully turbulent, the laminar region can remain to some extent in the experiments. Another is that, among the experimental data, there exist differences in Reynolds number and whether the transition is free or fixed. It is likely that the data scatter because of the sensitivity of the shock wave to these factors.

The surface pressure measurements on two different chord models were made at the uncorrected and the corrected Mach number of 0.8. Figure 4 indicates the effects of the airfoil aspect ratio. They are 1.2 (shown by circle symbols) and 2.0 (triangle symbols), respectively. The open symbols denote the data measured at the uncorrected Mach number, and the solid symbols at the corrected. As mentioned previously, when no Mach number correction is made, the shock waves are located forward. However, using the correction, the shocks move

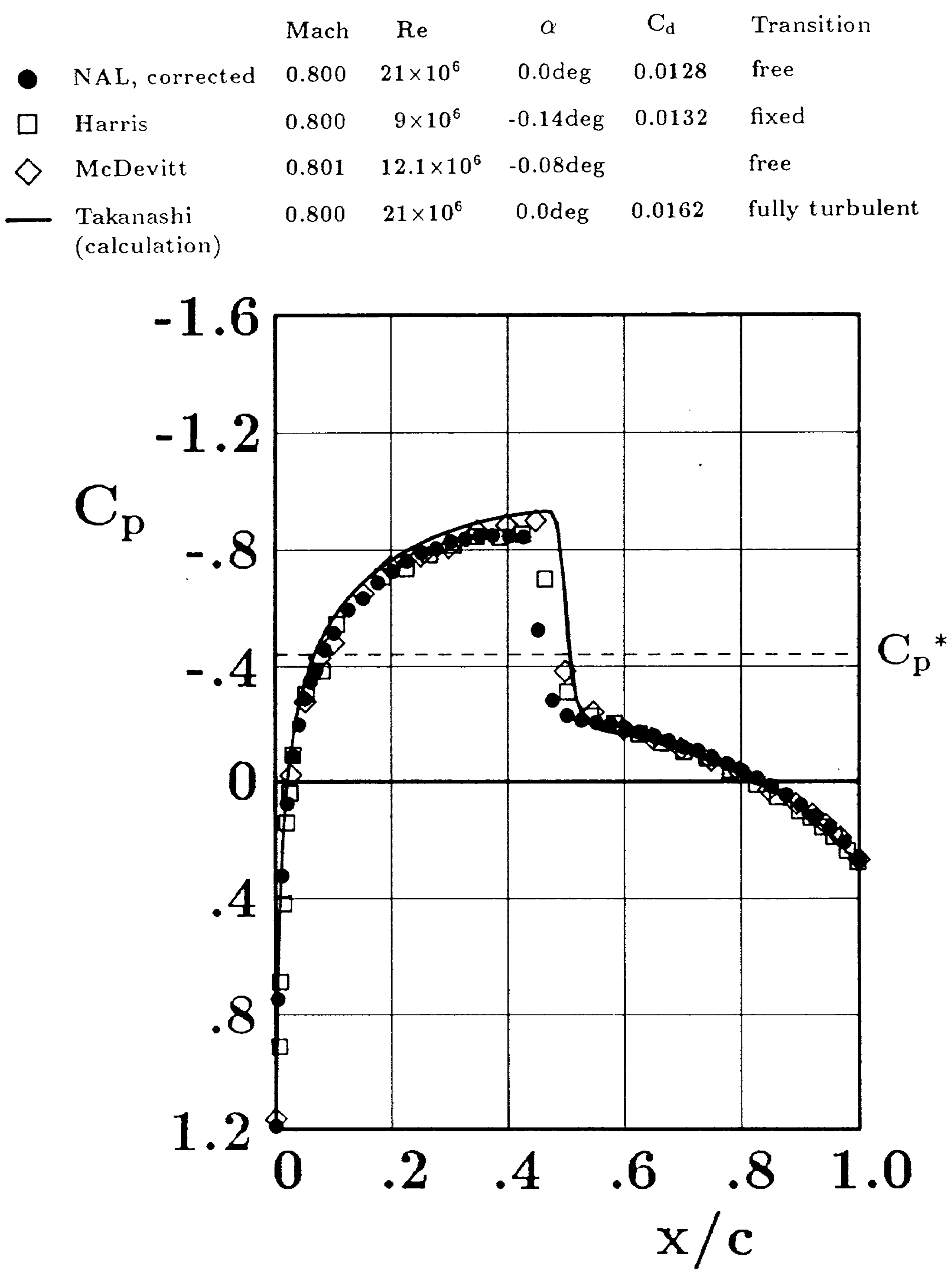

Figure 3. Pressure distributions at M = 0.8 and α = 0°

rearward into the good extent and, particularly, the data on the higher-aspect-ratio model are in very good agreement with the calculation. From the viewpoint of the pressure distributions at the mid-span, it results that the Murthy correction is more applicable by using the higher-aspect-ratio model. Since the sidewall interference is small and the value of the correction is also small, the flow around the high-aspect-ratio airfoil can keep the good two-dimensionality.

However, it is very difficult to use a high-aspect-ratio model because of the manufacture accuracy,

		Mach	Re	α	C_d	AR	Transition
●	NAL, corrected	0.800	21×10^6	0.0deg	0.0128	1.2	free
▲	NAL, corrected	0.801	21×10^6	0.0deg	0.0163	2.0	free
○	NAL, uncorrected	0.801	21×10^6	0.0deg	0.0089	1.2	free
△	NAL, uncorrected	0.801	21×10^6	0.0deg	0.0109	2.0	free
——	Takanashi (calculation)	0.800	21×10^6	0.0deg	0.0162		fully turbulent

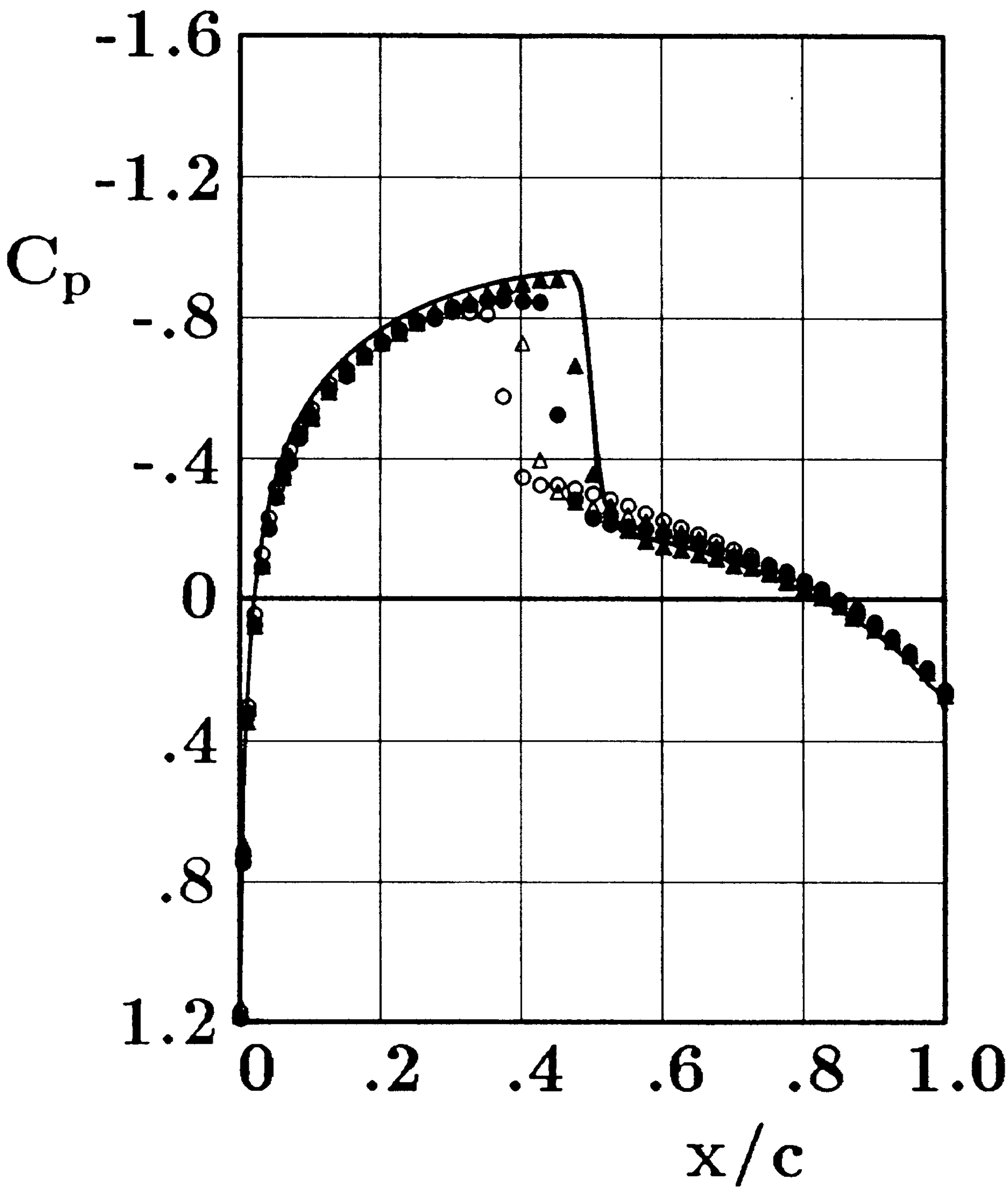

Figure 4. Effect of aspect ratio on pressure distribution at $M = 0.8$ and $\alpha = 0°$

the strength of the model and the productivity of the data at high Reynolds numbers. Therefore, in the NAL 2D-TWT, a high-aspect-ratio model cannot be usually used, so that it is necessary to find the good sidewall interference correction as soon as possible.

Up to the present, in the NAL 2D-TWT, the measured results have been assumed to be similar to the results measured at the Mach numbers which are the same as the setting Mach numbers, because

		Mach	Re	α	C_d	Transition
●	NAL, corrected	0.778	21×10^6	0.0deg	0.0089	free
□	Harris	0.779	9×10^6	-0.14deg	0.0093	fixed
——	Takanashi (calculation)	0.777	21×10^6	0.0deg	0.0097	fully turbulent

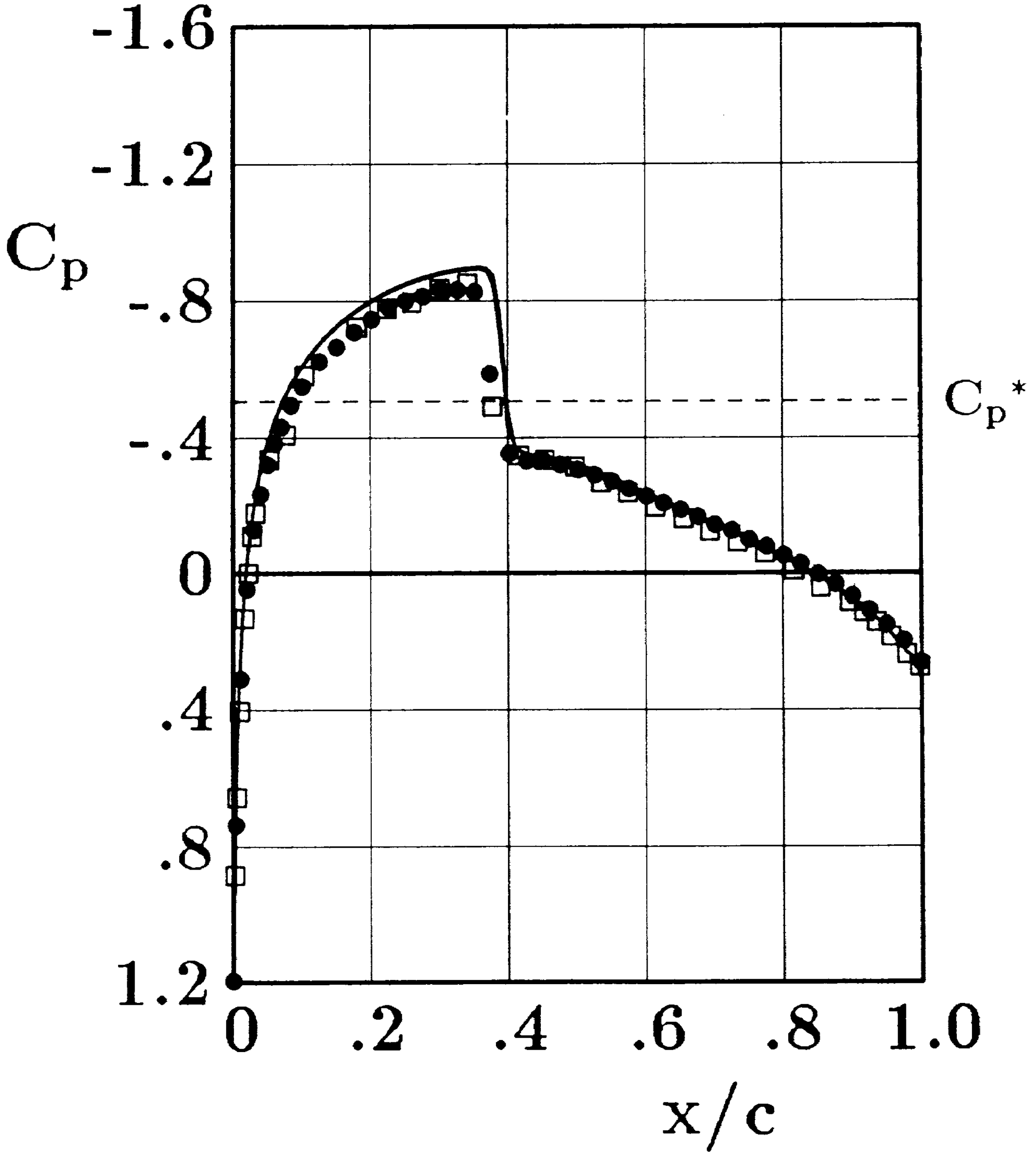

Figure 5. Pressure distributions at M = 0.777 and α = 0°

the correction values of the top and bottom walls interference are negligibly small. However, through the present investigation, it is found that the setting Mach number of 0.8 corresponds to the value of 0.777 by using the Murthy correction. The pressure distribution in this case is shown in Figure 5. Also the Harris data (M = 0.779)[9] and the calculation are described with it. This figure shows that all the data are in good agreement, so that it is confirmed that the NAL data at M_s = 0.8 are equivalent to the data at $M \simeq 0.777$.

In the present tests, in addition to the effects of the sidewall correction, the shock behavior at high Reynolds numbers is revealed. Although the shock

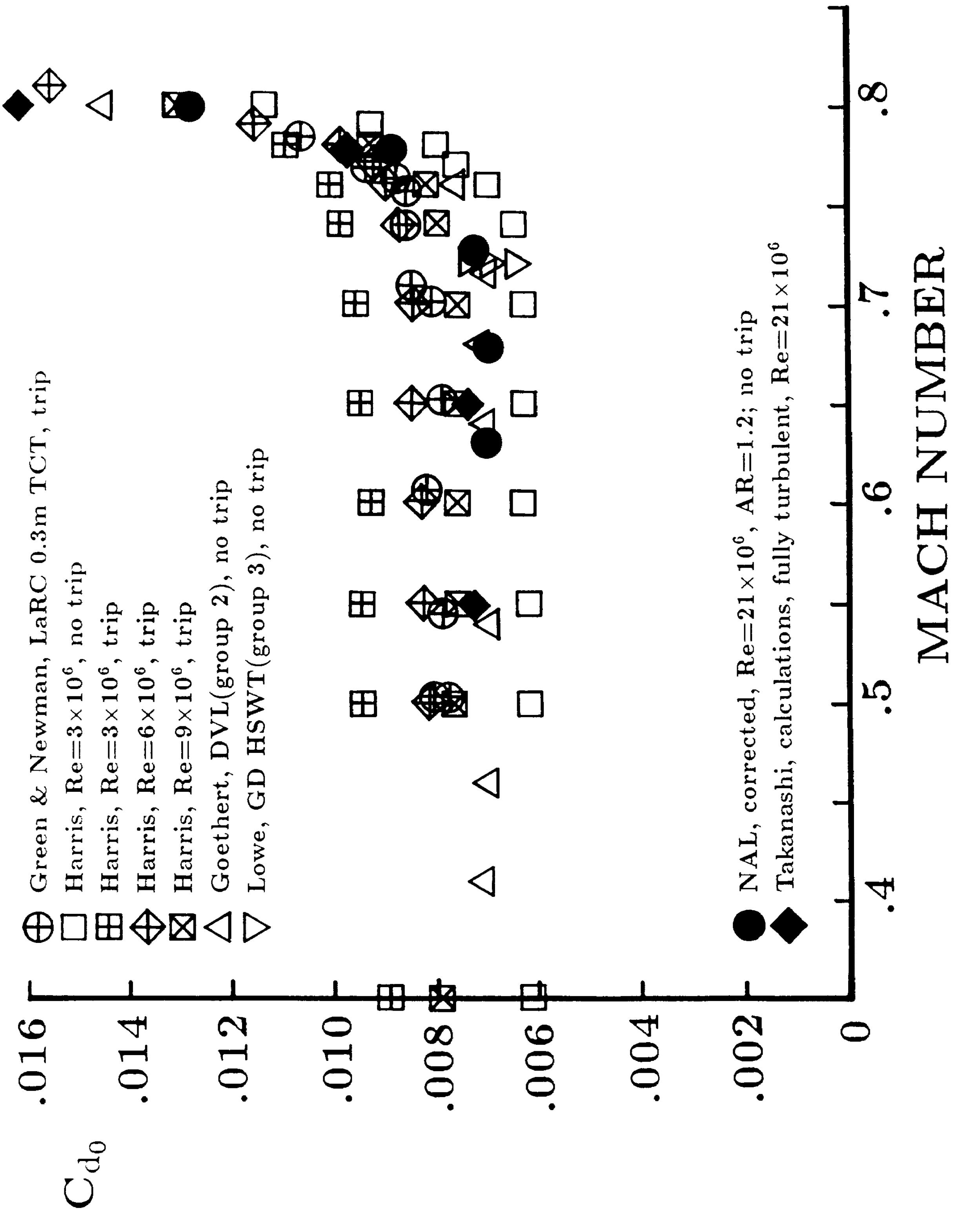

Figure 6. Minimum drag vs. Mach number

waves are located at the same position at Reynolds number of less than 21 x 10^6, they stand rearward abruptly at more than 21 x 10^6. Since there are hardly enough experimental data besides them, the phenomena cannot be fully elucidated. Therefore, this will be one of the most interesting subjects in future.

Drag Coefficient

As another transonic characteristics on the NACA0012 airfoil which is used to evaluate the NAL data corrected by the Murthy method, the section drag coefficient is considered. The Mach number correction requires the drag coefficient to be adjusted because of using the similarity rule. The adjustment actually applies only to the component of pressure drag in the drag coefficient because it is assumed that the skin friction drag is independent of Mach number. Really, as shown in Figure 6, in the region below the drag rise where the skin friction drag is considered to occupy the most of the drag coefficient, it is substantially constant.

The present results of the minimum drag coefficient (i.e. at zero angle of attack) plotted against Mach number are shown in Figures 6 and 7. Figure 6 shows the NAL's present results which are described in addition to the figure in McCrosky's report.[3] The solid circles represent the NAL corrected data at Re = 21 x 10^6, and the solid diamonds are the calculation resutls. Here, Harris' results scatter excessively because of the effect of the trip on the airfoil, so that it is too difficult to estimate the reasonable value. However, in regard to the drag divergence Mach number, all the data are in good agreement. Although the deviation of the values below the drag rise between the NAL experimental data and the calculation are slight, the values of them at M = 0.8 are in disagreement. The drag near the drag divergence Mach number is so sensitive to the difference of Mach number that the exact comparison is very difficult in the viewpoint of the

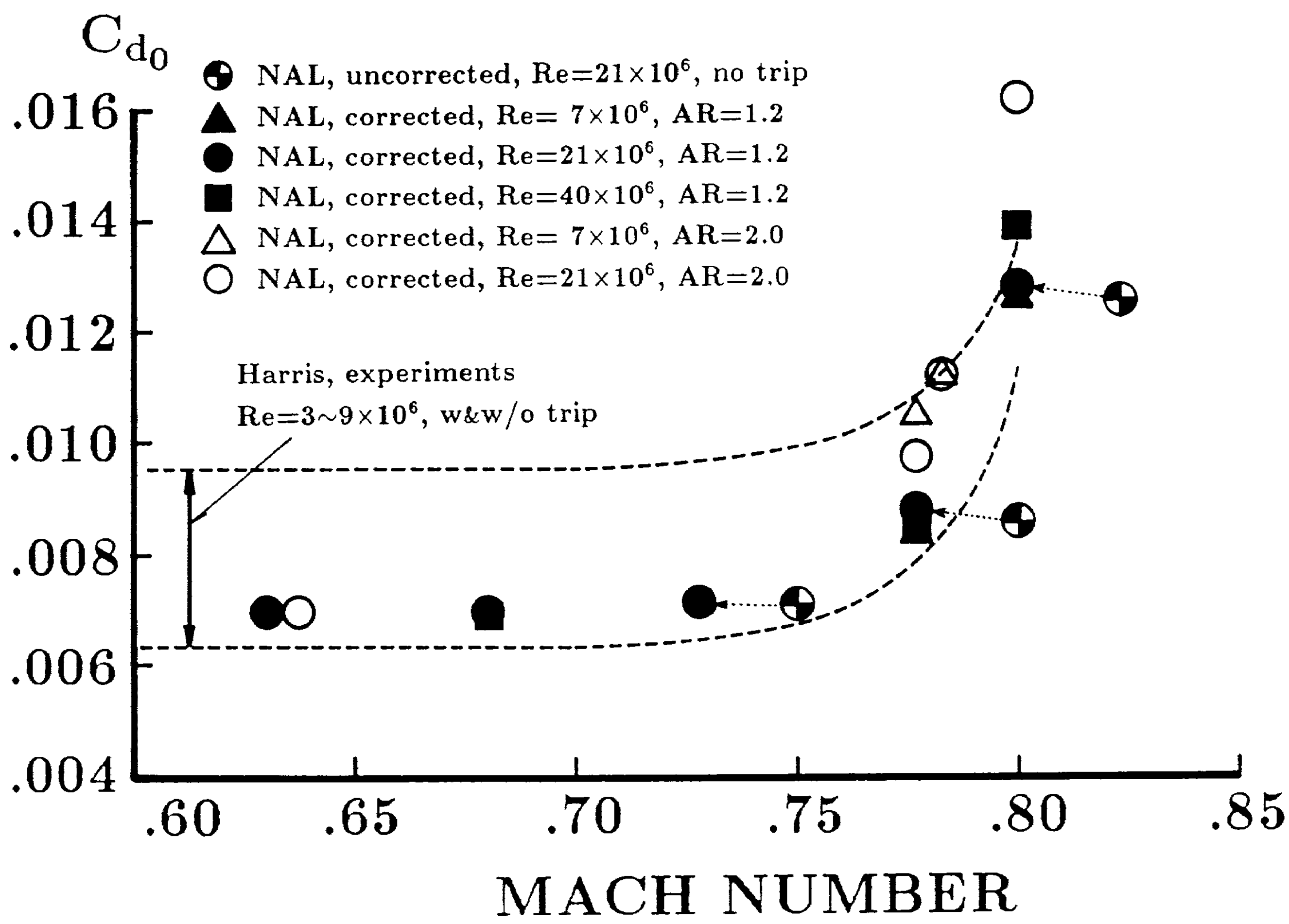

Figure 7. Minimum drag vs. Mach number ; including effects of Reynolds number and aspect ratio

accuracy of the measurement. The further discussion about the accuracy of the drag measurement will be given later.

Figure 7 shows the effects of aspect ratio and Reynolds number on the minimum drag coefficient. This figure also reveals that the NAL's data become close to Harris' by using the correction. Although the effects of Reynolds number (solid symbols) appear above the drag rise due to the difference of the shock location, no evident variation can be recognized. The data on the lower-aspect-ratio model are close to the lower boundary of Harris' results. If the value of the skin friction drag (i.e. the constant below the drag rise) is subtracted, the deviation between the NAL's and Harris' data diminishes. However, when the higher-aspect-ratio model is used, the values of the drag are higher than those of the lower-aspect-ratio model at the drag rise region. Considering that the two-dimensionality of the wake isn't so bad, and that the data on the higher-aspect-ratio model agree with the calculation better than those of the lower-aspect-ratio model, it isn't easy to evaluate which data represent the true value. Further investigation is necessary to evaluate them.

The drag polar curve at M = 0.777 is shown in Figure 8. The NAL experimental results represent the corrected value with the solid circles. The data are in good agreement with the calculation independently of the variation of Reynolds numbers. However, the uncorrected data (open circles) are

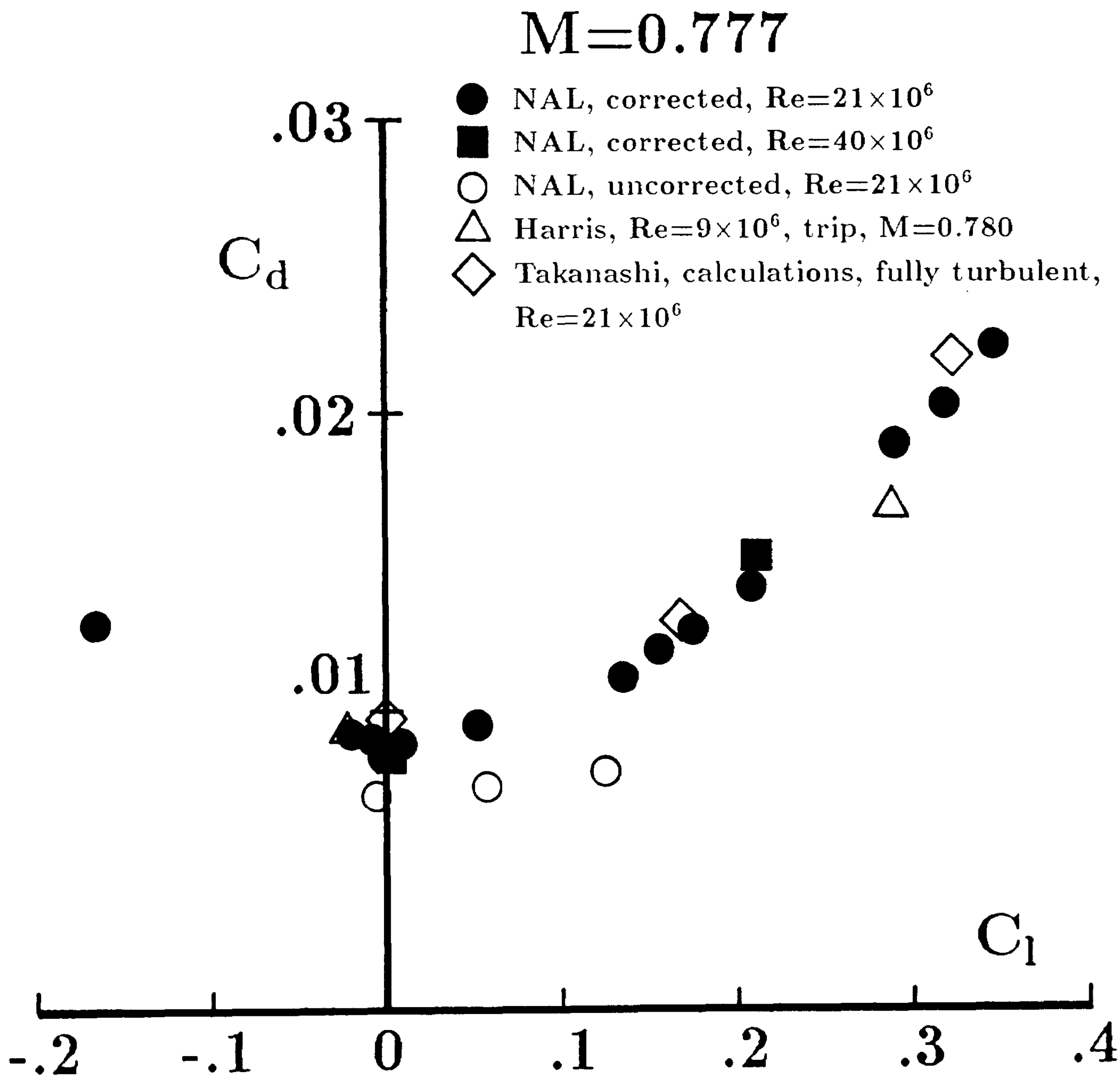

Figure 8. Drag polar curve at M = 0.777

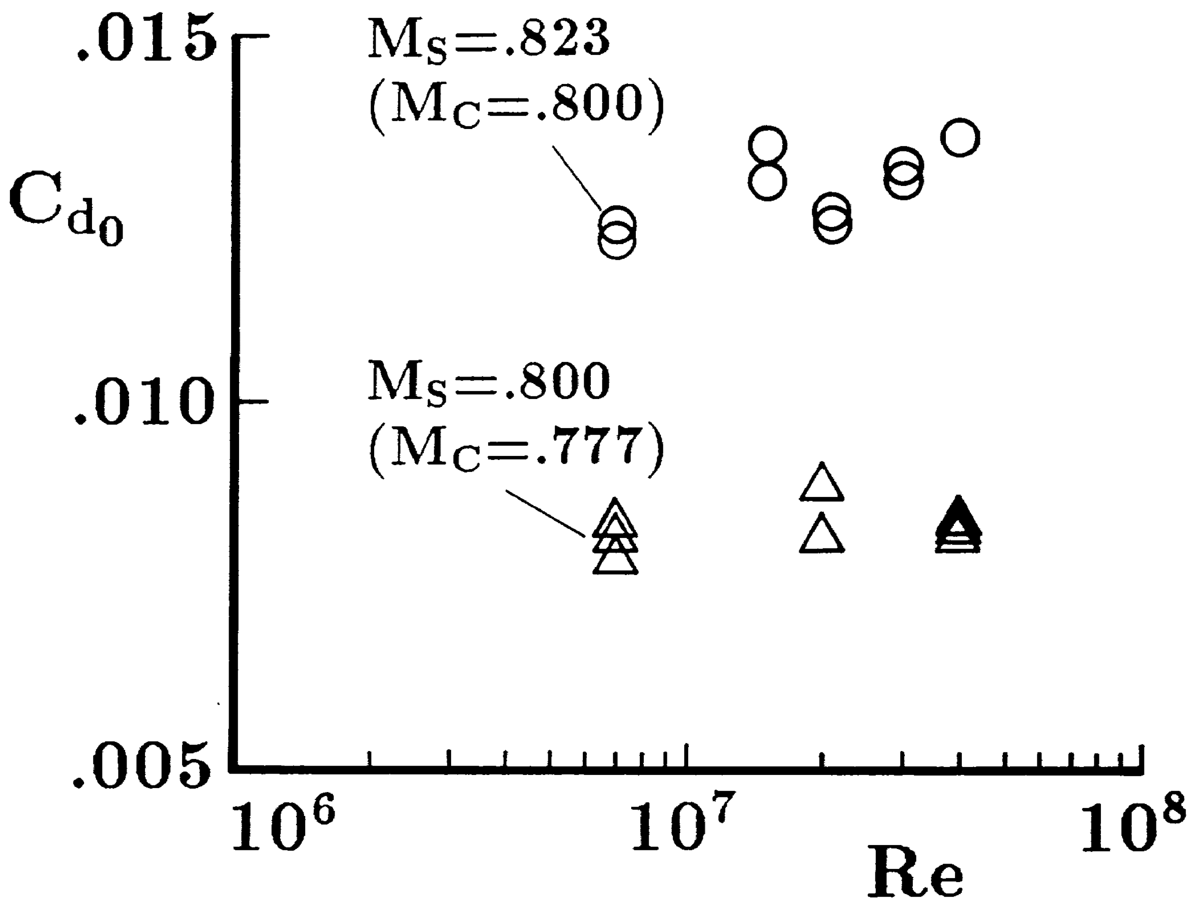

Figure 9. Repeatability of drag measurements in the NAL 2D-TWT

largely out of other data. From this fact, it is also found that the Murthy correction is very promising and that it gives good evaluation to the drag measurement results.

Although there is no direct relation with the discussion of the correction method, the repeatability of the drag measurement is discussed here. As mentioned previously, it is difficult to measure the drag precisely near the drag rise. Therefore, the present tests were conducted carefully so as to minimize the errors in Mach number particularly. As shown in Figure 9, the errors of the drag aren't more than 0.0007 regardless of Reynolds number, and it is satisfactory as a value measured by the method of the wake rake.

Pressure Distributions in Lifting Conditions

The pressure distribution measurements in lifting conditions were made to evaluate applicability of the Murthy correction. The lift coefficient is corrected according to the similarity rule in the same way as the pressure drag coefficient. The comparison of pressure distributions at $M = 0.8$ between the NAL's (solid circles) and Harris' data[9] (open squares) with prescribed normal force coefficient of nearly 0.3 are shown in Figure 10. Except the slight difference of the shock location, they are completely in agreement. Figure 11 shows the comparison between the NAL experimental results (solid circles) and those of the calculation (solid line) at M = 0.777. As recognized in Figures 4 and 5, the suction peak level of the experimental data upstream of the shock on the upper surface tends to be lower than that of the calculation, thereby the accordance of each lift coefficient is not so much significant in this case. Thus the similar distributions are selected. From Figures 10 and 11, it results that the Murthy correction is applicable even to the lifting conditions.

The angles of attack of the experimental data in these figures are setting values. If the value in Figure 11 is corrected by the Sawada method[2] as to the top and bottom walls interference, it becomes the

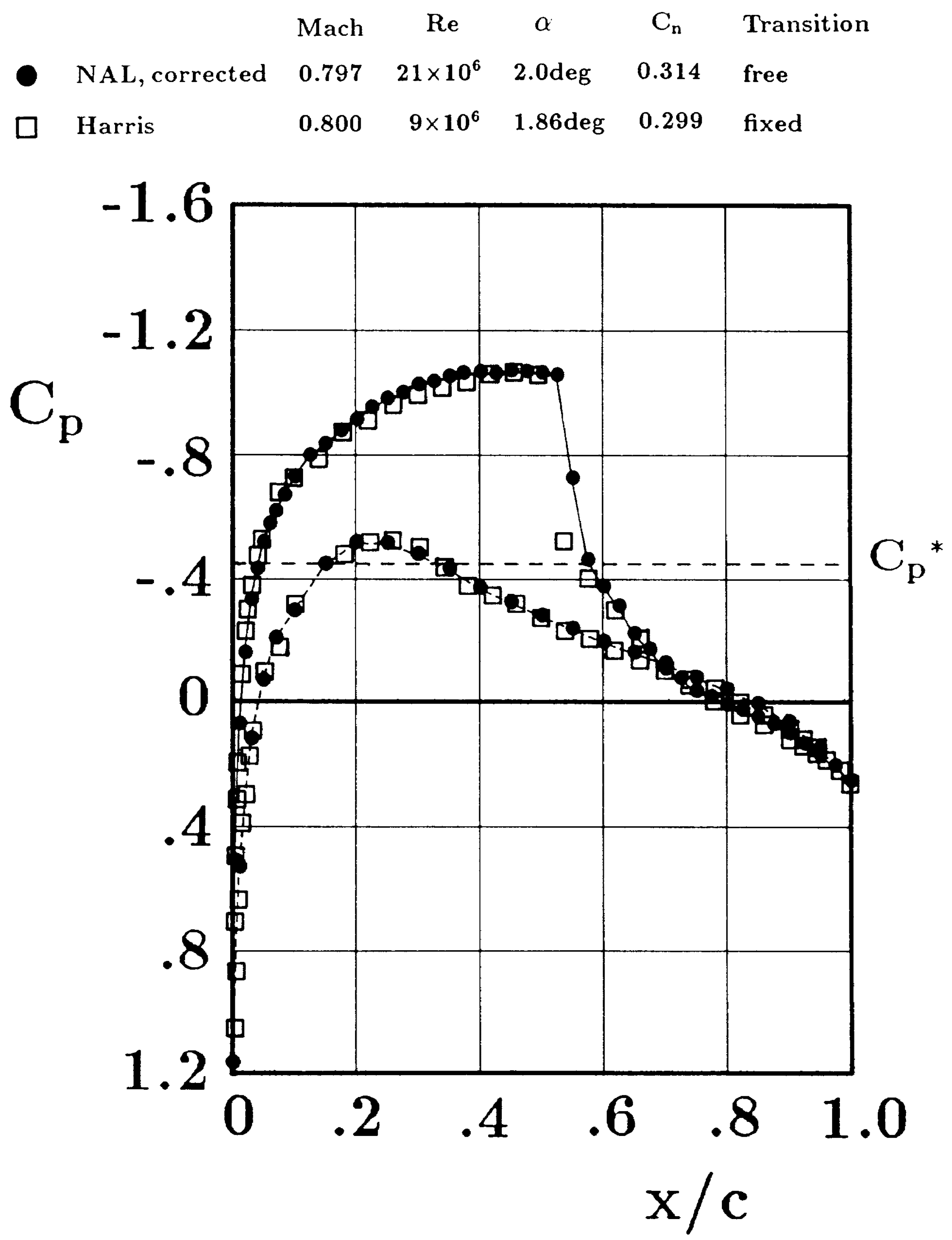

Figure 10. Comparison of pressure distribution with Harris' result (the LaRC 8'TPT) at M = 0.8 and $C_n \simeq 0.3$

value of 1.76°. The deviation between the experimental value and that of the calculation is 0.22°. It is not clear whether this is due to the sidewall interference, or something else. If the question is elucidated, a two-dimensional wind tunnel test corresponding with the free air (i.e. a truly two-dimensional test) will be possible. For that purpose, the comparative study between experiments and CFD must be continued.

5. Concluding Remarks

The tests on the two different chord NACA0012 airfoils (AR = 1.2 and 2.0) were conducted in the NAL Two-Dimensional Transonic Wind Tunnel to investigate the applicability of the Murthy sidewall

	Mach	Re	α	C_l	C_d	Transition
● NAL, corrected	0.775	21×10^6	2.0deg	0.287	0.0202	free
—— Takanashi (calculation)	0.777	21×10^6	1.54deg	0.327	0.0219	fully turbulent

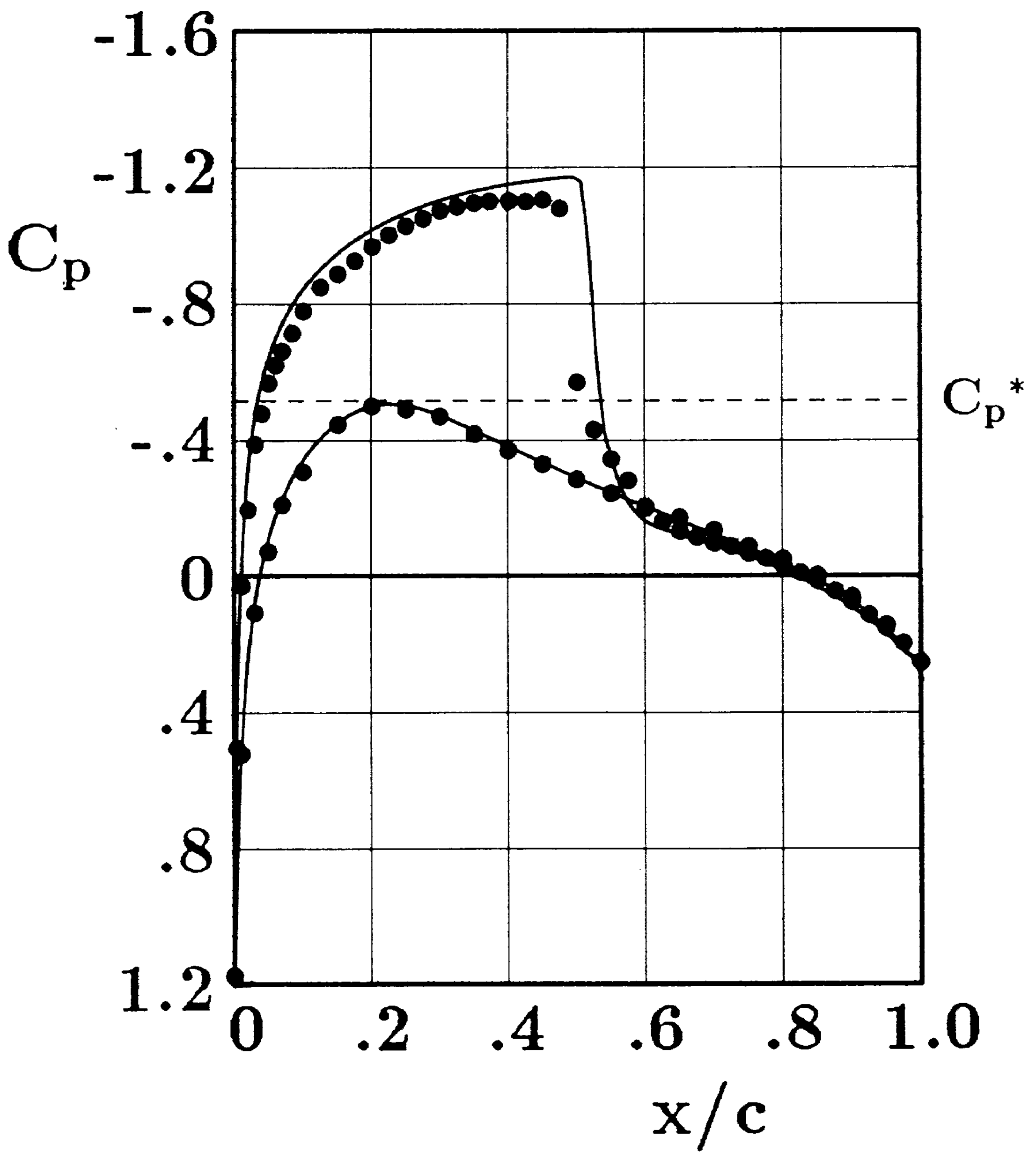

Figure 11. Comparison of pressure distribution in lifting condition with the Navier-Stokes calculation at $M = 0.777$

correction method. It was shown that the corrected data by the Murthy method give an excellent agreement with both the data of other wind tunnels and the Navier-Stokes calculation on the shock wave position, the drag, and the surface pressure distributions. The tests to investigate the effect of the difference in aspect ratio also give a good agreement with them.

Using the Murthy sidewall correction method, the results measured in the NAL 2D-TWT are given reasonable evaluation. Although further experimental data on other airfoils are necessary, this investigation is expected to give useful data for establishing the most suitable sidewall correction and elucidating the subtle differences in measured results among wind tunnels by comparative study with both CFD and other wind tunnels.

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