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Assessment of Mach Number Correction for Sidewall Interference in Transonic Airfoil Testing

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Effects of test section sidewalls have been investigated in a two-dimensional transonic wind tunnel. The facility has recently been modified to improve the flow two dimensionality for airfoil testing. A procedure where two Mach number corrections for top and bottom wall interference and for sidewall interference are combined is applied to data obtained in the modified facility. Corrected data for two different airfoils are compared with free-air computations and other experimental results, and satisfactory agreement is observed when a proper correction for sidewall effects is selected. Sidewall correction methods proposed are then applied to various configurations of airfoil model. It is found that the applicability depends on the ratio of test section height to airfoil chord rather than model aspect ratio for the range of aspect ratio tested. To reduce three-dimensional effects, a device for removing sidewall boundary layers has been installed upstream of the model. The reduction of the boundary-layer thickness by the device, however, has no effect on Mach number correction, and the significant usefulness of the device has not been determined. These results suggest that the optimum experimental arrangement for transonic airfoil testing should be reconsidered for each wind tunnel to provide truly two-dimensional data.

Nomenclature

AR	=	aspect ratio
b	=	airfoil semispan
C_l	=	lift coefficient
C_p	=	pressure coefficient
C_p^*	=	critical pressure coefficient
c	=	airfoil chord
H	=	shape factor of boundary layer on the sidewall
h	=	test section height
k	=	constant, Eq. (2)
k_2	=	constant, Eq. (3)
l	=	length in terms of airfoil chord
M	=	Mach number
p	=	static pressure
p_0	=	total pressure
Re	=	Reynolds number based on airfoil chord
w	=	test section width, $2b$
x	=	streamwise coordinates
y	=	spanwise coordinates
α	=	angle of attack
γ	=	ratio of specific heats
δ^*	=	displacement thickness of boundary layer on the sidewall

Subscripts

c	=	corrected value
F	=	corrected value for four wall effects
TB	=	corrected value for top and bottom wall effects
u	=	uncorrected value
∞	=	freestream condition

Introduction

IN transonic airfoil testing, the assessment of wind-tunnel wall interference has been a major concern to make highly reliable two-dimensional tests. To attain a high Reynolds number flow, a larger chord is preferable. The use of a larger chord model, however, leads to lower aspect ratio and lower ratio of test section height to airfoil chord. In this situation, the interference of wind-tunnel walls changes significantly the effective freestream Mach number and the effective angle of attack.

Because the flow produced in the test section is constrained by the top and bottom walls and the sidewalls, it is very difficult to simulate a flowfield in free air with wind-tunnel experiments. In a strict sense, even if the data obtained in wind tunnels are corrected for wall interference, the results never can present the truly two-dimensional flowfield around the airfoil model tested. Data comparisons using the same airfoil among various wind tunnels, however, help to find solutions for truly two-dimensional flowfields. McCroskey[1] made comparisons of aerodynamic characteristics for the NACA 0012 airfoil measured in dozens of wind tunnels and demonstrated a data scattering of shock position at $M = 0.8$ and $\alpha = 0$ deg. Although the majority of the results seems to lie between $x/c = 0.44$ and 0.48, the data are scattered over the range $x/c = 0.3$–0.55. For the experimental cases examined, the top and bottom walls are considered to have less effect on freestream Mach number, so that effects of sidewalls should be dominant. Nevertheless, even if the data with sidewall boundary-layer suction or those corrected for sidewall interference are selected, the sets of data still lie between $x/c = 0.4$ and 0.52. Therefore, it is very difficult to determine the shock position in truly two-dimensional flow from experimental data only.

Although computational fluid dynamics (CFD) may suggest the truly two-dimensional solution, the results depend on the scheme and turbulence model used. In the case without flow separation (or with small separation), however, selections of a reliable scheme and a turbulence model are expected to result in a smaller scattering of computational results. CFD has recently played an important role in the design of aircraft, and thus, demonstrating truly two-dimensional reliable data in experiments is of critical importance to the validation of CFD codes. In this paper, data obtained in the National Aerospace Laboratory of Japan (NAL) two-dimensional transonic wind tunnel (TWT2)[2,3] have been compared with computational results by

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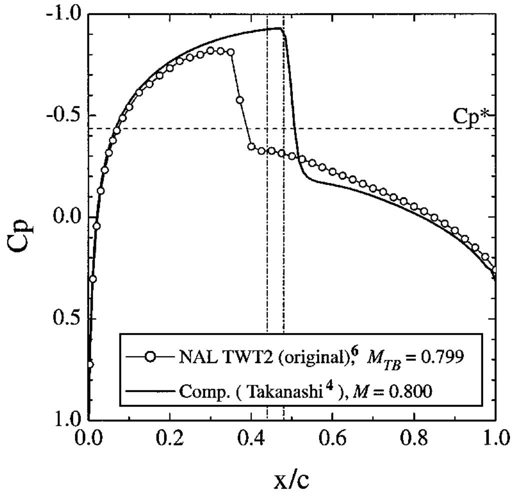


Fig. 1 Pressure distribution on a NACA 0012 airfoil model (model 1) corrected for top and bottom wall interference in the original NAL TWT2: $\alpha = 0$ deg and $Re \simeq 21 \times 10^6$ (case A).

Takanashi[4] and other experimental data. The main objective is to establish a wall interference correction procedure suitable for the NAL facility. Differences in effects of test section walls on angle of attack can be eliminated by comparing data with the same lift coefficient, and thus, only the Mach number correction is focused on in this study.

In the NAL facility, Sawada's method[5] has been used to correct measured data for top and bottom wall interference, and no special care was first taken in terms of sidewall effects. A pressure distribution on a NACA 0012 airfoil model corrected for top and bottom wall interference[6] is shown in Fig. 1. (Chordwise C_p distributions presented here without a note of spanwise location are derived from pressure data measured at the midspan.) The shock wave is located at 39% of the chord, and this location is far upstream of the range between the dash–dotted lines where the majority of experimental results lies as described in Ref. 1. Sudani et al.[7] then compared data for a shockless lifting airfoil (named BGK 1)[8] obtained in the NAL facility with data[9] of the Institute for Aerospace Research (IAR, Canada) Two-Dimensional High Reynolds Number Test Facility[10] (Fig. 2). The data obtained in the NAL and IAR facilities are corrected for top and bottom wall interference by Sawada's[5] and Mokry and Ohman's[11] methods, respectively. It has been confirmed that the two methods give nearly equal corrections to Mach number when they are applied to the same experimental data. Nevertheless, the two data sets indicate slight differences in shock position, and moreover, both data sets are completely different from two-dimensional computations. These results imply that sidewall effects cause the slight discrepancies between the two experimental data sets obtained in the NAL and IAR facilities and the serious differences between the experimental results and computations. When a model with a relatively high aspect ratio is used, the wind-tunnel sidewalls are considered to have little effect on pressure measurements at the midspan. Moreover, many of two-dimensional facilities have a device for sidewall boundary-layer suction. However, the sidewall effects must change the effective Mach number to a greater or lesser degree, so that clarifying the effects and establishing a correction procedure for the sidewall interference are essential, especially at transonic speeds where high accuracy in Mach number should be attained.

To establish a correction procedure for wall interference suitable for the NAL facility, Sudani et al.[6,12] first applied Murthy's sidewall correction[13] to NACA 0012 airfoil data and showed that the shock position was dramatically improved. It was then demonstrated that the application of Murthy's correction to BGK 1 airfoil data removed the discrepancies between the data sets in the NAL and IAR facilities. Furthermore, Sudani et al.[14] determined the minimum aspect ratio to achieve a nearly two-dimensional region at the midspan using flow visualization data by oilflow and liquid crystal methods.

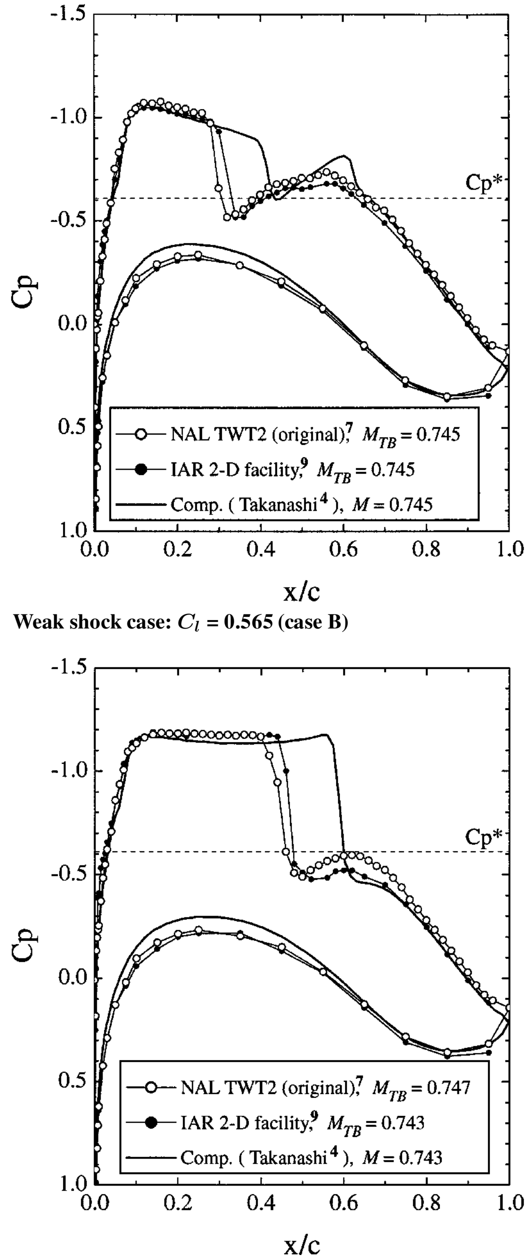


Fig. 2 Pressure distributions on a BGK 1 airfoil model (model 2) corrected for the top and bottom wall interference in the original NAL TWT2: $Re \simeq 21 \times 10^6$.

On the basis of these experimental results, the test section of the NAL facility was modified from 1995 to 1997 to improve the quality of the freestream and the two dimensionality of the flow over the airfoil model.[3] In this paper, major modifications of the NAL facility are first described. A correction procedure for wall interference suitable for the facility is then established, by combining a sidewall interference correction with the present top and bottom wall interference correction. The applicability of sidewall corrections already proposed to data obtained in various experimental arrangements is examined by comparisons with two-dimensional computations. In addition, effects of the sidewall boundary-layer suction on the flow two dimensionality and on Mach number correction are discussed. With the use of these new results, the optimum experimental arrangement of two-dimensional airfoil testing is also reconsidered. Data comparison between the two test sections of the same wind tunnel is expected to be very useful in finding truly two-dimensional solutions and in assessing the validity of wall interference corrections applied.

Table 1 Test sections of the original and modified NAL TWT2 and the IAR two-dimensional facility

Parameter	Facility: NAL TWT2 (original)	NAL TWT2 (modified)	IAR two-dimensional facility[10,15]
Test section			
w, m	0.3	0.45	0.381
h, m	1.0	0.8	1.524
Airfoil model			
c, m	0.25	0.2	0.254[a]
$2b$, m	0.3	0.45	0.381
Æ ($2b/c$)	1.2	2.25	1.5
h/c	4.0	4.0	6.0
Top and bottom walls			
Type	Slotted	Slotted	Porous
Open area ratio/porosity	3%	6%	19.3%[a]
Sidewalls			
δ^*, mm ($M = 0.7$–0.8, $Re \simeq 21 \times 10^6$)	4.7	3.7	3.8
$2\delta^*/w$	0.031	0.016	0.013
Boundary-layer suction (location in respect to airfoil model)	Vicinity	Upstream	Vicinity

[a]Values in a series of tests for a BGK airfoil model.[9]

Modifications of the Facility

The test section of the modified NAL TWT2 (the NAL 0.8 × 0.45 m High Reynolds Number Transonic Wind Tunnel) is illustrated in Fig. 3. Major modifications are as follows[3]: 1) The width of the test section has been increased to make the model aspect ratio higher. 2) A flexible nozzle has been newly installed to raise the freestream Mach number attainable up to 1.4 and to improve the flow quality at low supersonic speeds. 3) A sting-strut support system has been newly installed for three-dimensional models. 4) The diffuser has been telescoped to approach the model easily in the test section. 5) An ejector for sucking sidewall boundary layers effectively has been newly installed. 6) The settling chamber has been modified for reducing the pressure fluctuation in the freestream. 7) A gate valve has been newly installed between the air reservoirs and the settling chamber for our safety when working in the test section. 8) The wind-tunnel control system and the data acquisition system have been renovated.

The details of the test section before and after the modification are described in Table 1, together with those of the IAR facility.[10,15] To use a high aspect ratio model, the width of the test section is expanded to 450 mm, and the chord of airfoil model usually used is changed from 250 to 200 mm. The aspect ratio is nearly doubled, and the two dimensionality is, therefore, expected to improve dramatically. To avoid an enormous increase in mass flow rate of the freestream, however, the height of the test section had to be decreased to 800 mm, so that the ratio h/c could not be increased. The decrease in chord leads to the decrease in Reynolds number attainable, but this is not considered to deteriorate the performance of the facility significantly. The freestream Mach number can be varied from 0.2 to 1.4, and the Reynolds number based on airfoil chord ($c = 200$ mm) can be varied from 6×10^6 to 32×10^6 at $M = 0.8$. The duration for the maximum Reynolds number is approximately 6 s, during which pressure and wake drag measurements are possible at an angle of attack.

A series of tests for initial calibration measurements of freestream Mach number distributions has been carried out in the empty tunnel. Static pressure distributions were measured using pressure rails installed on the top and bottom walls, and local Mach numbers were calculated. Distributions of the Mach number deviation from the intended value ($M = 0.8$) before and after the modification of the facility are shown in Fig. 4. Before the modification, the flow accelerates abruptly from the model location. Measurements of total pressure deficit in the wake for calculating section drag coefficient had been performed using a traversing rake at $x/c = 3.0$, and the flow acceleration is no less than 0.008 at the location of the wake measurements. Moreover, the sharp deviation of Mach number in the vicinity of the model is thought to decrease the reliability of wall

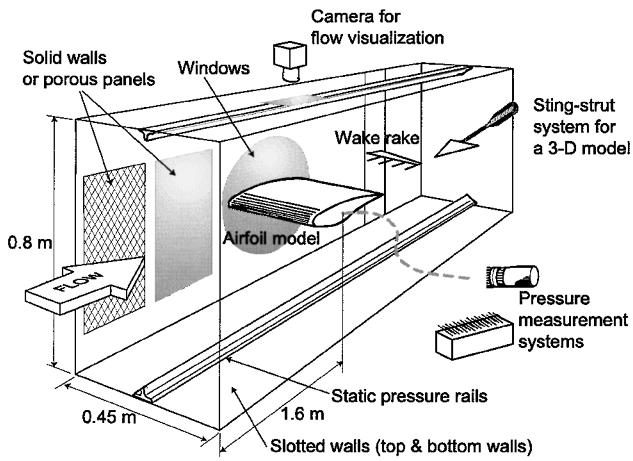


Fig. 3 Schematic of test section of the modified NAL TWT2 (NAL 0.8 × 0.45 m High Reynolds Number Transonic Wind Tunnel).

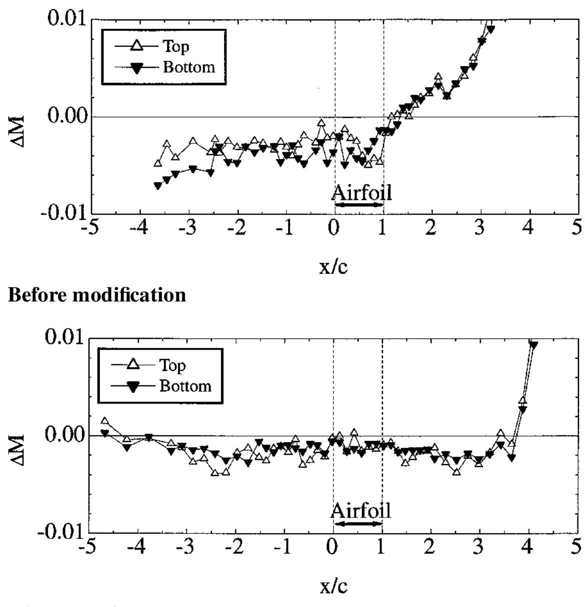


Fig. 4 Local Mach number distributions on the top and bottom walls in the empty tunnel before and after the facility modification: $M_\infty = 0.8$ and $Re \simeq 21 \times 10^6$.

interference corrections. After the modification, the uniformity of Mach number distribution has been improved dramatically by elaborate adjustments of the wall inclination. In a wide region including the location of the wake measurements ($x/c = 3.0$), the freestream produced in the modified facility is of satisfactory quality.

Experimental Setup and Numerical Procedure

Airfoil Models, Measurements, and Test Conditions

Six models in Table 2 were used to measure pressure distributions on the airfoil surface. Two airfoils were selected, a conventional airfoil (NACA 0012) and a shockless lifting airfoil (BGK 1) designed for shockless flow at $M = 0.75$ and $C_l = 0.63$. The uncorrected freestream Mach number was derived from pressures measured in the settling chamber and the plenum chamber. Pressure distributions on airfoil models were acquired by measuring pressure differences between the model surface and the plenum chamber using a pressure scanning system. Uncertainties for M_u and C_p are estimated at $\pm 0.1\%$ and ± 0.01, respectively.

Experiments were carried out mainly for the three test conditions in Table 3. Experimental data in case A in other facilities can be obtained in Ref. 1 for the comparison of shock position. Cases B and C were selected from test conditions where a series of BGK 1 airfoil tests in the IAR facility and pressure distributions corrected for the top and bottom wall effects can be obtained from Ref. 9. In the present series of tests in the modified NAL TWT2, a device for sidewall boundary-layer suction, consisting of two porous plates on each sidewall (Fig. 3.), was not activated except for experiments to investigate effects of the device. (Results are discussed in section "Sidewall Boundary Layer Suction.") The downstream porous plates, however, were installed in the test section throughout the series, and the roughness effects increased the boundary-layer displacement thickness at the model location from 3.7 to 4.2 mm. (Values of the sidewall boundary-layer thickness were derived from the Preston tube measurement in the empty tunnel.)

Numerical Procedure

All computations presented were performed using a two-dimensional code developed by Takanashi.[4] The governing equations are the Reynolds averaged thin-layer Navier–Stokes equations, and fully turbulent flows are simulated by means of the Baldwin–Lomax turbulence model. The computational scheme is a MUSCL-type higher-order total variation diminishing finite difference method, which is based on Roe's flux difference splitting method. A C-type grid system is used, which produces good resolution for computing the wake. To remove the dependence of the numerical solution on the number of cells, both the number of cells in the flow direction and that in the direction perpendicular to the airfoil surface are doubled until airfoil aerodynamic characteristics computed indicate no change. As shown in Fig. 1, the shock wave computed tends to be located slightly downstream of the range where the majority of experimental results lies. However, this code was validated by elaborate comparisons with theoretical values for a supercritical airfoil[16] and is expected to provide reliable data for discussing truly two-dimensional solutions.

Table 2 Airfoil models

Model	Airfoil	c, mm	$2b$, mm	Wind tunnel
1	NACA 0012	250	300	Original
2	BGK 1	250	300	Original
3	NACA 0012	200	450	Modified
4	BGK 1	200	450	Modified
5	BGK 1	250	450	Modified
6	NACA 0012	150	450	Modified

Table 3 Experimental conditions

Case	Airfoil	M	C_l	Re
A	NACA 0012	0.8	0	21×10^6
B	BGK 1	0.745	0.565	21×10^6
C	BGK 1	0.743	0.734	21×10^6

Mach Number Correction

To correct data obtained in the modified NAL TWT2 for top and bottom wall interference, Sawada's[5] method is applied, and a corrected Mach number M_{TB} is obtained. This method is based on subcritical linearized flow theory and calculates corrections to aerodynamic characteristics using static pressure distributions measured on the top and bottom walls. As mentioned in the Introduction, the method was applied to some of data sets[17] obtained in the IAR facility for comparison with Mokry and Ohman's[11] method, which is usually used in the facility. The comparison showed that the two methods provided nearly equal corrections in terms of Mach number.

In the next procedure, a sidewall correction method (Murthy's[13] or Barnwell[18]–Sewall's[19] method) is applied to Mach number M_{TB}. The correction methods are based on the negative blockage effect caused by the decrease in boundary-layer thickness at the model location due to a favorable pressure gradient induced by the airfoil model. A corrected Mach number M_F for all four wall interference is given by

$$\frac{1 - M_{\mathrm{TB}}^2 + k}{M_{\mathrm{TB}}{}^{\frac{4}{3}}} = \frac{1 - M_F^2}{M_F{}^{\frac{4}{3}}} \tag{1}$$

where

$$k = (\delta^*/b)\left(2 + 1\big/H - M_{\mathrm{TB}}^2\right)(k_2/\sinh k_2) \tag{2}$$

$$k_2 = 2\pi b\sqrt{1 - M_{\mathrm{TB}}^2}\Big/l \tag{3}$$

Because the correction $(M_c - M_u)$ by Sawada's[5] method or the sidewall corrections is insensitive to the slight change in Mach number applied, M_u, the reversed procedure results in nearly equal corrected values, M_F.

For the sidewall boundary-layer characteristics δ^* and H, measured values at the model location in the empty tunnel are used. The constant k_2 depends on Mach number, airfoil span, and a length scale l representing the model chord. Therefore, this constant includes the effect of model aspect ratio. For $k_2 > 0$, Eqs. (1–3) represent Murthy's[13] correction. For $k_2 = 0$ ($l \to \infty$), the correction is identical to Barnwell[18]–Sewall's[19] correction, which is independent of airfoil aspect ratio. In Ref. 13, two assumptions of $l = c$ and $2c$ were investigated, and these are based on that the effects of the airfoil model on the sidewall boundary layer are distributed over a distance of one and two chord lengths, respectively.

The pressure coefficient should be also corrected to account for the changes in dynamic pressure and static pressure associated with the Mach number correction as follows:

$$p_{\infty,c} = p_0\left\{1 + [(\gamma - 1)/2]M_F^2\right\}^{-\gamma/(\gamma - 1)} \tag{4}$$

$$C_{p,c} = \left(2\big/\gamma p_{\infty,c} M_F^2\right)(p - p_{\infty,c}) \tag{5}$$

The lift coefficient is then recalculated using corrected pressure coefficients.

Results and Discussion

Correction for Four Wall Effects

Establishing a correction procedure suitable for the modified NAL TWT2 is first attempted using NACA 0012 airfoil data at zero lift coefficient around a Mach number of 0.8. The selection of this airfoil and this test condition is the most appropriate for the assessment of Mach number correction because the chordwise location x/c of the shock wave increases almost linearly with increasing freestream Mach number. Furthermore, because of the zero lift coefficient, differences in the top and bottom wall effects are of insignificance.

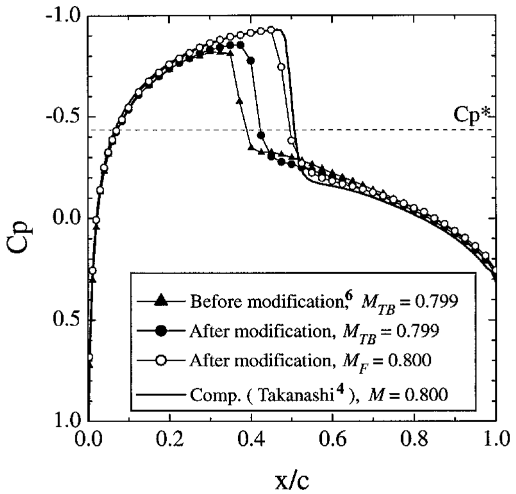


Fig. 5 Pressure distributions on NACA 0012 airfoil models (models 1 and 3) corrected for wall interference: α = 0 deg and $Re \simeq 21 \times 10^6$ (case A).

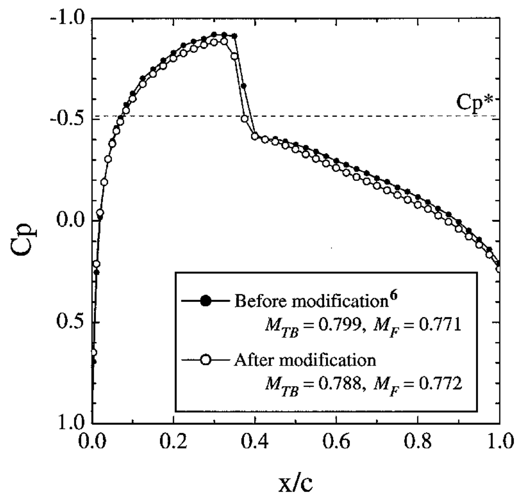


Fig. 6 Pressure distributions on NACA 0012 airfoil models (models 1 and 3) corrected for all four walls interference before and after the facility modification: α = 0 deg and $Re \simeq 21 \times 10^6$.

As shown in Fig. 5 (or also Fig. 1), the data corrected only for top and bottom wall interference before the facility modification are completely different from the computation. After the modification, the wall pressure distribution in the empty tunnel has been dramatically improved, as already shown in Fig. 4. Moreover, because the aspect ratio of the airfoil model usually used has been much higher, sidewall effects would be negligibly small, and the shock wave has been expected to shift downstream to a location close to that of the computation. The shock wave location, however, is moved slightly downstream (closed circles in Fig. 5), and the improvement is not satisfactorily sufficient. The result indicates that the sidewall interference still affects strongly pressure measurements at the midspan even for an aspect ratio of 2.25.

The application of a sidewall correction is then attempted. It is accepted that, for a relatively high aspect ratio, applying Barnwell[18]–Sewall's[19] correction, which includes no effect of aspect ratio, tends to yield an overcorrected value, and that Murthy's[13] correction instead should be promising. In the present situation ($\text{Æ} = 2.25$), Murthy's correction with $l = 2c$ has been expected to be the best because of high aspect ratio. However, the application of this correction to the NACA 0012 data gives a considerably conservative value. (The result is not plotted.) The applicability of the corrections needs to be reconsidered.

Contrary to expectation, applying Barnwell[18]–Sewall's[19] correction to the NACA 0012 airfoil data indicates the best agreement between the experimental and computational results (open circles). Barnwell–Sewall's correction is derived under the assumption of the linear variation of the crossflow velocity across the width of the tunnel. This assumption is more accurate when the airfoil aspect ratio is sufficiently low.[20] When a high aspect ratio is used and Murthy's[13] correction is applied, on the other hand, the increase in l in the correction procedure improves the accuracy of the linear crossflow assumption. Accordingly, Murthy's correction with a greater l is considered to be also applicable to the data after the facility modification. This means that the effects of the airfoil model on the sidewall boundary layer are distributed over a distance much longer than that assumed by Murthy. Sudani et al.[6] applied Murthy's[13] correction with $l = 2c$ to data with $\text{Æ} = 1.2$ obtained in the original facility and mentioned that the correction was very promising. The reexamination of the results, however, has showed that an assumption of $l = 2c$ gives a significantly conservative correction.[6] In Fig. 6, the data obtained in the original facility are again corrected with Barnwell[18]–Sewall's[19] method and compared with the corrected data in the modified facility. A good agreement on shock position is observed between the two distributions. This comparison demonstrates that the experimental data in Fig. 1 are those at a freestream Mach number of 0.771–0.772. Because Barnwell-Sewall's correction is independent of airfoil aspect ratio, the difference in sidewall correction ($M_{\text{TB}} - M_F$) between the data sets obtained in the original and modified facility is made by that in $\delta^*/2b$, that is, the boundary-layer displacement thickness to tunnel width. The value of $\delta^*/2b$ before the modification was much larger than after the modification. This is the reason why the shock wave measured in the original facility without sidewall interference correction is located far upstream, compared with those of other wind tunnels, as presented in Fig. 1.

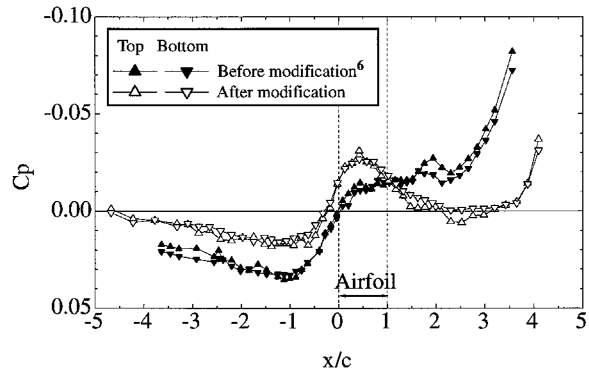
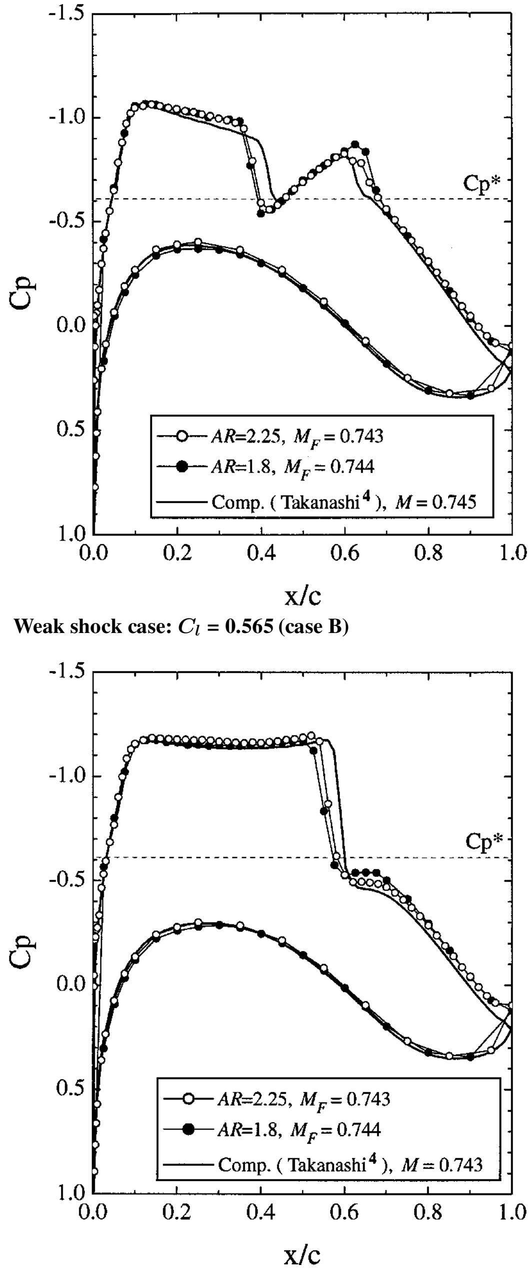


Fig. 7 Comparison of top and bottom wall pressure distribution with a NACA 0012 airfoil model installed in the test section between before and after the facility modification: $M_F \simeq 0.772$, α = 0 deg, and $Re \simeq 21 \times 10^6$.

Figure 6 also indicates that there are slight differences in pressure level from upstream of the shock to the trailing edge. This is probably due to flow acceleration in the vicinity of the airfoil model. The detailed data calculated in the procedure of the top and bottom wall interference correction demonstrate that the freestream is sharply accelerated at the model location in the original facility. This can be also seen from top and bottom wall pressure distributions, as shown in Fig. 7. With the modified facility, the pressure coefficient decreases abruptly from $x/c = -0.5$ and increases back to zero at $x/c = 2.5$. With the original facility, on the other hand, the pressure coefficient does not increases back downstream of the airfoil and decreases further sharply from $x/c = 2.5$. A great improvement to wall pressure distribution with an airfoil model installed in the test section has been produced by the modification of the facility and leads to a more reliable correction for top and bottom wall interference.

To investigate the applicability of Barnwell[18]–Sewall's[19] correction to high-lift cases, the correction is applied to BGK 1 data obtained in the modified facility. Figure 8 shows pressure distributions on different chord models for cases B and C. Compared with Fig. 2, great improvements are shown in shock position and pressure level on the lower surface. The main shock waves for both cases, however,



Weak shock case: C_l = 0.565 (case B)

Strong shock case: C_l = 0.743 (case C)

Fig. 8 Pressure distributions on BGK 1 airfoil models (models 4 and 5) with different chord lengths corrected for all four wall effects: $Re \simeq 21 \times 10^6$.

are still located upstream of those of the computations. This indicates that the application of Murthy's[13] correction with $l = 2c$ would never result in satisfactory agreement between the experimental and computational C_p distributions. Barnwell[18]–Sewall's[19] correction is also proven to be more applicable to high-lift cases, and this means that the sidewall effects are not significantly related to airfoil aspect ratio for the experimental arrangement adopted in the modified facility. This conclusion is also supported by the experimental fact that the data corrected by Barnwell-Sewall's method for two different chords demonstrate good agreement.

A close observation of Fig. 8 reveals slight differences between the two sets of data for different chords. For the larger c, the flow tends to accelerate downstream of the main shock. Especially in case B, the secondary shock seems to be stronger and be located more downstream than that for the smaller c or the computation. Figure 9 shows that the pressure on the top wall decreases more quickly and does not increase back sufficiently for the larger c (closed triangles). In the procedure of Sawada's[5] correction,

Table 4 Mach number correction for sidewall effects in the NAL TWT2: $M_u \simeq 0.8$

		Correction	
Wind tunnel	Æ	Barnwell[18]–Sewall[19]	Murthy[13] ($l = 2c$)
Original	1.2	−0.028[a]	−0.023
Modified	1.8	−0.016[a]	−0.010
Original	2.0	−0.028	−0.017[a]
Modified	2.25	−0.016[a]	−0.008
Modified	3.0	−0.016	−0.006

[a]Better correction for each Æ.

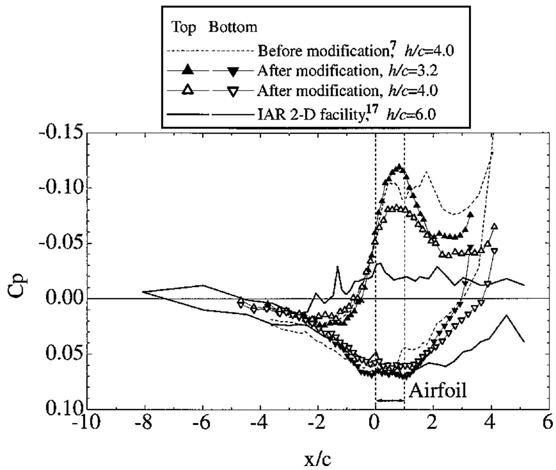


Fig. 9 Comparison of top and bottom wall pressure distributions with a BGK 1 airfoil model installed in the test section: $M_{TB} \simeq 0.745$, C_l = 0.565, and $Re \simeq 21 \times 10^6$ (case B).

therefore, the rate of increase in local freestream Mach number is considerably higher. Thus, the differences are probably attributed to that the flow for the larger c accelerates crucially in the model location.

In Fig. 9, a significant improvement to wall pressure distribution by the facility modification is clearly seen when the smaller c is used (open triangles), as in the case with the NACA 0012 airfoil models. Although the pressure deviation on the walls from the static value is not so small as that of the IAR facility, a Mach number correction by Sawada's[5] method should become more reliable by the improvement. Furthermore, the application of Barnwell[18]–Sewall's[19] correction is also promising for such high-lift cases. A correction procedure for four-wall interference has been satisfactorily established for the modified NAL TWT2.

Applicability of Corrections

After the facility modification, it had been predicted that the shock position for case A would shift significantly downstream even without any sidewall correction because the aspect ratio used was nearly doubled and because the displacement thickness of sidewall boundary layers was reduced. Even if a sidewall correction had been needed, Murthy's[13] correction should have been appropriate because it is considered to be applicable to high aspect ratio. However, the application of the correction was insufficient to resolve the disagreement between experimental and computational results. Barnwell[18]–Sewall's[19] correction instead is more applicable, as discussed in the preceding section.

To examine the applicability, both data sets obtained in the original and modified facility have been analyzed again. Mach number correction by Murthy's[13] and Barnwell[18]–Sewall's[19] methods for various aspect ratios are given in Table 4. For Æ = 1.2, 1.8, and 2.25, the Barnwell–Sewall correction is more applicable, as already demonstrated in Figs. 6 and 8. From these results, it can be expected that the suitable correction for Æ = 2.0 should also be the same correction. However, Ref. 6 indicates that Barnwell–Sewall's method overcorrects the data for Æ $= 2.0$ and that the data corrected by Murthy's[13] method with $l = 2c$ are in good agreement with computations. It had been believed that the applicability of the two corrections simply depends on airfoil aspect ratio only. The experimental results, however, imply that the applicability of the corrections has no significant relation to airfoil aspect ratio for the range of aspect ratio tested and that the applicability depends on another parameter. An experiment using an airfoil with much higher aspect ratio was then attempted to support the inference, and results are shown in Fig. 10. Data at $M_{\mathrm{TB}} = 0.813$ show the best agreement with the computation. A correction of -0.013 is derived from Murthy's method with $l = 4c$. In this case, Barnwell[18]–Sewall's[19] method would overcorrect the data and locate the shock downstream of the computation. Furthermore, despite a considerably high aspect ratio of 3, Murthy's[13] correction with $l = 2c$ is not sufficient as demonstrated by closed circles.

A hypothesis is then proposed that the constraint of top and bottom walls affects the development of sidewall boundary layers unless the walls are located sufficiently apart from the model. Figure 11 shows experimental points plotted against Æ and h/c, indicating which correction is the most applicable. The IAR facility data [BGK 1 (Ref. 9) and NACA 0012 (Ref. 21) airfoil data] are assessed using measured characteristics of sidewall boundary layers,[15] and the data in the NASA Langley Research Center 0.3-m Transonic Cryogenic Tunnel (0.3-m TCT) are assessed from the discussion in Ref. 13. It can be seen that Barnwell[18]–Sewall's[19] correction is more applicable to $h/c \leq 4.0$. Murthy's[13] correction with $l = 2c$, on the other hand, is more applicable to $h/c \geq 6.0$. When the ratio h/c is not sufficiently high, the influence of the airfoil model on sidewall boundary layers

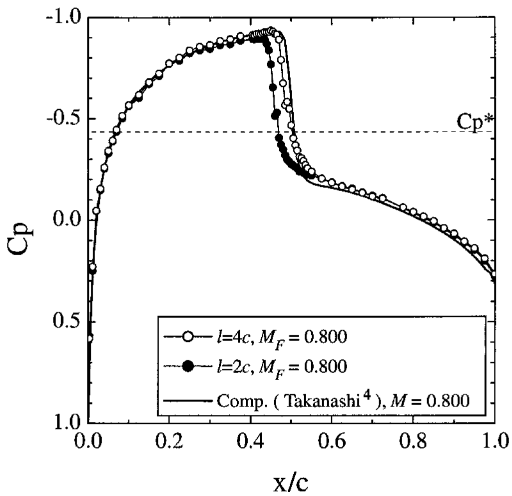


Fig. 10 Pressure distributions on a NACA 0012 airfoil model (model 6) corrected by Murthy's[13] method: $\alpha = 0$ deg and $Re \simeq 21 \times 10^6$ (case A).

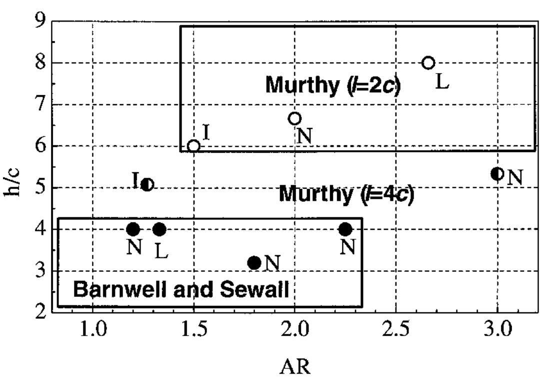


Fig. 11 Applicability of sidewall interference corrections to data with various aspect ratios and ratios of tunnel height to airfoil chord: N, data in the original and modified NAL TWT2; I, data[9,21] in the IAR two-dimensional facility; L, data[13] in the Langley 0.3-m TCT facility.

may be heightened by the severe constraint of the top and bottom walls. As mentioned before, the effects of the airfoil model on the sidewall boundary layer are distributed over a distance much longer than that assumed by Murthy for a small h/c. For $4 < h/c < 6$, Murthy's correction with $l = 4c$ seems to be the most applicable irrespective of Æ. It can be seen from all of the data that the effects of sidewalls are increased with decreasing h/c and that the vertical variation in δ^* must be considered in the correction procedure for higher accuracy. To clarify effects of h/c on the development of sidewall boundary layers, that is, the relation between the constraint of the top and bottom walls and the sidewall interference, three-dimensional computations or boundary-layer measurements over a wide range of the sidewalls are strongly suggested.

The sidewall correction methods discussed might be somewhat simple because they assume a one-dimensional growth of sidewall boundary layers. However, the selection of a proper correction method and the use of relatively high aspect ratio and h/c can produce data with reliable accuracy in transonic airfoil testing as the first step to seek truly two-dimensional solutions.

Sidewall Boundary-Layer Suction

The original NAL TWT2 was equipped with a device for removing sidewall boundary layers. Porous plates were mounted in the vicinity of the model, and the diameter of the plates was approximately twice the airfoil chord. The device, however, had usually not been activated because it caused serious Mach number nonuniformity at the model location.[22] Flow visualization studies[14] performed in the original facility suggested that the suction upstream of the model should be effective to extend the two-dimensional region. The suction device has been moved upstream (from $x/c = -4.1$ to -2.1) as one of the major modifications of the facility. A dramatic improvement to Mach number distribution in the vicinity of the airfoil model, is described in Ref. 22, although the Mach number is slightly decreased at the location of the wake rake ($x/c = 3.0$).

Pressure distributions at different spanwise locations for a NACA 0012 airfoil model (model 3) have been examined with and without the sidewall boundary-layer suction (Fig. 12). Without the suction, at $y/b = 0.5$, that is, a location one-quarter span from the sidewall, the shock position and the pressure level upstream of the shock agree with those at the midspan. At a location closer to the sidewall ($y/b = 0.75$), the pressure is increased slightly just upstream of the shock. The shock is thought to be curved upstream near the sidewall. In the central region, however, a nearly two-dimensional region is attained to a satisfactory extent. With the suction through the downstream porous plates only (Fig. 3.), the pressure distribution at $y/b = 0.75$ demonstrates a significant improvement to the two dimensionality. Furthermore, it has been confirmed that the displacement thickness of the sidewall boundary layer is reduced by approximately 40% with the suction. Nevertheless, the shock position measured at the midspan has no change. According to Barnwell[18]–Sewall's[19] or Murthy's[13] sidewall correction, the reduction in δ^* of such an amount leads to a smaller correction to Mach number, and thus, the shock should be moved downstream. These experimental results imply that the local change in δ^* in the test section has no effect on Mach number correction and support the inference that the effects of the airfoil model on the sidewall boundary layer are distributed over a distance considerably longer than expected.

Reconsideration of Airfoil Testing Techniques

Aspect Ratio and Airfoil Chord

To avoid sidewall effects, a higher aspect ratio is preferable. An aspect ratio of 2 or more was recommended by flow visualization studies.[14] For a high aspect ratio, however, the airfoil chord should be small, and this makes the attainable Reynolds number lower. Even if a Reynolds number required is achieved with a small chord model, the reservoir pressure needs to be higher, and the duration becomes shorter (for a blowdown facility). Furthermore, it becomes difficult for the manufactured model to have adequate strength and coordinates with necessary accuracy, especially for a thin airfoil.

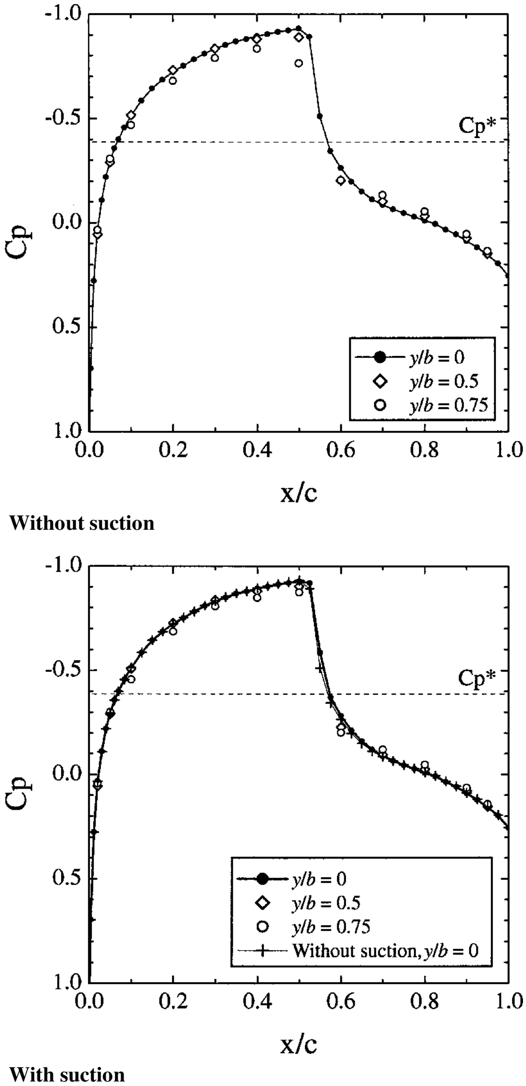


Fig. 12 Pressure distributions on a NACA 0012 airfoil model (model 3) with and without sidewall boundary-layer suction: M_u = 0.833, α = 0 deg, and $Re \simeq 21 \times 10^6$.

A chord of 150 mm or more is recommended from our experience because the accuracy needed can be easily attained at a moderate cost of manufacture. In the modified NAL TWT2, a chord of 200 mm and an aspect ratio of 2.25 ($2b = 450$ mm) has been selected for usual use.

Ratio of Test Section Height to Airfoil Chord

When the NAL facility was modified, the tunnel height had to be reduced to 800 mm to maintain the same duration as that before the modification. The determination of this tunnel height results in the same h/c and is not expected to weaken the top and bottom wall interference. Figure 9 shows that there is an abrupt change in wall pressure compared with the IAR facility ($h/c = 6$). The local freestream Mach number calculated in the procedure of Sawada's[5] correction increases in the flow direction at the model location, whereas it remains nearly constant for $h/c = 6.67$ (Ref. 12). Thus, the application of top and bottom wall corrections to data for a higher h/c can provide more reliable data of corrected Mach number. Furthermore, as discussed in section "Applicability of Corrections," the sidewall effects is significantly reduced for $h/c \geq 6$. From these considerations, it is probable that the present value of h/c (=4 with $c = 200$ mm) is insufficient to avoid strong wall interference, and a ratio around 6 is recommended when wall interference is excessively severe.

Sidewall Boundary-Layer Suction

A device for removing sidewall boundary layers has been expected to improve the flow two dimensionality over the airfoil model. In the NAL facility, no significant changes have been actually observed in shock position and drag coefficient at the midspan.[22] For a low aspect ratio model, slight extension of the two-dimensional region may be achieved as shown by the oil-flow visualization studies.[14] The use of the suction device, however, causes the nonuniformity of freestream Mach number and the inapplicability of schlieren pictures. Taking into consideration the disadvantages, one can not conclude at present that a suction device is necessary for two-dimensional airfoil testing if the aspect ratio used is relatively high. If needed, a device for thinning sidewall boundary layers should be installed farther upstream, probably at the entry of the test section as in the IAR facility.[10] Because it is impossible to remove sidewall boundary layers completely, the effects must be corrected to a greater or lesser degree. The sidewall boundary-layer suction would make the data assessment more complicated. To reach a decision about the validity of sidewall boundary-layer suction, further investigations are needed in a wide range of test conditions and for a variety of device configurations (suction location, suction amount, etc.).

Conclusions

The NAL two-dimensional facility has been modified to improve the two dimensionality and the flow quality for airfoil testing. Experiments using two different airfoils have been conducted to investigate effects of wind-tunnel sidewalls.

A procedure where Sawada's[5] top and bottom wall interference correction and Murthy's[13] or Barnwell[18]–Sewall's[19] sidewall correction are combined is applied to data for NACA 0012 and BGK 1 airfoil models obtained in the modified NAL TWT2. The data corrected for four wall effects are in good agreement with two-dimensional computations and other experimental data, and a correction procedure to Mach number is established with reliable accuracy. For the experimental arrangement usually adopted in the modified facility, Barnwell–Sewall's correction is more applicable than Murthy's[13] correction including aspect ratio effects. This implies that the effects of the airfoil model on the sidewall boundary layer are distributed over a distance much longer than that assumed by Murthy. It is found that the applicability of sidewall corrections depends on the ratio of tunnel height to airfoil chord rather than airfoil aspect ratio for the range of aspect ratio tested. The sidewall effects are presumably heightened by the severe constraint of the top and bottom walls when the ratio h/c is not sufficiently high and are undesirably increased with decreasing h/c. Although the aspect ratio adopted in the modified facility is sufficient for achieving two-dimensional flow around the midspan, it is probable that the ratio of tunnel height to airfoil chord is insufficient to avoid strong wall interference. A proper airfoil chord must be chosen for each experiment. The optimum configuration of an airfoil model for each facility should be reexamined to obtain experimental data that are considered close to truly two-dimensional data.

A device for removing sidewall boundary layers has been installed upstream of the airfoil model in the modified NAL TWT2. The movement of the suction area upstream of the airfoil model leads to the satisfactory uniformity of freestream Mach number in the model location, although the Mach number is slightly decreased at the wake measurement location. No significant improvement, however, is produced to shock position measured at the midspan. This means that the local change in boundary-layer thickness in the test section has no effect on Mach number correction. The disadvantages of the Mach number nonuniformity and the inapplicability of flow visualization suggest that the use of a suction device should be reconsidered.

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