Aerodynamic Characteristics of Busemann Biplane by Wake Measurements in Low-Speed Wind Tunnel†

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In this study, the basic aerodynamic characteristics of a Busemann biplane model in a low-speed wind tunnel were clarified by the wake survey technique. A low-speed wind tunnel of the blowdown type was used, and it had an exit nozzle size is in 1.5 m × 1.5 m. A wake measurement system and six-component balance system were installed in the test section. A five-hole probe was used in the wake measurement system to measure the velocity distribution. The Busemann biplane model was designed to cruise at Mach 1.7. The total length of the model was 742 mm, and the width was 556 mm. The shape of the fuselage was 76 mm square on a side, and the nose had a double-conical shape. The flow velocity was 20 m/s, and the Reynolds number derived from the mean chord length was 1.4 × 10^5. The results of the study are summarized as follows. Based on the visualization of the wake flow, vortices were generated from the tips of the upper and lower wing elements of the biplane at an attack angle of α = 0°, but the vortices had an opposing rotation direction. Analysis of the wake data showed that locally induced drag was not generated at the wing tip. At α = 6° and 8°, the profile drag increased, probably due to the influence of the flow separation from the upper surface of the upper wing element. However, the total lift coefficient increased with an increasing angle of attack, even at α > 8°. Therefore, it can be concluded that the biplane lift is mainly generated by the lower wing elements.

Key Words: Biplane, Wake Measurements, Aircraft Design, Wind Tunnel Testing, Aerodynamics

1. Introduction

The Busemann biplane was proposed with the aim of cancelling the wave drag of a diamond shaped airfoil at the zero-lift condition in a supersonic flow. This airfoil eliminates wave drag by splitting it into two elements and placing these elements at locations that cancel the shock waves. Therefore, the Busemann biplane is considered to be one of the best candidates for reducing the sonic boom of supersonic flight.

It is well known that various studies of a supersonic biplane have been performed at Tohoku University for the past 15 years or so.1)1 The concept of a modified Busemann biplane with low-boom supersonic flow was proposed by Kusunose et al. The team reanalyzed the configuration of a classic Busemann biplane using modern CFD tools to understand the basic wave cancellation concept. Then they designed a 2-D supersonic biplane that has the desired aerodynamic performance with an inverse design technique.3–5) The designed 2-D biplane demonstrated a remarkable drag reduction during supersonic flow. Since then, supersonic biplane concepts have mostly been investigated by 2-D numerical flow simulation, and valuable aerodynamic results have been obtained.6–9)

In recent years, the complex 3-D flow field caused by interference between the shock wave generated from the fuselage and the biplane main wing has been investigated numerically and experimentally to obtain more detailed aerodynamic data for aircraft design.10–13) Moreover, a low-boom fuselage design with reference to the concepts of a twin body and a Sears–Haack body has been investigated by 3-D numerical simulation.14,15) The concept of this body is relevant to biplane design.

However, in an aircraft design using the biplane concept, an understanding of the aerodynamic characteristics at low speeds should be obtained, in addition to reducing the wave drag at supersonic speeds, for the assessment of performance, such as during take-off and landing. For example, wake turbulence, wing tip vortices, and high-lift devices are important topics for study with a low-speed wind tunnel, including measurements of basic aerodynamic forces. However, as far as we know, it seems that low-speed wind tunnel testing of a 3-D biplane model has not yet been fully discussed.

In low-speed wind tunnel testing, a wake measurement technique based on the conservation of momentum was proposed (the technique is called the wake integration method16)), and it has been used for the aerodynamic design of aircraft.17–19) The technique is capable of determining the lift and drag, and also decomposing the drag to components (the profile drag and the induced drag). In addition, this technique can calculate distributions of the local lift coefficient and the local drag coefficient along the spanwise direction of the model. Hence, it is considered to be a powerful tool.

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for wind tunnel testing and basic aerodynamic investigations of the aircraft. Also the wake measurement in the low-speed testing can be expected to contribute to basic aerodynamic knowledge of biplanes.

In this study, a 3-D Busemann biplane model based on a previous CFD study are constructed and tested. The basic aerodynamic characteristics of the 3-D model in a low-speed wind tunnel are clarified with the wake survey technique. The aim of this study is to contribute to the basic knowledge of the aerodynamic characteristics of a Busemann biplane.

2. Experimental Setup

2.1. Low-speed wind tunnel

The wind tunnel is the blowdown open circuit type, and its test section has no solid boundaries. The exit configuration is square in 1,500 mm × 1,500 mm. An AC motor (75 kW) controlled by an inverter drives the axial fan. The maximum wind speed in the tunnel is 27 m/s. The test section is equipped with a six-component balance system, which features a tandem strut to set the angle of attack. In addition, a wake survey system is installed downstream of the balance system.

2.2. Test model

A model Busemann supersonic biplane was prepared by referring to the 3-D configuration of the research by Odaka et al. This shape is considered as one of the first studies of the 3-D configuration of a Busemann supersonic biplane. Figures 1(a) and 1(b) show the front and top views, respectively, of the 3-D Busemann biplane model used in the experiment. Also, the coordinate system is shown in the figure, where the x-axis is in the freestream flow direction, the y-axis is in the model width direction, and the z-axis indicates the direction perpendicular to the x and y axes.

The model of the Busemann biplane was designed to cruise at Mach 1.7 by employing appropriate spacing between the upper and lower wing elements. The total length of the model was 742 mm, and the width was 556 mm. The fuselage was square and measured 76 mm on a side, and the nose had a double-conical shape. The original model formed a square cutting surface at the rear of the fuselage. A part of a quadrangular pyramid was installed at the rear, as shown in Fig. 1(b), to avoid a large pressure loss in the wake due to the cutting surface. The model was made of aluminum.

Details of the model’s main wing (upper and lower wing elements) dimensions are shown in Table 1. The aspect ratio is $AR = 5.3$, the mean aerodynamic chord length is $c_{ref} = 105$ mm, and the wing thickness is $t = 0.05c$, where $c$ is the local chord length (of each section). The spacing between the upper and lower wing elements is located at 0.505c that is the design Mach number 1.7. The projected wing area $S = 57,076$ mm² of the model is defined as the reference area.

2.3. Experimental conditions and method

The experimental conditions are shown in Table 2. The flow velocity was 20 m/s, and the Reynolds number based on the mean chord length was $1.4 \times 10^5$. As shown in Table 2, the aerodynamic data were measured by means of a wake survey and the balance system.

A five-hole probe was used to measure the static pressure and the stagnation pressure in the measurement region. All three components of wake velocity were calculated with the pressure data. The probe was made of stainless steel, the angle of the total pressure hole was 90° relative to the wind tunnel uniform direction, and the four static pressure conditions and method.

Table 1. Detailed dimensions of the main wing.

| Overall width (b) | 556 mm |
| Wing root chord length ($c_r$) | 150 mm |
| Wing tip chord length ($c_t$) | 37.5 mm |
| Aspect ratio (AR) | 5.3 |
| Taper ratio ($c_r/c_t$) | 0.25 |
| Mean aerodynamic chord length ($c_{ref}$) | 105 mm |
| Wing thickness (t) | 0.05c |
| Vertical spacing between wing elements | 0.505c |

Table 2. Experimental conditions and method.

| Flow condition | 20 m/s |
| Reynolds number | $Re = 1.4 \times 10^5$ |
| Experimental method (Wake survey system) | |
| Traverse velocity | 5 mm/s (Interval for analysis: vertical and horizontal 5 mm) |
| Probe | 5 holes probe (The inner diameter of holes 0.2 mm, the total diameter 2.0 mm) |
| Probe position | 3.5 times aerodynamic mean chord length from wing root |
| Measurement surface | 320 mm × 300 mm |
| Angle of attack | $0°, 2°, 4°, 6°, 8°$ |
| Experimental method (Balance system) | |
| Angle of attack | $−10° +20°$, interval 2° |
holes were mounted on a 45° cutting surface. The inner diameter of each hole was 0.2 mm, and the total diameter of the hole pattern was 2.0 mm.

Pressure was measured with pressure transducers (Fujikura Ltd., PSM 200KPG). The electrical signals obtained from the pressure were collected by a personal computer through a data logger (Keyence, NR500). The sampling rate was 1.0 kHz.

An overview of the wake measurement is shown in Fig. 2. The coordinate system is the same as that in Fig. 1. A 2-D traverse system (IAI Corp., ISA-SXM and ISA-MXM) was located 1.5 m downstream from the wind tunnel exit. The coordinate system is controlled to produce attack angles of \( -3 \leq \alpha \leq 3 \) and yaw angles of \( 0 \leq \beta \leq 2 \) by a remote-control unit. The aerodynamic forces are measured by changing the angle of attack from 0° to 20° at 2° intervals.

Because the air stream in the open test section is free to expand, the wake and solid blocking effects are small during wind tunnel testing. Thus, wind tunnel correction methods are applied only to the effect of the strut on the drag coefficient during the first processing step.

3. Wake Analysis

An arbitrary closed surface that surrounds a model is defined in the wake integration method. Figure 4 shows an overview of the control volume (specifically, the x-z surface), where \( P_t \), \( U \), and \( \rho \) are the total pressure, velocity, and density, respectively; the subscript \( \infty \) denotes the free-stream condition; \( S_1 \) is the control volume surface area upstream; and \( S_2 \) is the downstream area, including the wake region \( W_A \). The experiment was performed under the following assumptions:

[1] The wake data were measured downstream of the model.
[2] The flow field at the measurement station is a steady flow.

\[ \text{Unit [mm]} \]

\[ \text{Wake analysis interval (Horizontal)} \]

\[ \text{5 holes probe} \]

\[ \text{Traverse velocity: 5 mm/sec} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Vertical)} \]

\[ \text{Test model} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Horizontal)} \]

\[ \text{5 holes probe} \]

\[ \text{Traverse velocity: 5 mm/sec} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Vertical)} \]

\[ \text{Test model} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Horizontal)} \]

\[ \text{5 holes probe} \]

\[ \text{Traverse velocity: 5 mm/sec} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Vertical)} \]

\[ \text{Test model} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Horizontal)} \]

\[ \text{5 holes probe} \]

\[ \text{Traverse velocity: 5 mm/sec} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Vertical)} \]

\[ \text{Test model} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Horizontal)} \]

\[ \text{5 holes probe} \]

\[ \text{Traverse velocity: 5 mm/sec} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Vertical)} \]

\[ \text{Test model} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Horizontal)} \]

\[ \text{5 holes probe} \]

\[ \text{Traverse velocity: 5 mm/sec} \]

\[ \text{Flow direction} \]

\[ \text{Wake analysis interval (Vertical)} \]
The flow field around model is a parallel flow along the freestream flow direction.

Shear stress at the measurement station is ignored.

Small perturbation method was employed to expand the general lift and drag equations in the integration form of the momentum equation.

The lift and drag of the aircraft model can be calculated by measuring the flow variables inside the model wake. The local profile drag \( c_{dp} \) and the local induced drag \( c_{di} \) are given by Eqs. (1) and (2) under low Mach number flow conditions (\( \rho = \rho_{\infty} \) is assumed).

\[
c_{dp}(y) = \left[ \int_{z_{wa}} P_{\infty} \frac{\Delta s}{R} dz - \int_{z_{wa}} \frac{P_{\infty}}{2} \left( \frac{\Delta s}{R} \right)^2 dz \right] / q_{\infty} c(y)
\]

(1)

\[
c_{di}(y) = \left[ \int_{z_{wa}} \frac{\rho_{\infty}}{2} \psi \xi dz \right] / q_{\infty} c(y)
\]

(2)

Where \( R, P_{\infty}, \Delta s, \psi, \xi, c, q_{\infty}, \) and \( Z_{wa} \) are the gas constant, freestream static pressure, perturbation entropy, stream function, vorticity, local biplane wing chord length, freestream dynamic pressure and local integral region of \( z \)-axis in wake area, respectively. The profile drag coefficient \( C_{dp} \), the induced drag coefficient \( C_{di} \) and the total drag coefficient \( C_D \) are shown by Eqs. (3), (4) and (5).

\[
C_{dp} = \left[ \int \int_{W_A} P_{\infty} \frac{\Delta s}{R} dydz - \int \int_{W_A} \frac{P_{\infty}}{2} \left( \frac{\Delta s}{R} \right)^2 dydz \right] / q_{\infty} S
\]

(3)

\[
C_{di} = \left[ \int \int_{W_A} \frac{\rho_{\infty}}{2} \psi \xi dydz \right] / q_{\infty} S
\]

(4)

\[
C_D = C_{dp} + C_{di}
\]

(5)

Where \( W_A \) is the integral area over the model wake region.

The local lift coefficient \( c_l \) is obtained by

\[
c_l(y) = \frac{2}{U_{\infty} \Gamma(y)}
\]

(6)

Where \( \Gamma(y) \) means the circulation. \( U_{\infty} \) is the freestream velocity. The total lift coefficient \( C_L \) is obtained as follows:

\[
C_L = \left( \frac{\rho_{\infty} U_{\infty}}{W_A} \int y \xi dydz \right) - \rho_{\infty} U_{\infty}^2 \int \frac{w}{U_{\infty}} \frac{\Delta u}{U_{\infty}} dydz \right) / q_{\infty} S
\]

where \( w \) and \( \Delta u \) are the velocity component of the \( z \)-axis, and perturbation velocity, respectively.

The detailed process of the equation expansion is described in references (16), (18), and (19).

4. Results and Discussion

4.1. Visualization of wake flows

This section discusses the qualitative characteristics of the wake of the model, such as flow vectors, total pressure loss, and vorticity.

(a) Flow vectors

Figures 5(a)–(c) show the velocity vector of the wake flow. The angle of attack is 0°, 4°, and 8° in the three figures, respectively. The vertical axis \( y \) and the horizontal axis \( z \) are normalized by the model width \( b \). The wake survey was done on the half span of the model. Where, half of the front view is shown in the upper part of Fig. 5(a). The hori-
zontal axis shows the range $y/b = 0.0 - 0.55$, and the vertical axis shows the range $z/b = -0.225 - 0.225$. The center of the model (the aircraft axis) in the case of $\alpha = 0^\circ$ is located at the origin of the coordinates.

Figure 5(a) at $\alpha = 0^\circ$ shows that the upper wing element generates a wing tip vortex rotating from the upper surface to the lower surface at the wing tip, while the lower wing element generates a wing tip vortex in the opposite rotational direction. The concept of the Busemann biplane is to generate a shock wave between the wing elements, which cancels it. The main wing is designed as two wings, with the upper wing element having a negative camber and the lower wing element having a positive camber. Therefore, wing tip vortices with an opposite direction of rotation are generated at the wing tips of the upper and lower wing elements. In this case, the absolute strength of each vortex is expected to be the same.

In Fig. 5(b) at $\alpha = 4^\circ$, only a single wing tip vortex with counterclockwise rotation can be observed. This may be because the upper and lower wing elements act as a single wing due to the increased angle of attack. Then, at an attack angle of $\alpha = 8^\circ$, as shown in Fig. 5(c), the size of the wing tip vortex becomes progressively larger. This trend of the biplane shows characteristics qualitatively similar to those of a monoplane aircraft.20)

(b) Total pressure loss

Figures 6(a)–(c) show the total pressure loss of the wake. The vertical axis, the horizontal axis, and the angle of attack are the same as in Fig. 5. The color bar $(P_{T} - P_{\infty})/q_{\infty}$ describes a dimensionless value where $P_{T}$, $P_{\infty}$, and $q_{\infty}$ are local total pressure, freestream total pressure, and freestream dynamic pressure, respectively.

From the pattern shown in Fig. 6(a) at $\alpha = 0^\circ$, the total pressure losses can be observed at locations (around $y/b = 0.1 - 0.5$, $z/b = \pm 0.07$) that match the upper and lower wing elements in the width direction. The total pressure losses indicate the influence of the main wing. Also, an area with higher total pressure loss (as shown by red) can be observed around $y/b = 0 - 0.1$ due to the influence of the fuselage and the strut. Such a pattern was also seen in a previous study.17)

Figure 6(b), at $\alpha = 4^\circ$, shows no significant qualitative difference from Fig. 6(a), but the area of total pressure loss at the upper wing element is slightly larger than that in Fig. 6(a) due to the effect of the increased angle of attack. An area of higher total pressure loss, which is presumed to be due to the influence of the fuselage, is also observed.

At $\alpha = 8^\circ$ in Fig. 6(c), an area of total pressure loss from the upper wing element is seen around $y/b = 0.15 - 4.0$. This shows that the flow separated from the wing surface of the upper element as the angle of attack increased. Furthermore, a total pressure loss is also observed at the wing tip. This might indicate the influence of a strong wing tip vortex due to the higher angle of attack of $\alpha = 8^\circ$. In a more detailed observation, a weak but smaller total pressure loss is also found below the highest loss due to the strong wing tip vortex. This loss is further discussed below.

(c) Vorticity

The vorticity of the wake flow is shown in Figs. 7(a)–(c). The vertical and horizontal axes and the angles of attack are the same as those in Fig. 5. The color bar $(\xi - b)/U_{\infty}$ shows the value of the dimensionless vorticity.

At $\alpha = 0^\circ$ in Fig. 7(a), the velocity vector trend is qualitatively the same as that in Fig. 6(a). Therefore, two vorticities that are rotating in opposite directions (the two served clearly at the wing tips.

As shown in Fig. 7(b), at $\alpha = 4^\circ$, the negative vorticity at the wing tip is no longer seen, and only an intense positive vorticity is found. This is because two vorticities with posi-
tive intensity are generated from the upper and lower wing elements as the angle of attack increases, resulting in the combination of these two vorticities.

In Fig. 7(c) at \( \alpha = 8^\circ \), a wing tip vorticity with higher intensity than that at \( \alpha = 4^\circ \) is observed. The vorticity is considered to be due to the vortex caused by the pressure difference between the upper surface of the upper wing element and lower wing surface of the lower wing element. Also, as a unique phenomenon, we can see another weaker positive vorticity under the high-intensity vorticity at the wing tip. This weak vorticity is presumed to be generated from the upper single wing element as the angle of attack increases.

In fact, the weak vorticity was not observed in our preliminary experiment\(^{21} \) of the wingtip plate of the biplane.

### 4.2. Local aerodynamic characteristics

This section discusses the distribution of the lift, profile drag, and induced drag along the width direction of the model. Because the flow separation from the upper wing element is clearly shown in Fig. 6, the establishment of assumption \([2]\) in Section 3 is somewhat irrefutable. However, the wake data are assumed to be an average over time. Accordingly, a more detailed analysis of the wake data was attempted with the wake integration method as a first step.

(a) Local lift coefficient

The local lift coefficient distribution along the width direction of the model is shown in Fig. 8. The vertical axis shows the normalized lift coefficient \( l_{c} \) and the horizontal axis shows the normalized distance \( y/b \) of the spanwise direction from the center of the fuselage. The measurement attack angles are \( \alpha = 0^\circ, 2^\circ, 4^\circ, 6^\circ \), and \( 8^\circ \).

First, the local lift coefficients of higher angles of attack increase at any cross-section \( y/b \). At \( \alpha = 0^\circ \), the figure shows that the lift along the width direction is almost negligible. Although wing tip vortices were generated at the wing tips of the upper and lower wing elements (see Fig. 5(a)), the local lift coefficient is almost \( c_{l} \times c/c_{ref} = 0 \) near the wing tip \( y/b = 0.5 \). As a result, the two main wings of the biplane functions as a single symmetrical wing.

At \( \alpha = 2^\circ \), the local lift coefficient is increased from the coefficient of \( \alpha = 0^\circ \) due to the increased angle of attack. The local lift coefficient approaches \( c_{l} \times c/c_{ref} = 0 \) as \( y/b \) moves towards 0.5, and it increased slightly around the wing tip. This trend is different from that seen with other angles of attack. One of reasons for this trend is a transition of the flow pattern. It is expected that the negative vorticity from the upper wing element will disappear around \( \alpha = 2^\circ \). Then, it affects the equilibrium of the local aerodynamic forces. But further investigation of the details of this phenomenon is needed.

At \( \alpha = 4^\circ \), a slightly wavy variation of the local lift coefficient is observed in the range of \( y/b = 0.1–0.2 \). Although it is not clear at \( \alpha = 4^\circ \), this may be the effect of the flow separation from the upper surface of the upper wing element de-
As shown in Fig. 7(a) at the width direction of the local cross section of the width direction from the model center. The phenomenon is more evident during total pressure loss at $\alpha = 8^\circ$ (see Fig. 6(c)). Also, the distribution of the local lift coefficient at $\alpha = 6^\circ$ and $8^\circ$ shows the same trend as that at $\alpha = 4^\circ$ qualitatively.

(b) Local profile drag coefficient

Figure 9 shows the local profile drag coefficient at the attack angles of $\alpha = 0^\circ$, $2^\circ$, $4^\circ$, $6^\circ$, and $8^\circ$. The vertical axis $c_{d0} \times c/c_{ref}$ shows the dimensionless local profile drag coefficient. The horizontal axis shows the normalized distance $y/b$ of the local cross section of the width direction from the model center.

In Fig. 9, it can be observed that the value of the local profile drag coefficient becomes large in the range of $y/b = 0.1$ to $0.5$ at all angles of attack. This is due to the influence of the model support strut installed in the balance system. In addition, the local drag coefficient decreases as the angle of attack increases because the area of the strut projected onto the measurement surface decreases. At $\alpha = 0^\circ$, $2^\circ$, and $4^\circ$, the profile drag $c_{d0} \times c/c_{ref}$ gradually decreases along the width direction in the range from $y/b = 0.1$ to $0.5$.

At $\alpha = 6^\circ$ and $8^\circ$, an increase of the local profile drag coefficient can be observed in the range of $y/b = 0.2$ to $0.3$, probably due to the influence of the flow separation seen in Fig. 6(c). In addition, an increase of the local profile drag was observed around the wing tip ($y/b = 0.5$). This is considered to be caused by the higher total pressure loss shown in Fig. 6(c) due to the strong wing tip vortex.

(c) Locally induced drag coefficient

Figure 10 shows the locally induced drag coefficients. The vertical axis $c_{d0} \times c/c_{ref}$ shows the dimensionless locally induced drag coefficient, and the horizontal axis shows the dimensionless distance $y/b$.

In Fig. 10, the locally induced drag is almost negligible in the width direction of $y/b = 0.45$ at all angles of attack. As shown in Fig. 7(a) at $\alpha = 0^\circ$, vorticities with both positive and negative intensity were observed at the tips of the upper and lower wing elements ($y/b = 0.5$). However, the intensity of the two vorticities cancelled each other out, and the locally induced drag was almost negligible around $y/b = 0.5$.

As the angle of attack increased to $\alpha = 2^\circ$, $4^\circ$, and $6^\circ$, the locally induced drag gradually increased at the wing tip. Then, the locally induced drag rapidly increased to a maximum value of $c_{d0} \times c/c_{ref} = 0.1$ at $\alpha = 8^\circ$. It was found that the trend of a biplane is qualitatively similar to that of a monoplane.

Here focusing on the peak location of the induced drag, the peak of locally induced drag seems to move outward in accordance with the increase of the angle of attack from $\alpha = 2^\circ$ to $6^\circ$, also it suddenly moves inward at the attack angle of $\alpha = 8^\circ$. This is presumed to be due to the transition of the wingtip vortex formation process with the increase in the angle of attack. Namely, the vortex (positive vorticity) on the upper surface of the lower wing element at the angle of attack $\alpha = 0^\circ$ goes around to the outward at the wing tip of the upper wing element together with increasing of the angle of attack. Also, at an angle of attack of $\alpha = 8^\circ$, it is conjectured that the vortex moved rapidly inward due to the influence of the weak wing tip vortex (see Fig. 7(a)) after the vortex reached the upper surface of the upper wing element.

### 4.3. Total lift and drag coefficient

The basic aerodynamic characteristics (total lift and total drag coefficients) of the Busemann biplane model are shown in Figs. 11 and 12. The figures show the experimental results for the balance system and the wake measurement.

Figure 11 shows the total lift coefficient $C_L$ versus the angle of attack $\alpha$. The figure shows that the total lift coefficient increases linearly with the angle of attack. The qualitative trend of the total lift coefficient of the biplane is the same as that of a monoplane aircraft\cite{22,23} for Reynolds numbers above $10^5$. As discussed for the total pressure loss (see Fig. 6(c)) and the profile drag (see Fig. 9 at $\alpha = 6^\circ$ and $8^\circ$), the flow separated from the upper surface of the upper wing element. However, the total lift coefficient increased linearly with an increasing angle of attack, and the model did not stall at $\alpha \geq 6^\circ$. This is because the flow on the upper surface of the lower wing element did not separated, and the lift of the biplane is mainly generated by the lower wing elements. Although it is limited to this range of attack angles, the flow separation from the upper wing element does not have a sig-
significant effect on the stall characteristics. The same trend can be seen in Ref. 24). Also, in the previous study, the stall angle of the double wedge airfoil (10% airfoil thickness, non-sophisticated airfoil) shows only about $\alpha = 2^\circ$ lower than that of NACA 0012. Therefore, it is considered that the stall angle of the model shows relatively good characteristics.

The results of the wake measurement and the balance system agree quantitatively, though the data are slightly different from the results of $\alpha = 2^\circ$. Moreover, the lift slope of the wake measurement was $dC_L/d\alpha = 0.0912$, which shows good agreement with the value of 0.0956 obtained by the balance system, quantitatively.

Here begins the description of the drag of the strut before discussing the total drag coefficient by wake measurement. Table 3 shows the total drag coefficient of the strut by the wake measurement. The wing area of the model is employed as the reference area. As the angle of attack increases from the table, the drag of the strut decreases because the strut has been moved to down ($z$-axis minus direction) and the projected area to the wake measurement surface decreases. In the wake measurement, it is difficult to separate the wake of the strut and that of the model, therefore, the total drag of the strut was removed from the total drag of the wake measurement in this study as the first step.

Figure 12 shows the total drag coefficient $C_D$ versus the angle of attack $\alpha$ by the wake measurement and the balance system. The data of the wake measurement shows the total drag after removal of the strut drag, which was measured separately as described above. In the balance measurement, the drag of the expose part of the strut is removed at each angle of attack. The total drag coefficient obtained by the balance system shows a minimum value at $\alpha = 0^\circ$. Then, the total drag rapidly increases with attack angles other than $\alpha = 0^\circ$. One of the reasons for this rapid increase in the total drag may be due to the fact that the shape of the biplane model is simple, with flat planes as described in Section 2.1. Thus, a large flow separation occurs on the upper surface of the upper wing element above $\alpha = 6^\circ$, as seen in Fig. 9. Additionally, the wake measurements show the same trend as that of the balance system, qualitatively.

Table 4 shows the differences in total drag coefficient obtained by the two measurement methods, where $|\Delta C_D|$ shows the difference in total drag between the wake measurement and the balance system in terms of counts. In accordance with the accuracy of the present wind tunnel setup, the difference became larger when the angle of attack was greater than $\alpha = 4^\circ$, with a maximum difference of 154 counts. This may be explained by the unsteady flow caused by the flow separation and the method for compensating for the struts.

According to Fig. 12 and Table 4, the data from wake measurement are generally in agreement with the data from the balance system quantitatively.

### 5. Conclusion

In this study, the basic aerodynamic characteristics of a 3-D biplane model in a low-speed wind tunnel were clarified with the wake survey technique. The results of the study are summarized in this section.

Visualization of the wake flow revealed that wing tip vortices are generated from the wing tips of the upper and lower wing elements of the biplane at an attack angle of $\alpha = 0^\circ$, but their direction of rotation is opposite. Therefore, the analysis of the wake data showed that locally induced drag was not generated at the wing tip. At $\alpha = 6^\circ$ and $8^\circ$, the profile drag increased, possibly due to the influence of the flow separation from the upper surface of the upper wing element. However, the total lift coefficient increased with an increasing angle of attack, even a $\alpha > 8^\circ$. Therefore, the lift of the biplane...
is mainly generated by the lower wing elements. The total lift and drag coefficients obtained by wake measurement are generally in agreement with the results from the balance system quantitatively. In addition, the trend of the total lift and the total drag coefficient of the biplane is qualitatively similar to that of a monoplane.

Consequently, our results contribute to the basic knowledge of the aerodynamics of the Busemann biplane at low speeds.

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