Control of Delta-Wing Vortex by Micro-Fin-Type Leading-Edge Flap

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Abstract

The present study examined the effects of micro leading-edge flaps on the vortex characteristic changes of a double-delta wing through pressure measurements of the wing upper surface and PIV measurements of the wing-leeward flow region. The experimental data were collected and analyzed while changing the deflection angle of the leading-edge flaps to investigate the feasibility of using micro leading-edge flaps as flow control devices. The test results revealed that the leading edge modification could greatly alter the vortex flow pattern and the wing surface pressure of the delta wing, which suggested that the leading-edge flaps could be used as an effective device for the control of delta-wing vortex flow.

Key Word : delta-wing vortex, micro leading-edge flap, flow control, wing surface pressure, PIV measurement

Introduction

Vortex flows over delta wings contribute a large amount of aerodynamic force generation; therefore, many studies have been carried out to search for the ways of controlling vortex flows to enhance the stability and controllability characteristics of delta wings, or even further to replace the conventional control surfaces. With the help of the recent development of micro-electro-mechanical-systems(MEMS) techniques and synthetic jet systems, the research effort in this field has been accelerated and various new methods of controlling vortex structures around a delta wing have been tested. The techniques investigated in the previous researches can be mainly divided into two categories: a) direct flow-momentum control utilizing blowing jets located at apex, leading edge or trailing edge of delta wings, b) shape or configuration modification of delta wing itself. The literatures for the latter category include a swept-back angle variation method[1], apex-flap methods[2-4], leading-edge vortex-flap methods[5-8], and trailing-edge vortex-flap methods[9,10]. The results of the previous researches[11,12] indicated that the shape modification was a more effective method for altering vortex flow than the direct flow-momentum control method using jets. The blowing jet methods were found to be less effective since the vortex flows generated from delta wings maintained fairly strong and stable structures up to certain degree of angle of attack.

Among the shape modification methods to control aerodynamic forces of delta wings, there have been some efforts to implement micro leading-edge flaps and generate aerodynamic moments for flight control. Lee at el. [5] conducted an experimental study exploring the possibility of independently controlling these moments using distributed MEMS actuators located on leading

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edges of a 56.5-deg sweep delta wing. They provided wing-surface pressure data, direct force measurement data and flow visualization results, and concluded that asymmetric vortex pairs were formed by appropriate actuation of the distributed micro-actuators, which led to the generation of significant aerodynamic forces. Matsuno et al.[12] conducted aerodynamic force measurements and flow visualization experiments to test small leading-edge flaps for controlling the leading-edge vortices and for suppressing the self-induced oscillation that occurred on a 45-deg delta wing. The results of their work indicated that the flaps could reduce the unsteady moment in post-stall high angle-of-attack regimes. Kaiden and Nakamua[13] numerically simulated the physical phenomena of leading-edge separation-vortex movement with respect to the change in the incidence angle of leading-edge micro flap.

In the present study, the effects of micro leading-edge flaps and their deflection angle variation on the vortex formation, interaction, and breakdown characteristics of a double-delta wing with the apex strake, were investigated through the pressure measurements of the wing upper surface and particle image velocimetry(PIV) measurements of the wing-leeward flow region. The main objective of this work is to investigate whether micro leading-edge flaps can effectively generate significant asymmetry in the vortex flow system such that they can be used as a flight control device.

Experimental Set-up and Procedures

Wind Tunnel

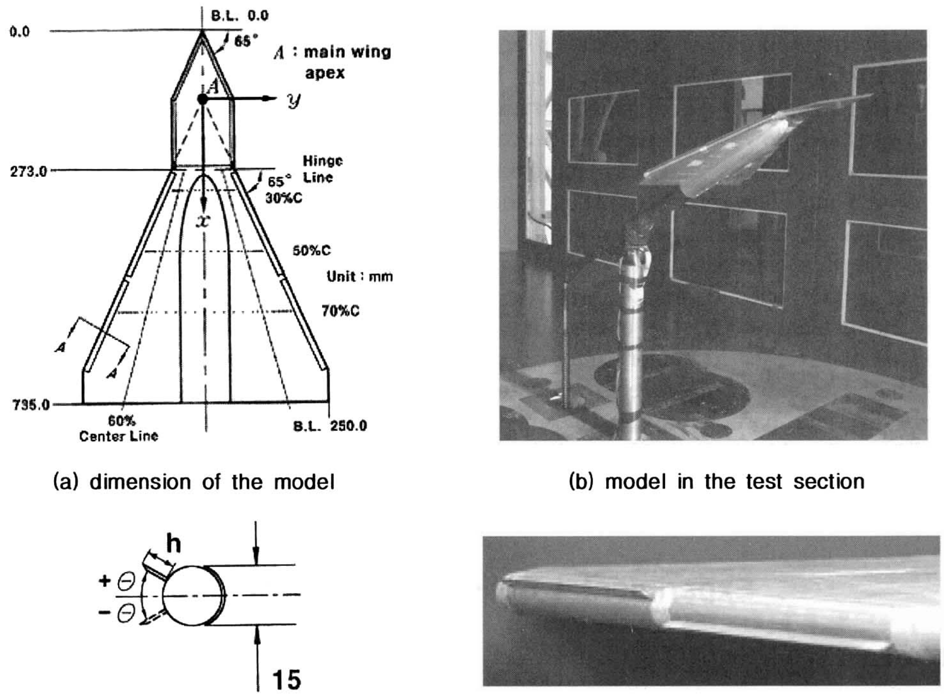
The experimental apparatus was set up at the Korea Air Force Academy Low-Speed Wind Tunnel(KAFA LSWT). The medium-scale test facility is a closed-circuit atmospheric tunnel having a test section of 3.5m(W) × 2.45m(H) × 8.7m(L) with a maximum free stream velocity of 92m/sec. The contraction ratio is 7.26:1, flow angularity is less than 0.1 deg, and the axial turbulence intensity(u'/U) is 0.04% at the free stream velocity of 74m/sec.

Experimental Model

The experimental model had the main wing with rounded leading edge, 65-deg sweep angle, the root chord of 600mm, the trailing edge span of 500mm, and the thickness of 15mm. The strake was the flat plate wing with a planform of 65-deg/90-deg sweep cropped-delta shape, and had the thickness of 6.35mm and the root chord of 273mm. The leading and side edges of the strake were symmetrically beveled. The apex portion of the delta wing can be removed and replaced with the strake to form a double-delta wing configuration with the root chord of 735mm. Figure 1(a) shows the model geometry of the integrated configuration. The lower wing surface was mounted with a fuselage-like structure that served as the housing for the pressure tubes and model support. Figure 1(b) shows the model installed in the wind tunnel. Figure 1(c) defines the flap deflection angle Θ, and Fig. 1 (d) shows the picture of the leading block equipped with leading edge flaps. In the present study, x represents the coordinate along the wing centerline measured from the wing apex, y represents the coordinate along the wing local semi-span measured from the wing centerline, and z represents the height above the upper wing surface.

The main wing was equipped with three spanwise rows of upper-surface static pressure taps for pressure measurements. The pressure rows were located at the 30%, 50% and 70% wing chord(c) stations, measured from the main wing apex. There were 23-35 pressure taps on each chord station along the entire span. The spanwise locations of the pressure taps ranged from $y/s=0.0$ to $y/s=\pm0.75$(30% chord station), $y/s=\pm0.85$(50% chord station), and $y/s=\pm0.95$(70% chord station), s being the local semi-span.

The static pressure on the wing upper surface was measured by the PSI 8400 pressure measuring system. The measurement rate was 0.2 sec/measure, and the wing surface pressure data in this study was an ensemble average of 300 pressure signals from each pressure tap. The



(a) dimension of the model (b) model in the test section

(c) definition of flap deflection angle (d) photograph of micro leading-edge flap

Fig. 1. Geometry Definition of Experimental Model

free stream velocity was 40m/s, which corresponds to a Reynolds number of 1.64×10^6. The angle of attack(α) tested ranged from 24 degs to 40 degs. The uncertainty analysis for pressure measurements was carried out to assess the data accuracy and confidence level, following the method suggested in AIAA Standard[14]. It was estimated that the pressure measurement uncertainty for the present study was less than ±24.86Pa at 95% confidence level, which was about 0.36% F.S.

The PIV System used in this study consisted of a double-pulse Nd:YAG laser(Vlite-200) with a maximum pulse energy of 2200 mJ at a repetition rate of 10Hz, a 8-bit digital CCD with 2048×2048 pixels, and a PC equipped with the DaVis FlowMaster software and a synchronization board developed by LaVision GmbH for the system synchronization, control, data acquisition and post-processing. Aerosol Generator was used for the seeding of DEHS($C_{26}H_{50}O_4$) particles. The particles were filled in the tunnel beforehand and the generator was turned off during the actual measurements. The PIV data in this study was an ensemble average of 20 instantaneous velocity fields.

The estimation of the uncertainty of PIV velocity measurements requires many factors to be examined, such as the uncertainty in the determination of the geometrical parameters and the fabrication tolerances of the camera device, lenses and laser system, as well as the uncertainty in the determination of the average particle displacement in the interrogation region[15].

The total uncertainty consists of bias error and precision error. The estimation of the bias error requires the information about the sensitivity of the various factors affecting the measurements, which, in turn, requires an extensive experimental test in a carefully prepared environment. However, the bias error can be minimized with the careful calibration of measuring instruments[16]. The precision error was estimated by calculating the standard deviation of 30 sample records with the coverage factor of 2 at 95% confidence level. The estimated uncertainties of the PIV velocity components in the region near the strake vortex were 12.5 and 13.3% for v and w velocity components, respectively, based on the maximum velocity components measured. The uncertainties appear large because the vortex formation meanders, causing the instability of the vortex location. However, in the regions where the flow behaves well (away from the vortex systems), the uncertainties for each velocity components decreased less than 2.2%.

Results and Discussion

Figure 2 shows the effects of the flap height(h) on the pressure distributions of the wing upper surface at x/c=0.5 and x/c=0.7 locations when the angles of attacks were 24 and 32 degs. The deflection angles of the flaps were set to zero. According to Reference 17, the maximum height of the boundary layer thickness was expected to be less than 0.2 mm. The flap heights of 2, 4, 6mm were considered to be large enough to alter the boundary layer. The figure indicates that the flap height variation did not affect much the pressure distributions at x/c=0.5 and x/c=0.7 locations when the angle of attack was 24 degs. For the angle of attack of 32 deg case, the flap height variation started to change the pressure distribution. However, even at the angle of attack 32 degs, the flap heights of 4mm and 6mm cases resulted in almost identical results; therefore, the height of 4mm was considered as the critical point and this height was selected as the primary experimental condition. The flap height of 4mm also corresponded to that used in the previous research results such as Lee et al.[5] and Kaiden et al.[13] (root chord-to-flap height ratio=0.67%).

Figure 3 shows the wing-upper-surface suction pressure for several angles of attack and two flap angle positions, +20 degs and -20 degs. The flap height was fixed at 4mm As expected, the suction pressure distribution generally increased as the angle of attack increased up to a certain point. It can be estimated that the wing experienced the stall condition between 32-deg

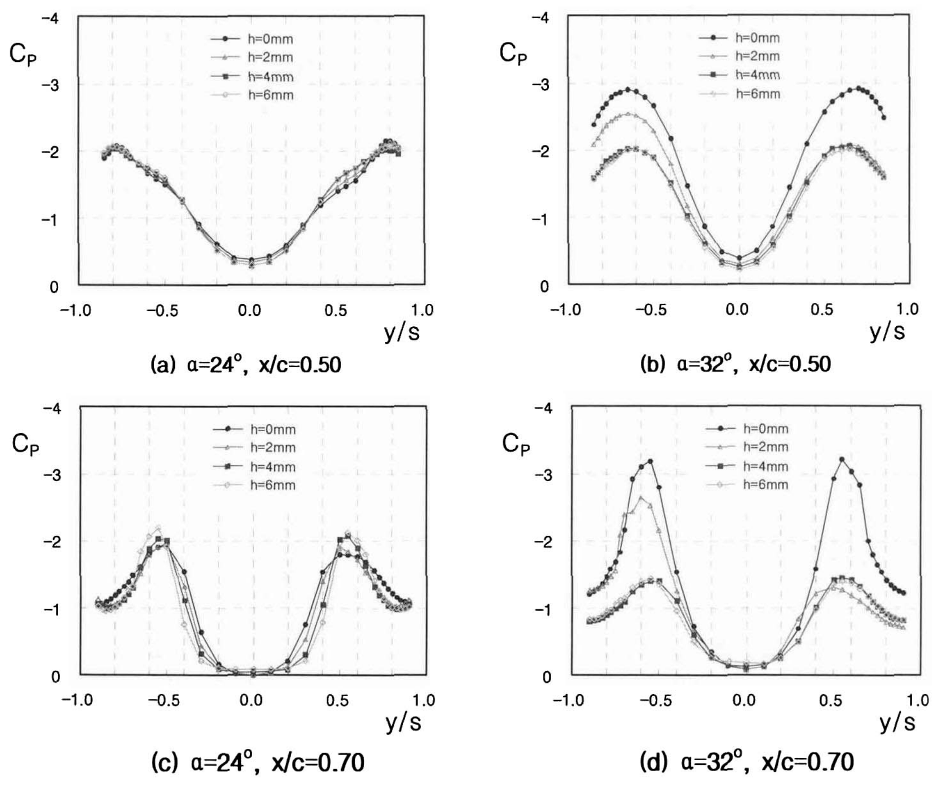
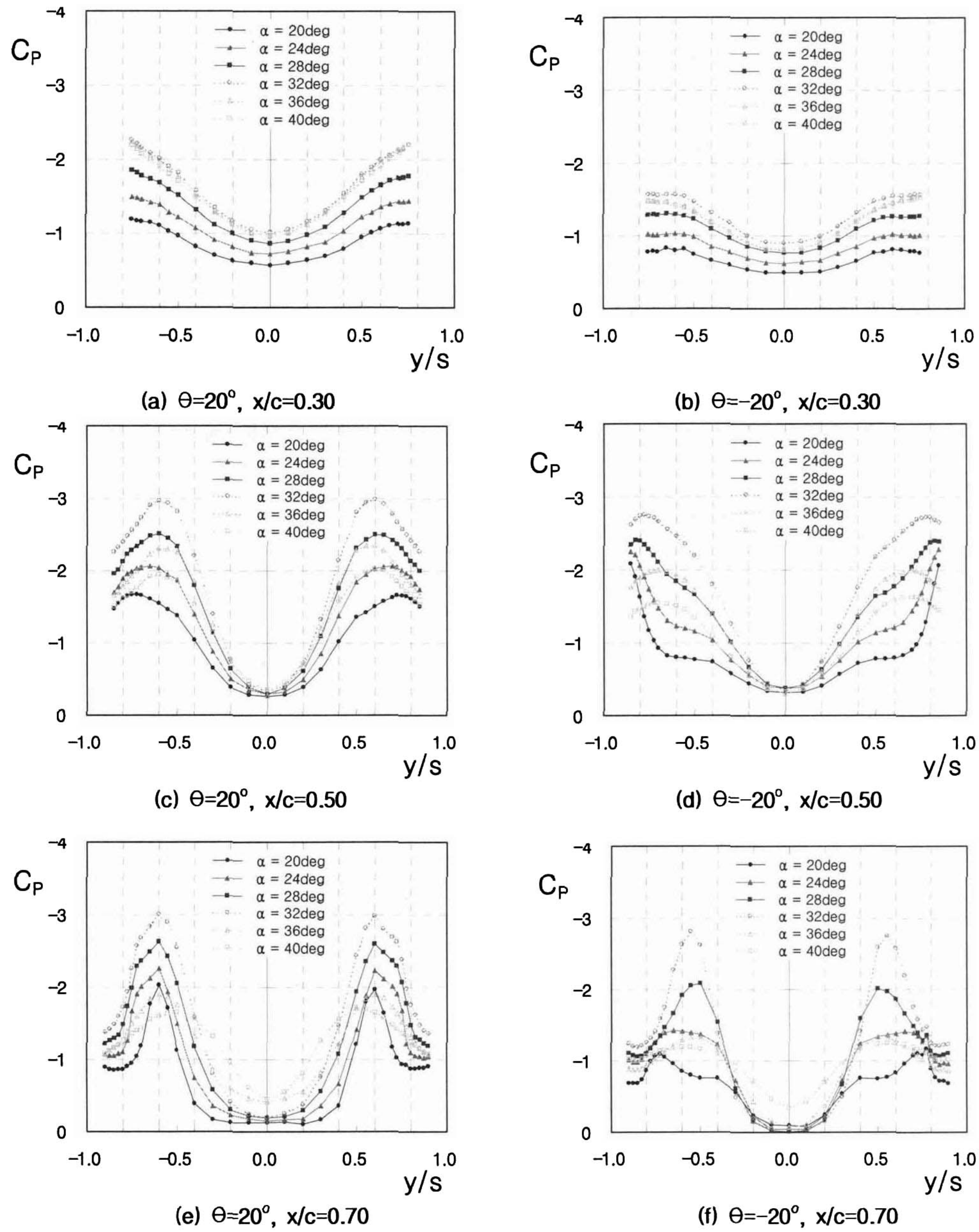


Fig. 2. Effect of flap height(h) on the wing-upper surface pressure distributions

and 36-deg angles of attack for both of $\Theta=20^{\circ}$ and $\Theta=-20^{\circ}$ cases. However, the pressure distributions remained fairly constant beyond the stall angle of attack at x/c=0.30 where the vortex system was not fully interacted yet, whereas sudden pressure change after the stall angle of attack was observed at x/c=0.50 and x/c=0.70 locations where the vortex system was fully interacted. As the flap deflection angle varied from 20° to -20°, the overall magnitude of the suction pressure distribution decreased. This implies that the positive flap deflection-angle has a favorable effect on the aerodynamic forces compared to the negative flap deflection angle. It can be inferred that the positive flap angle tends to increase the effective leading edge radius since the flow separates after the rounded leading edge portion, whereas the negative flap angle imposes



(a) $\Theta=20^{\circ}$, x/c=0.30 (b) $\Theta=-20^{\circ}$, x/c=0.30

(c) $\Theta=20^{\circ}$, x/c=0.50 (d) $\Theta=-20^{\circ}$, x/c=0.50

(e) $\Theta=20^{\circ}$, x/c=0.70 (f) $\Theta=-20^{\circ}$, x/c=0.70

Fig. 3. Effect of angle of attack and flap deflection angle on the wing-upper surface pressure distributions

early separation at the lower portion of the rounded leading edge, which results in decrease of the effective leading edge radius.

Figure 4 compares the wing-surface pressure distributions for the angles of attack of 24- and 32-deg cases. The flap height was 4mm. At 24-deg angle of attack when the vortex system was fairly stable, the pressure distributions changed with a consistent trend with respect to the flap deflection angle change. As the flap deflection increased to the positive direction, the magnitude of the suction pressure at the outboard wing surface increased, whereas the opposite trend showed when the deflection angle increased to the negative direction. The same trends were observed for both of 24- and 32-deg angle-of-attack cases. However, different trends were observed for x/c=0.5 and x/c=0.7 cases. At x/c=0.5, the peak position of the suction pressure

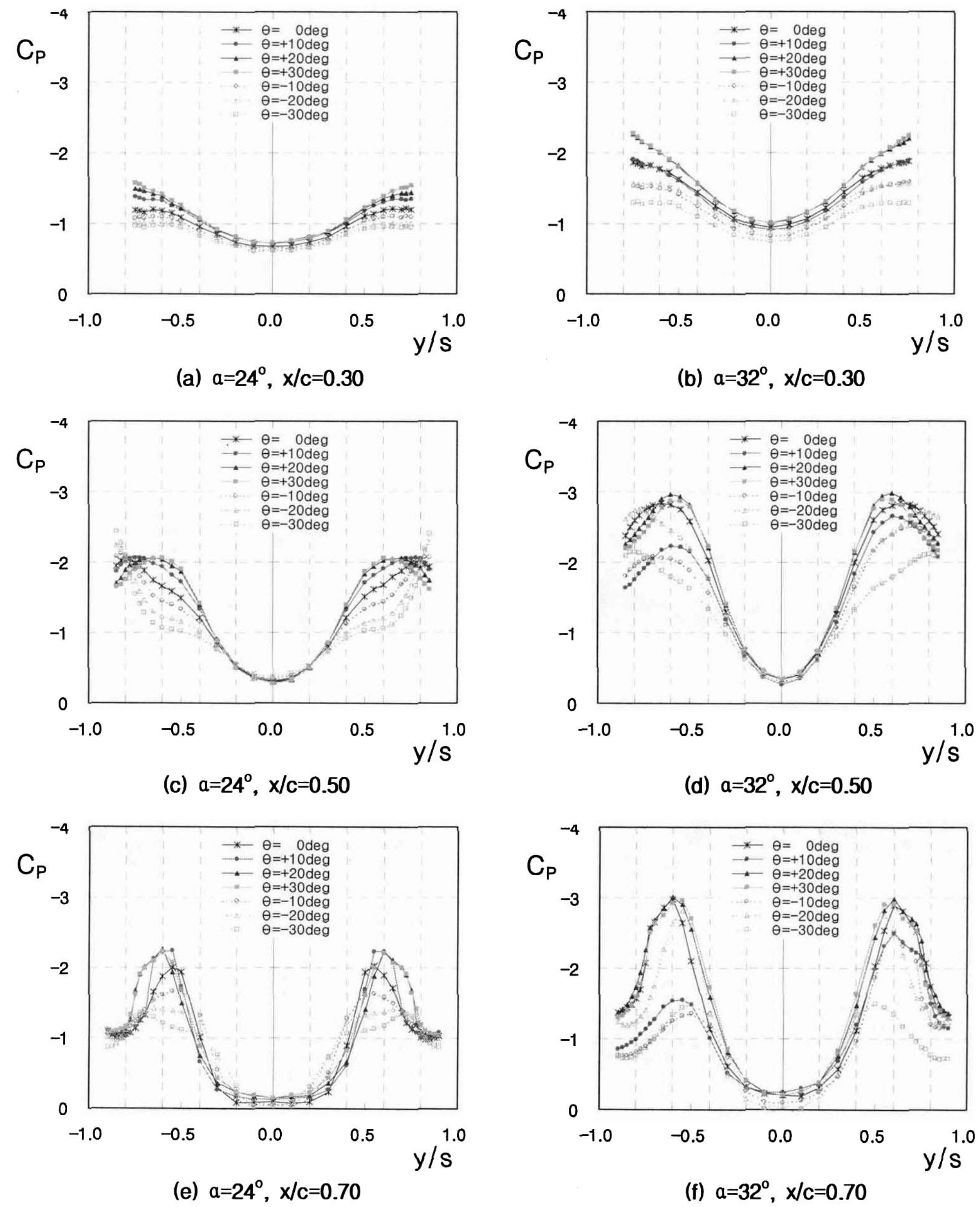
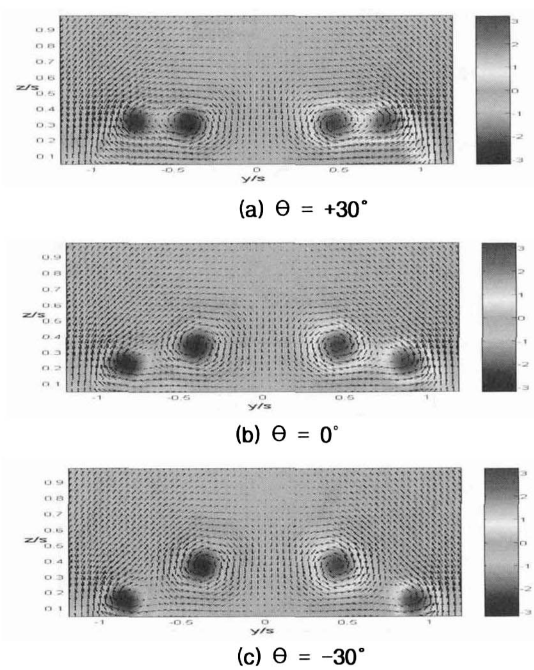


Fig. 4. Effect of flap deflection angle on the wing-upper surface pressure distributions

distribution moved into the inboard direction as the flap angle increased to the positive direction, and into the outboard direction as the flap angle increased to the negative direction. In particular, Fig. 4(c) shows very strong suction peaks developed at the leading edge region for Θ=-30-deg case, which indicated that concentrated vortex systems existed near the wing leading-edge region. This result was also confirmed with the PIV measurements. At x/c=0.7, the pressure distribution did not change much for positive flap angles, whereas the magnitude of suction pressure peak decreased for negative flap angles.

Figures 5 and 6 show cross-flow velocity vectors obtained from the PIV measurements at x/c=0.5 and x/c=0.7, respectively. The axial vorticity distribution was also calculated based on the velocity vector field and plotted as color contours. The vortex pair in the left-half of the wing rotate in clockwise direction, and the vortex pair in the right-half of the wing rotate in counter clockwise direction. The vortices in the inboard position are vortices generated by the strake and those in the outboard position are vortices generated by the wing in each frame of Fig. 5. As seen in Fig. 5 of the PIV results(x/c=0.5), the vortex structures and positions were greatly altered with the flap angle variation. The coiling process between the wing and the strake vortices was accelerated for Θ=+30 case. The vortex pair were observed as positioned parallel to the wing upper surface, which produced a pressure distribution with a smoothed pressure peak as shown in Fig. 4(c). For the case of Θ=-30, the coiling process was delayed compared to that for Θ=0 case, and the concentrated wing vortices pair existed near the leading edge region. These wing vortices caused very sharp pressure peaks at the leading edge regions(y/s=-1.0 and y/s=1.0) as observed in Fig. 4(c).

Figure 6 shows the PIV results for x/c=0.7. As for the case of x/c=0.5, the coiling process between the wing and the strake vortices was accelerated and the two vortices merged together and aligned vertically for Θ=+30 case. The vertically positioned vortices produced sharp suction pressure peaks as observed in Fig. 4(e). The coiling process was again delayed for Θ=-30 case, and the vortices still maintained the relative positions nearly parallel to the wing upper surface,

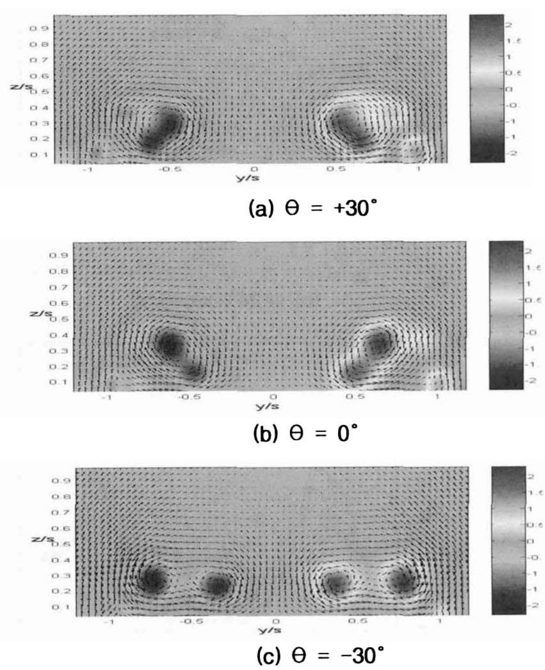


(a) Θ = +30°

(b) Θ = 0°

(c) Θ = -30°

Fig. 5. Comparison of cross-flow velocity and axial vorticity fields for different flap angle positions(α=24°, x/c=0.50)



(a) Θ = +30°

(b) Θ = 0°

(c) Θ = -30°

Fig. 6. Comparison of cross-flow velocity and axial vorticity fields for different flap angle positions(α=24°, x/c=0.70)

which resulted in the smoothed suction pressure peaks as seen in Fig. 4(e).

Based on their research using 56.5-deg delta wing with 2mm MEMS flap, Lee et al.[5] reported that the wing having a negative flap angle generated positive rolling moment, whereas the positive flap angle generated negative rolling moment. Even though the present study dealt with a more complicated geometry of the double delta wing with strake, the results of the present study showed the same trend as Lee's results. It was found that not only the strength of the wing and the strake vortices but also the interaction process between the two vortices were noticeably altered with the flap angle variation. The flap angle variation to a positive direction caused the accelerated coiling and merging process between the wing and the strake vortices, whereas the flap angle variation to a negative direction delayed the interaction between the wing and the strake vortices.

Conclusion

The present study examined the effect of a micro leading-edge flap on the flow characteristics of the vortex system and pressure distributions over a double delta wing with strake. The flap deflection angle variation to a positive direction caused the accelerated coiling and merging process between the wing and the strake vortices, whereas the flap angle variation to a negative direction delayed the interaction between the wing and the strake vortices and kept them near the wing surface longer.

The pressure measurement results indicated the overall lift of the wing generally increased when the flap angle was on the positive side. Thus it is expected that an asymmetric micro flap angle setting between the right- and left-half wings would generate a fairly large amount of rolling moment which, in turn, can be used for flight control. PIV measurements made it possible to investigate the flow structure qualitatively through which the characteristics of the pressure distribution variation with respect to the change in flap angles could be physically explained.

Acknowledgement

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