NUMERICAL OPTIMIZATION OF LEADING-EDGE DEFLECTION ANGLES FOR AN SST CONFIGURATION AT LOW SPEED

Zhong Lei*, Dong-Youn Kwak**
***Tokyo University of Science, Suwa, **Japan Aerospace Exploration Agency**

Keywords: *SST, High-Lift Device, Aerodynamics, Optimization, Responds Surface Method*

Abstract

A procedure for designing deflection angles of multi-segmented flaps in the design optimization of a supersonic wing has been developed to improve aerodynamic performance at taking-off and landing conditions. The geometric shape and computational mesh modification was conducted automatically by a newly-developed tool for all cases with different combination of the leading-edge deflection angles. Flow field around the configuration was simulated and aerodynamic forces were calculated by solving the Reynolds-averaged Navier-Stokes equations with MPI parallel programming. In the design space, a response surface of the 2nd- polynomial was constructed initially by using a number of design variations sampled by an optimal Latin hypercube method, to approximately estimate aerodynamic force and improved by adding cases progressively in the design process. The response surface method largely reduced the number of sampling cases needed in design. Optimal solution of the leading-edge deflection angles, that the drag was the minimum at the specified design lift, was searched on the response surface. Optimization studies were conducted for a cranked-arrow wing with multi-segmented flaps of a supersonic transport configuration. It was confirmed that aerodynamic performance was improved by the optimization design, and the design method was efficient to reduce design cost.

* Contact address: Faculty of Systems Engineering, Tokyo University of Science, Suwa. 5000-1 Toyohira, Chino-shi, Nagano-ken, Japan 391-0292
Email: lei@rs.suwa.tus.ac.jp

1 Introduction

The delta wing planform has advantages to largely reduce wave drag generated by shock wave in supersonic flight, and continue producing lift up to very high angles of attack thus remarkably delay stall in the low speed regime. A modification to the pure delta wing frequently utilized to improve low-speed performance is the so-called cranked-arrow planform. The cranked-arrow wing consists of a highly swept delta wing in the inner part and less swept wing in the outer part. It can provide better performance at low speeds and high angles of attack, and also has fewer shifts in aerodynamic center between subsonic and supersonic speeds. However, this kind of wing with low aspect ratio and large sweep angle of the leading edge is known for having poor aerodynamic efficiency lift-to-drag ratio for taking-off and landing conditions.

For the highly swept wing, one of the most important phenomenons is the strong vortical flow over the upper surface at moderate or high angles of attack, which generated large additional vortex lift. However, the leading-edge separation vortices reduced the effective lifting span and dramatically increased the induced drag. The additional thrust required to overcome the increased drag and the low speed performance deficiencies, generates an unacceptable engine noise and large fuel consumption.

In order to meet economic viability and environmental compatibility, the next generation supersonic transport (SST) is required to have sufficient low speed performance in take-off and landing situations [1]. A high-lift system is desirable to enhance the lift as well as the lift-to-drag ratio. In the past, the linearized theory method and wind tunnel test have been employed for estimation and optimization of the aerodynamic performance for fighter and SST configurations to determine deflection angles of the simple hinged-flap system [2]. However, for complex flow with boundary layer separation, vortical flow around the high lift devices of a SST configuration, the linear method cannot provide sufficient accuracy in estimating aerodynamic performance. Wind tunnel testing is capable to determine performance of scaled models, but it is limited by scale effect, has high cost and time consumption. On the other hand, with the development of computer and numerical algorithms, computational fluid dynamics (CFD) has been widely accepted as an important role for aircraft design to shorten the overall design process and reduce cost. CFD is capable of simulating flow-field at lower cost as compared with wind tunnel testing, and providing more reasonable accuracy than linear methods.

However, in the design process, the number of objective estimation for different sets of design variables is generally very large and takes long time. In the design of high-lift devices using CFD, a large number of geometries with different combination of design variables are required for parametric study or shape optimization to improve aerodynamic performance [3]. For a real problem, the optimization design process must be carried out rapidly in acceptable cost. By using numerical simulations to estimate aerodynamic performance, the difficulties are typically involves: (a) geometric shape and computational mesh modification, (b) numerical simulation for the complex flow, (c) optimization in the design space. An efficient method with lower cost and reliable accuracy is desirable in a real design process.

To reduce the cost in the design process, an efficient optimization method was proposed in this study. The high fidelity CFD was used and carried out on a parallel high performance computer to estimate the aerodynamic performance of the SST configuration. An automatic tool was developed, and time cost for shape modification and grid generation was dramatically reduced. A polynomial-based response surface model was applied to approximate the objective functions in the design space. A limited number of numerical simulations were needed to construct a database for the response surface model. The response surface method can not only approximately estimate values of the objective function but also provide further information of sensitivity and contribution of each design parameter with respect to the objective function.

In this study, a cranked-arrow wing with multi-segmented leading-edge (LE) vortex flap was designed. The objective of this study is to develop an efficient optimization method to design the deflection angles of the multi-segmented LE vortex flaps to improve aerodynamic performance of an SST configuration, and investigate effects of the flap segment on the aerodynamic performance at take-off and landing conditions.

2 High Lift System

To improve aerodynamic performance of the next generation SST at takeoff and landing situations, a high-lift system is desirable to enhance the lift as well as the lift-to-drag ratio. Increased aerodynamic performance can reduce engine power setting and therefore community noise. One approach to improve the high-lift performance of a highly-swept wing at subsonic conditions was suggested to use a LE vortex flap [4]. As shown in Fig. 1 [5], a suitable vortex flap locates the LE vortex system on the forward facing area of a deflected flap, thus creates a low pressure region on the flap surface and results in a net thrust component. The trailing-edge (TE) flap can be used to increase the effective camber of the wing and therefore the lift. Thus, deflection of the TE flap reduces the angle of attack required to produce the design lift, decrease the LE separation, and results in less drag.

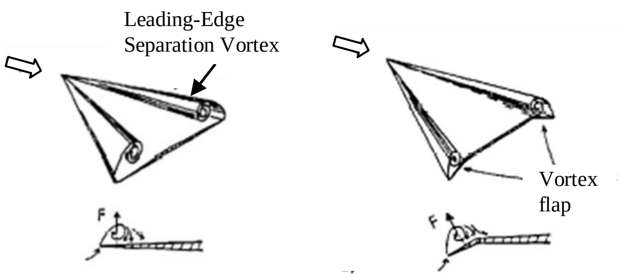
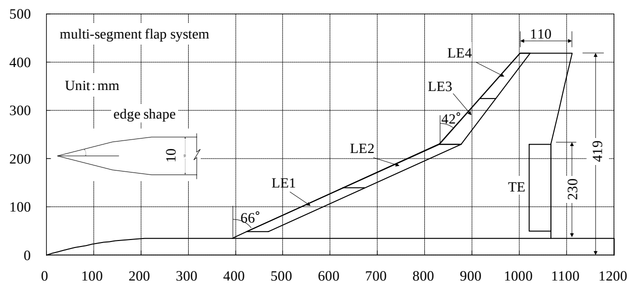


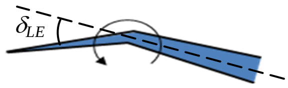
Fig. 1 Concept of vortex flap (from Rinoie and Stollery [5])

Owing to the spanwise variation of upwash, it was expected to obtain more effects of the LE vortex flap by locally deflecting the leading edge of the wing. Rao [6] used flap segmentation to reduce flap area while achieving the same L/D as the simple flap without segmentation. He also was the first to explore using vortex flap deflections on individual wing panels to produce roll control. It showed that segmented LE flaps had effects to improve the lift-to-drag ratio and reduce pitch instability.

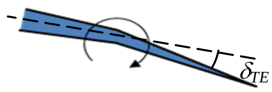
The geometric characteristics of the configuration model used in this study were shown in Fig. 2. The baseline model was designed preliminarily for a supersonic experimental airplane in Aviation Program group of Japan Aerospace Exploration Agency (JAXA) [7]. It consisted of an axisymmetric body and a cranked-arrow wing with segmented LE and TE flaps. The nose of the body was ogival and the circular cylinder had a diameter 70 mm. The mean aerodynamic chord (MAC) of the wing was 459mm. The leading edge was cranked at 55% semi-span station of the wing. The inboard wing was designed as a subsonic leading edge with a sweepback angle 66° to reduce the wave drag at the design Mach number 2. The leading edge of the outboard wing was swept back by an angle 42° to increase the aspect ratio and improve the low-speed performance. The aspect ratio (AR) of the wing was 2.42. In the study of the segmented flap system, the wing was a flat wing with a constant thickness 10mm, and all edges were sharpened by a vertical angle of 30° normal to the edges. The configuration with zero flap



(a) LE and TE flaps



(b) LE deflection



(c) TE deflection

Fig. 2 The model of SST high-lift configuration.

deflections is called baseline in this paper. The range of lift coefficient of interesting is from 0.4 to 0.6 for an SST at take-off and landing conditions.

Due to structure constraints, the number of segments was restricted. The LE flap systems were divided by 4 segments along the leading edge of the wing as shown in Fig. 2. Each deflection of the LE flaps was obtained by simply rotating the flap along the hinge line to the specified deflection angle. Four design parameters were set to be the responding deflection angle of each segment. Leading-edge flap segments were numbered consecutively from inboard to outboard. Deflections of the TE flaps were fixed by specified angles, 10° for the inboard TE flap and 0° for the outboard TE flap. All of the flaps were segmented with streamwise cuts. To prevent mesh cells from negative volume and skewness in generation of structured mesh, the gaps between adjacent flap segments and main wing were set to be 5 mm in spanwise to ease the mesh generation. In other words, the LE flap of the computational model was short by a total of 40mm (about 5% of the semi-span) than that of the real one. This resulted in a little difference in the computation as compared with the corresponding experiments.

3 Design Consideration

The design procedure is shown in Fig. 3. The first stage was to generate a database to construct an initial response surface model. An optimal Latin hypercube method [8] was used to sample sets of design variables in the design space. It retained the design space topology with relatively few design variations uniformly distributed in the specified design space. Then, the geometric shape and computational mesh modification was conducted by an automatic tool for all cases with different combination of the LE deflection angles. The third step was to compute the flow field around the configuration and calculate the aerodynamic forces by CFD. Once the database was generated, coefficients of the 2^{nd}-polynomial could be determined by the least squares method, and the response surface model was then constructed to approximate aerodynamic forces. The "best" design, where the drag was the minimum at the specified design lift, was searched on the response surface by using an optimization solver. The response surface could be updated by adding new designs into the database to further improve the aerodynamic performance.

In the design space, the points necessary to construct the response surface are typically close to the real optimum solution. For these reasons, the number of design variations and the cost of numerical simulation for the flow field are in general, rather small. This approach always produces better performance progressively to the optimum solution.

3.1 Flap Modification

Usually, in the design process, a large number of shape and mesh generation have to be conducted for different configurations. Especially for CFD using a structured mesh, it spends designer's much labor for hand operation and severally affects the design progress. For example, it usually costs about one month to generate a new structured mesh for a complete aircraft configuration. Even though the CFD solver using structured meshes has those difficulties for complex configuration, it may provide higher accuracy in calculating flow field and aerodynamic performance.

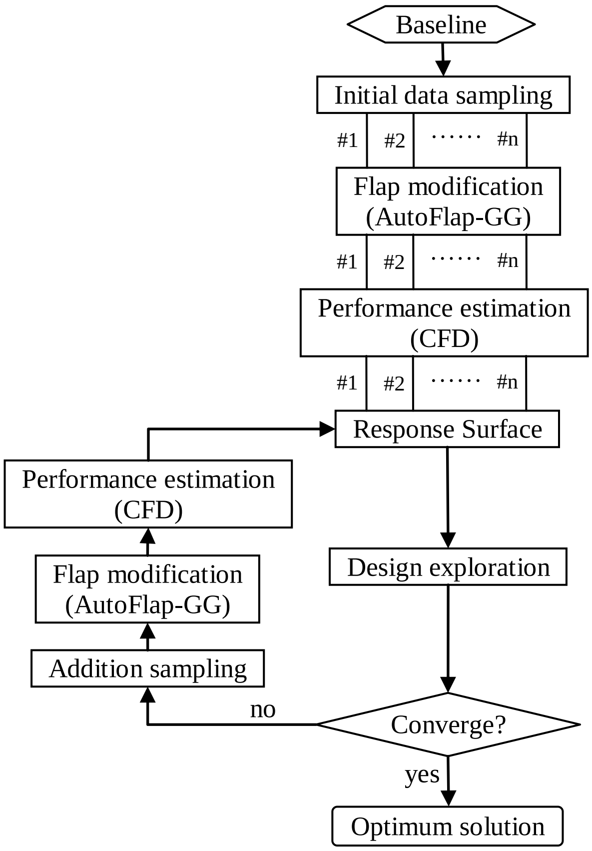
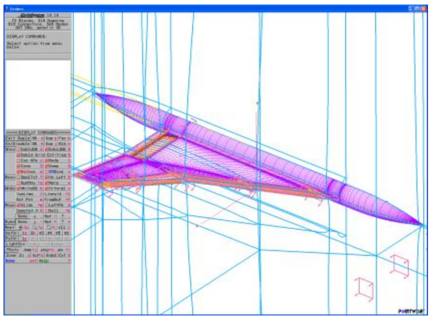


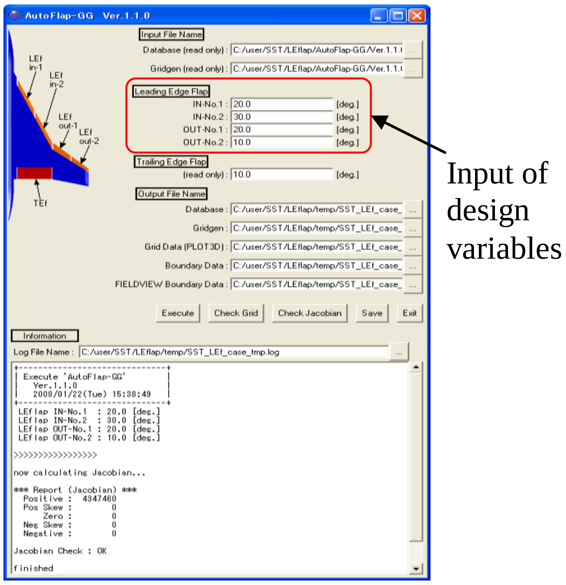
Fig. 3 Flow chart of the design procedure.

To shorten time for flap modification, an automatic mesh generator [9] was developed to modify the surface shape and computational mesh automatically for a supersonic transport configuration as the LE and TE flaps were deflected. In this AutoFlap-GG, the unique script language Glyph included in the grid generation software, Gridgen [10], was used to automatize the process of shape modification and mesh generation.

As the deflection angles of the LE flaps were changed, it usually costs about one week to modify the shape and mesh around the SST configuration. By using the automatic tool, what needed to do is only to input specified values of the design variables in the control panel as shown in Fig. 4. Then, the script automatically executes all of the process to modify the shape and generate a mesh only in a few minutes. Therefore, the total time taken in the design process is dramatically reduced. The automatic process also minimizes the numerical error due to mesh differences generated by hand operation in the flap modification.



(a) Interface of mesh generation

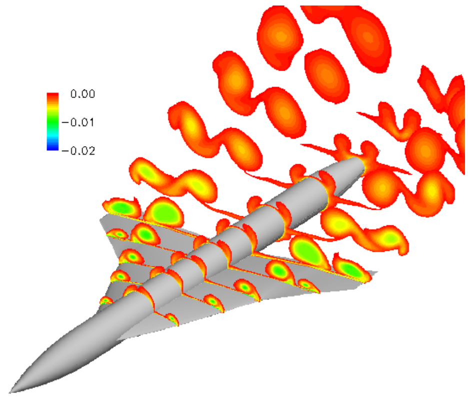


(b) control panel for automatic process

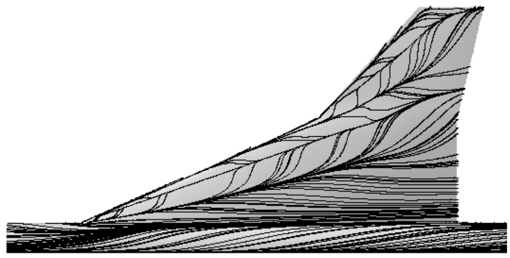
Fig. 4 AutoFlap-GG: an automatic tool of shape and mesh modification for SST high-lift configuration.

An H-H topology grid system was generated around a half model of the configuration with a 5m width and 5m height. Grids were extended to 3m both upstream to the nose and downstream to the body tail. Grid points were carefully distributed in the normal direction of the surface to resolve the vortex structure. The spatial grid near the surface was forced normal to the configuration surface and the normal spacing of the first point from the surface was set to 1.0×10^{-5} MAC. The final mesh was divided into 72 blocks, and contained about 4.3 million grid points.

3.2 Flow Simulation



(a) total pressure distributions



(b) oil-flow pattern on the upper

Fig. 5 CFD for flow simulation at angle of attack 12°

Numerical simulation of high lift flows by solving the Reynolds-averaged Navier-Stokes (RANS) equations plays an important role in the design process of an aircraft, and is now often used in analysis and design of complex configurations. It can provide details of flow structure and help designer to understand flow physics. Although CFD has not such limitations of wind tunnel testing, the use of CFD in the design is strongly dependent on the computer power and the computing accuracy.

In this study, an in-house flow solver, AeroDynamic Computational System (ADCS) [11], which was developed in JAXA's supersonic transport project, was used to simulate the flow field and estimate the aerodynamic performance. Computation has been validated in good agreement with experimental flow visualization, aerodynamic characteristics and surface pressures as compared with wind tunnel experiments [12]. The compressible RANS equations were descretized on structured mesh using a finite difference method with multi-block technique. To simulate effect of turbulence, Menter's shear stress transport model was used. It used the third-order TVD (total variation diminishing) of Chakravarthy-Osher scheme for convection terms, central difference for other terms, and a diagonalized implicit method LU-ADI for time integration. On the solid surface, the no-slip condition was applied to the velocities and the normal derivative of pressure was forced to be zero.

As an example of numerical results for the baseline configuration without flap deflections at the angle of attack 12°, distributions of total pressure loss are shown Fig. 5(a) at several stations. It can be seen that both the inboard and outboard LE vortices are continuously fed with vorticity by separated shear layer from the leading edge, bent at the kink and wing tip, and eventually convects longitudinally downstream. The large the separation, the more the total pressure is lost. As the leading-edge flaps are deflected, the level of total pressure loss is reduced, and the total pressure is less lost. Computed oil-flow patterns on the wing upper surface are given in Fig. 5(b). Converged lines represent separation and diverged ones represent reattachment. The primary and secondary vortices can be clearly identified by the limiting streamlines on the surface. The inner vortices are originated from the intersection point of the wing leading edge and the body. And the outer vortices are originated from the kink of the inboard and outboard wing. Both inner and outer vortices are separated from the leading edge.

3.3 Response Surface Model

Instead of a large number of numerical simulations, the response surface method can dramatically reduces the number of design variations need to estimate the objective function in the design process. In this study, to understand effects of the LE vortex flap, the deflection angles of the LE flap segments were defined as design variables, while the TE flap was fixed to a specified deflection angle 10°. The aerodynamic forces were dependent on the angle of attack and the deflection angles of the LE flap segments.

The basis of the optimization method was that measurement of aerodynamic performance of the SST configuration with LE and TE flaps varies as a polynomial function of the deflection angles that define the design space. In this study, the 2nd-polynomial, as the following, was used to construct a response surface model and approximate points not calculated by CFD.

$$f = a_0 + \sum_{i=1}^{n} a_i x_i + \sum_{i=1}^{n} a_{ii} x_i x_i + \sum_{i=1}^{n-1} \sum_{j=i+1}^{n} a_{ij} x_i x_j + \varepsilon \qquad (1)$$

where x_i is the deflection angle, i and j is the number of the LE flap segment, a_i and a_{ij} are the coefficients to be determined, and ε is the error.

The approximate function f was called response surface of the objective with respect to the design variables. The maximum (or minimum) could be computed using an optimization solver allowing for the search of the "best" design. Because aerodynamic forces were much more sensitive to the angle of attack than the deflection angle of LE flap, the approximation of response surface was divided into two steps to obtain better accuracy of the polynomial. Flow field around each configuration with a set of design variables were simulated by CFD at three different angles of attack, i.e., 8°, 10° and 12°. First, the force coefficient was approximated by the different polynomials as a function of the deflection angles of LE flap segments corresponding to each angle of attack, respectively. Then, the aerodynamic forces were approximated by a parabolic function which was depended on the angle of attack. Finally, the optimum point with a minimum drag coefficient (C_D) at the design lift coefficient ($C_{L,\mathrm{des}}$=0.5) was searched on the above response surface by Newton's gradient method.

4 Results

The design method was applied to design the leading-edge deflection angles of the cranked-arrow wing of an SST configuration at low speeds. Flow conditions were referred to the experiments carried out by Kwak, et al [12] in a

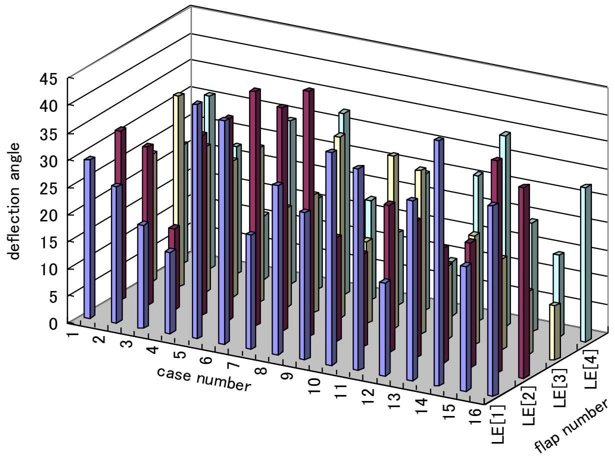


Fig. 6 Initial sampling points for constructing the response surface.

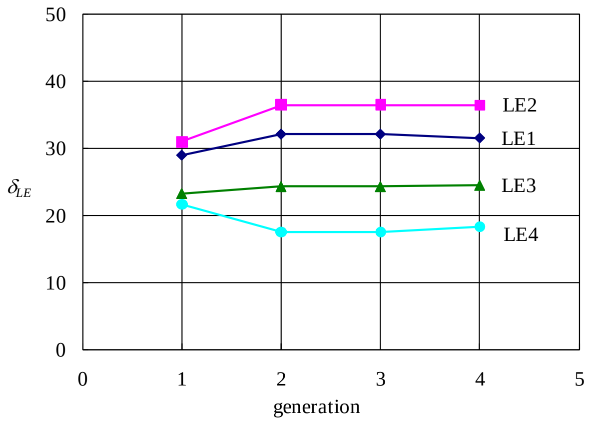


Fig. 7 Convergence history of the optimum LE deflection angles.

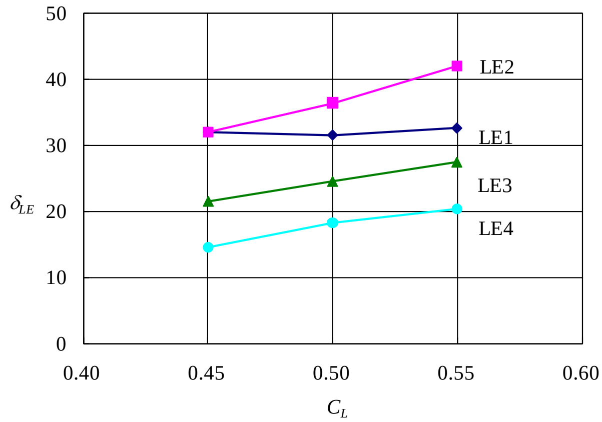


Fig. 8 The LE deflection angles vs. $C_{L,\mathrm{des}}$

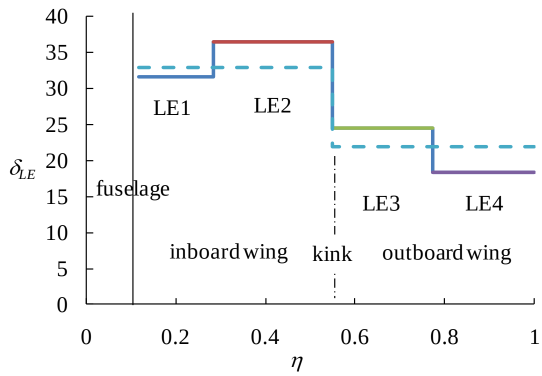


Fig. 9 Optimum deflection angle schedule of the LE flap segments at C_L=0.5. Solid line: all LE flaps were deflected separately; Dash line: both LE inboard and outboard flaps were not segmented.

low speed wind tunnel of JAXA. Wind tunnel tests were ranged in -4° ≤ α ≤ +30° angles of attack at the air speed of freestream U_∞ =30m/sec and Reynolds number 0.945×10^6 based on MAC. In this study, the optimization design was subjected to find a set of design variables, which has the minimum drag coefficient C_D at the design lift coefficient $C_{L,\mathrm{des}}$=0.5. It would result in the maximum lift-to-drag ratio and improve the aerodynamic performance.

According to the past experience, ranges of the deflection angles were set to 15°~45° for the inboard segments, and 10°~35° for the outboard segments. 16 sets of design variables were selected by the optimal Latin hypercube method, and uniformly distributed in the specified design space, as shown in Fig. 6.

The 16 points of the first generation were used to construct the initial response surface. Then the optimum point was explored by the optimizer and added to the database as the second generation. The design cycle finished as the optimum point converged. Convergence history of the design variables are shown in Fig. 7. The optimization design converged rapidly, and only three new generations were added in the design process.

In Fig. 8, the optimum deflection angles of the LE flap segments were compared at different values of the design lift coefficient. In the range $0.45 \le C_L \le 0.55$, besides the inboard LE1 flap segment, the deflection angles of other segments were increased with the design lift coefficient C_L. The leading-edge vortex was

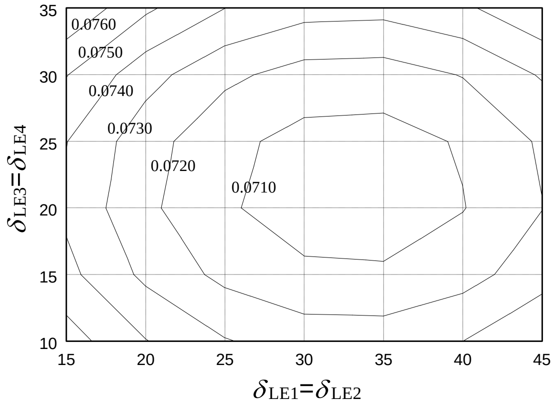
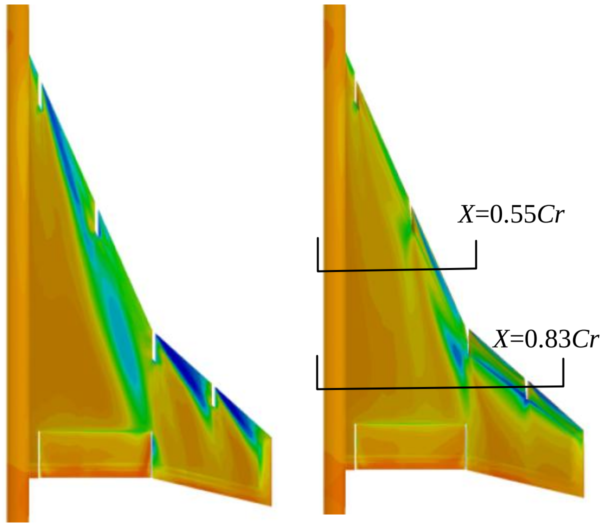


Fig. 10 Drag contour map for LE deflection angles at C_L=0.5.

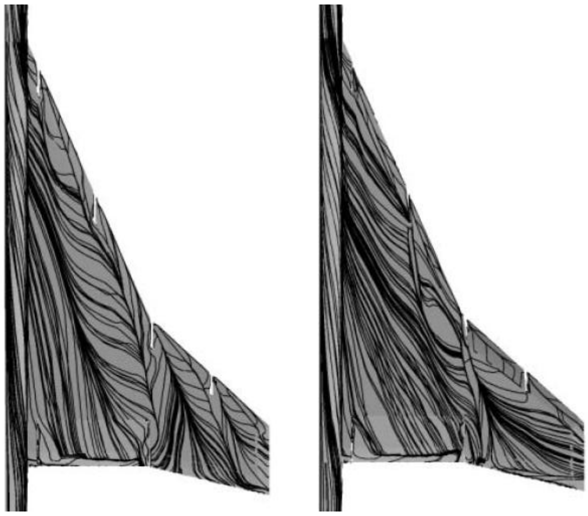
originated from wing apex and fed by the vorticity of the leading-edge separation. At low angles of attack, because vortex generated from the LE1 inboard segment was weak in strength and small in size, the LE1 inboard segment can be more deeply deflected to obtain more potential flow on the wing than others.

The optimum schedules of the LE deflection angles at the design lift coefficient C_L =0.5 were shown in Fig. 9. The deflection angle of the LE2 was larger than that of the LE1 to suppress large separation. The deflection angles of outboard segments LE3 and LE4 were smaller than those of the inboard ones because the outboard leading edge was less swept. To discuss segmentation effect on the aerodynamic performance, optimum result of the four LE flap segments were compared with that of the two LE flap segments, when the inboard and outboard flaps were not segmented. It showed that the deflection angle became smaller for the LE1 segment and larger for the LE2 segment as the inboard flap was segmented. Similarly, the deflection angle became larger for the LE3 segment and smaller for the LE4 segment. As calculated at the same design C_L=0.5, the difference was 2.2 drag counts between the drag coefficients of the configurations with and without segmentation. It hoped that more flap segments would have much more benefit to the aerodynamic performance. Furthermore, in Fig. 10, a contour map for LE deflection angles at C_L=0.5 were provided by estimating aerodynamic forces on the response surface. It indicated that an optimum solution did exist in



(a) without deflection (b) with optimum deflection

Fig. 11 Pressure distributions on the upper side surface at the angle of attack 10°.



(a) without deflection (b) with optimum deflection

Fig. 12 Oil-flow patterns calculated on the upper side surface at the angle of attack 10°.

(a) without deflection at X=0.55Cr (b) with optimum deflection at X=0.55Cr

(c) without deflection at X=0.83Cr (d) with optimum deflection at X=0.83Cr

Fig. 13 Total pressure distributions at the angle of attack 10°

the design space. Near the solution of minimum drag, drag coefficient was more sensitive to the outboard flap than the inboard flap. Because the swept angle of the outboard leading edge was smaller than that of the inboard, it should have larger influence on vortex generation. It also found that the contour lines were close a cluster of ellipse, and indicated that interference effects between inboard and outboard LE flaps was generally small for this wing in the range of the deflection angles specified in this study.

As shown in Fig. 11 and Fig. 12, large separation was observed both on the inboard and on the outboard in the case of no LE deflection. The vortex became stronger as it convected outward along the leading edge. Distributions of total pressure lose at streamwise stations X=0.55Cr and X=0.83Cr were shown in Fig. 13. Without LE deflection, large vortices were generated and induced secondary vortices near the leading edge. By deflection of the LE flaps, the LE vortices generated on each LE flap were significantly suppressed, largely reduced both in size and in strength. Low pressure regions were observed on the upper surface of the deflected flap segments. All of LE vortices were restricted on the flap upper surfaces and generated a net thrust component. It indicated that typical vortex flaps were achieved by the optimum LE deflection angles.

However, it was observed that small separations were additionally generated from the hinge lines of the LE2 and LE3 flaps, and they were even larger than the LE separations. The hinge line separations were located on the downstream of the hinge lines and resulted in drag thus obstructed more flap deflection. It is suggested that the hinge line separation should be suppressed in order to obtain better aerodynamic performance.

5 Conclusions

An efficient design method was proposed for a multi-segmented flap system of supersonic transport at take-off and landing conditions. Results showed that this optimization method was efficient and dramatically reduced cost in the design process.

An automatic tool was developed for the LE flap segments to modify geometric shape and computational mesh as deflection angles changed. It dramatically reduced cost and labor of hand operation. Flow field around the configuration was simulated and aerodynamic forces were calculated by solving the Reynolds-averaged Navier-Stokes equations with MPI parallel programming. In the design space, a response surface based on the 2nd- polynomial approximation was constructed by using a number of design variations sampled by an optimal Latin hypercube method. The response surface model rapidly estimated aerodynamic force and optimum solutions were explored. At the same time, it was updated and improved accuracy by adding new generations progressively in the design process. The response surface method largely reduced the number of sampling cases needed in design.

A cranked-arrow wing with multi-segmented flaps of a supersonic transport configuration was optimized by the proposed design method. It was confirmed that aerodynamic performance was improved by the optimization design, and the design method was efficient to reduce design cost. Leading edge vortices were largely suppressed as the LE flap segments were deflected by the optimum deflection angles, but the hinge line separation was found to influence on the flap deflection. The multi-segmented LE flap had better aerodynamic performance than the simple LE flap.

Acknowledgments

The authors gratefully acknowledge support for this research from colleagues working in the SST team of JAXA.

References

[1] Antani, D.L. and Morgenstern, J.M. HSCT high-lift aerodynamic technology requirements. *AIAA paper* 92-4228.

[2] 1998 NASA High speed research program aerodynamic performance workshop. *NASA CP*-1999-209682.

[3] Herrmann, U. Low-speed high-lift performance improvements obtained and validated by the EC-project EPISTLE, *24th International Congress of the Aeronautical Sciences*, ICASE-2004-411, August 29 – September 3, Yokohama, Japan.

[4] Rao, D.M. Exploratory subsonic investigation of vortex flap concept on arrow-wing configuration. *Supersonic Cruise Research'79*, Part I, NASA CP-2108, 1980.

[5] Rinoie, K. Experimental studies of vortex flaps and vortex plates, Part 1 0.53m span 60° delta wing. *NAL TR-1140T*, published by National Aerospace Laboratory of Japan, March 1992.

[6] Rao, D. M., Segmented vortex flaps. *AIAA* 83-0424, January 1983.

[7] Yoshida K and Makino Y. Aerodynamic design of unmanned and scaled experimental airplane in Japan. *ECCOMAS 2004*, Jyväskylä.

[8] Mckay, M.D., Beckman and R.J., Conover, W.J. A comparison of three methods for selecting values of input variables in the analysis of output from a computer code. *Technometric* Vol.21, No.2, 1979, pp.239-245.

[9] Nagata, Y. and Lei, Z. Automatic grid generator for flap deflection of supersonic transport configuration. JAXA-RM-09-003, published by Japan Aerospace Exploration Agency, Sept. 2009.

[10] Pointwise inc., Gridgen User Manual, Version 15, 2006.

[11] Lei, Z. Flow simulation of a supersonic transport configuration at low-speed and high-lift conditions. *Journal of Aircraft*, Vol.45, No.5, pp1514-1521, 2008.

[12] Kwak, D.-Y., Rinoie, K., and Noguchi, M. Experimental research of aerodynamics on an SST configuration with high lift devices. *International Council of the Aeronautical Sciences*, Paper 2006-5.11.3, Hamburg, Germany, Sept. 2006.

Copyright Statement