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Airfoil Design for Martian Airplane Considering Using Global Optimization Methodology

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Abstract : To design airfoils for novel airplanes, new knowledge of aerodynamics is required. In this study, modified Parametric SECtion (PARSEC) which is a airfoil representation is applied to airfoil design using a multi-objective genetic algorithm to obtain an optimal airfoil for consideration in the development of a Martian airplane. In this study, an airfoil that can obtain a sufficient lift and glide ratio under lower thrust is considered. The objective functions are to maximize maximum lift-to-drag ratio and to maximize the trailing edge thickness. In this way, information on the low Reynolds number airfoil could be extracted efficiently. The optimization results suggest that the airfoil with a sharper thickness at the leading edge and higher camber at the trailing edge is more suitable for a Martian airplane. In addition, several solutions which has thicker trailing edge thickness were found.

Key Words : Airfoil Representation, Aerodynamic Design, Martian Airplane

1. Introduction

Because computational fluid dynamics (CFD) has been widely used to design airfoils, more efficient airfoil representations are required. One popular representation method for automated airfoil/wing design is the PARSEC airfoil representation [1]. However, its applicability to other conditions such as low Reynolds number should be investigated, because this method was originally parameterized for transonic flow. The performance of the PARSEC airfoil representation is still low around the leading edge, it may be difficult to represent thin airfoil/wings that have a higher camber for low Reynolds number flows. To design the generic airfoil, modification of PARSEC method has been proposed [2] and better design result was shown for low Reynolds number.

Recently, many Mars exploration projects have been carried out around the world. For example, in Japan, the Institute of Space and Aeronautical Science (ISAS) /Japan Aerospace Exploration Agency (JAXA) proposed the Mars Exploration with a Lander–Orbiter Synergy (MELOS) [3] project. This project includes a Martian airplane to explore Mars, although many other exploration methods (by the orbiter, by the lander, etc.) are considered. The Martian airplane gives the possibility to obtain high-quality and wide-scale geological information; however, the development of the Martian airplane requires the use of a brand-new airfoil under unknown aerodynamic conditions. In previous study [2], several solutions which achieve better aerodynamic performance. However, they do not have enough thickness especially around their

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trailing edge. Such designs have structural and manufactural problem.

In this study, modified PARSEC method is applied to airfoil design using a multi-objective genetic algorithm (MOGA) to obtain an optimal airfoil for low Reynolds number flows for application on a Martian airplane in consideration of the trailing edge thickness effect. The performances of designed airfoils are compared with Ishii airfoils [4], which are the high-performance airfoils for the hand-launch glider.

2. Airfoil Representation Methods

The original PARSEC method [1] represents a supercritical airfoil based on a polynomial function. This representation is effective to design the transonic airfoil. However, it is have several problems to apply to not transonic flow. Although the center of leading edge radius r_{le} is originally defined on a camber line, the original PARSEC method postulates that r_{le} is on the chord line. Therefore, the representation performance of an airfoil with the original PARSEC method with a high camber around leading edge is low. To improve this representation performance around the leading edge, the airfoil's thickness and camber should be separately defined, as shown in Figs. 1 and 2. The thickness distribution is defined by Eq. (1).

$$z = 2\sqrt{2r_{le}x} + \sum_{n=1}^{5} a_n x^{\frac{2n-1}{2}} \qquad (1)$$

The camber distribution is defined by Eq. (2).

$$z = \mathrm{sgn}(r_c)\sqrt{2|r_c|x} + \sum_{n=1}^{5} b_n x^n \qquad (2)$$

The a_n and b_n values of Eqs. (1) and (2) are determined from 12 parameters, as shown in Fig. 1 and 2.

It is important to control the leading-edge (LE) camber (which can control the LE suction) and the trailing-edge (TE) camber (which can maintain Kutta’s condition) for the purpose of drag reduction (or l/d improvement). A square root term is added to the camber’s representation (see Eq. (2)) to improve the representation performance around the leading edge. This square root term includes the radius of the camber at the leading edge, r_c. The term “sgn” in Eq. (2) denotes the sign (signum) function. The large coefficient of the first term of Eq. (2) gives weight to the design of the LE camber.

In this study, the modified PARSEC method was used to describe a case in which r_c does not become negative (“Case1”) and a case in which r_c does become negative (“Case2”). The design results obtained for Case1 and Case2 were compared, and the effect of the representation of the camber around the leading edge was investigated.

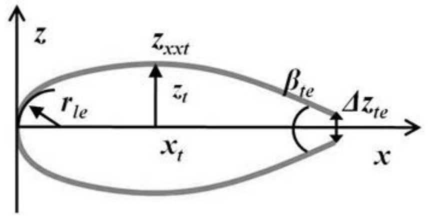


Fig. 1 Definition of thickness distributions

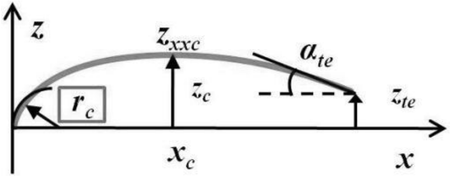


Fig. 2 Definition of camber distributions

3. Optimization Method

3.1. Non-dominated Sorting Genetic Algorithm-II (NSGA-II)

Genetic algorithms are popular heuristic optimization techniques that use operators such as selection, crossover, and mutation. The non-dominated sorting genetic algorithm-II (NSGA-II) [5] is employed in this study. NSGA-II is characterized by non-dominated sorting and crowding distance sorting. Based on this sorting combining elitism, the new generation is filled until the population size exceeds the current population size.

3.2. Parallel Coordinate Plot (PCP)

A parallel coordinate plot (PCP) is a statistical visualization technique used to convert high-dimensional data into two-dimensional graphs. [6] To generate a PCP, the attribute values in the design problem, such as the design variables, objective functions, and constraint values, have to be normalized for comparison along the same axis. After normalization, the axes are arranged in consistent parallel lines. In general, the distances between each line and the next are equivalent. Using a PCP, it is easy to inspect the design problem at a glance.

3.3. Computational Fluid Dynamics (CFD)

In this study, the aerodynamic evaluation was carried out using a two-dimensional Reynolds-averaged Navier-Stokes solver (RANS). A lower-upper symmetric Gauss-Seidel (LU-SGS) implicit method is employed for time integration, and a third-order-accurate upwind differential scheme [7] with a monotone upstream-centered scheme for conservation laws (MUSCL) method is employed for the flux evaluation. The Baldwin-Lomax model is used as a turbulent model. Because the convergence of the fully turbulent solver is faster than that of the laminar flow solver, the turbulent flow solver is employed in the MOGA process. After the optimum solutions are obtained, their performances are also investigated using the laminar solver, because this study considers low Reynolds number flow. In this study, a structured grid is employed for space discretization. The grids are automatically created by an algebraic method; the structured grid for the RANS is 128 × 61 (see Fig. 3).

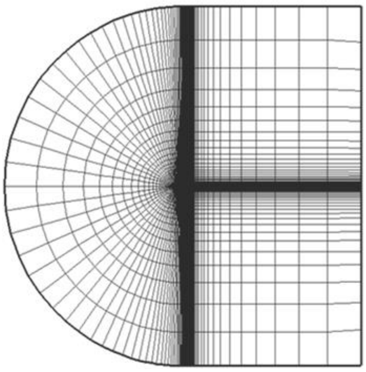


Fig. 3 Grid resolutions for MOGA evaluation

4. Formulations

In this study, the proposed method is applied to airfoil design for the Martian airplane. To acquire an airfoil that achieves a high glide ratio and higher structural performance under the unknown conditions of the Martian atmosphere, two objective functions are considered: one is to maximize the maximum lift-to-drag ratio (*maxl/d*), and another is to maximize thickness at 75% chord length (*th75*.)

$$\begin{cases} \text{Maximize } th75 \\ \text{Maximize } maxl/d \\ \quad \text{Subject to } t=7\%\text{c } t_{TE}=0.01 \end{cases} \tag{3}$$

Because Δz_{te} is fixed to 0.01 and the airfoil thickness is fixed to 7%c, numbers of parameter 10 are used in the proposed representation. The computational conditions are Mach number M = 0.2 and Reynolds number $Re = 2.3 \times 10^4$.

5. Results

5.1. Exploration Results by Means of NSGA-II

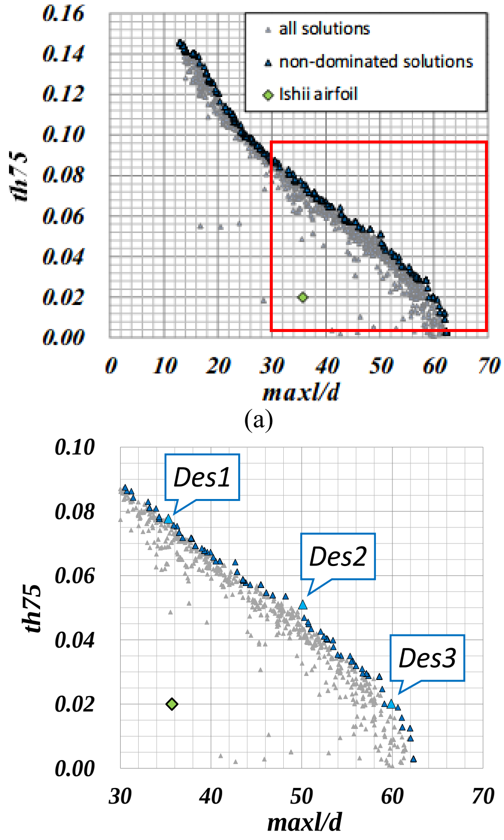
All solutions and non-dominated solutions obtained by the MOGA are shown in Fig. 4(a). Figure 4(b) shows a close up view of the solutions that achieve high aerodynamic performance, and Des1, Des2, and Des3 are selected for investigation of geometries and flow fields. *th75* of Des1 is four times higher than that of the Ishii airfoil while the *maxl/d* values are equal. The *maxl/d* of Des3 is 1.7 times higher than that of the Ishii airfoil while Des3 has the same th75 as Ishii airfoil.

Figure 6(a) shows all solutions obtained by MOGA colored according to the value C_l, and Fig. 6(b) shows the solutions that are colored according to the value C_d. According to Fig. 6(a), it is difficult to find a clear relationship between C_l and objective functions. On the other hand, Fig. 6(b) shows the relationship between C_d and objective functions, i.e., higher *maxl/d* indicates lower C_d. This suggests that C_d is important for the maximization of *maxl/d.*

5.2. Design Examples

Airfoil geometries, cambers and pressure coefficient (C_p) distributions of the selected airfoils are shown in Fig. 6. According to Fig. 6, all the selected airfoils have sharper leading edge than that of the Ishii airfoil. Des3 shows the highest *maxl/d* among Des1, Des2, Des3, and Ishii airfoil, and it also shows the smallest thickness distributions. However, Des3 has the highest camber around the trailing edge, and it is similar to Des1 and Des2. Therefore, the airfoil that has high camber around the trailing edge can achieve high aerodynamic performance while th75 is large.

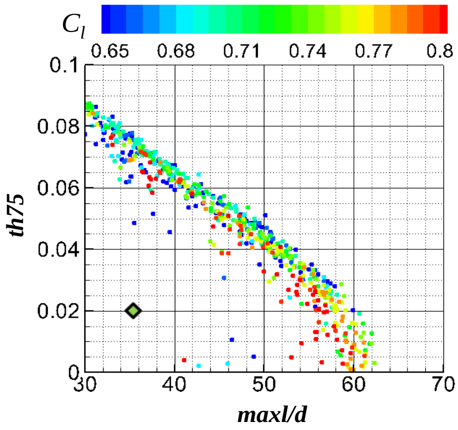
l/d vs. angles of attack (*AoA*) of the Des1-3 and the Ishii airfoil are investigated and compared in Fig. 7. Des1 has the largest *th75* among the selected designs and the Ishii airfoil. *l/d* of Des1 gradually increases. It has the capability to carry out robust aerodynamic performance for the wing of the Martin airplane.



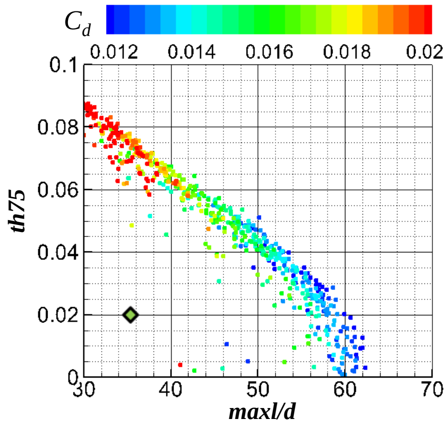
(a)

(b)

Fig. 4 Solutions obtained by MOGA. (a) All solutions, and (b) Close up view of those whose *maxl/d* is over 30.0 and selected solutions

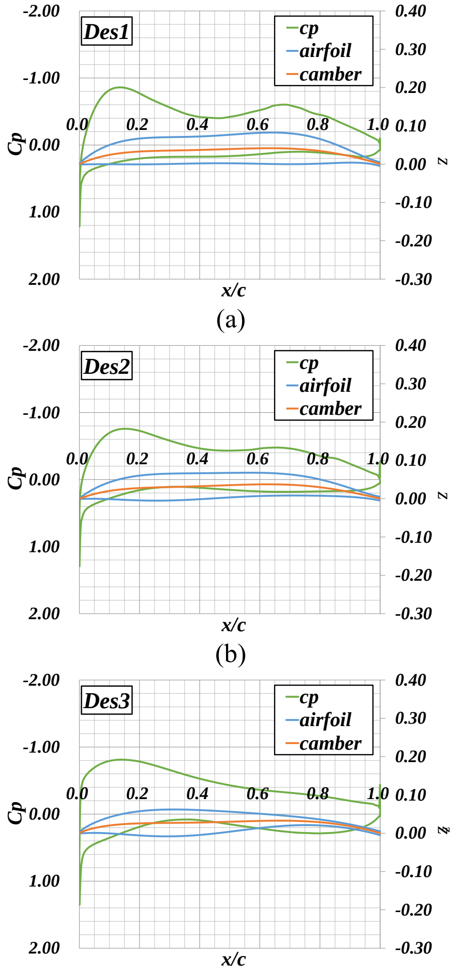


(a)



(b)

Fig. 5 All solutions. (a) Colored by C_l, and (b) Colored by C_d



(a)

(b)

(c)

Fig. 6 Comparison of C_p distribution among Des1, Des2, and Des3. (a) Des1, (b) Des2, and (c) Des3

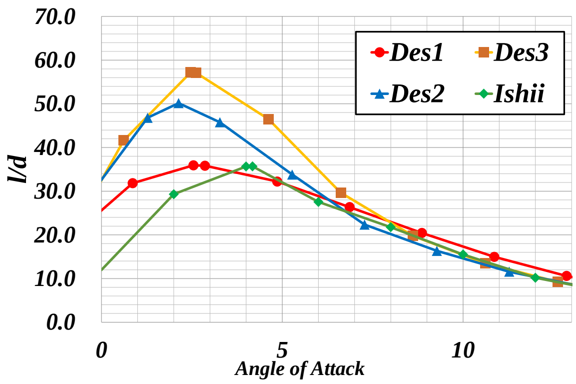


Fig. 7 *l/d* against AoA of Des1, Des2, Des3, and the Ishii airfoil

4. Conclusions

In this study, airfoil design optimization was carried out to improve the aerodynamic performance with a thick trailing edge. To solve this design problem, maximization of the maximum lift-to-drag ratio and the thickness at the 75% airfoil chord was firstly solved by MOGA. According to the design result, the design achieved high aerodynamic performance when the trailing edge thickness was four times larger than that of the Ishii airfoil.

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