Topology optimization of compliant adaptive wing leading edge with composite materials

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Abstract
An approach for designing the compliant adaptive wing leading edge with composite material is proposed based on the topology optimization. Firstly, an equivalent constitutive relationship of laminated glass fiber reinforced epoxy composite plates has been built based on the symmetric laminated plate theory. Then, an optimization objective function of compliant adaptive wing leading edge was used to minimize the least square error (LSE) between deformed curve and desired aerodynamics shape. After that, the topology structures of wing leading edge of different glass fiber ply-orientations were obtained by using the solid isotropic material with penalization (SIMP) model and sensitivity filtering technique. The desired aerodynamics shape of compliant adaptive wing leading edge was obtained based on the proposed approach. The topology structures of wing leading edge depend on the glass fiber ply-orientation. Finally, the corresponding morphing experiment of compliant wing leading edge with composite materials was implemented, which verified the morphing capability of topology structure and illustrated the feasibility for designing compliant wing leading edge. The present paper lays the basis of ply-orientation optimization for compliant adaptive wing leading edge in unmanned aerial vehicle (UAV) field.

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1. Introduction

The morphing wing is a novel aircraft wing that can improve aircraft aerodynamics performances for different missions by changing its shape. A multitude of reports shows that the morphing wing can obviously increase the lift and decrease the drag. It is a significant to the morphing wing that changing wing leading edge shape. The morphing wing of aircraft F-111 has been designed and tested by air force research labs (AFRL) at the Wright Patterson air force base (WPAFB). The superior aerodynamic performances of the morphing wing have been proved by the nice maneuverability of aircraft F-111. The leading edge of this wing was designed using conventional rigid-link mechanism. Whereas, the weight and complexity of this mechanism limited the wide application of morphing wing leading edge. The compliant mechanism, which gains some or all of its mobility from the elastic deformation, has advantages of lighter weight and simpler structure compared with conven-
tional rigid-link mechanism.3,4 So far, compliant mechanism has been extensively highlighted in morphing wing area.

In the recent years, topology optimization and finite element method were generally applied to designing the distributed compliant mechanisms.5,6 In 1998, a compliant morphing wing leading edge was firstly designed, fabricated and tested using compliant mechanism. This compliant leading edge realized the change from 0° to 6° in camber.7 Wind tunnel test results showed 51% increase in lift-to-drag ratio and 25% enhancement in the lift coefficient. Lu and Kota8 had proposed a design method to minimize LSE between the deformed and desired curves of compliant mechanisms. The compliant adaptive wing leading edge was achieved based on this method.10,11 de Gaspari and Ricci12 presented a two-level approach to optimize morphing wings with compliant structures. However, most of the existing compliant mechanisms are made of homogeneous metal material, which cannot achieve the high performance of compliant mechanisms. It remains a great challenge to develop the compliant adaptive wing leading edge with the capabilities of large changes and high load-carrying.

Composite materials are considered as good candidates of obtaining special performance due to their anisotropy, light weight, high specific strength and outstanding designability characteristic. Composite materials have been widely applied in the fields of aviation and aerospace. Yin and Ananthasuresh13 provided a gradually formed continuous peak function interpolation used in topology optimization with multiple materials. Ren et al.14 investigated topology optimization of micro compliant mechanisms with two materials. Xu et al.15 studied topology optimization of composite material with respect to minimization of the sound power radiations using the extended SIMP model. Li et al.16 studied a topology optimization of the compliant mechanisms with different composite materials by using the method of SIMP. Saxena17 introduced topology design of compliant mechanisms with multiple materials using genetic algorithm and the barrier assignment approach. These researches mainly focus on the building of multiple materials interpolation scheme model for the micro compliant mechanisms. The obtained topology results using the materials interpolation scheme were realized in theory due to the difficulty for manufacturing these results with multiple materials. As is well known, the laminated fiber reinforced epoxy composite plate was easily prepared and its mechanical performances can be remarkably improved by using different fiber ply-orientations. Little work has been reported on compliant adaptive wing leading edge using the laminated plates of fiber reinforced epoxy composite.

In the present paper, an equivalent constitutive relationship of laminated glass fiber reinforced epoxy composite plates is presented. The models of topology optimization of compliant adaptive wing leading edge with glass fiber reinforced epoxy composite are built using SIMP method. An objective function is used to minimize the LSE between deformed and desired shape. The compliant wing leading edges with different fiber ply-orientations are generated by optimality criteria (OC) method and sensitivity filtering technique. The corresponding physical model of composite and experiments are designed and manufactured. Meanwhile, the morphing capability of compliant wing leading edge and influences of fiber ply-orientations on topology structures are demonstrated.

2. Equivalent constitutive relations of laminated composite plates

Elastic properties of laminated fiber reinforced composite plates depend on each single layer material and ply-orientations. Therefore, the typical characteristics of laminated composite plates are anisotropy and outstanding designability. In the present paper, a symmetric laminated plate with orthotropic glass fiber reinforced epoxy as a single layer was adopted due to no coupling effect of tension and bend. A typical symmetric laminated composite plate is shown in Fig. 1. $\sigma_r$, $\sigma_y$, $\epsilon_r$, and $\epsilon_y$ are the exerted equivalent stresses of the laminate plate. Supposing that no slip occurs between the layers, the strain of each layer corresponding to the stress should be equal to $\epsilon_r$, $\epsilon_y$, $\epsilon_{xy}$ in the middle plane based on the symmetric laminated plate theory.

In laminated composite plates design fields, the exerted force of symmetric laminated composite plate is equal to the sum of forces from each single layer. Based on the symmetric laminated plate theory, the relationship of exerted stress and strain can be expressed as

$$
\begin{align*}
\bar{\sigma}_r &= \sum_{q=1}^{M} (t_q - t_{q-1}) \mathbf{Q}_{q} \begin{bmatrix} \epsilon_{0}^{r} \\ \epsilon_{0}^{y} \\ \epsilon_{0}^{xy} \end{bmatrix} \\
\bar{\epsilon}_{xy} &= \begin{bmatrix} \epsilon_{11} & \epsilon_{12} & \epsilon_{16} \\ \epsilon_{12} & \epsilon_{22} & \epsilon_{26} \\ \epsilon_{16} & \epsilon_{26} & \epsilon_{66} \end{bmatrix} \begin{bmatrix} \epsilon_{0}^{r} \\ \epsilon_{0}^{y} \\ \epsilon_{0}^{xy} \end{bmatrix} \\
D_{rs} &= \frac{t_r}{t} \sum_{q=1}^{M} \mathbf{Q}_{q} 
\end{align*}
$$

where $t$ is total thickness of laminated plate, $t_q$ thickness of $q$th layer, $M$ the total number of layers and $\mathbf{Q}_{q}$ an element of stiffness matrix in $q$th layer in global coordinate system.

According to the above hypothesis and symmetric laminated theory, the equivalent elastic constitutive relations of the whole symmetric laminated plate can be expressed as

$$
\begin{align*}
\begin{bmatrix} \sigma_s \\ \sigma_y \\ \tau_{xy} \end{bmatrix} &= \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{bmatrix} \epsilon_{s} \\ \epsilon_{y} \\ \epsilon_{xy} \end{bmatrix} \\
D_{rs} &= \frac{t_r}{t} \sum_{q=1}^{M} \mathbf{Q}_{q}, \quad (r, s = 1, 2, 6)
\end{align*}
$$

3. Definition of wing leading edge optimization problem

In this section, the topology optimization problem of compliant wing leading edge will be defined. It is envisaged that the compliant wing leading edge is used in a UAV. The wing profile is set as the NASA1050 profile with a chord length $c = 400$ mm. The wing leading edge domain is located on
the first quarter chord \( c/4 = 100 \text{ mm} \). The spanwise space between consecutive ribs is 200 mm and each rib thickness is 1.6 mm in our design. The ribs are prepared by glass fiber reinforced epoxy and the properties of a single layer composite are listed in Table 1. A schematic diagram of the wing leading edge structure is shown in Fig. 2. The ribs fixed in the middle of wings are not only significant components of bearing loads from skins, but also generate desired deformation with appropriate driving force. In this paper, the first quarter rib is highlighted to identify the design domain of wing leading edge.

Boundary condition of topology optimization of leading edge is shown in Fig. 3(a). The domain between initial wing leading edge shape and straight line \( AB \) is defined as design domain. Each end of straight line \( AB \) is fixed. External force \( F \) is loaded at midpoint of straight line \( AB \). Moreover, aerodynamic loading from leading edge surface is simulated by these arrowed lines. The aerodynamic loads are calculated assuming \( Ma = 0.5 \) air speed and an angle of attack of \( 6^\circ \). Aerodynamic pressure loads are replaced with equivalent concentrated forces along leading edge in this paper.

### 4. Implement of leading edge optimization with composite materials

#### 4.1. Objective function of wing leading edge

The optimal aerodynamic shape during different missions is the objective of designing compliant adaptive wing leading edge. In Fig. 3(b), the initial and target shapes are given. The actuator provides a force input to the system and the compliant mechanism changes the shape from its initial curve to target curve in terms of the structural elastic deformation. Therefore, displacements of 14 output points in shape curve are used to evaluate the differences between deformed and target shape curves. The minimizing LSE between deformed and target displacement at these output points is viewed as the objective function. The objective function of wing leading edge is shown as

\[
f(X) = \sum_{l=1}^{n} \sqrt{\left(u_{x_l}(X) - u'_{x_l}\right)^2 + \left(u_{y_l}(X) - u'_{y_l}\right)^2}
\]

where \( n \) is the number of given output points along the shape change boundary. \((u_{x_l}(X), u_{y_l}(X))\) and \((u'_{x_l}, u'_{y_l})\) denote the \( l \)th

<table>
<thead>
<tr>
<th>Output points number</th>
<th>Coordinates ( x ) (mm)</th>
<th>Coordinates ( y ) (mm)</th>
<th>Weight factor</th>
<th>Target displacement (mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>89.5</td>
<td>39.5</td>
<td>0.01</td>
<td>-0.8</td>
</tr>
<tr>
<td>2</td>
<td>72.5</td>
<td>36</td>
<td>0.01</td>
<td>-1.3</td>
</tr>
<tr>
<td>3</td>
<td>54.0</td>
<td>31.3</td>
<td>0.02</td>
<td>-4.6</td>
</tr>
<tr>
<td>4</td>
<td>19.7</td>
<td>17.8</td>
<td>0.04</td>
<td>-7.8</td>
</tr>
<tr>
<td>5</td>
<td>7.8</td>
<td>10.2</td>
<td>0.08</td>
<td>-10.6</td>
</tr>
<tr>
<td>6</td>
<td>2.5</td>
<td>5.3</td>
<td>0.12</td>
<td>-12.6</td>
</tr>
<tr>
<td>7</td>
<td>0.38</td>
<td>1.3</td>
<td>0.14</td>
<td>-13.5</td>
</tr>
<tr>
<td>8</td>
<td>86.8</td>
<td>-13.4</td>
<td>0.01</td>
<td>-0.5</td>
</tr>
<tr>
<td>9</td>
<td>67.9</td>
<td>-12.9</td>
<td>0.01</td>
<td>-2.5</td>
</tr>
<tr>
<td>10</td>
<td>46.4</td>
<td>-11.8</td>
<td>0.02</td>
<td>-6.0</td>
</tr>
<tr>
<td>11</td>
<td>26.3</td>
<td>-9.8</td>
<td>0.04</td>
<td>-9.5</td>
</tr>
<tr>
<td>12</td>
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<td>0.08</td>
<td>-12</td>
</tr>
<tr>
<td>13</td>
<td>4.1</td>
<td>-4.3</td>
<td>0.12</td>
<td>-13</td>
</tr>
<tr>
<td>14</td>
<td>0.87</td>
<td>-2.2</td>
<td>0.14</td>
<td>-13.4</td>
</tr>
</tbody>
</table>

![Fig. 2](image2.jpg)  
**Fig. 2** Schematic diagram of three consecutive wing ribs.

![Fig. 3](image3.jpg)  
(a) Boundary conditions of wing leading edge  
(b) Initial and target shapes

![Fig. 2](image4.jpg)  
**Fig. 3** Schematics of boundary conditions and output points.

| Table 1 Elastic constants of glass fiber reinforced epoxy single layer. |
|---------------------------|-------------------|-------------|-------------|-------------|-------------|
| Elastic constant          | \( E_1 \) (GPa)   | \( E_2 \) (GPa) | \( G_{12} \) (GPa) | \( v_{12} \) |
| Value                     | 39                | 8.4         | 4.2         | 0.26        |

| Table 2 Coordinates, weight factors and target displacement of output points. |
|---------------------------|-------------------|-------------|-------------|-------------|
| Output points number      | Coordinates \( x \) (mm) | Coordinates \( y \) (mm) | Weight factor | Target displacement (mm) |
| 1                         | 89.5              | 39.5        | 0.01        | -0.8        |
| 2                         | 72.5              | 36          | 0.01        | -1.3        |
| 3                         | 54.0              | 31.3        | 0.02        | -4.6        |
| 4                         | 19.7              | 17.8        | 0.04        | -7.8        |
| 5                         | 7.8               | 10.2        | 0.08        | -10.6       |
| 6                         | 2.5               | 5.3         | 0.12        | -12.6       |
| 7                         | 0.38              | 1.3         | 0.14        | -13.5       |
| 8                         | 86.8              | -13.4       | 0.01        | -0.5        |
| 9                         | 67.9              | -12.9       | 0.01        | -2.5        |
| 10                        | 46.4              | -11.8       | 0.02        | -6.0        |
| 11                        | 26.3              | -9.8        | 0.04        | -9.5        |
| 12                        | 11.3              | -6.8        | 0.08        | -12         |
| 13                        | 4.1               | -4.3        | 0.12        | -13         |
| 14                        | 0.87              | -2.2        | 0.14        | -13.4       |
output point deformed and target displacement, respectively. \( x_j \) is the \( j \)th weight factor in the \( j \)th output point. Coordinates, target displacements and weight factors of the output points are listed in Table 2.

### 4.2. Topology optimization model based on SIMP method

Topology optimization methods have been approached in general ways: the load-path approach using a framework of beam elements, the SIMP approach and level-set methods. In SIMP method, material density \( x_i \) of the \( i \)th element is set as design variable with values between 0 and 1. The relationship of elastic matrix \( E_i \) and element density \( x_i \) in optimization process can be written as

\[
E_i = x_i^p E_0
\]

where \( p \) is a penalization value and \( E_0 \) elastic matrix of initial solid element. Meanwhile, based on the same interpolation model described in Eq. (4), elastic modules along the main directions of single layer within laminated fiber reinforced composite plate can be expressed as

\[
\begin{bmatrix}
E_1 \\
E_2 \\
G_{12}
\end{bmatrix}
= x_i^p
\begin{bmatrix}
E_0^1 \\
E_0^2 \\
G_{12}^0
\end{bmatrix}
\]

where \( E_1, E_2, G_{12} \) are elastic modulus of fiber, epoxy resin matrix and shear modulus in optimization process, respectively. \( E_0^1, E_0^2, G_{12}^0 \) denote elastic constants of initial solid element. Note that for a given element \( x_i = 1 \), it implies that the element fully consists of the composite material, while \( x_i = 0 \) means that the element is hole.

In views of the above boundary conditions and objective function, topology optimization model of compliant wing leading edge structure can be expressed as

Find : \( X = [x_1, x_2, \ldots, x_N]^T \)

Min : \( f(X) = \sum_{i=1}^{N} \sqrt{(u_{x,i}(X) - u_{x,i}^0)^2 + (u_{y,i}(X) - u_{y,i}^0)^2} \)

\[
\begin{aligned}
KU &= F \\
kV &= L \\
\sum_{i=1}^{N} x_i \bar{v}_i &\leq h V_0 \\
0 < x_{\text{min}} &\leq x_i \leq 1
\end{aligned}
\]

where \( K \) is the global stiffness matrix of the laminated composite plates, \( U \) the global displacement vector, \( L \) the adjoint load vector, \( X \) design variable vector, \( x_i \) design variable of element material density, \( \bar{v}_i \) element volume after optimization, \( h \) desired volume ratio, \( V_0 \) the initial volume of the design domain, \( F \) the external force vector, \( V \) the adjoint displacement vector, \( x_{\text{min}} \) the minimum limit of element relative density and \( N \) the total number of discrete elements.

### 4.3. Sensitivity analysis

The sensitivity of objective function is provided using the adjoint matrix method in this section. The sensitivity of objective function in Eq. (6) with respect to a change in design variable \( x_e \) is expressed as

\[
\frac{\partial f(X)}{\partial x_j} = \sum_{i=1}^{N} \left( u_{x,i}(X) - u_{x,i}^0 \right) \frac{\partial u_{x,i}(X)}{\partial x_j} + \left( u_{y,i}(X) - u_{y,i}^0 \right) \frac{\partial u_{y,i}(X)}{\partial x_j}
\]

The displacement \( u_j \) of a specified degree of freedom \( j \) can be written as

\[
u_j = L^T U = V^T K U
\]

Assuming design-independent loads, the sensitivity of the output point displacement \( u_j \) with respect to a change in design variable \( x_i \) is written as

\[
\frac{\partial u_j}{\partial x_i} = -V_i^T \frac{\partial K}{\partial x_i}
\]

where \( k_i \) is the \( i \)th element stiffness matrix, \( v_i \) and \( u_i \) denote the \( i \)th element displacement and adjoint displacement vectors, respectively. In this study, plane quadrilateral mesh is used and element stiffness matrix is described as

\[
k_i = \int \int B_i^T D B_i \mathrm{d}x \mathrm{d}y
\]

where \( B \) is the strain matrix, \( T \) is element thickness (thickness of laminate plates) and \( D \) is the equivalent constitutive relations of laminated plates found by the Eq. (2). Substituting Eq. (10) into Eq. (9), the sensitivity of the output point displacement \( u_j \) with respect to a change in design variable \( x_i \) is obtained as

\[
\frac{\partial u_j}{\partial x_i} = -p(x_i)^{-1} V_i^T k_0 u_i
\]

where \( k_0 \) is element stiffness matrix of initial solid material. Its value can be obtained in \( x_i = 1 \) condition using Eq. (10).

The sensitivities of the material volume \( \sum_{i=1}^{N} x_i \bar{v}_i \) with respect to the element densities \( x_i \) are given by

\[
\frac{\partial}{\partial x_i} \sum_{i=1}^{N} x_i \bar{v}_i = \bar{v}_i
\]

### 4.4. Optimality criteria method and sensitivity filtering

In order to satisfy the constraint conditions, the Lagrangian function of the optimization problem Eq. (6) is followed:

\[
g = f(X) + \lambda \left( \sum_{i=1}^{N} x_i \bar{v}_i - h V_0 \right) + \mu^T (KU - F)
\]

\[
+ \sum_{i=1}^{N} w_i (x_{\text{min}} - x_i) + \sum_{i=1}^{N} \psi_i (x_i - x_{\text{max}})
\]

where \( \lambda, \mu, w_i, \psi_i \) represent the Lagrange multipliers, meanwhile, \( \hat{\lambda}, \hat{\mu}, \hat{\psi} \) and \( \hat{\mu} \) are scalar fields and vectors, respectively.

Based on the SIMP and Kuhn–Tucker conditions, optimization problem Eq. (7) is solved by means of a standard optimality criteria method. The heuristic updating scheme is given by

\[
x_i^{e+1} = \begin{cases} 
\min((1 + m)x_i^e, 1), & \text{if } ((1 + m)x_i^e, x_{\text{min}}) \leq (B_j)^T x_i^e \\
(B_j)^T x_i^e, & \text{if } ((1 + m)x_i^e, x_{\text{min}}) > (B_j)^T x_i^e \\
\max((1 + m)x_i^e, x_{\text{min}}), & \text{if } ((1 + m)x_i^e, x_{\text{min}}) < (B_j)^T x_i^e \\
\max((1 + m)x_i^e, x_{\text{min}}), & \text{if } ((1 + m)x_i^e, x_{\text{min}}) < (B_j)^T x_i^e
\end{cases}
\]
where $m$ is a positive move limit, $\eta$ is a numerical damping coefficient and $B_k^i$ is obtained from the optimality condition as:

$$
B_k^i = \frac{\partial f(X)}{\partial x_i} + \frac{\partial}{\partial x_i} \left( \sum_{j=1}^{N_k} B_j \frac{\partial f(X)}{\partial x_j} \right)
$$

(15)

where the Lagrangian multiplier $\lambda$ must be chosen so that the volume constraint is satisfied; the appropriate value can be found by means of a bisection algorithm.

In order to ensure the existence of solutions to the topology optimization problem and to avoid the formation of checkerboard patterns, a filter to the sensitivities on the design must be imposed. A whole range of filtering methods is thoroughly described by Sigmund. The sensitivity filter modifies the sensitivities as follows:

$$
\frac{\partial f(X)}{\partial x_i} = \frac{1}{X} \sum_{z \in N_e} H_z \frac{\partial f(X)}{\partial x_z}
$$

(16)

where $N_e$ is the set of elements $z$ for which the center-to-center distance $\text{dist}(z, i)$ to element $i$ is smaller than the filter radius $r_{\text{min}}$ and $H_z$ is a weight factor defined as

$$
H_z = \max(0, r_{\text{min}} - \text{dist}(z, i))
$$

(17)

Design domain shown in Fig. 3(a) is discretized with plane quadrilateral mesh. Displacement field is obtained by finite element analysis. The topology optimization problem has been solved by OC method and sensitivity filtering technique. The solution flowchart for topology optimization of compliant wing leading edge with symmetric laminated composite plates is shown in Fig. 4.

5. Optimization results and experiments

It is well-known that mechanical properties of fiber reinforced composite mainly depend on the fiber directions. In order to investigate the influences of fiber ply-orientations on topology structures, different topology structures of wing leading edges can be achieved with different fiber ply-orientations based on the model described by Eq. (6). The topology optimization results of wing leading edge with specific fiber ply-orientations and 30% volume constraint are shown in Fig. 5. The image of Fig. 5(a) is those of the isotropic material (aluminum alloy). The images of Fig. 5(b–d) are that of the laminated composite plates under different laminated sequences: $[0/45/0/45]$, $[-45/45/-45/45]$, and $[90/45/90/45]$, respectively. Comparing the results of isotropic material shown in Fig. 5(a) with the results of specific fiber ply-orientations shown in Fig. 5(b–d), it can be seen that the topology structure is largely changed using isotropic and laminated composite plates. Investigating Fig. 5(b–d), topology structures of laminated composite plates are dependent on fiber laminated sequences. Meanwhile, materials (black pixels) of topologic images are mainly distributed on fiber directions.

Taking the topology structure of the laminated sequence $[90/45/90/45]$, (displayed in Fig. 5(d)) as an example, the solution and iterative process are illustrated for the optimization problem described by Eq. (6). The displacement curves of the six key output points (5–7, 12–14 shown in Fig. 3(b)) and objective function in Eq. (6) are shown in Fig. 6. From Fig. 6(a), the six displacement curves gradually stabilize after iteration 464 and the corresponding stable values are $-11.3$, $-12$, $-12.4$, $-11.7$, $-12.1$, $-12.3$ mm, respectively, which are consistent with the desired values listed in Table 2. The objective function gradually reaches 1.38 after iteration 464 (shown in Fig. 6(b)). Therefore, deformed shape of compliant wing leading edge with laminated composite plates can achieve the desired shape approximately.

A full-size experimental wing leading edge model (shown in Fig. 5(d)) was constructed using the laminated glass fiber reinforced epoxy composite plates. Fig. 7(a) shows the overall configuration of the experimental platform. Compliant wing leading edge is fixed on the platform and aerodynamic loads are simulated by concentrated forces of weights. Deformed shape of compliant wing leading edge is demonstrated in Fig. 7(b) under imposing force of 150 N. The initial shape is indicated by the black solid line. The maximal displacement
of wing leading edge is approximately \(-11.0\) mm at the output point 7. However, the desired maximal displacement at the same point is \(-13.5\) mm. The errors may be caused by extraction of topology image and tolerance of manufacturing. The model fully returns to its initial configuration when the actuation is removed. The morphing experiment verifies that the compliant wing leading edge with symmetric composite laminated plate has a capability of achieving the desired shape and the method presented in this paper is an effective way of designing optimal compliant wing leading edge.

6. Conclusions

(1) The approach for designing compliant adaptive wing leading edge with symmetric laminated composite plate is presented based on topology optimization. Equivalent elastic constitutive relation of symmetric laminated composite plate is obtained by using laminated plate theory. Topology optimization model of compliant wing leading edge is built to minimize the LSE between deformed curve and desired aerodynamics shape.

(2) Well-developed OC methods and sensitivity filtering technique are used to solve the optimization problem and ensure the existence of solutions. The topology results show that different fiber ply-orientations for laminated composite plate affect their topology structures.

(3) Taking the topology structure obtained using the fiber laminated sequences [45/90/45/90]s as an example, the solution and iterative process of the optimization problem are illustrated. Displacements of the key output points are consistent with the desired values. Comparing the deformed and desired shape, the compliant wing leading edge with symmetric composite laminate plates is able to approximately achieve the desired shape. The corresponding morphing experiment verifies the morphing capability of topology structure and illustrates the feasibility for designing compliant wing leading edge.

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References


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