Aeroelastic passive control optimization of supersonic composite wing with external stores

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Abstract. This paper provides a study on passive aeroelastic control optimization, by means of aeroelastic tailoring, of a composite supersonic wing equipped with external stores. The objective of the optimization is to minimize wing weight by considering the aeroelastic flutter and divergence instability speeds as constraints at several flight altitudes. The optimization variables are the composite ply angle and skin thickness of the wing box, wing rib and its control surfaces. The aeroelastic instability speed is set as constraint such that it should be higher than the flutter speed of a metallic base line model of supersonic wing having previously published. A finite element analysis is applied to determine the stiffness and mass matrix of the wing and its multi stores. The boundary element method in the form of doublet lattice method is used to model the unsteady aerodynamic load. The results indicate that, for the present wing configuration, the high modulus Graphite/Epoxy composite provides a desired higher flutter speed and lower wing weight compare to that of Kevlar/Epoxy composite as well as the base line metallic wing materials. The aeroelastic boundary thus can be enlarged to higher speed zone and in the same time reduce the structural weight which is important for a further optimization process.

1. Introduction
The importance to optimize structural weight has been the goal since the beginning of air vehicle design concept. The effort has been conducted through structural optimization by implementation of structural design concept such as semi-monocoque design, improvement on structural analysis using finite element method to accurately estimate stress and displacement level of highly redundant complicated structure, enhancement on manufacturing capability to produce just necessary structural thickness, and advancement on material technology such as to use composite and more recently smart material.

On the other hand, reducing structural weight may induce some adverse effects that relate to comfort and safety aspects. One important adverse effect is that the air vehicle may become sensitive to vibration or flutter where, at this point, aeroelasticity plays important part. The interaction between structural weight and stiffness under steady and unsteady aerodynamic loading is thoroughly investigated in aeroelasticity[1]. For the reason of safety issue, aeroelasticity becomes necessary procedure in aircraft design. All civil and military aircraft standard regulations include aeroelastic stability and response assessment as part of the requirement [2 – 4]. A huge number of studies have been devoted to increase the accuracy and efficiency of aeroelastic estimation through analytical, numerical and experimental investigations [1].

Advancement in composite technology itself opens a new branch in aeroelasticity called aeroelastic tailoring which is defined as, according to Ref. 5, a technique to passively control structural deformation by altering the structure directional stiffness to improve its aeroelastic performance. The first implementation of aeroelastic tailoring is perhaps as early as 1949 where Munk design a propeller blade made by wood [6]. Realizing that the wood has a non-isotropic material properties, Munk rotated the fiber orientation of the blade wood material in order to gain a beneficial effect of the blade deformation under its selected operation condition. Aeroelastic tailoring is also called passive...
aeroelastic control [6] as the composite fiber orientation can be used to passively control aeroelastic mode shapes which can improve critical flutter and divergence speed. Without aeroelastic tailoring, the swept forward wing concept may not be accomplished into a safe high speed aircraft [5].

Many articles have contributed important review on the direction of aeroelastic tailoring [5 - 7]. Recently, Ref. 6 presents a comprehensive state of the art and promising technology of passive aeroelastic control that may lead to a new structural design. Some of previous optimization studies investigate the wing composite laminate with fiber orientations of 0°, ±45° and 90° arrangements[6] or, in other word, the variation on the fiber orientation between laminates is restricted to be 0°, 45° or 90°. In the present work, the optimization is performed in such a way that the variation of the fiber orientation can be as small as 5°. With this more relax restriction, it is expected that the aeroelastic performance can be further improved.

2. Aeroelasticity and Optimization

2.1. Structural model
The present work performed the aeroelastic analysis based on two numerical simulation models: structural model and aerodynamic model. The structural model of the wing is constructed based on finite element method where combination of quadrilateral shell elements and bar elements are used. The in-plane membrane and lateral bending characteristics of the quadrilateral elements are exploited to model the composite laminate for each layer with different material orientation direction. The launcher of the external stores is modeled as bar element where its attachment to the wing box is carefully set to simulate accurate boundary conditions. In the present work the composite materials used is Kevlar/Epoxy. The materials data is taken from [12] as shown in Table 1. Note that the Kevlar/Epoxy is selected since its modulus elasticity $E_1$ is near the baseline model material which is Aluminum, but with lighter density.

2.2. Aerodynamic model
The aerodynamic model of the wing for the supersonic region is based on the so-called ZONA51 method, a boundary element method for unsteady aerodynamics of multiple lifting surfaces oscillating in supersonic flow [13]. For the subsonic region, the doublet lattice method of MSC/Nastran is used [13]. The wing lifting surface is divided in chordwise and spanwise directions into a number of trapezoidal panel elements. The parallel side of the trapezoidal element is set in line with the incoming flow. It is noted that even though the present wing is designed for the supersonic region, the flutter analysis is required to be conducted at the subsonic region as well to ensure the instability does not occur in the complete flight design envelope. To ensure compatibility between the structural finite element and aerodynamic boundary element model, an infinite surface spline method is used such that the 6 degrees of freedom structural deformations at each point is related to 6 degrees of freedom aerodynamic forces of each trapezoidal panel [13].

2.3. Aeroelastic Solution
Prior to the aeroelastic solution, a modal analysis is performed using SOL 103 module of MSC/Nastran to calculate the vibration frequencies and mode shapes. SOL 145 module of MSC/Nastran is then used to solve the aeroelastic solution, where in the present case, the British PK method of the flutter solution is selected. The results of SOL 145 module are in the form of a set of frequency and structural damping for each speed. The flutter speed and frequency are then obtained by interpolating the structural damping to zero. This aeroelastic procedure is repeated for several altitudes covering the required operational flight zone. Note that a certain negative altitude for aeroelastic analysis is required in the military and civil standard regulations [2 - 4] to take into account the uncertainty of atmospheric properties among others as discussed in detailed in [9].
Table 1. Composite material properties (adapted from [12])

|                        | S-glass/epoxy | Kevlar/epoxy | HM Graphite/epoxy |
|------------------------|---------------|--------------|-------------------|
| Elastic Properties:    |               |              |                   |
| $E_1$, GPa             | 55            | 80           | 230               |
| $E_2$, GPa             | 16            | 5.5          | 6.6               |
| $G_{12}$, GPa          | 7.6           | 2.1          | 4.8               |
| $\nu_{12}$            | 0.26          | 0.31         | 0.25              |
| Tensile Strengths:     |               |              |                   |
| $\sigma_1$, MPa       | 1800          | 2000         | 1100              |
| $\sigma_2$, MPa       | 40            | 20           | 21                |
| $\sigma_{12}$, MPa    | 80            | 40           | 65                |
| Compressive Strengths: |               |              |                   |
| $\sigma_1$, MPa       | 690           | 280          | 620               |
| $\sigma_2$, MPa       | 140           | 140          | 170               |
| Physical Properties:   |               |              |                   |
| $\alpha_1$, $10^{-6}$/°C | 2.1            | -4.0        | -0.7              |
| $\alpha_2$, $10^{-6}$/°C | 6.3            | 60        | 28                |
| Volume fraction        | 0.7           | 0.54         | 0.7               |
| Thickness, mm          | 0.15          | 0.13         | 0.13              |
| Density, Mg/m³         | 2.0           | 1.38         | 1.63              |

2.3. Optimization

The optimization is performed with the objective to minimize the wing weight by considering the flutter speed as constraints. The constraint is set such that it should be higher than the flutter speed at the flight altitudes of the supersonic Aluminum base line wing model of Ref. 8. The optimization variables are the composite ply angle, skin thickness and material of the wing box, wing rib and its control surfaces. In the present work, the optimization is performed first to find the optimum material orientation for each layer of the wing skin. The increment of the angle orientation of each layer is selected as 5°. The second step is to find the optimum composite layer thickness with the increment of $1/8$ of the thickness of the base line model. The third step is to find the optimum wing rib thickness that it is found to be sensitive to the flutter speed result. The reduction of the rib thickness is performed by considering the spanwise distance from the wing root. In this case, the wing is divided into four spanwise zones where its zone has the same wing rib thickness.

3. Validation of the Flutter Analysis Procedure

To demonstrate the present aeroelastic procedure, comparison with two benchmark cases are conducted. The first case is to check the flutter speed at subsonic flow by referring to the well-known BAH wing model. A more detailed data of the BAH wing can be found in Ref. 14. The second case is to check the flutter speed at supersonic flow by comparing to the Hedgepeth wing model [15]. The results of both cases are presented in Tables 2 and 3 which show a good agreement with those of the benchmark models. Note that the critical eigenvalue in Table 3 is another non-dimensional representation of flutter frequency.

Table 2. Comparison of subsonic flutter solution procedure results

| Parameter          | Bisplinghoff et al.[14] | Present procedure result |
|--------------------|--------------------------|--------------------------|
| Flutter speed, V (m/s) | 386.69                  | 339.85                   |
| Flutter frequency, f (Hz) | 2.960                    | 2.976                    |
### Table 3 Comparison of supersonic flutter solution procedure results

| Parameter           | Hedgepeth [15] | Present procedure result |
|---------------------|----------------|--------------------------|
| Critical eigen value| 492            | 493.96                   |
| Flutter speed, V (m/s)| 570.36       | 577.59                   |

#### 4. Baseline Configuration

To demonstrate the enhancement of the present aeroelastic passive control, the results are compared with a baseline configuration which is a metallic wing with multiple external stores presented in [8]. The baseline model has a wing span of 7.5 m, aspect ratio of 5, taper ratio of 0.5 and leading edge swept-back angle of 30°. The wing section has a diamond shape airfoil with the wedge angle of 10°. The wing box portion in is 65% of chord-length as shown in figure 1, whereas the leading edge and trailing edge are 15% and 20%, respectively. The AMRAAM AIM-120A external stores are positioned at two locations along the wing span and Sidewinder AIM-9M at wing tip. A detailed wing sizing has been conducted in [8] for this configurations with a flight load factor of $n_x = 5.5$ and safety factor of 1.5. With this load factor, the wing sizing was performed resulting to the total wing and external store weight of 811 kg [8]. The flutter analysis performed at five altitudes shows that the aircraft flutter Mach numbers are $M=3.01$, $M=2.42$, $M=1.98$, $M=1.65$ and $M=1.44$ at the altitudes of 30 kft, 20 kft, 10 kft, sea level and -7.9 kft, respectively.

![Figure 1](image)

**Figure 1:** Double wedge airfoil (Taken from [8] with permission)

#### 5. Kevlar/Epoxy Laminate Configuration

A preliminary work using Kevlar/Epoxy composite laminate for the surface skins of wing box and control surfaces has been conducted in [10]. Ref. 10 basically performed the first step of the present optimization procedure, where the optimum fiber orientation layer are found to be $[45/20/0]$, for the wing skin and $[90/60/45]$, for flap and aileron control surfaces. The second step is then performed in the present work to find the optimum thickness of the composite layer. The thickness of the skin layer is decreased with the increment of $1/8$ of the thickness of the base line model. Some of important iteration results are shown in Table 4. Note that $r_{wing}$ in Table 4 indicates the wing skin thickness ratio with respect to that of baseline model. Similarly, $r_{surface}$ is the control surface skin thickness ratio with respect to that of baseline model. The clean wing weight is to represent the wing weight without external stores and its launchers.

| Parameter | Hedgepeth [15] | Present procedure result |
|-----------|----------------|--------------------------|
| Critical eigen value | 492 | 493.96 |
| Flutter speed, V (m/s) | 570.36 | 577.59 |
In Table 4, the result of Baseline is adapted from [8], and the results of Case 1 and 2 are from Ref. 10. Cases 3 until 5 (and a number of iterations not showing in Table 4) are the optimization of the present work. Case 1 until 5 results indicates a changes between the flutter mode of the baseline and composite wing. This is as expected that the composite laminate alter the bending and torsional modes in such a way that increase the flutter boundary with a less weight. Case 2 gives the desired highest flutter speed as reported in Ref. 10, however the wing weight is not the lowest one. It is Case 5 that gives the lowest weight and in the same time satisfies the constraint that its flutter speed is higher than the baseline configuration. As shown in Table 4, the flutter boundary of Case 5 in all altitudes is enlarge by approximately 113% with the wing weight reduction by 51%. A more detailed result for Case 5 are presented in figure 2 and Table 5 where the first 8 modes are shown. The results show that there are almost no pure bending or torsional modes. Obviously this is due to the influence of the external store along the wingspan and the composite laminates.

Table 5. Natural frequency and mode description of Case 5 configuration

| Mode | Frequency, \( f \) (Hz) | Mode Description |
|------|-----------------|------------------|
| 1    | 2.223           | First wing bending |
| 2    | 3.753           | Combination of Wing bending and torsion |
| 3    | 5.389           | Comb. of Wing torsion and Station 2 external store pitch |
| 4    | 5.817           | Comb. of Wing torsion and bending |
| 5    | 8.437           | 2nd Wing bending |
| 6    | 10.276          | 2nd Wing torsion and station 3 external store |
| 7    | 12.297          | Comb. of Wing torsion and control surface deflection |
| 8    | 17.266          | Comb. of Wing bending and station 1 external store |
The third step of the optimization procedure is conducted in the present work by varying the thickness of the wing rib. Started from Case 5 configuration, the wing rib is reduced by taking into account the four spanwise zone along the wingspan. The thickness is reduced for each zone with the incremental reduction ratio of $\frac{1}{8}$ with respect to the baseline configuration. A number of iterations are conducted where three cases of the results are presented in Table 6.

Figure 2. Vibration Mode Shapes of Case 5 Configuration
Table 6. Flutter Mach numbers for the Step 3 iteration

| Case | Flutter Mach number | Flutter mode | Clean Wing Weight (kg) |
|------|---------------------|--------------|------------------------|
|      | -8 kft 0 kft 10 kft 20 kft 30 kft |              |                        |
| Baseline | 1.437 1.651 1.983 2.420 3.008 | 4          | 289.1                   |
| 5       | 1.618 1.862 2.244 2.744 3.415 | 5          | 141.7                   |
| 5.1     | 1.618 1.862 2.244 2.744 3.415 | 5          | 136.2                   |
| 5.2     | 1.607 1.849 2.228 2.725 3.392 | 6          | 131.8                   |
| 5.3     | 1.628 1.874 2.258 2.761 3.437 | 6          | 127.0                   |

Table 6 shows that Case 5.3 is perhaps the optimum configuration for the present wing configuration where the wing weight is the lowest and in the same time the flutter boundary is enlarged by more than 112.4%. A further reduction may be difficult to perform since the strength and manufacturing requirement should be satisfied. The flutter plot for the structural damping and frequency for sea level is shown in figure 3 which shows that Mode 6 is the dominant modes that induced the destructive flutter mode. There is a possible hump flutter mode represented by Mode 3 that may induced a limit cycle oscillation. Figure 4 shows the flutter mode for each altitude. It shows that the flutter mode is very dangerous since the structural damping curve is very stiff near the flutter speed that indicates a very rapid changing of the structural damping with the increase of flight speed. This information would be useful to design the velocity range of the aircraft to assure a safe flight mission profile. In addition, for an explosive type flutter, the flight flutter test should be conducted with a well prepared planning, sufficient real time monitoring of aeroelastic stability and fast, accurate flutter speed predictions.

Figure 3. Structural damping g and frequency plot of Case 5.3 at sea level
6. Conclusion

The present work shows that Kevlar/Epoxy laminate can be used to reduce the wing structural weight and enlarge the aeroelastic stability boundary. The results show that the external store and the anisotropic behavior of the composite laminate influence the vibration mode shapes and therefore the flutter mode characteristic. By exploiting the laminate thickness and fiber orientation of the wing skin, control surface and rib, the lowest wing weight that can be achieved is 56% less than the baseline model with the flutter speed improved to more than 12%. The optimum configuration has an explosive flutter type that needs careful arrangement during flight flutter test.

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Figure 4. V-g graph of flutter mode of Case 5.3 at several altitudes
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