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The Design of a High Aspect Ratio HALE Aircraft Composite Wing. Part I: Static Strength Analysis

Bambang K. Hadi
Muhammad A. Ghofur
IndraPermana
Lightweight Structures Research Group
Faculty of Mechanical and Aerospace Engineering
Institute of Technology Bandung
Jl. Ganesha 10, Bandung 40132
Indonesia

ABSTRACT

High Altitude Long Endurance (HALE) UAV is a type of aircraft that is able to fly in high operational altitude (±65,000 ft) and longer duration (±36 hours). This aircraft uses a high aspect ratio wing, up to 25. In this paper, the design of a high aspect ratio wing will be performed. The Global Hawk RQ-4A was used as a test case. Static strength analysis with Tsai-Wu failure criteria will be used. The carbon/epoxy T300/5208 is selected and the results will be compared with the use of aluminum 2024-T3. Two laminate configurations were used in this study. They were $(0^\circ/0^\circ/+45^\circ/-45^\circ)_s$ *and a quasi-isotropic* $(0^\circ/+45^\circ/-45^\circ/90^\circ)_s$. *The aim of the study was to find a minimum weight while fulfilling the strength criteria. Finite Element Method was used extensively during the study, using NASTRAN software. The results showed that the wing with laminate configuration of* $(0^\circ/0^\circ/+45^\circ/-45^\circ)_s$ *was 30% lighter than the one with quasi-isotropic configuration, and was 60% lighter than the aluminum wing.*

Keywords: *composite wing, HALE UAV, high aspect ratio, static strength analysis, Tsai-Wu failure criteria*

Introduction

High Altitude Long Endurance (HALE) UAV was designed to be able to fly in high altitude up to 65,000 feet and long endurance up to 36 hours. In order to achieve the goal, the aircraft should have a high aspect ratio wing, up to 25. A high aspect ratio wing set a challenge to the wing structural designer, since it produces a high bending moment and the problem of wing rigidity; while at the same time it should be light. Mostly, HALE wing used composite materials in her main structures.

The purpose of this study is to perform a preliminary design of a high aspect ratio wing using composite materials. Carbon/epoxy T300/5208 was used in the design and Tsai-Wu failure criteria was applied in analyzing the static strength of the wing. Two laminate configurations were used, namely $(0^o/0^o/+45^o/-45^o)_s$ and a quasi-isotropic $(0^o/+45^o/-45^o/90^o)_s$. An aluminum 2024/T3 was also studied as a comparison to the composite wing. The objective of the study is to find the minimum weight while fulfilling the strength criteria.

Several studies have been conducted on using composite materials for the design of aircraft wing. Ali and Farhood [1] concluded that using composite materials in designing a wing box will reduce the weight up to 40% compared to aluminum wing box. Ainsworth et al [2] also concluded that the optimum rib-spacing in a wing box should be between 28–32 inch, and the use of composite materials reduced the weight up to 30% compared to the aluminum one. Chedrik, et al [3] studied the use of composite materials in designing a wing with aspect ratio of 12.5 and stated that the use of composite materials will solve the problem of high aspect ratio wing. Jweeget al [4] concluded that a composite taper wing was 32% lighter than the corresponding aluminum wing, while for the case of rectangular wing, the composite wing was 32% lighter. Liu [5] identified an effective method to optimize a composite wing in order to achieve a minimum weight; while Xu [6] focused on the effect of wing skin thickness on the stiffness and strength of a composite wing.

Recently, Naidu and Adali [7] designed a wing box structure for Medium Altitude Long Endurance UAV. However, they only considered a sandwich structure, using isotropic and composite materials. Romeo, et al [8] used multi-objective optimization procedure to design composite wing box of solar powered HALE UAV. The wing was modeled as a beam structure. The spar also used a sandwich structure. The latest, Lim et al [9] optimized wing design of Solar-HALE aircraft, by considering interaction between aerodynamics and structures. They achieved 25.3% of weight reduction. However, they did not provide the detail of composite lay-up configuration in their paper. Therefore, the novelty of this paper is that it will generate detail composite lay-up configuration.

Methodology

A static finite element method was used extensively during the design processes. The method solves a linear algebra equation in the form of Eq 1:

$$\{F\} = [K]\{d\} \tag{1}$$

Where {F} is the force matrix, [K] is the global stiffness matrix and {d} is the displacement matrix. A detail of Eq.1 can be found in a standard textbook on finite element method. A NASTRAN software was used extensively during the study.

A Tsai-Wu failure criterion was used to predict the failure of a laminate. When a layer (lamina) failed according to Tsai-Wu criteria, then the whole laminate was considered to be failed (first ply failure assumption). A Tsai-Wu failure criterion is given by Eq 2 and Eq 3:

$$A_{11}\,\sigma_{11}^2 + 2A_{12}\sigma_1\sigma_2 + A_{22}\sigma_2^2 + A_{66}\sigma_6^2 + B_1\sigma_1 + B_2\sigma_2 = 1 \quad (2)$$

where:

$$A_{11} = \frac{1}{\chi_t \chi_c} B_1 = \frac{1}{\chi_t} - \frac{1}{\chi_c}$$

$$A_{22} = \frac{1}{Y_t Y_c} B_2 = \frac{1}{Y_t} - \frac{1}{Y_c} A_{66} = \frac{1}{S^2} \quad (3)$$

Here, X is the unidirectional composite ply strength in fiber direction and Y is the strength in perpendicular fiber direction. S is the shear strength. Subscript t and c are tensile and compressive strengths respectively.

The wing geometry in this paper was adopted from Global Hawk RQ-4A wing as the baseline geometry. This aircraft is a modern HALE UAV aircraft that utilized high aspect ratio wing. The Global Hawk was chosen since it has high wing loading, compared to other HALE-UAV, such as solar HALE. Therefore, it gives more challenges to designer. Nevertheless, the methodology presented in this paper can be used to design any arbitrary aircraft wing having high aspect ratio. The external wing geometry specifications used in this paper were the same as the Global Hawk RQ-4A wing. Since the internal structures of the actual Global Hawk wing was unknown, the current structures was designed according to the standard aircraft wing structural design configuration. The current structure consists of 3 main components: 23 ribs with 80 cm rib spacing, 2 spars at 15% and 60% wing chord, and skin panels. The rib spacing was chosen to have the maximum buckling strength of the panels, while the spars position was chosen to accommodate the maximum torsional stiffness according to the design practices in the wing structural design.

The wing geometry specifications and internal layout are shown in Table 1 and Figure 1 respectively.

Table 1: Wing Geometry Specification

Wing Span, S	35.41 m
Wing Area, A	50.2 m^2
Aspect Ratio, AR	25
Swept Angle	5.9°
Root Chord	2.1 m
Root Tip	0.735 m
Airfoil	LRN

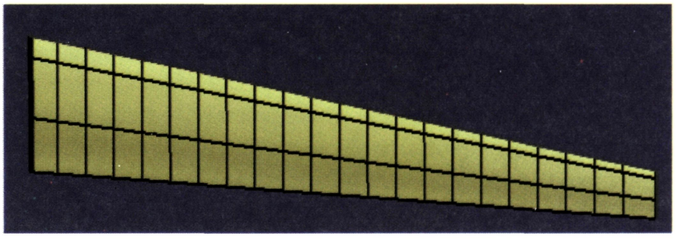

Figure 1: Wing Geometry and Internal Structures. The dimension is based on Table 1

Carbon/epoxy T300/5208 and aluminum was used during the design process. The carbon fiber was chosen since it has high specific strengths and modulus. Moreover, this material is the standard material for high specification HALE UAV aircraft.

The material characteristics were given in Table 2 and Table 3 for aluminum and carbon/epoxy respectively.

Table 2: Aluminum T-2024 Properties

	Aluminum 2024-T3	**Unit**
Elastic Modulus	73.1	GPa
Poisson Ratio	0.33	
Density	2780	Kg/m3
Tensile Yield Strength	345	MPa

Table 3: Carbon/epoxy T-300/5208 Properties

	T-300/5208	**Unit**
Fiber Volume	70%	
Elastic Modulus 11	181	GPa
Elastic Modulus 22	10.3	GPa
Poisson Ratio 12	0.28	
Shear Modulus 12	7.17	GPa
Shear Modulus 23	5	GPa
Shear Modulus 13	7.17	GPa
Density	1600	Kg/m^3
Tension Stress Limits 11	1500	MPa
Tension Stress Limits 22	40	MPa
Compress Stress Limit 11	1500	MPa
Compress Stress Limit 22	246	MPa
Shear Stress Limit	680	MPa
Bonding Shear Stress Limit	50	MPa

From the data sheet, it was found that the maximum take-off weight (MTOW) of the Global Hawk was 11,612 Kg, with a load factor of 3. The load was taken by the wing box with the area of 11.3 m^2. Using the safety factor of 1.5, the maximum pressure on the wing box was found to be 22.7 MPa. The load factor was taken from [10], while the safety factor number is the standard safety factor for aircraft structural design and analysis. In the finite element model, the pressure was given on the lower wing box. The wing root was fixed along 20 points in the wing root which represented the bolted joint between the outer wing and inner wing. Fig 2 shows the finite element model in the wing root and the applied constant pressure along the lower wing box. All the element was modeled as shell elements.

Along the span, the wing was divided into 9 sections. Each section was designed to have the same skin thickness. This will ease the manufacturing procedures. Fig 3 shows the 9 section along the wing span.

Results and Discussion

The design procedure was as follows. First, the wing thicknesses were set to be the same for all sections from wing-root to wing-tip. Then, the finite element model was run and check the failure indices (FI) using Tsai-Wu criteria for composite structures and von Misses criteria for aluminum structures. If the value of FI is less than 1.0, it means the structure is safe. On the other hand, if the value of FI is more than 1.0, the structure is considered to be failed. If the FI is less than 1.0, we will reduce the thickness so that the FI is approaching 1.0, but it is still less than 1.0. However, if the FI is more than 1.0, the wing thickness will be increased, so that the FI for that particular element is less than 1.0. The iteration was done several times to make sure that the FI for all structural elements are less than 1.0. The above procedure was carried out on all laminate configurations which are $(0°/0°/+45°/-45°)_s$, $(0°/90°/+45°/-45°)_s$ and aluminum T-2024 structures.

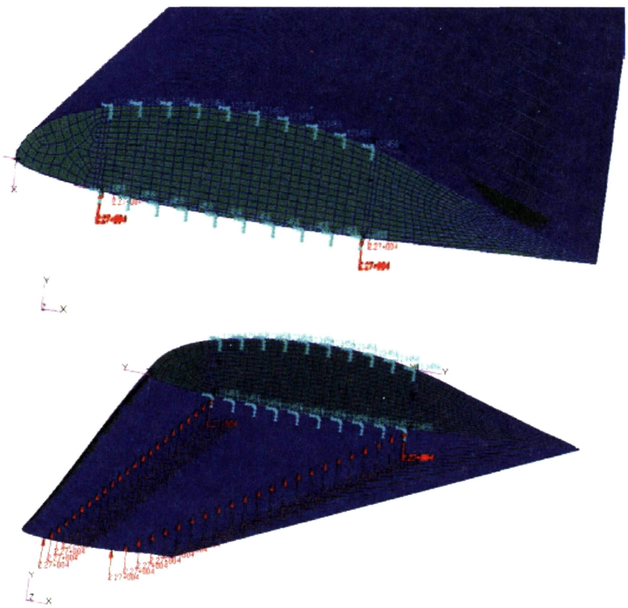

Figure 2: Fixed boundary conditions along the wing root and applied constant pressure on the lower wing box

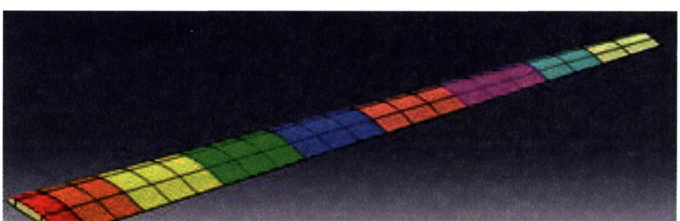

Figure 3: The 9 sections along the wing span. Each section has the same skin thickness

Table 4, 5 and 6 shows an example of Tsai-Wu FI values for all structural elements: ribs, skins and spars element for the case of $(0°/0°/+45°/-45°)_s$ laminate configuration.

Table 4: Tsai-Wu Failure Indices for Ribs Elements

Section	Rib	
	Thickness (mm)	Tsai-Wu FI
1	50	0.725
2	12	0.508
3	8	0.575
4	8	0.511
5	4	0.496
6	4	0.475
7	2	0.430
8	2	0.384
9	2	0.378

Table 5: Tsai-Wu Failure Indices for Skins Elements

Section	Skin	
	Thickness (mm)	Tsai-Wu FI
1	60	0.828
2	14	0.876
3	13	0.806
4	11	0.760
5	9	0.716
6	6	0.820
7	4	0.796
8	3	0.560
9	2	0.892

Table 6: Tsai-Wu Failure Indices for Spars Elements

Section	Spar	
	Thickness (mm)	Tsai-Wu FI
1	60	0.828
2	16	0.735
3	13	0.648
4	11	0.576
5	9	0.526
6	6	0.534
7	4	0.445
8	4	0.385
9	3	0.753

Table 4 to 6 also show that all structural elements are safe. Fig 4 shows the distribution of Tsai-Wu Failure Indices (FI) for the whole wing and its structural components: skins, ribs and spars.

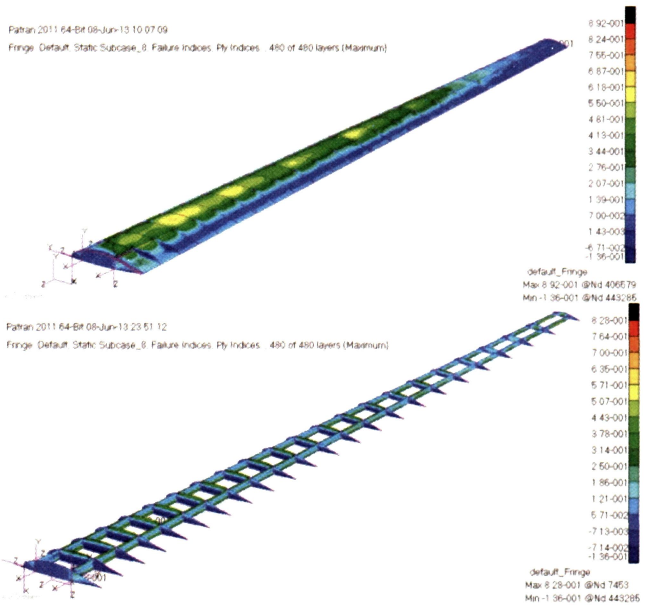

Figure 4: Tsai-Wu Failure Indices (FI) for the wing and its components: skins, spars and ribs

The same procedures were carried out for the wing with quasi-isotropic $(0^{o}/90^{o}/+45^{o}/-45^{o})^{s}$ and aluminum materials. Careful calculations were done to make sure that all structural elements were safe. Finally, the total weight was calculated. Table 7 to 9 shows the total weight of the wing for different laminate configuration.

Table 7: Wing Total Weight for $(0^{o}/0^{o}/+45^{o}/-45^{o})_{s}$

Ribs		Skins		Spars	
Mass (kg)	Volume (m^3)	Mass (kg)	Volume (m^3)	Mass (kg)	Volume (m^3)
91.61	0.057	453	0.283	102.7	0.064
Total Weight: 647 kg					

Table 8: Wing Total Weight for $(0^{o}/90^{o}/+45^{o}/-45^{o})_{s}$

Ribs		Skins		Spars	
Mass (kg)	Volume (m^3)	Mass (kg)	Volume (m^3)	Mass (kg)	Volume (m^3)
137.8	0.086	635.7	0.401	152.5	0.095
Total Weight: 926 kg					

Table 9: Wing Total Weight for Aluminum

Ribs		Skins		Spars	
Mass (kg)	Volume (m^3)	Mass (kg)	Volume (m^3)	Mass (kg)	Volume (m^3)
231.7	0.080	1060	1.149	260	0.094
Total Weight: 1552 kg					

Table 7-9 shows the total weight comparison of the wing for different laminate configuration and compared with the one having aluminum wing. It clearly shows that the wing having orthotropic laminate configuration of $(0°/0°/+45°/-45°)_s$ has the least weight compared to other configurations. The orthotropic wing configuration was 30% lighter than the quasi-isotropic configuration, and 60% lighter than the aluminum configuration. Table 8 and 9 also show that even using a conservative quasi-isotropic laminate configuration still produces a wing with 40% lighter than the aluminum configuration. Therefore, it can be concluded that using composite materials especially carbon/epoxy will substantially reduce the total weight of the structures.

Conclusion

The paper provides some steps in designing a high aspect ratio wing such as HALE UAV aircraft using composite materials. Finite element method was used extensively during the work, with Tsai-Wu failure criteria for composite materials and von Misses for aluminum. The paper concluded that orthotropic laminate configuration of $(0°/0°/+45°/-45°)_s$ for the wing components: skins, spars and ribs will produce the least total weight of the wing, compared to quasi-isotropic configuration and aluminum configuration. This is the first part in the study of using composite materials in the design of high aspect ratio wing. The second part of the study is the analysis of the wing for buckling analysis and flutter speed.

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