Original Article

Conceptual design and simulation validation based finite element optimisation for tubercle leading edge composite wing of an unmanned aerial vehicle

Ernie Illyani Basri\textsuperscript{a}, Faizal Mustapha\textsuperscript{a}, Mohamed Thariq Hameed Sultan\textsuperscript{a,b,c,∗}, Adi Azriff Basri\textsuperscript{a}, Mohd Firdaus Abas\textsuperscript{a}, Mohd Shukry Abdul Majid\textsuperscript{d}, Kamarul Arifin Ahmad\textsuperscript{a,e,∗}

\textsuperscript{a} Department of Aerospace Engineering, Faculty of Engineering, Universiti Putra Malaysia, 43400 UPM Serdang, Selangor Darul Ehsan, Malaysia
\textsuperscript{b} Laboratory of Biocomposite Technology (BIOCOMPOSITE), Institute of Tropical Forestry and Forest Products (INTROP), Universiti Putra Malaysia, 43400 UPM Serdang, Selangor Darul Ehsan, Malaysia
\textsuperscript{c} Aerospace Malaysia Innovation Centre (944751-A), Prime Minister’s Department, MIGHT Partnership Hub, Jalan Impact, 63000 Cyberjaya, Selangor, Malaysia
\textsuperscript{d} School of Mechatronic Engineering, Universiti Malaysia Perlis, Pauh Putra Campus, 02600 Arau, Perlis, Malaysia
\textsuperscript{e} Aerospace Manufacturing Research Centre (AMRC), Universiti Putra Malaysia, 43400 UPM Serdang, Selangor, Malaysia

A R T I C L E   I N F O

Article history:
Received 7 February 2019
Accepted 22 July 2019
Available online 12 August 2019

Keywords:
Laminates composite wing
Rib-reinforced
Monofoam
Finite element analysis

A B S T R A C T

A finite element model is developed to determine deformation and stresses on a composite wing of unmanned aerial vehicle (UAV) with a tubercle design at the leading edge of the wing. Tubercles, commonly known as protuberances found on the leading edge of a whale pectoral flipper, offering great performance from an aerodynamic perspective. This paper deals with a first order shear deformation theory (FSDT) approach to discover the UAV laminates composite wing model of tubercle leading edge (TLE) with rib-reinforced so that the equivalent stiffness and material properties are obtained from the simulation of finite element analysis using ANSYS. Another structural configuration of design replicating the idea of monocoque concept, whereby foam is used at the leading and trailing edges of the wing. Styrene acrylonitrile (SAN) core foam is used representing high strength-to-weight ratio with its superiority in the mechanical properties of polymeric sandwich composites. The updated static structural analysis from rib-reinforced can be applied to update the wing stiffness distribution of monocoque-foam. The optimum design is concluded from the tabulated deformation and stresses of both wings, where monocoque-foam showed better performance with a reduction in 50.72% of deformation and 35.88% of stress, compared to rib-reinforced design.

© 2019 The Authors. Published by Elsevier B.V. This is an open access article under the CC BY-NC-ND license (http://creativecommons.org/licenses/by-nc-nd/4.0/).

∗ Corresponding authors.
E-mails: thariq@upm.edu.my (M.T. Sultan), aekamarul@upm.edu.my (K.A. Ahmad).
https://doi.org/10.1016/j.jmrt.2019.07.049
2238-7854/© 2019 The Authors. Published by Elsevier B.V. This is an open access article under the CC BY-NC-ND license (http://creativecommons.org/licenses/by-nc-nd/4.0/).
1. Introduction

Finite element method has been widespread use in the applications of aerospace, mechanical and civil engineering for predicting accurate response of the product design. It is widely known as a computer-aided mathematical technique to determine numerical solutions subjected to abstract equations of calculus that predicts the behaviour of physical systems under external influences [1]. The finite element method has proven to be a powerful tool in various applications of aerospace engineering; particularly it has the advantage of being applicable to shells or plate of irregular geometry that contain composite material properties. Previous studies related to finite element analysis software and outcome of analyses conducted encompasses the scope of UAV wing.

James et al. [2] aimed to reduce the weight of light UAV GA (W)-2(modified) airfoil using Brazier load. A linear static analysis is conducted using NASTRAN software to estimate the bending stress on the carbon/epoxy composite of internal components. The static analysis proved that the buckling factor increase subjected to the effect of crimp on buckling stress due to the presence of stringer in the web of rib. Hence, the reduction of the weight of 2.85 kg from the composite wing was achieved by removing 15.77% from inter-spars ribs solely. Paradies and Ciresa [3] performed static and dynamic finite element analyses using ANSYS on a wing with thin Jedelsky profile. With the aid of load (as air velocity) from aerodynamic forces acting on the surface of the composite wing skin, the deformation was found at the distance of 500 mm from the wing root. Mazhar and Khan [4] studied the finite element analyses subjected to the aerodynamic load that were applied on the wing as pressure functions using artificial neural network in CFD analysis. The analysis involves the effects of geometric parameters, component location, loads and materials using ANSYS. The simulation resulted in a 35% weight reduction of using composite instead of aluminium. Shabeer and Murtaza [5] investigated the optimal design structure of UAV wing of static structural analysis using NASTRAN. In this case, the wing skin was made of composite materials, whereas the other internal components were made of alloys. The obtained stress was at the main spar which less than the yield strength of Aluminium. The similar analysis was carried out with different orientation or sequence of composite skin plies. The ply sequence of [0/90/+45/−45/90/0] showed better performance in terms of displacement and maximum stress at the root of the wing. Kanesan et al. [6] investigated the composite wing deflection for NACA4415 airfoil at different locations by using ABAQUS. The static analysis was carried out by applying distributed pressure load on the bottom skin that directly on top of main and aft spars. Higher deflection is observed near the tip of the wing.

Composites, a combination of numbers of constituent materials offer excellent strength-to-weight ratio with great manufacturing feasibility of complex parts, special contours and appearance [7–9]. The construction of laminated composite materials involves the lamination of thin shell as individual component, known as lamina. Hence, elastic behaviour of composite material should be taken into consideration to analyse the effect of laminated composite corresponding to transformation between principal axis and orientated coordinates outlined. Then, the laminate will response as a result of imposed boundary conditions, which is support and loading conditions [10]. However, in the finite element of composite structures, some modelling errors are commonly encountered which may lead to inaccuracy in the response prediction [11]. The reason of the discrepancy may be subjected to the conditions of laminas in the laminate with deformation compatibility and stress-strain relations with equilibrium. In this case, one can use an approach namely, the first order shear deformation theory (FSDT) to determine the effective and realistic simplifying assumptions that reduce three-dimensional elastic problems to two-dimensional one [11–13]. The process of the approach is shown in Fig. 1.

Referring to Fig. 1, the approach is represented by the linear variation of displacement across the thickness [15]. In this case, the normal stress is ignored as the assumption only known as the plane stress condition. The composite lamina stiffness matrix (Q) which describing the stress–strain relations in coordinates aligned with the principal material directions. For stress–strain analysis of laminated composite shell, the inverse form of the lamina stress–strain relation (Q) needed to be obtained, at the condition of lamina off-axis coordinate system. Equilibrium, kinematics (strain–displacement relations) and the constitutive relation of stress–strain are combined to obtain the stress resultant for the composite laminate at one point [16]. For the constitutive relations, the stress resultant is determined in form of matrix [A_{ij}], [B_{ij}], [D_{ij}] and [G_{ij}]. In ANSYS, the SHELL181 and SHELL821 element use an equivalent energy method to calculate shear factors subjected to strain energy. This is due to

![Fig. 1 – Theoretical approach for analysing laminated composite [14].](image-url)
transverse shear stress resultant is equal to strain energy that based on the true transverse shear stresses predicted by the 3D elasticity theory.

Composite has been widely applied in aeronautical applications due to its novel structural technologies that may reduce structural weight. There are numbers of papers studied in the field of structural optimisation comprises material selection subjected to high strength-to-weight ratio, optimisation algorithms, shape improvement, etc. [17]. In this study, structural optimisation with shape improvement of a composite UAV tubercles wing is investigated to determine the response of the structure that significantly improves the performance of the wing structure. Typically, implementing tubercles design at the leading edge of UAV wing has proven its aerodynamic performances, such as improvement on airfoil performance, effectively reduce noise and separation bubble size [18–22]. However, the study on tubercle leading edge has only available from aerodynamic perspective instead of structural optimisation. This will lead to critical acknowledge of the reliability and robustness of the wavy pattern at the leading edge of the wing. Despite that, the internal structure of wing is also a key role to determine the optimal design of structures in manipulating shape and materials characteristics in a variety ways. Practically, the internal structures of a wing

![Flowchart of structural optimisation of composite UAV wing of TLE.](image)

**Fig. 2** – Flowchart of structural optimisation of composite UAV wing of TLE.

![Side view of spherical tubercles.](image)
![Parameters for the leading edge of the wing.](image)

**Fig. 3** – (a) Side view of spherical tubercles. (b) Parameters for the leading edge of the wing.
consists of spars, ribs, and stringers mainly to support the distribution load and concentrated wing of the aircraft [23]. The design internal structures can vary, whereby the variation leads towards the idea of changing and shape and size that contributed to unique design optimisation. Namely, one can change by having the rib-reinforced, as support members to spars. The other alternative to the structural configuration is by implementing the idea of a monocoque structure. In this case, the ribs that arranged along the wingspan of internal wing structure can be replaced with foam structure to help in the withstanding of loads. The monocoque-foam structure also able to help in stiffening up the load carried by the wing skin without having to increase the skin thickness. Hence, both design features will pave the way of interdisciplinary of wing design particularly in structural aspects.

In this paper, the finite element analysis is carried out to study the structural optimisation with design improvement of a composite UAV tubercles wing. A rib-reinforced design of internal wing structure with the assignment of the composite is developed and the static structural analysis is performed using ANSYS Workbench 17.0. Then, followed by a monocoque-foam-reinforced design of internal wing structure. The optimum design is concluded according to the results of stress and displacement. In particular, this paper is structured into sections and subsections. The methodology of modelling the tubercle leading edge (TLE) of UAV wing is elaborated in the next section. The following subsections include the modelling of composite structures and structural analyses on different design of internal wing components. The result of the analyses in explained in Section 4. Conclusion will be provided in Section 5.

2. Methodology

The main methodology of this study comprises the three main stages: design module, ANSYS Composite Prepost (ACP) Pre module and Static structural module. The flowchart of the structural optimisation of UAV composite wing model of tubercle leading edge (TLE) with the internal component design features of rib-reinforced, is depicted in Fig. 2.

2.1. Stage 1: design module

Geometrical modelling of the baseline airfoil of NACA4415 is essential in developing a wing model for tubercle design. By using SolidWork, the span length is set to 5.1527 m and the chord length is 0.5886 m. For the parameters for the leading edge of the wing, the optimum amplitude \( A = 0.025c \) and wavelength, \( \lambda = 0.25c \) was chosen due to its remarkable effect on the airfoil performance [24–26]. The proposed optimal ratio of amplitude and wavelength for NACA4415 airfoil with spherical pattern at the leading edge of the wing, as in Fig. 3(a) and (b).

The geometrical modelling of the TLE wing structural is imported into the static structural domain in ANSYS Workbench 17.0. Since the UAV is assumed to be symmetrical only half of the wing is modelled in order to reduce the total number of elements used in the analysis. Thus, computational time also can be reduced. The design of TLE wing along the span is depicted in Fig. 4.

2.1.1. Structural configurations for rib-reinforced

The major structural components of the TLE wing with rib-reinforced includes one part of wing skin, two spars, two outboard ribs, six leading ribs, five middle ribs and seven trailing ribs. The arrangement of the ribs in conjunction with the spars, as in Fig. 5.

These structures are made up of individual members of spars with the combination of L-shape and U-shape. The attachment of major components such as spar to skin, rib to skin, and spar to rib are attached with adhesive joints. This type of joint is considered as a tied constraint with zero thickness between the tied surfaces.

2.1.2. Structural configurations for monocoque-foam-reinforced

Meanwhile, the major structural components of the TLE wing with monocoque-foam reinforced includes one part of wing skin, two spars, one outboard rib, one leading rib, two middle ribs, two trailing ribs, one leading monocoque and one trailing monocoque. The concept of monocoque-foam is replacing the
all the ribs that supported the spar with the foam materials as auxiliary elements to withstand the load. The configuration of the monocoque in conjunction to the spars, as in Fig. 6.

Similar spars shape configuration with the rib-reinforced design are used. The spar to skin, foam to skin, rib to skin, spar to rib and spar to foam are attached using adhesive joints. In ANSYS, all components are joined using tied-surface function.

2.1.3. Mesh generation

The accuracy of predicting the real-world behaviour from the finite element model can be obtained from the finite element mesh. The geometric model is subdivided into smaller domains (elements) subjected to mesh refinement. The process of generating mesh is the key step in validating finite element model for the analysis of composite structures. The shell elements (SHELL181) are used for the analysis of composite shells or plates [27]. In the case of laminated shell, the orientation of each lamina is defined from the given rotation angle relative to orientation for the entire shell section. Moreover, the properties of each lamina are defined by the linear elastic behaviour for lamina under plane stress condition. There are guidelines for generating mesh in order to ensure accuracy of analysis results [27]:

i. The mesh should represent the geometry of the computational domain and applied loads accurately.
ii. The mesh should adequately represent the large displacement or stress gradient in the solution.
iii. The mesh should contain elements with small number of aspect ratio.

Since the wing is assumed to be symmetrical, this help in reducing the total number of element used in the analysis. Subsequently, the grid generation for the wing including both internal and external components is performed. In this case, a grid dependency test is carried out for both structures in order to select the best mesh for the simulation. The best mesh of approximately 450,000 quadrilateral elements is selected for the dependency tests on maximum total deformation. The maximum skewness of mesh is in good quality of 0.804. The mesh element of UAV wing with rib-reinforced is depicted in Fig. 7.

2.2. Stage 2: ACP(Pre) module

As in Fig. 2, the subsequent stage is ACP(Pre) module, whereby the overall process of constructing composites on the wing is carried out in the ACP domain of ANSYS Workbench 17.0. The updated geometrical modelling is converted into a shell model.

2.2.1. Composite construction for rib-reinforced

In ACP, each lamina of the composite will be defined in terms of fabric, material used and its thicknesses except for brackets that attached to the fuselage and some other minor components. The construction of composite components mostly used carbon fibre fabric, unidirectional carbon fibre, Kevlar, honeycomb core with epoxy resin as matrix. The assignment replicates the actual manufacturing process of the composite. The wing’s structural components with their corresponding composite laminate are given in Table 1. The elastic properties of material used and the strength properties are also shown in Tables 2 and 3.

The commencement stage of preparing a laminated composite is the definition of base materials as fabric, mechanical properties, ply type and additional failure criteria. The ply arrangement and notation in the laminate is presented by considering the shell elements to model the wing structure.
Table 1 – Details of composite laminates of wing structure.

<table>
<thead>
<tr>
<th>Wing structure</th>
<th>Total no. of layers</th>
<th>Types of composite material</th>
</tr>
</thead>
<tbody>
<tr>
<td>Skin</td>
<td>5</td>
<td>Carbon fibre fabric, Kevlar veil, honeycomb</td>
</tr>
<tr>
<td>Main spar</td>
<td>8</td>
<td>Carbon fibre fabric ±45°, 0/90, carbon unitape</td>
</tr>
<tr>
<td>Aft spar</td>
<td>6</td>
<td>Carbon fibre fabric ±45°, 0/90, carbon unitape</td>
</tr>
<tr>
<td>Ribs (leading and trailing)</td>
<td>5</td>
<td>Carbon fibre fabric ±45°, Kevlar veil</td>
</tr>
</tbody>
</table>

Source: Ref. [6].

Table 2 – Elastic properties materials.

<table>
<thead>
<tr>
<th>Material</th>
<th>Density (kg/m³)</th>
<th>Elastic modulus, $E_{11}$ (GPa)</th>
<th>Elastic modulus, $E_{22}$ (GPa)</th>
<th>Poisson ratio, $v_{12}$</th>
<th>Shear modulus, $G_{12}$ (GPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Carbon fibre fabric/epoxy</td>
<td>1600</td>
<td>70</td>
<td>70</td>
<td>0.1</td>
<td>5</td>
</tr>
<tr>
<td>Carbon Unitape/Epoxy</td>
<td>1600</td>
<td>140</td>
<td>10</td>
<td>0.3</td>
<td>5</td>
</tr>
<tr>
<td>Kevlar/Epoxy</td>
<td>1400</td>
<td>78.5</td>
<td>5.52</td>
<td>0.34</td>
<td>2.07</td>
</tr>
<tr>
<td>Honeycomb</td>
<td>48</td>
<td>128.7</td>
<td>12.6</td>
<td>0.261</td>
<td>1.6</td>
</tr>
</tbody>
</table>

Source: Ref. [6].

Table 3 – Strength properties of materials.

<table>
<thead>
<tr>
<th>Material</th>
<th>Tensile strength in fibre direction, $X_t$ (MPa)</th>
<th>Tensile strength in fibre direction, $X_c$ (MPa)</th>
<th>Tensile strength in transverse direction, $Y_t$ (MPa)</th>
<th>Tensile strength in transverse direction, $Y_c$ (MPa)</th>
<th>Shear strength, $S$ (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Carbon fibre fabric/epoxy</td>
<td>600</td>
<td>570</td>
<td>570</td>
<td>570</td>
<td>90</td>
</tr>
<tr>
<td>Carbon Unitape/Epoxy</td>
<td>1500</td>
<td>1200</td>
<td>50</td>
<td>250</td>
<td>70</td>
</tr>
<tr>
<td>Kevlar/Epoxy</td>
<td>1380</td>
<td>276</td>
<td>29.6</td>
<td>137.9</td>
<td>43.4</td>
</tr>
<tr>
<td>Honeycomb</td>
<td>2.344</td>
<td>4.07</td>
<td>2.344</td>
<td>4.07</td>
<td>6</td>
</tr>
</tbody>
</table>

Source: Ref. [6].

Firstly, it is important to consider a laminate consists of $n$ plies as in Fig. 8. In Ref. [27], each ply of the laminate (refer as lamina) has a thickness of $t_k$. The thickness of the laminate, $h$ can be derived as

$$h = \sum_{k=1}^{n} t_k$$  \(1\)

The location of the mid-plane is $h/2$ from the top and the bottom surface of the laminate. The $z$-coordinates of each ply, $k$ surface are given by:

**Ply 1**: (top surface) $$h_0 = -\frac{h}{2}$$  \(2\)

(bottom surface) $$h_1 = -\frac{h}{2} + t_1$$  \(3\)

**Ply k**: ($k = 2, 3, \ldots, n - 2, n - 1$)  \(4\)

(top surface) $$h_k-1 = -\frac{h}{2} + \sum_{i=1}^{k-1} t_i$$  \(5\)

(bottom surface) $$h_k-1 = -\frac{h}{2} + \sum_{i=1}^{k} t_i$$  \(6\)
2.2.2. Composite construction for monocoque-foam-reinforced

A similar process will be applied on the composite construction for monocoque-foam wing design. However, the leading and trailing edges of the wing will be used foam replacing the rib-reinforced design. The material properties of the styrene acrylonitrile (SAN) foam are tabulated in Table 4. Once assignment of materials updated, the wing model will be converted to subsequent stage.

2.3. Stage 3: static structural module

This is a crucial stage, where the laminated composite of UAV wing for both designs will be analysed to determine the response of the structures subjected to the external influence of load applied. From the FSDT approach, the applied loads acting on the laminate are related to the mid-plane strains and curvatures. Both mid-plane and curvatures are assumed to be constant across the thickness. Hence, the stresses and strains in each lamina of the thin laminated structure are able to be calculated. However, several assumptions should be made prior to performing the analysis [30]:

- Each lamina is orthotropic and homogeneous.
- Each lamina is linearly elastic structure.
- Deformations are continuous and small through the laminate.
- A line, which is straight and perpendicular to the middle surface remains straight and perpendicular to the middle surface during deformation.
- The laminate is thin and is loaded in its plane (plane stress), whereby out-of-plane (normal) direct stress is zero.
- No slip occurs between the lamina interfaces.

In this study, the wing is modelled into shell element of SHELL181, in ANSYS. The incorporated FSDT is subjected to 6 degrees of freedom (DOF) at each node, where the translations in the x, y and z directions and rotations about the x, y and z-axes.

2.3.1. Boundary condition

A boundary condition for the wing model is set from a known value for a displacement or an associated load [1]. In this context, the boundary conditions are subjected to DOF constraint at specified model boundaries to define rigid support points. The values of displacement or force DOF at certain nodal points of the model are specific for certain cases. In this
Fig. 11 – Boundary condition of fixed support at the bottom part of the bracket.

Fig. 12 – Boundary condition of fixed support at the spars set to be symmetrical.

Table 4 – Elastic and strength properties materials [28,29].

<table>
<thead>
<tr>
<th>Material</th>
<th>Density (kg/m³)</th>
<th>Elastic modulus, $E_{11}$ (GPa)</th>
<th>Elastic modulus, $E_{22}$ (GPa)</th>
<th>Poisson ratio, $v_{12}$</th>
<th>Shear modulus, $G_{12}$ (GPa)</th>
<th>Tensile strength in fibre direction, $X_t$ (MPa)</th>
<th>Tensile strength in transverse direction, $Y_c$ (MPa)</th>
<th>Shear strength, $S$ (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>SAN foam</td>
<td>81</td>
<td>0.107</td>
<td>0.143</td>
<td>0.3</td>
<td>0.23</td>
<td>1.38</td>
<td>0.80</td>
<td>0.99</td>
</tr>
</tbody>
</table>

2.3.3. Failure criterion

In this study, the failure criterion is considered in order to identify the ability of the wing structure withstanding to the load given. The Tsai–Wu criterion is applied in order to examine the existence of a failure surface between compressive and tensile strength of a lamina [32]. For a plane state of stress, the calculation and reduction of the Tsai–Wu criterion can be written as [33]:

$$F_1 \sigma_1 + F_2 \sigma_2 + F_{11} (\sigma_1)^2 + F_{22} (\sigma_2)^2 + F_{66} (\tau_{12})^2 + 2 F_{12} \sigma_1 \sigma_2 \geq 1$$  \hspace{1cm} (9)

where $\sigma_1$ is stress in fibre direction, $\sigma_2$ is stress in transverse direction and $\tau_{12}$ is shear stress. This equation can be used to assess failure in a composite lamina and the expression of coefficients can be expressed as follows:

**Coefficients for longitudinal strength:**

$$F_1 = \frac{1}{\sigma_{1T}} - \frac{1}{\sigma_{1C}}$$  \hspace{1cm} (10)

$$F_{11} = \frac{1}{\sigma_{1T} - \sigma_{1C}}$$  \hspace{1cm} (11)

**Coefficients for transverse strength:**

$$F_2 = \frac{1}{\sigma_{2T}} - \frac{1}{\sigma_{2C}}$$  \hspace{1cm} (12)

$$F_{22} = \frac{1}{\sigma_{2T} - \sigma_{2C}}$$  \hspace{1cm} (13)

**Coefficients for shear strength:**

$$F_{66} = \frac{1}{\tau_{12}}$$  \hspace{1cm} (14)

**Interaction coefficient:**

$$F_{12} \equiv -\frac{1}{2} (F_{11} F_{22})^{1/2}$$  \hspace{1cm} (15)

The tensile and compressive stresses in this failure criterion can be distinguished due to appropriate coefficients. It also can be easily incorporated in automated computational procedures.

Most importantly, the margin of safety (MoS) can be employed in Tsai–Wu criterion, which expressed as:

$$\text{MoS} = SR - 1$$  \hspace{1cm} (16)

analysis, two important boundary conditions required to be specified.

The first is the attachment of the bracket to the fuselage. The bottom surface of the bracket was defined as fixed due to the fuselage not included in the model. All the displacements and rotations of the bottom surface of the brackets were set to zero, $UX = UY = UZ = ROTX = ROTY = ROTZ = 0$. The first boundary condition is depicted in Fig. 11.

The second is the symmetrical condition of the wings that were defined as the boundary condition. The spars of the wing were set to be symmetrical about the $YZ$-plane. The second boundary condition is in Fig. 12.

2.3.2. Loading condition

One of the most critical aspects in the design of the wing is the assignment of lift load [31]. In this study, the calculation of the loads on the wing is subjected to the pressure that is distributed at the bottom skin of the wing. The directions inferred as the arrow labels in the nodal coordinate system, where $FX = FY = 0$ and $FZ = F$. The concentrated load is specified directly on top of main and aft spars. The loading conditions are subjected to corresponding pressure values at four different locations, as shown in Table 5 and Fig. 13. In this case, the pressure was assigned based on the distance from $X$-axis, from the root of the wing.
Table 5 – Wing loading condition.

<table>
<thead>
<tr>
<th>Total load (kg)</th>
<th>Spar</th>
<th>Distance from the root of lower skin (cm)</th>
<th>Pressure (Pa)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>0–30</td>
<td>30–60</td>
</tr>
<tr>
<td>40</td>
<td>MS</td>
<td>444.44</td>
<td>388.89</td>
</tr>
<tr>
<td></td>
<td>AS</td>
<td>166.67</td>
<td>166.67</td>
</tr>
<tr>
<td>41</td>
<td>MS</td>
<td>444.44</td>
<td>388.89</td>
</tr>
<tr>
<td></td>
<td>AS</td>
<td>166.67</td>
<td>166.67</td>
</tr>
<tr>
<td>42</td>
<td>MS</td>
<td>444.44</td>
<td>388.89</td>
</tr>
<tr>
<td></td>
<td>AS</td>
<td>166.67</td>
<td>166.67</td>
</tr>
<tr>
<td>43</td>
<td>MS</td>
<td>444.44</td>
<td>388.89</td>
</tr>
<tr>
<td></td>
<td>AS</td>
<td>166.67</td>
<td>166.67</td>
</tr>
<tr>
<td>44</td>
<td>MS</td>
<td>444.44</td>
<td>388.89</td>
</tr>
<tr>
<td></td>
<td>AS</td>
<td>222.22</td>
<td>166.67</td>
</tr>
<tr>
<td>45</td>
<td>MS</td>
<td>444.44</td>
<td>444.44</td>
</tr>
<tr>
<td></td>
<td>AS</td>
<td>222.22</td>
<td>166.67</td>
</tr>
</tbody>
</table>

Therefore, the obtained composite stack-ups of the FE analysis can be used for further optimisation of the wing structure by changing the design of leading edge and the presence of monocoque-foam-reinforced. For the subsequent analyses, the structural analysis of UAV composite wing model of TLE has variation in terms of maximum displacement, maximum von Mises stress and elastic strain.

### 3.2. Structure response on rib-reinforced of TLE wing

Table 6 provides visualisation of the numerical simulation results.

As in Table 6, the maximum displacement for the structural configuration of the TLE wing with rib-reinforced is 7.903 mm occurs at the tip of the wing. Meanwhile, the maximum von Mises stress of the wing is 77.194 MPa, which the stress is subjected to all layers of the laminated composite wing. In details, almost half of the stress subjected to the first layer of the skin, which is 47.033 MPa at the root of the wing skin. However 77.194 MPa occurred at the main spar nearer to the U-corner. On the other hand, 0.30569% of maximum shear elastic strain occurred at the aft spar, which is nearer to the root of the wing.

### 3.3. Structure response on monocoque-foam-reinforced of TLE wing

Table 7 provides visualisation of the numerical simulation results.

As in Table 7, the maximum displacement for the structural configuration of TLE wing with monocoque-foam-reinforced is 3.9002 mm occurs at the tip of the wing. Meanwhile, the maximum von Mises stress of the wing is 49.5 MPa, which the stress is subjected to all layers of the laminated composite wing. On the other hand, 0.24161% of maximum shear elastic strain occurred at the aft spar, which is nearer to the root of the wing.

### 4. Discussion

The wing of the TLE UAV is attached to the fuselage which represented as solid bracket and both spars. The end of both spars...
is to be considered as a thin walled rectangular of box and L-shape acting as the support structures of the wing under given load. In this study, the rib-reinforced is subjected to the additional support structure of the ribs on the spars using the given load, to be compared with the additional support structure of the monocoque-foam subjected to the spars. The difference between both reinforced conditions is observed according to the stress, displacement and strain. Additional observation on the skin composite structure of the wing will be explained.

### 4.1. Comparison of the results of design composite structure with Kanesan et al. [6]

According to Autio et al. [35], the error results given by commercial finite element method program should be less than 20%. This also supported by Nurhaniza et al. [36], for which the authors has the range of 10–25% of comparison the results from other finite element software. By comparing the deflections results, the registered difference percentage are in the range of 4–6%. Thus, the results are acceptable to validate finite element of composite UAV wing. The deflection results comparing both finite element (FE) analysis with Ref. [6] is shown in Table 8.

### 4.2. Comparison of the results using different reinforced conditions

From the observation, the maximum stress acting tangent to the skin of the wing, which calculated in the result 77.194 MPa, represented for the rib-reinforced condition. However, based on the analysis of monocoque-foam reinforced condition, the result showed a reduction of 35.88% difference of maximum von Mises stress proved that monocoque-foam-reinforced design provides better structural performance as compared to the rib-reinforced condition.
Table 8 – Deflection results based on total load.

<table>
<thead>
<tr>
<th>Total load (kg)</th>
<th>Deflection result (mm)</th>
<th>Kanesan et al. [6]</th>
<th>FE analysis</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>40</td>
<td>12.666</td>
<td>12.095</td>
<td>4.51</td>
<td></td>
</tr>
<tr>
<td>41</td>
<td>12.930</td>
<td>12.311</td>
<td>4.79</td>
<td></td>
</tr>
<tr>
<td>42</td>
<td>13.330</td>
<td>12.628</td>
<td>5.27</td>
<td></td>
</tr>
<tr>
<td>43</td>
<td>13.727</td>
<td>12.945</td>
<td>4.22</td>
<td></td>
</tr>
<tr>
<td>44</td>
<td>13.799</td>
<td>13.019</td>
<td>5.65</td>
<td></td>
</tr>
<tr>
<td>45</td>
<td>13.953</td>
<td>13.148</td>
<td>5.77</td>
<td></td>
</tr>
</tbody>
</table>

Table 9 – Structural response between rib-reinforced and monocoque-foam-reinforced for TLE wing.

<table>
<thead>
<tr>
<th>Structural parameters</th>
<th>Design configurations</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total displacement (mm)</td>
<td>Rib-reinforced</td>
<td>Monocoque-foam reinforced</td>
</tr>
<tr>
<td>Maximum stress (MPa)</td>
<td>77.194</td>
<td>49.5</td>
</tr>
<tr>
<td>Shear elastic strain (mm/mm)</td>
<td>0.0030569</td>
<td>0.0024161</td>
</tr>
<tr>
<td>Margin of safety (MoS)</td>
<td>0.125</td>
<td>0.875</td>
</tr>
</tbody>
</table>

Fig. 15 – Stress versus displacement along wingspan for top and bottom skin.

4.3. Composite structure behaviour on the wing skin

From the observation, both reinforced design condition used the same layered structures of material combination on the wing skin. On general basis, the location of the pressure acted on the bottom part of the wing is in distributed manner implying the maximum stresses would occur at the top and bottom part of the combination composite materials of the skin. For the static load, the relationship of shear–strain and stress–displacement for the composite skin is well explained in Figs. 15 and 16.
From Fig. 15, the graph showed the bottom part of wing skin where the loading acted on the skin has the highest displacement at distance of 200 mm from the root of the wing with the highest equivalent stress of 1.5759 MPa. In specific, 15.481 MPa of stress acted on the first layer of carbon fibre at the strain of 0.0539%. This showed that less stress acted on the skin because the load subjected to the wing which supported by the spars and ribs, as shown in Fig. 16.

5. Conclusion

In this study, the modelling of NACA 4415 UAV composite wing of TLE with the rib-reinforced model is designed in a configuration of rib supporting the spars of the wing. From the finite element analysis, it was found that 7.903 mm displacement at the tip of wing and 77.194 MPa of maximum equivalent von Mises stress at the root of wing. From the optimisation perspective, another structural configuration of the wing of TLE with monocoque-foam reinforced, whereby the foam was used at the trailing and leading edge. From the finite element analysis, the displacement is 3.9302 mm at the wing tip and the maximum equivalent von Mises stress is 49.5 MPa at the root of the wing. Therefore, the monocoque-foam reinforced design configuration showcased the better structural performance with a reduction in 50.72% of deformation and 35.88% of stress, compared to rib-reinforced design. The design of monocoque-foam reinforced is proved with better structural performance. For further study, the monocoque-foam reinforced structure concept needs for further study on Fluid-Structure interaction (FSI). FSI for tubercles composite wing with monocoque-foam reinforced is new and novel research which can be used to exhibit the capabilities to predict wing characteristic accurately. It also provides significant insight into several numerical issues encountered in order to conduct this computation.

Conflicts of interest

The authors declare no conflicts of interest.

Acknowledgments

This work is supported by UPM under GP-IPS grant, 9647200. The authors would like to express their gratitude and sincere appreciation to Department of Aerospace Engineering, Faculty of Engineering, Universiti Putra Malaysia and Laboratory of Biocomposite Technology, Institute of Tropical Forestry and Forest Products (INTROP), Universiti Putra Malaysia (HICOE) for the close collaboration in this work.

REFERENCES


