Design of the Wing of a Medium Altitude Long Endurance UAV

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Abstract: Over time, the need to extend the endurance of fixed-wing UAVs has been a challenge and requirement for the aviation industry. As a result, there are many ongoing researches on how to address this challenge while developing new concepts for an aircraft wing. This paper presents the methodology applied in the design of the wing of a lightweight unmanned aerial vehicle (UAV) that is intended for intelligence and surveillance (ISR) missions. A conceptual design was developed with the aircraft having a hybrid wing for the purpose of improving its structural design. The wing has two spars and thirteen ribs a side with an all-composite structure. Initial calculations were performed theoretically, thereafter, analyzed using the finite element analysis tool.

Index Terms: Wing Design, Unmanned Aerial Vehicle, Aircraft Structure

1. Introduction

Engineering design involves a process which is usually iterative in the form of a continual feedback of information to check the assumptions made at the initial stage [1]. The methods adopted in the design process of an aircraft usually involve the following activities [2]:

1. Problem solving through engineering or mathematical calculations; and
2. Choosing a preferred method among many alternatives.

Increase in demands in the rate of production and performance of aircraft has led to the research for newness in the design and manufacturing of aircraft wings with the use of UAVs being more pronounced these days. Therefore, it is pertinent to consider the characteristics, assembly and manufacturing method for the wing which are some of the key drivers for the design of next generation wings. The wing of a UAV according to its mission and/or intended role varies in airfoil shape, chord dimensions, span, surface area and geometry [3]. Fig 1 shows a typical example of an aircraft wing.

Aircraft are majorly classified by their wing configuration. The conceptual design of the wing adopted for this UAV is a “V-tail high wing cantilever monoplane of straight constant chord with tapered outer sections”. High wing aircraft are inherently stable as their center of mass is located below the center of lift and with high aspect ratio, they usually experience less induced drag and reduced fuel consumption [5].

However, to get a critical design objective, it was important to study similar UAVs in the same weight category. The design methodology adopted involves:

1. Calculating the design speeds and maneuver flight envelope.
2. Determining the critical loads which act on the aircraft and sizing the wing structure based on those loads.
3. Producing a CAD model.
**Fig. 1. Typical aircraft wing [4]**

### Nomenclature

- $\alpha_{2A}$: aileron lift curve slope
- $\alpha_{WB}$: wing-body lift curve slope
- $ac$: further nomenclature continues down the page inside the text box
- $b$: wingspan
- $\bar{c}$: standard or geometric mean chord (SMC)
- $c$: mean aerodynamic chord (MAC)
- $cg$: center of gravity
- $C_D$: drag coefficient
- $C_l$: lift coefficient
- $C_{M_0}$: zero lift pitching moment coefficient
- $g$: acceleration due to gravity
- $h$: location of cg as a fraction of MAC
- $H_o$: location of aerodynamic center as a fraction of MAC
- $I_x$: mass moment of inertia in roll
- $k_r$: roll radius of gyration
- $K_g$: gust alleviation factor
- $l_i$: distance from aircraft cg to tail aerodynamic center
- $L_x$: rolling moment due to aileron deflection
- $L_p$: rolling moment due to roll rate
- $L_T$: horizontal stabilizer or tail plane load
- $L_{WB}$: wing-body load
- $L_{WO}$: increment of wing load due to control deflection
- $m$: maximum take-off weight
- $n$: load factor
- $\dot{p}$: maximum roll acceleration
- $S$: wing area
- $T$: thrust
- $U_{de}$: derived gust velocity
- $V_A$: design maneuvering speed
- $V_C$: design cruise speed
- $V_D$: design dive speed
- $V_F$: design flap speed
- $V_{Sl}$: design stall speed
- $z_t$: distance of thrust axis above cg
2. Literature Review

Aircraft design has evolved over the years from the first flight of the Wright brothers (Orville and Wilbur) in 1903. The design of the wing is very important for an aircraft as it directly affects the aerodynamic shape, drag, range, maximum speed, and maneuverability [2].

Naidu and Adali (2014) designed the wing of a lightweight Medium Altitude Long Endurance (MALE) Unmanned Aerial Vehicle (UAV) intended for intelligent and surveillance (ISR) missions. The aircraft was designed and optimized to incorporate a relatively high aspect ratio to decrease drag and extend the flight time. The wingbox was modelled using FEA for a maximum loading condition of +3.2g [8].

Grodzki and Lukaszewicz (2015) designed the wing of an unmanned aerial vehicle as a multistage task which included: airfoil selection, geometrical calculations, structural design, materials selection, numerical analysis, and elaboration of technology. The simulation was carried out in SolidWorks environment while creating variety of composite structures, analyzing them and comparing their strength properties to provide valuable information in the early design stage [3].

Yu (2018) compared several types of wing such as the beam-type, single-block type, and multi-ventral plate type and chose the beam-type wing. Thereafter, performed detailed design and optimization to reduce the weight of the wing. CATIA was employed to design the 3D model for a potential realistic production [18].

Chauhan et. al (2017) attempted to formulate the structural design process for a UAV wing and subsequent optimization for SAE India Aero Design Challenge 2017. The design commenced with the identification of the structural design parameters and challenge requirements. Based on the engineering values, a mathematical interface was coded in MATLAB to calculate the mechanical equivalents for the wing at individual sections. The structural design of the wing was modelled in SolidWorks with the final mass calculated in the Preliminary Design Phase. For initial estimates, the static structural analysis of the layout was performed theoretically. However, the final design of the wing was imported into ANSYS Finite Element solver for static structural analysis. Successive iterations were performed with the final simulation of the wing which gave a safety factor of 3.73 [19].

A comparative study was carried out on UAV designs with intended weight category of similar configuration and statistical data such as the Bayraktar Tactical UAS, Elbit Systems Hermes 900 designed in accordance with (IAW) STANAG 4671, Thales Watchkeeper WK4450 and TAI Anka. The statistical data from the prototypes with empirical formulae for determining geometrical parameters as presented by Raymer and Gudmundsson [6,7] were used to derive the initial data of the proposed aircraft.

3. Design Analysis

Loads are applied to all major structural components of the aircraft. The structural load analysis helps in calculating the loads acting on the structure of the aircraft for flight maneuvers [9]. Therefore, there is a need to determine the wing loads for a proper structural design. To calculate the load actions analysis of the UAV, we must first establish its center of gravity location and subsequently determine the mass moments of inertia in roll, pitch, and yaw. The overall location of the cg was based on mass breakdown of each component and their respective cg locations taking the nose of the aircraft as the datum reference.

The n-V diagram was constructed for the most critical flight envelope which is the operating empty weight at mean sea level, see Fig 2. The diagram was produced in accordance with STANAG 4671 which has an airworthiness requirement for fixed wing UAV systems of maximum take-off weight of more than 150kg and less than 20,000kg [10]. Table 1 shows the fundamental data for the proposed aircraft as derived from initial sizing. The constraint analysis method was used to determine the appropriate wing area based on the intended aircraft role and mission.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximum take-off weight, MTOW</td>
<td>650 kg</td>
</tr>
<tr>
<td>Operating empty weight, OEW</td>
<td>332.95 kg</td>
</tr>
<tr>
<td>Wing area, S</td>
<td>10.36 m²</td>
</tr>
<tr>
<td>Mean aerodynamic chord, MAC</td>
<td>0.9 m</td>
</tr>
<tr>
<td>Cruise altitude</td>
<td>15,000 ft</td>
</tr>
<tr>
<td>Endurance</td>
<td>24 hrs</td>
</tr>
</tbody>
</table>

The positive and negative maneuvering load factors as specified by STANAG are 3.8 and -1.5 respectively. This was used in deriving the n-V diagrams and calculations. The requirements for the Equivalent Airspeeds (EAS) are calculated as follows:

- Design cruising speed,
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\[ V_C = 2.4\sqrt{\frac{Mg}{S}} \]  
(1)

Design dive speed,

\[ V_D = 1.25V_C \]  
(2)

Design stall speed,

\[ V_{S1} = \sqrt{\frac{2W}{\rho C_{L_{\max}}S}} \]  
(3)

where the maximum positive lift coefficient, \( C_{L_{\max}} \), was derived as 1.416 and air density at sea level, \( \rho \), is given from standard atmosphere table as 1.225\( \text{kg/m}^3 \).

Design maneuvering speed,

\[ V_A = V_{S1}\sqrt{n} \]  
(4)

STANAG prescribed that gust velocities of 15.2\( \text{m/s} \) at \( V_C \) must be considered at altitudes between sea level and 20,000\( \text{ft} \), and 7.6\( \text{m/s} \) for altitudes from 20,000\( \text{ft} \) to 50,000\( \text{ft} \) while 7.6\( \text{m/s} \) at \( V_D \) must be considered at altitudes between sea level and 20,000\( \text{ft} \), and 3.8\( \text{m/s} \) for altitudes from 20,000\( \text{ft} \) to 50,000\( \text{ft} \) [10]. However, the gust load factors were derived using the procedure outlined in CS-VLA which is an airworthiness code applicable to single-engine aircraft with a certified take-off weight of not more than 750\( \text{kg} \).

Aircraft mass ratio,

\[ \mu_g = \frac{2(m/s)}{\rho C_{a_{WB}}} \]  
(5)

where the wing-body lift curve slope and the geometric mean chord were gotten as 5.175/\( \text{rad} \) and 0.942\( \text{m} \) respectively.

Gust alleviation factor,

\[ K_g = \frac{0.88\mu_g}{5.3+\mu_g} \]  
(6)

Gust load factor,

\[ n = 1 \pm \frac{1/2\rho V_a V_{aWB} K_g U_{de}}{m g S} \]  
(7)

At \( V_C \), maximum positive gust load factor \( n_3 = 4.84 \), and negative \( n_4 = -2.84 \). At \( V_D \), maximum positive gust load factor \( n_3 = 3.40 \), and negative \( n_4 = -1.40 \).

The equivalent airspeeds at sea level, transition altitudes, cruise altitude and the service ceiling calculated theoretically using Equation (1-7) are detailed in Table 2.

<table>
<thead>
<tr>
<th>Table 2. Design equivalent airspeed</th>
</tr>
</thead>
<tbody>
<tr>
<td>SL</td>
</tr>
<tr>
<td>-----</td>
</tr>
<tr>
<td>OEW</td>
</tr>
<tr>
<td>( V_{S1} ) (m/s)</td>
</tr>
<tr>
<td>( V_A ) (m/s)</td>
</tr>
<tr>
<td>( V_F ) (m/s)</td>
</tr>
<tr>
<td>( V_C ) (m/s)</td>
</tr>
<tr>
<td>( V_D ) (m/s)</td>
</tr>
<tr>
<td>MTOW</td>
</tr>
<tr>
<td>( V_{S1} ) (m/s)</td>
</tr>
<tr>
<td>( V_A ) (m/s)</td>
</tr>
<tr>
<td>( V_F ) (m/s)</td>
</tr>
<tr>
<td>( V_C ) (m/s)</td>
</tr>
<tr>
<td>( V_D ) (m/s)</td>
</tr>
</tbody>
</table>
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3.1. Load action analysis

The loads that will act on the wing have to be calculated to enable proper sizing since the wing carries 95% of the total loads acting on the aircraft. As such, the failure of the wing structure will invariably lead to the failure of the whole aircraft. All the loads were calculated theoretically using known empirical formulae with the help of Microsoft excel spreadsheet. Each flight condition was analyzed with the aircraft in various configurations of altitude, velocity, mass, and load factor of the flight envelope. The largest wing-body load was obtained as 27,488N at a sudden maximum aileron deflection at $V_A$ (maneuvering speed) for maximum take-off weight at sea level. The following equations from Denis Howe [12] were used to determine the tail load on the aircraft at symmetric loading except otherwise stated:

\[ L_T = \left[ M_o + n g m (h - H_o) c + T z_{c} - K_{y} \ddot{\theta} \right] / l'_{t} \]  

where the zero-lift pitching moment, $M_o$, is given by [13]:

\[ M_o = 0.5 \rho V^2 S C_{M_o} \]  

The moment created by the lift from the wing-body, $M_{WB}$, is given by:

\[ M_{WB} = n g m (h - H_o) c \]  

In this case, $h = 0.188$ for the forward CG position, $H_o = 0.25$ and load factor ($n$) for steady trimmed level flight = 1.

For steady level flight, it was assumed that thrust is equal to drag. Hence,

\[ T = D = 0.5 \rho V^2 S C_D \]  

Lift Coefficient [13]

\[ C_L = \frac{L}{0.5 \rho V^2 S} = \frac{mg}{0.5 \rho V^2 S} \]  

Fig. 2. Maneuvering and gust envelope at operating empty weight
where the thrust offset below the aircraft center of gravity, $z_t = 0.34m$

Moment due to thrust,

$$M_{zt} = T \times z_t \quad (13)$$

**Note:** $K_{y} \dot{\theta}^{2}$ was assumed to be negligible for a steady level flight since there is no angular acceleration. Distance from the aircraft CG to tail aerodynamic center, $l'_{i} = 1.842m$.

To achieve equilibrium of forces in a steady level flight

$$L_{WB} + L_T - n g m = 0 \quad (14)$$

The following method was used to calculate the aileron loads for case 1 (full deflection at VA):

Rolling moment due to roll [14],

$$L_p = C_{lp} (\rho V_o S b^2 / 4 l_x) \quad (15)$$

where variation of roll moment coefficient with roll rate $C_{lp}$ was gotten from AVL [17] as -0.5573, yaw moment of inertia $I_x = 17.41kgm^2$, and maneuvering speed $V_A = 51.921m/s$ at sea level.

Rolling moment due to aileron deflection [14],

$$L_\zeta = C_{l\zeta} (\rho V_o S b / 2 l_x) \quad (16)$$

where variation of roll moment coefficient with aileron $C_{l\zeta} = 0.0051$ from AVL.

Maximum roll acceleration

$$\dot{p}_{max} = (\rho V_o S b L_\zeta / 2) (m k_x^2) \quad (17)$$

where the roll radius of gyration $k_x = 0.1637$

Maximum steady rate off roll,

$$p_{max} = -(V_o / b) (L_\zeta / L_p) \zeta \quad (18)$$

The value used for the aileron deflection ($\zeta$) is $20^\circ = 0.349rad$

The increment of the wing load due to the control deflection ($L_{WO}$) was calculated according to the method stated by Denis Howe and added to the wing-body load of the symmetric loading at trimmed steady level flight as detailed below:

$$L_{WO} = \rho V_o^2 S a_{2\alpha} \zeta / 2 \quad (19)$$

where aileron lift curve slope, $a_{2\alpha} = 3.42/rad$

Total wing-body load

$$Total \ L_{WB} = L_{WB} + L_{WO} \quad (20)$$

### 3.2. Load Distribution

Having obtained the design load on the wing, the next stage was to distribute the external forces in an appropriate way over the surface which generates them. These air-loads must be distributed in combination with the inertia distributions. Therefore, the span-wise lift distribution using the Schenk’s method [15,16] was established. An ideal semi-elliptical distribution, wing chord, and an average was plotted, see Fig 3.
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Since the wing is wetted with the incorporation of missile attachment as shown in Fig. 4, it was necessary to determine the inertia load due to the wing, fuel, and missile mass. The wing experiences shear and bending loads during maneuver or gust under the action of inertia and aerodynamic lift forces or air-load [13].

Therefore, the next phase was to project its shear force and bending moment diagram as shown in Fig 5 and Fig 6 respectively. The calculations of the shear force and bending moment are given below for the 3.8g steady rotary case of the wing semi-span.

Having gotten the limit wing load, we derived the ultimate wing load as

\[ L_w = 1.5 \times L \]  \hspace{1cm} (21)

where 1.5 is a factor of safety as specified in STANAG [10]. The shear force is obtained by integrating the distributed load over the semi-span of the wing using Equation (22).
where $L_W$ is the aerodynamic lift acting on the entire wing has given in Equation (21). The maximum value for the total shear force was gotten as $12,555.28\text{N}$.

The bending moment due to lift or inertia forces was calculated by integrating the shear force due to lift or inertia loads over the semi-span of the wing using Equation (23).

$$BM_{lift(inertia)} = \int_{0}^{b} SF_{lift(inertia)} \, dy$$  \hfill (23)

The value of the total bending moment was gotten as $35,714.92\text{Nm}$. Table 3 shows the result of the total shear force and bending moment.

### Table 3. Shear force and bending moment calculations result

<table>
<thead>
<tr>
<th>SF due to lift</th>
<th>SF due to inertia</th>
<th>Total SF</th>
<th>BM due to lift</th>
<th>BM due to inertia</th>
<th>Total BM</th>
</tr>
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<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>507.224</td>
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<td>126.806</td>
<td>-7.266</td>
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<td>-12931.593</td>
<td>35714.924</td>
</tr>
</tbody>
</table>

Fig. 5. Shear force diagram
4. CAD Layout

The detailed design of the wing as shown in Fig. 7 reflects design modifications resulting from verification and validation of the finite element analysis. The CAD model of the entire wing assembly was designed using CATIA V5. The design approach emphasized structural simplification and reduced parts to minimize cost. The assembly drawing of the wing is shown in Fig. 4 which represents the final CAD model of the wing structure.

5. Results

Having derived the shear force and bending moment results, the Strand7 Finite Element Analysis Software was used to validate the results. The result of the analysis for the shear force and bending moment for the 3.8g steady rotary was obtained as shown in Fig. 8.
Having derived the results from Strand7 software and Excel spreadsheet, the results were compared. Table 4 shows the comparison of the results.

Table 4. Comparison of results

<table>
<thead>
<tr>
<th>Load case</th>
<th>Maximum Shear Force</th>
<th>Maximum Bending Moment</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Excel (N)</td>
<td>Strand7 (N)</td>
</tr>
<tr>
<td>Steady rotary</td>
<td>12,555.28</td>
<td>11,977.92</td>
</tr>
</tbody>
</table>

6. Conclusion

At the initial design phase, a design environment for the lifting surfaces was created using the Athena Vortex Lattice software. Thereafter, the critical design case was established as the maximum aileron deflection at maximum take-off weight and maneuvering speed with a wing-body load of 27,488N. Furthermore, the maximum shear force and bending moment were derived as 12,555N and 35,715Nm respectively. Finally, the mass estimation for the proposed all composite structure was done using CATIA which weighed 56.64kg.

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References


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