Design of a two seats light sport aircraft

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Abstract. Although flying machines have made their tremendous technological advancement since the first successfully flight of heavier-than-air aircraft, its benefits to the greater community are still belittled. One of the reasons for these drawbacks is because the relatively high cost needed to fly on the typical light aircraft. A smaller and lighter plane, widely known as Light Sport Aircraft (LSA) which has the potential to attract more people of the greater community to actively participate in flying the LSA for recreational, business trip or other personal purposes. In this paper, we propose a new LSA design with some simple, yet importance analysis required in the aircraft conceptual design stage.

1. Introduction

Leonardo Da Vinci once wrote ‘nothing is wanting and nothing is superfluous’ as he contemplating on the flying seagulls [1]. In the early history of flight, human have the thought that the dream of flying can only become a reality if they are able to imitate the birds. The idea of flying for recreational has emerged in the very beginning of the first human successful flight in 1903. However, due to many reasons such as economical and immature technology, this dream is not easily come true until the last few decades.

The design of airplane has change evolved significantly since in the Wright’s Brother era, where we can witnessed the evolution from the strut and wire biplane to the highly sopisticated smooth fuselagejet engine aircraft. The complete aircraft design process consists of three distinct phases, which are conceptual design, preliminary design, and detail design [2,3]. The design process commonly starts with specific requirements, sets forth either by potential customers or the desire to implement innovative ideas and technology [4,5].

Design requirements contain several parameters such as aircraft range and payload, takeoff and landing distances, and speed requirements. Usually, the conceptual design contains seven important tasks as depicted in Fig. 1. At the end of the design process, the overall shape, size, weight, and performance of the aircraft are estimated, and ready to proceed to the next design phases.

2. Project Brief

Light sport aircraft (LSA) is normally referred to as an aircraft that has 2 seats, with the maximum gross weight of 1320 lb and the stall speed of not more than 45 knots [6].
In this paper, we present the conceptual design process of a two-seats LSA, and for ease of reference, we called this aircraft as ZA230. The project starts with the requirement given in Table 1. These requirements are developed from current market needs for LSA and the appropriate aircraft standard for LSA design. Thus, from this set of requirements we develop all the necessary tasks.

| Requirement | Crude Weight Estimation |
|-------------|-------------------------|
| Critical Performance Parameters; $C_L_{max}$, $L/D$, $W/S$, $T/W$ |
| Configuration Layout |
| Better Weight Estimation |
| Performance Analysis |
| Optimization |
| Iterate |
| No |
| Yes |
| Preliminary Design |

**Figure 1.** Steps in conceptual design.

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**Table 1. Requirements for Light Sport Aircraft.**

| Requirement                                         | Value       |
|-----------------------------------------------------|-------------|
| Maximum speed                                       | 120 knots   |
| Range                                               | 340 mile    |
| Service ceiling                                     | 20,000 ft   |
| Stall speed                                         | 40 knots    |
| Landing distance (clear 50 ft obstacle)             | 1,500 ft    |
| Take off distance (clear 50 ft obstacle)            | 800 ft      |

**3. First Weight Estimation**

The first weight or crude weight estimation is based on the available statistics for the existing or ever built light sport aircrafts. Here we want to see the trend of the weight for this aircraft. The gross weight buildup, $W_0$, of the aircraft consists of crew ($W_c$), payload ($W_p$), fuel ($W_f$) and empty weight, summarizes in Eq. (1). Express all terms in Eq. (1) as fractions of $W_0$, and rearrange, we have Eq. (2) to solve for $W_0$.

\[
W_0 = W_c + W_p + W_f + W_e \tag{1}
\]

\[
W_0 = \frac{W_c + W_p}{1 - W_f/W_0 - W_e/W_0} \tag{2}
\]

From Eq. (2), $W_c$ and $W_e/W_0$ are already known. So we have to estimate $W_f/W_0$ based on the typical aircraft mission profile depicted in Fig. 2. The fuel needed or cruising flight is estimated by using Brequet equation [5]. For the takeoff run, we assume fuel burned is 3%, while for climb, the
aircraft consumed 1.5% fuel. The descent flight as indicated by segment 4-5, we assume the covered horizontal distance is part of the required range, thus there is no fuel difference in this segment. For the landing run, we assume only 0.5% fuel burned. Allowance is also made for trapped fuel of 6%. By putting together all these assumptions, we write the fuel weight ratio as given in Eq. (3).

\[
\frac{W_f}{W_o} = 1.06 - 1.008e^{L/D_{MP}}
\]  

(3)

![Figure 2. Typical aircraft mission profile.](image)

The range, \(R\) for the LSA is 340 miles, specific fuel consumption, \(c\) is \(2.02 \times 10^{-7}\) lb/(ft.lb), typical propeller efficiency \((\eta_p)\) is 0.85, and lift to drag ratio, \(L/D\) is 14, from Eq. (3) we have \(\frac{W_f}{W_o} = 0.088\). Similarly, we estimate the empty to gross weight ratio for ZA230 from statistics. It is based on historical data of similar class aircrafts as shown in Fig. 3. From this figure, it is reasonable to select \(\frac{W_e}{W_o} = 0.55\) for ZLA.

![Figure 3. Historical weight for light sport aircraft.](image)

ZA230 is a two seats aircraft, for simplicity we assume both occupants have the average of 200lb weight and each of them is allow to carry 40lb luggage. Thus we have the payload \(W_p = 40lb + 40lb = 80lb\). Substitute all these values in Eq. (2), we have the following gross weight,
\[ W_0 = \frac{W_c + W_p}{1 - W_f/W_0 - W_e/W_0} \]
\[ = \frac{400 + 80}{1 - 0.088 - 0.55} \]
\[ = 1270 \text{ lb} \]

By using this gross weight, we can estimate the fuel weight as follows,

\[ W_e = 0.088W_0 = 0.088(1270) = 112 \text{ lb} \]

4. Estimation of Crucial Performance Parameters
There are four main parameters that determine the performance of the aircraft; maximum lift coefficient, maximum lift to drag ratio, wing loading and thrust to weight ratio. The subsequent sections discuss the estimation of these parameters.

4.1. Maximum Lift Coefficient, \( C_{L_{\text{max}}} \)
The maximum lift coefficient depends on the selected wing airfoil. Abundance of airfoils are available for this purpose. Although at this stage it is not quite critical to have the detail of the chosen wing airfoil, it is necessary for us to estimate the expected maximum lift coefficient for our aircraft. To have the high lift device, we simply use a simple plain flap with fixed slats, that is typically provides \( C_{L_m} = 3.4 \) [8].

4.2. Maximum Lift to Drag Ratio, \( L/D \)
The lift to drag ratio is also estimated based on the statistical value. For ZA230 light sport aircraft, it is reasonable to have \( L/D \) of 14, similar to many existing light aircraft [9].

4.3. Wing Loading, \( W/S \)
The wing loading is estimated based on two constraints; stall speed, \( V_s \) and landing distance, \( L_g \). For steady flight, stall speed can be expressed in Eq. (5).

\[ V_s = \sqrt{\frac{2\rho_{\infty}W/S}{C_{L_{\text{max}}}}} \]

The requirement specifies \( V_s = 40 \text{ kts} = 67.5 \text{ ft/s} \). The air density at sea level is \( \rho_{\infty} = 0.002377 \text{ slug/ft}^3 \), solve for \( W/S \) in Eq. (5), we have,

\[ W/S = 0.5\rho_{\infty}V_s^2C_{L_{\text{m}}} \]
\[ = 0.5(0.002377)(67.5)^2(3.0) \]
\[ = 16.2 \text{ lb/ft}^2 \]

Next we compute \( W/S \) from landing distance constraint. The total landing distance (\( L_g \)) is composed of approach (\( L_a \)), flare (\( L_f \)) and ground roll (\( L_r \)) distances. We consider the typical approach angle for
light sport aircraft of 10o. The ground roll distance can be approximated by using Eq. (7).

\[ L_r = 53.5 \sqrt{\frac{W}{S}} + 12.3 \frac{W}{S} \]  

(7)

Since the allowable landing distance is stated in the requirement is 1100 ft, and we can compute \( L_a \) and \( L_f \), the allowable distance for \( L_r \) is 392.3 ft. Thus we have the wing loading of \( \frac{W}{S} = 15 \text{ lb/ft}^2 \) as computed from Eq. (8).

\[
\frac{W}{S} = \left[ \left( \frac{-100.1}{\sqrt{C_{lm}}} + \sqrt{\left( \frac{100.1}{\sqrt{C_{lm}}} \right)^2 + \frac{172.8}{C_{lm}} (L_r) \frac{86.4}{C_{LM}}} \right)^2 \right] = 15 \text{ lb/ft}
\]

(8)

Now we have two different wing loadings, \( W/S = 19 \text{ lb/ft}^2 \) and \( W/S = 15 \text{ lb/ft}^2 \) obtained from stall speed and landing distance constraints respectively. Notice from Eq. (7), smaller \( W/S \) will cause smaller ground roll, which is clearly desirable. Thus, the wing loading for ZA230 is selected as \( W/S = 15 \text{ lb/ft}^2 \). By using this wing loading and gross weight calculated in Eq. (4), the wing area is computed as,

\[
S_w = \frac{W_o}{W/S} = \frac{1270}{15} = 84.7 \text{ ft}^2
\]

(9)

4.4. Power Required, \( P_R \)

The power required, \( P_R \) is determines the takeoff distance (\( L_{TO} \)), rate of climb (\( ROC \)), and maximum velocity (\( V_m \)) requirements. Each of these requirement should be evaluated in order to obtain the design value of \( T/W \). We begin the evaluation based on the required take off distance \( L_{TO} = 1100 \text{ ft} \). Recall for the selected wing loading and maximum lift coefficient, we can calculate the thrust-to-weight ratio as follows,

\[
\frac{T}{W} = \frac{15.9W/S}{[L_{TO} - 0.22V_s^2 \sin^{-1}(1 - 50/0.22V_s^2)]C_{lm}} = 0.3
\]

(10)

Then, the required power, \( P_R \) to take off with the gross weight of \( W_o = 1270 \text{ lb} \) can be calculated as,

\[
P_R = 0.77TV_s = 0.77 \frac{T}{W}W_oV_{\infty} = (0.77)(0.3)(1270)67.5 = 19802.5 \text{ ft.lb/s}
\]

(11)

Take the typical propeller efficiency of 0.8, the power from the engine should be at least,
Next is to estimate the power required based on the rate of climb requirement, $RC = 500 \text{ ft/min}$ (8.33 ft/s) at sea level. The maximum $RC$ of a propeller driven airplane can be written as,

$$RC = \frac{\eta_p P}{W} - \left( \frac{2}{\rho_\infty} \sqrt{\frac{W}{3C_{D_o} S}} \right)^{1/2} \frac{1.155}{L/D}$$  \hspace{1cm} (13)

For typical single engine airplane, the zero-lift drag coefficient is approximately, $C_{D_0} = 0.017$ and with the earlier assumed $L/D = 14$ and $C_{LM} = 3.5$, then we have $K$ as follows,

$$K = \frac{1}{4C_{D_0}(L/D)^2} = \frac{1}{4(0.017)(14)^2} = 0.075$$  \hspace{1cm} (14)

We can also calculate the wing aspect ratio, $AR$ as follows. For light aircraft, it is reasonable to have Oswald efficiency, $e = 0.6$.

$$AR = \frac{1}{\pi e K} = \frac{1}{\pi(0.6)(0.075)} = 7.07$$  \hspace{1cm} (15)

Solve Eq. (14) for power required, we have,

$$P = \left( RC + \left( \frac{2}{\rho_\infty} \sqrt{\frac{W}{3C_{D_o} S}} \right)^{1/2} \frac{1.155}{L/D} \right) \frac{W_0}{\eta_p}$$  \hspace{1cm} (16)

$$= \left( 8.33 + \left( \frac{2}{0.00237} \sqrt{\frac{0.075}{3(0.017)}} (15)^{1/2} \frac{1.155}{14} \right) \frac{1270}{0.8} \right)$$

$$= 31066.5 \text{ ft. lb/s}$$

$$= 31066.5 / 550 = 56.5 \text{ hp}$$

The final constraint for $P_R$ estimation is the maximum speed requirement, $V_{max} = 115 kts$ (191 ft/s). The specified maximum speed is assumed to occur at 3000 ft, and at this flight phase, thrust is equal to drag, therefore,
Consider propeller efficiency of 80%, we can estimate the power required to meet the maximum speed as follows,

\[
T = D = \frac{1}{2} \rho \omega V_o^2 S C_{D_0} + \frac{2KS}{\rho \omega V_o^2} (W/S)^2
\]

\[
= \left( \frac{1}{2} \right) (0.002175)(191)^2(84.7)(0.017) + \frac{2(0.075)(84.7)}{0.002175(191)^2}(15)^2
\]

\[
= 93.2 \text{ lb}
\]

From this power analysis, clearly the rate of climb specification is the determining factor of the required power from the engine, where in this case, the engine should be able to produce a maximum power of 56.5 hp or more. Several engines are commercially available to meet our requirement, where the most common is Rotax 582 (65 hp). The power loading can be expressed as \( \frac{W_o}{P} = \frac{1270}{40.4} = 31.44 \text{ lb/Hp} \).

5. Aircraft Layout

The parameters that we have estimated so far are sufficient to allow us to produce the first conceptual layout of the airplane. As we progress into more details analysis in later design stages, the layout will be revised, modified and improved accordingly. We begin with the determination of several design aspects of the overall airplane configuration. There are a lot of arguments on advantages and disadvantages of certain design feature. As the design process involves much of compromises, the deciding factor is much depends on what the airplane is intended to perform.

In this regards, we choose conventional design features of high wing, tractor type, tricycle landing gear and T-tail for our LSA design. The overall proposed aircraft layout is shown in Fig. 4a, Fig. 4b and Fig. 4c. By using Rotax 582 engine and having two people on board, the centre gravity \((c.g)\) is estimated is at \( \bar{x} = 6.72 \text{ ft} \) location.

5.1. Wing Configuration

ZA230 aircraft must be able to fly at very low speeds for take off and landing, and yet it must also able to offer acceptable cruise performance. In this regards, we must be able to provide high lift wing with the wing area as small as possible. The task is to delay the wing stall to a higher angle of attack, and we use the common solution to this problem which is to add flap on the wing. The chosen wing planform is rectangular, with unity taper ratio.

On top of that, the leading edge slat is also added to the wing, where it can prevent the stall up to approximately 30° angle of attack [7]. Note that is a wise decision to choose high-wing configuration because it offers a larger space for large wing flaps needed for high lift. The estimated wing area is
84.7 ft\(^2\), and aspect ratio, \(e = 0.6\), thus for a rectangular wing planform we have the wing span and wing chord of 24.5 ft and 3.5 ft respectively. We use half span for both flap and aileron.

### 5.2. Fuselage

For ZA230, the fuselage is to accommodate engine, people, fuel and baggage. The two seats are arranged in side by side configuration. Behind the seats are the space for an external fuel tank and baggage. For the design simplicity and weight saving, the remaining fuselage section is narrowed down to a straight tail boom. Arbitrarily we choose the length of the aircraft from the nose to the tail mean aerodynamic centre is 20.2 ft. This is a reasonable number if we compare the length to the wing span, as it has a typical ratio for light aircraft.

### 5.3. Tails

The T-tail configuration is selected where the horizontal tail is located somewhere on the vertical tail root and on the end of the fuselage. This configuration is favored because it provides substantially good undisturbed airflow over the horizontal tail, thus increasing the pitch control effectiveness that is desirable for LSA aircraft [2,4]. Furthermore, this configuration enables to have smaller vertical tail because the placement of the horizontal tail acts as winglet and thus increasing of the effective aspect ratio.

The tail sizing is based on the chosen of typical tail’s volume ratio for GA aircraft, defined as follows,
\[ V_{ht} = \frac{l_{ht} S_{ht}}{c S} = 0.7 \quad (19a) \]
\[ V_{vt} = \frac{l_{vt} S_{vt}}{b S} = 0.04 \quad (19b) \]

From the c.g location and Eq. (19), we can compute the tail arms \((l_{ht} \text{ and } l_{vt})\) and both areas for horizontal and vertical tails \((S_{ht} \text{ and } S_{vt})\). Tail is similar in terms of shape with the wing but it has different role. Unlike the wing, whose function to generate lift, only small aerodynamic forces are required on the tail to maintain stability and control. Thus, smaller aspect ratio can be allocated for the tail as shown in Fig. 4.0, together with several main dimensions.

5.4. Landing Gear
The landing gear is the tricycle configuration which is most commonly used on many aircrafts. Tricycle configuration has the advantages of higher stability on the ground and allows the aircraft to land at fairly crabbing attitude. For structural and space reasons, we place the main landing gear at the wing’s centre and at almost c.g location.

Similarly, the nosewheel is located at the position as shown in Fig. 4b because of the strut is directly connected to the main structure. The distance between nose wheel and main landing gear is estimated to be 4.20 ft. Both main landing use the same tire.

6. Better Weight Estimation
Now we already have the configuration layout of the aircraft. By having the layout, we can have a better weight estimation when compared to the method used in Section 3.0. This method is also based on the historical data, but it is referring to aircraft layout, which is the actual shape and size of the aircraft. The estimation of major aircraft components are approximated from certain ratios as given in Table 2.0. Based on the aircraft shown in Fig. 4.0, we need to calculate the exposed wing and tail areas and the fuselage wetted area. The calculation of the wing and tail areas is quite straightforward, but we need to simplify the fuselage’s shape in order to calculate the wetted area.

Based on these areas and the initial gross weight estimation, we have a better aircraft empty weight of 665.1 lb as given in Table 3. In the first weight estimation, we have chosen the empty to gross weight ratio of \(W_e/W_o = 0.55\), where from this ratio, we found that our aircraft gross weight is 1270 lb. We can check the aircraft empty weight is \(W_e = 0.55 \times 1270 \text{ lb} = 698 \text{ lb}\). As expected, we have empty weight reduction, which is about 5% when compared to weight obtained in the previous crude estimation.

Recall the gross weight equation given in Eq. (1), with the better empty weight of 665.1 lb, we now can have the new gross weight of,

\[ W_o = W_e + W_p + W_f + W_e \quad (20) \]
\[ W_o = 200 + 80 + 111.76 + 665.1 = 1256.86 \text{ lb} \]
Table 2. Estimation of aircraft empty weight.

| Component Buildup      | Multiplier                        | Weight    |
|------------------------|----------------------------------|-----------|
| Wing weight            | $= 2.5 \times$ Exposed wing planform area | 175.0 lb  |
| Horizontal tail weight | $= 2.0 \times$ Exposed horizontal tail area | 26.0 lb   |
| Vertical tail weight   | $= 2.0 \times$ Exposed vertical tail area | 8.5 lb    |
| Fuselage weight        | $= 1.4 \times$ Fuselage wetted area | 156.8 lb  |
| Landing gear weight    | $= 0.014 \times$ Gross weight    | 17.8 lb   |
| Installed engine weight| $= 1.4 \times$ Engine weight      | 154.0 lb  |
| All else empty         | $= 0.1 \times$ Gross weight       | 127 lb    |

Aircraft empty weight: 665.1 lb.

With this new gross weight, we return to Table 2 to recalculate new landing gear and all else empty weights, follows by another new the empty and gross weights. We have to iterate the calculation for $W_e$, $W_f$ and $W_o$ until convergence is obtained for the gross weight. In this case we have the converged empty, fuel and gross weight of 663.2 lb, 110.3 lb and 1253.5 lb respectively as shown in Table 3.

Table 3. Iteration for Gross Weight.

| Iteration | $W_e$ (lb) | $W_f$ (lb) | $W_o$ (lb) |
|-----------|------------|------------|------------|
| 1         | 665.0      | 112.0      | 1257.0     |
| 1         | 663.6      | 110.6      | 1254.2     |
| 2         | 663.3      | 110.4      | 1253.7     |
| 3         | 663.2      | 110.3      | 1253.5     |
| 4         | 663.2      | 110.3      | 1253.5     |

Note that the weight estimation will continuously get improved as we move to later stages of the design process. This is apparent because as we know the actual components to be fixed on the aircraft, the corresponding weight buildups are more accurate.

7. Performance Analysis
The final step in the aircraft conceptual design stage is to examine whether the proposed aircraft configuration gives the required performance requirement. Here we only analyse the most three important performance parameters which are stall speed, take off distance and landing distance. From the new value of the gross weight, the updated wing loading and power loading are as follows,

$$W/S = 1253.5/84.7 = 14.8 \text{ lb/ft}^2$$  \hspace{1cm} (21)

$$W/P = 1253.5/40.4 = 31.03 \text{ lb/ Hp}$$  \hspace{1cm} (22)
Based on this new wing loading, the stall speed can be calculated by Eq. (5),

\[
V_s = \sqrt{\frac{2}{0.002377}(14.8)\frac{1}{3.4}} = 60.5 \frac{ft}{s}
\]

\[= 38.9 \text{kts} \quad (23)\]

Recall that the specification given in the requirement in Section 2 for the stall speed is 40 kts. We have just calculated, for the proposed design the stall speed is 38.9 kts, thus satisfying the requirement. For the landing distance, we can calculate by using Eq. (24), where it is the summation of the approach, flare and ground roll distances. Note that Eq. (24c) is similar with Eq. (7), where we rewrite it in such a way that the lift coefficient, gravitational, air density and ground friction appear in the equation.

\[
L_a = \frac{50 - 0.00032V_s^2}{0.05} = \frac{50 - 0.00032(60.51)^2}{0.05} = 976 \text{ ft} \quad (24a)
\]

\[
L_f = \frac{(1.23V_s)^2 \sin \theta_a}{0.2g} = \frac{(1.23(60.51))^2 \sin 3^\circ}{0.2(32.2)} = 45 \text{ ft} \quad (24b)
\]

\[
L_r = 100 \sqrt{\frac{W/S}{C_{lm}} + \frac{1.32W/S}{g\mu C_{lm}}} \quad (24c)
\]

\[
= 100 \sqrt{\frac{14.8}{3.4} + \frac{1.32(14.8)}{(32.2)(0.002377)(0.4)(3.4)}} = 396 \text{ ft}
\]

In Eq. (24b), \( g \) is the gravity (32.2 ft/s^2), \( \rho \) is the air density at the sea level (0.002377 slug/ft^3), and \( \mu \) is the typical ground friction coefficient (0.4). The approach angle \( \theta_a \) is taken as 3\(^\circ\) for the typical value. From Eq. (24), the landing distance is 1417 ft. We satisfied the landing distance requirement, as it is stated in the specification that the landing distance must not greater than 1500 ft.

The take off distance comprises of the ground roll \( L_r \) and airborne distance \( L_a \), as given in Eq. (25). The obstacle height angle, \( \theta_b \) calculated for our case is 20\(^\circ\). Note that we have the take off distance of 500 ft. Again, as stated in the requirement, the take off distance should not greater than 800 ft, we clearly satisfied this requirement by having the take off distance of 500 ft.
\[ L_a = \frac{6.96V_s^2}{g} \sin \theta_b \]  
\[ = \frac{6.96(60.51)^2}{32.2} \sin 20^\circ \]  
\[ = 271 \text{ ft} \]  

\[ L_r = \frac{1.21(W/S)}{g\rho(T/W)C_{Lm}} \]  
\[ = \frac{1.21(14.8)}{32.2(0.002377)(0.3)(3.4)} = 229 \text{ ft} \]  

8. Conclusion

We have discussed the essential steps on how to develop a LSA design. In the conceptual aircraft design stage, the analysis are clearly presented and discussed methodologically. It started with a set of the requirement, and then based on the historical data of aircrafts of the similar class to that we are intend to design, we select the most appropriate empty to gross weight ratio. From here we can make initial estimates of several important performance parameters. Then the aircraft layout is proposed, with in the consideration of many configuration options.

This is followed by the better weight estimation based on the size of the aircraft. Finally the better weight estimation guides us to calculate the performance parameters. We found the proposed light sport aircraft design is able to meet the performance criteria. Since a lot of improvement needed to be done on the analysis, such as the calculated take off distance is far exceed the requirement, optimization is needed for the analysis, and this is left for future work. In this context, we may redesign the aircraft in order to increase the take off distance, where this may be possible by having lower lift coefficient. Consequently, having lower lift coefficient is simply using a more simple and of course less cost.

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