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Design of a Micro Class Unmanned Aerial Vehicle

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Abstract - *This document details the comprehensive design, analysis, performance, and the manufacturing process used to build the aircraft named "Falcon". The main objective is to design and manufacture an Unmanned Aerial Vehicle capable of transporting the highest possible load with lower weight and the aircraft being able to achieve excellent performance. In conjunction to this, it fulfills its mission by following the SAE design stipulations and proved to be an aircraft airworthy to operate.*

Key Words: Fuselage, Wing, Payload, Airfoil, Empennage, Unmanned Aerial Vehicle

1. INTRODUCTION

The final aircraft design consists of a monoplane and rectangular-tapered high wing. The aircraft's fuselage has a Warren truss structure with an empennage supported by a tail boom that is united to the truss structure. The propulsion system is constituted by a 1100kV brushless direct current motor and a single 2-blade propeller.

2. LITERATURE REVIEW

The very first piece of literature that had to be carefully studied was the SAE Aero Design Challenge 2020 Rules. These rules provided a set of parameters and boundaries which should not be violated. Since the rules differ from year to year and between the events, the author has tried his level best to fulfill the various requirements. Other literature that has been researched is the reports and results from previous universities that have competed in SAE Aero Design Competitions. There were plenty of proven ideas that can be found in this kind of research as well as ideas that were found to have been not so good. In fact, in some cases bad design ideas lead to catastrophic failure and damage of the airplane. The author has prioritized learning from previous mistakes as much as possible. Research on various aircraft manufacturing techniques has also been done and is also of great importance.

3. GENERAL METHODOLOGY

The Engineering Design Process as shown in Fig -1 is iterative, meaning that the process can be repeated as many times as needed, optimizing along the way to learn from failure and discover new design possibilities to arrive at the most optimum solutions, simultaneously keeping in mind the time and financial constraints. The aircraft design was made considering situations that could put aircraft in risks like strong winds, battery discharging, excessive weight and construction delays.

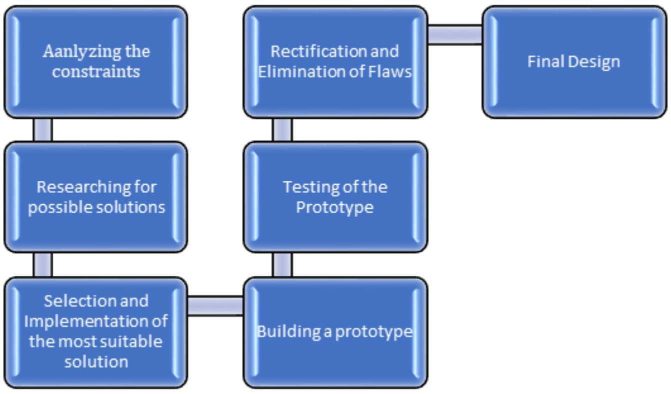

Fig -1: Engineering Design Process

4. DESIGN APPROACH AND CALCULATIONS

4.1 Wing Planform

Various wing planforms were compared in order to achieve the best possible wing configuration and maximum lifting capabilities within the given dimensional constraints. Initially, elliptical wing planform was found to be ideal as it has lowest induced angle of attack, induced drag and stall characteristics as compared to other planforms but its manufacturing is quite tedious. Whereas rectangular planform is easy to manufacture, but its stalling characteristics are high. Tapered wing planform characteristics are near to elliptical one but the wing span increases for the desired wing area. So, a combination of rectangular and taper was selected in order to achieve desired area with the minimum wing span. Lift distribution of rectangular-taper planform was nearly the same as that of the elliptical planform.

Rectangular-taper planform has the following perks:

1. It reduces wing span as compared to taper wing.
2. Due to its end taper portion, Oswald efficiency factor increases.
3. Neutral point shifts backwards as it increases static margin of plane, hence stability also increases.
4. Tip losses are reduced due to end tapered portion.

4.2 Airfoil Selection

In order to meet the requirement of achieving high payload fraction of about 0.6, the research for an airfoil with low Reynolds number and high lift producing traits was carried out.

Table -1: Airfoil Comparison

Parameters	E216	E423	S1223	S1210
$C_{L\,Max}$	1.6	1.7	2.1	2.05
Stall Angle	13.5	15	10.5	14.5
C_L/C_D	68.5	51.9	54.5	59.3
$C_{D\,Min}$	0.019	0.022	0.021	0.02
Camber (%)	4.7	9.5	8.1	6.7
Thickness (%)	10.4	12.5	12	12

From the above 2D coefficient values, the airfoils were selected for further detailed analysis using XFLR5 and Ansys.

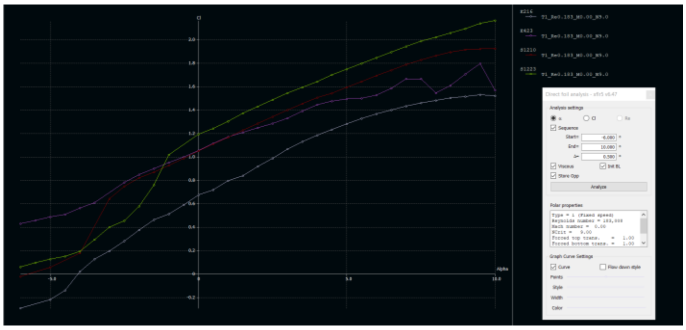

Fig -2: C_L v/s Alpha

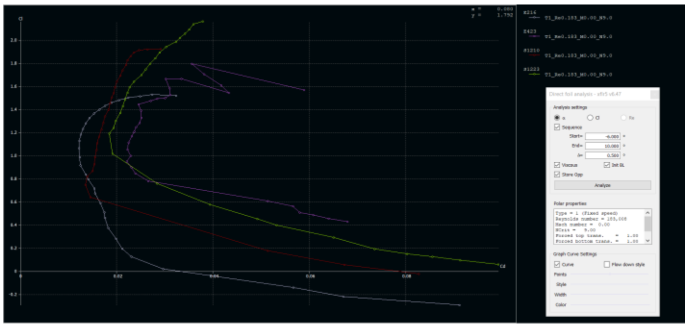

Fig -3: C_L v/s C_D

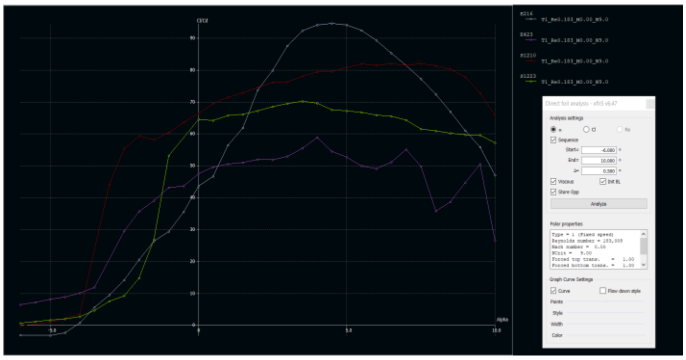

Fig -4: C_L/C_D v/s Alpha

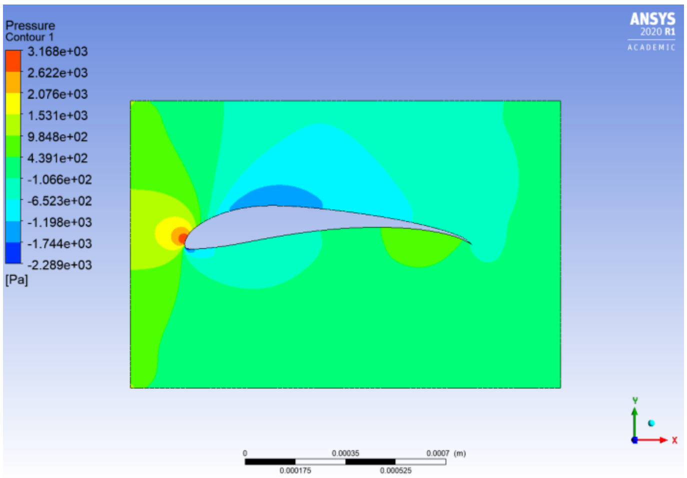

Fig -5: Pressure Contour

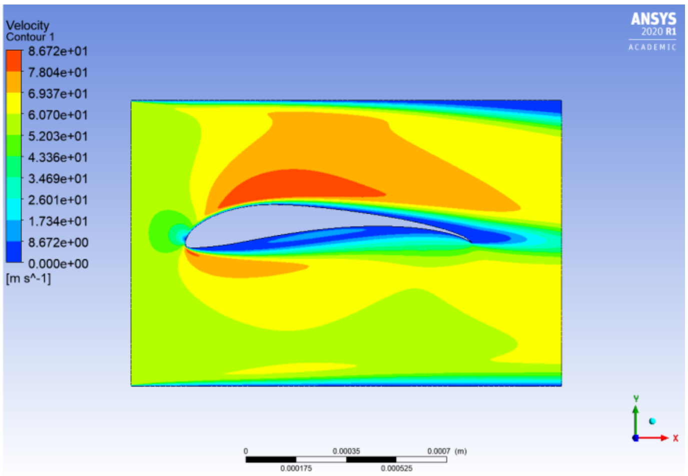

Fig -6: Velocity Contour

4.3 Wing Traits

To achieve maximum payload fraction, the empty weight of the plane was assumed to be 750 gm and payload fraction as 0.55, so the payload to be lifted was 950 gm. The total lifting weight of the plane came out to be 1700 gm.

$$\text{Wing Area} = \frac{\text{weight}}{\text{wing loading}} = \frac{1700}{0.6713} = 2532.3\ \text{cm}^2$$

By considering factor of safety, wing area was taken as 2550 cm^2.

Wing loading is the structural concept as it gives the distribution of load over wing. According to the type of particular aircraft which is categorized as moderate speed flyer, wing loading of about 16 to 22 oz./sq. ft. is suggested.

$$\text{Wing Loading} = \frac{\text{weight}}{\text{wing area}} = \frac{1700}{2550} = 0.6666\ \text{gm/cm}^2$$

0.6666 gm / cm^2 = 21.84 oz./sq. ft. which is in range of 16-22 oz./sq. ft.

For good gliding properties, aspect ratio of 6 was selected.

$$L = \frac{1}{2} \times \rho \times V^2 \times C_L \times S_W$$

which gives us C_L as 0.6317.

Now,

$$\text{Aspect Ratio} = \frac{(\text{span})^2}{\text{wing area}}$$

which gives us wing span as 123.69 cm.

Also,

$$\text{Aspect Ratio} = \frac{\text{span}}{\text{chord}}$$

which gives us chord length as 20.615 cm.

For meeting the requirements of mentioned payload fraction, a mean aerodynamic chord (MAC) of 0.2061m and cruising velocity of 13 m/s were opted.

$$R_n = \frac{\rho v C_W}{\mu}$$

which gives the Reynolds number as 221820.18.

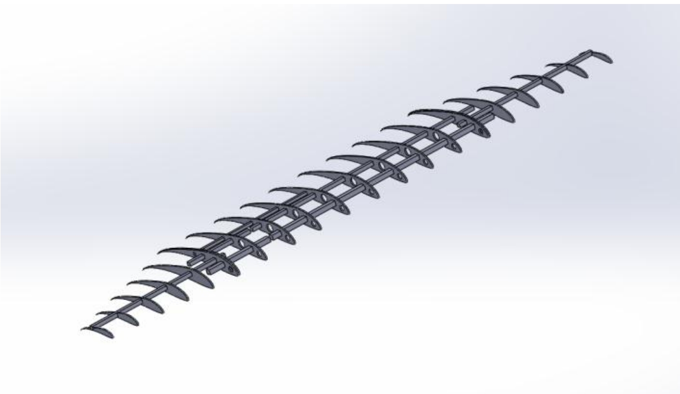

Fig -7: Wing – CAD

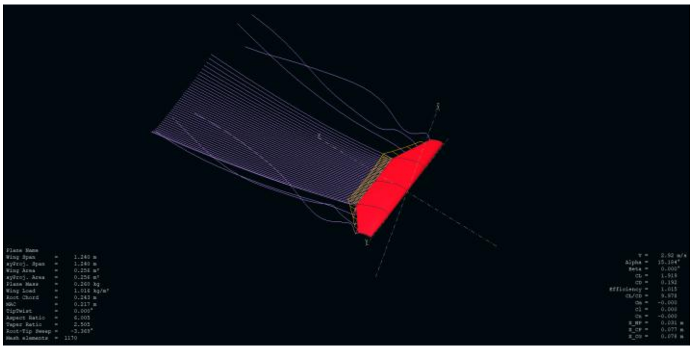

Fig -8: Wing Performance

4.4 Fuselage

In order to decrease drag, the fuselage was designed in shape of airfoil. Since tail boom was supposed to be used to attach tail with fuselage, unsymmetrical airfoils were ruled out as these airfoils have thin trailing edge and it would have been difficult to rigidly mount the tail to the end of the fuselage. Thus, shape of fuselage was designed in the shape of N-24 airfoil. The length and width of the fuselage being 55.2cm and 6.8cm respectively, for easy accommodation of electronics. The structure of fuselage needed to be strong and lightweight, so that it could sustain static as well as dynamic forces during flight. The base was constructed by using Warren truss structure which has excellent strength to weight ratio.

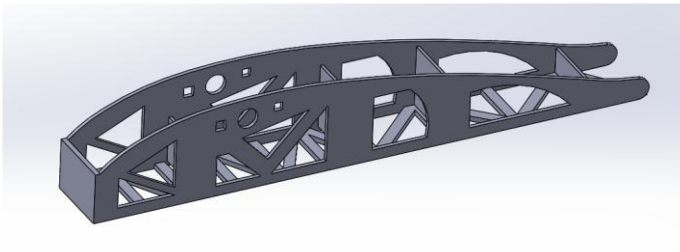

Fig -9: Fuselage

4.5 Empennage and Stability

The empennage of the aircraft is responsible for providing sufficient balancing moments in level flight. Thus, the tail was designed which provided adequate stability and control of the aircraft. A conventional tail with a tail-boom configuration was chosen for the aircraft. This design has the lightest structural weight compared to other configurations.

Using an iterative methodology, the area of horizontal and vertical stabilizers was found to be 510cm^2 and 183.6cm^2 respectively, of which the area of elevator and rudder was 153cm^2 and 61.2cm^2 respectively.

For horizontal stabilizer: $V_H = \frac{(S_H \times L_H)}{(S_W \times MAC)}$

And for vertical stabilizer: $V_V = \frac{(S_V \times L_V)}{(S_W \times W_S)}$

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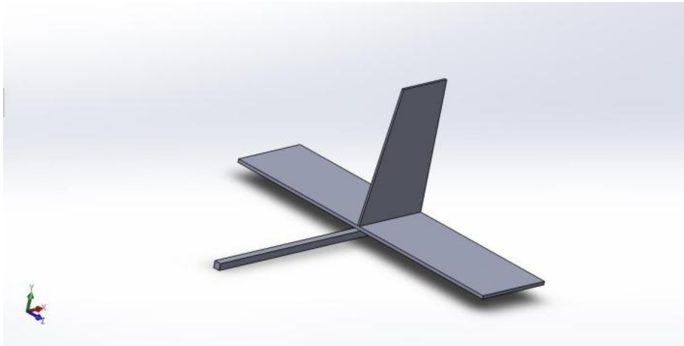

Fig -10: Empennage with tail boom – CAD

The aircraft's stability depends mainly on the position of center of gravity (CG) with respect to the neutral point (NP) and the tail sizing.

The lift of the wing acting through the center of pressure is in front of the center of gravity of the aircraft. This causes a destabilizing motion (increase in lift to increase in angle of attack causes a nose up moment, further increasing angle of attack). This is counteracted by the moment produced by the lift of the horizontal stabilizer, acting behind the center of gravity.

The neutral point of the aircraft was calculated using the given formula,

$$\frac{x_{np}}{c} \cong \frac{1}{4} + \frac{1 + {}^{2}/_{AR}}{1 + {}^{2}/_{AR_h}}\left(1 - \frac{4}{AR + 2}\right)V_h$$

Various values of Static Margin were selected in the range of 0.1 to 0.5 in order to achieve the kind of stability which this aircraft requires. The Neutral Point of the aircraft was calculated at a distance of 45.9% of the Man Aerodynamic Chord form the Centre of Gravity. To achieve the required stability, a Static Margin of 0.13 was chosen. With the given formula, position of Centre of Gravity of the aircraft was calculated and was located at a distance of 33% of Mean Aerodynamic Chord.

$$\text{Static Margin} = \frac{(X_{np} - X_{cg})}{C}$$

5. ANALYSIS

5.1 Wing Analysis

After selecting the different wing parameters, CFD analysis was conducted on the wing using Ansys at a constant velocity of 13 m/s in order to determine the lift and drag generated by the wing. The results showed that a net force of 16.924N was generated. This confirmed that it was possible to lift 1000 gm of payload and achieve payload fraction around 0.6.

It was estimated that the wing produced a net drag force of about 2.3169 N. This drag constituted of all the drags including viscous drag which is not possible to predict by conventional methods.

Forces

Zone	Forces (n) Pressure	Viscous	Total
wing	(2.1424592 16.930762 0.0012259603)	(0.17451748 -0.0066788875 -0.00049785073)	(2.3169767 16.924083 0.0007281096)
Net	(2.1424592 16.930762 0.0012259603)	(0.17451748 -0.0066788875 -0.00049785073)	(2.3169767 16.924083 0.0007281096)

Forces - Direction Vector (1 0 0)

	Forces (n)			Coefficients		
Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
wing	2.1424592	0.17451748	2.3169767	3.4978926	0.2849265	3.7828191
Net	2.1424592	0.17451748	2.3169767	3.4978926	0.2849265	3.7828191

Fig -11: Drag Table

Forces

Zone	Forces (n) Pressure	Viscous	Total
wing	(2.1424592 16.930762 0.0012259603)	(0.17451748 -0.0066788875 -0.00049785073)	(2.3169767 16.924083 0.0007281096)
Net	(2.1424592 16.930762 0.0012259603)	(0.17451748 -0.0066788875 -0.00049785073)	(2.3169767 16.924083 0.0007281096)

Forces - Direction Vector (0 1 0)

	Forces (n)			Coefficients		
Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
wing	16.930762	-0.0066788875	16.924083	27.64206	-0.010904306	27.631155
Net	16.930762	-0.0066788875	16.924083	27.64206	-0.010904306	27.631155

Fig -12: Lift Table

5.2 Wing Spar Analysis

After performing CFD analysis on the wing, structural analysis was performed on the system of spars. A slight deflection was noted, but from the experimental iterative methods and performance of the prototype, the author noticed that the system of spars could sustain the forces without deflecting, meaning the wing design was safe.

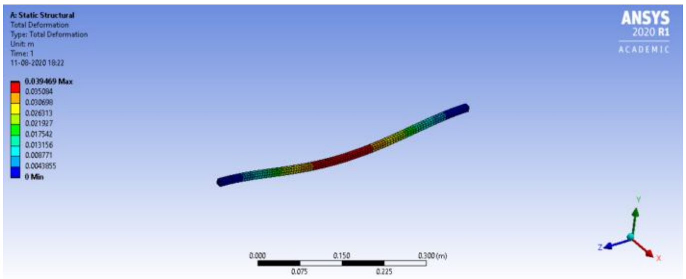

Fig -13: Total Deformation

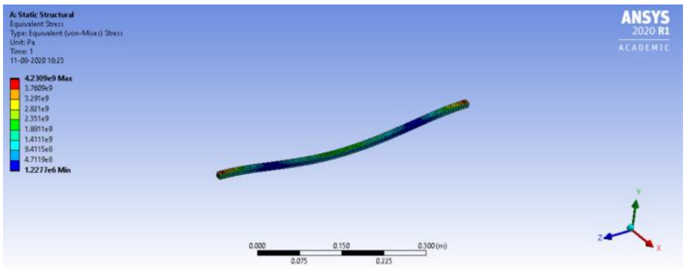

Fig -14: Equivalent (von-Mises) Stress

5.3 Fuselage Analysis

Likewise, CFD analysis was also performed on the fuselage to determine the drag generated which was found around 0.326N.

Forces - Direction Vector (1 0 0)

	Forces (n)			Coefficients		
Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
fuselage	0.28203537	0.044792164	0.32682754	0.46046591	0.073130063	0.53359598
Net	0.28203537	0.044792164	0.32682754	0.46046591	0.073130063	0.53359598

Fig -15: Drag Table

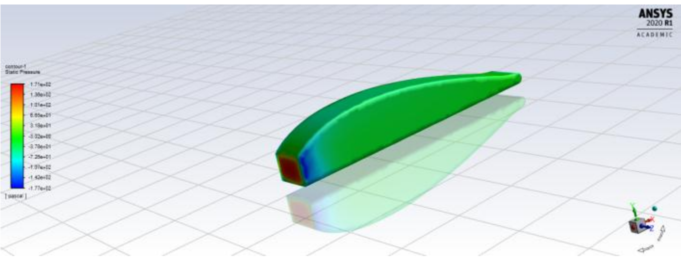

Fig -16: Fuselage – CFD

6. AVIONICS

As the plane is hand launched, it is very important to achieve lift as soon as it leaves hand. Hence, a combination of motor and propeller is needed to achieve the required thrust. Thrust was calculated with the help of the following equation.

Thrust = Mass × Acceleration + Drag Force + Mass × g sin

From the many tests of glider, time required to touch plane on the ground when it leaves is about 2 seconds.

From the kinematical equation: $v = u + at$

we got acceleration about 4.33 m/s^2 assuming initial velocity, u = 0. From Newton's second law, i.e. F = ma, thrust calculated equals to 750gm and drag force = 150gm. Thus, thrust required to take off is about 850gm to 900gm.

6.1 BLDC Motor and Propeller Selection

By calculating the required thrust, selection of BLDC was done. It was done on the basis of power output produced by motor.

$$\text{Power} = \text{Force} \times \text{Velocity}$$

For thrust of 1300gm, power output of 156.283 Watts is needed. Higher the RPM, higher the thrust. Next parameter that needs to be considered was propeller pitch. Lower the pitch higher the thrust, but lower the RPM.

Finally, the parameter that connects RPM and voltage together is kV rating. kV rating is the measure of RPM of motor per unit voltage. So, lower the kV rating means higher the thrust with less speed.

For a 3 cell battery and voltage=11.1V,

$$\text{RPM} = \text{kV}_{\text{rating}} \times \text{Voltage}$$

kV rating of 1100 was selected which can produce RPM of 12,210 at 11.1V. Propeller size was selected on the basis of load test report provided by motor manufacturer.

As dynamic thrust is more than that of static thrust, so propeller of 10 × 3.8 was selected.

$$\text{Power} = \text{Thrust} \times \text{Velocity}$$

Also, $$\text{Power} = \text{Voltage} \times \text{Current}$$

Maximum current drawn from the motor = 15A.

6.2 Servo Sizing

Servo motors are used to control the control surfaces of the plane. The selected servo motor should provide sufficient torque. Servo motor of 9gm was selected for flight which was readily available in market and can also sustain during flight.

Maximum current drawn from the servo = 2A.

6.3 ESC Selection

A 30A ESC with BEC (Battery Elimination Circuit) was selected as integral weight also reduces.

6.4 Battery Selection

The battery was selected on the basis of time of flight and discharge rate required by the motor and other avionic applications. By keeping margin of safety, we assumed the time of flight as 3 minutes.

$$\text{mAh} = \frac{\text{time of flight} \times \text{current} \times 1000}{60}$$

From the above formula, mAh rating of battery was calculated as 850, but the Li-Po batteries are capable to discharge about 80% of its maximum capacity. By taking factor of safety, the Li-Po battery was selected of 1300mAh.

C-rating:

Total discharge rate required = Discharge required by motor + Discharge by servos

$$\text{Safe discharge current} = \frac{\text{mAh} \times \text{C rating}}{1000} \text{ ampere}$$

By considering all the factors of safety, a 3s battery of 1300mAh with 25C rating which safely supplied 32.5A current and required time of flight was selected.

7. FLIGHT TESTING

Manufacturing of the aircraft was carried out and the final weight of the aircraft was found to be 0.9kg which was quite high. The flaws were identified such as faults in manufacturing technique and material selection, and proper optimization was done to omit furthermore defects.

Fig -17: Aircraft Model

8. CONCLUSION

The author successfully eliminated all the flaws and designed a compact, lightweight, reliable, and an airworthy Micro Class Unmanned Aerial Vehicle having excellent flight characteristics along with the ability to carry high payload. The wing profile and geometry, aircraft sizing, propulsion system, control system, and aircraft assembly through analysis and trade studies promptly support the aforementioned sentence, and shows that the aircraft has excellent performance characteristics.

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