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DESIGN AND PRODUCTION OF SMALL TAILLESS UNMANNED AERIAL VEHICLE

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ABSTRACT

The military has shown the most recent interest in small UAVs (SUAVs) for many reasons. We developed a tailless small UAV (flying wing) for military application (SAKR 2) and short range reconnaissance. SUAV is much more portable than its large counterparts and requires only one operator. A smaller reconnaissance plane can assess ground targets at a closer range without being detected. Most SUAVs use electric motors for advantages as it is affected less by the environmental circumstances, lower weight, less vibration and having lower cost. Also an SUAV is less expensive and can be considered a disposable asset. Our design philosophy is based upon improving aerodynamic features, surveillance capability, and controlling the systems autonomously. To fulfill the mission requirements we designed a tailless platform flying wing with 120 cm wingspan, flying at a maximum speed of 30 m/s and about 3 kg take-off weight. Two identical prototypes were fabricated and tested under several flight conditions using remote control to examine flight stability and performance. A commercial autopilot and short range video link was integrated to the SUAV to complete the unmanned system. The designed SUAV succeeded to perform 20 minutes flying in stabilized mode with circle pattern and appeared to have accepted navigational accuracy using FY-3ZT autopilot.

KEYWORDS

SUAV, Vacuum Bagging, Flying wing, Autopilot.

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INTRODUCTION

Unmanned Aerial Vehicles (UAVs) have proven their usefulness in military reconnaissance in recent military conflicts. Their practical applications have been expanding to more than military uses [1]. Various sizes of UAVs are designed to different levels of performance depending on their application. UAVs can be categorized into four different groups: large, medium, small, and micro as shown in Fig. 1. The age of the small UAV started in 1987, when Dr Paul, MacCready's company (AeroVironment) developed the Pointer, the first hand-launched, backpack-carried UAV. Pointer combined the airframe technology of a high-performance model sailplane with electric motor propeller, consumer video camera and radio datalink. The Pointer system was too large to be conveniently carried by soldiers and was normally hauled around in a Hummer vehicle or the like, and so the Army asked AeroVironment if the company could come up with a more portable solution.

In 1999, the US Army bought four AeroVironment Pointer small UAVs [2] for testing in the service's AeroVironment developed a half-sized control system and a cut-down version of the Pointer called the Raven. The Raven has the same configuration and central pod of the Pointer, but a shorter tail and a wing reduced to 52% span of 1.34 m. Encouraged by such successes, AeroVironment is also working on a larger version of the Pointer, named the "Puma", with greater endurance and payload. Several Companies compete to produce this category of Back man SUAV for military application.

Having some notional concepts in mind, base on the imposed specifications, the components, including all the subsystems were chosen. Considering those sketches and the selected components, design and analysis work could determine the vehicle weight and size. Aerodynamic and stability analysis would enhance the initial layout. The flight tests outcome would be used for further modifications to make the design iteration go on. The last but not the least, making a historical data base from successful developed SUAVs can always suggest a very helpful initial estimation of the selected parameters.

Necessary design requirements must be fulfilled such as Portability and Reliability The aircraft and transmission equipment must both be man/backpack portable so that they can be transported and used easily anywhere that surveillance is necessary. In order to reduce the deployment time, the aircraft must be also quickly assembled. Due to the expensive payload carried by the SUAV and the important data stored during the mission by the SUAV, it is desirable to design a reliable SUAV has low observability capable of accomplishing several missions without failures and without being caught by the enemy, thus the feature of small size of the UAV must be pursued throughout the design process.

Also the weight of the SUAV must be consistent with the physical resistance and strength of a soldier to transport it during a battle and to throw it for the launch. Therefore the feature of light weight must be pursued during the design process. Another important constraint is the UAV autonomy where the mission profile requires this SUAV to be handled and driven by soldiers, not professional pilots and most of the time during a battle, therefore it must be either easily remotely controlled from the Ground Central Unit or fly completely autonomous missions using GPS waypoint

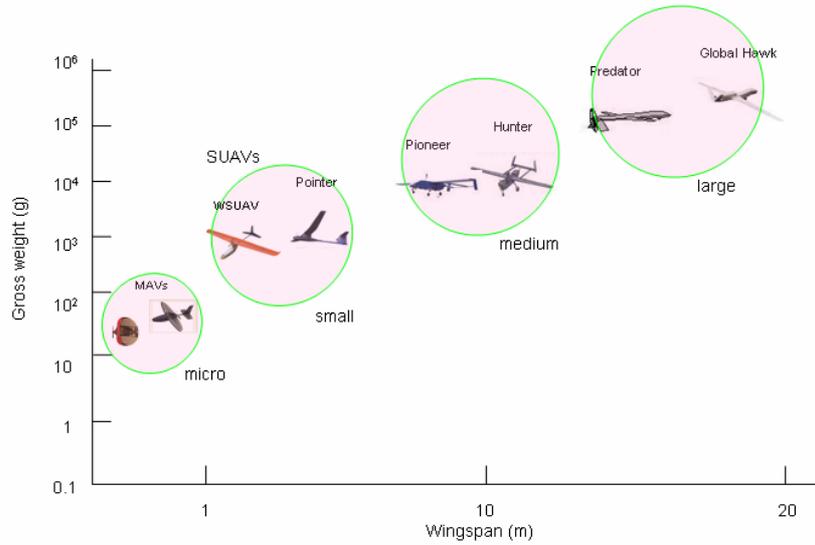


Fig. 1. Classification of UAVs with respect to size and weight.

navigation. The SUAV also will immediately return to its launch point by selecting the “home” command.

THEORETICAL FORMULATION

Fig. 2 displays the typical flow chart that governs the overall SUAV design process. Aircraft performance and mission requirements, among others, needs to be heavily considered in the conceptual and preliminary design process. From this result, the prototype aircraft can be tested. Modification can be implemented to improve overall performance. The optimized design can lead to avionic systems, such as autopilot and video systems.

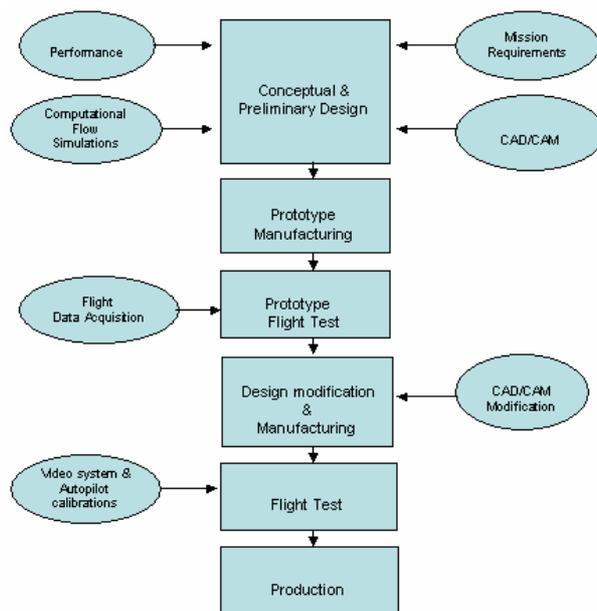


Fig. 2 Design and manufacturing process [3].

Designing Process

To design a small, lightweight, affordable man-portable/backpackable, hand launched SUAV capable of providing reconnaissance, surveillance, remote monitoring imagery and having the ability loiter about a ground target over a hill or behind a building without being detected and without exposing human personnel to harm. RAYMER method [4] is chosen as the main design algorithm. Iterative process of designing includes analytical activities along with empirical investigations.

SAKR 2 Mission Requirements

Table 1. SAKR 2 mission requirement.

Maximum Takeoff Weight	< 3.5 kg
Payload weight	< 1 kg
Assembly	< 90 seconds
Endurance	< 45 min
Payloads	Interchangeable CCD camera and Autopilot
Data transmission range	5 km
Cruise speed and climb rate	cruise speed 20 m/s climb rate 2 m/s
Take-off and landing	Increase hand-lunching success with less danger, Short field landings (optional parachute recovery).
Wing span	Maximum 1.3 m
Propulsion	Electric motor
Airframe	Robust construction, easy repairable, inexpensive manufacturing cost, major parts separately replaced (modular changeability).
Operation	One man operable
Mission altitude	150 m from the sea level

SAKR 2 Preliminary Sizing

The typical mission profile for this class of aircraft is shown in Fig. 3 . Mission profile of our designed SUAV is composed of 6 different steps: After the take off, the airplane climb to the target altitude (ascent); after reaching this altitude there is cruise segment necessary to reach the mission target, then a phase of loitering around the target, finally a cruise to come back to the launching site, a descent and a landing, with or without a parachute.

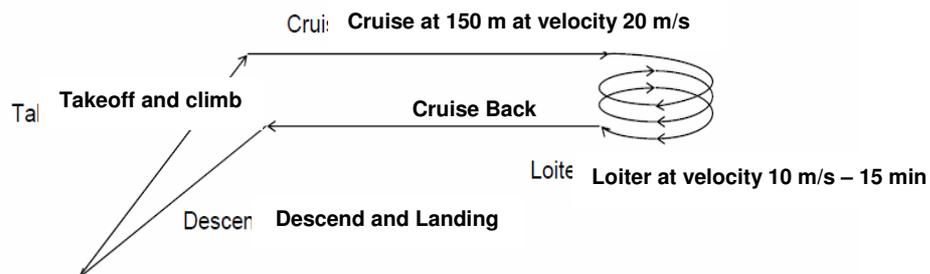


Fig. 3. Mission profile of a SAKR 2.

A variety of possible configurations were analyzed to determine the configuration best suited to the set of previously established requirements. The configurations selected for analysis, as seen in Figure below, consisted of conventional, pod and boom, twin tail, tailless and canard.

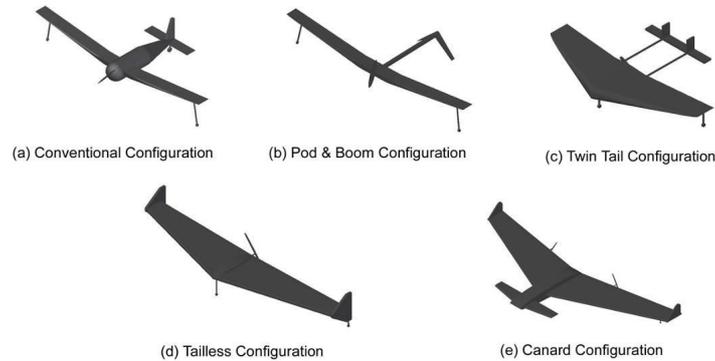


Fig. 4. Possible configuration for the designed SUAV.

A brief synopsis of the strength and weaknesses of each of the configurations was performed to aid the construction of a decisions matrix, from which the most appropriate platform design would be chosen. Each configuration was assessed in terms of the following characteristics: Weight (20%), Aerodynamic Efficiency (20%), Controllability (15%), Stability (14%), Manufacturability (12%), Maneuverability (9%), Innovation (7%), Aesthetics (3%). A decision matrix was performed and concluded to choose a tailless configuration for our configuration design [5].

Table 2. Configuration decision matrix.

Criteria	Max Possible Weighted Score	Conventional	Pod & Boom	Twin Tail	Tailless	Canard
Weight	20	12	14	7	20	16
Aerodynamic Efficiency	20	14	13	8	20	14
Stability	15	15	15	15	6	3
Controllability	15	15	15	12	7	11
Manufacturability	12	10	8	6	12	6
Manoeuvrability	8	3	3	1	6	8
Innovation	7	2	2	4	7	6
Aesthetics	3	2	2	0	3	0
Total	100	73	72	53	81	64

Using successful designs of previous famous hand launched SUAVs, an historical statistical database for SUAVs flying wings is created here and the main characteristics for this class of aircraft are tabulated and reported in Table 3 [6-9].

This table reports per each row the primary conceptual design characteristics for 10 different successful SUAV designs. This table is very important start to have a feeling for the order of magnitude of these characteristics for this class of aircraft. To

Table 3. UAV previous successful designs.

Name		Colorado	Jackaroo	Bommerang	Smartone	DevilRay	T4	T10	Crow	Skydot	Wasp III
Span	m	1.727	1.5	1.4	1.2	2.4	1.8	1.8	0.7	0.8	0.72
Length	m	0.75	0.65	0.55	0.6	0.9	0.7	1.2	0.32	0.4	0.38
Weight	kg	6.8	3.5	2	0.9	6.5	6	6	0.7	0.5	0.43
W _{PL}	kg	2	1.75	0.5	0.3	2.5	2.3	2	0.2	0.15	0.12
Power	W	530	430	320	260	600	530	600	200	230	180
V _{st}	m/s	3.716	---	---	9	---	---	---	---	---	---
Endurance	hr	1	1.5 --- 2.5	1.5	1	1.5--2.5	1	0.75	0.75	0.75	0.75

better define a feasible range of values and to provide some initial numbers to be used in the sizing of the proposed SUAV, a plot, called a constraint diagram to get an optimal design point, showing the relationship between thrust-to-weight ratio (T/W) and wing loading (W/S) for the required mission was developed using the equations outlined below. In particular the term thrust-to-weight is associated with jet-engined aircraft. For propeller-powered aircraft the equivalent term is expressed in terms of horsepower, and is the horsepower-to-weight ratio [7].

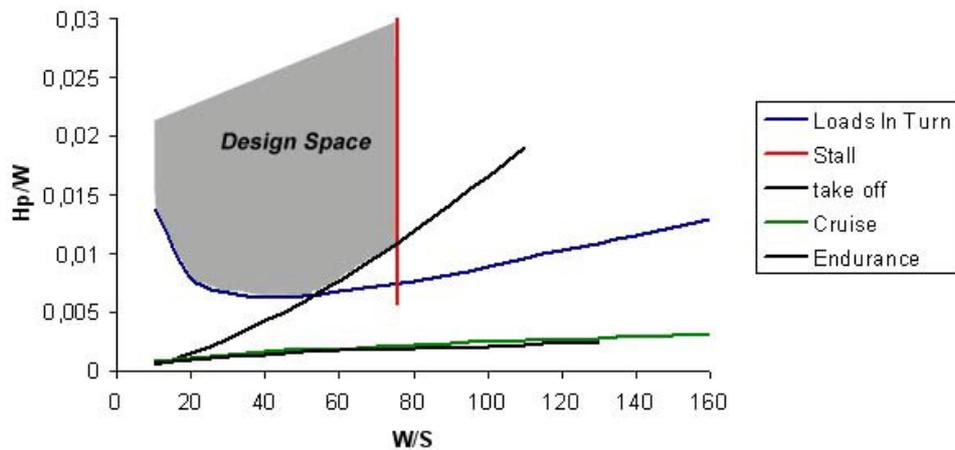


Fig. 5. Constraint diagram.

Fig. 5 shows the constraint diagram obtained plotting the design equations. The area enclosed within each constraint segment is very useful because each point of the design space in grey, represents a possible aircraft design for a certain class of aircraft and for the mission profile in consideration[8]. The main constraint equations for each phase of flight of the mission profile are given as:

- Maximum Load/Turn

$$\frac{HP}{W} = \frac{1}{550\eta_p} \left[\frac{1}{2} \rho V^3 C_{D_0} \left(\frac{S}{W} \right) + 2K \frac{n^2}{\rho V} \left(\frac{W}{S} \right) \right] \quad (1)$$

- Endurance

$$\frac{HP}{W} = \frac{4}{550\eta_p} \left[C_{D_0}^{\frac{1}{2}} \left(\frac{K}{3} \right)^{\frac{1}{2}} \left(\frac{2W}{\rho S} \right)^{\frac{1}{2}} \right] \quad (2)$$

- Cruise

$$\frac{HP}{W} = \frac{2}{550\eta_p} \left[C_{D_0}^{\frac{1}{2}} K^{\frac{1}{2}} \left(\frac{2W}{\rho S} \right)^{\frac{1}{2}} \right] \quad (3)$$

- Takeoff

$$\frac{HP}{W} = \frac{2.44}{550\eta_p} \frac{1}{g d_{to}} \left[\frac{1}{\rho_{SL} C_{Lmax}} \left(\frac{W}{S} \right)^{\frac{1}{2}} \right] \quad (4)$$

- Stall

$$\left(\frac{W}{S} \right)_{stall} = \frac{1}{2} \rho V_{stall}^2 C_{Lmax} \quad (5)$$

Using the greater knowledge about this class of aircraft derived from analyzing comparative aircraft and the historical assessment shown in Table 4, it is possible to make some first crude assumptions about the parameters the constraint equations require, the loiter velocity for Small reconnaissance and surveillance UAV is usually constrained by the camera's parameters and target altitude of 150-200 m the maximum speed to avoid blurry images is around 20 m/s. The mission profile for the SUAV considered usually requires a maximum load factor n related to the maximum level turn bank angle (45 deg) used during the loiter, during this kind of maneuver the load factor is around 2.5. The parasite drag that consists mostly of a skin friction component and a component due to pressure depends on the particular layout of the aircraft and its protuberances but at this early point of the design it is possible to assume that it should range between 0.025 and 0.04. A value of 0.035 was considered. For the lack of a more accurate Oswald efficiency factor for this class of aircraft, a typical value for standard aircraft was chosen, equal to 0.8 [9].

Finally, since the proposed SUAV is hand launched, an equivalent takeoff distance was considered, it represents in this case the equivalent distance that an aircraft should cover in order to reach the required velocity to takeoff if the required velocity to takeoff is the stall velocity and the acceleration is the acceleration that a soldier can give to the aircraft running with a weight of about 3.5 kg in his hand. This distance was estimated to be approximately 14 meters. A sensitivity analysis is later performed in order to take into consideration the impact of any errors due to the assumptions made.

Table 4. Initial assumed designed values predicted from historical SUAV design.

Parameter	$\rho @ 150$ (kg/m ³)	η_p	V_{cr} (m/s)	n	C_{D_0}	AR	e	C_{lmax}	d_{to} (m)	V_{st} (m/s)
Value	1.2	0.7	20	2.5	0.035	3	0.8	1.2	14	8

This table reports per each row the primary conceptual design parameters based on historical successful designs.

Initial sizing of payload weight fraction

In order to have a first estimate of the Weight at Takeoff of the airplane, it is useful to consider the historical values of the Payload Weight Fraction reported in Table 5 and highlighted in Table 3.

Table 5. Payload weight fraction for previous successful flying wings.

	Colorado	Jackaroo	Bommerang	Smartone	DevilRay	T4	T10	Crow	Skydot	Wasp
Weight TO (kg)	6.8	3.5	2	0.9	6.5	6	6	0.7	0.5	0.43
Weight PL (kg)	2	1.75	0.5	0.3	2.5	2.3	2	0.2	0.15	0.12
W_{PL}/W_{TO}	0.294	0.5	0.25	0.33	0.38	0.38	0.33	0.285	0.3	0.27

As the payload is a driving factor behind the design and sizing of the aircraft. After a first research, the initial Payload Weight was estimated to be, for a medium-high price camera and autopilot including cables and extra power source:

$$W_{PL} = 0.9 \text{ kg} \tag{6}$$

Therefore, using the Historical Values, it is possible to estimate an initial value of the Weight at Takeoff of the proposed SUAV:

$$W_{TO} = \frac{W_{PL}}{0.3319} = \frac{0.9}{0.3319} \approx 3 \text{ kg} \tag{7}$$

Design point selection

By using the Horsepower to weight ratio vs. wing loading diagram it is possible to determine the wing surface area and the power of the UAV using the conceptual Takeoff weight. The basic idea for selecting the design point that the aircraft with a higher hp/W will accelerate more quickly, climb more rapidly, reach a higher maximum speed and sustain higher turn rates. On the other side, a bigger engine will consume more fuel or battery, which will drive up the aircraft’s takeoff weight to perform the required mission.

The wing loading (W/S) affects stall speed, climb rate, takeoff and landing distances and turn performances. The wing loading also determines the design lift coefficient and affects the drag because of the wetted area and wing span. A low Wing Loading generally means a larger wing; this may improve the performance, but the additional drag due to the larger wetted area and the additional empty weight due to the larger structure of the wing will increase takeoff weight to perform the required mission. A compromise between hp/W and W/S must be found that better matches and optimizes the required mission of the aircraft.

It is noted that the cruise and endurance constraints for this kind of mission are less important than the others. Considering the requirements of the proposed mission, a lightweight, long endurance and short takeoff distance airplane is highly desirable. Looking at the design space in Fig.6 as an initial estimate, the following combination of values seemed to be a good compromise that respects the constraints:

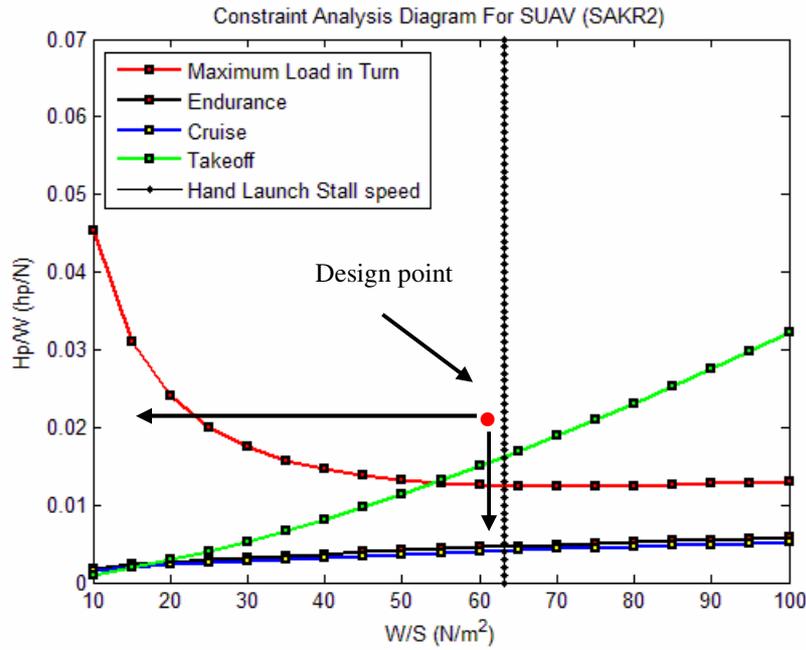


Fig. 6. SAKR 2 Constraint diagram.

$$\frac{W}{S} = 62 \text{ N/m}^2 \tag{8}$$

$$\frac{hp}{W} = 0.018 \text{ hp/N} \tag{9}$$

From the estimated weight obtained from payload fraction, now we can calculate SAKR 2 wing area as follows:

$$S = \frac{3(kg)}{62} = 0.48 \text{ m}^2 \tag{10}$$

Wing and control Surface sizing

A moderate wing aspect ratio of 3 was selected from historical trend, as a compromise between low induced drag and vulnerability of the wing to damage. Hence the wing span can be estimated as follows:

$$b = \sqrt{AR \times S} = \sqrt{3 \times 0.48} = 1.2 \text{ m} \tag{11}$$

And the mean geometric chord is:

$$\bar{c} = \frac{0.48}{1.2} \approx 0.4 \text{ m} \tag{12}$$

A taper ratio of 0.5 was chosen ($\lambda =$) from successive designs to improve the lift distribution and the maximum lift distribution of the wing. Also the geometric characteristics of the designed wing can be calculated after choosing root and tip chords dimensions based on MAC equals 0.4 m as follows:

The sweep angle of half root chord

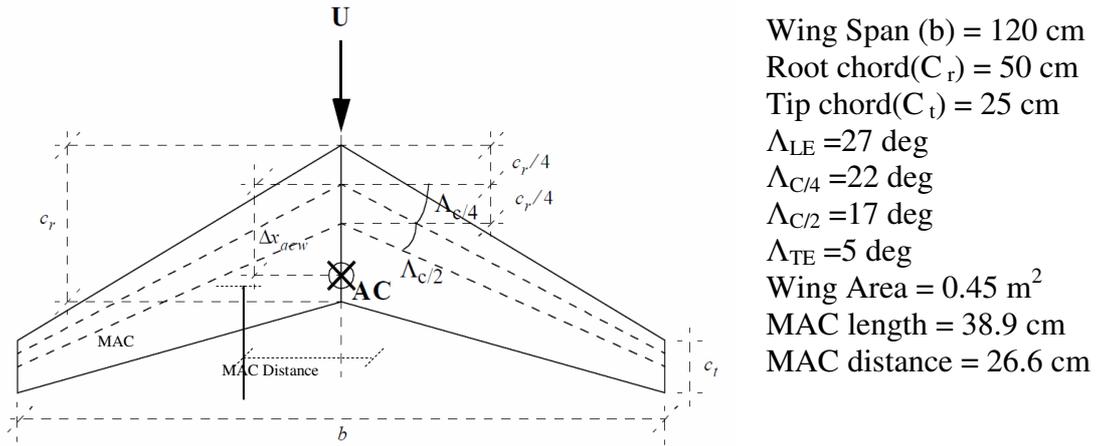


Fig. 7. SAKR2 Wing geometric characteristics.

The sweep angle of half root chord

$$\Lambda_{c/2} = \text{atan} \left[\tan(\Lambda_{c/4}) - \frac{(1-\lambda)}{A_w(1+\lambda)} \right] \quad (13)$$

$$\Lambda_{TE} = \text{atan} \left[\tan(\Lambda_{c/4}) - \frac{3(1-\lambda)}{A_w(1+\lambda)} \right] \quad (14)$$

Sizing the winglets is difficult especially when we decided to make the winglets act as vertical fin. The tail volume coefficient method [10], calculated by hand, uses heuristic data to determine the minimum size of the vertical fins and winglets necessary to provide yaw stability. The coefficients for the proposed Small UAV were found by researching comparatives SUAVs, through the use of available pictures, in order to check the order of magnitude was correct. The aspect ratio was chosen to minimize the dimensions of the aircraft.

The sizing is dependent on the wing area, the wingspan, the area of the vertical tail, the distance between the c.g. of the aircraft and the aerodynamic center of the vertical tail. The tail volume coefficient for most previous SUAV falls in the range of 0.03 to 0.06. The coefficient used in SAKR 2 is 0.04, a value selected to fit within the range and minimize the size and weight of the winglets. The following equations were used to determine the areas of the vertical fin including endplates:

$$S_{vT} = \frac{c_{vT} b S_{ref.Wing}}{L_{vT}} = \frac{0.04 \times 1.2 \times 0.45}{0.22} = 0.104 \text{ m}^2 \quad (15)$$

This coefficient selection corresponds to a vertical tail area of 0.104 m². We chose to have 2 vertical fins rather than one, which when combined with an approximated taper ratio of 0.5 with Curved leading and trailing edge, make the winglets of 0.05 m² and 18 cm tall. The dimensions range of control surfaces were defined by statistics analysis. Elevons width turned out to be in the range of 1/10 to 1/4 of the main root

chord of the wing. Further evaluations showed that 1/8 ratio was more suitable for our designed model.

Aerodynamic Analysis

Operating lift and drag coefficients

It is possible at this point to calculate an operating lift and drag coefficients. The aircraft should be designed so that it flies the design mission at or near the operating lift coefficient. In level flight the lift must equal the weight, so the required lift coefficient can be found as follows based on calculated wing loading and cruise speed:

$$\frac{1}{2}\rho V^2 S C_L = W \rightarrow C_L = \frac{W}{\frac{1}{2}\rho V^2 S} = \frac{3}{\frac{1}{2} \times 1.225 \times 20^2 \times 0.45} = 0.25$$

$$C_D = C_{D_o} + C_{D_i}$$

The induced Drag Coefficient, C_{D_i}

$$C_{D_i} = \frac{C_L^2}{\pi e AR}$$

The parasite drag coefficient was assumed to be $C_{D_o} = 0.035$, Oswald's factor $e = 0.8$, $AR = 3$ and under the assumption of considering angles of attack lower than the stall angle, it is possible to draw an initial draft of the Lift to Drag versus the angle of attack curve.

$$C_D = 0.035 + \frac{0.25^2}{\pi \times 0.8 \times 3} = 0.0432$$

Airfoil selection

Finding the ideal airfoil involves researching an airfoil database (such as the UIUC database [11]) and performing an analysis in XFOIL [12]. XFOIL uses the Full-Inverse method, in which geometry is introduced and a pressure distribution is calculated over the airfoil surface from given Reynolds and Mach numbers. From the pressure distribution, the lift, drag, and pitching moment coefficients are computed at varying angles of attack. Inputs to XFOIL are the coordinates to create the airfoil shapes along with a specific Reynolds number and Mach number. XFOIL in turn outputs the lift, drag, and pitching moment coefficients as the angle of attack varies. The data are then plotted in order to compare the five best airfoils shown in Fig. 9 displayed by using Profili2 software [13]. We can calculate the Reynolds-number (Re) at altitude 100 m, speed of 20 m/s and chord length equals 38.5 cm.

$$Re = \frac{\rho V \bar{c}}{\mu} = \frac{1.167 \times 20 \times 0.385}{1.749 \times 10^{-3}} = 523012 \quad (16)$$

The selection is performed based on effective aerodynamic parameters of two reflex airfoils (E186, S5010) and three cambered airfoils (E174, E197 and SD7080) from Fig.8 and Fig.9 such as lift coefficient, pitching moment coefficient and drag polar curve.

At a trimmed flight condition, where all forces and moments are in equilibrium. For the reflexed camber line makes the moment coefficient positive, which means, that the moment around the c/4 point is working in the tail heavy direction. Therefore, the center of gravity has to be located in front of the c/4 point to balance the moment M^* by the lift force L^* . The larger the moment (-coefficient) of the airfoil, the larger the distance between c/4 and the c.g. for equilibrium. To have a stable swept wing we chose S5010 airfoil with zero pitching moment coefficient at the calculated Reynolds no. Choosing airfoil 'reflex' instead of wing washout to attain longitudinal stability does not reduce effective span and easier to jig wing on constant waterline.

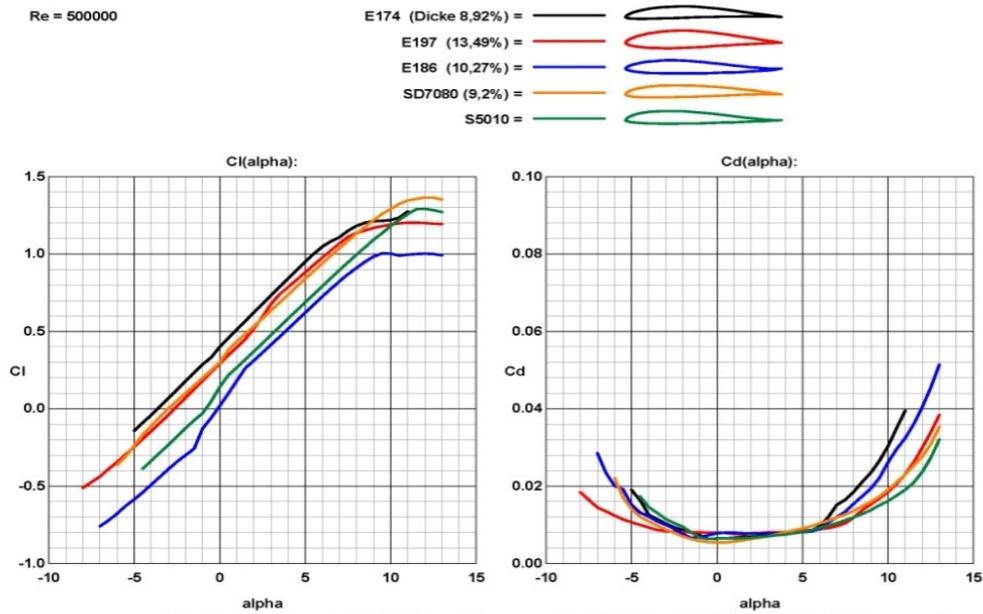


Fig. 8. Lift Curve slope and drag polar for the selected airfoils.

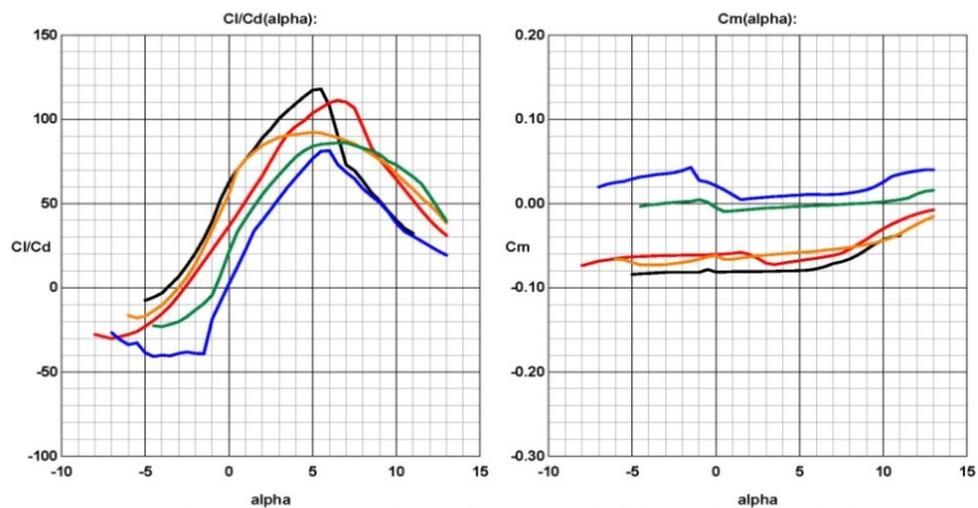


Fig. 9. CL Vs CD data for the selected airfoils.

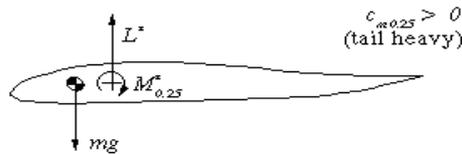


Fig. 10. Reflex Airfoil equilibrium behavior.

The airfoil now chosen, two key aerodynamic parameters for the 3D wing were zero lift angle of attack, C_{l_0} and change in lift with change in alpha, $dC_l/d\alpha$. C_{l_0} is mostly dependant on airfoil camber. $dC_l/d\alpha$ is essentially linear up to stall and is strongly affected by quarter chord sweep and aspect ratio. Both of these parameters directly determine the lifting performance of the aircraft. The planform points were entered in XFLR5, panel mesh was created, and vortex lattice results calculated at the T.O. speed of 9 m/s.

The maximum lift coefficients for a wing configuration without the use of high-lift devices like flaps and slats at a Reynolds number around 500,000 is fixed by the current technology and a typical value is 1.2. The stall velocity historically can be estimated around 8 m/s, CL max of the planform can be calculated by using this equation:

$$C_{L_{max}} = 0.9c_{l_{max}} (\cos \Lambda_{0.25c}) = 0.9788$$

The Lift curve slope of the wing can be calculated using the following empirical equation [14]:

$$C_{L_{\alpha w}} = \frac{2\pi A_w}{2 + \sqrt{\frac{A_w^2}{k^2} (1 + \tan(\Lambda_{c/2})^2) + 4}}$$

where $k = \frac{C_{l_{\alpha}}}{2\pi}$ is the ratio of wing profile section lift curve slope to $2\pi/\text{rad}$

$$C_{L_{\alpha w}} = \frac{2\pi \times 3}{2 + \sqrt{\frac{9}{1^2} (1 + \tan(17)^2) + 4}} = 0.05739/\text{°}$$

Neutral Point and Stability

XFLR5 was used to calculate the AC location on the planform, in order to determine where the CG should be placed for positive SM. The AC was calculated at approximately 24.2% of the MAC, which is in agreement with general theory that holds that the AC is roughly 25% of MAC. The center of gravity must be located in front of the neutral point. After calculating the MAC and the spanwise location of the mean chord b_{MAC} . The n.p. of our swept wing can be found by drawing a line, parallel

to the fuselage center line, at the spanwise station y. The chord at this station should be equal to MAC. The n.p. is approximately located at the c/4 point of this chord line. Instead of using the graphical approach, the location of the neutral point can also be calculated by using one of the following formulas, depending on the taper ratio [15]:

$$X_N = \frac{c_x}{4} + \frac{2b}{3\pi} \tan(\Delta_{C/4}), \text{ if taper ratio} > 0.375$$

$$X_N = \frac{0.5}{4} + \frac{2 \times 1.2}{3\pi} (\tan(22)) = 0.243 \text{ m}$$

Therefore, to remain statically stable, the CG must be forward of X = 24.3 cm, where X=0 is at the LE vertex of the planform. In our case, the SUAV must maintain a static margin of no less than 5%, requiring the CG to be forward of X = 23 cm. This was determined to be a good compromise between flying qualities and performance since an increased static margin improves stability, but reduces performance due to greater trailing edge deflections required for trim. To achieve static longitudinal stability, we have to locate CG position properly, the following equation was used:

$$SM = \frac{(AC - CG)}{MAC} * 100 \tag{17}$$

SM is generally in the range of 5% - 20%, to track down the best place. Component internal arrangement for four different static margins of 5, 8, 10, 15 and 20 percent was done and stability status in pitch axis was studied carefully [16].

SM (%)	Calculated parameters (mm)			
	0.05	0.1	0.15	0.2
MAC (mm)	388.8	388.8	388.8	388.8
AC (1/4 chord of MAC) (mm)	97.2	97.2	97.2	97.2
CG distance (mm)	155.3	174.7	194.2	213.6

For testing new prototypes it is recommended to adjust CG to be at 15% of MAC, the final configuration was frozen with SM of 20%.

Propulsion system

We performed a comprehensive study on available motors and propellers in the market by focusing on their power consumption, weight, voltage, current drain and their cost.

Motor, ESC, battery and propeller Selection

According to the preliminary sizing and selection of the design point the power to weight ratio was calculated

$$\frac{hp}{S} = 0.018 \text{ hp/N}$$

So the required motor power for the designed SUAV can be calculated as follows:

$$\text{Motor power} = (3 * 9.8)N * 0.018 \frac{hp}{N} = 0.529 \text{ hp} = 396.75 \text{ W}$$

Required motor power at cruise

Deciding to use electric propulsion system, we can calculate the required motor power during cruise for SAKR 2 as the following relation ship exists [7]:

$$Power = IV = TV = DV = \frac{1}{2}\rho V^2 S(C_{D_o} + C_{D_i})V \tag{18}$$

Substituting, where in cruise $C_L = \frac{2W}{\rho V^2 S}$

$$Power = \frac{1}{2}\rho V^3 S C_{D_o} + \frac{1}{2} \frac{\rho V^3 S}{\pi e AR} \frac{4W^2}{\rho^2 S^2 V^4} = \frac{1}{2}\rho V^3 S C_{D_o} + \frac{2W^2}{\pi e AR \rho V S}$$

This shows the relationship between the power and the weight. Using the same parameters used in the constraint equations, the power required in cruise is P= 77.3 W, and considering an efficiency of the motor and of the propeller of 0.8, so the power consumed in cruise is P=96.6 W. Using a suitable battery with an operating voltage of 16.4 V and a brushless electric motor of 500 Watt max. power, the endurance associated with this power is:

$$I = \frac{P}{V} = \frac{96.6}{16.4} = 5.89, \quad A = \frac{C(Ah)}{h} \rightarrow h = \frac{C(Ah)}{I} = 23.5 \text{ min}$$

Flight duration is mainly dependent upon a battery’s performance. Representative of the new higher energy density batteries are: disposable LiSO2 (Lithium Sulphur Oxide), rechargeable LiON (Lithium Ion), and rechargeable Lithium-Polymer [17]. These batteries display good discharge rates at high and low-temperatures. This new lithium technology has proven to be successful in military SUAVs with respect to duration, reliability, and cost Lithium-Polymer is the battery of choice for the SAKR 2 because of its high energy density and cost effective, reusable characteristics. We decide to use Li-Poly battery type.

Table 6. Available Accessories selection from market.

				
Model	37/48 Deluxe	Thunder power 4600	Electricfly SS-45	
Kv (rpm/V)	770	---	5V/2A BEC	
power	450 W	4600mAh (35C)	3-4 LiPol	APC 12x8 Electric
Voltage	8.5-18 V	14.8 V	---	pusher folded and
Current	30 A (Peak 38 A)	---	---	unfolded propeller
Diameter	37.2 mm	160mm x 24.5 mm x 40 mm	60 mm x 35 mm	
Weight	172 g	460 g	65 g	

Flight duration experiments

A test stand shown in Fig 14 is designed to measure the max static thrust for the chosen for two motors (37/48 Deluxe and 0.25 Rimfire) and also measures the

battery time duration at half throttle position or cruising phase while fixing thrust static load. Using a wattsup meter to analyze motor power data [18], Volt watch to monitor each cell capacity while performing the test and Tachometer to measure the propeller rpm.

Thrust measurement set-up uses a load cell of capacity suitable for SUAV class propellers (here 7.5 kg load cell was used). DC electric motor clamped between two wooden ballets mounted on one end of load Cell, other end of load cell is clamped to rigid support, this load cell works as cantilever beam as shown in Fig 11. Thrust produced by propeller acts normal to the vertical axis of load cell and equal amount of force is registered at one end of load cell as shear force. RPM is measured using optical sensor as explained in previous section.

A 12X8 APC propeller [12 inch is diameter and 6 inch is pitch] was used in order to measure thrust characteristics and validation for experimental set-up. The throttle on the radio-controlled transmitted was gradually increased to 100% by 20% graduations. The throttle setting was slowly increased to the first required point and the aircraft was allowed some time to settle before the reading on the load cell was taken for that setting. The throttle was also decreased to 0 before the next throttle setting.

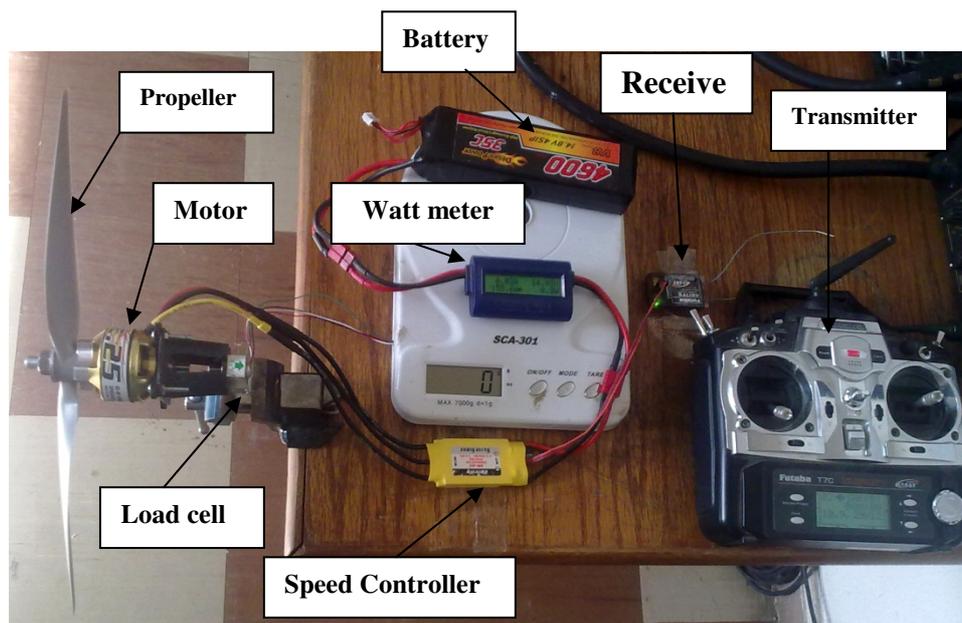


Fig. 11. Test rig arrangement for tested motor.

Measurement of the strain gauge was accurate up to 100 N, hence the upper and lower limits of the calibration curve is also shown to reflect the possible deviations from the curve. From the graph shown it is clear that at 60 % throttle position gives 1 kg of thrust which considered being the SUAV situation for cruise phase, sustained the parameters accompanied with this phase like altitude and speed. At the maximum throttle location gives thrust of 1976 gm (full maximum power).

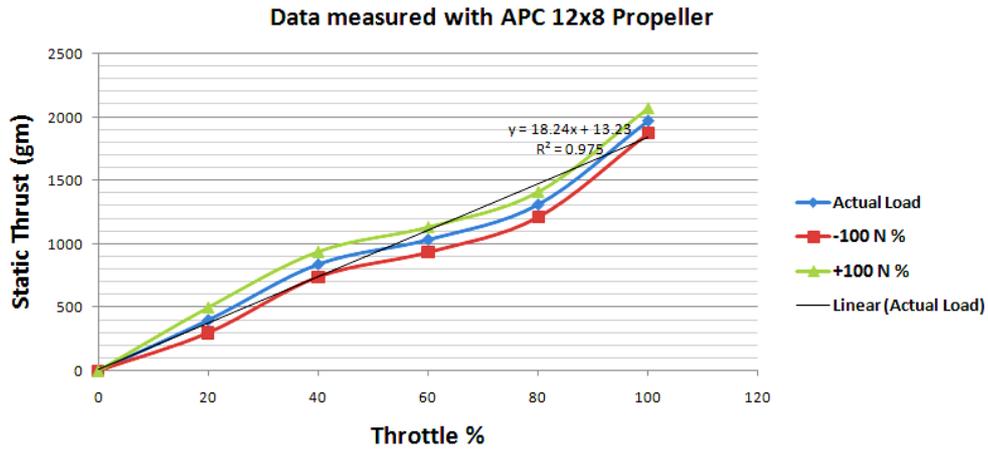


Fig. 12. Static thrust calibration curve.

We can measure the battery endurance related to propulsion system by fixing static load condition in cruise phase. The maximum flight endurance is estimated by battery capacity (W_h), P_{bat} , and total power consumption of the propulsion system at constant load value [19].

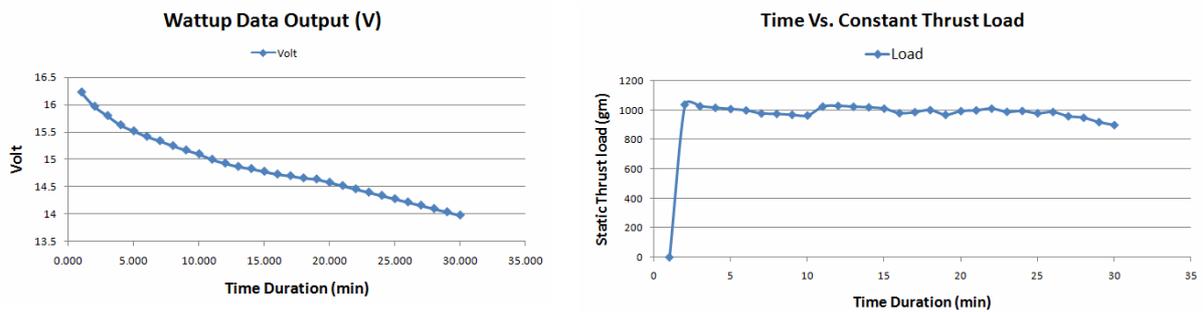


Fig. 13. Volt and Static thrust consumed Vs. Time.

From the figure above, it is clear that the fixed static load sustained equals to 1000 gm for 30 min with a decrease in battery voltage.

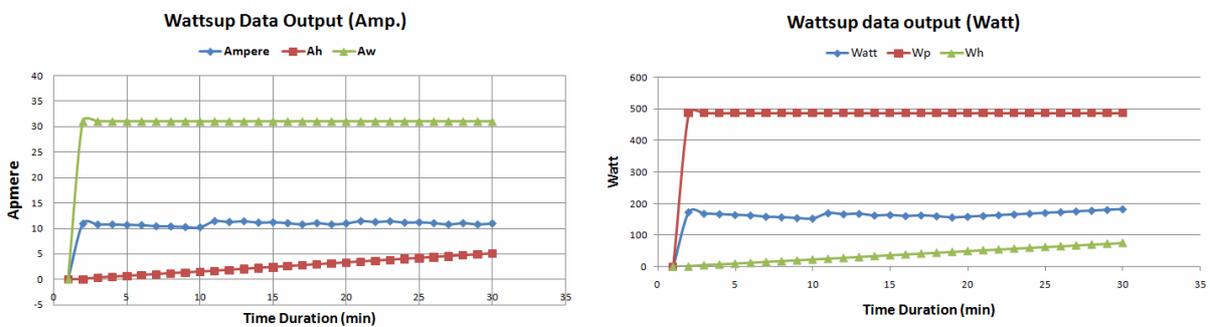


Fig. 14. Ampere and Watt consumed Vs. Time.

The calculated value of endurance at cruise, 23.5 minutes, partially meets given requirements of 45 minutes. To Confirm the power system endurance equation, from Fig. 13 it is clear the maximum Ah at constant throttle position equals 5 Ah with the capacity of battery equals 4600 mAh, leads to a duration of 27.6 min.

Weight breakdown Analysis

Traditional method was used to estimate the take-off weight, considering an empirical equation derived from gathered SUAV information with the same configuration i.e. flying wing.

Weight & balance analysis was done due to the selected configurations and sized dimensions. This resulted in determining approximate CG position. The components internal placement was done and is finalized according to the proper location of the relevant C.G.

Table 7. Weight breakdown analysis.

		Part name	Weight
SAKR 2 SUAV	Airframe	Fuselage	260
		Wing	700
		winglets	100
		Bolts and plastics	60
	Payload	Autopilot	140
		Camera	80
		Transmitter	145
		Harnesses and cables	35
		Autopilot battery	260
	battery	Thunder power (4600 mAh)	460
	Servos	Slim heavy duty	420
	Motor	Propeller	19.3
		Deluxe motor	172
Electric fly ESC		65	
Total weight (W)			2916

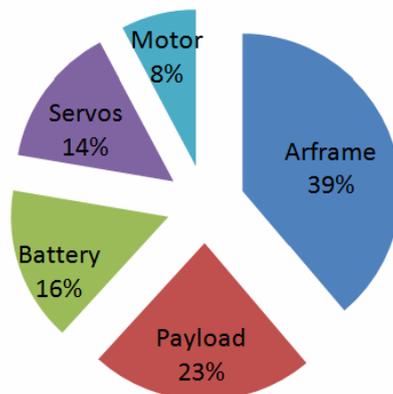


Fig. 15. SAKR 2 SUAV Weight breakdown diagram.

PROTOTYPE INTEGRATION AND FABRICATION

Construction

Important features in selecting materials and fabrication techniques like low weight, high strength, low cost, availability of the materials etc. Considering these parameters, we decided to build our SUAV airframe out of foam and the control surfaces with balsa wood light ply and the fuselage was manufactured by composite

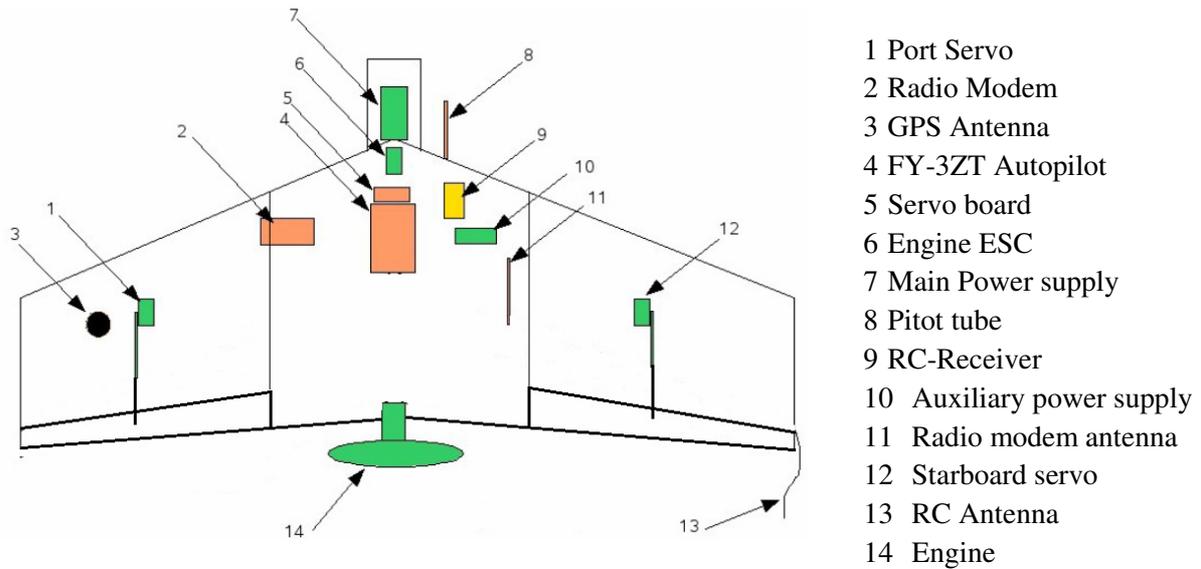


Fig. 16. SAKR 2 Component internal placement.

material (unidirectional carbon fiber fabric to reinforce the fuselage bottom to withstand hard skidding landing) with the use of vacuum bagging process. Our fabrication process can be summarized in the following steps:

- a) Complete CAD drawing is done by using INVENTOR software.

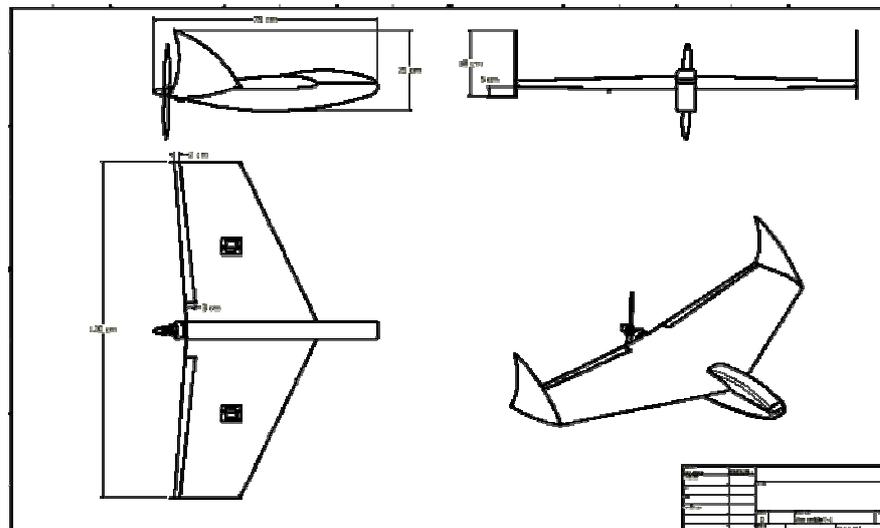


Fig. 17. INVENTOR 3D view Drawing for SAKR 2.

- b) The wing's tip and root airfoils and fuselage bulkheads were cut out by the use of laser cut.



- c) Foam blocks were prepared according to Layout drawings and airfoil sections were attached to the sides of the blocks and the wings were cut with a hot wire.



- d) Foam coated with thin layer of balsa wood was assembled according to INVENTOR drawings and the wings were glued to be one part and the wing was coated by monocoat material then the servos and components were installed properly.



Vacuum Bagging Setup

There are several items used in the vacuum bagging process. They are mainly used for preparing a well sealed vacuum environment, collecting excess resin from the composite material, having a desired surface quality, and taking the composite product out of the mold easily. In the market, vast amount of these items having different characteristics are available.

Here, some of the main items that can be used in a basic application of vacuum bagging are listed and described below [20]:

- Release agent*: is a kind of chemical that is applied to the tool surface for preventing the sticking of the laminate to the mold surface.
- Peel ply*: is a tightly woven fabric, often nylon, and impregnated with some type of release agent. It is used to give the laminate a rough surface, rather than a smooth finish.
- Release film*: is a thin plastic which has been treated so it will not bond to the laminate. There are small holes on it allowing the matrix material to pass through.

- d) *Breather*: is a thick layer of cloth used for collecting the excess resin from the laminate that passed through the peel ply and the release film. It also provides the uniform vacuum distribution.
- e) *Vacuum bagging film*: is a relatively thick plastic layer which is used for isolating the laminate. It provides the necessary vacuum environment.
- f) *Sealant tape*: is a putty-like sticky material. It is used for sticking the vacuum bagging film to the tool surface, and it prevents air leakage.
- g) *Vacuum pump*: is used for supplying necessary vacuum pressure, we use a 1/3 hp vacuum pump with the necessary accessories like the sniffer.

The bagging procedures are done with a new product from DIATEX (Vacuo Peel) to save time and get more smooth products.



Fig. 18. Vacuum bagging process used in manufacturing.

The fuselage skin was manufactured as a sandwich structure. The female molds of left and right skin are used as the mold surface. A 1-mm-thick styrofoam layer is placed in between two layers of e-glass fabric (2 oz). The aim of using a styrofoam layer is to increase the stiffness of the fuselage skin. Production steps for SAKR 2 fuselage is as follows:

- a) Prepare molds using CNC Machine and laying up fabric with preparing bagging process.



- b) Laying up a strip of carbon fabric (unidirectional) with approximately 7 cm width is laid up on the internal lower surface of the fuselage skin after the two parts assembled together.



c) Fixing bulkheads, fuselage longerons and motor firewall.



Fig. 19. Fuselage manufacturing process using Vacuum bagging.

The layers are placed in plastic vacuum bag and the air is pulled out, by a vacuum pump, to a suction pressure of 30 mmhg. The layers are cured for twelve hours at room temperature. Table 8 represents the material used in production for 2 prototypes of the designed SAKR 2 SUAV.

Table 8. Properties of composite material and adhesive used in fabrication.

Material type	Density	Amount used
E-glass fabric	56 g/m ²	----
E-glass fabric	112 g/m ²	----
Styrofoam	31 kg/m ³	1 layer with 1 mm thickness
Carbon fabric (unidirectional) 3K	84 g/m ²	1 layer strip with 7 cm width
Epolam 2015 epoxy	1.15	90 gm
Epolam 2015 Hardener	0.95	30 gm
Peelply (Breather+perforated film+Bleeder)	---	2 m x 1.5 m width
Bagging film	Temp 260 c ^o	1.2 m x 2 m

At the end of the process, the edges of the composite skin are trimmed to the actual wing geometry. In order to have an access to the equipment, further structure reinforcement is pending for more flight tests' results.

SAKR 2 FLIGHT TESTS

Autopilot

The autopilot chosen was the FY-3ZT. The chosen autopilot is the smallest and lightest fully featured micro autopilot on the market – ideal for surveillance and reconnaissance applications. The system uses an external GPS unit for inertial navigation and wireless modems communications between the ground station and

autopilot. The FY-3ZT Autopilot can guide mini- and micro-UAVs autonomously and/or receive dynamic user commands through the ground station and RC radio while providing live video feeds to the user. It uses three-axis rate gyros and accelerometers for attitude estimation, as well as differential and absolute air pressure sensors for airspeed and altitude measurement. The FY-3ZT autopilot includes features such as Waypoint setting while in flight, automated altitude control, Ground speed control, activation of circling at given points, Control of circling radius, automated Return to Base (RTB) and automated take-off. FY-3ZT works with Data Radio frequency 433 MHz, Half Duplex, as the Transmission Range of the FY602 (433 MHz) has a far longer tested range of 5 kilometers [21].

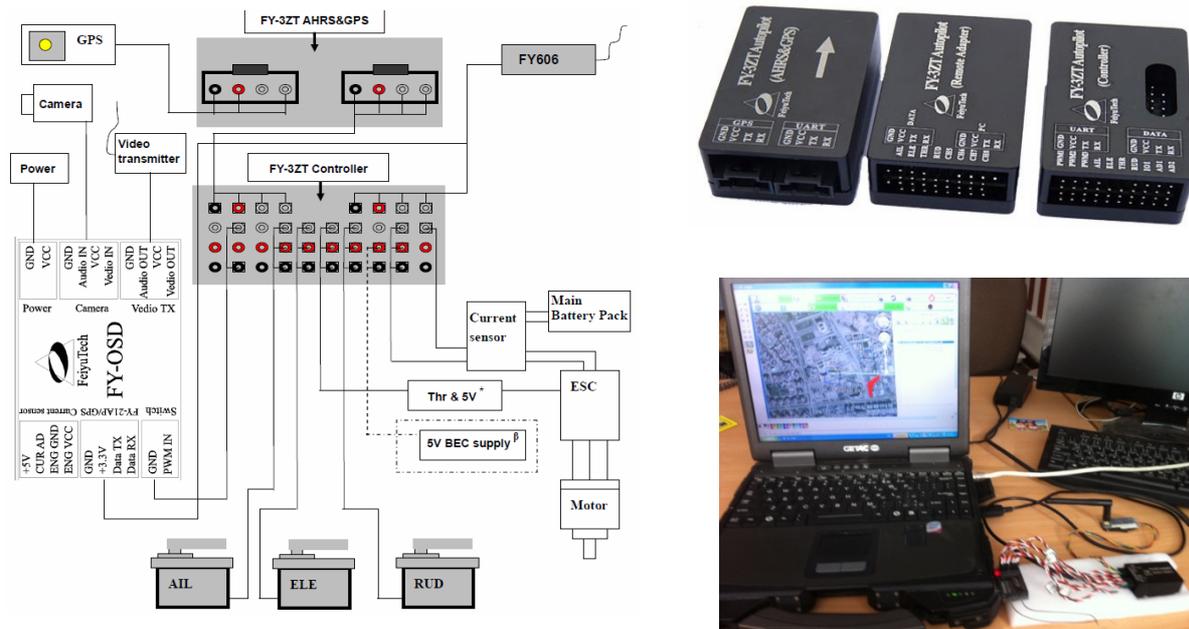
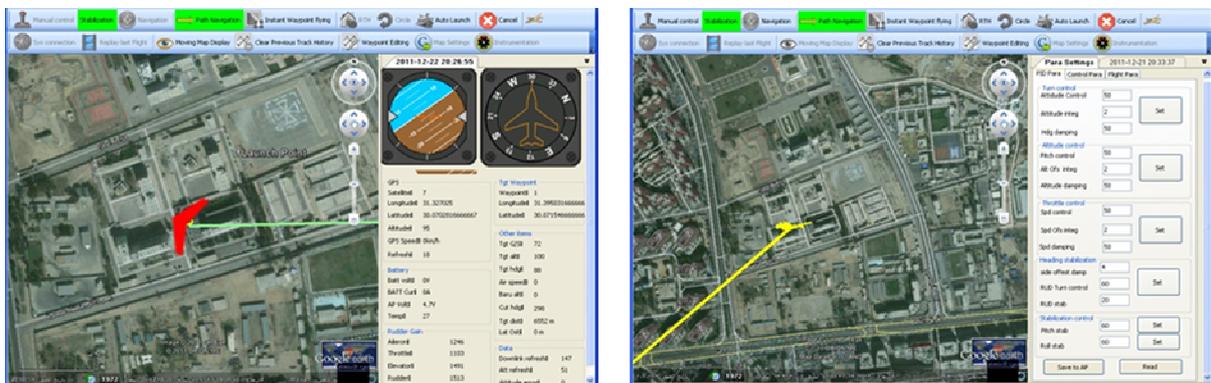


Fig. 20. FY-3ZT Autopilot connection diagram.

Figures shown below explain the main interface for the used autopilot that indicates the status of the SAKR 2 in pitch and roll and heading (a), PID setting for required control for the designed SAKR 2 (b).



a)

b)

Fig. 21. GUI for ground control station for SUAV.

Camera System Installation

Channel 2.4 GHz Tx/Rx video system with CCD camera is used. A high power 1500mW Tx on 2.4 GHz ensures a very long signal range. The system includes SONY CCD 1/3inch Camera, 2.4 GHz 8 channel Video Transmitter (1500mW), 2.4 GHz 12 channel Video Receiver.



Camera specification

Input voltage: 9~12.6V

Size: 38*38mm

Flight time: Approx 60min/100mah 3S

FPV Tx Specification

Channel: 8-Ch, AV synchronization

Power: 1500mW

Weight: 18g

Size: 73x41x14mm

Fig. 22. FPV Hobbyking long range Camera system.

Flight test which lead to performance analysis is the last step of the design cycle. These flight tests revealed the design problems and provided an iterative process to optimize and trim the vehicle. Mostly after each flight test a new refinement was done on our SAKR 2 to improve the maneuvering capability, reducing the size and takeoff weight and increasing the endurance.

Since SAKR 2 is small in size and trimming its flight status in the initial real flights was difficult, it was decided to have 2-times scaled, exactly identical to the main one with different vertical fin configuration, to start the first flight tests and study the stability and maneuverability more precisely. Initial flight tests were performed to find the proper CG position. The CG location was varied in each test to trace the appropriate one with respect to stability issues. Initial tests of the central vertical fin position model revealed poor stability in roll axis. To refine the problem, the design was modified to move the vertical fins to act as end plates.



Fig. 23. SAKR 2 Tested configurations.

The first SAKR 2 flight was accepted using landing gear arrangement in order to have rough estimation about upsticking speed and takeoff distance. During test we observed that the ground run phase take 20 meters to be airborne with speed of 10 m/s and makes slightly mixed positive yaw with negative roll during climb phase

specially with increasing the throttle, this move was trimmed by the pilot during flight. The first modification after flight test is to increase the incidence angle by two degree to decrease ground run and to be more easily to hand launched. Also, changing the installation angle of the motor mount will reduce the motor torque. The problem solved by setting the motor up by 2 degree. These changes resulted in reducing the stall speed of our vehicle and solve the yawing reaction with increasing power. We performs several tests with hand launch takeoff, we observed high performance with almost zero trim. We managed to fly over 20 minutes in 100 m local altitude.



Fig. 24. SAKR 2 Flight test.

Conclusion

The design of a flying wing need not imply a difficult, time-consuming process and, ultimately, an unstable and uncontrollable aircraft. We describe in this paper how by using project requirements a design area which includes those requirements can be determined. Once the airfoil and aircraft geometry are established, stability and lift and drag analysis are performed to analyze aircraft tendency to be stable one. By utilizing these suggested methods a flying wing aircraft can be created from a few relatively simple requirements. The modifications in production molds concerns assembly and disassembly techniques for SAKR 2 has to be fulfilled in order to acquire manback UAS.

Future work has to be taken in consideration to complete the unmanned aerial system and to predict flying qualities for the designed SUAV. Some of this future work is wind tunnel testing for the model to have accurate stability derivatives and to build the control model for SAKR 2 in order to take a step forward for designing and implementing our own autopilot. Also using XFLR5, DATCOM and AVL software for calculating aerodynamic model is very useful for a quick start to complete controller modeling and making the necessary simulations before flight tests.

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