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Conceptual Design Methodology of Day Flight Solar UAV

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ABSTRACT

The staff and the students of the aeronautical departments of King Abdul Aziz University of Saudi Arabia and Tokai University in Japan have collaborated to design and manufacture a solar powered UAV called Sun Falcon I, designed for non-stop day and night operations. The first conceptual design steps called for particular considerations and inputs from such performance features as the operational cruise velocity, altitude, payload, endurance, rate of climb, and payload. This was then followed by estimating from first principles such vehicle sizing features as probable wing span, aspect ratio, chord length, nominal lift and drag values, and so on to the geometry of the solar panels acceptable on the wing. Lift, drag and weight of the UAV had to be estimated quite accurately followed by the determination of operational power needed for the actual flight as well as power management schedules established to recover continuous power from solar panels to replenish the onboard batteries for night operations. The temperature charts from Saudi Meteorology were inspected closely to make sure that the UAV operations were convenient throughout the year.

Nomenclature

α	Angle of attack
θ	Instantaneous inclination of the vehicle
A	Area
B	Full span of the main wing
c	Airfoil chord length
C_l	Sectional lift coefficient
C_L	Lift coefficient,
C_p	Pressure coefficient
D	Drag
Eff	Efficiency
E	Energy
E_b	Total required battery energy
E_g	Generated energy by solar panel
E_s	Specification energy of battery module
g	Gravitational constant
L	Lift force per unit length
M	Mach number
M_m	Module mass of solar panel
P	Power
Re_c	Reynolds number
R/C	Rate of Climb (dH/dt)
Rad	Radiation
S_w	Main wing area
S_p	Solar panel area
s	Semi span
V	Velocity
T	Thrust
V_∞	Free stream velocity
W	Weight of airplane
x, y	Axis of Coordinate system
ρ_∞	Free stream density
γ	Specific heat
sub	scripts
ave	Average
req	Required
init	Initial
solar	Solar conditions
cruise	Cruising conditions

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Keywords

Solar UAV's, Aircraft Performance, Energy and Power requirements of UAv's, long duration UAv's

1. INTRODUCTION

Unmanned Air Vehicles are becoming increasingly useful for both civilian missions as well as more demanding theatre deployment in military conflicts. In civil operations they could be used quite proficiently in such roles as coastal patrol for narcotics prevention and other border violations, surveillance of protracted areas, reconnaissance in military operation or other platform perching roles and over the hill inspection for military purposes. They could also be used to look for minerals, oil and other water foot prints in Saudi environment or even control large scale movement of people in Hajj seasons. The military operations could involve use of such vehicles for reconnaissance, aerial watch of sensitive installations and guarding a military encampment against intrusions. More sophisticated versions of large size UAV's could actually be used as pilot-less interceptors against or other military platforms on bombing missions. The UAV's depending on their designated mission come in all shapes and sizes. In extremely low Reynolds regimes their deployment is considered to be in Micro Unmanned Air Vehicle (MUAV) regime where they tend to mimic the flight of the small sized insects where their application may be reserved primarily for sophisticated intelligence purposes with designed navigational capability programmed to fly in extremely close environments, stair well cases, and highly secure and prohibited areas. A very large number of such micro sized vehicles with simple flight capabilities in a swarm may be used to overwhelm a target or damage the power plant of a more sophisticated flight platform. The aerodynamics of such vehicles even when modeled on 2D airfoils, as discussed by Yuan and Khalid [1-3], is quite complex often involving the presence of a separation bubble with telling consequences upon aerodynamic behavior. At such minuscule size and navigational imperatives the propulsion requirements become increasingly difficult to implement and there is a tendency to move towards a flapping wing concept where the power would now be recovered from the flapping or beating of even smaller wings in order to facilitate the stability and control of such a vehicle.

Elsewhere, at immediately higher Reynolds number the realm of ordinary UAV's becomes operative where the size of the vehicle may stretch from the dimensions of a stretched palm to a couple of meters. Again it is the mission and payload requirements which would determine the size, overall weight and the propulsion features of such a vehicle. A large number of commercial companies have sprung up all over the world to cater for the UAV's in this market to address different missions as outlined above. At much higher flight Reynolds numbers the role of Predator in Afghanistan has proven that such unmanned vehicles can be used with deadly economy against selected individuals or adversary groups with reduced risk to home military. Indeed at these full scale aircraft flight Reynolds numbers UAV aircraft like the Stealth Bomber and Global Hawk hold sway which can respectively be used for large scale bombing over global distances or as remote surveillance platforms for monitoring and controlling the aerial activity of a distant adversary.

The Sun Falcon project calls for the design and manufacture of a flight vehicle to operate at intermediate Reynolds numbers of about $4x10^5$ at a cruise velocity of about13.9m/s at an altitude of about 100 m. It is basically a demonstrative project to show that such a vehicle can be designed within the resources and manpower facilities of universities to address an important civilian and military requirement. The design strategies can easily be picked up by local manufacturers to compete against in a burgeoning international market. While in some ways the conceptual design was modified by referring the very comprehensive and detailed strategies by Noth et al [4], the design cycles have been focused upon a defined mission that is limited in day flight to arrive at the Sun Falcon configuration with a minimum of iterations. Whereas the paper by Harasani et al [5] deals mostly with aerodynamic design considerations, the present paper has included a detailed outline of the power management strategies as well as a discussion of the aerodynamic imperatives.

Such a vehicle would be designed to remain airborne for long hours including night flight with solar panels installed strategically on the wing surface to consume the batteries to operate during the day and store sufficient energy for flight into the night hours. However, in this study, Sun Falcon was conceptually designed only for day flight with meeting an above-mentioned requirement as the first step forward the design of UAV that has longer endurance time. A more advanced version of the present design, Sun Falcon 2 would address more prolonged flight operations lasting many days and nights with a flight schedule also a part of its design to keep the operations within a prescribed region.

2. Preparations for Concept Design

2.1 Aerodynamic coefficients

The initial design characteristics of the Sun Falcon were estimated simply by using the linear theoretical equations. The preliminary design cycle used in this initial stage of the study is shown in Figure 1. In order to work through the various boxes in the design cycle, such aerodynamic features as the lift, skin friction, induced drag and power requirements for a given cruise velocity and aspect ratio were estimated from relations as follows:

$$C_L = 2\pi\alpha \quad (1)$$

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

$$C_f = \frac{1.328}{\sqrt{Re}} \quad (3)$$

$$C_{D_{total}} = \frac{D}{\frac{1}{2}\rho V^2 S} = \frac{C_{D_i} + C_f}{0.85} \quad (4)$$

2.2 Estimation of weight

Other weights for the airframe, battery and cell were estimated as follows

$$Total\ Weight\ of\ Solar\ Modules\ (\text{kgf}) = w_{sm} * S_{sm} \quad (5)$$

Where, w_{sm} and S_{sm} are a weight of solar module per unit area and a total area of solar modules, respectively. We assumed that $w_{sm} = 2.0(\text{kgf/m}^2)$ because of an *empirical value*.

$$Weight\ of\ battery\ (kgf) = 1 \quad (6)$$

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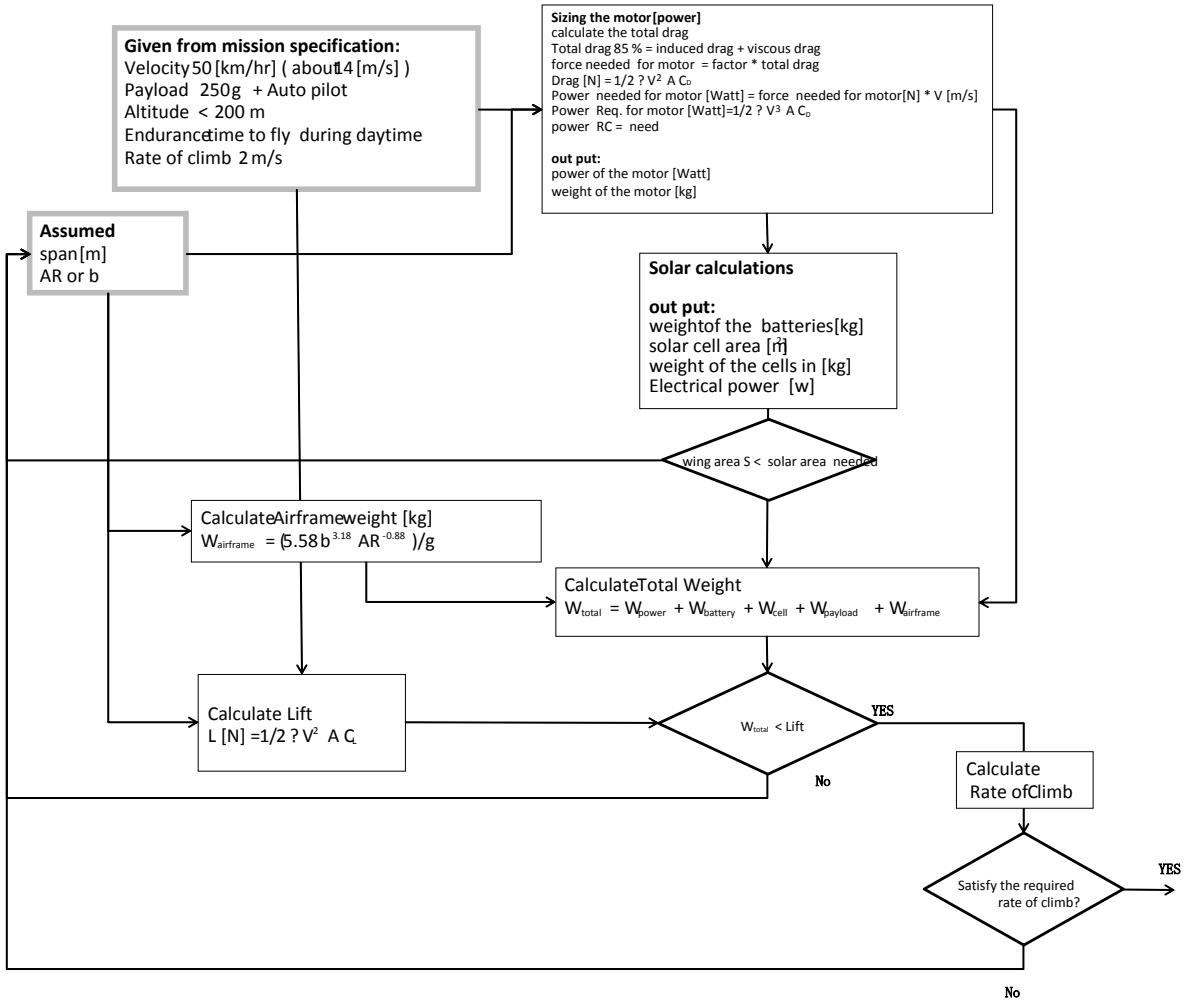

Fig 1: Preliminary Design Cycle for the Sun-Falcon

$$Weight\ of\ Airframe\ (kgf) = 5.58b^2 \frac{AR^{-0.88}}{g} \quad (7)$$

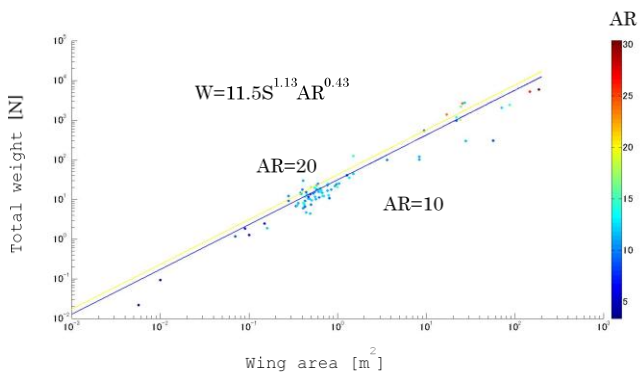

Fig.2 Estimation model for total weight of solar UAV

The next equation is derived from a statistical fit obtained from the data shown in Figure 2 for some 84 Solar UAVs. The overall weight of the Sin Falcon 1 as a whole was estimated using the equation:

$$W = 11.5S^{1.13}AR^{0.43} \quad (8)$$

The estimated weight ratio of solar panels is 2.0kg/m^2 and the weight turns out to be approximately twice the weight of the ordinary panels used in solar UAVs, Thus the airframe weight can be estimated directly by subtracting the half weight of solar panels(1.22kg) and weight of batteries, avionics, motor, and propeller (0.8kg) from the total weight. The weight distribution from various UAV components is appropriately itemized in Table 1. Therefore, the airframe weight of SUN-FALCON is estimated to be mere 2.10kg. If now the payload requirement is estimated to be 0.6kg, the total weight of the SUN FALCON is estimated to increase to some 6.53kg.

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Table 1 Required Performance and Specifications for Sun-Falcon

Mission	
Cruising Speed [km/hour]	50
Cruising Altitude [m]	100
Payload [kg]	0.6
Endurance Time [hours]	10
Rate of Clime R/C [m/s]	2
Configuration of Main Wing	
Cruising C_L [-]	0.4
Wing Span [m]	3.5
Aspect Ratio [-]	7.8
Main Wing Area [m^2]	1.57
Given Data	
Efficiency of Motor and Propeller [-]	0.7
Efficiency of Solar Cell [-]	0.17
Minimum Average of Daily Horizontal Radiation [Wh/m^2]	4300
Energy Density of Battery [Wh/kg]	200

Cruising Power and Energy	
Total Drag [N]	4.03
Required Thrust [N]	4.03
Required Total Energy[Wh]	800
Solar Cells and Module	
Number of Row [-]	3
Number of Column [-]	26
Solar Module Area [m^2]	1.28
Solar Module Weight [kg]	2.44
Battery	
Number of Li-ion Battery [-]	16
Total Battery Weight [kg]	0.69
Weight	
Airframe Weight [kg]	2.29
Total Weight [kg]	6.82

2.3 Energy Requirements and Battery Capacity

The energy requirements for the prototype will be based upon the available power generation attributes of the solar panels. The minimum value of average daily horizontal radiation4,300 [Wh/m^2] for day time in Saudi Arabia was obtained from a radiation measurement chart[5]. The energy balance in terms of the portion generated by the solar panels and the amount used up under fight conditions is appropriately shown Iin Figure 3. It is understandable that the remaining portions will be duly used for charging up the battery and storage for use when normal radiation replenishment is unavailable.

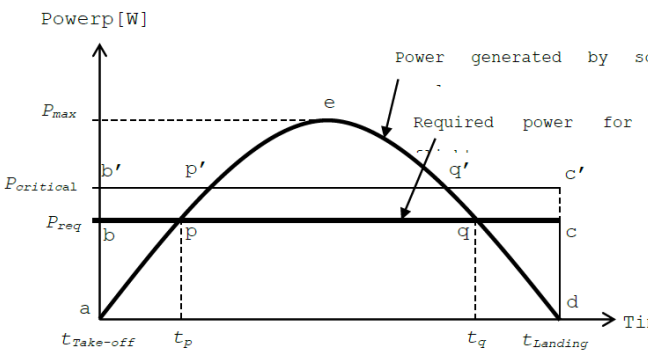

Fig.3 Distribution of Energy Balance

In the diagram depicted in Figure 3, the flight condition is ideally simplified such that the plane takes off at the time of sun rise and lands at the time of sun set. The energy consumption requirements under flight, E_m, and the available supply of energy from solar panels E_g, are represented by the rectangular area "abcd", and "aeda", respectively. The sinusoidal curve "aed" represents the continuous power generated by the solar panels, while the straight line "bc" is power required during cruising flight, which is considered to be constant for the present simple case scenario. Up to point P on the sine curve, the energy requirement during take- off phase is more than the amount being generated by the solar panels. The excess amount must be supplied from the energy stored in solar cells. Solar cell power gets over required one at the point p. In fact under actual flight, owing to heavy climb loads, the plane needs more power than P_{req} as shown by the line "bc". Similarly during the descent phase would not consume the full P_{req} immediately before landing. Maximum value of power supplied by solar panels is P_{max}, which may be determined as following. Since,

$$E_g = \int_{t_{Take-off}}^{t_{Landing}} P_{max}\, sin\left(\frac{t - t_{take-off}}{t_{Landing} - t_{Take-off}}\pi\right) dt \qquad (10)$$

And therefore,

$$P_{max} = E_g * \left(t_{Take-off} - t_{Landing}\right) * \pi/2 \qquad (11)$$

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As the power generated by the solar panels for the take-off phase is less than the total power required for flight, the energy represented by a triangle "abp" is actually the shortfall which must be topped up by the energy already stored in the charged battery. Another shortfall represented by "qcd" occurs during the landing phase and must this time be balanced by the surplus energy gathered in the regime "peqp". It must be cautioned here that as the charged battery is used during the landing phase, the requirement of energy depicted by "qcd" must not exceed the total energy gathered in the quarter "pepq" as the battery would have been exhausted below the acceptable level. As explicit in Fig.2, P_req is determined directly by the actual power being consumed by motor and it understandably approaches the value of P_critical when the surplus energy stored in the battery is equals to the energy shortfalls during various flight phases. P_critical in fact shows the limitation for charging up the battery, and it is obvious therefore that P_reqshould not exceed P_critical.

Here, P_critical is determined to be 0.725*P_max and considering the diagram in Figure 3 it is observed that energy requirement over and above what is duly available from direct solar cells,

$E_{Over} = peqp = E_1 = E_2,$

$$E_1 = \Delta abp\ (Battery\ Capacity) \quad (12)$$

$$E_2 = \Delta qcd\ (Battery\ Capacity)$$

and the condition about the P_{req} can be state as

$$P_{req} \leq P_{critical} \approx 0.725 * P_{max}\,. \qquad (13)$$

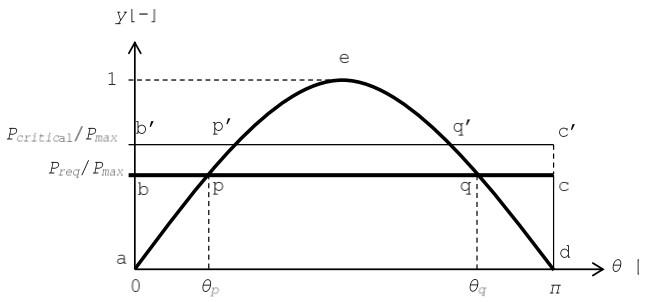
Additionally, a pragmatically designed battery must cater for some overshoots in flight times and an optimized battery capacity along these lines than is

$$E_b = E_1 + E_{extra} \qquad (14)$$

Where E_1 can be obtained by above consideration about geometrical relation between E_g and E_1,and E_{extra} means the energy for extra flyingtime.

To obtain the pragmatic battery energy E_b, based on Fig.4, areas of domains for each energy are calculated as below.

First, the area surrounded by the curve of $\sin\theta$ and the abscissa, A_{cell}, is



$$A_{cell} = \int_0^{\pi} \sin\theta\, d\theta = 1 - \cos\pi = 2\,. \qquad (15)$$

Fig.4 Geometric relation based on the energy balance

Next,

$$\theta_p = \sin^{-1}\left(\frac{P_{req}}{P_{max}}\right), \qquad (16)$$

then, an area for the minimum battery capacity E_1, A_{abp}, is

$$A_{abp} = \frac{P_{req}}{P_{max}}\theta_o - (1 - \cos\theta_o)\,. \qquad (17)$$

And, moreover, an area for the battery capacity for extra running time t_{extra}, A_{extra}, is

$$A_{extra} = \frac{P_{req}}{P_{max}} \cdot \frac{t_{extra}}{t_{endurance}}\pi \qquad (18)$$

Where $t_{endurance}$ means endurance time. According to Fig.3 various quadrants of areas under graph refer to individual requirement of energy and thus substituting aforementioned relations, the pragmatic battery capacityE_b can be calculated by

$$E_b = E_1 + E_{extra}$$

$$= \frac{A_{abp}}{A_{cell}}E_c + \frac{A_{extra}}{A_{cell}}E_g$$

$$= \left(\frac{P_{req}}{P_{max}}\theta_o - (1 - \cos\theta_o) + \frac{P_{req}}{P_{max}} \cdot \frac{t_{extra}}{t_{endurance}}\pi\right)\Big/2 \times E_g$$

$$= \left(\frac{P_{req}}{P_{max}}\sin^{-1}\left(\frac{P_{req}}{P_{max}}\right) - \left[1 - \cos\left\{\sin^{-1}\left(\frac{P_{req}}{P_{max}}\right)\right\}\right] + \frac{P_{req}}{P_{max}} \cdot \frac{t_{extra}}{t_{endurance}}\pi\right)\Big/2 \times E_g\,. \qquad (19)$$

The power requirements can then be stated as follows:

$$P_{req} = \frac{T_{req} \cdot V_{cruise}}{Eff_{motor} \cdot Eff_{propeller} \cdot Eff_{gear\ head}} \qquad (20)$$

$$E_g = Eff_{solar} \cdot Rad_{ave} \cdot A_{solar} \qquad (21)$$

2.4 Energy for climb

The calculations of the rate of climb are closely related to the power and available thrust. Starting from the equation of motion:

$$\frac{W}{g}\frac{dV}{dt} = T - D - W\sin\theta \qquad (22)$$

$$\frac{W}{g}\frac{d\theta}{dt} = L - W\cos\theta \qquad (23)$$

Where,*dV/dt* is the acceleration (alpha). Therefore the required thrust is given by:

$$R/C = \frac{dH}{dt} = V\left(\frac{T-D}{W} - \frac{1}{g}\frac{dV}{dt}\right) \qquad (24)$$

$$T = D + W\left(\frac{R/c}{V} + \frac{1}{g}\frac{dV}{dT}\right) \qquad (25)$$

Required Time to Cruising Altitude is therefore given by

$$T_{cruise} = \frac{H}{R/C} \qquad (26)$$

The average acceleration is given by :

$$\frac{dV}{dt}_{ave} = \frac{(V_{cruise} - V_{init})}{T_{cruise}} \qquad (27)$$

Energy required to climb is calculated from the equation:

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$$E_{climb} = \int \frac{\frac{dV}{dt}_{ave} \cdot T_{cruise}}{Eff_{motor} \cdot Eff_{gear\ head} \cdot Eff_{prop}} dt \quad (28)$$

Above relationship were used in a preliminary exercise for an airfoil based two dimensional based wing whose C_L and C_D characteristics were estimated from simple lifting linear relationship[6].

3. Design cycle for concept design

3.1 Flow chart

For a more exhaustive design procedure it must be understood that almost all parameters related with aerodynamic characteristics, design geometry, weight and performance inter connect and influence each other. Invariably it was seen that a small change in one parameter had more global effect on performance. Nevertheless an appropriate flow chart, Figure 5, was duly prepared to make sure that each individual aspect of the design satisfied the aerodynamic, performance, power, weight and energy relationships. Each loop in the flow chart was singularly mean tested within its own domain before the outcomes were passed on to adjacent loop for further scrutiny and final convergence.

As shown in Fig. 4, the parameters in group I, II, and III can be instantly loaded and varied. If an individual failed to satisfy a particular test at some juncture A, B, C, or D, it was necessary to refine by changing the values in group I, II, and III. For example, the main wing area is compared with total area of solar panel in discrimination section A. Section B interrogates the energy balance between the required energy based on the total drag and the energy generated by solar panel as above-mentioned relation (13). When these discriminants are positively satisfied only then the total weight of airplane can be determined with some certainty. Sections C and D are

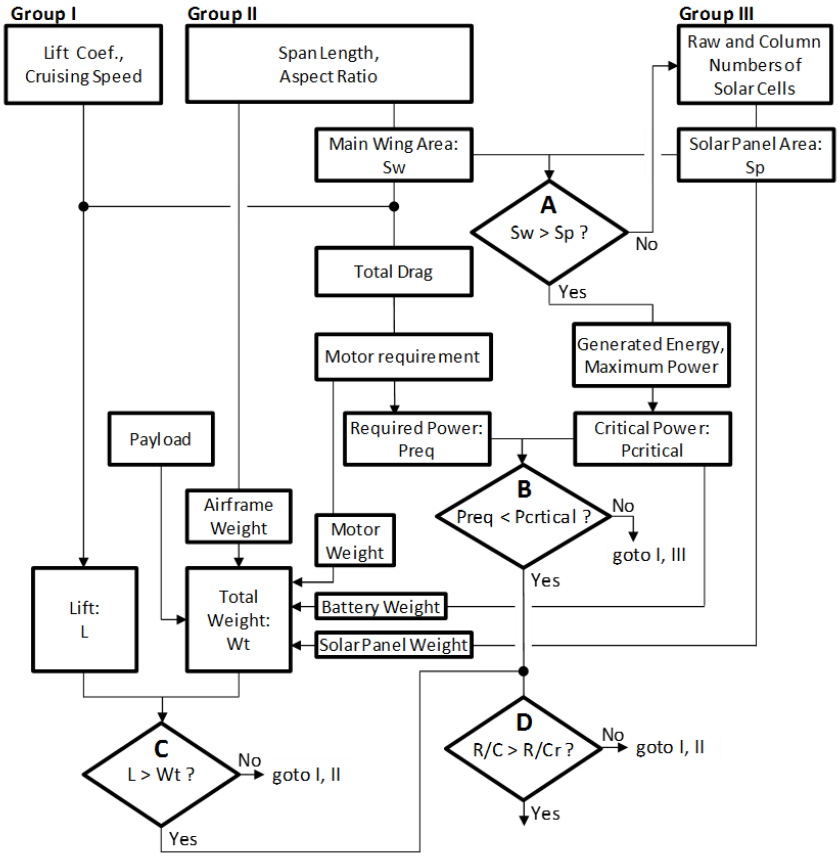

Fig. 4 Flow chart of Design cycle for Sun-Falcon

designed to ascertain if indeed the lift is larger than the total weight and whether the rate of clime of relation (24) satisfies requirement R/Cr, respectively.

3.2 Parametrical results of concept design

The first set of configuration design along with geometry, weight, propulsion and performance as obtained from the latest flow chart diagram in Figure 5 are shown in table 1. This set of parameters satisfies all of discriminants in the flow chart above mentioned. As shown in Table 1, the UAV has a span length of 3.5 m and an AR 7.8. These values were obtained for a planar shape main wing and were considered to be most appropriate geometry owing to the restriction concerning the structural maintenance of solar modules and their efficiency in generating most power. The main wing was untapered with no sweep-back.

4. Initial Flight Tests

The Sun Falcon 1 project was completed in June 2013, with the first successful test flights conducted during June 6th to June 9th 2013. The UAV in flight is shown in Figure 6. Some of the lessons learnt from the flight tests included the following:

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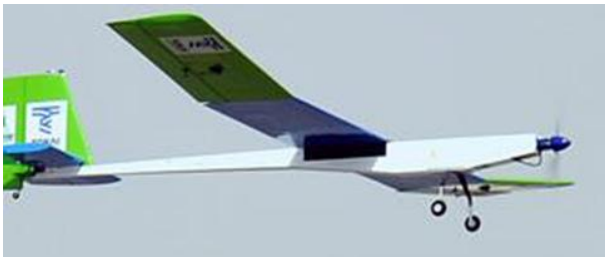

Figure 5. Sun Falcon 1 in full flight during the first Test Runs

- The overall weight of the UAV was monitored with meticulous care throughout the fabrication stages. The energy balance too was established accurately during the design phases.
- It is proposed that the S9352 HV high torque Futaba servos should be used in order to avoid a situation where the RC glider would not respond when controlled in flight. The use of a closed-loop system on the rudder and elevator controls was strongly recommended as the servos were more than 1300 mm away from the controls' hinge point.
- It was recommended to replace Li-po batteries in place of Li-ion version as they would be more suitable for the longer duration flight.
- It was learnt that the brushless motor's capacity of 810 kV was too high a kV for the 17x12 propeller, and this created the heat due to the overload from the propeller. After conducting numerous on-ground tests with the same 810 kV motor and a smaller propeller and also with a 350 kV motor it was decided to fly Sun Falcon 1 with a 350 kV motor, and a 17 x 12 propeller with a Li-po or Li-ion battery.
- For the overall design concept, the first flight tests confirmed that the prototype performed according to intended design. Further collection of data on such parameters as speed, altitude, power required by motor, propeller revs/sec, energy generated by solar modules and so on, during cruising flight and their respective comparison against design values will determine the need to further fine tune our design formulae. Weight estimate techniques certainly warrant modification as the weight of airframe as the actual weight was somewhat heavier than design one.

5. CONCLUSION

The Sun Falcon wing has been designed starting from the rudimentary analysis of a conceptual configuration based on classic methods. More accurate design procedures, based on accurate CFD analysis were adopted towards the final design iterations. The design at hand provides for a cruise velocity of about 17 m/s at an altitude of 100 m with a minimum endurance of about 10 hours although continuous recharging of batteries in Saudi climate would provide for virtually non-stop day/night operations. The longer duration flight has been designed to be made possible by efficient operations of the solar panels reputed to be capable of consistent charging of the batteries from prolonged daylight sun exposures. While the staff and students from both institutions participated in the overall design over long distance teleconference meetings, the majority of the manufacturing was carried out at the Tokai University in Japan.

The continuing collaboration between the King Abdul Aziz University of Saudi Arabia and the Tokai University of Japan calls for a second generation design and manufacture of Sun Falcon 2 capable of continuous flight operations lasting many days and nights. The above lessons learnt particularly in the area of weight management would be extremely important in order to keep the weight within design limits. Sun Falcon II would have to support larger wings in order to accommodate a bigger solar panel area. The flight tests from Sun Falcon 1 would be used to calibrate precisely the number of solar panels required. Equally, the motor capacity would be worked out accurately based on the data from Sun Falcon 1 to avoid heating issues. For better control during flight, due attention would be focused on the closed loop system on the rudder for a better response. Further study of data from flight tests and continuing wind tunnel experiments suggest that the V shaped tail design would offer better prospects. It has further been recommended that Li-ion batteries would be better suited for the longer duration flight intended for Sun Falcon 2, than the first set of Li-po batteries installed on Sun Falcon 1.

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