# Conceptual and preliminary design approach of a high altitude, long endurance solar-powered UAV

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# Conceptual and Preliminary Design Approach of A High Altitude, Long Endurance Solar-Powered UAV

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***Abstract*—In this paper, a description of newly developed conceptual and preliminary design approaches is introduced, to design a high altitude long endurance solar powered unmanned aerial vehicle. The conceptual design approach is based on representing the mass and the power requirement of each aircraft element as a fraction, in order to produce the total mass equation. The fractions have been gathered statistically from available data of existing aircraft of the same type. The mass equation will be solved for the expected range of the aspect ratio and span of the wing to generate a possible design space. The optimal design is then concluded from the design space as the minimum weight. This approach has been validated using existing data of related aircraft. In the preliminary design tool, the aircraft shape and the wing geometry are designed using the main characteristics of aircraft which were obtained from a previous design stage. An appropriate twist and sweep of the wing are then found using an optimisation tool which contains the aerodynamic and the structure models. The outcome at this stage must be a flyable aircraft geometry capable of meeting the mission requirements. Moreover, a case study of designing a solar powered aircraft is introduced using the developed design approaches.**

***Keywords—solar aircraft, conceptual design; preliminary design; high altitude aircraft; optimisation***

## I. INTRODUCTION

High altitude long endurance solar-powered unmanned aerial vehicles (HALE SP UAVs) operate within the stratosphere (17-25 km altitude) and use only the solar irradiance as a source of power [1]. The available solar irradiance depends on the time of the day, the day of the year, the latitude, and the position of the solar cell panels. For long endurance missions, the aircraft can fly continually if the energy collected during the daytime is enough to operate the aircraft for 24 hours [1, 2]. Thus, the power and the mass of the aircraft are the starting point of the design [3]. The flight profiles of these aircraft is at high altitudes and moderate flight speeds, which represents low Reynolds number operating conditions [4]. Therefore, the viscous effect is dominant and can influence the aerodynamic performance [5]. The required lift coefficient is high when compared to more conventional lower-altitude aircraft, due to reduced air density. Therefore, high altitude aircraft have an extreme wing span, which suggests that the aircraft will be quite flexible, so it is crucial to employ the elastic influence in the design tool [6]. As shown above, sustained flight at high altitudes represents a substantial challenge because of the numerous inter-related engineering disciplines required for the analysis.

The design of aircraft involves three design stages; conceptual, preliminary and detailed design. In this paper only the first and the second stages will be described briefly to design a high altitude long endurance solar-powered unmanned air vehicle.

## II. THE CONCEPTUAL DESIGN STAGE:

A mathematical model has been developed to estimate the optimal aircraft weight, its main characteristics and the weight & power of each aircraft element for given mission requirements. The concept of this methodology is based on representing the mass and the power of each aircraft element as a fraction of the total mass or power to produce the total mass equation. The fractions are retrieved statistically from available data of existing related aircraft. Then the mass equation will be solved for the expected design space which represents the aspect ratio and span of the wing. The optimal design is then concluded from the design space as the minimum weight. The optimal weight as well as the corresponding level flight power, aspect ratio and the wing span will be used to size all the aircraft components. This approach has been validating using an existing baseline aircraft called SHAMPO and the results approved its capability to provide a good estimation of the weight and the main characteristics of aircraft. This methodology is detailed in these references [7-9] and summarized in Fig.1.

- Define the mission requirements such as the operational location, altitude, start and end date of the mission and the payload characteristics.
- Evaluate the available solar energy and the daytime hours at the operational location and altitude.
- Suggest wing configuration shape and then suggest a suitable wing section profile (aerofoil).
- Define the aerofoil characteristics such as the aerodynamic performance at the expected flight conditions with differing Reynolds number.
- Design the aircraft components using the conceptual design tool. The main characteristics of aircraft will be

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found such as the aspect ratio and the wing span as well as the weight and the power of each element in the aircraft.

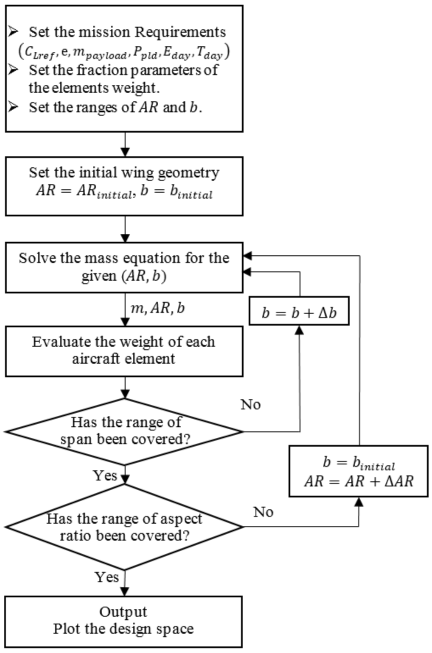


Figure 1. Conceptual design approach to explore the design space

## III. THE PRELIMINARY DESIGN STAGE:

In this stage, the aircraft shape and the wing geometry will be designed using the main characteristics of the aircraft which is obtained from the previous design stage. The optimal twist and sweep of the wing will be found using the optimisation tool which is an iterative process. The outcome from this stage is a flyable aircraft geometry capable of meeting the mission requirements. The design steps of each iteration of optimisation process can be summarise as following:

- Define the aircraft geometry as obtained from the previous stage.
- Evaluate the non-spar element weight of aircraft using empirical equations.
- Suggest the distribution of the inboard weights, such as the propulsion system, power storage and the payload in the span-wise and the chord-wise directions.
- Suggest the sweep and the twist distribution for the wing.
- Evaluate the critical aerodynamic loads using the aerodynamic solver (Tornado VLM) then find the entire load distribution in the span-wise direction.
- Size the spar of the wing to withstand the critical load using the composite structure model. In addition, the mechanical properties of the wing are obtained. Figure 2 shows the overview of the wing sizing process.
- Find the shape of the wing whilst in flight and its aerodynamic performance parameters. This process can be achieved using the interaction between the aerodynamic and the structural influences until a quasi-static equilibrium is achieved. A quasi 3D aerodynamic solver has been developed and used to evaluate the aerodynamic performance including the profile drag of aircraft. A linear finite beam element model has been adopted to evaluate the elastic deformation of the wing.
- Analyse the aircraft stability; only the static stability will be constrained during the optimisation process.

The individual design models will be briefly presented next:

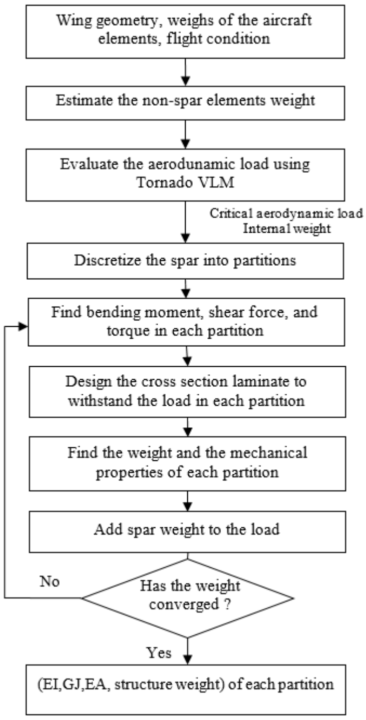


Figure 2. Overview of the wing sizing methodology

### A. *Aerodynamic Model (Quasi-3D Aerodynamic Solver)*

Low-order analysis tools are employed to facilitate efficient computations, which are important when there are multiple optimization loops for the various engineering analyses. This model consists of two parts as shown in Fig.3: a 3D inviscid flow model (vortex lattice method) and a 2D viscid/inviscid flow solver. Additional supplementary tools are employed such as an aerofoil section geometry generator. The vortex lattice method is used to evaluate the lift force and the induced drag for the wing geometry. A two-dimensional panel method coupled with an integral boundary-layer method are used to assess 2D profile drag in a strip-wise sense. Span-wise integration then leads to the evaluation of the profile drag of the entire wing. The computational methods are developed and written in a MATLAB environment. The model has been validated with experimental data and the results show an acceptable estimation as detailed in references [9-11].

### B. *Composite Structure Model*

This model can design the wing structure for the given critical expected load in addition to evaluating the elastic deformation of the wing using linear finite beam elements method. The wing structure is divided into two categories; the non-spar elements and the main spar. The weight of the non-spar elements can be estimated using empirical equations as functions of the wing geometry. These equations were accomplished by NASA for solar powered high-altitude UAVs. The spar can be sized by using a numerical method [8, 12]. The spar has been represented as a composite rectangular thin-walled beam and assumed to withstand the entire load with no contribution from the secondary wing components. The spar is discretised into partitions. Then, each partition will be designed to withstand the exerted load by finding the numbers of the laminate on each side to prevent any expected failure with a margin of safety. The overview of the cross section design methodology is explained in Fig.4. The orientations of laminate of each side of the spar are inspired by available data of existing structures of related aircraft. A linear finite beam element has been employed to calculate the elastic deformation of the wing under the given load distribution while only a quasi-static equilibrium is considered. The model has been written in MATLAB. The stress analyses and the elastic deformation results have been validated using a high order commercial package ANSYS. The results showed a good agreement with that obtained by ANSYS. The weight estimation model also has been validated by designing a wing structure of existing high altitude aircraft called Helios P03. The tool has demonstrated its capability of obtaining a good estimation of the weight.

### C. *Optimisation Tool*

A design optimization framework has been developed, under a MATLAB environment, combining aerodynamic, structural and stability analysis. A Canonical Genetic Algorithm has been used in the optimization process to vary the geometric variables until achieving the optimal design which is represented by the minimum drag coefficient. The optimiser code has been developed based on the principle of genetic evolutionary processes [13]. The communication shape of the optimisation tool is presented in Fig. 5.

## IV. AIRCRAFT CASE STUDY

A basic surveillance mission is selected to design a high-altitude long endurance solar-powered UAV. The mission is to survey the Iraqi marshes during the period 1st April to 10th September whilst carrying a 100 kg payload at an altitude of 17 km. The reference available solar energy and the daytime hours are calculated. A swept-back wing geometry is suggested as well as the wing section during the conceptual design stage. The results of the conceptual design tool were a wing span of 72 m and wing aspect ratio of 20. In addition, the power and mass of the required fuel cells, propulsion units, avionics and other electrical instruments are estimated. Then, the output of the conceptual design is used to perform the preliminary design stage.

The inboard weight distribution has been suggested carefully and then the required sweep to achieve a specific static margin is calculated; this sweep angle came out as 7 degrees.

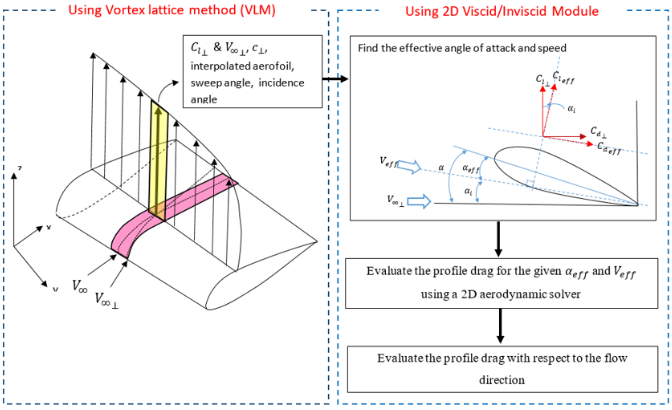


Figure 3. Procedure of the Quasi 3D Aerodynamic model

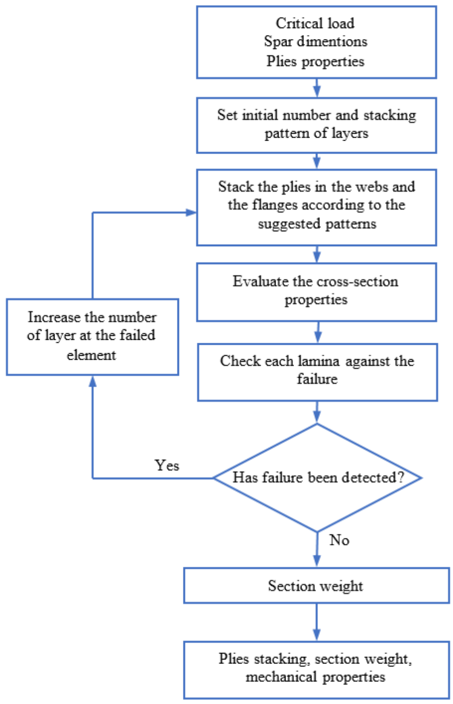


Figure 4. Cross-section design methodology

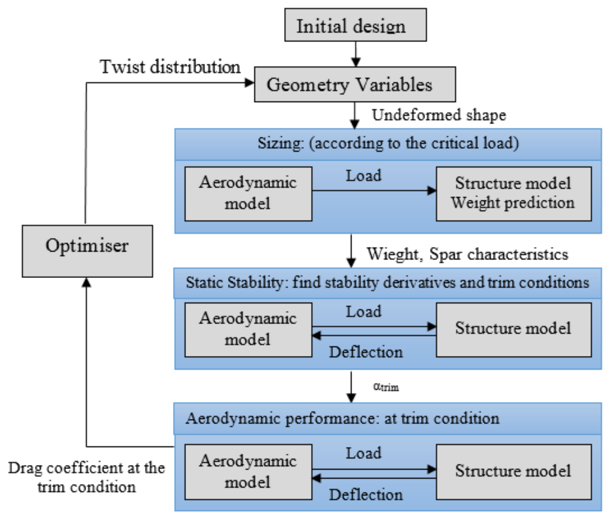


Figure 5. Optimisation Architecture

Eight variables are used to represent the twist distribution along the semi-span of the wing. The optimisation tool has been used to find the optimal twist distribution to achieve minimum drag at a trim condition. The optimisation problem can be written in the following form:

*Minimise:* *Drag coefficient*
*Variables:* *Twist distribution (8 variables along Semi-span)*
*Subject to:* *Lift force at trim condition=aircraft weight*
*Statically stable*

The final result of the optimisation came out with a flyable geometry which meets the given mission requirements. The aeroplane is trimmed at 8.5 degrees angle of attack, 0.85 lift coefficient and 0.0206 total drag coefficient. The gradient of the pitching moment coefficient is -0.277 and the zero lift pitching moment coefficient about the centre of gravity is 0.028, indicating a longitudinally stable aeroplane. The final configuration is shown in Fig.6.

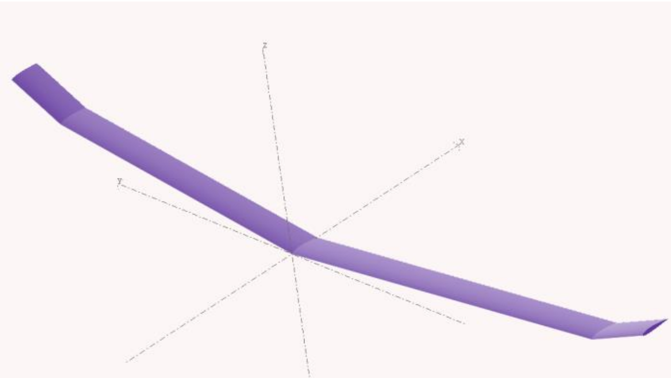

Figure 6. Optimal airplane configuration

## CONCLUSION

In this paper, two design approach have been briefly described; the conceptual and the preliminary design approaches. The source code for each approach is written in-house and validated using existing data of related aircraft. The main contribution of this paper includes collecting all the design tools which have been accomplished by the authors and combining them in one sequence of design methodology.

## ACKNOWLEDGMENT

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