ICEUBI2013 University of Beira Interior Faculty of Engineering Covilhã | Portugal
November 27 28 29 International Conference on Engineering
Engineering for Economic Development

Aircraft Design Methodology Using Span and Mean Wing Chord as Main Design Parameters

Pedro Gamboa, Miguel Silvestre, Pedro Albuquerque
Department of Aerospace Sciences, Faculty of Engineering, University of Beira Interior, Covilhã, Portugal
pgamboa@ubi.pt, mars@ubi.pt, pffa@ubi.pt

Conference Topic – CT 02

Abstract

The present work describes an aircraft design methodology that uses the wingspan and its mean aerodynamic chord as main design parameters. A simple weight representation model based on semi-empirical formulae is used to estimate the prototype structural weight as a function of its wing planform shape. The energy mass - either fuel or batteries - is estimated from the user defined UAV mission profile. A *Microsoft Excel*® workbook that calculates the performance of each wingspan and mean aerodynamic chord combination for the user defined mission and vehicle's performance requirements has been developed. In order to demonstrate this methodology, the results for a case study using the design specifications of the Air Cargo Challenge 2013 are shown.

Key Words: Aircraft Design, Parametric Study, Design Optimisation, UAV, Air Cargo Challenge

Symbols/Acronyms

Symbol	Description	Unit	Symbol	Description	Unit
α_i	Induced angle of attack	-	h	CG position (fraction of the mean aerodynamic chord)	-
β	Symmetrical of the zero lift angle	-	h_n	NP position (fraction of the mean aerodynamic chord)	-
δ	Thrust setting	-	I	Electric Current	A
η_{motor}	Motor efficiency	-	K_w, K_{ht}, K_{vt}, K_{fus}	Ratio of the reference component weight to the reference vehicle's weight	-
η_{Prop}	Propeller efficiency	-	K_n	Static Margin (fraction of the mean aerodynamic chord)	-
λ	Taper ratio	-	L	Tail arm	m
Λ	Aspect ratio	-	L_{fus}	Fuselage length	m
ν	Kinematic viscosity of air	-	M	Mach Number	-
ξ_l, ξ_m, ξ_n	Adjustable damping coefficients	-	n	Load Factor	-
τ	Correction factor for non-elliptical lift distribution	-	N	Rotational frequency	min^{-1}
b	Wingspan	m	P_{motor}, P_{prop}, P_{req}	Motor, propulsive and required power	W
c	Aerofoil chord	m	q	Dynamic pressure	Pa
C_f	Local friction coefficient	-	Q	Fuselage interference factor	-
C_f^{total}	Total friction coefficient	-	R_{bat}	Battery internal resistance	Ω
C_l, C_d, C_m	2D lift, drag and pitching moment coefficients	-	R_{ESC}	Electronic speed control internal resistance	Ω
C_{m_0}	2D Pitching moment coefficient on the aerodynamic centre	-	Re_x	Local Reynolds number	-
C_L, C_D, C_M	3D lift, drag and pitching moment coefficients	-	S	Reference area	m^2
C_{M_0}	3D Pitching moment coefficient on the aerodynamic centre	-	t	Aerofoil thickness	m
C_P	Propeller power coefficient	-	U	Electric voltage	V
D	Aerodynamic drag	N	V	Velocity	m/s
D_{fus}	Fuselage diameter	m	x	Characteristic length	m
e	Oswald coefficient	-	W	Weight	N
e_{spec}	Specific energy	J/kg	W_{ener}	Energy weight	N
E	Energy	J	W_{pay}	Payload weight	Ng
$\mathcal{F}$	Fuselage form factor	-	W_{struct}	Structural weight	Ng
g	Gravity acceleration	m/s	W_{syst}	Systems weight	N

Sub/superscript	Meaning	Sub/superscript	Meaning
$(.)_{fus}$	Refers to the fuselage	$(.)^{ref}$	Refers to reference variables
$(.)_{ht}$	Refers to the horizontal tail	$(.)_w$	Refers to the wing
$(.)_{vt}$	Refers to the vertical tail		

Acronym	Description	Acronym	Description
ACC	Air Cargo Challenge	NP	Neutral Point (Aerodynamic Centre of the full aeroplane)
CG	Centre of Gravity	R&I	Research and Innovation
ESC	Electronic Speed Control	R&D	Research and Development
DTOW	Design Take-off Weight	TRL	Technical Readiness Level
MDO	Multidisciplinary Design Optimisation	UAV	Unmanned Aerial Vehicle

1. Objective

Besides the exponential improvement in terms of digital data storage and computers' processing capacity [1], optimisation algorithms have had very significant progresses throughout the last decades.
These advances are fostering industrial applications of multidisciplinary optimisation [2], which are becoming central design tools that go far beyond the scope of elementary R&I and R&D with low TRL[1].
Earlier quest for maximum performance has been superseded by the need for an equilibrium among performance, life-cycle cost (including manufacture), reliability, maintainability, among others. This is true for every aircraft, with the relevance of each discipline varying with respect to its mission, the most relevant aspects being: the best performance (e.g. fighters), the maximum energetic efficiency (e.g. airliners) or the lowest manufacturing and/or operating costs - relevant for all aircraft categories.
Parametric studies have been used for several decades and are an interesting approach based upon which traditionally used graphical methods can be used to find the maximum or minimum of a multivariable function. However, despite the advantages of the graphical method, which obviates the local extrema problem, the truth is that this approach becomes inadequate for problems with more than 3-4 dimensions (Figure 1).

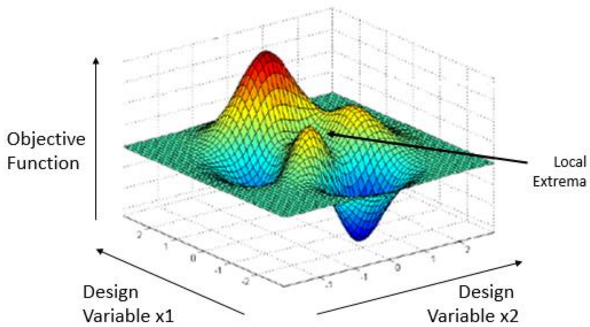


Figure 1 - Typical graphical output of a parametric study with two design variables

This work aims to develop a comprehensive methodology for the design optimization of an unmanned aerial vehicle together with a computational tool – developed in Microsoft Excel® - in order to materialize the concept.
The main design variables are the wing span and the wing mean geometric chord, with the objective(s) function(s) being defined by the user in accordance with the intended mission profile and performance requirements. The implemented parametric study is briefly described in section 2.

2. Methodology

Obtaining the correct size of an aircraft is essential to produce a high performance design. Size and mass also have a close correlation with costs. The design methodology developed is based on an extensive parametric study developed in-house in a spreadsheet whose primary design parameters are the wing span (b) and the wing mean chord (c). Other design parameters allowed to vary in the study are the wing aerofoil cruise lift coefficient, the centre of gravity (CG) position, the tail arm, the aerofoil, the motor and the propeller among others. In the implemented tool, low fidelity models have been developed for various analysis of the vehicle: aerodynamics, stability and control, propulsion, weight and balance and flight performance. The design process is based on the mission profile and performance requirements. Aerodynamic performance is evaluated at every *Re* required and aerofoil data

[1] Technical Readiness Level

is either interpolated or extrapolated as necessary from a few combinations of *Re* and angles of attack. Lift and drag coefficients are computed considering trimmed conditions in each and every analysis point. The tail is automatically sized for stability and control by setting the desired tail volume coefficients or the required static margins. The propulsion model matches the propeller and the motor for a given speed and throttle setting, for every point analysed, and provides thrust and power consumption. The implemented weight model scales the airframe component weights of a known similar aircraft vehicle based on wing span, wing mean chord, aerofoil relative thickness, load factor, maximum weight and fuselage size, among others. Systems' weights, such as motor, batteries, solar cells, etc., are provided by the user or modelled simply by scaling parameters, e.g. using the specific energy of battery to scale its weight for the required energy to perform the mission.

2.1. General Approach

One of the most relevant design variables is unknown – the aircraft's DTOW will depend on the mission profile and on the wing span to mean aerodynamic chord combination. Additionally, and in order to encompass the power source differences that are commonly found on today's vehicles – either reciprocating engine or electrical motor – both possibilities have been analysed. As such, an iterative procedure is put into place as depicted in Figure 2, where the energy to accomplish the required mission greatly determines the DTOW.

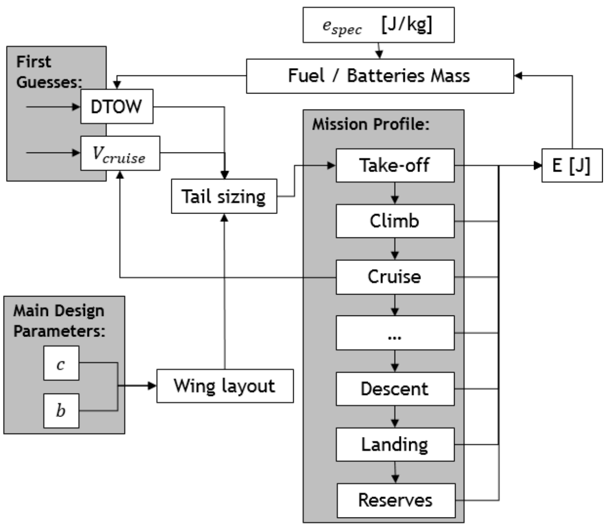


Figure 2 - Scheme featuring the iterative procedure for determining the DTOW, cruise velocity and tail sizing using the wing chord and wing span as main design parameters.

2.2. Stability/Tail Sizing

The tail sizing is done at each iteration step because the best horizontal tail length, area and aspect ratio will depend on the cruise velocity which will vary with DTOW and will comply with the static stability requirements. The user will have to define the minimum desired static margin (K_n) (1), which is the distance between the CG (h) and the aeroplane's NP[2] (h_n) – all variables are non-dimensional, in fraction of the wing's mean aerodynamic chord [3].

$$K_n = h_n - h \tag{1}$$

Furthermore, the CG position will be asked for, with the horizontal tail size and distance from the wing obeying the consequent neutral point position, which is fully defined as the static margin and CG positions are known.

[2] Neutral Point

2.3. Weight Model

The DTOW is the sum of the aircraft structure, systems, energy and payload weights, where: structure refers to the structural components, like the wing, tail, fuselage and landing gear; systems refers to all devices required for flight that are not structural, such as the motor, propeller, ESC, cables, servomechanisms and the receiver; energy refers to the weight of the power source used, either fuel, in the case of a reciprocating engine or a battery in the case of the electrical motor; and payload refers to all cargo and devices that might be transported by the UAV but which are not required for flight.

$$DTOW = W_{syst} + W_{struct} + W_{ener} + W_{pay} \tag{2}$$

The type of propulsion, payload weight and systems weight, together with the intended mission profile are user inputs. Conversely, the structural weight will depend on the energy weight which in turn will depend on the mission profile and has to be dully estimated. For a typical mission profile with a cruise/loiter phase the energy weight is iterated until the distance/time required is attained.

2.3.1. Power Supply Weight

The approach for determining the amount of energy required for successful mission completion has already been illustrated in Figure 2.
As already stated, two power sources have been considered and built in the present tool – reciprocating (fuel) and electrical (batteries). Once the specific energy (e_{spec}) of the fuel or battery is known, it is trivial to obtain the power supply (energy) weight:

$$W_{ener} = E.\, e_{spec}.\, g \tag{3}$$

2.3.2. Structural Weight

An empirical approach for estimating the structural weight of the main components - the wing, the horizontal stabilizer, the vertical stabilizer and the fuselage - has been adopted [4] to allow for variations in the lifting surfaces aerofoil's relative thickness, areas, aspect ratios and taper ratios as well as on the fuselage length, diameter and distance between the aerodynamic centres of the wing and the horizontal stabilizer.
The weight formulation is based on reference [5]. Accordingly, the structural weight of the aircraft under analysis is the sum of the reference aircraft[3] structural weight and the main structural components weight variations (4).

$$Wstruct = Wstruct^{ref} + \Delta W_w + \Delta W_{ht} + \Delta W_{vt} + \Delta W_{fus} \tag{4}$$

The weight increments (ΔW_i) in the main structural components, in accordance with reference [5], can be estimated as follows:

$$\Delta W_w = K_w W_w^{ref} \left[\left(\frac{S_w}{S_w^{ref}}\right)^{0.758} \left(\frac{\Lambda_w}{\Lambda_w^{ref}}\right)^{0.6} \left(\frac{\lambda_w}{\lambda_w^{ref}}\right)^{0.04} \left(\frac{t/c}{(t/c)^{ref}}\right)^{-0.3} \left(\frac{nW}{n^{ref} W^{ref}}\right)^{0.49} - 1 \right] \tag{5}$$

$$\Delta W_{ht} = K_{ht} W_{ht}^{ref} \left[\left(\frac{S_{ht}}{S_{ht}^{ref}}\right)^{1.344} \left(\frac{\Lambda_{ht}}{\Lambda_{ht}^{ref}}\right)^{-0.448} \left(\frac{nW}{n^{ref} W^{ref}}\right)^{0.414} - 1 \right] \tag{6}$$

[3] Aircraft structure using similar materials and manufacturing approaches

$$\Delta W_{vt} = K_{vt} W_{vt}^{ref} \left[\left(\frac{S_{vt}}{S_{vt}^{ref}} \right)^{1.31} \left(\frac{\Lambda_{vt}}{\Lambda_{vt}^{ref}} \right)^{0.437} \left(\frac{nW}{n^{ref} W^{ref}} \right)^{0.376} - 1 \right] \quad (7)$$

$$\Delta W_{fus} = K_{fus} W_{fus}^{ref} \left[\left(\frac{S_{wet}}{S_{wet}^{ref}} \right)^{1.086} \left(\frac{L}{L^{ref}} \right)^{-0.051} \left(\frac{L_{fus}/D_{fus}}{L_{fus}^{ref}/D_{fus}^{ref}} \right)^{-0.072} \left(\frac{nW}{n^{ref} W^{ref}} \right)^{0.177} - 1 \right] \quad (8)$$

where K_w, K_{ht}, K_{vt}, and K_{fus} are the ratios of the reference component weight to the reference vehicle's weight, for the wing, horizontal tail, vertical tail and fuselage, respectively.

2.4. Aerodynamics Model

In all conventional subsonic aircraft with medium/high aspect ratio wings ($\Lambda > 6$) the major single contribution to the overall aerodynamic performance comes from the wing aerofoil and therefore its careful selection is paramount. For this reason, the driving parameter of the design study is the cruise/loiter aerofoil lift coefficient (C_l). This value is user defined but is bounded by the aerofoil's stall lift coefficient and the cruise speed interval given by the user, according to the vehicle's operational requirements.

2.4.1. Lifting Surfaces

For the aerofoils analysis – and given the low Reynolds number expected for UAV operations – the XFOIL software has been chosen. The user selects a number of different aerofoils which are run in XFOIL for a set of Reynolds numbers.
Then, when the aerofoil aerodynamic coefficients (C_l, C_d and C_m) for a specific Reynolds are needed – at each step of the iteration featured on Figure 1 – the stored matrices data from XFOIL are thus either interpolated or extrapolated.
In order to obtain the lifting surfaces aerodynamic coefficients (3D), the following procedure has been adopted: an iterative solution for finding both the induced angle of attack (α_i) and the 3D lift coefficient (C_L), using the system of equations (9).

$$\begin{cases} \alpha_i = \frac{C_L}{\pi \Lambda (1+\tau)} \\ C_L = C_l \cos \alpha_i \end{cases} \quad (9)$$

The 3D drag coefficient (C_D) is the sum of the profile drag (C_d) with the lift induced drag (10):

$$C_D = C_d + \frac{C_L^2}{\pi \Lambda e} \quad (10)$$

The Oswald span efficiency factor (e) is defined according to wing planform geometry. Finally, and assuming a wing without twist, taper or sweep, the lifting surface pitching moment coefficient at zero lift can be assumed equal to the aerofoil pitching moment coefficient at zero lift (11):

$$C_{M_0} \approx C_{m_0} \quad (11)$$

2.4.2. Fuselage

In order to determine the fuselage parasite drag, the equivalent skin-friction method is adopted because a well-designed aircraft in subsonic cruise will have parasite drag that is mostly skin-friction drag plus a small separation pressure drag [5]. For laminar ($Re_x < 1000$) and turbulent flow ($Re_x \geq 1000$), respectively:

$$C_f = \frac{1.328}{\sqrt{Re_x}} ; \quad (12)$$

$$C_f = \frac{0.455}{(\log_{10} Re_x)^{2.58}(1+0.144M^2)^{0.65}}\ ; \tag{13}$$

Equations (12) and (13) refer to the local friction coefficient and must be integrated along the characteristic length to obtain the total friction coefficient.
Several corrections for the local Reynolds number ($Re_x = Vx/\nu$) may be found in references [5] and [6] which account for early transition on rough surfaces.
The total viscous drag can be computed from equation (14), where (q) is the dynamic pressure, (S_{wet}) the fuselage surface area (wetted area), (C_f^{total}) is the total friction coefficient, ($\mathcal{F}$) is the fuselage form factor and (Q) the interference factor, which accounts for the fact that parasite drag is increased due to the mutual interference with the lifting surfaces and other components. This effect is usually negligible in the case of the fuselage ($Q = 1$) [5].

$$D_{fus} = qS_{wet}C_f^{total}\mathcal{F}Q \tag{14}$$

The form factor ($\mathcal{F}$) is a function of the fuselage characteristic dimensions [5]:

$$\mathcal{F} = 1 + \frac{60}{\left(\frac{L_{fus}}{D_{fus}}\right)^3} + \frac{\frac{L_{fus}}{D_{fus}}}{400} \tag{15}$$

2.4.3. Miscellaneous

In what concerns to miscellaneous aerodynamic drag, it is at least worthwhile to consider the landing gear contribution for the overall drag since most UAVs do not have retractable landing gears and they do not have streamlined shapes like the lifting surfaces or the fuselage. Its drag is best estimated by comparison to test data for a similar landing gear configuration [5]. In case that data is not available, the gear drag can be estimated as a summation of the wheels, struts and other components using typical drag coefficients for each component as featured in reference [5].

2.5. Propulsion Model

A scheme with the propulsion model for the electrical motor case is shown in Figure 3. An analogous approach can be developed for the reciprocating engine case, but is not shown in the present document. The thrust setting (δ) and current (I) are initially guessed and thereafter iterated, while the user has to know the idle voltage (U_{bat}^0) as well as the internal resistances (R_{bat}) of the battery and the ESC (R_{ESC}). Finally, the required power (P_{req}), is the product of the aeroplane drag (D) by its velocity (V) for the flight condition under study, which comes from the flight mechanics analysis and vehicle mission.
Three iteration cycles have been built. One is optional and is only used if one wants to establish a maximum current and correct the thrust setting if this limit is exceeded. A second iteration cycle makes sure the electric current is corrected so that the shaft power (P_{motor}), equals the absorbed propeller power (P_{prop}/η_{prop}) which is a must since there is no slippage between the two. A last iteration corrects the thrust setting to ensure the available propulsive power (P_{prop}) equals the required power (P_{req}). In the diagram of Figure 3 (η_{motor}) refers to the motor efficiency and (η_{prop}) to the propeller's efficiency.
Several different alternatives are possible for the propeller module in order to obtain the propeller efficiency and power coefficient as a function of the propeller advance ratio, which will enable the calculation of the propeller power. For example, by increasing degree of complexity, possible propeller analysis programs are: PropSelector, JavaProp and QPROP. The first is relatively straightforward to use - making use of experimental propeller data [7] - and does not need many input parameters while the latter has got a relatively sophisticated and accurate aerodynamic model - with an advanced Blade-Element/Vortex Method [8] - thus requiring more inputs including detailed information regarding the propeller shape.

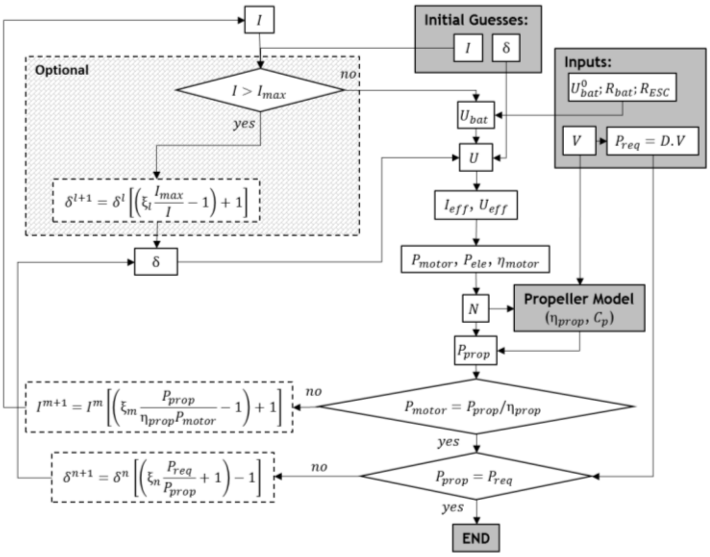


Figure 3 - Propulsion model implemented for the electrical power source case

3. Case Study

3.1. Air Cargo Challenge 2013

In order to make a case study, the ACC'13 regulations have been adopted. This competition, created in 2003 by students from the IST[4], is an International biannual competition destined to the academic community with engineering, science and technology background.
Each team has the assignment of designing, building and flying a radio-controlled aircraft which goal is to lift the highest useful payload possible in a 60 meters runway [9]. Furthermore, each group has to provide written and oral support to its decisions. The final score is a weighted sum of the design report, technical drawings, oral presentation and actual payload demonstrated in flight, with bonuses and penalties also being used.
The aircraft must perform a pattern flight – usually without any demanding manoeuvres, because the models are essentially freighters - which has to be completed for the lifted payload to be valid. The design drivers/specifications have several differences in each edition but the main ones for the Air Cargo Challenge 2013 are shown in Table 1 [9]. The propulsion is provided by the AXI 2826/10 brushless out-runner electric motor and a commercially available propeller.

Table 1 - Design drivers for the Air Cargo Challenge 2013

Maximum load factor	2.5
Maximum current	$40A$
Battery	$> 2500\ mAh$
Motor	AXI Gold 2826/10

Maximum take-off distance	$60m$
Transportation box (outer dimensions)	$1m x 0.4m x 0.4m$
Goal	Lift maximum cargo

The maximum load factor indicated in Table 1 is supposed to size the wing structure when it is suspended at the wing tips with the aircraft weighting the maximum take-off weight. This load factor value can be regarded as equivalent to a flight load factor of 2.5 for a similar wing root bending moment.
In the ACC, the competition airfield conditions are quite important so they were considered in optimizing the aircraft for maximum payload capability. In this exercise an altitude of 40m and an average wind speed of 1.5m/s about 0.5m above ground level were assumed.

[4] Instituto Superior Técnico – Universidade de Lisboa

In this type of case study, where a cruise/loiter phase does not exist, the energy (batteries) weight is kept constant while the payload weight is iterated for the performance parameters to be fully met.

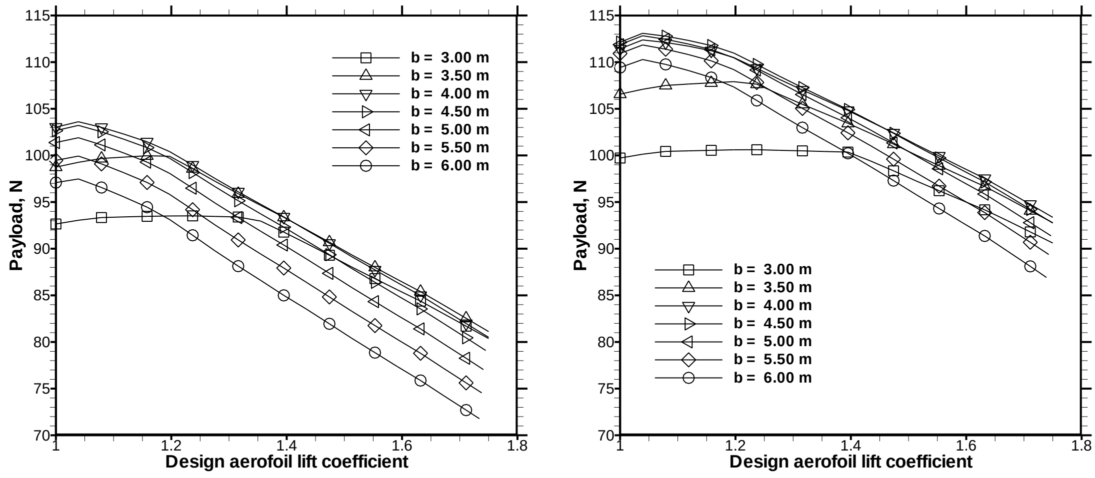
3.2. Results

A parametric study is performed for four design parameters: wing span; wing mean geometric chord; aerofoil design lift coefficient; and propeller size (diameter and pitch). The wing span values vary between 3m and 6m and the chord values vary between 0.25m and 0.4m. The (C_l) values are varied between 1.0 and 1.75. Two propellers are used: 14”x7” and 15”x4”; with their characteristic curves obtained by PropSelector. The aerofoil adopted is the S1223 [10] which is one of the most widely used aerofoils in this kind of competition and has a relative thickness of 12%.
The weight of the vehicle is calculated using equations (2) through (8), using as reference parameters those of a model designed for the ACC’11. These parameters are summarized in Table 2.

Table 2 – Reference values for weight estimation

K_w	0.469	W_{ht}^{ref}	$0.981N$
W_w^{ref}	$6.267N$	S_{ht}^{ref}	$0.181m2$
S_w^{ref}	$1.512m^2$	K_{vt}	0.037
Λ^{ref}	11.667	W_{vt}^{ref}	$0.49N$
$(t/c)^{ref}$	0.14	S_{vt}^{ref}	$0.104m^2$
n^{ref}	3	K_{fus}	0.421
W^{ref}	$133N$	W_{fus}^{ref}	$5.639N$
K_{ht}	0.073	L_{fus}^{ref}	$1.9m$

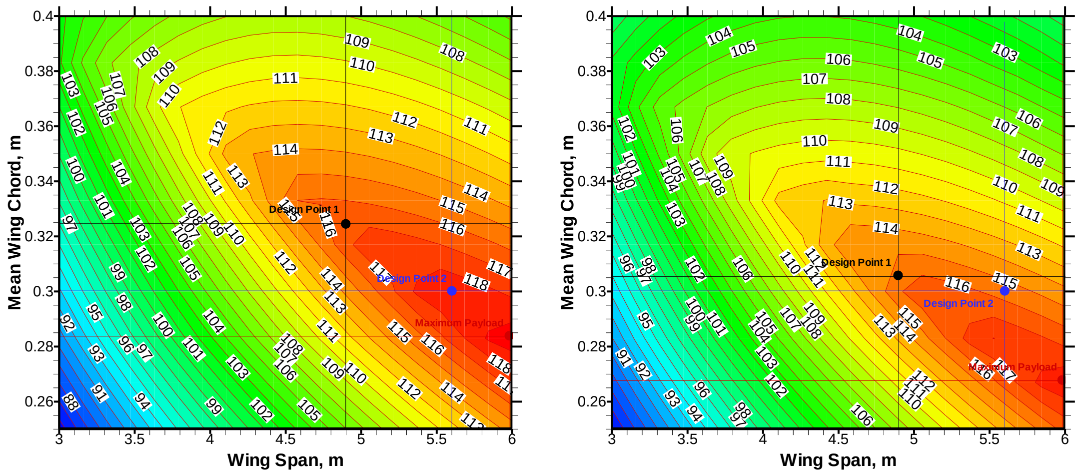
The taper ratio of the wing is considered unchanged in this study. Also, the ratios of horizontal tail aspect ratio, vertical tail aspect ratio, fuselage wetted area and fuselage fineness ratio are considered unity in equations (5) through (8).
Four performance parameters are selected to size the payload capability of each parametric design: maximum take-off length of $(60m)$ for a lift-off speed of 1.14 times the stall speed; minimum rate of climb after take-off of $(0.7m/s)$; minimum of 2 full circuits around the airfield; and minimum of 30º bank angle turn.



(a) propeller 14”x7” (b) propeller 15”x4”

Figure 4 – Payload weight results as a function of design aerofoil lift coefficient and wing span for the two propeller sizes and a wing mean chord of 0.36m

Since there is a requirement in the competition regulations that the aircraft must be disassembled and transported in a closed box with outer dimensions $(1m\ x\ 0.4m\ x\ 0.4m)$ it is necessary to build the wing into panels of about $(0.98m)$ long and with maximum chord length of about $(0.38m)$. Selecting a wing planform with a rectangular central panel and tapered middle and outer panels approximating an elliptical chord distribution with the tip chord length with 60% of the root chord, the maximum mean chord is around $(0.36m)$.
For this upper limit of wing mean chord and the 15”x4” propeller, a parametric study is performed for different wing spans and design aerofoil lift coefficients as shown in Figure 4. Clearly the 15”x4”propeller is superior to the 14”x7” for the given requirements. The limiting performance requirement at the lower lift coefficient values (and lower span values) is the take-off manoeuvre while at the higher lift coefficient values (and higher span values) the rate of climb requirement becomes critical. It is apparent that the best design occurs near $(C_l = 1.05)$ for a span 4m with the 14”x7” propeller and for a span of 4.5m with the 15”x4” propeller. The overall maximum from Figure 4 occurs for $(b = 4.5m, c = 0.36m, C_l = 1.04)$ and the propeller is the 15”x4” with a payload weight of $(113.09N)$.
Based on the results of Figure 4, performing a new parametric study for the span and chord design parameters in the ranges mentioned above and aerofoil lift coefficients of (1.05) and (1.25), with the 15”x4” propeller, the optimum aircraft size becomes apparent. The results for the payload capability of the designs obtained with those design parameters are shown in Figure 5. The optimum design in each graph of Figure 5 is represented by the red dot. The overall maximum from both graphs occurs for $(b = 6m, c = 0.283m, C_l = 1.05)$ with a payload weight of 119.6N.



(a) $C_l = 1.05$ (b) $C_l = 1.25$

Figure 5 – Payload weight results (in N) for the Air Cargo Challenge 2013 parametric study for various combinations of wing span, mean wing chord and aerofoil design lift coefficient using the 15”x4” propeller

Selecting a wing with five $(0.98m)$ panels in order for it to fit inside the transportation box, the maximum span possible is $(4.9m)$. Taking into consideration that limit, the design points for each of the graphs of Figure 5 are highlighted with the black dot. With these dimensions, the overall design point is $(b = 4.9m,\ c = 0.325m,\ C_l = 1.05)$ which produces a payload weight of $(116.5N)$. Another possible design point can be selected if a larger span is chosen. From Figure 5, it is clear that the payload can still be increased for larger spans and smaller chords. By reducing the wing chord, the effective wing thickness is also reduced allowing a sixth panel to fit inside the transportation box. An overall common design point may thus be adopted, shown with the blue dot in Figure 5, which increases the payload capacity at $(C_l = 1.05)$ and maintains it at $(C_l = 1.25)$. This final design point is $(b = 5.6m,\ c = 0.3m,\ C_l = 1.05)$ for a payload weight of $(118.7N)$, for the lowest aerofoil lift coefficient under analysis.
Other values of (C_l), more propeller sizes and even different aerofoils could be used to refine the parametric study and determine the design point more exactly.

4. Conclusions and Future Work

A methodology for performing parametric studies in aircraft design, where the wing span and the wing mean chord are the primary study parameters, was developed and implemented. The approach, which uses simple analysis models for the aerodynamics, weight, propulsion and performance analyses, was applied to a new design for the Air Cargo Challenge 2013 regulations showing its capability to determine the best UAV size and corresponding performance figures for maximum payload capacity.

The authors of this article are implementing an analogous methodology by developing a computational program in FORTRAN which is expected to enable a much faster and broader analysis thus capable of obtaining better solutions. Additionally, this tool may likely be much easier to improve in the future as it will be developed in independent working blocks.

Once all the different subroutines are working independently with an adequate complexity level, a truly MDO[5] approach will be the impending challenge. Several optimisation methods shall be studied (either gradient based or non-gradient based (Heuristic) methods). At least two of them shall be applied in order to obviate the local maxima/minima problem. The goal is to widen the optimisation spectrum by making the different disciplines interact in such a way that a weighted combination of them enable a better overall aeroplane design at the cost of not having the optimum for each discipline (e.g. aerodynamics, structures, etc.).

A final enhancement to such a tool may include an assessment of the profitability of morphing wing concepts, including a combination of up to three such concepts.

Acknowledgements

The work presented herein has been partially funded by the European Community’s Seventh Framework Programme (FP7) under the Grant Agreement 314139. The CHANGE project (Combined morphing assessment software using flight envelope data and mission based morphing prototype wing development) is a Level 1 project funded under the topic AAT.2012.1.1-2. involving 9 partners. The project started on August 1st 2012.

References

[1] Alfaris, Anas – Multidisciplinary System Design Optimization (MSDO), Lecture 2, 2010.

[2] Weck, Olivier de, Willcox, Karen – Multidisciplinary System Design Optimization (MSDO), Lecture 2, 2010.

[3] Etkin, Bernard – Dynamics of Flight – Stability and Control, JOHN WILEY & SONS, INC, 1996.

[4] Mestrinho, J., Gamboa, P., Santos, P. – Design Optimization of a Variable-Span Morphing Wing for a Small UAV, Structures Dynamics and Materials Conference – AIAA, Denver, Colorado, 2011.

[5] Raymer, D. P., Aircraft Design: A Conceptual Approach, pp. 280-284, pp. 404-405, 2nd edition, AIAA Education Series, AIAA, 1992.

[6] Corke, Thomas C., Design of Aircraft, pp.103-107, Prentice Hall, 2002.

[7] Prop Selector, Software voor de modelvliegliefhebber, http://www.hoppenbrouwer-home.nl/ikarus/software/propselector.htm. (26/07/2013)

[8] QPROP – Propeller/Windmill Analysis and Design – MIT, http://web.mit.edu/drela/Public/web/qprop/ (26/07/2013)

[9] Regulations for the Air Cargo Challenge 2013, Universidade da Beira Interior, http://acc2013.ubi.pt/?page_id=2. (25/07/2013).

[10] Selig, M.S., and Guglielmo, J.J. High-Lift Low Reynolds Number Airfoil Design. *Journal of Aircraft*, Vol. 34, No. 1, (1997), 72-79.

[5] Multidisciplinary Design Optimisation