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Design Methodology for Low Speed High Altitude Long Endurance Unmanned Aerial Vehicles

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Abstract

High Altitude Long Endurance (HALE) Unmanned Aerial Vehicles (UAV's) are being considered increasingly to perform an extensive range of tasks. Recent years have seen greater acceptance of UAV's for a wide variety of applications.

The focus of this research is the creation of a conceptual design methodology specifically for the design of these aircraft. The primary challenge lies in the fact that there have been very few built (especially in the low speed regime), so there is a small parametric database. In addition, they have been "one-off" designs (none have gone into production), with many not achieving their published design goals.

Key issues to be addressed are the lack of data on extremely high aspect ratio (22 < AR < 35) wing design, weight, and downwash angle prediction at the tailplane, propulsion system performance at altitude, low Reynolds number (Re < $1*10^6$) drag data for airfoils, tail volume coefficients for HALE UAV's.

The present code (written to operate in MS Windows) and methodology are flexible enough to consider a variety of possible configurations. In addition, the code is robust by allowing for the inclusion of future engine and wing profile drag data as it becomes available.

Nomenclature :

AR	Wing Aspect Ratio
CG	Center of Gravity
C_{Do}	Zero Lift Drag Coefficient
$C_{L\max}$	Maximum Lift Coefficient
d_{to}	Takeoff Distance
e	Oswald Efficiency Factor
g	Gravitational Acceleration
HALE	High Altitude Long Endurance
HP	Horsepower
K	$\frac{1}{\pi A \mathrm{Re}}$
n	Load Factor (in g's)
Re	Reynolds Number
RoC	Rate of Climb
S	Wing Planform Area
SFC	Specific Fuel Consumption
UAV	Unmanned Aerial Vehicle
V	Cruise Velocity at Altitude
V_{cr}	Cruise Velocity
$V_{\max L/D}$	Velocity of Maximum Lift to Drag Ratio
$V_{\min PR}$	Velocity of Minimum Power Required
V_{so}	Velocity of Stall
W	Weight
W_{PL}	Payload Weight
W_{TO}	Takeoff Weight
W_{TOGW}	Takeoff Gross Weight
$\frac{W}{S}$	Wing Loading
η_p	Propeller Efficiency
ρ	Air Density

1. Introduction - General Background

High Altitude Long Endurance (HALE) Unmanned Aerial Vehicles (UAV's) are increasingly being considered to perform a wide range of tasks.

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Recent years have seen greater value of UAV's in the military reconnaissance arena, first in the Persian Gulf war, with the use of small UAV's, and more recently over Bosnia with medium sized UAV's.

The NASA ERAST program uses HALE UAV's for environmental sensing and monitoring. The military has investigated using HALE UAV's for Theater Ballistic Missile defense, as well as general battlefield reconnaissance. Proposals have been made to use HALE UAV's for communications relay, long term surveillance (with a degree of flexibility not available from satellites), border surveillance, powerline monitoring, forest fire detection , and many others.

By removing the pilot, UAV's reduce risk, not only to the potential pilot, but also to politically sensitive situations. Removal of the pilot provides an overall simplification of the aircraft (with possible exception to actuators and flight control systems). It also provides for a greater variability of possible configurations by removing the need for a pilot to see out and sit up, removal of complex environmental controls, the need to accommodate other activities necessary for human subsistence, etc.

The justification for using Low Speed HALE UAV's is that the low speed regime can be better for environmental sampling, and they are less likely to disturb the composition of the air around them. Lower speed implies the regime in which reciprocating internal combustion engines are used, although it does not exclude the use of electric motors. These motors typically have lower first cost and lower fuel consumptiona. Despite having lower reliability, they still have lower maintenance costs than turboprops or turbofans.

High altitude flight is desirable for providing a wider field of view for environmental sensing and military reconnaissance. It is also the lower end of the regime of interest for Ozone layer composition/chemistry tests. The air is more stable which supplies a more stable sensor platform, it is also the altitude of lower windspeed, and it is above commercial airways (for the time being).

Long endurance is desired so that full solar day sensing can be carried out. When used in the disaster relief or Cellular Telephone relay role, the airplane is required to be on station for a maximum number of hours. In addition, in the military reconnaissance role it is desirable to watch developing problem areas for a long period of time. Of course in the anti-ballistic missile role, it would be advantageous to stay on station awaiting the launch of enemy missiles for as long as possible.

Having introduced a range of possible missions for a Low Speed HALE UAV, it is possible to make intelligent assumptions regarding some of the more basic performance parameters thereby enabling the preliminary sizing of the aircraft to begin.

2. Preliminary Constraint/Performance Equations

The preliminary design is of particular importance when the aircraft being sized is expected to perform functions that have been hitherto unattained by even the most recent of attempts. If the aircraft being sized was going to perform a more conventional task, than the methodology to follow is reasonably well defined and documented, and the design drivers are usually more readily apparent or surface quickly when analyzing parametric data.

For a High Altitude Long Endurance Unmanned Aerial Vehicle (HALE UAV) the design drivers are not as readily apparent. It is necessary to perform a constraint analysis on the aircraft based upon the general aircraft mission to be performed, for each possible set of mission requirements. This is achieved by defining what areas of the flight regime will be the most limiting on the overall design. For this aircraft, the obvious areas of limitation are Cruise, Endurance, and Minimum Rate of Climb/ Maximum Ceiling. An area that is not as obvious is the Maximum Load Factor.

Below are the equations necessary to perform the constraint analysis. The resulting constraint diagram will define the design space and subsequently be used to determine the initial design point of the aircraft. The Constraint Diagram provides multiple possibilities from which to launch the iterative conceptual design process.

All of the equations that follow are in terms of the horsepower to weight ratio as a function of the wing loading. In general, the equations are in terms of the lift coefficient. The conversion factor of 1/550 is present in all equations to convert from units of ft.lbf./s to horsepower.

2.1 Maximum Load/Turn

A quick derivation based on the assumption of a parabolic drag polar yields the equation:

$$\frac{HP}{W} = \frac{(1.2)^3}{550\eta_p} C_{D_o} \left(\frac{2n}{C_{L_{max}}}\right)^{3/2} \left(\frac{1}{\rho}\frac{W}{S}\right)^{1/2}$$

This equation is written in terms of the maximum lift coefficient and assumes a constant velocity turn. In this equation, the density selected was at cruise altitude. Propeller efficiency (η_p) is considered a constant value. Load factor (n) is provided depending on how robust a structure is necessary or desired, and what g loading the aircraft is to be designed for in a turn. K is a function of Oswald Efficiency factor and aspect ratio. The velocity used is 1.2 times the stall speed of the aircraft in a clean configuration.

As you can see from this equation, for a given design (fixed η_p, C_{D_o}, K), maximum load will be most sensitive to changes in Altitude (ρ), maximum load factor (n), and maximum lift coefficient ($C_{L_{max}}$).

2.2 Endurance

$$\frac{HP}{W} = \frac{4}{550\eta_p} C_{D_o}^{1/4} \left(\frac{K}{3}\right)^{3/4} \left(\frac{2}{\rho}\frac{W}{S}\right)^{1/2}$$

Again, in the endurance equation, for a fixed design, the design is sensitive to the zero lift drag coefficient (C_{Do}), the aspect ratio (through K), and the altitude (ρ)

2.3 Cruise

$$\frac{HP}{W} = \frac{2}{550\eta_p} C_{D_o}^{1/4} K^{3/4} \left(\frac{2}{\rho}\frac{W}{S}\right)^{1/2}$$

In this equation, only aspect ratio (through K) and altitude have any effect on the sensitivity of the design to variation of the cruise parameters.

2.4 Rate of Climb (RofC)/Ceiling

$$\frac{HP}{W} = \frac{1}{550\eta_p}\left[\mathrm{RoC} + \left(\frac{C_{D_o}}{C_{L_{max}}^{3/2}} + K C_{L_{max}}^{1/2}\right)\left(\frac{2}{\rho}\frac{W}{S}\right)^{1/2}\right]$$

In this equation, the parameters that affect sensitivity to a specific design are the altitude, aspect ratio, and the maximum lift coefficient ($C_{L_{max}}$). The reason that the maximum lift coefficient is used instead of the maximum lift drag ratio lift coefficient is that the true limiting factor in terms of lift coefficient at such a high altitude will most likely be stall speed for a given wing loading/power loading. Therefore, for the constraint diagram, the most limiting factor for the design is used.

To use this equation to calculate the absolute ceiling of the aircraft, we simply set the Rate of Climb (*RofC*) to zero.

2.5 Takeoff Distance

$$\frac{HP}{W} = \frac{2.44}{550\eta_p}\frac{1}{gd_{to}}\left(\frac{1}{\rho_{SL} C_{L_{max}}}\frac{W}{S}\right)^{3/2}$$

For the takeoff distance constraint equation the greatest degree of sensitivity is found in the takeoff distance specified and the maximum lift coefficient. The density is assumed to be that at sea level. The takeoff propeller efficiency should be that appropriate for a cruise maximized propeller unless a variable pitch propeller is assumed. The takeoff velocity is assumed to be 1.2 multiplied by the stall speed.

2.6 Stall Condition

$$\frac{W}{S} = \frac{\rho}{2} C_{L_{max}} V_{So}^2$$

This equation is sensitive to the assumed values of maximum lift coefficient and stall speed. The density is assumed at sea level for the constraint diagram since no other reasonable assumption could be made. Thus, this constraint is in effect a maximum allowable stall speed. This is chosen as a constraint for a UAV as it is often found that a low speed long endurance UAV will be built without assuming the drag and weight penalties of a permanent undercarriage.

3 Methodology

3.1 Mission Specifications

At this point in the conceptual design process it is necessary to fix some of the basic aircraft constants in order to perform the constraint analysis. Before this can be done, a better understanding of the mission specifications must be formulated. Figure 3.1 shows an example of the data necessary to complete the constraint analysis. In order to complete the constraint process, for a low speed HALE UAV of virtually any configuration, the data required in Fig. 3.1 can be known, assumed, or guessed !

It is reasonable to expect that the Cruise and Maximum altitudes would be known for any HALE UAV application. Aspect Ratio is another term that should have a minimum value above 20 for long endurance and high altitude flights. By allowing for the specification of Cdo, a variety of configurations can be considered early on, as long as some idea of comparative values of Cdo exist for the different configurations.

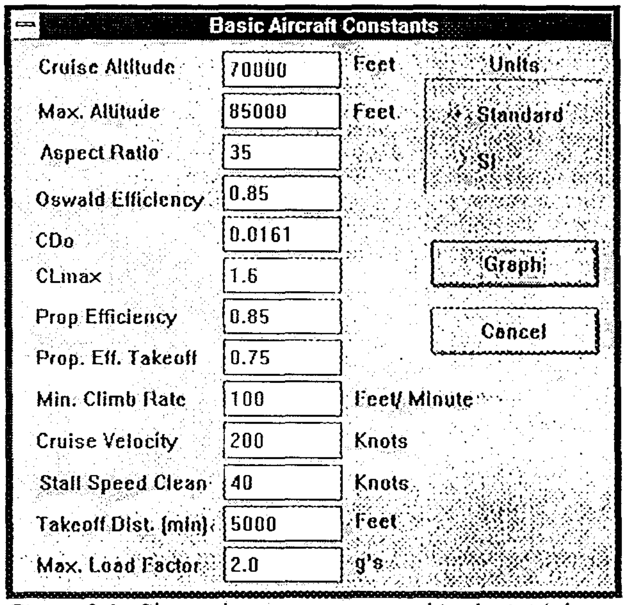

Figure 3.1 Shows the parameters used in the initial sizing of the aircraft and constraint diagram.

Upper cruise velocity will be limited by Mach number at altitude over the wing, and the lower cruise velocity will be bounded by stall at altitude. The maximum load factor, which as seen in Section 2.1 is based on the turn condition, or number of g's in gust loading will also normally be known. When all of this data has been entered, the constraint diagram is available.

3.2 Constraint Diagram

Figure 3.2a shows an example of a constraint diagram resulting from the above input parameters. It should be pointed out that the equations used for the constraint diagram were derived from basic physics and the relationships between drag, lift, thrust, and weight.

You can see from the figure that the upper limit of wing loading for the design point is based on stall for the data shown in Figure 3.1. The lower limit for Wing Loading is restricted by Cruise at a specified velocity at altitude, and the lower limit for thrust (HP) to weight ratio results from the maximum load curve.

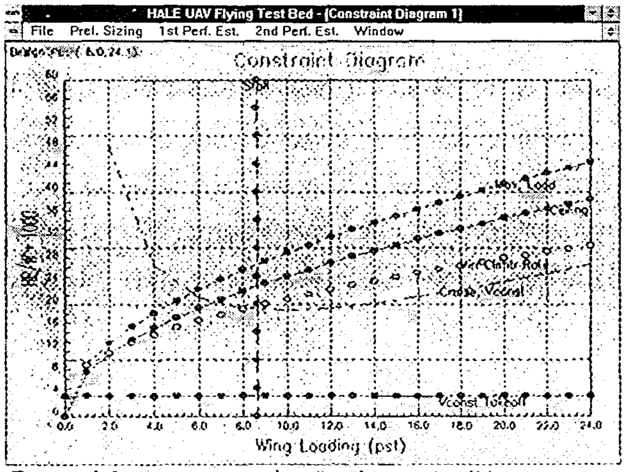

Figure 3.2a is an example of a Constraint Diagram resulting from data input in Fig. 3.1.

The code/methodology is configured so that the user may select the design point by clicking the mouse at the location on the graph that the design point is desired or by selecting the menu option and entering the design point manually into the dialog box. Alternatively, if the user is interested in determining how variations in any of the basic aircraft parameters affects the resulting Constraint Diagram, it is a simple matter to select the preliminary constants menu option and enter in new constants. This results in the opening of another new Constraint Diagram window. The two constraint diagrams can be tiled such that they may be viewed simultaneously to facilitate the comparison of the variations in the curves based on any changes made in the constants.

This process can be repeated until the user is content with the choices of basic constants (and consequently the mission requirements), and the resulting constraints.

It should be clear from observing Figure 3.2a what the constraining factors are for the mission for any given set of basic aircraft constants. By varying these parameters, it becomes even more clear to the designer which are the constraining design factors.

Additional information becomes available upon the selection of the design point. This information is now available to the designer and can be seen graphically in Figures 3.2b and 3.2c. In these figures, the designer can observe and experiment with the manner in which variations in gross takeoff weight affect the final wing planform area, and how variations in cruise velocity affect the final cruise lift coefficient.

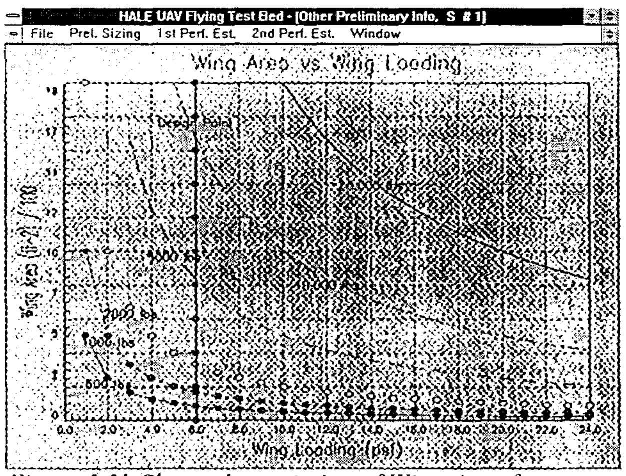

Figure 3.2b Shows the variation of Wing Area for a given Takeoff Weight providing useful feedback for the designer.

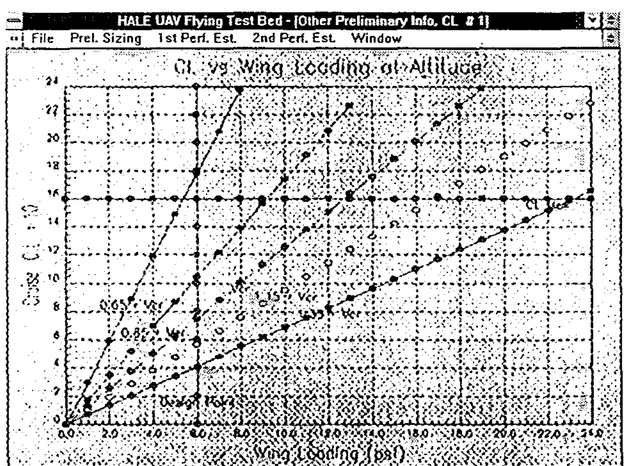

Figure 3.2c shows the variation of Lift Coefficient for a given Cruise Velocity.

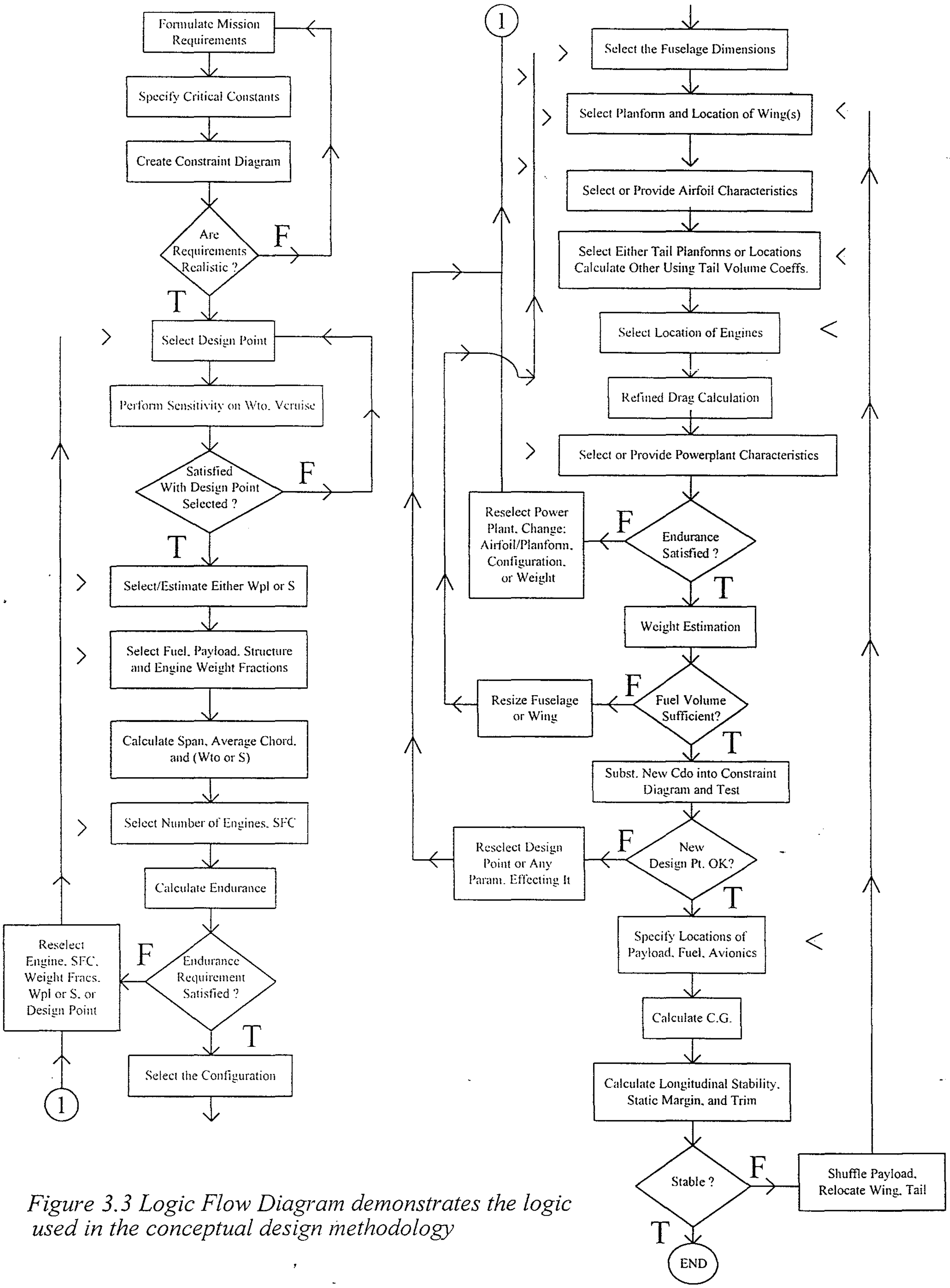

Figure 3.3 Logic Flow Diagram demonstrates the logic used in the conceptual design methodology

3.3 Logic Flow Diagram

Figure 3.3 is the Logic Flow Diagram for the present methodology. Since this paper represents a "work in progress", some aspects of the methodology may still change, but the general concept will remain the same.

As you look at Figure 3.3, you will notice that the first seven steps in the design process have already been discussed in the previous sections. Using the constraint diagram to determine which set of requirements are realistic has allowed for the selection of the design point. At this time a sensitivity analysis was presented for Wto and Vcruise. Then, if the designer is still satisfied with the design point, it is time to estimate values for either Payload Weight or Wing Planform Area. It is reasonable to assume that an estimate of one of these two values will be known based upon the assumed requirements of the mission.

3.4 Preliminary Sizing

The next step in the process is to select values for the weight fractions ($\frac{W}{W_{TOGW}}$) of the parameters specified in Figure 3.1. Although not many HALE UAV's have been built, there is enough information available to make sensible estimates for these values. It should be noted that these values are only used for the early preliminary sizing stages, and that they are replaced later in the conceptual design process with more detailed weight estimates.

The next step is to calculate the basic planform parameters. These values are only used for a gross estimate of the drag of the wing. This estimate along with the basic aircraft constants provided is used to verify that the desired endurance is satisfied at this early stage.

Before the endurance can be calculated though, the Specific Fuel consumption of the engine(s) must be specified, along with the number of engines. The Horsepower at which the engine must operate in cruise is calculated and presented to the designer (based on the design point selected and the Payload weight or Wing Area already specified), so that the fuel consumption can be specified for the conditions of operation of the engine. This provides the designer with the flexibility to estimate fuel consumption at a given Horsepower, or to use occasionally obtainable data from an engine manufacturer.

If the endurance is not satisfied, the designer can return for the reselection of *any* of the parameters entered thus far. Additionally, the designer is presented with the amount of time the aircraft has spent in climbing to the specified cruise altitude.

It should be noted that since the program is written in Borland Turbo C++ for Windows, that the code is "event driven". This means that it is highly modular, and at almost any juncture in the execution of the program, any of the modules can be invoked. What this means to the designer is that at any point in the design process, when it is discovered that changing of any one parameter is desired, there is nothing preventing the designer from going back several steps in the design process (and the program) and making that change within the same execution of the program. This feature would be impossible to include in a traditional FORTRAN code without overwhelming the designer with too many command line options.

3.5 More Detailed Sizing

Up to this point in the methodology, it was unnecessary to specify the configuration, by working under the assumption that the designer has some idea of the comparative drag of the various possible configurations to achieve the task.

The designer is now required to select a class of configuration from the window seen in Figure 3.5a, and subsequently a configuration type from a second window that opens when the configuration class selection is made.

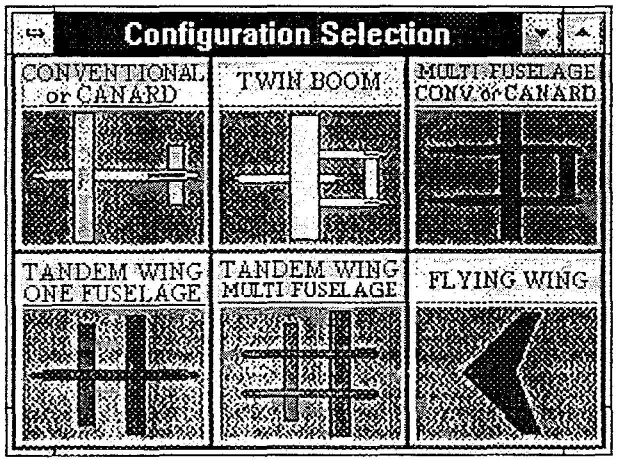

Figure 3.5a shows the Configuration Class Selection window.

Once the configuration class and type are selected, a default aircraft (complete with all of the major external features) is provided to the designer. Calculations are performed to estimate the fuselage length and tail size and location. A default location for the engine(s) for each configuration is specified based on the number of engines previously specified. In addition, a default airfoil is specified for the main wing and tailplane.

Any of the physical characteristics of this default aircraft can be changed by either clicking the mouse on the part of the aircraft of interest, or by clicking on a separate button dedicated to each part of the aircraft (see Figure 3.5b). This action produces a dialog box created specifically for that particular characteristic of the selected configuration. For more complex configurations, physical characteristics such as the distance between multiple fuselages or booms may also be specified. Figure 3.5c demonstrates the method used for modifying the default aircraft provided by the program, using the planform parameter dialog box as an example.

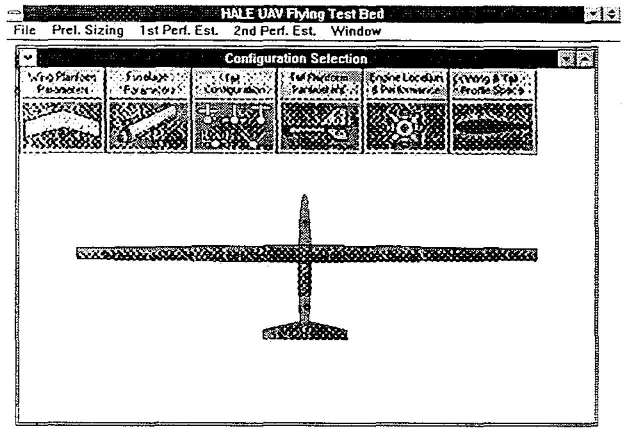

Figure 3.5b is a snapshot of the default aircraft and configuration selection window for the conventional configuration.

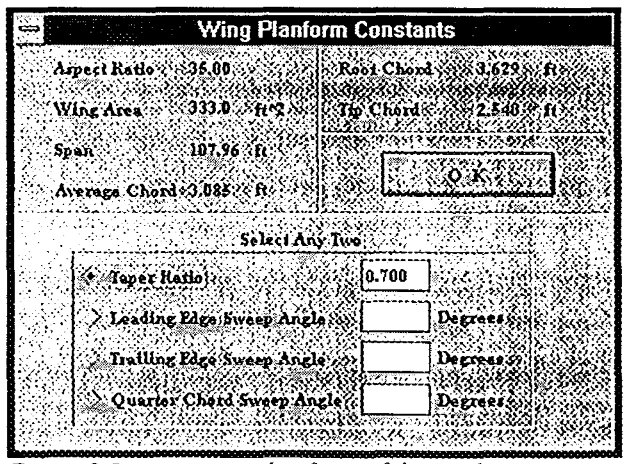

Figure 3.5c is an example of one of the configuration detail dialog boxes.

In addition, there is a dialog box to determine the fuselage length, fuselage cross sectional shape and dimensions, fuselage nose shape, fuselage tail shape, and fineness ratio.

For the tail, there are separate dialog boxes for tail configuration, as well as for all of tail planform and profile parameters.

Similar dialog boxes make it possible for the designer to supply an airfoil and engine performance data of the designer's choice. The airfoil dialog box allows the designer to specify their own lift coefficient versus drag coefficient curves from data stored in an external file. Once the filename where this data is stored is provided to the program, after the data has been read in, the program prompts the designer to determine whether or not to plot the curves immediately, or to return to the airfoil specification dialog box.

The engine performance dialog box allows for similar input of Specific Fuel Consumption versus Horsepower curve data for a given engine.

Specification of these curves allows for a greater degree of flexibility in executing the program, as the variation in airfoil drag with changes in lift coefficient can be compensated for by using the curves rather than prompting the user every time there is a change in the cruise lift coefficient. Similarly, it is not necessary to prompt the user for a new specific fuel consumption every time there is a change in cruise horsepower, as the data are available directly from the curves.

After entering this information, the program defaults back to the main configuration window. This enables the designer to enter or change any relevant information before moving on to the next step. When the user opts to close this window, a drag calculation is carried out.

4.0 Difficulties Encountered

All of the data necessary for a more detailed drag calculation are now in place. This presents the first real difficulty encountered in the present methodology. The most ideal method for drag calculation of low speed HALE UAV aircraft has yet to be determined. Included in this difficulty is the calculation of wing profile drag, wing planform drag, and downwash angles at the tail.

The primary difficulty arises from a total lack of any three dimensional data available for extremely high aspect ratio wings. Most data charts or tables cut off after an aspect ratio of 12. In the event of an assumed elliptical lift distribution, the drag due to lift, and downwash angles at the tail can both by calculated in a straightforward manner. In the instance when the lift distribution is even slightly more complex, this analytical solution is no longer available.

With regards to the profile drag, there is a growing body of information in the relevant low Reynolds number regimes. Since this body of information is continually growing, the code was written to enable the inclusion of this information as it becomes available. Thus, the designer is not constrained by any outdated or computationally intensive drag calculation techniques, and must only have available Lift versus Drag Coefficient data for the profile of choice.

In terms of calculating the drag due to lift, many methods would be appropriate, without being too computationally intensive. Lifting line theory, Vortex Panel and Vortex Lattice methods have all been considered, and further study is ongoing to determine which is the most appropriate for the lift induced drag calculation of the wing. The overall drag of the wing is expected to contribute 70% of the total drag of the entire aircraft.

The problems with using any of these methods arise from obtaining realistic values for drag with less conventional configurations. Any values obtained using these methods would have to be verified against either more detailed computational analyses or experimental results, both of which are not readily available for lower Reynolds numbers and alternative configurations.

After the drag of the aircraft is calculated, the endurance is presented to the designer again. If the value is satisfactory, then the second real difficulty emerges.

There is very little data on estimating wing weights for extremely high aspect ratio wings. There are very few data points, and their values also vary widely, so it is impossible to obtain a realistic estimate for a value of wing weight per unit area without performing more detailed calculations.

Values are known for the Rutan Voyager (0.5 psf) and the Boeing Condor(2 psf), both of which carried substantial fuel in their wings. In addition a reasonably sized database for glider wing weight is also available, but the Aspect Ratio of these wings is rarely over 22. The advantage of the glider wings, though, is that they can provide dry wing weight estimation for wings with similar aspect ratios.

Additionally, weight estimation techniques for some of the less conventional configurations are difficult if not impossible to find. Traditional parametric methods do not provide reasonable results.

5.0 Closing the Loop, the End of the First Iteration

Once the weight estimation is performed, the result of a modified drag calculation is substituted back into the constraint diagram and tested to determine if the power loading still satisfies the constraint curves based on the new value for drag. If not, the design point must be reselected and the process repeated.

If the design point still satisfies all of the constraints, the designer is now prompted to specify the location of the payload, fuel (a calculation of present fuel volume available in the specified location is made), and avionics in the fuselage/wing. The center of gravity is subsequently calculated and the longitudinal stability, static margin and trim conditions are calculated with the assistance of the downwash angles at the tailplane as calculated from one of the drag estimation techniques used.

Then if the aircraft is stable, the designer has successfully completed a single iteration of the design process. This will provide a wonderful basis for comparison to all of the other configurations/possibilities which may be created and evaluated alongside this one using the same execution of the program/methodology.

If the aircraft is unstable, then the designer can return to practically any point in the design process. The most logical place to return to in order to make the aircraft more stable, though, would be to the location of the various parts of the payload, or to shift the location of the main wing.

6.0 Conclusions - Future Work

It has been pointed out that the most major difficulties with the methodology lie with the aerodynamic prediction and the wing weight estimation. There is work ongoing to overcome these difficulties and complete the methodology, and the synthesis model. The engine performance data can be provided at any time for better estimation of aircraft performance.

Given the overall flexibility of the code, it provides the designer with enough information to make intelligent decisions at every step in the conceptual design methodology. The event driven program execution of the code allows the designer to break out of the design loop and change any parameter at virtually any point in the design process. In addition, the ability to compare the relative merits of many different configurations simultaneously, based on the what is essentially identical mission requirements, provides the designer with an exceptional opportunity to vastly improve their understanding of the driving parameters in the design.

Once the problems with wing weight estimation and aerodynamic prediction have been overcome, this code will enable a quick, visual, educational approach to the conceptual design of low speed HALE UAV's.

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