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Multidisciplinary Design Optimization of an Extreme Aspect Ratio HALE UAV

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Development of High Altitude Long Endurance (HALE) aircraft systems is part of a vision for a low cost communications/surveillance capability. Applications of a multi payload aircraft operating for extended periods at stratospheric altitudes span military and civil genres and support battlefield operations, communications, atmospheric or agricultural monitoring, surveillance, and other disciplines that may currently require satellite-based infrastructure. The central goal of this research was the development of a multidisciplinary tool for analysis, design, and optimization of HALE UAVs, facilitating the study of a novel configuration concept. Applying design ideas stemming from a unique WWII-era project, a "pinned wing" HALE aircraft would employ self-supporting wing segments assembled into one overall flying wing. When wrapped in an optimization routine, the integrated design environment shows potential for a 17.3% reduction in weight when wing thickness to chord ratio, aspect ratio, wing loading, and power to weight ratio are included as optimizer-controlled design variables. Investigation of applying the sustained day/night mission requirement and improved technology factors to the design shows that there are potential benefits associated with a segmented or pinned wing. As expected, wing structural weight is reduced, but benefits diminish as higher numbers of wing segments are considered. For an aircraft consisting of six wing segments, a maximum of 14.2% reduction in gross weight over an advanced technology optimal baseline is predicted.

I. Introduction

Development of High Altitude Long Endurance (HALE) aircraft systems has long been part of a vision for a low cost communications/surveillance capability [1, 2, 3, 4]. Applications of a multi payload aircraft operating for extended periods at stratospheric altitudes span military and civil genres and support tactical battlefield operations, communications, atmospheric monitoring, precise agricultural and wildfire monitoring, surveillance, and other disciplines requiring satellite-based infrastructure or high resolution imagery [5]. Currently, the Defense Advanced Research Projects Agency (DARPA) is requesting proposals for an aircraft that can sustain flight for multiple years and act as a pseudo-satellite for intelligence, surveillance, and reconnaissance missions [6]. Design of this and any type of air vehicle represents a substantial challenge because of the vast number of engineering disciplines required for analysis. In addition, some tools and analysis methods used in the design of aircraft with more conventional missions may not be applicable to certain types of HALE vehicles. In the modern competitive environment surrounding the manufacture of aircraft systems, oftentimes simply meeting the customer's requirements may not win a contract. Instead, the proposed system must also represent the optimum vehicle for the customer needs [7]. This focus on finding an optimal solution places some additional requirements on the design process itself.

Searching for an overall optimal solution involves broadening the trade space and allowing a large number of variables. These high degree of freedom environments are not handled well by a sequential design process [8]. Also, with highly multivariate design spaces, analyzing the sensitivities to each variable individually and relating this information to a whole system sensitivity is a daunting task. One method for mitigating many of the challenges associated with designing complex aeronautical systems is to compile the individual disciplines and analysis methods into one environment, allowing for better organization of data flow, and more efficient communication. This may be accomplished on a small scale by simply bringing codes together on one machine, or in a larger sense by allowing physically separated flight science groups to wrap their analyses for remote use. Once assembled, multidisciplinary analysis, design, and optimization techniques can be applied in the hopes of allowing more broad design spaces and providing a clearer view of system drivers and sensitivities.

With an integrated HALE design environment in place, it is possible to perform parametric studies to investigate areas of potential improvement over current concepts. In essence, we are looking for the active constraints on the design, or the design drivers. Wing design and propulsion systems are the two main aspects of HALE vehicles that

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are driven by mission requirements, and consequently present the greatest opportunity for system improvements. Accordingly, when considering new or revolutionary design concepts, these two areas should be of primary focus. Generally, HALE aircraft of the past and present, like the *U2, Helios,* or *Global Hawk* exhibit high aspect ratio wings that allow the aircraft to achieve the altitudes of interest. Sacrifices are made, however, because with a high aspect ratio planform comes high wing bending moments, unfavorable dynamic structural responses, and large deflections. In the propulsion arena, many current designs feature distributed propulsion, advanced propeller design, and a strong coupling between propulsion and flight controls. Closely related to propulsion is the energy source for the aircraft. Most modern internal combustion architectures cannot satisfy the persistent operation requirements of current HALE missions like *Vulture* or the Communications Relay posed by the AIAA in 2007 that stipulate months to years of continuous flight [9]. As a result, much effort has been devoted to development of environmental energy collection, high energy density storage devices, and other alternative energy concepts [6].

The intent of this research is to evaluate revolutionary changes to HALE aircraft architecture, using state of the art propulsion and energy concepts while breaking new ground for wing design. Wing aspect ratio is the primary characteristic of wing design for the purposes of this study, and the NASA / AeroVironment *Helios* aircraft set the current threshold for demonstrated all-electric flight with an aspect ratio of 31. With a new wing concept, it may be possible to push this envelope of high aspect ratio platforms, while simultaneously mitigating problems associated with highly flexible aircraft structures. An integration of architecture-independent design codes into an optimization environment enables identification of constraints that emerge when exploring extreme-aspect-ratio concepts. These constraints take the form of structural and energy requirements such as max stress or minimum specific energy storage density, as well as mission operation requirements that take into account things like available runways and hangers for aircraft with extremely long wingspan. One goal of this paper is to find the area of diminishing returns for wing aspect ratio if such behavior exists, and discuss why and how certain constraints become active. In addition, the work diverges from combustion-based sizing methods and focuses on generalizing the design process for energy-optimized systems and all-electric aircraft.

II. Extreme Aspect Ratio Concept

Developing a revolutionary concept for increasing aspect ratio without paying the penalties commonly associated with doing so is primarily inspired by a somewhat obscure Air Force research effort in the early 1950s that was itself inspired by German scientist Dr. Richard Vogt who emigrated to the U.S. after WWII. The initial concept was that the range of a bomber may be increased by adding “free-floating” fuel carrying wing segments that are pinned to the bomber wingtips [10]. Also, the U.S. military wanted to examine the feasibility of utilizing the long range capabilities of bombers like the B-36 Peacemaker or the B-29 Superfortress to tow, carry, or otherwise transport smaller and more maneuverable fighter aircraft like the F-84 to foreign combat zones. The first concepts involved the smaller aircraft docking in the bomb bay of the bomber for parasitic flight to and from the target. In this scenario, the extra aircraft adds weight and parasitic drag to the bomber, while taking up bomb bay space, and not contributing anything aerodynamically positive in return [11]. A follow-on effort designated MX-1016 “Tip Tow” moved the parasite fighter from under the host aircraft to the wingtip [10]. Figure 1 shows a Boeing B-29 in flight with two EF-84 aircraft coupled at the wingtips [12].

Figure 1 - Early Air Force Wingtip Coupled Flight [12]

With the new configuration, the parasitic aircraft now contribute additional wingspan to the bomber, reducing the induced drag [13]. The wingtip coupling mechanisms underwent some revision under a new project called "Tom-Tom" involving clamps or jaws on the wingtips of a B-36 [12].

Applying the idea of wingtip coupling to a flying wing HALE aircraft represents a revolutionary step in the field. Conceptually, each segment of the flying wing would lift its own weight and comprise a generally self-sufficient aircraft system. Stringing some number of individually moderate aspect ratio wing segments together to form an extreme aspect ratio platform allows each segment to benefit from lower structural loads while simultaneously reaping benefits of an extremely high aspect ratio planform. Figure 2 shows what one such vehicle might look like.

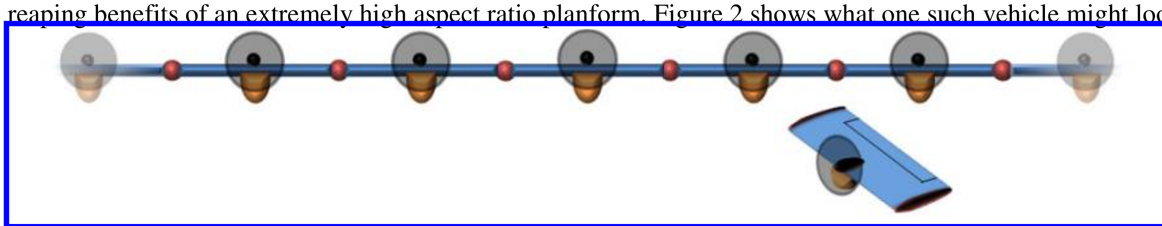

Figure 2 - Extreme Aspect Ratio Concept

In addition to lowering the structural weight fraction of the aircraft, a "pinned wing" concept may offer the ability to orient solar cells favorably towards the sun. Also, with a conventional high aspect ratio flying wing configuration, natural frequencies of the structure can be so low that they approach control response frequencies. If these come too close together, an aileron deflection, or step input may induce structural resonance rather than the desired change in flight condition. Considering that the individual wing segments of the extreme aspect ratio concept have low-to-moderate aspect ratios, they will be more rigid and exhibit higher natural bending mode frequencies. For this study, however, the dynamic behavior of the coupled system is undetermined.

III. Problem Statement

The central goal of this research is the development of a multidisciplinary tool for analysis, design, and optimization of High Altitude Long Endurance (HALE) UAVs. Current and projected future missions for this type of aircraft platform focus on its ability to provide sustained support for surveillance, communications, or other science missions, and act as an "atmospheric satellite". Accordingly, the baseline mission profile consists of a climb to some stratospheric cruise/loiter altitude, where the aircraft begins mission operations and enters an extended cruise or loiter flight mode, shown in Figure 3.

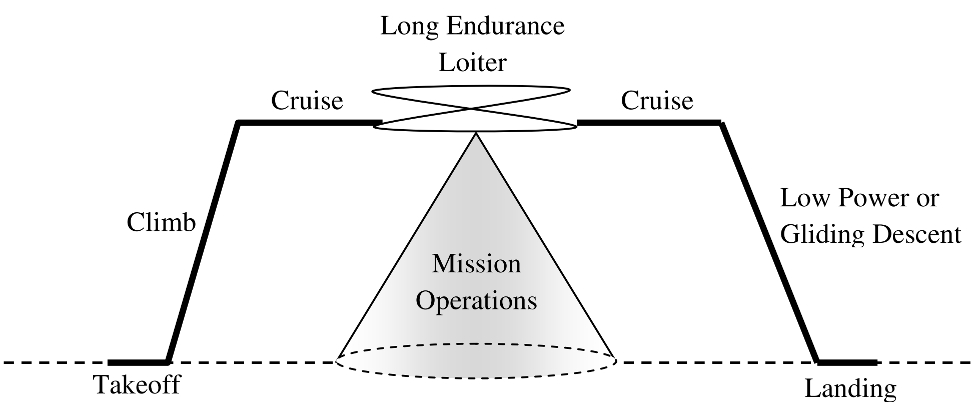

Figure 3 - HALE UAV Mission Profile

With an integrated design and analysis environment in place, certain parameters may be adjusted to approximate current or past proven aircraft configurations in an effort to calibrate the tool. Next, optimization techniques are applied to a baseline platform, here represented by flying wing aircraft similar to *Helios* or the *Vulture* concept. Investigation of performance trends and the effect of new technologies, as well as system sensitivities from parametric studies may be performed at this stage. Considerations are made throughout the development of the tool to allow a wide range of missions, payload systems, and potential solutions.

Following the analysis and optimization of a proven platform aircraft, the design tool is adjusted to study the effect of implementing a segmented wing concept. Again, the individual flight science modules, or discipline analysis codes are developed to be flexible and applicable to both single and multiple segment wings. As before, parametric studies and optimization of the segmented platform reveal design drivers and best configurations, which may be compared with the results from the proven aircraft study, as well as actual flown aircraft results.

One of the attractive aspects of a segmented wing platform is the ability to build it up from identical and self sufficient segments, allowing more flexibility for payloads and missions. As such, the optimization objective is to develop a platform of identical segments that minimizes aircraft weight while meeting mission, performance, and technology constraints. In reality, the goal is minimization of total cost, which may be composed of both acquisition and operating cost. For the included studies, aircraft cost is assumed to be well represented by aircraft weight, as suggested in [14]. Also, in a report prepared for the Suborbital Science Office Earth Science Enterprise of NASA, it is proposed that a breakthrough in reducing acquisition cost of UAV science missions may be possible with a new generation of small HALE aircraft with simplified operational requirements [15].

IV. Methodology

Development of the multidisciplinary analysis, design, and ultimately optimization tool begins with a buildup of individual disciplines, each capable of analyzing both a baseline flying wing/cantilever configuration and a segmented wing platform. What follows is a description of the approach to optimization, and a look into the integrated multidisciplinary environment.

A. Optimization Architecture

With aircraft weight as the central objective for minimization, parameters describing the aerodynamic, structural, propulsive, and energetic qualities of the aircraft are varied to determine preferred configurations that meet applicable constraints. As the main body of code was developed in MATLAB, the built-in optimizer "fmincon" is used to minimize a constrained multivariate function. The general form of the optimization problem is given in (1).

$$\min_{\vec{x}} f(x) \text{ such that } \begin{cases} c(\vec{x}) \leq 0 \\ ceq(\vec{x}) = 0 \\ lb \leq \vec{x} \leq ub \end{cases} \tag{1}$$

Where *c* represents a set of inequality constraints, *ceq* represents the equality constraints, supplemented by *lb* and *ub*, the lower and upper bounds enforced on the set of design variables $\vec{x}$.

When performing an optimization, fmincon defaults to attempt to use a trust-region-reflective algorithm, which requires a user supplied gradient for the objective function. Developing an analytical gradient for the entire design process for an aircraft from conceptual design through mission analysis and preliminary sizing is both difficult and beyond the scope of the work herein. In addition, several of the analysis modules utilize pre-compiled binaries or executables, further discouraging any attempt to find the gradient. Instead, an optimization algorithm must be used that numerically estimates gradient and Hessian functions. The Optimization Toolbox in MATLAB offers Active-Set optimization for problems such as this. For Active-Set optimizations, MATLAB implements sequential quadratic programming (SQP) to choose search directions by mimicking Newton's method [16]. Sequential Quadratic Programming approximates the objective function, generally $W_{tot}(\vec{x})$, as a quadratic function. The method then linearizes constraints locally and applies Quadratic Programming to approximate the solution. SQP looks to the Broyden-Fletcher-Goldfarb-Shanno (BFGS) method for updating the Hessian [17].

Fundamentally, the process employed for designing the UAV is an iterative process, meaning that from a systems perspective, there is feedback inherent in the data flow structure. Specifics about the feedback quantities are discussed in the next section, but how they are handled effects the optimization environment. Two methods were considered and tested for solving the resulting system of nonlinear equations. The first is a simple iterative scheme that converges all of the system feedback given initial guesses for each feedback quantity. This scheme provides an intuitive method for solving a system, but no guarantee that a solution exists. An attempt was made at developing a convergence criteria using Newton's Method, but an analytical representation of the whole system is not feasible.

The alternative to FPI is a process called Optimizer Based Decomposition (OBD) where data links in the system of equations are broken and replaced by new design variables and constraints [18, 19]. Specifically, for the problem of interest here, only the feedback data links are decomposed, and we designate the process Partial OBD (POBD). Allowing the optimizer to simultaneously handle the regular design variables and the requirement for a converged system decreases the run time per iteration and increases the probability of closing the design, or achieving convergence. In addition, the constraints on convergence may be held strict or loosened depending on the desired fidelity of the solution. A simplified and purely theoretical system is presented in Figure 4 to illustrate the interaction between disciplines. In this system representation, active column elements are inputs to a module and rows are outputs. Reference [20] gives a thorough description of the processes for using a diagram like this, but completing a quick system trace reveals that active cells in the lower triangle represent feedback data paths. A POBD process operates on these cells to eliminate the need for iterative solving.

Geom.	X		X		
	Aero	X			X
	X	Mission Analysis	X	X	X
X	X	X	Weights	X	X
				Cost	
					Perf.

Figure 4 - Theoretical Aircraft Design Structure Matrix

A more detailed Design Structure Matrix (DSM) is presented in the next section, and represents the actual multidisciplinary system implemented for this study.

Design Variables

Parameters selected as design variables for this optimization are products of the approach to conceptual design, mission analysis, and preliminary aircraft design methods employed. As a quick preview, several key characteristics of the aircraft are specified up front, and an iterative design process (hopefully) converges to a final configuration. A complete list of inputs to the design environment comprises the set of all potential design variables, only some of which are actually selected as design variables for the optimizer, while others represent technology factors, mission characteristics, or configuration identifiers. The design of experiments for a complex multivariate optimization problem such as this begins with selecting a basic set of design variables, leaving the rest as constants that define aspects of the configuration. Table 1 shows the basic set of design variables used.

Table 1 – Initial Optimizer Design Variables

	Design Variable
1	Total aspect ratio
2	Wing loading (lb/ft^2)
3	Power to weight (W/lb)
4	Wing thickness-to-chord ratio
5	Percent of S_{ref} covered by solar panels

Along with the fundamental design variables of Table 1, there are several other parameters of the design which may be more effective as optimizer-controlled design variables than predetermined quantities. Table 2 shows these parameters, including the variables necessary for implementing POBD.

Table 2 - Additional Design Variables (* = OBD variable)

6^*	Total weight (lb)
7^*	Wing weight (lb)
8	Cruise Altitude (ft)
8	Payload weight (lb)
9	Payload power requirement (W)

Lastly, there are several characteristics of the aircraft that are discrete numbers. The optimization methods employed do not handle such data types, so when designing the experiments, each of the variables in Table 3 must be specified for a set of parametric studies or optimization runs. Results may then be compared between the discrete values to assess potential benefits against additional complexity.

Table 3 - Discrete Variables

Number of wing segments
Number of battery/payload pods
Number of spar cross sections

Optimizer Constraints

Equation (1) states that our system may be subject to either inequality or equality constraints which may act on the design variables themselves or certain determined quantities within the system. Decisions about which metrics to use as constraints are a bit more vague with a HALE UAV platform than with aircraft designed for more conventional missions. Take, for example, the recent Broad Agency Announcement delivered by DARPA requesting proposals for a HALE UAV. The requirements supplied by DARPA are simple and few [6]:

- 5 years uninterrupted operation
- 1000 lb, 5 kW payload
- 99% probability of station-keeping
- High probability of mission success

From a preliminary design point of view, the first two bullets are the only requirements. Where other categories of aircraft may have a set of point performance requirements explicitly laid out for them, in our case the designer must work diligently to flow down these two top-level requirements into subsystems requirements and ultimately design constraints.

Considering another recent HALE platform that broke new ground for its kind, a general goal for qualifying as 'high altitude' can be defined. The high altitude configuration of *Helios* (HP01) was designed with the intent of demonstrating flight at 100,000 ft.

From these reference programs, the central desired capabilities of future HALE UAVs is distilled into the first two constraints imposed on the optimization environment herein. First, we impose the requirement that the all-electric aircraft must achieve a sustainable energy balance for repeatable day/night operation. Persistent multi-day operation necessitates the inclusion of both energy generation and energy storage systems. This first constraint requires that for a given flight profile, the aircraft must be able to generate enough power during daytime operation to not only sustain flight and payload operations, but to do so with enough excess energy produced to power the aircraft through the night. In addition, the aircraft must be able to support the weight of a system capable of storing this excess energy, along with any associated power management systems. When considering batteries as the storage medium, current technologies result in as much as 30-50% of the total aircraft weight taken up by energy storage, meaning that the persistent operation requirement is a major design driver.

Supporting the payloads of interest for programs like *Vulture* or *Helios* requires stratospheric flight altitudes, and as previously stated, a good benchmark for future platforms is flight at 100,000 ft altitude. This becomes the second constraint imposed on the design environment, stating that the absolute ceiling of the aircraft must be at least 100,000 ft.

In order to keep the solutions controlled to a reasonable domain, a third constraint is imposed that defines a maximum wingspan. Initial studies showed that without this constraint, optimal configurations sometimes exhibited wingspans of nearly 500 ft, almost twice that of the Airbus *A380*. A constraint value of 300 feet is used for the majority of the study herein, and was chosen to be similar to the *Helios* aircraft with some room to grow.

The fourth and fifth constraints are products of the POBD of the design system. As implemented, two feedback variables have been offloaded onto the optimizer: total weight and the structural weight of the wing. For each function call, the optimizer provides initial guesses for these weights, allowing the design environment to calculate dimensional values for things like required energy or power, wetted area, solar panel area, etc… In addition, the structures module must account for the weight of the wing when sizing the spar. This catch-22 of needing a guess of structural wing weight in order to calculate the structural wing weight characterizes the feedback loop that was decomposed. Accordingly, convergence of the design is enforced by imposing equality constraints that require the calculated wing structural weight and total aircraft weight be within a certain tolerance of the guessed values. Table 4 summarizes the fundamental constraints imposed on the design and optimization environment. These constraints remain the same whether implementing single segment baseline configurations or multi-segment XAR concepts.

Table 4 - Fundamental Design Constraints

	Constraint	Type
1	Multi-day energy balance	Inequality
2	Absolute ceiling	Inequality
3	Wingspan	Inequality
4	Total weight compatibility	Equality
5	Wing structural weight compatibility	Equality

The HALE UAV optimization problem is expressed in standard form below to provide a general summary of the variables, constraints, and objective. When implemented, design variables, objective function values, and constraint values are all individually linearly scaled to have a magnitude on the order of 10^0. This process is important because it helps to ensure well-conditioned Lagrange multipliers used in evaluation of constraints under methods like Kuhn-Tucker (KT) conditions [17].

$$minimize \quad W_{tot} = f(\vec{x})$$

$$where \qquad \vec{x} = \begin{cases} Aspect\ ratio \\ Wing\ loading \\ Power\ to\ weight \\ Wing\ t/c \\ \%\ solar\ coverage \\ W_{tot} \\ W_{wing} \\ Payload \\ Technology \\ etc\ ... \end{cases}$$

$$subject\ to \qquad \frac{E_{batt\_req} + E_{payload} + E_{systems} - E_{solar}}{mag(E)} < 0$$

$$1 - \frac{h_{absolute\_ceiling}}{100{,}000} < 0$$

$$\frac{b}{300} - 1 < 0$$

$$\frac{\left|W_{tot\_guess} - W_{tot\_calc}\right|}{mag(W_{tot})} = 0$$

$$\frac{\left|W_{wing\_guess} - W_{wing\_calc}\right|}{mag\left(W_{wing}\right)} = 0$$

Lastly, the tool includes many parameters that may be rearranged to alter the optimization problem. These additional characteristics, when implementing the optimization described above, are either set as inputs to the system, or determined as outputs. However, with minor adjustments, the aircraft may be optimized for a different objective, and/or subject to alternative constraints. For example, additional constraints may be set for the number of motors, or the solar cell efficiency or energy storage density may be introduced as design variables and objectives for minimization.

B. Multidisciplinary Integration

Development of a multidisciplinary analysis, design, and optimization tool begins with the identification of which disciplines are involved, and what inputs and outputs are associated. Much of the aircraft design process involves coupled systems, feedback, and indirect dependencies that pose significant challenges to analytical modeling or sequential design. There are many approaches to initial concept design, sizing, and weight estimation for an aircraft, but many of the traditional methods have significant shortcomings when applied to the systems of interest here. The majority of classical preliminary design methodologies have three central tasks: point performance analysis (constraint diagram), mission analysis, and weight estimation (Figure 5).

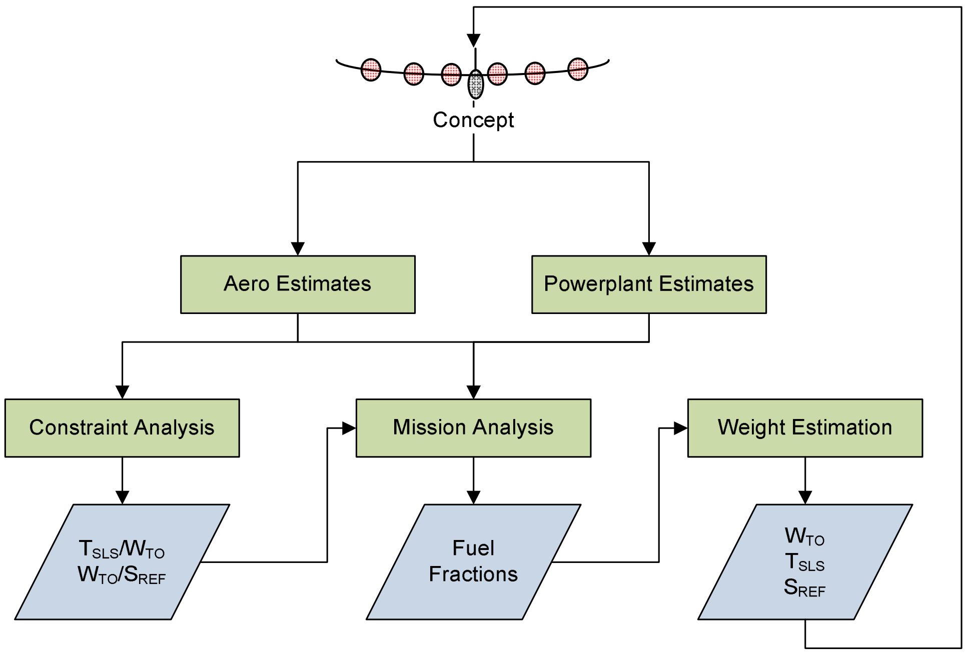

Figure 5 - Classical Aircraft Sizing Process

When considering a revolutionary concept such as an all-electric HALE UAV under this sizing architecture, several problems arise. First, new propulsion and energy systems which depart from the internal combustion arena will surely beget unconventional configurations as seen with the development of the AeroVironment family of vehicles under the NASA Environmental Research Aircraft and Sensor Technology (ERAST) effort that led up to the Helios Prototype [21]. Sizing an aircraft according to Figure 5 requires some knowledge of weight trends, generally in the form of historical regressions or expert opinion concerning empty or structural weight fractions. Care must be taken in selecting these parameters, but with resources describing structural optimization of HALE aircraft, conventional sizing methods may still apply [1, 22, 23]. Alternatively, structural weight estimation may be achieved using more complex physics-based tools that find worse-case loading situations and size the major structural elements of the aircraft accordingly. This "bottom-up" method for estimating empty weight fractions is employed in the final MDO tool and is described later in this section.

The second, and more pronounced problem sizing an alternative fuel aircraft with the model of Figure 5 is the mission analysis. Currently, the traditional sizing algorithm requires a portion of the aircraft to "burn up" during the mission in the form of fuel weight; if the aircraft has no combustion cycle and consequently completes the mission with no weight change, the process of Figure 5 breaks down. This inflexibility to alternative methods of converting energy to power is the motivator for developing a new sizing process with one fundamental difference. Our new method of initial design will focus more directly on the energy of the aircraft without inherently selecting the form that energy occupies. For example, the mission analysis of our new design methodology does not calculate the fuel fraction required for climb. Instead we develop the total energy requirement for the climb (actually the total energy normalized by aircraft weight). At the completion of our mission analysis we will have developed the total mission specific-energy requirement represented in units of energy per pound of aircraft. This approach allows us to then apply any set of energy sources to the airframe including but not limited to batteries, fuel cells, photovoltaic generation, and conventional internal-combustion based power plants. A similar approach was taken in developing an Architecture Independent Aircraft Sizing Method (AIASM) [24], and is supported by power system analysis given in [25].

With a sizing method applicable to electric-powered aircraft and a design perspective centered on energy, the foundation of a valid conceptual design environment has been laid. Building the MDAO capability around our new approach to sizing follows as it would for any optimization environment. A functional decomposition, or multilevel breakdown, of the aircraft system leads to the central disciplines that will be involved in the design process [8]. Figure 6 shows the specific areas of analysis that comprise the MDAO environment in the form of an N-squared diagram. As pictured, the analysis modules have been arranged to minimize feedback, though it is still present. Feedback in the system is represented by links in the lower triangle of the matrix. As previously mentioned, these areas of feedback are disconnected and the requirement for design convergence is enforced by the controlling optimizer.

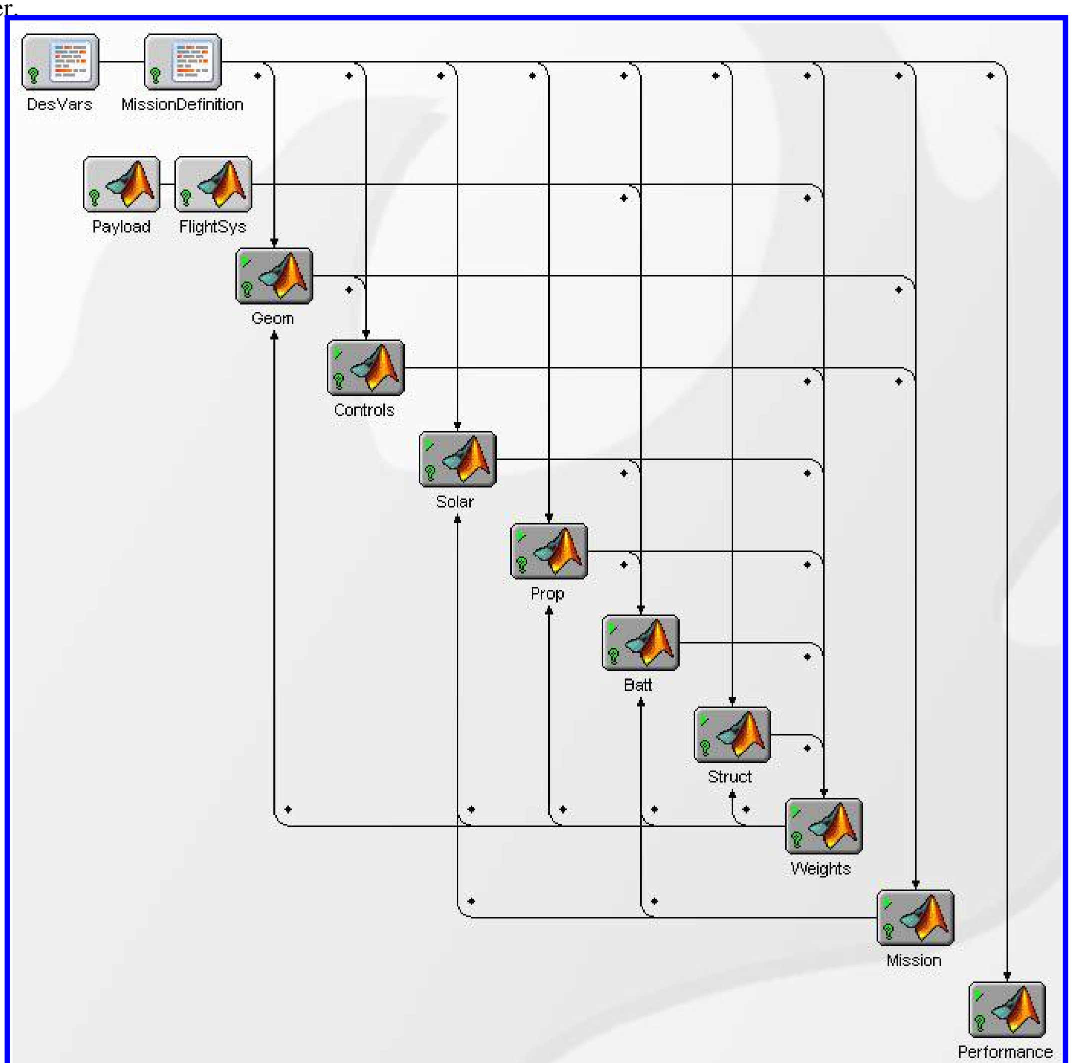

Figure 6 - Multidisciplinary Design Analysis and Optimization Architecture

V. Optimization Results

A. Single Wing Segment Aircraft Baselines

Optimal configurations have been found for two different sets of initial assumptions. First, a platform using somewhat outdated technology to perform a medium endurance mission was developed. Based on the *Helios* aircraft of the 1990's, this first baseline was optimized for roughly 15 hours of mission endurance, and is shown as the green aircraft (II) in Figure 7. The second aircraft configuration used advanced technology and was optimized for multiple-day missions. Shown below in red (Figure 7 III) this aircraft achieves a sustainable 24-hour energy balance.

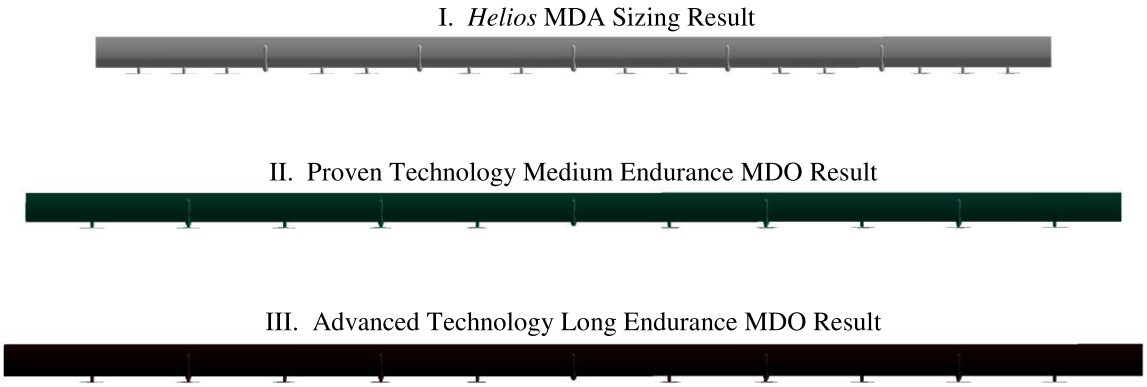

Figure 7 - *Helios* Proxy Alongside Two Optimal Baselines

Note that the grey metallic aircraft (I) is not simply a CAD model of *Helios*, but is the resulting geometry from the MDA environment when the aspect ratio, wing loading, power to weight ratio, and wing thickness are set to *Helios* values and the tool is tasked with sizing the aircraft.

B. Additional Wing Segments

Rather than presenting specific results one at a time for each case, results are discussed here comparing the set of two to six wing segments. Optimum configurations for the various pinned wing platforms are all selected as the minimum weight solution from a large set of optimization runs. The weight breakdowns for the best solutions are compared in Figure 8.

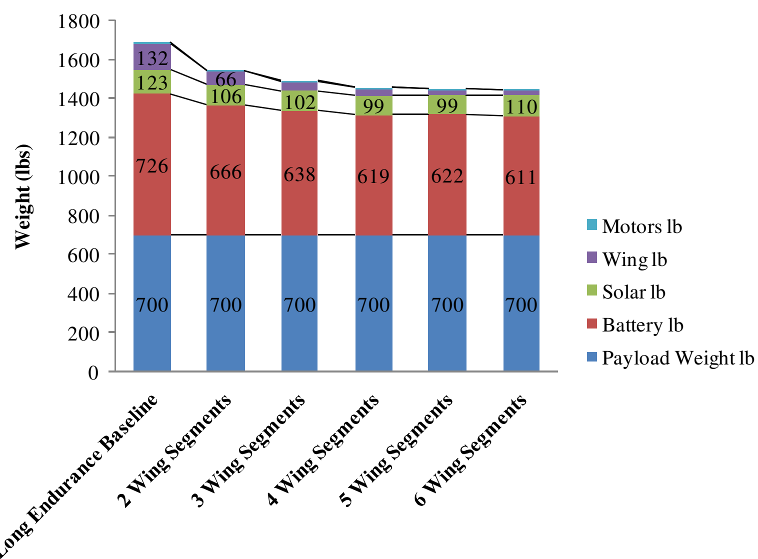

Figure 8 - Weight Breakdown Comparison of Multiple Wing Segment Configurations

The most noticeable aspect of this chart is that with an increasing number of wing segments, the weight of the optimal aircraft converges to similar values and benefits of additional segments disappear. Of all the parameters and characteristics of each optimal aircraft configuration, one of the more interesting ones to explore when looking at increasing the number of wing segments is the wing thickness. We expected that increasing the number of segments would decrease the structural loading on the spar, and Figure 8 confirms this. What happens with the higher number of segments is that the wall thickness of the spar that is required to support the loads decreases to the minimum allowable value of 0.005 inches. The aerodynamic analysis process employed is sensitive to wing thickness and higher thickness results in a higher C_{Do}. Since wing thickness is a design variable, and spar wall thickness is constrained by a lower bound, the optimizer decreases the wing thickness to chord ratio to reduce power required, battery weight, solar weight, etc. A thinner wing results in a smaller spar cross section and increased principal stresses, so we imagine that there exists a Pareto front of low spar weight and low wing thickness. One segment designs showed a tendency toward high thickness because the aerodynamic benefits of a thin wing were outweighed by the large increases in structural weight. Figure 9 tracks the wing thickness for different numbers of wing segments, and we see the upper bound of 30% thickness to chord ratio for a single segment wing. Also apparent is the ability of the optimizer, when analyzing a multiple segment configuration, to decrease the thickness some amount before seeing the structural penalty. Compared to the single segment, t/c is halved for two segments.

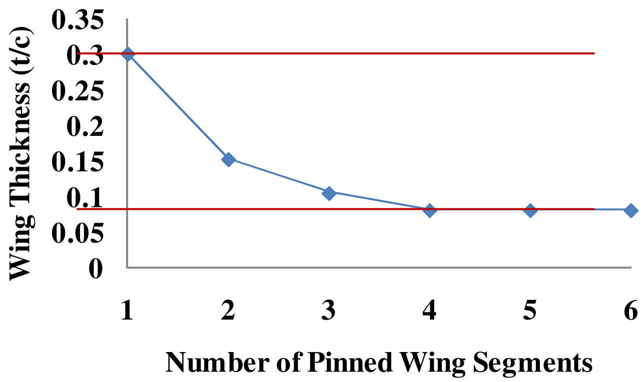

Figure 9 - Optimal Wing Thickness for Multiple Wing Segments

Next, we look the upper and lower bounds for the thickness to chord design variable as red lines on Figure 9. It seems that for more wing segments, the lower structural loads facilitate thinner and thinner wings, eventually encountering the lower bound on the variable. Keep in mind that Figure 9 actually represents the wing thickness as a percent of chord length, and that we haven't quoted any actual maximum thickness values. This means that the optimizer may have arrived at a minimum thickness to chord ratio, but can still further decrease the actual thickness value by decreasing the chord. Figure 10 shows that if the aircraft has more than 4 wing segments, there is a sharp decline in chord length, and with the wingspan at the upper limit of 300 ft for all cases, this translates directly to a sharp decrease in S_{ref} (Figure 10).

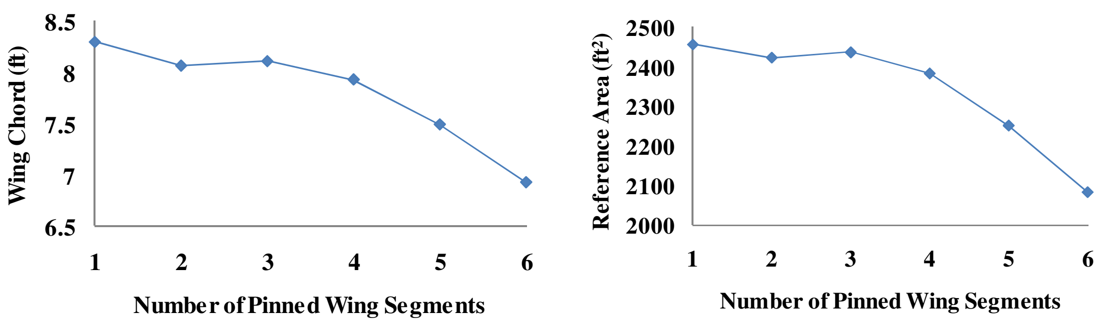

Figure 10 - Optimal Wing Chord, S_{ref} for Multiple Wing Segments

Another result of the change lower bound on thickness to chord ratio becoming active is apparent in the percent of the wing requiring solar cell coverage. The actual area of solar cells for each case is generally similar, so decreasing the wing area simply means that a greater percent of the wing must be covered (Figure 11).

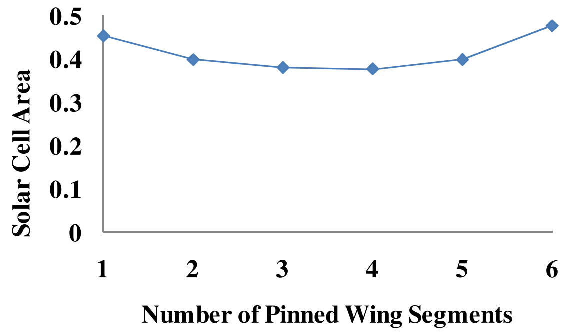

Figure 11 - Optimal Solar Coverage (Percent)

While Figure 8 shows slight improvements in structural weight between 3, 4, and 5 segments, the reality is that the gross weight of the aircraft is nearly the same (Figure 12). When reference area is decreased for an aircraft with a given weight, the result is a higher wing loading, seen in the graph to the right in Figure 12.

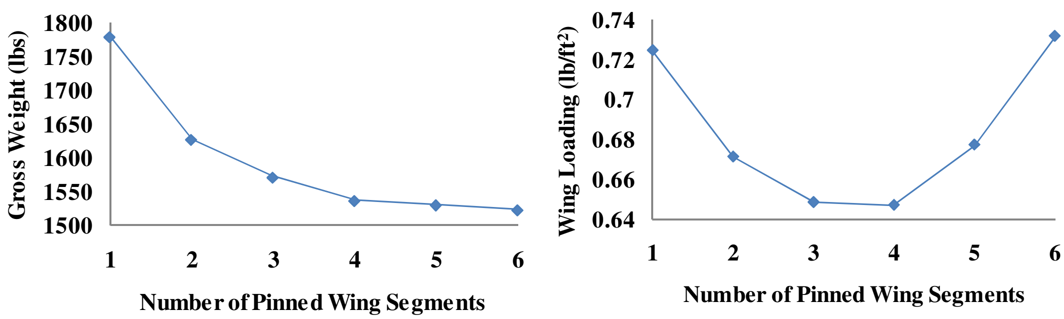

Figure 12 - Optimal Gross Weight, Wing Loading for Multiple Wing Segments

We might expect that higher wing loading would increase required power, battery weights, etc, but the results in Figure 8 and Figure 12 do not show this happening. In fact, the optimizer has increased the aspect ratio, made possible within the wingspan constraint because of the decreased chord length for higher numbers of wing segments. Figure 13 tracks aspect ratio for each configuration.

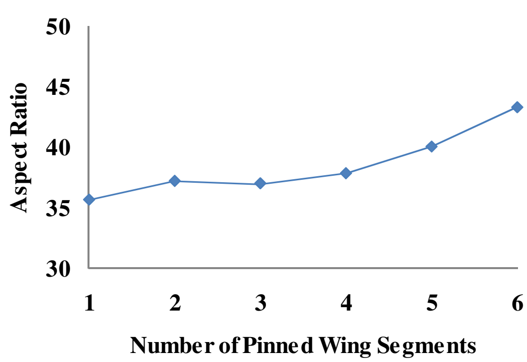

Figure 13 - Optimal Aspect Ratio for Multiple Wing Segments

VI. Conclusion

The central goal of this research was the development of a multidisciplinary tool for analysis, design, and optimization of HALE UAVs, facilitating the study of a novel configuration concept. Applying design ideas stemming from a unique WWII-era project, a "pinned wing" HALE aircraft would employ self-supporting wing segments assembled into one overall flying wing. The research effort began with the creation of a multidisciplinary analysis environment comprised of analysis modules, each providing information about a specific discipline. As the modules were created, attempts were made to validate and calibrate the processes against known data, culminating in a validation study of the fully integrated MDA environment. Using the NASA / AeroVironment *Helios* aircraft as a basis for comparison, with generalized *Helios* payload and mission data, the included MDA environment sized a vehicle to within 5% of the actual maximum gross weight. Because certain mission specifics for *Helios* were unclear or non-specific, like battery weight or cruise profile, several sensitivity studies were completed. In addition to showing correct trends and system responses to certain variable perturbations, these studies identified some of the stronger drivers of aircraft weight, like energy storage technology or mission definition (cruise altitude and endurance). With reassurance that the tool provides reasonable results and correct trends, the whole process was wrapped in an optimization environment for further study.

Because of the parametric approach to the problem at hand, comparison of optimum results is focused on evaluating the change from some baseline configuration. First, the *Helios*-based validation case was re-posed as an optimization problem to see if any improvements were possible. Here, improvements may be in the form of reduced gross weight, or enhanced mission capabilities. For *Helios* technology and a medium-endurance mission (14-15 hours) the MDO tool shows a 17.3% reduction in gross weight, largely due to a thicker wing for structural weight reduction, and an optimized planform for low power flight, reducing battery weight. Next, more advanced technology parameters were used with the *Helios*-style aircraft to perform a long-endurance mission with requirements similar to the DARPA *Vulture* program. With 10% more efficient solar cells, a 92% net power train efficiency, and advanced batteries, the aircraft was able to achieve day/night operation at a gross weight nearly identical to the optimized medium-endurance configuration. All subsequent optimizations consider advanced technology components and a long endurance mission, so this last case is used as a baseline for comparison of the segmented wing concept.

When applying the MDO tool to a multi-segment flying wing architecture, we expect that the reduced aspect ratio of each individual wing section should reduce the overall structural weight of the aircraft. Indeed, moving from the baseline to a two segment wing allows a 50% reduction in wing structural weight in addition to lowered battery and solar weights. Overall, the two segment wing provides an 8.5% reduction in gross weight over the long-endurance baseline for the same mission and payload requirements. It was shown that increasing the number of wing segments provides diminishing benefit, and that when certain constraints are activated, like the lower bound of t/c, some solution trends may change but the resulting gross weight is not greatly affected. For the six segment configuration, the MDO tool predicts a 14.2% decrease in weight compared to the long endurance baseline.

Upon conception of this research effort, it was thought that implementing a pinned wing concept may facilitate aircraft of extremely high aspect ratio and great improvement over a one-piece wing. We have shown that in fact the concept does allow for reduced structural weight, low wing loading, and optimal aspect ratios in the high 30s to low 40s. However, with advancements in energy storage technologies, highly efficient propulsion devices, and a careful spanwise mass distribution applied to a one-piece flying wing, many of the same benefits are possible. The added complexity inherent in a pinned wing configuration, and the associated risk detract from the potential benefits. Where we initially expected drastic improvements, results from the MDO tool suggest that actual benefits may not be quite as outstanding.

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