CONCEPTUAL DESIGN OF A VERTICAL TAKEOFF AND LANDING UNMANNED AERIAL VEHICLE WITH 24-HR ENDURANCE

William J. Fredericks*

NASA Langley Research Center, Hampton, VA 23681

This paper describes a conceptual design study for a vertical takeoff and landing (VTOL) unmanned aerial vehicle (UAV) that is able to carry a 25-lb science payload for 24 hr and is able to land and take off at elevations as high as 15,000 ft without human intervention. In addition to the science payload, this vehicle must be able to carry a satellite communication system, and the vehicle must be able to be transported in a standard full-size pickup truck and assembled by only two operators. This project started with a brainstorming phase to devise possible vehicle configurations that might satisfy the requirements. A down select was performed to select a near-term solution and two advanced vehicle concepts that are better suited to the intent of the mission. Sensitivity analyses were also performed on the requirements and the technology levels to obtain a better understanding of the design space. This study found that within the study assumptions the mission is feasible; the selected concepts are recommended for further development.

INTRODUCTION

This study was undertaken to explore the possibility of combining two diametrically opposed requirements, specifically, a long-endurance flight with vertical takeoff and landing. With today's technology levels, a rotorcraft cannot achieve a 24-hr flight. As a point of reference, the Boeing A160 program has a goal of 24-hr endurance. The Boeing A160 utilizes a two-speed transmission to keep the rotor at its optimal revolutions per minute (RPM) as the forward speed, altitude, and weight of the vehicle change. As of June 2010, the A160 has demonstrated an endurance of 18.7 hr.

This conceptual design study was initiated to consider other vehicle concepts that may be capable of meeting these requirements. The study considered both near-term concepts and more advanced concepts. A small unmanned aerial vehicle (UAV) company, MLB, was subcontracted to design the near-term concepts and to provide guidance on the assumptions going into the advanced vehicle concepts that only a company with aircraft-manufacturing experience can provide. The advanced vehicle concepts that were considered included aggressive configurations as well as advanced technologies such as electric propulsion. MLB and NASA Langley Research Center used different analysis tools but validated these tools on the V-Bat, which is a vertical takeoff and landing (VTOL) UAV that is in production by MLB. The focus of the analysis tools was on the vehicle performance; assumptions were made regarding the weights and performance of the subsystems.

STUDY OBJECTIVES AND REQUIREMENTS

The intent of this study was to design vehicles that do not require the extensive ground infrastructure that current long-endurance unmanned aerial systems (UAS) necessitate. All UAS

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* Aerospace Engineer, Aeronautics Systems Analysis Branch

over 50 lb that are capable of off-airport operation require some sort of catapult to launch the vehicle and some sort of retrieval mechanism, whether it be a catch wire or a net. This requires multiple operators and multiple vehicles to deploy and retrieve the UAS. Figure 1 shows an example of the ground infrastructure that is required to launch a UAS in this size class. The objective of this study was to design a UAS that can be transported in a full-size pickup truck and that is capable of being assembled and launched by only two operators. This also led to the goal of a maximum takeoff weight (MTOW) of 250 lb. This MTOW was selected as the maximum weight that two operators could handle unassisted. Further, the wing span was limited to 20 ft in order to enable it to fly in and out of confined areas.

Figure 1. Aeromech Fury on its catapult.

The vehicle was also required to carry two payloads. The science payload was defined as weighing 25 lb, consuming 400 W of electricity, and occupying 2,500 in^3 of volume. The science payload was assumed to only consume electrical power during the loiter phase of the mission. The second payload was a satellite communication system that is capable of transmitting live video. The smallest off-the-shelf system found was the ViaSat VMT-1220LA. This system weighs approximately 35 lb and consumes 275 W of electrical power. It has three components of the following approximate dimensions: the antenna (8 in. by 18 in. by 14 in.), the antenna control unit (8 in. by 11 in. by 14 in.), and the modem (8 in. by 5 in. by 14 in.). The fuselage was sized to package these components along with the engine and fuel tank, while allowing adequate room for the structure.

The performance requirements were based on the accomplishment of two missions. The primary mission was the loiter mission. In this mission, the vehicle takes off vertically, climbs to 18,000 ft, performs a 350 km dash to the science objective at a speed of 100 kt, loiters for 20 hr, dashes 350 km back at 100 kt, and descends and lands vertically. Total time for this mission is 24 hours. The alternate mission is a sensor-placing mission. This mission is the same as the primary mission, except for the loiter phase. For this mission, the loiter phase is replaced by multiple vertical takeoffs and landings at altitudes as high as 15,000 ft in order to push sensors into the ground. Note that in this case “multiple” is defined as the maximum number of takeoffs and landings that can be accomplished without exceeding the allocated fuel burn for the loiter mission. The alternate mission is not intended to become the active constraint for the vehicle sizing.

In addition to the previously mentioned requirements, operational requirements had to be considered as well. For the down select, qualitative assessments were made in the following areas: low acoustic, visual, and electronic signatures; ground handling; and ease of assembly in the field.

STUDY ASSUMPTIONS

In this study, assumptions were made regarding the empty weight of the aircraft. Performing a structural analysis to determine the empty weight of these proposed vehicles was outside the scope of this study. The V-Bat was used as a baseline; it has a structural weight fraction of 30 percent (structural weight relative to MTOW). As can be seen in Figure 2, the V-Bat has a structurally simple configuration. The advanced

Figure 2. The V-Bat produced by MLB.

concepts for this study, which are discussed later, were assumed to have a 40 percent structural weight fraction. Further, the wing aspect ratio was limited to 15 because the structural mass fraction for vehicles in this class is nearly insensitive to aspect ratio when the aspect ratio is less than 15. With a wing aspect ratio greater than 15, the structural mass fraction becomes a significant contributor to structural weight.

The propulsion system was assumed to be a diesel engine for two reasons. First, the availability of gasoline is scarce in many parts of the world. Also, diesel engines can be more fuel efficient than gasoline engines. The Cosworth AE1 diesel engine was used as a baseline. It has a brake specific fuel consumption (BSFC) of 0.55 lb/(hp·hr).Unlike a gasoline engine, the BSFC for a compression ignition engine was assumed to be constant for various throttle settings. The AE1 has a better BSFC at partial power (0.48 lb/(hp·hr) at 50-percent power). The power-to-weight ratio of the AE1 is 2.4 lb/hp. Turbocharging was added to provide improved high-altitude performance at the expense of increased weight, and the power-to-weight ratio that was used was 2.65 lb/hp. This assumption was used for the NASA-designed vehicles; however, for the MLB design a gasoline engine was assumed. The power-to-weight ratio was assumed to be 1.5 lb/hp and the BSFC to be 0.55 lb/hp-hr. This assumption was necessary because of the high disk loading for the V-Bat (see Concept Introduction below), which is highly sensitive to engine weight due to the size required of the engine. The assumptions are only 10% more aggressive than a production Rotax 914 turbo 4-stroke gasoline engine and it represents what is believed achievable with slight improvements to existing commercial production 4-stroke engine technology.

This study considered the use of hybrid electric propulsion systems to enable distributed propulsion concepts. The electric motor/generators were assumed to have a power-to-weight ratio of 0.5 lb/hp and a conversion efficiency of 90 percent. Batteries were assumed to be 350 W·hr/kg. Sion Power produces a lithium sulfur battery with this level of performance. No maximum battery discharge rate was considered. The Sion Power battery can output 400 W/kg, but battery technology is progressing very rapidly in this area. Note that the NASA designed aircraft that are presented later in this paper require a discharge rate approximately four times higher than what the Sion batteries provide.

Other propulsion systems were considered as well. The power-to-weight ratios for fuel cells are too low. The fuel consumption for small turbine electric generators is too high. For this application, an internal combustion engine turning an electric generator best suited the mission.

Hover analysis was performed by using an actuator disk momentum method. The model was calibrated to small manned helicopters. Downloading of the vehicle exposed to the rotor wash was assumed to be equal to the rotor-disk loading times the planform area under the rotors.

DESIGN SPACE EXPLORATION AND INITIAL SIZING

The Brequet endurance equation was used to obtain a rough estimate of the performance that would be required to achieve 24-hour endurance. Figure 3 provides the combination of BSFC, fuel fraction, and lift-to-drag (L/D) ratio that is required for the mission. Only the loiter segment is included in this analysis; the fuel that is required for takeoff, climb, and so on is not considered. This analysis also assumes a constant thrust-specific fuel consumption (TSFC). The TSFC is given in equation 1. The velocity was assumed to be 100 kt, and the propeller efficiency was assumed to be 70 percent. The conversion from ft·lb/s to horsepower is 550.

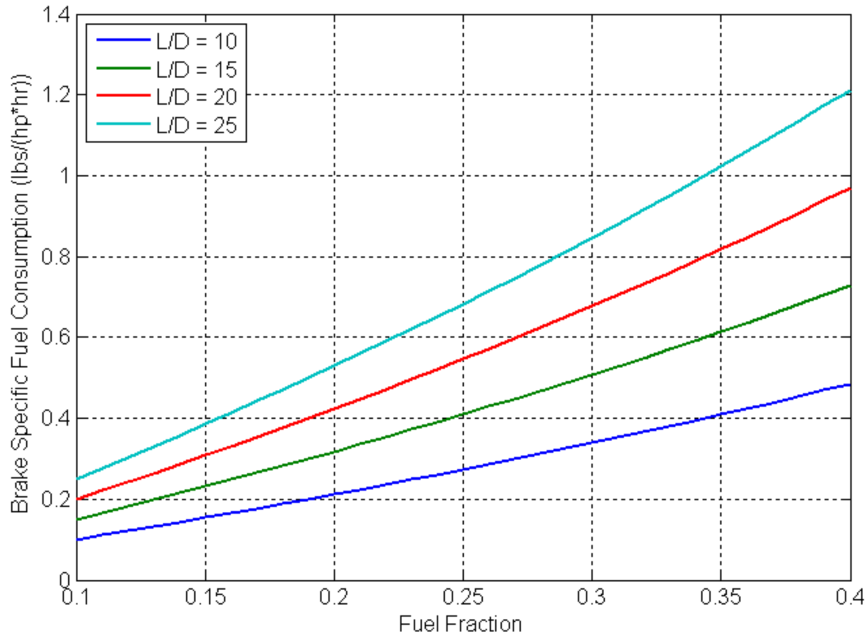

Figure 3. Vehicle characteristics required to achieve 24-hour endurance.

$$TSFC = \frac{BSFC * V}{550 * \eta_{Prop}} \tag{1}$$

As can be seen in Figure 3, an infinite number of combinations exist for the BSFC, L/D ratio, and fuel fraction that will meet the loiter requirement. Because we are only examining the loiter segment of the mission, this figure depicts only the minimum performance levels that are required. Interestingly, the performance levels of the vehicles that are designed in this study align very closely to the values that are predicted in this initial design space mapping.

CONCEPT INTRODUCTION

Early in this study, seventeen vehicle concepts were brainstormed. Wetted areas and disk loadings were calculated for these vehicle concepts. Further, each concept was qualitatively assessed by examining its strengths and weaknesses.

At the midterm review a down select to three concepts was performed and they were carried to the end of the design study. The first concept is the near-term concept, which carried a lower risk and could be developed in less time. The second and third concepts are more advanced concepts that better suit the intent of the requirements.

Scaled-Up V-Bat

MLB first sized a scaled-up V-Bat to meet the mission requirements. The V-Bat that is currently flying has an endurance of roughly five hr. Simply scaling up this vehicle to meet the payload and endurance requirements provided the lowest risk and the nearest term concept.

However, this vehicle produces a great deal of noise due to the high disk loading. Also, on landing the V-Bat frequently tips over and requires a person to set the aircraft back up on its tail. Thus, this vehicle is not well suited to autonomously placing sensors in the ground. For these reasons, this concept does not fully meet the intent but was carried through to show a possible near-term solution.

An open-rotor tail-sitter concept was not formally analyzed, but this concept was expected to have a performance similar to that of the scaled-up V-Bat.

Tazenflugel

Like the scaled-up V-Bat, this concept is also a tail sitter. The wings are mounted on a bearing surface; in vertical flight, the wings spin to generate the lift. The fuselage is nonrotating. This vehicle uses a hybrid electric propulsion system. The diesel engine, which turns an electric generator, is mounted in the fuselage. Batteries are use to supplement the power that is required at takeoff. At the wing tips are propellers that are driven by electric motors. Figure 4 shows an engineering visualization of the concept. The name of this concept means “spin wing” in German.

Figure 4. Tazenflugel concept depicted in flight and prior to takeoff.

While in vertical flight, pitch and yaw control is achieved via swashplateless technology like the “heliflap.” The aileron deflection is phased with the rotation angle of the wing rotor. Roll control is handled by tangentially deflecting the tail control surfaces. The dynamic pressure on the tails in hover is generated by the wing-rotor wash. The amount of control power for these surfaces is small, but there is little control power required about this axis.

This aircraft has the lowest disk loading (0.9 lb/ft^2) and the potential for the lowest noise. Further, this concept has the smallest wetted area and the best L/D ratio of all of the concepts. However, because the propellers are not directly connected to the engine, efficiency suffers because shaft power must be converted to electrical power and back to shaft power.

Like many other tail-sitter aircraft, tip-over is a concern. Should the stance of this aircraft not be sufficient to prevent tip-over, landing-gear legs would need to be designed to extend from the aft fuselage, which would add additional weight.

Dos Samara

This design utilizes outboard wing panels, which spin to generate thrust to lift the vehicle in vertical flight. In horizontal flight, the outboard wing panels lock. A pusher propeller is located on the tail to provide forward thrust in horizontal flight. Electric motors power the outboard wing panels in vertical flight. The batteries are used as counter weights to balance the outboard wing panels. Figure 5 depicts an engineering visualization of the concept. Because this vehicle lands level and is close to the ground on landing, the risk of the vehicle tipping over is greatly

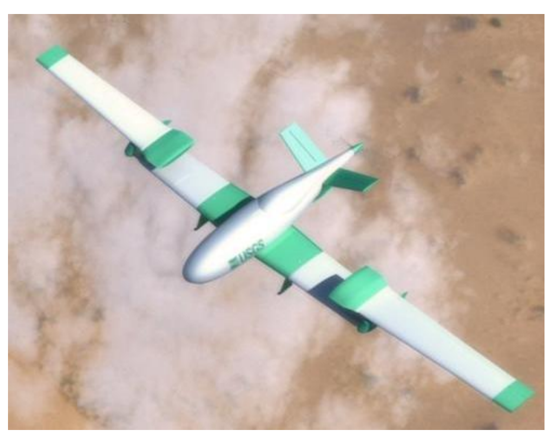

Figure 5. Dos Samara in flight.

Figure 6. Samara seed.

diminished. In addition, no field-of-view issues result from changing the fuselage orientation from horizontal to vertical and back. This vehicle is named after the Samara seed, which is shown in Figure 6.

While in vertical flight, this concept also uses swashplateless technology to control pitch via the aileron. Roll is controlled by collective on each rotor via the aileron. The tail propeller gimbals from side to side to control yaw.

The disk loading of the Dos Samara is 1.6 lb/ft^2, which is only 1.8 times greater than that of the Tazenflugel concept, but this vehicle also has the potential for low noise. The disk loading is still significantly lower than that of most helicopters. In addition, the tip speed of the wing rotors is half that of the average helicopter. Noise generated is a function of velocity to the fifth power, so even minor reductions in tip speed provide dramatic reductions in the noise that is generated.

NUMERIC ANALYSIS TOOLS

To produce a more accurate mission analysis, a time-step-based mission analysis tool was written. This code simulates the fuel burn in every segment of the actual mission to compute the total fuel burn as the vehicle flies the mission. The code assumes that all accelerations are zero and that the velocities are constant within each time step. The largest source of error is the vehicle transition from vertical to horizontal flight and vice versa. However, a transition that takes roughly 10 s uses a negligible amount of fuel relative to a 24-hour mission.

A vehicle sizing code was also written to size the vehicles. An initial guess for the MTOW is input by the user so that the tool can size the vehicle. Then, the mission analysis code is called to fly the mission. If the vehicle runs out of fuel, the MTOW is increased, the vehicle is resized, and the mission is flown again. This process is repeated in the code until the vehicle converges.

These codes were written in Matlab[*]. MLB independently wrote similar codes to conduct their vehicle analyses.

AERODYNAMIC ANALYSIS

The propulsion system performance and weight estimations were assumptions that were entered into the analysis codes, but the aerodynamic performance was computed. Drag polars were generated from a conceptual geometry. Vehicle Sketch Pad (VSP)[†] was used to model these vehicle concepts to perform the drag buildup.

The first step was to analyze the geometry in a vortex lattice tool. This study used XFLR5[‡], an open-source tool intended for use with remote-control aircraft. This tool calculates the local C_l at each location along the wings and tails for a given flight condition. This tool also provides the induced drag. Wrapped in this tool is XFOIL, a tool that was developed at the Massachusetts Institute of Technology (MIT) for computing two-dimensional drag polars. By using this two-dimensional airfoil data, the local C_d can be numerically integrated to provide the profile drag. At this point, only the drag from the wings and the tails has been taken into account.

The wetted areas for each component were used to estimate a drag coefficient. This process estimates the skin friction drag by calculating the skin friction drag over a flat plate with the same Reynolds number and then making a correction to account for the thickness of the body. The

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[*] A programming language developed by MathWorks http://www.mathworks.com/products/matlab/

[†] A conceptual level geometry modeling software created by the Aeronautics Systems Analysis Branch at the NASA Langley Research Center

[‡] An aerodynamic analysis tool that uses a vortex lattice based method http://xflr5.sourceforge.net/xflr5.htm

Reynolds number is computed by using a characteristic length for that part. The point at which transition from laminar to turbulent flow occurs must be estimated. A good initial guess is the location at which the flow trips from laminar to turbulent in the two-dimensional airfoil data at the same Reynolds number. Then, the flat-plate skin friction coefficients are calculated by using equations (2) and (3).

$$c_{f-\text{Laminar}} = \frac{1.328}{\sqrt{Re\#}} \tag{2}$$

$$c_{f-\text{Turbulent}} = \frac{0.074}{\sqrt[5]{Re\#}} \tag{3}$$

To obtain the equivalent flat-plate area, the wetted area of the component is multiplied by the weighted average skin friction coefficient $C_{f\text{-}Average}$. The $C_{f\text{-}Average}$ is determined by using the ratio of the laminar wetted area to the turbulent wetted area, respectively. For example, if $C_{f\text{-}Laminar}$ = 0.002, $C_{f\text{-}Turbulent}$ = 0.005, and 25 percent of the wetted area has laminar flow, then $C_{f\text{-}Average}$ = 0.00425.

Now, the equivalent flat-plate area must be scaled up to account for the super velocities that are generated as a result of the thickness of the component; this is referred to as the "form factor." Wings and fuselages have separate form-factor regressions. The wing formfactor scales with wing thickness to chord, and the fuselage form factor scales with the fineness ratio (length/diameter). Figures 6 and 7 in reference 1 contain body and wing form factors, respectively.

The final step in this skin friction calculation is to divide the scaled-up flat-plate area by the reference wing area to yield the lift independent drag (C_{Do}) of that component. Then, the C_{Do} are summed for each component. Table 1 and Table 2 show this analysis.

Table 1. Tazenflugel Wetted Area Drag Buildup

Component	Wetted area	Mean chord	Avg *Re#*	% Lam	% Turb	T/C	Length/Diam	Component C_{Do}
Wing	49.9	1.37	767,172	50%	50%	12%		0.0089
Engine Pods	3.5	2.00	1,117,512	25%	75%		5.00	0.0007
Fuselage	26.6	7.10	3,967,167	15%	85%		4.73	0.0047
Tails	12.0	0.85	474,942	50%	50%	15%		0.0026
Totals	91.9							0.0169

Table 2. Dos Samara Wetted Area Drag Buildup

Component	Wetted area	Mean chord	Avg *Re#*	% Lam	% Turb	T/C	Length/Diam	Component C_{Do}
H tail	6.3	0.85	474,942	50%	50%	15%		0.0014
V tail	2.3	0.75	419,067	50%	50%	15%		0.0005
Inboard Wing	39.1	1.7	949,885	50%	50%	17%		0.0075
Fuse	23.7	7	3,911,291	15%	85%		4.67	0.0043
Wing	17.9	1.11	620,219	50%	50%	15%		0.0037
Pod	2.4	2	1,117,512	25%	75%		6.67	0.0004
Batteries	15.3	1.6	894,009	50%	50%	20%		0.0032
Totals	107.0							0.0209

Any areas in which the flow separates must be estimated and added to the drag buildup. For the analyzed vehicles, zero separated flow is assumed. The C_P is assumed to be $^{-}1$ in the separated-flow regions. This C_P is multiplied by the aft-facing projected area of the separated-

flow regions; this number is divided by the reference wing area to yield an additional C_{Do} increment. The sum of the skin friction C_{Do} and the separated flow C_{Do} is the total whole aircraft C_{Do}.

By using a vortex lattice tool, a drag polar can be built that accounts for both the induced and the profile drag of the aircraft. The C_{Do} of this drag polar is subtracted from the C_{Do} of the wetted area drag buildup to yield the net parasite drag. This subtraction step is applied to prevent double-bookkeeping the skin friction drag of the wings and tails that is included in the two-dimensional airfoil drag polars. As shown in Figure 7 and Figure 8, the net parasite drag is applied uniformly over the drag polar. An extra 10 percent is also added to account for the cooling drag of the engine. The total drag is the sum of the induced, profile, parasite, and cooling drag. For these drag buildups, the total C_{Do} were estimated to be 0.019 and 0.023, and the best L/D ratios were estimated at 24.0 and 21.2 for the Tazenflugel and Dos Samara vehicles, respectively.

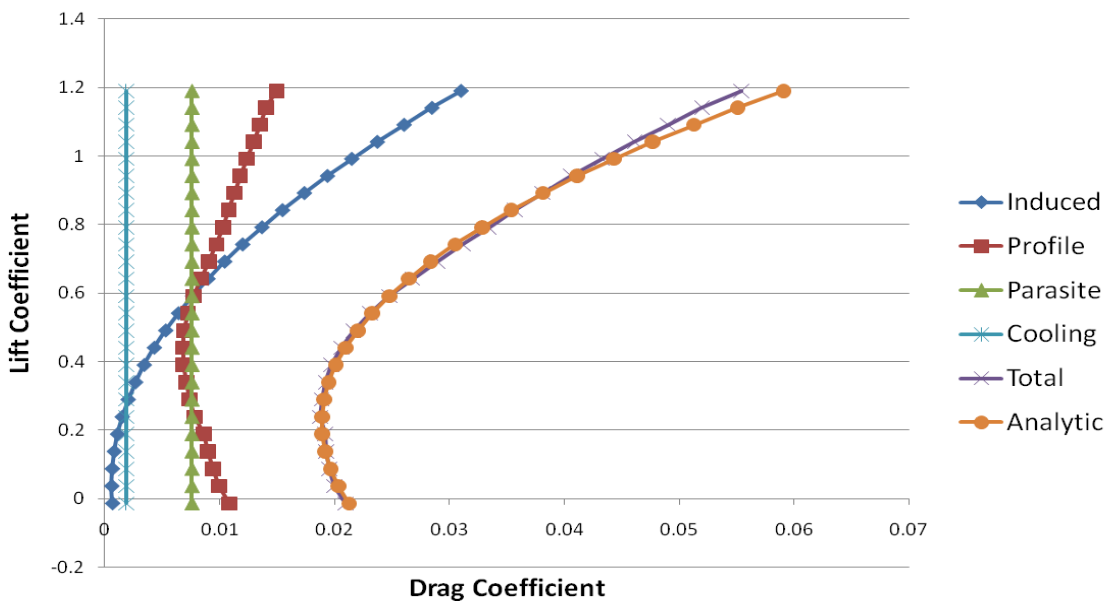

Figure 7. Tazenflugel drag polar.

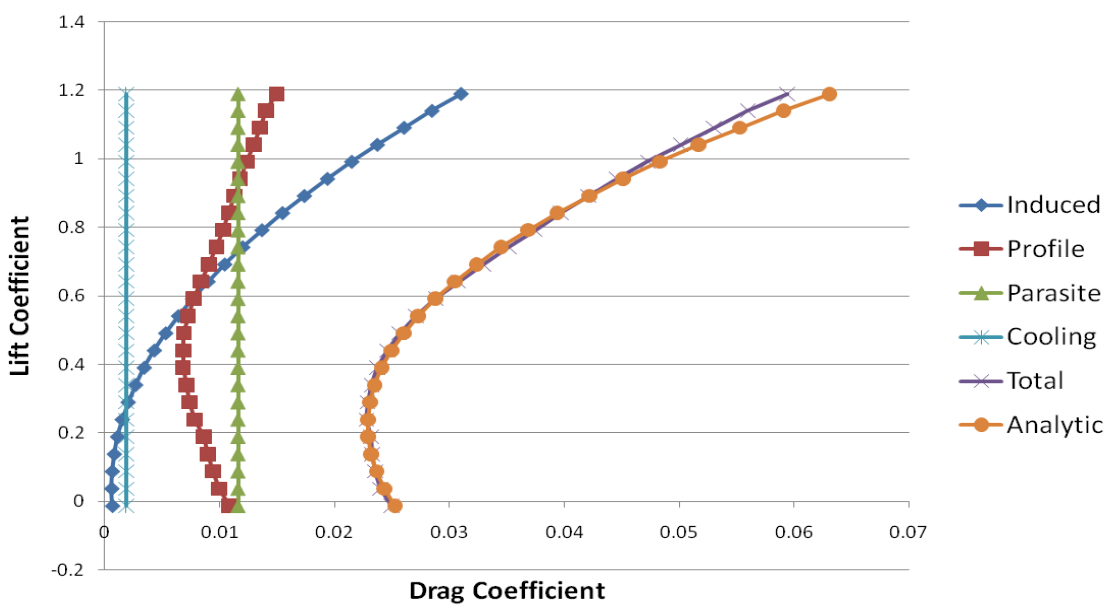

Figure 8. Dos Samara drag polar.

These drag polars were of a fixed initial geometry. To allow the sizing code to vary the wing area and the aspect ratio, an analytic model of the drag polars was generated. The drag of the fuselage remains constant because the fuselage is sized to fit the required payloads and internal components. As the wing area is varied, the L/D ratio varies. As the wing aspect ratio is varied, the induced drag and, therefore, the L/D ratio vary as well. Equation (4) is the textbook method for defining a simple drag polar, where C_{Do} is defined by equation (5). See the Notation section at end of this paper for variable descriptions.

$$C_D = C_{Do} + \frac{C_L^2}{\pi * AR * e} + k * C_L \tag{4}$$

$$C_{Do} = C_{Do1} + C_{Do2} * \frac{S_{Baseline}}{S} \tag{5}$$

The analytic models that are given in equations (4) and (5) were used to select the values given in Table 3 to best model the drag polars (Figure 7 and Figure 8). The analytic models were plotted over the drag polars to show how well they compare.

Table 3. Coefficient Values Used to Define Analytic Aerodynamic Model

Variable	Tazenflugel	Dos Samara
C_{Do1}	0.0134	0.0134
C_{Do2}	0.0076	0.0116
e	0.51	0.51
k	-0.019	-0.019
$S_{Baseline}$	27.46	27.46

SIZING RESULTS

The tools, the analysis, and the assumptions that are described above were used to size and optimize the vehicles based on wing area and engine size. Table 4 shows the results of the analysis. As expected, the scaled-up V-Bat was the largest vehicle. It is the near-term brute-force concept that requires a 43-hp engine for the vehicle to hover. For this vehicle, the assumed engine power-to-weight ratio is 1.5 lb/hp. Note that the Tazenflugel model has a better L/D ratio than the Dos Samara model but is 25 lb heavier. This is because on the Dos Samara vehicle the propeller is directly connected to the engine; the Tazenflugel vehicle is not quite as efficient because the engine turns the generator, and then electric motors turn the propellers.

For this set of mission requirements, a hybrid electric propulsion system is superior to a conventional system in which an engine directly turns a propulsor. The amount of power that is needed to lift the vehicle off the ground is vastly greater than the power that is needed for cruise. Electric motors have a much better power-to-weight ratio than internal combustion motors. The downside to electric propulsion is the battery weight, but the power that is required for takeoff is only needed for a short time; therefore, only a small amount of energy and thus, a relatively small amount of battery weight is needed.

Table 4. Vehicle Sizing Results

	Scaled-up V-Bat*	Tazenflugel	Dos Samara
Wing loading (lb/ft^2)	5.5	10.5	10
Wing area (ft^2)	57.4	26.95	25.8
Wing span (ft)	20	20	19.7
Aspect ratio	7.0	14.8	15.0
Best L/D ratio	13.0	24.6	21.5
CL for best L/D	0.65	0.71	0.79
Engine power loading (lb/hp)	7.2	38	37
Engine power (hp)	43.0	7.4	7.0
Disk loading (lb/ft^2)	15.9	0.9	1.6
Payload (lb)	60	60	60
Avionics (lb)	5	5	5
Fuel consumed (lb)	88.1	63.9	54.9
Battery weight (lb)	N/A	10.7	6.3
Engine weight (lb)	65*	19.7	18.5
Generator weight (lb)	1.4	3.7	3.5
Electric motor(s) weight (lb)	N/A	6.7	6.4
Airframe weight (lb)	94	113.4	103.4
MTOW (lb)	313	283	258

SENSITIVITY ANALYSIS

To better understand the design space, sensitivity analyses were performed on both the requirements and the technology assumptions. In the tables that appear in this section, the yellow highlighted values are the baseline values that were used for the vehicle-sizing results. A MTOW that appears in red italics indicates that the vehicle did not converge within the assumptions that were established in this study. For these cases, the red italicized value is the minimum MTOW that would converge (the structural weight would be lighter for such a vehicle; the percentage of airframe weight at which the vehicle did converge is given for informational purposes).

Requirements Sensitivities

The sensitivity of the vehicle sizing was evaluated with respect to the following requirements: dash speed, cruise altitude, vertical-flight rate of climb, and the deice system weight.

Table 5 gives the sensitivity results of the vehicle sizing to the minimum speed requirement as the aircraft flies to and from the research area. The Tazenflugel could not converge at the 110 kt speed requirement. If we assume that the airframe weight is only 38.7 percent of the MTOW, then the Tazenflugel vehicle would converge at an MTOW of 347 lb. If the span requirement was relaxed to 21 ft, then the Tazenflugel vehicle would converge with a 40-percent airframe weight at an MTOW of 306 lb.

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* For this vehicle, an engine power-to-weight ratio of 1.5 lb/hp is assumed.

Table 5. Dash Speed Sensitivity

Dash speed (kt)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
110	*347*	305
100	283	258
90	278	254
80	276	251

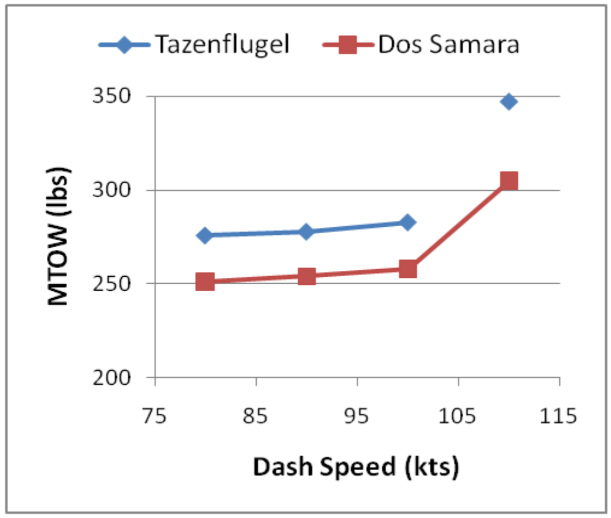
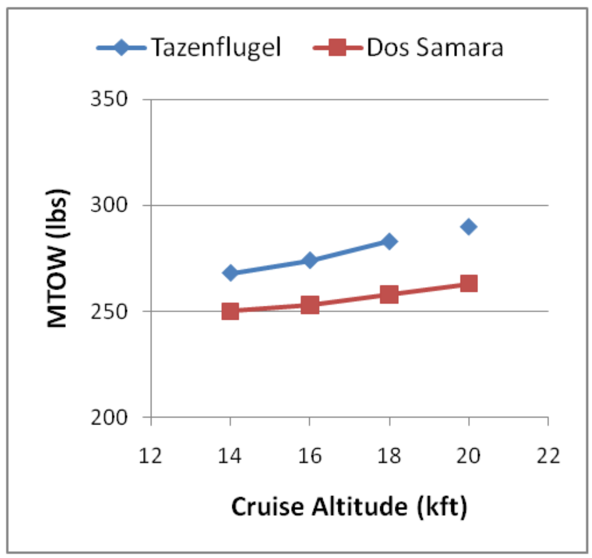


Table 6 shows the sensitivity to cruise altitude. This altitude is maintained for the dash to the research area, during the loiter period at the research area, and for the dash back. The Tazenflugel vehicle does not converge at an altitude of 20,000 ft. If we assume that the airframe equals only 39.2 percent of the weight of the vehicle, then it converges with a MTOW of 290 lb.

Table 6. Cruise Altitude Sensitivity

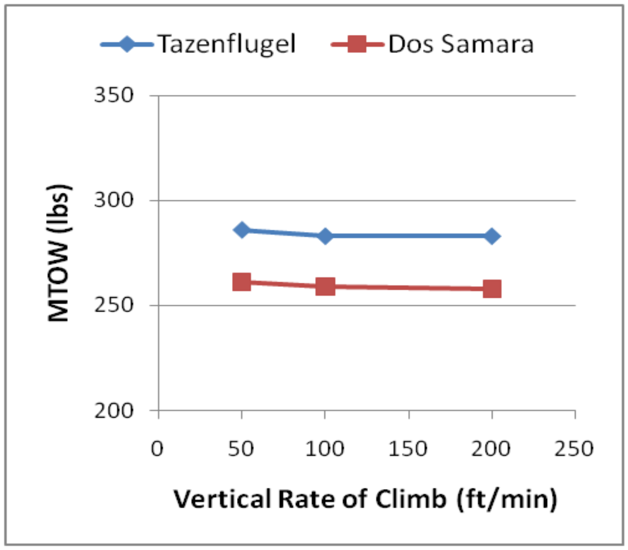
Cruise altitude (kft)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
20	*290*	263
18	283	258
16	274	253
14	268	250



The vehicle is required to climb vertically to 200 ft and then transition to horizontal flight and climb at a rate of 500 ft/min to the cruise altitude. Table 7 shows the sensitivity to the rate of climb that is required while in vertical flight. Note that the vehicle sizing is insensitive to this requirement. As the vertical rate of climb is reduced, the electric motors can be smaller because less power is required. However, more time is spent in vertical flight, so the amount of energy that is consumed is greater, which requires more battery weight. These two characteristics traded nearly one for one.

Table 7. Vertical Flight Rate of Climb Sensitivity

Rate of climb (ft/min)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
200	**283**	**258**
100	**283**	**259**
50	**286**	**261**



The impact of adding a deicing system to the aircraft was also investigated. Because no data are available on deicing systems for an aircraft of this size, the deicing system was assumed to weigh 1 or 2 percent of the MTOW. The drag polars were unchanged. (This data can be used to assess the weight impact of any system to be added to the aircraft.) Table 8 shows the impact of adding the deicing system. The Tazenflugel vehicle sizing would not converge with any additional weight, so the span constraint was relaxed and convergence was easily achieved. Table 9 shows the wing span of the corresponding designs in Table 8. All of the spans remained within 21 ft. The middle column shows the results for the Dos Samara vehicle with no span constraint, and the right column shows the results for the Dos Samara with the span constrained to 20 ft.

Table 8. Deicing System Weight Sensitivity

System weight (% MTOW)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)	Span constraint Dos Samara MTOW (lb)
2	**299**	**272**	**281**
1	**289**	**266**	**266**
0	**281**	**258**	**258**

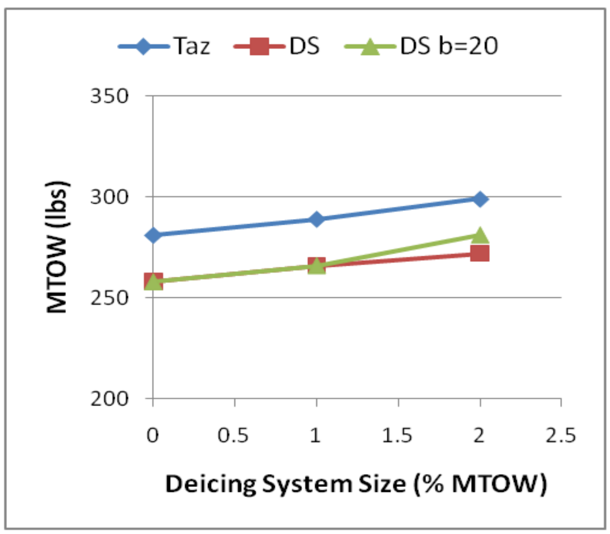


Table 9. Wing Span Corresponding to Table 8

System weight (% MTOW)	Tazenflugel span (ft)	Dos Samara span (ft)	Span constraint Dos Samara span (ft)
2	**20.67**	**20.20**	**20.00**
1	**20.32**	**19.97**	**19.97**
0	**20.10**	**19.67**	**19.67**

Technology-Level Sensitivities

The sensitivity of the major technology-level assumptions that were used in this study was evaluated. These include satellite communication system weight, combustion motor weight, BSFC, electric motor/generator weights, electric motor/generator conversion efficiencies, and battery energy density.

Table 10 shows the vehicle-sizing sensitivity to the use of lighter satellite communication systems. As would be expected, reducing the payload weight significantly reduced the overall size of the vehicle. For this evaluation, the assumption was made that the volume and the power that were required for the satellite communication system were unchanged.

Table 10. Satellite Communication System Weight Sensitivity

SatComm weight (lb)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
35	**283**	**258**
25	**250**	**229**
15	**222**	**204**

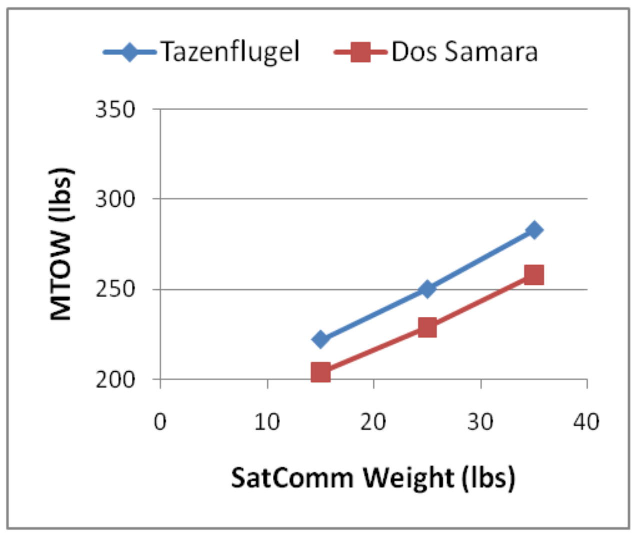


Table 11 demonstrates the benefit of using an engine with a better power-to-weight ratio. Note that a turbocharger is not included in this power-to-weight ratio; an additional 0.25 lb/hp is still included.

Table 11. Combustion Engine Power-to-Weight Sensitivity

Engine power to weight (lb/hp)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
2.4	**283**	**258**
1.7	**266**	**244**
1.0	**252**	**232**

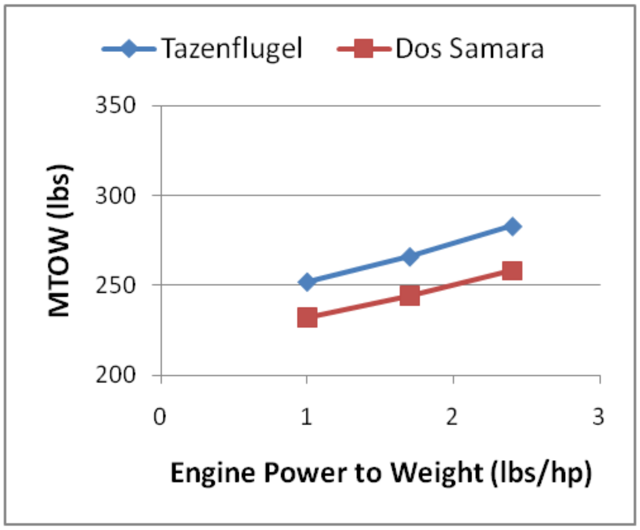


Table 12 demonstrates the effect of using a more fuel-efficient engine.

Table 12. Brake-Specific Fuel Consumption Sensitivity

BSFC (lb/(hp·hr))	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
0.55	**283**	**258**
0.45	**252**	**233**
0.35	**229**	**215**

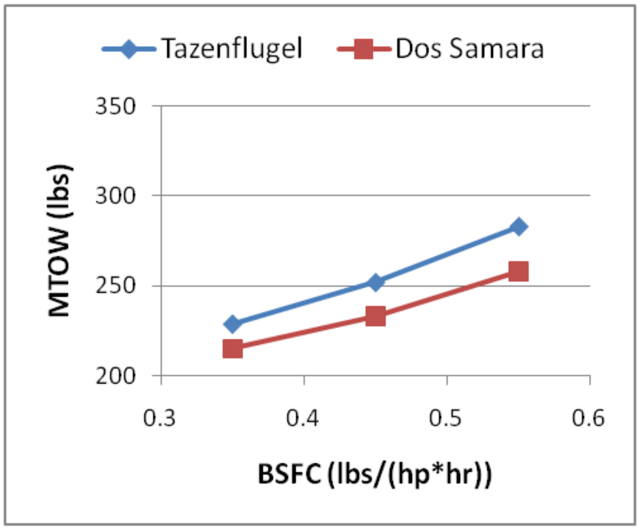


Table 13 shows the benefit of using electric motors and generators with higher power-to-weight ratios. The same power-to-weight ratios and efficiency assumptions were used for both the electric motors and the generator that is driven by the engine.

Table 13. Electric Motor/Generator Weight Sensitivity

Power/ weight (hp/lb)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
2.0	**283**	**258**
4.0	**267**	**244**
6.0	**261**	**237**

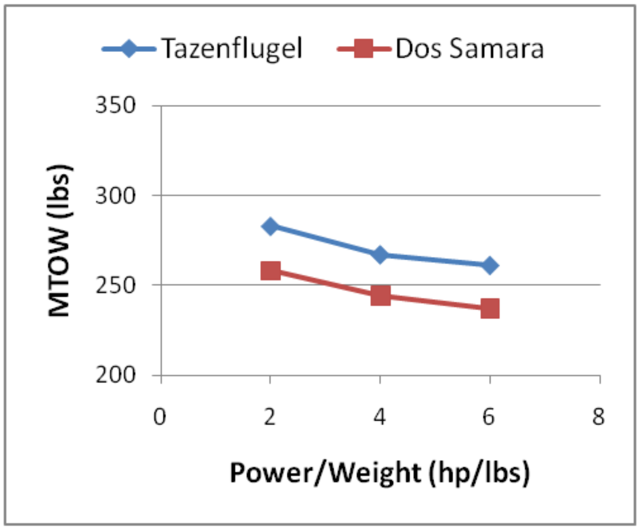


Table 14 quantifies the improvements that were made to the vehicles with the use of more efficient electric motors and generators. Note that the Dos Samara is nearly insensitive to electric motor efficiency because the electric motors are only used in vertical flight to turn the outboard wing panels; the propeller on the tail is directly connected to the engine. On the other hand, the Tazenflugel must in all phases of flight convert shaft power to electric power and then electric power back to shaft power to turn the propellers. If the conversion efficiency for the electric motors and generators is assumed to be 96 percent, then the Tazenflugel is the better vehicle. This is because the Tazenflugel has less drag, and the penalty for making the electric conversions is reduced.

Table 14. Electric Motor/Generator Conversion Efficiency Sensitivity

Efficiency (%)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
90	**283**	**258**
93	**267**	**256**
96	**252**	**254**

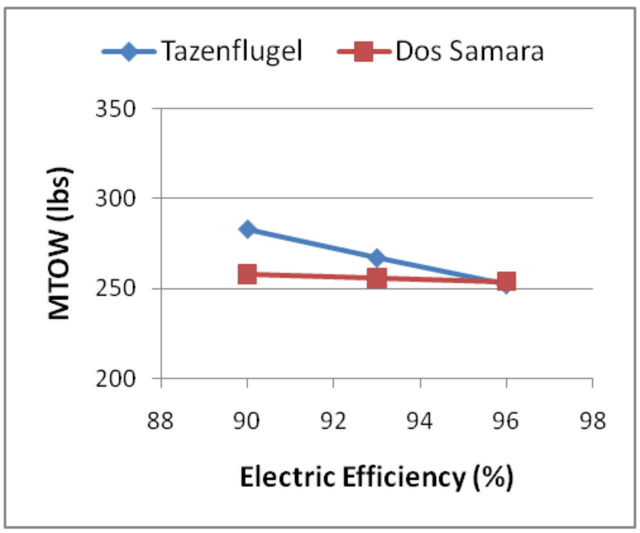
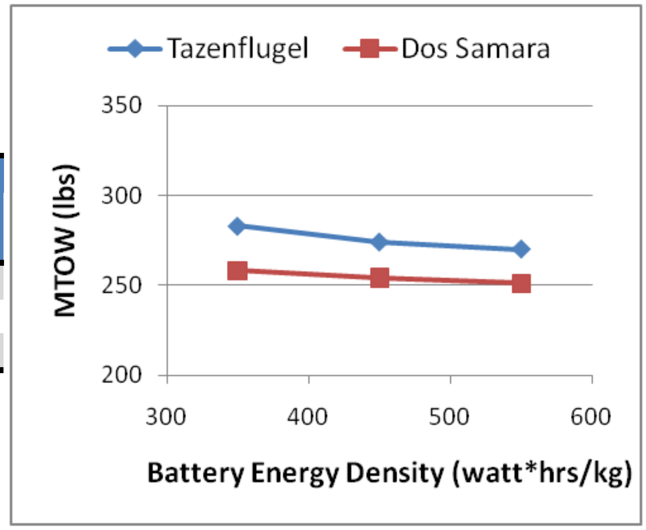


Table 15 shows the benefit of using higher performance (energy density) batteries. Note that these designs were not reoptimized. With reoptimization, one would expect minor reductions (i.e., 0 to 5 lb) in MTOW beyond the values listed in Table 15. One would expect smaller combustion engines and larger battery capacity after optimization.

Table 15. Battery Energy Density Sensitivity

Battery (W·hr/kg)	Tazenflugel MTOW (lb)	Dos Samara MTOW (lb)
350	**283**	**258**
450	**274**	**254**
550	**270**	**251**



CONCLUSIONS

The technology sensitivity analysis and qualitative assessments were used to rank the technologies according to investment priority (see Table 16). Interestingly, note that for both vehicles the priority ranking is the same, with one exception. The electric motor efficiency is the lowest priority for the Dos Samara. This ranking would be useful in guiding investment funds should the need arise to make a vehicle smaller than the vehicles sizes as they are presented based on the assumption of this study. Investment in priority 1, satellite communication system weight reductions, will have the greatest benefit in reducing vehicle size. Another factor to consider that was not considered in this ranking is the cost that is associated with advancing the technology. If all six of these technologies were applied at the most optimistic values, the Tazenflugel would have an MTOW of 142 lb, and the Dos Samara would have an MTOW of 146 lb.

Table 16. Technology Priority Ranking

	Tazenflugel	Dos Samara
1	Satellite comm weight	Satellite comm weight
2	BSFC	BSFC
3	Engine P/W	Engine P/W
4	Electric motor efficiency	Electric motor P/W
5	Electric motor P/W	Battery energy density
6	Battery energy density	Electric motor efficiency

The Tazenflugel vehicle is on the edge of the feasible design space for the given requirements and assumptions in the study. Even slight penalties imposed on the vehicle prevent it from converging. This is primarily because the wing span requirement is an active constraint. As the vehicle gets heavier, its aspect ratio must decrease. With reduced aspect ratio the induced drag increases.

The Dos Samara is a more robust concept because the vehicle still has some room to increase in size before the span constraint becomes active. In addition, because the engine is directly connected to the propeller, which is more efficient, the vehicle can be smaller.

This study still leaves areas of uncertainty that should be evaluated prior to committing to full-scale development. The major areas of risk include the assumption that the airframe structural weight equals 40 percent of the MTOW and the dynamics of converting from hover to forward

flight and vice versa. However, this study indicates that the requirement set is feasible, even with the unconventional concepts, and warrants further investigation.

NOTATION

AR	Wing aspect ratio
BSFC	Brake specific fuel consumption
C_D	Total drag coefficient
C_{Do}	Zero lift drag coefficient
C_{D1}	Wing-related zero lift/drag coefficient
C_{D2}	Fuselage and other non-wing-related zero lift/drag coefficient
e	Oswald's induced drag factor
k	Wing profile drag factor
L/D	Lift to drag ratio
MIT	Massachusetts Institute of Technology
MTOW	Maximum takeoff weight
RPM	Revolutions per minute
S	Wing area
TSFC	Thrust specific fuel consumption
UAV	Unmanned aerial vehicle
UAS	Unmanned aerial system
VSP	Vehicle Sketch Pad
VTOL	Vertical takeoff and landing
π	3.141593

REFERENCES

[1]Feagin, R. C., "Delta Method: An Empirical Drag Buildup Technique," NASA CR-151971, 1978.