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High Altitude Long Endurance UAV Analysis of Alternatives and Technology Requirements Development

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March 2007

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Abstract

The objective of this study was to develop a variety of High Altitude Long Endurance (HALE) Unmanned Aerial Vehicle (UAV) conceptual designs for two operationally useful missions, hurricane science and communications relay, and compare their performance and cost characteristics. Sixteen potential HALE UAV configurations were initially developed, including heavier-than-air (HTA) and lighter-than-air (LTA) concepts with both consumable fuel and solar regenerative (SR) propulsion systems. Through an Analysis of Alternatives(AoA) down select process, the two leading consumable fuel configurations, one each from the HTA and LTA alternatives, and an HTA SR configuration were selected for further analysis. Cost effectiveness analysis of the consumable fuel configurations revealed that simply maximizing vehicle endurance can lead to a sub-optimum system solution. An LTA concept with a hybrid propulsion system consisting of solar arrays and a hydrogen-air proton exhange membrane fuel cell was found to have the best performance; however, an HTA diesel-fueled wing-body-tail configuration emerged as the preferred consumable fuel concept because of the large size and technical risk of the LTA concept. The two study missions could not be performed by even the best HTA SR concept. Mission and SR technology trade studies were conducted to enhance understanding of the potential capabilities of such a vehicle. With near-term technology, SR-powered HTA vehicles are limited to operation in favorable solar conditions, such as the long days and short nights of summer at higher latitudes. Energy storage system specific energy and solar cell efficiency were found to be the key technology areas for enhancing HTA SR performance.

1.0 Introduction

High Altitude Long Endurance (HALE) air vehicles have been the focus of significant research and development efforts for decades (refs. 1-6). The state of the art has been advanced to enable higher operational altitudes, longer durations with greater payloads, and increased autonomy. Applications for these vehicles include scientific data collection, communications relay, and surveillance and reconnaissance missions. A wide variety of air vehicles, both operational and technology demonstration types, have been developed or are currently under development. Examples of high altitude and/or long endurance vehicles include the Lockheed U-2; Boeing Condor; Northrop Grumman RQ-4 Global Hawk; AeroVironment Pathfinder, Helios, and Global Observer; Scaled Composites Voyager and Global Flyer; AC Propulsion SoLong; Lockheed Martin High Altitude Airship; and the European "Solar Impulse" effort to build a solar powered airplane to fly around the world. The desire to extend the endurance of these vehicle types has led to research in solar regenerative (SR) propulsion systems relying on a solar photovoltaic array coupled to an energy storage system (ESS). SR propulsion systems are theoretically capable of propelling air vehicles to endurances of many months.

The purpose of this study was to develop a variety of HALE Unmanned Aerial Vehicle (UAV) conceptual designs for two operationally useful missions, compare their performance and cost characteristics, and quantify the technology improvements required (if any) to enable these missions. Lighter-than-air (LTA) and heavier-than-air (HTA) concepts utilizing both SR and non-regenerative propulsion systems were analyzed. A secondary goal of this study was to develop and demonstrate a design and analysis capability for HALE UAV concept technical and feasibility assessments. Figure 1 shows the study flow. The initial effort, termed Phase I, consisted of requirements derivation given the two design missions, the identification and analysis of a set of sixteen potential configurations, and a down select to the best HTA and LTA configurations. Phase II of the study consisted of an operational and life cycle cost analysis utilizing the feasible down-selected configurations. In addition, technology and mission requirements trade studies were performed for the preferred HTA SR configuration.

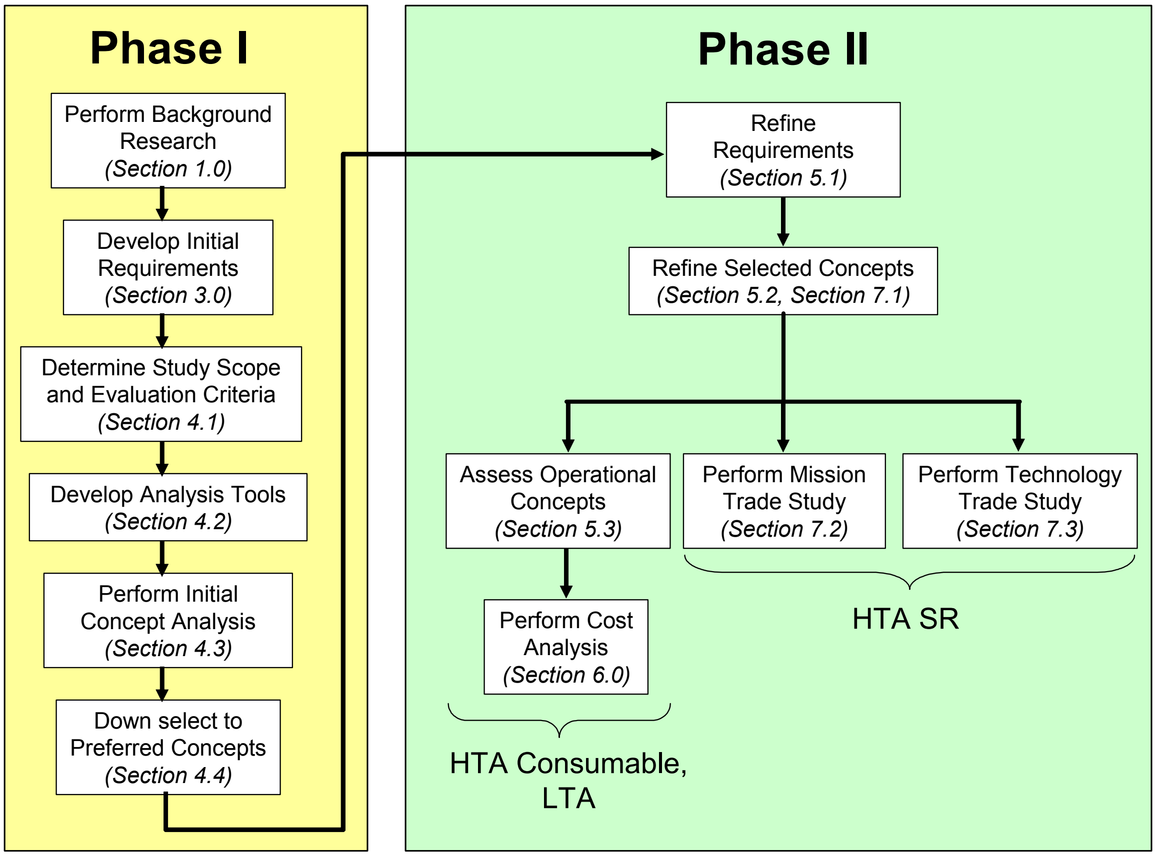


Figure 1. HALE UAV concept study flowchart.

In order to enable the exploration of a wide variety of potential concepts this study was focused on the conceptual phase of aircraft design. Succeeding phases of preliminary and detailed design require a more narrow focus on a single concept. The goal of aircraft conceptual design is to identify the best system concept from a large set of potential solutions to meet a given requirements set. This process is often termed an Analysis of Alternatives (AoA). During the conceptual design phase, the requirements set is not generally fixed; indeed, a primary activity during conceptual design is performing requirements trade studies to identify design drivers and conflicts within the requirements set. The products of this design phase are, therefore, not only preferred system concepts, but also refined requirements that are properly balanced. A balanced set of requirements helps to enable the successful execution of the subsequent design, development, and production phases from a cost, schedule, and performance perspective. In addition, risk areas requiring technology investment are identified during the conceptual design phase. For this study, a technology constraint was specified that all vehicle, payload, and ground operations technologies be at a Technology Readiness Level (TRL) ≥ 5 by the end of fiscal year 2008 (FY08). TRL 5 is defined as component or breadboard validation in a relevant environment (refs. 7 and 8).

Two conceptual design tools were utilized to perform this study; a HALE Multi-disciplinary Design Optimization (MDO) code for the HTA concepts, and an Airship Design and Analysis Code (ADAC) for the LTA concepts. Inputs to these codes generally take the form of the required mission parameters (e.g. payload, loiter altitude, dash speed), propulsion system characteristics (e.g. power, fuel flow), and configuration information (e.g., type of planform, number and placement of engines, structural design criteria). Outputs of the codes generally include a sized vehicle and estimates of its aerodynamic characteristics, mass properties, and performance. The conceptual designer can select an objective function, specify constraints, and select which design variables are allowed to vary in order to arrive at an optimum solution. Given the lack of detailed information available to the designer during the conceptual phase, assumptions for various input parameters must be made. These assumptions are generally based on historical data or expert judgment. For the conceptual phase it is more important to utilize consistent assumptions across the concept options than it is to focus entirely on absolute accuracy, since the purpose is to produce accurate relative comparisons amongst the potential solution concepts. Accuracy is needed to the extent that the potential solution concepts must be feasible and analytically substantiated to provide adequate decision support.

This report is organized based on the study process flow in Figure 1 and the section of the report corresponding to each step in the process is shown in the figure. The initial requirements are presented in Section 3.0, beginning with a discussion of the hurricane science and communications relay missions. Vehicle design requirements were derived from these mission areas for use in the AoA. Section 4.0 presents the AoA, starting with a discussion of the scope and evaluation criteria. Next, the tools and processes are described, including tool validation information. The final part of Section 4.0 presents the sixteen Phase I concepts and the down select decisions. Section 5.0 presents the operational concept study, beginning with a refinement of the mission requirements. These refined requirements, used for all subsequent analyses, are referred to as the Phase II requirements. Based on the Phase II requirements, configuration updates were performed on the down-selected concepts and the operational concept study modeling assumptions and results are presented. Section 6.0 presents the life cycle cost analysis performed for the Phase II concepts. Section 7.0 describes the HTA SR mission requirements and technology trade studies, and Section 8.0 presents the overall study conclusions.

2.0 Symbols

AR_{ht} – Horizontal Tail Aspect Ratio

AR_{vt} – Vertical Tail Aspect Ratio

AR_{w} – Wing Aspect Ratio

b_{ht} – Horizontal Tail Span

b_{s} – Wingspan

b_{vt} – Vertical Tail Span

C_{D} – Coefficient of Drag

C_{D0} – Coefficient of Zero Lift Drag

C_{L} – Coefficient of Lift

C_{mo} – Pitching Moment Coefficient

D_{p} – Propeller Diameter

Fus_D – Fuselage Diameter

Fus_L – Fuselage Length

Λ_{LE} – Wing Leading Edge Sweep

L/D – Vehicle Lift-to-Drag Ratio

N_{pyl} – Number of Pylons

P_{AR} – Power Available from Solar Regenerative System

P_{RL} – Power Required from Solar Regenerative System for Loiter

P_{sep} – Pod/Nacelle Separation Distance

PL_{sep} – Pylon Separation Distance

Pod_D – Pod/Nacelle Diameter

Pod_L – Pod/Nacelle Length

Re – Reynolds Number based on Mean Aerodynamic Chord

S_{aux} – Auxiliary Tracking Array Area

S_{ht} – Horizontal Tail Area

S_{pyl} – Pylon Area

S_{vt} – Vertical Tail Area

S_w – Wing Area

t/c – Airfoil Thickness-to-Chord Ratio

W/S – Wing Loading

Abbreviations

ADAC – Airship Design and Analysis Code

AoA – Analysis of Alternatives

SFC – Specific Fuel Consumption

CECOM – Communications and Electronics Command (U.S. Army)

CI – Compression Ignition

CONUS – Continental United States

COTS – Commercial Off The Shelf

DOE – Design of Experiments

EO/IR – Electro-Optical/Infrared

ERAST – Environmental Research Aircraft and Sensor Technology

ESS – Energy Storage System

HALE – High Altitude Long Endurance

HTA – Heavier than Air

IC – Intermittent Combustion

LCC – Life Cycle Cost

LH_2 – Liquid Hydrogen

LTA – Lighter than Air

MDO – Multi-disciplinary Design Optimization

NAST – NPOESS Airborne Sounder Testbed

NOAA – National Oceanic and Atmospheric Administration

NPOESS – National Polar Orbiting Environmental Satellite System

O&S – Operations and Support

PEM – Proton Exchange Membrane

$\%P_{regen}$ – Percentage of Power Required Supplied by the Solar Regenerative System (P_{AR}/P_{RL})

PMAD – Power Management and Distribution

RDT&E – Research, Development, Test and Evaluation

ROM – Rough Order of Magnitude

RSE – Response Surface Equation

RTB – Return to Base

SI – Spark Ignition

SP – Specific Power

SR – Solar Regenerative

TAS – True Air Speed

TOGM – Takeoff Gross Mass

TRL – Technology Readiness Level

UAV – Unmanned Aerial Vehicle

3.0 Initial Requirements

The two reference missions considered for this study were hurricane science and communications relay. HALE UAVs have been candidates for both of these mission types in past studies. A recent NASA study of the use of HALE UAVs for hurricane science is detailed in an unpublished white paper by M. Avery et al. (ref. 9). According to this paper, the current Earth observing capability consists primarily of satellites and ground networks. Although aircraft missions also play an important role, their usefulness is limited by constrained durations, limited observation envelopes, and crew safety issues. A HALE UAV platform has the potential to overcome these constraints and provide measurements that complement the current space and ground based systems. Another past NASA study effort, conducted in support of the Environmental Research Aircraft and Sensor Technology (ERAST) program, produced a hurricane science mission demonstration plan that highlighted hurricane formation (tropical cyclogenesis) and hurricane intensity forecasting as areas which would benefit from the application of HALE UAVs as measurement platforms (ref. 10). Because hurricanes form over tropical oceans where data are sparse, additional in situ measurements are required to complement satellite data in order to understand hurricane formation (cyclogenesis). The ERAST mission plan states that hurricane intensity is related to the vertical temperature profile from the top of the storm to the sea surface. A HALE UAV capable of operating above the storm cloud canopy would be able to provide in situ measurements of that vertical profile by releasing dropsondes which fall through the storm. In addition, the high operating altitude would enable observation of the storm interaction with the lower stratosphere, where measurements are currently unavailable. Both studies provided a consistent background and point-of-departure for developing hurricane science mission requirements in the current study.

The communications relay mission was selected to provide a commercial (and military) complement to the science driven hurricane mission. HALE UAV platforms have the potential to serve as effective, low cost communications relay systems due to their long endurance, large ground footprint (compared to cell phone towers), flexibility, and relatively low acquisition and operating costs (compared to satellites).

The vehicle requirements evolved over the course of the study. The initial set of requirements was derived from an examination of the two mission areas. Both threshold (minimum acceptable) and goal vehicle requirements were identified. A subset of these requirements was then used for Phase I, and, based on Phase I results and additional input, a refined set of requirements was developed for Phase II.

The hurricane science design mission profile is shown in Figure 2. The air vehicle launches from the operating base, assumed to be located near the Eastern coast of the U.S., cruise-climbs to the optimum transit altitude and speed, and transits to the surveillance location, assumed to be 15° N latitude, 30° W longitude for the worst case (near the Cape Verde Islands where many tropical depressions form in the eastern Atlantic). Initial transit distance to the surveillance location is roughly 5000 km. On-station loiter time is 14 days (threshold) to 164 days (goal), and is terminated by a tracking task. The tracking task is to "escort" the hurricane for up to 14 days while maneuvering above the hurricane to make measurements and deliver expendable devices. The vehicle then returns to base at its optimum transit altitude and speed. Total transit times are required to be 48 hours or less (24 hours each way, assuming no headwinds), resulting in total airborne endurance of 30 days (threshold) to 180 days (goal). The six-month goal endurance was selected to match the length of the Atlantic hurricane season (June 1 through November 30). The threshold loiter altitude of 21 km enables the vehicle to stay above most storm generated turbulence and to provide a reasonable field-of-view for the sensor package. This requirement was revisited prior to Phase II, and the threshold value modified to 18 km to support the SR requirements and technology study (see Table 24). The dash speed requirement of 151 km/h true airspeed (TAS) is based on worst case 99^{th} percentile winds aloft of 126 km/h plus a fast hurricane movement of 25 km/h. The "99^{th} percentile winds" is a statistical value indicating the wind speed which is exceeded only 1% of the time. Winds aloft data (from the National Climatic Data Center) for numerous points located in the Atlantic hurricane area of interest (see Figure 3) were statistically analyzed to determine the average and 99^{th} percentile winds during the hurricane season. Figure 4 shows the average and 99^{th} percentile winds aloft data for the tropical locations depicted in Figure 3. The worst case 99^{th} percentile wind speed at 21 km in these randomly selected data samples occurred over Kingston, Jamaica in August at 35 m/s (126 km/h). The vehicle is required to operate during the hurricane season (June 1 through November 30) in the latitude range of 10° N to 30° N.

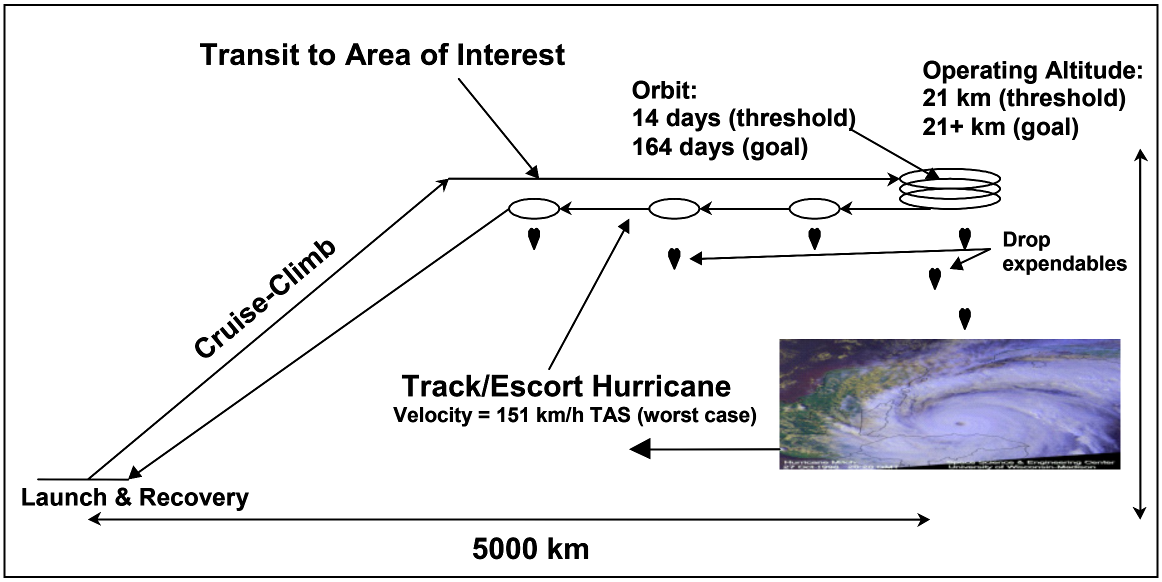


Figure 2. Hurricane science design mission profile.

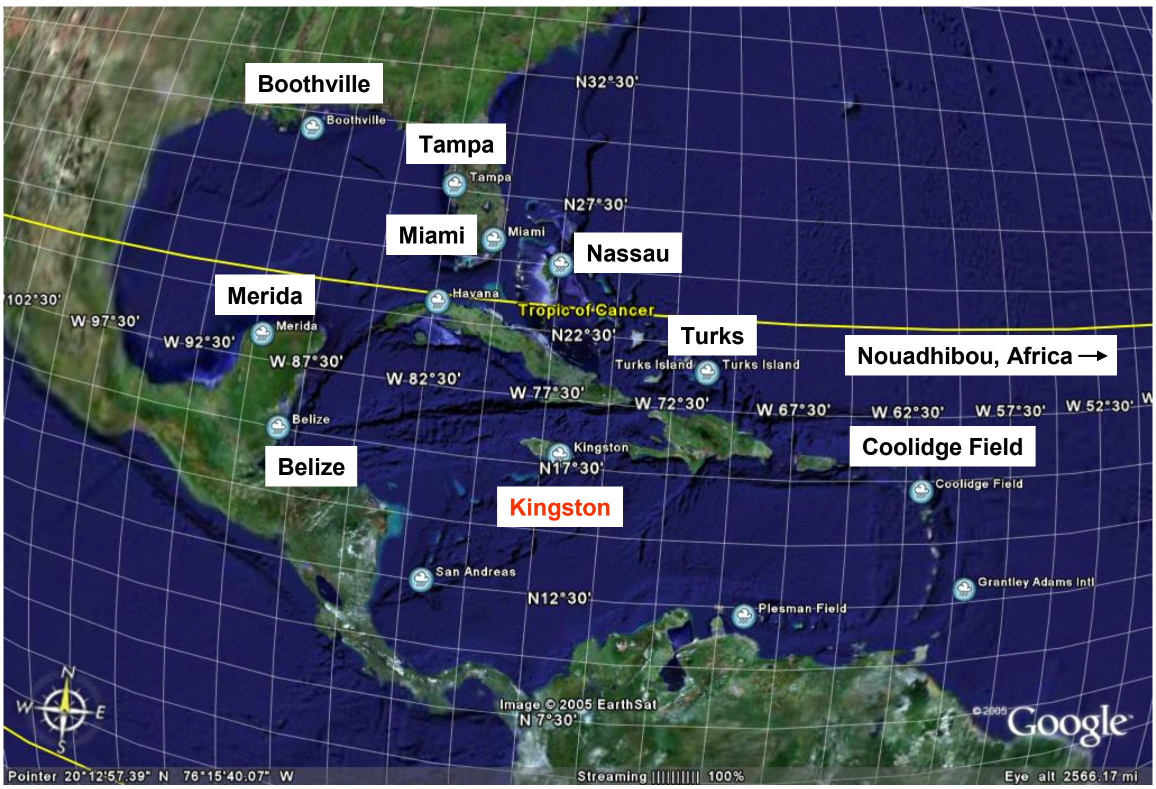


Figure 3. Tropical locations of winds aloft data sampling for hurricane science mission.

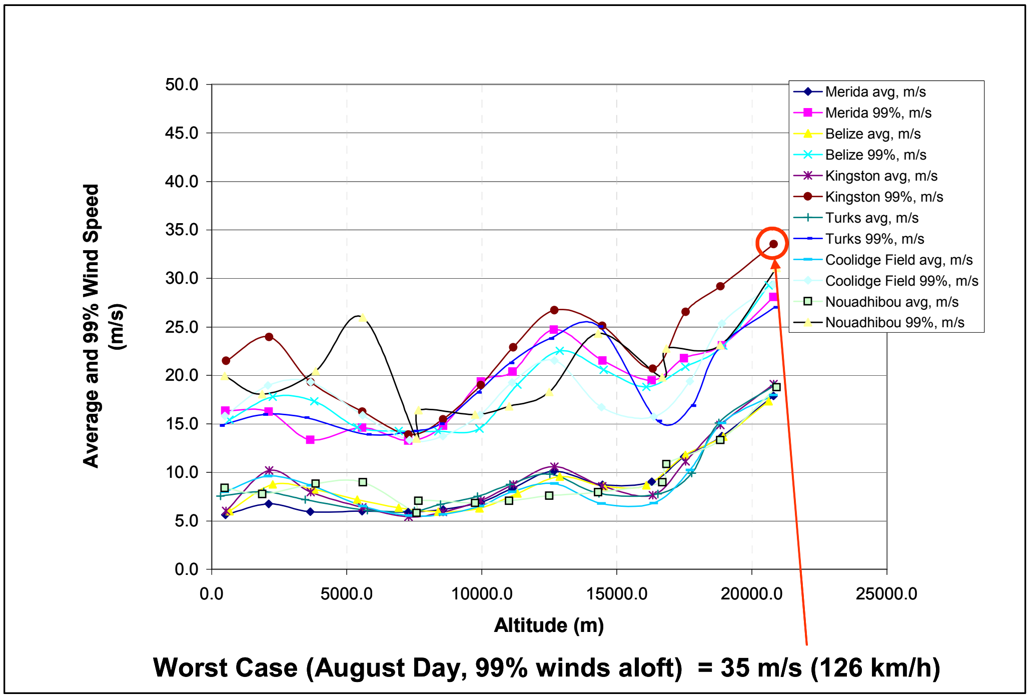


Figure 4. Winds aloft data for hurricane science mission.

Both the Avery white paper (ref. 9) and the ERAST planning document (ref. 10) supported the development of a notional payload for the hurricane science mission, consisting of both fixed and expendable devices. Fixed instrumentation was assumed to include an active Doppler radar, a passive microwave instrument, and an electro-optical/infrared (EO/IR) imaging sensor. The vehicle payload capacity to accommodate these fixed sensors was required to be between 200 kg (threshold) and 500 kg (goal). Power required was estimated to range from 1 kW (threshold) to 2.5 kW (goal). The volume required was 1 m^3 (threshold) to 3 m^3 (goal). The expendable portion of the payload consists of a large number of small dropsondes, and/or small, maneuverable UAVs. The total mass of the expendables could range from 175 kg (threshold) to 350 kg (goal), resulting in a total payload mass requirement of 375 kg (threshold) to 850 kg (goal). The volume required to carry the dropsondes and small UAVs was 4 m^3 to 8 m^3. Therefore, the total required payload volume ranged from 5 m^3 to 11 m^3, respectively. There is a fairly wide range between the threshold and goal payload requirements due to uncertainty during the early stages of the study. As the study progressed, the uncertainty associated with the payload definition was reduced. A subset of these requirements was utilized for the Phase I analysis. Section 5.1 of this report presents a refined set of payload requirements that was used during Phase II of the study.

The communications relay mission profile is shown in Figure 5. The air vehicle launches from the operating base, assumed to be Las Cruces, NM, cruise-climbs to the optimum transit altitude and speed, and transits to the operating location, assumed to be in the northern part of Maine (this location provides worst case combination of transit distance, winds aloft, and latitude for the continental United States (CONUS)). The Las Cruces area is attractive for an operating base due to favorable weather and relative isolation from commercial air traffic. Transit distance from Las Cruces, NM to northern Maine is roughly 3500 km. The mission requirements limit the outbound and return transit times to 24 hours or less (assuming no headwinds). Based on a maximum required total airborne endurance of 14 days (threshold) to 180 days (goal), the minimum resulting on-station loiter time requirement is 12 days (threshold) to 178 days (goal). Once loiter is complete, the vehicle returns to base at the optimum transit altitude and speed. The threshold loiter altitude of 18 km is required to stay above commercial traffic and the jet stream. The goal altitude of 21 km was selected to provide a larger field-of-view and is also compatible with the hurricane science mission loiter altitude requirements. The vehicle must be capable of loitering at any time of year in the latitude range of 25° N to 47° N (Miami to Caribou). The dash speed requirement of 201 km/h TAS is based on worst case 99th percentile winds aloft of 191 km/h plus a small maneuver margin of 10 km/h. Winds aloft data (from the National Climatic Data Center) for southern, middle, and northern CONUS latitudes (see Figure 6) were statistically analyzed to determine the average winds and the 99th percentile winds. Figure 7 shows the average and 99th percentile winds aloft data for the northern points depicted in Figure 6. The worst case 99th percentile wind speed at 21 km in the randomly selected data subset occurred over Caribou, Maine in January at 53 m/s (191 km/h).

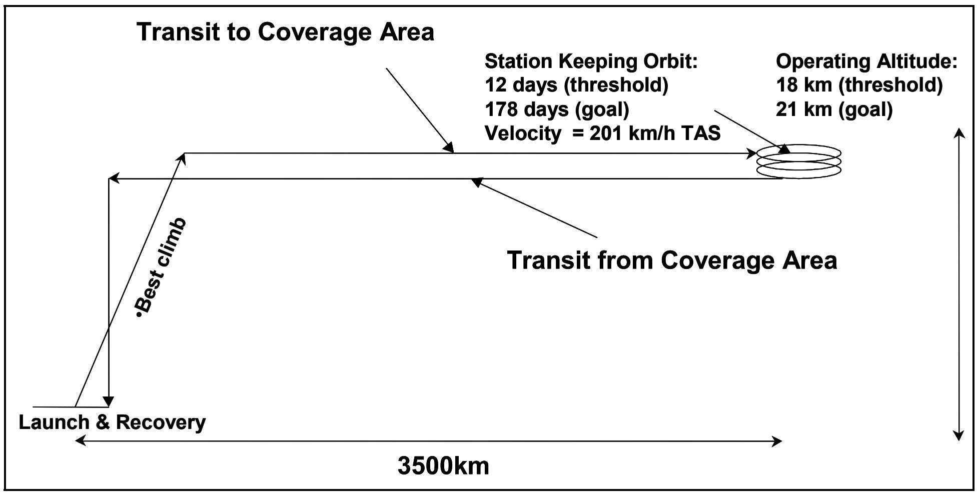


Figure 5. Communications relay mission profile.

Figure 6. CONUS winds aloft data sampling locations.

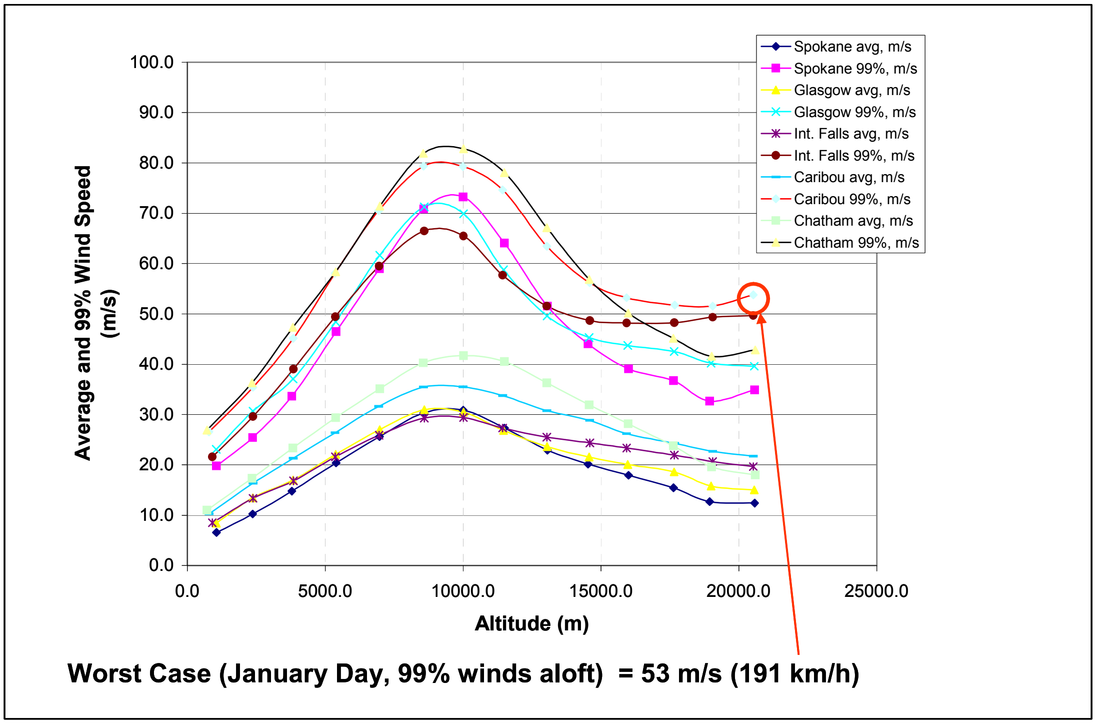


Figure 7. Winds aloft data for selected CONUS locations.

The communications relay payload requirements were derived from a variety of sources (refs. 11-16). There was some variation in payload characteristics among the sources and the requirements presented here are a representative subset of the information gathered. The payload will deliver high frequency communications (28 GHz-48 GHz) since these frequencies support broadband communications such as cell phone service, digital TV, and video-on-demand. A frequency of 48 GHz has been allocated to high altitude platform services by the International Telecommunication Union. The threshold payload mass requirement is 136 kg and the goal is 200 kg. These masses are somewhat larger than communications payloads assumed for previous high altitude aircraft concepts, but lower than those assumed for high altitude airships, which are less sensitive to payload mass. This mass would provide adequate capability for a communications relay platform. The payload power requirement of 1 kW (threshold) to 1.5 kW (goal) is representative of other concepts and would provide adequate capability. The volume requirement of 0.1 m^3 is consistent with both Rockwell Collins projections and Army Communications and Electronics Command (CECOM) information. The communications relay mission requires a consistent orbit and antenna pointing capability which translates into a maneuverability requirement. Based on work by J. Thornton et al. (ref. 15), the vehicle must be able to maintain station within a position cylinder having a 4000 m radius and 3000 m height 99.9% of the time.

For both missions the payloads must be designed to operate in a high altitude solar radiation environment and be protected from the low temperatures, vibration, and acceleration experienced by the vehicle. The expected operational temperature range for high altitude missions is -40°C to +50°C. This represents the broadest range expected based on information from Rockwell Collins, Army CECOM, and the Japanese Pathfinder experiment. The operational pressure range is from sea level to 21 km. Although the payload was unpressurized in the Japanese Pathfinder communications relay experiment, some communication equipment vendors do recommend limited pressurization.

4.0 Analysis of Alternatives

This section describes the process, implementation, and results for the AoA. Section 4.1 presents the AoA scoping activity and development of the evaluation criteria employed in the AoA. The scoping activity is required to identify an appropriate set of concepts for consideration given the large number of potential solution options available. The evaluation criteria are needed to provide a consistent and relevant basis for comparing the various concepts and determining which ones to carry forward to the next phase of study. Section 4.2 describes the analysis tools and processes used, including their modification and validation to support this study. Section 4.3 describes the sixteen concepts that emerged from the scoping effort. The concepts are grouped into HTA and LTA sections, and the HTA section is sub-divided into consumable fuel and SR concepts. Each group is presented through a general discussion of design data that is applicable to the entire group, and then the unique features of each concept in the group are detailed. For the HTA consumable fueled concepts these unique features are associated with the choice of propulsion system. For the HTA SR concepts several unique planform configurations are detailed. Hybrid solar-consumable propulsion system concepts are explored as well in Section 4.3. Except for the "aeroship" concept, all of the LTA designs are based on the same airship configuration, having varying propulsion systems that mirror those used for the HTA concepts. Finally, the concept down select results and rationale are discussed in Section 4.4.

4.1 Scope and Evaluation Criteria

The scope of the AoA included HTA fixed wing configurations, LTA configurations, and hybrid designs. HTA and LTA configurations were both used to perform propulsion system trade studies (focusing on energy storage and energy conversion options). In addition, an HTA configuration trade study for the SR propulsion options was performed. The decision to limit the scope to selected HTA and LTA configurations and propulsion options purposefully omitted other possibilities. This process of eliminating options from consideration was accomplished using the traditional morphological matrix approach. The rows of the matrix represent a functional decomposition of the elements required to form a solution. In the case of aircraft design, row topics could be wing configuration, propulsion, fuel type, recovery mechanism, etc. The columns of the matrix consist of alternatives for each of the row elements. A fully populated matrix contains all of the possible configuration options. Richard Weber and Sridhar Condoor provide an excellent description of this process in reference 17. The decomposition of the Power & Propulsion and Configuration categories is shown in Table 1.

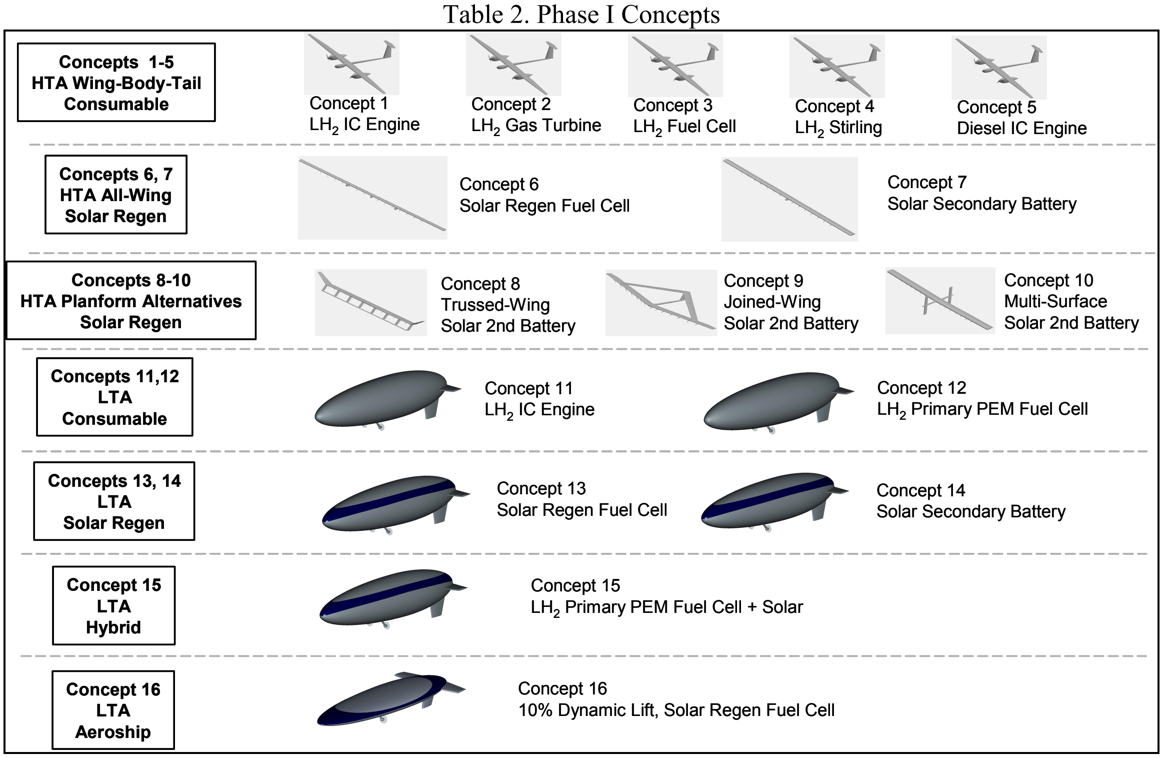
Table 1. Morphological Decomposition Matrix

Power and Propulsion	Power source	None (implies fuel)	Beamed	Solar	Primary Battery	Nuclear	
	Energy conversion	IC Engine	Gas Turbine	None	Photovoltaic + Electric Motor	Stiriling Heat Engine	Fuel Cell + Electric Motor
	Energy storage	None	Altitude	Battery	Fly wheel	Regen. Fuel Cell	Hybrid
	Thrust generation-propulsors	Propellor	Rotary Wing	Jet	Rocket	Flapping wing	
	Auxiliary power generation	Mechanical power extraction	Electrical Power Exctration	Bleed extraction	Stand alone		
	Fuel	None	Hydrocarbon	Hydrogen			
Configuration	Variable Geometry	None	Span	Sweep	Dihedral	Chord	Aux. surfaces
	Rotorcraft	None	Helicopter	Autogyro	Tiltrotor		
	Fixed Wing	None	W-B-T/Canard	Bi-plane	All wing	Three surface	Joined wing
	Airship (LTA)	None	Dirigible	Blimp	Hybrid	Powered Balloons	

During the process of creating and discussing the HALE UAV morphological matrix, many solution alternatives were considered and rejected. For example, rotorcraft appears under the Configuration sub-heading in Table 1, with solution options of helicopter, autogyro, and tiltrotor. Since there was no requirement for VSTOL or hover in the study missions, the added weight and complexity of a rotorcraft solution was undesirable. Other potential solution options considered and rejected during the process were formation flight, tip-joined vehicles, beamed power, nuclear power, powered balloons, and flywheels. These options were rejected due to low TRL, perceived high cost, high risk, or safety concerns.

The configurations selected for Phase I of the study were grouped into three categories; two HTA categories and one LTA category. Table 2 presents the sixteen concepts studied during Phase I. HTA, consumable fuel concepts consisted of high aspect ratio wing-body-tail configurations with multiple propulsion options utilizing either liquid hydrogen (LH_2) or diesel fuel. The LH_2-fueled propulsion options studied were spark ignition (SI) engine, gas turbine engine, proton exchange membrane (PEM) fuel cell, and Stirling cycle heat engine. A diesel-fueled compression ignition (CI) engine option was also included in the HTA consumable concepts. HTA SR planform configurations included all-wing, joined-wing, trussed-wing, and a variable geometry multi-surface arrangement. Two energy storage options were studied, regenerative fuel cells and secondary batteries. The LTA concepts consisted of both consumable fuel and SR airships, plus an "aeroship" that obtains lift from both buoyant forces and aerodynamic forces. Several hybrid propulsion options were explored for both the LTA and HTA configurations, including several combinations of solar arrays with consumable fuel systems. HTA hybrid propulsion concepts were not included in the AoA for reasons that will be detailed in Section 4.3.3.

Table 2. Phase I Concepts



The relative merits of the sixteen concepts were compared using a set of metrics developed as evaluation criteria. They are summarized below:

1. *Endurance, days* – total mission endurance for both the hurricane science mission and the communications relay mission. Endurance is the key performance parameter and highest priority requirement.

2. *Takeoff Gross Mass, kg* – takeoff gross mass for both missions, this metric is a good indicator for the size and cost of the concepts.

3. *Wingspan (HTA) or Length and Width (LTA)*, *m* – indicative of ground handling challenges.

4. *Volume,* m^3 – applies to the LTA concepts only. To ensure compatability with existing airship hangar facilities a volume constraint of 415,000 m^3 was assumed.

5. $\%P_{regen}$ – percentage of the total power required supplied by the SR propulsion system on the worst day of the mission (worst day for SR concepts implies shortest length of daylight, lowest sun angles, highest wind, etc.). 100% indicates that the system is energy balanced for the day/night cycle. This metric was required to compare SR concepts for which there were no combinations of input parameters resulting in a viable concept.

6. *Takeoff and Landing Robustness %* – percentage of the mission timeframe (hurricane science mission is June through November, communications relay mission is year-round) that the concept can takeoff and land from its operating base factoring in cloudiness and ground level winds (LTA ground operations were constrained to times of winds less than 2.5 m/s).

7. *Ground Footprint* – this metric contains two parts, spot factor and support required. Spot factor is a measure of the vehicle overall size and ground footprint. It is determined by imagining a rubber band stretched around the top view of the vehicle and calculating the area enclosed. This area is divided by the area of a reference vehicle assigned a spot factor of one. The Global Hawk RQ-4B was utilized as the reference vehicle (the RQ-4B is a new derivative of the Advanced Concept Technology Demonstrator RQ-4A with a larger payload, wingspan, and improved performance). For example, a HALE UAV concept with spot factor = 2 has twice the enclosed area as compared to the RQ-4B. Support required is a subjective rating of the amount of ground support equipment and crew required to operate the vehicle. Once again, Global Hawk was selected as the reference vehicle and assigned a value of 0. A qualitative rating scale ranging from minus three (much better than RQ-4B) to three (much worse than RQ-4B) was used to score the following categories: fuel handling; ground crew size; propulsion system uniqueness and complexity; hangaring; maintenance requirements; deployability; and safety. The scores were summed for each concept, with a score greater than zero being worse than RQ-4B and less than zero better.

8. *Growth Factor* – the number of kilograms the overall configuration mass grows to achieve the same performance with the addition of one extra kilogram of zero fuel mass. This factor indicates the relative sensitivity of each concept to mass growth during development. Lower growth factor is better, indicating a more robust design.

9. *Risk* – a subjective estimate of the overall vehicle development and operational risk, again referenced to the Global Hawk RQ-4B. A qualitative rating scale ranging from minus three (much less risky than RQ-4B) to three (much riskier than RQ-4B) was used to score the following categories: structure and materials; propulsion system; subsystems; vehicle integration; and test program. The risks in each category were then summed, and concepts with total risk scores less than or equal to Global Hawk (<=0) were assigned a color code of green, concepts with total risk scores between one and six were assigned yellow, and concepts with total risk scores equal to or greater than seven were assigned red.

For each concept analyzed, a table was generated summarizing these metrics and the color coding was applied to the estimated risk levels (see Table 5 as an example).

4.2 Tools and Processes

This section describes the two primary conceptual design tools used in this study; a HALE Multi-disciplinary Design Optimization (MDO) code for the HTA concepts, and an Airship Design and Analysis Code (ADAC) for the LTA concepts. Various enhancements and modifications made to the HALE MDO code analysis environment and the ADAC source code are described below, as well as code validation information. The process used to conduct the AoA is also briefly summarized.

The primary tool used to design and analyze the HTA vehicle concepts was a HALE MDO code developed by AeroVironment Inc. and delivered to NASA Dryden Flight Research Center under contract in June of 2004. This HALE MDO code provides high-level, conceptual analysis and sizing of lightweight, low wing loading aircraft designed specifically for HALE missions. The code has the capability to address both consumable and SR propulsion systems. Consumable systems (such as combustion engines or fuel cells) with either conventional fuel (gasoline, diesel) or liquid hydrogen can be modeled. Regenerative energy storage options include a secondary battery or a fuel cell with electrolyzer. Solar aircraft analysis is supported through calculation of incident solar energy based on latitude, time of day, and time of year. Given appropriate user inputs, the code estimates total vehicle mass and overall aerodynamic and propulsion system performance. For consumable propulsion concepts, the performance analysis is conducted at various climb and loiter flight conditions and the total endurance is calculated. For SR systems, the energy balance (solar energy in versus energy out) is analyzed and the state-of-charge of the energy storage system is determined at the end of one diurnal cycle (24 hours).

The HALE MDO code was validated for use in this HALE concept study with data from existing vehicles and past conceptual design studies. The Scaled Composites Voyager aircraft was used as one of the consumable fuel validation cases. An input deck of aerodynamic, mass, and propulsion data from Voyager was created and the around-the-world mission was modeled. This around-the-world flight had an endurance of 9 days and covered a distance of 42,400 km at an average altitude of 3353 m. The code is limited to the mission profile segments of climb and fixed altitude cruise or loiter. It was difficult, therefore, to model the actual cruise-climb flight profile of Voyager. In addition, the code is not able to model the shutdown of one engine during the mission as done during the Voyager flight. After SFC and cruise altitude adjustments were made to overcome these limitations, the code predicted an endurance of 8.2 days and a range of 40,400 km, a 9% under prediction of endurance and 5% under prediction of range. Given the approximations necessary to model the Voyager mission, these results were considered acceptable. In addition to validating overall performance estimates, outputs from individual subroutines of the code were also compared to existing vehicles and results of other design studies. Data used to evaluate accuracy of the subroutines included data from the Boeing Condor HALE UAV, the AeroVironment Pathfinder and Helios solar HALE UAVs, and a HALE propulsion study conducted by Boeing (ref. 6). For cases in which significant discrepancies were found between the code output and other data sources (e.g., fuselage/pod mass, propeller mass, cooling drag), appropriate calibration factors were determined and applied for the study analyses.

The HALE MDO code was designed with the intention of using an external sizing-optimization driver to iterate the configuration and meet specified design constraints. At NASA Langley the code was integrated into a commercially available software integration framework. Custom models and user interfaces were developed separately for consumable fuel and SR concepts based on the required inputs. Figure 8 shows the analysis model for consumable fuel concepts. An external driver component called "Power_Wtsizing" was written to run the code iteratively and size the power plant, find the appropriate propeller RPM inputs to match power required, set the tank boil-off rate (LH_2 fuel only), and converge the vehicle mass while meeting design constraints. "Power_Wtsizing" can also be used to apply technology or calibration factors to adjust code-estimated component masses. Trade study and optimization features of the integration framework can be executed to explore the design space and optimize design variables.

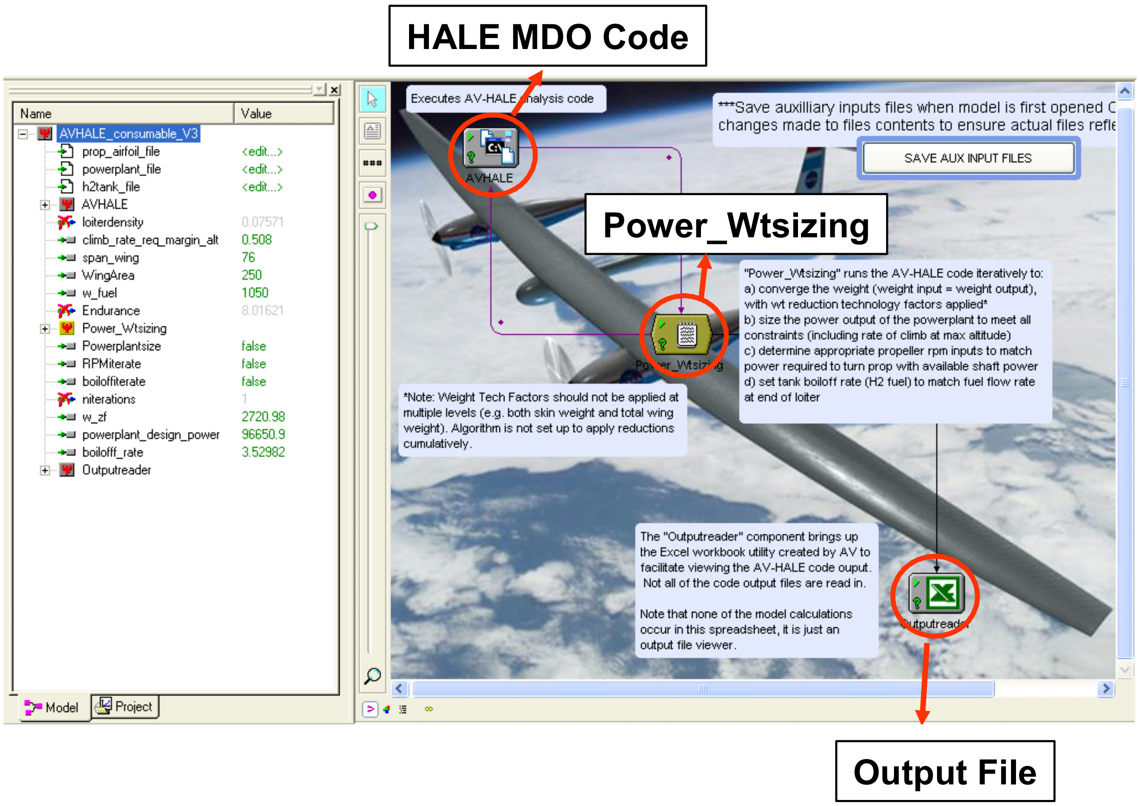


Figure 8. Integrated HTA, consumable fuel MDO model.

Analysis models for the SR concepts contain additional components not needed for consumable fuel cases. A representative SR model with a regenerative fuel cell propulsion system is shown in Figure 9. A program "ArraySum" was written to extend solar energy calculations to non-horizontal and movable solar arrays. This program steps through a 24-hour period and sums the total incident solar radiation for each array on the aircraft. Array surfaces can be defined with any tilt relative to the aircraft, or the tilt angle can be optimized at each time step to maximize the amount of energy collected. Effects of aircraft heading and bank angle are also included in the solar radiation calculations. The "ArraySum" program does not calculate the actual power and energy output of the aircraft arrays considering solar cell efficiency, etc. Instead, it determines approximate power and energy multiplication factors for the complete array system compared to a flat, wing-only array. These factors are then applied to the HALE MDO code internal solar energy calculations. The solar radiation analysis in "ArraySum" is performed with the solar position and intensity utility, "SOLPOS," from the Department of Energy's National Renewable Energy Laboratory (ref. 18). "Power_Wtsizing" converges the vehicle mass and sizes the propulsion system components (electric motor, propeller, and fuel cell if present) to meet mission constraints. "Power_Wtsizing" also sizes the amount of H_2 reactant carried (energy storage capacity) for regenerative fuel cell systems to match the energy storage required. For the regenerative fuel cell model, as shown in Figure 9, the "Electrolyzersizer" component iterates the electrolyzer size (number of cells) to determine the minimum size required to achieve an energy-balanced system, or if that is not possible, the size that minimizes the gap between available energy and required energy. In the case of battery energy storage, a “Batterysizer” component replaces “Electrolyzersizer” and optimizes the battery mass (energy storage capacity) to achieve an energy balance or a minimum energy deficit.

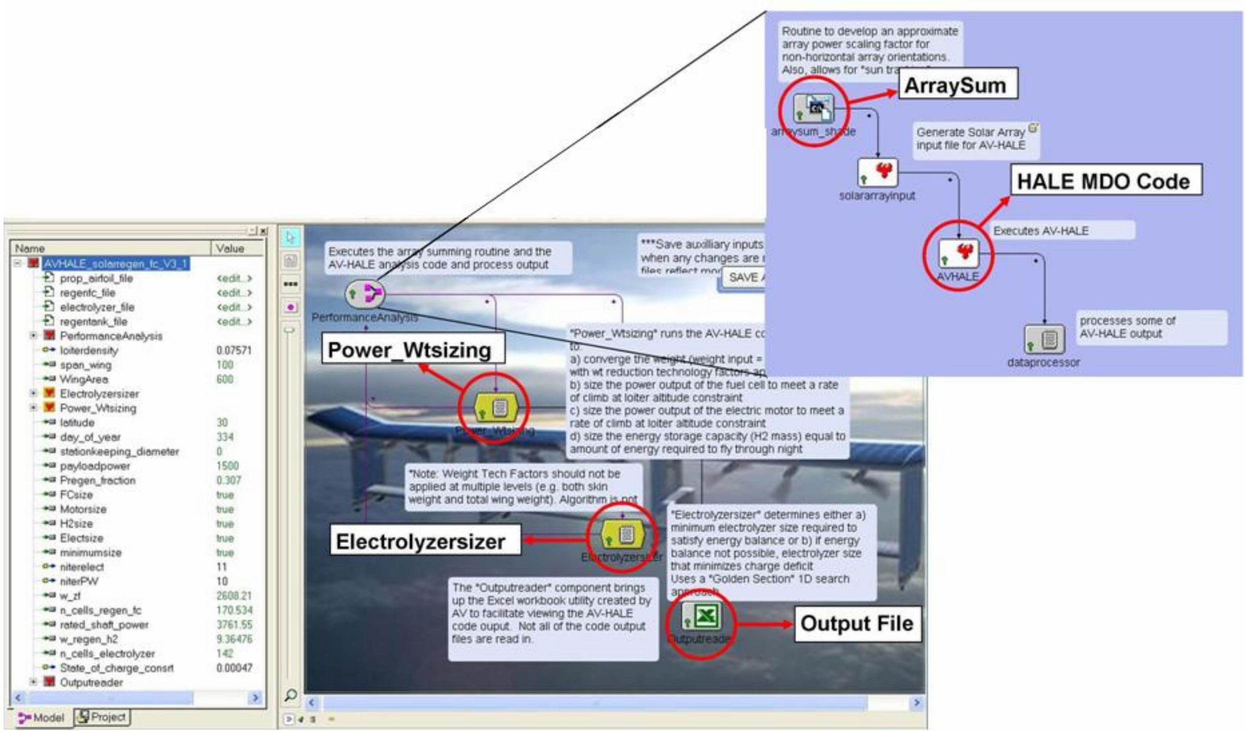


Figure 9. Integrated HTA, SR MDO model.

The primary tool used to design and analyze the LTA vehicle concepts was the Airship Design and Analysis Code (ADAC) (ref. 19). ADAC is capable of performing rapid, vehicle level feasibility studies for HALE airship vehicles. ADAC was specifically designed to assess the feasibility of long endurance LTA vehicles required to perform station-keeping missions at altitudes between 16 and 22 km. Written utilizing Microsoft Excel®/Visual Basic, this code provides a graphical user interface through the Excel® application. The user is prompted to enter a range of input parameters, including wind speed data; type of lifting gas; altitude, payload and airspeed requirements; and propulsion system type. The code then utilizes a unique algorithm to size the airship. This algorithm determines the volume of the gas envelope required to lift the payload only, then adds the mass of the fabric required to enclose the gas volume. Masses related to the hull volume or fabric area such as the suspension, stiffening, tail structure, and the ballonet are calculated and added to the fabric mass to determine a basic hull mass. Next, the propulsion system is sized based upon required thrust and mission energy to meet the airspeed requirements. The updated mass of the propulsion system is considered as additional payload mass, but tracked separately. The algorithm calculates a new lifting gas volume required to lift the updated payload and basic hull mass and then continues this iterative process until the hull volume convergence criterion is met. The converged solution provides the user with the design volume, envelope area, solar cell area, geometric parameters, mass parameters, and mission performance.

ADAC has been validated and calibrated at low altitudes using existing blimp data (ref 19). Extrapolation to high altitude long endurance missions involves characterizing the likely advanced materials, power systems, and structural needs. This challenge was met by consulting discipline experts for perspective and guidance on choosing and implementing parameterized models of the systems and technologies. As a result, ADAC has undergone important enhancements over the course of this study to include a wider range of power systems, to allow the addition of solar cells to consumable fuel systems, to add aerodynamic lift to buoyant systems, and to consider the hydrogen production rate limitations of electrolyzers supplied with excess solar power. Considering important effects such as the extreme cold, UV exposure, lifting gas leakage rates, and unattended engine operation, remains, however, outside the scope of ADAC.

The process used to conduct the AoA was characterized by a series of steps to refine and narrow the field of alternatives. Initial configurations were selected using the morphological matrix process described previously. Sketches were made using a specialized rapid geometry modeling software, Vehicle Sketch Pad (ref. 20). Those sketches were then sized using the analysis codes discussed above, and initial results were provided to discipline experts in the areas of propulsion, aerodynamics, subsystems, and structures. Feedback from the discipline experts in the form of modified input parameters was incorporated to produce a configuration update. The AoA metrics presented in Section 4.1 were then assessed for all sixteen updated configurations. The three best performing configurations (one HTA consumable, one HTA SR, and one LTA) were then selected for Phase II.

The initial requirements included both threshold and goal values for payload mass, payload power and loiter altitude. To simplify the Phase I sizing studies, discrete values within these ranges were utilized. For the hurricane science mission, 400 kg, 1.5 kW, and 21 km were assumed for payload mass, payload power, and loiter altitude, respectively. For the communications relay mission, 200 kg, 1.5 kW, and 18 km were assumed for payload mass, payload power, and loiter altitude, respectively.

4.3 AoA Concept Descriptions and Results

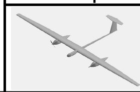
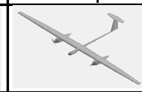
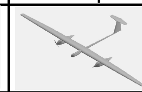
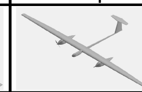
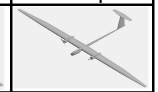
This section details the sixteen Phase I concepts including geometry, propulsion, aerodynamics, mass, and performance data. The concept descriptions are grouped into four sections; Section 4.3.1 presents the five HTA consumable fuel concepts; Section 4.3.2 presents the five HTA SR concepts; Section 4.3.3 contains a discussion of HTA hybrid concept options; and Section 4.3.4 presents the six LTA concepts.

4.3.1 Heavier-Than-Air Consumable Fuel Concepts (Concepts 1-5)

The primary discriminator between the five HTA concepts described in this section is the power and propulsion system. This section begins with a discussion of geometry, aerodynamic, mass and performance information for all five concepts. The power and propulsion data are then presented separately for each concept.

The five concepts analyzed in the HTA consumable fuel category utilized a relatively conventional wing-body-tail configuration. This vehicle has a very high wing aspect ratio of approximately 25, a large wingspan of 80 m, and twin engines contained in two wing pods. Table 3 presents the primary geometric parameters for the first five concepts. Consistent with the FAA wingspan limit, wingspan was constrained to 80 m or less; all five concepts optimized at the maximum span. Wing areas ranged from 250 m^2 to 267 m^2, resulting in wing aspect ratios in the range of 24-26. The most demanding mission for these concepts was the hurricane science mission, and all data shown in this section are for vehicles sized for that mission. The communications relay mission performance was obtained by analyzing the hurricane science mission vehicle on the communications relay mission profile with 200 kg less payload.

Table 3. Primary Geometric Parameters for Concepts 1-5

			Concept 1	Concept 2	Concept 3	Concept 4	Concept 5
							
Parameter	**Symbol**	**Units**	IC Engine LH_2 Fuel	Gas Turbine LH_2 Fuel	PEM Fuel Cell LH_2 Fuel	Stirling engine LH_2 Fuel	IC Engine Diesel Fuel
Wingspan	b_s	m	80	80	80	80	80
Wing Area	S_w	m^2	250	250	260	247	267
Wing Aspect Ratio	AR_w	-	25.6	25.6	24.6	25.9	24.0
Wing Loading	W/S	kg/m^2	18.5	17.9	18.9	17.9	19.1
		lb/ft^2	3.77	3.60	3.84	3.58	3.82
Propeller Diameter	D_p	m	4.2	4.2	4.4	4.1	4.3
Pod/Nacelle Separation	P_{sep}	m	25	25	25	25	25
Pod/Nacelle Length	Pod_L	m	11.8	11.8	10.9	10.5	8.4
Pod/Nacelle Diameter	Pod_D	m	2.9	2.9	2.7	2.6	2.1
Fuselage Length	Fus_L	m	29	29	29	29	29
Fuselage Diameter	Fus_D	m	1.6	1.6	1.6	1.6	1.6
Horiz. Tail Span	b_{ht}	m	10.6	10.6	11.1	10.5	11.4
Horiz. Tail Area	S_{ht}	m^2	22.6	22.6	24.6	22	26.1
Horiz. Tail Aspect Ratio	AR_{ht}	-	5	5	5	5	5
Vert. Tail Span	b_{vt}	m	6.7	6.7	7.0	6.6	7.22
Vert. Tail Area	S_{vt}	m^2	22.6	22.6	24.6	22.0	26.1
Vert. Tail Aspect Ratio	AR_{vt}	-	2	2	2	2	2

The aerodynamic analysis subroutine in the HALE MDO code contains a mix of empirical and analytical handbook methods. It was validated using airfoil test results and data from the AeroVironment Pathfinder. The reasonableness of the empirical airfoil model was also checked using the Wortmann FX-67-K170 (t/c = 0.17) as a benchmark. Overall, the differences between the HALE MDO code airfoil results and the Wortmann section data indicate the HALE MDO code results represent a more conservative, robust design which can still achieve high performance. A span efficiency of 0.88 was assumed for induced drag calculations, accounting for the aerodynamic interference of the two nacelles/pods and the fuselage with the wing. Figure 10 shows the full vehicle drag polars at loiter conditions for all five concepts. Concept 5 has the lowest drag at the cruise C_L of 1.0 and the smallest C_{D0}. For Concept 5 the diesel fuel is stored in wing tanks resulting in much smaller engine nacelles since they do not have to accommodate large LH_2 tanks as in the other four concepts. Figure 11 shows the variation of L/D with C_L for all five concepts. At the loiter C_L of 1.0, L/D ranges from 34.5 to 36.5. Concept 5 has the highest L/D over the range of lift coefficients which would be expected during loiter, climb, or transit operations.

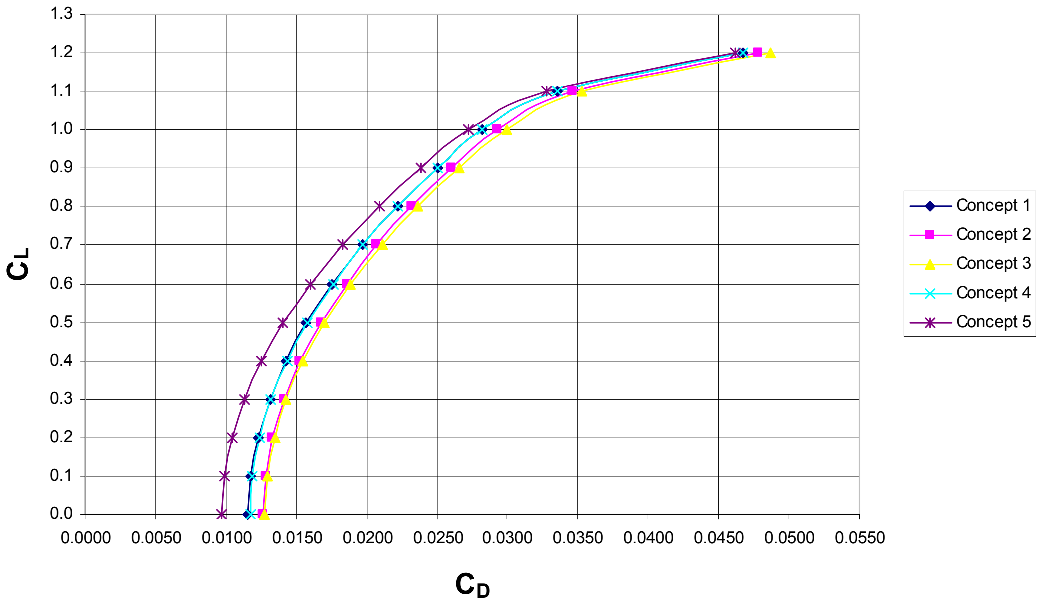


Figure 10. Drag polars for Concepts 1-5 at loiter conditions, Re~1.1M.

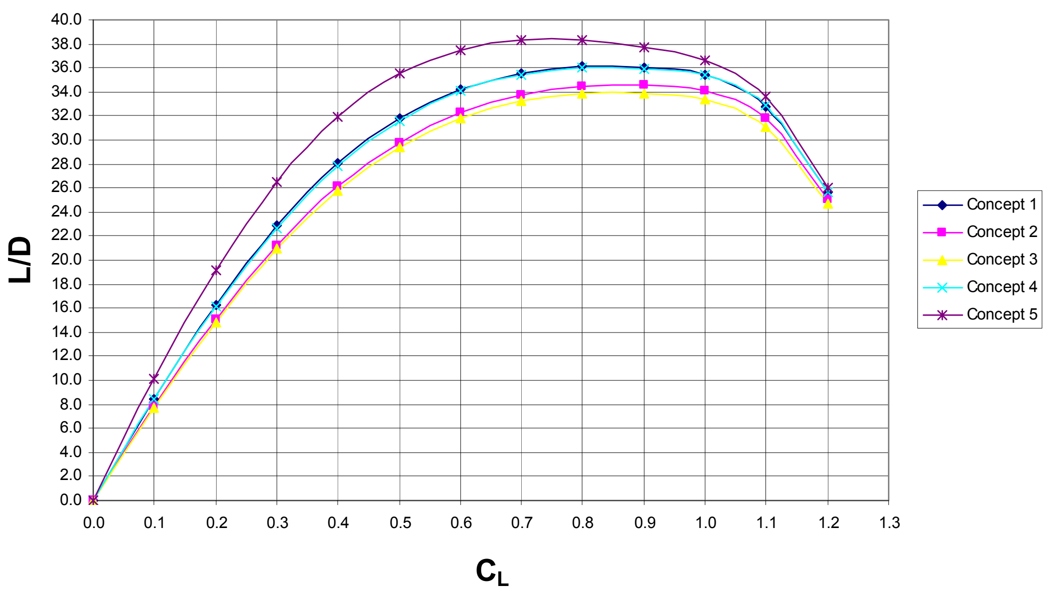


Figure 11. L/D versus C_L for Concepts 1-5 at loiter conditions, Re~1.1M.

Determining a mass estimate for these concepts required making assumptions for the payload mass, fuel mass, subsystem mass, and structural sizing criteria. The structural sizing criteria listed below were used to estimate the amount of material required for the wing, fuselage, and tail structure assuming carbon-epoxy composite construction.

- Wing tip maximum deflection = 25% of wing semi-span
- Aileron effectiveness fraction at cruise velocity (at wing tip) = 0.50
- Positive uniform gust load factor = 3.8 g

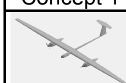
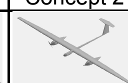
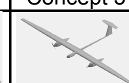
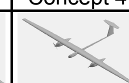
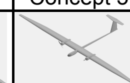
- Negative load factor for taxi bump = 1.5 g
- Wing structure factor of safety = 1.5
- Tailboom maximum deflection = 5% of tailboom length
- Mounting/Installation mass = 6% of component mass

In addition to the uniform load factors listed above, the HALE MDO code also considers both positive and negative sinusoidal gust loads. Subsystems assumptions are listed below:

- Avionics power required = 300 W
- Avionics mass = 15 kg
- Backup battery mass = 12 kg
- Servo mass = 1% of takeoff mass
- Landing Gear mass = 1.8% of takeoff mass

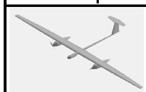
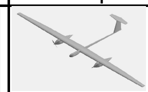
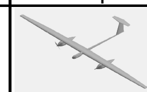
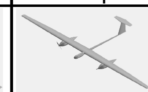
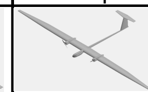
As discussed previously, payload mass was set to 400 kg for these concepts. The parametric study capability of the model was utilized to vary fuel mass, wing span, and wing area to find the maximum endurance subject to a wingspan constraint of 80 m. Given the wingspan constraint, there is an optimum point beyond which additional fuel results in a heavier vehicle with less endurance. As the wing area is increased to lift the extra weight of the fuel and fuel system, the span constraint causes a decrease in wing aspect ratio, resulting in a reduction of L/D and reduced endurance. Mass data for Concepts 1-5 sized for the hurricane science mission are presented in Table 4.

Table 4. Mass Data for Concepts 1-5

	Concept 1	Concept 2	Concept 3	Concept 4	Concept 5
					
Component Masses (kg)	IC Engine LH_2 Fuel	Gas Turbine LH_2 Fuel	PEM Fuel Cell LH_2 Fuel	Stirling engine LH_2 Fuel	IC Engine Diesel Fuel
STRUCTURES:					
Wing	838	822	826	819	903
Tail (includes horizontal and vertical)	83	82	86	80	97
Tailboom	126	124	131	122	137
Fuselage/body	55	54	55	53	56
Landing Gear	87	81	89	79	92
Pods/nacelles	398	386	339	305	204
TOTAL STRUCTURE:	1587	1549	1526	1458	1489
PROPULSION:					
Engines, Turbochargers, Radiators	673	343		834	580
PEM Fuel Cell and Motors			1138		
Engine Accessories, Mounts	184	167	194	168	75
Propellers	85	87	93	83	89
Fuel system	303	291	264	220	68
TOTAL PROPULSION:	1245	888	1689	1305	812
EQUIPMENT:					
Servos	48	45	49	44	51
Wire/Electrical	81	81	81	81	81
Avionics	15	15	15	15	15
Backup Battery	12	12	12	12	12
TOTAL EQUIPMENT:	156	153	157	152	159
(STRUC + PROP + EQUIP) EMPTY MASS:	2988	2590	3372	2915	2460
Payload	400	400	400	400	400
Zero Fuel Mass	3388	2990	3772	3315	2860
Fuel	1440	1490	1150	1100	2250
Takeoff Gross Mass	**4828**	**4480**	**4922**	**4415**	**5110**

Takeoff mass for the hurricane science mission ranged from a low of 4420 kg for the Stirling engine Concept 4 to a high of 5110 kg for the diesel-fueled Concept 5. Takeoff masses for the communications relay mission are 200 kg less than shown in Table 4, reflecting the reduced payload mass required.

Table 5. Comparitive AoA Metric Results for Concepts 1-5

			Concept 1	Concept 2	Concept 3	Concept 4	Concept 5
							
			IC Engine LH_2 Fuel	Gas Turbine LH_2 Fuel	PEM Fuel Cell LH_2 Fuel	Stirling engine LH_2 Fuel	IC Engine Diesel Fuel
Evaluation Criteria	Endurance, days	Hurricane Science	7.9	6.3	7.6	5.0	5.7
		Communications Relay	10.0	9.1	9.9	5.8	6.5
	TOGM, kg	Hurricane Science	4830	4480	4920	4420	5110
		Communications Relay	4630	4280	4720	4220	4910
	Wingspan (HTA) or Length and Width (LTA), m		80	80	80	80	80
	Volume (LTA), m^3		n/a	n/a	n/a	n/a	n/a
	$\%P_{regen}$ (SR)	Hurricane Science	n/a	n/a	n/a	n/a	n/a
		Communications Relay	n/a	n/a	n/a	n/a	n/a
	Takeoff and Landing Robustness %	Hurricane Science	97	97	97	97	97
		Communications Relay	97	97	97	97	97
	Ground Footprint	Spot Factor	4.1	4.1	4.1	4.1	4.1
		Support Required	6	6	7	7	3
	Growth Factor		7.6	8.2	9.4	6.7	8.5
	Risk		2	2	5	5	1

Results for the metrics described in Section 4.1 are presented in Table 5 for each of the five concepts. The LH_2 intermittent combustion (IC) engine-powered Concept 1 had the maximum endurance, followed closely by the PEM fuel cell-powered Concept 3. Concept 4 (LH_2-fueled Stirling engine) had the least endurance. None of the concepts met the threshold endurance requirement for either mission. As discussed previously, all five concepts optimized at the maximum allowable wingspan of 80 m. This large span drove the spot factor for these concepts to over 4, indicating that these vehicles require approximately four times the hangar area compared to the Global Hawk RQ-4B. The scores for the qualitative "support required" metric show the diesel-fueled Concept 5 as the most desirable; however, all of these concepts would require more support than Global Hawk due to their large sizes and unconventional propulsion systems. The concepts utilizing relatively unproven propulsion systems (PEM fuel cell and Stirling engine in aircraft applications) have the highest risk, whereas the lowest risk concept is the diesel-fueled Concept 5. Additional details on the assumptions and results for each concept are presented below.

Concept 1 - LH_2-Fueled Spark Ignition Intermittent Combustion Engine

Concept 1 has a wing aspect ratio of 25.6, a wingspan of 80 m, and twin engines contained in two wing pods which are each sized by the spherical LH_2 tank diameter (see Figure 12).

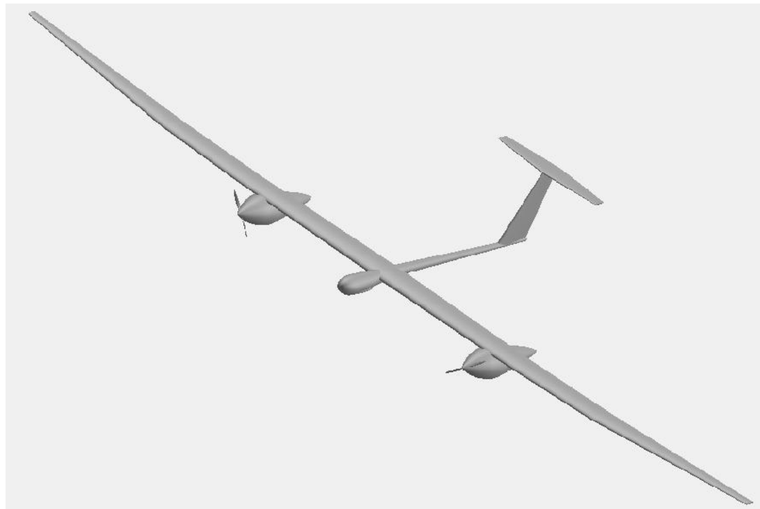


Figure 12. Concept 1: LH_2-fueled propulsion system.

This concept utilizes a spark ignition, IC engine, fueled with LH_2. The two primary metrics of interest for the propulsion system (specific fuel consumption (SFC) and specific power (SP)) were estimated using actual engine data with adjustments for the 21 km operating altitude and the use of LH_2 fuel, as discussed below.

Multiple stages of turbocharging are required for operation at 21 km due to atmospheric pressure lapse. In addition to the turbocharging, intercooling and aftercooling is required for the compressed air stream. Due to the low air density at loiter conditions, these components are larger than their lower altitude counterparts. Although operation of an IC engine at high altitude is challenging, past efforts have been successful. During the NASA ERAST program of the 1990's, significant progress was made with an operational aircraft using a doubly turbocharged Rotax 912 capable of producing 43.3 kW at 19.8 km. Another aircraft capable of long endurance, high altitude operation was the Boeing Condor developed and flown during the 1980's. The Condor used two six-cylinder Teledyne Continental Motors engines each producing 131 kW at 19.8 km. Without a detailed power plant analysis, adding up the masses of the major individual components of a HALE power plant can significantly under-predict total system mass. Additional components beyond the core engine and turbomachinery include the heat rejection system (radiators), lubrication system, gearbox, and other accessories necessary for operation. Examining the ERAST Rotax 912 system in detail, the core engine and turbomachinery masses account for only 65% of the total system installed mass, while on the Boeing Condor these items consist of only 63% of the total installed mass. The core engine mass accounts for only 48% of the Rotax system mass and 31% of the Boeing Condor system mass. This historical information was utilized as a guide to estimate the ancillary mass required for the concepts in this study.

As previously mentioned, the available data (Boeing Condor and ERAST Rotax 912) were for systems designed for operation at 19.8 km. Since the hurricane science mission requires operation at 21 km, the feasibility of operating a 2-stage turbocharged system at 21 km was evaluated. For a 2-stage system at 19.8 km, compressing air to 101.4 kPa at the inlet requires each stage to have a pressure ratio of 4.23. At 21 km the pressure ratio increases to 4.63. Although these turbocharger pressure ratios are high, higher pressure ratios were demonstrated during the Boeing Condor program. In fact, the Condor propulsion system included a compressor with a pressure ratio capability of over 5.7:1 (ref. 21).

In addition to adjustments for operating altitude, adjustments must be made to account for the use of hydrogen fuel. Unlike conventionally fueled SI or CI engines, where the liquid fuel is sent into the cylinders and then vaporized, hydrogen must be sent into the engine in gaseous form due to its extremely low boiling point. This hydrogen gas displaces the air in the cylinders, effectively reducing air mass flow for the given geometric volume. This air mass flow reduction accounts for approximately a 10% reduction in maximum power. Hydrogen fuel also has a very fast flame propagation speed (~5 times that of gasoline) and thus equivalence ratios greater than 0.65 produce detonation. For this reason, hydrogen-fueled IC engines must operate at a very lean equivalence ratio. This equivalence ratio reduction causes approximately a 30% power reduction. Because of these two factors, hydrogen-fueled IC engines typically produce only about 60% of the power that their gasoline-fueled counterparts produce. In order to compensate for this power reduction, either a larger power plant (i.e., more core displacement) or increased inlet pressure is needed. Although using hydrogen reduces the power output of a given size engine, the benefit of hydrogen fuel is the large reduction in SFC associated with the higher specific energy (W-h/kg) of hydrogen compared to hydrocarbon fuels. For long endurance missions, the additional power plant mass required for hydrogen-fueled propulsion is more than offset by the reduction in fuel mass required.

Table 6 shows the progressive weight adjustments made to the Rotax 912 system to account for the unique requirements of Concept 1. The first column is the baseline 19.8 km Rotax 912, the second column is the same IC engine scaled to operate at 21 km, and the third is the 21 km engine scaled to operate with excess air for use as a hydrogen power plant. The resulting mass growth is reflected in the decreased SP (from 387 W/kg to 341 W/kg).

Table 6. Mass Adjustments Required for 21 km Operation and Hydrogen Fuel

Description (all masses in kg)	At 19.8 km	With growth to 21 km	21 km with H_2 mass flow
Rotax 912 Engine Core	53.5	53.5	53.5
Low Pressure Turbo	13.6	16.3	22.7
Low Pressure Intercooler	1.5	1.7	2.9
High Pressure Turbo	5.4	6.4	7.3
High Pressure Intercooler	2.3	2.8	4.6
Exhaust Manifold	4.5	4.5	4.5
Intake Manifold	5.4	5.4	5.4
Turbo Air Inlet	0.79	0.79	0.79
Oil Pump Stack	2.7	2.7	2.7
Oil Tank (dry)	2.6	2.6	2.6
Oil Cooler	0.66	0.66	0.66
Oil (in Cooler)	0.15	0.15	0.15
Coolant Radiator	2.5	2.5	2.5
Coolant (in Radiator)	0.54	0.64	0.64
Primary Alternator	3.9	3.9	3.9
Secondary Alternator	3.9	3.9	3.9
IC Servo and Mechanism	2.3	2.3	2.3
AC Servo and Mechanism	2.3	2.3	2.3
Wiring Harness	2.2	2.2	2.2
Fluids in Oil, Fuel and Coolant Lines	1.1	1.1	1.1
Total	112	116	127
Specific Power	387 W/kg	373 W/kg	341 W/kg
Engine Core % Total	47%	46%	42%
Air System % Total	30%	33%	38%
Misc. % Total	22%	21%	19.5%

In addition, there is an accompanying loss of power due to the reduced equivalence ratio issue, resulting in a final SP estimate for the LH_2-fueled SI IC engine of 222 W/kg, with a SFC of 80 g/(kW-h). Table 7 summarizes the propulsion inputs utilized for this concept. The turbocharing system results in a flat-rated power output; that is, no variation in power with altitude. Due to a lack of detailed engine performance data, SFC was assumed to be essentially constant throughout the 50% to 100% power range and also invariant with altitude. Waste heat rejection can be an issue given the extremely low air density at 21 km, and heat exchanger volumetric sizing, although not addressed in detail for this study, is an important consideration.

Table 7. Propulsion Inputs for Concept 1 (LH_2-Fueled IC Engine)

Specific Power = 222 W/kg		
H_2 Fuel Lower Heating Value = 33300 W-h/kg		
Altitude (m)	Fraction of rated sea level power	SFC g/(kW-h)
0	0.5	80
	1	80
21000	0.5	80
	1	80

The hurricane science mission total endurance for Concept 1 was 7.9 days, with a 253 km/h true air speed during loiter. These figures were 10.0 days and 197 km/h for the communications relay mission. With an 80 m wingspan, and a 29 m total length (fuselage + tailboom), this vehicle has a spot factor of 4.1, which is about four times that of the Global Hawk RQ-4B. The support required was also judged to be greater than Global Hawk due to the unconventional fuel and large size which will complicate ground handling and hangaring (note that a score of 6 in this metric is not six times worse than Global Hawk, here the scores are additive, not multiplicative). The growth factor was 7.6, indicating a large sensitivity to mass growth. The risk was scored only moderately higher than Global Hawk, with higher risk due to the challenge of producing the extremely lightweight structure and the unconventional propulsion system. These metrics are summarized in Table 5.

Concept 2 – LH_2-Fueled Gas Turbine Engine

Concept 2 utilizes the same airframe configuration as Concept 1, but uses a LH_2-fueled gas turbine engine instead of an IC engine propulsion system. Gas turbine engines operate at considerably less than stoichiometric fuel-to-air ratios; therefore, large amounts of air are needed for their operation. At 21 km this can be addressed with large propellers and air intakes. Many studies have been performed to develop estimates for the mass and SFC of such a propulsion system design for high altitude aircraft. One such engine is a scaled derivative of the T406 Allison engine (ref. 22). Table 8 summarizes the propulsion inputs utilized for this concept based on this scaled derivative engine.

Table 8. Propulsion Inputs for Concept 2 (LH_2-Fueled Gas Turbine)

Specific Power = 425 W/kg		
H_2 Fuel Lower Heating Value = 33300 W-h/kg		
Altitude (m)	Fraction of rated sea level power	SFC g/(kW-h)
0	0.3	116
	1	116
21000	0.3	116
	1	116

Compared to the IC engine of Concept 1 (Table 7), the gas turbine engine has significantly better specific power, but increased SFC.

The hurricane science mission total endurance for Concept 2 was 6.3 days, with a 243 km/h TAS during loiter. These figures were 9.1 days and 189 km/h for the communications relay mission. Overall size, spot factor, support required, and risk are identical to Concept 1. Growth factor was slightly greater at 8.2, indicating a larger sensitivity to mass growth. These metrics are summarized in Table 5.

Concept 3 –LH_2-Fueled PEM Fuel Cell and Electric Motor

This concept utilizes the same airframe configuration as the previous two concepts, but uses an electric propulsion system consisting of a PEM fuel cell and electric motor. Hydrogen (stored as liquid) and atmospheric air are the fuel cell reactants. Compared to combustion engines, fuel cells typically have lower specific power (higher mass), but due to their higher conversion efficiencies they also have lower specific fuel consumption. Stack size and reactant (fuel and air) flow rate depend on the current-voltage operating point of the stack. Operation at low current density and high cell voltage yields a more efficient stack, thus lowering fuel consumption, but at the expense of increased stack mass. Conversely, operating at high current density minimizes stack mass but increases fuel consumption. Another factor that affects fuel cell performance, especially for HALE UAV applications, is operating pressure. Higher pressures improve performance, but at the expense of increased mass and power penalties due to the added compressors. Most PEM H_2-air stacks are designed to operate at ~100 kPa as dictated by the commercial market (ground-based applications).

In order to achieve an optimized fuel cell-powered vehicle, trade studies of performance, fuel consumption, and operating pressure would need to be performed. Unfortunately, only a limited amount of published data were available on state-of-the-art fuel cell stack and power plant performance due to the proprietary and highly competitive nature of the technology, making it difficult to truly assess the characteristics of these systems over a range of operating conditions and power levels. However, a few data points were available to help guide the analysis in this study. A previous, unpublished study by George Turney examined the overall specific power of a fuel cell subsystem for a high altitude aircraft (19.8 km) (ref. 23). Although Turney considered an alkaline fuel cell, the specific power of the PEM fuel cell technology at that time was fairly close to the alkaline system, and thus his results can serve as a data point for consideration. The overall specific power of the system, which included the fuel cell, ancillaries, turbocharger, etc., was estimated to be 185 W/kg with a SFC of 61 g/(kW-h). A second data point was generated through a proprietary Boeing/Giner Electrochemical Systems study. Data from this study correlate well with the Turney results. As with the combustion engines, the PEM fuel cell turbomachinery and heat exchanger masses will increase at higher altitudes. The change in altitude from

19.8 km to 21 km resulted in roughly a 20% increase in those component masses for the combustion engine systems. Since these components will be similar for the fuel cell system, a 20% increase in turbomachinery and heat exchanger mass was also assumed for the fuel cell system to account for operation at 21 km rather than 19.8 km. Based on Turney's study of the 19.8 km system, the fuel cell power plant constitutes approximately 2.1 kg/kW of the total system mass-to-power ratio of 5.4 kg/kW (185 W/kg), leaving 3.3 kg/kW for the remaining system components. Increasing this mass by 20% yields a mass-to-power ratio of 4.0 kg/kW. Adding this value to the fuel cell power plant results in a total mass-to-power ratio of 6.1 kg/kW, or a specific power of 164 W/kg. Therefore, 164 W/kg was used in this study for sizing the fuel cell system at 21 km. The 61 g/(kW-h) SFC in the Turney study was based on the output of the electric motor, not the fuel cell, and included the motor losses. In the HALE MDO code, the electric motor is modeled separately from the fuel cell and the SFC input is for the fuel cell alone. The Turney study SFC was adjusted to 57 g/(kW-h) for input to the HALE MDO code to remove the impact of motor losses. The propulsion inputs utilized for this concept are summarized in Table 9.

Table 9. Propulsion Inputs for Concept 3 (LH_2-Fueled PEM Fuel Cell and Electric Motor)

Specific Power = 164 W/kg		
H_2 Fuel Lower Heating Value = 33300 W-h/kg		
Altitude (m)	Fraction of rated sea level power	SFC g/(kW-h)
0	0.5	57
	1	57
21000	0.5	57
	1	57

Compared to the IC engine of Concept 1 (Table 7), the PEM fuel cell has lower specific power, but a significantly lower SFC.

The hurricane science mission total endurance for Concept 3 was 7.6 days, with a 251 km/h TAS during loiter. These figures were 9.9 days and 195 km/h for the communications relay mission. The overall size and spot factor are identical to Concept 1. The growth factor was greater at 9.4, indicating a larger sensitivity to mass growth. Also, support required was assessed slightly higher than the previous two concepts due to the relative uniqueness and complexity of the propulsion system. The risk was assessed higher for the same reasons. These metrics are summarized in Table 5.

Concept 4 - LH_2-Fueled Stirling Engine

This concept utilizes the same airframe configuration as the previous three concepts. The propulsion system is a Stirling engine using LH_2 and atmospheric air as reactants. Stirling engine use has increased in recent years and greater understanding of unsteady gas dynamics has produced significant improvements in efficiency and specific power. High altitude operation coupled with material improvements in Stirling converters could lead to excellent overall efficiencies for this application. With heater head temperatures approaching 1100 K and an environmental temperature of a little over 200 K at altitude, very high temperature ratios (>4) across the converter are possible, which improves both Carnot and fraction of Carnot efficiency. Recently designed Stirling converters have achieved greater than 60% of the Carnot efficiency, and at the temperature ratios possible with the very cold upper atmosphere, overall efficiencies of 47% (not including the burner or power conversion) may be achievable. One challenge with Stirling converters is transferring the heat into the device. In order to keep the heat transfer area reasonable, the air passed over the heater head into the combustor should be near a pressure of one atmosphere. To accomplish this several stages of turbocharging can be used as for IC engines. The major downside to Stirling converters that has curtailed their use in aircraft is their relatively low specific power. Using superalloy materials, a specific power of 200 W/kg should be achievable for a 50 kW system. This is about two thirds the value of a spark ignition engine. Adding similar ancillaries to the system as needed for the IC engine systems (radiator, intercoolers, etc.), specific powers of approximately 162 W/kg are likely. This combination of high power plant mass but reasonable efficiency could still lead to an attractive system since the burners can operate on jet fuel, diesel, or hydrogen. The estimated SFC assuming LH_2 is 170 g/(kW-h) at sea level and 101.6 g/(kW-h) at 21 km. The lower ambient temperature at altitude results in a larger temperature differential, increasing the cycle efficiency and reducing SFC. The propulsion inputs utilized for this concept are summarized in Table 10.

Table 10. Propulsion Inputs for Concept 4 (LH_2-Fueled Stirling Engine)

Specific Power = 162 W/kg		
H_2 Fuel Lower Heating Value = 33300 W-h/kg		
Altitude (m)	Fraction of rated sea level power	SFC (g/kW-h)
0	0.5	170
	1	170
21000	0.5	101.6
	1	101.6

Compared to the IC engine of Concept 1 (Table 7), the Stirling engine has lower specific power and higher SFC.

The hurricane science mission total endurance for Concept 4 was 5.0 days, with a 242 km/h TAS during loiter. These figures were 5.8 days and 188 km/h for the communications relay mission. Overall size and spot factor are identical to Concept 1. The growth factor was less at 6.7, indicating a smaller sensitivity to mass growth. Support required and risk were assessed to be the same as the PEM fuel cell concept due to the relative uniqueness and complexity of the propulsion system. These metrics are summarized in Table 5.

Concept 5 – Diesel-Fueled Compression Ignition Intermittent Combustion Engine

The propulsion system for this concept is a conventional CI engine using diesel fuel. The airframe configuration layout is the same as the previous four concepts except that the diesel fuel is stored in the wing rather than in spherical tanks like the LH_2 fuel. Table 11 contains a list of two- and four-stroke aviation CI engines, including power output and mass. The only engine in Table 11 that has not been built is the TCM 186, a general aviation class engine proposed by Teledyne Continental Motors. The rest of these engines were built from the early 1930's to the present. The average specific power for the four-stroke engines is 714 W/kg whereas for the two-stroke engines it is 935 W/kg. The average air-cooled engine has a specific power of 781 W/kg whereas the average liquid-cooled engine has a specific power of 926 W/kg. These specific powers do not include the mass of the cooling package. Cooling system data from reference 24 was adjusted for 21 km altitude operation resulting in an additional mass of 0.16 kg per kW.

Table 11. Historical Two- and Four-Stroke Diesel Engines

Make	Model	Config	Cycle	Cooling	Number Cyl.	Bore (mm)	Stroke (mm)	Disp. (liters)	Comp Ratio	Power (kW)	RPM	Mass (kg)	Specific Power (W/kg)	Year
Packard	DR980	Radial	4	air	9	122	152	16.1	16/1	174	2050	231	753.2	1930
Guiberson	A980	Radial	4	air	9	122	152	16.1	14.7/1	155	2050	231	671.0	1931
Deschamps		30deg V	2	liquid	12	152	229	50.5	16/1	1000	1750	1089	918.3	1934
Bristol	Phoenix	Radial	4	air	9	146	190	28.75	14/1	318	2000	494	643.7	1934
Zborojovka	ZOD	Radial	2	air	9	120	130	13.2	15/1	207	1600	297	697.0	1935
Hispano	Clerget 14F2	Radial	4	air	14	140	160	34.5	15/1	518	2200	600	863.3	1935
Salmson	SH18	Radial	2	air	18	118	150	29.5	16/1	481	1700	567	848.3	1935
Mercedes	OF2	60 deg V	4	liquid	12	165	210	53.9	15/1	592	1790	935	633.2	1935
Junkers	204	Opposed	2	liquid	6	120	2*210	28.75	17/1	570	1800	750	760.0	1935
Junkers	205	Opposed	2	liquid	6	105	2*160	16.6	16/1	444	2200	510	870.6	1936
Junkers	207 Turbo	Opposed	2	liquid	6	105	2*160	16.6	16/1	780	3000	649	1201.8	1938
Napier	Nomad	Flat	2	liquid	12	152	187.3	41	16/1	1984	2050	1624	1221.7	1953
McCuloch	TRAD-4180	Radial	2	air	4	98.4	98.43	3	15/1	150	2850	149	1006.7	1970
TCM	186	Opposed	2	liquid	4	108	110	4.031	13/1	186	3500	174	1069.0	1990's

CI engines differ from SI engines in two important ways with respect to their use for HALE applications. First, by use of higher compression ratios, CI engines are more efficient at removing energy from the fuel that is injected into the cylinders. A typical efficiency value for a naturally aspirated 4-stroke SI engine is about 34% compared to almost 40% for a CI engine. This increase in efficiency means more of the useful work is extracted from the cycle resulting in less energy in the exhaust for turbocharging. The second difference with respect to HALE applications is that CI engines use excess air and thus the exhaust temperatures are further reduced. SI engines operate near ideal stoichiometric conditions, or equivalence ratios of 1.0, to match the fuel-to-air ratio. CI engines require excess air to operate lean with a maximum equivalence ratio of about 0.6. This lower equivalence ratio prevents engine smoking, which is not environmentally acceptable. Typical exhaust mass averaged temperatures for a SI engine are around 1200 K, whereas for a CI engine they are close to 800 K. At the design operating point, this is typically not a problem as the turbocharger efficiency is at or close to its highest operating efficiency. However, this becomes a problem at part power since there is insufficient energy to run the turbomachinery, reducing the turndown capability of the power plant. Adding the mass required for the ancillary equipment, in addition to the two stage turbocharger and intercoolers, results in a specific power of 263 W/kg. SFC for this system is estimated to be 182.5 g/(kW-h). The diesel engine inputs are summarized in Table 12.

Table 12. Propulsion Inputs for Concept 5 (Diesel Engine)

Specific Power = 263 W/kg		
Diesel Fuel Lower Heating Value = 11600 W-h/kg		
Altitude (m)	Fraction of rated sea level power	SFC g/(kW-h)
0	0.5	182.5
	1	182.5
21000	0.5	182.5
	1	182.5

Compared to the IC engine of Concept 1 (Table 7), this diesel engine has slightly higher specific power but much higher SFC due to the lower energy content of the diesel fuel. However, diesel fuel has fewer technical and safety challenges when considering ground handling and infrastructure issues.

Aerodynamic, structural, and mass assumptions were similar to Concept 1, although in this case LH_2 tanks are not required. Therefore, the nacelle size is no longer a function of the required LH_2 tank diameter, resulting in a smaller and more aerodynamically favorable nacelle as shown in Figure 13. The smaller pod/nacelle size is also evident from the dimensions given in Table 3. The aerodynamic benefits of this smaller nacelle can be seen in Figure 10 and Figure 11. Again, the required cooling area has not been addressed in detail for this study, and the surface area required may be greater than shown in Figure 13.

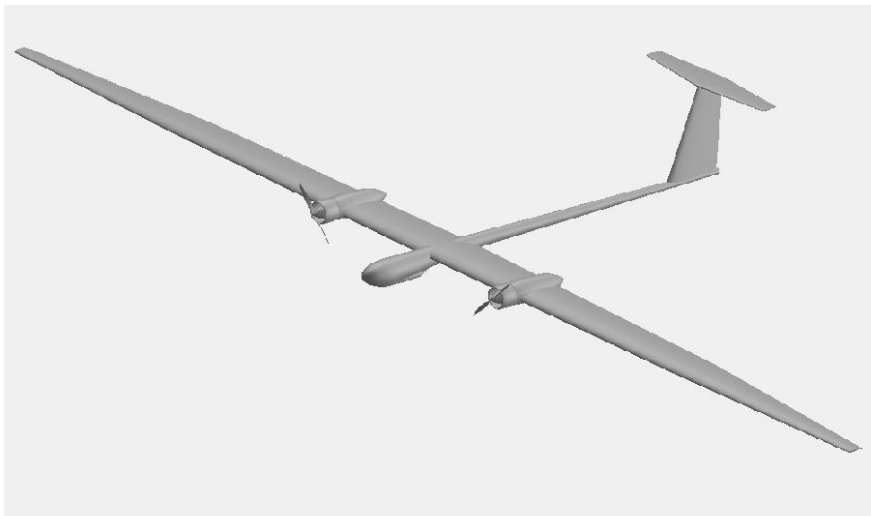


Figure 13. Concept 5: Diesel-fueled CI engine propulsion system.

The hurricane science mission total endurance for Concept 5 was 5.7 days, with a 250 km/h TAS during loiter. These figures were 6.5 days and 195 km/h for the communications relay mission. Overall size and spot factor are identical to Concept 1. The growth factor was higher at 8.5, indicating a larger sensitivity to mass growth. Support required and risk were assessed to be significantly less than the PEM fuel cell and Stirling concepts, and also less than the gas turbine and SI IC engine concepts due to the conventional propulsion system and fuel. These metrics are summarized in Table 5.

4.3.2 Heavier-Than-Air Solar Regenerative Concepts (Concepts 6-10)

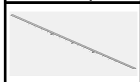
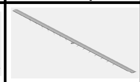
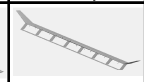
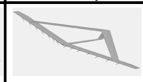
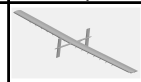
The basic idea of SR propulsion is that the sun can be the sole energy source for the vehicle. During the day some of the energy collected by the solar cells is used to power the propulsion system, payload, and other on-board systems. Excess energy collected above that required to operate the vehicle is used to charge an energy storage system. At night the vehicle is powered by discharging the energy storage system. If the system is balanced over a diurnal cycle (energy collected from the sun equals energy required to fly plus losses), then the vehicle can theoretically remain aloft indefinitely. There are a number of different ways to store energy such as secondary (i.e., rechargeable) batteries, flywheels, regenerative fuel cells, or even altitude. During the 1990’s, the NASA ERAST program established a goal to demonstrate 96-hour endurance with an SR HALE vehicle. The AeroVironment Helios was originally intended to be used for that demonstration. The goals for the ERAST program and Helios design were later changed due to budget constraints and gaps in available technology. Two days (48 hours) of continuous flight using SR propulsion was demonstrated in 2005 with the small SoLong UAV built by AC Propulsion. Although this demonstrated the potential for SR propulsion, the SoLong UAV is a low altitude platform with very little payload capability. Energy storage for the SoLong UAV is provided by Li-Ion batteries, but the successful multi-day flight demonstration depended heavily on utilization of up-drafts and thermals which permitted the propulsion system to be turned off for significant periods of time.

Sizing and analysis of the SR concepts was performed using the analysis code described in Section 4.2. Each concept was sized separately for the hurricane science mission and communications relay mission, and results are presented for both. Preliminary analysis of the SR concepts quickly revealed that none of them would have sufficient performance to conduct either the hurricane science mission or the communications relay mission. The SR propulsion system, given the assumptions made for the analysis, could not provide the amount of power on a continuous basis needed to operate the vehicle. Since the missions were not feasible for any of the concepts, traditional metrics such as endurance and takeoff mass could not be used to compare the HTA SR concepts. Instead, a new metric was developed to compare the "degree of feasibility" among the different concepts. That is, the performance of a given concept was considered better than another if it resulted in a higher level of mission feasibility. The metric used to measure feasibility was $\%P_{regen}$, the percentage of total power required supplied by the SR propulsion system on the worst day of the mission (see metric definitions in Section 4.1).

$\%P_{regen}$ is calculated from the ratio of two parameters, P_{AR} and P_{RL}. P_{AR} is the power available from the SR system which can be provided continuously with no net loss in "state-of-charge" at the end of 24 hours. In other words, this is the power level which can be energy balanced for a given SR power system. The SR power system consists of the solar arrays, energy storage system, and associated auxiliary equipment. The value of P_{AR} depends on flight latitude and time of year, solar array size, solar array efficiency, power management and distribution efficiency, energy storage system efficiency, and energy storage system capacity. P_{RL} is the power required from the SR power system during loiter; comprised of power needed for propulsion and power needed for the payload and aircraft systems. The payload power required and aircraft system power required are fixed inputs to the analysis. The propulsion power required is a function of the total aircraft mass, the aerodynamic efficiency (lift-to-drag ratio), flight speed, propeller efficiency, and motor efficiency. If P_{AR} divided by P_{RL} is greater than 1.0 (100%), the specified mission is feasible since the SR power system can provide the power required to fly the vehicle and maintain a diurnal cycle energy balance. Values less than 1.0 indicate the mission is infeasible. Note that P_{RL} is not independent of the characteristics of the SR power system since total aircraft mass includes the mass of the SR power system. Maximizing P_{AR} does not necessarily maximize the ratio of P_{AR} to P_{RL} because increasing solar array size or energy storage system capacity also increases P_{RL}. Maximizing the ratio, $\%P_{regen}$, was the objective used to size the SR system and optimize the vehicle designs.

Geometric parameters for each of the HTA SR concepts are summarized in Table 13. Wing area and span were optimized, with a maximum span limit of 100 m (80 m for Concept 9, see Concept 9 discussion below). The FAA 80 m span limit was not applied to the HTA SR concepts in general since these vehicles already require special ground handling and accommodations. There is still a practical limit to wingspan, however. A limit of 100 m was selected following consultation with experts familiar with this class of vehicle. In all cases the wing area is approximately 600 m^2. In comparison, the AeroVironment Helios had a wingspan of 75 m and wing area of 180 m^2. The wings for these concepts are even larger than a Boeing 747 wing, which has a span of 64 m and area of 540 m^2.

Table 13. Summary of Geometric Parameters for HTA SR Concepts

			Concept 6	Concept 7	Concept 8	Concept 9		Concept 10
								
			All-Wing Fuel Cell	All-Wing Secondary Battery	Trussed-Wing Secondary Battery	Joined-Wing Secondary Battery		Multi-Surface Secondary Battery
Parameter	**Symbol**	**Units**				Fwd	Rear	
Wingspan	b_S	m	100	100	97	80	56	100
Wing LE Sweep	Λ_{LE}	degrees	0	0	0	10	-33	0
Wing Area	S_W	m^2	600	600	576	280	210	590
Wing Aspect Ratio	AR_W	-	16.7	16.7	16.3	22.8	14.9	16.9
Wing Loading	W/S	kg/m^2	(4.35/3.28)*	(5.35/3.64)*	(6.23/4.60)*	(5.49/3.73)*		(6.44/5.75)*
		lb/ft^2	(0.89/0.67)*	(1.09/0.75)*	(1.28/0.94)*	(1.12/0.77)*		(1.32/1.18)*
Propeller Diameter	D_P	m	3	3	3	3		3
Pod/Nacelle Separation	P_{sep}	m	33.5	-	-	-		-
Pod/Nacelle Length	Pod_L	m	(10.0/8.8)*	-	-	-		-
Pod/Nacelle Diameter	Pod_D	m	(1.7/1.5)*	-	-	-		-
Fuselage Length	Fus_L	m	7.6	5	-	25		30
Fuselage Diameter	Fus_D	m	1.3	1.3	-	1.3		1.3
Number of Pylons	N_{pyl}	-	-	-	8	-		-
Pylon Separation	PL_{sep}	m	-	-	11	-		-
Pylon Area	S_{pyl}	m^2	-	-	12	-		-
Vertical Tail Span	b_{vt}	m	-	-	-	3.9		-
Vertical Tail Area	S_{vt}	m^2	-	-	-	14.8		-
Vertical Tail Aspect Ratio	AR_{vt}	-	-	-	-	1		-
Tracking Array Area	S_{aux}	m^2	-	-	-	-		(74/174)*

* Hurricane Science/Communications Relay

Mass data for the SR concepts are shown in Table 14 and 15 for the hurricane science mission and communications relay mission, respectively. The analysis code estimates aircraft mass using a combination of analytical methods, empirical factors, and user inputs. Wing structural mass is based on analysis of the amount of material required in the wing to accommodate a series of load cases. The structural arrangement for the wing was assumed to be a tubular spar of carbon fiber composite construction, ribs, and a non-structural skin as used in the AeroVironment Helios design. Additional assumptions used for structural sizing were:

- Wing tip maximum deflection = 25% of wing semi-span
- Aileron effectiveness fraction at cruise velocity (at wing tip) = 0.50
- Positive uniform gust load factor = 3.8 g
- Negative load factor for taxi bump = 1.5 g
- Wing structure factor of safety = 1.5
- Torque box fraction of chord = 0.25
- Mounting/Installation mass = 6% of component mass

Assumptions used for airframe subsystems include:

- Avionics power required = 300 W
- Avionics mass = 15 kg
- Backup battery mass = 12 kg
- Servo mass = 1% of takeoff mass
- Landing Gear mass = 1.2% of takeoff mass

Table 14. Mass Data for HTA SR Concepts, Hurricane Science Mission

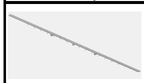
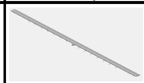
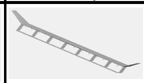
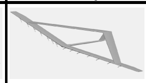
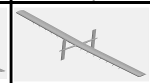
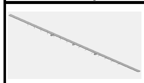
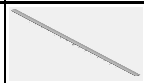
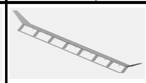
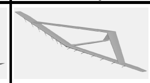
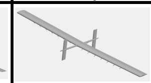
	Concept 6	Concept 7	Concept 8	Concept 9	Concept 10
					
Component Masses (kg)	All-Wing Fuel Cell	All-Wing Secondary Battery	Trussed-Wing Secondary Battery	Joined-Wing Secondary Battery	Multi-Surface Secondary Battery
STRUCTURES:					
Wing	775	813	936	573	865
Landing Gear	31	38	43	32	46
Center Pod/Fuselage	58	61		58	64
Propulsion Pods	118				
Tailboom/Vertical Tail				58	
Auxiliary Arrays and Booms					120
TOTAL STRUCTURE:	982	912	979	721	1095
PROPULSION:					
Solar Array	303	303	429	247	348
Propulsion Mounts	43	81	92	67	100
Secondary Batteries		1071	1234	871	1383
H_2 Tank	55				
O_2 Tank	25				
Fuel Cell & Electrolyzer	323				
H_2 Reactant	9				
O_2 Reactant	74				
Motors	162	187	191	160	181
Propellers	80	95	106	87	102
TOTAL PROPULSION:	1074	1737	2052	1432	2114
EQUIPMENT:					
Servos	26	32	36	27	38
Wire/Electrical	99	99	97	81	99
Avionics	15	15	15	15	15
Backup Battery	12	12	12	12	12
Array Tracking Mechanism					25
TOTAL EQUIPMENT:	152	158	160	135	189
(STRUC + PROP + EQUIP) EMPTY MASS:	2208	2807	3191	2288	3398
Payload	400	400	400	400	400
Zero Fuel Mass	2608	3207	3591	2688	3798
Fuel	0	0	0	0	0
Takeoff Gross Mass	**2608**	**3207**	**3591**	**2688**	**3798**

Table 15. Mass Data for HTA SR Concepts, Communications Relay Mission

Component Masses (kg)	Concept 6	Concept 7	Concept 8	Concept 9	Concept 10
					
	All-Wing Fuel Cell	All-Wing Secondary Battery	Trussed-Wing Secondary Battery	Joined-Wing Secondary Battery	Multi-Surface Secondary Battery
STRUCTURES:					
Wing	705	716	833	516	826
Landing Gear	24	26	32	22	41
Center Pod/Fuselage	41	43		42	49
Propulsion Pods	80				
Tailboom/Vertical Tail				55	
Auxiliary Arrays and Booms					221
TOTAL STRUCTURE:	850	785	865	635	1137
PROPULSION:					
Solar Array	303	303	429	247	414
Propulsion Mounts	27	42	57	35	82
Secondary Batteries		539	761	439	1157
H_2 Tank	32				
O_2 Tank	15				
Fuel Cell & Electrolyzer	200				
H_2 Reactant	5				
O_2 Reactant	37				
Motors	116	124	132	105	148
Propellers	42	46	56	43	67
TOTAL PROPULSION:	777	1054	1435	869	1868
EQUIPMENT:					
Servos	20	22	26	18	34
Wire/Electrical	99	99	97	81	99
Avionics	15	15	15	15	15
Backup Battery	12	12	12	12	12
Array Tracking Mechanism					25
TOTAL EQUIPMENT:	146	148	150	126	185
(STRUC + PROP + EQUIP) EMPTY MASS:	1773	1987	2450	1630	3190
Payload	200	200	200	200	200
Zero Fuel Mass	1973	2187	2650	1830	3390
Fuel	0	0	0	0	0
Takeoff Gross Mass	**1973**	**2187**	**2650**	**1830**	**3390**

Predicted aerodynamic characteristics for loiter conditions are presented in Figure 14. Maximum L/D is in the range of 28 to 32. Variation in L/D among the concepts is largely due to differences in wetted area because of additional components such as fuselage booms, wing pods, and pylons. As mentioned previously, the aerodynamic analysis contains a mix of empirical and analytical handbook methods. A number of the SR concepts employ an all-wing layout. Care must be taken in applying empirical airfoil predictions to the all-wing configurations because the airfoil design is constrained by the need to provide positive pitching moment at zero lift ($C_{mo} > 0$). The empirical methodology used does adjust for the required pitching moment of the airfoil and good agreement in lift and drag characteristics between the airfoil model and test data for the AeroVironment Pathfinder airfoil has been demonstrated. (Although Helios is the basis for much of the SR modeling, the expected aerodynamic conditions are more consistent with the lower altitude Pathfinder design than the 30.5 km Helios design condition.) Because the pods of the SR configurations are smaller relative to the wing than those of the consumable configurations, the impact on the wing aerodynamics is expected to be less. A span efficiency of 93% was therefore used for induced drag calculations (higher than used for the consumable configurations).

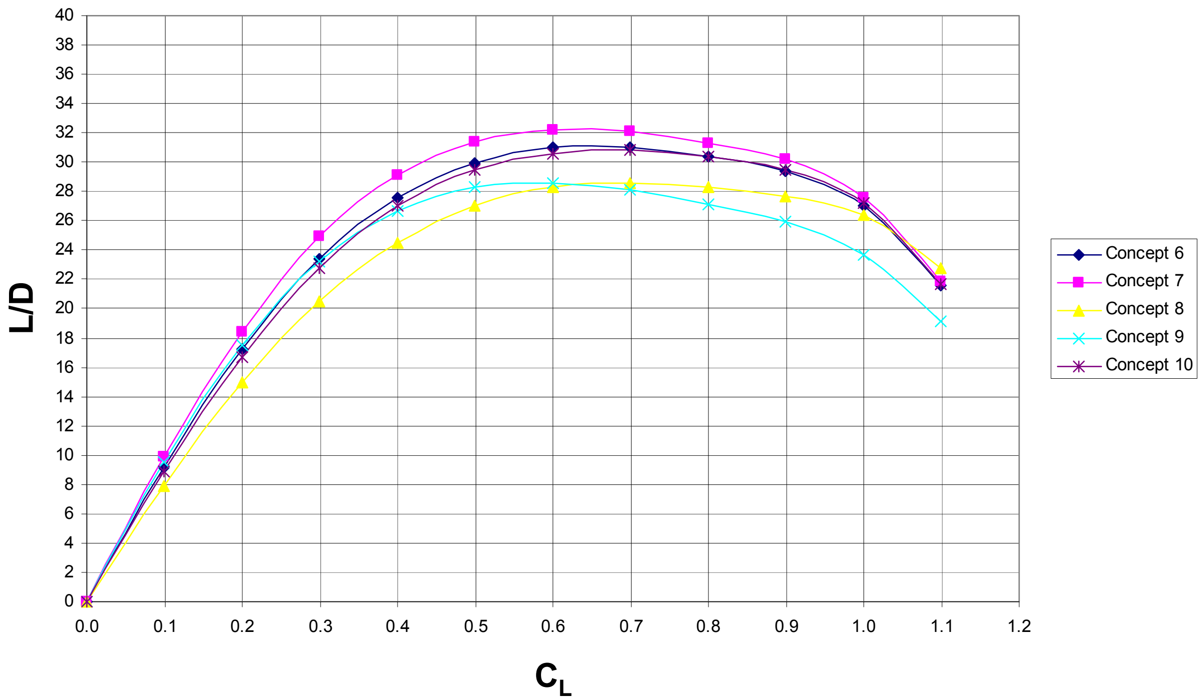
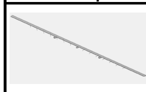
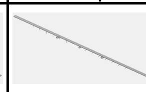
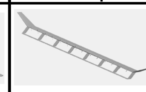
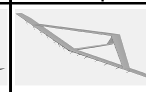
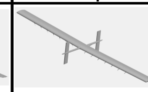


Figure 14. Predicted aerodynamic performance of SR concepts (at loiter).

Table 16 contains the AoA metric results for the SR concepts. When examining these results, it must be remembered that the designs are not feasible designs that could be built to accomplish the mission. Since $\%P_{regen}$ varies among the concepts, comparison of characteristics such as takeoff mass between concepts is not valid. It is, in fact, the percentage of power supplied by the SR system ($\%P_{regen}$) which is the most important metric to examine. More in-depth discussion of the concept assumptions and results follows.

Table 16. AoA Metric Results for SR Concepts

Evaluation Criteria		Concept 6	Concept 7	Concept 8	Concept 9	Concept 10
						
		Solar Regen Fuel Cell	Solar Regen Secondary Battery	Trussed-Wing Secondary Battery	Joined-Wing Secondary Battery	Multi-Surface Secondary Battery
Endurance, days	Hurricane Science	n/a	n/a	n/a	n/a	n/a
	Communications Relay	n/a	n/a	n/a	n/a	n/a
TOGM, kg	Hurricane Science	2610	3210	3590	2690	3800
	Communications Relay	1970	2190	2650	1830	3390
Wingspan (HTA) or Length and Width (LTA), m		100	100	97	80	100
Volume (LTA), m^3		n/a	n/a	n/a	n/a	n/a
$\%P_{regen}$ (SR)	Hurricane Science	31	36	31	29	35
	Communications Relay	26	36	35	29	40
Takeoff and Landing Robustness %	Hurricane Science	60	60	60	60	60
	Communications Relay	80	80	80	80	80
Ground Footprint	Spot Factor	2.5	1.8	2.7	3.8	(5.6/5.9)*
	Support Required	4	-3	-3	-3	-3
Growth Factor	Hurricane Science	2.8	3.3	3.6	3.1	3.9
	Communications Relay	2	2.2	2.5	2.1	3.4
Risk		8	8	6	8	8

*(Hurricane Science/Communications Relay)

Concept 6 – All-Wing Configuration with Solar Regenerative Fuel Cell Propulsion

Concept 6 represents a baseline approach for the SR vehicles. The all-wing design (see Figure 15) has heritage in the family of solar-electric aircraft built by AeroVironment before and during NASA's ERAST program (Pathfinder, Pathfinder Plus, Helios). All of these vehicles utilized distributed electric propulsion systems with numerous propellers driven by electric motors. Use of a regenerative fuel cell system was researched under the ERAST program and such a system was designed for the Helios aircraft, although never completed. In a solar regenerative fuel cell system, the energy is stored as hydrogen and oxygen reactants. At night the reactants are combined in a fuel cell producing heat, water, and electricity to power the vehicle. During the daytime excess energy from the sun is used to electrolyze the water back into H_2 and O_2 which is then stored for use at night, forming a closed-loop system. Regenerative fuel cell systems can utilize separate systems for the power generation (fuel cell) and water electrolysis (electrolyzer) or a single unitized system which performs both functions. For Concept 6 separate fuel cell and electrolyzer systems were used due to the lower level of technology maturity for the unitized system.

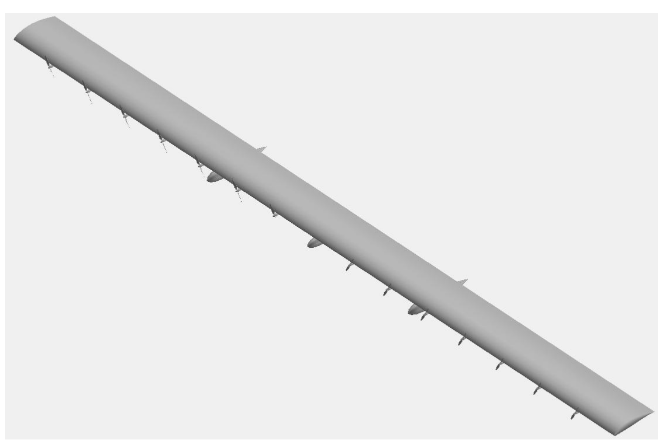


Figure 15. Concept 6: All-wing configuration with solar regenerative fuel cell propulsion.

Propulsion system assumptions for Concept 6 were based on characteristics of the AeroVironment Helios design and test data from NASA Glenn Research Center. The solar array incorporates high efficiency, bi-facial silicon solar cells of the type used on Helios. Solar cell reference efficiency was assumed to be 20%. This efficiency is representative of an individual solar cell at a reference condition. The HALE MDO code accounts for variation in efficiency due to cell temperature and includes an array power scaling factor for wiring losses and other miscellaneous losses associated with installation of individual solar cells into a solar array. Fuel cell and electrolyzer assumptions are summarized in Table 17. Mass characteristics are based on Giner and Lynntech systems designed for Helios (ref. 25). For simplicity in modeling and sizing the system, stack and ancillary masses have been combined into a single mass per cell value. The fuel cell and electrolyzer polarization data (voltage versus current density) were derived from tests at NASA Glenn on the Lynntech Gen IV system (ref. 26). Estimates for reactant tank mass were based on tanks designed and built for the Helios system. Technology assumptions based on Helios might be considered somewhat conservative since that system was designed a few years ago and fuel cell technology has been advancing rapidly in recent years. However, the system designed for Helios was never flight tested, and there are often unexpected increases in mass and reductions in efficiency as a system is brought to an operational condition.

Table 17. Regenerative Fuel Cell System Assumptions

	Fuel Cell		**Electrolyzer**	
Active Area per cell	224 cm^2		161 cm^2	
Total Mass per cell	0.49 kg		0.55 kg	
Polarization Data	Current Density (A/cm^2)	Voltage (V)	Current Density (A/cm^2)	Voltage (V)
	0.0	1.04	0.0	1.48
	0.025	0.91	0.15	1.59
	0.30	0.78	0.50	1.69
	0.50	0.74	0.75	1.71

Although Concept 6 borrows significantly from Helios heritage, the vehicle size is much larger. Best overall performance of the design was obtained at the maximum allowed wingspan (100 m). Despite the large wingspan, the wing aspect ratio is relatively low compared to Helios. Optimum wing area was found to be approximately 600 m^2 resulting in an aspect ratio of only 16.7. The resulting wing loading is similar to Helios. For the hurricane science mission $\%P_{regen}$ is only 31%, and it is even less at 26% for the communications relay mission. If there are no changes in the power required, this result implies that the power output of the SR propulsion system needs to increase by a factor of more than 3 (without any increase in mass or solar array area) in order for the missions to become feasible. Closing such a large gap in feasibility would require significant improvement in technology as will be explored further in Section 7 of this paper.

Concept 7 – All-Wing Configuration with Solar Secondary Battery Propulsion

Concept 7 differs from Concept 6 primarily in the type of energy storage system used. For Concept 7, rather than storing energy in the form of H_2 and O_2, energy is stored in rechargeable batteries (also referred to as secondary batteries). Secondary batteries are superior to regenerative fuel cell systems in terms of roundtrip efficiency, defined as the amount of energy extracted from the system when it is discharged compared to the amount of energy expended to charge the system. Roundtrip efficiencies for regenerative fuel cell systems are on the order of 50% whereas some batteries can achieve efficiencies greater than 90%. The drawback of secondary batteries is the higher mass required to store a given amount of energy (i.e., lower specific energy, W-h/kg). Because of their efficiency, however, batteries need not have a specific energy as high as regenerative fuel cells in order to result in better overall aircraft performance. Lithium-Ion is one type of rechargeable battery which is currently used widely in a variety of applications. A different type of battery that theoretically provides opportunity for much higher specific energy is Lithium-Sulfur (ref. 27). Near-term projected Li-S technology served as the basis for the battery assumptions used in Concept 7. After accounting for depth-of-discharge and power management efficiencies, a specific energy of 252 W-h/kg and roundtrip efficiency of 82% was used in the analysis.

Table 16 shows that the Li-S battery-based energy storage system provides a higher feasibility ($\%P_{regen}$) than the regenerative fuel cell approach for both the hurricane science and communications relay missions. A $\%P_{regen}$ of 36% is obtained for both of the design missions. As noted above, from a performance standpoint the choice between secondary batteries and regenerative fuel cells for energy storage represents a trade-off between the high specific energy of the fuel cell-based system and the high efficiency of the battery system. Interestingly, for this study the specific energy of the regenerative fuel cell system was not significantly higher than that assumed for the Li-S battery technology. Note that because some elements of a regenerative fuel cell system are sized by power requirements and some elements are sized by energy storage requirements, the specific energy of a regenerative fuel cell system will vary with the design requirements. For Concept 6, the effective specific energy of the regenerative fuel cell system was 350 W-h/kg for the hurricane science mission and 300 W-h/kg for the communications relay mission. Although this is slightly more than the battery system, the roundtrip efficiency was significantly less at ~46%. Another benefit of secondary batteries is that the system is less complex than a regenerative fuel cell system. Regenerative fuel cell systems consist of many different components such as fuel cell stack, electrolyzer stack, reactant tanks, valves, plumbing, and other ancillaries. A battery energy storage system consists mainly of battery packs, wiring, and a power management and distribution module. Because of the better overall performance of the secondary batteries for the study missions and their simplicity, Li-S secondary battery energy storage was assumed for the remaining SR concepts.

Concept 8 – Trussed-Wing Configuration with Solar Secondary Battery Propulsion

Concepts 8 through 10 represent an attempt to evolve beyond the Helios-like, all-wing configuration and use more unconventional designs to address some of the known issues with past HALE SR vehicles. One well known issue with the Helios design was its high degree of flexibility. Figure 16 illustrates the large amount of wing dihedral experienced by Helios due to the lightweight, flexible structure. Another issue is the performance of near-horizontal, wing solar arrays during operation at mid to high latitudes in winter. The low sun elevation at these conditions greatly diminishes the effectiveness of the Helios-like solar array arrangement.

Figure 16. AeroVironment Helios in flight. (NASA Photo ED03-0152-2 by Carla Thomas, ref. 28)

One way to limit flexibility, while also providing for more favorably oriented solar arrays, is to use a trussed-wing structure as in Concept 8 (Figure 17). Although not visible in the figure, this design also incorporates wire cross-bracing to add to the structural stiffness. In addition to providing rigidity to the structure, the pylons in Concept 8 provide a vertical surface for solar arrays. When operating at high latitude in winter months, vertical arrays can provide a more optimum angle relative to the sun than horizontal (wing) arrays. It was believed that this additional vertical array surface could improve the performance of the SR propulsion system.

The trussed-wing design could not be directly accommodated in the HALE MDO code, necessitating a series of simplifying assumptions and modifications to the analysis. The truss elements of the wing introduce additional drag. Adjustments for pylon drag were made based on pylon wetted area. Since a detailed structural layout was not performed the total length of wire bracing was not known. The drag of the wire bracing was accounted for with a simple 5% drag penalty rather than a detailed build-up. The structural sizing methodology of the analysis code does not address this type of structural arrangement. Since the main benefit of the trussed-wing would likely be to increase rigidity rather than to reduce structural mass, the mass predicted by the HALE MDO code for a conventional wing of the same size was used as a basis for the mass estimate of the trussed-wing. The estimated mass for a conventional wing was assumed to be equivalent to the combined mass of the main wing and lower structural element in the trussed-wing design. Additional mass was added for the vertical pylons based on pylon area. For this configuration there is the potential for significant shading of the pylon solar arrays by each other and the wing surface, depending on the relative orientation of the vehicle and sun. This effect was accounted for with shading factors determined for a series of vehicle-to-sun orientations.

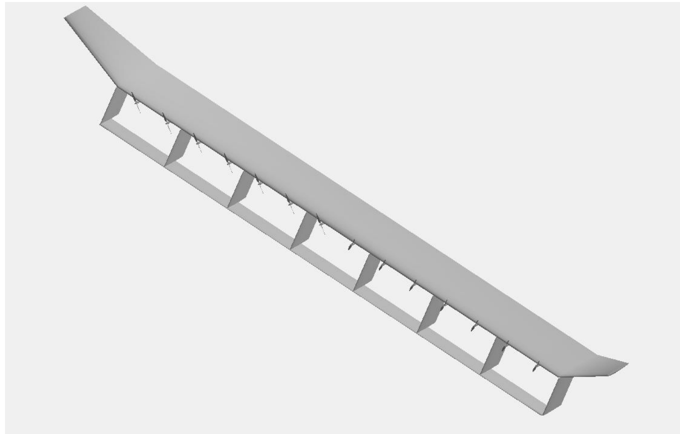


Figure 17. Concept 8: Trussed-wing configuration with solar secondary battery propulsion.

As evident in Table 16, Concept 8 did not show a benefit over the more conventional Concept 7 design for the AoA metrics assessed. Feasibility of both the communications relay mission and hurricane science mission were slightly less. One drawback of the Concept 8 design is a reduction in aerodynamic performance due to the drag of the truss structure. Figure 14 shows the decrement in maximum lift-to-drag ratio associated with the higher wetted area and wire bracing. The pylons increased drag, but the pylon vertical solar arrays did not provide the anticipated energy benefit. Although offering the potential for a more direct solar incidence angle, performance of vertical arrays suffer from a directionality issue. For example, a vertical array facing east at sunrise would benefit from a near normal sun angle and produce much more solar power than a horizontal array. However, by the afternoon that array is facing away from the sun and collecting no energy. With both sides of the pylons in Concept 8 covered with solar cells, half of the pylon solar cells are receiving no solar energy at any given time. If the vehicle were able to fly in a single direction, the solar cells could be placed more optimally. Because the design missions are loiter missions, the aircraft cannot be optimized to fly in a single direction. Furthermore, with the vehicle repeating a circular pattern throughout the day, the vertical arrays spend only a fraction of the time at the optimal angle with the sun. Flying a more oblong pattern with the majority of the time spent in the most favorable solar orientation would improve energy collection. Unfortunately, the shape of the pattern that can be flown and still maintain the required loiter accuracy will depend on the magnitude and direction of winds encountered during the mission. It is not possible, therefore, to design the vehicle assuming a specific pattern can be flown to maximize solar energy collection. At the worst case solar design conditions for the hurricane science mission (30°N, November 30), for a circular flight pattern a horizontal array collects more energy than a vertical array. As a result, the average energy collected per square meter of array area is 18% less for Concept 8 than Concept 7. However, the pylon arrays do provide a 42% increase in total array area for a given wing size. The net result of a less effective array system (on average) and increased array area is a 17% increase in energy collected per square meter of wing area. But because of the additional mass of the pylon arrays, removing them marginally improves $\%P_{regen}$ for the hurricane science mission. At the more stringent solar design conditions of the communications relay mission (47°N, December 21), a vertical array can collect roughly 30% more energy than a horizontal array. Despite that increase, the average solar energy collected per square meter of array area for Concept 8 is approximately equal to that of a horizontal array because at any given time half of the pylon array area is on the "back side" facing away from the sun. The pylon arrays do provide an overall net benefit for the communications relay mission, increasing $\%P_{regen}$ from 32% to 35%.

The simple modeling performed for the AoA did not permit full understanding of the potential structural benefits of the trussed-wing layout utilized in Concept 8. A number of the areas in which the trussed-wing design would show potential for improvement over the cantilever wing designs were not addressed in the high level analysis conducted for the AoA. In some respects the analysis conducted highlighted the penalties of the design without fully exploring the benefits. Even so, these benefits would not be enough to overcome the limitations of current SR propulsion technology and achieve feasibility of the hurricane science or communications relay missions.

Concept 9 – Joined-Wing Configuration with Solar Secondary Battery Propulsion

Another unconventional layout considered was a joined-wing configuration. The joined-wing was expected to provide an increase in structural rigidity and perhaps a reduction in total structural mass compared to the all-wing arrangement. Another motivation for the joined-wing approach was to obtain a large amount of solar array area in a more compact design. Because "compactness" was one of the desires for the joined-wing concept, the span of Concept 9 was limited to 80 m. Implementation of the joined-wing approach in an SR vehicle is illustrated in Figure 18. Modeling this type of design with the available analysis tools required an extensive set of simplifying assumptions. Predicted wing spar mass was checked with a structural analysis code developed specifically for joined-wing configurations based on inextensible beam theory (ref. 29). Good agreement was obtained between the predicted mass and the more detailed structural analysis. The simplified modeling did not fully address all of the penalties associated with a joined-wing design such as a heavier vertical tail than a conventional design.

Overall, the results for Concept 9 make it the least desirable SR concept. Feasibility ($\%P_{regen}$) is lowest of the secondary battery concepts. Despite the lower wingspan, the spot factor is also higher than Concept 7 due to the nose-to-tail length. Past evaluations of joined-wing designs have shown structural mass benefits compared to conventional wing-body-tail designs. It is not clear, however, that a joined-wing design would have structural mass benefits compared to an all-wing design in which the tail mass and fuselage mass have been eliminated. Additionally, splitting a single large wing into two smaller chord wings can have negative aerodynamic impacts. Because induced drag is a function of the total area and largest span of the two wings, not the individual aspect ratios (ref. 30), two joined wings with higher individual aspect ratios do not necessarily have lower induced drag than a single, larger chord wing of the same span. The lower chord Reynolds number of the two smaller wings does, however, lead to higher profile drag. As with Concept 8, the scope of the AoA analysis did not reveal all the beneficial aspects of this concept, but a more thorough analysis would not result in mission feasibility ($\%P_{regen} \geq 100\%$).

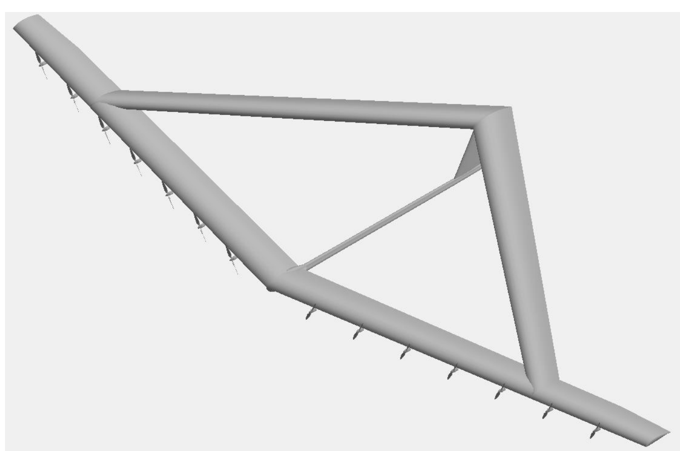


Figure 18. Concept 9: Joined-wing configuration with solar secondary battery propulsion.

Concept 10 – Multi-Surface Configuration with Solar Secondary Battery Propulsion

The amount of solar energy collected during the day is greatly reduced by non-optimal array orientation. Solar energy collection is maximized when the array is normal to the incident solar rays and decreases with the sine of the incidence angle. As noted in the discussion of Concept 8, because the aircraft heading and orientation relative to the sun is continually changing, an array that is fixed on the aircraft will be in an optimum orientation for only a fraction of the time during the day. Concept 10, shown in Figure 19, was developed in an attempt to address the problem of solar array orientation. The basic idea of this concept is to have arrays which re-orient throughout both the loiter pattern and the day to maximize the solar energy collected (the arrays only vary in roll angle, not pitch and yaw). Unlike the vertical arrays in Concept 8, a sun-tracking array will always perform better than a horizontal array no matter what the latitude and time of year. The magnitude of the benefit, however, does vary with time of year and latitude, with a maximum in wintertime at high latitudes when it is most needed. For the communications relay mission worst solar conditions (47° N, December 21) and a circular loiter pattern, a sun-tracking array can collect more than 2.5 times the energy of a horizontal array assuming it is positioned at the optimum roll angle at each point in time. It is not possible to roll the entire wing to perform the sun-tracking function since the wing still must produce sufficient lift in the "up" direction to maintain level flight. In Concept 10 auxiliary surfaces are used which are not intended to provide any contribution to lift or control of the vehicle, but rather whose sole purpose is to be positioned for maximum energy collection. The stability and controllability of the vehicle will vary with auxiliary surface position, however, with the maximum impact on longitudinal characteristics when the auxiliary surfaces are in the horizontal position and maximum impact on lateral characteristics when they are in the vertical position. Addressing the stability and control issues introduced by the auxiliary surfaces was beyond the scope of this study. Although these surfaces provide additional energy with much greater effectiveness than the horizontal wing array, they also add mass and drag to the configuration without any lift benefit. The mass and drag of these surfaces were accounted for in the analysis by modeling them as tail surfaces. An additional mass penalty of 25 kg was added to account for the mechanism required to rotate the surfaces, and 250 W was included to power this rotation mechanism. The size of the auxiliary surfaces was optimized for the worst case solar conditions of each mission.

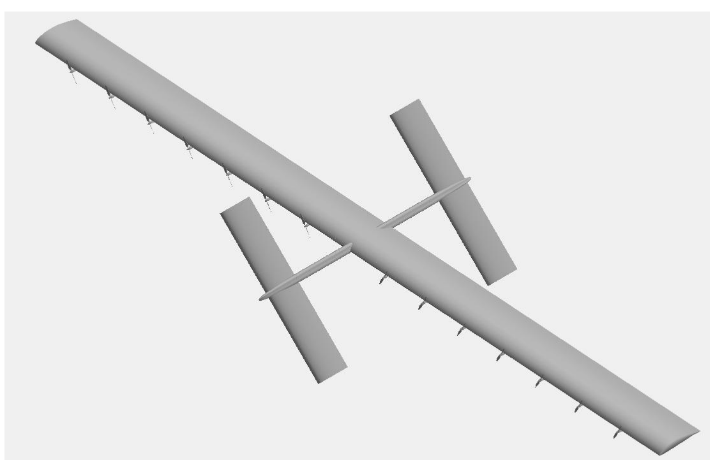


Figure 19. Concept 10: Multi-surface configuration with solar secondary battery propulsion.

For the hurricane science mission, the feasibility of Concept 10 is slightly worse than Concept 7 indicating that the additional energy collected by the auxiliary surfaces is counteracted by their penalties. The increase in solar energy collected (sum of wing and auxiliary arrays) is 12% compared to a fixed horizontal array of the same total area. In the case of the communications relay mission, the increase in solar energy collected is 58% and is sufficient to offset the mass, drag, and power penalties of the surfaces, resulting in feasibility that is slightly greater than Concept 7 (40% versus 36%). The net impact is relatively small despite such a large improvement in solar energy collection. The spot factor for this concept is much larger than the rest of the SR concepts due to the auxiliary array surfaces and booms.

4.3.3 Heavier-Than-Air Solar-Consumable Hybrids

At the beginning of the AoA it was initially expected that hybrid propulsion concepts (a combination of consumable and solar propulsion systems) would provide an attractive alternative. However, prior to developing a series of hybrid propulsion based HTA concepts for the AoA, a more generic assessment of hybrid propulsion systems was conducted. The results of this assessment, discussed below, led to HTA solar-consumable hybrids being excluded from the AoA.

One type of hybrid approach is to supplement a consumable propulsion system with solar energy (with or without energy storage). The additional energy collected from the sun could decrease the rate of on-board fuel use and thereby extend the endurance. Another type of hybrid system is a consumable fuel power generation system augmenting an SR propulsion system. For example, if the energy storage system were completely discharged before sunrise, a consumable system could be started and used until the amount of solar power available was sufficient to power the vehicle. Although hybrid propulsion was approached from both of these mindsets, in reality they are two ends of a single continuum of consumable plus solar configurations. The optimum split between solar and consumable energy for a given design depends on the performance characteristics of each system and the mission requirements. A consumable system can deliver a certain amount of energy per kilogram of mass based on the fuel energy content and the efficiency of the system in converting that fuel energy to useful work. A solar array or SR system can also be considered to deliver a certain amount of energy per kilogram of mass. However, in addition to depending on the system characteristics, the energy delivered per kilogram depends on the mission latitude, time of year, and mission endurance. The energy contained in a kilogram of fuel can be extracted only once, whereas the amount of energy provided by a kilogram of solar cells increases the longer the vehicle is aloft.

The potential endurance benefit from adding solar arrays to a consumable concept can be assessed in simple terms by comparing the energy output of solar cells versus an equivalent mass of fuel. For example, if the solar array has a mass of 100 kg it must be able to provide more energy than could be extracted from 100 kg of fuel (accounting for the efficiencies of both systems). Otherwise, adding more fuel instead of a solar array would provide better performance. (Only performance aspects are being considered here. There is also a cost aspect to this comparison since the solar array is only purchased once and the fuel is purchased for each mission.) Figure 20 illustrates the potential benefit of a hybrid system based on this simple comparison (horizontal solar array versus LH_2 fuel). In Figure 20 the required "breakeven endurance" in days is plotted against mission latitude and day of year. "Breakeven endurance" indicates how long the aircraft must be aloft for the solar array to collect more energy than could be extracted from an equivalent mass of consumable fuel (accounting for additional tank mass). For this simple comparison it is assumed that all of the solar energy collected can be used interchangeably with energy from the consumable system. The breakeven endurance is very sensitive to the mission latitude and time of year. In favorable solar conditions, the breakeven endurance is in the 4 to 5 day range. At the communications relay mission worst case latitude and time of year (47°N, December 21), however, the endurance must be 20 days before the solar array is able to collect enough energy to be superior to LH_2 fuel.

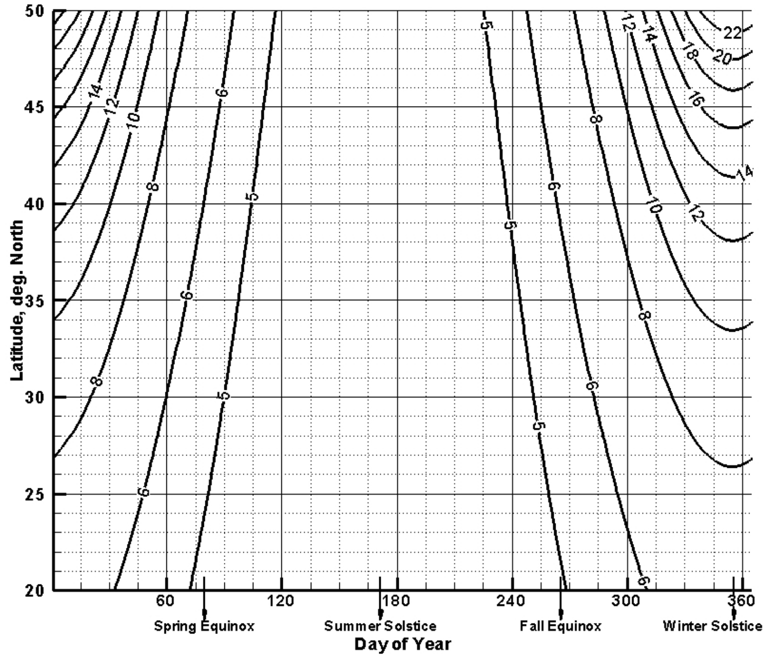


Figure 20. Breakeven endurance (in days) for use of solar array on consumable fuel configuration.

Beyond the simple comparison above there are other important issues to consider. First, the solar energy is collected in the form of electricity whereas some of the consumable propulsion concepts generate mechanical work. If the consumable system is mechanical (such as an IC engine) an additional system (additional mass) is needed to make the two sources of energy compatible and realize the full benefits of the solar energy collected. Otherwise, the solar energy cannot be used for propulsion, but only for electrical power demands such as avionics and payload power. Another issue is the "turn down" or turn off capability of the consumable propulsion system. As noted in Section 4.3.1, high altitude, multiple turbocharged IC systems cannot operate efficiently at low part power. In order to be able to turn off an IC system, a capability for air restart most be provided. An on-board starter system adds mass and complexity to the consumable system. If the consumable system cannot be throttled back and/or turned off when the solar energy is available, there will be little reduction in fuel consumption. Because of these issues, an electric power-based consumable fuel configuration such as Concept 3 (LH_2-fueled PEM fuel cell and electric motor) is most attractive for a solar hybrid system. A third issue in the case of LH_2 fuel configurations is the design of the LH_2 tank. The problem of LH_2 boil-off is mitigated in the consumable concepts by fuel being consumed constantly throughout the mission. That is, the boil-off rate need only be less than the rate at which fuel is consumed to avoid loss of fuel or tank overpressure. If the hybrid approach is successful in reducing fuel flow for part of mission, then the LH_2 tank insulation must be increased to further restrict boil-off.

The potential benefits of adding a solar array are relatively small even if issues such as LH_2 tank boil-off are ignored. At favorable solar conditions, a solar array added to Concept 3 would be able to provide about 20% of the required energy over the period of a day. It takes about 4 days of flight for the resulting amount of fuel saved to equal the mass of the solar array. At 10 days of endurance, the extra energy provided is equivalent to about 1.3 day's worth of fuel, i.e., endurance can be increased by 1.3 days. Therefore, the maximum benefit of the hybrid approach applied to a 10-day Concept 3 type vehicle is about a 13% increase in endurance. Note that this benefit grows with increased endurance since the cumulative energy provided by the solar array increases while the mass is fixed. Figure 21 shows the percentage endurance increase versus latitude and mission start date assuming 10-day endurance.

Although a potential 13% increase in endurance is significant, the benefit is less for less favorable solar conditions and adding a solar array becomes an endurance penalty at high latitude, winter conditions. Because of the wide variation in benefit from the solar array, unless the operation of the vehicle will be limited to a specific latitude and time of year, the extra endurance from solar energy cannot be assured for a given mission. Any endurance requirement for the vehicle would have to be met under the worse case solar conditions.

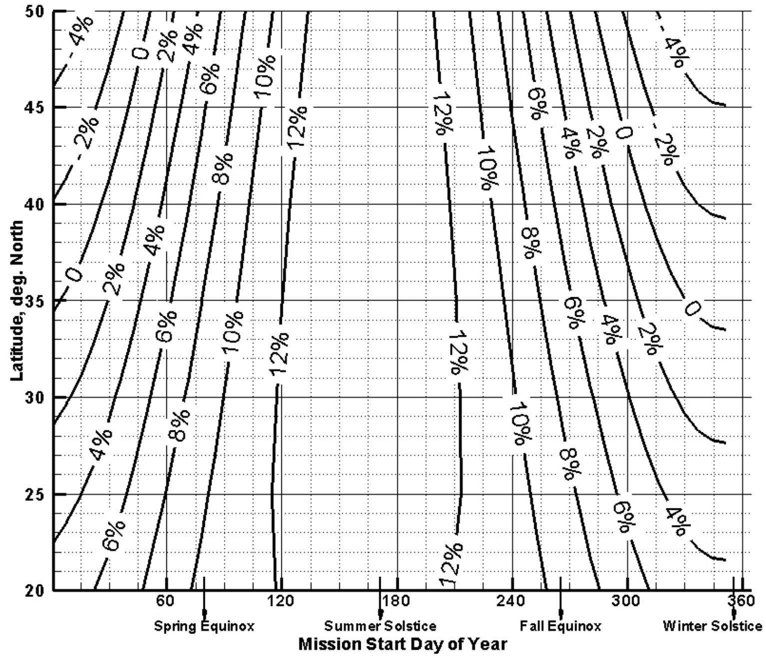


Figure 21. Percentage increase in endurance from adding a solar array to a 10-day consumable configuration.

When considering a solar-consumable hybrid approach a choice must be made whether or not to include an ESS. Without energy storage, it is possible that not all of the solar energy available will be useable. During mid-day the power output of the solar array could be more than the power required to operate the vehicle. The excess power would be lost, reducing the effective energy collection of the array. Also, the maximum power capability of the consumable fuel propulsion system has to be sized assuming no solar power so that requirements such as minimum climb rate can be met at night. With energy storage, supplemental power can be provided at a constant level throughout the day and night enabling the consumable system to be sized for a lower power level. This can also alleviate the turndown problem discussed in Section 4.3.1 since the consumable system can be held at a constant output level and does not have to be throttled down when the sun rises. An energy storage system, on the other hand, adds a significant amount of mass to the supplemental solar power system and reduces the effective energy per kilogram per day that is provided. Using the SR system assumptions of Concept 7, at high latitude, summer solstice conditions (most favorable solar day), the energy supplied by the SR system per day is effectively 500 W-h/kg of system mass (array+battery). When this is compared to the consumable fuel system, the breakeven endurance at which the SR system provides the same amount of energy as extracted from an equal mass of fuel is about one month. Adding an SR system to a consumable fuel configuration is only beneficial for concepts capable of very long endurance.

As previously mentioned, another approach to a hybrid system is to use a consumable capability to augment an SR configuration. However, once a consumable system is added to an SR configuration, it becomes a limiting factor in the endurance. The design mindset immediately changes from the “indefinite endurance" mindset of an SR system to the fuel flow rate and fuel load mindset of a consumable system. This type of hybrid approach then reduces to the one just discussed, a consumable system supplemented with an SR power supply. The difference is in how the total power requirement is split between the consumable system and SR system. As noted above, given the propulsion system assumptions of Concept 3 and Concept 7 and the mission requirements, the optimum consumable-SR split is an "all consumable" system.

Because of the results of the initial exploration of hybrid options, no HTA hybrid concepts were developed for the AoA. The use of hybrid propulsion was still considered for the LTA concepts, however, because the very long endurance of the LTA concepts increases the effective "energy content" of the solar energy systems.

4.3.4 Lighter-Than-Air Concepts (Concepts 11-16)

Analysis of the LTA airship concepts (see Table 2) differed from that of the HTA configurations due to the physical characteristics of the vehicles and the emphasis of the codes used to design and analyze the vehicles. For example, the generous volume and surface area of the airships eliminate critical airplane sizing constraints such as volume required for equipment and payload and area required for solar cells. The buoyant lift of the airship presents the potential for high altitude long endurance missions, because in light wind relatively small amounts of power are required to enable the airship to maintain station or to move between stations. In strong winds, however, the power required to move such a big vehicle through the air becomes prohibitively large, station keeping ability is lost, and the endurance can decrease to less than one day. The effect of strong winds can be offset by adding lifting gas volume to carry a more powerful propulsion system, but at some point the vehicle becomes excessively large from a cost and/or operational standpoint. In contrast, a HTA vehicle generates sufficient lift at loiter speeds that are generally much greater than the wind speed.

LTA Design Considerations

For this study, the maximum design speed of the airships was set by the highest wind speed encountered during any mission. The propulsion system was sized to meet the power required at that speed, but a volume constraint was applied to limit the vehicle to an acceptable size. The maximum volume constraint was 415,000 m^3, which is about 80% of the size of existing hangar facilities. For comparison, the volume of the Goodyear blimp Eagle is 5740 m^3 (59 m long and 15 m wide) and the volume of the Hindenburg was 212,000 m^3 (245 m long, 41 m wide). The required transit speed from base to the mission site defined a minimum design cruise speed, which could possibly exceed the mission maximum wind speed. However, for both missions of this study the transit speed was less than the mission maximum wind speed. The 99th percentile wind speed was used to set the maximum design speed for the airships. This speed is not encountered during many simulated mission sorties, but usually exists for short durations when it is encountered. The time-integrated wind speed at all times during the mission is an indicator of the total energy required for the mission. The power requirement as a function of time can be met using combinations of consumable fuel, solar power, or regenerative systems.

The critical airship design variables and the first-order design impacts of these variables are shown in Table 18.

Table 18. Critical Airship Design Considerations

Design Consideration	First Order Design Impact
Hangar Size	Maximum airship size limit
99^{th} percentile wind speed	Allowable maximum mission wind speed
Maximum mission wind speed	Mission power required
Time-integrated wind speed	Mission energy required
Propulsion system type, mission power required, and mission energy required	Fuel or reactant mass, solar cell area, propulsion system mass, vehicle mass and volume
Fabric, Structure, Lifting Gas, Altitude, Payload	Vehicle mass and volume

LTA Fabric Material

The hull and ballonet fabric type and thickness affect a significant portion of the mass of the vehicle; so it is important to ensure that this part of the design is reasonably modeled. The basic problem is to determine the fabric type and thickness based on the amount of structural stiffening and internal pressure required to maintain the vehicle shape. Complications to this problem include the need for seams and stitching; UV protection; coatings to prevent lifting gas leakage; avoiding cracks, wrinkles, and delamination; temperature tolerance; minimizing elongation under strain; and applying reasonable factors of safety. The vehicle shape is maintained by a combination of internal pressure and structural stiffening. It was assumed that a maximum internal pressure of one half the dynamic pressure would be adequate to maintain the shape of the vehicle. At 18 km and 200 km/h (communications relay mission), half the dynamic pressure is 90 Pa. In addition, daytime heating of the lifting gas also exerts pressure on the hull. A thermal equilibrium code developed by ILC Dover was used to estimate the pressure from the heating of the gas inside the airship. Among other factors, it includes direct and reflected energy with consideration of Earth's albedo. Applying this code to the airship design results in an equilibrated day/night temperature ratio, and hence pressure ratio, of 113%. At 18 km, the ambient pressure is 7.17 kPa. Thus, the minimum internal pressure to maintain shape at night is 7.17 + 0.090, or 7.26 kPa, the maximum daytime pressure is 8.21 kPa, and the total fabric design pressure differential is 1.03 kPa.

The hull fabric carries the hull tensile structural loads and also encloses the ballonets. The fabric is woven from yarns made of multiple fibers, and is typically coated to protect the fibers from UV and weather. The hull is manufactured in fabric strips (gores), which are bonded together by adhesives or thermal welding. The gores of material are usually joined using a tape strip of approximately 5 cm in width, overlaying a butt joint. Welding is occasionally used, but this process is normally done with films for zero pressure balloons, which have significantly lower tensile stresses.

The nature of fibers and fabrics is such that typical stress terms used for isotropic materials are not applicable; the yarn thickness is not easily measured and the fabric may consist of significant voids in the weave. After the manufacturing process, the actual fiber allowable stress is substantially below the ideal fiber allowable stress for the following reasons:

- Fibers must be woven to allow for bi-axial strength needed to prevent aneurism-type failures under load. Weaving significantly reduces the effective strength because fibers are not straight, and therefore do not carry all the load in the stress direction.

- Yarn consists of individual fibers that must be coated and supported in a matrix for bonding, and the fibers must have space between them for weaving. A tighter weave creates more curvature in the fibers, which reduces the working strength and stiffness.

- In a laminate, all loads are transferred to and from the fiber by the adhesion to other fibers, either by an adhesive matrix or welding.

- The inflation of an airship can be a fairly harsh process; the fabric creases and unfolds, and slides over itself during this process.

- Woven fabrics need to be designed to tolerate local punctures and failures. If all the fibers are loaded to their limit and one is broken, the adjacent fibers must carry the extra load. This can cause them to fail, by the so-called zipper effect, and the fabric can be ripped apart.

Two lightweight fabric materials commonly known for high specific strength are Kevlar® and Spectra®. Kevlar® has poor abrasion resistance and flex cracking characteristics, which limits airship applications due to the need for inflation and handling. Spectra® fiber has one of the highest strength-to-weight ratios of any man-made fiber and is also highly resistant to flex fatigue and UV light (ref. 31). Other fabrics considered in Phase I were sufficiently limited in one or more categories as to be eliminated. Therefore, Spectra® was assumed for the Phase I concepts.

LTA Propulsion and Power Considerations

Many of the differences among the airship concepts are associated with the choice of power and energy systems to meet the vehicle power demand as a function of time throughout the mission. These choices set the masses of the propulsion system, fuel tanks, fuel, solar cells, regenerative equipment, and batteries. For this reason, it is important to track the power demands and energy balance of the vehicle over short time intervals as the vehicle is subjected to changing wind speed and solar energy flux. For this study, wind data was reduced to one-hour intervals over the 180 day mission so that the vehicle power as a function of time could be calculated hourly. The power is time integrated over the entire mission to ensure that the energy balance cycle closes at either the end of the mission or at the most demanding point of the mission. This contrasts with the HTA SR vehicle designs which close the energy balance cycle once every 24 hours. The power systems used for the LTA concepts included the LH_2-fueled IC engine (Concept 11), the LH_2-fueled PEM fuel cell system (Concept 12), the SR fuel cell system (Concepts 13 and 16), the SR secondary battery system (Concept 14), and a solar PEM fuel cell hybrid system (Concept 15). The propulsion system assumptions used for the LTA concepts were identical to those in the HTA analysis except that thin film flexible solar arrays were utilized for the LTA concepts.

The power balance during an illustrative three-day hypothetical mission (consisting of 72 hourly segments) is shown in Figure 22 (a)-(d). The charts show the power required for fixed station keeping as the wind speed changes with time; the amount of solar power available to meet the power requirement, recharge an energy storage system, or both; and the net amount of power required from the fuel cell (or internal combustion engine, as the case might be). Each case in Figure 22 (a)-(d) can be paired with a complementary chart of the same letter in Figure 23 (a)-(d). Figure 23 (a)-(d) shows how the normalized fuel level or energy storage level changes with time for each type of power system. For the consumable fuel concepts, the fuel amount is calculated and minimized using an iterative process such that there is no fuel remaining after the last time step of the mission.

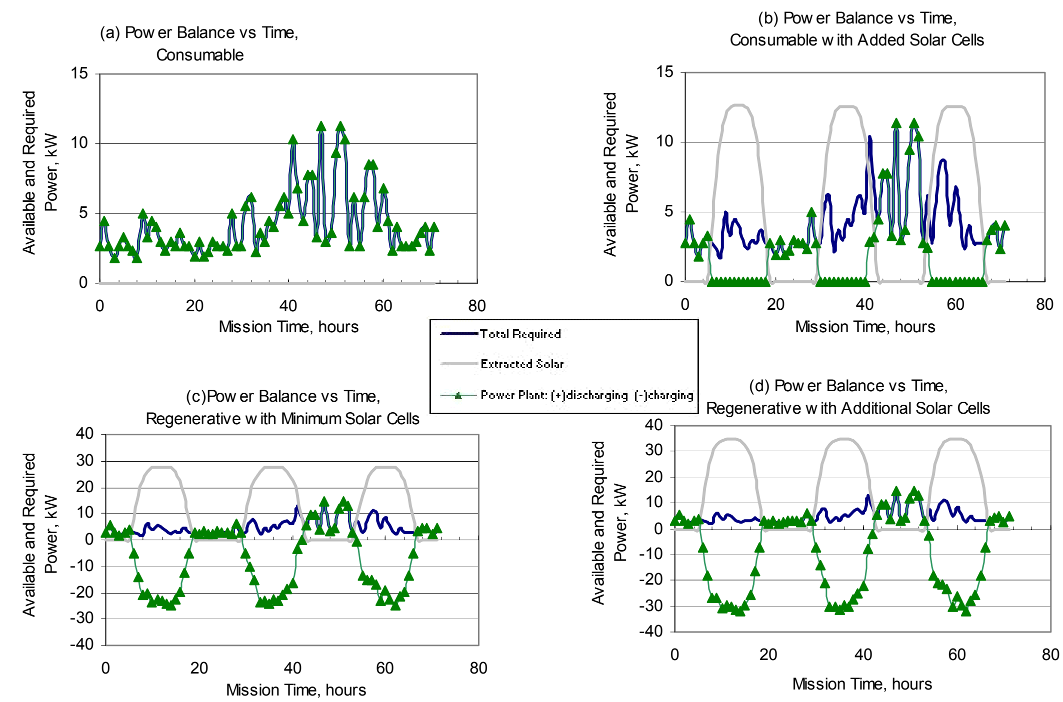


Figure 22. Power balance versus time for an illustrative three-day hypothetical mission.

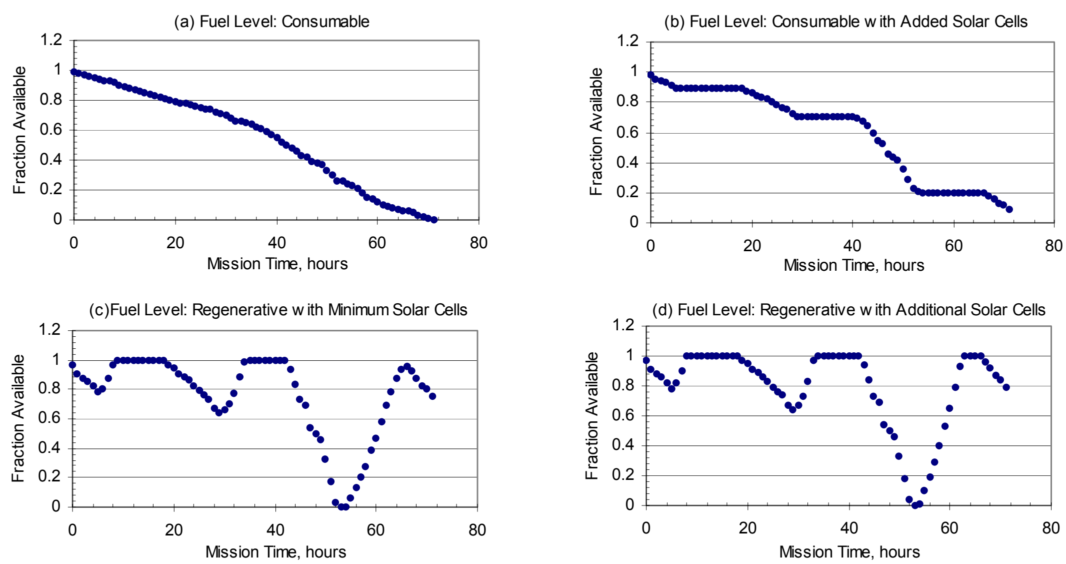


Figure 23. Fuel level versus mission time for an illustrative three-day hypothetical mission.

In case (a) an internal combustion engine or primary fuel cell converts stored energy to power and exactly meets the power demand with no solar power supplement. Fuel is consumed throughout the mission such that the fuel level monotonically decreases from 1 at the beginning to 0 at the completion of the mission. Concepts 11 and 12 are represented by case (a).

In case (b) a combustion engine or primary fuel cell converts stored energy to power and exactly meets the power demand with no solar power supplement at night. During the morning and evening the power demand is met with a combination of consumable fuel and solar power, but during most of the day the solar power meets the entire power demand. Fuel is consumed throughout the mission such that the fuel level decreases from 1 at the beginning to 0 at the completion of the mission; but during most of the day fuel is not consumed. These portions of the mission are flat segments in Figure 23. The solar cells enabled a significant reduction in fuel mass, and reduced the total vehicle mass by close to 20%. Concept 15 is represented by case (b).

In case (c) a regenerative fuel cell or secondary battery converts stored energy to power and exactly meets the power demand with no solar power supplement at night. During the morning and evening, the power demand is met with a combination of fuel cell power and solar power, but during most of the day the solar power meets the entire power demand. The difference between this case and case (b) is that excess solar power is required during the day to regenerate reactants in the case of the regenerative fuel cell, or recharge the battery in the case of a secondary battery energy storage system. The use of a regenerative fuel cell decreases the amount of H_2 required to complete the mission. The need for H_2 is eliminated completely in the case of battery energy storage. However, to be effective the reactant mass savings must be greater than the mass associated with additional equipment such as a larger solar array and either a battery or the electrolyzer and water storage tanks. For the regenerative fuel cell case, reactants are processed throughout the mission such that the reactant level decreases from 1 at times of complete recharge to 0 at the point of highest persistent energy demand during the mission. An equivalent analogy for the batteries would show that the battery energy is consumed and replenished throughout the mission in a similar manner.

The solar array for case (c) was sized using a method to minimize the amount of solar cell area needed to exactly meet the mission energy demand. Since the airship generally has excess area available for additional solar cells, the effect of adding additional solar cells during the initial design iteration phase was investigated. By comparing case (c) with case (d), in Figure 22, it is evident that extra solar energy is collected which results in more fuel cell charging (more negative values). It was found that the additional energy collected could be used to reduce the depletion of the reactants during discharge cycles. This decreases reactant mass, reactant tank mass, and the mass of the fuel cell stack. Although this excess energy cannot directly increase the electrolyzer recharge rate, it decreases the amount of H_2O which needs to be converted back into reactants, decreasing mass by decreasing the number of electrolyzers needed. This effect is most notable when comparing the final recharge cycle in Figure 23 for cases (c) and (d). In case (d), the fuel cell is recharged and the electrolyzers are not needed for four time steps before discharge begins, whereas in case (c) the electrolyzers have not quite recharged the system before discharge begins. Concepts 13, 14, and 16 are represented by case (d).

LTA Concepts

The mass properties for Concepts 11-16 are presented in Table 19 (hurricane science mission) and Table 20 (communications relay mission). The fuel cell and electrolyzer ancillary masses for Concepts 12, 13, 15, and 16 are accounted for in the fuel cell stack row since the available data included the ancillary masses with the main fuel cell stack. Corresponding geometry, propulsion, and mission performance data are shown in Table 21 and Table 22.

Table 19. Mass Data for LTA Concepts 11-16, Hurricane Science Mission

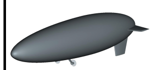
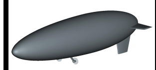
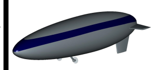
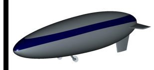
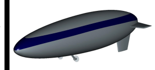
	Concept 11	Concept 12	Concept 13	Concept 14	Concept 15	Concept 16
LTA Masses (kg)						
Item	IC Engine LH_2 Fuel	PEM Fuel Cell LH_2 Fuel	Solar Regen Fuel Cell	Solar 2nd Battery	Hybrid: PEM FC LH_2 + Solar	Aeroship Regen Fuel Cell
Solar Cell Mass	0	0	313	175	168	422
Hull: Fabric	4732	4241	2813	3906	3733	4534
Hull: Suspension	50	50	50	50	50	50
Hull: Nose Reinforce	50	50	50	50	50	50
Hull: Access/Maintenance	50	50	50	50	50	50
Hull: Other	0	0	0	0	0	807
Hull: Ballonet	1893	1696	1125	1562	1493	1814
Hull: Tail Structure+Fabric	802	719	477	662	633	769
Hull Mass	7577	6806	4566	6281	6008	8073
Propulsion: Mech. System	1475	763	529	709	681	659
Propulsion: Fuel + Tank (IC only)	8917	0	0	0	0	0
Propulsion Mass	10392	763	529	709	681	659
Mission Payload	412	412	412	412	412	412
Fuel Cell Stack	0	1807	1366	0	1608	1707
Fuel Cell Ancillaries	0	0	0	0	0	0
H_2 Fuel + Tank	0	5799	390	0	3992	532
O_2 Oxydizer + Tank	0	0	731	0	0	998
Water Tank	0	0	22	0	0	30
Electrolyzer Stack	0	0	92	0	0	92
Electrolyzer Ancillaries	0	0	0	0	0	0
PMAD	0	6	5	5	6	5
Batteries	0	0	0	6203	0	0
Energy System Mass	0	7612	2606	6208	5606	3364
Total Mass	18381	15593	8426	13784	12876	12930

Table 20. Mass Data for LTA Concepts 11-16, Communications Relay Mission

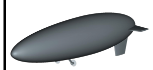
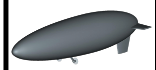
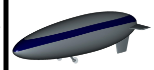
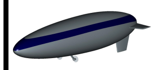
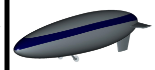
	Concept 11	Concept 12	Concept 13	Concept 14	Concept 15	Concept 16
LTA Masses (kg)						
Item	IC Engine LH_2 Fuel	PEM Fuel Cell LH_2 Fuel	Solar Regen Fuel Cell	Solar 2nd Battery	Hybrid: PEM FC LH_2 + Solar	Aeroship Regen Fuel Cell
Solar Cell Mass	0	0	378	174	125	466
Hull: Fabric	2439	2923	3585	5456	2745	5078
Hull: Suspension	50	50	50	50	50	50
Hull: Nose Reinforce	50	50	50	50	50	50
Hull: Access/Maintenance	50	50	50	50	50	50
Hull: Other	0	0	0	0	0	902
Hull: Ballonet	976	1169	1434	2183	1098	2031
Hull: Tail Structure+Fabric	413	496	608	925	465	861
Hull Mass	3978	4738	5777	8714	4459	9022
Propulsion: Mech. System	2691	1798	2165	3173	1699	2424
Propulsion: Fuel + Tank (IC only)	4084	0	0	0	0	0
Propulsion Mass	6775	1798	2165	3173	1699	2424
Mission Payload	212	212	212	212	212	212
Fuel Cell Stack	0	4299	5688	0	4059	6372
Fuel Cell Ancillaries	0	0	0	0	0	0
H_2 Fuel + Tank	0	3335	1783	0	2535	2086
O_2 Oxydizer + Tank	0	0	3343	0	0	3912
Water Tank	0	0	100	0	0	117
Electrolyzer Stack	0	0	92	0	0	92
Electrolyzer Ancillaries	0	0	0	0	0	0
PMAD	0	6	5	5	6	5
Batteries	0	0	0	24413	0	0
Energy System Mass	0	7639	11012	24418	6600	12585
Total Mass	10965	14387	19544	36692	13095	24709

Table 21. Geometry, Propulsion and Performance Data for Concepts 11-16, Hurricane Science Mission

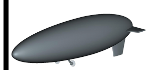
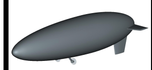
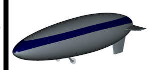
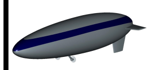
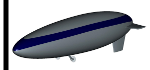
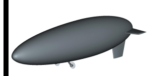
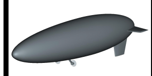
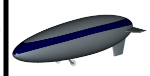
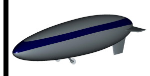
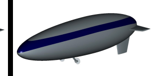
	Concept 11	Concept 12	Concept 13	Concept 14	Concept 15	Concept 16
						
Item	IC Engine LH_2 Fuel	PEM Fuel Cell LH_2 Fuel	Solar Regen Fuel Cell	Solar 2nd Battery	Hybrid: PEM FC LH_2 + Solar	Aeroship Regen Fuel Cell
Geometry						
Design Lifting Gas Volume, m^3	281877	239126	129209	211382	197449	178458
Hull Envelope Area, m^2	25610	22950	15225	21139	20199	24535
Maximum Skin Thickness, mm	0.1524	0.1524	0.1524	0.1524	0.1524	0.1524
Solar Cell Area, m^2	0	0	1601	894	861	2155
Vehicle Length, m	187.5	177.5	144.6	170.4	166.5	197.2
Vehicle Width, m	53.6	50.7	41.3	48.7	47.6	65.7
Vehicle Height, m	53.6	50.7	41.3	48.7	47.6	26.3
Propulsion						
Drag @ Max Power Req, N	7810	7063	4849	6550	6283	6073
C_D, total drag coefficient	0.02763	0.02788	0.02885	0.02807	0.02818	0.02914
Total Power Required, kW	327.4	296.3	204.0	274.9	263.8	255.0
Number of Fuel Cells or Batteries	0	54	12	13970	48	14
Number of Electrolyzers	0	0	6	0	0	9
Max Continuous Discharge Time (hrs)	0	0	17.0	22.0	0	17.0
Max Continuous Recharge Time (hrs)	0	0	13	13	0	13
Propulsion Mass Factor, kg/kW	4.505	2.533	2.533	2.533	2.533	2.533
Engine SFC, kg/kW/h	0.08	0	0	0	0	0
Mission						
Start Date	1-Jun	1-Jun	1-Jun	1-Jun	1-Jun	1-Jun
End Date	30-Nov	30-Nov	30-Nov	30-Nov	30-Nov	30-Nov
Max. Mission Wind Speed, m/s	42.4	42.4	42.4	42.4	42.4	42.4
Max.Design Speed, m/s	42	42	42	42	42	42

Table 22. Geometry, Propulsion and Performance Data for Concepts 11-16, Communications Relay Mission

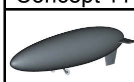
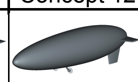
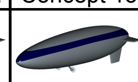
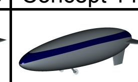
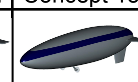
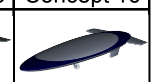
	Concept 11	Concept 12	Concept 13	Concept 14	Concept 15	Concept 16
						
Item	IC Engine LH_2 Fuel	PEM Fuel Cell LH_2 Fuel	Solar Regen Fuel Cell	Solar 2nd Battery	Hybrid: PEM FC LH_2 + Solar	Aeroship Regen Fuel Cell
Geometry						
Design Lifting Gas Volume, m^3	104286	136834	185887	348988	124547	211512
Hull Envelope Area, m^2	13198	15818	19403	29529	14857	27479
Maximum Skin Thickness, mm	0.1524	0.1524	0.1524	0.1524	0.1524	0.1524
Solar Cell Area, m^2	0	0	1933	891	639	2383
Vehicle Length, m	134.6	147.4	163.2	201.4	142.8	208.7
Vehicle Width, m	38.5	42.1	46.6	57.5	40.8	69.6
Vehicle Height, m	38.5	42.1	46.6	57.5	40.8	27.8
Propulsion						
Drag @ Max Power Req, N	10719	12655	15261	22426	11948	17101
C_D, total drag coefficient	0.02570	0.02532	0.02489	0.02403	0.02545	0.02559
Total Power Required, kW	597.4	704.9	849.7	1247.8	665.6	951.9
Number of Fuel Cells or Batteries	0	129	46	54985	122	52
Number of Electrolyzers	0	0	4	0	0	5
Max Continuous Discharge Time (hrs)	0	0	22.0	19.0	0	23.0
Max Continuous Recharge Time (hrs)	0	0	15	15	0	15
Propulsion Mass Factor, kg/kW	4.505	2.533	2.533	2.533	2.533	2.533
Engine SFC, kg/kW/h	0.08	0	0	0	0	0
Mission						
Start Date	1-May	1-May	1-May	1-May	1-May	1-May
End Date	28-Oct	28-Oct	28-Oct	12-Oct	28-Oct	27-Oct
Max. Mission Wind Speed, m/s	71.6	71.6	71.6	69.5	71.6	71.6
Max.Design Speed, m/s	56	56	56	56	56	56

Values for the AoA metrics are shown in Table 23 for each hurricane science and communications relay airship concept in this study. Mass, geometry and key mission parameters are shown for each concept, as optimized for each mission. Because the airship size increases as wind speed and altitude increase, it is difficult to operate a vehicle designed for the hurricane science mission on the communications relay mission and vice versa. For example, the volume and amount of lifting gas for a vehicle designed to operate at 18 km must be increased by 1.6 times to operate at 21 km, excluding all consideration of the increase in every subsystem mass as required to enclose the additional gas and power the vehicle. This is unlike a conventional HTA vehicle design in which the wing area and cruise speed might be optimized for sufficient performance in both missions. As a result, the airship hurricane science and communications relay missions are different designs.

Another factor to consider is that an airship with 180 days of endurance during the summer months is not capable of the same endurance during winter months when winds are stronger and the available solar energy decreases. The hurricane science mission is approximately 180 days long during the summer, starting June 1 and ending November 30; all concepts meet that endurance requirement. For the year-round coverage required by the communications relay mission, however, the endurance of each concept varies depending on the start date of the mission. If a mission start date of May 1 is chosen, all concepts can meet the 180 day mission endurance requirement except Concept 14 (secondary batteries), which has an endurance of 165 days. If a mission start date of January 1 is chosen, none of the communications relay concepts can meet the 180 day mission endurance requirement. The primary reason is that strong winter winds require sustained high power levels and the vehicle cannot be sized to meet those power levels without violating the maximum volume constraint. Also, the available solar energy is close to a minimum at this time of year due to the short days, which leads to high reactant use through long discharge cycles and short recharge cycles. The solar angle is also close to a minimum during this time of year, but this can be mitigated by mounting the solar array strip closer to the side of the airship than the top. Geometric and mass data shown in Table 23 for the communications relay mission correspond to a May 1 mission start date. This mission start date was selected so the majority of concepts would meet the 180 day mission endurance requirement. Some of the communications relay vehicles are smaller than the corresponding hurricane vehicles since during the summer months the winds near 50° N latitude are more benign than the winds near 20° N latitude. Also, the lower mission altitude is beneficial to the communications relay mission vehicle design size and mass.

Concept 16 augments the buoyant lift with aerodynamic lift to present a hybrid-lift vehicle in addition to the hybrid-propulsion vehicle. It was found that the "aeroship" design volume was highly dependent on the vehicle lift-to-drag ratio. Assuming a total of 10% aerodynamic lift, an optimistic L/D of 5, and using the same propulsion system design parameters as for Concept 13, the aeroship size was significantly higher than the basic airship model. In order to generate aerodynamic lift, the shape is flatter and broader and some additional lifting surface is needed, resulting in increased structural mass. This additional mass is reflected in the "Hull:Other" rows of Table 19 and Table 20. The aeroship shape is also different from that of the airships. The non-circular cross section requires more fabric to enclose a given volume of lifting gas than does a circular cross section, resulting in increased hull fabric weight as well. For realistic, lower values of L/D, the vehicle size rapidly exceeded the design volume constraint. In addition, the increment in fuel mass necessary to maintain a minimum flight velocity in light winds was not modeled for the Concept 16 design, so the aeroship size was underestimated. Based on the predicted sizes and masses, the aeroship concept is a less attractive option than other airship designs.

Table 23. AoA Metric Results for LTA Concepts 11-16

			Concept 11	Concept 12	Concept 13	Concept 14	Concept 15	Concept 16
								
			IC Engine LH_2 Fuel	PEM Fuel Cell LH_2 Fuel	Solar Regen Fuel Cell	Solar Regen Secondary Battery	Hybrid: PEM FC LH_2 + Solar	Aeroship Solar Regen Fuel Cell
Evaluation Criteria	Endurance, days	Hurricane Science	180.0	180.0	180.0	180.0	180.0	180.0
		Communications Relay	30-180**	36-180**	1-180**	1-165**	36-180**	1-179**
	TOGM, kg	Hurricane Science	18381	15593	8426	13784	12876	12930
		Communications Relay	10965	14387	19544	36692	13095	24709
	Wingspan (HTA) or Length and Width (LTA), m		135x39	147x42	163x47	201x58	143x41	209x70x28
	Volume (LTA), m^3		104286	136834	185887	348988	124547	211512
	$\%P_{regen}$ (SR)	Hurricane Science	n/a	n/a	100	100	n/a	100
		Communications Relay	n/a	n/a	100	100	n/a	100
	Takeoff and Landing Robustness %	Hurricane Science	97	97	62	62	97	62
		Communications Relay	72	72	58	58	72	58
	Ground Footprint	Spot Factor	(xx/38.0)*	(xx/21.0)*	(29.0/40.0)*	(40.0/40.0)*	(21.0/18.0)*	(31.0/40.0)*
		Support Required	11	12	9	1	12	7
	Growth Factor		2.3	2.4	2.7	2.8	2.5	2.8
	Risk		3	6	8	7	6	8

*(Hurricane Science/Communications Relay)

**The lower endurance limit was set by the maximum endurance for missions starting January 1 for vehicles constrained to volumes less than 415,000 m^3. The upper endurance limit was set by the maximum endurance for missions starting May 1 (Communications Relay) or June 1 (Hurricane Science) with volume constraint of 415,000 m^3

4.4 Concept Down Select

Based on the AoA metric results, a concept down select was performed. The primary discriminator used was endurance, with consideration given to the other metrics as well, particularly takeoff gross mass and risk. Although none of the HTA consumable concepts met the threshold endurance requirement for either mission, Concept 1 (LH_2-fueled IC engine propulsion) showed the greatest endurance, closely followed by Concept 3 (LH_2-fueled PEM fuel cell and electric motor). However, Concept 3 ranked higher in risk, due primarily to the relatively complex and unproven propulsion system. In addition, the takeoff gross mass for Concept 3 was higher in both missions than Concept 1. Therefore, Concept 1 was selected for Phase II of the study.

As previously described, none of the HTA SR concepts could close for either of the two study missions (See Table 16). That is, the SR system could not provide sufficient energy even on the most favorable day-night cycle of the required mission period. Concept 7 (all-wing, secondary battery) and Concept 10 (multi-surface, secondary battery) had the highest $\%P_{regen}$ values; however, in neither case was the SR system able to supply more than half of the power required. Although Concept 10 had better performance than Concept 7 on the communications relay mission, the performance benefit was too small to justify the added complexity of the variable geometry surfaces. These surfaces did not provide a net benefit for the hurricane science mission due to the lower latitudes of the operational area. Therefore, Concept 7 was selected for Phase II of the study. For this concept, technology and mission requirements trade studies were conducted for Phase II since an operational and cost analysis of an infeasible concept was deemed of little value.

The LH_2-fueled IC engine and the LH_2-fueled PEM fuel cell propulsion systems provided excellent endurance for the LTA concepts (Concepts 11 and 12). The endurance of these concepts was sufficient to complete the full hurricane science mission, and, depending on the time of year, a large part of the communications relay mission. However, the feasibility of operating the IC engine of Concept 11 continuously over the entire mission is questionable. The oil supply required by the IC engine for lubrication and cooling would deplete over time. The engine would have to be designed to minimize this use of oil or the oil would have to be replenished during the mission. In addition, the mechanical components of the IC engine would have to perform over endurances not typical for aviation applications. Concept 12, utilizing a PEM fuel cell, avoids these issues. None of the LTA SR concepts were attractive due to their lack of endurance and large size and mass for the communications relay mission. Unlike the HTA hybrid concepts, the LTA hybrid PEM fuel cell plus solar array system of Concept 15 showed promising performance. (A hybrid IC engine plus solar array concept was not investigated due to the extreme endurance issues with the IC engine noted above.) As discussed in Section 4.3.3, the cost-benefit relationship for LTA hybrids differs from their HTA counterparts due to the very long endurance of the airship relative to the HTA vehicles. The performance of Concept 15 was similar to that of Concept 12, but Concept 15 was smaller in size and mass. Therefore, Concept 15 was selected for Phase II of the study.

5.0 Operational Concept Study

This section marks the beginning of Phase II of the study and opens with a discussion of the requirements evolution from Phase I to Phase II. A more detailed hurricane science mission payload was defined and attrition and maintenance requirements were added to support the Phase II analysis. Next, the Phase II concepts are discussed, including the down-selected concepts from Phase I re-sized for the Phase II requirements (Concept 1, HTA LH_2-fueled IC Engine and Concept 15, LTA LH_2-fueled PEM fuel cell plus solar cell hybrid) and two additional concepts developed to support the Phase II analysis: "Concept-1 small" and "Concept 5-small." (Concept 5 is the HTA diesel-fueled concept.) The additional concepts are significantly reduced endurance versions of their Phase I counterparts. Results of the operational study are presented for these four concepts, resulting in the fleet size, fuel burn, and maintenance data utilized for the life cycle cost estimates.

5.1 Refined Requirements

The requirements utilized for Phase I were refined prior to entering Phase II based on the results of the concept studies and additional inputs obtained during Phase I. In addition, requirements for operational metrics such as turn around time, attrition intervals, and maintenance plans were developed. The evolution of requirements from Phase I to Phase II is summarized in Table 24.

Table 24. Summary of Requirements Evolution

	Hurricane Science Mission				Communications Relay Mission			
	Initial	Phase I	Phase II Threshold	Phase II Goal	Initial	Phase I	Phase II Threshold	Phase II Goal
Endurance (days)	30-180	30-180	30	180	14-180	14-180	14	180
Payload Mass (kg)	200-500	400	200	350	136-200	200	136	200
Payload Power (kW)	1-2.5	1.5	1	2.5	1-1.5	1.5	1	1.5
Loiter Altitude (km)	21-21+	21	18	21	18-21	18	18	18
Dash Speed (km/h)	150	150	110	150	200	200	200	200
Mission Dates	June-Nov	June-Nov	June-Nov	June-Nov	Year-Round	Year-Round	Year-Round	Year-Round
Latitudes (°N)	10-30	10-30	10-30	10-30	25-47	25-47	25-47	25-47
Operating Base HTA	n/a	Jacksonville	Jacksonville		n/a	Las Cruces	Las Cruces	
Operating Base LTA	n/a	Jacksonville	Lakehurst		n/a	Las Cruces	Lakehurst	
Turn Around Time (hrs)	n/a	n/a	8		n/a	n/a	8	
A check interval (hrs)	n/a	n/a	HTA: 720 , LTA: 1 mission		n/a	n/a	HTA: 720 , LTA: 1 mission	
A check time (hrs)	n/a	n/a	HTA: 48, LTA:120		n/a	n/a	HTA: 48, LTA:120	
C check interval (hrs)	n/a	n/a	HTA: 7200, LTA: 20000		n/a	n/a	HTA: 7200, LTA: 20000	
C check time (hrs)	n/a	n/a	336		n/a	n/a	336	
Attrition Interval (hrs)	n/a	n/a	HTA: 20000, LTA: 40000		n/a	n/a	HTA: 20000, LTA: 40000	
Assured Coverage	n/a	n/a	No		n/a	n/a	Yes	

The initial hurricane science mission payload mass requirement, consisting of estimates for both fixed and expendable elements, was 200 kg (threshold) and 500 kg (goal). A value of 400 kg was selected for use in the Phase I concept studies. To support Phase II, a specific payload build-up was performed utilizing refined estimates for each element. Table 25 shows the build-up for the hurricane science mission payload used for Phase II.

Table 25. Refined Hurricane Science Mission Payload Buildup

Payload Element	Heritage/Source	Mass (kg)	Volume (m^3)	Power (W)
336 dropsondes + receiver	Vaisala	58.6	0.651	60
High-Resolution EO/IR Sensor	NAST-I	127	0.368	970
Multi-channel Microwave Radiometer	NAST-M	100	0.34	600
Active Doppler Radar	Honeywell RDR-4000	57	0.234	800
	Total	**342.6**	**1.593**	**2430**

The number of dropsondes required was estimated by assuming use of one per hour over a 14-day mission, for a total of 336 dropsondes. Actual concept mission endurance times, and hence on-station times, vary widely from approximately three days for Concept 4 up to 176 days for Concept 15. A three-day loiter with 336 dropsondes would allow an average of 4.5 drops per hour. For a 176 day loiter, this decreases to only 1.9 drops per day. However, there will most likely be days or weeks of no storm activity during which no drops are needed. A more detailed payload and operational mission concept study is required to determine the optimal payload for a given vehicle and mission plan. Dropsonde mass was assumed to be 0.1 kg and the size was assumed to be 8 cm in diameter and 30 cm high. The total mass of the dropsondes is 33.6 kg and the mass of the data receiver is 25 kg. Although dropsonde mass and size are smaller than current systems (0.4 kg and 8 cm x 40 cm) (ref. 32), these values are consistent with the TRL constraint for this study and efforts to realize these targets are on-going. The high resolution EO/IR sensor specifications were estimated from the NAST-I (NPOESS Airborne Sounder Testbed – Infrared) instrument, and the microwave sensor from the NAST-M (microwave) instrument. The active Doppler radar was assumed to be of the Honeywell RDR-4000 class. This next-generation weather radar is much lighter, smaller, and requires less power than its predecessors. It is currently being installed by Airbus on the A380 and by Boeing on the 777 and C-17. Based on this payload build-up, the total payload mass, power, and volume utilized for the goal hurricane science mission values for Phase II were 350 kg, 1.6 m^3, and 2.5 kW. Threshold values remained at 200 kg and 1.0 kW, representing an estimate of the minimum useful payload mass and power.

The communications relay mission payload mass and power requirements were unchanged for Phase II. The threshold loiter altitude requirement for the hurricane science mission was reduced from 21 km to 18 km. The goal loiter altitude for the communications relay mission was also reduced from 21 km to 18 km. A loiter altitude of 18 km was used for the communications relay mission analysis in Phase I and the benefit of the higher goal altitude of 21 km did not appear to be worth the size and mass penalty for this application. The threshold dash speed requirement for the hurricane science mission was lowered from 150 km/h to 110 km/h, representing a reduction from 99^{th} percentile winds to 90^{th} percentile (plus a 25 km/h speed to track the moving hurricane). The communications relay dash speed requirements were unchanged. Note that the base of operations for the LTA vehicles was moved to Lakehurst, NJ due to the existence of airship hangars. The HTA operational locations were unchanged. Assured coverage, the last row in Table 24, indicates that a redundant system is required to minimize coverage gaps due to an operational failure.

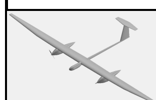
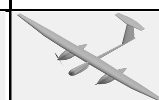
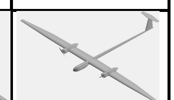
The remaining Phase II requirements changes represent additions needed to support the operational concept and cost analysis. A two-level maintenance concept was assumed, with operational level ("A-check") and depot level maintenance ("C-check"). The assumptions utilized for these maintenance actions, including the interval between checks and time required to perform the check, are shown in Table 24. The A- and C-check times shown represent clock time, not total labor hour time. For example, the HTA 48 hour A-check represents a two day effort. For the cost analysis, this two day A-check was assumed to require three mechanics per eight hour shift and two shifts per day for a total of 96 labor hours. The LTA A-check of 120 hours represents 5 days. For the cost estimate, making similar assumptions in terms of shift loading, the total labor hours are 240. The C-check time is 14 days for both the LTA and HTA concepts.

5.2 Phase II Configuration Descriptions

5.2.1 Heavier-Than-Air Consumable Fuel Concepts

Concept 1 was re-sized to the hurricane science mission Phase II goal requirements, which, compared to Phase I requirements, reduced payload mass by 50 kg but increased payload power by 1 kW to 2.5 kW. The resulting 80 m span vehicle had a slightly reduced takeoff gross mass. Hurricane science mission endurance increased from 7.9 days to 8.1 days. Endurance capability on the communications relay mission remained at 10 days. To support the operational concept study, Concept 1 was also re-sized to a 4-day endurance for the hurricane science mission. The resulting vehicle, termed "Concept 1-small," had a 46 m wingspan, had less than half the mass of the 80 m wingspan vehicle, and used about one-third the fuel. The final HTA vehicle utilized to support the operational concept study was a re-sized version of Concept 5, the diesel-fueled IC engine configuration. "Concept 5-small" was also sized to a 4-day endurance for the hurricane science mission, resulting in a vehicle with a fuel load less than half of Concept 5, a gross mass a little more than half of Concept 5, and a 58 m wingspan. These two additional "small" concepts were developed to assess the sensitivity of the system life cycle cost to vehicle endurance. Table 26 shows the primary geometric parameters for the three HTA concepts used in the operational concept study. The methodology used to size these concepts was the same as used in Phase I.

Table 26. Primary Geometric Parameters for Phase II HTA Vehicles

			Concept 1	Concept 1-small	Concept 5-small
					
Parameter	**Symbol**	**Units**	IC Engine LH_2 Fuel (8-day endurance)	IC Engine LH_2 Fuel (4-day endurance)	IC Engine Diesel Fuel (4-day endurance)
Wingspan	b_s	m	80	46	58
Wing Area	S_w	m^2	254	105	140
Wing Aspect Ratio	AR_w	-	25.2	20.2	24.0
Wing Loading	W/S	kg/m^2	18.8	21.3	18.1
		lb/ft^2	3.81	4.36	3.71
Propeller Diameter	D_p	m	4.2	4.2	3.5
Pod/Nacelle Separation	P_{sep}	m	25	25	25
Pod/Nacelle Length	Pod_L	m	11.8	6.3	8.4
Pod/Nacelle Diameter	Pod_D	m	2.9	2.1	2.1
Fuselage Length	Fus_L	m	29	18.3	29
Fuselage Diameter	Fus_D	m	1.6	1.6	1.6
Horiz. Tail Span	b_{ht}	m	10.8	8.3	6.9
Horiz. Tail Area	S_{ht}	m^2	23.4	13.9	9.5
Horiz. Tail Aspect Ratio	AR_{ht}	-	5	5	5
Vert. Tail Span	b_{vt}	m	6.8	5.3	4.4
Vert. Tail Area	S_{vt}	m^2	23.4	13.9	9.5
Vert. Tail Aspect Ratio	AR_{vt}	-	2	2	2

Drag polars for the three concepts are shown in Figure 24 and L/D versus C_L is shown in Figure 25. Concept 1-small had the highest drag coefficient and lowest loiter L/D due to a proportionately larger nacelle and fuselage size compared to Concept 1 and Concept 5-small. Table 27 shows the mass characteristics for these three concepts. Concept 1-small, the 46 m span LH_2-fueled vehicle, had the lowest gross mass of 2270 kg. The larger, diesel-fueled Concept 5-small was slightly heavier with a gross mass of 2610 kg. This difference was due to the higher diesel fuel mass required for the 4-day endurance hurricane science mission (970 kg of diesel compared to 500 kg of LH_2). Concept 5-small had the lowest empty mass, 1290 kg compared to 1420 kg for Concept 1-small. Note that the 80 m wingspan vehicle sized for maximum endurance (8.1 days) required almost triple the fuel (1460 kg of LH_2) compared to Concept 1-small. That is, nearly triple the fuel load was required to double the endurance for this concept. A summary of the mass and performance characteristics for all three concepts is given in Table 28, including the average transit speeds and communications relay mission performance.

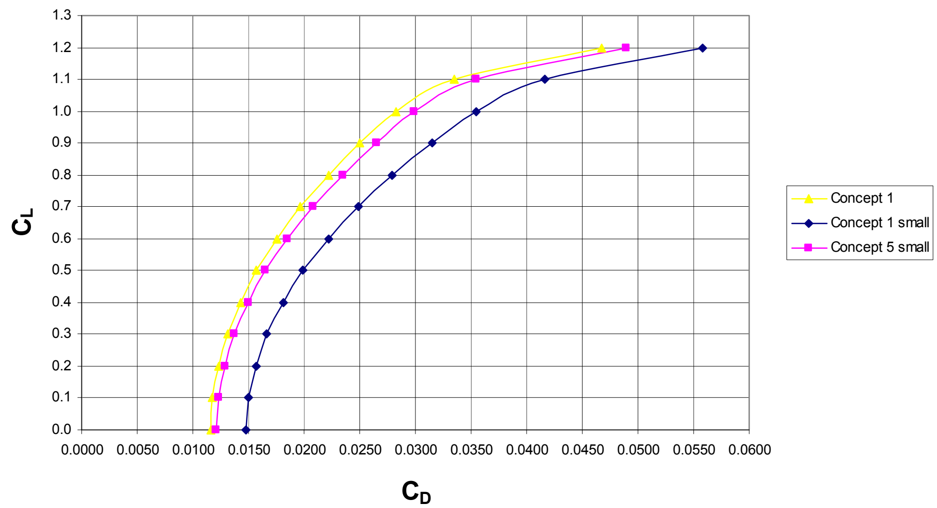


Figure 24. Drag polars for Phase II HTA concepts at loiter conditions.

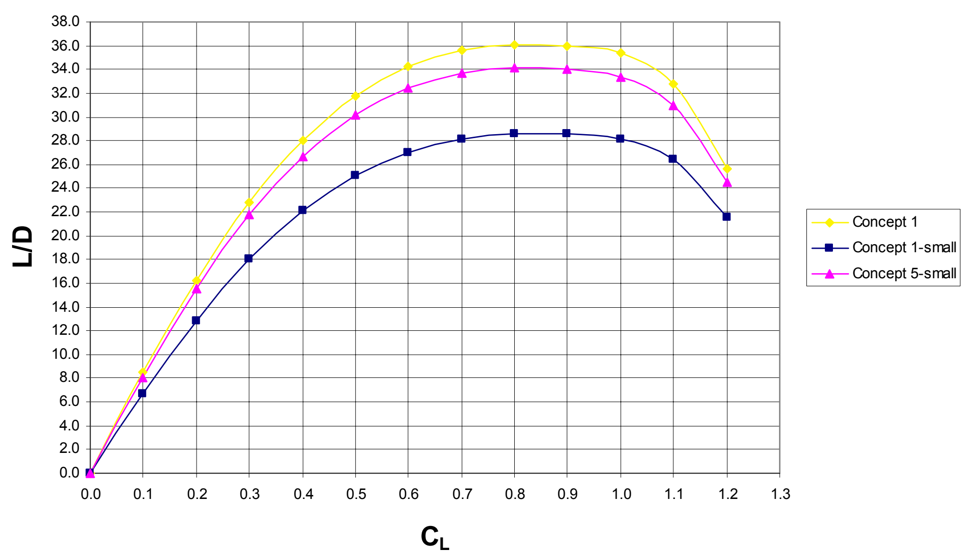


Figure 25. L/D versus C_L for Phase II HTA concepts at loiter conditions.

Table 27. Mass Data for Phase II HTA Concepts

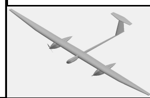
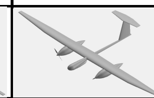
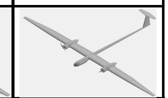
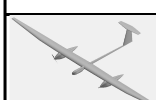
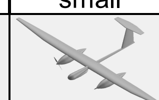
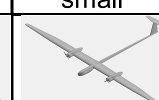
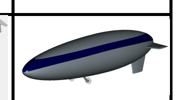
	Concept 1	Concept 1-small	Concept 5-small
			
Component Masses (kg)	IC Engine LH_2 Fuel	IC Engine LH_2 Fuel	IC Engine Diesel Fuel
STRUCTURES:			
Wing	828	287	361
Tail (horizontal and vertical)	84	45	23
Tailboom	127	37	82
Fuselage/body	55	44	45
Landing Gear	86	41	47
Pods/nacelles	395	161	164
TOTAL STRUCTURE:	1575	615	722
PROPULSION:			
Engines, Turbochargers, Radiators	671	412	329
Engine Accessories, Mounts	183	93	49
Propellers	84	48	46
Fuel system	308	151	29
TOTAL PROPULSION:	1246	704	453
EQUIPMENT:			
Servos	48	23	26
Wire/Electrical	81	51	62
Avionics	15	15	15
Backup Battery	12	12	12
TOTAL EQUIPMENT:	156	101	115
(STRUC + PROP + EQUIP) EMPTY MASS:	2977	1420	1290
Payload	350	350	350
Zero Fuel Mass	3327	1770	1640
Fuel	1460	500	970
Takeoff Gross Mass	**4787**	**2270**	**2610**

Table 28. Performance Summary for Phase II HTA Concepts

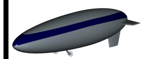
	Concept 1	Concept 1-small	Concept 5-small	Concept 15
				
Propulsion/Fuel Type	IC Engine LH_2 Fuel	IC Engine LH_2 Fuel	IC Engine Diesel Fuel	Hybrid PEM FC + Solar
TOGM Hurricane Science Mission (kg)	4790	2270	2610	20380
Fuel Mass Hurricane Science Mission (kg)	1460	500	970	4104
Endurance - Hurricane Science (days)	8.1	4.0	4.0	180.0
Endurance - Communications Relay (days)	10.0	5.6	5.0	36-180
Average Transit Speed - Hurricane Science (km/h)	250	260	234	103
Average Transit Speed - Communications Relay (km/h)	188	190	173	140

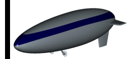
5.2.2 Lighter-Than-Air Concept

The down selected LTA design, Concept 15, was also further refined for the cost and operations study. Most of the changes were based on an in-depth review of the hull fabric parameters and the hull construction techniques. In addition to Spectra®, fabrics considered for Phase II included 710 polyester by Celanese Corporation and Vectran™ HS. 710 polyester is a standard fabric used for airships, aerostats, and logging balloons. Compared to 710 polyester, Vectran™ HS offers superior specific strength, but it is more expensive, must be protected with a UV resistant coating, and production availability is limited. Further investigation into the properties of Spectra® revealed some limitations in airship applications. Spectra® creeps under load, does not accept coatings for bonding seams and for laminating to non-porous films, and the seams cannot be welded due to loss of fiber orientation and strength at higher temperatures (ref. 33). Reference 34 details the use of coated Vectran™ HS fabric at 200 denier in the Mars Pathfinder Lander Air Bags. The areal mass for the basic fabric was 91.5 g/m^2 and the areal mass for the coated fabric was 145.8 g/m^2, with a maximum tensile ribbon stress of 71.6 kg/cm. The coating material was silicone rubber, which protected the fibers from abrasion, UV, and provided a short term barrier to leakage. According to a NASA Dryden Flight Research Center sponsored study for the Exo-Atmospheric Trans-solar wind aircraft (EXAT) program, a 0.013 mm (0.5 mil, thousandth of an inch) polyester film is recommended to prevent leakage of the lifting gas in an airship. At a specific gravity of 1.38, a 0.013 mm polyester film has an areal mass of 17.0 g/m^2, the adhesive bonding adds about 6.8 g/m^2 additional mass. Adding the masses of coatings and bonding to the Vectran™ HS coated fabric mass results in an areal mass of 170.0 g/m^2. For Phase II, it was assumed that Vectran™ HS would be used for the hull and ballonet fabric. In addition to other stiffening factors, the hull mass was calculated as a base hull mass plus 15% for seams, 10% for structure interface, and 40% for ballonet. These allowance factors, although optimistic by current standards of airship design, correlate well with a design proposed in the NASA Dryden EXAT program study, in which the ballonet and seam allowance for an extremely high altitude airship (46 km) was estimated at 65% of the hull base mass.

Another change to the Concept 15 design for Phase II was the use of hydrogen as the lifting gas instead of helium. Hydrogen has numerous benefits. Due to safety considerations, hydrogen is not used for manned airships; however, it is reasonable to assume that it would be acceptable in future unmanned airship applications. H_2 has a specific lift (Newtons of lift per cubic meter of lifting gas) which is 8% greater than He, resulting in a smaller vehicle if all other factors are held constant. Also, since the H_2 molecule is bigger than atomic He, leakage is less of a problem. Since H_2 is stored on board as fuel, it is possible to release additional H_2 as needed should the lifting gas level be depleted due to leakage. H_2 is less expensive than He and is more readily available. Precautions would be necessary to prevent combustion of the lifting gas, but safe engineering solutions to this problem are possible. The final results of the Concept 15 design refinement are shown in Table 28 and Table 29 for both the hurricane science and communications relay missions. Once again, the fuel cell and electrolyzer ancillary masses are accounted for in the fuel cell stack row since the available data included the ancillary masses with the main fuel cell stack.

Table 29. Mass, Geometry and Performance Data for Phase II LTA Concept 15

Masses (kg)		Concept 15 
Item	**Mission**	Hybrid: PEM FC LH_2 + Solar Airship
Solar Cell Mass	Comm Relay	121
Hull: Fabric	Comm Relay	3010
Hull: Suspension	Comm Relay	50
Hull: Nose Reinforce	Comm Relay	100
Hull: Access/Maint	Comm Relay	50
Hull: Other	Comm Relay	0
Hull: Ballonet	Comm Relay	1204
Hull: Tail Structure+Fabric	Comm Relay	487
Hull Mass	Comm Relay	4901
Propulsion: Mech. System	Comm Relay	1631
Propulsion: Fuel + Tank (IC only)	Comm Relay	0
Propulsion Mass	Comm Relay	1631
Mission Payload	Comm Relay	212
Fuel Cell Stack	Comm Relay	3895
Fuel Cell Ancillaries	Comm Relay	0
H2 Fuel + Tank	Comm Relay	2446
O2 Oxydizer + Tank	Comm Relay	0
Water Tank	Comm Relay	0
Electrolyzer Stack	Comm Relay	0
Electrolyzer Ancillaries	Comm Relay	0
PMAD	Comm Relay	6
Batteries	Comm Relay	0
Energy System Mass	Comm Relay	6347
Total Mass (kg)	Comm Relay	13211
Solar Cell Mass	Hurricane	177
Hull: Fabric	Hurricane	4236
Hull: Suspension	Hurricane	50
Hull: Nose Reinforce	Hurricane	100
Hull: Access/Maint	Hurricane	50
Hull: Other	Hurricane	0
Hull: Ballonet	Hurricane	1695
Hull: Tail Structure+Fabric	Hurricane	686
Hull Mass	Hurricane	6817
Propulsion: Mech. System	Hurricane	653
Propulsion: Fuel + Tank (IC only)	Hurricane	0
Propulsion Mass	Hurricane	653
Mission Payload	Hurricane	362
Fuel Cell Stack	Hurricane	1542
Fuel Cell Ancillaries	Hurricane	0
H2 Fuel + Tank	Hurricane	4127
O2 Oxydizer + Tank	Hurricane	0
Water Tank	Hurricane	0
Electrolyzer Stack	Hurricane	0
Electrolyzer Ancillaries	Hurricane	0
PMAD	Hurricane	6
Batteries	Hurricane	0
Energy System Mass	Hurricane	5675
Total Mass (kg)	Hurricane	13684

		Concept 15 
Item	**Mission**	Hybrid: PEM FC LH_2 + Solar Airship
Geometry		
Design Lifting Gas Volume, m^3	Comm Relay	116393
Hull Envelope Area, m^2	Comm Relay	14201
Maximum Skin Thickness, mm	Comm Relay	0.1270
Minimum Solar Cell Area, m^2	Comm Relay	618
Vehicle Length, m	Comm Relay	139.6
Vehicle Width, m	Comm Relay	39.9
Vehicle Height, m	Comm Relay	39.9
Propulsion		
Drag @ Max Power Req, N	Comm Relay	11464
C_D	Comm Relay	0.02555
Total Power Required, kW	Comm Relay	638.7
Number of Fuel Cells or Batteries	Comm Relay	117
Number of Electrolyzers	Comm Relay	0
Max Continuous Discharge Time (hrs)	Comm Relay	0
Max Continuous Recharge Time (hrs)	Comm Relay	0
Propulsion Mass Factor, kg/kW	Comm Relay	2.533
Engine SFC, kg/kW/h	Comm Relay	0
Mission		
Start Date	Comm Relay	1-May
End Date	Comm Relay	28-Oct
Max. Mission Wind Speed, m/s	Comm Relay	71.6
Max.Design Speed, m/s	Comm Relay	56
Geometry		
Design Lifting Gas Volume, m^3	Hurricane	194381
Hull Envelope Area, m^2	Hurricane	19989
Maximum Skin Thickness, mm	Hurricane	0.1270
Minimum Solar Cell Area, m^2	Hurricane	905
Vehicle Length, m	Hurricane	165.7
Vehicle Width, m	Hurricane	47.3
Vehicle Height, m	Hurricane	47.3
Propulsion		
Drag @ Max Power Req, N	Hurricane	6076
C_D	Hurricane	0.02826
Total Power Required, kW	Hurricane	252.9
Number of Fuel Cells or Batteries	Hurricane	46
Number of Electrolyzers	Hurricane	0
Max Continuous Discharge Time (hrs)	Hurricane	0
Max Continuous Recharge Time (hrs)	Hurricane	0
Propulsion Mass Factor, kg/kW	Hurricane	2.533
Engine SFC, kg/kW/h	Hurricane	0
Mission		
Start Date	Hurricane	1-Jun
End Date	Hurricane	30-Nov
Max. Mission Wind Speed, m/s	Hurricane	42.4
Max.Design Speed, m/s	Hurricane	42

5.3 Operational Modeling Assumptions and Results

The objective of the operational modeling task was to calculate the required fleet size for each of the Phase II concepts to meet both the hurricane science and communications relay missions over a twenty-year operational period. Estimates of required fleet size were needed to support the production and operations and support (O&S) elements of the life cycle cost (LCC) analysis. The overall mission requirement was to provide a single station of continuous coverage during the six months of the hurricane season while simultaneously providing continuous, year-round coverage of one station supporting communications relay. Four different concepts of operations were considered: single vehicle, serial flight, multi-vehicle, and air-to-air refueling. Single vehicle assumes that one vehicle has the endurance necessary to complete the mission without refueling. Serial flight assumes that the vehicle is relieved on station by another vehicle when low fuel levels require a return to base. In this manner the entire mission can be covered without interruption utilizing several vehicles rotating as needed. Multi-vehicle assumes combining several different types of vehicles to cover the missions. For example, the LTA vehicle may be more suitable for the hurricane science mission and a HTA vehicle might be better for the communications relay mission. Air-to-air refueling is an alternative to serial flight that enables continuous station coverage but requires a capability to refuel in flight.

The multi-vehicle and air-to-air refueling options were not investigated in depth due to the negative impact these operational concepts would have on the total life cycle cost. The multi-vehicle option would necessitate two separate development, production, and operations efforts, greatly increasing overall cost to obtain a small performance benefit. The air-to-air refueling option would require the development of a re-fueling vehicle (tanker aircraft) and associated re-fueling system, increasing the development cost and risk. The large distances from the operating bases to the operational locations would require the tanker aircraft to have a long range capability, and large amounts of fuel would be consumed by the tanker aircraft transiting to and from the operational location. Alternatively, a separate operating base for the tanker aircraft could be established. Establishing, operating, and maintaining an additional operating base would greatly add to the total system cost.

For single vehicle or serial flight concepts of operation, the mission timeline begins with the first vehicle taking off and transiting to the area of interest to begin the loiter segment. When the fuel level requires a return to base (RTB), the first vehicle transits back and is recovered. For the single vehicle scenario, the total time on station requirement is met by the first vehicle and no additional flight operations are required. If the loiter endurance is less than required, a second vehicle would be launched in time to arrive at the station prior to the departure of the first vehicle (serial flight). The first vehicle returns to base and post-flight and pre-flight processing are performed. The first vehicle is launched again in time to relieve the second vehicle on station prior to its return to base. The cycle then continues for the duration of the mission period. If the on-station loiter endurance time is greater than the sum of the return to base transit time, the post-flight and pre-flight processing time ("turn around time"), and the out-going transit time, then this cycle can be supported with only two vehicles. A third vehicle is required if the loiter time is less than the sum of the transit times and turn around time. Figure 26 depicts this serial flight cyclical mission timeline for two vehicles, Air Vehicle 1 (AV#1) and Air Vehicle 2 (AV#2).

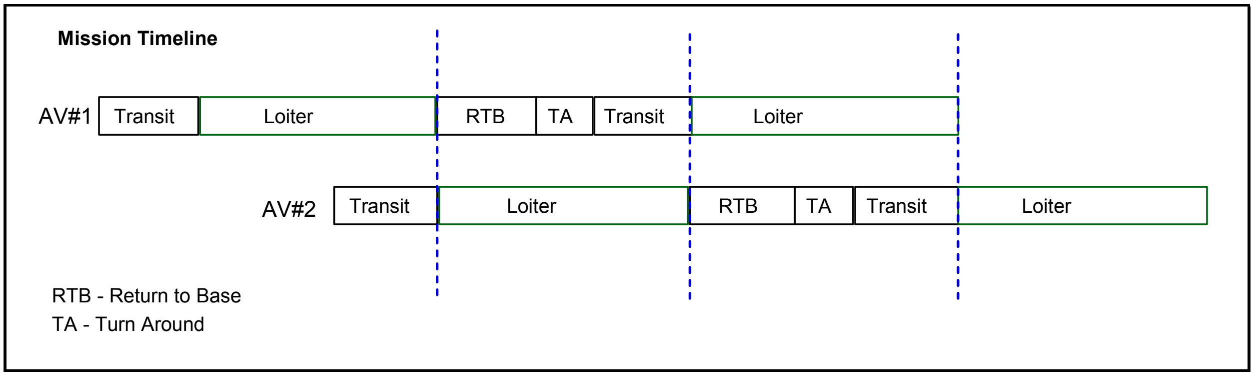
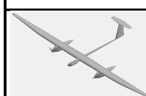
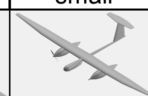
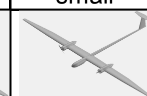
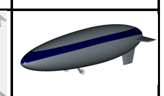


Figure 26. Two vehicle cycle to support a single station with continuous coverage.

A spreadsheet model was developed to facilitate the fleet size estimates. Inputs to this model include the annual coverage requirement (six months for the hurricane science mission and twelve months for the communications relay mission), the operational life (twenty years), the distance from the operating base to the loiter station, the average transit velocity, the total endurance of the vehicle, and the vehicle turn around time. In the case of the LTA communications relay vehicle, total endurance was assumed to be 180 days regardless of the mission start date. This simplifying assumption was required because the fidelity of the spreadsheet model was not adequate to simulate the dynamically varying mission endurance of the LTA configurations. From these inputs, the loiter time is calculated as the total endurance minus the time required for both transit legs. Total transit time (RTB+Transit) is then added to the turn around time (TA). If this value (RTB+TA+Transit) exceeds the loiter time three baseline vehicles are needed, otherwise two baseline vehicles are adequate. Based on the maintenance requirements assumed for the A-check and C-check intervals shown in Table 24, the number of A-checks and C-checks required are then calculated. If the time required for an A-check exceeds the difference between the loiter time and the RTB+TA+Transit time, an additional vehicle will be required since the A-check will prevent the vehicle from launching in time to relieve the on-station vehicle. A similar logic is applied to the C-check time. In addition to the vehicles needed to cover operational and maintenance requirements, vehicles will also be needed to cover attrition. Total flight hours are calculated and then divided by the assumed attrition interval (20,000 hours for HTA and 40,000 hours for LTA) to determine the number of vehicles required to replace attrited assets. The attrition interval would most likely be greater for the longer endurance vehicles due to the fewer number of launch and recovery cycles compared to the shorter duration vehicles; however, the same interval was assumed for all HTA vehicles due to lack of attrition data sufficient to quantify these assumptions. Finally, if "assured coverage" is specified the number of baseline vehicles is doubled. Assured coverage is required for the communications relay mission to reduce the occurrence of coverage gaps. An on-station spare vehicle is utilized to provide this assured coverage through dual redundancy. The final input is fuel consumed per sortie, enabling the model to calculate the total fuel consumed annually and over the twenty-year operational period.

The outputs from the operational concept model that were utilized as inputs to the cost model include total number of vehicles required, number of A-checks and C-checks required, total number of flight hours, and total fuel consumed. The totals for these parameters for both missions over the twenty-year operational period are presented in Table 30 for the Phase II concepts. The results show that the longer endurance vehicles result in smaller total fleet sizes; however, the relation between endurance and fleet size is not linear. That is, doubling the endurance does not correlate to a 50% reduction in the fleet size. Concept 1 has twice the endurance of Concept 1-small (8 days compared to 4 days), but only 20% more vehicles are required for the Concept 1-small fleet. Once the loiter time exceeds the transit times plus turn around time, the second air vehicle will sit on the ground ready to launch and the cost effectiveness of additional loiter time is reduced. Furthermore, the cost of additional loiter time is high (a larger, heavier vehicle). The 4-day endurance vehicles do increase the total flight hours required and therefore have increased maintenance and attrition costs. However, the mass results shown in Table 27 indicate that Concept 1-small is much lighter and therefore should be less costly to procure than Concept 1. The trade-off between procurement cost and operational cost will be presented in Section 6.0, Cost Analysis. Table 30 shows that Concept 15, the hybrid PEM Fuel Cell + Solar Airship, has the smallest fleet size, total fuel required, total flight hours, and total maintenance checks. These benefits result directly from the long endurance of this design. However, the size and mass of Concept 15 will negatively impact the procurement costs. In addition, the simplifying assumption that Concept 15's endurance for the communications relay mission does not vary with the time of year makes these results somewhat optimistic.

Table 30. Operational Modeling Results for Hurricane Science and Communications Relay Missions (20 Years of Operation)

	Concept 1	Concept 1-small	Concept 5-small	Concept 15
				
Propulsion/Fuel Type	IC Engine LH_2 Fuel	IC Engine LH_2 Fuel	IC Engine Diesel Fuel	Hybrid: LH_2 PEM FC + Solar Airship
Total # of vehicles	35	42	45	18
Total kg of fuel	3,364,000	2,557,000	5,868,000	270,300
Total flight hours	525,400	628,900	686,500	441,200
Total # of A-Checks	768	935	968	101
Total # of C-Checks	72	87	94	25

6.0 Cost Analysis

This section presents details of the rough order of magnitude (ROM) cost analysis conducted to compare the relative costs of the Phase II concepts. Section 6.1 describes the process and assumptions used to perform a full life cycle cost (LCC) analysis for the four Phase II concepts presented in the previous section. The LCC includes estimates for the development, production, and operational phases of a notional program. The results of the analysis are presented in Section 6.2.

6.1 Life Cycle Cost Analysis Process and Assumptions

The purpose of the cost analysis was to obtain ROM LCC estimates to compare concepts on a relative basis. Producing an accurate cost estimate for an individual concept on an absolute basis is beyond the scope of this effort. Estimates were made for minimum cost (Min), maximum cost (Max), and most likely cost. The magnitude of the range between Min and Max is indicative of the uncertainty of the cost estimate. The Max values should be used as conservative ROM cost estimates to support future planning efforts. When available, actual cost data from vendors and previous NASA programs were utilized.

The cost estimating process followed a relatively simple flow which started with the creation of basic programmatic assumptions. The program schedule was assumed to begin with a two-year phase of risk reduction efforts performed by two competing contractor teams. Following the risk reduction phase, a single contractor would be competitively selected for full scale development, lasting five years, followed by a production phase, the length of which is concept dependent. Finally, a twenty-year period of mission operations was assumed. Figure 27 depicts the top-level schedule assumed for the purposes of the LCC estimates. This example is for Concept 1 with a 35 unit production quantity. The LCC estimate was sub-divided along the traditional boundaries of Research, Development, Test and Evaluation (RDT&E), Production, and Operations and Support (O&S). Assuming a fiscal year 2007 (FY07) start, RDT&E spans seven years from FY07 through FY13. Production begins in FY14 and lasts through FY20 (this will vary among the concepts depending on the production quantity and assumed production rate), and the twenty-year O&S phase is from FY14 through FY34. The cost estimate was performed using constant year FY06 dollars.

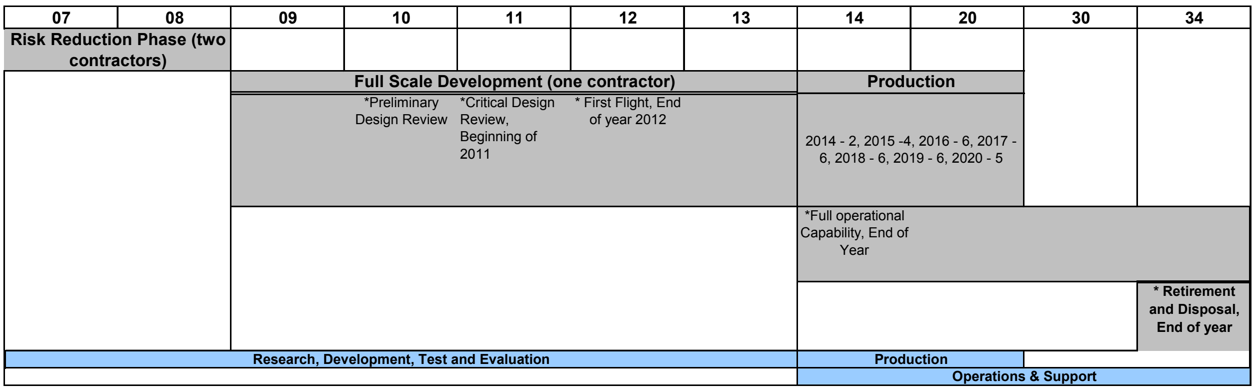


Figure 27. Schedule assumptions to support the life cycle cost estimate.

The labor rate assumptions were based on FY06 average rates. For example, a full time engineer was assumed to have a fully burdened rate of between \$230K (Min) and \$250K (Max) per Work Year (WY). Table 31 shows the assumptions made for the avionics costs. The values in Table 31 represent 1st unit recurring costs and do not include any development or testing costs, which are accounted for in the RDT&E estimate. Most of these values were based on representative equipment that is not identified here to protect proprietary pricing information. The large range between the Min (\$334K) and Max (\$8.3M) estimate is indicative of the cost uncertainty present in these early avionics estimates. The Min values assume the use of commercial-of-the-shelf (COTS) hardware with minimal modifications and maximum software re-use. The Max values represent use of unique hardware and software for this application. The most likely value of \$1.7M indicates the assumption that some unique hardware and software will be required, but an emphasis is placed on maximizing the use of COTS equipment.

Table 31. Avionics Cost Assumptions

	FY06 $K		
Avionics	**Min**	**Most Likely**	**Max**
Flight Control and Mission Management Computer	210	1,000	4,000
GPS Receiver	10	50	100
Line of Sight Communications	10	50	100
Over the Horizon Communications (low bandwidth)	10	250	2,000
Over the Horizon Communications (high bandwidth)	10	250	2,000
Transponder	2	5	20
Navigation lights	1	2	2
Servos	80	90	100
Backup Battery	1	1	1
Total	**334**	**1,698**	**8,323**

Table 32 shows the assumptions made for the fixed payload and the propulsion systems, both IC engine and PEM fuel cell. The communications relay payload cost was assumed to be the same as the fixed payload for the hurricane science mission. Also, the LH_2 IC engine cost was assumed to be the same as the diesel-fueled IC engine. These assumptions were necessary because available data were insufficient to discriminate between these items.

Table 32. Payload and Propulsion Cost Assumptions

	FY06 $K		
Payload (Hurricane Science Mission)	**Min**	**Most Likely**	**Max**
High Resolution EO/IR Sensor	500	750	1,000
Microwave Radiometer	500	750	1,000
Active Doppler Radar	200	500	700
Total	**1,200**	**2,000**	**2,700**
Propulsion System IC Engine			
Core Engine	12	15	20
Motor	5	6	7
Radiators	5	6	7
Turbochargers	8	9	10
Propeller	12	15	20
Total	**42**	**51**	**64**
Propulsion System PEM Fuel Cell			
Electric Motors (2)	54	60	60
PEM Fuel Cell + Turbochargers and Radiators	1,502	1,669	1,836
Propellers (2)	24	30	40
Total	**1,580**	**1,759**	**1,936**

Table 33 shows the cost assumptions made for the ground control station and the launch and recovery element. Again, the range between Min and Max is indicative of the amount of unique hardware required versus COTS.

Table 33. Ground Control Equipment Cost Assumptions

	FY06 $K		
	Min	**Most Likely**	**Max**
Ground Control Station	1,000	2,000	3,000
Launch and Recovery Element	500	1,000	1,500

With the exception of the servos, Tables 31 through 33 include the primary elements of the cost estimate that are fixed across the different concepts. The discriminators between the concepts are based in a large part on the mass differences, materials used, production quantities, fuel consumption, and maintenance requirements. The basic assumptions for the RDT&E phases of the program are listed in Table 34. (WY is a work year and represents one fully burdened year of labor.)

Table 34. Summary of RDT&E Cost Estimate Assumptions

2 Year Risk Reduction	5 Year Full Scale Development
20 Full time WY per contractor team	80 full time WY
1 PM and 1 Support WY per contractor team	5 PM and 5 Support WY
Wing Section Fab and Test (or fabric for LTA)	Wing/body for structural testing
Engine or PEM Fuel cell prototype	(2) Wind tunnel models
LH_2 tank	Full up propulsion system for ground testing
Software Integration Lab	Expanded software integration lab
	Iron bird for flight control integration and testing
	2 full up pre-production test articles
	1000 hours for flight testing and certification

The production cost estimates were based on the production quantities resulting from the operational concept study. The primary reference utilized to estimate recurring production costs was a RAND study published in 1991 for the United States Air Force (ref. 35). This reference provides cost estimates in terms of dollars per pound of structural material. The data were obtained from Boeing, General

Dynamics, Grumman, Lockheed, McDonnell Douglas, Northrop, Rockwell and LTV, and came from primarily military applications such as the AV-8B, B-1, F-14, F-15 and V-22. Civilian applications were included as well, such as the 737, DC-10 and L-1011. Although these configurations are different than HALE UAVs, the underlying cost drivers for structural materials are similar. Materials addressed include titanium, steel, aluminum, graphite/epoxy and copper wiring, among others. These recurring dollars per pound estimates include the cost of manufacturing labor (both fabrication and assembly labor), raw material, and support labor (sustaining engineering, tooling, and quality assurance). The data provided in reference 35 are in 1990 dollars for 100 units and 1000 lb of structure. For this study the cost estimates were converted into 1st unit estimates in 2006 dollars by backing up the learning curve and adding 1990 to 2006 inflation. An 85% learning curve (15% reduction in unit recurring cost every time the production quantity doubles) was assumed. Given the slope of the 85% learning curve and the value for 100 units, the value for any other production quantity can be determined. These conversions are illustrated in Table 35 using graphite/epoxy as an example. In the "Most Likely" column of Row 1 is the original estimate from the RAND report. Row 2 shows the result of backing up the 85% learning curve from the 100th unit to the 1st unit. Next, Row 3 shows the conversion from 1990 dollars to 2006 dollars by applying inflation. The values in Row 3 were then multiplied by the composite component masses to determine the recurring production costs of the composite structure. The same approach was taken for the other materials used in the structure. The avionics, payload, and propulsion cost estimates were added to the airframe cost estimate to obtain the 1st unit recurring cost. The 85% learning curve was then applied to the production quantity to determine total recurring production costs. In addition, the labor costs to support production varied depending on the production quantity and rate.

Table 35. Conversion of Graphite/Epoxy Estimate to 1st Unit, 2006 Dollars

	FOR 100 UNITS, 1000lbs of material	**Min**	**Most Likely**	**Max**
1	Graphite/Epoxy $/lb in FY'90 $ *	$822.00	$1,548.00	$2,133.00
	1st unit cost using Learning Curve Effects on $/lb, 1000lbs of material			
2	Graphite/Epoxy $/lb in FY'90 $	$2,420.00	$4,557.00	$6,279.00
	Apply inflationary factors to convert to FY06 $			
3	Graphite/Epoxy $/lb in FY'06 $	$3,814.00	$7,182.00	$9,896.00
	(Converted to $/kg)	**$8,408.00**	**$15,834.00**	**$21,817.00**

*As shown in Reference 35

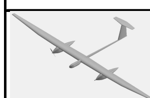
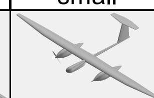
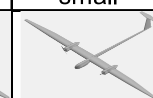
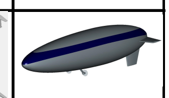
The O&S cost estimates cover twenty years of operations; including operating personnel, maintenance, consumables, and facility costs. For the HTA vehicles, eight ground personnel were assumed to support the launch and recovery operations, and four personnel were assumed to support the mission operations. For the LTA concept, twenty ground personnel were assumed to support the launch and recovery operations, and four personnel were assumed for the mission operations center. These assumptions include personnel for both the hurricane science and communications relay missions. Four weeks of operator training was assumed to occur at the beginning of the operational phase, and refresher training was assumed to occur once every three years. Operational facility costs were estimated assuming the rental of existing space over the twenty-year period of operations. Square footage requirements were estimated based on the vehicle spot factors, and the cost per square foot and utility costs were estimated based on empirical data. Iridium satellite communications support was assumed for the duration of the mission period, at a maximum cost of $100K/month for two channels. Maintenance costs were based on the required number of A-checks and C-checks from the operational concept model. Costs for equipment and spares were also estimated. The fuel costs were estimated by multiplying the total amount of fuel required (obtained from the operational concept model) by fuel cost per kilogram. The other major consumable elements are the dropsondes for the hurricane science mission. Recall from Section 5.1 that the number of dropsondes (336) was derived assuming a drop rate of one per hour over a 14-day mission.

Actual operations will vary widely from this assumption, depending on the frequency of storm activity and vehicle endurance capability. The HTA vehicles will be rotating on station every four to eight days and could carry a new load of dropsondes each sortie. However, many of these sorties will not require use of any dropsondes because of lack of storm activity. The LTA vehicle was also sized to carry 336 dropsondes, which cannot be replenished during the 180 day mission. Therefore, for costing purposes, a total of 336 dropsondes were assumed to be expended during each six-month hurricane season. Finally, maintenance training costs were estimated assuming an eight week training program once every three years.

6.2 Life Cycle Cost Analysis Results

The LCC analysis results are presented in Table 36. The values presented are the maximum values and therefore the most conservative, although at this early conceptual stage the cost uncertainty is high.

Table 36. Life Cycle Cost Analysis Results (Max values)

	Concept 1	Concept 1-small	Concept 5-small	Concept 15
				
Propulsion/Fuel Type	IC Engine LH_2 Fuel	IC Engine LH_2 Fuel	IC Engine Diesel Fuel	Hybrid: LH_2 PEM FC + Solar Airship
Total # of vehicles	35	42	45	18
Max RDT&E (FY06 CY $M)	337	251	255	336
Max Production (FY06 CY $M)	1226	913	970	750
Max O&S (FY06 CY $M)	196	206	200	180
Max Total LCC (FY06 CY $M)	**1759**	**1370**	**1425**	**1266**
Max Operations ($/flight hour)	373	328	291	408
Max Average Unit Flyaway Cost (FY06 CY $M)	35.0	21.7	21.6	41.7

Concept 1-small and Concept 5-small had the lowest estimated RDT&E costs, which correlates directly with their lower mass estimates. The estimates for total production costs are a function of the production quantities, production schedules, and concept mass estimates. The Concept 15 production quantity is only 18 vehicles resulting in the lowest total production cost, even though the average unit fly-away cost is the highest. The O&S cost estimates are similar for all four concepts. However, Concept 15 has a slight advantage in this category due to its minimal fuel consumption and fewer required maintenance actions. This lower O&S cost, combined with the lower production cost, results in Concept 15 having the lowest overall estimated LCC. The HTA vehicle with the lowest estimated LCC is Concept 1-small, followed closely by the diesel-fueled Concept 5-small. The estimated LCC of Concept 1 is significantly greater, proving that maximizing endurance for the HTA vehicles does not result in the most cost effective system given the mission and operational assumptions made to support this study. Another interesting result is the Max Operations ($/flight hour) metric, which shows the diesel-fueled Concept 5-small to be the least expensive to operate at $291/flight hour. This is due mainly to the relatively low cost of diesel fuel compared to LH_2.

7.0 Solar Regenerative Mission Requirements and Technology Study

In Phase I of the study, HTA SR concepts were analyzed for two sets of mission requirements which were deemed useful for communications relay and hurricane science. None of the concepts evaluated were able to perform either of the two missions. The concept and mission combination closest to feasibility was Concept 10 executing the communications relay mission. In that case, the SR propulsion system was able to provide 40% of the power required (where 100% would indicate that the mission is feasible). For Phase II, the mission requirements were revisited and new "threshold" missions were defined for both mission types (see Section 5.1). These less demanding missions still could not be accomplished by an SR configuration. Although these results eliminate consideration of an SR concept for the study missions in the target timeframe, they do not provide any insight into what missions could be accomplished with these vehicles or what technology advances are required to make the study missions feasible. Additional analysis was therefore conducted to determine: a) what mission requirements can be met by a HTA SR concept using baseline technology assumptions, and b) what technology advances are required to achieve feasibility of the hurricane science and communications relay missions.

7.1 Study Configuration

Concept 7 was the preferred HTA SR concept following the down select conducted at the end of Phase I. Although a number of unconventional configurations were included in the AoA, the analysis indicated that the more conventional Helios-type configuration performed as well or better in most cases. The only exception to this was the high latitude performance of the variable geometry, multi-surface configuration (Concept 10). However, the slight benefit in capability for Concept 10 was outweighed by the greater risk and uncertainty inherent in an unconventional design. The secondary battery energy storage system was found to offer better capability than a regenerative fuel cell system for the baseline technology assumptions.

Following the down select to Concept 7 some slight modifications were made to the configuration geometry and analysis approach to make the analysis more realistic. The revised geometry is shown in Figure 28. Two landing gear pylons were added that had not been explicitly included in the previous geometry. The outboard landing gear pylons are 50 m apart as in the Helios design. Another modification to the geometry was the addition of 4° of dihedral to the outer wing. This dihedral was added to avoid striking the wingtip or outboard propellers on the ground during ground operations. Although specified in the initial requirements, minimum dash speed requirements (based on expected winds aloft) were not explicitly addressed for the SR concept sizing and analysis in Phase I. The analysis model for Concept 7 was later modified to permit consideration of dash speed requirements in configuration sizing. Note that this requirement was only used in sizing the airframe structure and propulsion system maximum thrust. The mission analysis was not performed with the aircraft flying at the dash speed, but rather at the optimum loiter speed. In the case of sustained high winds, the aircraft could need to fly at the dash speed for a full 24-hour solar cycle to maintain station keeping and the SR system would need to provide the higher power level required for this higher flight speed. For the hurricane science threshold mission, the required dash speed is approximately the same as the loiter speed (2% more) so the impact on the analysis results of flying at the dash speed for extended periods of time would be small. However, for the communications relay mission, the dash speed is almost twice the loiter speed.

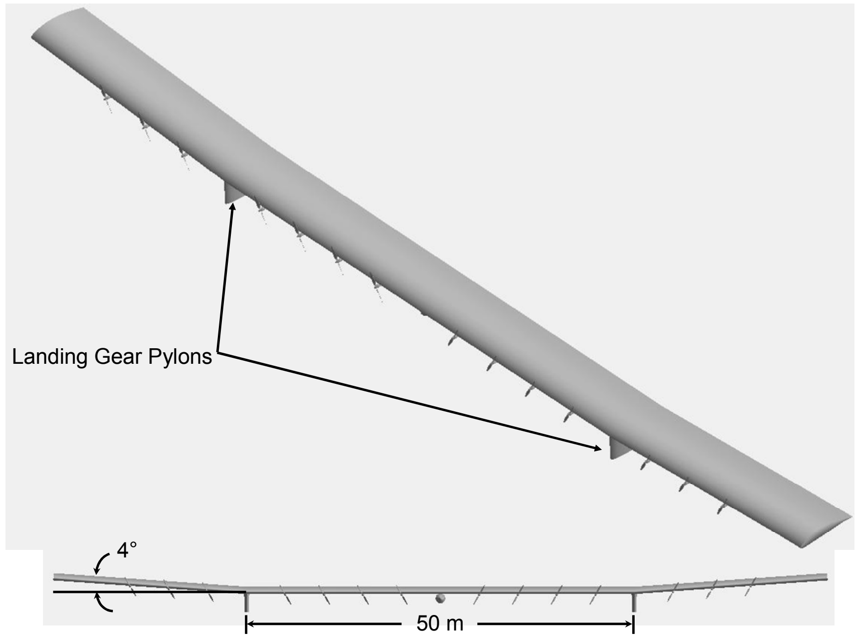
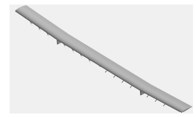
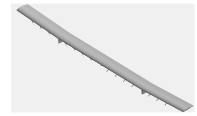


Figure 28. Refined geometry for Concept 7.

Changing to the relaxed, "threshold" Phase II requirements for the hurricane science mission (see Table 24) had a significant positive impact on feasibility for Concept 7, increasing $\%P_{regen}$ from 35% to 52%. Unfortunately, this is still well short of being able to accomplish the mission. Despite the less demanding Phase II threshold mission requirements, for the communications relay mission $\%P_{regen}$ decreased from 32% to 25% due to the revisions made to the concept geometry and analysis approach. Addressing the dash speed requirement of 200 km/h in the sizing had a large negative impact on the feasibility for this mission. Requiring the vehicle to fly a full diurnal cycle at the dash speed instead of at optimum loiter speed would make the communications relay mission even less feasible than 25%. Mass data for the Phase II Concept 7 configurations (hurricane science and communications relay) are shown in Table 37.

Table 37. Mass Data for Phase II HTA SR Concept

	Concept 7	Concept 7
		
Component Masses (kg)	Hurricane Science Threshold Mission	Communications Relay Threshold Mission
STRUCTURES:		
Wing	740	1127
Landing Gear	32	33
Center Pod/Fuselage	47	49
Propulsion Pods	37	56
TOTAL STRUCTURE:	856	1265
PROPULSION:		
Solar Array	269	299
Propulsion Mounts	69	52
Secondary Batteries	967	542
Motors	135	256
Propellers	59	75
TOTAL PROPULSION:	1499	1224
EQUIPMENT:		
Servos	27	28
Wire/Electrical	99	99
Avionics	15	15
Backup Battery	12	12
TOTAL EQUIPMENT:	153	154
(STRUC + PROP + EQUIP) EMPTY MASS:	2508	2643
Payload	200	136
Zero Fuel Mass	2708	2779
Fuel	0	0
Takeoff Gross Mass	**2708**	**2779**

7.2 Mission Requirements Trade Study

Since HTA SR vehicles have been designed for multi-day flight in the past with current technology (such as the original AeroVironment Helios design), the existence of feasible combinations of mission requirements was expected. The HTA SR mission requirements trade study was conducted to provide more insight into feasibility across a broad range of mission requirements. Specifically, objectives of the study were to: determine the mission capabilities of a baseline, near-term technology HTA SR vehicle; evaluate the sensitivity of feasibility to mission requirements; and explore potential trade-offs among mission requirements

7.2.1 Study Approach

Because of the number of parameters defining the mission, and the ranges of interest for those parameters, performing an exploration of the mission trade space with a full analysis at each point of interest was felt to be computationally prohibitive. Analysis of the Concept 7 model for a new set of mission requirements takes 1.5-2 minutes, including sizing of the motors, propellers, and energy storage system. Although this is relatively quick for a single point, it is too long to permit extensive exploration of the mission requirements trade space. For example, performing a trade between two mission variables (e.g., latitude and day of year) with 10 values each would require 100 analysis points, taking perhaps three hours to complete. This analysis time is with design parameters such as wingspan and wing area held fixed. With widely varying mission requirements, the optimum wing geometry is likely to change. Finding the optimum wing geometry for each combination of mission parameters in the above example would multiply the number of runs and associated execution time by several times or more.

To facilitate execution of the study within a reasonable amount of analysis time, a "meta-model" of the Concept 7 analysis model was developed using response surface methodology. Developing a response surface approximation requires conducting a "design of experiments" (DOE) in which numerous test cases (with varying inputs) are defined, performing the full analysis for each case in the DOE, and performing a multivariate regression to fit a response surface equation (RSE) to the output data. Although this represents a significant up-front investment, once the response surface equation is determined an approximate result for a given case can be obtained in a fraction of a second. Since the performance capabilities of an SR configuration result from a complex interaction among a multitude of variables, care must be exercised in developing the response surface equation to ensure a reasonably accurate approximation.

Table 38. Mission Requirements Study Parameters and Ranges of Interest

	Initial Parameter Set	**Final Parameter Set**
Mission Requirements		
Latitude	0° to 50° North	0° to 50° North
Day of Year	1 to 365	1 to 365
Payload Mass	0 to 500 kg	0 to 200 kg
Payload Power	0 to 4 kW	0 to 4 kW
Loiter Altitude	15 to 21 km	15 to 18 km
Minimum Dash Speed	25 to 65 m/s	25 to 45 m/s
Loiter Altitude Rate-of-Climb	0.13 to 0.51 m/s (25 to 100 ft/min)	fixed at 0.51 m/s
Design Variables		
Wing Area	100 to 700 m^2	determined by aspect ratio
Wingspan	50 to 120 m	fixed at 100 m
Wing Aspect Ratio	determined by area and span	14 to 25
Array Coverage	20 to 80% of wing area	fixed at 80%
Loiter C_L	0.5 to 1.2	optimized during analysis

Table 38 lists the parameters included in the mission requirements trade study. Since the accuracy of a response surface approximation increases as the size of the trade space considered is decreased, effort was made to minimize the number of parameters considered. The initial parameter set included 7 mission variables and 4 design variables, only a subset of the possible variables that could be considered. Even so, attempts to develop a sufficiently accurate response surface approximation for the initial 11 variables were unsuccessful. To improve the results of the response surface approach, the variables considered were reduced to only those of most importance and the ranges to those of most interest. In the mission requirements category, the upper bounds considered for payload mass, loiter altitude, and minimum dash speed were lowered to focus the response surface approximation towards areas of the trade space with $\%P_{regen} \geq 100\%$. Rate-of-climb was dropped from the parameter list and held fixed at 0.51 m/s (100 ft/min) as assumed in the Phase I mission analysis. Whether or not this requirement impacts the design depends on the dash speed requirement, which can be the determining factor in propulsion system size rather than rate-of-climb. Loiter C_L was removed from the design variable list by modifying the analysis model to loiter at optimum C_L for any given design. Array coverage was initially included as a design variable because it was found that the optimum array size is not always the maximum possible depending on solar cell efficiency and mass. However, the cases for which this is true are rare. Moreover, when designs are pushed to the limits of feasibility as in this study, maximizing energy collection is critical and optimum array size tends to be the maximum possible. Array coverage was therefore dropped from the parameter list and held fixed at the assumed maximum of ~80% of wing area. Similarly, based on prior results, it was expected that the wingspan which maximizes the mission capabilities would be the maximum span allowed. Wingspan was fixed at 100 m; the same maximum span constraint used in the Phase I analysis. The remaining seven parameters, six mission requirements and one design variable, were carried forward. The wing area design variable was replaced with wing aspect ratio simply because of the way the analysis model was structured. Since wingspan was fixed, wing area and aspect ratio were directly related variables.

Given the overall accuracy desired for the study, the goal for the response surface development was an approximation with a nominal $\%P_{regen}$ accuracy of ±5 percentage points. A DOE method with 500 random cases (that is, a set of test cases for which variable inputs are set at random values) was sufficient to achieve this level of accuracy. A slight improvement in the RSE accuracy was obtained by removing cases with $\%P_{regen} < 50\%$ from the dataset (~40 points). A plot of the residual for the selected third-order polynomial RSE approximation is shown in Figure 29. The error in the P_{regen} fraction is generally less than ±0.05 (i.e., predicted $\%P_{regen}$ is within ±5 percentage points of the actual).

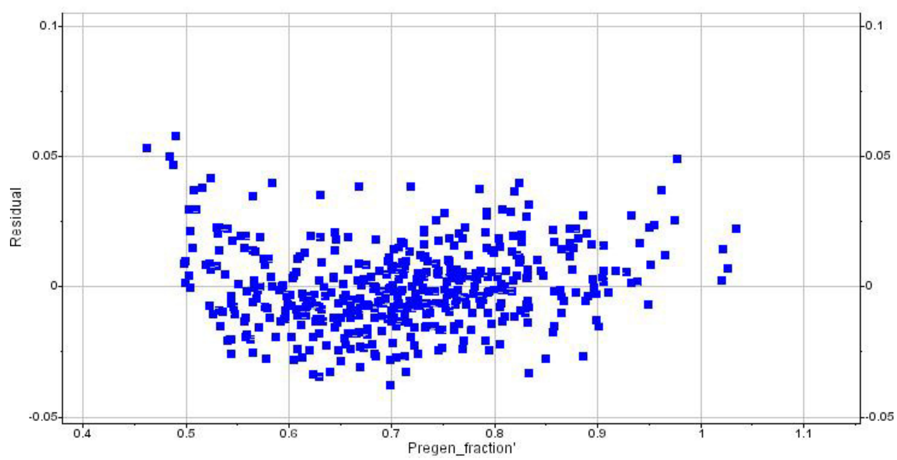


Figure 29. Response surface approximation residual for 460 point random dataset.

Another check of the RSE accuracy is the predicted $\%P_{regen}$ for the threshold missions. The RSE result for the hurricane science threshold mission is 53% compared to 52% for the full analysis. Since the communications relay threshold mission has a dash speed requirement beyond the range used in the DOE, it is not possible to use the RSE for that mission. However, with dash speed reduced to 45 m/s the RSE predicts a $\%P_{regen}$ of 29% compared to 30% for the full analysis.

7.2.2 Study Results

Given that only a few of the 500 random cases analyzed for the DOE resulted in feasible missions, the trade space of feasible missions was expected to be small. This was indeed the case as will be shown in the results below. For the exploration of the mission trade space, wing aspect ratio (or equivalently wing area) was optimized at each analysis point. Each point, therefore, represents a vehicle specifically designed for those mission requirements, not the "off-design" performance of a fixed vehicle design.

Impact of Payload Mass on Feasibility

A series of latitude and day of year "feasibility contours" for various payload masses is plotted in Figure 30. The feasibility contours are constructed from the set of latitude and day of year combinations for which $\%P_{regen}$=100% (the SR system is able to provide 100% of power required to fly the vehicle based on a 24-hour energy balance). The operational envelope for which feasible missions are possible encompasses the area "inside" these contours. The contours in Figure 30 were determined with the least stringent values for the other mission requirement areas; a loiter altitude of 15 km, a payload power requirement of 0 kW, and a minimum dash speed of 25 m/s (essentially no dash speed requirement).

Year round capability is not possible for any payload mass. Even with no payload, an energy balance is possible only for mission days from about March to September. A payload of 200 kg can be accommodated only near the summer solstice at high latitude (>43°). (The feasibility contours would be expected to peak at the most favorable solar day (June 21)). The slight skewing of the contours away from this shape is likely due to the response surface approximation.)

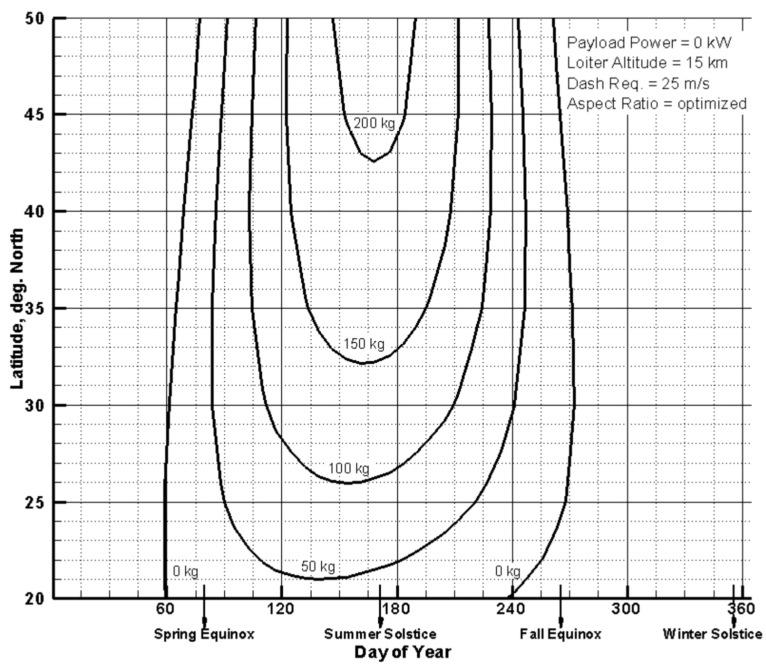


Figure 30. Latitude and Day of Year feasibility for various payload mass requirements.

Impact of Loiter Altitude on Feasibility

The impact of the loiter altitude requirements on latitude and day of year capability can be examined in Figure 31. These feasibility contours were determined assuming no payload and a minimum dash speed of 25 m/s. Increasing the loiter altitude requirement from 15 km to 16 km increases the minimum feasible latitude to 25° and reduces the range of feasible mission days by 2-3 months at higher latitudes.

The maximum possible loiter altitude (most favorable solar day) with no payload mass or power, and a minimum dash speed requirement of only 25 m/s, is 16.8 km.

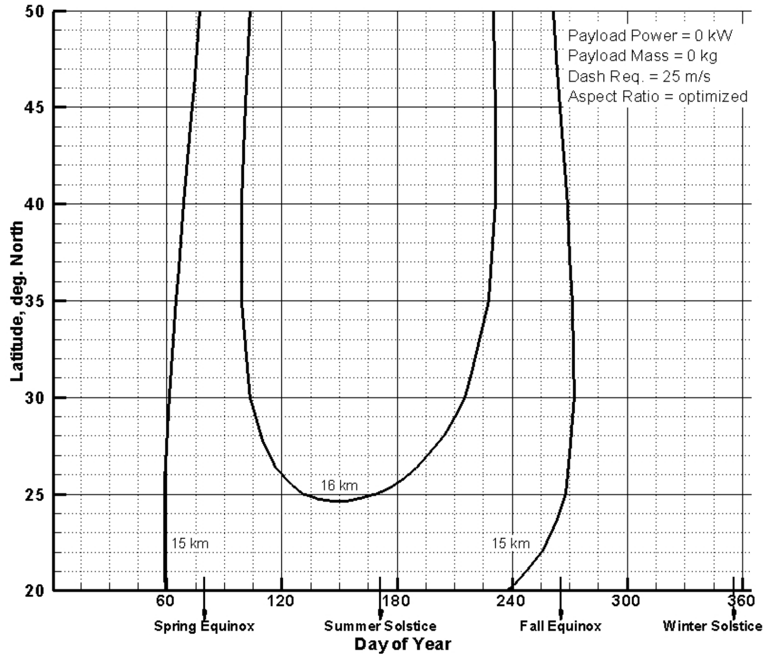


Figure 31. Latitude and Day of Year feasibility for various loiter altitude requirements.

Impact of Minimum Dash Speed on Feasibility

Sensitivity of latitude and day of year capability to the dash speed requirement is shown in Figure 32.

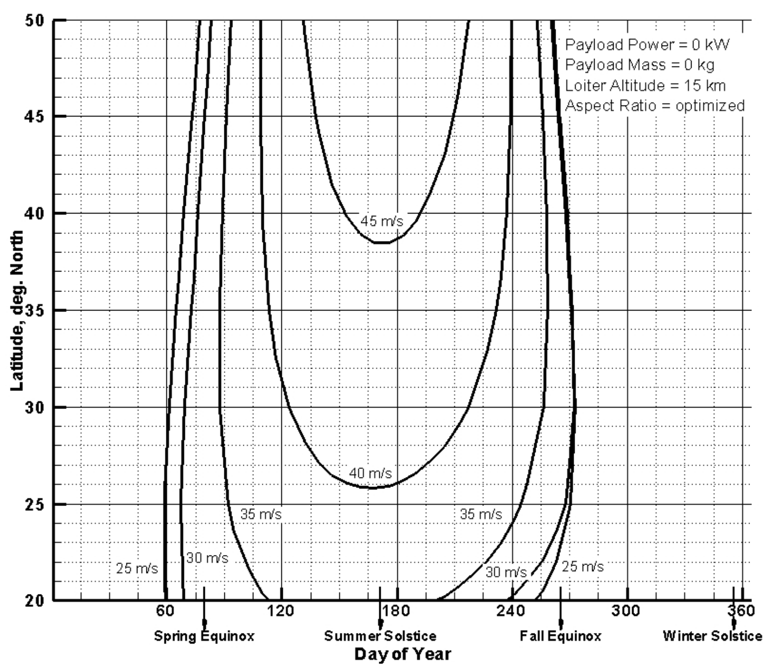


Figure 32. Latitude and Day of Year feasibility for various dash speed requirements.

The dash speed requirement generally evolves from the expected winds aloft and the need for station keeping against those winds. Increasing the required dash speed capability has a significant negative impact on the other capabilities that can be achieved. There is little difference between a requirement of 25 m/s and 30 m/s due to the fact that these have little to no impact on the design (about the same or less than loiter speed). Requiring a dash capability of 45 m/s restricts feasibility to high latitude around the summer solstice, even when flying at 15 km with no payload. Note that for these contours the dash speed requirement sets the maximum speed capability of the vehicle, but does not influence the mission analysis. The SR propulsion system energy balance does not include any flight time above the nominal loiter speed. Sizing the energy storage system to account for time spent flying at these elevated speeds would further degrade the latitude and day of year capabilities.

Altitude, Payload Mass, and Payload Power Trades

In Figure 33 the feasibility of various altitude requirements are plotted against payload mass and power rather than latitude and day of year. The remaining requirements are set at their least tasking values (minimum dash speed 25 m/s, most favorable solar conditions of 50° latitude on June 20). A trade-off between payload power, payload mass, and altitude capability is clearly evident from this plot. Increasing altitude 1 km reduces payload mass capability by roughly 100 kg or payload power capability by close to 2 kW.

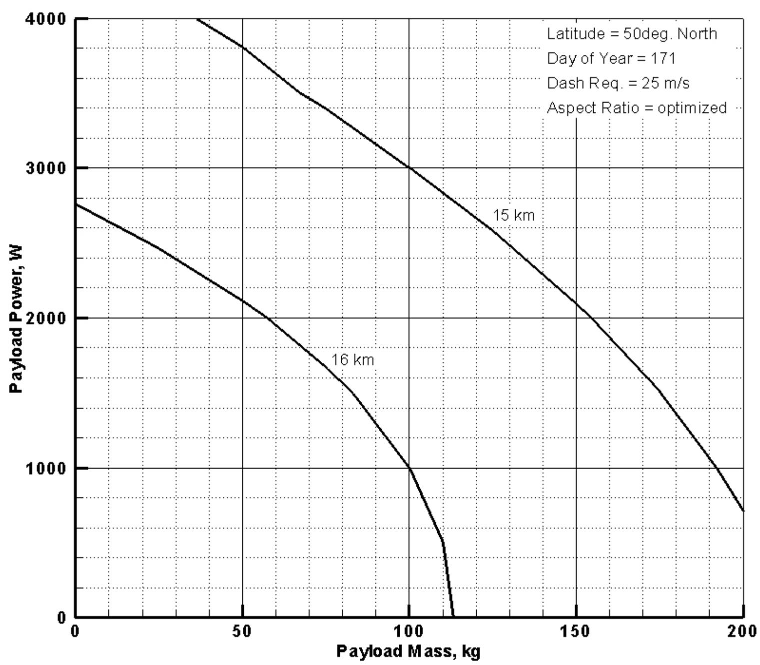


Figure 33. Payload Mass and Payload Power feasibility for various altitude requirements.

Example Feasible Mission

The mission requirements trade study indicates that the capabilities of an SR configuration with the baseline technology assumptions are very limited. Fortunately, however, they are not non-existent. Although the hurricane science and communications relay reference missions cannot be realized, there are capabilities available which could be useful for other applications. One example of a feasible mission is shown in Table 39.

Table 39. Possible SR Concept Feasible Mission

Loiter Altitude	15 km
Payload Mass	50 kg
Payload Power	0.5 kW
Minimum Dash Speed	35 m/s
Latitude	>30° North
Mission Duration	May 1 to July 31

Optimum wing aspect ratio for this mission (as determined using the RSE approximation) is 22.5, corresponding to a wing area of ~450 m^2. Such a vehicle would be able to operate in most of the continental United States during the summer months. There may be a number of scientific uses for this vehicle such as monitoring coastlines, in situ atmospheric sampling, wildfire detection, etc.

Requirement Trade-Offs

One way to assess sensitivity to mission requirements is to determine what changes in requirements bring about a certain change in feasibility ($\%P_{regen}$). Some of this information can be inferred from examining Figures 30 through 33. A more specific assessment of these sensitivities was also conducted by establishing a baseline case similar to the mission in Table 39 and individually varying the mission parameters to create a 10% decrease in feasibility (feasibility was decreased rather than increased because effecting a 10% increase in feasibility was not possible for all parameters). Table 40 contains the approximate equivalent mission requirement changes. Note that sensitivity of feasibility to day of year depends on the latitude and conversely sensitivity to latitude depends on the day of year examined. This table can be used to assess potential trade-offs between requirements or capabilities. For example, Table 40 indicates that minimum dash speed could be increased by 15 m/s if the altitude requirement was relaxed by 1 km.

Table 40. Equivalent Mission Requirement Changes from a Feasibility Perspective

Requirement Area	**Baseline**	**Δ for 10% reduction in feasibility**
Loiter Altitude	15 km	+1 km
Payload Mass	50 kg	+100 kg
Payload Power	0.5 kW	+2 kW
Minimum Dash Speed	25 m/s	+15 m/s
Latitude (at June 20)	30° N	10° expansion South
Day of Year Envelope (at 30°N)	June 20	150 day symmetric expansion around June 20

7.2.3 Mission Requirements Trade Study Conclusions

Given near-term technology assumptions and projections, it will not be possible for an HTA SR configuration to perform either the hurricane science or communications relay mission. In fact, mission capabilities are far from those required for the two missions. Utility in a communications relay application is severely hindered by the fact that year round capability is not possible at any latitude. Only missions which take advantage of the long days and short nights of summer to relax the demands placed on current energy storage system technology are feasible. Even at favorable solar conditions, payload mass and power have to be kept to a minimum to achieve feasibility. Despite latitude, time of year, and payload limitations, there may still be useful missions which could be accomplished with near-term SR concepts. A number of important scientific measurements can be obtained with very lightweight, low power payloads. And, there are likely scientific investigations for which the required mission timing and location match well with the vehicle capabilities. Although a notional set of feasible mission requirements has been defined, as with any aircraft there are trades which can be made among the various requirements.

7.3 Technology Trade Study

The SR technology trade study was conducted to provide insight to the sensitivity of mission feasibility to technology advancements in various areas and to determine the level of technology advancement required to make the hurricane science and communications relay threshold missions feasible. More specifically, the study objectives were: determine technology advances which will enable the threshold missions; evaluate the sensitivity of mission feasibility to technology assumptions; and identify technology areas which are most important to mission feasibility

7.3.1 Study Approach

The concept described in Section 7.1 was used as the baseline for the technology trade study. Based on the analysis of Phase I, the preferred concept utilized a secondary battery energy storage system. Whether a secondary battery approach will continue to offer an advantage over regenerative fuel cells as technologies for both systems advance is not known. Rather than investigate these two types of systems separately in the technology study, the energy storage system characteristics were defined in a more generic sense by the effective specific energy (W-h/kg) and roundtrip efficiency. The specific type of system which provides the assumed characteristics is not explicitly modeled in the analysis.

As with the mission requirements study, the number of parameters of interest for the technology trade study was too large to permit a full analysis at each point examined in the trade space. Response surface methodology was therefore used for this study as well. A random DOE was again found to lead to a sufficiently accurate response surface approximation. Subsets of 1000 random cases, excluding extremely high or low $\%P_{regen}$ cases, were used in multiple response surface equation regressions in an effort to obtain the best approximation possible. Also, the ranges of the input parameters were tailored to each of the missions to increase the number of random cases in the area of most interest (i.e., around $\%P_{regen}$=100%).

7.3.2 Hurricane Science Mission

The input parameters and ranges used for the hurricane science mission technology trade study are shown in Table 41. Note that the airframe drag and mass "tech factors" are simply multipliers which are applied to the values predicted in the analysis model. For example, a mass tech factor of 0.9 implies a technology has been applied which reduces that total airframe mass by 10% (for the same design gross mass, etc.).

Table 41. Hurricane Science Mission Trade Study Parameters and Ranges

Technology Variables	
Solar Cell Reference Efficiency	0.10 to 0.75
Solar Array Mass	0 to 1.5 (kg/m^2)
Energy Storage System Roundtrip Efficiency	0.3 to 1.0
Energy Storage System Specific Energy	100 to 1000 (W-h/kg)
Airframe Mass Tech Factor	0.75 to 1.5
Airframe Drag Tech Factor	0.75 to 1.5
Design Variables	
Wing Aspect Ratio	10 to 35

A plot of the residual for the final third-order polynomial RSE approximation is shown in Figure 34. Error in the P_{regen} power fraction is generally less than ±0.05 (i.e., predicted $\%P_{regen}$ is within ±5 percentage points of the actual). There are a few points with much higher error; however, these cases are outside the area of primary interest. Using the RSE approximation, the predicted $\%P_{regen}$ for the baseline technology case is 48% compared to the 52% obtained from the full analysis.

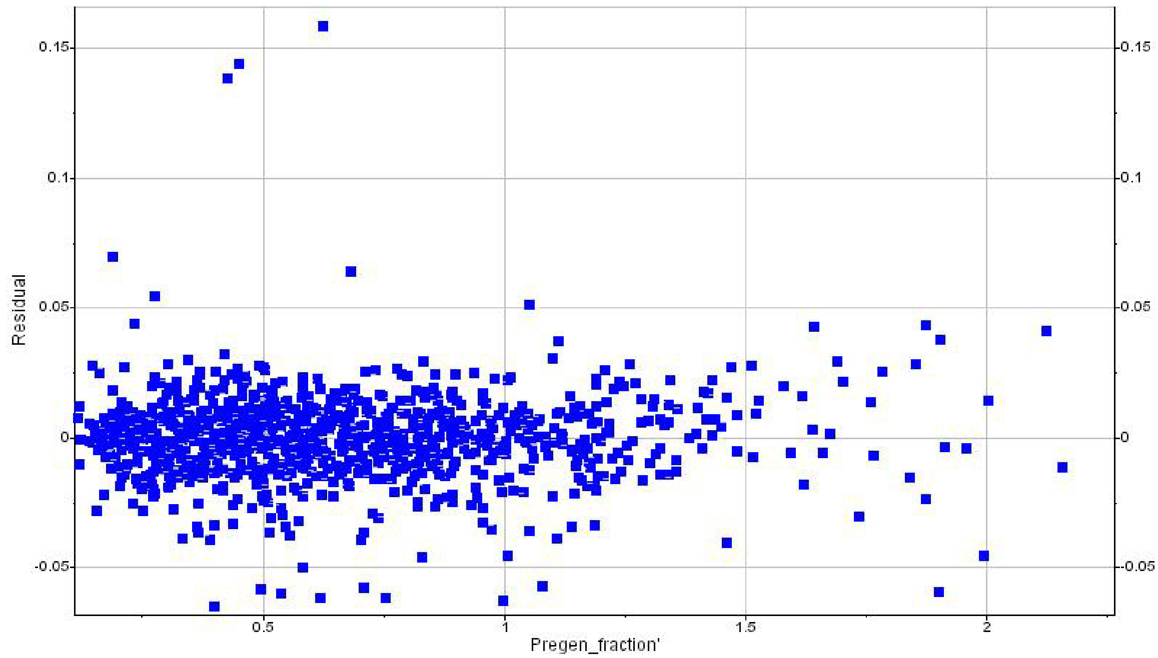


Figure 34. Hurricane science mission response surface approximation residual.

Hurricane Science Mission Technology Sensitivities

One way to assess the sensitivity of feasibility ($\%P_{regen}$) to advanced technology is to examine the amount of technology improvement required to increase feasibility by a certain amount. Table 42 summarizes the amount of technology improvement for each area which will individually increase feasibility by 1% (i.e., $\%P_{regen}$ increases from 48% to 49%). The wing aspect ratio (equivalently wing area) was optimized for each point.

Table 42. Sensitivity of Hurricane Science Mission Feasibility to Technology Improvements

Technology Area	**Baseline Value**	**Δ required to increase $\%P_{regen}$ by 1 point**
Solar Cell Reference Efficiency	20%	+1.2 pts (+6%)
Solar Array Mass	0.67 kg/m^2	-0.07 kg/m^2 (-11%)
Energy Storage System Roundtrip Efficiency	82%	+12 pts (+15%)
Energy Storage System Specific Energy	252 W-h/kg	+8 W-h/kg (+3%)
Airframe Mass Tech Factor	1.0	-0.03 (-3%)
Airframe Drag Tech Factor	1.0	-0.02 (-2%)

For the baseline technology assumptions (with secondary battery energy storage), feasibility is most sensitive to increasing ESS specific energy, reducing airframe mass, and reducing airframe drag. Feasibility is least sensitive to reductions in solar cell mass and increases in energy storage system roundtrip efficiency. Given that a 50 percentage point increase in $\%P_{regen}$ is necessary to make the mission feasible, the magnitude of technology improvement required for a one percentage point increase is discouraging. Clearly mission feasibility cannot be achieved by technology advances in any one of these areas alone; a multi-disciplinary technology approach will be necessary to achieve feasibility. Although the above sensitivities have been determined from small changes around the baseline values, the behavior of the results can be very non-linear. That is, the magnitudes of the sensitivities can change when the baseline values are changed. The extent and impact of this non-linear behavior will become evident from the mission feasibility contour plots which are presented below.

Hurricane Science Mission Technology Trade-Offs

In Figures 35 through 38 plots of $\%P_{regen}$ are used to assess the relative impacts of technology advances. The wing aspect ratio for each point was selected to maximize $\%P_{regen}$.

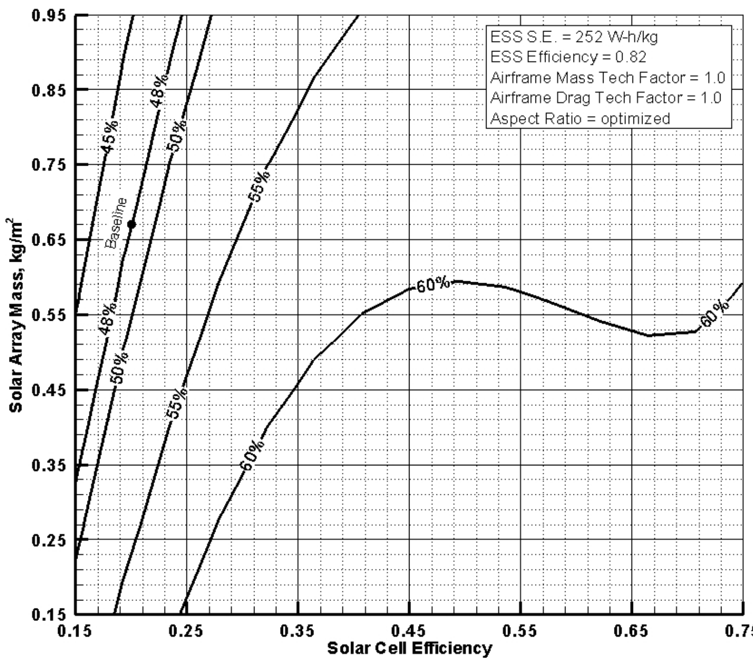


Figure 35. Variation of hurricane science mission feasibility with solar array technology.

<u>Solar Array</u>: The variation of feasibility with solar array technology is presented in Figure 35. It is clearly evident from this graph that increasing solar cell efficiency from the baseline has a much greater impact on feasibility than reducing array mass. Above about 50% solar cell efficiency, however, there is little to be gained from further increases in efficiency. (Note that the waviness evident in the 60% P_{regen} contour is associated with the response surface approximation. Using a polynomial equation to approximate a relatively flat "plateau" in the response can result in this type of behavior.) This fact can be seen even more clearly in Figure 36. In Figure 36, rather than plotting $\%P_{regen}$ contours the direct relationship between $\%P_{regen}$ and solar cell efficiency is plotted for a series of fixed solar array mass values. Above 40-45% solar cell efficiency the curves become very flat indicating little impact on mission feasibility. The close spacing between the curves of different solar array mass levels also reinforces that solar array mass only has a small impact on hurricane science mission feasibility.

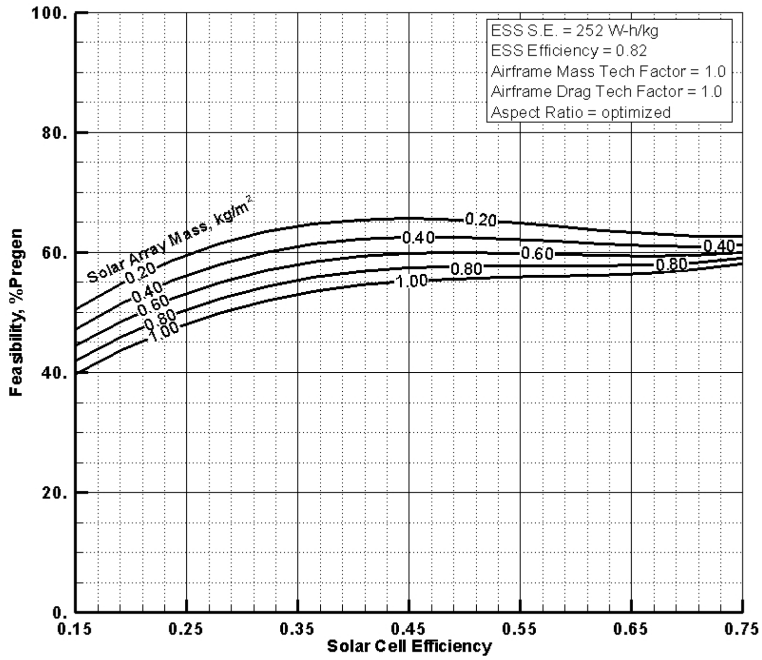


Figure 36. Variation of hurricane science mission feasibility with solar cell efficiency.

One influence on the relationship between feasibility and solar cell efficiency is the energy storage specific energy. To make use of the additional solar energy collected by higher efficiency cells, the energy storage capacity must grow. With a low specific energy system, the increase in mass required to store additional energy causes the energy required for flight to grow nearly as fast as the increase in energy available, leading to little benefit in terms of $\%P_{regen}$ and resulting in the nearly flat curves of Figure 36. If the solar array area were optimized for each case rather than set to the maximum possible, as done in this study, there would likely be a greater sensitivity to solar cell efficiency at high values. In that case increases in solar cell efficiency could be used to reduce array size and mass instead of increasing collected energy. However, note from Figure 36 that solar array mass has only a minimal influence on feasibility. Therefore, even if solar array size were optimized to reduce array mass the change in feasibility would be small.

Energy Storage System: Figure 37 shows the variation in feasibility with energy storage system technology. At the low specific energy of today's energy storage systems, feasibility is much more sensitive to increases in specific energy than improvements in efficiency. This is true not only at the relatively high efficiency of the baseline battery system (indicated by the dot on Figure 37), but also at the low efficiencies associated with regenerative fuel cell systems. There is a point, however, at which further improvement in specific energy has little value. For example, given a roundtrip efficiency of 0.50, similar to what might be achieved with a regenerative fuel cell system, there is little benefit from increasing specific energy above about 600 W-h/kg. Above that point, feasibility is best advanced by improvements in efficiency. For the parameter ranges considered, energy storage system technology has potential for a much greater impact on feasibility than solar array technology. The maximum $\%P_{regen}$ in Figure 37 is over 95% versus a maximum in Figure 36 of only ~65%.

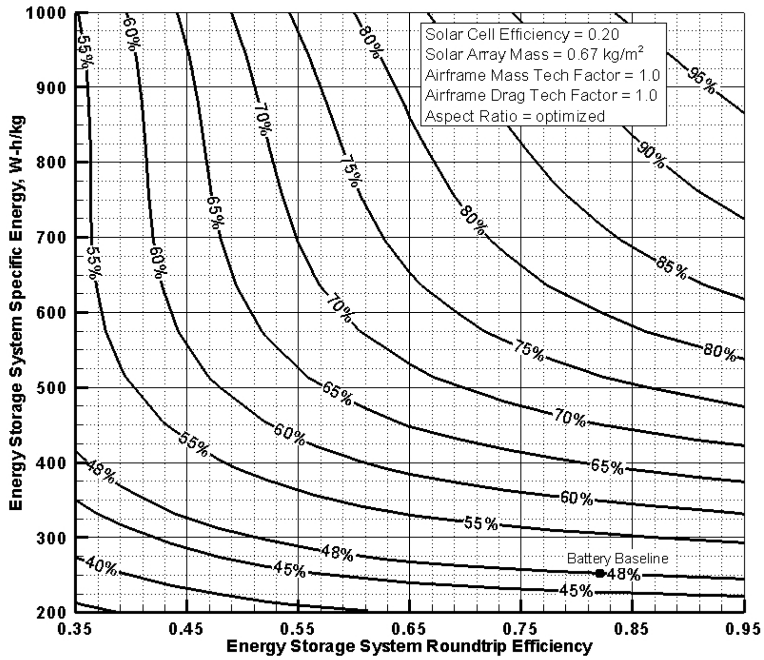


Figure 37. Variation of hurricane science mission feasibility with energy storage system technology.

Airframe Mass and Drag: The impact of airframe mass and drag technology on feasibility is shown in Figure 38. In this context "airframe mass" includes all of the structure mass (wing, pods, pylons, etc.) plus all of the non-propulsion related system masses (control surfaces, servos, avionics, etc.) Basically, "airframe mass" is the total vehicle mass minus the payload mass and propulsion system mass (which includes propellers, motors, energy storage system, and solar array). Figure 38 indicates feasibility is slightly more sensitive to drag reduction than airframe mass reduction, as also indicated in Table 42. In contrast to Figures 35 and 37, the contour lines in Figure 38 are straight and parallel. This indicates the relative impacts of mass and drag reduction are essentially unchanged throughout the range considered (±25%).

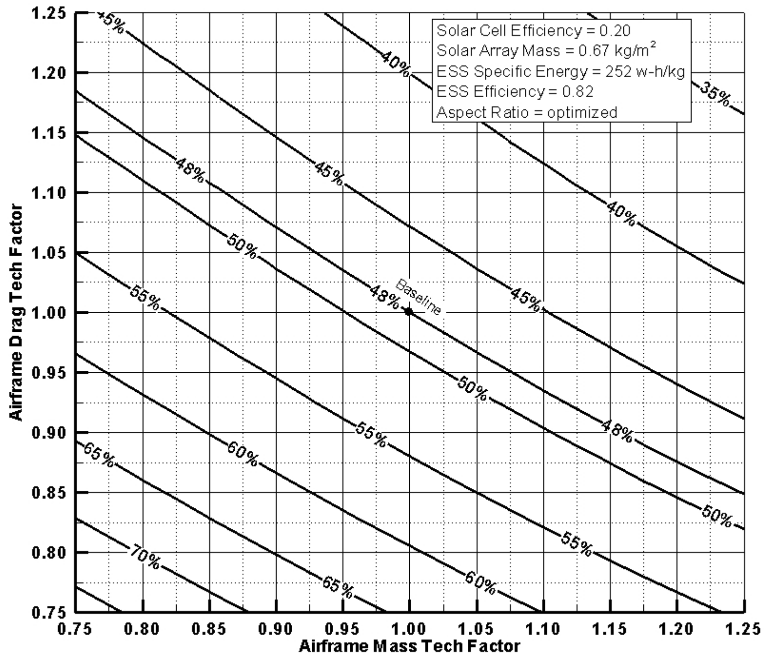


Figure 38. Variation of hurricane science mission feasibility with airframe technology.

Hurricane Science Mission Advanced Technology Solution

The results of the previous section demonstrate that the hurricane science threshold mission will not become feasible through advances in only one technology area. A combination of advances must be realized to enable this mission. There are infinite combinations of solar cell, energy storage, and airframe technology advances which result in mission feasibility. In selecting a "pathway" to feasibility the ultimate objective would be to minimize the risk and level of effort required (cost). Hundreds of different technology sets leading to feasibility can be identified in a relatively short amount of time using the response surface approximation. The difficulty, however, is in comparing the relative cost and risk among them to select a preferred solution. One approach attempted for this study was to define "degree-of-difficulty" curves for each technology area. These curves can be used to represent the relative difficulty of achieving advances in different areas and the typical increase in difficulty as advances are made (e.g., decreasing airframe mass by 10% is more than twice as hard as decreasing airframe mass by 5%). The preferred technology set can then be selected as the one that minimizes the total "degree-of-difficulty." Although the necessary set-up for this approach was performed, the data needed to generate suitable degree-of-difficulty curves were not readily available. Without input from subject matter experts in each of the technology fields, the degree-of-difficulty curves are highly subjective in nature, which limits the usefulness of this approach. Absent rigorously defined degree-of-difficultly curves, the hypothetical advanced technology solution was instead obtained from a subjective "balancing" of the required technology advances across the various areas.

A hypothetical advanced technology solution for the hurricane science threshold mission is given in Table 43. This combination of technology advances results in a $\%P_{regen}$ of 101%. Note that the values presented are not intended to be representative of any specific existing or envisioned technology. Improvement is primarily assumed in solar cell efficiency and energy storage system specific energy. The assumed solar cell efficiency of 35% is comparable to efficiencies which are currently being demonstrated in research laboratories. Array mass is actually assumed to increase to account for the extra mass typically associated with high efficiency cells. A modest increase in energy storage system roundtrip efficiency has been assumed, although the performance is fairly insensitive to this increase. The key technology assumption for this hypothetical scenario is a 500 W-h/kg specific energy. This would require significant advances in battery technology. A specific energy of 500 W-h/kg would be easier to achieve with regenerative fuel cell technology, albeit at the expense of lower efficiency. The trade-off between ESS specific energy and efficiency will be explored further below. A modest reduction of only 10% has been assumed for airframe mass and there is no reduction assumed in airframe drag.

Table 43. Hypothetical Advanced Technology Assumptions for Hurricane Science Threshold Mission

Technology Area	Baseline Value	Adv. Tech Value
Solar Cell Reference Efficiency	20%	35%
Solar Array Mass	0.67 kg/m^2	0.80 kg/m^2
Energy Storage System Roundtrip Efficiency	82%	90%
Energy Storage System Specific Energy	252 W-h/kg	500 W-h/kg
Airframe Mass Tech Factor	1.0	0.90
Airframe Drag Tech Factor	1.0	1.0

As noted above, the technology assumptions in Table 43 represent one of the infinite possible combinations which result in mission feasibility. To aid in visualizing other possible combinations, the contour plots of Figures 35, 37, and 38 are repeated below using the advanced technology assumptions.

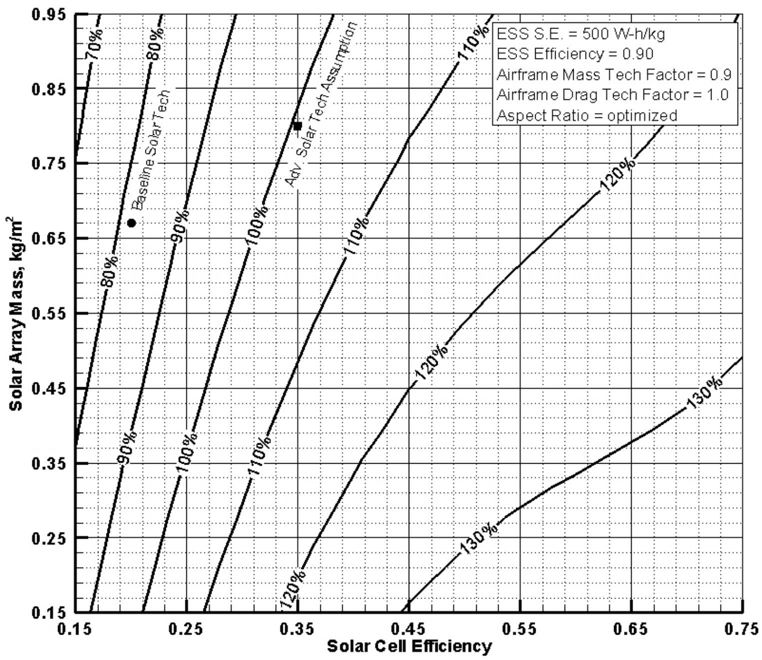


Figure 39. Variation of hurricane science mission feasibility with solar array technology (advanced ESS and airframe technology assumptions).

<u>Solar Array</u>: Figure 39 shows there is a fairly wide trade space in solar array technology which will result in feasibility (%$P_{regen} \geq 100\%$) given the advanced energy storage and airframe technology assumptions of Table 43. As in Figure 35, feasibility is more sensitive to solar cell efficiency than to array mass. However, a solar cell efficiency as low as 21% can still lead to feasibility if solar array mass can be reduced to 0.15 kg/m^2. The sensitivity of feasibility to efficiency decreases as efficiency is increased, but a region in which additional increases are not beneficial is never reached as occurred in Figure 35. This is likely the result of the higher energy storage system specific energy. Collecting more energy is only

beneficial if the energy storage system can grow to accommodate that energy without greatly increasing the vehicle mass and energy required. Therefore, the system level benefits accrued from increases in solar cell efficiency depend on the characteristics of the energy storage system.

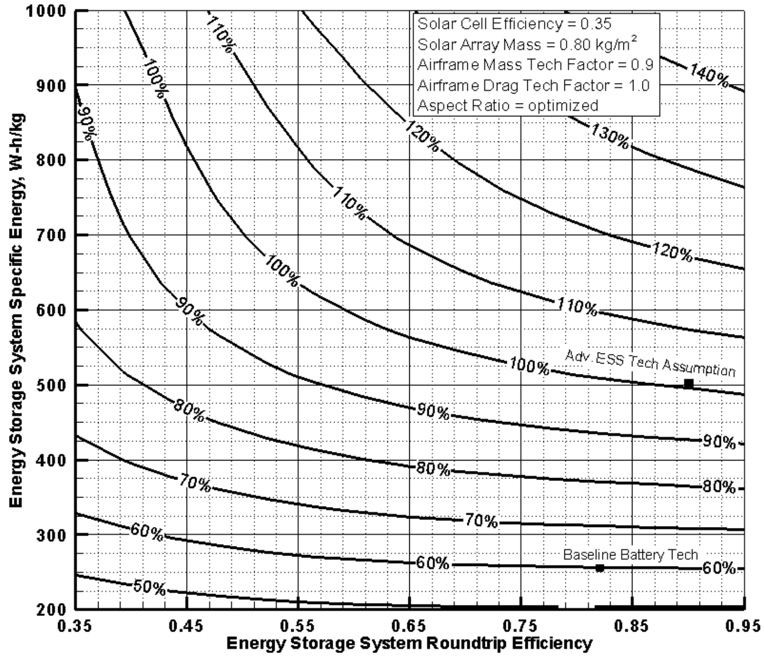


Figure 40. Variation of hurricane science mission feasibility with energy storage technology (advanced solar array and airframe technology assumptions).

Energy Storage: The relationship between required energy storage system specific energy and efficiency is presented in Figure 40. These contours have a shape very similar to those contained in Figure 37 for the baseline technology case. The technology assumptions made in Table 43, 90% efficiency and 500 W-h/kg specific energy, could possibly be viewed as representative of an advanced lightweight battery. However, since the sensitivity of feasibility to energy storage system efficiency is relatively low, there are also lower efficiency options available. For example, an advanced regenerative fuel cell system capable of 55% roundtrip efficiency at a specific energy of 650 W-h/kg would also achieve feasibility. Such an advance in regenerative fuel cell technology may in fact be more readily achieved than the 500 W-h/kg, 90% efficient generic battery assumption.

Airframe Mass and Drag: Figure 41 illustrates the combinations of airframe mass and drag reduction which result in hurricane science threshold mission feasibility given the propulsion system assumptions of Table 43. As in Figure 38, the impacts of mass and drag reductions are similar. A 6% reduction in drag with no reduction in mass provides the same level of feasibility as a 10% reduction in mass with no reduction in drag.

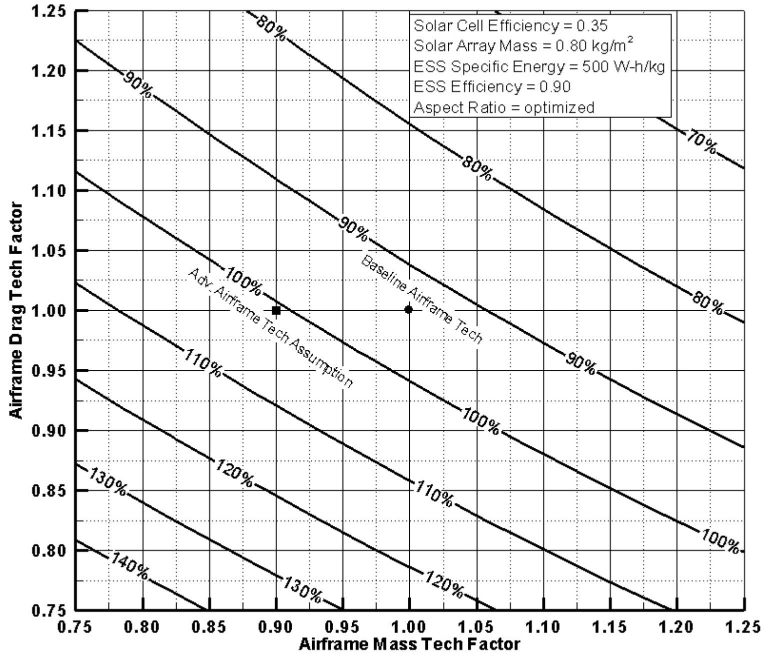


Figure 41. Variation of hurricane science mission feasibility with airframe technology (advanced solar array and energy storage technology assumptions).

Multi-System Technology Trades: The curves presented in Figures 39 through 41 allow one to quickly investigate possible technology trade-offs within the solar array, energy storage, and airframe areas individually. They can also be used in an approximate way to assess trade-offs among those different systems. Suppose, for example, that the energy storage technology assumptions in Table 43 are too aggressive and the level of energy storage performance expected corresponds to the 80% P_{regen} curve in Figure 40 (perhaps it is a 50% efficient regenerative fuel cell at 450 W-h/kg). The 20% gap in feasibility has to be closed by additional advances in the solar array and/or airframe areas. One approach to closing that gap would be to select a combination of solar array characteristics on the 110% curve in Figure 39 (e.g., 40% efficiency and 0.65 kg/m^2) and a combination of airframe technologies on the 110% curve in Figure 41 (e.g., 15% reduction in airframe mass and 5% reduction in drag). Although using the curves in this manner is not an exact approach to finding a feasible combination of technology assumptions, the combination of advances assumed above does result in a %P_{regen} value of 100% using the full analysis model.

7.3.3 Communications Relay Mission

The input parameters and ranges used for the communications relay mission technology trade study are shown in Table 44. Because the communications relay mission is more demanding of the SR propulsion system than the hurricane science mission, the parameter ranges cover more aggressive technology assumptions.

Table 44. Communications Relay Mission Trade Study Parameters and Ranges

Technology Variables	
Solar Cell Reference Efficiency	0.10 to 1.0
Solar Array Mass	0 to 1.5 (kg/m^2)
Energy Storage System Roundtrip Efficiency	0.3 to 1.0
Energy Storage System Specific Energy	100 to 1500 (W-h/kg)
Airframe Mass Tech Factor	0.50 to 1.5
Airframe Drag Tech Factor	0.50 to 1.5
Design Variables	
Wing Aspect Ratio	10 to 35

Removing cases with low and high $\%P_{regen}$ output was found to slightly improve the response surface accuracy for this trade study. A plot of the residual for the selected third-order polynomial RSE approximation is shown in Figure 42. As in the hurricane science mission study, the error is generally less than ±0.05 (i.e., predicted $\%P_{regen}$ is within ±5 percentage points of the actual) with a few points having higher error. Using the RSE approximation, the predicted $\%P_{regen}$ for the baseline technology case is 27% compared to the 25% obtained from the full analysis.

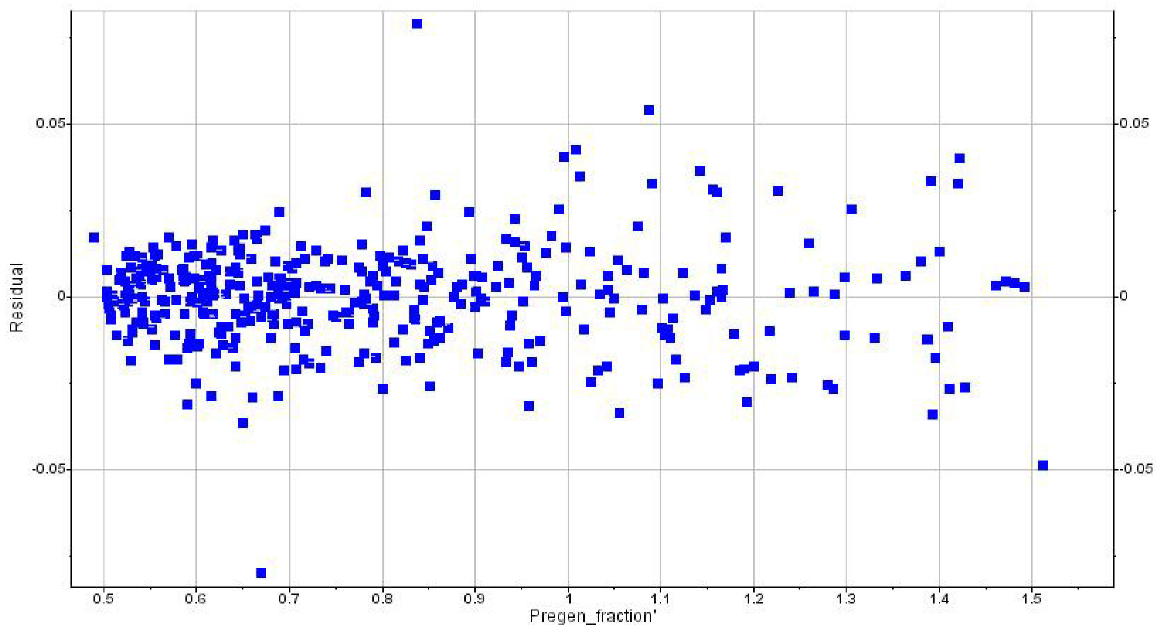


Figure 42. Communications relay mission response surface approximation residual.

Communications Relay Mission Technology Sensitivities

Table 45 shows the amount of technology improvement for each area which will individually increase feasibility by 1% (i.e., $\%P_{regen}$ increases from 27% to 28%). The wing aspect ratio (equivalently wing area) was optimized at each point.

Table 45. Sensitivity of Communications Relay Feasibility to Technology Improvements

Technology Area	Baseline Value	**Δ required to increase $\%P_{regen}$ by 1 point**
Solar Cell Reference Efficiency	20%	+2.0 pts (+10%)
Solar Array Mass	0.67 kg/m^2	-0.19 kg/m^2 (-28%)
Energy Storage System Roundtrip Efficiency	82%	+18 pts (+22%)
Energy Storage System Specific Energy	252 W-h/kg	+38 W-h/kg (+15%)
Airframe Mass Tech Factor	1.0	-0.08 (-8%)
Airframe Drag Tech Factor	1.0	-0.04 (-4%)

In all areas the improvement in technology required to increase $\%P_{regen}$ is more for the communications relay mission than for the hurricane science mission (compare to Table 42). As with the hurricane science mission, feasibility is most sensitive to improvements in airframe mass and drag. For the communications relay mission solar cell efficiency has a larger impact on feasibility than energy storage system specific energy, whereas the reverse is true for the hurricane science mission. Solar array mass has essentially no impact on feasibility. In fact, if the mass of the solar array is removed entirely, the feasibility increases from 27% to just 31%. Also, almost no improvement in feasibility is possible through increases in energy storage system efficiency. Increasing the energy storage system efficiency from the baseline value of 82% to the limit of 100% is necessary to achieve a 1% increase in $\%P_{regen}$. The low baseline $\%P_{regen}$ value of 27% and the low sensitivity of $\%P_{regen}$ to technology advances are indicative of the extremely demanding requirements the communications relay mission places on an SR type configuration. Significant advances in all of the above technology areas will be necessary to make the communications relay threshold mission feasible. Sensitivities are presented here around the baseline technology assumptions; the extent and impact of the non-linearity of these sensitivities away from the baseline will become evident from the mission feasibility contour plots which are presented below.

Communications Relay Mission Technology Trade-Offs

Technology trades have been investigated by plotting contours of constant feasibility ($\%P_{regen}$) for variations in technology parameters. The wing aspect ratio for each point was selected to maximize $\%P_{regen}$.

<u>Solar Array</u>: The variation of feasibility with solar array technology is presented in Figure 43. As with the hurricane science mission, increasing solar cell efficiency from the baseline has a much greater impact on feasibility than reducing array mass. Unlike the hurricane science mission, however, the benefit of solar cell efficiency increase is retained across the full range of values. The fundamental reason solar cell efficiency is more important for the communications relay mission is the significantly lower amount of solar energy available at the high latitude, wintertime design conditions. The short day and low sun angles cut the amount of solar energy collected by a horizontal array by roughly half compared to the hurricane science mission design conditions. Some of this loss in collected solar energy could possibly be offset through changes in mission operations and aircraft geometry to optimize the solar incidence angles for the array. In this analysis, however, the same conventional wing array arrangement and circular station keeping pattern used in the hurricane science mission is assumed. The shortage of incident solar energy makes high solar cell efficiency critical to feasibility. Though a key technology area, solar array improvements alone are not sufficient to make the mission feasible. Even at 100% efficiency and no array mass, the $\%P_{regen}$ is less than 60%.

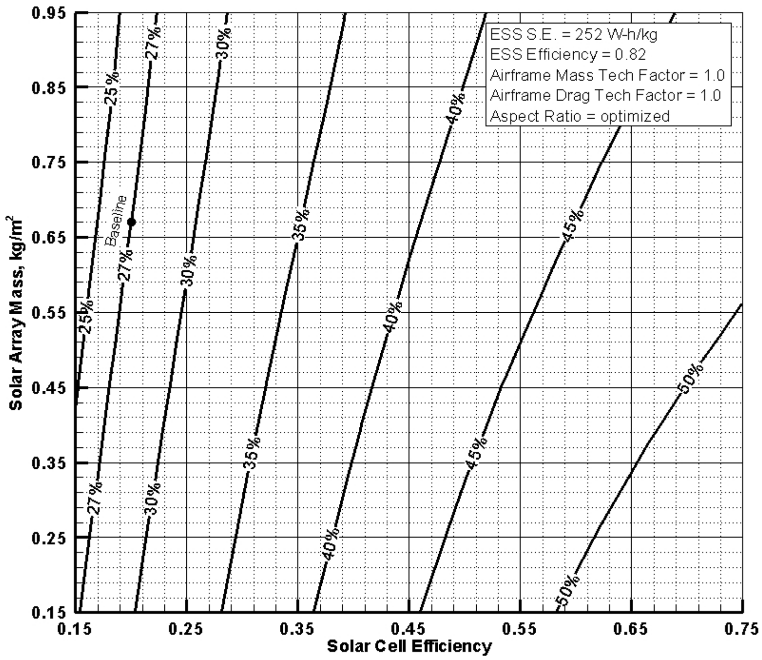


Figure 43. Variation of communications relay mission feasibility with solar array technology.

Energy Storage System: Figure 44 shows the variation in feasibility with changes in energy storage system technology. For moderate to high efficiency energy storage systems, increasing specific energy has a higher impact on feasibility than increasing efficiency. However, as with the hurricane science mission case, above a specific energy of about 600 W-h/kg there is little to be gained from further increases in specific energy and only efficiency improvements can increase feasibility. The contours in Figure 44 actually indicate that additional increases in specific energy can slightly hurt feasibility. This is, of course, not realistic and the slight bend of the contours can be attributed to approximating a "flat" surface with the polynomial RSE. In the hurricane science mission case, energy storage system technology was found to have a greater potential for positive impact on feasibility than solar array technology. The reverse is true for the communications relay mission. With the baseline solar array technology assumptions used in Figure 44, the energy collected by the solar array is quite limited and the corresponding energy storage system fairly small. The energy storage system mass is therefore a smaller fraction of total vehicle mass than in the hurricane science mission case (20% versus 36%) and improvements in specific energy (ESS mass reductions) have less overall impact.

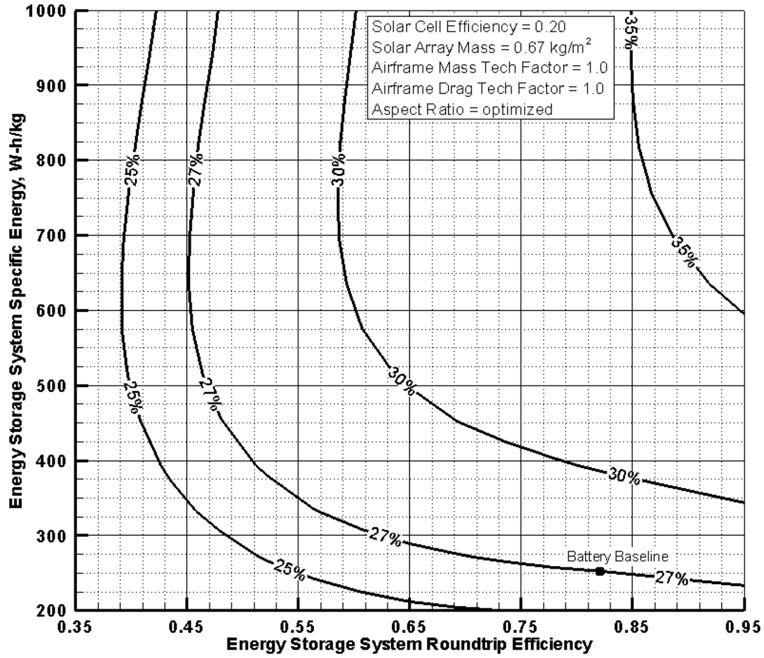


Figure 44. Variation of the communications relay mission feasibility with the energy storage system technology.

Airframe Mass and Drag: The contour lines illustrating the impact of airframe technology on communications relay mission feasibility in Figure 45 are similar in shape to those in Figure 38 for the hurricane science mission. Drag reduction has about twice the impact of mass reduction throughout the range investigated.

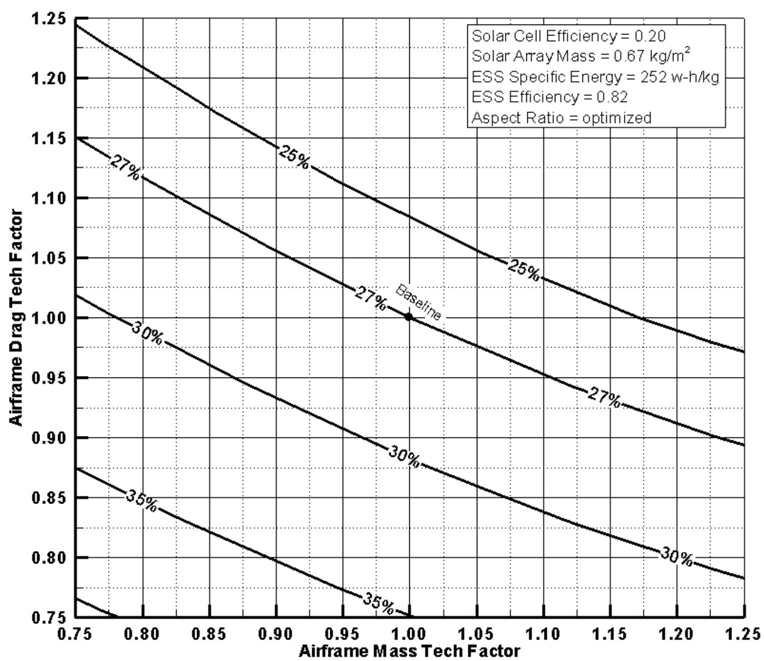


Figure 45. Variation of communications relay mission feasibility with airframe technology.

Communications Relay Mission Advanced Technology Solution

A subjective "balancing" of assumed advances across the technology areas has been performed to arrive at a hypothetical set of technology advances which enables the communications relay threshold mission. Selecting a reasonable set of assumptions was more difficult for the communications relay mission than the hurricane science mission because the required technology advances are so large. It was necessary to push the technology assumptions in every area to achieve feasibility. The hypothetical advanced technology solution for the communications relay threshold mission is given in Table 46. This combination of technology advances, which results in a $\%P_{regen}$ of 100%, is just one of many that would make the communications relay threshold mission feasible. Note that the values presented are not intended to be representative of any specific existing or envisioned technologies.

Table 46. Hypothetical Advanced Technology Assumptions for Communications Relay Mission

Technology Area	**Baseline Value**	**Adv. Tech Value**
Solar Cell Reference Efficiency	20%	45%
Solar Array Mass	0.67 kg/m^2	0.40 kg/m^2
Energy Storage System Roundtrip Efficiency	82%	90%
Energy Storage System Specific Energy	252 W-h/kg	750 W-h/kg
Airframe Mass Tech Factor	1.0	0.75
Airframe Drag Tech Factor	1.0	0.85

The technology assumptions in Table 46 are aggressive. The solar cell efficiency of 45% is beyond what has been demonstrated to date with multi-junction cells. Note that high effective efficiencies may be possible from combining the solar cells with other electricity producing elements. For example, one concept is to use thermoelectric cells in combination with the solar cells. Some of the heat generated by the inefficiency of the solar cells could then be converted to electrical power by the thermoelectric cells, resulting in a higher combined efficiency, but such concepts lead to higher array mass. In the above technology set a reduction in solar array mass has also been assumed in addition to the increase in efficiency. The simultaneous reduction in mass and increase in efficiency is counter to trends associated with current types of solar cells and array concepts. The required energy storage system has battery-like high efficiency with a specific energy greater than that projected for advanced regenerative fuel cell systems (having lower efficiency), and many times greater than current battery capabilities. Although the baseline airframe is already "clean" and very lightweight, the drag has been reduced by 15% and the mass by 25%. It should not be inferred that there are known research efforts to achieve the technology levels assumed in Table 46. These aggressive assumptions illustrate the extreme difficulty associated with meeting the communications relay mission requirements using a HTA SR platform.

Sensitivities and technology trades around the selected hypothetical advanced technology set are presented in Figures 46 through 48.

<u>Solar Array</u>: Advances in solar array technology, especially solar cell efficiency, are critical for the communications relay mission. As shown in Figure 46, with advanced energy storage and airframe technology assumptions the required solar cell efficiency for feasibility is greater than 35%, even if the solar array mass can be reduced to essentially zero. With more reasonable solar array mass assumptions, solar cell efficiencies greater than 50% are needed.

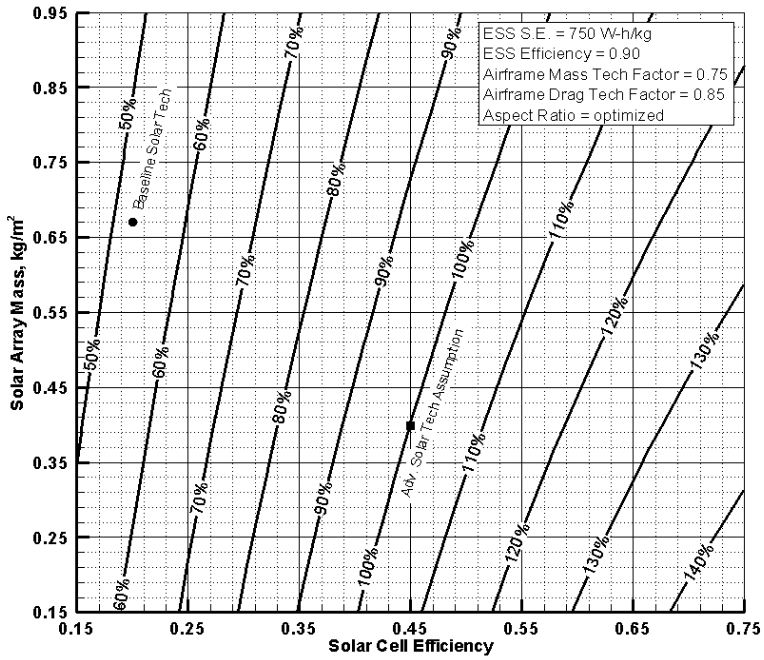


Figure 46. Variation of communications relay mission feasibility with solar array technology (advanced ESS and airframe technology assumptions).

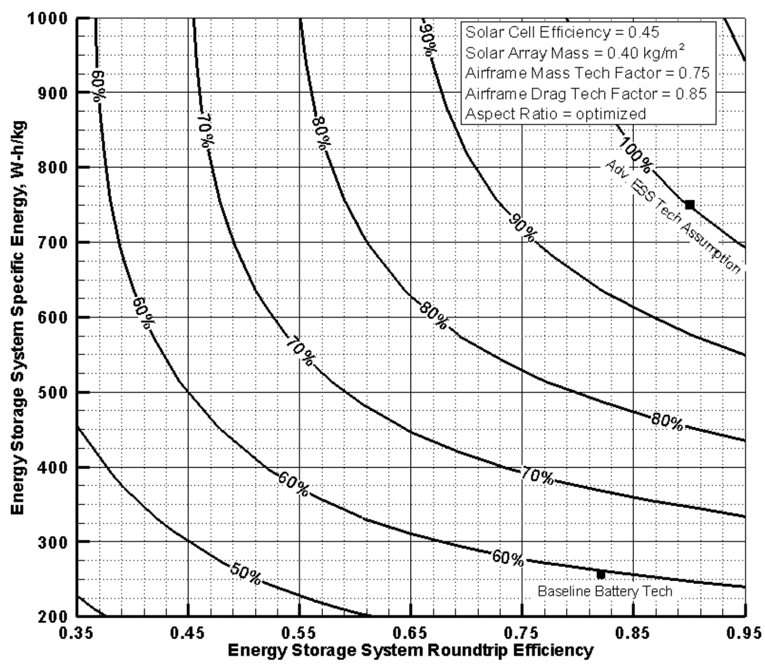


Figure 47. Variation of communications relay mission feasibility with energy storage technology (advanced solar array and airframe technology assumptions).

Energy Storage: Figure 47 shows that energy storage system technology must also be advanced significantly for the communications relay mission to become feasible. The required combination of high efficiency and high specific energy is beyond any currently envisioned technologies. With the advanced technology assumptions made for the solar array and airframe, Figure 47 indicates the energy storage system specific energy needs to be at least 700 W-h/kg. The roundtrip efficiency needs to be greater than 80% even if a specific energy of 1000 W-h/kg can be achieved.

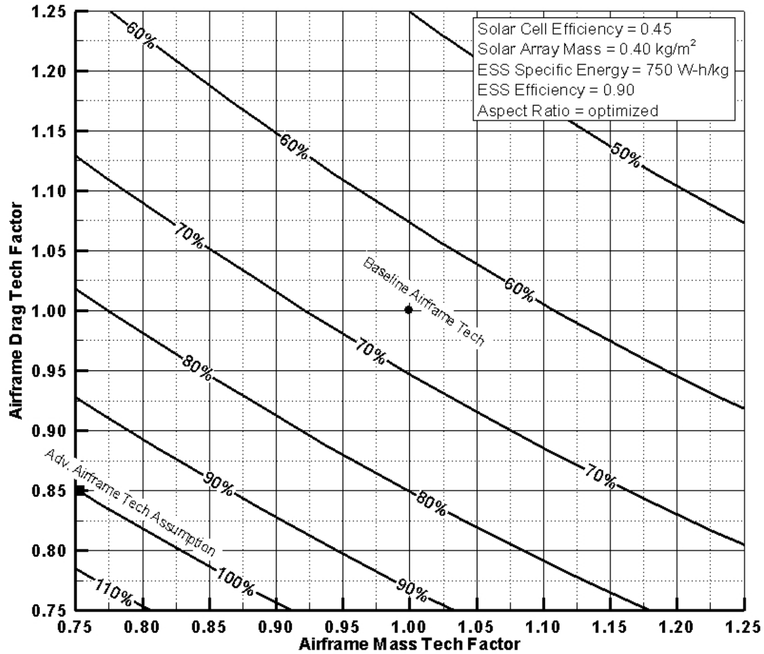


Figure 48. Variation of communications relay mission feasibility with airframe technology (advanced solar array and energy storage technology assumptions).

Airframe Mass and Drag: Despite the significant technology advances assumed for the propulsion system, the communications relay mission will not be feasible without significant advances in airframe technology as well. As shown in Figure 48, combining the advanced technology propulsion system with a baseline technology airframe results in a $\%P_{regen}$ of only ~65%. Technology must advance in all areas of the vehicle design in order for the communications relay mission to become feasible. Only through a multi-disciplinary research portfolio will this difficult mission ever become a reality for HTA SR aircraft.

<u>Multi-System Technology Trades</u>: There is little flexibility for trading technology assumptions among the solar array, energy storage, and airframe systems because the required technology assumptions are already near the limits of the ranges considered in the trade study. Additional advances in solar array technology offer the best opportunity for reducing the required technology in other areas. If, for example, solar cell efficiency could be increased to 65% with a solar array mass of 0.59 kg/m^2 (120% P_{regen} contour in Figure 46), the required advances in energy storage and/or airframe technology could be relaxed somewhat. If relaxing energy storage technology alone, combinations of specific energy and efficiency along the 80% P_{regen} contour in Figure 47 would become feasible. At a specific energy of 1000 W-h/kg, a roundtrip efficiency of 55% would now be acceptable (as perhaps could be realized with a very lightweight regenerative fuel cell system). The $\%P_{regen}$ predicted by the full analysis for this combination of technology assumptions is 98%. Figures 46 through 48 can, therefore, be used to assess the feasibility of a wide variety of technology assumptions around those in Table 46 with reasonable accuracy.

7.3.4 Technology Trade Study Conclusions

Solar cell efficiency and energy storage system specific energy are the key propulsion system technologies for improving feasibility of HTA SR concepts. However, the best mix of technology investments and goals for SR aircraft research depends on the target mission (especially the latitude and time-of-year requirements). Missions requiring high latitude flight during winter are largely limited by the amount of solar energy that can be collected and benefit greatly from solar cell efficiency improvements. The ability to efficiently collect solar energy is less critical for missions in more favorable solar conditions, and in that case feasibility can be hindered by the mass associated with storing the energy that is collected. This important interaction between solar cell efficiency and energy storage system specific energy is examined more explicitly in Figures 49 and 50. In these figures the variation in feasibility ($\%P_{regen}$) with combined changes in solar cell efficiency and energy storage system specific energy is shown for the hurricane science mission and communications relay mission, respectively. The technology assumptions for the other four technology areas (solar array mass, energy storage system roundtrip efficiency, airframe mass, and airframe drag) are held fixed at their baseline values.

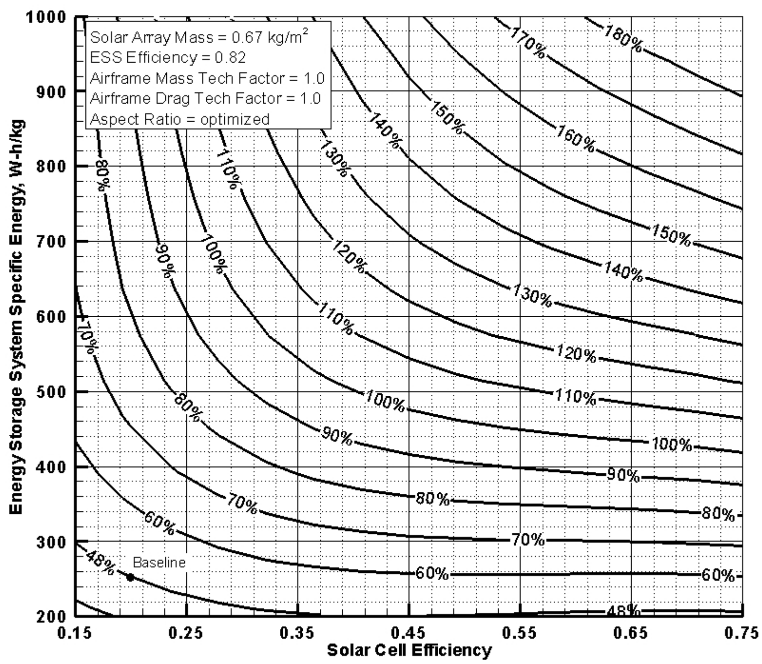


Figure 49. Variation of hurricane science mission feasibility with solar cell efficiency and ESS specific energy.

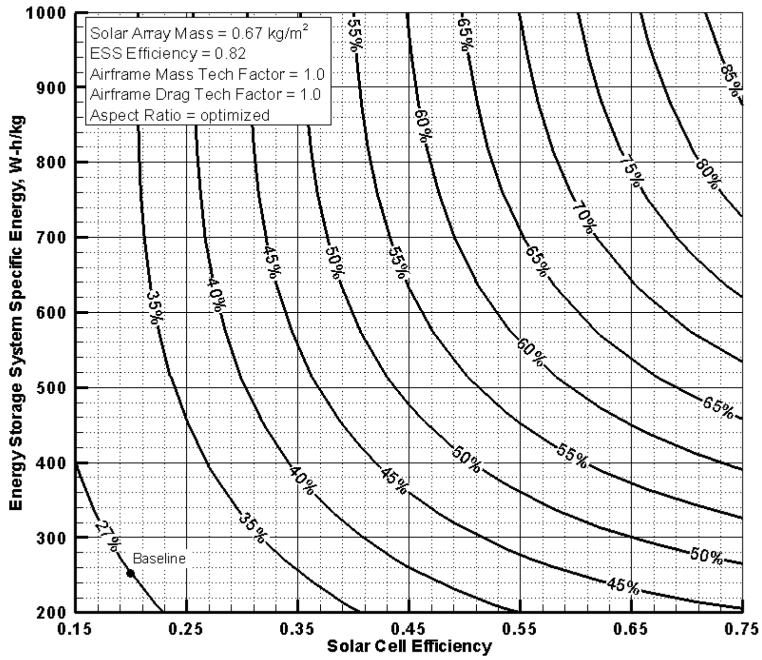


Figure 50. Variation of communications relay mission feasibility with solar cell efficiency and ESS specific energy.

Both figures illustrate a couple of general trends. First, solar cell efficiency and energy storage system specific energy must both be improved to achieve substantial increases in mission feasibility. Second, for both technology areas there are regions of the trade space in which little or no increase in feasibility is obtained from further improvement. While these general trends apply to both missions, there are also some clear differences. The regions of "diminishing returns" differ for the two missions. In Figure 49 (hurricane science mission), at low specific energy the contour lines are almost horizontal, implying increases in solar cell efficiency have little impact on feasibility. The contour lines at low specific energy are more angled in Figure 50 indicating solar cell efficiency improvements do increase mission feasibility for the communications relay mission. At high specific energies, the contour lines become vertical more quickly in Figure 50 than in Figure 49. This indicates that the relative importance of solar cell efficiency versus energy storage system specific energy is higher for the communications relay mission than the hurricane science mission. The difference in the relative importance of energy storage specific energy and solar cell efficiency can also be observed by examining the increase in feasibility from improvement in just one technology area. Improvement in solar cell efficiency alone increases $\%P_{regen}$ by ~12 points for the hurricane science mission and ~22 points for the communications relay mission. Improvement in energy storage system specific energy alone increases $\%P_{regen}$ by ~42 points for the hurricane science mission and only ~8 points for the communications relay mission. Another important difference between Figures 49 and 50 is the $\%P_{regen}$ levels. For the hurricane science mission, the $\%P_{regen}$ obtained from a combination of very high solar cell efficiency and very high specific energy is over 180%. The same combination of technologies for the communications relay mission results in a $\%P_{regen}$ of only ~90%, or in other words even with very high solar cell efficiency and energy storage system specific energy the mission is still infeasible. This is primarily due to the scarcity of available solar energy for high latitude, wintertime conditions.

Overall, the hurricane science mission requires technology advances which seem plausible given the current state of technology and research efforts that are underway. The high latitude, wintertime requirements of the communications relay mission, however, necessitate a set of airframe and propulsion advances which are revolutionary compared to current technology.

8.0 Conclusions

The purpose of this study was to develop a variety of HALE Unmanned Aerial Vehicle conceptual designs for two operationally useful missions (hurricane science and communications relay), compare their performance and cost characteristics, and quantify the technology improvements required (if any) to enable these missions. A total of sixteen concepts were developed for the study, including heavier-than-air (HTA) and lighter-than-air (LTA) configurations with solar-regenerative (SR) and non-regenerative (consumable fuel) propulsion systems. Several hybrid (solar+consumable) propulsion options were also explored. A capability to perform technology and mission feasibility studies for HTA and LTA HALE UAVs has been demonstrated.

None of the HTA consumable concepts examined can meet the threshold endurance requirement for either of the two missions (30 days for the hurricane science mission and 14 days for the communications relay mission). The 80 m wingspan, LH_2-fueled IC engine powered concept (Concept 1) has the greatest endurance of eight days for the hurricane science mission and 10 days for the communications relay mission. The endurance capability of Concept 1 is nearly matched by the 80 m wingspan, LH_2-fueled PEM fuel cell and electric motor powered concept (Concept 3). However, Concept 3 has higher risk, due primarily to the relatively complex and unproven propulsion system. Because the goal mission endurance could not be met with a single HTA vehicle, in Phase II of the study multi-aircraft operational concepts were examined. The operational and life cycle cost effects of a serial flight approach were compared for Concept 1 and two, "downsized" four-day endurance concepts. These four-day endurance vehicles, a 46 m wingspan, LH_2-fueled IC engine powered concept (Concept 1-small) and a 58 m wingspan, diesel-fueled concept (Concept 5-small), have lower estimated life cycle cost than the eight-day endurance Concept 1 (22% lower for Concept 1-small and 19% lower for Concept 5-small). This proves that maximizing endurance for the HTA vehicles does not result in the most cost effective system solution.

All of the LTA concepts are able to meet the hurricane science mission goal endurance of 180 days and exceed the communications relay threshold endurance for most mission start dates. (LTA endurance is sensitive to mission start date due to seasonal changes in winds aloft.) The LTA concepts with the best endurance for the communications relay mission are the LH_2-fueled PEM fuel cell powered concept (Concept 12) and the PEM fuel cell solar hybrid concept (Concept 15), both having endurances ranging from 36 to 180 days. For the communications relay mission, the SR LTA concepts (Concepts 13, 14 and 16) range from 36% to 155% larger than the largest consumable-fueled concept (Concept 12). In addition, the risk associated with the SR concepts is higher than the consumable options. A fundamental analysis of hybrid propulsion (solar+consumable) indicated that although hybrid HTA concepts would not be competitive, the extreme endurance of LTA concepts would make hybrid propulsion a viable option. The performance of Concept 15 (LH_2-fueled PEM fuel cell plus solar array hybrid) is similar to that of Concept 12, but Concept 15 is smaller in size and mass (17% smaller for the hurricane science mission and 9% smaller for the communications relay mission). Compared to the HTA consumable concepts, Concept 15 has lower overall production cost since the production quantity is only 18 vehicles. Concept 15 also has lower operations costs due to its minimal fuel consumption and fewer required maintenance actions. This lower operations and support cost, combined with the lower production cost, results in Concept 15 having the lowest overall estimated life cycle cost of all the concepts.

None of the HTA SR concepts are feasible assuming near-term technology. That is, the SR propulsion system is not able to collect, store, and deliver a sufficient amount of energy to keep the vehicle aloft for a full diurnal cycle (24 hours). For the mission worst case solar days, the SR system is capable of providing at most half of the energy required. A mission requirements trade study was conducted which indicated that given near-term technology, HTA SR concepts are limited to missions consisting of minimally useful payloads and operation at mid to high latitude, summer conditions. Assuming high altitude operations (altitude > 15 km), wintertime missions are not possible at any latitude even with no payload. Solar cell efficiency and energy storage system specific energy are the key technology areas requiring improvement to enable enhanced mission capabilities for HTA SR vehicles. The technology advances required to enable the SR powered HTA vehicles for the threshold hurricane science mission are reasonable; such as, a solar cell efficiency of 35% (baseline was 20%), an energy storage system specific energy of 500 W-h/kg and efficiency of 90% (baseline was 252 W-h/kg and 82% efficiency), and a 10% reduction in baseline airframe mass. Revolutionary advances are required, however, for the communications relay mission; for example, a combination of a solar cell efficiency of 45% accompanied by a 40% reduction in solar array mass, an energy storage system specific energy of 750 W-h/kg and efficiency of 90%, a 25% reduction in airframe mass, and a 15% reduction in airframe drag.

In the near term, the hurricane science and communications relay mission requirements can best be met with consumable propulsion systems. HTA SR concepts are not viable for these missions, and for the communications relay mission SR propulsion greatly increases the size and mass of LTA vehicles with little performance benefit. Although LTA vehicles have the greatest potential for extreme, multiple month endurance, the mission requirements can also be met by serial flight of lower endurance vehicles. In fact, maximum endurance is not necessarily the optimum from a system risk and life cycle cost perspective. Balancing cost, risk, and performance, Concept 5-small (HTA, 58 m wingspan diesel-fueled propulsion) is the best near-term solution.

References

1. Hall, D. W.; Fortenbach, C.D.; Dimiceli, E.V.; and Parks, R.W.: *A Preliminary Study of Solar Powered Aircraft and Associated Power Trains.* NASA CR-3699, December 1983.

2. Youngblood, J. W.; Talay, T.A.; and Pegg, R.J.: Design of Long Endurance Unmanned Airplanes Incorporating Solar and Fuel Cell Propulsion. AIAA 84-1430, June 1984.

3. Foch, R. J.; and Ailinger, K.G.: Low Reynolds Number, Long Endurance Aircraft Design. AIAA 92-1263, February 1992.

4. Romeo, G.; Frulla, G.; Cestino, E.; and Corsino, G.: HELIPLAT: Design, Aerodynamic and Structural Analysis of Long-Endurance, Solar-Powered Stratospheric Platform. *J. of Aircraft*, vol. 41, No. 6, Nov-Dec 2004.

5. Colozza, A.: *Initial Feasibility Assessment of a High Altitude Long Endurance Airship.* NASA CR 2003-212724, December 2003.

6. Atreya, S.; Mata, M.; Jones, R.; and Kohout, L.: Power System Comparisons for a High Altitude Long Endurance (HALE) Remotely Operated Aircraft (ROA). AIAA 2005-7401, September 2005.

7. Mankins, J. C.: Technology Readiness Levels: A White Paper. April 6, 1995. Office of Space Access and Technology, NASA, April 1995. URL: http://www.hq.nasa.gov/office/codeq/trl/trl.pdf. Accessed 12/15/2006.

8. Moorhouse, D. J.: Detailed Definitions and Guidance for Application of Technology Readiness Levels. *J. of Aircraft*, vol. 39, No.1, 2002, pp. 190-192.

9. Avery, M.; Harrison, W.; Velden, C.; Hawkins, J.; Vanek, M.; Vipavetz, K.; Guynn, M.; Buccellato, S.; Paddock, D.; and Qualls, G.: Decadal Survey Submission: An Autonomous Aerial Observing System for the Exploration of the Dynamics of Hurricanes (unpublished). May 2005.

10. Elsberry, R.; Harr, P.; Ritchie, E.; Emanual, K.; Gordon, R.; and Schoenung, S.: ERAST Tropical Cyclone Science Mission Demonstration Plan (unpublished). December 1997.

11. Volacom/Boeing UltraLEAP Phase 1 mid-term review (unpublished), October 2002.

12. Correspondence from Valentine Connell, director of engineering for Sky Station International, Inc., to James Bagwell, Communication Project Manager, NASA Lewis, November 1996.

13. Correspondence with Cameron Littke from Rockwell Collins, October 2005.

14. Pinkey (MAJ), F.; Hampel, D; DiPierro, S.; Abbe, B.; and Sheha, M.: UAV Communications Payload Development. MILCOM 97 Proceedings, vol. 1, November 1997, pp. 403-407.

15. Thornton, J.; Grace, D.; Spillard, C.; Konefal, T.; and Tozer, T.: Broadband Communications from a High-Altitude Platform: The European HeliNet Programme. *Electronics and Communication Engineering Journal*, vol. 13, No. 3, June 2001, pp. 138-144.

16. Oodo, M.; Tsuli, H.; and Miura, R.: Experiments on IMT-2000 Using Unmanned Solar Powered Aircraft at an Altitude of 20 km. *IEEE Transactions on Vehicular Technology*, vol. 54, No. 4, July 2005.

17. Weber, R.; and Condoor, S.: Conceptual Design Using a Synergistically Compatible Morphological Matrix. IEEE 1998 FIE Conference Paper, Session T2D, Paper #0-7803-4762-5/98, May 1998.

18. National Renewable Energy Laboratory (NREL) Renewable Resource Data Center (RRDC). *Solar Position and Intensity,* http://rredc.nrel.gov/solar/codesandalgorithms/solpos. Accessed 12/15/06.

19. Ozoroski, T.; and Mas, K.: A PC-Based Design and Analysis System for Lighter-Than-Air Unmanned Vehicles. AIAA-2003-6566, September 2003.

20. Gloudemans, J. R.; Davis, P. C.; and Gelhausen, P. A.: A Rapid Geometry Modeler for Conceptual Aircraft. AIAA-1996-0052, January 1996.

21. Benway, R. B.: Design and Development of a Light Weight, High Pressure Ratio Aircraft Turbocharger. SAE Technical Paper Series #871041, April 1987.

22. Patterson, C.: Unmanned High Altitude Long-Endurance Aircraft. AIAA 89-2011, July 1989.

23. Turney, G.: Fuel Cells for Propulsion of High-Altitude Aircraft (unpublished). NASA Glenn Research Center Aero Propulsion Analysis Office (A.P.A.O.) Note. December 1989.

24. Freen, P. D.; et al.: *Lightweight Two-Stroke Cycle Aircraft Diesel Engine Technology Enablement Program*. NASA CR-174923, vol. 1, August 1985.

25. Steinbroner, M.: *Something New Under the Sun*. Evaluation Engineering, March 2002. http://www.evaluationengineering.com/archive/articles/e0302data.pdf. Accessed 12/15/06.

26. Garcia, C. P.; Chang, B.; Johnson, D.; Bents, D.; Scullin, V.; and Jakupca, I.: *Round Trip Efficiency of NASA Glenn Regenerative Fuel Cell System*. NASA TM-2006-214054, January 2006.

27. Tudron, F. B.; Akridge, J. R.; and Puglisi, V. J.: *Lithium-Sulfur Rechargeable Batteries: Characteristics, State of Development, and Applicability to Powering Portable Electronics. Sion Power, Inc*., http://www.sionpower.com. Accessed 12/15/06.

28. NASA Dryden Flight Research Center. *Dryden Aircraft Photo Collection*: http://www.dfrc.nasa.gov/gallery/photo/index.html. Accessed 12/15/06.

29. Gallman, J. W.; Smith, S. C.; and Kroo, I. M.: Optimization of Joined-Wing Aircraft. *J. of Aircraft*, vol. 30, No. 6, Nov-Dec 1993, pp. 897-905.

30. Munk, M. M.: *General Biplane Theory*. NACA-TR-151, 1922.

31. Honeywell. *Honeywell Advanced Fibers and Composites, Spectra Fiber 2000 Product Specification*: http://www.honeywell.com/sites/sm/afc/spectra_fiber.htm. Accessed 12/15/06.

32. Avery, M. A.: Technical Memo: In Situ Measurements from Sondes (unpublished). NASA Langley Research Center, February 2006.

33. Zhai, H.; and Euler, A.: Material Challenges for Lighter-Than-Air Systems in High Altitude Applications. AIAA 2005-7488, September 2005.

34. Cadogan, D.; Sandy, C.; and Grahne, M.: Development and Evaluation of the Mars Pathfinder Inflatable Airbag Landing System. Acta Astronautica, vol. 50, No. 10, May 2002, pp. 633-640.

35. Resetar, S. A.; Rogers, J. C.; and Hess, R. W.: Advanced Airframe Structural Materials. A Primer and Cost Estimating Methodology. RAND R-4016-AF, 1991.

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14. ABSTRACT
An Analysis of Alternatives and a Technology Requirements Study were conducted for two mission areas utilizing various types of High Altitude Long Endurance (HALE) Unmanned Aerial Vehicles (UAV). A hurricane science mission and a communications relay mission provided air vehicle requirements which were used to derive sixteen potential HALE UAV configurations, including heavier-than-air (HTA) and lighter-than-air (LTA) concepts with both consumable fuel and solar regenerative propulsion systems. A HTA diesel-fueled wing-body-tail configuration emerged as the preferred concept given near-term technology constraints. The cost effectiveness analysis showed that simply maximizing vehicle endurance can be a sub-optimum system solution. In addition, the HTA solar regenerative configuration was utilized to perform both a mission requirements study and a technology development study. Given near-term technology contraints, the solar regenerative powered vehicle was limited to operations during the long days and short nights at higher latitudes during the summer months. Technology improvements are required in energy storage system specific energy and solar cell efficiency, along with airframe drag and mass reductions to enable the solar regenerative vehicle to meet the full mission requirements.

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