

Impact of Technology and Mission Variations on Aircraft Designed for Advanced Air Mobility

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ABSTRACT

NASA is conducting investigations in Advanced Air Mobility (AAM) aircraft and operations, including the development of Urban Air Mobility (UAM) aircraft designs that can be used to focus and guide research activities in support of AAM. This report is an investigation of the impact of technology and mission variations on several of the NASA AAM concept aircraft: quadrotor, quiet single main rotor, side-by-side, and tiltrotor configurations, with turboshaft and electric propulsion variants for each. First, the mission and aircraft models of the baseline designs were reassessed and updated, including rotor geometry optimization, update of the rotor performance models, and disk loading optimization. For these eight designs, technology and mission excursions were performed. Relative to the calibration cases that can be considered examples of good design practice, the impact of the weight technology factors is significant. For the electric aircraft, there is a very large impact of battery specific energy (Wh/kg), and correspondingly a very large impact of mission range. The vision of Advanced Air Mobility is driven by missions that will enable new transportation capabilities. Hence it is appropriate to compare Concept Vehicles of different lift and propulsive architectures, all designed to accomplish the same UAM mission. It is also useful however to consider specific missions that can take advantage of the strengths of individual aircraft configurations. So alternate designs were also developed for the concept vehicles: for turboshaft aircraft, longer unrefueled range, including faster cruise speed for the tiltrotor; for electric aircraft, shorter range and more realistic battery weight.

INTRODUCTION

NASA is conducting investigations in Advanced Air Mobility (AAM) aircraft and operations. AAM missions are characterized by ranges below 300 nm, including rural and urban operations, and passenger carrying as well as cargo delivery. Urban Air Mobility (UAM) is a subset of AAM and is the market segment that is projected to have the most economic benefit and be the most difficult to develop. NASA is developing UAM VTOL aircraft designs that can be used to focus and guide research activities in support of aircraft development for emerging aviation markets (Refs. 1–15, Figure 1). These NASA concept vehicles encompass relevant UAM features and technologies, including propulsion architectures, highly efficient yet quiet rotors, and aircraft aerodynamic performance and interactions. The configurations adopted are generic, intentionally different in appearance and design detail from prominent industry arrangements, while capturing the essential technology features. UAM operations are enabled by vertical take-off and landing (VTOL) capability, power and energy requirements are minimized by using low disk-loading rotors, and short range requirements permit consideration of non-traditional propulsion concepts.



Figure 1. NASA Advanced Air Mobility concept aircraft.

This report is an investigation of the impact of technology and mission variations on the NASA AAM concept aircraft. In order to focus on that objective without distraction, aircraft with unsolved dynamics issues are not considered — no two-bladed rotors, no stopped rotors, no hingeless rotors operating at high advance ratio. Similarly, aircraft with unsolved aerodynamics issues are not considered — no low disk-loading tilt-wings, no low speed wings, no ducted rotors. Thus

design excursions are conducted in this report for four aircraft: quadrotor (quad, Ref. 5), quiet single main rotor (QSMR, Ref. 6), side-by-side (side, Ref. 5), and tiltrotor (tilt, Ref. 10) configurations (Figure 2). Both turboshaft and electric propulsion variants are examined.



Figure 2. Aircraft for investigation of technology and mission variations: Quadrotor (quad), quiet single main rotor (QSMR), side-by-side (side), and tiltrotor (tilt) aircraft.

NDARC (Refs. 16–17) is used for aircraft sizing and mission performance evaluation. The comprehensive analysis CAMRAD II (Refs. 18–19) is used for rotor geometry optimization, to generate rotor performance models for NDARC.

The work begins with a reassessment and update of the baseline designs (Refs. 5, 6, 10). The design mission consists of carrying 5 occupants over 75 nm un-refueled range, with vertical takeoff at 6000 ft ISA and best-range speed cruise at 10000 ft ISA, and a one-engine inoperative requirement. Climb to cruise altitude is conducted at 900 fpm. There is a reserve segment of 20 min after the last cruise segment.

The baseline designs incorporate unified weight technology factors (based on small helicopter weights), systems and equipment (based on operating helicopter configurations), aerodynamics (based on established models for airframes and advanced rotor performance), new models (such as thermal management systems), and updated models (advanced turboshaft engine and battery performance).

Based on the original development of these four concept vehicles, a low tip speed gives minimum noise for FAA certification conditions (Refs. 7, 14). Rotor and aircraft noise were not further considered in the present work. The blade configurations (airfoils and blade tip geometry) were maintained from the original development. With the updated mission and aircraft models, the tasks of sizing (NDARC), rotor twist and taper optimization (CAMRAD II), rotor performance modeling (CAMRAD II), and disk loading optimization (NDARC) were iteratively executed. The result

was updated baseline designs for the four configurations, both turboshaft and electric propulsion variants (eight designs).

For these eight designs, technology and mission excursions were performed. The technology excursions include:

- weight technology (with and without weight reductions due to technology, including high and low calibrations)
- systems and equipment weight
- airframe drag
- rotor design blade loading and rotor profile power
- turboshaft engine weight and specific fuel consumption
- electric motor weight and efficiency
- wire weight
- thermal management systems design heat rejection (motor and battery)
- torque margin for rpm-controlled multicopter
- battery weight (installed usable Wh/kg).

The mission excursions include:

- range (together with battery weight for electric propulsion)
- reserve time
- rate-of-climb to cruise altitude
- one-engine inoperative requirement
- rate-of-climb at cruise
- takeoff and cruise altitude
- tiltrotor cruise speed.

The impacts of these technology and mission variations were generally consistent for the concept aircraft considered. Consequently, results for each variation are presented in this paper for only one of the configurations. Complete results for all variations and all configurations are in Ref. 20.

DESIGN TOOLS

The process and tools for development of the NASA concept vehicles are described in Reference 7.

The concept vehicles were sized using NDARC (NASA Design and Analysis of Rotorcraft), which is a conceptual/preliminary design and analysis code for rapidly sizing and conducting performance analysis of new aircraft (Refs. 16–17). NDARC has a modular architecture, facilitating its extension to new aircraft and propulsion types, including non-traditional propulsion systems. The design task sizes the vehicle to satisfy a set of design conditions and missions. The aircraft size is characterized by parameters such as design gross weight, weight empty, component dimensions, drive system torque limit, fuel tank capacity, and engine power. The analysis tasks include off-design mission analysis and flight performance calculation for point operating conditions. To achieve flexibility in configuration modeling, NDARC constructs a vehicle from a set of components, including fuselage, rotors, wings, tails,

transmissions, and engines. For efficient program execution, each component uses surrogate models for performance and weight estimation. Higher fidelity component design and analysis tools as well as databases of existing components provide the information needed to calibrate these surrogate models, including the influence of size and technology level. The reliability of the synthesis and evaluation results depends on the accuracy of the calibrated component models.

Performance analyses were conducted with the comprehensive rotorcraft analysis CAMRAD II (Refs. 18-19). CAMRAD II is an aeromechanics analysis of rotorcraft that incorporates a combination of advanced technologies, including multibody dynamics, nonlinear finite elements, and rotorcraft aerodynamics. The trim task finds the equilibrium solution for a steady state operating condition, and produces the solution for performance, loads, and vibration. The CAMRAD II aerodynamic model for the rotor blade is based on lifting-line theory, using steady two-dimensional airfoil characteristics and a vortex wake model. CAMRAD II has undergone extensive correlation with performance and loads measurements on rotorcraft (Ref. 19). CAMRAD II is used in the optimization of the rotor geometry for the aircraft sizing conditions, and to develop calibrated rotor performance models for NDARC.

TECHNOLOGY

Weight

The design gross weight is the mission takeoff weight. The structural design gross weight is taken as the design gross weight, with an ultimate load factor of 4. The maximum takeoff weight, as a parameter used in the weight models rather than operations, is equal to the design gross weight (full fuel and maximum payload).

The maximum number of occupants (perhaps including the pilot) is 5. With a weight of 200 lb each, the maximum payload is 1000 lb.

The weight allocation for vibration reduction is 3% of the aircraft empty weight, consistent with the use of low hub moment rotors (flapping blades).

The designs include a contingency weight equal to 5% of the aircraft empty weight. This factor is based on the 5% error in estimating the total scalable aircraft weight using the NDARC weight models.

NDARC parametric weight models (Ref. 16) are used for fuselage, flight controls, landing gear, rotor hub and blades, gear box, drive shaft, wing, and propeller. The parametric models are calibrated by comparing actual and estimated weights for specific aircraft. In the absence of weight statements for certified, production AAM/UAM aircraft, the small aircraft in the weights data base are used for calibration: Bell B412, Sikorsky S-76C++, Eurocopter SA-365 Dauphin, AugustaWestland A109, MBB Bo-105, Bell OH-58D,

Eurocopter AS350, McDonnell Douglas MD500M (Figure 3). These aircraft have maximum weights from 3000 to 11900 lb. Larger rotorcraft (UH-60A, UH-60M, RAH-66, V-22) are included in the calibration when information is not available for small helicopters. Calibration for tiltrotors is based on the XV-15 (though it was not a production design) and the V-22 (though it is not small). The baseline calibration is based on the mean values for several aircraft; design excursions will include high calibration (mean + standard deviation) and low calibration (mean - standard deviation).



Figure 3. Small rotorcraft for weight model calibration.

Allocations for System and Equipment Group weights are based on current helicopters, with 10–50% reduction for technology. The weights are furnishings and equipment group = 150 lb; flight control and hydraulic group = 50 lb; environmental control group = 75 lb; instruments group = 20 lb; avionics group = 110 lb; electrical group = 75 lb. The total weight is 480 lb.

Drag

The aircraft drag buildup is specified in terms of drag coefficients, which are scaled with the appropriate component area. All aircraft have the same well-designed and built low-drag fuselage, and fixed landing gear. The rotors have faired hubs and low-drag pylons. Fuselage and wing vertical drag coefficients produce the aircraft download in hover. Figure 4 shows the drag buildup for the baseline aircraft. The rotor hubs and support contribute much of the drag. Rotor hub, rotor support, and wing drag scale with aircraft size, hence the higher drag of the electric aircraft.

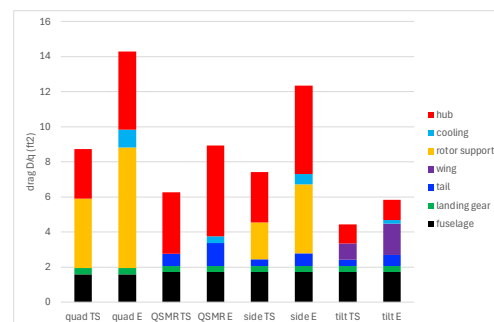


Figure 4. Drag buildup for baseline aircraft.

Aircraft and Rotor Models

The aircraft and rotor characteristics are generally based on the original designs (Refs. 5, 6, 10), with updated blade loading and airfoil decks, optimized twist and taper, and updated rotor performance models.

The design blade loading is $C_w/\sigma=0.09$, chosen based on stall margin for good maneuver capability in forward flight. The reference atmosphere for this blade loading is the takeoff condition, 6000 ft altitude ISA. The rotor tip speed is 450 ft/sec, for low noise. The blade airfoils for all the rotors (except the tiltrotor) are VR12 inboard and SSCA09 outboard, with transition over 85-95%R (Ref. 21). The VR12 airfoil has good stall characteristics, consistent with the design blade loading choice. The thin airfoil at the tip is not needed for the low tip speeds of the present designs, but was used in the original developments (Refs. 5–6) to permit examination of the effects of tip speed. The airfoil decks were constructed specifically to support these concept aircraft designs (Ref. 21).

For the tiltrotor, the design blade loading is $C_w/\sigma=0.13$, higher than for the other configurations since the aircraft is in airplane mode for high speed forward flight (Ref. 22). The cruise tip speed is 225 ft/sec, 50% of hover for efficient operation. The blade airfoils are NACA-64 series sections with thickness $t/c = 28/18/12/8\%$ from root to tip (XV-15 airfoils, Ref. 23; used at radial stations $r/R=0.2, 0.55, 0.8, 0.95$ respectively). A stall delay model is used for hover and edgewise flight performance.

By examining the impact of design blade loading and rotor profile power on the concept vehicles, the potential gain from further optimizing airfoils can be assessed.

The quadrotor has collective control, cross-shafting, and flapping rotors. The quadrotor hubs are at lateral and longitudinal stations $\pm 1.35R$ (35% separation). The rear rotors are 0.35R above the front rotors, for improved performance and noise levels (reduced rotor/rotor interference). The rotors have 5 blades. The flap frequency is 1.03/rev (4%R equivalent hinge offset), for low hub moments and hence low rotor weight. The hinge and pitch horn are arranged with pitch-flap coupling $\delta_3=45$ deg, which reduces flapping relative the shaft by about 30% in forward flight. The support arm weight is 7.5 lb/ft (an estimate that gives a reasonable fraction of empty weight). The optimized blade geometry has -14 deg linear twist and 0.95 linear taper ratio.

The QSMR has conventional helicopter control, with articulated blades (flap-lag-pitch motions). The original design uses the NOTAR anti-torque system (NDARC model described in Ref. 6); the baseline design for this investigation has a low-tip-speed tail rotor. The main rotor has 6 blades, with tips drooped (20 deg) and tapered (60%) at 0.94R for noise reduction. The flap frequency is 1.035/rev (4.5% equivalent hinge offset). The optimized blade geometry has -14 deg linear twist. The tail rotor design constraints are 7

lb/ft² disk loading, $C_w/\sigma=0.12$, and 450 ft/sec tip speed; with 6 blades and $\delta_3=45$ deg.

The side-by-side aircraft has collective and longitudinal cyclic rotor control, with articulated blades (flap-lag-pitch motions). The rotors are overlapped by 15% (span = 85% rotor diameter) for optimum cruise performance, with rotors rotating outboard blades forward. The rotors have 4 blades, with tips swept (15 deg) and tapered (60%) at 0.94R. The flap frequency is 1.035/rev (4.5% equivalent hinge offset). The rotor support (wing) is a non-lifting, faired structure; the structural weight is estimated as for a tiltrotor wing, with the takeoff requirement and torsion stiffness constraint (Ref. 16). The optimized blade geometry has -16 deg linear twist.

The tiltrotor has collective and longitudinal cyclic control for helicopter mode flight, and fixed wing controls for airplane mode flight, with hingeless rotors. The rotors have 6 blades. The flap frequency is 1.07/rev (for blade weight estimation). The optimized blade geometry has -29 deg linear twist (nonlinear twist is not needed with cruise at 50% hover tip speed) and 0.95 linear taper ratio. The design wing loading is 60 lb/ft², with the wing span obtained from the fuselage width, rotor radius (the rotor hubs are at the wing tips), and the rotor-fuselage clearance of 0.60 ft.

Turboshaft Engine

An advanced technology turboshaft engine model is used (from Refs. 24-25). The weight model is $W = 255. + 0.0484P + 0.00000353P^2$, for weight in lb from the power P (MRP) in hp. Specific fuel consumption decreases with increased power. At the design size (660 hp MRP), the engine weight is 288 lb (0.44 lb/hp) and $sfc = 0.50$ lb/hp-hr. Transmission losses are 2%. Fuel flow is increased by 5% to account for engine degradation. Accessory losses are 20 hp. Engine quieting requires 25 lb per engine (Ref. 6).

Electric Motor

The electric-powered aircraft use high-speed/low-torque motors, with a transmission from the motor to the low-speed/high-torque rotors. The combination of motor and transmission can be considered equivalent to an electric motor specifically designed to carry the steady and vibratory loads of the rotors. The NDARC parametric motor weight model (Ref. 16) is $W = 0.5382Q^{0.8129} + 0.1P$, for weight in lb the peak torque Q (ft-lb) and power P (hp). The second term is the weight of the electronic speed control system. There is a separate model for the thermal management system. The motor efficiency is assumed to be constant at 95%, independent of torque and speed. A more complete model of the motor performance would use an efficiency map, but constant efficiency used for hover and cruise captures the effect on aircraft sizing.

Wiring weight is 0.62 lb/ft, with the wiring length calculated from the distance between motors and the battery.

With rpm-controlled rotors, a torque margin of 25% is needed for good response to control in gusts and maneuvers (Ref. 26). This margin is implemented by limiting the power available to 75% (instead of 95%) in all design mission segments and flight conditions.

Battery

Light and efficient batteries are crucial to producing good designs for electric aircraft. The battery technology considered is based on the MoliceL INR-21700-P45B Li-ion battery. The battery model uses discharge characteristics (voltage as function of depth-of-discharge and current) for efficiency calculations. Margins for maximum charge and discharge are established to prolong battery life (in terms of discharge-charge cycles): charge to within 5–10% of full capacity and discharge to 15–20% capacity. The model considers 80% usable energy: depth-of-discharge = 0.05 to 0.85. The NDARC convention is that the battery capacity refers to the usable energy, with the input specific energy accounting for minimum and maximum depth-of-discharge limits. The installed specific energy is reduced by packaging and conditioning requirements, with a separate model for the thermal management system.

The MoliceL INR-21700-P45B Li-ion battery has specific energy of 242 Wh/kg, and energy density of 643 Wh/L. The maximum burst discharge current is 10C. With 80% usable energy, the cell specific energy is 194 Wh/kg. Assuming 50% for the battery management system, the installed usable specific energy is 129 Wh/kg.

The baseline designs with electrical propulsion use an installed usable battery specific energy of 400 Wh/kg. Cell specific energy (uninstalled) is 650 Wh/kg; 520 Wh/kg with 80% usable energy; and assuming 30% for the battery management system, the installed usable specific energy is 400 Wh/kg.

Thermal Management System

The thermal management system model gives the power, mass flow, and thrust as a function of rejected heat (Ref. 27). The system weight depends on the design rejected heat. For the baseline aircraft, the design rejected heat is 2% of the battery power capacity, and 5% of the motor power.

BASELINE DESIGNS

Baseline designs were developed for the four AAM concept vehicle configurations: quadrotor, QSMR, side-by-side, and tiltrotor; for both turboshaft and electric propulsion systems. Figure 5 shows key attributes of the eight aircraft (weight in lb, power in hp, speed in knots, ROC in ft/min).

An important metric of the design is the growth factor, which quantifies the sensitivity of the aircraft to weight changes due to design maturity, technology introduction, or mission variations (Table 1). The structural weight growth

factor is the design gross weight change produced by a unit change of payload or system weight, here evaluated for increments of ± 200 lb. The turboshaft aircraft all have growth factors of about 2. The electric aircraft have much larger growth factors, significantly varying with configuration, largely due to the high battery weight.

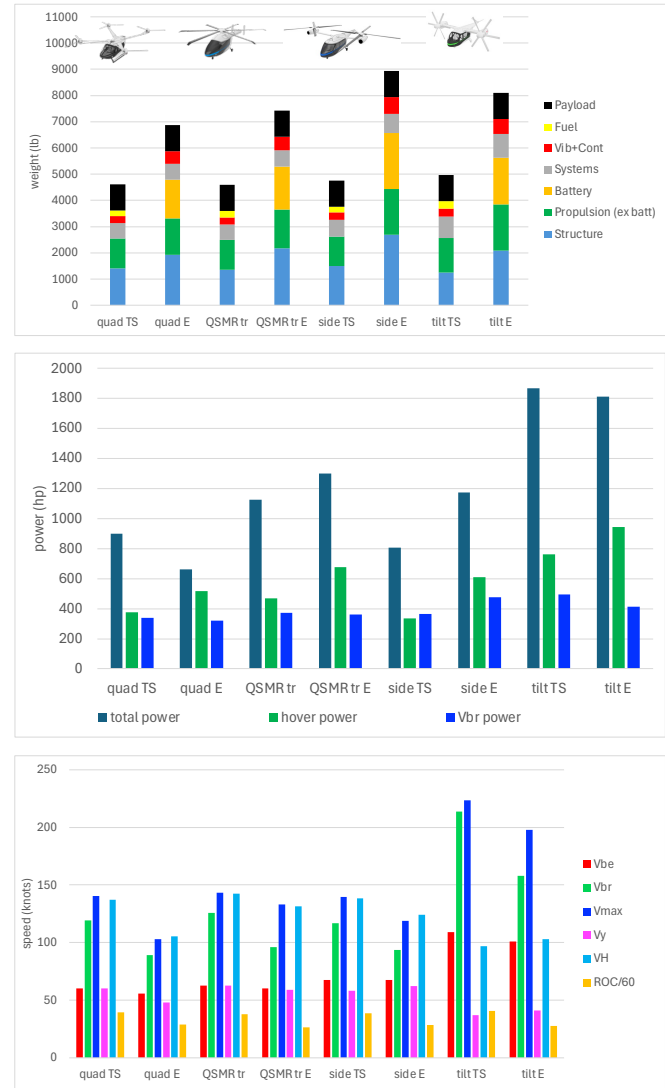


Figure 5. Baseline AAM Concept Vehicles (weight, power, and speeds).

Table 1. Growth factors

	turboshaft	electric
quadrotor	1.75	3.07
QSMR	1.89	4.00
side-by-side	1.97	5.27
tiltrotor	2.05	3.92

The turboshaft aircraft all have about the same design gross weight, fuel weight, and empty weight. The electric aircraft have higher weight empty than the corresponding turboshaft designs, due to the large battery weight and aircraft growth factor. The differences in weights for the electric designs are a consequence of the large growth factor together with differences in sensitivity to various technology and mission parameters.

The total installed power is larger than the hover power, because of the one-engine-inoperative requirement, and for turboshaft engines also reflecting the lapse rate between sea level (for the reference power) and the operating altitudes (6000 ft takeoff, 10000 ft cruise). The aircraft generally have about the same speeds for best endurance V_{be} , best range V_{br} , and maximum V_{max} , with best range and maximum speeds somewhat lower for electric aircraft. The tiltrotor best range (cruise) and maximum speeds are significantly higher, even though cruise speed is fallout for these designs, not a requirement. This higher speed is due to the increased efficiency in cruise of the tiltrotor configuration, which follows from the rotor performance in axial flight and the lower aircraft drag.

VARIATIONS

The impact of technology and mission variations are examined for the concept aircraft (quadrotor, QSMR, side-by-side, and tiltrotor configurations), by changing the appropriate parameters of the NDARC weight and performance models. Some parameters, notably battery weight and mission range, produce very high growth of the aircraft size, even divergence of the sizing iteration. The design variations are conducted for both turboshaft and electric vehicles, except of course for the propulsion-system-specific parameters.

TECHNOLOGY

Weight technology factors

Weight technology is accounted for by using factors on the weight estimation equations. These factors first are calibrated based on specific aircraft, close to the target design. In the absence of weight statements for certified UAM designs, the calibration is based on small helicopters (Figure 3). The calibrations use the mean values over several helicopters (Figure 6), as well as low values (mean minus standard deviation) and high values (mean plus standard deviation). The impact of technology is estimated to be a 5% to 25% reduction in component weight (Figure 6), particularly due to advanced materials and design practices.

The baseline aircraft are compared to designs obtained considering mean calibration factors (as well as low and high calibration values) but no technology advances (Figure 7). The designs using low calibration but no technology advances show some increase in size (weight and power) compared to the baseline designs, which use mean calibration together with technology advances. The low calibration is

representative of good design practice, hence is important as an attainable design case, even in the absence of demonstrations or research for weight reduction by technology. The high calibration can be considered poor design practice. With high calibration and no technology advances, the designs do not close for the electric QSMR and side-by-side configurations.

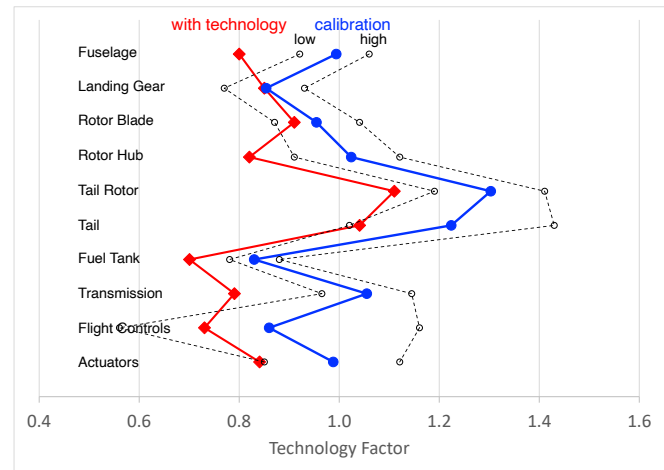


Figure 6. Weight technology factors.

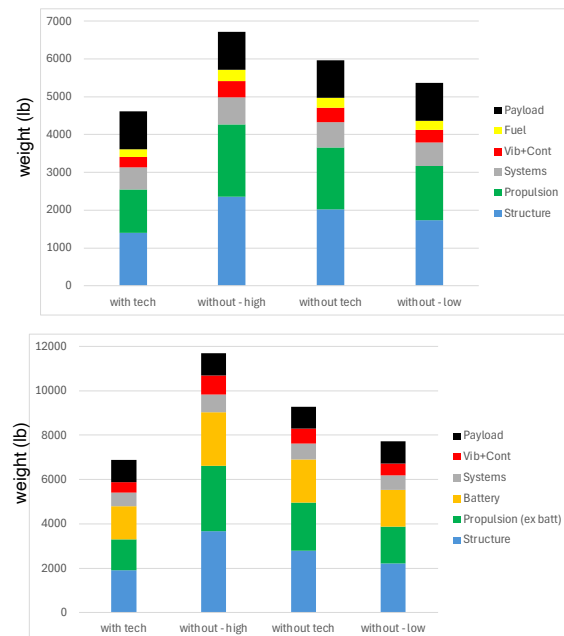


Figure 7. Quadrotor, turboshaft (top) and electric (bottom) — impact of weight technology factors. Designs with technology, compared to only calibration (no technology factors): high, baseline, and low calibration.

Systems and equipment

The systems and equipment weight is changed by -200 lb to 200 lb, relative to the baseline fixed weight of 480 lb. There is a moderate impact of weight changes, reflecting the aircraft growth factor (Figure 8).

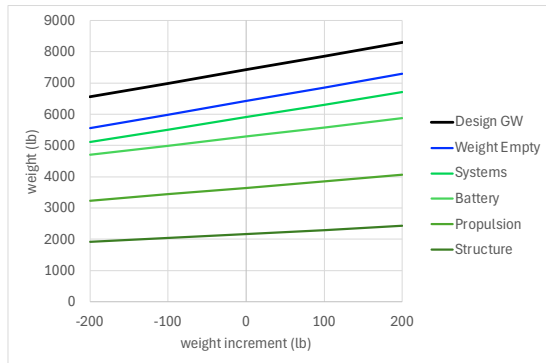


Figure 8. QSMR, electric — impact of systems and equipment weight.

Aircraft drag

Figure 4 shows the aircraft drag buildup. These total drag values are relatively high (except for the tiltrotor) for aircraft of this size. The airframe drag is varied by $\Delta D/q = -2.0$ to 2.0 ft². There is little impact of aircraft drag change, since cruise efficiency is not a major driver of the size given the short range of the design mission (Figure 9).

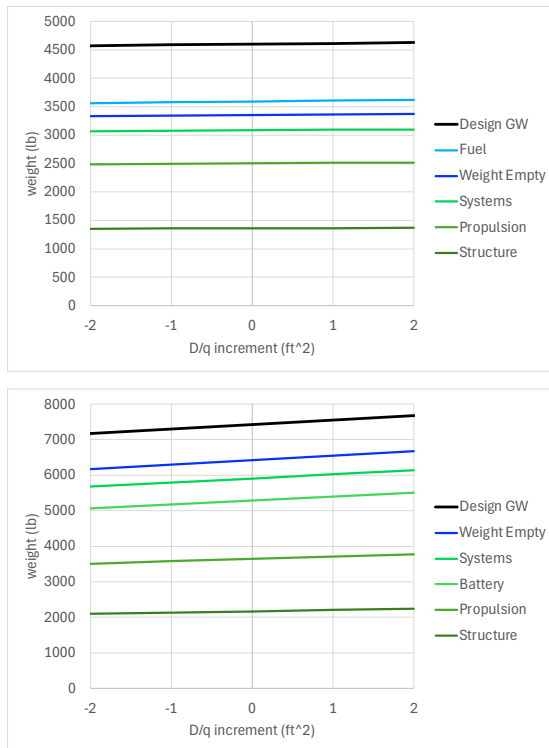


Figure 9. QSMR, turboshaft (top) and electric (bottom) — impact of aircraft drag.

Blade loading and profile power

The design rotor blade loading, $C_w/\sigma=0.09$, is significantly higher than for current helicopters. The blade loading is varied from 0.09 to 0.07 (baseline 0.13 to 0.11 for tiltrotors). The mean blade drag coefficient in cruise is high, since the low tip speed results in high advance ratios, increasing the stall on the retreating blade of edgewise rotors. The profile power is varied from 100% to 80% . The prescribed changes in blade loading and profile power reflect possible further optimization of the rotor design, such as through airfoil modifications. There is a small impact for turboshaft configurations, more for electric due to the larger growth factor (Figure 10). However, the impact might be larger if the performance model was re-calibrated to account for increased solidity, and the disk loading was re-optimized.

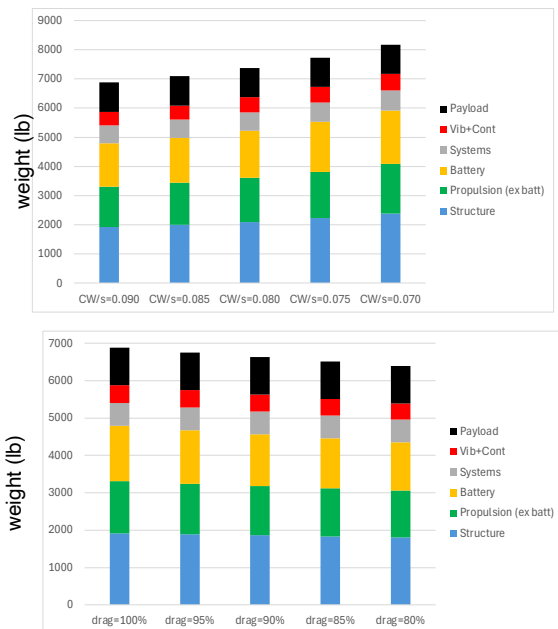


Figure 10. Quadrotor, electric — impact of blade loading (top) and profile power (bottom).

Turboshaft engine

The turboshaft engine weight (which scales with power) is varied from 80% to 120% . The engine specific fuel consumption is varied from 80% to 120% . There is a small impact of weight variation, and little impact of specific fuel consumption, since cruise efficiency is not very important given the short range of the design mission (Figure 11).

Electric motor

The motor weight (which scales with torque) is varied from 80% to 120% of the baseline. The motor efficiency (constant, independent of speed and power) is varied from 0.95 to 0.75 (Figure 12). There is a small impact of weight variation, since the designs use light-weight motors; and a moderate impact of motor efficiency, since efficiency has direct influence on energy used and hence battery weight.

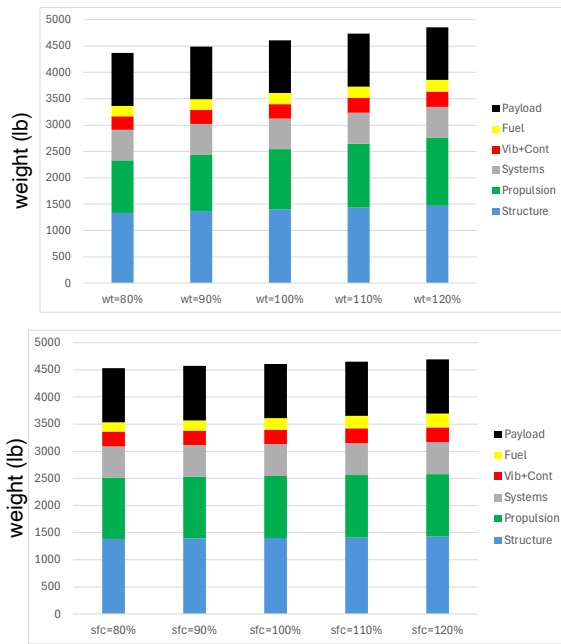


Figure 11. Quadrotor — impact of turboshaft engine weight (top) and specific fuel consumption (bottom).

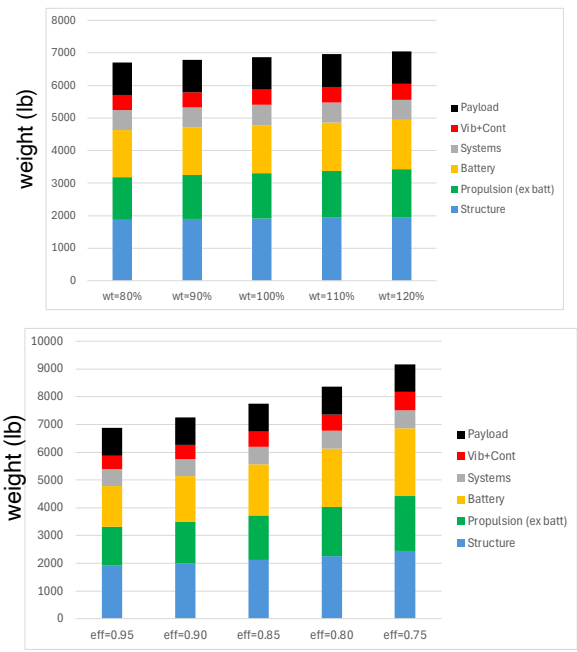


Figure 12. Quadrotor — impact of electric motor weight (top) and efficiency (bottom).

The wiring weight is varied from 100% to 300% of the baseline, and the thermal management system design rejected power is varied from 100% to 300% of the baseline, for both motor and battery (Figure 13). There is little impact of wire weight and thermal management system variations, because of the small fraction of empty weight given the design assumptions.

Motor torque margin

For the rpm-control hexacopter configuration (see Ref. 20 for comparison of quadcopter and hexacopter designs), the usable fraction of available power is varied from 55% to 95%, relative to the baseline of 75% (25% control margin). There is a moderate impact on weight of requiring 75% usable fraction available power (25% control margin), a large impact of requiring 55% fraction (Figure 14).

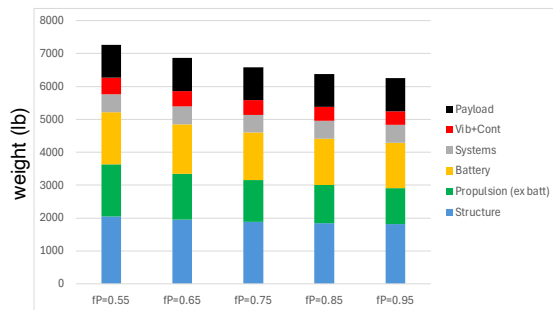


Figure 14. Hexacopter, rpm-controlled: — impact of electric motor torque margin.

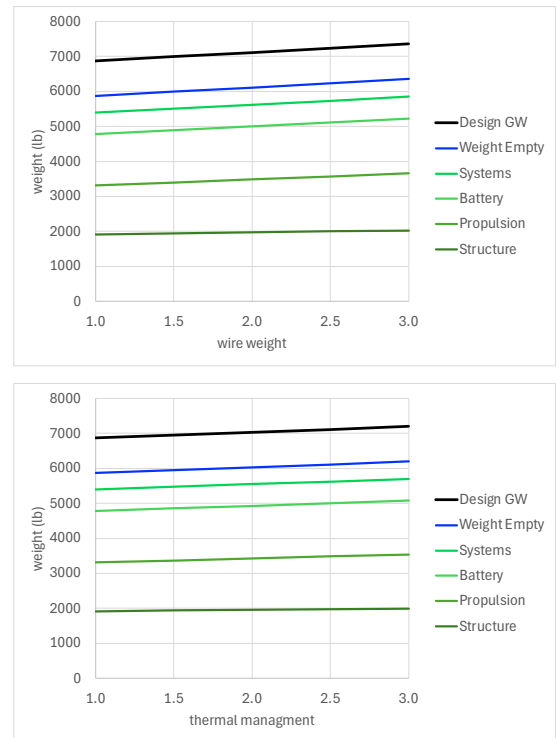


Figure 13. Side-by-side: impact of wire weight factor (top) and thermal management systems factor (bottom).

Battery

The baseline battery specific energy is 400 Wh/kg (usable installed). The specific energy is varied from 400 Wh/kg to about 200 Wh/kg (Figure 15). There is a very large impact of battery specific energy (Wh/kg) on the electric aircraft. The designs do not close for current state-of-the-art battery weights. The aircraft weight growth can be characterized by doubling of the design gross weight relative the baseline weight (Table 2)

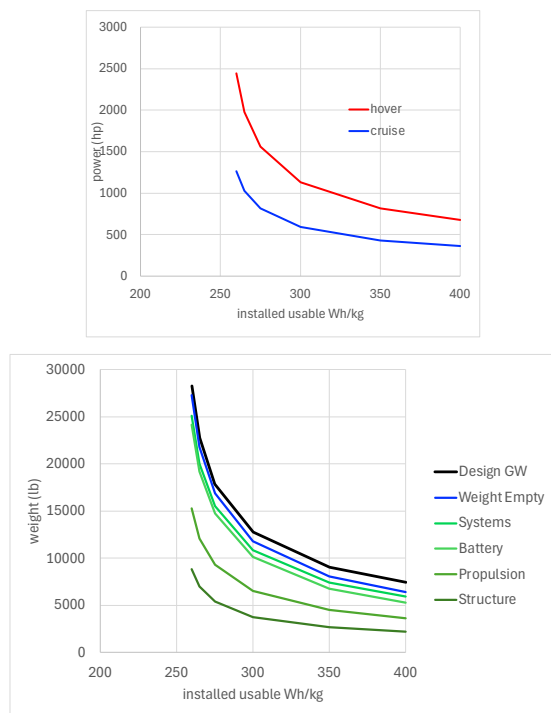


Figure 15. QSMR — impact of battery weight.

Table 2. Impact of battery weight — 75 nm range, relative 400 Wh/kg.

	DGW double	design not close below
quadrotor	250 Wh/kg	225 Wh/kg
QSMR	250 Wh/kg	260 Wh/kg
side-by-side	320 Wh/kg	310 Wh/kg
tiltrotor	275 Wh/kg	250 Wh/kg

MISSION

Range

The baseline design mission has a range of 75 nm (in two segments). The range is varied from 25 to 500 nm for the turboshaft configurations, and from 25 to 100-200 nm for the electric configurations (Figures 16–17). For turboshaft propulsion, there is a small impact of range up to 150 nm, a moderate impact at 500 nm. The fuel required increases with range of course, but even at 500 nm the aircraft size (weight and power) does not diverge. For electric propulsion, the battery weight is so large that increasing range drives

divergence of the designs. The achievable mission range with electric propulsion is sensitive to the battery specific energy, so the mission range is varied for a set of battery weights (400 Wh/kg to 150 Wh/kg) (Figures 18–19).

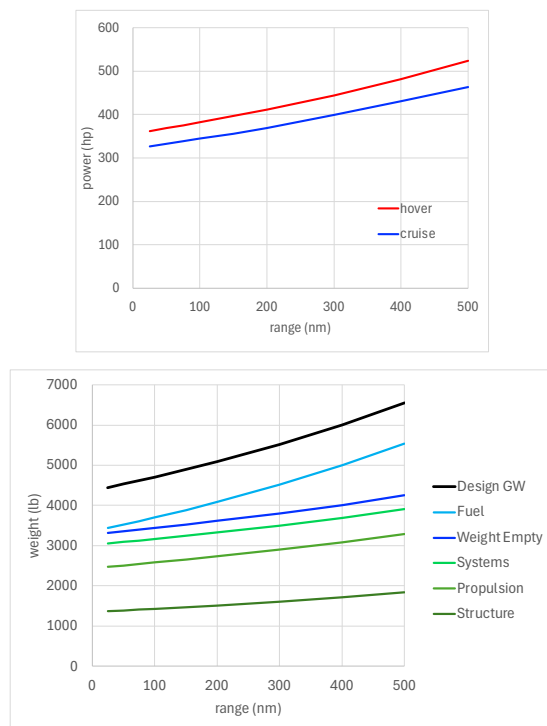


Figure 16. Quadrotor, turboshaft — impact of mission range.

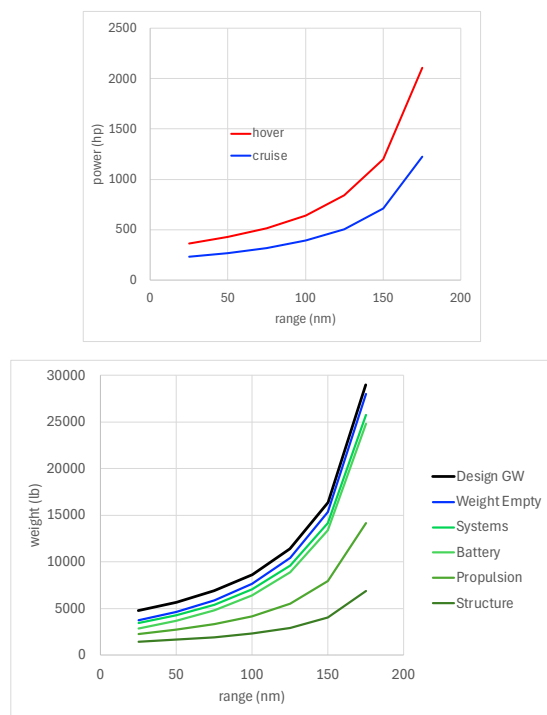


Figure 17. Quadrotor, electric — impact of mission range.

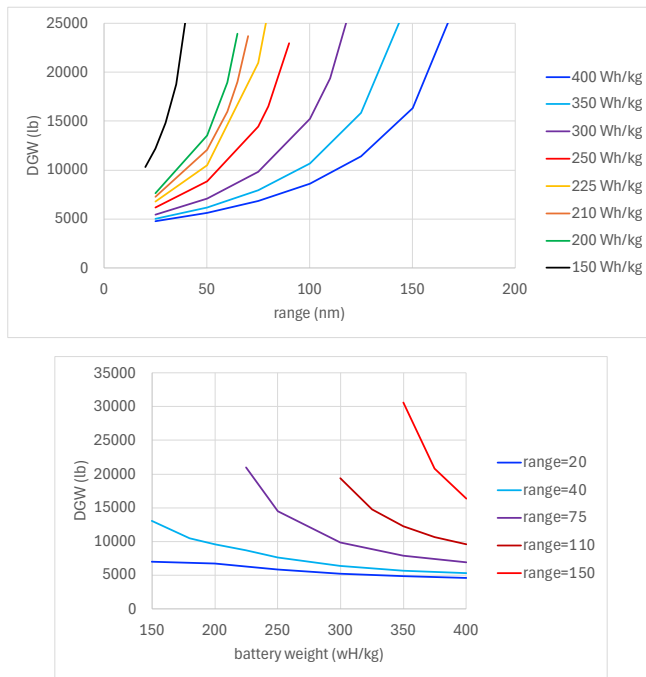


Figure 18. Quadrotor, electric — impact of battery weight and mission range on design gross weight.

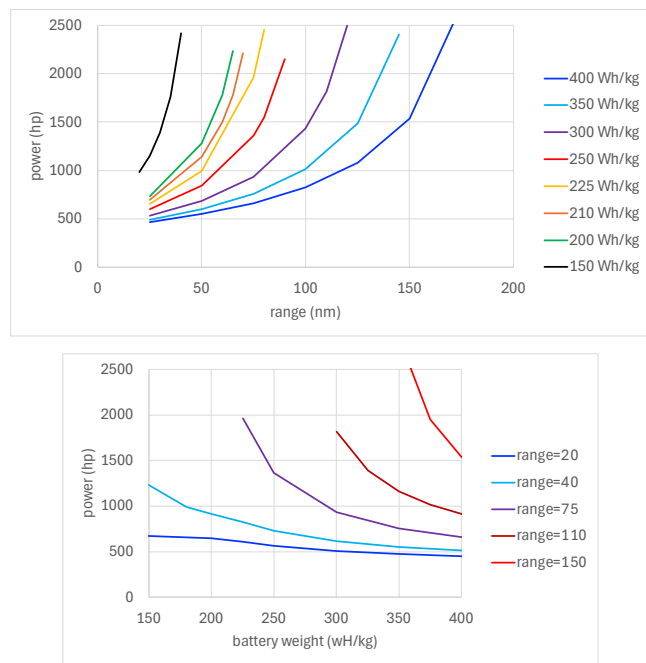


Figure 19. Quadrotor, electric — impact of battery weight and mission range on power

The aircraft weight growth can be characterized by doubling of the design gross weight relative the baseline weight. Battery specific weight and mission range are crucial trades for electric vehicle design.

Table 3. Impact of range —battery specific energy 400 Wh/kg, relative 25 nm range,

	DGW double	design not close above
quadrotor	120 nm	175 nm
QSMR	100 nm	130 nm
side-by-side	80 nm	105 nm
tiltrotor	125 nm	200 nm

Table 4. Impact of range —battery specific energy 200 Wh/kg, relative 25 nm range,

	DGW double	design not close above
quadrotor	55 nm	70 nm
QSMR	45 nm	45 nm
side-by-side	35 nm	35 nm
tiltrotor	40 nm	40 nm

Reserve

The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. Alternate requirements are 30 min for instrument flight rules (IFR) rotorcraft; or 30 min daytime VFR, 45 min nighttime or IFR for airplanes. The reserve segment time is varied from 0 to 45 min (Figure 20). For turboshaft propulsion, there is little impact. For electric propulsion, there is a moderate impact of 20 min reserve, and a large impact of 45 min reserve (for 75 nm range), because of battery weight growth.

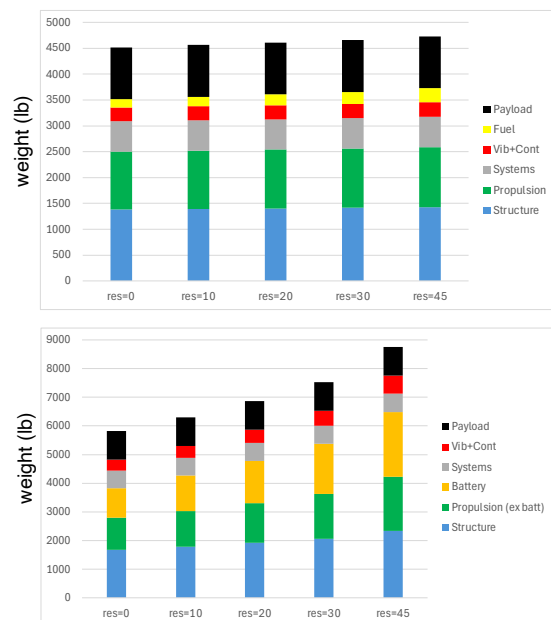


Figure 20. Quadrotor, turboshaft (top) and electric (bottom) — impact of mission reserve time.

Climb and one-engine-inoperative

The design mission requires climb to cruise altitude at 900 ft/min, and the design condition required one-engine-inoperative at DGW, hover, 6000 ft ISA. Typically one of these requirements sets the engine size. The rate of climb is varied from 900 ft/min to 200 ft/min (with even lower ROC, the range would be covered before the cruise altitude is reached); with and without the OEI requirement (Figures 21–22). There is little impact of rate-of-climb to cruise altitude with the one-engine-inoperative requirement, although 900 ft/min climb requires a bit more power than OEI. Without the OEI requirement, the engine is sized by climb for rates above about 500 ft/min (except for the tiltrotor). Generally there is a moderate impact of the one-engine-inoperative requirement.

Cruise climb

The design condition is for 500 ft/min climb at cruise altitude, best-range speed, and weight of the mission cruise segment. The cruise rate of climb is varied from 500 ft/min to 0 (no requirement) (Figure 23). There is little impact of the requirement for climb rate in cruise conditions, since this design requirement does not size the engine.

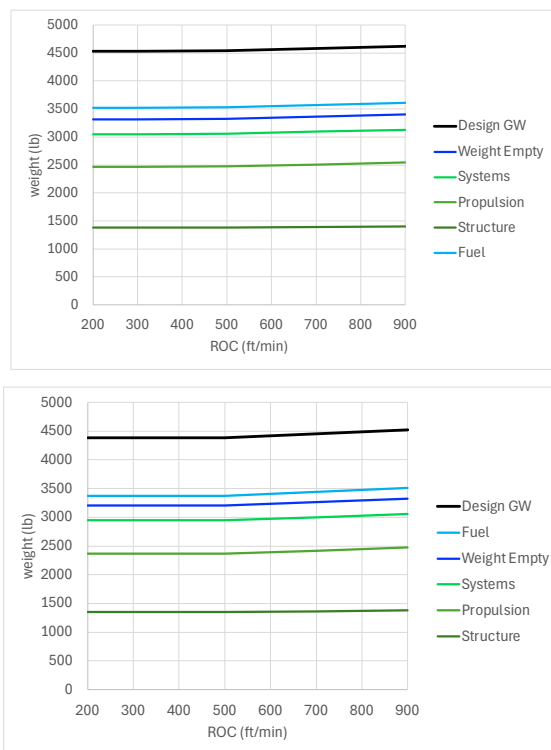


Figure 21. Quadrotor, turboshaft — impact of rate of climb to cruise altitude, with (top) and without (bottom) one-engine-inoperative requirement.

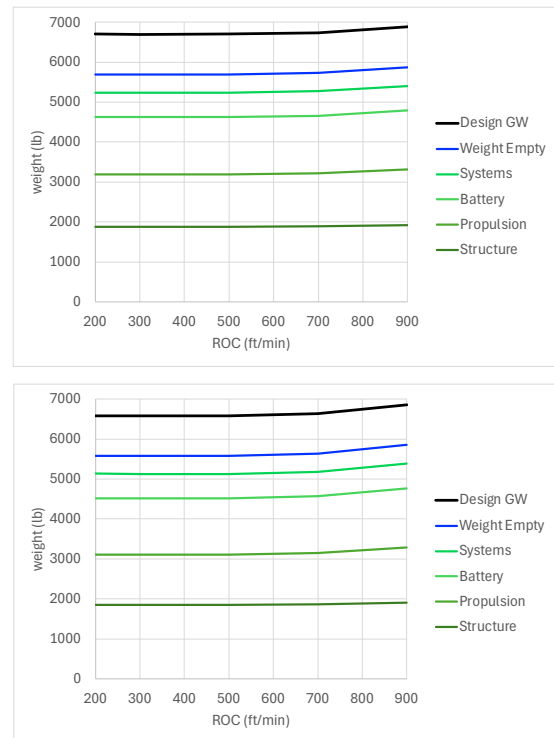


Figure 22. Quadrotor, electric — impact of rate of climb to cruise altitude, with (top) and without (bottom) one-engine-inoperative requirement.

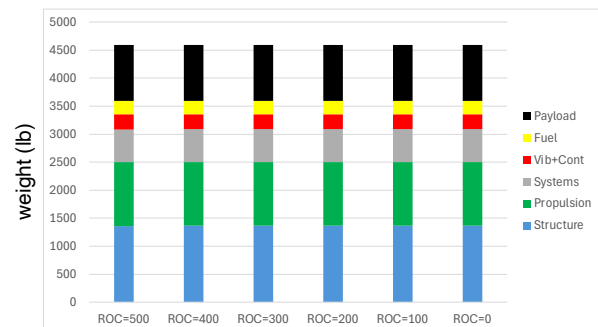


Figure 23. QSMR, turboshaft — impact of rate-of-climb at cruise altitude.

Altitude

The design mission has takeoff at altitude of 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA). As an alternative, takeoff at sea level ISA and cruise at 4000 ft ISA is considered; and designing the aircraft to accomplish both missions (Figure 24). There is a small impact of takeoff and cruise altitude.

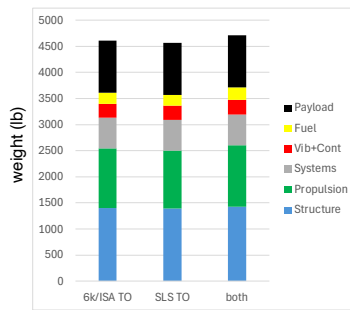


Figure 24. Quadrotor, turboshaft — impact of takeoff and cruise altitude.

Tiltrotor cruise speed

The mission has the aircraft cruise at the best-range speed to minimize the fuel or energy consumed, which is appropriate for the relatively short range requirement. Such a mission does not take advantage of the potential for high speed of the tiltrotor configuration. For a mission with 250 nm range, the cruise speed is varied from 200 knots to 300 knots (Figure 25). The rotor disk loading and wing loading are optimized for each design cruise speed, to minimize aircraft weight and power. For the tiltrotor with turboshaft propulsion, there is a small impact of cruise speed up to 250 knots, and a moderate impact at 300 knots.

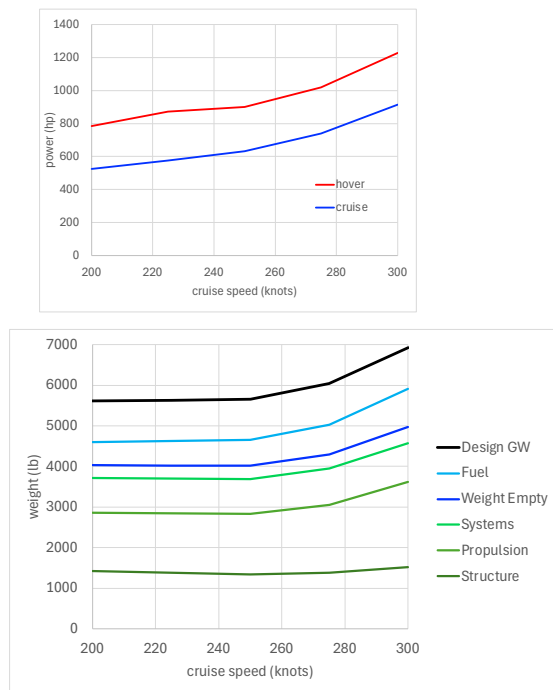


Figure 25. Tiltrotor, turboshaft — impact of cruise speed, for 250 nm range.

ALTERNATE DESIGNS

The vision of Advanced Air Mobility is driven by missions that will enable new transportation capabilities.

Hence it is appropriate to compare concept vehicles of different lift and propulsive architectures, all designed to accomplish the same UAM mission. It is also useful however to consider specific missions that can take advantage of the strengths of individual aircraft configurations. Alternate designs were developed for the four AAM concept vehicle configurations: quadrotor, QSMR, side-by-side, and tiltrotor; for both turboshaft and electric propulsion systems:

Turboshaft aircraft: Longer unrefueled range — 250 nm instead of 75 nm. Except for the tiltrotor, this range is accomplished in several hops with separate takeoff, cruise, and landing segments.

Tiltrotor: Faster cruise speed — 275 knots instead of cruise at fallout best-range speed. The entire cruise distance (250 nm) could be flown in a single segment of less than one hour duration.

Electric aircraft: Shorter range — 40 nm instead of 75 nm; and more realistic battery weight — 250 Wh/kg instead of 400 Wh/kg (installed usable).

Figure 26 shows key attributes of the eight aircraft (weight in lb, power in hp, speed in knots, ROC in ft/min).

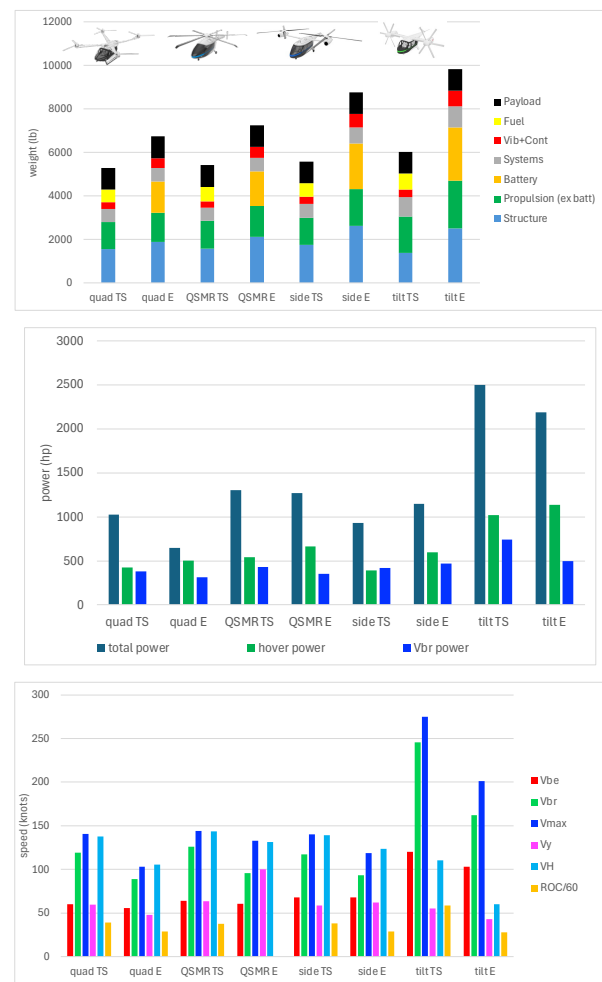


Figure 26. Alternate AAM Concept Vehicles.

CONCLUSIONS

This work has explored the impact of technology and mission variations on the NASA AAM concept aircraft: quadrotor, quiet single main rotor, side-by-side, and tiltrotor configurations, with turboshaft and electric propulsion variants for each. Generally the impact of technology and mission variations is consistent for these concept aircraft.

These results show a moderate impact of

- Systems and equipment weight, reflecting the weight growth factor
- One-engine-inoperative requirement
- For turboshaft: small impact of mission range up to 150 nm, moderate impact at 500 nm
- For electric: moderate impact of motor efficiency
- For rpm-control hexacopter: moderate impact of requiring 75% usable fraction of available power (25% control margin)
- For tiltrotor with turboshaft: small impact of cruise speed up to 250 knots, moderate impact at 300 knots

The short design range and endurance minimizes impact of efficiency in cruise.

Structural weight efficiency (represented in the design method by technology factors) is crucial for reasonable aircraft size, for all aircraft and propulsion configurations. Good design practice is required for all aircraft components. Technology advances (through materials and design) are important to minimize weight and power, hence cost.

Batteries are heavy, so any technology or mission variation that increases energy requirement has substantial impact on aircraft size, to the point where designs do not close for high range or low battery specific energy. Hence for the electric aircraft, there is a large impact of battery specific energy (Wh/kg), and mission range.

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NOTATION

AAM	Advanced Air Mobility
DGW	design gross weight
E	electric aircraft
ISA	international standard atmosphere
MCP	maximum continuous power
MRP	maximum rated power
OEI	one-engine inoperative
quad	quadrotor

QSMR	quiet single main rotor
ROC	rate of climb
sfc	specific fuel consumption
side	side-by-side aircraft
SL	sea-level
SLS	sea-level standard
tilt	tiltrotor
TO	takeoff
TS	turboshaft aircraft
UAM	Urban Air Mobility
VTOL	vertical take-off and landing
A	rotor disk area, πR^2
A_{blade}	total blade area
C_T	rotor thrust coefficient, $T/\rho A V_{tip}^2$
C_W	aircraft weight coefficient, $W/\rho A V_{tip}^2$
D/q	drag divided by dynamic pressure
GW	gross weight (operating weight+payload+fuel)
P	power
Q	torque
R	rotor blade radius
T	rotor thrust
V	speed
V_{be}	best endurance speed (maximum 1/fuel flow)
V_{br}	best range speed (99% high-side maximum V/fuel flow)
V_{max}	maximum speed (power required = 90% MCP)
V_{tip}	rotor tip speed
W	weight
WE	aircraft empty weight
WMT0	maximum takeoff weight
W_{pay}	payload weight
ρ	air density
σ	rotor solidity, A_{blade}/A