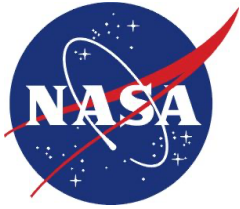


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Impact of Technology and Mission Variations on NASA Advanced Air Mobility Concept Vehicles

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June 2025

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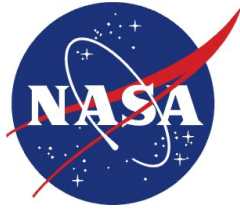
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Abstract

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. The technology excursions include: growth factor; weight technology factors (with and without weight reductions due to technology, including high and low calibrations); systems and equipment weight; airframe drag; rotor design blade loading; rotor profile power; turboshaft engine weight and specific fuel consumption; motor weight and efficiency; wire weight; thermal management systems design heat rejection (motor and battery); torque margin for rpm-controlled multicopter; and battery weight (installed usable Wh/kg). The mission excursions include range (together with battery weight for electric variants); reserve time; rate-of-climb to cruise altitude, with and without one-engine inoperative requirement; rate-of-climb at cruise; takeoff and cruise altitude; and tiltrotor cruise speed. 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. The NASA Revolutionary Vertical Lift Technology (RVLT) project 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. And 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 (but no hybrid propulsion architectures), plus NOTAR and tail rotor variants for the QSMR and a hexacopter for multicopter rpm control.



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.

The tools and workflow of the conceptual design process are illustrated in Figure 3. 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, and to produce blade airloads for acoustics. AARON/ANOPP2 (Refs. 20–22) is used for rotor noise evaluation.

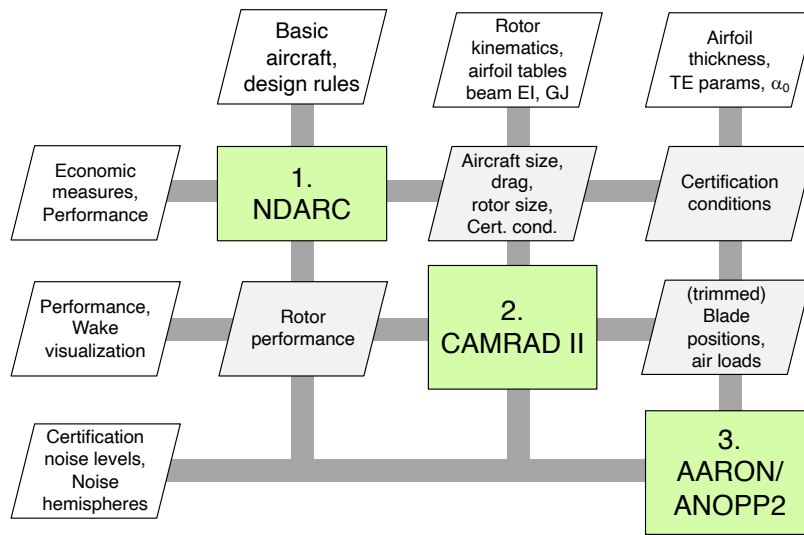


Figure 3. Tools and workflow of the conceptual design process.

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. The baseline designs incorporate unified 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:

- growth factor
- weight technology factors (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, with and without one-engine inoperative requirement
- rate-of-climb at cruise
- takeoff and cruise altitude
- tiltrotor cruise speed.

Design and Analysis Tools

The process and tools for development of the NASA concept vehicles are described in Reference 7. Figure 3 illustrates the central tools in terms of an eXtensible Design Structure Matrix (XDSM) diagram. The primary tools for performing physical calculations are on the diagonal of the matrix as green rectangles, inter-connections are gray lines, and data entities are parallelograms. Data transfer and inter-tool design process are managed by scripts written in Python. In general this toolchain can be integrated as part of a larger process. Overall inputs are in the white parallelograms at the top, and overall outputs are white parallelograms at the left. Guided design space exploration such as parameter sweeps, optimization drivers, vehicle type comparison may be implemented, along with other subsystem design (e.g. propulsion, flight control). Vehicle and subsystem design of higher fidelity could be added in order to verify earlier conceptual

design assumptions, or capture phenomena that are inadequately addressed in earlier steps of the conceptual design process.

Rotorcraft Sizing and Analysis NDARC

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.

Comprehensive Analysis CAMRAD II

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).

For interacting rotors, a freely convecting wake is necessary to accurately predict performance, as well as blade-vortex-interaction airloads that are responsible for significant noise. For example, quadrotor and side-by-side aircraft have rotors that interact to varying degrees in various stages of flight. Similarly, for practical UAM rotors larger than a couple of feet in radius, elastic and kinematic motion should be considered to calculate the rotor trim and blade-vortex interactions. Blade airloads and velocities will in general depend on blade motion, induced velocity, and wake convection. Whether or not the aerodynamic model of a comprehensive analysis code is adequate for evaluating the noise sources, a comprehensive analysis model of the structure is required. Building a comprehensive analysis model is generally a necessary step in the development of a meaningful high-fidelity simulation, as vehicle trim, rotor trim, and blade kinematic and elastic motion are usually computed by a comprehensive analysis that is coupled to CFD airloads calculations.

CAMRAD II can be used in the optimization of the rotor geometry for the aircraft sizing conditions, and to develop calibrated rotor performance models for NDARC; as well as to calculate rotor and hub loads for structural design, assess rotor and blade stability and whirl flutter, and provide rotor blade airloads for mid-fidelity acoustics calculations. The performance calculations for calibration of the NDARC rotor models considered first an isolated rotor, in particular to define profile power including the influence of stall. Then calculations for the multi-rotor system were used to calibrate the rotor-rotor interference effects on induced and profile power. Other applications of CAMRAD II (load, stability, airloads calculations) were not conducted for the present work.

Acoustic Analysis AARON/ANOPP2

In order to predict the acoustic metrics, a tool that predicts thickness, loading, and broadband noise, acoustic propagation, and observer noise is necessary. The Aircraft Noise Prediction Program 2 (ANOPP2, Refs. 20–21) and AeroAcoustic Rotor Noise (AARON) tools provide the acoustic calculations in the toolchain. AARON is the user code to perform rotorcraft calculations with ANOPP2. A Python script provides a simplified interface that is geared toward generating the certification noise and other typical rotorcraft calculations with a manageably small number of inputs. AARON/ANOPP2 calculates the loading noise with a compact loading model, using the blade section loading and motion as well as the aircraft operating condition from the comprehensive analysis. Consistently, thickness noise is calculated with a compact monopole model. An airfoil self-noise model extended to rotating blades is used for broadband noise (BPM model, Ref. 22). AARON/ANOPP2 can also calculate the loading and rotational noise from a higher-fidelity solution for the blade surface pressures and geometry, such as from RANS CFD calculations. The present work relies on previous applications of the acoustic analysis for design choices.

Baseline Aircraft Designs

The baseline aircraft designs were developed starting from the original NDARC models (Refs. 5, 6, 10). The updated 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. The technology level and systems and equipment were assessed and updated. In particular, the baseline designs incorporate unified 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).

The CAMRAD II models were constructed from geometry, weight, and operating condition information extracted from the sized NDARC designs. The rotor articulation (flap-lag hinges, or hingeless; characterized by blade flap frequency from NDARC), control system geometry, and blade tip taper/sweep/droop were fixed. The aerodynamic analysis used appropriate airfoil tables (described below). Jobs were constructed for full aircraft (with appropriate trim and control schemes) and single rotor (for performance calibration), for hover and cruise conditions from NDARC.

CAMRAD II sweeps of blade twist and taper were conducted, and the optimum geometry identified based on aircraft size (weight and power) from NDARC. With this optimum geometry, single rotor power was calculated for hover (thrust sweep) and forward flight (speed sweeps at several thrust levels). The induced power factor and mean drag coefficient from these calculations were used to construct a calibrated rotor performance model for NDARC. Power calculations for the full aircraft were used to calibrate the NDARC model for rotor/rotor interference.

Using this calibrated rotor performance model, the aircraft was re-sized. The disk loading was optimized, based on aircraft size (weight and power). If the resulting aircraft geometry and operating conditions were sufficiently different from the starting designs, the cycle was repeated: the tasks of rotor twist and taper optimization (CAMRAD II), rotor performance modeling (CAMRAD II), and aircraft re-sizing (NDARC) were repeated.

Descriptions of the baseline aircraft were extracted from the NDARC results, and the corresponding CAMRAD II models were updated. These eight designs — quadrotor, QSMR, side-by-side, and tiltrotor, with turboshaft and electric propulsion variants — are the foundation for examining the impact of technology and mission variations.

UAM Design Requirements

Following an initial air taxi vehicle study, which explored vehicle technology themes using aircraft of various sizes designed for several candidate missions, the RVLT project performed a focused study to better understand a particular urban air mobility market. A design mission was developed accounting for the existing geography, population patterns, infrastructure, and weather in twenty-eight markets across the United States of America (Ref. 3). The resulting mission is to carry six passengers (including the pilot, if not autonomous; 1200 lb payload) on two 37.5-nm flights (total 75-nm range without recharging or refueling), with a 20 min reserve (Figure 4 and Table 1). Vertical takeoff altitude is 6000-ft (ISA), and cruise is at best-range speed, 4000-ft above ground level (AGL). This mission is intended to be used as a sizing requirement. The actual operational missions flown by the aircraft will be different, driven by economics, air traffic, and other aspects of a particular flight.

After considering various cabin layouts, it was concluded that a practical load is 5 occupants (including any pilot), with space for baggage in the cabin. At 200 lb per occupant, the design payload is thus 1000 lb. The reserve requirement is 20 min flight after the last cruise segment, corresponding to visual flight rules (VFR) rotorcraft operations. The design mission sizes the gross weight, power, transmission, and fuel tank. The first design condition is cruise climb, and the second design condition is one-engine-inoperative flight. These design conditions size the power and transmission.

In order to consistently compare the designs, all aircraft are sized to this mission (Figure 4 and Table 1). Aircraft sized for flight ranges separately appropriate for turboshaft and electric propulsion will also be presented.

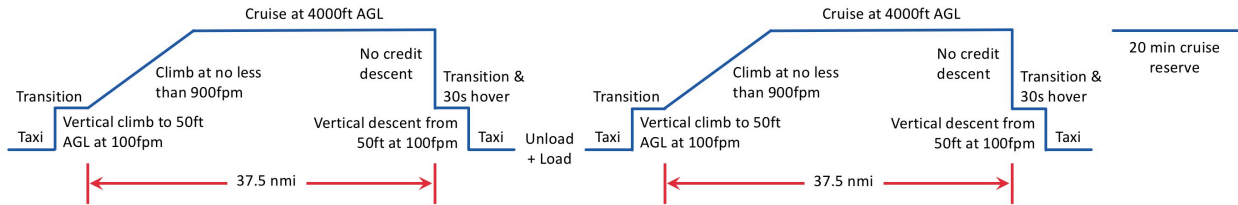


Figure 4. UAM design mission.

Table 1. UAM design mission and conditions.

| | | weight | distance | time | altitude ft / ISA | speed | ROC ft/min | power available | tiltrotor tilt/V _{tip} |
|--|------------------|-----------|----------|--------|----------------------|------------|---------------|--------------------|------------------------------------|
| design mission — 5 occupants (1000 lb payload) | | | | | | | | | |
| 1 | taxi | DGW | | 15 sec | 6000 | 0 | | 25% MRP | 90/450 |
| 2 | takeoff | | | | | vertical | 100 | 95% MRP | 90/450 |
| 3 | transition | | | 10 sec | 6050 | hover | | 95% MRP | 90/450 |
| 4 | climb | | | | | best range | 900 | 95% MRP | 0/450 |
| 5 | cruise | | 37.5 nm | | 10000 | best range | | 95% MCP | 0/225 |
| 6 | transition | | | 10 sec | 6050 | hover | | 95% MRP | 90/450 |
| 7 | hover | | | 30 sec | 6050 | hover | | 95% MRP | 90/450 |
| 8 | descent | | | | | vertical | -100 | 95% MRP | 90/450 |
| 9 | landing | | | 15 sec | 6000 | 0 | | 25% MRP | 90/450 |
| 10 | taxi | | | 15 sec | 6000 | 0 | | 25% MRP | 90/450 |
| 11 | takeoff | | | | | vertical | 100 | 95% MRP | 90/450 |
| 12 | transition | | | 10 sec | 6050 | hover | | 95% MRP | 90/450 |
| 13 | climb | | | | | best range | 900 | 95% MRP | 0/450 |
| 14 | cruise | | 37.5 nm | | 10000 | best range | | 95% MCP | 0/225 |
| 15 | reserve | | | 20 min | 10000 | best end | | 95% MCP | 90/450 |
| 16 | transition | | | 10 sec | 6050 | hover | | 95% MRP | 90/450 |
| 17 | hover | | | 30 sec | 6050 | hover | | 95% MRP | 90/450 |
| 18 | descent | | | | | vertical | -100 | 95% MRP | 90/450 |
| 19 | landing | | | 15 sec | 6000 | 0 | | 25% MRP | 90/450 |
| design conditions | | | | | | | | | |
| 1 | cruise climb | segment 5 | | | 10000 | best range | 500 | 100% MRP | 0/225 |
| 2 | OEI | DGW | | | 6000 | hover | | 100% CRP | 90/450 |
| off-design conditions | | | | | | | | | |
| 1 | WMTO | maximum | | | 6000 | hover | | 95% MRP | 90/450 |
| 2 | hover | DGW | | | 6000 | hover | | 95% MRP | 90/450 |
| 3 | V _{br} | DGW | | | 6000 | best range | | 100% MCP | 0/225 |
| 4 | V _{be} | DGW | | | 6000 | best end | | 100% MCP | 0/225 |
| 5 | V _{max} | DGW | | | 6000 | maximum | | 90% MCP | 0/225 |
| off-design conditions — noise certification | | | | | | | | | |
| 6 | V _y | WMTO | | | SL/77°F | best climb | best rate | 100% MRP | 90/450 |
| 7 | V _H | WMTO | | | SL/77°F | maximum | | 100% MCP | 90/450 |

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 (Table 2). 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 XV-15 (though it was not a production design) and the V-22 (though it is not small). The calibration factors used for the present designs are given in Table 3. 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). Table 3 also gives technology factors, which include the impact of design approach and advanced materials. The NDARC input technology factors (parameters TECH_xxxx) are the product of the calibration and technology factors in Table 3.

The weight equations model separately non-boosted controls (which do not see aerodynamic surface or rotor loads), boost mechanisms (actuators), and boosted controls (which are affected by aerodynamic surface or rotor loads); see Ref. 16. The load path goes from pilot, to cockpit controls, to non-boosted controls, to boost mechanisms, to boosted controls, and finally to the component. The parametric equations are developed from weight data for hydraulic-powered actuators. It is assumed that the sum of boost mechanisms and hydraulics weights (with the same values for technology factors) is also a reasonable estimate for the weights of electrical actuation systems.

Allocations for System and Equipment Group weights are based on current helicopters. The weights are given in Table 4. Also shown are the technology factors that have been included in these weights. The total weight is 480 lb.

Drag

Table 5 summarizes the aircraft drag buildup, 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. The fuselage geometry is given in Table 6, for the configurations with tail and for the quadrotor (without tail). The drag of the fixed landing gear is $D/q = 0.35 \text{ ft}^2$. The rotors have faired hubs and low-drag pylons. The quadrotor hub drag coefficient includes the rotor support arms (so the support arm D/q scales with rotor radius squared). The side-by-side rotor support (wing) is a faired structure that does not generate lift. 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.

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. 23). 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. 23).

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. 24). 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. 25; 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 aircraft trim scheme obtains zero net force and moment on the aircraft by adjusting aircraft controls (collective, lateral cyclic, longitudinal cyclic, and pedal, connected to rotors and fixed wing controls as appropriate for the rotorcraft type) and aircraft attitude (pitch and roll angles). For symmetric aircraft, trim can use only the longitudinal loads and controls.

The quadrotor has collective control, cross-shafting, and flapping rotors. Rotor speed control is examined as an excursion (the fixed collective pitch value is based on the hover performance with collective control and design tip speed). The quadrotor hubs are at lateral and longitudinal stations $\pm 1.35R$ (35% separation, arm length 1.91R). The rear rotors are 0.35R above the front rotors, for improved performance and noise levels (reduced rotor/rotor interference). The center of gravity is 0.8 ft forward (relative the mid-point between rotors), for further performance improvement in forward flight trim. 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/ft^2 , 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. 26-27). The weight model is

$$W = 255. + 0.0484P + 0.00000353P^2$$

for weight in lb from the power P (MRP) in hp. Table 7 gives the performance characteristics, including IRP (30 min) and MRP (10 min) ratings. The power values scale with the engine design power (MRP). The mechanical limit is actually a torque limit, at the specification speed 15000 rpm. Specific fuel consumption decreases with increased power. At the design size (660 hp MRP), the engine weight is 288 lb (0.44 lb/hp).

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, hence have a transmission from the motor to 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). This regression equation is based on 64 motors, from 20–550 hp, 100–10000 rpm, with average error of 25% for high torque-to-weight motors. The second term is the weight of the electronic speed control system. There is a separate model for the thermal management system. Table 8 gives the performance characteristics, including IRP (30 min) and MRP (10 min) ratings. The power values scale with the engine design power (MRP). The peak power is actually a torque limit, at the specification speed 6000 rpm. The base speed is where the power limit and torque limit intersect. 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. 28). This margin is implemented by limiting the power available to 75% (instead of 95%) in all design mission segments and flight conditions (Table 1).

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 discharge characteristics are shown in Figure 5. The internal resistance reduces efficiency at high discharge rates. Current delivery limits for cells are specified as a C-rate (capacity/hr). The battery model uses these curves (V/V_{ref} 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 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. 29). 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.

Cost

Purchase cost of aircraft is roughly (to about 20% accuracy) driven by aircraft empty weight, installed power, and complexity, plus the costs of electronic systems. For electric propulsion, the cost of batteries are explicitly included in the purchase cost estimate. Data are available for maintenance cost of helicopters flying traditional missions, but not for unconventional aircraft types engaged in air taxi operations. A significant component of operating costs is the cost of fuel or energy. If the mission range is small enough so that electric propulsion is feasible, energy costs are generally smaller for the all-electric propulsion configuration, even though the aircraft weight is larger.

A method to estimate the purchase price of rotary wing aircraft was developed by Harris and Scully (Ref. 30), revised and extended by Scott (Ref. 31–32). The method gives the price within 20%, as $\text{Price} = K(\text{SF})$, with the size factor $\text{SF} = \text{WE}^{0.4638} P^{0.5945}$. Here WE is the weight empty and P is the installed power; the factor K depends on the aircraft type, and some measures of complexity. The equation $\text{Price} = K(\text{SF})$ is applicable to helicopters, tiltrotors, general aviation, and airlines (Ref. 31), and to commercial turboprops and unmanned aircraft (Ref. 32). The size factor SF appears as a universal scaling parameter for aircraft cost (Figure 6). The NASA concept vehicles have $\text{SF} = 1500$ to 6000 , which would imply a unit flyaway cost in the range \$3M to \$10M following the data in Figure 6. However, the UAM industry is projecting prices an order of magnitude less than existing rotorcraft and turboprops when high-rate manufacturing is employed and larger numbers of UAM vehicles are produced. There is no data yet to evaluate the factor K appropriate for UAM aircraft, but it is reasonable to use this size factor to estimate the relative costs of the concept vehicles.

Based on 2024 data, operating cost is calculated using fuel price of \$2.31 per gallon for jet A, and \$0.0400 per kWh for electricity. Battery cost is \$50 per kWh, corresponding to battery purchases within the next decade. Emissions trading scheme cost is \$0.065 per kg CO₂. For electric propulsion, the power is reduced by a factor of 0.5 in the size factor. Cost of mission equipment and flight control electronics is \$5000 per lb.

Table 2. Small rotorcraft for weight model calibration.

| Aircraft | crew | pass | empty weight lb | maximum weight lb | rotor diameter ft | |
|---------------------------|------|------|--------------------|----------------------|----------------------|---|
| Bell B412 | 1 | 13 | 6789 | 11900 | 46 |  |
| Sikorsky S-76C++ | 2 | 13 | 7005 | 11700 | 44 |  |
| Eurocopter SA-365 Dauphin | 2 | 10 | 6896 | 9480 | 39 |  |
| AgustaWestland A109 | 1 | 7 | 3505 | 6283 | 36.09 |  |
| MBB Bo-105 | 1 | 4 | 2813 | 5512 | 26.4 |  |
| Bell OH-58D | 1 | 6 | 3829 | 5500 | 35 |  |
| Eurocopter AS350 | 1 | 5 | 2588 | 4960 | 35.1 |  |
| McDonnell Douglas MD500M | 2 | 3 | 2550 | 3000 | 27.33 |  |
| NASA/Army/Navy/Bell XV-15 | 2 | | 570 | 15000 | 12.5 |  |

Table 3. Technology factors for weight models

| | calibration | | technology |
|------------------------------|----------------------|---------|------------|
| | mean | std dev | |
| Fuselage | | | |
| basic | 0.994 | 0.069 | 0.80 |
| crashworthiness | 1.000 | | 0.90 |
| crash factor | 15% (6% tilt) | | |
| wire strike | 25 lb | | 0.80 |
| Landing Gear | 0.853 | 0.081 | 1.00 |
| Rotor | | | |
| blade | 0.954 | 0.083 | 0.95 |
| hub and hinge | 1.023 | 0.101 | 0.80 |
| auxiliary thrust | 1.009 | 0.128 | 1.00 |
| tail rotor | 1.303 | 0.109 | 0.85 |
| Wing | 0.953 | 0.100 | 0.80 |
| Tail | | | |
| horizontal | 1.224 | 0.204 | 0.85 |
| vertical | 0.976 | | 0.85 |
| Fuel Tank | | | |
| tank | 0.829 | 0.053 | 0.85 |
| plumbing factors | 80 & 2.0 | | 0.25 |
| Propulsion Group | | | |
| gear box | 1.055 | 0.090 | 0.75 |
| rotor shaft | 1.055 | 0.090 | 0.75 |
| drive shaft | 0.982 | 0.249 | 0.75 |
| rotor brake | 1.234 | 1.071 | 0.95 |
| gearbox loss | 2% | | |
| Engine Group | | | |
| cowling | 0.923 | 0.084 | 0.65 |
| pylon | 1.000 | | 0.85 |
| support | 1.482 | 0.010 | 0.65 |
| air induction | 1.482 | 0.010 | 0.65 |
| accessories | 0.959 | 0.103 | 0.50 |
| momentum drag | 75% | | 0.90 |
| exhaust loss | 2% | | |
| exhaust system | 2.5%P | | |
| accessory power | 20 hp | | |
| Engine quieting | 40 lb/eng | | 0.625 |
| Flight controls | | | |
| rotary wing | | | |
| boosted | 0.859 | 0.301 | 0.85 |
| boost mechanisms | 0.988 | 0.135 | 0.85 |
| non-boosted | 0.959 | 0.186 | 0.40 |
| rotary wing, collective only | | | |
| boosted | 100% | | |
| boost mechanisms | 50% | | |
| non-boosted | 33% | | |
| fixed wing | | | |
| boost mechanisms | 0.604 | 0.099 | 0.85 |
| non-boosted | 0.503 | 0.002 | 0.40 |
| conversion | | | |
| boost mechanisms | 1.000 | | 0.85 |
| non-boosted | 1.000 | | 0.40 |
| hydraulics | | | |
| rotary wing fraction | 82%mb | 0.040 | |
| fixed wing fraction | 70%mb | | |
| conversion fraction | 44%mb | | |
| Trapped fluids | 10 lb (TS), 5 lb (E) | | |

Table 4. Systems and Equipment Group weights ($N_{\text{pass}} = 5$ occupants).

| | | tech factor |
|---|--|-------------|
| furnishings and equipment group seats (25) + other | $30 * N_{\text{pass}} = 150 \text{ lb}$ | 1.00 |
| flight control group and hydraulic group | | |
| cockpit controls | 10 lb | 0.70 |
| automatic flight control system | 40 lb | 0.65 |
| environmental control group | $15 * N_{\text{pass}} = 75 \text{ lb}$ | 0.90 |
| air conditioning, bleed air heating | | |
| instruments group | 20 lb | 0.65 |
| indicators, transmitters and amplifiers, installation; sensors | | |
| avionics group | 110 lb | 0.50 |
| electronics and sensors, equipment and installation | | |
| electrical group | $50 + 5 * N_{\text{pass}} = 75 \text{ lb}$ | 0.60 |
| power supply, conversion, distribution; lights; occupant electrical | | |

Table 5. Aircraft component drag.

| component | reference area | drag coefficient | |
|------------------------|----------------|---------------------------|----------------------|
| fuselage | wetted | 0.0045 | low drag fuselage |
| vertical | projected | 0.40 | for download |
| landing gear | | $D/q = 0.35 \text{ ft}^2$ | fixed, faired |
| rotor hub | rotor disk | 0.0015 | faired hub |
| quadrotor | rotor disk | 0.0030 | support arm |
| tiltrotor | rotor disk | 0 | |
| tail rotor | rotor disk | 0.0030 | |
| pylon | pylon wetted | 0.0150 | low drag pylon |
| tiltrotor | pylon wetted | 0.0070 | |
| tiltrotor vertical | pylon wetted | 0.30 | helicopter mode drag |
| tiltrotor spinner | spinner wetted | 0.0150 | |
| wing | wing | 0.0090 | tiltrotor |
| wing-body interference | wing | 0.0020 | tiltrotor |
| vertical | wing | 2.0 | for download |
| side-by-side | wing | 0.0500 | |
| horizontal tail | tail | 0.0110 | |
| vertical tail | tail | 0.0150 | |

Table 6. Fuselage geometry.

| | | with tail | quadrotor |
|----------------|-----------------|-----------|-----------|
| length | ft | 30.0 | 21.3 |
| width | ft | 6.85 | 6.85 |
| height | ft | 6.5 | 6.5 |
| wetted area | ft ² | 382 | 354 |
| projected area | ft ² | 158 | 140 |

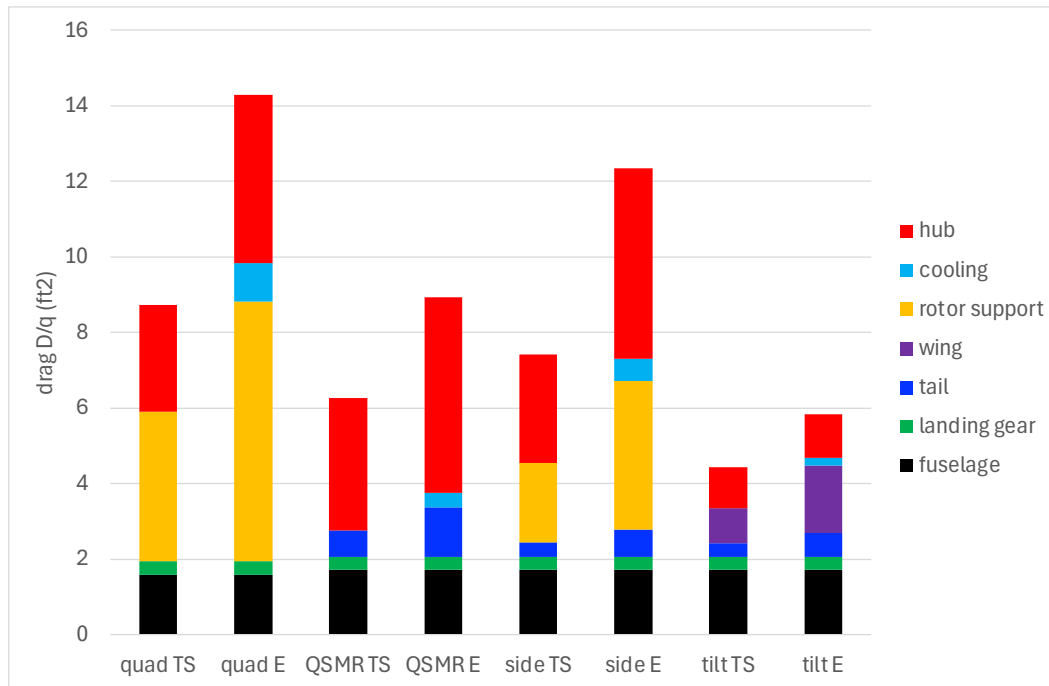


Figure 4. Drag buildup for baseline aircraft.

Table 7. Turboshaft engine performance.

| rating | power P_0 (hp) | mechanical limit (hp) | engine speed (rpm) | sfc_{OC} (lb/hp-hr) |
|--------|------------------|-----------------------|--------------------|-----------------------|
| MCP | 574 | 830 | 15000 | 0.4995 |
| IRP | 602 | 830 | 15000 | 0.4995 |
| MRP | 660 | 830 | 15000 | 0.4995 |
| CRP | 723 | 830 | 15000 | 0.4995 |

Table 8. Electric motor performance.

| rating | power (hp) | peak power (hp) | base speed (rpm) | efficiency |
|--------|------------|-----------------|------------------|------------|
| MCP | 320 | 500 | 3840 | 95% |
| IRP | 400 | 500 | 4800 | 95% |
| MRP | 480 | 500 | 5760 | 95% |
| CRP | 500 | 500 | 6000 | 95% |

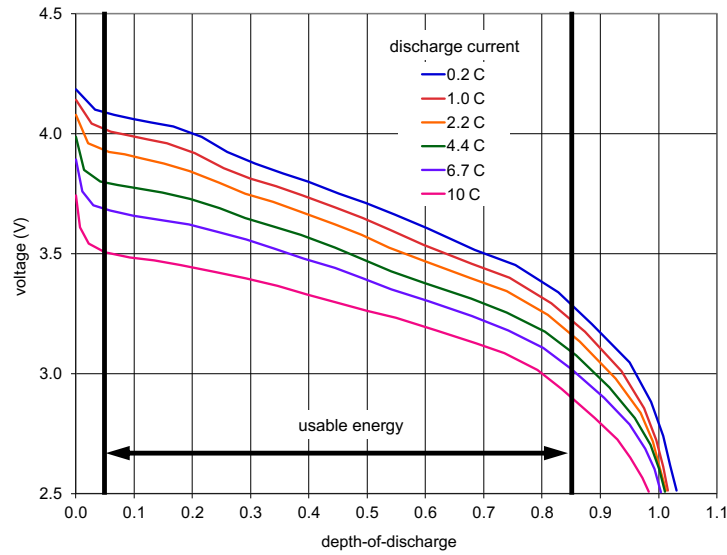


Figure 5. Battery discharge characteristics (Molicel INR-21700-P45B Li-ion battery).

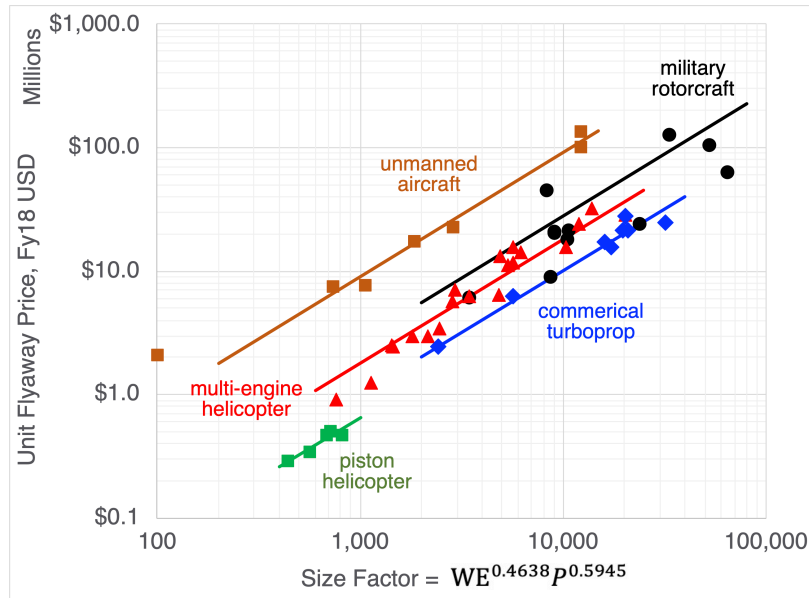


Figure 6. Unit flyaway price for several classes of rotary wing and propeller aircraft (Ref. 32).

Baseline Designs of AAM Concept Vehicles

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 7 shows key attributes of the eight aircraft (weight in lb, power in hp, speed in knots, ROC in ft/min). Table 9 gives the size, weight, performance, and mission characteristics; and Table 10 gives the weight statements at the group level.

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:

| | 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 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.

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.

Rotor-Speed Controlled Multicopter

The quadrotor is the representative multicopter configuration for this investigation. The baseline quadrotor designs have collective control and transmission shafts connecting all rotors and engines. For turboshaft propulsion, the two engines are in the fuselage, with shafts from the engines to each rotor. For electric propulsion, there are four motors, each at the end of the rotor support arm connected to the rotor, with inter-connect shafts along the support arms to a central gearbox on the fuselage. Hence with one-engine inoperative, power and control can be maintained to all four rotors. One-engine inoperative on the electric quadrotor means losing one of the four motors, hence less total installed power is required compared to the turboshaft quadrotor.

A major configuration variant for multicopters is the use of rotor-speed control of the rotor thrust, instead of collective pitch control. With fixed-pitch rotors, the pitch control system (about 100 lb) and inter-connect shafts (250–400 lb) are absent, reducing the aircraft weight substantially (growth factor of 3.0). However, with rpm-controlled rotors, a torque margin of 25% is needed for good response to control in gusts and maneuvers (Ref. 28), which results in motors sized for larger power and torque capability. In the one-engine inoperative state of a multicopter using rotor-speed control, equilibrium trim (zero net force and moment on the aircraft) requires reducing thrust and rpm on an opposite rotor, still using it for control but not for aircraft lift. Consequently, the quadrotor is not a viable configuration for rotor-speed control, as control and equilibrium flight can not be maintained with just three of the four rotors. Instead, the hexacopter configuration must be considered.

Figure 8 shows the weight (lb) and power (hp) for electric multicopters with collective or rotor-speed control, both quadrotor and hexacopter configurations. For reference, the turboshaft aircraft are also shown. The quadrotor aircraft with rpm-control is shown, since it can be sized in NDARC (with power lost from just one motor in the OEI condition), even though it can not be trimmed or flown.

The hexacopter has less weight but more power than the corresponding quadrotor aircraft. The influence of the torque margin requirement will be examined considering the hexacopter, but otherwise it is assumed that the quadrotor is representative of multicopters.

QSMR with NOTAR Anti-Torque System

The original design of the quiet single main rotor (QSMR) helicopter uses the NOTAR anti-torque system (the NDARC model of the NOTAR is described in Ref. 6). For this investigation, the baseline design has a tail rotor for anti-torque and yaw control, with a low tip-speed so it is assumed that it produces less noise than the main rotor.

Figure 9 shows the weight (lb) and power (hp) for the turboshaft QSMR designs, with NOTAR and tail rotor for anti-torque. The NOTAR design has about the same power, and bit more weight than the tail-rotor design. Only the tail-rotor configuration is considered further in this investigation, so it is not necessary to deal with any potential issues of the NOTAR design, particularly regarding acoustics and handling qualities.

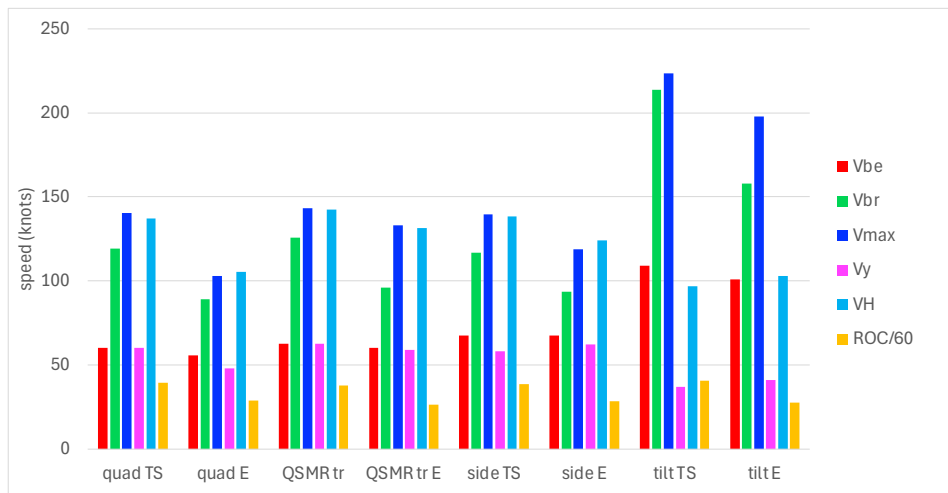
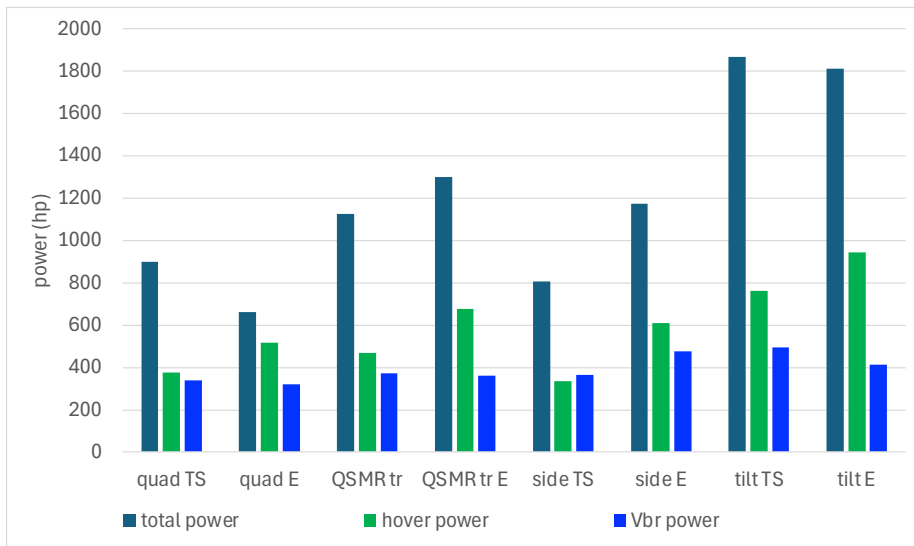
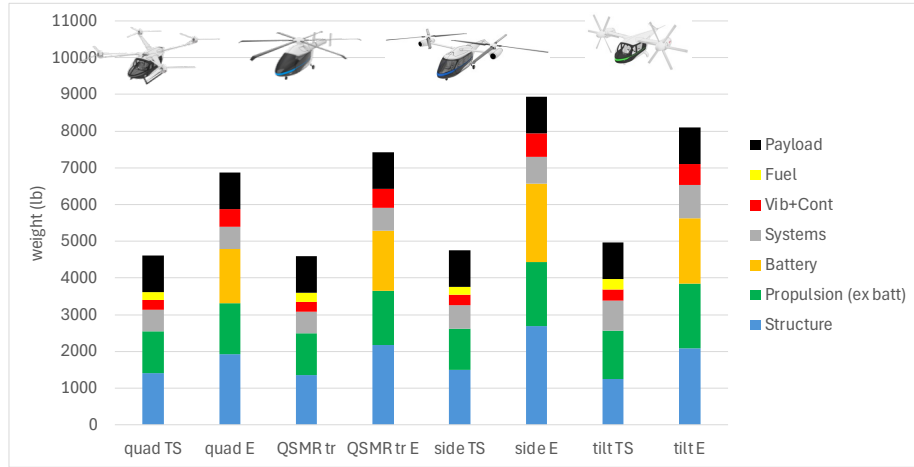


Figure 7. Baseline AAM Concept Vehicles: key attributes.

Table 9. Baseline AAM Concept Vehicles: size, weight, performance, and mission characteristics.

| | quad TS | quad E | QSMR | QSMR E | side TS | side E | tilt TS | tilt E |
|---------------------------------------|---------|--------|--------|--------|---------|--------|---------|--------|
| Size | | | | | | | | |
| number main rotors | 4 | 4 | 1 | 1 | 2 | 2 | 2 | 2 |
| disk loading lb/ft ² | 3.5 | 3 | 3.5 | 3 | 3.5 | 3.5 | 12 | 8 |
| design C_w/σ | 0.09 | 0.09 | 0.09 | 0.09 | 0.09 | 0.09 | 0.13 | 0.13 |
| blade radius ft | 10.25 | 13.51 | 20.46 | 28.07 | 14.73 | 20.17 | 8.13 | 12.70 |
| solidity (T-weighted) | 0.097 | 0.083 | 0.097 | 0.083 | 0.097 | 0.097 | 0.229 | 0.153 |
| number blades | 5 | 5 | 6 | 6 | 4 | 4 | 6 | 6 |
| tip speed ft/sec | 450 | 450 | 450 | 450 | 450 | 450 | 450 | 450 |
| number engines | 2 | 4 | 2 | 2 | 2 | 2 | 2 | 2 |
| total power MRP SLS hp | 899 | 662 | 1123 | 1299 | 805 | 1173 | 1868 | 1810 |
| power MCP SLS hp | 782 | 441 | 977 | 866 | 700 | 782 | 1625 | 1207 |
| sfc MCP SLS lb/hp-hr | 0.50 | | 0.50 | | 0.50 | | 0.49 | |
| engine weight lb | 555 | 219 | 567 | 323 | 550 | 295 | 607 | 434 |
| engine weight/power lb/hp | 0.62 | 0.33 | 0.50 | 0.25 | 0.68 | 0.25 | 0.32 | 0.24 |
| drive limit hp | 505 | 606 | 532 | 680 | 539 | 858 | 1145 | 1250 |
| fuel tank capacity lb | 210 | | 238 | | 216 | | 283 | |
| battery capacity kWh | | 435 | | 485 | | 633 | | 528 |
| aircraft drag D/q ft ² | 8.7 | 14.3 | 6.3 | 8.9 | 7.4 | 12.3 | 4.4 | 5.8 |
| $D/q/(W/1000)^{2/3}$ | 3.2 | 3.9 | 2.3 | 2.4 | 2.6 | 2.9 | 1.5 | 1.5 |
| hover download DL/W | 0 | 0 | 0.057 | 0.030 | 0.060 | 0.058 | 0.127 | 0.080 |
| Weight | | | | | | | | |
| DGW lb | 4621 | 6881 | 4602 | 7429 | 4771 | 8945 | 4979 | 8109 |
| weight empty lb | 3401 | 5875 | 3354 | 6424 | 3545 | 7936 | 3686 | 7100 |
| structure lb | 1404 | 1919 | 1363 | 2171 | 1506 | 2687 | 1251 | 2080 |
| propulsion (ex batt) lb | 1138 | 1392 | 1141 | 1475 | 1118 | 1741 | 1320 | 1764 |
| battery lb | | 1477 | | 1643 | | 2146 | | 1790 |
| systems lb | 587 | 617 | 582 | 621 | 637 | 727 | 821 | 897 |
| Performance | | | | | | | | |
| hover power hp | 375 | 516 | 469 | 677 | 336 | 611 | 763 | 943 |
| cruise power (V_{br}) hp | 339 | 319 | 372 | 360 | 364 | 476 | 495 | 412 |
| best range speed knots | 119 | 89 | 126 | 96 | 117 | 93 | 214 | 158 |
| best endurance speed knots | 60 | 56 | 63 | 60 | 68 | 68 | 109 | 101 |
| maximum speed knots | 140 | 103 | 143 | 133 | 139 | 119 | 223 | 198 |
| climb speed V_y knots | 60 | 48 | 63 | 59 | 58 | 62 | 37 | 41 |
| maximum speed V_H knots | 137 | 105 | 143 | 132 | 138 | 124 | 97 | 103 |
| Mission | | | | | | | | |
| payload lb | 1000 | 1000 | 1000 | 1000 | 1000 | 1000 | 1000 | 1000 |
| range nm | 75 | 75 | 75 | 75 | 75 | 75 | 75 | 75 |
| block time min | 46 | 58 | 45 | 57 | 47 | 56 | 26 | 32 |
| block speed knots | 98 | 77 | 101 | 79 | 95 | 80 | 174 | 140 |
| productivity $W_{pay}V/W_0$ ton-kt/lb | 0.0144 | 0.0066 | 0.0150 | 0.0062 | 0.0133 | 0.0050 | 0.0234 | 0.0099 |
| trip cost \$ | 799 | 800 | 857 | 1001 | 789 | 1054 | 697 | 697 |
| DOC \$/ASM | 1.85 | 1.85 | 1.99 | 2.32 | 1.83 | 2.44 | 1.61 | 1.62 |
| purchase cost \$M | 5.3 | 3.5 | 5.9 | 5.0 | 4.9 | 5.0 | 8.2 | 6.3 |
| maintenance cost \$/hr | 750 | 664 | 818 | 840 | 718 | 889 | 1067 | 989 |

Table 10. Baseline AAM Concept Vehicles: weight statement.

| | quad TS | quad E | QSMR | QSMR E | side TS | side E | tilt TS | tilt E |
|---------------------------|-------------|-------------|-------------|-------------|-------------|-------------|-------------|-------------|
| DGW | 4621 | 6881 | 4602 | 7429 | 4771 | 8945 | 4979 | 8109 |
| WEIGHT EMPTY | 3401 | 5875 | 3354 | 6424 | 3545 | 7936 | 3686 | 7100 |
| STRUCTURE | 1404 | 1919 | 1363 | 2171 | 1506 | 2687 | 1251 | 2080 |
| wing group | | | | | 262 | 529 | 187 | 383 |
| rotor group | 652 | 1005 | 477 | 937 | 454 | 993 | 239 | 573 |
| blade assembly | 247 | 404 | 299 | 536 | 318 | 663 | 133 | 298 |
| hub & hinge | 97 | 195 | 179 | 401 | 136 | 330 | 107 | 275 |
| rotor support | 308 | 405 | | | | | | |
| empennage group | | | 138 | 265 | 26 | 58 | 35 | 72 |
| fuselage group | 463 | 604 | 491 | 676 | 503 | 766 | 481 | 666 |
| alighting gear | 228 | 297 | 195 | 268 | 200 | 303 | 205 | 284 |
| engine section/nacelle | 44 | 14 | 45 | 24 | 48 | 39 | 89 | 102 |
| air induction | 17 | | 17 | | 13 | | 14 | |
| PROPULSION GROUP | 1138 | 2869 | 1141 | 3118 | 1118 | 3887 | 1320 | 3554 |
| engine system | 674 | 260 | 692 | 357 | 674 | 332 | 760 | 480 |
| fuel system (battery) | 61 | 2030 | 66 | 2197 | 61 | 2887 | 76 | 2388 |
| drive system | 403 | 579 | 383 | 565 | 383 | 668 | 483 | 686 |
| SYSTEMS AND EQUIP | 587 | 617 | 582 | 621 | 637 | 727 | 821 | 897 |
| flight controls | 107 | 128 | 99 | 123 | 152 | 219 | 308 | 418 |
| cockpit controls | 10 | 10 | 10 | 10 | 10 | 10 | 10 | 10 |
| automatic flight controls | 40 | 40 | 40 | 40 | 40 | 40 | 40 | 40 |
| system controls | 57 | 78 | 49 | 73 | 102 | 169 | 258 | 368 |
| instruments group | 20 | 20 | 20 | 20 | 20 | 20 | 20 | 20 |
| hydraulic group | 8 | | 9 | | 11 | | 51 | |
| electrical group | 94 | 104 | 96 | 108 | 95 | 113 | 89 | 98 |
| avionics (MEQ) | 110 | 110 | 110 | 110 | 110 | 110 | 110 | 110 |
| furnish & equipment | 150 | 150 | 150 | 150 | 150 | 150 | 150 | 150 |
| environmental control | 75 | 75 | 75 | 75 | 75 | 75 | 75 | 75 |
| anti-icing group | 23 | 30 | 24 | 35 | 23 | 40 | 18 | 26 |
| VIBRATION | 102 | 176 | 101 | 193 | 106 | 238 | 111 | 213 |
| CONTINGENCY | 170 | 294 | 168 | 321 | 177 | 397 | 184 | 355 |
| FIXED USEFUL LOAD | 10 | 5 | 10 | 5 | 10 | 10 | 10 | 10 |
| OPERATING WEIGHT | 3411 | 5880 | 3364 | 6429 | 3555 | 7946 | 3696 | 7110 |

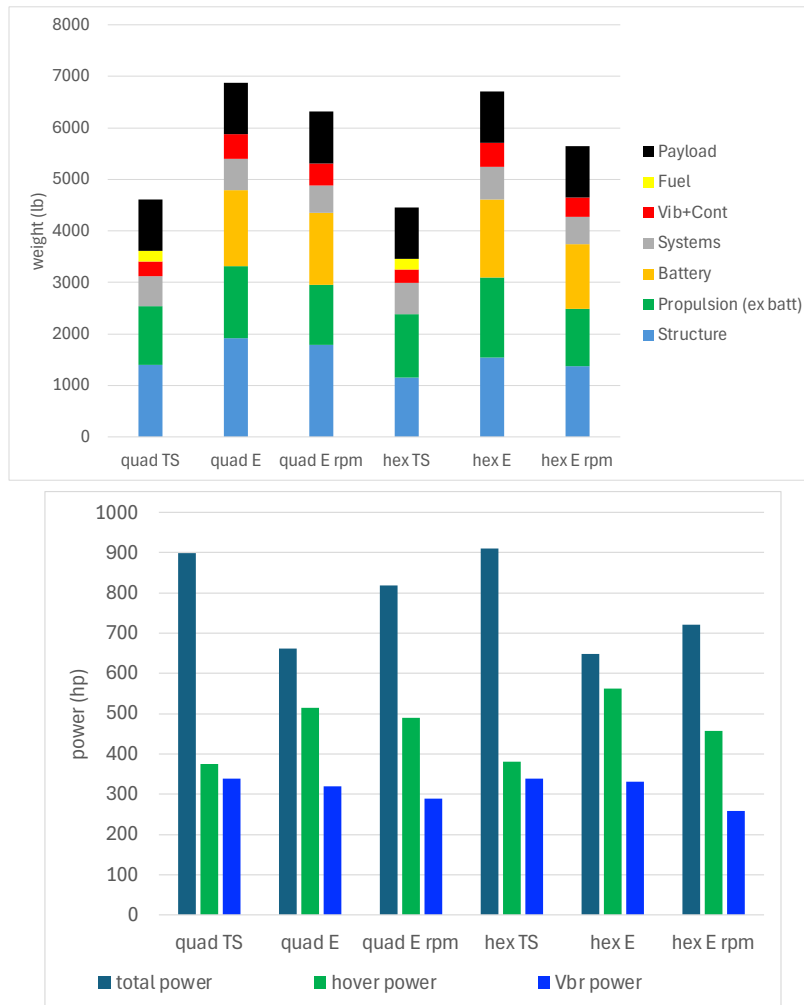


Figure 8. Electric multicopter with collective or rotor-speed control (weight in lb, power in hp).

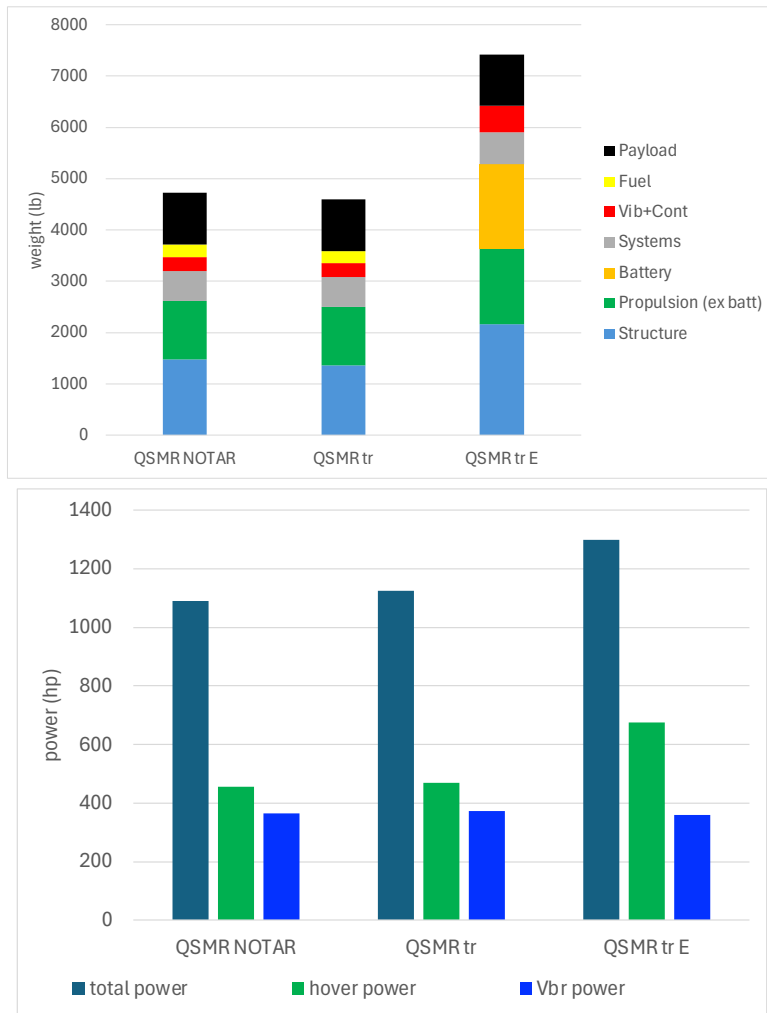


Figure 9. Quiet Single Main Rotor (QSMR) helicopter, with NOTAR and tail rotor anti-torque (weight in lb, power in hp).

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. Here there is no attempt to quantify how much weight and power growth is too much. Rather it is sufficient to identify such parameters and observe the limits in the designs.

The design variations are conducted for both turboshaft and electric vehicles, except of course for the propulsion-system-specific parameters.

The aircraft structural weight growth factors are obtained from the design gross weight change produced by payload changes of ± 200 lb.

Technology

Weight technology factors. The technology factors used in the component weight estimates (Table 3) are based on something between research objectives and wishful thinking. So it is important to compare the baseline aircraft to designs obtained only considering calibration factors. Also, the baseline calibration is based on the mean values for several aircraft, so design excursions include high calibration (mean + standard deviation) and low calibration (mean – standard deviation) cases.

Rotor support or wing. Separately, the impact of structural weight variation is examined for the quadrotor rotor support (1/3 to 2 times baseline weight per length), side-by-side helicopter rotor support (50% to 175% baseline technology factor), and tiltrotor wing (75% to 150% baseline technology factor).

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.

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.

Blade loading. 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 . For tiltrotors, the blade loading is varied from 0.13 (baseline) to 0.11 .

Profile power. The mean blade drag coefficient in cruise is high, since the low tip speed results in high advance ratios, increasing the stall on retreating blade of edgewise rotors. The profile power is varied from 100% to 80% .

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% .

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 . The wiring weight is varied from 100% to 300% of the baseline. The thermal management system design rejected power is varied from 100% to 300% of the baseline, for both motor and battery.

Motor torque margin. For the rpm-control hexacopter configuration, the usable fraction of available power is varied from 55% to 95% , relative to the baseline of 75% (25% control margin).

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. The sizing iteration does not converge if the battery weight is too large, due to low specific energy.

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. 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).

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.

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.

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).

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.

Cruise speed. The mission has the aircraft cruise at the best-range speed (Table 1) 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. The rotor disk loading and wing loading are optimized for each design cruise speed, to minimize aircraft weight and power.

Presentation

For a particular parameter variation, generally four plots are presented:

- 1) Power (hp) as a function of the independent parameter: for hover and cruise (V_{br}) mission segments.
- 2) Aircraft weights (lb) as a function of the independent parameter: structure, + propulsion, + battery (electric only) + systems, weight empty, + fuel (turboshaft only), and design gross weight.
- 3) Aircraft group weights (lb) as a bar chart: structure, propulsion, battery (electric only), systems, vibration and contingency (subtotal is weight empty), fuel (turboshaft only), and payload (total is design gross weight).
- 4) Power (hp) as a bar chart: installed power (sea level static for turboshaft), and power required for hover and cruise (V_{br}) mission segments.

The line plots show the trends correctly (since the cases typically are not for uniform increments of the independent parameter), while the bar charts are better for visualizing the components. If the parameter is discrete, not continuous, only the bar charts are shown. Typically both turboshaft and electric propulsion results are shown in a single figure.

Impact of Technology and Mission Variations — Quadrotor

Technology

Weight technology factors (Figure 10). Designs with technology, compared to only calibration (no technology factors) — high, baseline, and low calibration.

Rotor support weight (Figure 11). Baseline weight per length varied from 2.5 to 15 lb/ft (baseline 7.5).

Systems and equipment (Figure 12). The systems and equipment weight changed by -200 lb to 200 lb, relative to the baseline fixed weight of 480 lb.

Aircraft drag (Figure 13). The airframe drag varied by $\Delta D/q = -2.0$ to 2.0 ft².

Blade loading (Figure 14). The main rotor design blade loading varied from $C_W/\sigma = 0.09$ (baseline) to 0.07 .

Profile power (Figure 15). The profile power varied from 100% to 80% .

Engine weight (Figure 16). The turboshaft engine weight and electric motor weight varied from 80% to 120% of the baseline.

Engine efficiency (Figure 17). The turboshaft engine specific fuel consumption varied from 80% to 120% of the baseline (0.50 lb/hp-hr MCP SLS). The motor efficiency varied from 0.95 (baseline) to 0.75 .

Electric motor (Figures 18). The wiring weight varied from 100% to 300% of the baseline. The thermal management system design rejected power (motor and battery) varied from 100% to 300% of the baseline.

Motor torque margin (Figure 19). For the rpm-control hexacopter configuration, the usable fraction of available power is varied from 55% to 95% , compared to the baseline of 75% (25% control margin).

Battery (Figure 20). The battery specific energy varied from 400 Wh/kg (baseline) to 225 Wh/kg.

Mission

Range (Figures 21–22). 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. For electric aircraft, the mission range is varied for a set of battery weights (400 Wh/kg to 150 Wh/kg).

Reserve (Figure 23). The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. The reserve segment time is varied from 0 to 45 min.

Climb and one-engine-inoperative (Figure 24–25). The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min, with and without the OEI requirement.

Cruise climb (Figure 26). The cruise rate of climb is varied from 500 ft/min (baseline) to 0 (no requirement).

Altitude (Figure 27). The design mission has takeoff at 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA); or takeoff at sea level ISA and cruise at 4000 ft ISA; or both.

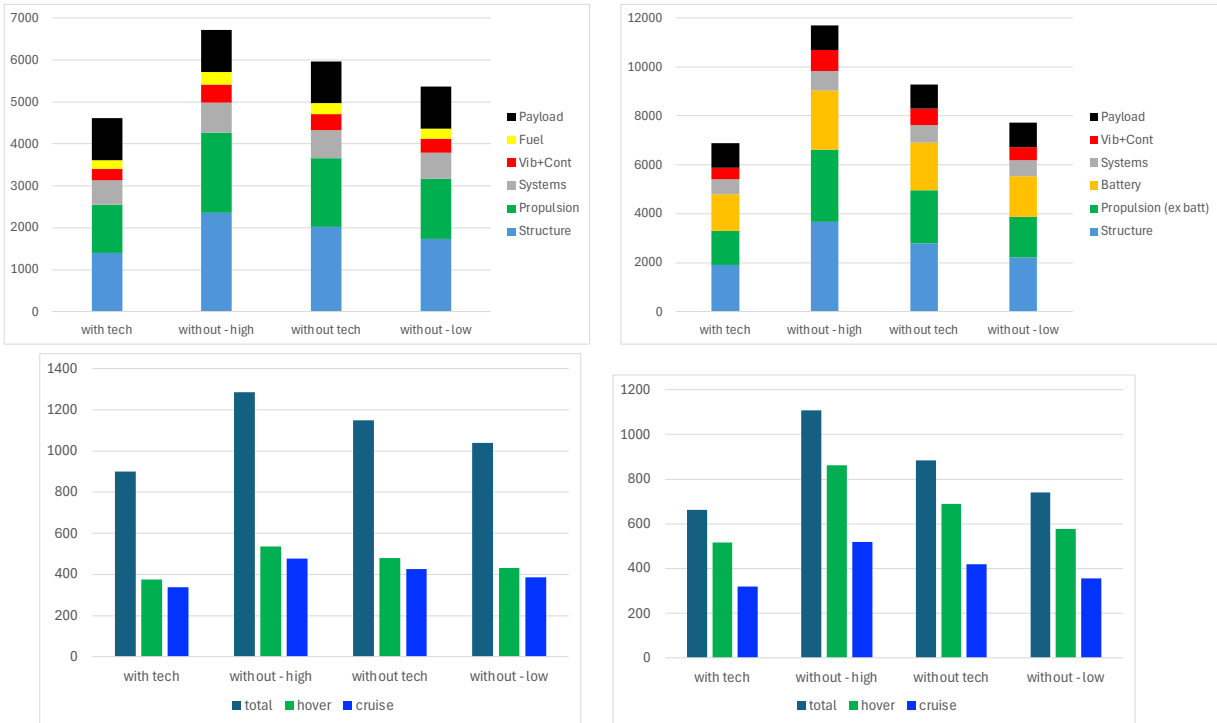


Figure 10. Quadrotor, turboshaft (left) and electric (right) — impact of weight technology factors. Designs with technology, compared to only calibration (no technology factors): high, baseline, and low calibration.

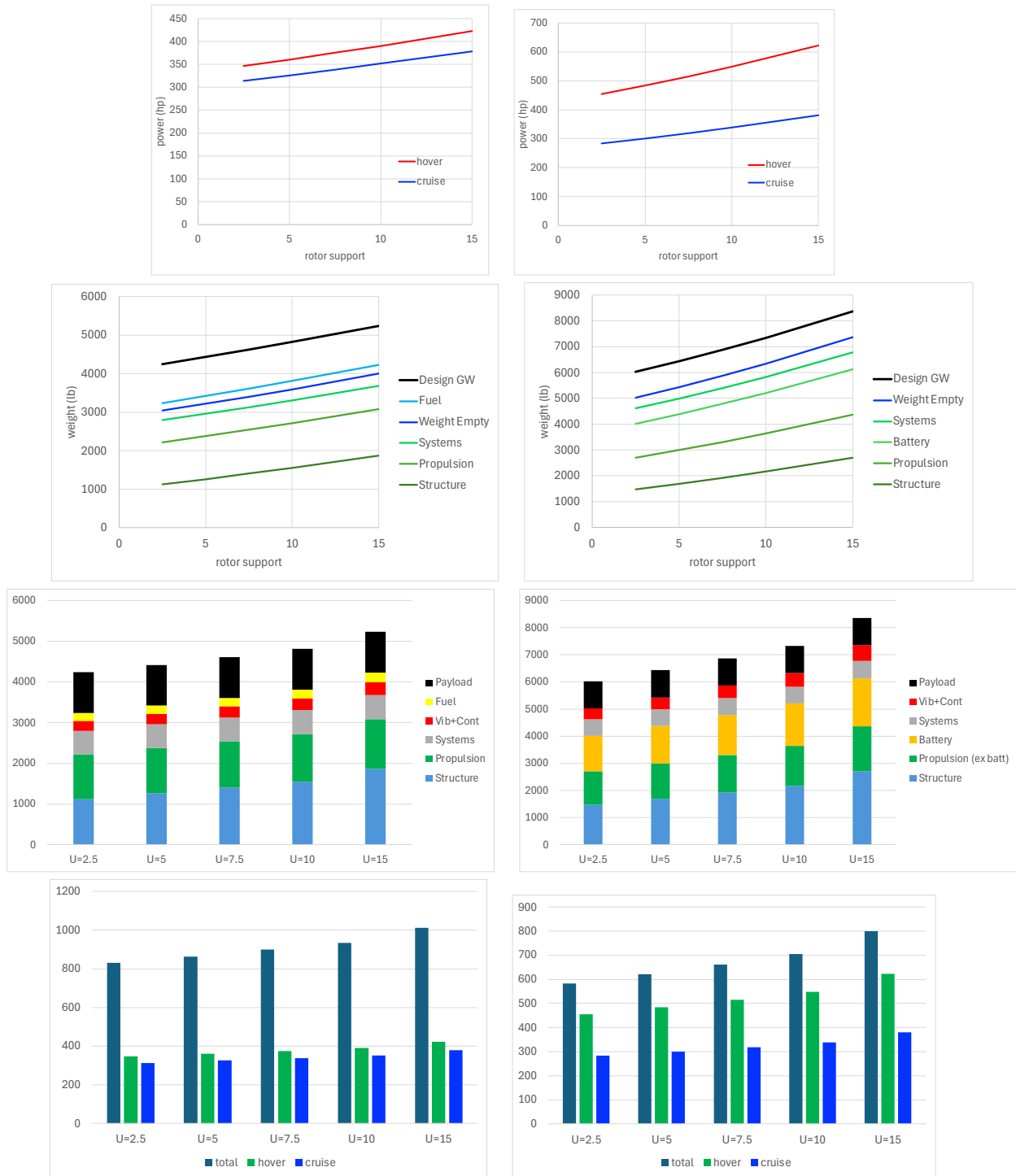


Figure 11. Quadrotor, turboshaft (left) and electric (right) — impact of rotor support weight. Baseline weight per length varied from 2.5 to 15 lb/ft (baseline 7.5).

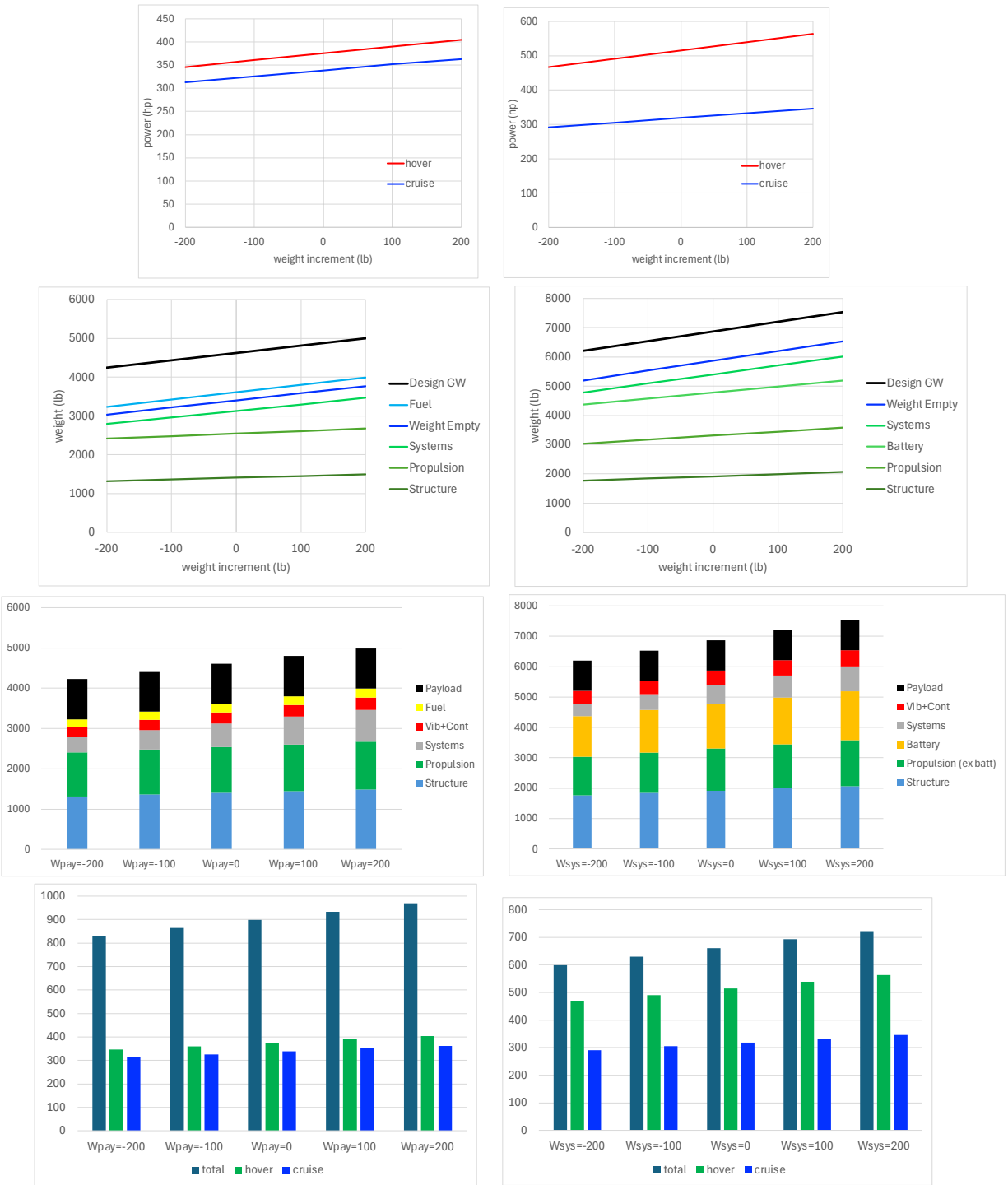


Figure 12. Quadrotor, turboshaft (left) and electric (right) — impact of systems and equipment weight. The systems and equipment weight changed by -200 lb to 200 lb, relative to the baseline fixed weight of 480 lb.

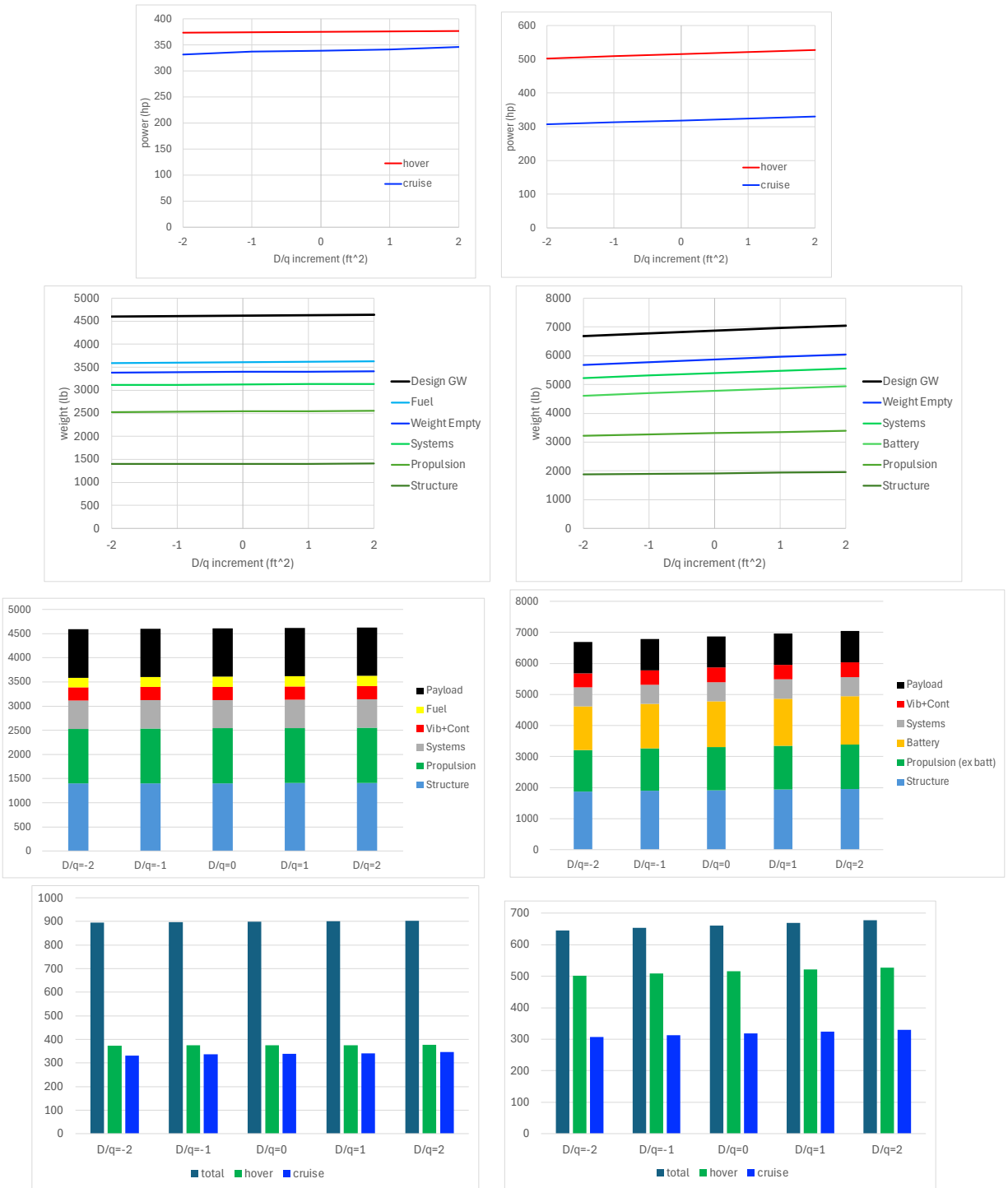


Figure 13. Quadrotor, turboshaft (left) and electric (right) — impact of aircraft drag. The airframe drag varied by $\Delta D/q = -2.0$ to 2.0 ft^2 .

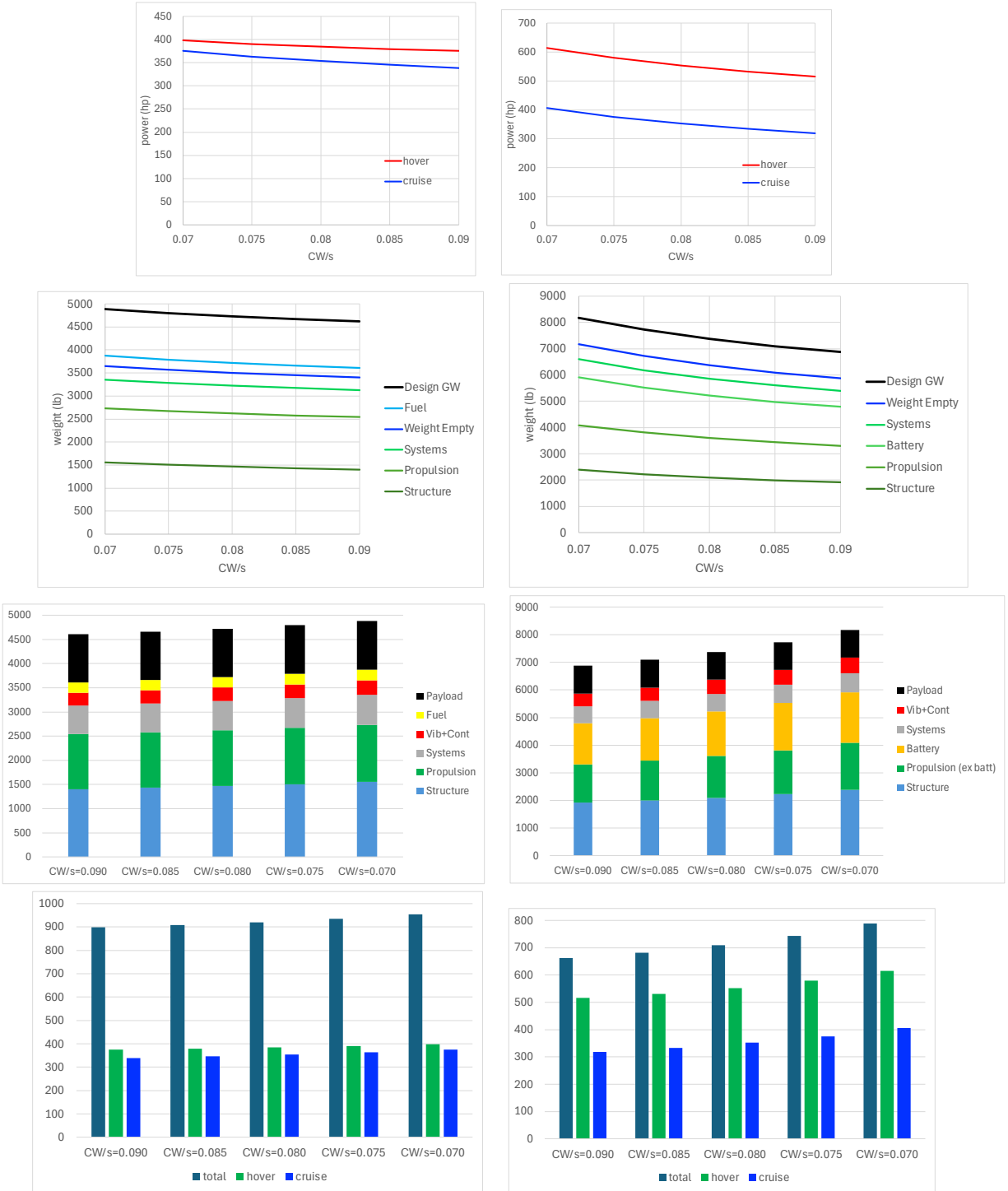


Figure 14. Quadrotor, turboshaft (left) and electric (right) — impact of main rotor design blade loading. The main rotor design blade loading varied from $C_W/\sigma=0.09$ (baseline) to 0.07.

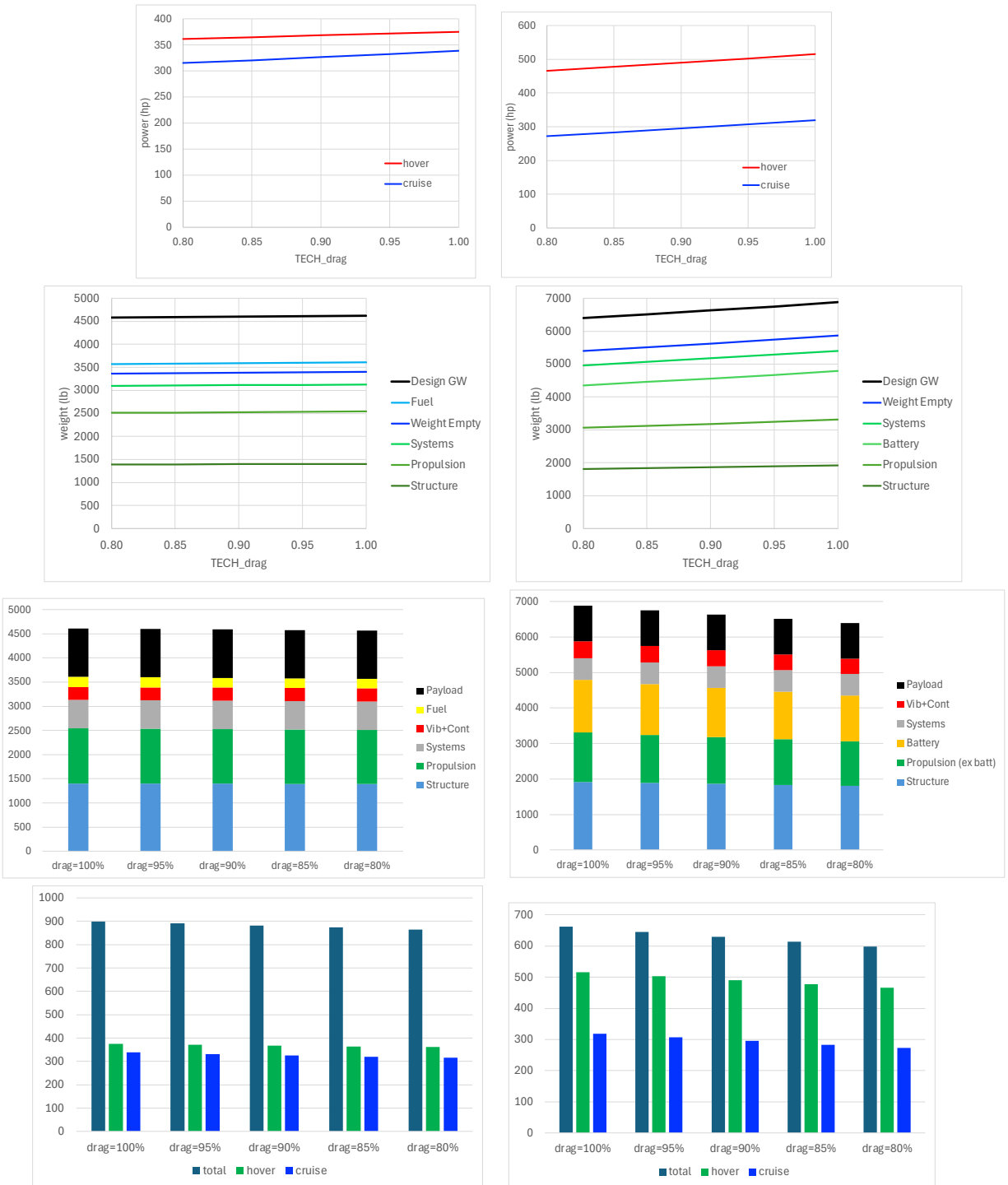


Figure 15. Quadrotor, turboshaft (left) and electric (right) — impact of rotor profile power. The profile power varied from 100% to 80%.

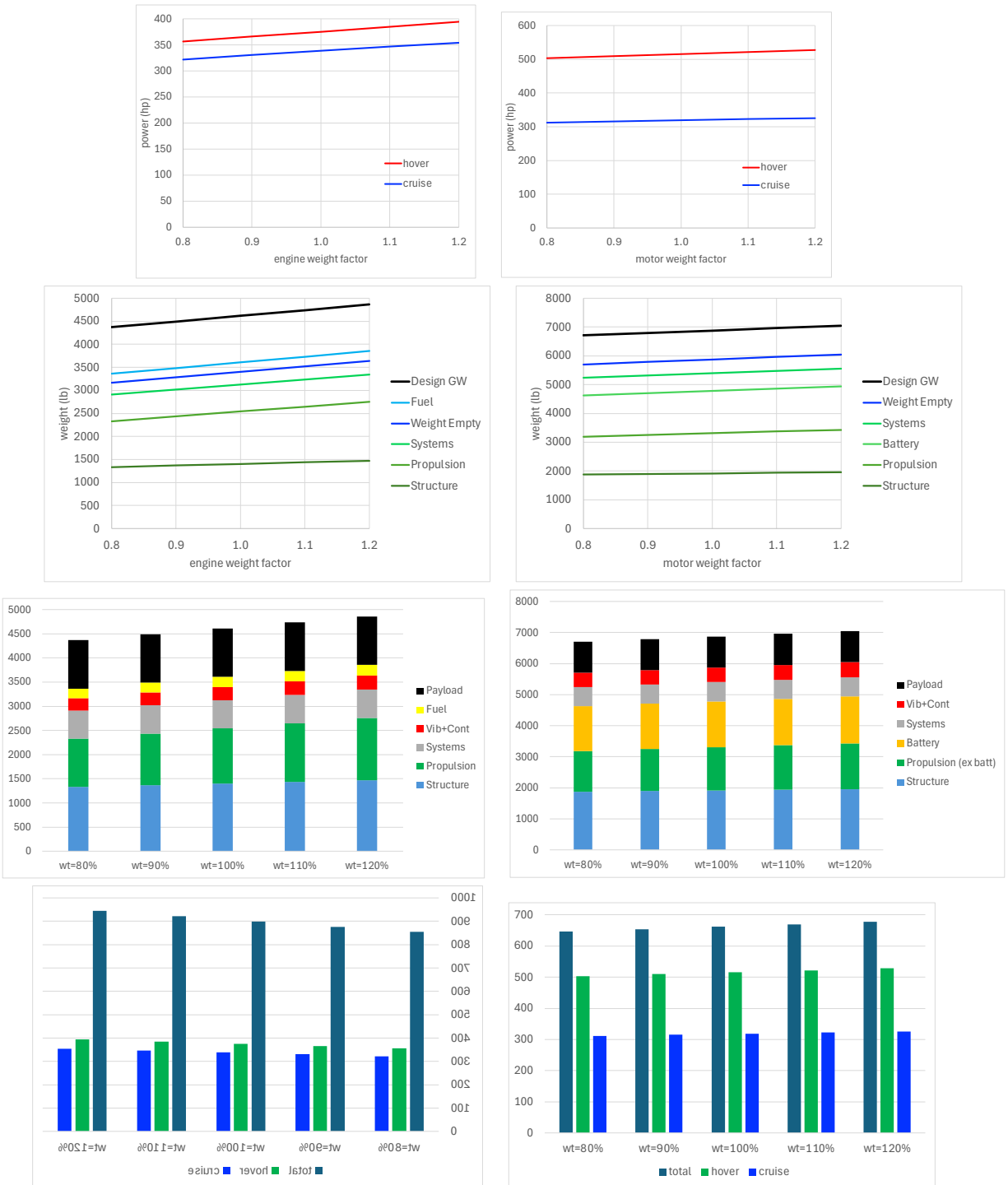


Figure 16. Quadrotor, turboshaft (left) and electric (right) — impact of engine weight. The turboshaft engine weight or electric motor weight varied from 80% to 120% of the baseline.

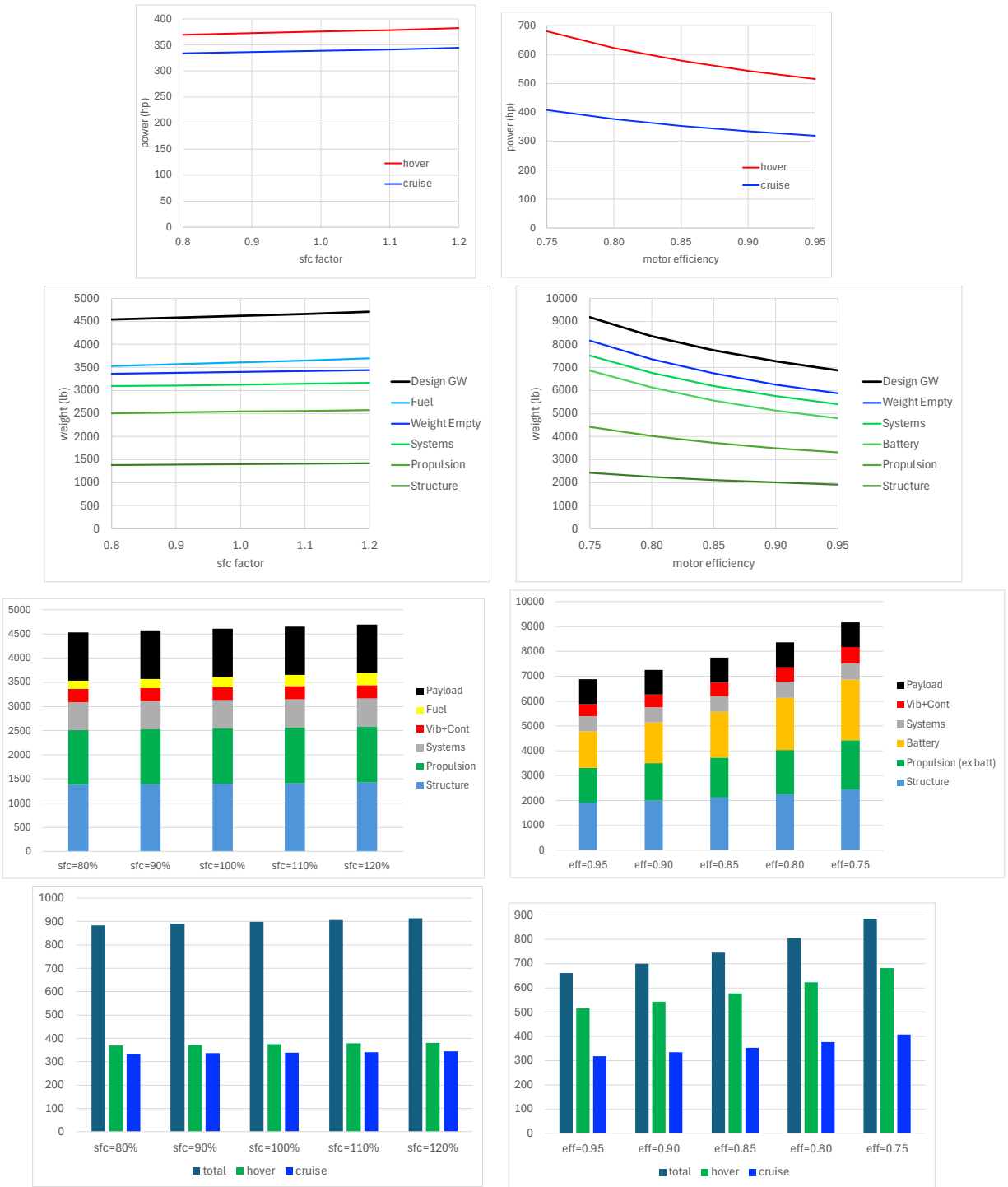


Figure 17. Quadrotor, turboshaft (left) and electric (right) — impact of engine efficiency. The turboshaft engine specific fuel consumption varied from 80% to 120% of the baseline (0.50 lb/hp-hr MCP SLS). The motor efficiency varied from 0.95 (baseline) to 0.75.

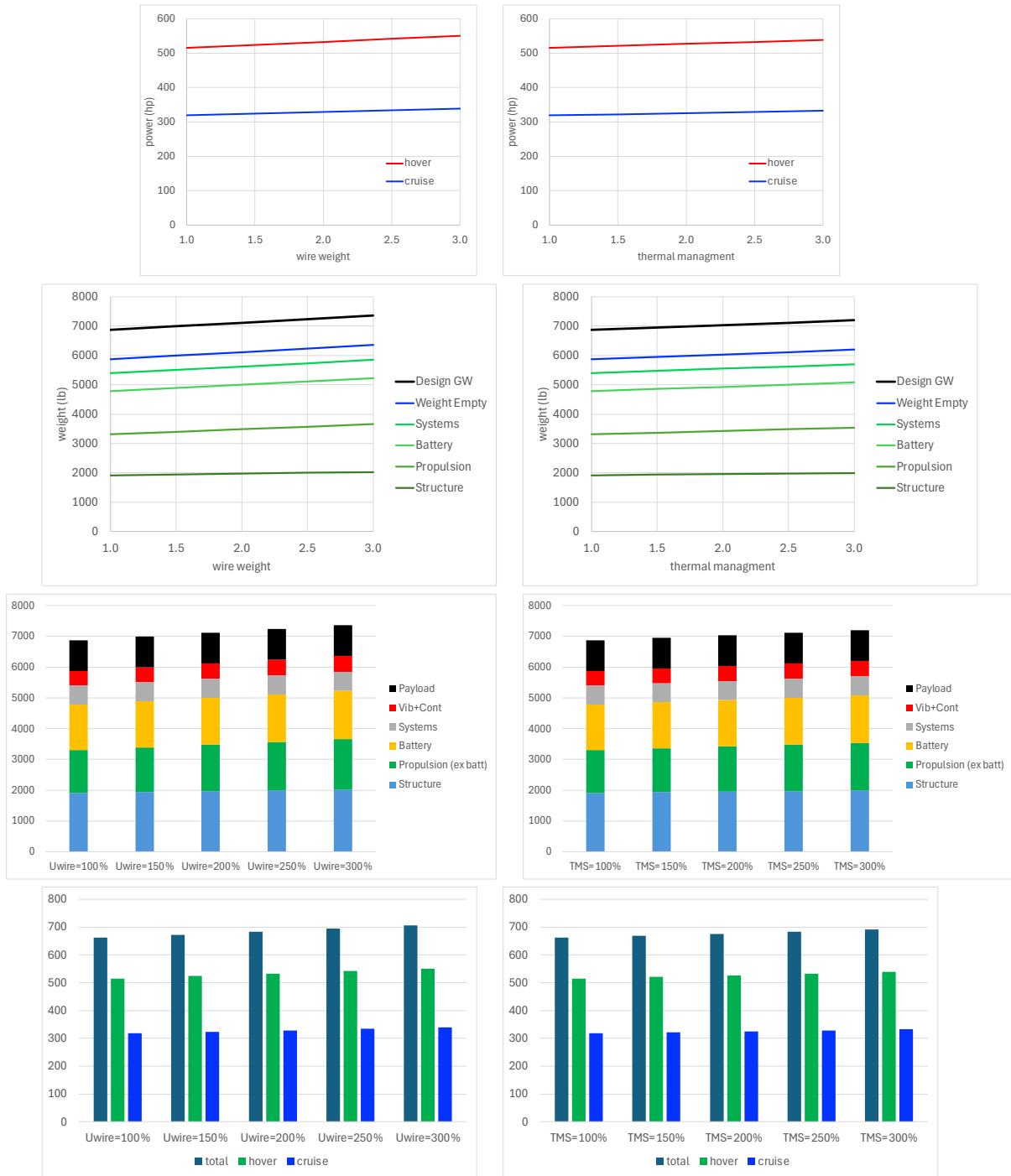


Figure 18. Quadrotor, electric — impact of wiring weight (left) and thermal management system (right). The wiring weight varied from 100% to 300% of the baseline. The thermal management system design rejected power (motor and battery) varied from 100% to 300% of the baseline.

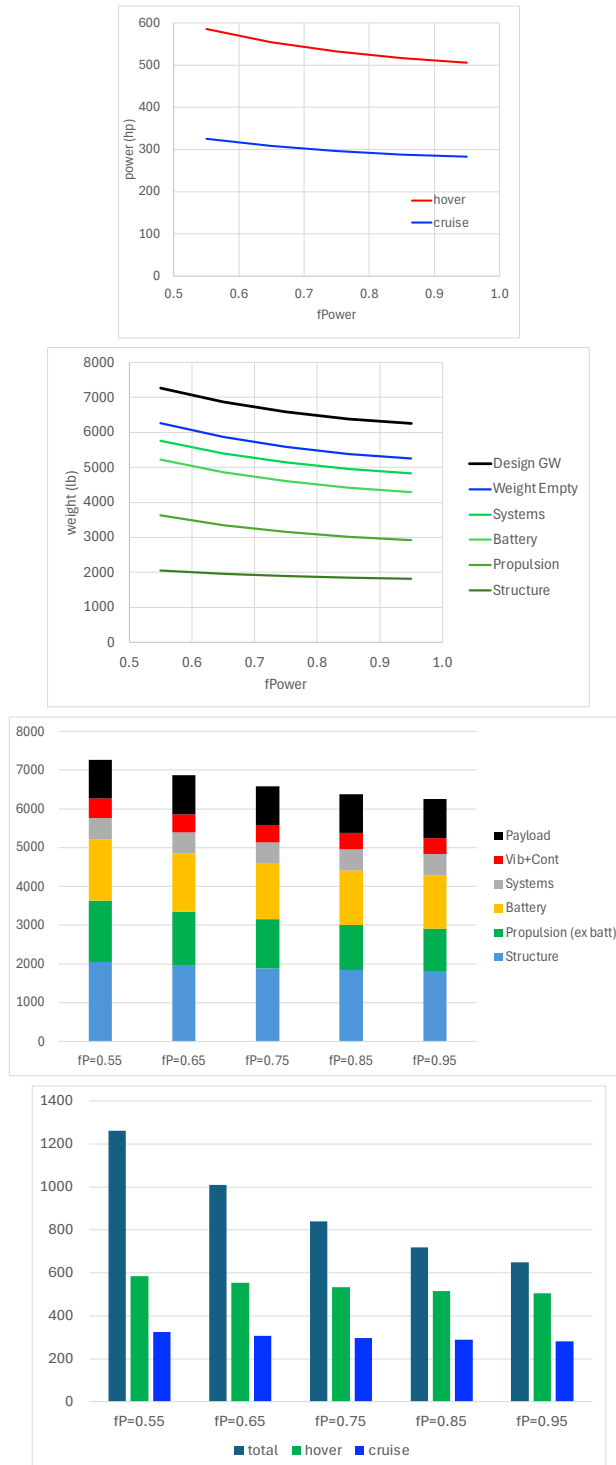


Figure 19. Hexacopter with rotor-speed control — impact of motor torque margin. For the rpm-control hexacopter configuration, the usable fraction of available power varied from 55% to 95%, compared to the baseline of 75% (25% control margin).

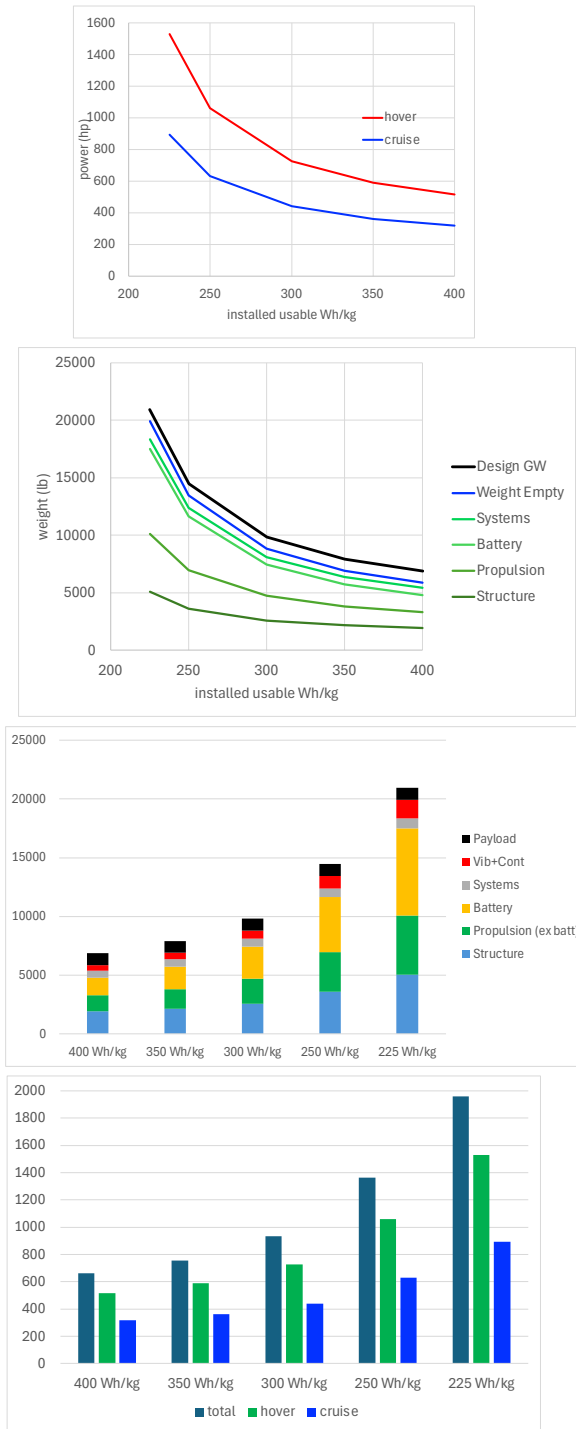


Figure 20. Quadrotor, electric — impact of battery specific energy. The battery specific energy varied from 400 Wh/kg (baseline) to 225 Wh/kg.

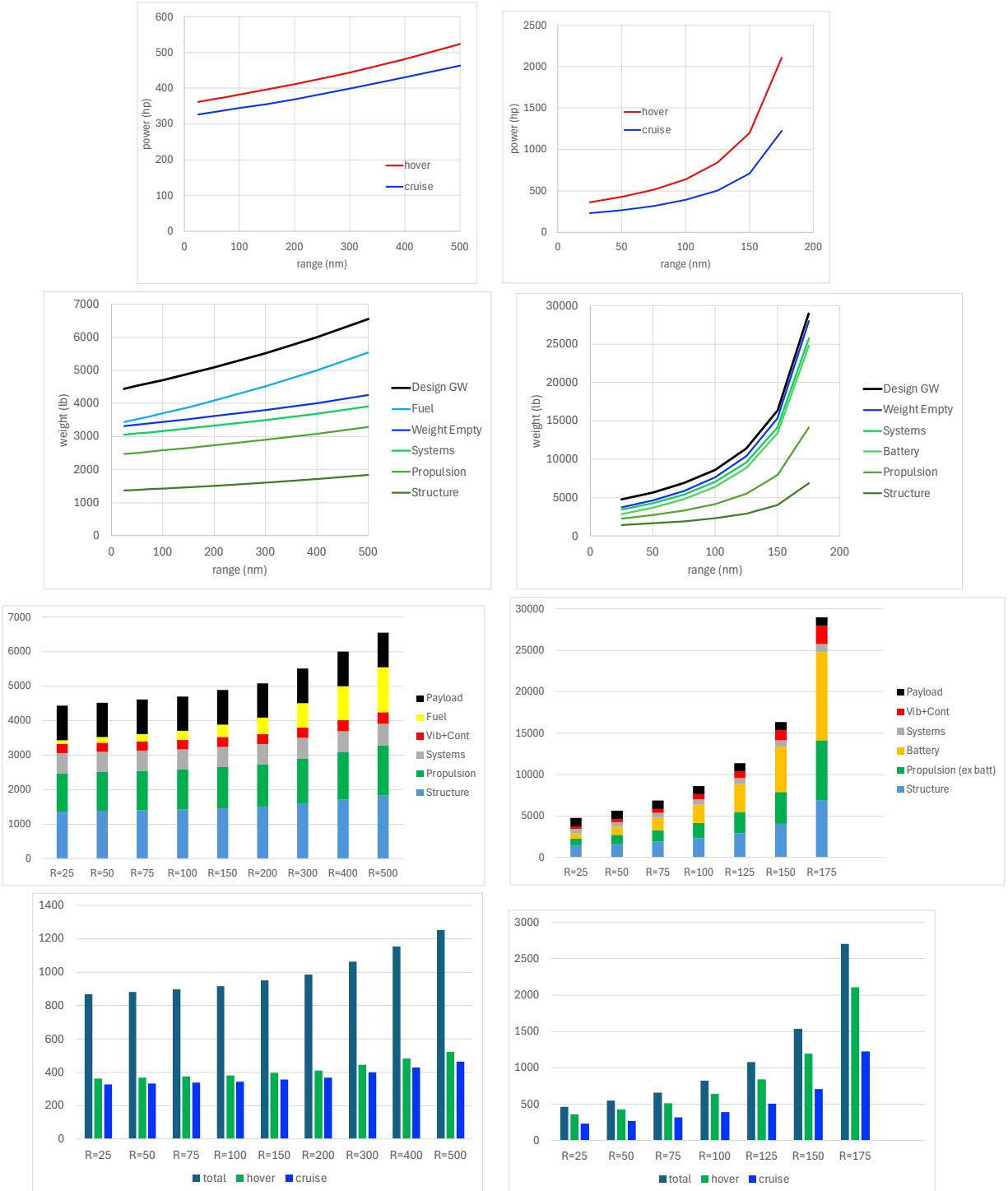


Figure 21. Quadrotor, turboshaft (left) and electric (right) — impact of mission range. The baseline design mission has a range of 75 nm (in two segments). The range is varied from 25 to 500 nm for the turboshaft configurations, and from 25 to 175 nm for the electric configurations.

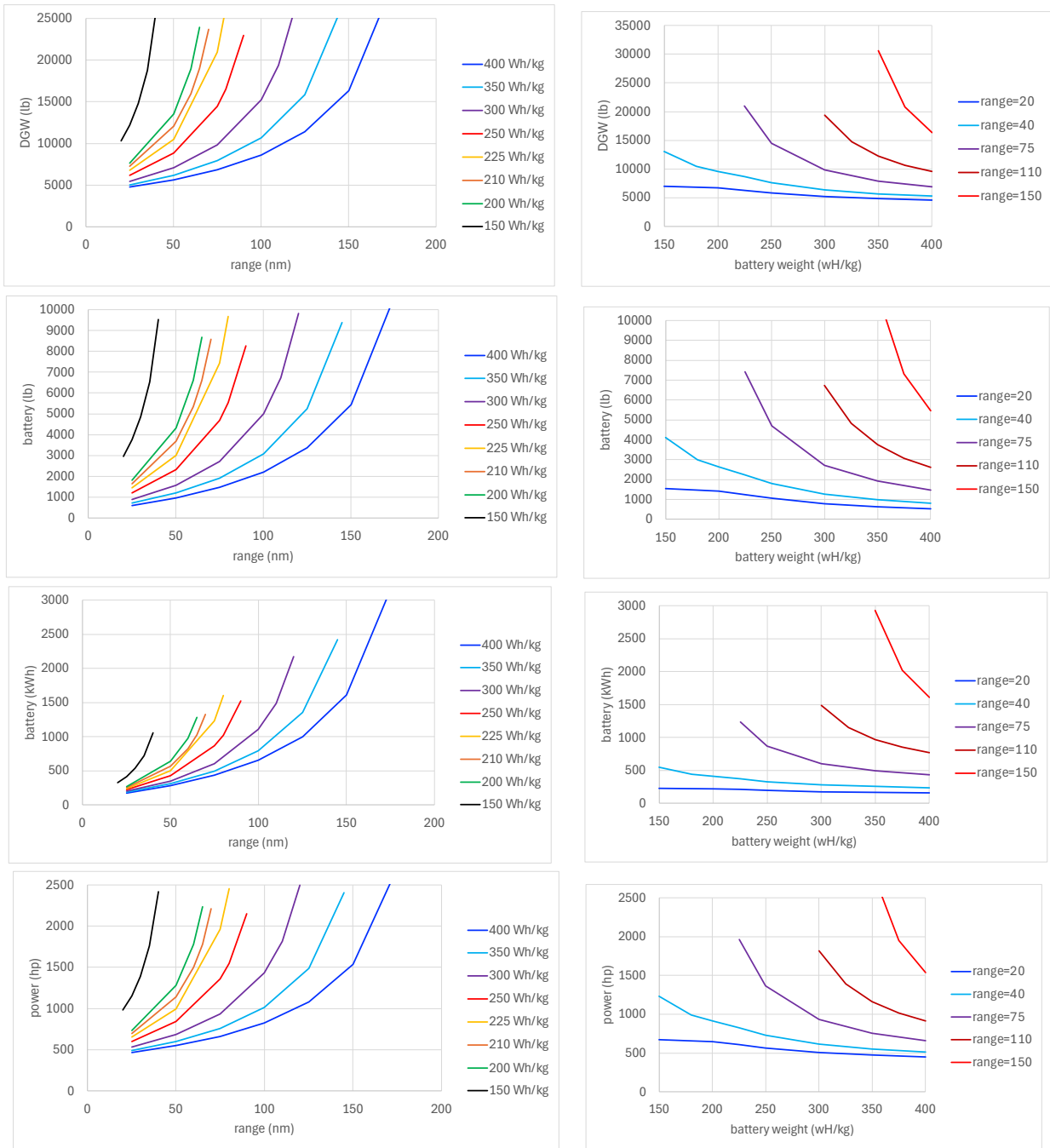


Figure 22. Quadrotor, electric — impact of mission range (20 to 175 nm) and battery weight (400 to 150 Wh/kg); design gross weight (lb), battery weight (lb), battery capacity (kWh), and installed power (hp). The baseline design mission has a range of 75 nm (in two segments), with battery weight 400 Wh/kg.



Figure 23. Quadrotor, turboshaft (left) and electric (right) — impact of mission reserve. The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. The reserve segment time is varied from 0 to 45 min.

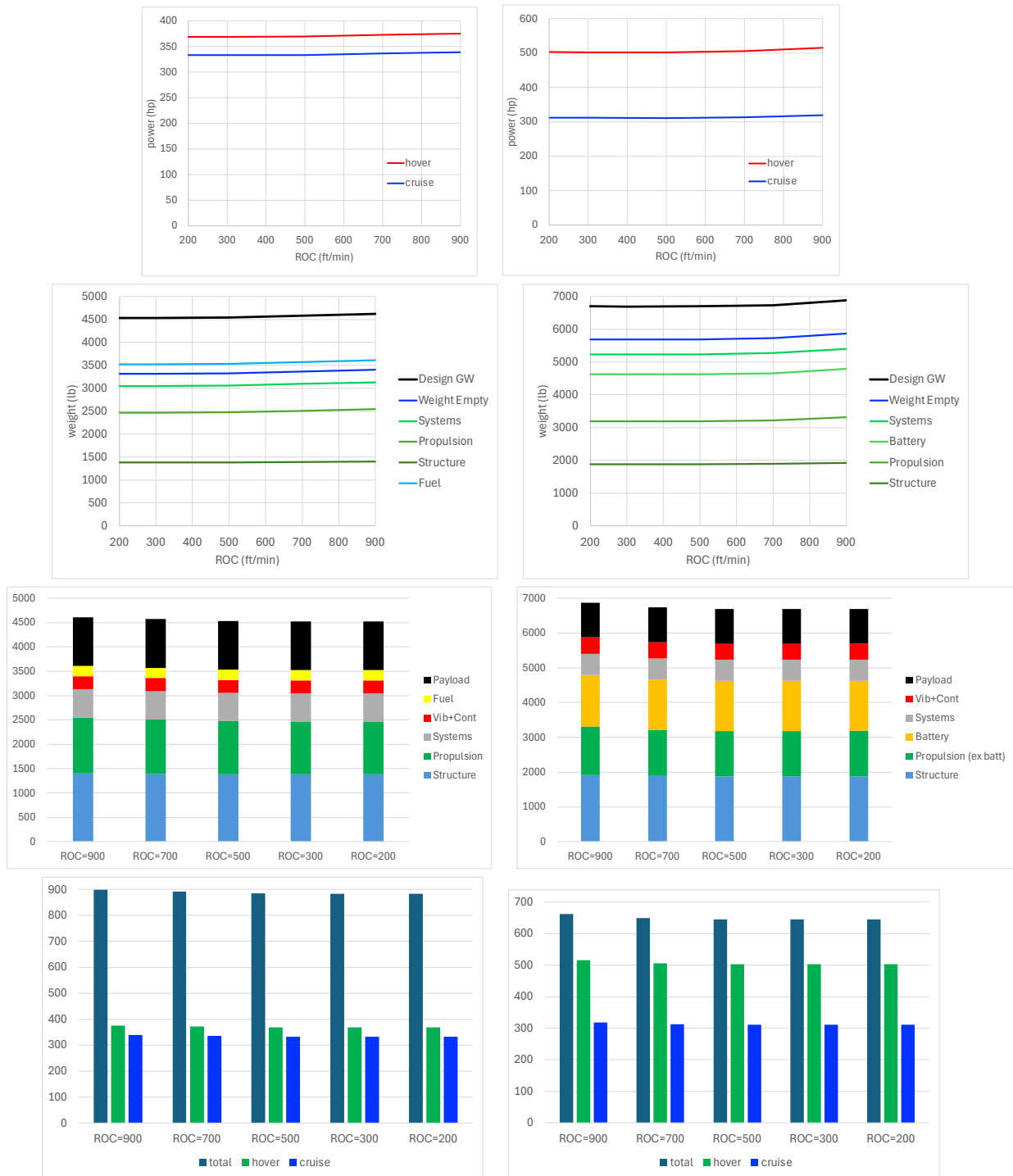


Figure 24. Quadrotor, turboshaft (left) and electric (right) — impact of mission climb to cruise altitude, with one-engine-inoperative requirement. The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min.

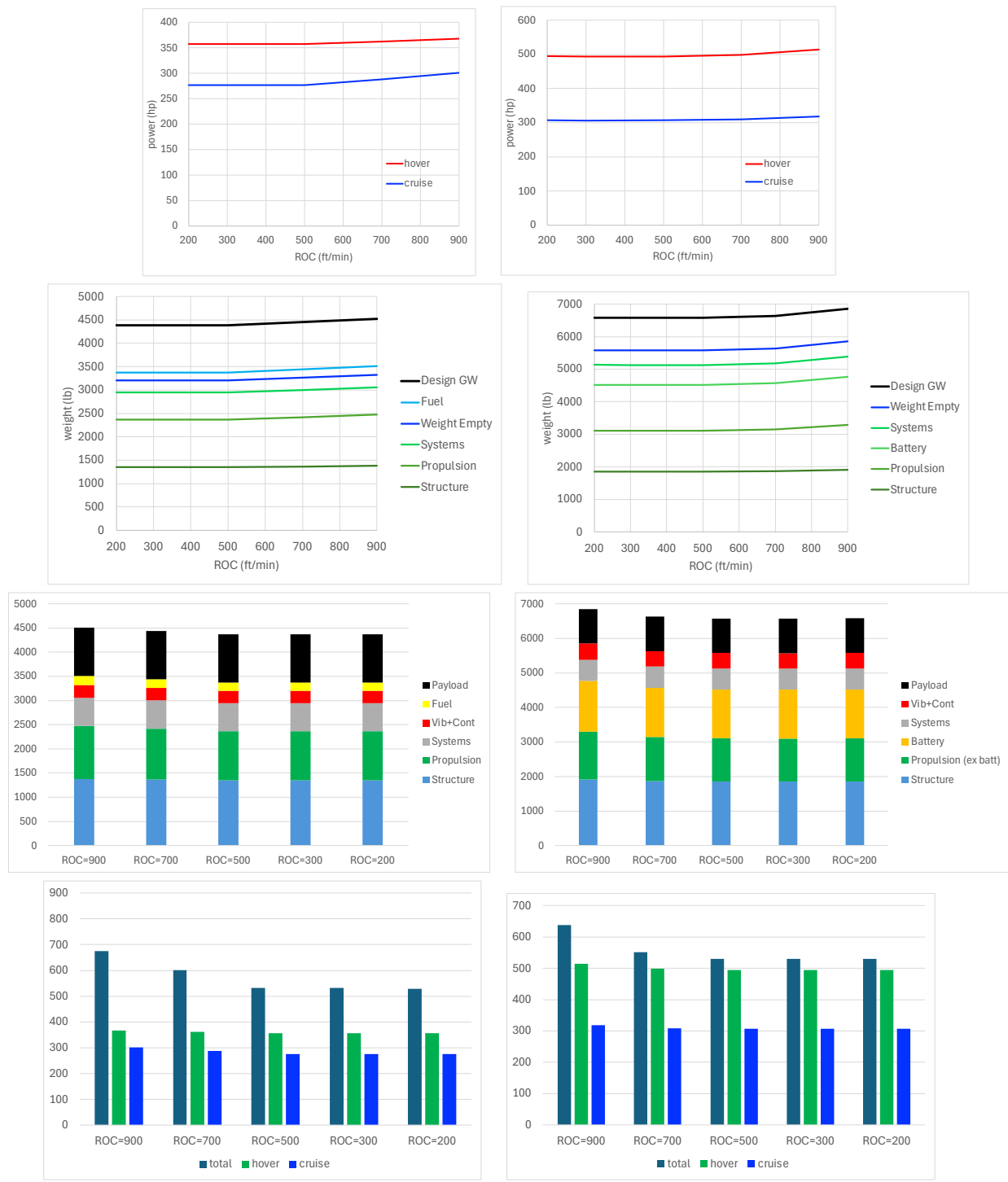


Figure 25. Quadrotor, turboshaft (left) and electric (right) — impact of mission climb to cruise altitude, without one-engine-inoperative requirement. The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min.

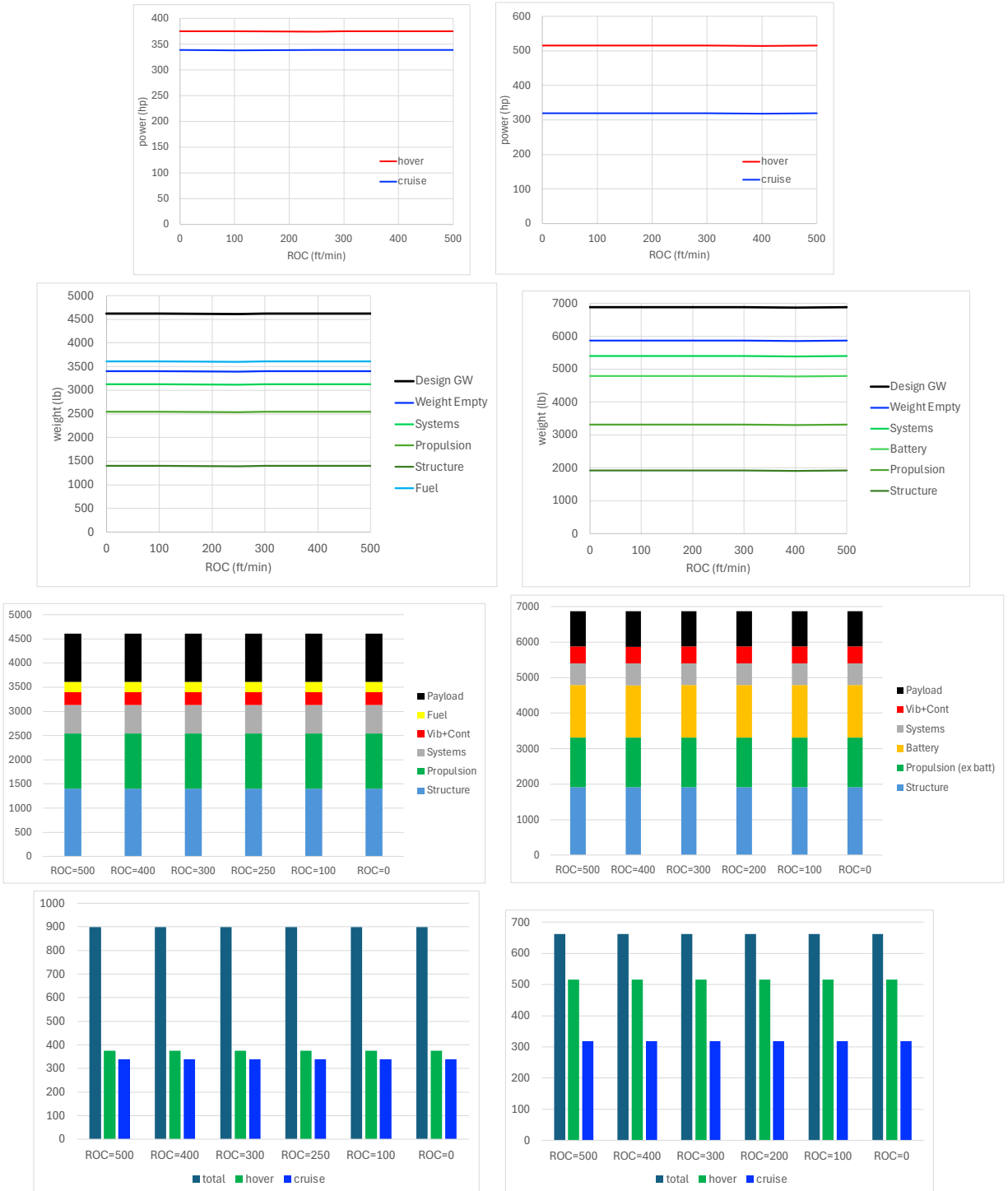


Figure 26. Quadrotor, turboshaft (left) and electric (right) — impact of cruise climb. The cruise rate of climb is varied from 500 ft/min (baseline) to 0 (no requirement).

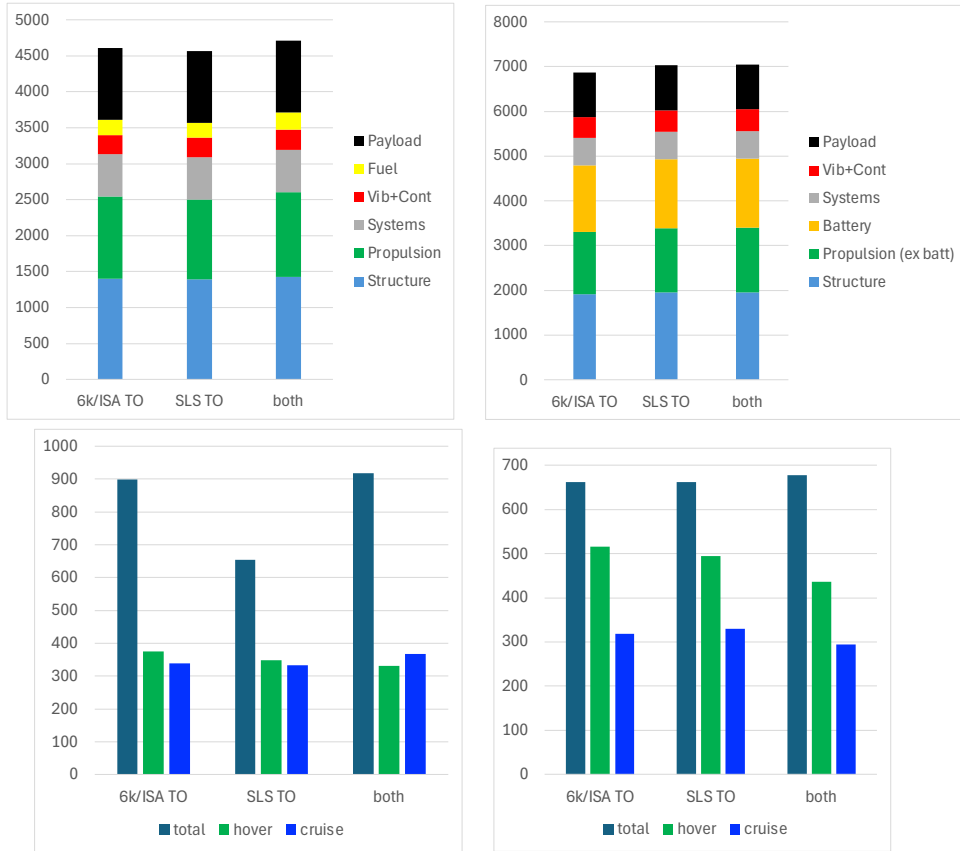


Figure 27. Quadrotor, turboshaft (left) and electric (right) — impact of operating altitude. The design mission has takeoff at 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA); or takeoff at sea level ISA and cruise at 4000 ft ISA; or both.

Impact of Technology and Mission Variations — QSMR

Technology

Weight technology factors (Figure 28). Designs with technology, compared to only calibration (no technology factors) — high, baseline, and low calibration.

Systems and equipment (Figure 29). The systems and equipment weight changed by -200 lb to 200 lb, relative to the baseline fixed weight of 480 lb.

Aircraft drag (Figure 30). The airframe drag varied by $\Delta D/q = -2.0$ to 2.0 ft².

Blade loading (Figure 31). The main rotor design blade loading varied from $C_W/\sigma = 0.09$ (baseline) to 0.07 .

Profile power (Figure 32). The profile power varied from 100% to 80% .

Engine weight (Figure 33). The turboshaft engine weight and electric motor weight varied from 80% to 120% of the baseline.

Engine efficiency (Figure 34). The turboshaft engine specific fuel consumption varied from 80% to 120% of the baseline (0.50 lb/hp-hr MCP SLS). The motor efficiency varied from 0.95 (baseline) to 0.75 .

Electric motor (Figures 35). The wiring weight varied from 100% to 300% of the baseline. The thermal management system design rejected power (motor and battery) varied from 100% to 300% of the baseline.

Battery (Figure 36). The battery specific energy varied from 400 Wh/kg (baseline) to 260 Wh/kg.

Mission

Range (Figures 37–38). 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. For electric aircraft, the mission range is varied for a set of battery weights (400 Wh/kg to 150 Wh/kg).

Reserve (Figure 39). The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. The reserve segment time is varied from 0 to 45 min.

Climb and one-engine-inoperative (Figure 40–41). The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min, with and without the OEI requirement.

Cruise climb (Figure 42). The cruise rate of climb is varied from 500 ft/min (baseline) to 0 (no requirement).

Altitude (Figure 43). The design mission has takeoff at 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA); or takeoff at sea level ISA and cruise at 4000 ft ISA; or both.

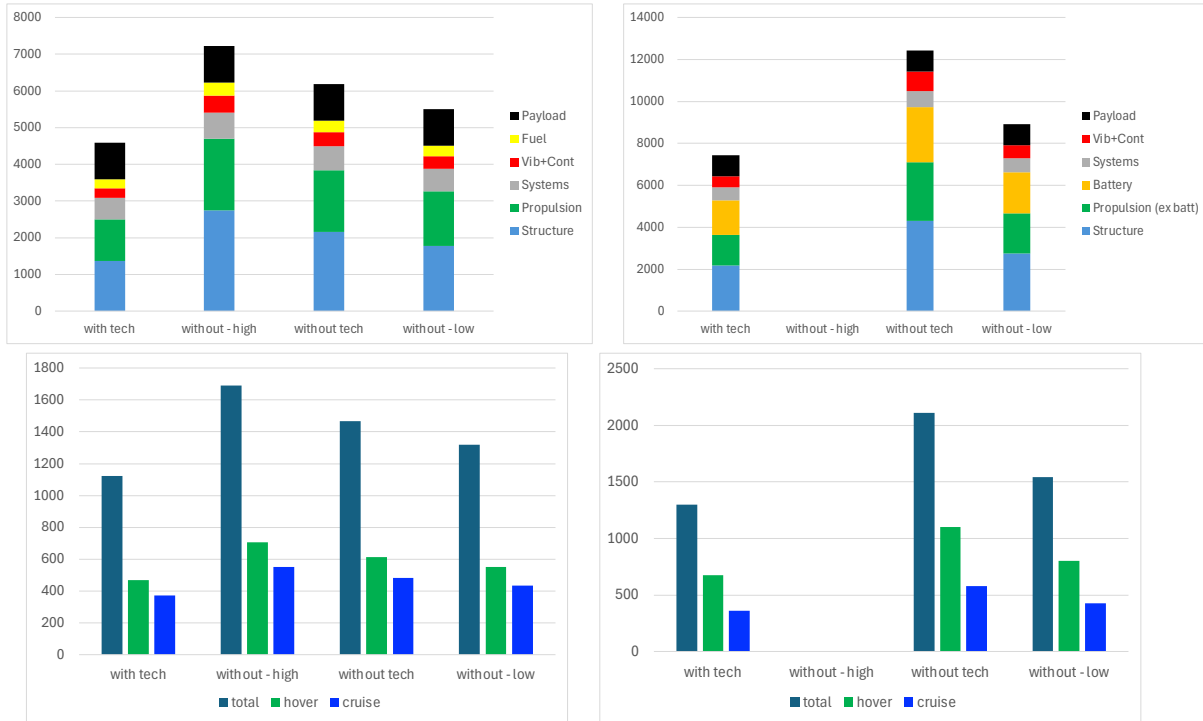


Figure 28. QSMR, turboshaft (left) and electric (right) — impact of weight technology factors. Designs with technology, compared to only calibration (no technology factors): high, baseline, and low calibration. Electric aircraft design with high calibration did not converge.

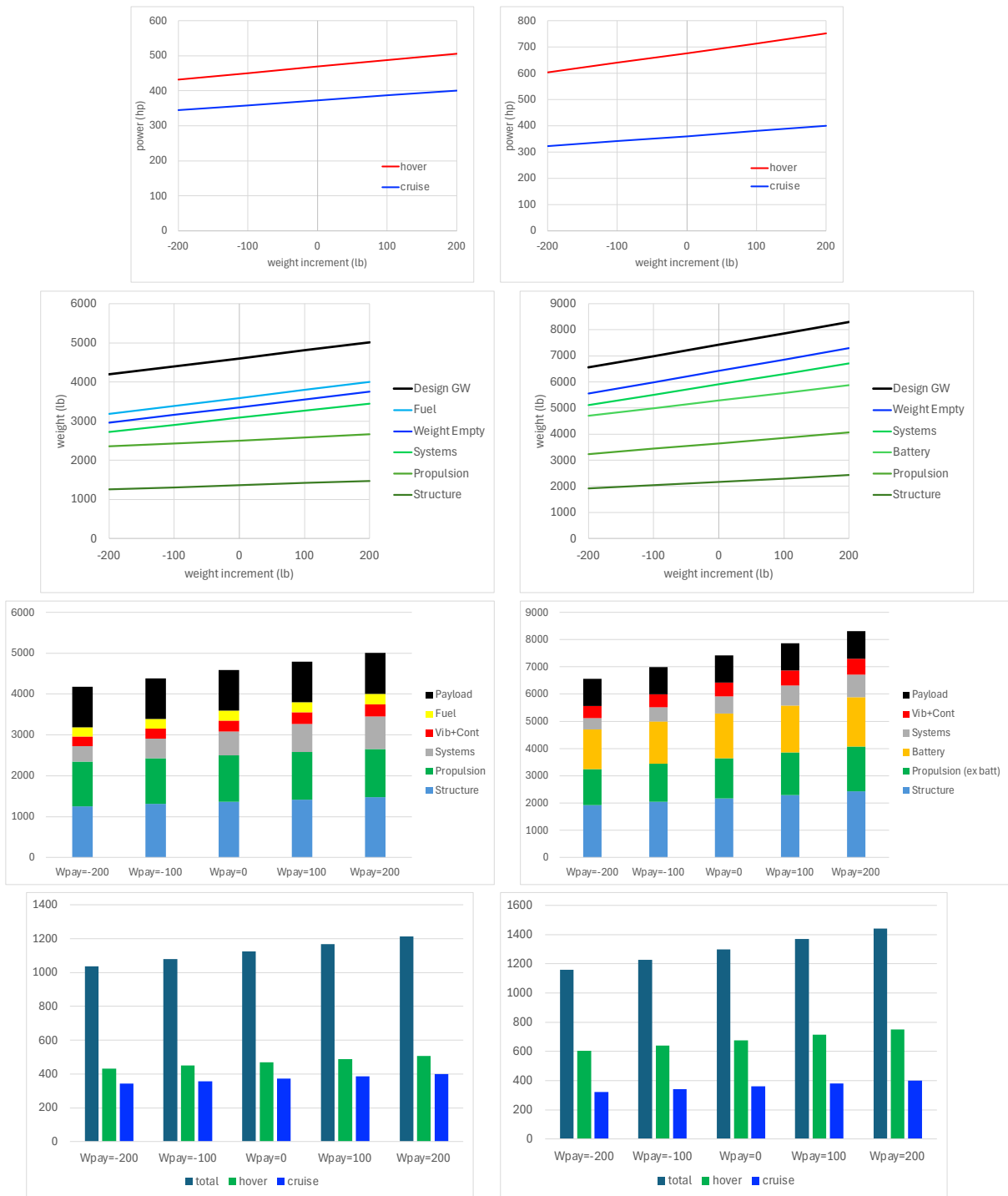


Figure 29. QSMR, turboshaft (left) and electric (right) — impact of systems and equipment weight. The systems and equipment weight changed by -200 lb to 200 lb, relative to the baseline fixed weight of 480 lb.

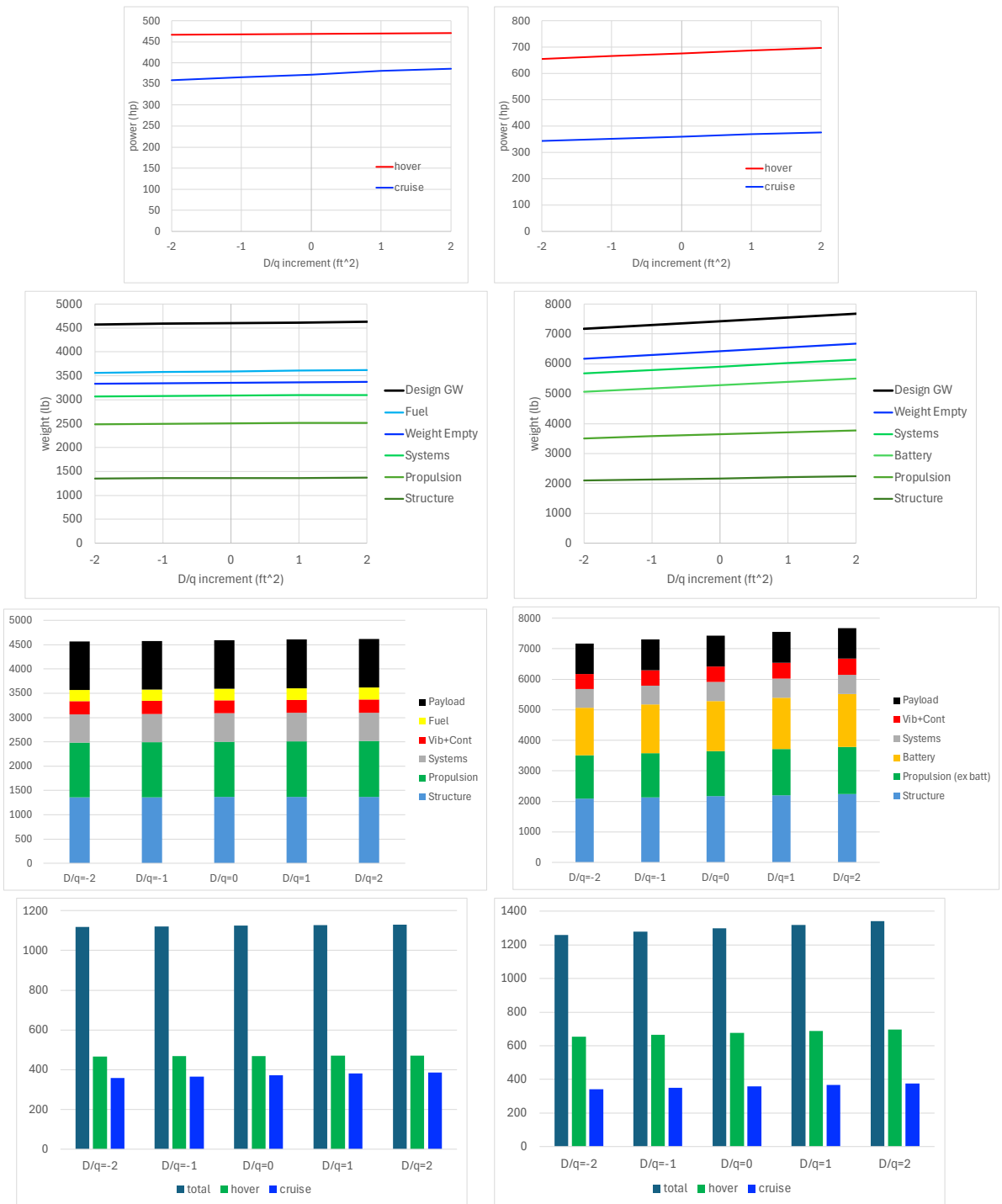


Figure 30. QSMR, turboshaft (left) and electric (right) — impact of aircraft drag. The airframe drag varied by $\Delta D/q = -2.0$ to 2.0 ft^2 .

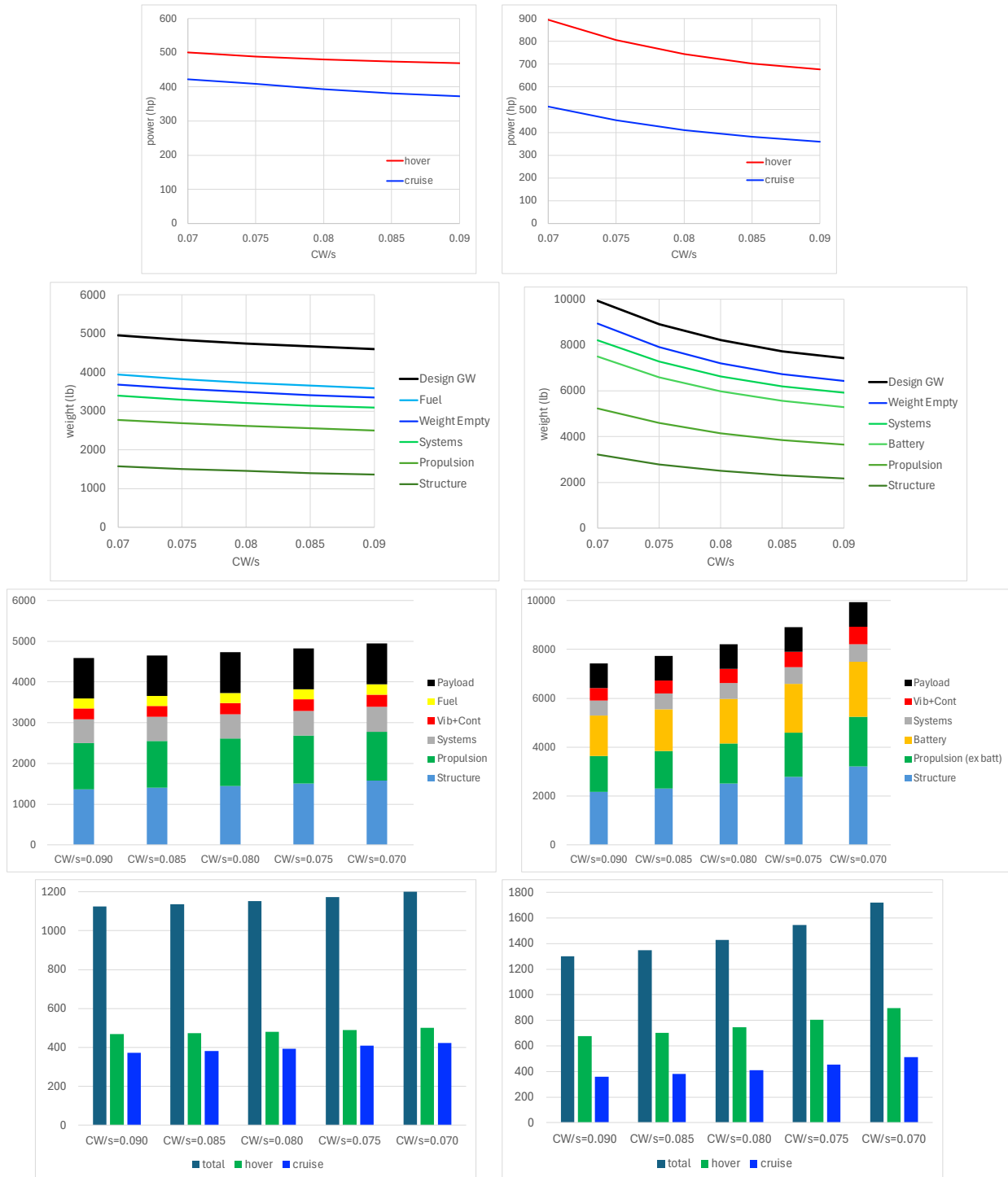


Figure 31. QSMR, turboshaft (left) and electric (right) — impact of main rotor design blade loading. The main rotor design blade loading varied from $CW/\sigma=0.09$ (baseline) to 0.07.

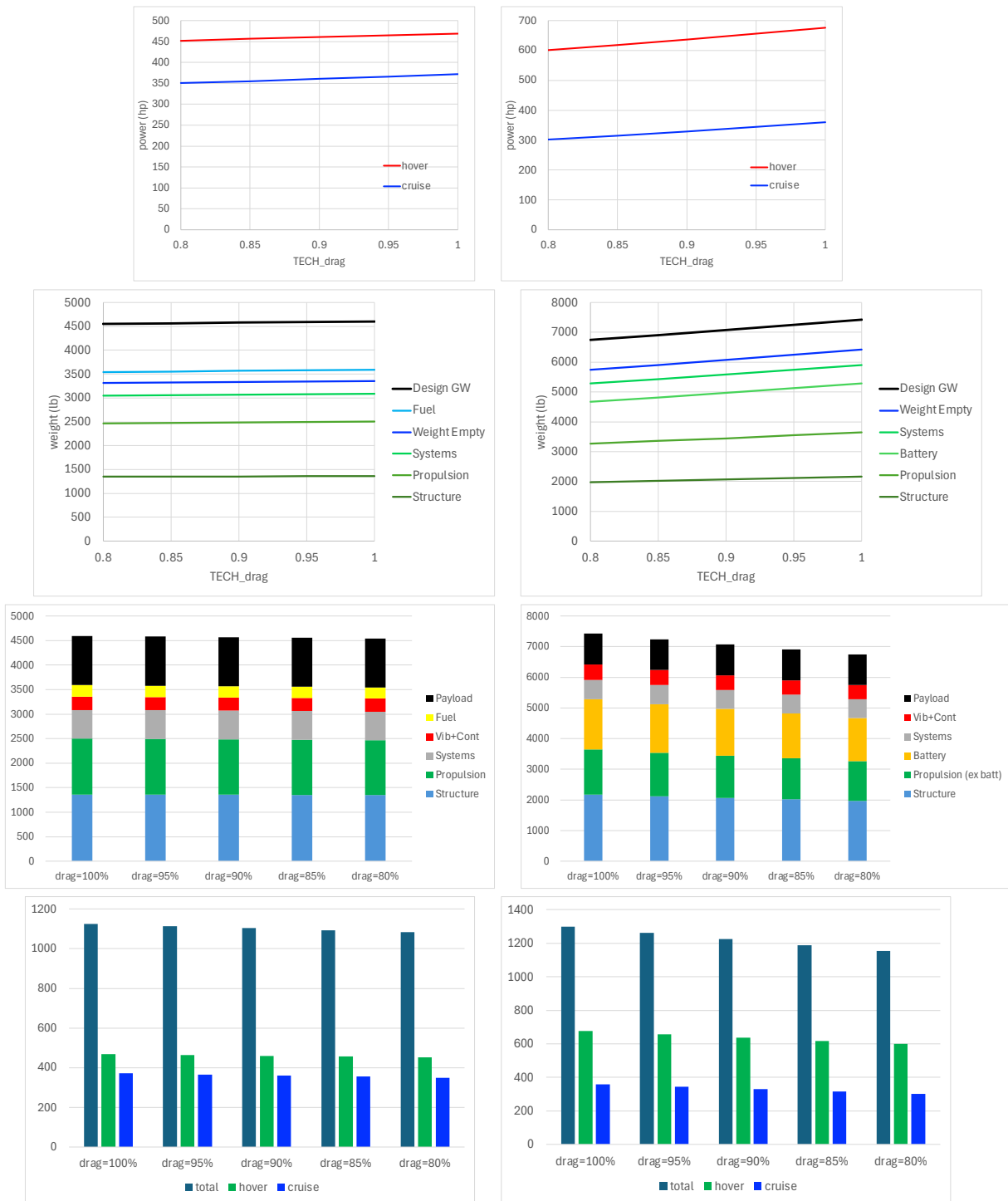


Figure 32. QSMR, turboshaft (left) and electric (right) — impact of rotor profile power. The profile power varied from 100% to 80%.

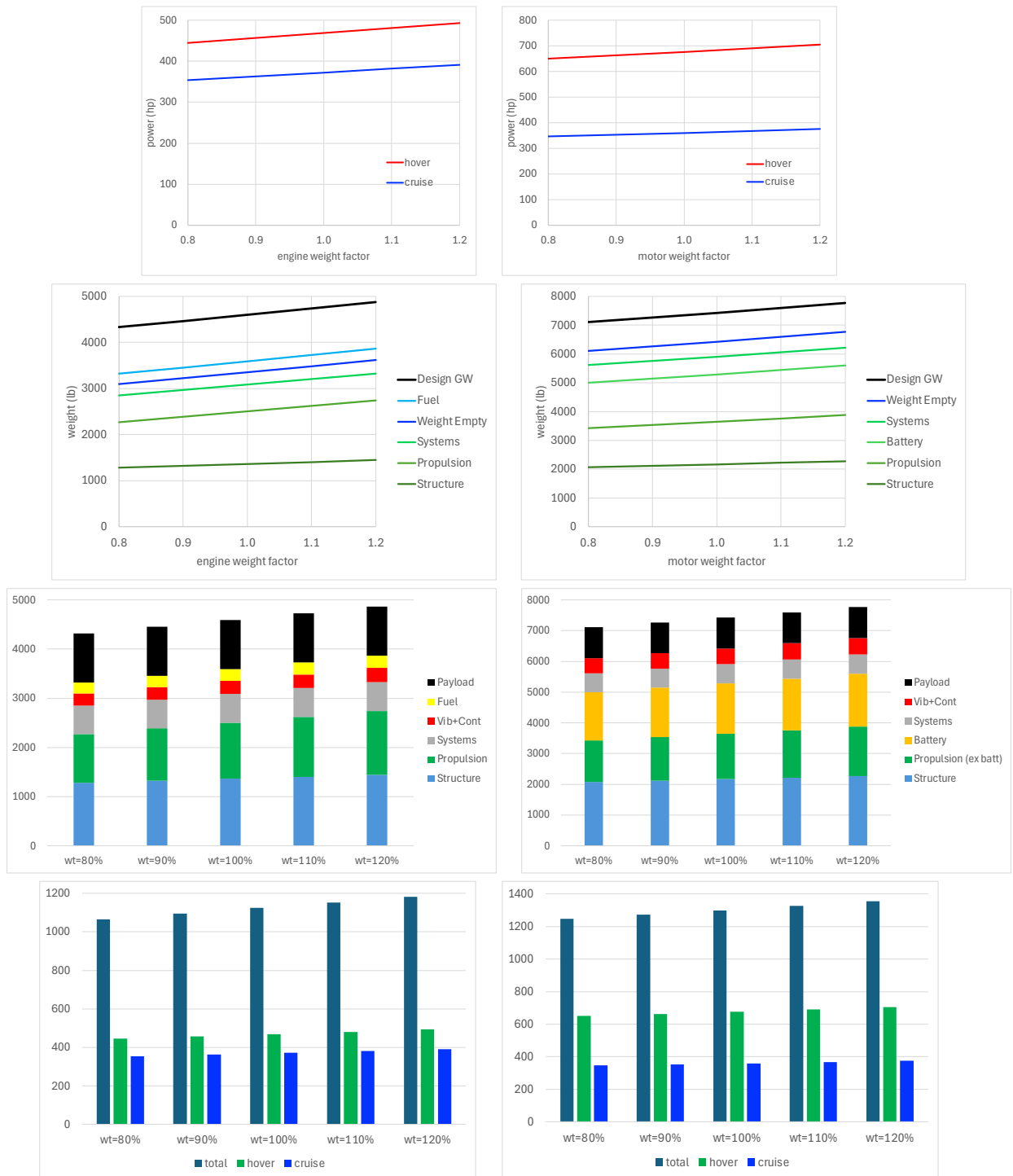


Figure 33. QSMR, turboshaft (left) and electric (right) — impact of engine weight. The turboshaft engine weight or electric motor weight varied from 80% to 120% of the baseline.

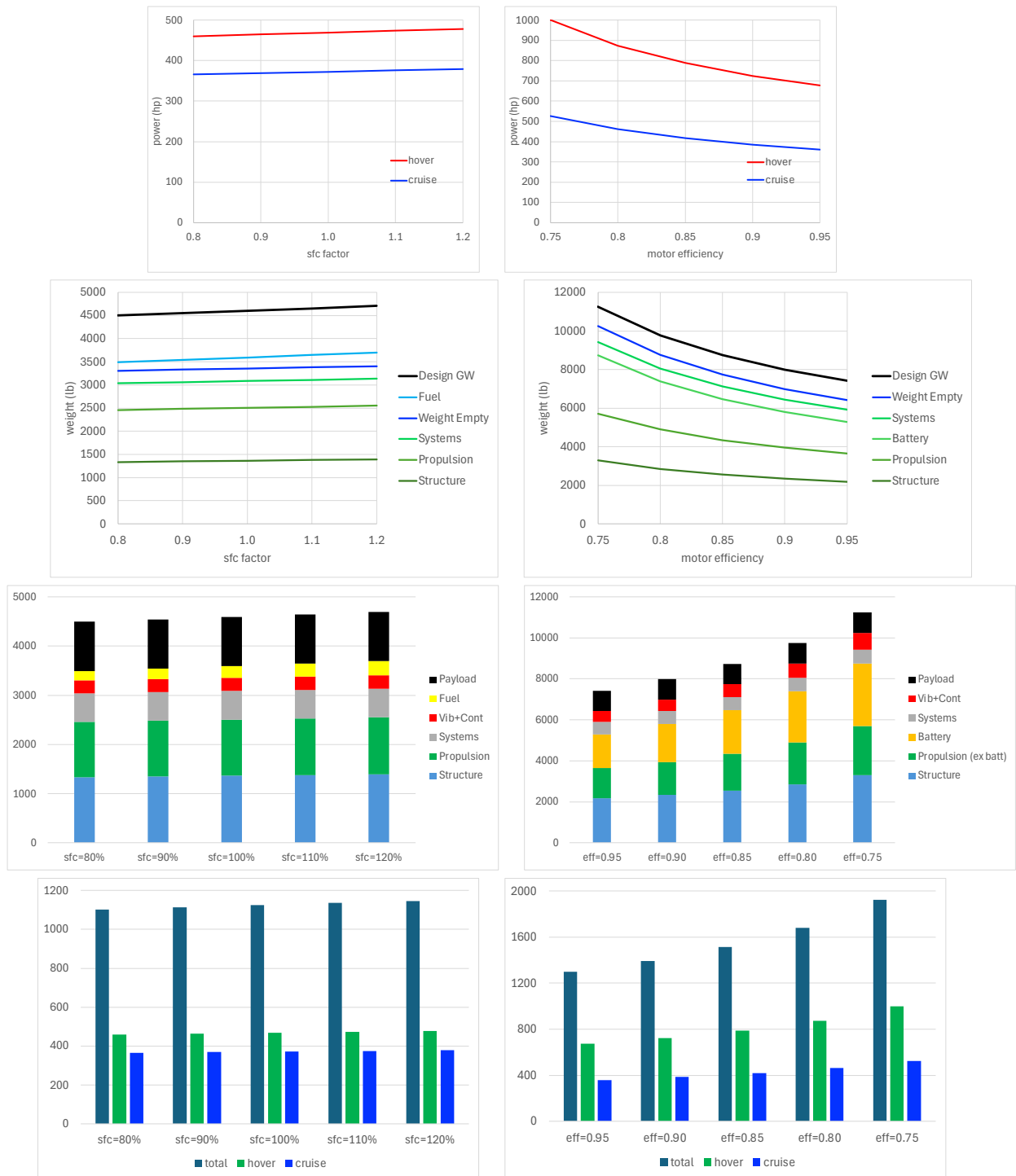


Figure 34. QSMR, turboshaft (left) and electric (right) — impact of engine efficiency. The turboshaft engine specific fuel consumption varied from 80% to 120% of the baseline (0.50 lb/hp-hr MCP SLS). The motor efficiency varied from 0.95 (baseline) to 0.75.

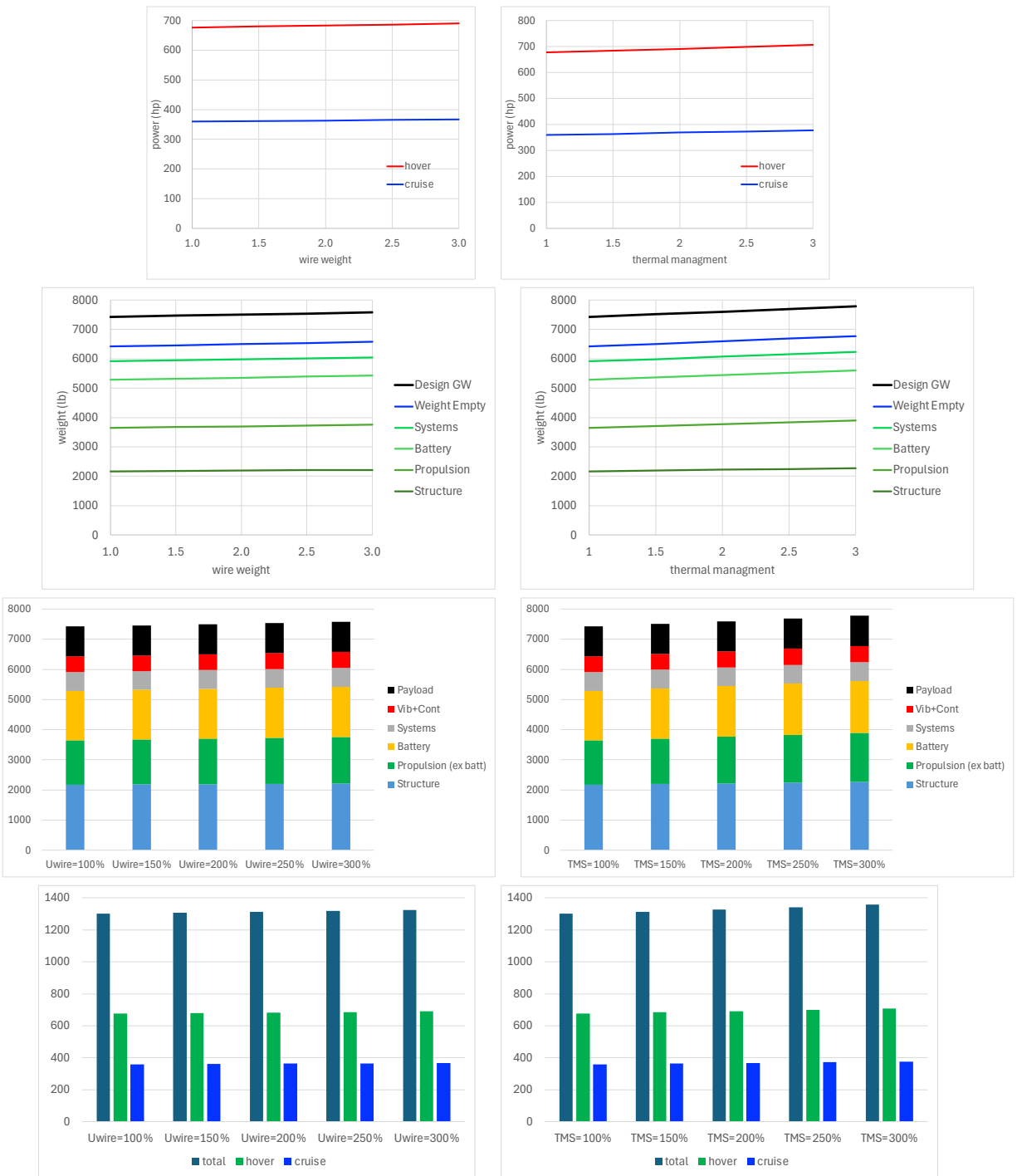


Figure 34. QSMR, electric — impact of wiring weight (left) and thermal management system (right). The wiring weight varied from 100% to 300% of the baseline. The thermal management system design rejected power (motor and battery) varied from 100% to 300% of the baseline.

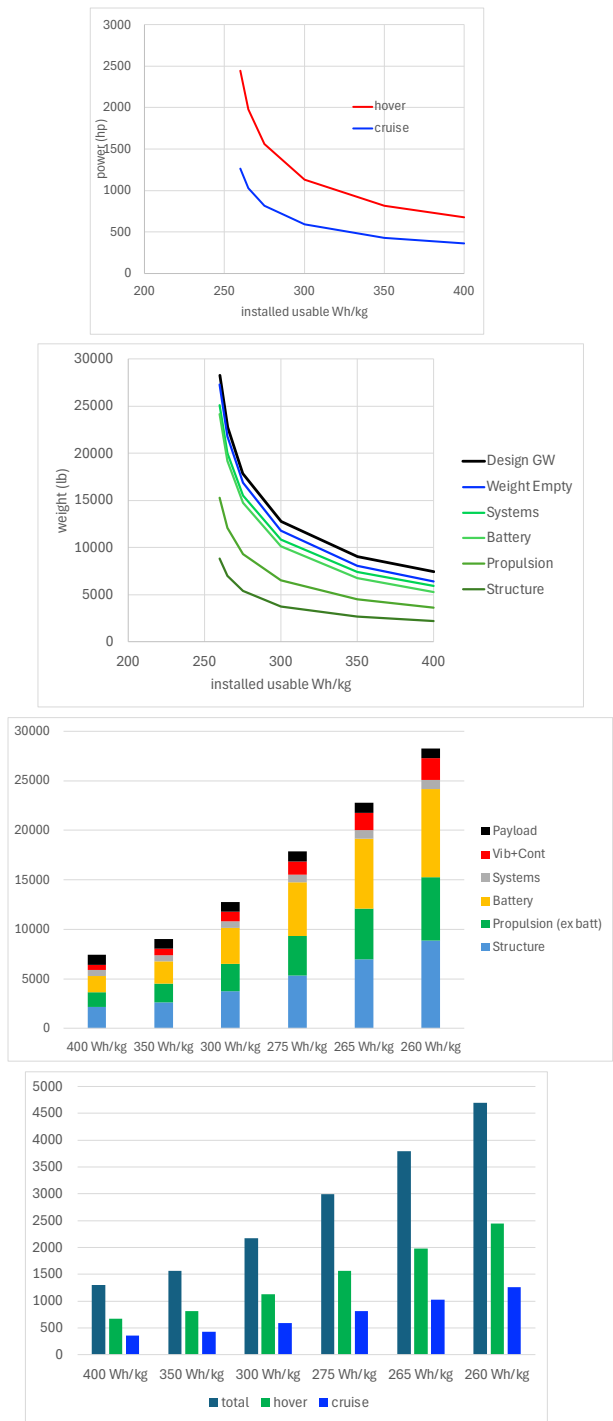


Figure 36. QSMR, electric — impact of battery specific energy. The battery specific energy varied from 400 Wh/kg (baseline) to 260 Wh/kg.

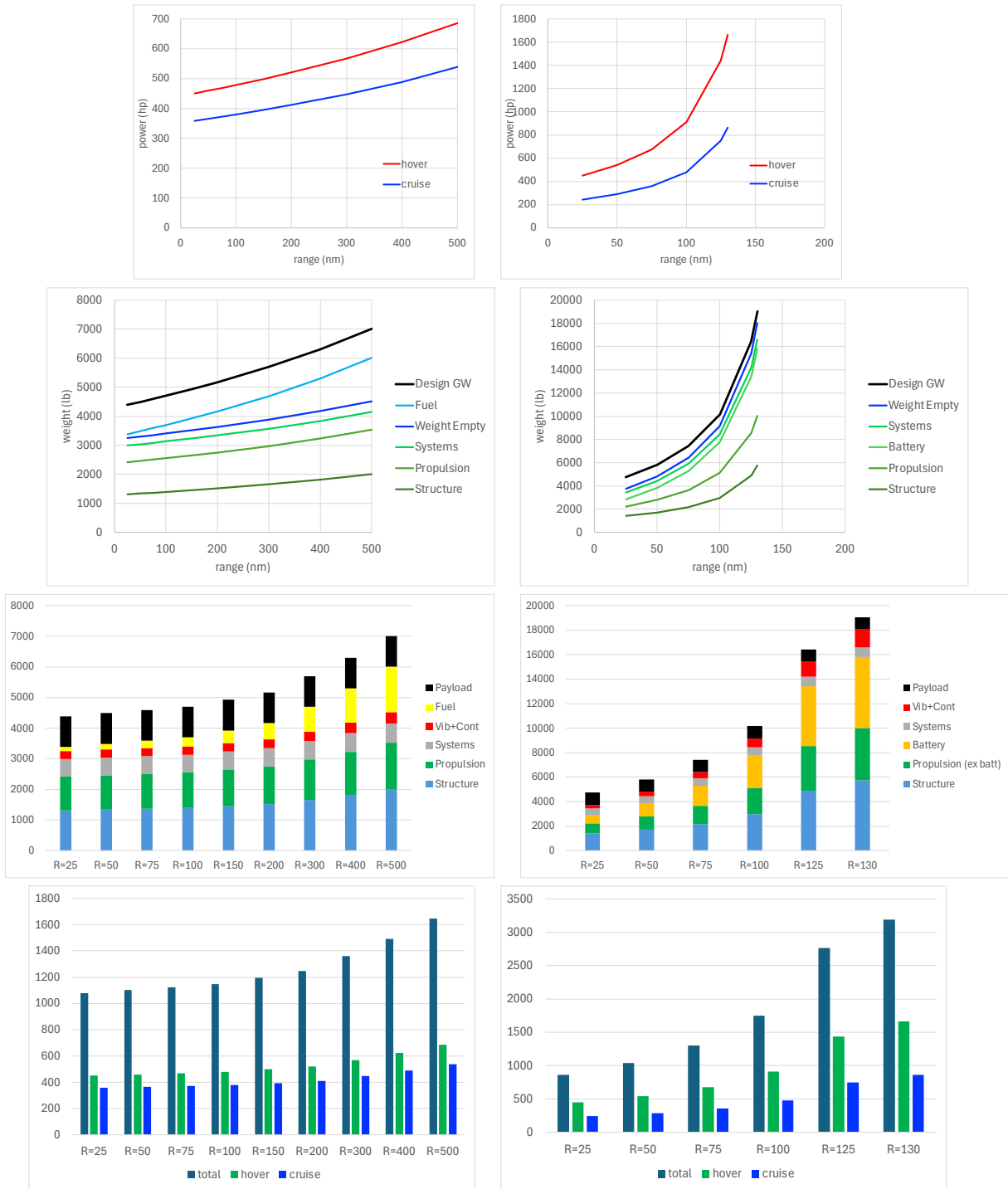


Figure 37. QSMR, turboshaft (left) and electric (right) — impact of mission range. The baseline design mission has a range of 75 nm (in two segments). The range is varied from 25 to 500 nm for the turboshaft configurations, and from 25 to 130 nm for the electric configurations.

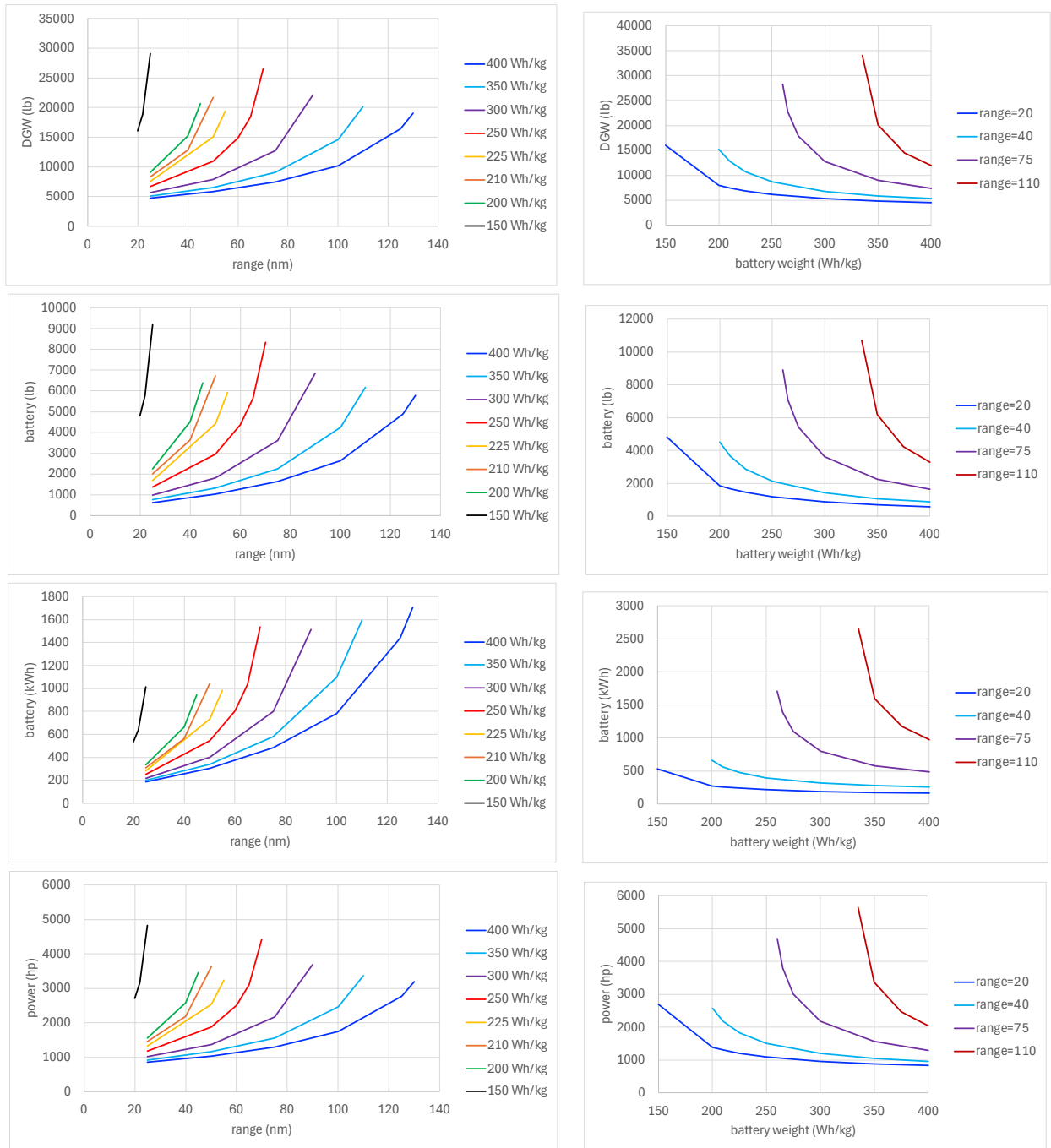


Figure 38. QSMR, electric — impact of mission range (20 to 130 nm) and battery weight (400 to 150 Wh/kg); design gross weight (lb), battery weight (lb), battery capacity (kWh), and installed power (hp). The baseline design mission has a range of 75 nm (in two segments), with battery weight 400 Wh/kg.

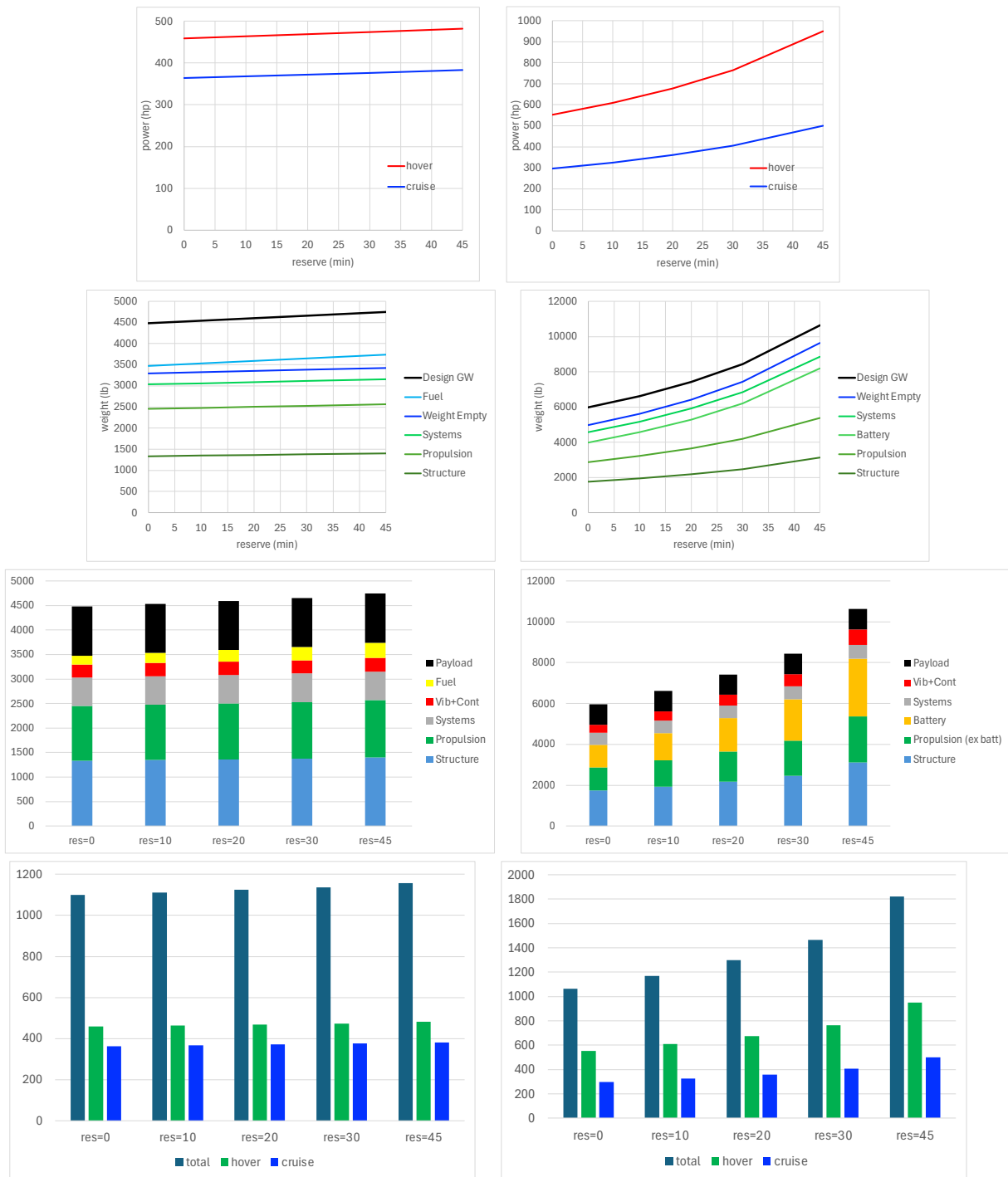


Figure 39. QSMR, turboshaft (left) and electric (right) — impact of mission reserve. The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. The reserve segment time is varied from 0 to 45 min.

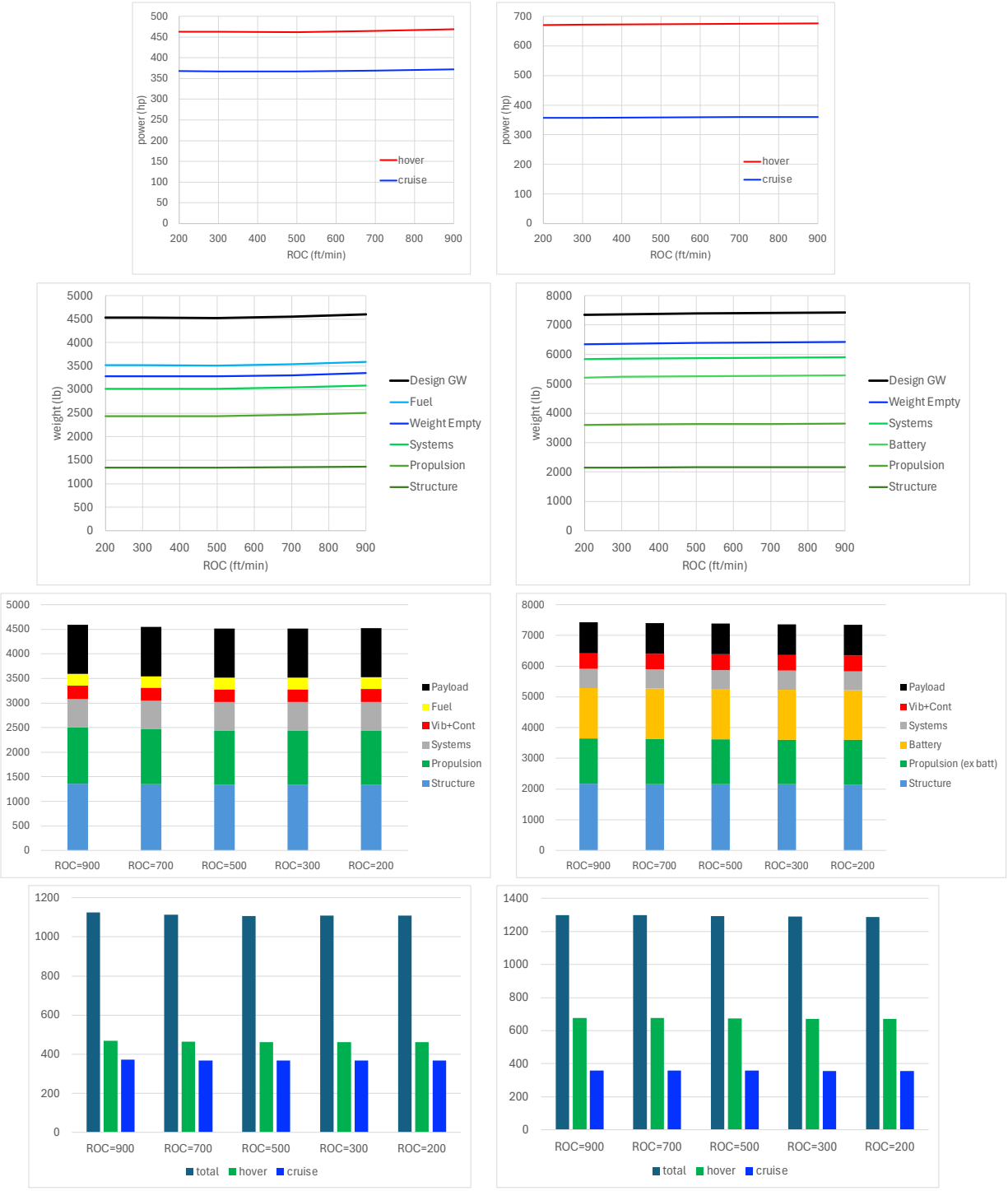


Figure 40. QSMR, turboshaft (left) and electric (right) — impact of mission climb to cruise altitude, with one-engine-inoperative requirement. The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min.

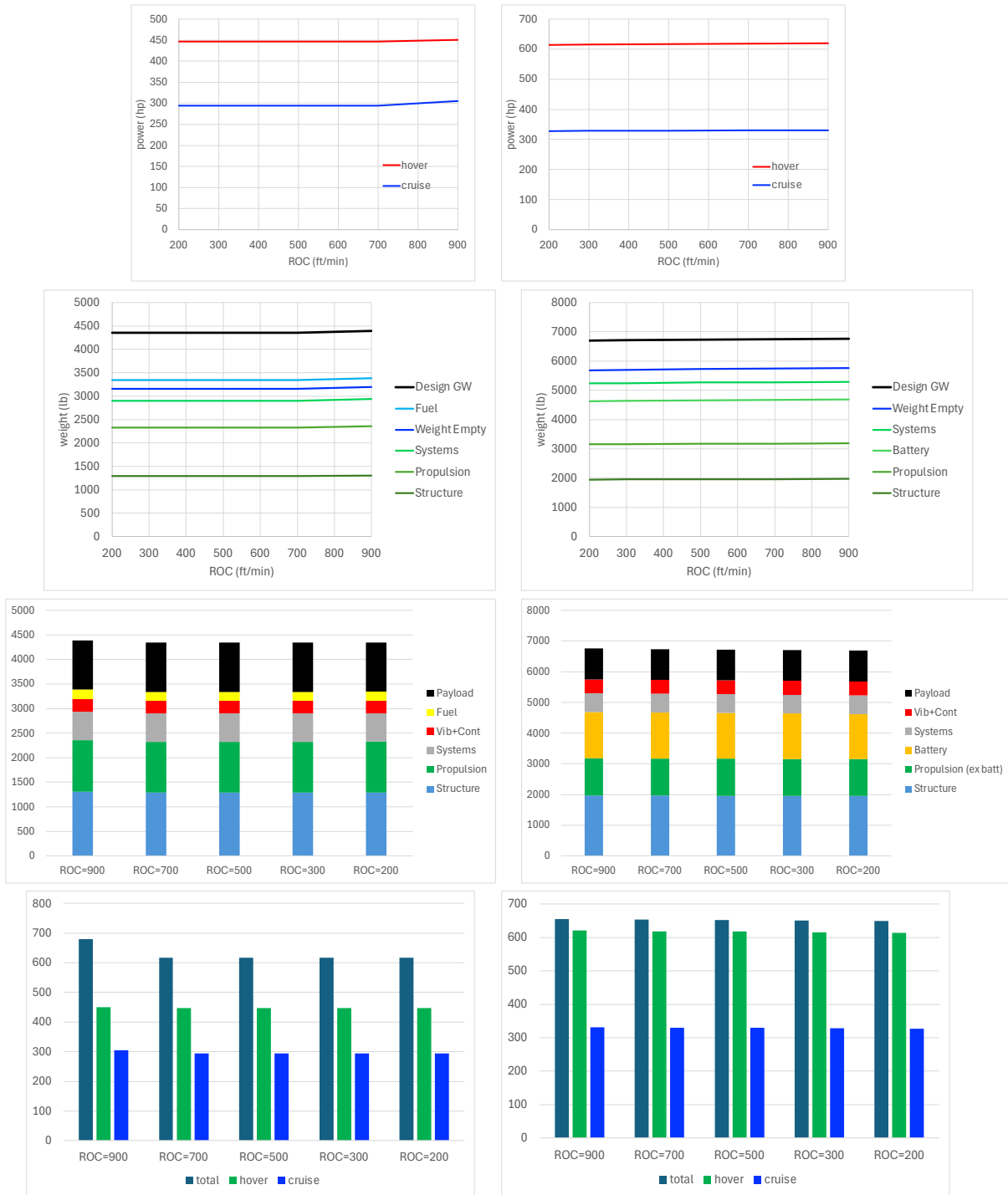


Figure 41. QSMR, turboshaft (left) and electric (right) — impact of mission climb to cruise altitude, without one-engine-inoperative requirement.

The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min.

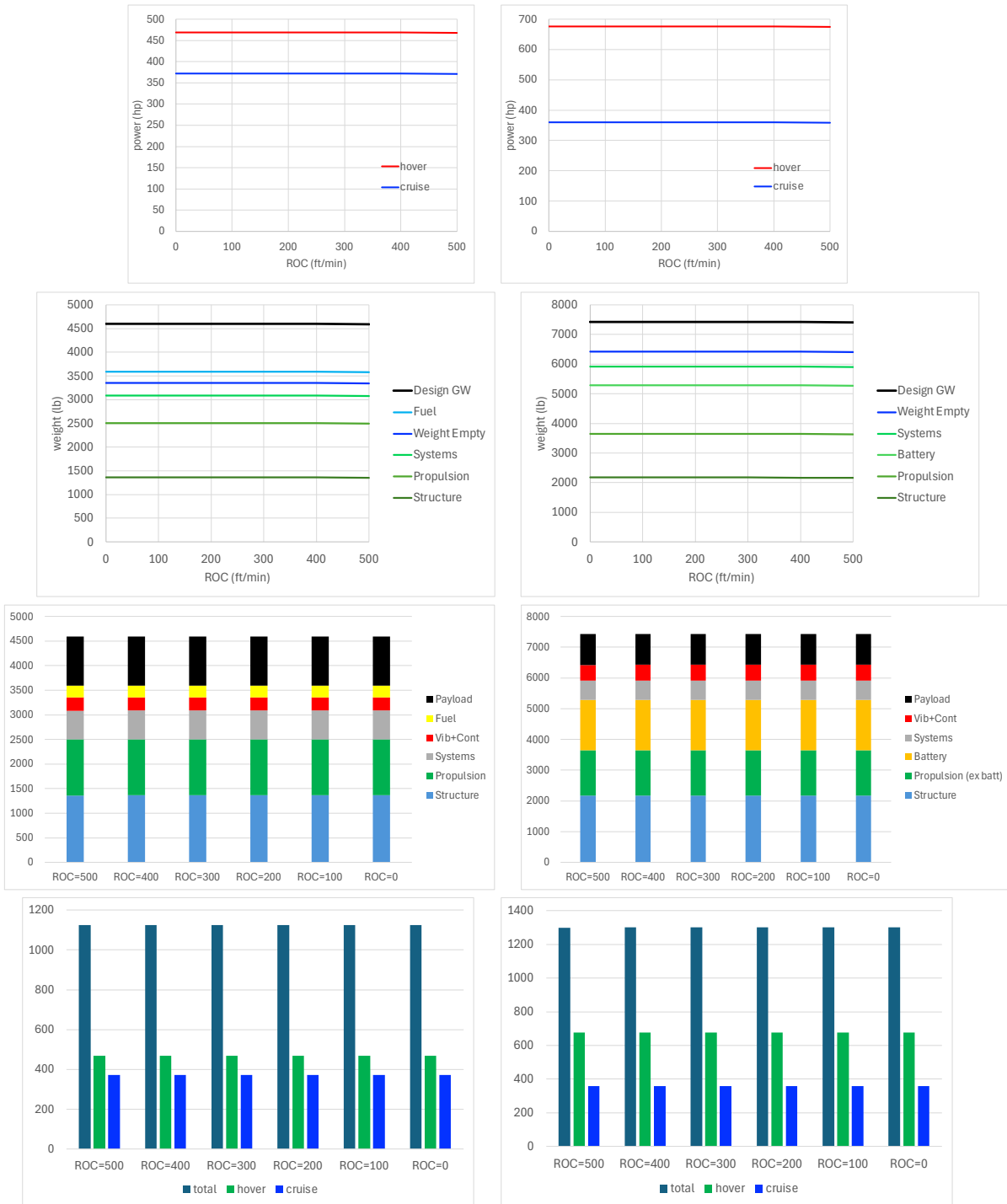


Figure 42. QSMR, turboshaft (left) and electric (right) — impact of cruise climb. The cruise rate of climb is varied from 500 ft/min (baseline) to 0 (no requirement).

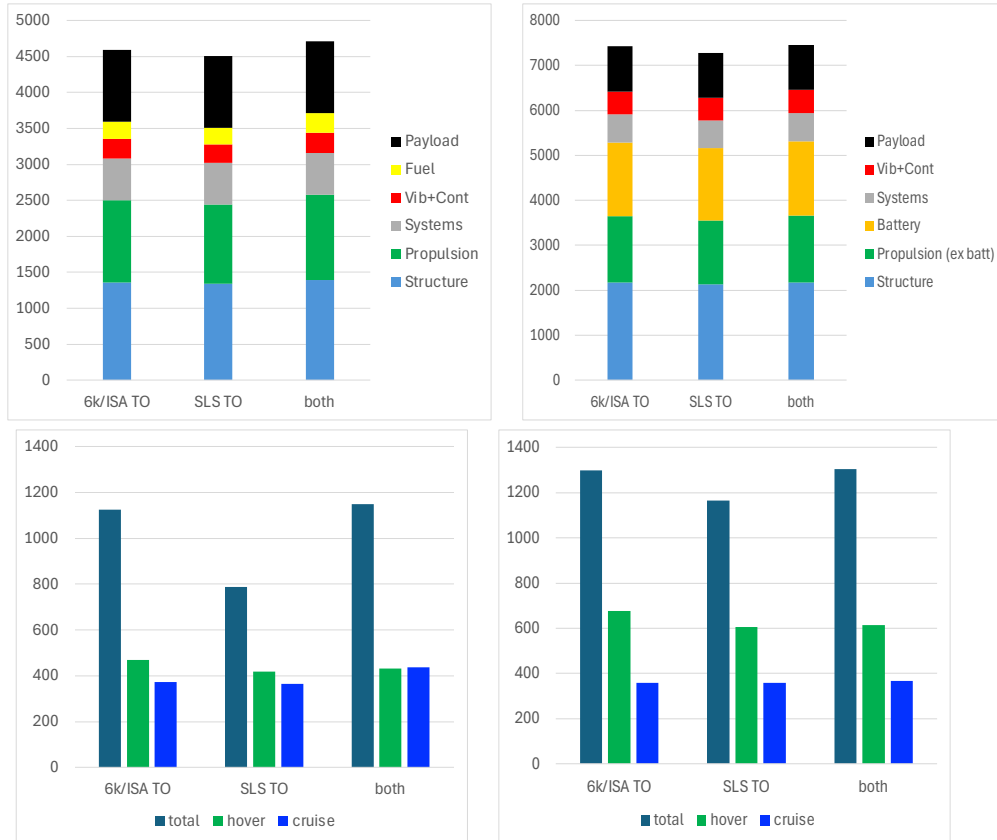


Figure 42. QSMR, turboshaft (left) and electric (right) — impact of operating altitude. The design mission has takeoff at 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA); or takeoff at sea level ISA and cruise at 4000 ft ISA; or both.

Impact of Technology and Mission Variations — Side-by-Side

Technology

Weight technology factors (Figure 43). Designs with technology, compared to only calibration (no technology factors) — high, baseline, and low calibration.

Rotor support weight (Figure 44). Technology factor varied from 50% to 175% of baseline (factor = 0.8).

Systems and equipment (Figure 45). The systems and equipment weight changed by –200 lb to 200 lb, relative to the baseline fixed weight of 480 lb.

Aircraft drag (Figure 46). The airframe drag varied by $\Delta D/q = -2.0$ to 2.0 ft^2 .

Blade loading (Figure 47). The main rotor design blade loading varied from $C_W/\sigma = 0.09$ (baseline) to 0.07.

Profile power (Figure 48). The profile power varied from 100% to 80%.

Engine weight (Figure 49). The turboshaft engine weight and electric motor weight varied from 80% to 120% of the baseline.

Engine efficiency (Figure 50). The turboshaft engine specific fuel consumption varied from 80% to 120% of the baseline (0.50 lb/hp-hr MCP SLS). The motor efficiency varied from 0.95 (baseline) to 0.75.

Electric motor (Figures 51). The wiring weight varied from 100% to 300% of the baseline. The thermal management system design rejected power (motor and battery) varied from 100% to 300% of the baseline.

Battery (Figure 52). The battery specific energy varied from 400 Wh/kg (baseline) to 310 Wh/kg.

Mission

Range (Figures 53–54). 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. For electric aircraft, the mission range is varied for a set of battery weights (400 Wh/kg to 150 Wh/kg).

Reserve (Figure 55). The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. The reserve segment time is varied from 0 to 45 min.

Climb and one-engine-inoperative (Figure 56–57). The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min, with and without the OEI requirement.

Cruise climb (Figure 58). The cruise rate of climb is varied from 500 ft/min (baseline) to 0 (no requirement).

Altitude (Figure 59). The design mission has takeoff at 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA); or takeoff at sea level ISA and cruise at 4000 ft ISA; or both.

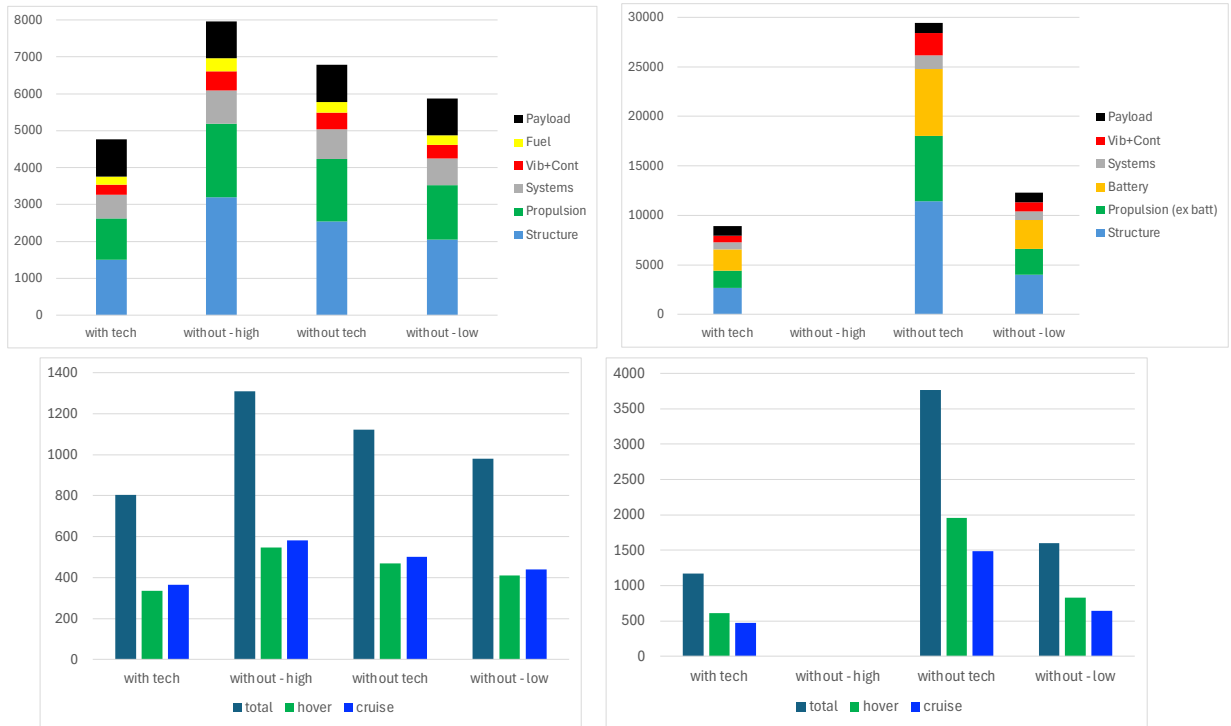


Figure 43. Side-by-side, turboshaft (left) and electric (right) — impact of weight technology factors. Designs with technology, compared to only calibration (no technology factors): high, baseline, and low calibration. Electric aircraft design with high calibration did not converge.

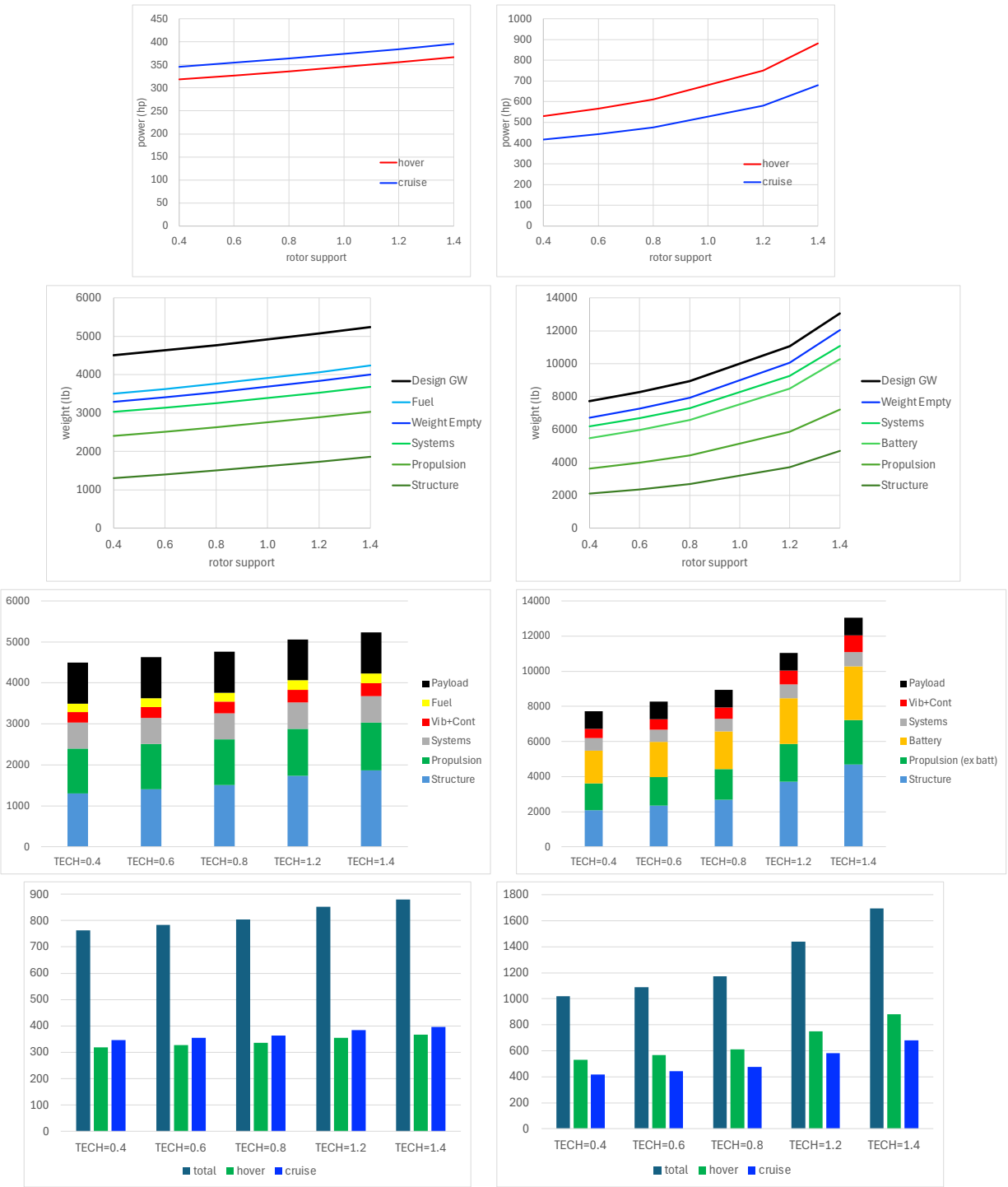


Figure 44. Side-by-side, turboshaft (left) and electric (right) — impact of rotor support weight. Technology factor varied from 50% to 175% of baseline (factor = 0.8).

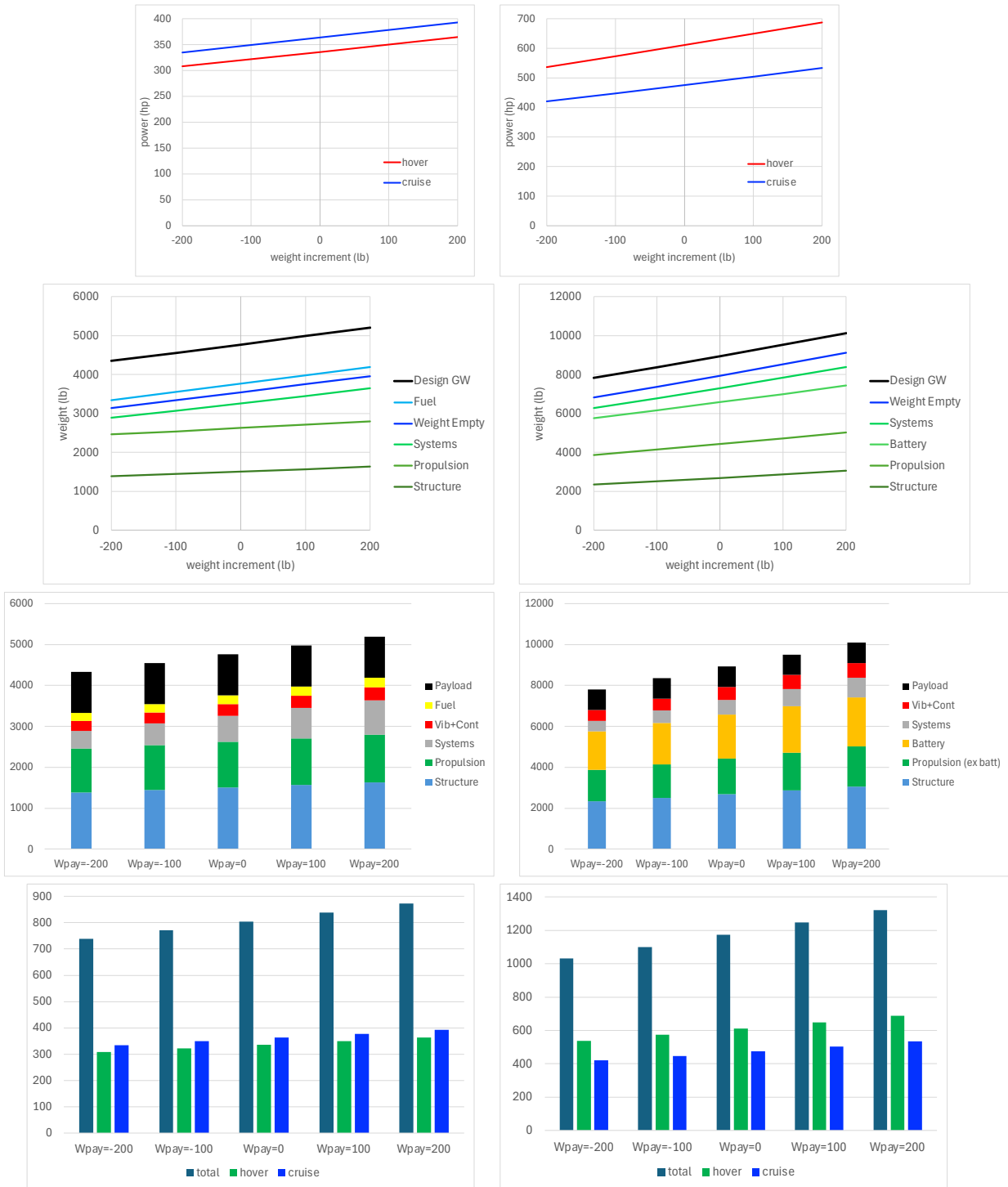


Figure 45. Side-by-side, turboshaft (left) and electric (right) — impact of systems and equipment weight. The systems and equipment weight changed by -200 lb to 200 lb, relative to the baseline fixed weight of 480 lb.

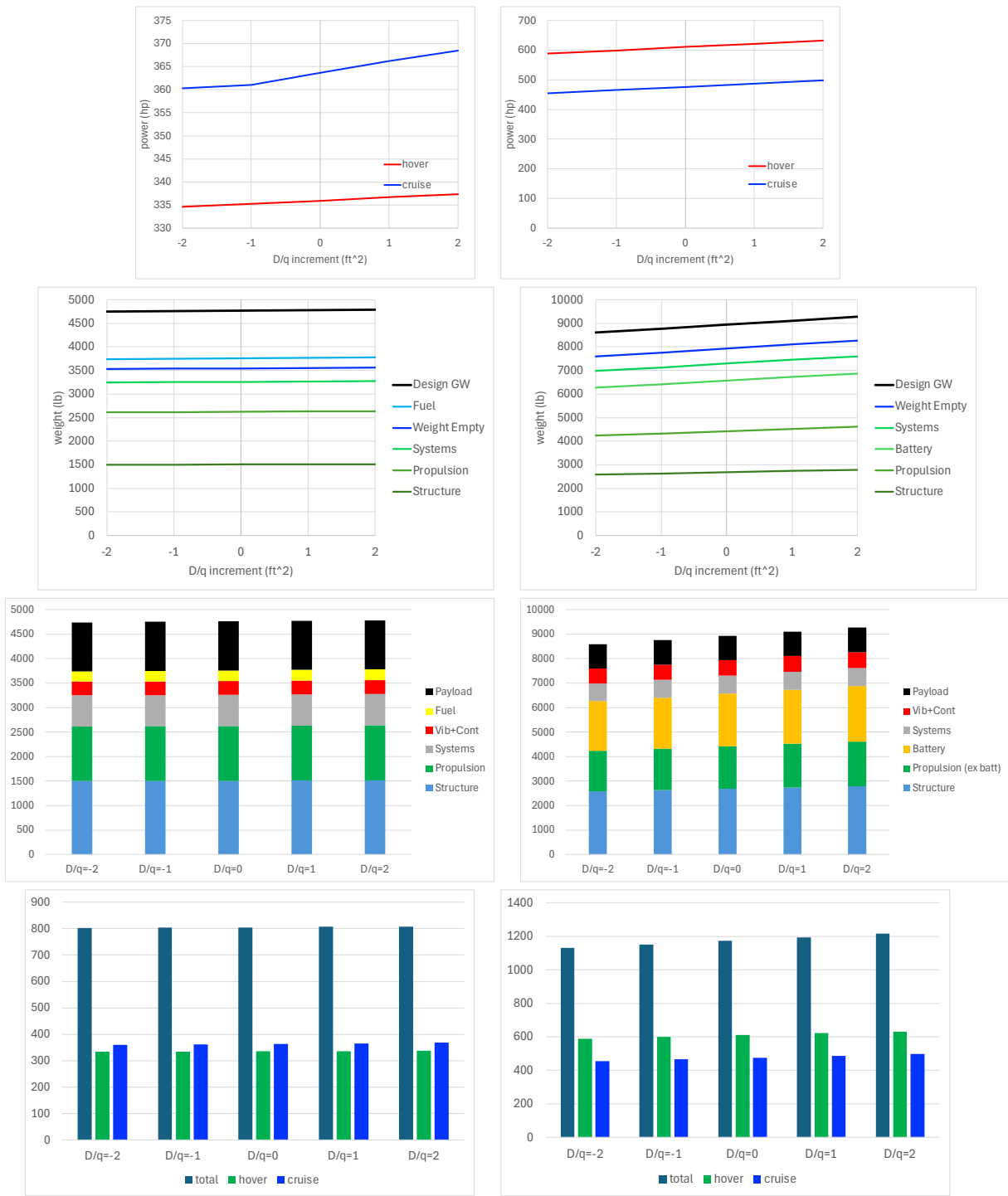


Figure 46. Side-by-side, turboshaft (left) and electric (right) — impact of aircraft drag. The airframe drag varied by $\Delta D/q = -2.0$ to 2.0 ft².

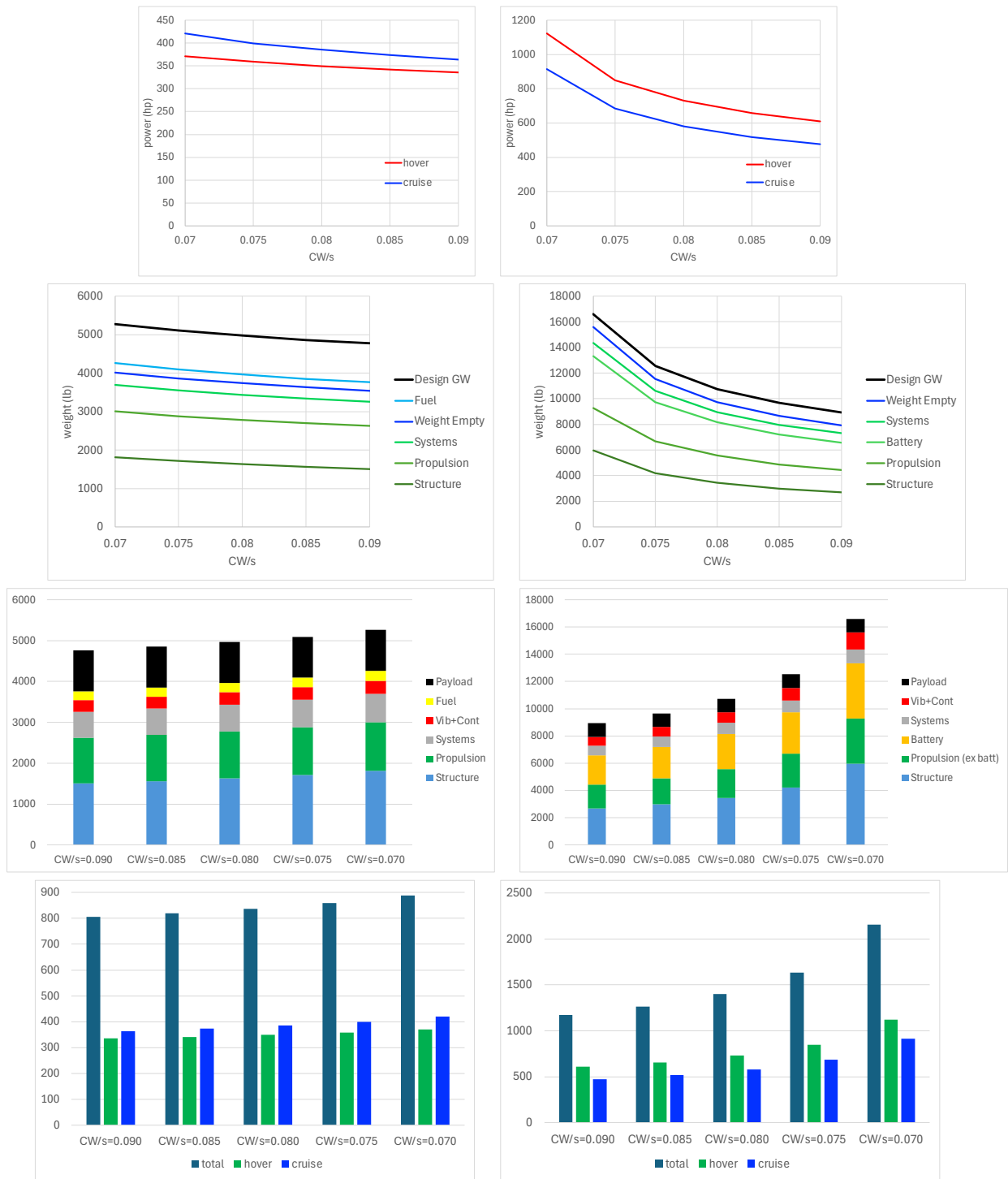


Figure 47. Side-by-side, turboshaft (left) and electric (right) — impact of main rotor design blade loading. The main rotor design blade loading varied from $C_w/\sigma=0.09$ (baseline) to 0.07.

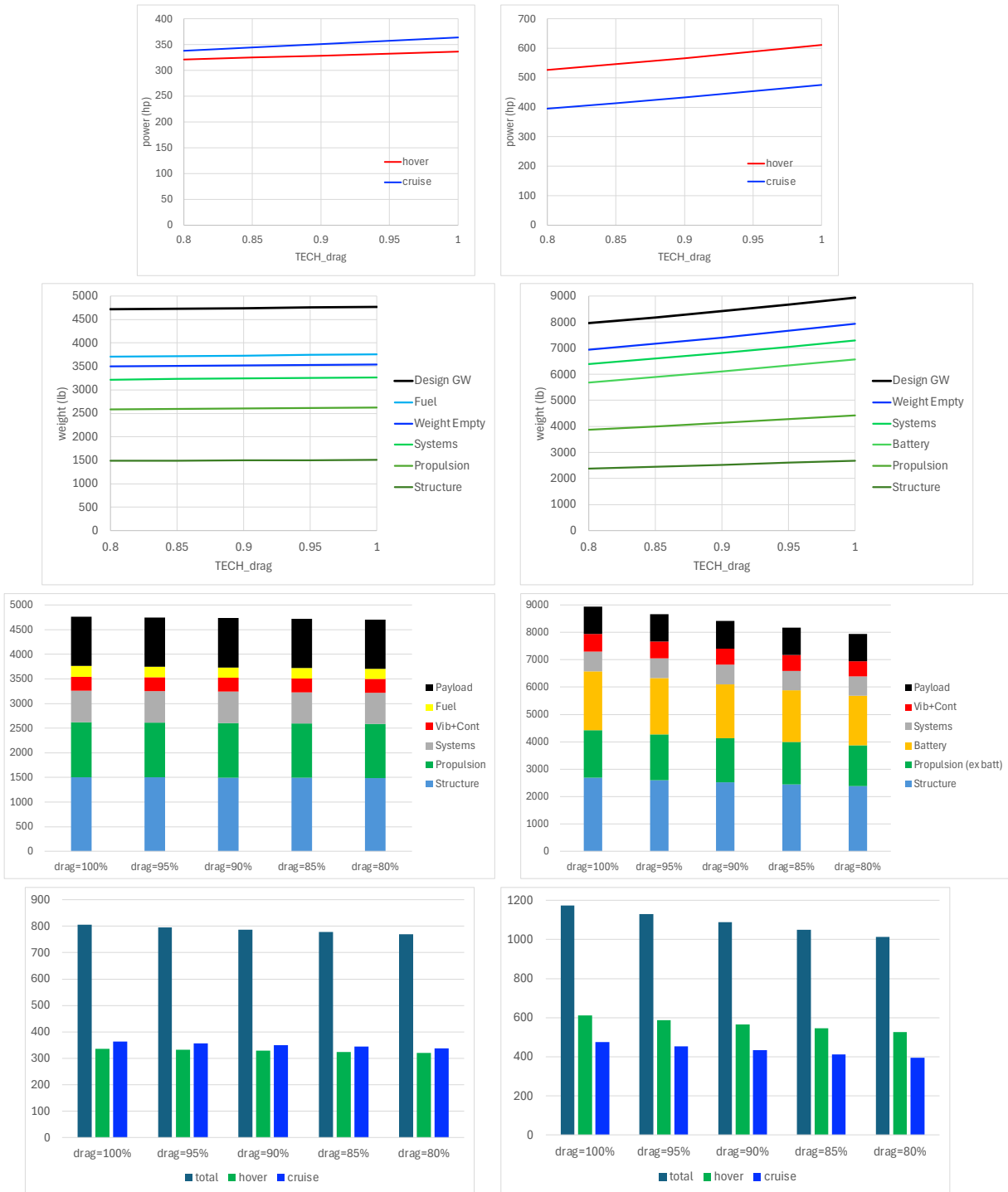


Figure 48. Side-by-side, turboshaft (left) and electric (right) — impact of rotor profile power. The profile power varied from 100% to 80%.

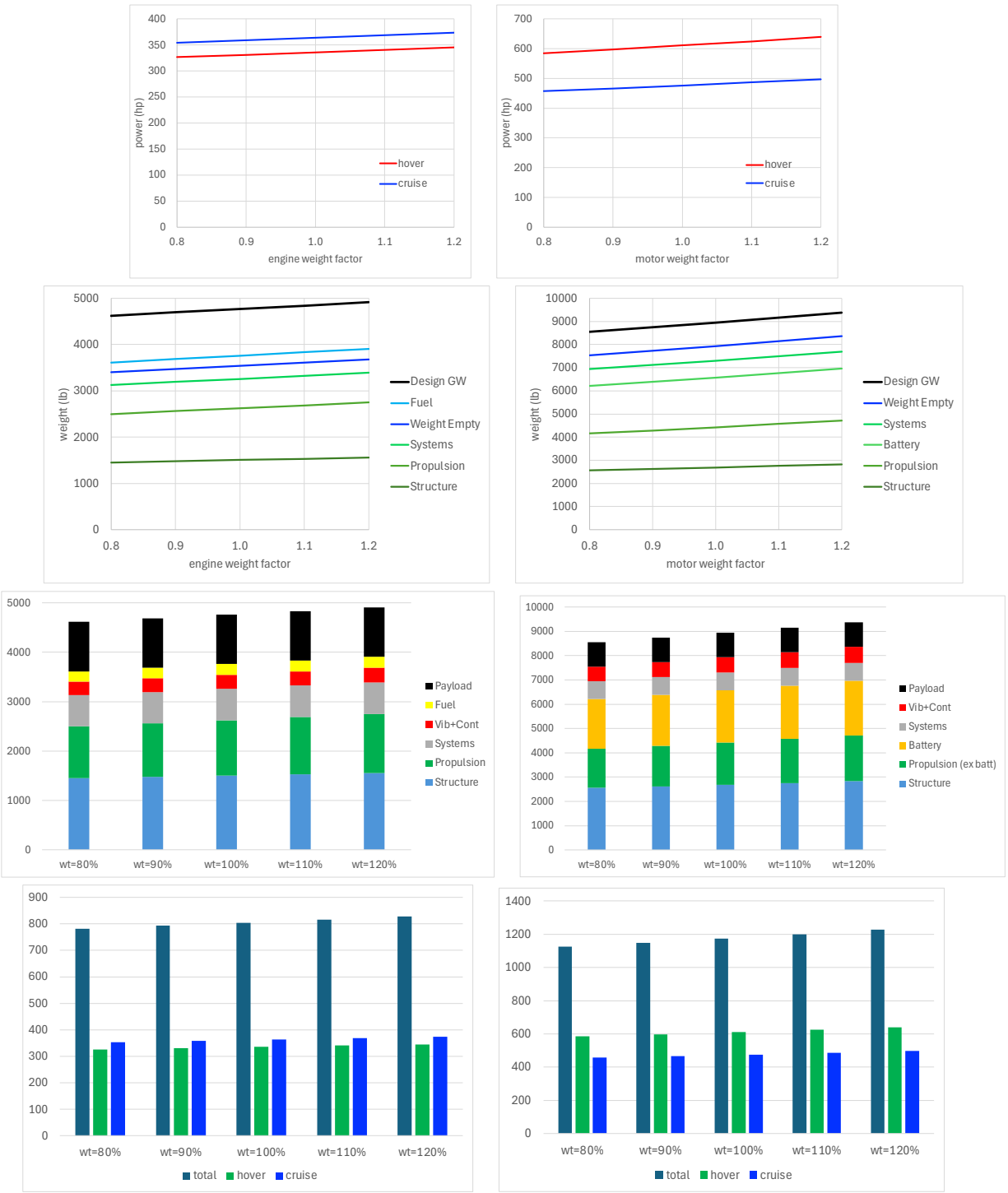


Figure 49. Side-by-side, turboshaft (left) and electric (right) — impact of engine weight. The turboshaft engine weight or electric motor weight varied from 80% to 120% of the baseline.

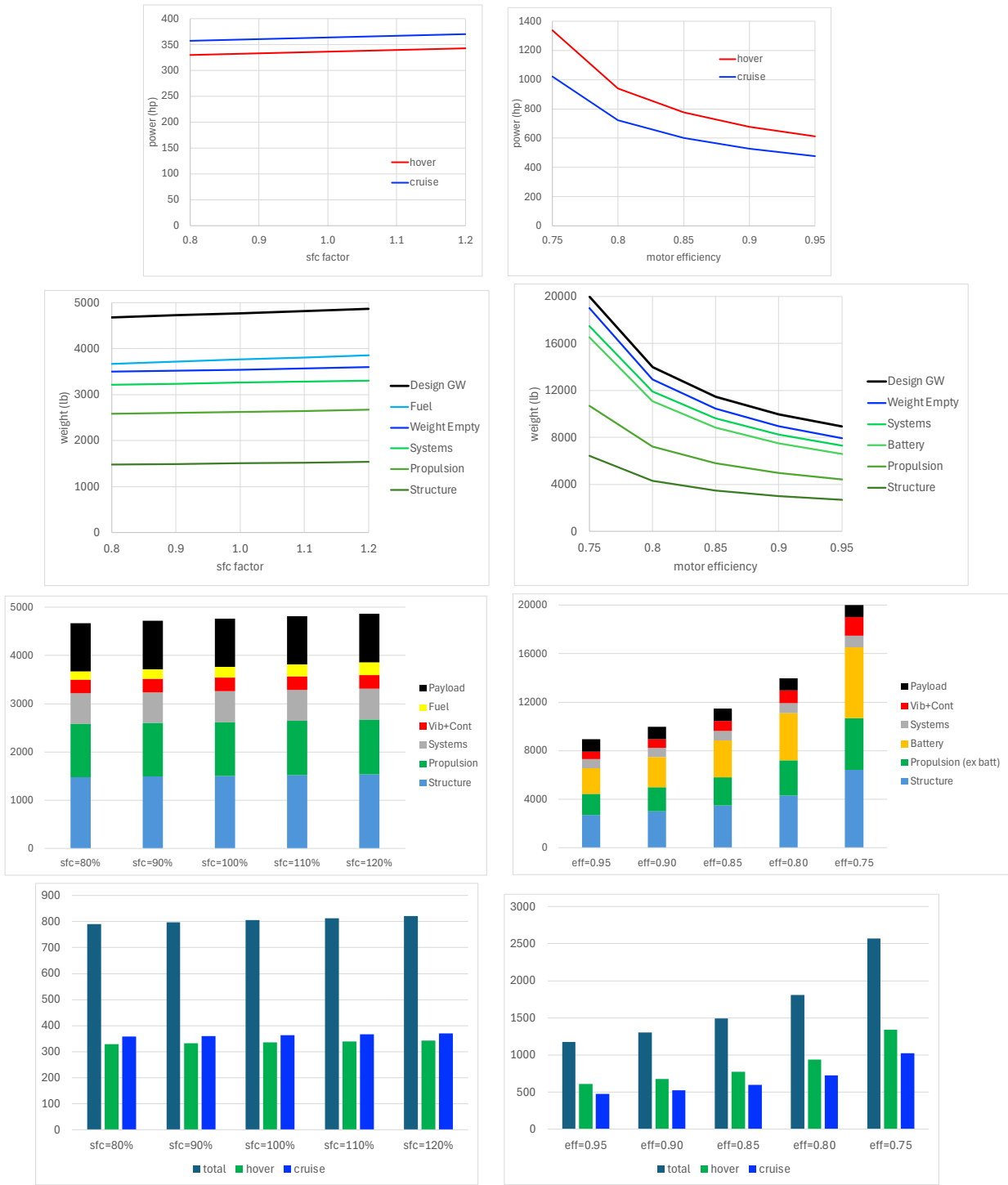


Figure 50. Side-by-side, turboshaft (left) and electric (right) — impact of engine efficiency. The turboshaft engine specific fuel consumption varied from 80% to 120% of the baseline (0.50 lb/hp-hr MCP SLS). The motor efficiency varied from 0.95 (baseline) to 0.75.

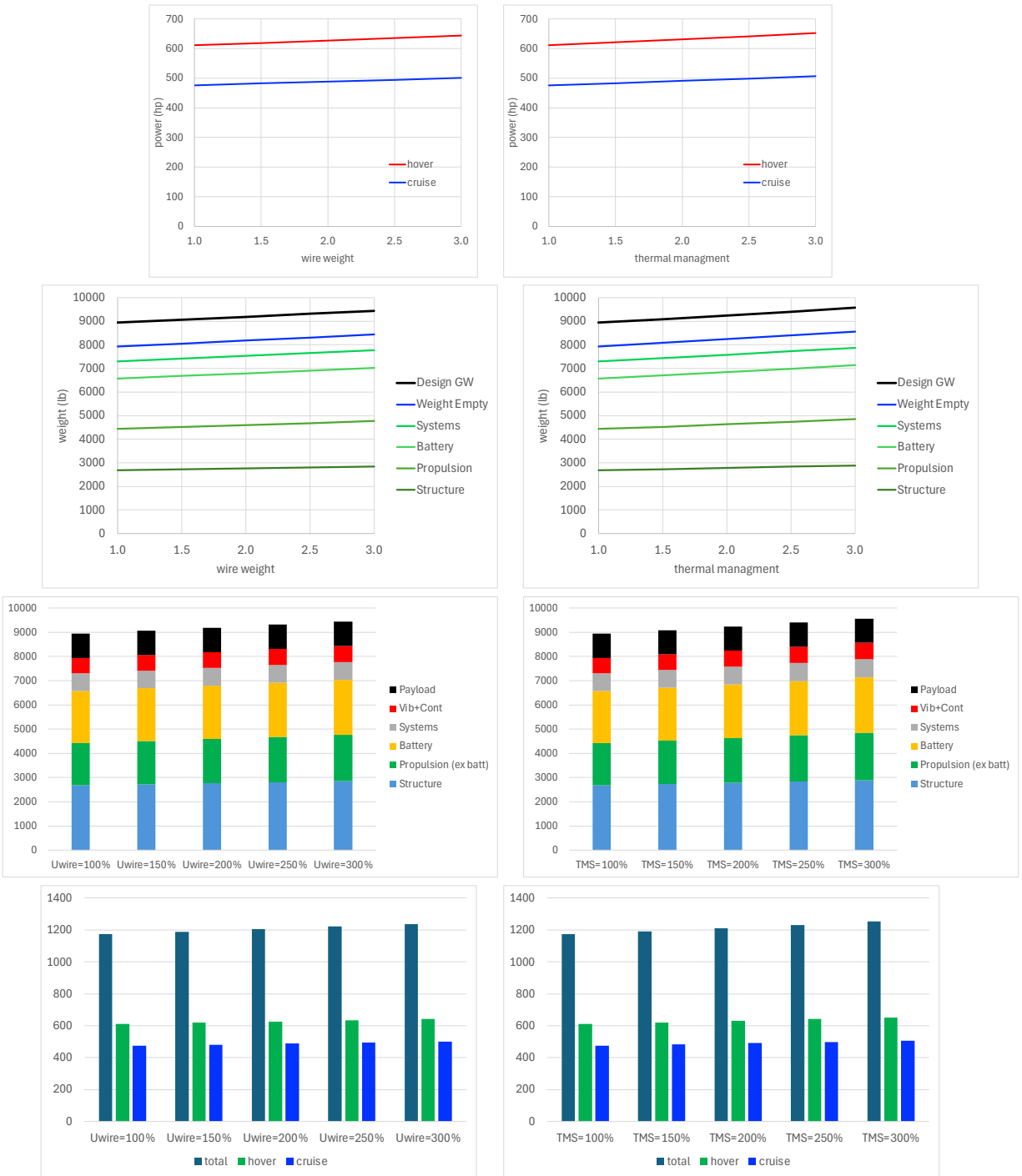


Figure 51. Side-by-side, electric—impact of wiring weight (left) and thermal management system (right). The wiring weight varied from 100% to 300% of the baseline. The thermal management system design rejected power (motor and battery) varied from 100% to 300% of the baseline.

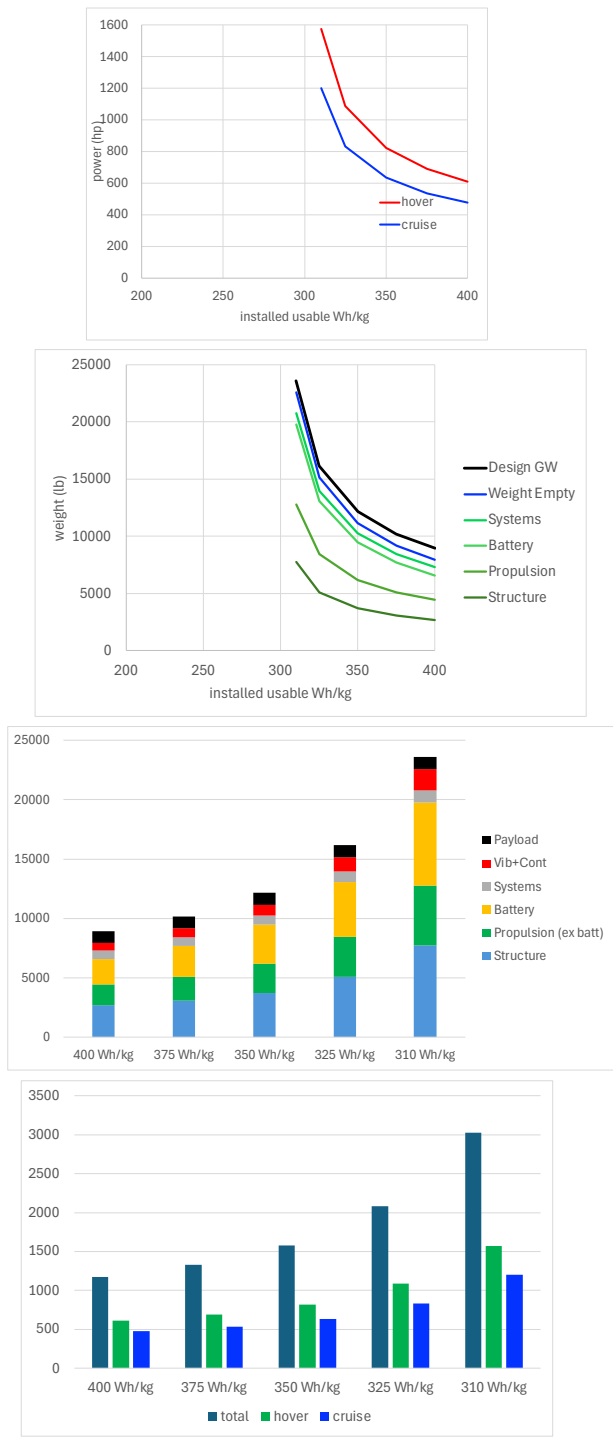


Figure 52. Side-by-side, electric — impact of battery specific energy. The battery specific energy varied from 400 Wh/kg (baseline) to 310 Wh/kg.

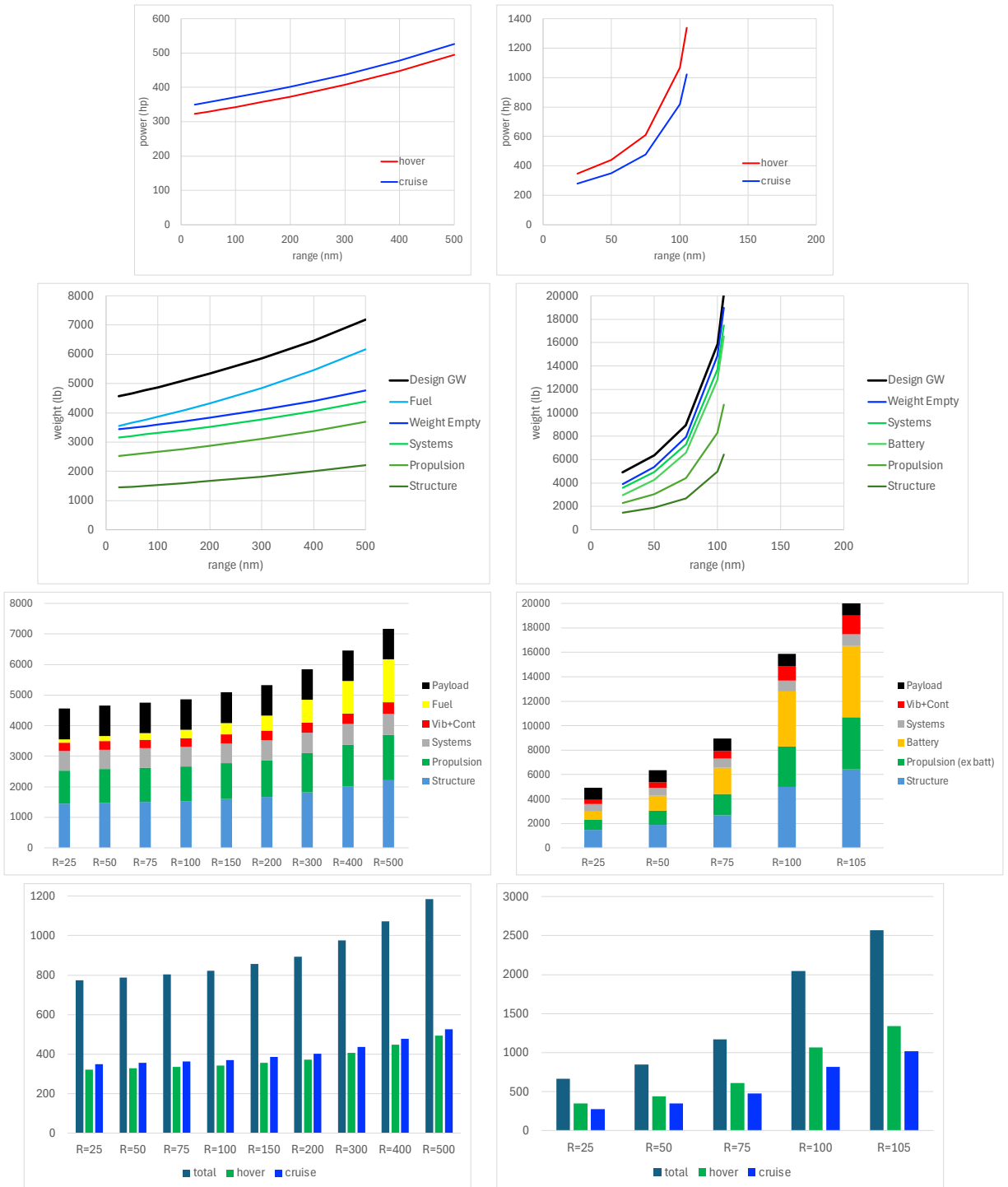


Figure 53. Side-by-side, turboshaft (left) and electric (right) — impact of mission range. The baseline design mission has a range of 75 nm (in two segments). The range is varied from 25 to 500 nm for the turboshaft configurations, and from 25 to 105 nm for the electric configurations.

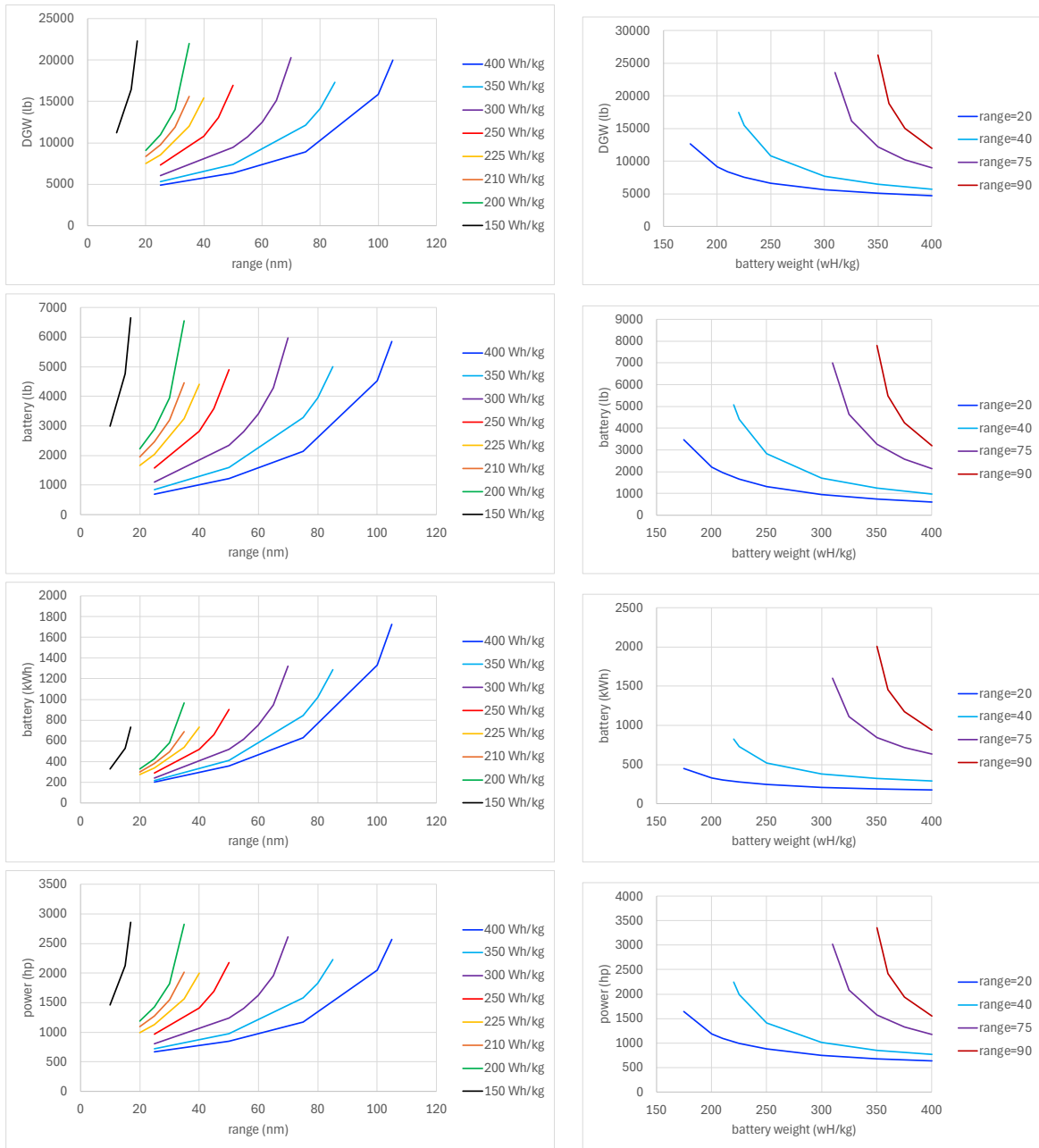


Figure 54. Side-by-side, electric—impact of mission range (20 to 155 nm) and battery weight (400 to 150 Wh/kg); design gross weight (lb), battery weight (lb), battery capacity (kWh), and installed power (hp). The baseline design mission has a range of 75 nm (in two segments), with battery weight 400 Wh/kg.

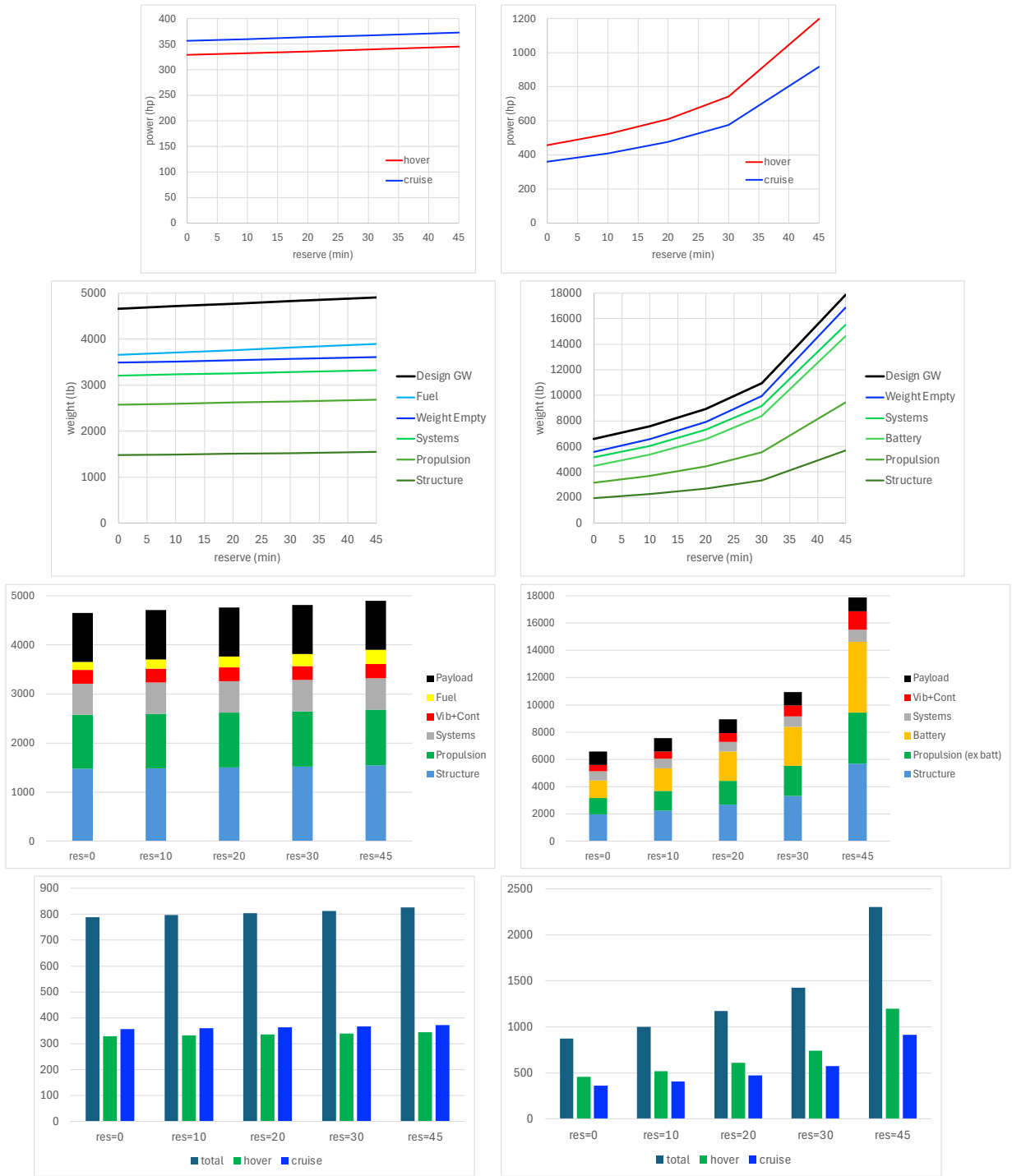


Figure 55. Side-by-side, turboshaft (left) and electric (right) — impact of mission reserve. The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. The reserve segment time is varied from 0 to 45 min.

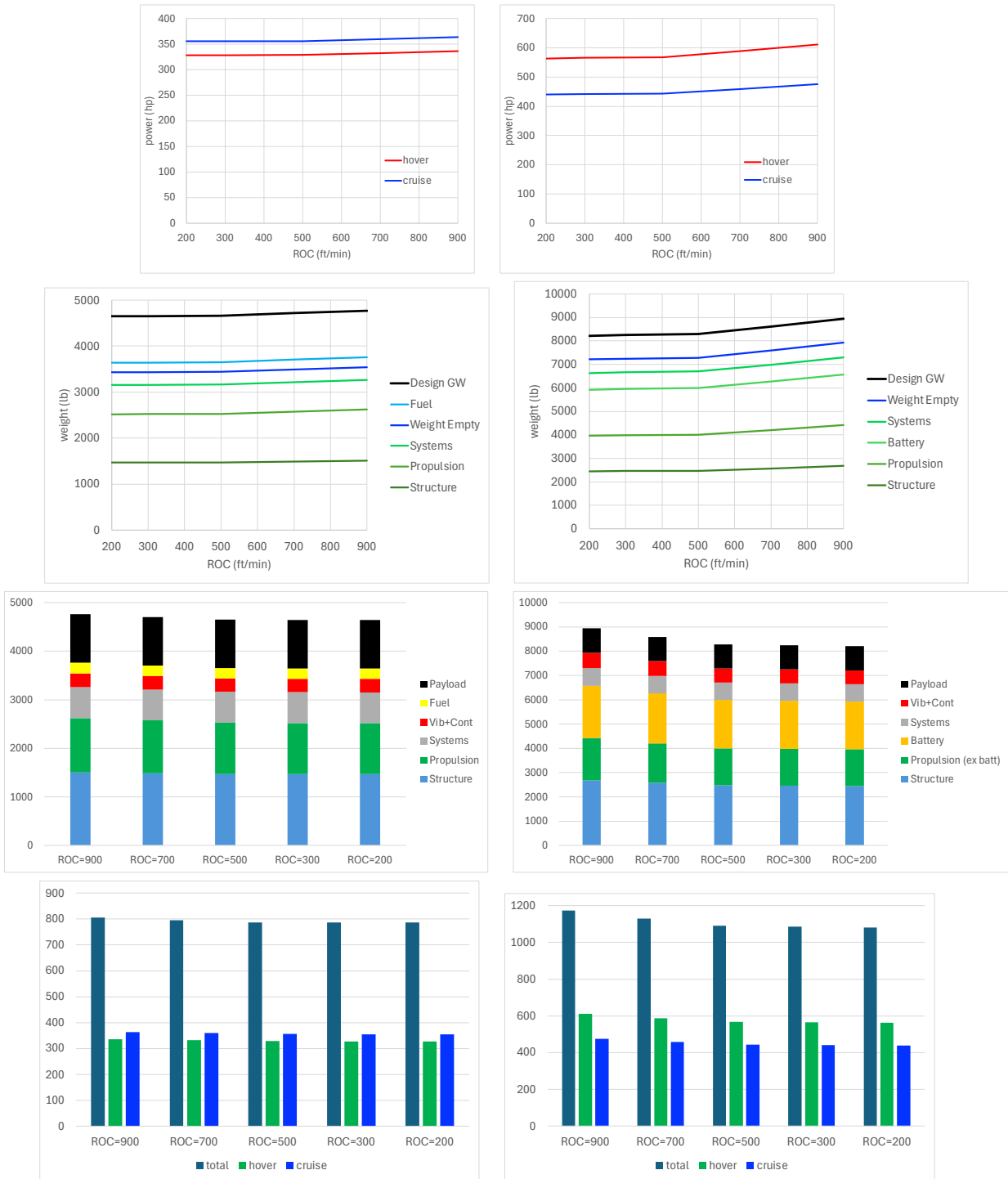


Figure 56. Side-by-side, turboshaft (left) and electric (right) — impact of mission climb to cruise altitude, with one-engine-inoperative requirement. The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min.

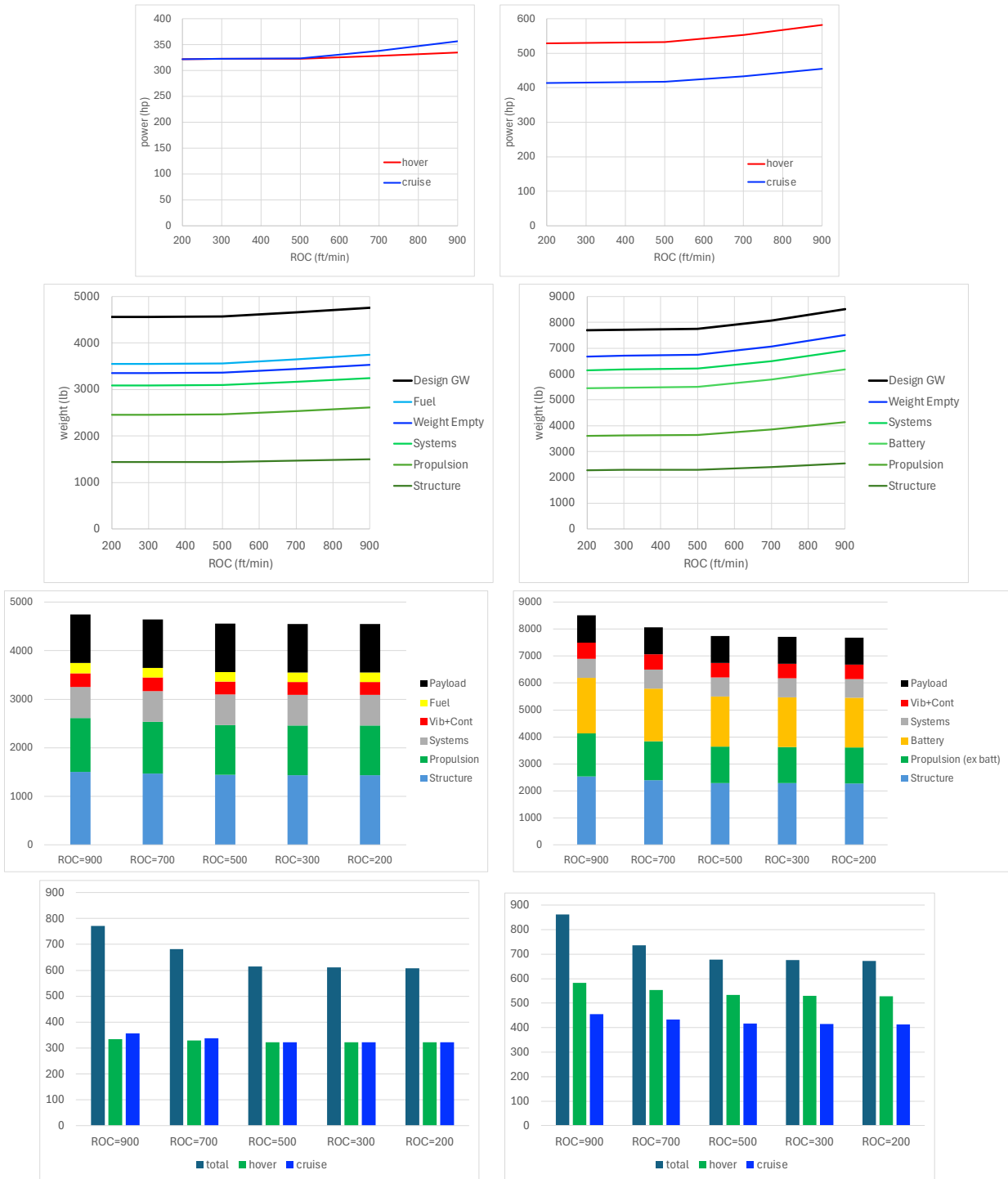


Figure 57. Side-by-side, turboshaft (left) and electric (right) — impact of mission climb to cruise altitude, without one-engine-inoperative requirement. The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min.

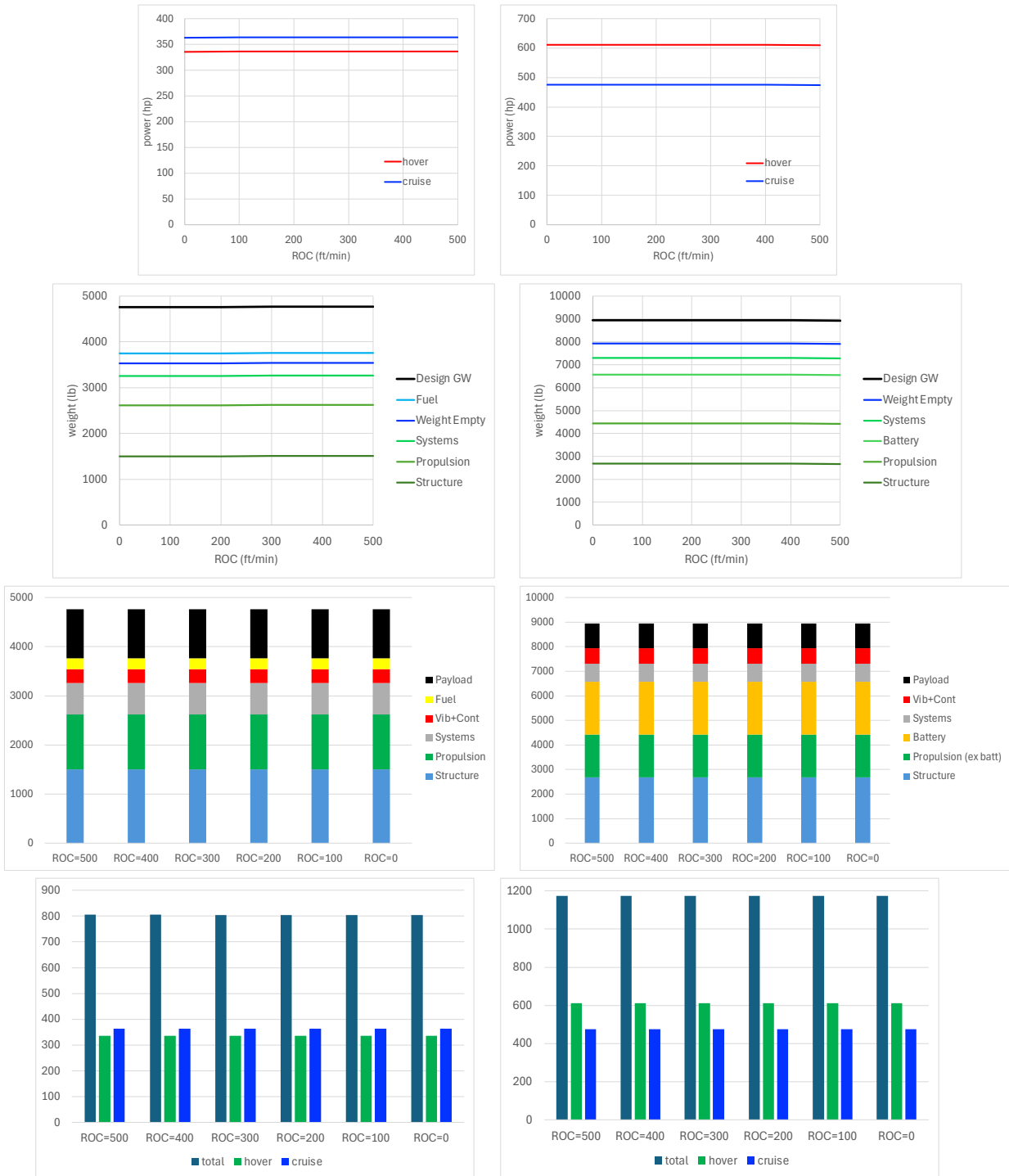


Figure 58. Side-by-side, turboshaft (left) and electric (right) — impact of cruise climb. The cruise rate of climb is varied from 500 ft/min (baseline) to 0 (no requirement).

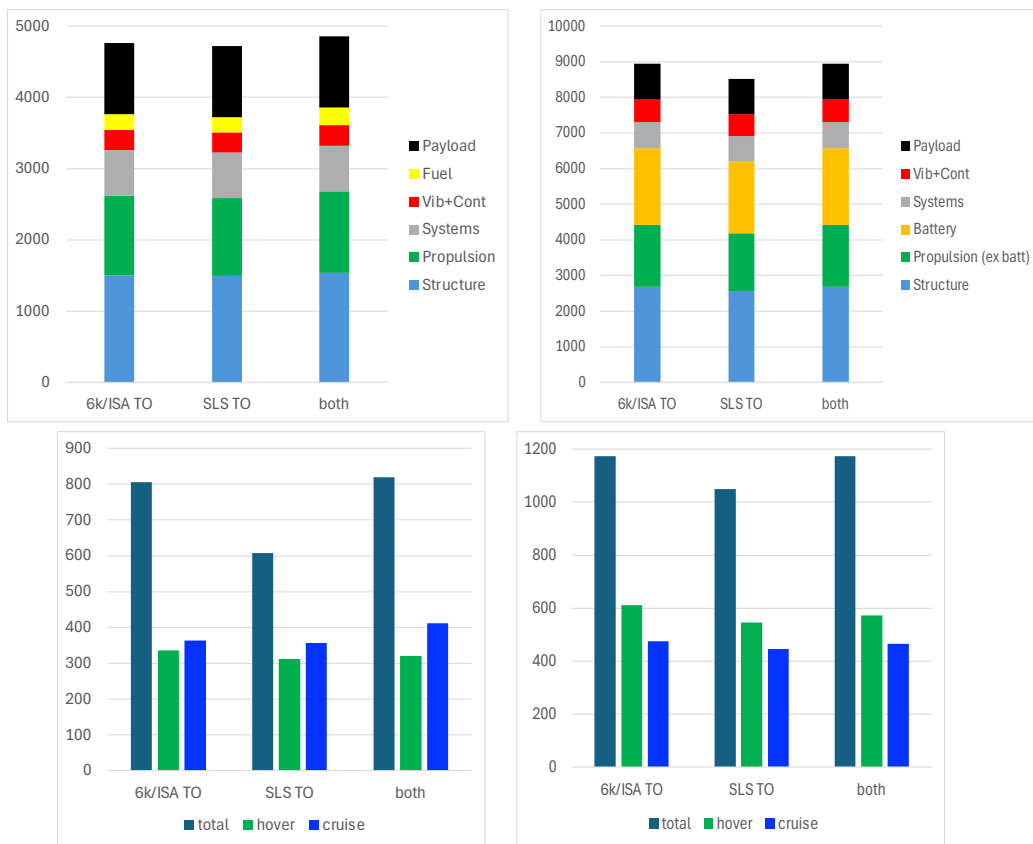


Figure 59. Side-by-side, turboshaft (left) and electric (right) — impact of operating altitude. The design mission has takeoff at 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA); or takeoff at sea level ISA and cruise at 4000 ft ISA; or both.

Impact of Technology and Mission Variations — Tiltrotor

Technology

Weight technology factors (Figure 60). Designs with technology, compared to only calibration (no technology factors) — high, baseline, and low calibration.

Wing weight (Figure 61). Technology factor varied from 75% to 150% (baseline 0.8).

Systems and equipment (Figure 62). The systems and equipment weight changed by -200 lb to 200 lb, relative to the baseline fixed weight of 480 lb.

Aircraft drag (Figure 63). The airframe drag varied by $\Delta D/q = -2.0$ to 2.0 ft^2 .

Blade loading (Figure 64). The main rotor design blade loading varied from $C_W/\sigma = 0.13$ (baseline) to 0.11.

Profile power (Figure 65). The profile power varied from 100% to 80%.

Engine weight (Figure 66). The turboshaft engine weight and electric motor weight varied from 80% to 120% of the baseline.

Engine efficiency (Figure 67). The turboshaft engine specific fuel consumption varied from 80% to 120% of the baseline (0.49 lb/hp-hr MCP SLS). The motor efficiency varied from 0.95 (baseline) to 0.75.

Electric motor (Figures 68). The wiring weight varied from 100% to 300% of the baseline. The thermal management system design rejected power (motor and battery) varied from 100% to 300% of the baseline.

Battery (Figure 69). The battery specific energy varied from 400 Wh/kg (baseline) to 250 Wh/kg.

Mission

Range (Figures 70–71). 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. For electric aircraft, the mission range is varied for a set of battery weights (400 Wh/kg to 150 Wh/kg).

Reserve (Figure 72). The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. The reserve segment time is varied from 0 to 45 min.

Climb and one-engine-inoperative (Figure 73–74). The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 200 ft/min, with and without the OEI requirement.

Cruise climb (Figure 75). The cruise rate of climb is varied from 500 ft/min (baseline) to 0 (no requirement).

Altitude (Figure 76). The design mission has takeoff at 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA); or takeoff at sea level ISA and cruise at 4000 ft ISA; or both.

Cruise speed (Figure 77). The cruise speed is varied from 200 knots to 300 knots, for a mission with 250 nm range.

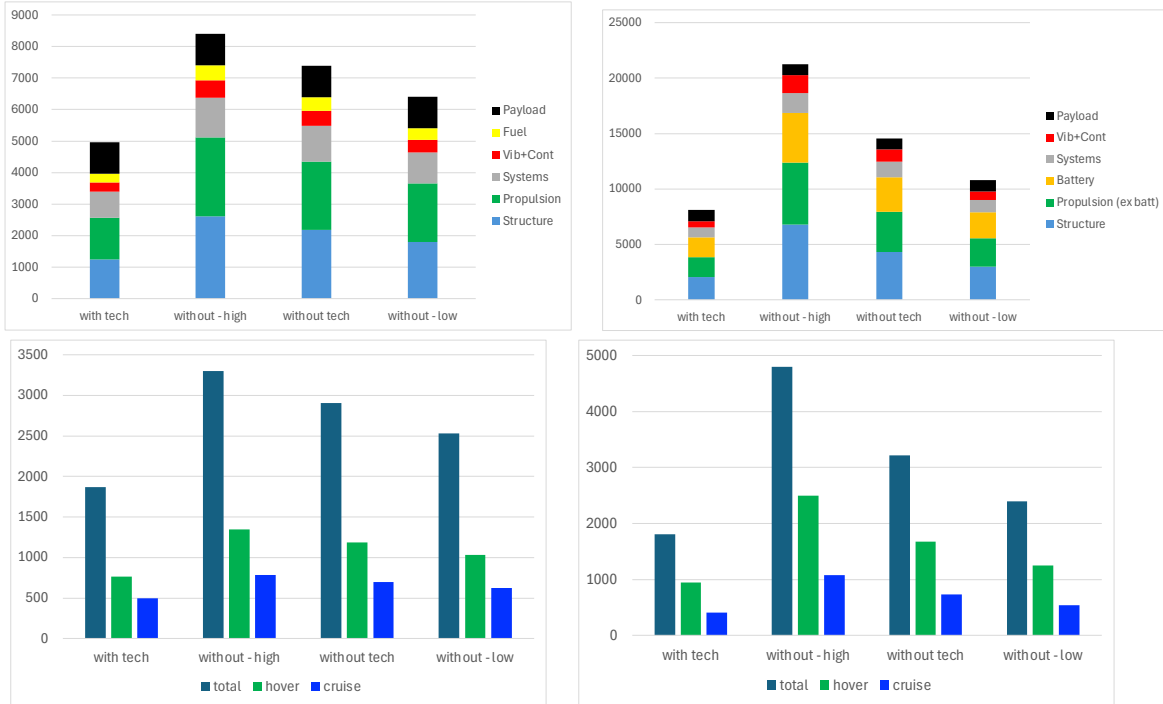


Figure 60. Tiltrotor, turboshaft (left) and electric (right) — impact of weight technology factors. Designs with technology, compared to only calibration (no technology factors): high, baseline, and low calibration.

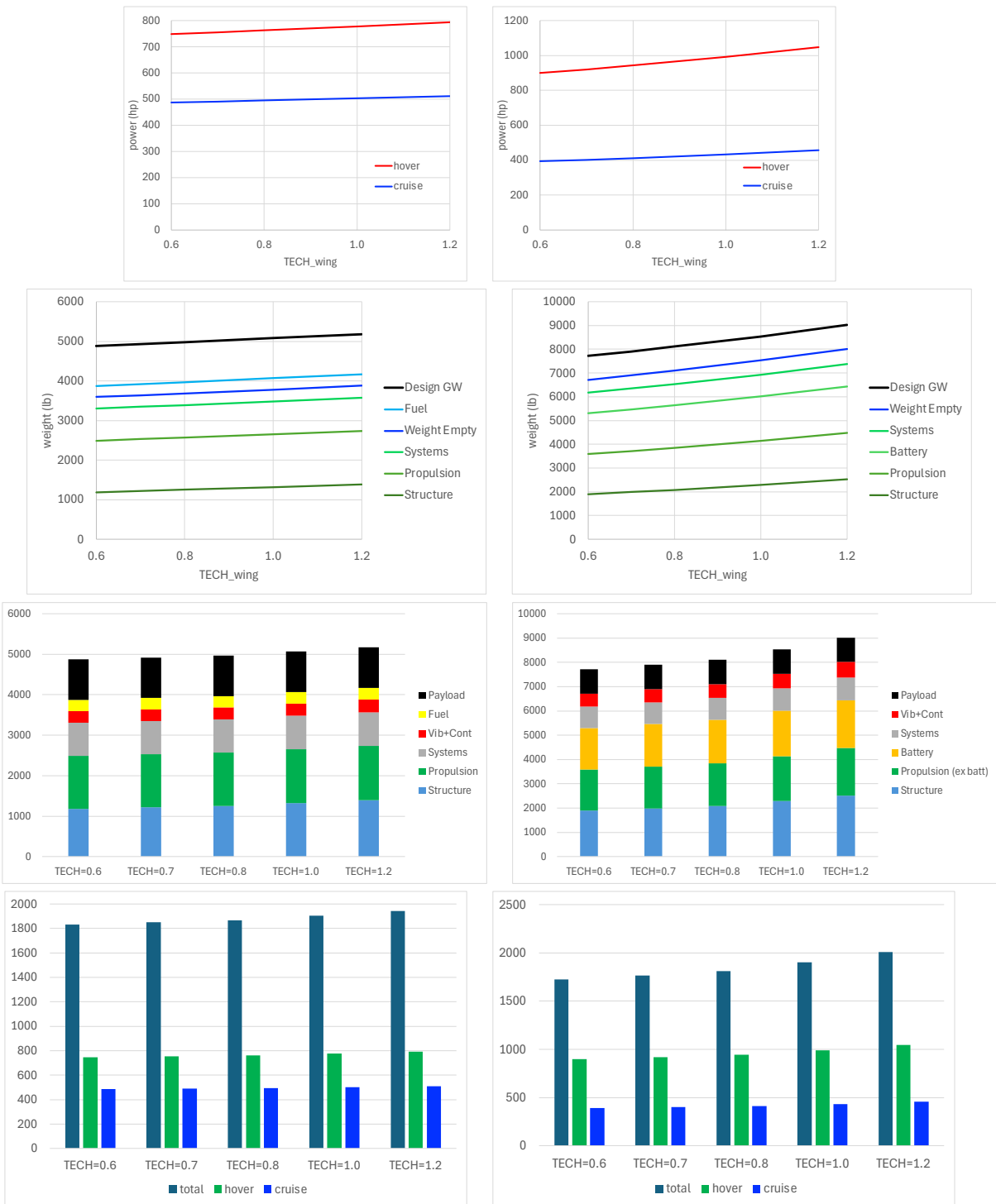


Figure 61. Tiltrotor, turboshaft (left) and electric (right) — impact of wing weight. Technology factor varied from 75% to 150% (baseline 0.8).

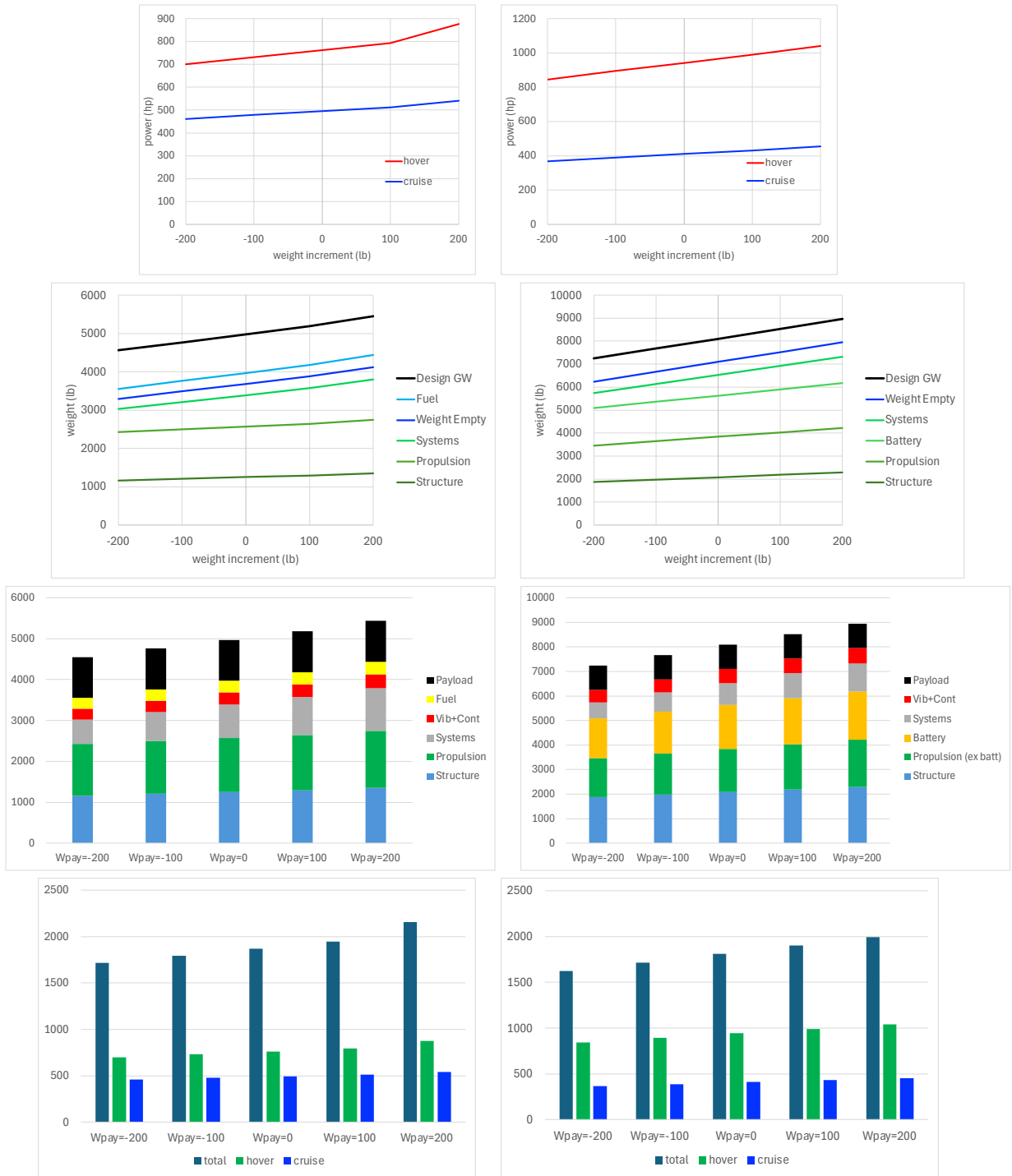


Figure 62. Tiltrotor, turboshaft (left) and electric (right) — impact of systems and equipment weight. The systems and equipment weight changed by -200 lb to 200 lb, relative to the baseline fixed weight of 480 lb.

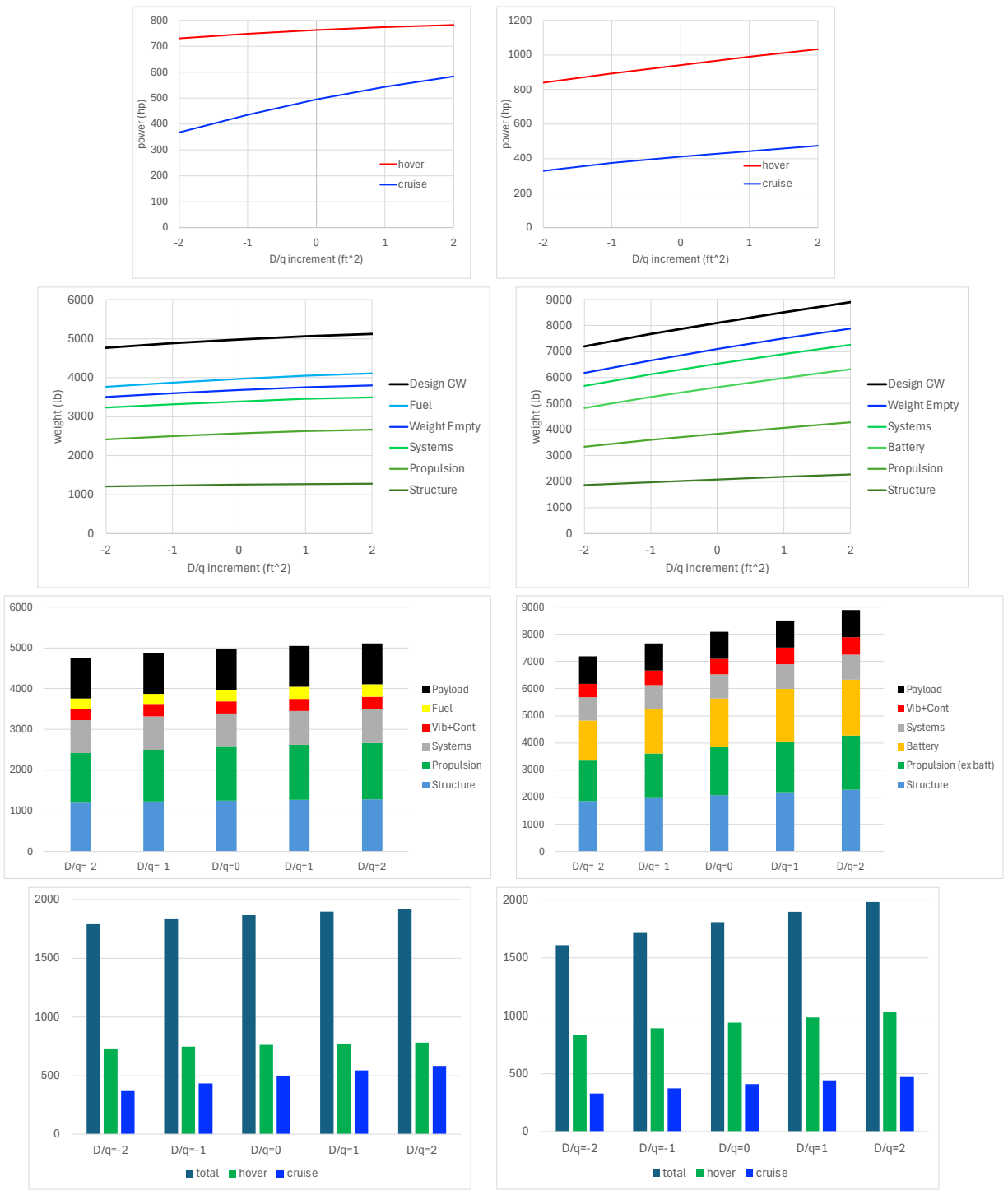


Figure 63. Tiltrotor, turboshaft (left) and electric (right) — impact of aircraft drag. The airframe drag varied by $\Delta D/q = -2.0$ to 2.0 ft².

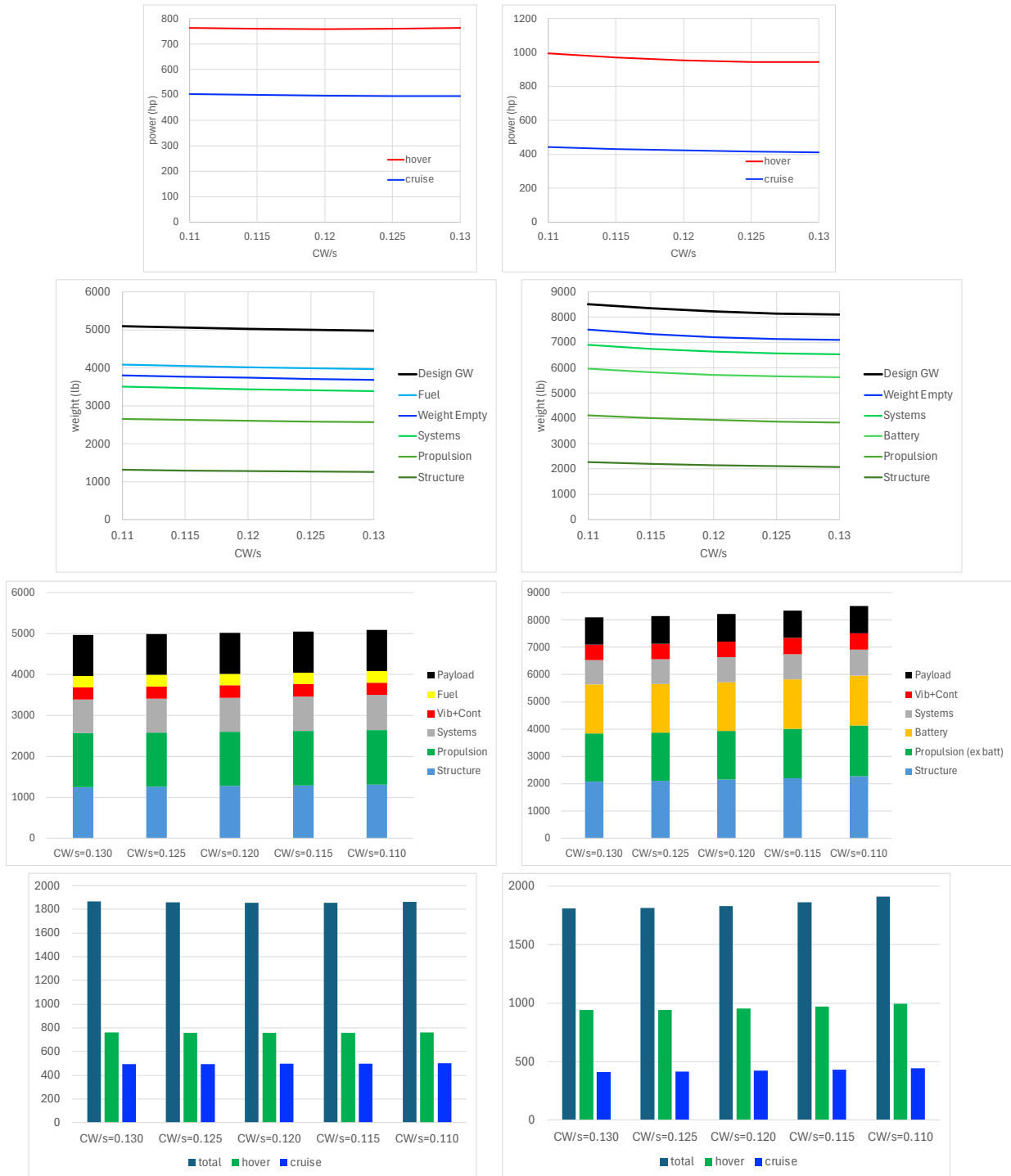


Figure 64. Tiltrotor, turboshaft (left) and electric (right) — impact of main rotor design blade loading. The main rotor design blade loading varied from $C_W/\sigma=0.13$ (baseline) to 0.11.

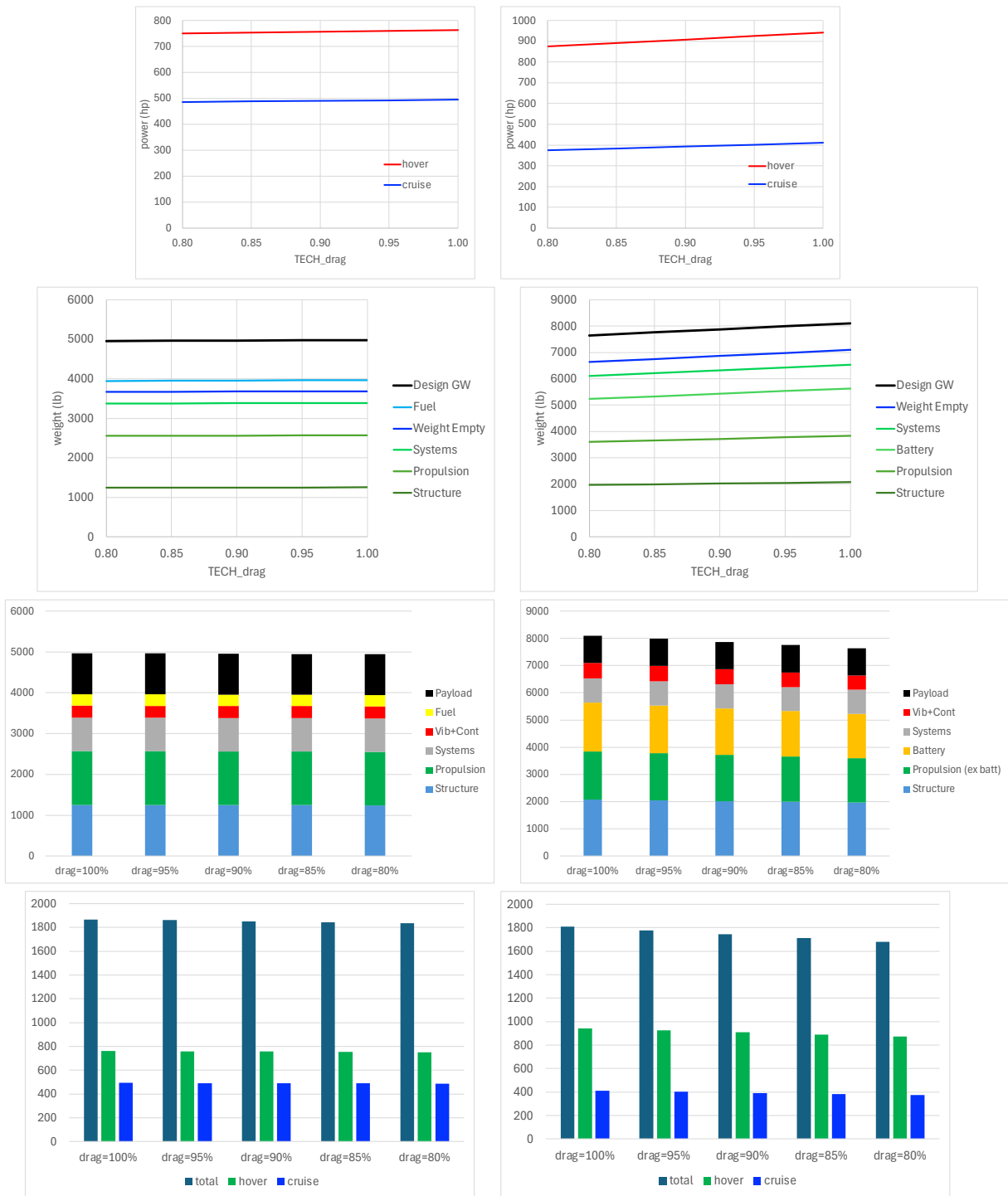


Figure 65. Tiltrotor, turboshaft (left) and electric (right) — impact of rotor profile power. The profile power varied from 100% to 80%.

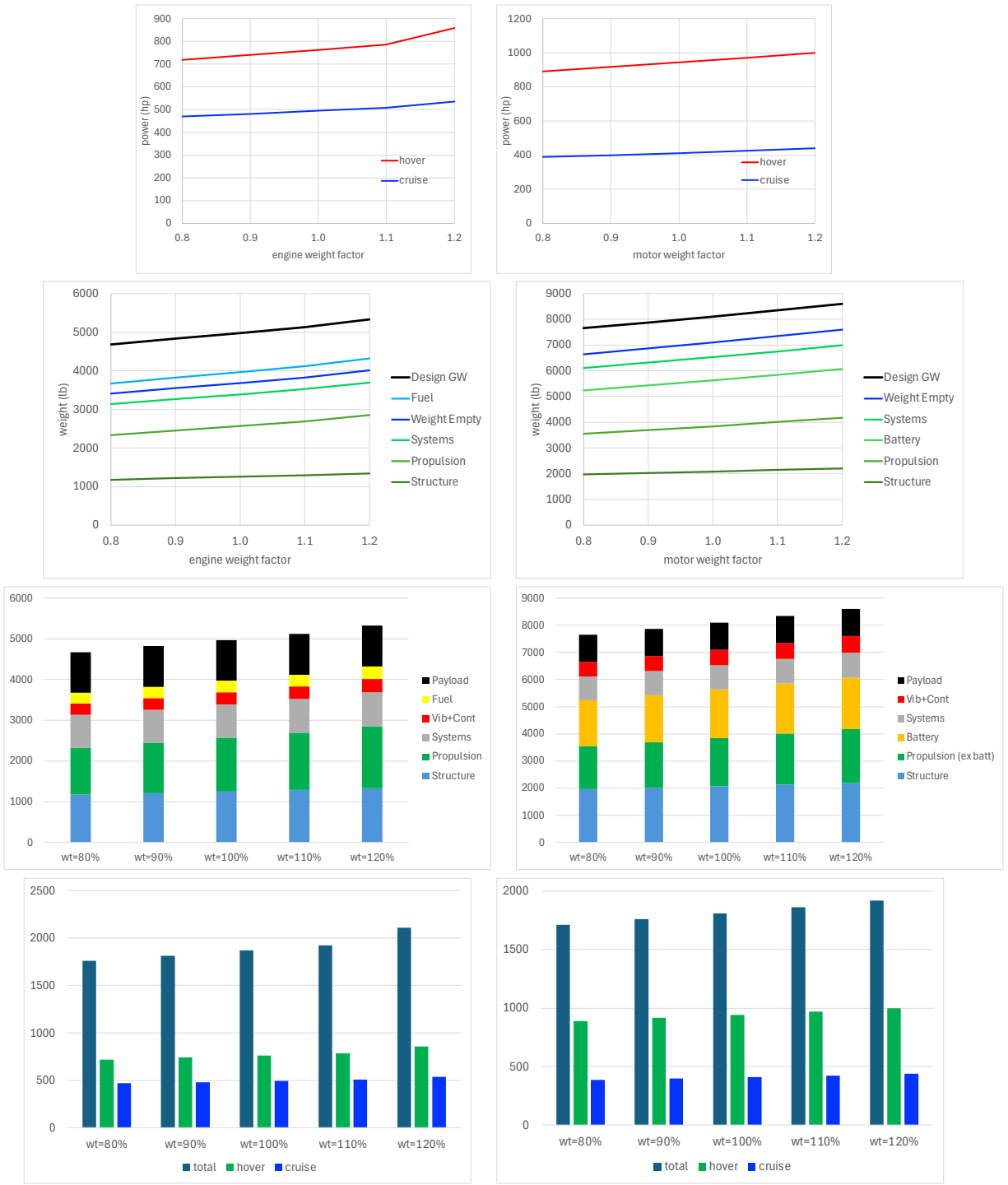


Figure 66. Tiltrotor, turboshaft (left) and electric (right) — impact of engine weight. The turboshaft engine weight or electric motor weight varied from 80% to 120% of the baseline.

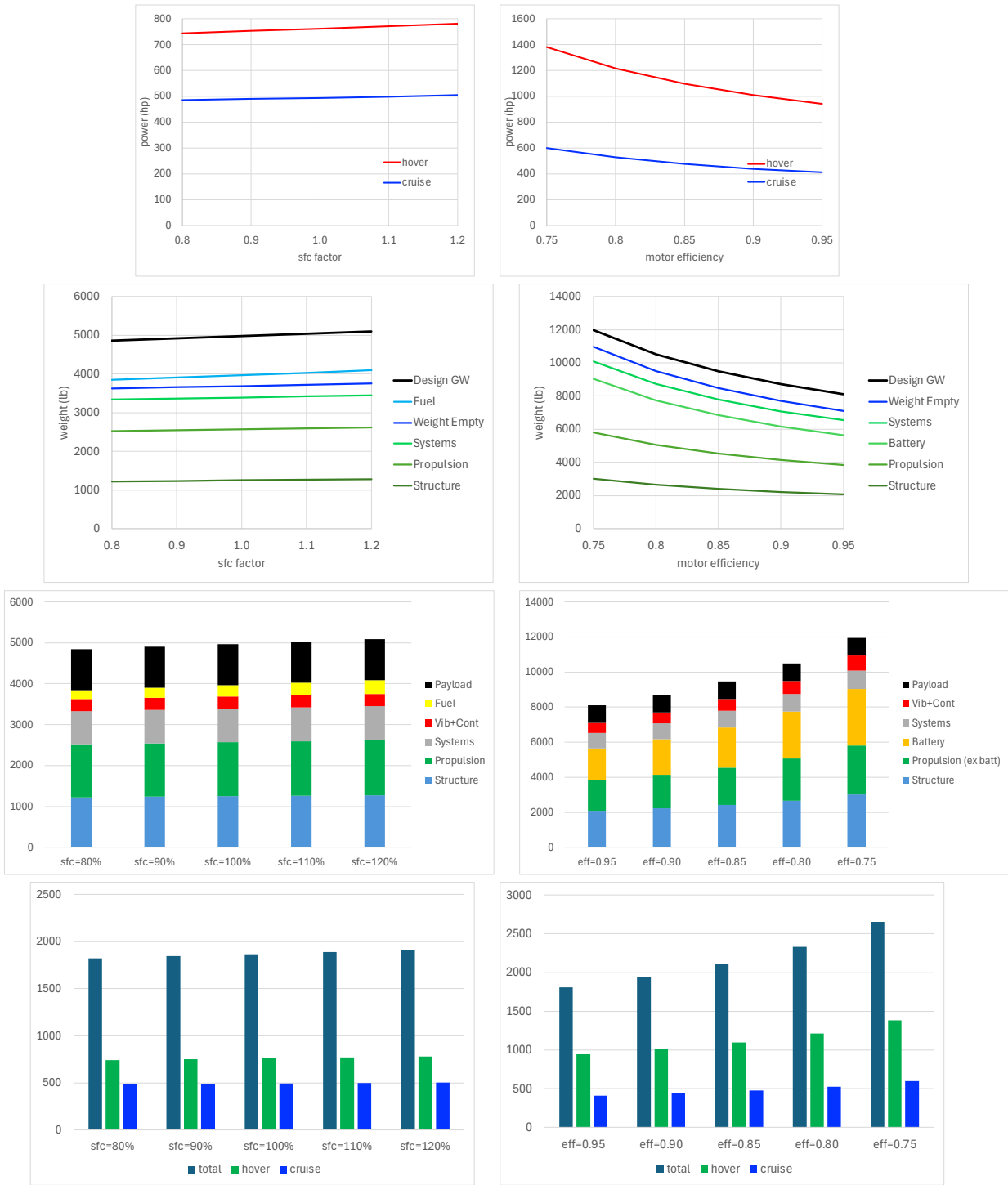


Figure 67. Tiltrotor, turboshaft (left) and electric (right) — impact of engine efficiency. The turboshaft engine specific fuel consumption varied from 80% to 120% of the baseline (0.49 lb/hp-hr MCP SLS). The motor efficiency varied from 0.95 (baseline) to 0.75.

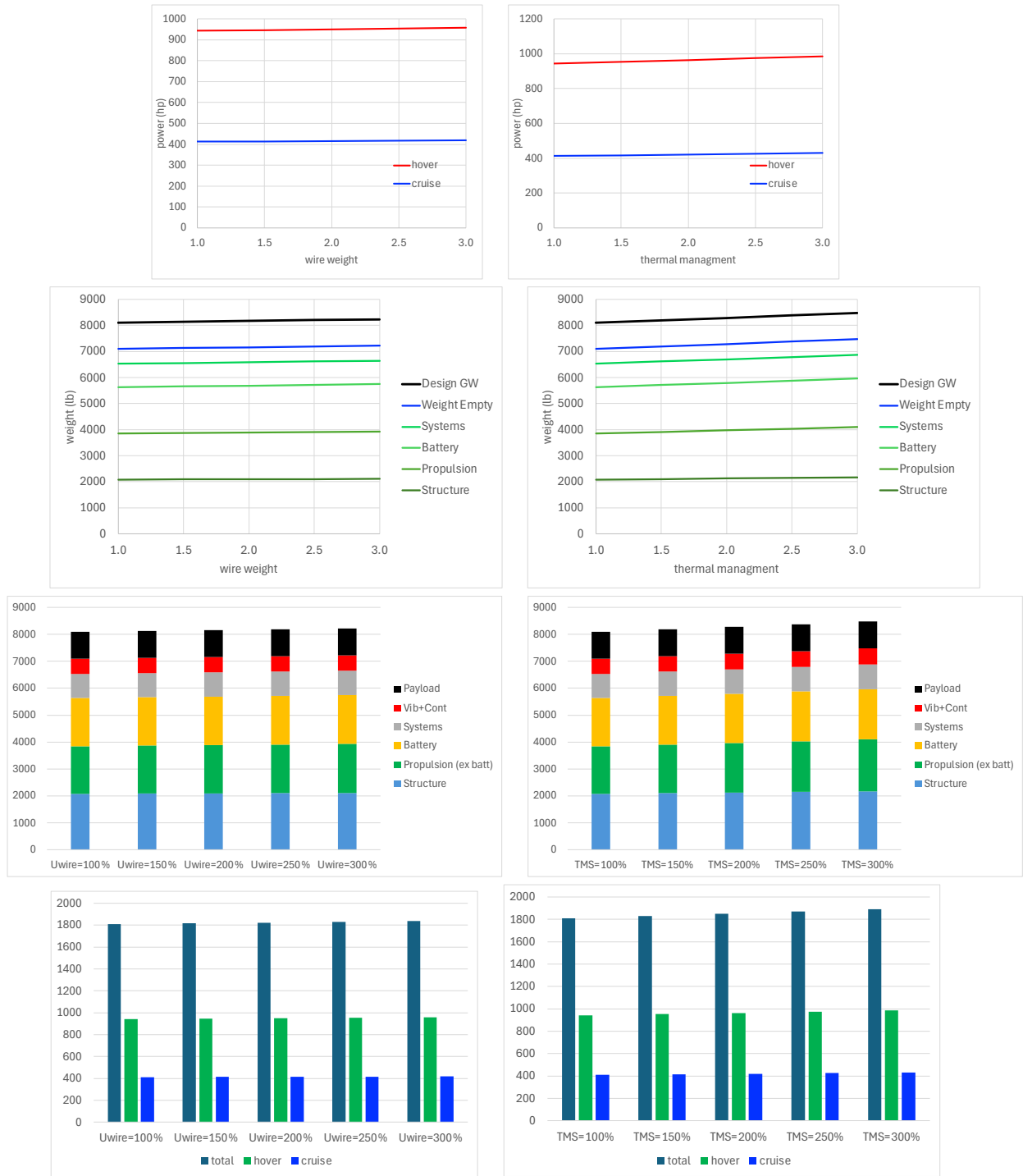


Figure 68. Tiltrotor, electric — impact of wiring weight (left) and thermal management system (right). The wiring weight varied from 100% to 300% of the baseline. The thermal management system design rejected power (motor and battery) varied from 100% to 300% of the baseline.

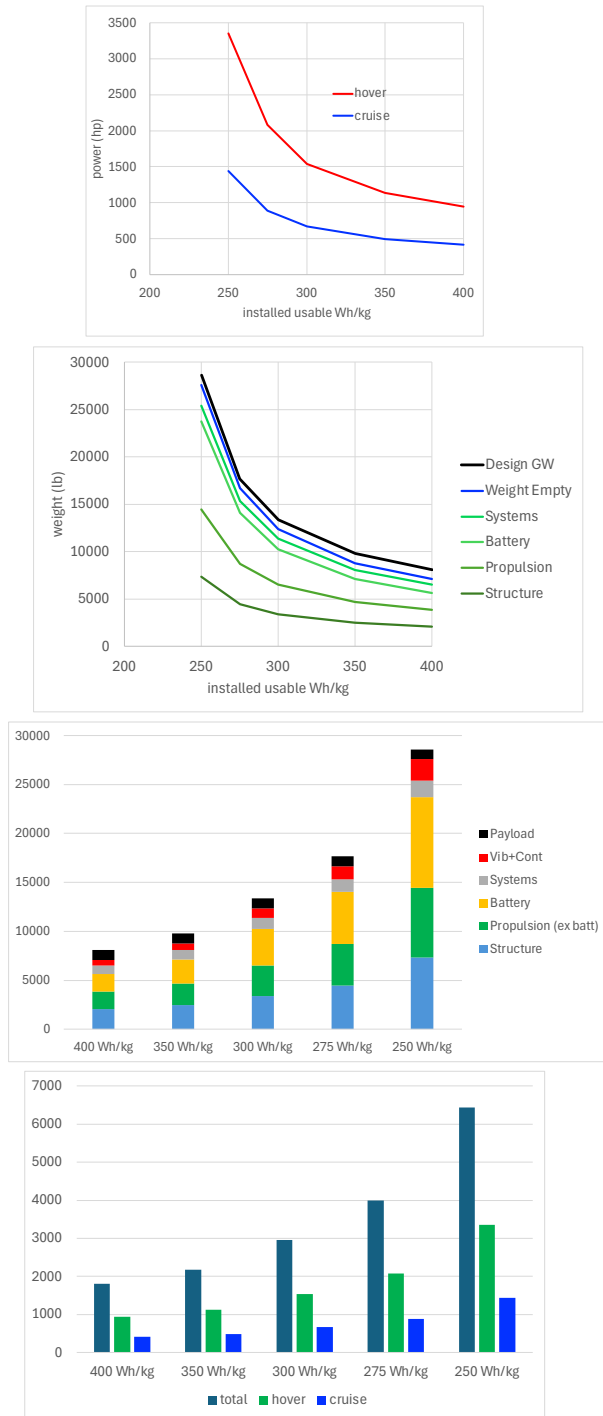


Figure 69. Tiltrotor, electric — impact of battery specific energy. The battery specific energy varied from 400 Wh/kg (baseline) to 250 Wh/kg.

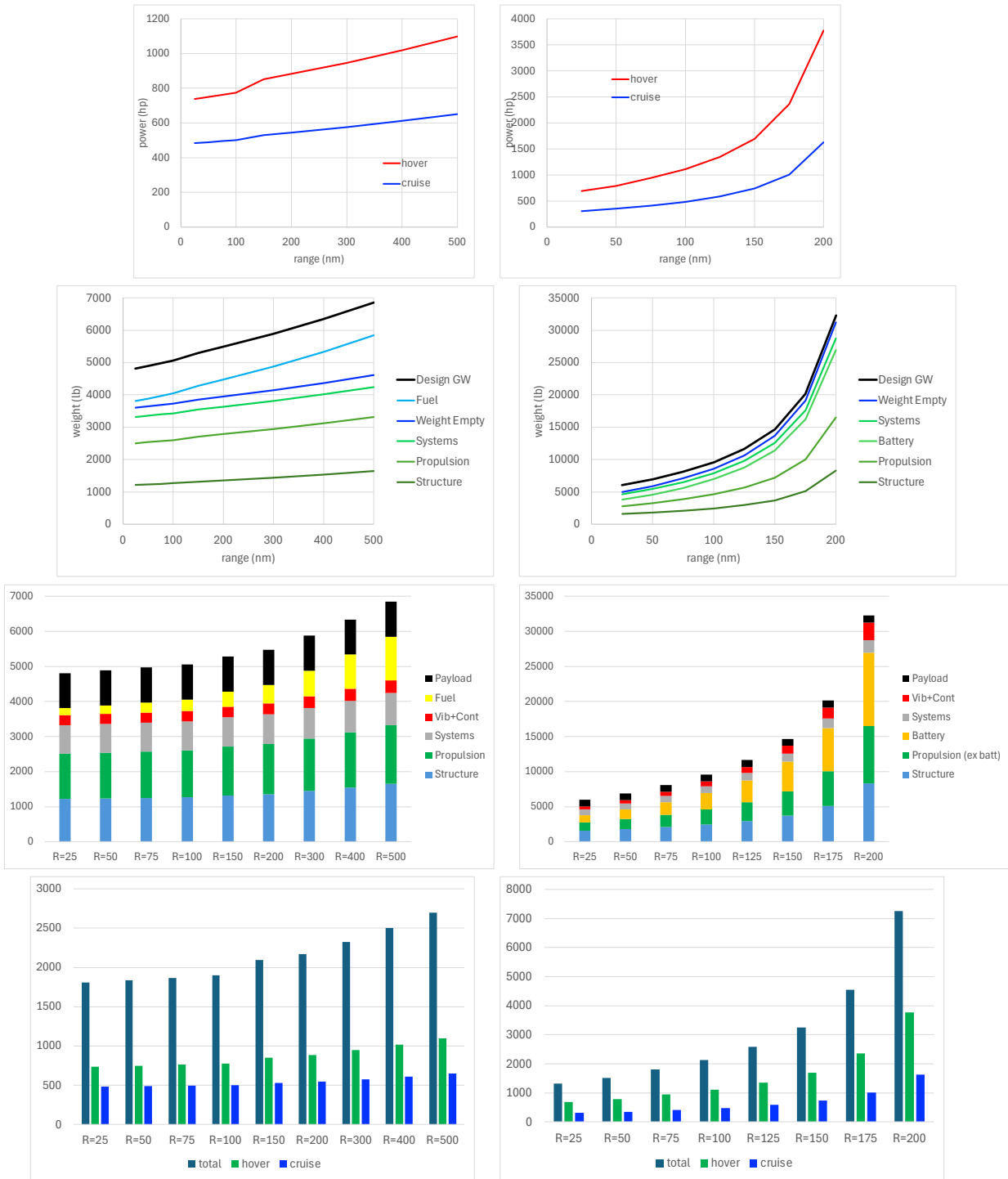


Figure 70. Tiltrotor, turboshaft (left) and electric (right) — impact of mission range. The baseline design mission has a range of 75 nm (in two segments). The range is varied from 25 to 500 nm for the turboshaft configurations, and from 25 to 200 nm for the electric configurations.

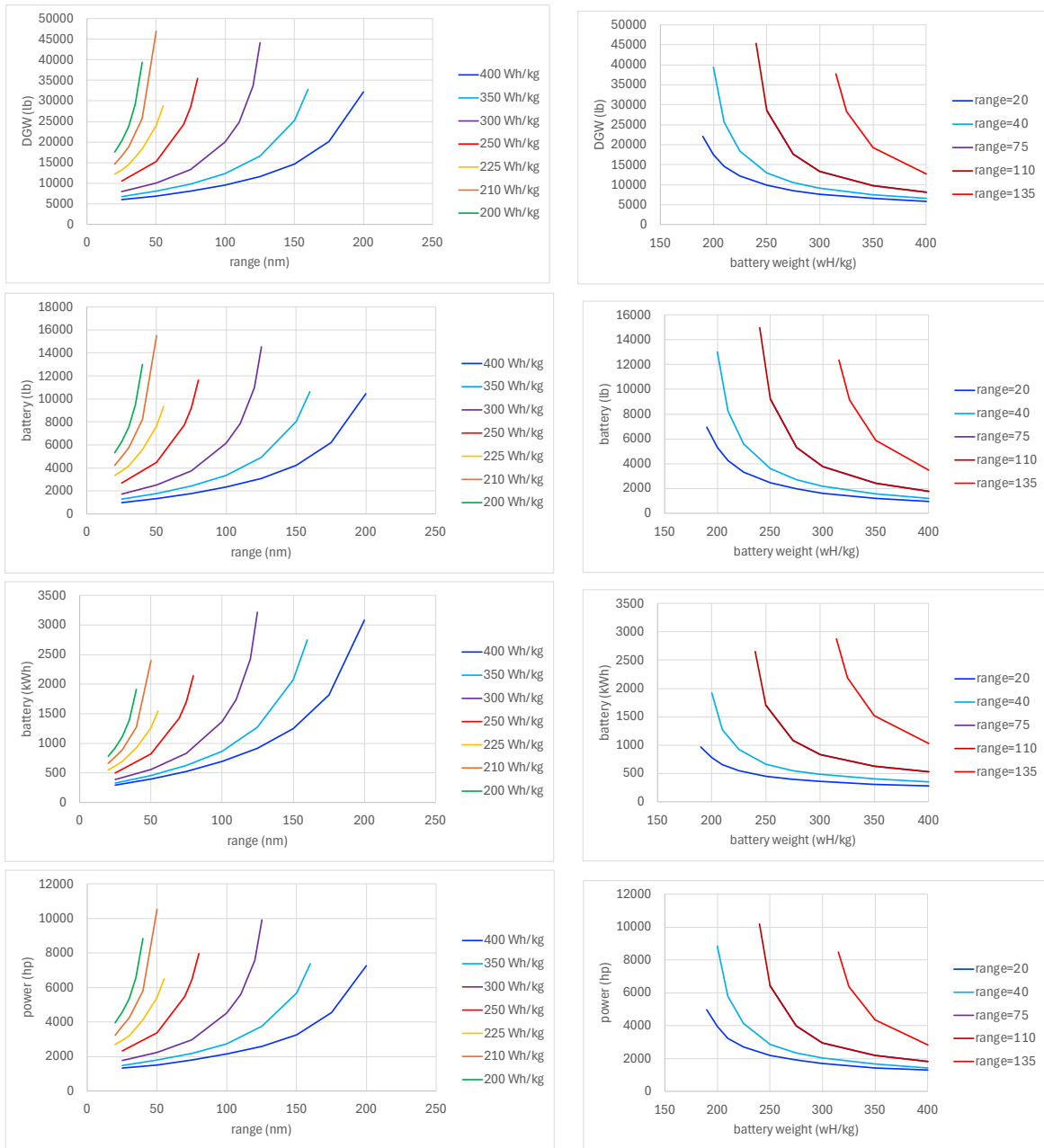


Figure 71. Tiltrotor, electric — impact of mission range (20 to 200 nm) and battery weight (400 to 200 Wh/kg); design gross weight (lb), battery weight (lb), battery capacity (kWh), and installed power (hp). The baseline design mission has a range of 75 nm (in two segments), with battery weight 400 Wh/kg.

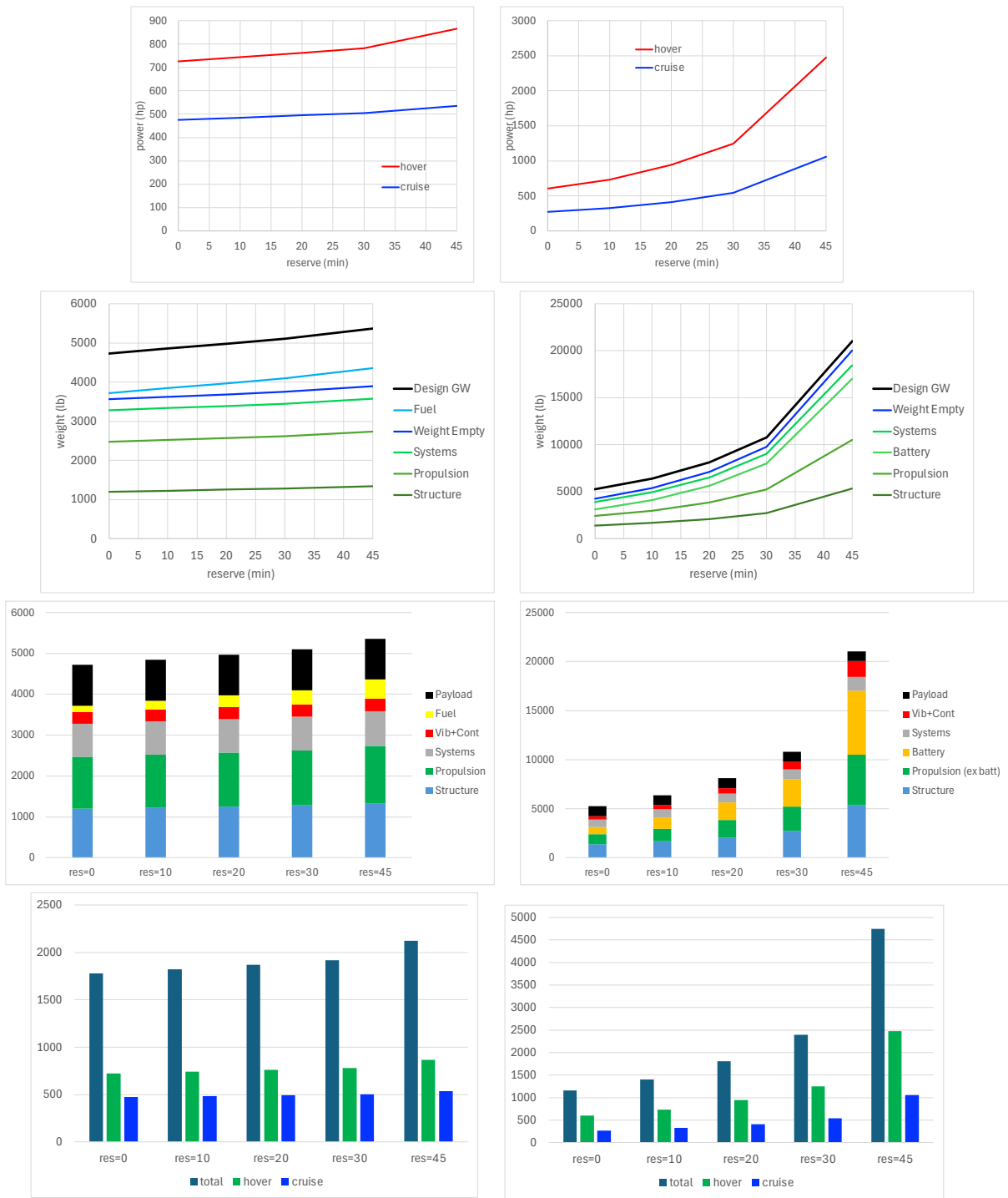


Figure 72. Tiltrotor, turboshaft (left) and electric (right) — impact of mission reserve. The baseline design mission has a reserve of 20 min, corresponding to visual flight rules (VFR) rotorcraft operations. The reserve segment time is varied from 0 to 45 min.

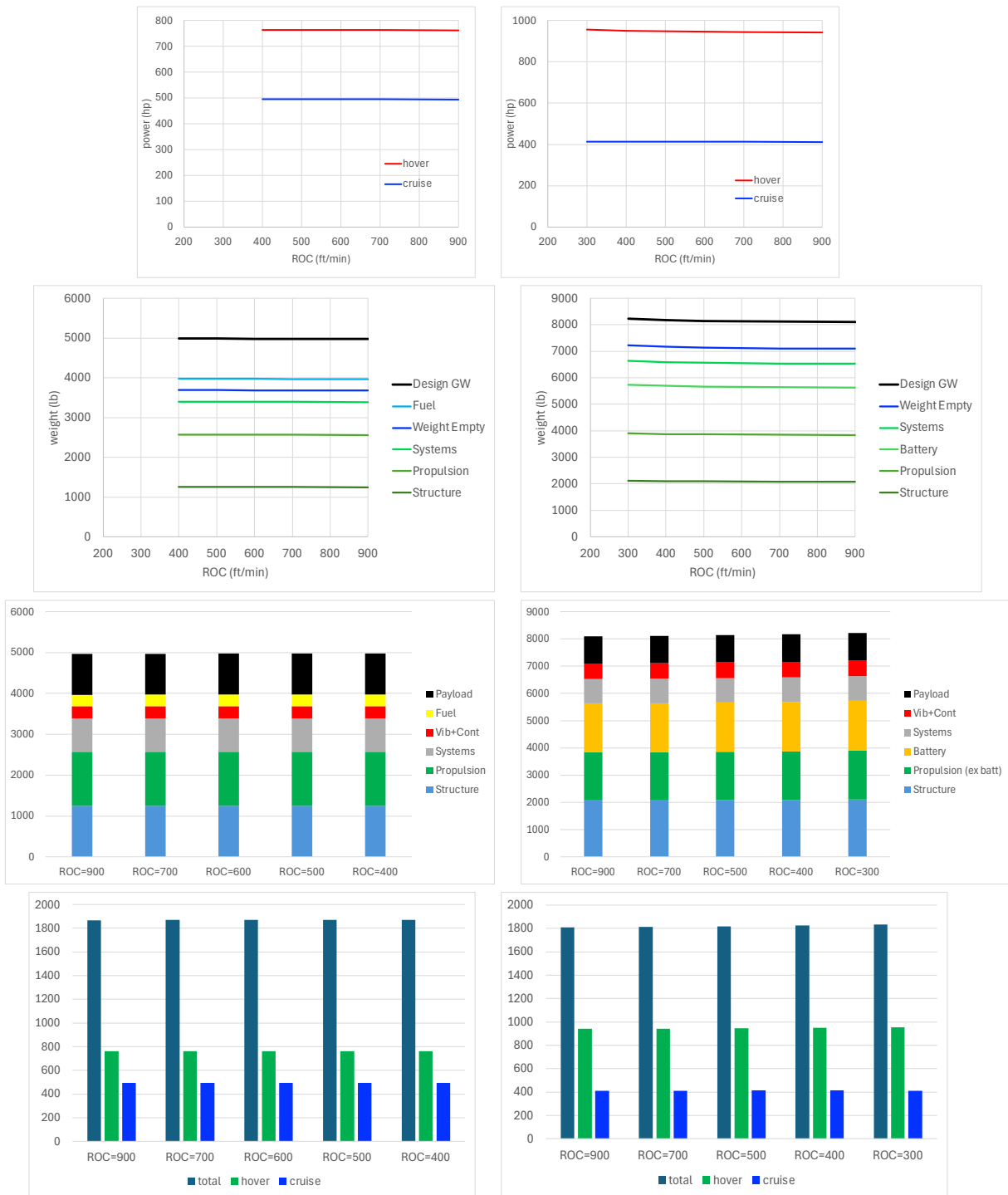


Figure 73. Tiltrotor, turboshaft (left) and electric (right) — impact of mission climb to cruise altitude, with one-engine-inoperative requirement. The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 300–400 ft/min.

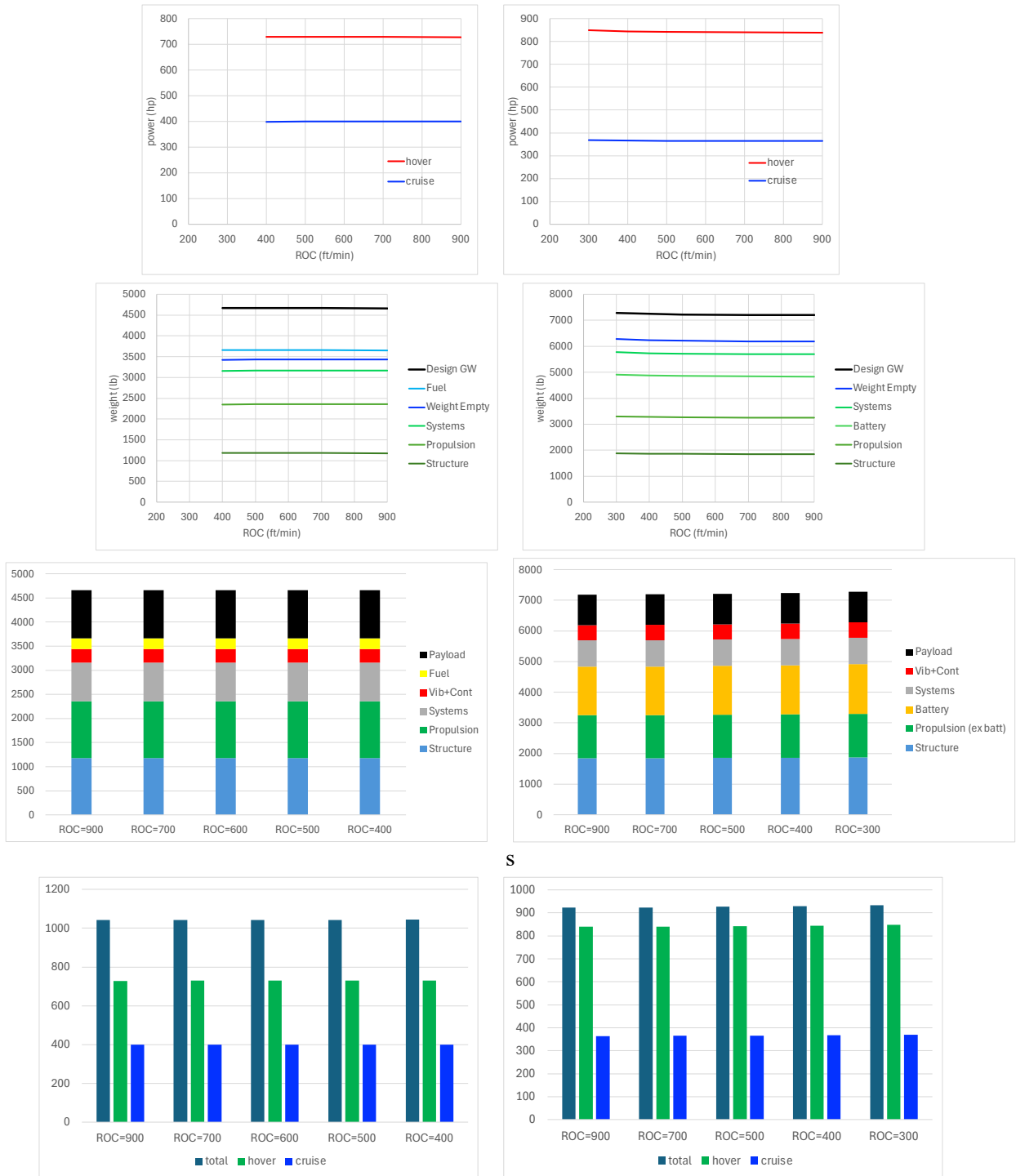


Figure 74. Tiltrotor, turboshaft (left) and electric (right) — impact of mission climb to cruise altitude, without one-engine-inoperative requirement. The rate of climb to cruise altitude is varied from 900 ft/min (baseline) to 300–400 ft/min.

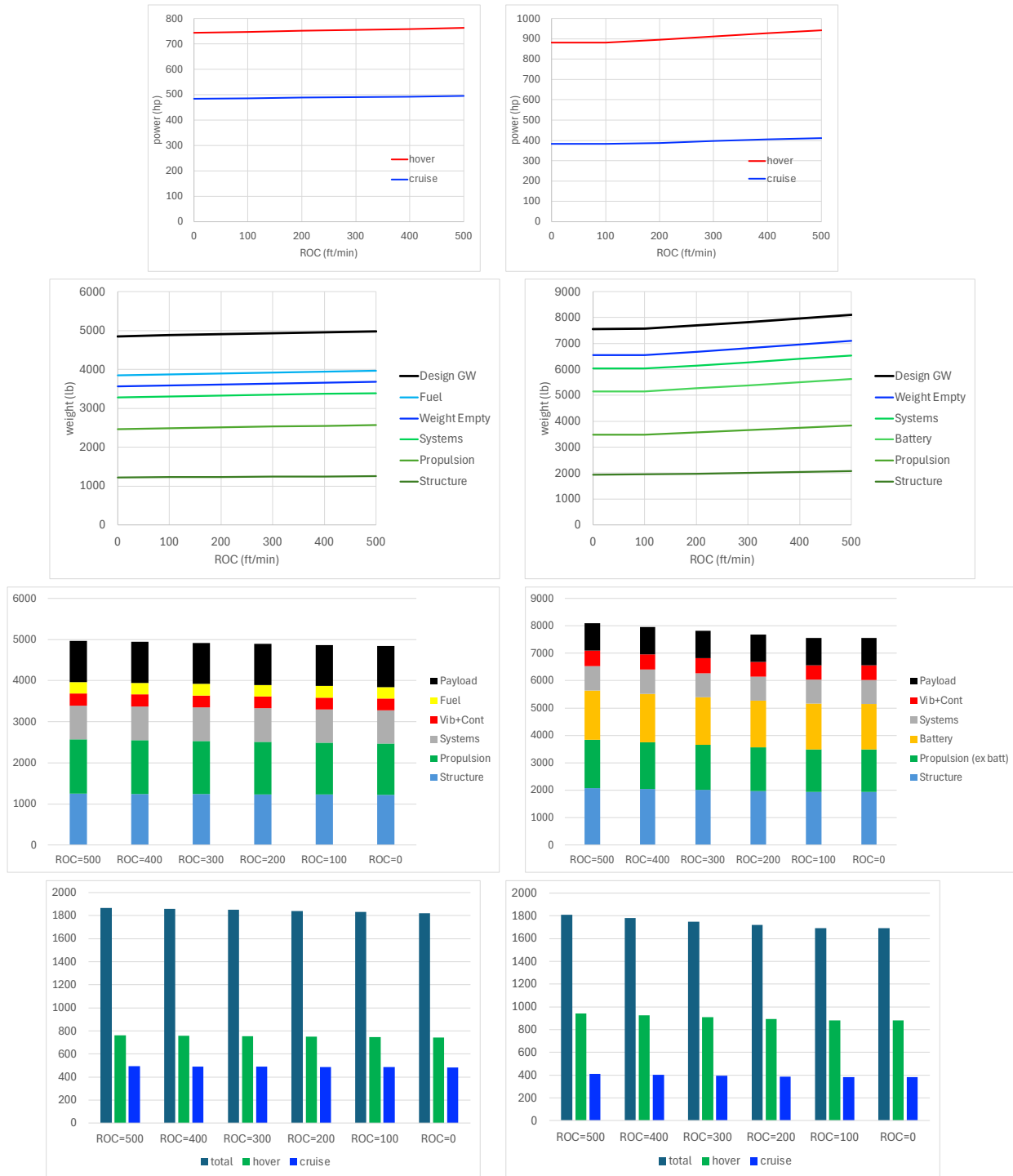


Figure 75. Tiltrotor, turboshaft (left) and electric (right) — impact of cruise climb. The cruise rate of climb is varied from 500 ft/min (baseline) to 0 (no requirement).

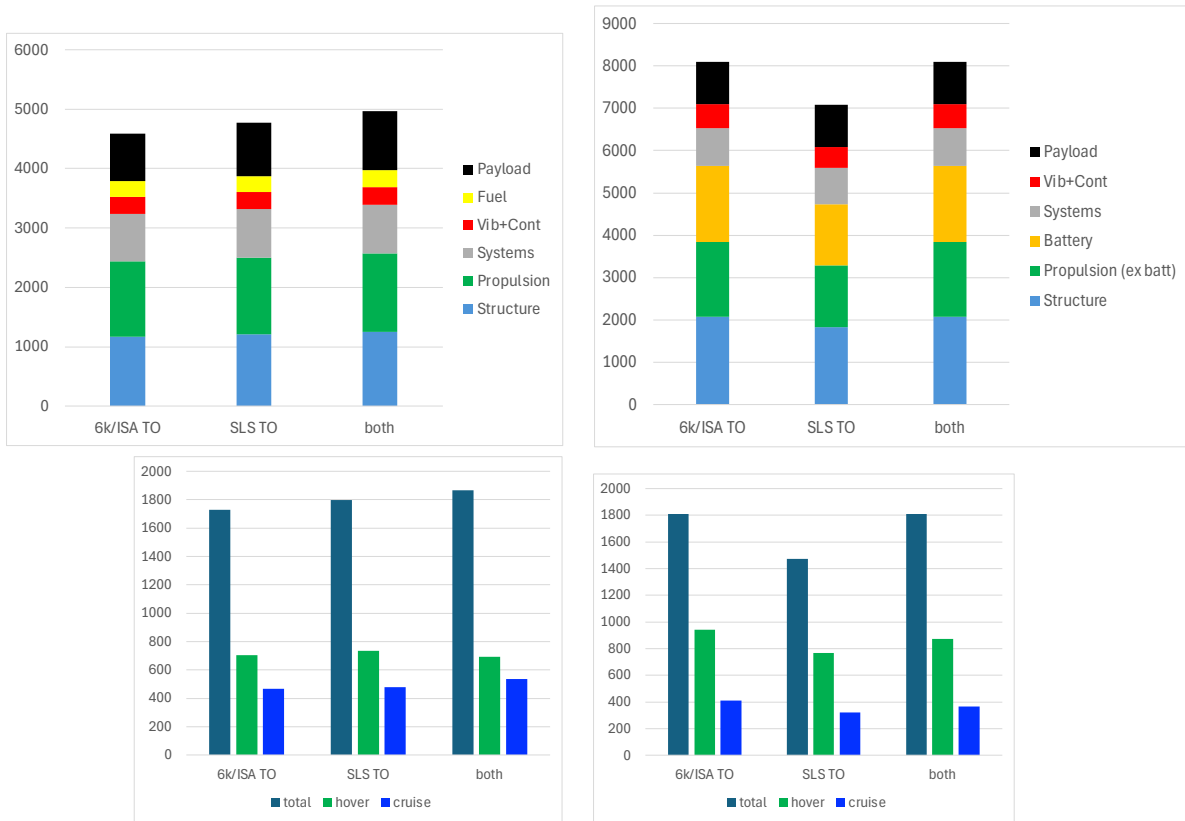


Figure 76. Tiltrotor, turboshaft (left) and electric (right) — impact of operating altitude. The design mission has takeoff at 6000 ft ISA, and cruise 4000 ft above ground level (10000 ft ISA); or takeoff at sea level ISA and cruise at 4000 ft ISA; or both.

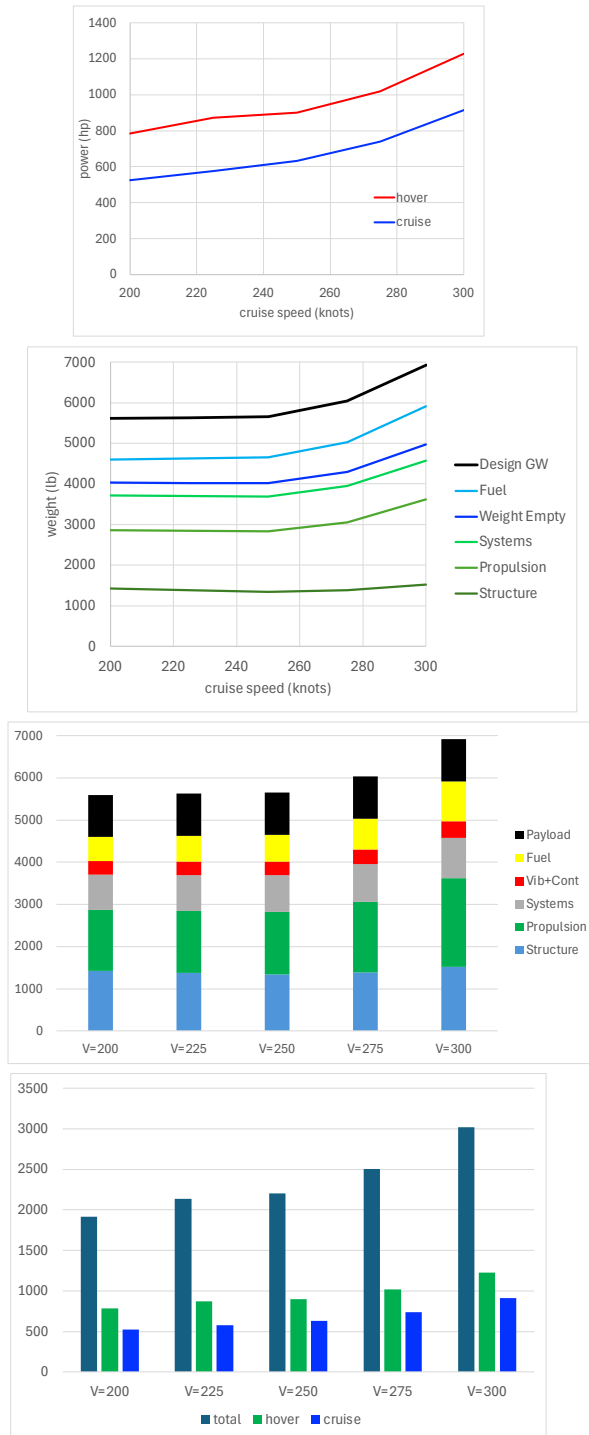


Figure 77. Tiltrotor, turboshaft — impact of cruise speed. The cruise speed is varied from 200 knots to 300 knots, for a mission with 250 nm range.

Observations

The impact of technology and mission variations is generally consistent for the concept aircraft considered — quadrotor, QSMR, side-by-side, and tiltrotor configurations, with turboshaft and electric propulsion.

Technology

Weight technology factors. The baseline designs include factors on the weight estimation equations that calibrate the model to the actual weights of small rotorcraft, plus weight reductions due to technology advances, particularly materials and design practices. These designs are compared to those with only the calibration — for high, baseline, and low calibration. 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.

Rotor support or wing. For the quadrotor and side-by-side configurations, the rotor support weight is important, reflecting the aircraft growth factor. The tiltrotor wing weight variation has a small impact on the aircraft, though more for the electric configuration.

Systems and equipment weight. Moderate impact of weight changes, reflecting the aircraft growth factor.

Aircraft drag. 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. Except for the tiltrotor, the baseline aircraft drag is high, because of the contributions from the hubs and rotor supports. For the tiltrotor, there is more impact of aircraft drag changes, as an increment of 2 ft² is a larger fraction of the total drag.

Blade loading or profile power. The prescribed changes reflect possible further optimization of the rotor design, such as through airfoil modifications. Small impact for turboshaft configurations, more for electric due to the larger growth factor. The impact is more for weight than for power. There is less impact for the tiltrotor, since the proprotor is in axial flight for cruise. 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.

Turboshaft engine. Small impact of weight variation. Little impact of specific fuel consumption, since cruise efficiency is not very important given the short range of the design mission.

Electric motor. Small impact of weight variation, since the designs use light-weight motors. Moderate impact of motor efficiency, since efficiency has direct influence on energy used and hence battery weight. Little impact of wire weight and thermal management system variations, even to three times the baseline weight, because of the small fraction of empty weight given the design assumptions.

Motor torque margin. For the rpm-controlled hexacopter configuration, moderate impact on weight of requiring 75% usable fraction available power (25% control margin), large impact of requiring 55% fraction.

Battery weight. 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 baseline designs use 400 Wh/kg (installed, usable) for the battery weight. The aircraft weight growth can be characterized by doubling of the design gross weight relative the baseline weight. For 75 nm range:

| | 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. For turboshaft propulsion, small impact of range up to 150 nm, 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 aircraft weight growth can be characterized by doubling of the design gross weight relative the baseline weight. For battery specific energy of 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 |

and for 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 |

Battery specific weight and mission range are crucial trades for electric vehicle design.

Reserve time. For turboshaft propulsion, little impact. For electric propulsion, moderate impact of 20 min reserve, large impact of 45 min reserve (for 75 nm range), because of battery weight growth.

Climb to cruise altitude and one-engine-inoperative requirement. Little impact of rate-of-climb to cruise altitude with one-engine-inoperative requirement, although 900 ft/min climb requires bit more power than OEI. Without OEI requirement, engine sized by climb for rates above about 500 ft/min (except for tiltrotor). Generally there is a moderate impact of the one-engine-inoperative requirement.

Cruise climb. Little impact of requirement for climb rate in cruise conditions, since this design requirement does not size the engine.

Altitude. Small impact of takeoff and cruise altitude.

Tiltrotor cruise speed. For the tiltrotor with turboshaft propulsion, small impact of cruise speed up to 250 knots, moderate impact at 300 knots.

Overall

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, and technology advances (through materials and design) are important to minimize weight and power, hence cost.

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

Batteries are heavy, so any technology or mission variation that increases the energy requirement of the electric configurations has a substantial impact on aircraft size, to the point where the designs do not close for high range or low battery specific energy.

Alternate Designs of AAM Concept Vehicles

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 (Figure 4). 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 78 shows key attributes of the eight aircraft (weight in lb, power in hp, speed in knots, ROC in ft/min). Table 11 gives the size, weight, performance, and mission characteristics; and Table 12 gives the weight statements at the group level.

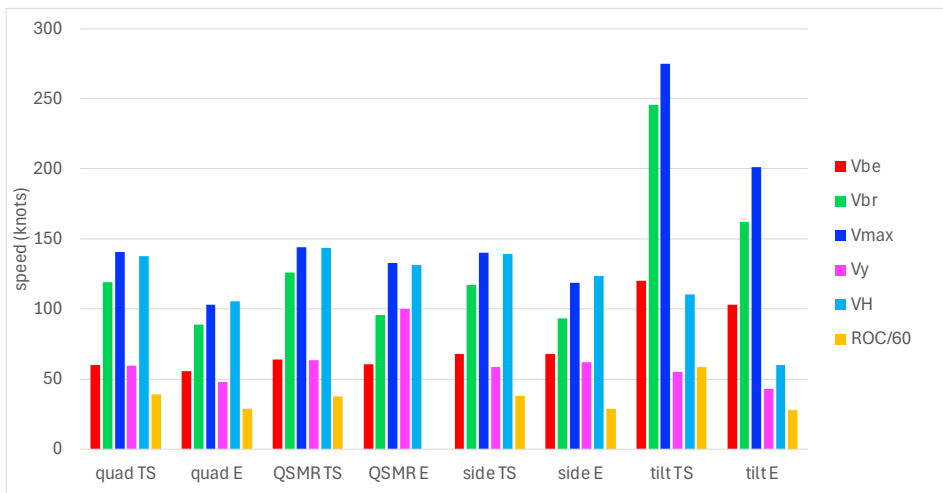
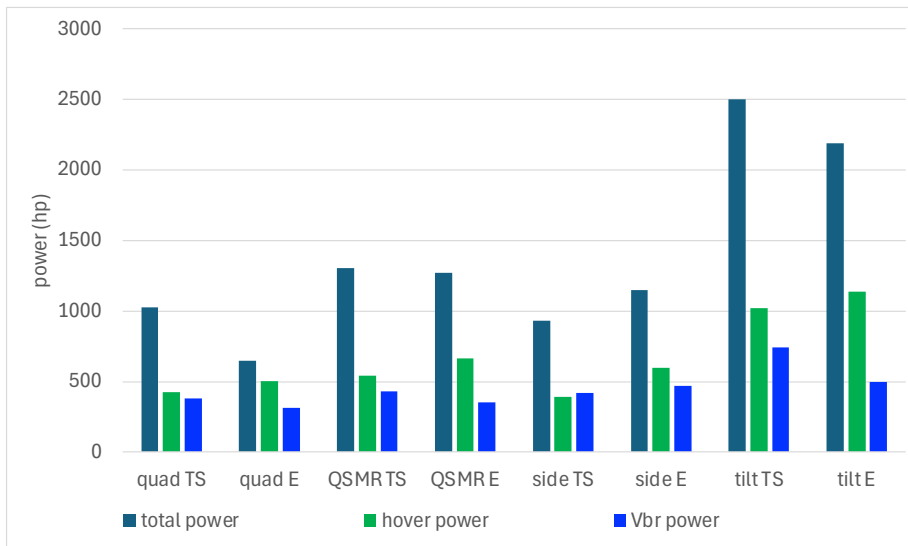
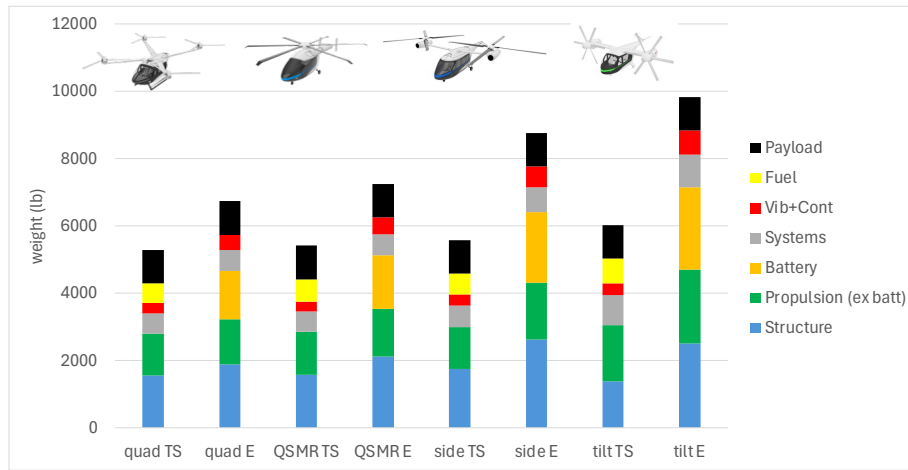


Figure 78. Alternate AAM Concept Vehicles: key attributes.

Table 11. Alternate AAM Concept Vehicles: size, weight, performance, and mission characteristics.

| | quad TS | quad E | QSMR | QSMR E | side TS | side E | tilt TS | tilt E |
|---------------------------------------|---------|--------|--------|--------|---------|--------|---------|--------|
| Size | | | | | | | | |
| number main rotors | 4 | 4 | 1 | 1 | 2 | 2 | 2 | 2 |
| disk loading lb/ft ² | 3.5 | 3 | 3.5 | 3 | 3.5 | 3.5 | 16 | 8 |
| design C_w/σ | 0.09 | 0.09 | 0.09 | 0.09 | 0.09 | 0.09 | 0.13 | 0.13 |
| blade radius ft | 10.98 | 13.37 | 22.22 | 27.75 | 15.94 | 19.97 | 7.75 | 13.99 |
| solidity (T-weighted) | 0.0967 | 0.0828 | 0.0967 | 0.0828 | 0.0967 | 0.0967 | 0.3059 | 0.1530 |
| number blades | 5 | 5 | 6 | 6 | 4 | 4 | 6 | 6 |
| tip speed ft/sec | 450 | 450 | 450 | 450 | 450 | 450 | 450 | 450 |
| number engines | 2 | 4 | 2 | 2 | 2 | 2 | 2 | 2 |
| total power MRP SLS hp | 1024 | 649 | 1302 | 1271 | 934 | 1151 | 2501 | 2188 |
| power MCP SLS hp | 445 | 108 | 566 | 424 | 406 | 384 | 1088 | 729 |
| sfc MCP SLS lb/hp-hr | 0.502 | 0.000 | 0.500 | 0.000 | 0.503 | 0.000 | 0.493 | 0.000 |
| engine weight lb | 561 | 216 | 576 | 317 | 557 | 290 | 642 | 514 |
| engine weight/power lb/hp | 0.55 | 0.33 | 0.44 | 0.25 | 0.60 | 0.25 | 0.26 | 0.23 |
| drive limit hp | 574 | 593 | 619 | 665 | 625 | 840 | 2035 | 1501 |
| fuel tank capacity lb | 586 | | 666 | | 619 | | 734 | |
| battery capacity kWh | | 263 | | 293 | | 387 | | 452 |
| aircraft drag D/q ft ² | 9.7 | 14.1 | 6.8 | 8.8 | 8.3 | 12.2 | 4.3 | 6.5 |
| $D/q/(W/1000)^{2/3}$ | 3.19 | 3.94 | 2.22 | 2.35 | 2.63 | 2.86 | 1.30 | 1.42 |
| hover download DL/W | 0.000 | 0.000 | 0.048 | 0.029 | 0.056 | 0.058 | 0.110 | 0.080 |
| Weight | | | | | | | | |
| DGW lb | 5302 | 6740 | 5431 | 7256 | 5589 | 8771 | 6040 | 9840 |
| weight empty lb | 3705 | 5735 | 3755 | 6252 | 3960 | 7762 | 4297 | 8830 |
| structure lb | 1557 | 1887 | 1583 | 2121 | 1744 | 2633 | 1385 | 2502 |
| propulsion (ex batt) lb | 1254 | 1345 | 1276 | 1422 | 1243 | 1686 | 1673 | 2202 |
| battery lb | | 1429 | | 1590 | | 2099 | | 2452 |
| systems lb | 598 | 615 | 596 | 619 | 656 | 723 | 896 | 968 |
| Performance | | | | | | | | |
| hover power hp | 428 | 506 | 543 | 662 | 390 | 599 | 1019 | 1139 |
| cruise power (V_{br}) hp | 382 | 313 | 429 | 352 | 419 | 467 | 740 | 498 |
| best range speed knots | 119 | 89 | 126 | 96 | 117 | 93 | 246 | 162 |
| best endurance speed knots | 60 | 56 | 64 | 60 | 68 | 68 | 120 | 103 |
| maximum speed knots | 141 | 103 | 144 | 133 | 140 | 119 | 275 | 201 |
| climb speed V_y knots | 60 | 48 | 63 | 100 | 59 | 62 | 55 | 43 |
| maximum speed V_H knots | 137 | 105 | 143 | 131 | 139 | 124 | 110 | 60 |
| Mission | | | | | | | | |
| payload lb | 1000 | 1000 | 1000 | 1000 | 1000 | 1000 | 1000 | 1000 |
| range nm | 250 | 40 | 250 | 40 | 250 | 40 | 250 | 40 |
| block time min | 143 | 31 | 137 | 30 | 147 | 30 | 62 | 17 |
| block speed knots | 105 | 78 | 109 | 80 | 102 | 80 | 242 | 144 |
| productivity $W_{pay}V/W_0$ ton-kt/lb | 0.0141 | 0.0097 | 0.0145 | 0.0063 | 0.0128 | 0.0051 | 0.0246 | 0.0082 |
| trip cost \$ | 2632 | 418 | 2830 | 525 | 2634 | 551 | 1974 | 438 |
| DOC \$/ASM | 1.83 | 1.82 | 1.97 | 2.28 | 1.83 | 2.40 | 1.37 | 1.90 |
| purchase cost \$M | 5.84 | 3.41 | 6.66 | 4.85 | 5.55 | 4.83 | 10.34 | 7.48 |
| maintenance cost \$/hr | 815 | 635 | 905 | 805 | 792 | 850 | 1282 | 1131 |

Table 12. Alternate AAM Concept Vehicles: weight statement.

| | quad TS | quad E | QSMR | QSMR E | side TS | side E | tilt TS | tilt E |
|---------------------------|-------------|-------------|-------------|-------------|-------------|-------------|-------------|-------------|
| DGW | 5302 | 6740 | 5431 | 7256 | 5589 | 8771 | 6040 | 9840 |
| WEIGHT EMPTY | 3705 | 5735 | 3755 | 6252 | 3960 | 7762 | 4297 | 8830 |
| STRUCTURE | 1557 | 1887 | 1583 | 2121 | 1744 | 2633 | 1385 | 2502 |
| wing group | | | | | 316 | 514 | 206 | 471 |
| rotor group | 738 | 985 | 588 | 910 | 553 | 969 | 255 | 731 |
| blade assembly | 291 | 395 | 363 | 522 | 383 | 648 | 147 | 380 |
| hub & hinge | 118 | 190 | 226 | 388 | 170 | 321 | 107 | 351 |
| rotor support | 329 | 401 | 166 | 258 | 32 | 56 | 36 | 116 |
| empennage group | | | 76 | 129 | 22 | 38 | 23 | 94 |
| fuselage group | 507 | 595 | 548 | 666 | 559 | 756 | 547 | 759 |
| alighting gear | 250 | 293 | 218 | 264 | 222 | 299 | 233 | 323 |
| engine section/nacelle | 45 | 13 | 46 | 24 | 51 | 38 | 106 | 123 |
| air induction | 17 | | 17 | | 13 | | 15 | |
| PROPULSION GROUP | 1254 | 2774 | 1276 | 3012 | 1243 | 3785 | 1673 | 4655 |
| engine system | 684 | 257 | 706 | 350 | 684 | 327 | 813 | 565 |
| fuel system (battery) | 103 | 1953 | 113 | 2111 | 106 | 2806 | 127 | 3244 |
| drive system | 467 | 565 | 457 | 550 | 454 | 653 | 733 | 845 |
| SYSTEMS AND EQUIP | 598 | 615 | 596 | 619 | 656 | 723 | 896 | 968 |
| flight controls | 112 | 127 | 104 | 122 | 162 | 217 | 364 | 478 |
| cockpit controls | 10 | 10 | 10 | 10 | 10 | 10 | 10 | 10 |
| automatic flight controls | 40 | 40 | 40 | 40 | 40 | 40 | 40 | 40 |
| system controls | 62 | 77 | 54 | 72 | 112 | 167 | 314 | 428 |
| instruments group | 20 | 20 | 20 | 20 | 20 | 20 | 20 | 20 |
| hydraulic group | 8 | | 10 | | 13 | | 63 | |
| electrical group | 97 | 103 | 99 | 107 | 99 | 112 | 93 | 104 |
| avionics (MEQ) | 110 | 110 | 110 | 110 | 110 | 110 | 110 | 110 |
| furnish & equipment | 150 | 150 | 150 | 150 | 150 | 150 | 150 | 150 |
| environmental control | 75 | 75 | 75 | 75 | 75 | 75 | 75 | 75 |
| anti-icing group | 26 | 29 | 28 | 34 | 27 | 39 | 21 | 32 |
| VIBRATION | 111 | 172 | 113 | 188 | 119 | 233 | 129 | 265 |
| CONTINGENCY | 185 | 287 | 188 | 313 | 198 | 388 | 215 | 442 |
| FIXED USEFUL LOAD | 10 | 5 | 10 | 5 | 10 | 10 | 10 | 10 |
| OPERATING WEIGHT | 3715 | 5740 | 3765 | 6257 | 3970 | 7772 | 4307 | 8840 |

Conclusions

The impact of technology and mission variations on the NASA AAM concept aircraft has been explored. The aircraft considered are quadrotor, quiet single main rotor, side-by-side, and tiltrotor configurations, with turboshaft and electric propulsion variants for each. The technology excursions include: growth factor; weight technology factors (with and without weight reductions due to technology, including high and low calibrations); systems and equipment weight; airframe drag; rotor design blade loading; rotor profile power; turboshaft engine weight and specific fuel consumption; motor weight and efficiency; wire weight; thermal management systems design heat rejection (motor and battery); torque margin for rpm-controlled multicopter; and battery weight (installed usable Wh/kg). The mission excursions include: range (together with battery weight for electric variants); reserve time; rate-of-climb to cruise altitude, with and without one-engine inoperative requirement; rate-of-climb at cruise; takeoff and cruise altitude; and tiltrotor cruise speed.

The impact of technology and mission variations is generally consistent for the concept aircraft considered.

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, and technology advances (through materials and design) are important to minimize weight and power, hence cost.

For the electric aircraft, there is a large impact of battery specific energy (Wh/kg), and correspondingly a large impact of mission range. The impact of the reserve time requirement is small for turboshaft propulsion; for electric propulsion, the impact is moderate at 20 min reserve, large at 45 min.

The impact of systems and equipment weight is moderate, reflecting the weight growth factor. The impact of the one-engine-inoperative requirement is moderate. For turboshaft propulsion, there is a small impact of mission range up to 150 nm, a moderate impact at 500 nm. For electric aircraft there is a moderate impact of motor efficiency. For the tiltrotor with turboshaft propulsion, the impact of cruise speed is small up to 250 knots, moderate at 300 knots. For the rpm-control hexacopter configuration, there is a moderate impact of requiring 75% usable fraction of available power (25% control margin).

Alternate designs were developed for the concept vehicles, considering specific missions that can take advantage of the strengths of individual aircraft configurations: for turboshaft aircraft, longer unrefueled range (250 nm instead of 75 nm); including faster cruise speed for the tiltrotor (275 knots); and for electric aircraft shorter range (40 nm instead of 75 nm) and more realistic battery weight (250 Wh/kg instead of 400 Wh/kg).

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Nomenclature

Acronyms

| | |
|-------|---|
| AAM | Advanced Air Mobility |
| ASM | available seat miles |
| CRP | contingency rated power (typically 2.5 min) |
| DGW | design gross weight |
| E | electric aircraft |
| IRP | intermediate rated power (typically 30 min) |
| ISA | international standard atmosphere |
| MCP | maximum continuous power |
| MRP | maximum rated power (typically 10 min) |
| NOTAR | no-tail-rotor anti-torque system |
| OEI | one-engine inoperative |
| quad | quadrotor |
| QSMR | quiet single main rotor |
| ROC | rate of climb |
| RVLT | Revolutionary Vertical Lift Technology |
| sfc | specific fuel consumption |
| side | side-by-side aircraft |
| SL | sea-level |
| SLS | sea-level standard |
| SOA | state-of-the-art |
| tilt | tiltrotor |
| TO | takeoff |
| TS | turboshaft aircraft |
| UAM | Urban Air Mobility |
| VTOL | vertical take-off and landing |

Symbols

| | |
|--------------------|--|
| A | rotor disk area, πR^2 |
| A_{blade} | total blade area |
| C | charge capacity (Wh or MJ) |
| C_T | rotor thrust coefficient, $T/\rho A V_{\text{tip}}^2$ |
| C_W | aircraft weight coefficient, $W/\rho A V_{\text{tip}}^2$ |
| D/q | drag divided by dynamic pressure |
| DL | download (fraction aircraft weight) |
| 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_H | maximum speed (100% MCP) |
| V_{max} | maximum speed (power required = 90% MCP) |
| V_{tip} | rotor tip speed |
| V_y | best rate-of-climb speed (100% MRP) |
| W | weight |
| W_E | aircraft empty weight |
| W_{MTO} | maximum takeoff weight |
| W_O | aircraft operating weight (empty weight+fixed useful load) |
| W_{pay} | payload weight |
| ρ | air density |
| σ | rotor solidity, A_{blade}/A |